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ENGINEERING DESIGN HANDBOOK. HELICOPTER ENGINEERING. PART ONE. PRELIMINARY DESIGN

Army Materiel Command
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ENGINEERING DESIGN HANDBOOK

HELICOPTER ENGINEERING

PART ONE

PRELIMINARY DESIGN

HEADQUARTERS, U.S. ARMY MATERIEL COMMAND AUGUST 1974

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DEPARTMENT OF THE ARMY
HEADQUARTERS UNITED STATES ARMY MATERIEL COMMAND
5001 Eisenhower Ave, Alexandria, VA 22333

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ENGINEERING DESIGN HANDBOOK
HELICOPTER ENGINEERING, PART ONE
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FOREWORD

The *Helicopter Engineering* Handbook forms a part of the Engineering Design Handbook Series which presents engineering data for the design and construction of Army equipment.

This volume, AMCP 706-201, *Preliminary Design*, is Part One of a three-part Engineering Design Handbook titled *Helicopter Engineering*. Along with AMCP 706-202, *Detail Design* and AMCP 706-203, *Qualification Assurance*, this part is intended to set forth explicit design standards for Army helicopters, to establish qualification requirements, and to provide technical guidance to helicopter designers in both the industry and within the Army.

This volume, AMCP 706-201, discusses the characteristics and subsystems which must be considered during preliminary design of a helicopter. Additionally, possible design problems encountered during helicopter design are discussed and possible solutions suggested. The volume is divided into 14 chapters and is organized as described in Chapter 1, the introduction to the volume.

AMCP 706-202 deals with the evolution of the vehicle from an approved preliminary design configuration. As a result of this phase, the design must provide sufficient detail to permit construction and qualification of the helicopter in compliance with the approved detail specification and other requirements. Design requirements for all vehicle subsystems also are included in AMCP 706-202.

The third volume of the handbook, AMCP 706-203, defines the requirements for airworthiness qualification of the helicopter and for demonstration of contract compliance. The test procedures used by the Army in the performance of those additional tests required by the Airworthiness Qualification Program to be performed by the Army also are described.

PREFACE

This volume, AMCP 706-201, *Preliminary Design*, is the first section of a three-part engineering handbook, *Helicopter Engineering*, in the Engineering Design Handbook series. It was prepared by Forge Aerospace, Inc., Washington, D.C., under subcontract to the Engineering Handbook Office, Duke University, Durham, N.C.

The Engineering Design Handbooks fall into two basic categories, those approved for release and sale, and those classified for security reasons. The Army Materiel Command policy is to release these Engineering Design Handbooks to other DOD activities and their contractors, and other Government agencies in accordance with current Army Regulation 70-31, dated 9 September 1966. It will be noted that the majority of these Handbooks can be obtained from the National Technical Information Service (NTIS). Procedures for acquiring these Handbooks follow:

a. Activities within AMC, DOD agencies, and Government agencies other than DOD having need for the Handbooks should direct their request on an official form to:

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(Requests for classified documents must be sent, with appropriate "Need-to-Know" justification, to Letterkenny Army Depot.)

Users of the handbook are encouraged to contact USAAVSCOM, St. Louis, Mo., Flight Standards and Qualification Division (AMSAV-EFI), with their recommendations and comments concerning the handbook. Comments should be specific and include recommended text changes and supporting rationale. DA Form 2028, Recommended Changes to Publications (available through normal publications supply channels) may be used for this purpose. A copy of the comments should be sent to:

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Alexandria, VA 22333

Revisions to the handbook will be made on an as-required basis and will be distributed on a normal basis through the Letterkenny Army Depot.

CHAPTER 1

INTRODUCTION

1-0 INTRODUCTION

This volume, *Preliminary Design*, is Part One of a three-part engineering design handbook series on helicopter engineering. The other two parts are AMCP 706-202, *Detail Design*, and AMCP 706-203, *Qualification Assurance*.

Preliminary design is the initial engineering activity in the development of any new weapon system. The capability defined by the U.S. Army, first in the Required Operational Capability (ROC), and then, more specifically, in a Request for Quotation (RFQ) or Request for Proposal (RFP), is the basis for the development of a new system. During preliminary design the characteristics necessary to provide the required mission capability are combined with the characteristics necessary to insure an acceptable cost of ownership. Since these characteristics sometimes tend to conflict, the preliminary design process comprises many tradeoffs and compromises, offering a challenge to the creative capability of the contractor's engineering organizations.

Detail design is that portion of the design cycle that commences upon award of a development contract. During detail design all subsystems and components must be defined completely, by specifications and/or drawings, so that the helicopter can be assembled and qualified. The requirements unique to the detail design are discussed in AMCP 706-202, while the requirements for qualification are described in AMCP 706-203.

Helicopters are defined, for the purposes of this handbook, as those aircraft which derive both lift and propulsive force from a powered rotary wing and have the capability to hover and to fly rearward and sideward, as well as forward. Existing configurations used by the Army include a single lifting rotor with an antitorque rotor, and tandem lifting rotors. A compound helicopter is a helicopter which incorporates fixed-wing surfaces to partially unload the lifting rotor and/or additional thrust producing devices. Such devices supplement the thrust-producing capability of the lifting rotor(s).

Army helicopters are classified with respect to the general mission they are developed to accomplish. Classes of interest are:

1. Attack Helicopter (AH). A fast, highly maneuverable, heavily armed machine used for combat fire support and helicopter escort missions. The attack helicopter may well be a compound vehicle, i.e., with auxiliary forward propulsion and/or a stub wing used to unload the main rotor in high-speed flight.

2. Cargo Helicopter (CH). Medium- and heavy-lift classes that are intended primarily for heavy load-carrying missions. The loads may be carried internally or externally. These helicopters generally have a wide range of center of gravity (CG) travel.

3. Observation Helicopter (OH). A small, light machine that may be useful for a variety of missions including surveillance, target acquisition, command and control, etc. Light armament may be installed.

4. Training Helicopter (TH). A small helicopter usually with seating only for instructor and student-pilot, or a helicopter of one of the other mission classes specifically assigned to the training mission.

5. Utility Helicopter (UH). A class that is assigned a wide variety of missions such as medical evacuation, transporting personnel, and/or light cargo loads. Speed and maneuverability are required in order to minimize vulnerability when operating over hostile territory.

This handbook discusses the design requirements applicable to Army helicopters for all missions under visual flight rule (VFR) operation, day or night. As such, the scope of this document has been limited to cover the basic aerial vehicles. Design requirements for mission-essential equipment—e.g., weapons, sensors, cargo-handling equipment—are beyond this scope and are not discussed, although the helicopter-integral interface requirements for such equipment are included. The design of power plants, batteries, generators or alternators, and similar components, are also beyond the scope of the handbook.

The US Army's mission assignments in land warfare result in an operational environment that is severe.

almost infinitely variable, and yet unique. The Army must be prepared to operate anywhere in the world in climate that is hot or cold, in atmosphere that is humid or dry, and on terrain that is rugged or level. The Army's materiel and equipment, including its helicopters, must be capable of effective performance in any of these environments. Furthermore, the helicopters must operate with the combat force, making regular use of unprepared or rapidly cleared sites. Servicing and organizational maintenance will be performed in similar surroundings. The Army helicopter, therefore, must be sufficiently rugged to perform its mission effectively under such diverse operating conditions, yet be mechanically simple in order to be maintained easily without specialized facilities and equipment or highly skilled personnel.

In addition to being able to perform its assigned mission effectively in a variety of operational environments, the Army helicopter must be designed for safety and survivability. Safety can be enhanced by design. Hazards must be identified and appropriate action taken to minimize both the probability of occurrences and the consequences of equipment failure and/or human error. Provision must be made for the protection of personnel following damage to the helicopter whether as a result of combat-related accidents, or normal peacetime operations. Therefore, as this handbook is concerned with design and development of helicopters for the U.S.Army, it discusses not only the methods of providing the performance characteristics essential to assigned missions, but also those additional design requirements and characteristics necessary for survivability and safety.

This preliminary design volume is organized in a functional manner. The design process requires the coordinated activities of a number of engineering disciplines. In order to facilitate the evaluation of the suitability or adequacy of a given design by specialists in each applicable discipline, this volume has been arranged to discuss the various disciplines in separate chapters. The integration of individual mission and design requirements into a complete and coordinated design is treated from two important, but different, points of view in separate chapters. This document may be used as a guide for the development and/or review of a proposed response to a Government solicitation.

In Chapter 2 the definition and determination of mission effectiveness are discussed. The performance capabilities must be combined with the availability for assignment to obtain a meaningful measure of mission effectiveness. The design alternatives available, meanwhile, permit a given level of effectiveness to be achieved at various levels of cost. The final criterion for selection of a candidate design is its cost effectiveness, with the cost of prime interest being life-cycle cost.

Chapter 3 discusses helicopter performance. Included are the fundamentals of rotary-wing aerodynamics and the inherent relationships between basic design parameters with specific mission performance requirements defined. A parametric analysis can be performed to select or define the geometric and aerodynamic characteristics of the design. With these characteristics known, detailed performance can be calculated over the entire operational envelope.

Chapter 4 treats structural design. The applicable design criteria are defined and discussed. The application of these criteria to a given design with specific performance results in the definition of loads pertinent to all structural components. The static strength adequacy of the components is then determined by stress analysis. A large number of helicopter components are subject to oscillatory or repeated loadings of sufficient magnitude that fatigue rather than static strength is the critical structural design consideration. Methods of fatigue life determination are discussed.

Chapter 5 discusses the dynamics of helicopters. Of particular concern are the potential instabilities of those parts of the helicopter—both rotating and nonrotating—that are subject to oscillating loads. Methods of analysis of the responses of these systems and the design requirements and techniques for preventing these instabilities are discussed.

Chapter 6 discusses handling qualities, or those characteristics of the helicopter which result from and depend upon the interrelation between its stability and control characteristics. Design requirements are defined separately for stability and control, but the intent is the achievement of flying qualities appropriate to the assigned mission of the helicopter.

Chapters 7, 8, and 9 discuss the design requirements pertinent to those subsystems not previously described in detail that are fundamental to the operation and performance of the helicopter. These are the drive system, the power plant and its supporting subsystems, and secondary power subsystems.

Chapter 10 discusses the subject of weight and balance which is essential to any aircraft. Included are discussions of weight and balance determination methods and weight control guidelines pertinent to prelir nary design.

Chapter 11 discusses maintainability requirements. Included are those considerations essential during the design phases to assure maximum maintainability of the helicopter. Chapter 12 discusses the characteristics of reliability and availability. Those design considerations and requirements that contribute to the achievement of required levels of reliability and availability also are reviewed.

In Chapter 13 the integrated design is discussed. The selection of an arrangement and configuration includes the consideration of all pertinent design requirements—whether they pertain to the complete helicopter, the individual subsystem, or the personnel who will operate and maintain it. The requirements applicable to the complete helicopter depend upon the assigned missions and the operational environment, and also upon the capability to be delivered to the assigned duty. Therefore transportability may impose restraints upon operational characteristics or upon the methods of manufacture and assembly of the helicopter.

The final chapter (Chapter 14) of this part of the handbook defines the documentation which may be required to define the proposed design. A given helicopter program will probably not require all the documentation listed. The intent of this chapter is to provide a definition of the information needed to permit a detailed evaluation of all possible characteristics of a proposed design.

Throughout this volume, the mandatory design requirements have been identified with the contractual language that makes use of the imperative word "*shall*". To assist in the use of this document in the planning of a helicopter program, the word "*shall*" has been italicized in the statement of each such requirement.

Since the preliminary design requirements for individual programs may vary from the level of detail described in this volume, the procuring activity will specify in its Request for Proposal the extent to which these requirements are applicable to the design of a given helicopter.

CHAPTER 2

MISSION EFFECTIVENESS

2-1 INTRODUCTION

The ultimate measure of the value of a helicopter is its effectiveness in performing its assigned mission(s). Until recent years quantitative treatment of the system value concept usually has resulted in considerable disagreement as to what constitute meaningful measurement criteria and evaluation techniques. The demand for greater performance and the corresponding escalation of costs, however, have necessitated the development of better, more useful mission effectiveness estimating methodologies.

Although most actual computations normally are performed as a part of requirements derivation and are refined further during concept formulation, an understanding of mission effectiveness provides design engineers and program management personnel with essential guidance during preliminary design. Their technical decisions must reflect careful consideration of the many parameters and trade-offs that contribute to the ultimate effectiveness of the resulting helicopter.

This chapter provides the general background information about and discusses the philosophy of the mission effectiveness concept, along with a number of approaches and methodologies used in quantification thereof. Detailed instructions relative to application of the techniques have been omitted, however, as being inappropriate for the preliminary design phase and hence beyond the scope of this handbook. Refs. 1-27 and AMCP 706-191 provide depth of detail with respect to specific analytical methods.

2-2 MILITARY SYSTEM REQUIREMENTS AND MISSION EFFECTIVENESS

The helicopter must interface with the other relevant elements, e.g., combat, logistic (or support), command, control, and communication subsystems, within the parent organization or system. In other words, a helicopter must perform its mission as a completely integrated component, providing support for and acting in conjunction and in balance with associated subsystems and components. This concept is depicted in Fig. 2-1.

The inputs to the helicopter are figurative rather than literal in this example; they refer to the requirements and constraints placed upon the vehicle by its association with the other elements. In a sense, the helicopter output is its mission performance. This output contributes in part to the overall output of the parent system, which, in turn, represents mission performance.

Mission effectiveness, then, becomes the measurement of this performance, be it for the parent system or any of its components. Actually, the component variables have the same relevance to this larger system as they do to the subsystem; the primary differences are the degree and level of definition. Although the subsequent paragraphs are oriented toward the helicopter, the overall applicability of the concept to weapon system design should be borne in mind.

2-2.1 MILITARY SYSTEM REQUIREMENTS

Military system requirements may emanate from several sources. Among these sources are:

1. The potential for improved capabilities offered by state-of-the-art advancements
2. Recognized deficiencies based upon operational experience
3. An identified threat for which no counterthreat exists
4. Concepts evolved from long-range studies of postulated future operations and from long-range technical forecasts.

The requirements normally will specify what types of missions and levels of performance the system should be capable of delivering. For example, an observation helicopter may be required to perform surveillance, artillery adjustment, command and control, and similar missions. In performing these missions, the helicopter should be capable of attaining given levels of range, speed, and payload for each type of mission. The specification also may require provisions for weapon, surveillance, and/or communication equipment installations.

It is the task of the designer to analyze and assimilate the specifications and to produce—through technical skill, trade-offs, and optimization—the system that has the highest probability of satisfying the military system requirements at the lowest possible cost (Ref. 1).

2-2.1.1 Operations Research (OR)

In most instances the development of quantitative design objectives from the sources listed in par. 2-2.1 is characterized by a series of broad and highly iterative analytical studies. The science of operations research (OR) is one of the principal tools in this effort.

The generalized procedural steps used in OR studies are:

1. Formulation of the problem
2. Establishment of measures of effectiveness
3. Determination and collection of relevant data
4. Construction, exercise, and analysis of a representative model
5. Derivation and interpretation of the solution(s)
6. Verification of the solution(s) by experiment
7. Recommendations and implementation (Refs. 2 and 3).

The actual analytical techniques and methodologies that are used in each step are far too numerous to catalog or define here. For example, Step 4 could be implemented by one or a combination of the following OR procedures:

1. Mathematical modeling
2. Simulation (manual or automated)
3. War gaming
4. Field experiments
5. Troop testing
6. Cost-effectiveness analysis
7. Human factors analysis and testing
8. Systems analysis studies.

Selection of the appropriate technique is dependent to a large degree upon the nature of the particular problem, the available resources, and the skill and philosophy of the analyst.

In general, OR attempts to project a quantified overview of the entire system so that its sensitivity to the particular subsystem or component, e.g., the helicopter, can be measured. Estimates of expected mission performance, requirements for operational and supporting force structures, and system costs are typical outputs of the analyses.

2-2.1.2 Budgetary Considerations

The continual demand for improved performance capabilities, which usually translate into increased complexity with an attendant growth in total system cost, has magnified the importance of budgetary considerations in recent years. The use of cost-effectiveness methodologies in OR normally allows the projection of a first-order estimate of the costs necessary to the postulated mission objectives.

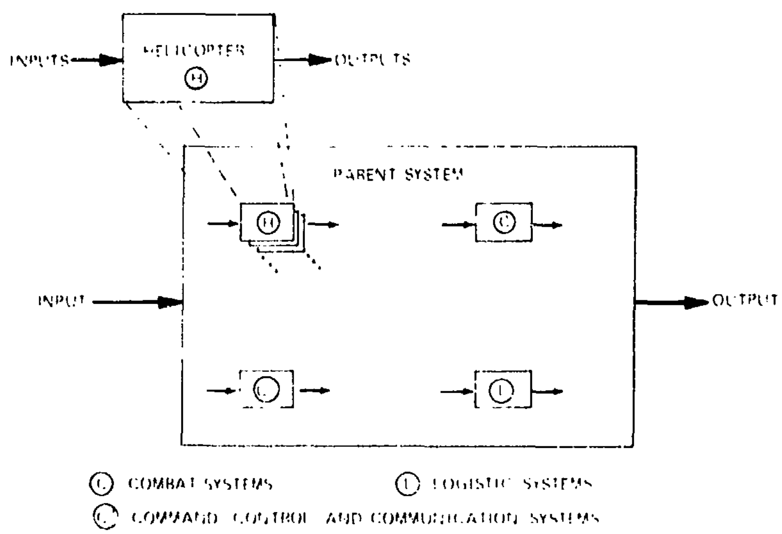


Fig. 2-1. Helicopter as a Subsystem

It is important that the fiscal requirements of the system in question be tempered by the demands of other systems within the overall military structure. In turn, the military requirements must be weighed against other national objectives. In other words, some degree of balance must be achieved within budgetary constraints. It is not unusual, therefore, to find that many desirable proposed military system requirements must be relaxed or abandoned because of budgetary considerations. This circumstance forces the establishment of priorities to insure that the essential characteristics of the system are retained without undue degradation of its capabilities.

Setting priorities is one of the most critical tasks in the formulation of military system requirements. The user must be made to realize that practical constraints exist. For example, a particular desired capability might result in severe weight penalties, exceed the current state-of-the-art, or be prohibitively costly. Thus, the trade-offs must be weighed before a firm statement of need is finalized. The user should be expected to temper the statement of requirements accordingly, but this should not imply a stifling influence upon the ultimate conceptual design. Rather, it demands that sound bases and rationales be developed to justify and validate each of the system requirements. Further, evidence must be presented to indicate that appropriate interfaces with, and sensitivities of, other systems and the budget have been analyzed.

2-2.2 DESIGN CYCLE

In the preliminary design phase, the user's mission requirements are translated into a general configuration. The purpose is to determine whether the military system requirements are obtainable within the constraints of money and state-of-the-art. A degree of risk or uncertainty usually is present in new designs. This risk can be tolerated when the budget will permit adequate supporting research and/or development concurrently.

Corporate design philosophy and experience influence the preliminary design phase to a considerable extent. Normally, a number of configurations are designed, with each incorporating the proprietary features for which the particular company is noted. In addition, each design reflects the company's interpretation and understanding of the military system requirements and design criteria.

Fig. 2-2 depicts conceptually the manner in which all elements of a design are brought together to satisfy the user's requirements. This process is repeated separately for each proposed configuration and consists of considerable trade-off, iteration, and compromise, even in the preliminary design phase. The result is the basis for an optimized configuration that can be translated into preliminary hardware design.

The next step is to compare the candidate designs in order to select the one that best appears to meet the user or military system requirements.

Practical or "real-world" constraints, e.g., cost, state-of-the-art, threat, and complexities arising from the trade-offs associated with hardware design, may render some of the military system requirements infeasible or incompatible. Provisions must be made to permit re-evaluation, modification, or revisions as necessary. Unreasonable adherence to unrealistic or obsolete requirements eventually may cause serious degradation of mission effectiveness by unduly compromising the helicopter.

2-2.3 MISSION-EFFECTIVENESS EQUATION

One standard approach to the problem of measuring the mission effectiveness of a helicopter is to consider four primary, measurable factors or variables—three operational and one economic. The operational factors, as defined in AMCP 706-134, are mission readiness, survivability, and overall performance; the economic variable is cost.

The operational factors are defined as follows (Ref. 5):

1. Mission readiness. A measure of the degree to which an item is operable and committable at the start of a particular mission, when the mission is called for at an unknown (random) point in time.

2. Survivability. A measure of the degree to which an item will withstand a hostile, manmade environment without suffering abortive impairment of its ability to accomplish its designated mission.

3. Performance (overall). A measure of how well an item executes its designated mission, i.e., the output resulting from the use of the item.

The economic factor, or cost, essentially is an expression of the resources—measured in dollars—required to design, produce, test, and operate an item during its life cycle.

The ultimate objective of the design cycle is the optimization of all helicopter system components to maximize the contribution of each factor to mission effectiveness. In this form the objective appears to be somewhat abstract and difficult to visualize in concrete, quantitative terms. The design phases, however, demand that mission readiness, survivability, performance, and cost be translated into terms that will guide

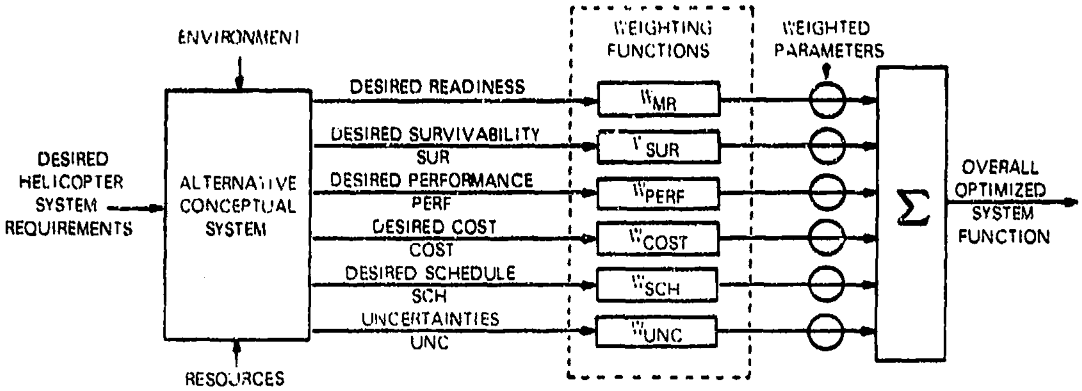


Fig. 2-2. Configuration Composition and Optimization (Ref. 4)

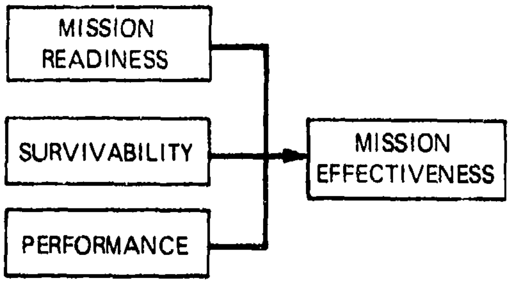


Fig. 2-3. Contribution of Operational Factors to Mission Effectiveness

the detailed development of a helicopter with maximized mission effectiveness.

2-2.3.1 Operational Factors

The operational factors—mission readiness, survivability, and overall performance—are the primary contributions to mission effectiveness. The relationship may be visualized by the schematic shown in Fig. 2-3.

Mathematically, this relationship may be illustrated by the following conceptual, probabilistic equation

$$\text{Mission Effectiveness} = f(\text{Mission Readiness}, \text{Survivability}, \text{Performance}) \quad (2\text{-}1)$$

Mission effectiveness is stated as the product of the factors, each of which can be expressed as a probability P. Each factor must be maximized independently with respect to the requirements, i.e., P approaches 1. Because in practice the ideal $P = 1$ can never be reached, the importance of maximizing the contribution of each factor during design is emphasized.

2-2.3.2 Economic Factor

The economic factor, or cost, associated with mission-effectiveness considerations conceptually assumes the role of a common denominator that relates the operational factors. In other words, a specific amount of mission effectiveness per dollar can be obtained from the contribution of specific values of mission readiness, survivability, and performance. Similarly, levels of each are achieved at a dollar cost, e.g., mission readiness per dollar, survivability per dollar. This concept is illustrated by Eq. 2-2:

$$\frac{\text{Mission Effectiveness}}{\text{Dollar}} = f\left(\frac{\text{Mission Readiness}}{\text{Dollar}}, \frac{\text{Survivability}}{\text{Dollar}}, \frac{\text{Performance}}{\text{Dollar}}\right) \quad (2\text{-}2)$$

Here again, the equation is conceptual and cannot necessarily be quantified. It does illustrate, however, the necessity of bringing all facets of the design effort into an overall degree of technical harmony and economic balance, so that the end result can be considered a successful configuration. For example, excessive performance at the expense of mission readiness clearly is undesirable, especially if the costs of achieving the uncalled-for level of capability are high.

Techniques have been developed that attempt to bring all of the relevant operational and economic factors into acceptable compatibility. A most useful concept in this regard is the application of the cost-effectiveness methodologies that are detailed in subsequent paragraphs.

2-3 OPERATIONAL FACTOR ANALYSIS

Eqs. 2-1 and 2-2 are conceptual and somewhat abstract. Each of the operational factors is far too complex and consists of too much detail to be reduced to such a simplistic mathematical model. Furthermore, the assumption of independence of terms implicit in these equations is questionable from a practical viewpoint; e.g., decisions concerning survivability generally will have considerable effect upon mission readiness and performance, and vice versa. Therefore, it is necessary to make a more detailed examination of the operational factors and the relevant subfactors in order to present a system overview to the helicopter designer.

2-3.1 MISSION READINESS

In the broadest sense, mission readiness can be construed to have two separate, identifiable facets. One may be termed "mission capability" and is defined to mean a configuration containing all of the equipment and facilities necessary to allow conduct of the mission(s) for which it was designed. The other facet is "availability" in the classical sense, i.e., readiness based upon the operational condition of the configuration. The designer is concerned primarily with enhancing or maximizing the availability of specified equipment on the basis of mission capability requirements.

2-3.1.1 Mission Capability (MC)

Mission capability (MC) refers to those design provisions that will permit the helicopter to be available for specific missions required of the military system. For example, it may be desirable that the helicopter have multimission capabilities, such as armed scout and medical evacuation; provisions for rapid and simplified conversion from one mission configuration to another promote readiness of the vehicle to accept new assignments.

Strategic deployability is another example of MC. In the event that deployment is desired at a distance beyond the design performance range, provisions either for rapid disassembly/assembly with maximum dimensions that will permit air transportability or for in-flight refueling equipment may be required. Note: Ferry range is a performance factor rather than a mission readiness factor.

Another aspect of MC might be a requirement for rapid response. The designer, therefore, must weigh self-start capability, freedom from extensive preflight checkout and warmup, etc.

As an MC effectiveness measure, the probabilistic approach does not apply. The helicopter either has the required provision $P = 1$, or it does not, symbolized as $P = 0$. The only realistic measurement that might be applied from the MC standpoint, therefore, is time; i.e., how rapidly can the design provisions be used when called upon?

If the effectiveness of MC items is to be measured, the user should endeavor to establish design criteria in the statement of military system requirements. The use of OR (par. 2-2.1.1) can assist in setting reasonable design objectives.

2-3.1.2 Availability

Availability in the classical sense has two primary subfactors: reliability and maintainability. Reliability, in turn, is related to mean-time-between-maintenance-actions (MTBMA), which is defined as the total functioning life, for a particular interval, of an item divided by the total number of maintenance actions within the population during the measurement interval. Maintainability is related to mean-time-to-repair (MTTR), and is defined as the total corrective maintenance time divided by the total number of corrective maintenance actions during a given period of time (Refs. 5 and 6).

Availability A, therefore, may be derived as a probability from the equation

$$A = f(\text{Reliability, Maintainability})$$

$$= \frac{\text{MTBMA}}{\text{MTBMA} + \text{MTTR}}, \text{ dimensionless} \qquad (2\text{-}3)$$

Fig. 2-4 depicts the typical trade-off between reliability and maintainability as it affects availability. It should be noted that increasing maintainability actually is the inverse of MTTR in this figure. Also, as the MTBMA, and therefore reliability, are improved for a given MTTR, the availability increases. Improved maintainability, or reduced MTTR, also improves availability (AMCP 706-134).

The designer's task, therefore, is clear-cut; any means available should be used to improve both subfactors, provided that unreasonable costs or degradation to the other aspects of mission effectiveness are not incurred. Where possible, the user should identify minimum limits of acceptable availability for the system as shown in Fig. 2-4.

Unfortunately, rapid technological advances and the corresponding demands for greater mission capabilities and higher levels of performance have resulted in marked increases in system complexity, i.e., more ele-

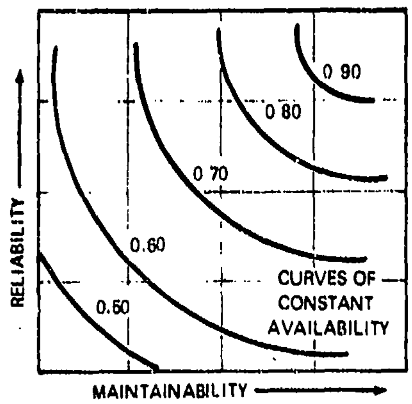


Fig. 2-4. Trade-offs in Reliability and Maintainability (AMCP 706-134)

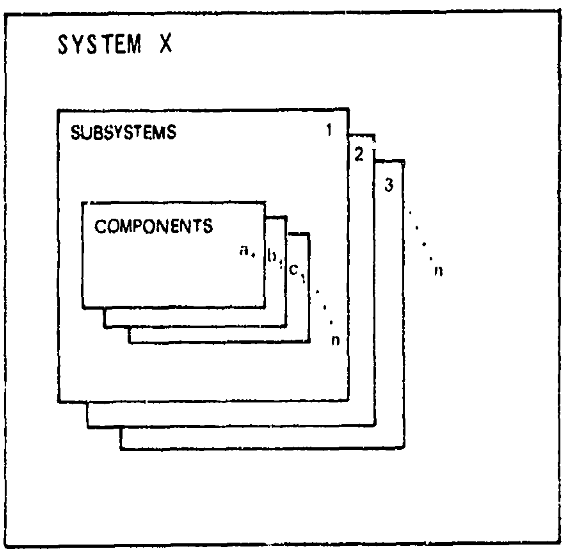


Fig. 2-5. Illustration of System Complexity

ments, which may result in losses in availability (Ref. 6). The proliferation of subsystems, components, etc., is illustrated in Fig. 2-5. This illustrative breakdown could be extended to the most basic element.

System X is composed of a series of subsystems, with each subsystem comprising a series of components, which in turn involve a series of basic elements. The overall system availability is computed as the product of each subsystem availability (assuming independent A_i for simplicity).

$$A_s = \prod_1^n A_i$$

$$= A_1 \cdot A_2 \cdot A_3 \cdots A_n, \text{ dimensionless} \quad (2\text{-}4)$$

where

A_s = system availability, dimensionless

A_i = subsystem availability, dimensionless

Theoretically, the subsystem availability could be computed as a function of the component availability A_{c_j}:

$$A_i = \prod_1^n A_{c_j}$$

$$= A_{c_1} \cdot A_{c_2} \cdot A_{c_3} \cdots A_{c_n}, \text{ dimensionless} \quad (2\text{-}5)$$

The implications of complexity are clear; the greater the number of components, the higher the probability of reduced availability. The designer's objective is to provide the desired levels of mission performance and other factors without an undue proliferation of subsystems and to provide improved availability through increased reliability and maintainability.

2-3.1.2.1 Reliability

Reliability may be defined as the probability that an item will perform satisfactorily for a specified period of time when operated under prescribed conditions (Ref. 6). In a sense, reliability is a measure of the deviation of an item from perfection; i.e., a perfect item would never fail and, thus, would have a reliability probability of $P = 1$. The usual history of failure rates is shown in Fig. 2-6. After initial debugging during the developmental test period, the failure rate for a given item tends to stabilize at a constant level until the item approaches the end of its designed lifetime, when the failure rate increases due to wearout. During the period of normal useful life, the failure rate is relatively constant and occurs by chance.

Generally, as operational time or complexity of an item increases, the failure rate becomes exponential. The complexity relationship is shown in Fig. 2-7 (Refs. 4 and 8).

The relationship between reliability R, failure rate, and mean-time-between-failures (MTBF), (Ref. 7), is given by:

$$R(t) = e^{-\lambda t}, \text{ dimensionless} \quad (2\text{-}6)$$

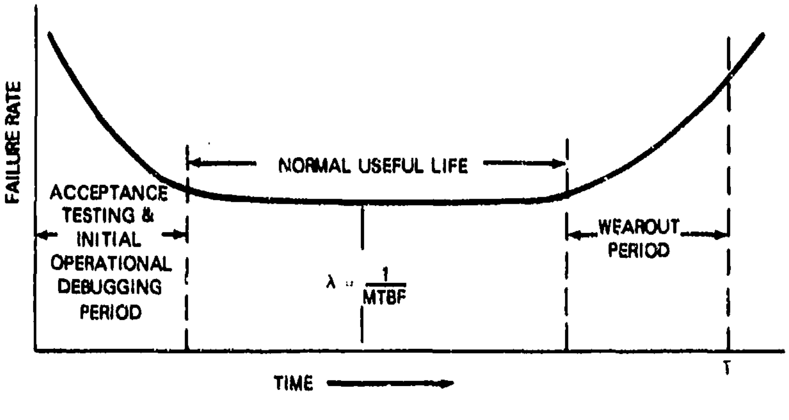


Fig. 2-6. Failure Rate History During Operating Lifetime of Items (Ref. 7)

where

t = time, hr

λ = failure rate = 1/MTBF, hr^{-1}

The effect of complexity upon a system is cumulative—dependent upon the collective reliabilities of its subsystems and components—in the same manner as for availability (Eq. 2-4).

$$R_s = \prod_{i=1}^{n} R_i = e^{-\lambda_s t}$$

$$= e^{-\lambda_1 t} \cdot e^{-\lambda_2 t} \cdot e^{-\lambda_3 t} \cdots e^{-\lambda_n t} \qquad (2\text{-}7)$$

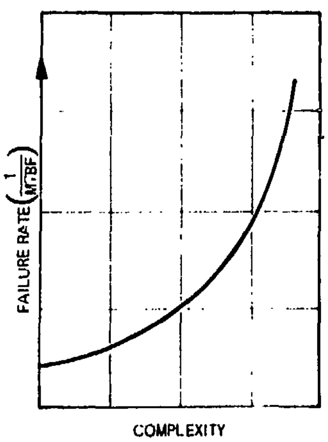


Fig. 2-7. Reliability as a Function of Complexity (Ref. 7)

where

R_i = subsystem reliability, dimensionless

λ_i = subsystem failure rate, hr^{-1}

λ_s = system failure rate, hr^{-1}

R_s = system reliability, dimensionless

The prediction of reliability for a system actually is a far more complex process than Eqs. 2-5, 2-6, and 2-7 would indicate. The implications of Eq. 2-7, however, are clear; the designer must maximize the degree of reliability of each subsystem to achieve satisfactory overall system reliability.

The user, too, has the burden of establishing definitive reliability design objectives as a part of the military system requirements. The problem is basically one of balancing the cost of achieving the desired level of reliability against the costs and effects of failing to perform an assigned mission because of in-service breakdown. The trade-offs are exceedingly difficult in that all potential operational applications and operating environments must be considered.

In order to satisfy established reliability requirements, the designer must optimize his reliability response in terms of weight, size, and cost. To assist in this process, he may select one or more of the reliability design techniques listed in Table 2-1.

Table 2-1 indicates that most techniques involve a weight penalty. The net result of increased system weight, especially when applied to helicopters, is an adverse effect upon weight-sensitive performance factors such as payload, range, and speed. Degradation of performance may lead in turn to reductions in helicopter survivability. This points up again the necessity for considering all elements of a system during design.

The quest for increased reliability continues far beyond the development of the first prototype. Improvements are sought throughout the service life of the system, up to the point at which improvements no longer are cost-effective. A continuing reliability program has other beneficial aspects as well. It provides:

1. A check on the accuracy of the initial design requirements, assumptions, decisions, and predictive techniques
2. An empirical data base for future design
3. A technique for improving initial design shortcomings.

The mechanisms for establishing and operating a reliability improvement program should be planned as an integral part of the helicopter system life cycle.

2-3.1.2.2 Maintainability

Maintenance is defined as all actions necessary for retaining an item in, or restoring it to, a specified condition. Maintainability, the companion term of reliability in determining system availability (Eq. 2-6), is a measure of maintenance effectiveness and is defined as a characteristic of design and installation expressed as the probability that an item will be retained in, or restored to, a specified condition within a given period of time when the maintenance is performed in accordance with prescribed procedures and resources (Ref. 5). Maintainability is related to the term MTTR (Ref. 6).

These definitions indicate the impact of maintenance considerations upon military operations. First, success in performing an assigned mission, i.e., mission effectiveness, is a function of maintainability; for example, a reconnaissance helicopter requiring excessive maintenance may miss vital time-sensitive mission objectives due to downtime. Second, ill-designed maintenance concepts lead to intolerable costs during the life cycle of the system. Third, faulty maintenance design provisions have tremendous impact upon the entire logistic support and technician training systems, requiring greater spare part stockage and distribution and the training of greater numbers of personnel (AMCP 706-134).

TABLE 2-1
SUMMARY OF RELIABILITY DESIGN TECHNIQUES

TECHNIQUE	DESCRIPTION	ADVANTAGES	DISADVANTAGES
REDUNDANCY	USES PARALLEL OR MULTIPLE SUBSYSTEMS/COMPONENTS TO PERFORM SOME TASK/FUNCTION.	ASSURES CONTINUITY OF OPERATION IN THE EVENT OF A SUBSYSTEM/COMPONENT FAILURE.	INCREASES: 1. SYSTEM COMPLEXITY 2. MAINTENANCE REQUIREMENTS 3. SIZE AND WEIGHT
DERATING	IS THE NORMAL OPERATION OF SUBSYSTEMS/COMPONENTS CONSIDERABLY BELOW MAXIMUM DESIGN OUTPUT CAPABILITY	OPERATES SIGNIFICANTLY BELOW DESIGNED STRESS LEVELS.	LESSENS OPERATING EFFICIENCY. INCREASES SIZE AND WEIGHT PER DEGREE OF OUTPUT (e.g., lb/hp)
DIAGNOSTIC AIDS	USES INSTALLED MONITORING DEVICES TO SENSE INCIPIENT FAILURES	WARNS OF POSSIBLE FAILURE BEFORE OCCURRENCE	INVOLVES COST OF MONITOR EQUIPMENT. REQUIRES SPECIALIZED MAINTENANCE OF MONITORS.
CONSERVATIVE DESIGN	UTILIZES TIME-PROVEN COMPONENTS IN LIEU OF NEW DEVELOPMENTS. UTILIZES EXTRA SAFETY MARGINS IN AREAS OF STRESS.	REDUCES CHANCE OF FAILURE DURING BREAKIN PERIOD – LESS EXPENSIVE PROVIDES ADDITIONAL ASSURANCE OF FAILURE-FREE OPERATION	OLDER COMPONENTS MAY NOT BE AS EFFICIENT – INCREASED WEIGHT. MIGHT INCUR A WEIGHT PENALTY.
COMPONENT DEVELOPMENT	REFINES DESIGN BY ITERATION, i.e., TESTING COMPONENT, THEN REDESIGN	PROVIDES MORE ASSURANCE OF RELIABLE INSERVICE OPERATION.	TESTING MAY NOT DUPLICATE ALL INSERVICE OPERATING CONDITIONS/ENVIRONMENTS. INCREASES TIME, COSTS REQUIRED.
SERVICING	PROMOTES EASE OF SERVICING/ROUTINE MAINTENANCE.	EXTENDS LIFE OF COMPONENTS/ITEMS, THUS ENHANCING RELIABILITY.	INCREASES TIME, LABOR REQUIRED.

TABLE 2-2
SUMMARY OF MAINTAINABILITY DESIGN CONSIDERATIONS

TECHNIQUE *	DEFINITION	IMPLICATION
ACCESSIBILITY	A MEASURE OF THE RELATIVE EASE OF ADMISSION TO THE VARIOUS AREAS OF AN ITEM (REF. 2).	HIGH ACCESSIBILITY, WHICH ALLOWS FOR MORE RAPID CHECKING, SERVICING, REMOVAL, AND REPLACEMENT, LOWERING OF MTTR.
IDENTIFICATION	ADEQUATE MARKING OF ITEMS, PARTS, ETC., TO FACILITATE REPAIR AND REPLACEMENT (REF. 1).	REDUCTION OF CIRCUIT TRACING AND RELATING ITEMS TO INSTRUCTIONS. LOWERING OF MTTR. PROMOTION OF SAFETY.
INTERCHANGEABILITY	ABILITY TO SUBSTITUTE FUNCTIONALLY AND/OR PHYSICALLY LIKE ITEM FOR DEFECTIVE ITEM.	ABILITY TO REPLACE ITEMS WITHOUT DIFFICULTY; LOWER MTTR.
SAFETY	PROVISION OF BUILTIN SAFETY PRECAUTIONS TO REDUCE SERVICE/REPAIR RISKS.	MORE UNDIVIDED ATTENTION TO TASK BY TECHNICIAN.
SIMPLIFICATION	REDUCTION IN THE DEGREE OF COMPLEXITY PRESENT IN SYSTEM/SUBSYSTEM.	FEWER MAINTENANCE TASKS TO PERFORM; LOWER MTTR (ALSO IMPROVED RELIABILITY).
STANDARDIZATION	ESTABLISHMENT OF DESIGN TECHNIQUES TO ACHIEVE THE GREATEST PRACTICAL UNIFORMITY.	REDUCTION OF NONSTANDARD ITEMS AND BURDEN ON LOGISTIC SUPPORT.
UNITIZATION AND MODULARIZATION	PACKAGING OF DIFFICULT-TO-MAINTAIN OR UNMAINTAINABLE COMPONENTS FOR EASY REMOVAL AND REPLACEMENT; RELATED TO INTERCHANGEABILITY AND STANDARDIZATION.	REDUCTION OF REPAIR TIME IN FIELD, THUS PERMITTING RAPID TURNAROUND.
IMPROVED MAINTENANCE EQUIPMENT	USE OF "QUICK-TO-USE" FASTENERS, TRANSPARENT PANELS, BETTER LUBRICANTS, ETC.	REDUCTION OF SERVICE TIME OR REQUIREMENT TO SERVICE.

* Ref. AMCP 706-134

The prime responsibility for reducing maintenance requirements lies with the system designer because features that significantly enhance maintainability must be incorporated into the design at the outset. Improvements introduced after fabrication, especially on complex systems, have only marginal or diminishingly effective results when measured against the efforts expended (Ref. 9).

Some of the principal maintainability design considerations available to the system designer are summarized in Table 2-2.

The complex and important subject of maintainability is discussed at length in Chapter 11. The purpose of introducing the subject here is to emphasize that the designer's successful application of maintainability techniques can have an impact far beyond that of pure mission effectiveness. A helicopter system that reduces spare part inventories, that can be serviced with ordinary mechanic-tools, that requires no specialized support equipment, and that requires no additional personnel with specialized training—all without sacrificing performance—can bring about measurable improvements in and savings for the entire military system.

2-3.2 SURVIVABILITY

Another primary factor of the mission-effectiveness equation (Eq. 2-1) is survivability which for purposes of this equation was defined in par. 2-2.3.[1]

Because mission accomplishment is of primary concern, it follows that protection of the lives and well-being of the crew and of the mission-sensitive equipment (fire controls, sensors, etc.) is mandatory. Also, if the concept of mission effectiveness extends over the lifetime of the helicopter, rather than only for a single

[1] The term survivability, from the mission-effectiveness point of view, should not be confused with "crash survivability" or "crash-worthiness" considerations. Although "crash survivability" is a vital factor in helicopter design, and normally is included as a portion of the survivability program during development, it does not contribute directly to mission effectiveness. Furthermore, quantitative treatment, if possible, is inconsistent with the basic approach to mission effectiveness.

sortie, any design feature that might prolong system usefulness by minimizing incurred damage must be considered. For purposes of discussion, those aspects of survivability having to do with combat environments will be referred to as "vulnerability".

Two primary aspects of vulnerability must be considered from the design point of view: (1) the avoidance of detection by hostile forces, and (2) the minimization of the possibility that crippling or catastrophic failure will be induced by hostile fire.

Existing design techniques and devices can reduce vulnerability; however, resultant effectiveness must be measured through interaction with a predicted threat environment, e.g., hostile defensive sensors and weapons, or weather.

Detection avoidance involves reduction of the helicopter signature in order to render the vehicle less noticeable to enemy sensors, both human and electronic. Several types of signatures are common to helicopters —acoustics, infrared (IR) radiation, and radar. Engineering design techniques have been developed that mitigate the undesirable effects of each type of signature (see Chapters 8 and 13); however, these techniques usually result in some loss of performance and/or a weight penalty. For example, a typical IR suppression device for a helicopter power plant might cool the exhaust duct and provide shielding to restrict the hostile viewing angle of the sensor. The penalties associated with such a device for a 1500-hp gas turbine engine might be as much as 50 lb, and could cause a loss of more than 2% in shaft horsepower. While the penalty seemingly is small, it should be accepted only after careful consideration of alternatives.

Under some conditions, both static and dynamic, camouflage and special nonreflective treatments for windscreens and curved surfaces may be specified to reduce the possibility of visual acquisition. There normally is a relatively minor weight penalty associated with these treatments.

The ability to operate a helicopter at extremely low altitudes, i.e., nap-of-the-earth, is another means of avoiding detection. This technique normally demands a high degree of maneuverability and excellent flight handling qualities, both of which are performance subfactors. This type of operation presents difficulties in navigation and in the avoidance of unforeseen obstructions. In some types of operations these difficulties can be allayed by using specialized navigation devices and terrain following/avoidance equipment, but this equipment imposes a weight penalty and increases system complexity (Ref. 10).

Once the helicopter has been detected by an enemy, a combination of both active and passive measures can be employed to reduce system vulnerability. Maneuverability (the ability to take rapid evasive action), the use of suppression-fire or counterfire, and the use of electronic countermeasures are typical active measures.

Various design techniques can be employed to reduce the vulnerability of a helicopter to hostile fire. Providing armor and shielding for the crew and for certain critical components is a highly effective means, but one that imposes major weight penalties. Redundant or backup subsystems and components, especially those that are separated widely within the structure, prevent catastrophic loss when severe battle damage is received by one system (see Chapters 9 and 13). Multiple load path design prevents immediate or progressive failure due to impact by distributing loads in other parts of the structure. An analogous technique is the use of ballistic-resistant components, e.g., bell cranks or rods that deform but do not fail when struck by fragments or high velocity debris. Self-sealing fuel cells and fire-retardant materials are other preventive measures (Refs. 11 and 12).

The degree to which vulnerability reduction capabilities should be incorporated into a design must be specified by the user in the military system requirements. Achieving balance among the available alternatives is an extremely difficult trade-off task. Where significant design differences exist, among alternatives, however, it is possible to obtain some measure of the relative vulnerability of each design by inserting each into models of threat environments. These models essentially are combat scenarios representing possible situations in which the helicopter might be expected to operate. Based upon empirical data and theoretical calculations, an evaluation of the relative vulnerability is possible. This technique is satisfactory, provided the threat environment is recognized to be artificial and may have little or no relevance to the eventual operating situation.

2-3.3 PERFORMANCE

Basically, performance is the output of a system. In the mission-effectiveness equation (Eq. 2-1), the performance factor is essentially a Figure of Merit in probabilistic form, representing an assessment of the relative worth of system output during the conduct of assigned tasks. The performance factor also may be considered as an overall indication of capabilities; i.e., the performance levels that are expected to be achieved when the system is committed to a mission.

2-3.3.1 Performance Requirements

Until the early 1960s, the overall performance capabilities of helicopters were relatively limited. Technological improvements—including reduction of parasitic drag, improved rotor systems, auxiliary propulsion, and lighter weight structures and engines—have resulted in considerable growth in almost all aspects of helicopter operational capability. Figs. 2-8 and 2-9 depict the improvements achieved in cruise speed and the reductions made in structural weight (Ref. 13).

The increase in the spectrum of obtainable performance has had a major impact upon military planning. Whereas, in the past, use of helicopters for certain missions was considered impossible because of performance deficiencies, new operational applications such as attack and heavy-lift missions now have become feasible. It is possible today to optimize configurations for particular classes of missions, rather than to use the one or two available helicopter types for the entire range of applications as used to be the practice.

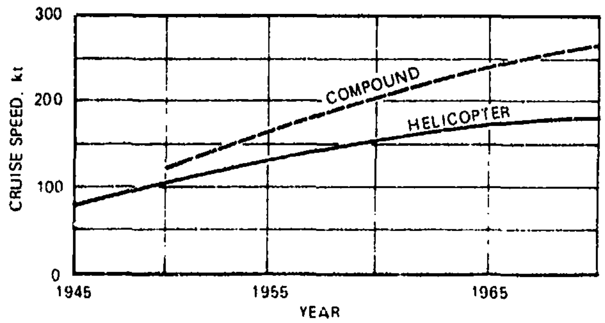


Fig. 2-8. Trends in Helicopter Cruise Speeds (Ref. 13)

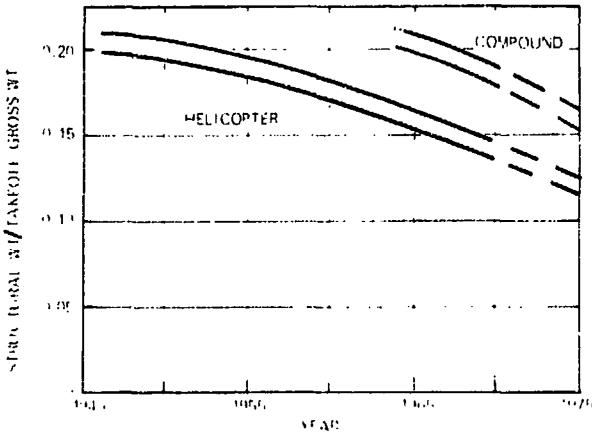


Fig. 2-9. Trend of Structural Weights for Helicopters (Ref. 13)

To take advantage of increased performance capabilities, it has become necessary for the user to be highly specific and definitive with respect to:

1. Identification of the classes of missions that lend themselves effectively to helicopter applications; e.g., attack, observation, utility tactical transport, and heavy-lift
2. Identification of the significant operational parameters for each class of mission
3. Statements of minimum acceptable performance for each class of mission
4. Statements of priorities to be considered in conduct of performance parameter trade-off analyses
5. Statements of requirements for mission augmenting/enhancing equipment to be installed; e.g., weapons, fire control systems, sensors, cargo-handling equipment, and navigation and communication equipment.

The statement of performance criteria evolves during the formulation of the military system requirements (par. 2-2.1). Normally, the criteria take the form of minimum acceptable levels of performance for each flight parameter that is significant for the mission class. Certain parameters are extremely sensitive to mission type; e.g., design-limit speed is an important parameter for the attack mission but of little value to the heavy-lift helicopter, and maneuverability may be significant to the attack and light observation missions but of lesser importance to the utility mission.

To augment the statement of requirements, the user may devise a mission profile (or several profiles for multimission capabilities) to assist the designer. A mission profile essentially is a synthesized scenario representative of the possible functions the helicopter is to perform. Fig. 2-10 depicts a hypothetical mission profile for a heavy-lift mission. The mission normally is designed to extract the maximum desired performance capability from the configuration.

2-3.3.2 Performance Effectiveness

Traditionally, the performance of helicopter systems has been described in terms of standard flight parameters; e.g., speed, rate of climb, range, and payload. With respect to military applications, however, the trend has been toward the establishment of more operationally oriented measures such as productivity, target acquisition and engagement capabilities, and maneuverability. The latter terms are quantified by manipulating the aforementioned flight parameters according to mission profiles (par. 2-3.3) or by inserting the candidate configuration into complex simulations.

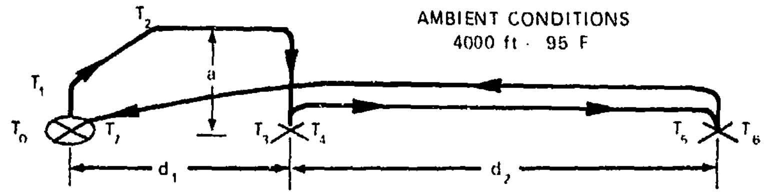
As mentioned previously, the flight parameters are dependent upon, and extremely sensitive to, the intended class(es) of mission(s). In some instances the entire design may be established on the basis of a single mission performance requirement; e.g., the heavy-lift mission may specify the vertical lift of a specific weight at 4000 ft altitude, 95°F, out-of-ground effect. Although details may vary during the parametric analysis, such a requirement probably will fix the amount of installed power and the general size and configuration of the helicopter system (Ref. 14). The designer must analyze each mission and military system requirement to determine the critical design parameters. This step is his point of departure.

Certain functionally oriented aspects of performance effectiveness, e.g., target acquisition and engagement capabilities, do not lend themselves to direct measurement. Regardless of the inherent and augmented capabilities of the helicopter, its ultimate effectiveness really is dependent upon the hostile environment —enemy battlefield array, organization, deployed offensive and defensive equipment, and battle plans. The enemy disposition only can be hypothesized from the best available intelligence. Considerable abstraction or deviation from available data may be necessary or desirable. If suitable enemy arrays can be constructed, it is possible to insert the helicopter or groups of helicopters represented by the inherent and augmented performance parameters into computer simulation models of varying degrees of complexity. Various techniques exist for treating significant unknown variables. Monte Carlo techniques (successive comparisons of computed or input probabilities with series of random numbers) commonly are used, but techniques also have been developed for providing complete distributions of possible results without recourse to Monte Carlo methods. The parametric approach can be used in cases in which some semblance of boundary conditions is known (Ref. 1).

This type of simulation process provides an assessment of the probability that targets can be acquired and engaged under certain sets of design conditions. This information not only is helpful in evaluating performance, but also can be used in design trade-off analyses.

The designer may employ user-supplied profiles to assist in the optimization of candidate configurations. When the user has specified the installation of performance-enhancing equipment, the designer may resort to sophisticated simulations (computer models) to determine maximum expected effectiveness. From such techniques it should be possible to derive a probabilistic Figure of Merit representative of the value of the particular configuration.

Another technique used to evaluate performance factors is an item-by-item comparison of each candidate configuration. Both the designer and the user normally employ some matrix of this type in their respective evaluations. Where significant differences in parameter levels exist, the selection often is easy. It would be assumed that a configuration having the highest levels for each parameter would provide the best overall level of performance (Ref. 4). Problems arise, however, when the order changes from one parameter to another, and/or when the differences in performance level are small. For example, Configuration A may have a 3% advantage in payload over Configuration B,



T_0 START ENGINES AT BASE
T_1 VERTICAL TAKEOFF WITH MAXIMUM FUEL AND CREW
T_2 CLIMB TO CRUISE ALTITUDE a FEET
T_3 FLY d_1 MILES AND LAND TO TAKE ON FULL, EXTERNAL PAYLOAD OF Y TONS
T_4 VERTICAL TAKEOFF
T_5 FLY d_2 MILES AND HOVER OUT-OF-GROUND EFFECT, DISCHARGE PAYLOAD
T_6 RETURN TO BASE
T_7 LAND WITH t MINUTES FUEL RESERVE

NOTE REPRESENTATIVE BOUNDARY CONDITIONS OR STATE OF OPERATION MUST BE STATED,e.g., CRUISE AT 99% AIRSPEED FOR BEST RANGE.

Fig. 2-10. Representative Mission Profile, Heavy-lift Helicopter

while B may be 2% faster than A. Thus, unless the user can indicate a priority of importance for each required parameter, the comparison method may be of marginal value.

2-4 ECONOMIC ANALYSIS

2-4.1 GENERAL

This paragraph discusses the use of economic analysis, often termed cost-effectiveness analysis, as an aid in design trade-off decisionmaking. Elements of an economic analysis are presented, along with detailed discussions of criteria cost concepts, cost estimation, and the use of simulation models in economic analysis.

2-4.2 ELEMENTS OF ECONOMIC ANALYSIS

The elements of an economic analysis are:

1. Statements of the design objective and of the effectiveness measures to be used to determine accomplishment of the objective
2. Specification of the candidate means (alternatives) by which the objective might be accomplished
3. Measurements of the costs associated with the potential use of each candidate means
4. A set of relationships (a model) that relates the cost of each candidate means to its corresponding effectiveness
5. A criterion or criteria with which alternatives are compared and upon which selection may be made.

Further discussion of the elements of an economic analysis are contained in Refs. 15 and 16.

2-4.3 FORMS OF CRITERIA

Normally, a fixed effectiveness or fixed cost form of criterion is established first, and then the sensitivity of the fixed level upon the most promising choices is examined. A fixed effectiveness criterion would minimize the cost to attain a given effect. A fixed cost criterion would maximize the effectiveness attainable at a fixed cost.

Fig. 2-11 presents continuous plots of effectiveness versus cost of ownership for two versions of a cargo helicopter design, where the objective is to choose between two alternative engines.

The measure of effectiveness is the payload range capability, defined as the maximum ton-miles attainable at the best-range speed. Each curve on the chart represents a continuous locus of points, with each point being a single airframe design configuration for a given engine. In general, an increase in ton-mile performance is attained for a given engine only with increasing sophistication of airframe design e.g., drag reduction or lighter weight structure , which, of course, is reflected in higher costs.

Note that the information given in Fig. 2-11 is insufficient to allow a specific design choice to be made until a criterion is applied. Some knowledge of what level of performance actually is needed must be incorporated. Thus, if a fixed effectiveness level of 5000 ton-miles of cargo-carrying capability is considered sufficient, a design using Engine A clearly would be less expensive than a design using Engine B to attain the same 5000-ton-mile capability. On the other hand, if a fixed effectiveness level of at least 7000 ton-miles of capability is required, only a configuration that uses Engine B can meet the requirement.

Consider also the adjustment of the fixed cost criterion level in determining the final choice. For example, if $5 billion initially is considered a reasonable fixed cost to the Army for a cargo helicopter fleet, then a cost approximating $6 billion using Engine B also should be considered because this alternative offers a substantial increase in cargo transport capability over that attainable from either the Engine A or B designs at the $5 billion level.

One additional criterion, dominance, is valid equally for decisionmaking. This is the rare but preferred situation in which one design always is more effective than all other choices at all cost levels.

Further treatment of criterion forms is found in Refs. 17 and 18.

2-4.4 COST ANALYSIS

This discussion provides the designer with a basic introduction to cost concepts, methods of cost estimations, and the role of cost estimation in the decision process.

The term "cost analysis" means many things. For the purposes of this handbook, cost analysis *shall* be concerned with the resources (usually expressed in dollars) required to design, produce, and operate Army helicopters. The results of such analyses, together with effectiveness, will aid designers in arriving at design trade-off decisions.

Cost analyses developed during a concept design phase normally are less accurate than those developed in a system procurement or operational phase. The reason is that planning, or design, involves projections for programs that reach further into the future and are specified less clearly than are programming or budgeting actions. In general, a far higher degree of uncertainty in cost data can be tolerated in design planning than would be acceptable in the later phases. Recognition of this fact is important to both the designer and the cost analyst in selecting an appropriate level of detail and precision to be applied in design decisions.

Two major sources of uncertainty exist in cost analysis:

1. Inadequate or inaccurate specification of the system being costed

2. Statistical inaccuracies in the cost-estimating relationships (CERs).

Cost estimates characteristically err on the low side, often underestimating total system costs by factors of two or more. Post-examination of several sets of program cost estimates (Ref. 19) has shown that the major reason for underestimation is a significant change in system specification. Thus, the system subjected to cost analysis is quite unlike the system that eventually is procured and enters the Army inventory. This fact should not discourage the designer from using costs to aid his decisions regarding design approaches, however. The important consideration in the use of cost in decisionmaking is that all data used to aid a given decision must be attained consistently and arrayed on a comparative basis. Thus, decisions made upon the basis of such data are valid within the context in which they are made. They will, in most cases, remain so for a given system even where other system factors change.

Ref. 20 presents a technique for quantitatively evaluating the impact of uncertainty in cost analysis.

2-4.4.1 Cost Concepts

2-4.4.1.1 Life Cycle Cost

One cost concept generally used to select among competing designs is life cycle cost, which is the summation of all expenditures required from conception of a system until it is phased out of operational use. The life cycle cost (also termed total system cost or "cradle-to-grave" cost) of an Army helicopter may be divided into the following time-phased parts: research and development, initial investment, and annual operating costs. Another way of dividing these costs is into nonrecurring and recurring expenditures. Briefly, these categories are defined as follows:

1. Nonrecurring:
   a. Research and development. Includes all costs of hardware, engineering, test, and other activities necessary to qualify a system for use
   b. Initial investment. Includes all costs of producing an item and procuring the operating personnel necessary for its use by the Army.
2. Recurring:
   a. Operations. Includes all costs—such as fuel, operations, personnel, replacement, and training—of operating the system throughout its service life
   b. Maintenance. Includes all costs of spares, re-

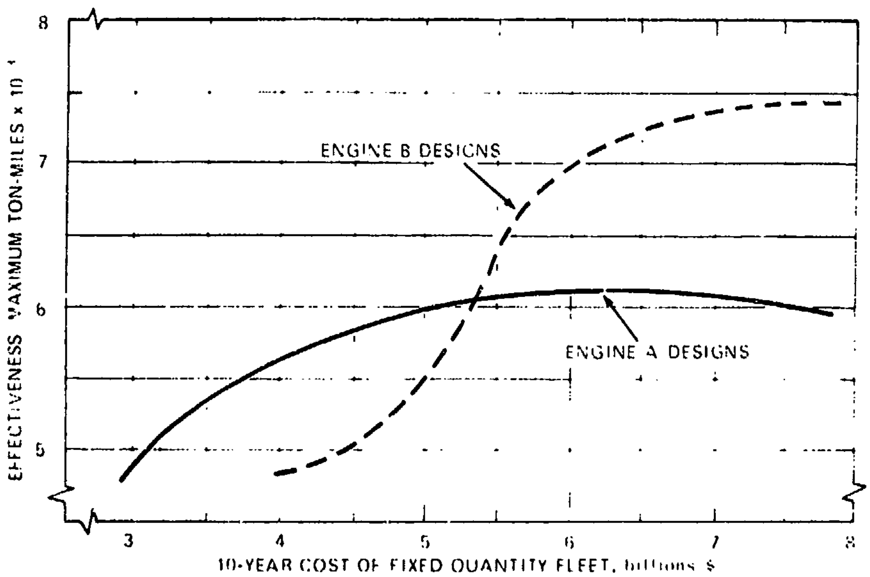


Fig. 2-11. Cost and Effectiveness for Various Designs

pair parts, maintenance personnel, etc., required to maintain a system in operational readiness.

For example, this cost concept might be used to aid selection of one from among two or more competing helicopter engines. Given equal survivability, availability, and performance factors, a determination would be made of which engine installation offers the least life cycle cost to the Army for a selected service life.

The designer should be warned, however, that use of the life cycle cost concept involves some difficulties. The first is the magnitude of the task of assembling sufficient historical data on similar systems and force structures to generate meaningful cost-estimating relationships for the new system concept. Such a task requires extensive and continual effort in data collection, analysis, and update. A second problem area, institutional constraints, involves the difficulty experienced by groups other than top-level Army agencies in obtaining access to relevant and sometimes sensitive historical information with which to assemble an adequate cost data bank. Finally, there is the problem of the perspective of the cost analysis organization; e.g., a private contractor cannot be faulted for displaying more concern for initial costs (the contractor's price to the Army), in the face of competition, than for the Army's costs of operating and maintaining the delivered item.

2-4.4.1.2 Incremental Cost

The concept of incremental cost involves the comparative use of the increment (difference) in costs between alternatives. This amounts to a "top-of-the-iceberg" analysis and eliminates the need for the large effort required for a total system cost comparison.

Thus, for example, if all engines being considered as candidates for a helicopter design already were qualified fully, the prior incurred R&D costs would not need to be considered in the selection process because there is no way to recover prior expenditures ("sunk" costs). By the same token, if all engines had demonstrated that they essentially were equal in maintenance factors, there likewise would be no reason to convert this information to dollar costs because the impact would be the same for each engine choice. In other words, the designer need use only costs that differ among the engines, e.g., initial procurement costs and annual operating costs. The total difference would allow the analyst to rank the engines in terms of incremental costs. For example, the incremental 10-yr Cost of Ownership for a 2000-engine fleet, assuming a baseline cost for Engine A, might be baseline cost plus $40 million for Engine B, and baseline cost minus $95 million for Engine C, where there is no need to evaluate the baseline cost in absolute terms.

Although it can be argued that such data provide no basis for determining the total cost of any of the program choices, nor do they give any indication of percentage differences among alternatives, it also can be argued that, in the absence of any better information, the incremental cost information can aid a selection among alternatives. Forms of selection criteria are discussed in par. 2-4.3.

2-4.4.1.3 Real Resource Utilization

Cost analysis also could be called "resource analysis" because it involves the estimation of the resource implications of alternative courses of action. Resource utilizations may be expressed in terms such as maintenance hours per flight hour, pounds of fuel consumed per flight hour, or number of quality control inspections required per component assembly. The utility of such measurements should not be ignored in making design decisions. The need to convert such physical resource measurements into dollar costs arises out of the desirability of putting differing resource measurements into commensurate terms as a basis for weighing the relative importance of each. The potential consumption of resources, expressed as dollar costs, allows unlike resource uses to be compared or totaled; a dollar consumed for fuel is equal in value to a dollar consumed to pay a mechanic. In this way, the designer can make comparative and unbiased choices among resource expenditures.

In some cases, however, the analyst will encounter physical resources that are constrained by other than dollar costs and that cannot be expressed meaningfully in dollar terms. Such a constrained resource might be the production capacity of an existing manufacturing facility. Management business risk decisions, insufficient time to build new facilities, or similar factors may exclude a given design alternative, even if no explicit limitation is placed upon dollar resources.

Another example is the consideration of the value of human life during a survivability trade-off study. The value of human life is not commensurate with other resource measurements and, hence, its value cannot be expressed in terms of payment of survivor benefits, or the costs of training and travel of replacement personnel. While dollar costs consumed for such activities are commensurate in every way with other dollar costs, their use would tend to mask the implications of the trade-off of human lives for dollars.

2-4.4.1.4 Joint Costs

Joint costs are generated when more than one system uses a specific resource. Normally, joint resources are those involved in the functions of command or support. Joint costs that can be considered as "overhead" may be classified as either fixed or variable. The use of fixed joint resources does not change with the systems or organizations using them; as a result, no incremental costs need be allocated to the system designs being compared. Variable joint costs, on the other hand, occur when a change in the system design results in changes in the joint resource requirements or uses. As such, an appropriate proportion of the change in the incremental variable joint cost properly should be allocated to the change in system design.

An example of such an unequal allocation of incremental joint costs in a design trade-off analysis is the difference in the number of support-services personnel necessary to support the maintenance personnel required by a fleet of one helicopter model versus a fleet of another design that requires fewer maintenance personnel to maintain an equal level of availability.

2-4.4.1.5 Other Cost Concepts

Additional cost concepts, such as discounted costs and time-phased costs, routinely ar[illegible] ecountered in force structure and programs. How[illegible] [illegible] is unlikely that they will be needed while analyzin[illegible] [illegible]or design trade-off planning. Concepts [illegible]d defin[illegible] with which the cost analyst sh[illegible]ld [illegible]e familiar [illegible]

1. Amortized costs. [illegible]ned by dividing [illegible] costs by the estimated service life of the system.

2. Constant dollar cost. A cost whose value throughout time may be affected by inflation or deflation, but these effects are segregated. The cost is "frozen" in time and expressed in dollars of a particular year.

3. Direct cost. That cost related solely to the system that generates it. This term often is used to distinguish operating unit costs from support costs (see item 6).

4. Discounted cost. U[illegible] to show the time preference for postponed commitments to expenditures. Discounted cost also may be termed present cost and is obtained by the formula

$$PC = C(1 + r)^{-m}\text{, dollars} \tag{2-8}$$

where

PC = present value of cost, dollars

C = cost at end of m time periods, dollars

m = [illegible]ber of tim[illegible] periods, [illegible]onless

r = [illegible] rate (expressed [illegible] [illegible]decimal) per time period, dimensionless

5. Inflated cost. That which uses an estimated inflation factor to increase present-day estimates of expenditures to be made in later years.

6. Indirect cost. Not identified with the unit or system that generates it. Indirect costs are joint costs; i.e., they reflect the use of resources that support more than one unit or system.

7. Investment cost. Nonrecurring costs required to take a system from the developmental stage to operating capability within a force. These consist of costs for initial procurement of the system and initial training, and allowances for maintenance and combat parts inventory.

8. Operating cost. Recurring cost required to keep a system in an active force.

9. Opportunity cost. Cost of resources consumed in following one course of action and, therefore, unavailable to any alternative course of action.

10. Price. The dollar amount paid to a seller. The price is only part of the cost to the buyer because the buyer also must pay the cost of ownership.

11. Residual value. The estimated current worth of remaining assets when a system is removed from the active inventory (also termed salvage value).

2-4.4.2 Cost Estimation

This paragraph defines cost-estimating relationships (CER) and describes how they are obtained and the purposes they serve. The objective is to supply the design engineer with sufficient knowledge to judge the value and individual validity of CERs as aids in making design trade-off decisions. A discussion of the statistical analysis techniques used to derive these relationships is beyond the scope of this handbook.

A CER may be defined as a statement of how the costs of a system are influenced by changes in one or more variables of that system.

In certain cases, a simple factor type of CER may be possible. For example, in estimating fuel consumption costs, a simple multiplier can be applied to the number of pounds of fuel consumed to obtain the annual cost of fuel. Quite often, however, CERs can become far more complex, especially where an interdependency exists between the number and size of engines selected for use on a single helicopter and the cost of maintenance of that vehicle.

2-4.4.2.1 Data Development

The most fundamental step in deriving cost-estimating relationships is the collection of data related to current models used in operations and environments similar to those for which the new design is being procured. Historical records are the most common source of such data. Other data are obtained from tests and specially designed field experiments.

Two types of data are collected. First is the information that defines the dependent variable, i.e., costs of various operations. Second are those data that define the independent or interdependent variables, i.e. those that describe how the costs were generated. The difficulty here is that most influential factors seldom are apparent. The principal task of the cost analyst is to assess the independent variables and to test the strength of the relationships between cost and the operations that generate cost. Those that prove to have both strong and reliable relationships are selected as the CERs to be used in predictive analyses. For example, in the case of helicopter depot maintenance costs, data might be collected on maintenance cost versus flying hours, numbers of takeoffs and landings, or helicopter gross weights. The relationship that offered the highest predictive reliability would be the preferred CER for estimating cost of depot maintenance for similar helicopter operations.

2-4.4.2 2 CER Display Formats

There are three principal methods used to display cost-estimating relationships: graphical formats, tabular formats, and mathematical formulas. The simplest form of mathematical CER is the multiplying factor. The CERs used to estimate the pay and allowances of military personnel normally are expressed with such a factor. Thus, the total cost for pay and allowances of direct military personnel associated with a given helicopter fleet equals the number of personnel assigned directly to the fleet times the average pay and allowance factors (dollars per man).

A similar CER presentation takes the form of the linear equation. For example, the cost of support personnel sometimes is expressed as a function of the number of direct personnel in the form

$$Y = aX + b \text{ , dollars} \tag{2-9}$$

where

Y = cost of support personnel, dollars
a = the amount by which Y increases for a unit increase in X, dollars
X = the number of direct personnel
b = constant cost, dollars

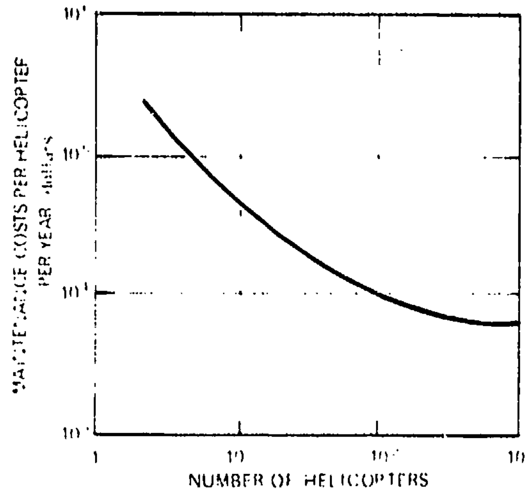


Fig. 2-12. Typical Curvilinear CER

In this example there is a minimum cost of support personnel b that must be assumed even if the number of direct assignees is minimal or zero. Obviously, this relationship also could be expressed graphically as a straight line (linear) variation of Y as a function of X with a Y intercept value of b. In either case, the interpretation should be straightforward. The choice of display format is strictly for the convenience of the analyst.

There are certain CERs—the nonlinear and often irregular relationships—that do not lend themselves readily to mathematical form. In this case the graphical format is preferred. One of the more common forms of graphical displays of CERs is that shown in Fig. 2-12. A typical curvilinear CER is plotted on log-log graph paper. Note that this curve is somewhat curvilinear rather than a straight line, and thus would prove difficult to express directly in mathematical form. Frequently, a simplification is applied to such a relationship that makes the assumption either that the curve is linear or that it can be dealt with as a series of linear segments. In any event, it often is easier in the case of irregular relationships to plot the results graphically and then read required values directly from the resulting plot.

The third common form of CER is the tabular pre-

TABLE 2-3
CONSTRUCTION COST AS MULTIPLE OF EQUIVALENT U.S. COST (U.S. = 1.0)

KOREA	0.90
FEDERAL REPUBLIC OF GERMANY	1.10
FRANCE	1.35
VIET NAM	1.40
PANAMA	1.60
PHILIPPINES	2.35

sentation. Several CERs lend themselves well to this format, as shown in Table 2-3.

Other less common forms of CERs are nomograms and multivariate equations; the former is a type of graphical display, while the latter can be used only when aided by computer.

2-4.4.2.3 CER Derivation

The first step in CER derivation is the collection of historical data on similar operations (par. 2-4.4.2.1). The second step is to assess the strength of the interdependencies between cost and the physical data that describe a system or operation. This step is aided principally by statistical analyses.

For example, assume that a significant degree of analysis has been made of several types of helicopters for three years of operations in both U.S. Army and related military operating organizations. The analyses have shown that, for a selected subset of the data that relate most directly to the design, depot maintenance costs are related to the continuous maximum power level of the engines, to the airframe empty weight, and to the maximum design gross weight. Now the factor must be selected that offers the best CER for prediction of depot maintenance costs. The data for this hypothetical example are shown in Table 2-4.

The next step is to plot the depot maintenance cost for the three years versus the independent variables. These plots are shown in Fig. 2-13, and it is apparent that the empty-weight-versus-cost relationship is the most consistent trend of those examined. The statistical technique for quantifying the quality of fit of a straight line drawn through these data is termed correlation analysis; a perfect fit yields a correlation coefficient of unity, and no correlation yields a zero correlation coefficient. A coefficient of variation is a second statistical measure and is a quantification of the degree of scatter of the data; i.e., the lower the coefficient of variation, the better the relationship. The coefficients of correlation and variation for the three relationships examined are shown in Table 2-5. In the cases examined the relationship between depot maintenance cost and helicopter empty weight offers the highest coefficient of correlation and the lowest coefficient of variation. Therefore, it clearly is the best of the three CERs for predicting depot maintenance costs on similar helicopters.

Another statistical technique, regression analysis, now is applied to this relationship. Regression analysis is used to express the trend line and the confidence bands. The trend line is the best fit linear relationship that can be put through the data points. In this case the trend line for CER is Cost ($/flt-hr) = 3.06 + 7.20 (empty weight $\times 10^{-3}$). The 95% confidence bands are shown in Fig. 2-14 and represent a statistical calculation that yields a subjective confidence that 95% of all measured values will fall between the bands.

2-4.4.2.4 Use of CERs

Like any predictive relationship, a CER is a tool for determining that which we wish to know based upon that which already is known or which can be measured. Where a sufficient body of historical data exists for analysis, a variety of CERs can be developed.

The degree to which costs can be used to aid decisionmaking is as sensitive to the validity of the CERs as it is to the validity of the physical relationships that describe the design alternatives. Similarly, it is important to avoid using the direct cost of producing an item of hardware as the sole cost criterion, and instead, to

TABLE 2-4
CER TEST DATA

HELICOPTER	DEPENDENT VARIABLES DEPOT MAINTANENCE, COST, $/flt-hr			INDEPENDENT VARIABLES		
	FY'68	FY'69	FY'70	GROSS WEIGHT, lb $\times 10^{-3}$	EMPTY WEIGHT, lb $\times 10^{-3}$	ENGINE POWER, hp $\times 10^{-3}$
CH A	7	8	8	11	9	2
CH B	6	5	5	11	7	2.5
CH C	15	-	18	28	21	5
CH D	4	3	4	11	6	2.5

consider the total cost implications of selecting one design over another.

Par. 2-4.5 describes, by example, how CERs can be combined with a knowledge of the effectiveness of various designs to aid design trade-off decisionmaking. Simulation modeling and the decision criteria related to cost-effectiveness techniques are discussed.

Further data concerning cost-estimating techniques and cost factors useful to the task of conducting helicopter design trade-off analyses are presented in Refs. 19 through 27.

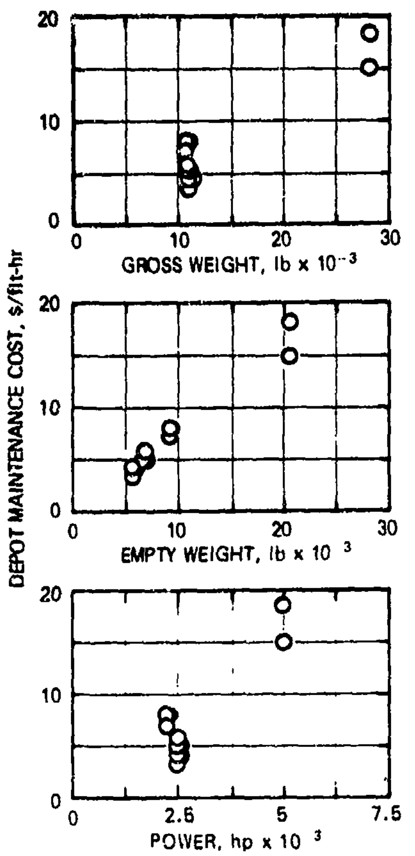


Fig. 2-13. Scatter Diagrams for Three Test CERs

TABLE 2-5
COEFFICIENTS OF CORRELATION AND VARIATION

	GROSS WT	EMPTY WT	POWER
COEFFICIENT OF CORRELATION	0.93	0.97	0.86
COEFFICIENT OF VARIATION	0.24	0.15	0.32

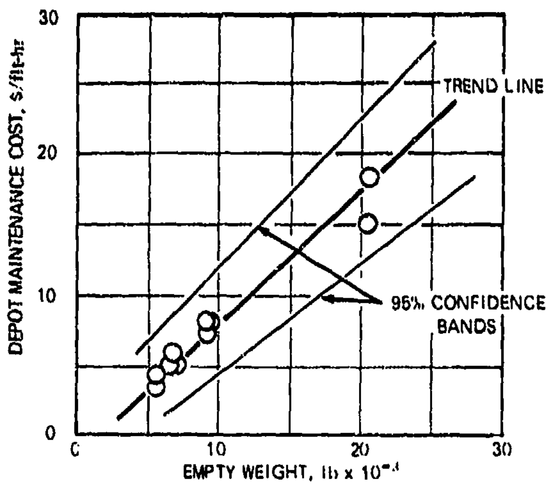


Fig. 2-14. Selected CER With Trend Line and Confidence Bands

2-4.5 SIMULATION MODELING: AN EXAMPLE

Any discussion of design trade-off analysis must include simulation modeling because of the importance of such modeling in relating unit effectiveness to total system cost.

This paragraph uses an example of choosing between two engines for a cargo helicopter. It is assumed that a choice of either engine design yields insignificant differences in the estimated performance and survivability of the basic helicopter. Differences exist only in the estimated vehicle availability and the initial investment cost.

There are five elements in cost-effectiveness analysis, as specified in par. 2-4.2:

1. Objective. To provide a capability to deliver one million flight hours of helicopter cargo airlift per year for a period of 10 yr. Thus, the measurement of effectiveness is the number of airlift hours available.
2. Measure of cost. Ten-year cost of ownership of the total fleet of identical cargo helicopters.
3. Design choices:
   a. Helicopter with Engine A
   b. Helicopter with Engine B.
4. Criterion of choice. That design which meets the objective at the least cost *shall* be preferred.
5. Model. The fifth element, the model, is developed herein.

The data available to begin the analysis are shown in Table 2-6. In addition, it is known that the mean cost of helicopter maintenance is \$240 per maintenance hour.

Two important facts should be noted for the construction of the model:

1. Helicopters are procured by the Army in fleet quantities and not as individual units. As such, a commitment of resources to one design or method can be assessed properly only in terms of the full costs and effects that the commitment yields.

2. The design of a helicopter is specified fully in a single unit. Therefore, measurements that characterize the capabilities of a given design are determined and expressed most readily for a single unit.

The difference in measurement levels between decisions regarding program resource allocations and those involving design information is both typical and proper. To the analyst, however, it presents a requirement to bridge a gap in information. The solution to this need is to construct a model or series of relationships that simulates the operational use of the helicopter fleet. The model takes information that is known—single-unit capability, or cost and maintenance data—and structures it to obtain a semi-realistic simulation of a total organization use. In this way the desired measures of fleet effectiveness and fleet cost are obtained.

A model that can be used in this way for the cargo helicopter design study is shown in logic flow form in Fig. 2-15.

The same can be expressed mathematically with two equations. An equation expressing the fixed effectiveness criterion is

$$E_{fe} = N_x H_x A_x \text{ , flt-hr/yr} \qquad (2\text{-}10)$$

where

E_{fe} = effectiveness criterion, fixed effectiveness, flt-hr/yr
N_x = number of helicopters in the fleet of type x helicopter
H_x = planned flight hours available per year per helicopter of design type x, hr
A_x = unit availability ratio for design type x helicopter, dimensionless

An equation expressing the 10-yr cost of ownership C_{10} is:

$$C_{10} = N_x(H_x) + 10\text{yr} \cdot E_{fe} \cdot C_{opr_x} + 10\text{yr} \cdot \frac{\text{maint-hr}}{\text{flt-hr}} \cdot E_{fe} \cdot C_{maint} \text{ , dollars} \qquad (2\text{-}11)$$

where

H_x = initial investment cost of one helicopter unit of design type x, dollars
C_{opr_x} = operating cost for design type x, dollars/flt-hr
C_{maint} = organizational cost of maintenance, dollars/maint-hr

A quantitative example of the cost calculations for the fixed effectiveness method is shown in Eqs. 2-12 through 2-14. This example is based upon the assumptions that $E_{fe} = 10^6$ flt-hr/yr and $H_x = 1000$ flt-hr/helicopter-yr. Combining Eqs. 2-10 and 2-11 and solving for C_{10} we obtain

$$C_{10} = \frac{10^3 H'_x}{A_x} + 10^7 C_{opr} + 10^7 \left(\frac{\text{maint-hr}}{\text{flt-hr}}\right) C_{maint} \text{ , dollars} \qquad (2\text{-}12)$$

Evaluating this equation for both designs, we obtain for Design A

$$C_{10_A} = \frac{(12+3)10^3}{12} [2.0 \times 10^6] + 10^7(110) + 10^7(1.25)\,240 = \$6.60 \times 10^9 \qquad (2\text{-}13)$$

and for Design B

TABLE 2-6
TYPICAL ANALYSIS DATA

	ENGINE A	ENGINE B
INITIAL INVESTMENT COST, MILLION \$	2	1.8
MEAN OPERATING COST, \$ PER FLT-HR	110	80
MTBMA, HR	12	15
MTTR, HR	3	5
RATIO OF MAINTENANCE HOURS TO FLIGHT HOURS	1.25	1.67

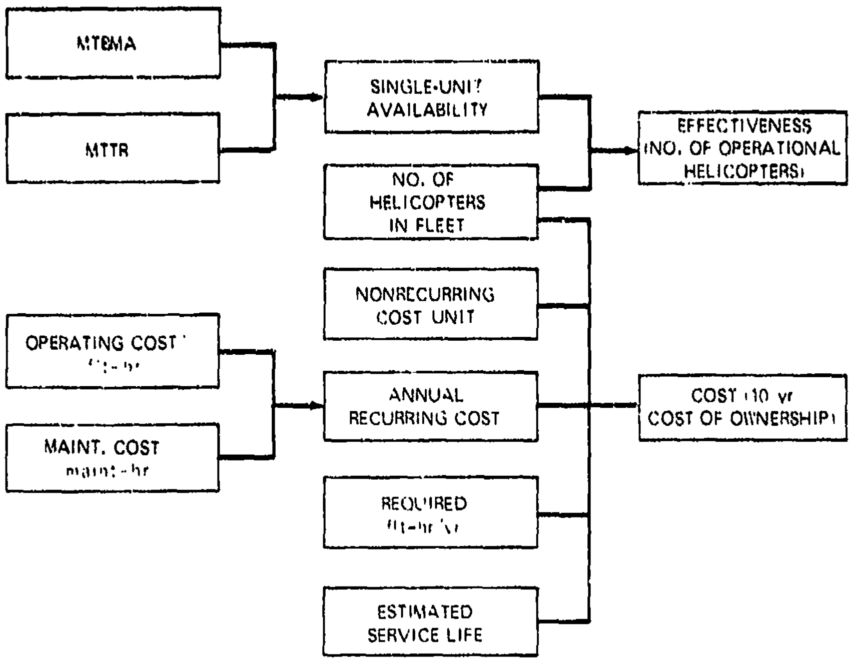


Fig. 2-15. Logic Flow Diagram for Design Trade-off Model

$$C_{10_B} = \frac{(15+5)10^3}{15}\,[1.8 \times 10^6] + 10^7(80) + 10^7(1.67)\,240$$

$$= \$7.20 \times 10^9 \qquad (2\text{-}14)$$

Thus, for the chosen level of effectiveness a \$600 million saving results if the helicopter design that uses Engine A is chosen.

The choice of Design A was not apparent readily from the original data. Indeed, the initial unit cost of Design A is 11% higher than for Design B. This example points out the danger of using only partial cost data to evaluate a design trade-off decision. This is further emphasized by the two 10-yr cost equations as evaluated for the two helicopter designs. The total initial investment cost of Design A (Eq. 2-13) is \$2.5 billion or approximately 38% of the total 10-yr cost; thus, the recurring costs contribution to the 10-yr cost of ownership amounts to 62%. For Design B these recurring costs are 67% of the total. In either case they have a significant impact and should not be ignored.

Although the example used was simple, the actual process of model building for more complex problems would be fundamentally the same. A more complex and perhaps more realistic model might be constructed by adding further elements such as actual crew replacement policies, part stockage, and fuel costs. Similarly, the effectiveness simulation might be portrayed more realistically by using a random selection of typical mission profiles or a probabilistic simulation of unscheduled maintenance action.

A discussion of such probabilistic models is beyond the scope of this handbook. However, Ref. 21 describes such a model used for helicopter design trade-off studies. In building a simulation model, the analyst should make sure that the addition of further detail and complexity not only will improve the simulation of reality, but also will improve the accuracy of the simulation, consistent with the needs of the problem and the inherent accuracy of the input data.

2-5 LIST OF SYMBOLS

A = availability, dimensionless

A_{c_j} = component availability, dimensionless

A_i = subsystem availability, dimensionless

A_s = system availability, dimensionless

A_x = unit availability ratio for design type x helicopters, dimensionless

a = amount by which Y increases for a unit increase in X, dollars

b = constant cost, dollars

C = cost at end of m time periods, dollars

C_{10} = 10 yr cost of ownership, dollars

C_{maint} = organizational cost of maintenance, dollars/maint-hr

C_{opr_x} = operating cost per hour for design type x, dollars

E_{fe} = effectiveness criterion, fixed effectiveness, flt-hr/yr

H_x = flight hours available per year from helicopter unit of type x, hr

II_x = initial investment cost of one unit of design type x, dollars

m = number of time periods, dimensionless

N_x = number of helicopters in the fleet of type x, dimensionless

P = probability, dimensionless

PC = present value of cost, dollars

R = reliability, dimensionless

R_i = subsystem reliability, dimensionless

R_s = system reliability, dimensionless

r = discount rate (expressed as a decimal) per time period, dimensionless

t = time, hr

X = number of direct personnel, dimensionless

Y = cost of support personnel, dollars

λ = failure rate = 1/MTBF, hr^{-1}

λ_i = subsystem failure rate, hr^{-1}

λ_s = system failure rate, hr^{-1}

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CHAPTER 3

PERFORMANCE

3-1 INTRODUCTION

This chapter describes procedures for the preliminary optimization of helicopter performance. In the preliminary design, performance evaluations necessarily are tentative, but proper use of historical data and available analytical techniques will minimize uncertainties and avoid difficulties and delays in the detail design and qualification assurance phases.

Aerodynamics and propulsion are the primary areas for performance optimization. Methods for evaluating these elements are described in this chapter. This introduction describes the evolution of the theoretical basis of helicopter performance evaluation, along with the major preliminary design problems and the potentials of helicopter performance.

Although the conception of the helicopter preceded that of the autogyro by at least several centuries, the discovery by Cierva of the optimum blade angle for autorotation of a freely rotating rotor was probably the most vital step toward practicable rotary-wing flight. It permitted achievement of a slow descent, making it possible for the helicopter to land safely in the event of engine failure.

The reports of Glauert (Ref. 1) and Lock (Ref. 2) formed the earliest theoretical bases for investigating the physical principles of rotary-wing flight. Analysis at first was confined to the autogyro, but Glauert later developed a theory of helicopter performance during vertical ascent (Ref. 3) that was extended to cover horizontal flight (Ref. 4) with the rotor axis vertical. Squire (Ref. 5) extended the analysis to flight with the rotor axis inclined forward to give a component of rotor thrust for propulsion. Wheatley (Ref. 6) made further fundamental contributions to the early literature on helicopter performance.

The era of the autogyro (Ref. 7) merged into that of the helicopter by Focke's demonstration (Ref. 8) in 1938 of adequate controllability in the hover mode. A series of papers by Bennett (Ref. 9) on rotary-wing aircraft summarized the state of knowledge in 1940; it was clear then that there were three main categories of rotary-wing aircraft:

1. The classic or "pure" helicopter that had no separate means of propulsion; i.e., all of the power was supplied to the rotor or rotors (Ref. 5)

2. The autogyro, whose rotor was kept in rotation during flight by aerodynamic forces only, the engine power being supplied to a propeller that provided a forward thrust component for translational flight. The rotor thus was wholly a lifting device.

3. The compound or hybrid helicopter, in which part of the power was supplied to the rotor for producing lift and part to a propeller for providing propulsion. The helicopter was enhanced greatly by the addition of a fixed wing to reduce the lift component provided by the rotor in translational flight, thereby enabling higher forward speeds to be achieved without encountering severe fluctuations in rotor lift. Such periodic fluctuations had been responsible for high rotor drag and inherent vibrational problems.

3-1.1 WORKING STATES OF A ROTOR

Glauert (Ref. 10) and Lock (Ref. 11) have analyzed the working states of a rotor in terms of two parameters F and f, thrust coefficients defined by the equation

$$\frac{dT}{dr} = 4\pi r\rho(V_D - \nu)^2 F$$

$$= 4\pi r\rho V_D^2 f \quad \text{, lb/ft} \qquad (3\text{-}1)$$

where

dT = thrust of the blade elements at radial distance r, lb (i.e., the thrust resulting from the momentum imparted to the air in an annulus of radius r and thickness dr)

V_D = rate of descent, fps

ν = rotor-induced velocity at the annulus, fps

ρ = density of air, slug/ft^3

Hence, the velocity of flow through the annulus is $(V_D - v)$ when $V_D > v$, and $(v - V_D)$ when $V_D < v$.

By using the empirical relationship between F and f given in Eq. 3-1 in the region where there is recirculatory flow through the disk, Bennett (Ref. 12) shows that a rotor autorotative is partially a propeller in the "vortex ring state" and partially a windmill in the "windmill brake state". The "windmill brake state", in which $V_D > v$, i.e., where there is an upwash, absorbs more torque from the air than can be expended in profile drag; therefore, the autorotative rotor at zero torque must expend excess torque in the "vortex ring state" where $V_D < v$, i.e., where there is a downwash.

In powered level flight or in a climb the rotor is working mainly in the "normal propeller state". V_D now becomes V, the helicopter speed, and the velocity of flow through the annulus at radius r is $(V + v)$. For uniform distribution of v over the disk—an ideal assumption—the rotor can be considered as an "actuator disk", a useful concept with which to compare the effects of nonuniform distribution and the effects of losses due to profile drag and unsteady flow.

Margler and Squire (Refs. 13 and 14) show that even in powered level flight there are regions of upwash as well as downwash. Thus, the blade elements are subjected to periodic variations in angle of attack due to the distribution of induced velocity and, therefore, experience periodic variations in torque that may fluctuate from a negative (windmill) value in an upwash to a strong positive (propeller) value in a downwash. Even with uniform distribution of v over the disk (a state that is unachievable in operation), a downwash results in an effective "wash-in" of angle of attack and, therefore, in increased loading toward the tip. Conversely, a uniform upwash results in an effective "wash-out" and, therefore, in decreased loading toward the tip.

The velocity distribution over the disk for any condition of flight can be calculated from Glauert's empirical values of F and f, using the method established by Ref. 12 for power-off descent. In the "vortex ring state", the rotor power ranges from zero in autorotation to nearly full power in the hover mode.

Stewart (Ref. 15) has determined in experimental flight the boundaries between the three main working states of a rotor for a typical helicopter. His work has shown that, as the forward speed increases, the rotor remains substantially in the normal propeller state up to higher sinking speeds. With the rotor in autorotation, the "windmill brake state" applies to lower sinking speeds as the forward speed increases. Thus, the rate of loss of potential energy in a glide is approximately equal to the power required for level flight at a given forward speed.

3-1.2 POWER REQUIRED FOR LEVEL FLIGHT

The energy equation of a helicopter is derived from the assumption that the power expended is equal to the power supplied from the engine or engines. This power may be transmitted wholly to the rotor or rotors (of a pure helicopter), wholly to a propulsive propeller or jet (of an autogyro), or partly to the rotor and partly to a propulsive propeller or jet (of a compound helicopter). The power is expended mainly in:

1. Induced drag of the rotor or rotors
2. Profile drag of the rotor blades
3. Body drag.

The induced power varies inversely with forward speed. The profile power increases from its value in the hover mode as the square of the forward speed and the power due to body drag increases as the cube of that speed.

The total power requirement during translational flight from hover to maximum speed initially decreases to a minimum value associated with best endurance, then increases at higher forward speeds. At these speeds the parasite power, which is expended to overcome body drag, becomes predominant. Typically, the power required for hovering is about twice that required for maximum endurance. Hence, at low translational speeds the helicopter operates in the region of the power curve where power decreases with forward speed.

It is evident that body drag must be kept to a minimum if high cruising and maximum speeds are to be attained. Use of a propeller or propulsive jet for forward propulsion would assist in maintaining a small fuselage angle of attack, and hence keep body drag to a minimum. The rotor drag at high speed can be minimized by using a fixed wing to off-load the rotor, thus avoiding periodic blade tip stall.

3-2 AERODYNAMICS

3-2.1 BASIC ANALYTICAL METHODS

In this paragraph the relationships of basic theories used in classical methods for calculating rotor performance are compared to show the considerations and processes required in building a realistic analytical performance model.

3-2.1.1 Hovering Flight

3-2.1.1.1 Momentum Theory

The most elementary analytical model for estimating the performance of a hovering rotor can be derived by applying the momentum and energy conservation laws to the air mass influenced by a rotor disk. The power applied by the disk equals the rate of change with time in kinetic energy of the airflow ($\Delta KE/\Delta t$), which is assumed to be steady and uniform over the disk area:

$$Tv = \frac{\Delta KE}{\Delta t} = \frac{\rho A v}{2} v_2^2 \quad \text{, ft-lb/sec} \qquad (3\text{-}2)$$

and the thrust is equal to the change of momentum across the disk

$$T = (\rho A v)v_2 \quad \text{, lb} \qquad (3\text{-}3)$$

where

A = rotor disk area, ft^2
T = rotor thrust, lb
v = rotor-induced velocity through the rotor disk, fps
v_2 = rotor-induced velocity infinitely downstream, fps

Hence, $v_2 = 2v$, indicating that the induced velocity infinitely far downstream is twice that at the actuator disk, and

$$v = \sqrt{\frac{T}{2\rho A}} \quad \text{, fps} \qquad (3\text{-}4)$$

Finally, the induced horsepower hp_i is given by

$$hp_i = \frac{Tv}{550} = \frac{T^{3/2}}{550\sqrt{2\rho A}} \qquad (3\text{-}5)$$

This expression for induced horsepower is the theoretical ideal. In reality, of course, the induced velocity is not constant over the disk area and has a rotational component. Also, wake vorticity, three-dimensional flow at the blade tip, and blade profile drag combine with other power losses to decrease the rotor efficiency when converting shaft power to thrust. To estimate a more reasonable lifting rotor power demand, profile drag losses and energy losses incurred because of the vorticity effects at the blade tips are taken into account and added to the ideal induced power of Eq. 3-5.

In general, profile power hp_o is given by

$$hp_o = \frac{\rho b}{1100} \int_0^R c(\Omega r)^3 c_d \, dr \qquad (3\text{-}6)$$

where

b = number of blades
c = blade chord at radius r, ft
Ω = rotor angular speed, rad/sec
c_d = section drag coefficient at radius r, dimensionless
R = rotor radius, ft

For a blade with constant chord c, but varying elemental drag coefficient c_d, we use the mean drag coefficient $\bar{C}_D$ and the profile power expression becomes

$$hp_o = \frac{\rho b c R^4 \Omega^3}{4400} \bar{C}_D \qquad (3\text{-}7)$$

To maintain consistency in this approach, the blade section lift coefficient c_l also is assumed constant along the blade (permitting the use of the mean lift coefficient $\bar{C}_L$) and the corresponding expression for rotor thrust is obtained. Normally, thrust T is given by

$$T = \frac{\rho b}{2} \int_0^R c(\Omega r)^2 c_l \, dr \quad \text{, lb} \qquad (3\text{-}8)$$

and for the conditions stated, this becomes

$$T = \frac{\rho b c R^3 \Omega^2}{6} \bar{C}_L \quad \text{, lb} \qquad (3\text{-}9)$$

Obviously, lift and drag coefficients do vary along the blade. However, if a relationship between the mean values of lift and drag coefficients can be established, the power and thrust relationships of Eqs. 3-7 and 3-9 can be exercised. Such a relationship, between mean lift and drag coefficients, is shown in Fig. 3-1 for one relationship between the mean lift coefficient $\bar{C}_L$ and the ratio C_T/σ, where C_T is the thrust coefficient and σ the rotor solidity. The additional refinement required to make this elementary analytical method suitable for approximating simple-geometry-rotor hover performance is the consideration of blade tip losses. The pressure differential along the blade span is dissipated at the

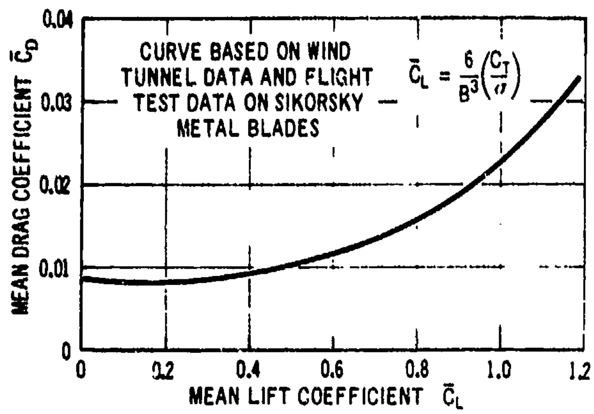


Fig. 3-1. Typical Mean Drag and Lift Coefficients

tip by the circulation of air flowing from the lower to the upper surface of the blade. This circulation persists over a portion of the tip area that, consequently, generates no lift. Semi-empirically, this condition is treated by assuming a tip loss factor B that reduces the effective radius for lift calculations. Typical values of B range from 0.93 to 0.98, depending upon rotor blade loading. This parameter is discussed further in par. 3-2.1.1.4.

The total power hp required by a hovering rotor includes the induced power of Eq. 3-5, modified to include the tip loss factor B, and the profile power of Eq. 3-7:

$$hp = \frac{T^{3/2}}{550B\sqrt{2\rho A}} + \frac{\rho bcR^4\Omega^3}{4400}\bar{C}_D \qquad (3\text{-}10)$$

For convenience, the thrust T, torque Q, and power P expressions can be nondimensionalized into coefficients C_T, C_Q, and C_P where

$$C_T = \frac{T}{\rho\pi R^2(\Omega R)^2}\text{ , dimensionless} \qquad (3\text{-}11)$$

$$C_Q = \frac{Q}{\rho\pi R^3(\Omega R)^2} = \frac{Q}{\rho\pi R^3(\Omega R)^2}\left(\frac{\Omega}{\Omega}\right)$$

$$= \frac{P}{\rho\pi R^2(\Omega R)^3} = C_P\text{ , dimensionless} \qquad (3\text{-}12)$$

where

Q = torque, lb-ft
P = power, ft-lb/sec

and Eq. 3-10 can now be reduced to the simple form

$$C_Q = C_{Q_i} + C_{Q_o} = C_P$$

$$= \frac{C_T^{3/2}}{B\sqrt{2}} + \frac{\sigma\bar{C}_D}{8}\text{ , dimensionless} \qquad (3\text{-}13)$$

where

C_{Q_i} = induced torque coefficient, dimensionless
C_{Q_o} = profile torque coefficient, dimensionless
σ = rotor solidity, $bc/(\pi R)$, dimensionless

Empirical corrections to the mean drag coefficient to account for Mach number effects and to the induced torque coefficient C_{Q_i} to account for nonuniform downwash can be introduced easily to make the theory as exact as is necessary for any specific rotor for which measured results are available.

3-2.1.1.2 Figure of Merit

An index of rotor efficiency for converting shaft power to thrust is given by the ratio of the theoretically ideal power to the actual power required. This index is called the rotor Figure of Merit M, expressed nondimensionally as

$$M = 0.707\frac{C_T^{3/2}}{C_Q}\text{ , dimensionless} \qquad (3\text{-}14)$$

For the ideal rotor, the Figure of Merit would equal unity; however, this value can be achieved only when blade tip and profile drag losses are nonexistent. One simple method of evaluating rotor performance is to examine a plot of thrust to power (power loading T_{hp}) vs thrust to rotor disk area (rotor disk loading w) for various Figures of Merit. This relationship is illustrated in Fig. 3-2. The value M = 0.75 may be considered representative of a relatively efficient rotor. The use of Figure of Merit for rotor design and evaluation is discussed further in par. 3-2.1.1.7.

3-2.1.1.3 Blade Element Theory

The blade element theory provides a more realistic method of analyzing hover performance by giving due consideration to blade profile drag losses. The rotor blade is analyzed as being made up of individual blade elements, each of which contributes to performance. Each spanwise element of the blade then may be con-

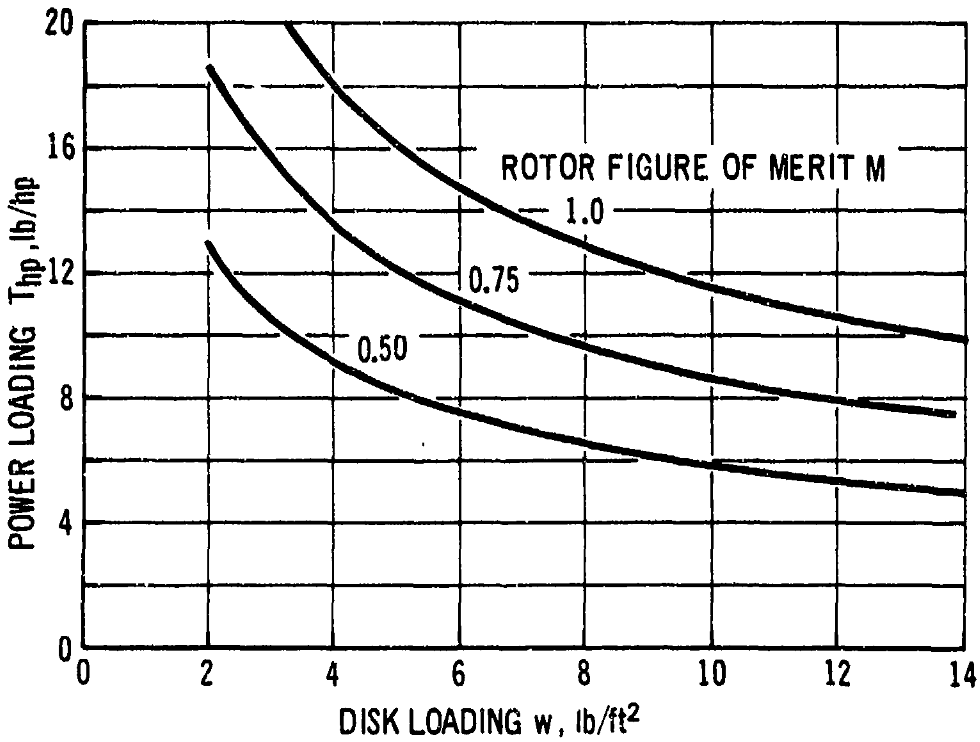


Fig. 3-2. Disk Loading vs Power Loading

sidered as a two-dimensional airfoil section, and the thrust and torque can be determined by integration. To derive a better representation of the inflow, and the resultant lift and inplane forces, at each blade element, this method is complemented by use of momentum theory. The rotor disk then is divided into elemental annuli and the change in momentum (elemental thrust), for each is equated to the blade element lift for a given blade pitch. This approach is depicted in Fig. 3-3 for a hovering rotor, where ϕ, α, and θ are defined below and V is the total velocity relative to the blade element.

$$\left.\begin{aligned} &\Delta T = \Delta L \cos\phi - \Delta D \sin\phi \\ &\text{or} \\ &\Delta T = \frac{\rho b c (\Omega r)^2}{2} \left[c_l \cos\phi - c_d \sin\phi\right] \Delta r \text{ , lb} \end{aligned}\right\} \quad (3\text{-}15)$$

where

L = lift, lb

D = drag, lb

ϕ = inflow angle, $\mathrm{Tan}^{-1}\left[\nu/(\Omega r)\right]$, rad

This represents the elemental thrust on a annulus of thickness Δr at a radius r. From momentum considerations the equivalent expression is

$$\Delta T = \rho(2\pi r \Delta r)\nu(2\nu) \text{ , lb} \quad (3\text{-}16)$$

Solving Eqs. 3-15 and 3-16 simultaneously gives

$$\nu = \left[\frac{b c r \Omega^2}{8\pi}(c_l \cos\phi - c_d \sin\phi)\right]^{1/2} \text{ , fps} \quad (3\text{-}17)$$

Both c_l and c_d vary with angle of attack α, and angle of attack is dependent upon induced velocity ν, i.e., $\alpha = \theta - \mathrm{Tan}^{-1}[\nu/(\Omega r)]$, where θ is the blade section pitch angle. Eq. 3-17 is transcendental in ν, requiring an iterative solution if nonlinear variations of c_l and c_d with α are considered. With the use of a digital computer, this is accomplished by assuming a value for ν, computing α and the corresponding c_l and c_d coefficients from appropriate two-dimensional airfoil characteristics (accounting for local stall and compressibility effects), thereby obtaining a new value of ν. This process then is repeated until convergence within the required tolerance is obtained. Again, tip loss is included by applying a "tip loss factor" that assumes complete loss of lift over a portion of the blade at the tip. Reasonable accuracy is achieved using this two-dimensional approximation of a tip loss factor. However, three-dimensional tip effects and the wake nonuniformity caused by a finite number of blades limit the usefulness of this method for detailed rotor design.

3-2.1.1.4 Tip Loss Factor

The value of the tip loss factor to be used in the calculations described is dependent upon the quantity being computed, i.e., thrust or power for a given pitch angle or thrust for a given power. The selected mode of application attempts to predict all of these quantities with the same tip loss factor. It is assumed that the lift and induced drag are integrated from root to radius BR (integrated from $r = 0$ to $r = 0.97R$, where B = tip loss factor = 0.97) while the profile drag is integrated from root to tip.

Two approximate formulas for B are offered (Ref. 16), one as a function of blade loading and one as a function of blade geometry (blade radius R, and tip chord c_{tip}):

$$B = 1 - \frac{\sqrt{2C_T}}{b}, \text{ dimensionless} \tag{3-18}$$

and

$$B = 1 - \frac{c_{tip}}{2R}, \text{ dimensionless} \tag{3-19}$$

These formulas yield similar factors for conventional main rotors where $B \approx 0.97$, and the adequacy of these factors is confirmed for most purposes by test. For tail rotors with low aspect ratio AR (AR = radius^2/blade area) and high solidity σ, both Eqs. 3-18 and 3-19 yield values for B that are too high for good correlation with test results.

3-2.1.1.5 Vortex Theory—Three-dimensional Considerations

The methods of analysis in pars. 3-2.1.1.1 and 3-2.1.1.3 consider three-dimensional effects only to the extent of including a semi-empirical tip loss factor. However, to evaluate the real situation, the rotor vortex pattern and the character of the generated wake structure must be analyzed because these factors affect the induced velocity distribution along a blade. The vortex theory provides the basis from which analytical treatment of three-dimensional considerations can be derived and completes the list of classical approaches used in the development of methods for calculating realistic rotor hovering performance.

The vortex pattern created beneath a hovering rotor by a blade consists of a helical vortex sheet, the diameter and helix angle of which change as the slipstream contracts. Fig. 3-4 illustrates the path of the vortex sheet at the blade tip. The vortex sheet starts behind the blade and rolls up into a pair of counter-rotating vortices near the blade tip and root. The induced velocity distribution at the rotor and the resultant rotor loading are determined by the combined influence of the vortex patterns generated by a number of blades. At the same time, the exact vortex pattern and the vortex strength are dependent upon the magnitude and distribution of the rotor loading. A complete analysis, therefore, requires consideration of the coupled effects of wake shape, wake vortex strength, and rotor lift. Ref. 17 provides a detailed description of the various theoreti-

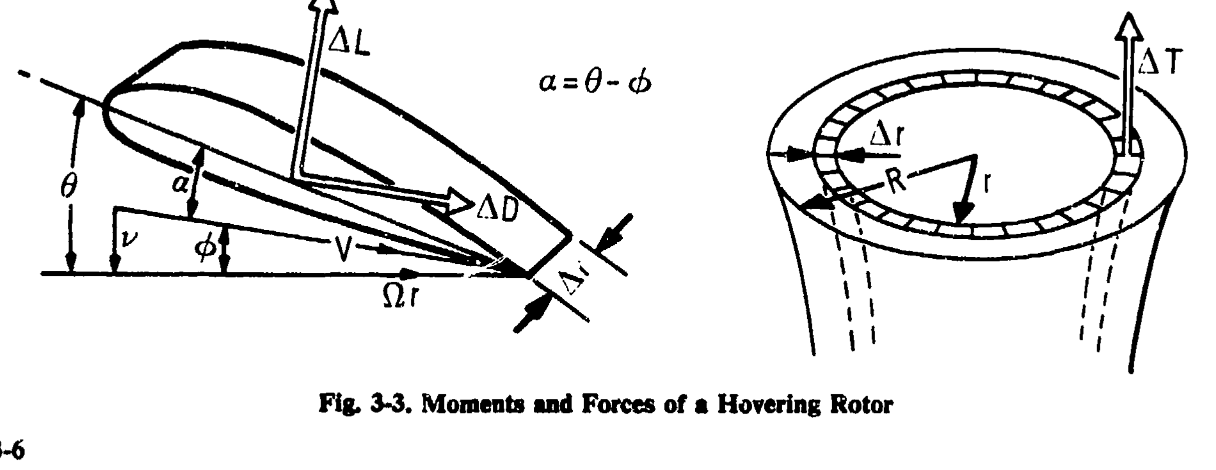


Fig. 3-3. Moments and Forces of a Hovering Rotor

cal approaches used to develop mathematical models of the vortex system and identify the modifications made to the theoretical concepts to encompass recent physical evidence of blade vortex interference and rotor wake characteristics.

The position of the vortex generated by each blade tip relative to the rotor tip path plane is dependent upon the contraction of the near wake. Consideration of this contraction, therefore, is most important in an analysis of the rotor blade interference caused by the proximity of the shed vortex to the following blades. The distortion in local angle of attack at the blade tip caused by the passage of a strong vortex generated by the preceding blade, as depicted in Fig. 3-5, produces severe profile drag losses not predicted by the previously discussed classical methods. Omission of this effect can result in highly optimistic performance estimates especially for high disk loading, high solidity rotors. Accurate definition of the wake geometry also is necessary to obtain correct axial and radial positioning of the vortex elements in the wake, which is essential for rotor geometry design optimization.

3-2.1.1.6 Theoretical Three-dimensional Prediction Method

Excellent agreement between theory and experimental results has been demonstrated with the Prescribed Wake-momentum Analysis (PWMA) (Ref. 18). This analysis uses the basic strip momentum theory, modified to include the effects of the near wake by adding a wake-induced "interference" velocity to the strip momentum calculated inflow. The wake geometry used by the PWMA is defined simply in terms of basic rotor parameters to which the wake is particularly sensitive. It has been investigated independently by analytical and experimental studies, all of which indicate basic agreement. An outline of the major iterative methods for use in the PWMA computer program is presented in Ref. 18. The airfoil characteristics used in conjunction with the PWMA are shown in Figs. 3-6 and 3-7. These data originate from two-dimensional wind tunnel tests of a production blade specimen using the NACA 0012 airfoil section.

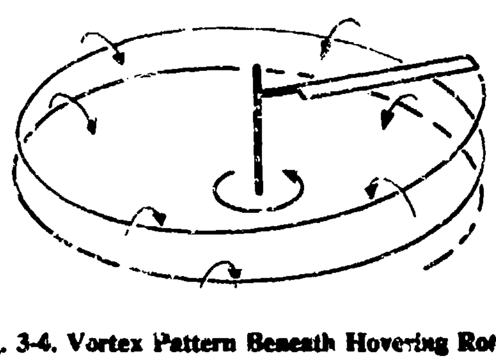


Fig. 3-4. Vortex Pattern Beneath Hovering Rotor

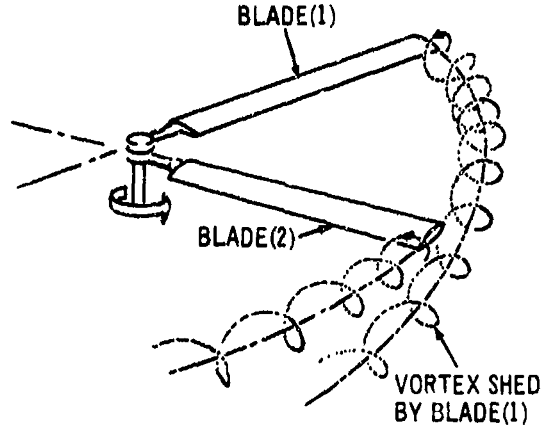


Fig. 3-5. Vortex Pattern

3-2.1.1.7 Empirical Prediction Method

A method that is considered to give excellent results in rapid prediction of rotor hovering performance at high rotor loadings is the Figure of Merit Ratio (FMR) method. This method is based upon empirical evaluation of isolated rotor whirl stand test data, and it is useful particularly for high rotor loading conditions where most other methods are highly optimistic. A significant result of this evaluation is that rotor solidity, rather than number of blades or individual blade geometry, appears to be the principal factor affecting rotor performance for a given tip Mach number and blade lift coefficient.

By equating rotor thrust (corrected empirically for blade tip losses) to rotor lift, it can be shown that the mean blade lift coefficient $\overline{C}_L$ is proportional to the parameter C_T/σ. This more readily measured parameter, together with rotor solidity σ, and tip Mach number M_t, is now used as a parameter in establishing rotor performance.

As noted in par. 3-2 1.1.2, the Figure of Merit for the ideal rotor would be unity. Realistically the maximum value of the Figure of Merit M_{max} must take into account blade profile drag and tip losses. Using the torque coefficient (C_Q) expression of Eq. 3-13 (which takes into account profile drag and tip losses), Eq. 3-14 can be rewritten as

$$M = \frac{0.707\, C_T^{3/2}}{\dfrac{C_T^{3/2}}{B\sqrt{2}} + \dfrac{\sigma \bar{C}_D}{8}}, \text{ dimensionless} \quad (3\text{-}20)$$

The maximum realistic Figure of Merit M_{max} can now be plotted as a function of C_T/σ for selected values of σ (Fig. 3-8) using the following empirical values:

$$B = 0.97$$
$$a = dc_l/d\alpha = 5.73, \text{ rad}^{-1}$$
$$\alpha = 6C_T/(\sigma a B^3), \text{ rad}$$
$$\bar{C}_D = 0.0087 - 0.0216\alpha + 0.4\alpha^2$$

The Figure of Merit Ratio (FMR) is now defined as the ratio between the actual Figure of Merit (taking into account tip Mach number M_t, blade root cutout, and other practical considerations) and the maximum Figure of Merit. The basic FMR is established empirically by normalizing all test data to −8 deg linear blade twist and 20% root cutout based upon derivatives developed from strip analysis considerations. Figs. 3-9, 3-10, and 3-11 present the resulting trends of FMR with solidity σ for several values of C_T/σ and tip Mach numbers of 0.55, 0.60, and 0.65, respectively. Actual data points for these trends are shown to illustrate the level of uncertainty inherent in the faired lines. Figs. 3-12 and 3-13 show the blade twist θ_1 and blade root

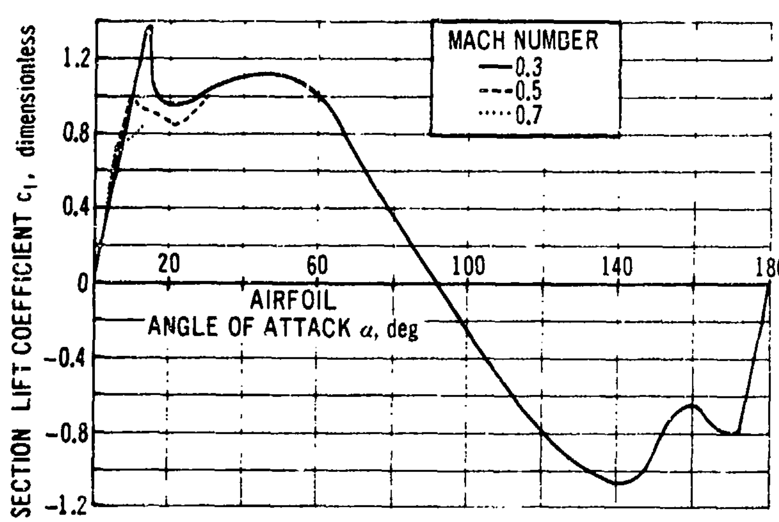


Fig. 3-6. NACA 0012 Airfoil Lift Coefficient

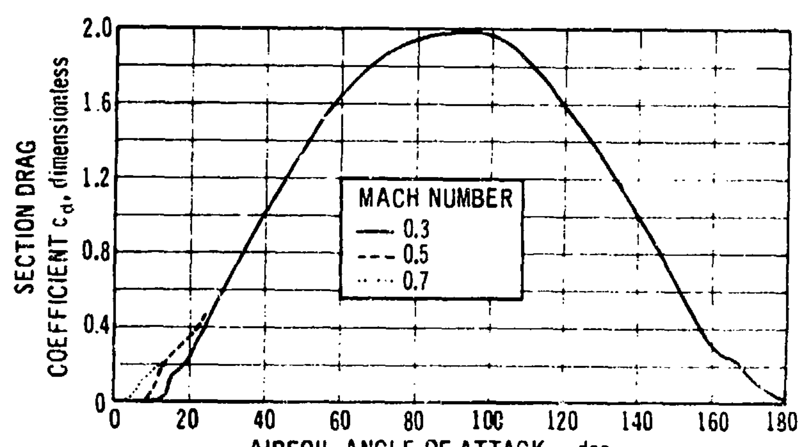


Fig. 3-7. NACA 0012 Airfoil Drag Coefficient

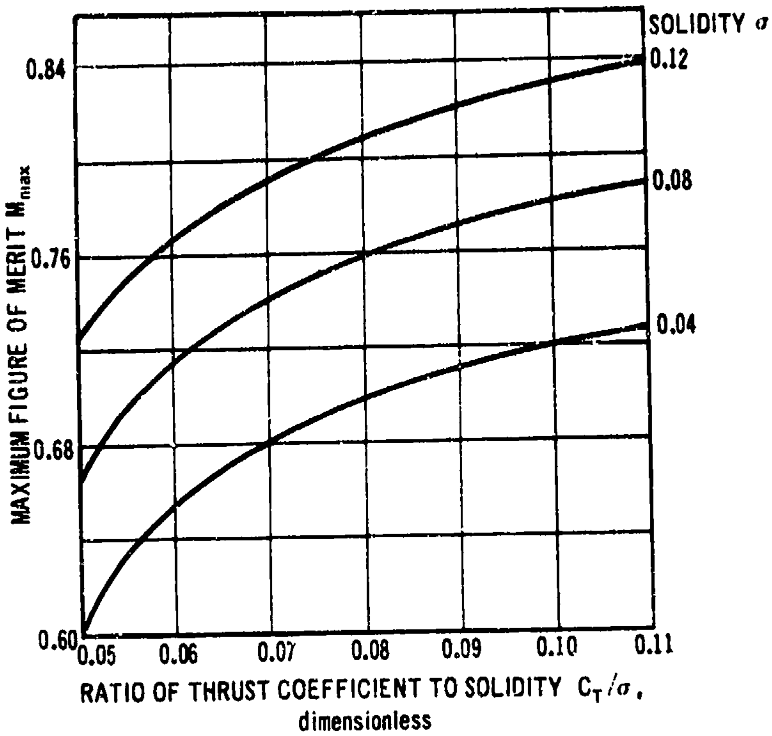


Fig. 3-8. Maximum Figure of Merit

cutout corrections. In reducing the test data, rotor whirl stand performance is adjusted for ground effect using the Cheeseman and Gregory correction (par. 3-2.1.1.8), and the results at discrete tip Mach numbers are cross-plotted to provide performance at the three Mach numbers presented.

There are inherent variants, which should be recognized whenever this method is used, and which contribute to the scatter on the plotted data points. These include:

1. Tip shape
2. Airfoil section
3. Leading edge abrasion strip geometry
4. Reynolds number.

Blade planform and section camber are two other important parameters that need to be considered in detail rotor design. Blade taper has an effect upon rotor performance similar to that resulting from increased blade twist. Both factors tend to provide a more uniform inflow distribution; this relieves the blade loading at the tip, which, in turn, reduces tip losses by decreasing the strength of the tip vortex. In addition, the combined power saving derived from lower induced and

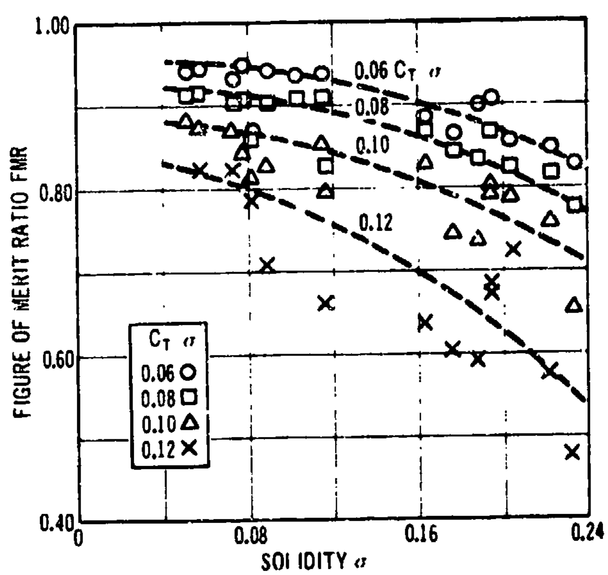


Fig. 3-9. Figure of Merit Ratio for M_t = 0.55

profile drag losses can be significant at high rotor loadings.

Section camber design considerations usually are

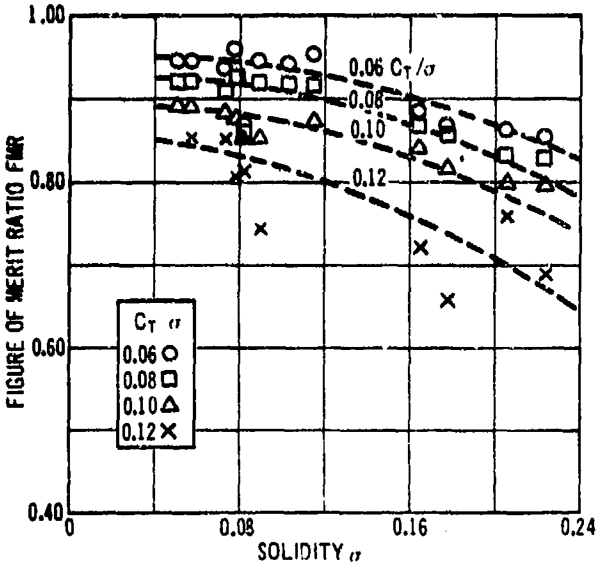


Fig. 3-10. Figure of Merit Ratio for M_t = 0.60

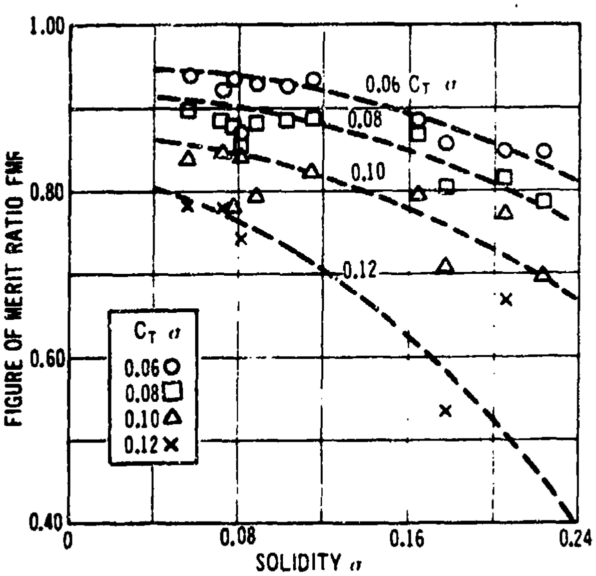


Fig. 3-11. Figure of Merit Ratio for M_t = 0.65

geared to meet specific performance requirements for a particular flight regime, and wind tunnel test programs are conducted to optimize the section characteristics sought. Section pitching moment characteristics normally are the undesirable side effects of camber in a rotating wing, insofar as rotor control load and blade stiffness requirements are concerned. However, means of minimizing inherent pitching moments have been demonstrated. Effects of various airfoils upon hov performance (horsepower required) can be appro imated by use of Eq. 3-7 with $\overline{C}_D$ adjusted for t. appropriate section.

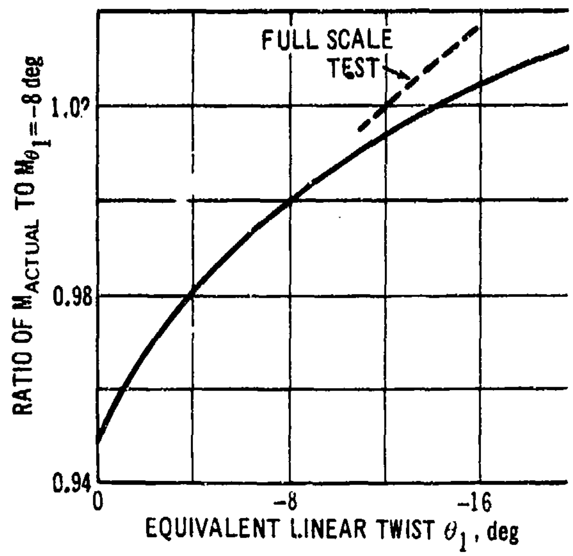


Fig. 3-12. Blade Twist Correction — Baseline: θ_1 = −8 deg

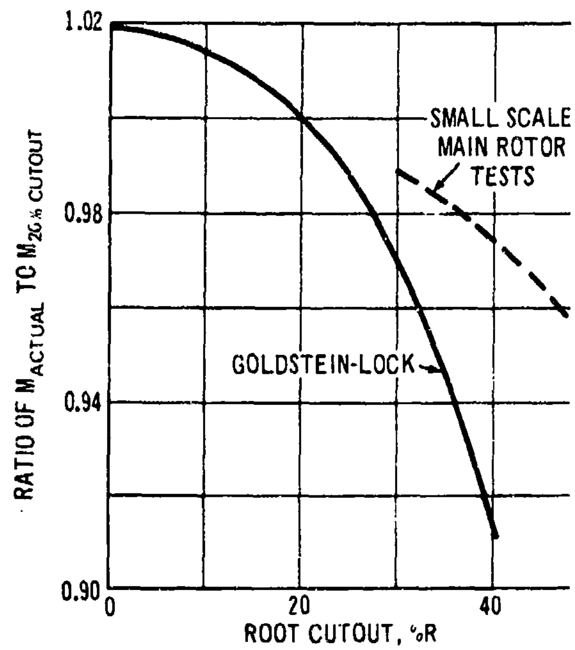


Fig. 3-13. Blade Root Cutout Correction — Baseline: 20% Cutout

3-2.1.1.8 Ground Effect

When a hovering rotor is near the ground, the velocity is resisted and reduced by the presence of the ground. This factor decreases the inflow to the rotor for a particular disk loading, resulting in a thrust increase for a given blade pitch angle. Conversely, at a constant thrust the power required to hover is reduced. The additional thrust attainable at a given power level permits greater payloads in cases where hover out-of-ground effect (OGE) is not necessary, and where translational flight can be initiated from the in-ground effect (IGE) wheel height. Ground effect corrections have been attempted theoretically using mirror image rotor systems and combinations of sources and sinks to simulate the obstruction of the wake. One such approach is that developed by Knight and Hefner in Ref. 19. In this analysis the rotor is replaced by a cylindrical vortex of strength Γ and the ground by an image vortex cylinder as shown in Fig. 3-14. The resulting induced velocities, of equal magnitude, act in opposite directions and thus cancel each other at the ground. This image vortex system can be solved by potential theory for an ideally twisted rotor (circulation constant along the blade radius and independent of the distance above the ground) to obtain the variation of the induced velocity along the blade for a range of rotor heights. Then, using the components of the ideal torque equation that vary with thrust and inflow velocity (the profile drag component $\sigma C_D/8$ is omitted), an incremental torque coefficient ΔC_Q can be calculated. The value of ΔC_Q out-of-ground effect is designated ΔC_{Q_∞} and a ground effect correction factor Λ is defined as

$$\Lambda = \frac{\Delta C_Q}{\Delta C_{Q_\infty}}, \text{ dimensionless} \tag{3-21}$$

where ΔC_Q and ΔC_{Q_∞} are given by

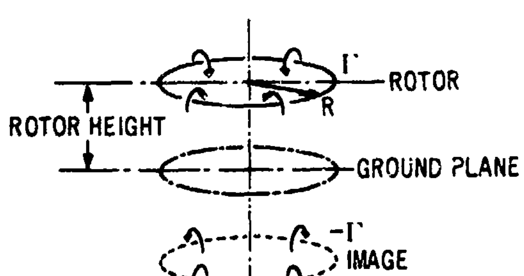


Fig. 3-14. Replacement of Rotor and Ground by Cylindrical Vortex and Image Vortex

$$\left.\begin{aligned}\Delta C_Q &= \frac{C_{T_I}^{3/2}}{B\sqrt{2}} + \frac{2}{3}\frac{\delta_1}{a}\frac{C_{T_I}}{B^2} + \frac{4\delta_2}{\sigma a^2}\left(\frac{C_{T_I}}{B^2}\right)^2, \text{ dimensionless}\\ \Delta C_{Q_\infty} &= \frac{C_{T_\infty}^{3/2}}{B\sqrt{2}} + \frac{2}{3}\frac{\delta_1}{a}\frac{C_{T_\infty}}{B^2} + \frac{4\delta_2}{\sigma a^2}\left(\frac{C_{T_\infty}}{B^2}\right)^2, \text{ dimensionless}\end{aligned}\right\} \tag{3-22}$$

where

C_{T_I} = thrust coefficient corrected for reduced inflow due to ground effect, dimensionless

a = slope of lift coefficient vs angle of attack curve ($dc_l/d\alpha$), rad^{-1}

δ_1, δ_2 = coefficients of equation $c_d = \delta_0 + \delta_1\alpha + \delta_2\alpha^2$, dimensionless

Calculation of Λ will provide a family of nondimensional curves for selected values of C_T/σ^2 (Fig. 3-15). These can be used readily to compute changes in power required to hover as a function of the rotor height above the ground Z; these curves can be approximated by $\Lambda = [Z/(2R)]^{1/3}$. This theory has been shown to be accurate for $Z/R \geq 1$; empirical methods, discussed in succeeding paragraphs, should be used at lower heights.

Theoretical ground effect corrections involve ideal rotors, and do not account for all factors that influence the rotor wake. To develop realistic analytical models, flight test data and small-scale investigations have been used to determine empirical modifications to theoretical ground effect equations. Cheeseman and Gregory (Ref. 20) have used a mirror image rotor system theoretical approach and single rotor helicopter test data to develop an empirical relationship. By use of the ideal power relationship, which states that the main rotor power equals the product of the rotor thrust and the induced velocity, the thrust augmentation ratio can be expressed as $T/T_\infty = v_\infty/v$ at a constant power. The velocity induced at the center of the rotor by its mirror image is

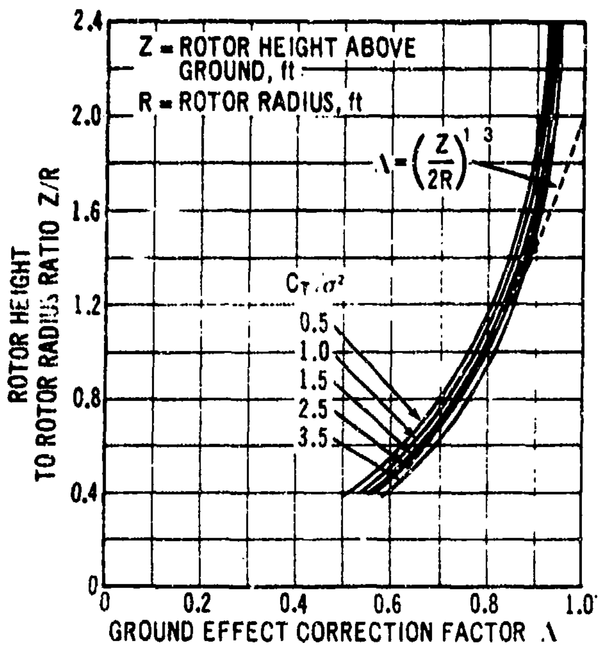


Fig. 3-15. Knight and Hefner Ground Effect Correction

$$\Delta\nu = \frac{A\nu}{16\pi Z^2}\ , \text{fps} \qquad (3\text{-}23)$$

Assuming ν_∞ and $\Delta\nu$ are constant over the disk, ν equals $(\nu_\infty - \Delta\nu)$ and

$$\frac{T}{T_\infty} = \frac{1}{1-(R^2/16Z^2)}\ , \text{dimensionless} \qquad (3\text{-}24)$$

To determine the effects of blade loading on ground effect, the OGE thrust T_∞ from blade element theory must be stated,

$$T_\infty = \frac{\rho abc\Omega^2 R^3}{2}(\theta/3 - \lambda/2)\ , \text{lb} \qquad (3\text{-}25)$$

where

a = slope of lift coefficient versus angle of attack curve $(dc_l/d\alpha)$, rad^{-1}

λ = inflow ratio, $(V\sin\alpha - \nu)/(\Omega R)$, dimensionless

Using Eqs. 3-23 and 3-25 and the inflow ratio λ relationship, Cheeseman derives the following equation:

$$\frac{T}{T_\infty} = 1 + \left\{\frac{0.25\eta_c a\sqrt{\sigma}}{\sqrt{2C_T/\sigma}}\right\} \times \left\{\frac{1}{16(Z/R)^2[1+(V/\nu)^2]}\right\}\ , \text{d'less} \qquad (3\text{-}26)$$

where

η_c = Cheeseman forward flight efficiency factor

In hover $V = 0$ and when $\left[(0.25\eta_c a\sqrt{\sigma})/\sqrt{2C_T/\sigma}\right] = 1$, Eq. 3-26 approximates Eq. 3-24. The Cheeseman equation correlates well for lightly loaded rotors. For higher disk loadings, however, the equation is somewhat optimistic and flight data should be used if available. Ref. 20 also includes an analysis of tandem ground effect.

Aircraft IGE vertical drag (par. 3-2.1.1.9) decreases from its OGE value. This decrease normally is contained in the empirical ground effect correlation. Flight tests of a helicopter at a series of wheel heights provide a good estimate of the IGE characteristics of that particular vehicle. These data then can be used, in conjunction with theoretically or empirically derived corrections, to define the ground effect characteristics for a helicopter similar in shape and vertical drag to the flight-tested aircraft.

3-2.1.1.9 Vertical Drag/Thrust Recovery

Vertical drag D_V is the drag upon the aircraft caused by rotor-induced velocities and is analogous to the parasite drag that occurs in forward flight. Vertical drag is added to gross weight to determine the total vertical thrust to be provided by the rotor. In vertical climb, the airflow velocity is increased, causing greater vertical drag. Thrust recovery is the thrust augmentation resulting from the proximity of the airframe to the rotor, which creates a partial "ground effect". Wake distortion around the fuselage usually is relatively minor; but when wings or bulky equipment are incorporated, the wake distortion becomes significant. The downwash is increased at outward radial stations. Additional power required due to the distorted wake is included in the thrust recovery term. In practice, the net vertical drag, which is the net effect of the download and the thrust recovery, ranges in value from 2% to 10% of the gross weight. The influence of the net vertical drag is even more important when considered in terms of payload, because the given percentages are doubled or tripled.

The analysis of vertical drag is based upon empirical methods at the present time. The purely theoretical analysis of an airframe as a three-dimensional body immersed in a nonuniform, accelerating, and separating flow would be a very difficult task. Therefore, a combination of simplified theoretical methods and test data is used in vertical drag prediction. The two basic means for evaluating vertical drag are the polar area moment of inertia ratio method and the strip analysis method.

1. Polar area moment of inertia ratio method. This method is based upon a simple velocity distribution beneath a hovering rotor over areas of characteristic shape.

The basic assumption underlying this approach is that the rotor has a parabolic spanwise blade loading, which is true for low linear-twisted blades. For a parabolic loading, the velocity distribution is triangular and the slipstream dynamic pressure varies parabolically along a blade, i.e., as a function of r^2. Since the vertical drag of an element is the product of the slipstream dynamic pressure, drag coefficient, and fuselage (or wing) element area, the drag is proportional to (area $\times$ r^2), or polar area moment of inertia I_p. The I_p ratio is now defined as the ratio between I_p and the polar area moment of inertia of the rotor disk, $(\pi R^4)/2$.

Section drag coefficients (normally difficult to determine directly), thrust recovery, and flow peculiarities are not calculated directly, but are accounted for by using test results for the variation of net vertical drag with the polar area moment of inertia. Fig. 3-16 presents the net vertical drag as a percentage of rotor thrust versus I_p ratio for the extreme body geometries—cylinders and flat plates—and an average helicopter fuselage. The usefulness of this figure can be enhanced by the addition of comparable lines for additional fuselage types.

Use of this method is rapid and includes the following steps:

a. Draw the planform of the helicopter.

b. Assume a nominal wake contraction to 90% at the fuselage, and divide the helicopter within this wake area into convenient geometric elements.

c. Calculate the polar area moment of inertia for each geometric element and add the results.

d. Calculate the polar area moment of inertia of the rotor disk, $\pi R^4/2$.

e. Obtain the ratio of airframe to rotor polar moment of inertia, I_p ratio (Item c divided by Item d).

f. Enter Fig. 3-16 at the total aircraft I_p ratio and read the total net vertical drag percentage on a curve representative of the fuselage shape under consideration.

The method also can be used by obtaining the polar area moment ratio of each element separately, entering Fig. 3-16 for each I_p ratio, and summing the elemental net vertical drag values. The advantage of the second method is that the relative weighting of cylindrical and flat plate elements is simpler. However, the total aircraft effect—including representative interference and thrust recovery effects—as determined by testing is lost.

The method as presented does not compensate for the effects of rotor geometry, rotor characteristics, and fuselage (element) vertical position; additional correction factors must be applied for any effect that will influence the wake geometry significantly.

A sample net vertical drag calculation for a typical compound helicopter is given in Fig. 3-17.

2. Strip analysis method. A vertical drag prediction method that inherently should be more accurate than the polar area moment of inertia method is strip analysis. This method consists of the determination of the slipstream geometry below the rotor and the estimation of element two- and three-dimensional drag coefficients produced by the interaction of this wake with the airframe. A thrust recovery factor is applied to the gross drag to give the net vertical drag.

Experience has shown that the appropriate drag coefficients are difficult to determine due to uncertainties in actual Reynolds numbers and to complexities in body shape. Correlation with test data has indicated that drag coefficients higher than normal two-dimensional steady-state drag coefficients must be used.

An accurate determination of the wake geometry is necessary for reasonable calculation of impingement velocities. Wake geometries may be determined using various techniques; e.g., analytical free wake velocity distributions; analytical velocity distribution at a blade, extended below the rotor using the results of smoke studies or wake velocity measurements (Ref. 21); or wake surveys using hot wire or pitot static probes. A typical isolated rotor wake distribution is shown in Fig. 3-18. Experiments have shown that the wake is distorted by the presence of a surface; the wake is expanded by the body, causing a greater portion of the airframe to be affected by the downwash, as shown by Fig. 3-19.

Use of the strip method involves the division of the aircraft planform into small elements. The velocity at each element is found from wake velocity charts, and representative drag coefficients are assumed. Using the conventional drag equation $D = C_D \rho V^2 A/2$, the total download can be found. This value is increased to account for interference and protuberances ($\approx 10\%$) and reduced to reflect the thrust recovery (≈30%), giving the net vertical drag.

The uncertainties of both of the methods cited are large, up to ±3% of the gross weight at a 95% confidence level. Small-scale data have been used to provide correction factors used in the methods, along with a limited amount of flight test data. In the latter case, vertical drag is derived from a comparison of isolated rotor whirlstand test data with flight test information. The uncertainty of prediction can be removed only when present methods are modified by results from coordinated full-scale vertical drag tests, out-of-ground effect, conducted expressly to examine the mechanisms of thrust recovery and download.

Uncertainties in the strip analysis method result from the difficulty of determining appropriate drag coefficients and the proper wake profile. The strip analysis method, however, is very useful in trending. When the approximate base vertical drag value is known, the effects of changes in wake geometry due to alterations in twist distribution, solidity, or fuselage shape can be evaluated with relatively good accuracy.

Wind and ground effect cause additional uncertainties due to wake distortion, and these factors are difficult to estimate accurately.

3-2.1.1.10 Power Train Effects

Main rotor power must be increased to allow for other sources of power consumption. Losses from these sources normally are stated in an overall power train efficiency factor η. Included in this factor is power absorbed by accessories, transmissions, transmission-driven oil pumps and cooler blowers, generators, hydraulic pumps, and tail rotor.

Power required to operate each piece of equipment is determined for particular flight conditions. The total efficiency can be summarized in two convenient formats, as shown in Fig. 3-20. The format of Fig. 3-20(A) gives a mean efficiency and assumes that power required by each component varies linearly with main rotor power at a given advance ratio, μ, where μ equals $V\cos\alpha/(\Omega R)$. Fig. 3-20(B) accounts more accurately for the variation in gearbox power requirements and in the tail rotor power absorption, but does not account fully for the variation of tail rotor power with velocity. If greater accuracy is required, the efficiency must be calculated for each combination of power level, airspeed, and atmospheric condition.

3-2.1.2 Forward Flight

3-2.1.2.1 Energy Methods

A relatively simple approach to forward flight rotor performance calculation is the energy method. The power absorbed by the rotor may be divided into the following elements:

1. Induced power, to sustain lift
2. Profile power, to overcome the drag of the rotor blades
3. Parasite power, to pull the fuselage and rotor head through the air
4. Climb power, to change the aircraft potential energy
5. Acceleration power, to change the aircraft or rotor kinetic energy
6. Power required to overcome vertical drag.

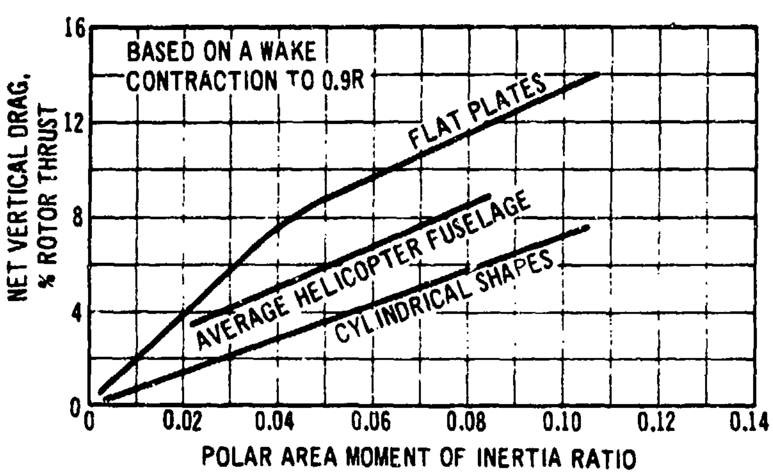
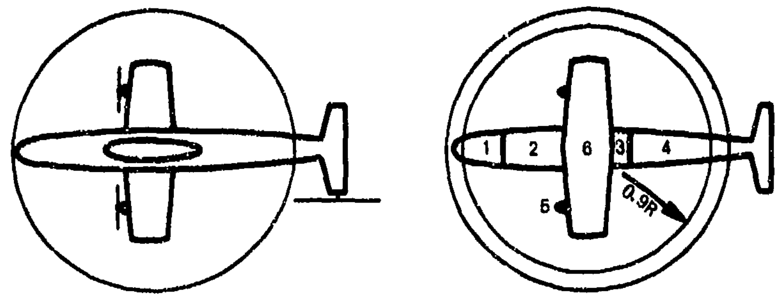


Fig. 3-16. Vertical Drag



AREA NO.	I_P, ft^4	I_P ROTOR, ft^4	I_P RATIO
1	0.0667 x 10^6	4.02 x 10^6	0.0166
2	0.0375 x 10^6	4.02 x 10^6	0.0094
3	0.0068 x 10^6	4.02 x 10^6	0.0017
4	0.0985 x 10^6	4.02 x 10^6	0.0245
5	0.0037 x 10^6	4.02 x 10^6	0.0009
6	0.0728 x 10^6	4.02 x 10^6	0.0181
		TOTAL	0.0712

METHOD 1 – USE TOTAL I_P RATIO, VERTICAL DRAG D_V = 7.7%

METHOD 2 – USE ELEMENTS

I_P RATIO OF FUSELAGE AND NACELLES	0.0531,	D_V = 3.8%
I_P RATIO OF WING	0.0181,	D_V = 3.3%
TOTAL	0.0712	7.1%

ADD 10% FOR INTERFERENCE D_V = 1.1 x 7.1% = 7.8%

Fig. 3-17. Sample Net Vertical Drag Calculation for Compound Helicopter

In steady, level flight the power required by the rotor can be grouped into the following elements:

1. Induced power to sustain a force normal to the flight path
2. Profile power to overcome the drag of the rotor blades
3. Parasite power to sustain a force along the flight path.

These are discussed separately in the paragraphs that follow.

3-2.1.2.1.1 *Induced Power*

The induced power in forward flight is the power required to accelerate air downward to create a force normal to the flight path. An induced velocity v is added to the free-stream velocity V, as shown in Fig. 3-21, such that the resultant velocity V' is given by

$$V' = \sqrt{(V - v\sin\alpha)^2 + (v\cos\alpha)^2}\ , \text{fps} \qquad (3\text{-}27)$$

Similar to the development of the induced power hp_i in hover (Eq. 3-5),

$$hp_i = \frac{T}{550}(v - V\sin\alpha) \qquad (3\text{-}28)$$

and can also be expressed as the change in KE of the air mass ($\rho A V'$) passing through the rotor

$$hp_i = \frac{1}{1100}\left[(V - v_2\sin\alpha)^2 + (v_2\cos\alpha)^2 - V^2\right]\rho A V' \qquad (3\text{-}29)$$

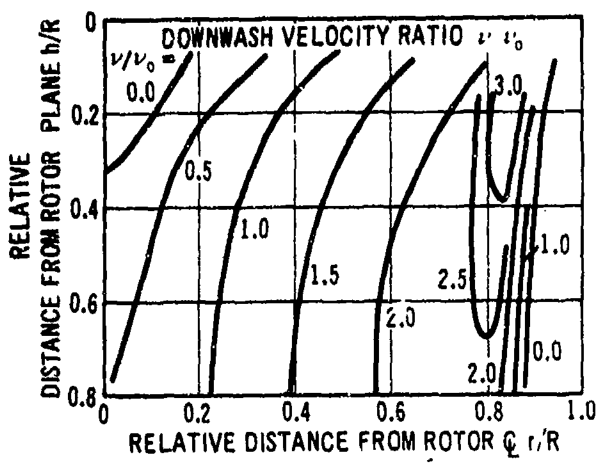


Fig. 3-18. Wake Profile, Out-of-ground Effect, Isolated Rotor

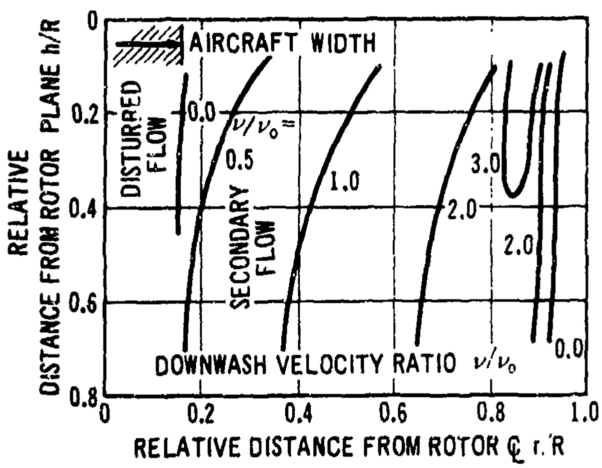


Fig. 3-19. Wake Profile, Out-of-ground Effect, With Aircraft (Estimated)

Since the thrust T equals the change in momentum perpendicular to the rotor plane,

$$T = \rho A V' \nu_2 \quad \text{, lb} \tag{3-30}$$

Combining Eqs. 3-28, 3-29, and 3-30 results in the same relationship between ν_2 and ν as for hover flight, namely, $\nu_2 = 2\nu$. Equating the right hand side of Eqs. 3-28 and 3-29, and substituting for ν_2, we obtain

$$\left(\frac{T}{2\rho A}\right)^2 = \nu^4 - 2V\nu^3 \sin\alpha + V^2\nu^2 \quad \text{, ft}^4\text{/sec}^4 \tag{3-31}$$

Nondimensionalizing by introducing the induced velocity in hover ν_0, where ν_0 equals $[T/(2\rho A)]^{1/2}$ (Eq. 3-4), we arrive at Wald's Equation (Ref. 22)

$$\left(\frac{\nu}{\nu_0}\right)^4 - 2\left(\frac{\nu}{\nu_0}\right)^3\left(\frac{V}{\nu_0}\right)\sin\alpha + \left(\frac{\nu}{\nu_0}\right)^2\left(\frac{V}{\nu_0}\right)^2 = 1 \quad \text{, dimensionless} \tag{3-32}$$

This equation had been plotted by Coleman, et al. (Ref. 23) and has the form shown in Fig. 3-22. The solution for ν in Eq. 3-32 is not straightforward. The thrust is assumed vertical for the computation of induced

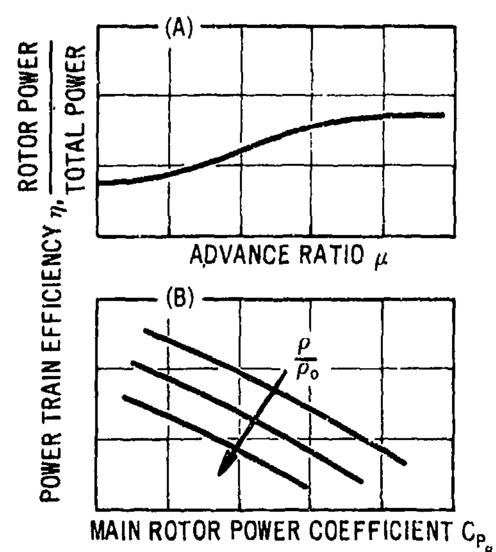


Fig. 3-20. Formats for Total Efficiency

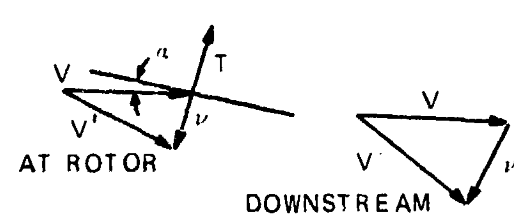


Fig. 3-21. Calculation of Resultant Velocity V' in Forward Flight

power, and the parasite power is left as a separate term. The profile and the parasite powers are discussed in pars. 3-2.1.2.1.2 and 3-2.1.2.1.3, respectively.

With the rotor angle of attack $\alpha = 0$, Eq. 3-27 becomes

$$V' = \sqrt{V^2 + \nu^2} \text{ , fps} \tag{3-33}$$

and Eq. 3-31 reduces to

$$\nu^4 + V^2\nu^2 - \left(\frac{T}{2\rho A}\right)^2 = 0 \quad \text{, ft}^4\text{/sec}^4 \tag{3-34}$$

Solving Eq. 3-34 yields

$$\nu = \left\{\frac{-V^2}{2} + \sqrt{\left(\frac{V^2}{2}\right)^2 + \left(\frac{T}{2\rho A}\right)^2}\right\}^{1/2} \text{ , fps} \tag{3-35}$$

When $T/(2\rho A) < V^2/2$, ν is approximately equal to $T/(2\rho AV)$. Using Eq. 3-28, we arrive at

$$hp_i = \frac{T^2}{1100\rho AV} \tag{3-36}$$

which is the induced power required to produce lift. Assuming that the thrust T is equal to the gross weight W_g and considering that the effective blade radius in the production of thrust is only BR, the induced power also can be expressed as

$$hp_i = \frac{W_g}{550}\frac{\nu}{\nu_0}\sqrt{\frac{W_g}{2\rho B^2 A^2}} \tag{3-37}$$

To make use of Eq. 3-37, ν/ν_0 is determined from Fig. 3-22.

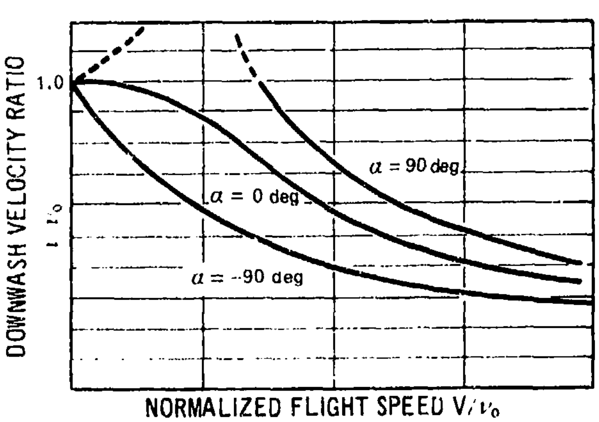


Fig. 3-22. Wald's Equation

3-2.1.2.1.2 Profile Power

The profile power hp_o in forward flight is computed assuming a mean drag coefficient $\overline{C}_D$. In general

$$hp_o = \frac{b}{1100\pi}\int_0^{2\pi}\int_0^R \frac{\rho c}{2}\overline{C}_D(\Omega r + \mu\Omega R\sin\psi)^3\,dr\,d\psi \tag{3-38}$$

where

μ = advance ratio using only the component of velocity normal to the blade in the plane of rotation, $V\cos\alpha/(\Omega R)$, dimensionless

ψ = azimuth angle of rotor blade from its downwind position, rad

Nondimensionalizing and accounting for the sign of the velocity in the reversed flow region, we have

$$C_{P_o} = \frac{\sigma}{2\pi}\int_0^{2\pi}\int_0^1 \overline{C}_D(x + \mu\sin\psi)^3\,dx\,d\psi$$

$$-2\int_0^{2\pi}\int_0^{-\mu\sin\psi} \overline{C}_D(x + \mu\sin\psi)^3\,dx\,d\psi \text{ , d'less} \tag{3-39}$$

where

C_{P_o} = profile power coefficient, dimensionless

x = generalized radial distance, r/R, dimensionless

Integrating Eq. 3-39 gives

$$C_{P_o} = \frac{\sigma\overline{C}_D}{8}\left(1 + 3\mu^2 + \frac{3}{8\mu^4}\right) \text{ , dimensionless} \tag{3-40}$$

The advance ratio μ used in Eqs. 3-38 and 3-39 is defined using only the component of velocity normal to the blade in the plane of rotation. Consideration of the radial and vertical velocity components in these equations would be very difficult; however, it has been shown that Eq. 3-40, accounting for these velocities, is approximately

$$C_{P_o} = \frac{\sigma \overline{C}_D}{8}(1 + 4.65\mu^2) \quad \text{, dimensionless} \qquad (3\text{-}41)$$

and, using the basic power coefficient equation and substituting for σ profile horsepower hp_o becomes

$$hp_o = \left(\frac{\rho bcR^4\Omega^3\overline{C}_D}{4400}\right)(1 + 4.65\mu^2) \qquad (3\text{-}42)$$

Eq. 3-42 equals Eq. 3-7 when $\mu = 0$.

The mean drag coefficient $\overline{C}_D$ is found from Fig. 3-1 when the mean lift coefficient $\overline{C}_L$ is determined by calculating thrust in a manner analogous to the power integral

$$T = \frac{b}{2\pi}\left[\int_0^{2\pi}\int_0^{BR} \frac{\rho c}{2}\overline{C}_L(\Omega r + \mu\Omega R \sin\psi)^2\, dr\, d\psi \right.$$

$$\left. -2\int_0^{2\pi}\int_0^{-\mu R\sin\psi} \frac{\rho c}{2}\overline{C}_L(\Omega r + \mu\Omega R \sin\psi)^2 dr\, d\psi\right] \text{, lb} \qquad (3\text{-}43)$$

Integrating Eq. 3-43 and using the basic thrust coefficient equation yields the mean lift coefficient $\overline{C}_L$

$$\overline{C}_L = \frac{6C_T/\sigma}{B^3 + \frac{3}{2}B\mu^2 - \frac{4}{3}\frac{\mu^3}{\pi}} \quad \text{, dimensionless} \qquad (3\text{-}44)$$

At high forward speeds a significant portion of the rotor disk is in a region of reversed flow, where the rotational speed of the inboard blade elements is less than the flight velocity. Hence the local velocity at the blade element V_L impinges the blade trailing edge. This occurs near $\psi = 270$ deg, within an area defined by $r = -\mu \sin\psi$ (Fig. 3-23). The effect of the reversed flow is incorporated into Eqs. 3-43 and 3-39 for thrust and profile power coefficients, respectively. The portion of the disk in reversed flow is assumed to have a negative lift equal in magnitude to the lift calculated for that region by Eq. 3-43. The profile power is not affected directly by the direction of flow, but a correction is required to maintain a positive drag contribution for all elements, assuring, in effect, that the inflow velocity cubed is always positive.

3-2.1.2.1.3 Parasite Power

The power required to overcome all drag not formed in producing lift is called parasite power. The method widely used to calculate this power is that of summation of component parasite drag contributions. In making an assessment of aircraft component drag by this method, it is convenient to use the equivalent flat plate concept, where the drag is expressed as the area f of a flat plate, with a drag coefficient of unity, that would produce the same drag force as the component under consideration. The drag force D normally is defined as

$$D = \frac{\rho V^2}{2}C_D S = qC_D S \quad \text{, lb} \qquad (3\text{-}45)$$

where

q = dynamic pressure, $(\rho V^2)/2$, psf
C_D = drag coefficient, dimensionless
S = component reference area, ft^2

and in terms of an equivalent flat plate area f,

$$D = qf \quad \text{, lb} \qquad (3\text{-}46)$$

The parasite power hp_p is, therefore,

$$hp_p = \frac{DV}{550} = \frac{qf_T V}{550} \qquad (3\text{-}47)$$

where

f_T = the total equivalent flat plate area of all components,

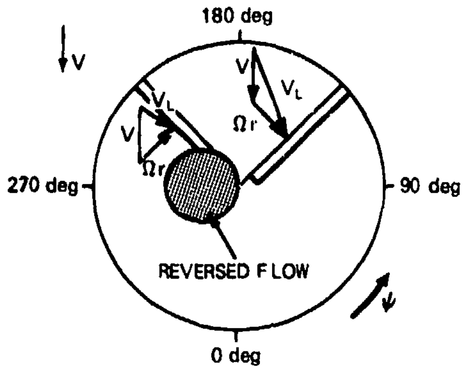


Fig. 3-23. Effect of Reversed Flow

including their mutual interference effects, ft^2

From Eqs. 3-45 and 3-46 the value of f for a particular component is

$$f = C_D S \quad , ft^2 \tag{3-48}$$

The value of the component drag coefficient C_D is dependent upon the reference area, e.g., wetted area, projected area, etc., under consideration and the relative location of the component, which may be under the influence of secondary flow rather than freestream flow. An excellent source of information concerning the drag coefficients of aircraft components is Ref. 24.

One major source of drag in a helicopter is the main rotor head/pylon assembly. Important facts derived from tests to date include:

1. Significant drag reduction is attainable with a properly configured fairing.
2. The rotor head/pylon combination must be considered as a whole. Isolated rotor head drag is not a true measure of total effect.
3. The rotor head wake and associated turbulence in the tail area can be influenced significantly by rotor head/pylon geometry.
4. A rotor head fairing must be sealed aerodynamically, particularly at the junction with the pylon, to be effective.

The total drag area of the aircraft usually is calculated for a level aircraft attitude. However, an estimate of the trim attitude for various forward flight conditions must be made, and drag area variation with fuselage angle of attack established. This is essential to realistically estimate the power requirements. The effect of fuselage lift (positive or negative) should also be established, since this can also affect the power requirements to maintain a given flight condition.

3-2.1.2.2 Blade Element Theory

The blade element theory for determining the characteristics of a lifting rotor in forward flight has been in the development stage for a number of years. The progress in the early years is summarized in Ref. 16, which also contains a basic treatment of the theory. This paragraph deals primarily with the theory and techniques developed by Wheatley (Ref. 25) and Bailey (Ref. 26). The Wheatley/Bailey method largely has been replaced by the more refined theories that now are possible with the aid of digital computers. However, it is the basis upon which the more sophisticated methods are founded and still is useful as a means for obtaining relatively quick answers when computer services are unavailable.

3-2.1.2.2.1 Wheatley Method

The general integral expressions for thrust, drag, and torque for a rotor using the blade element theory are exceedingly complex. By means of certain simplifying assumptions, Wheatley developed the theory to the extent that further refinement would make the expressions unmanageable. The assumptions used are:

1. Induced angles of attack α_i, inflow angles ϕ, and blade flapping angles β are small such that $\sin \alpha_i = \alpha_i$, $\sin \phi = \phi$, and $\sin \beta = \beta$
2. The radial component of the resultant air velocity at each blade element may be neglected.
3. The flapping hinge is located on the axis of rotation.
4. The profile drag coefficient and lift-curve slope are constant.
5. Blades are considered rigid.
6. Blades have linear twist; i.e., pitch is of the form $\theta = \theta_0 + (r/R)\theta_1$, where θ_1 is the difference between root and tip pitch angles.
7. The expression for blade flapping is of the form $\beta = a_0 - a_1 \cos \psi - b_1 \sin \psi - a_2 \cos 2\psi - b_2 \sin 2\psi$, where ψ is the azimuth angle of the blade from its downwind position.
8. Powers of μ of the fifth magnitude and above are neglected.

In addition to the given assumptions, approximate methods are introduced for dealing with blade tip losses and with the effect of reversed flow over the retreating blade.

To bring the resulting Wheatley expressions for rotor thrust torque and drag into a manageable form, the method of "t-coefficients" may be used (Ref. 26). For a given rotor tip loss factor B and a Lock number $\gamma = c\rho a R^4/I$, (where I is the mass moment of inertia of a blade about the flapping hinge, slug-ft^2), the theoretical equations for thrust coefficient, flapping coefficients, torque coefficients, and profile drag-lift ratio may be written as polynomials in terms of the inflow ratio λ, the advance ratio μ, and the blade pitch angle parameters θ_0 and θ_1. The coefficients in these expressions (the "t-coefficients") have been evaluated over a range of μ's for $B = 0.97$ and $\gamma = 15$. The Lock number γ is the ratio of the aerodynamic forces affecting blade flapping to the flapwise moment of inertia of the blade. The "t-coefficients" are relatively insensitive to changes in B and γ (Refs. 16 and 26).

3-2.1.2.2.2 *Iterative Procedure*

When using the Wheatley equations for a particular flight condition, an iterative solution is necessary. In general, λ and θ_0 are not known, and must be solved by iterating between the equations that contain the two known dependent variables.

To reduce much of the iterative work necessary in deriving forward flight performance, a method has been developed that uses charts to facilitate the procedure. The approach taken is a combination of blade element analysis and energy methods. The total power hp_T in unaccelerated steady level flight may be expressed as

$$hp_T = hp_i + hp_o + hp_p \tag{3-49}$$

where subscripts are

T = total
i = induced
o = profile
p = parasite

The total effective drag D_e is given by

$$D_e = D_i + D_o + D_p \quad \text{, lb} \tag{3-50}$$

or

$$\frac{D_e}{L_R} = \frac{D_i}{L_R} + \frac{D_o}{L_R} + \frac{D_p}{L_R} \quad \text{, dimensionless} \tag{3-51}$$

where

L_R = rotor lift, lb

For $L_R \approx T$, $\alpha \approx 0$, $T/(2\rho A) < V^2/2$, and $v < V$, using Eq. 3-36 we have

$$\frac{D_i}{L_R} = \frac{D_i V}{L_R V} = \frac{P_i}{L_R V} = \left(\frac{T^2}{2\rho A V}\right)\left(\frac{1}{L_R V}\right)$$

$$\approx \frac{L_R}{2\rho\pi R^2 V^2} = \frac{C_{LR}}{4} \quad \text{, dimensionless} \tag{3-52}$$

where

C_{LR} = rotor lift coefficient, $L_R/[(1/2)\rho V^2 \pi R^2]$, dimensionless

At speeds lower than $\mu = 0.10$ or at large rotor angles of attack, this expression for D_i/L_R breaks down because the induced velocity becomes large relative to the forward-flight speed. In either of these cases,

$$\frac{D_i}{L_R} = \frac{C_T}{2\mu(\mu^2 + \lambda^2)^{1/2}} \quad \text{, dimensionless} \tag{3-53}$$

From Eq. 3-46

$$\frac{D_p}{L_R} = \frac{\rho V^2 f}{2L_R} = \frac{f}{\pi R^2 C_{LR}} \quad \text{, dimensionless} \tag{3-54}$$

D_o/L_R may be obtained from the Bailey tables for any specific combinations of λ and θ and charts have been constructed from those results that relate D_o/L_R to C_{LR}/σ at specific levels of D_e/L_R. These charts are given in Ref. 16 for untwisted blades and in Ref. 27.

The basic steady forward flight performance may be evaluated as follows:

1. Given a required C_{LR}/σ and μ, assume D_e/L_R.
2. Calculate D_i/L_R and D_p/L_R.
3. Determine D_o/L_R from charts.
4. Sum the (D/L_R)'s to get D_e/L_R for comparison with that assumed.
5. Repeat Steps 1 through 4 for another assumed D_e/L_R and interpolate to find D_e/L_R for which output matches input.

3-2.1.2.2.3 *Blade Stall*

An additional important result of simple blade element theory is the prediction of section angle of attack over the rotor in order to evaluate the extent, if any, of blade stall. For rotor blades the maximum blade angle of attack occurs at the blade tip near $\psi = 270$ deg. Consequently, stall usually begins at that point and $\alpha_{1.0,270}$ can be used as a criterion for the onset of stall. At the retreating tip, the angle of attack is

$$\alpha_{1.0,270} = \theta + \frac{\lambda - \dfrac{\dot\beta}{\Omega} - \mu\beta}{1-\mu} \quad \text{, rad} \tag{3-55}$$

where

$\dot\beta/\Omega = -\theta_1 + 2b_2$, $\beta = a_0 + B_1 + a_2$.

Therefore, using the Bailey method, $\alpha_{1.0,270}$ can be found for any given θ and λ. The remaining portions of the rotor continue to function normally, but develop more than their share of thrust to compensate for the losses experienced in the stalled region, which yields less lift but more drag. Indications are that the profile power factors are increased in proportion to the degree by which the retreating blade tip angle exceeds the stalling angle, thus

$$hp_o = \frac{\sigma \overline{C}_D \rho \pi R^2 (\Omega R)^3}{4400} \left(1 + 4.65\mu^2 + \frac{\alpha_{1.0,270} - \alpha_{DD}}{\alpha_{st} - \alpha_{DD}}\right) \quad (3\text{-}56)$$

where

α_{DD} = angle of attack for drag divergence, rad

α_{st} = angle of attack where objectionable stall occurs, rad

The forward speed at which onset of blade stall first is noted generally is the speed for best range. This is due to the marked increase in profile power required at higher speed. Vibration levels also seem to increase at the higher speeds due to asymmetrical loading.

3-2.1.2.2.4 Numerical Methods

High-speed computers permit detailed iterative calculations using element theory (par. 3-2.1.2.2). The theory described in Refs. 25 and 26 can be used without the aid of advanced numerical methods in the low-speed flight regime, but is subject to inaccuracies in moderate- to high-speed flight regimes.

The current applied-element theory of the Generalized Rotor Performance (GRP) Computer Program eliminates assumptions of the simpler methods, including Mach number effects on c_l and c_d, reversed flow region characteristics, and some of the restricting small angle assumptions. This method does retain some assumptions, however, among which are:

1. Steady-state airfoil data
2. Two-dimensional flow at each blade section (no spanwise flow)
3. Constant rotational speed about the shaft axis
4. Uniform rotor inflow
5. Rotor blade rigid in bending and torsion.

Numerical methods exist to eliminate all of the given assumptions except 3, with an associated increase in program complexity. A nonrigid blade, for example, is useful for stress calculation as well as for aerodynamic analyses.

The basic numerical method is useful in level flight, climb, or descent calculations. In addition to rotor geometry values, the GRP computer program requires inputs for inflow, advance ratio, collective and cyclic pitch angles, and starting values of the flapping angle and its derivatives. Two-dimensional airfoil and span data (as a function of angle of attack and Mach number) also must be provided.

The GRP computer program is based upon the solution of the flapping motion from highly nonlinear equations for the sum of the moments about the flapping hinge in a shaft-axis system. These moments are the aerodynamic moment M_A, the weight moment M_W, the inertia moment M_I, and the elastic flapping hinge restraint M_E, such that $M_A + M_W + M_I + M_E = 0$. A numerical solution is found by finite differences and a step-by-step procedure, knowing that the steady-state flapping is periodic $\beta(\psi) = \beta(\psi + 2\pi)$.

Time-averaged values of rotor forces and moments are found by summing along a blade and averaging around the azimuth the aerodynamic loads on the blade. Iteration to specific lift and drag values can be accomplished by modifying the inflow and the collective pitch or longitudinal cyclic pitch using a Taylor series and partial derivatives with respect to the parameters cited, based upon the Wheatley-Bailey method. The flapping iteration is started with the new initial values for inflow and collective pitch or longitudinal cyclic pitch. A solution is reached when the drag and lift components are within specified tolerances. The basic flow chart is presented in Fig. 3-24.

3-2.1.2.2.5 Graphical Method–NASA Charts

The charts of Ref. 28, also known as the NASA Charts or Tanner Charts, are a series of curves developed from the Generalized Rotor Performance program (par. 3-2.1.2.2.4). The numerical procedure was used to calculate nondimensional rotor performance for linear blade twists θ_1 of -8, -4, and 0 deg, for advance ratios μ from 0.25 to 1.4, and advancing tip Mach numbers M_t between 0.7 and 0.9. Use of the charts is recommended where high-speed computers are not available. Sufficient information is obtainable from them to evaluate a rotor system. The charts are presented in terms of dimensionless coefficients of rotor lift C_L, drag C_D, and total torque C_Q, for various values of rotor control plane angle of attack α_C.

Fig. 3-25 illustrates a typical pair of charts. These curves are calculated for a nominal configuration consisting basically of rectangular-planform blades with a 25% cut-out, zero offset, and a tip loss factor of 0.97. A rotor solidity of 0.1 is used in the charts but corrections to other solidities can be calculated readily. Detailed examples of the use of these charts are given in Ref. 28.

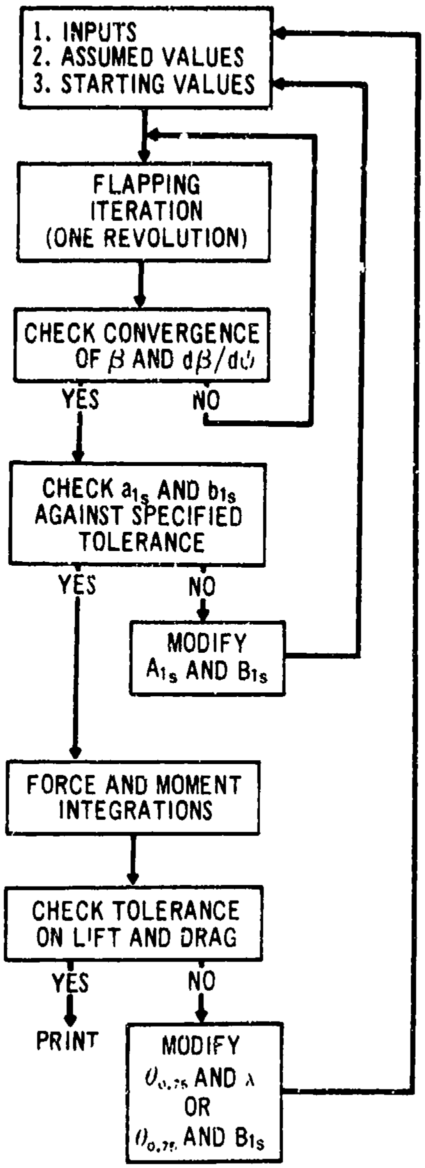


Fig. 3-24. Basic Flow Chart for Numerical Method

The NASA Charts represent the performance of an isolated rotor. To obtain total aircraft performance, airframe aerodynamics also must be considered. The airframe contributes lift and drag as functions of body pitch, yaw attitude, and flight velocity. Expressions for $d\psi_{fus}/dV$ and $d\alpha_{fus}/dV$ may be found by flight testing and then used to select the proper body attitude to enter curves showing the variation of lift and drag with body attitude (where ψ_{fus} and α_{fus} are the yaw angle and angle of attack, respectively, of the fuselage reference line). Examples of these curves are shown in Fig. 3-26.

The power required to operate aircraft accessories, transmissions, and the tail rotor can be combined into a single efficiency factor η (par. 3-2.1.1.10). Total power coefficient C_{P_T} is calculated by dividing the charted torque coefficient $C_{Q_{ch}}$ by the efficiency.

$$C_{P_T} = \frac{C_{Q_{ch}}}{\eta} \text{, dimensionless} \qquad (3\text{-}57)$$

Tail rotor performance can be obtained from the charts to establish this efficiency.

3-2.1.2.2.6 *Radial Flow Corrections*

The numerical methods and the charts described in pars. 3-2.1.2.2.4 and 3-2.1.2.2.5 do not consider the radial component of rotor velocity U_r parallel to the blade span. The radial component U_r (Fig. 3-27), which significantly increases the profile drag, can be expressed as

$$U_r = \mu\Omega R \cos\psi + \lambda\Omega R\beta \quad \text{, fps} \qquad (3\text{-}58)$$

For a first-order approximation

$$\lambda\Omega R\beta \approx 0 \quad \text{, fps} \qquad (3\text{-}59)$$

adding the tangential component U_t equal to $(\Omega r + \mu\Omega R \sin\psi)$, the resultant velocity U of U_r and U_t can be approximated

$$U \approx \left[(\Omega r + \mu\Omega R \sin\psi)^2 + (\mu\Omega R \cos\psi)^2\right]^{1/2} \quad \text{, fps} \qquad (3\text{-}60)$$

The integration and averaging of expressions for the rotor force perpendicular to the control axis H and the decelerating torque provide an expression for the profile torque coefficient C_{Q_o}

$$C_{Q_o} = \left(\frac{\sigma\bar{C}_D}{8}\right)\left(1 + n_1\mu^2 + n_2\mu^4\right) \quad \text{, d'less} \qquad (3\text{-}61)$$

and n_1 and n_2 are evaluated numerically from the complete expression. The sum of n_1 and n_2 ranges from 4.5 at $\mu = 0.0$ to 4.89 at $\mu = 0.50$. Ignoring the radial flow yields

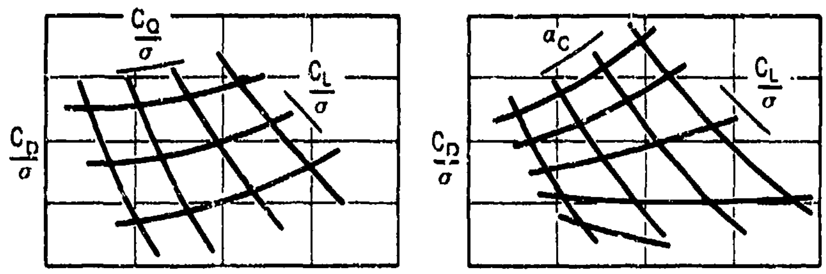


Fig. 3-25. Typical Charts for Estimating Performance

$$C_{Q_o} = \left(\frac{\sigma \overline{C}_D}{8}\right)(1 + 3\mu^2) \quad \text{, dimensionless} \qquad (3\text{-}62)$$

The differences between Eqs. 3-61 and 3-62 must be added to the profile power assumed by the NASA Charts and the numerical computation. The radial flow correction ΔC_{Q_o} then is equal to

$$\Delta C_{Q_o} = \left(\frac{\sigma \overline{C}_D}{8}\right)(n\mu^2)\text{, dimensionless} \qquad (3\text{-}63)$$

where n has the following values (Ref. 29):

μ	n
0.00	1.50
0.10	1.50
0.20	1.57
0.30	1.67
0.40	1.78
0.50	1.89
0.60	2.03
0.75	2.24
1.00	2.62

For a given value of drag coefficient, the value of ΔC_{Q_o} can be calculated and added to the value C_{Q_o} from the charts.

3-2.1.3 Tandem-rotor Interference

Aerodynamic interference caused by the overlap or intermesh of tandem-rotor systems increases the power required to produce a given total thrust over that required for the same rotors when isolated. Because the profile drag coefficient of a rotor operating at moderate thrust coefficients is nearly constant, this increase in total power is attributable to an increase in induced power only. For a given tandem-rotor overlap, the ratio of the induced power of the tandem-rotor system to the induced power of the two isolated single rotors provides an index of the level of aerodynamic interference. This ratio is referred to as the tandem rotor interference factor K such that:

$$K = \frac{C_{Q_T} - C_{Q_o}}{C_{Q_s} - C_{Q_o}} = 1 + \frac{\Delta C_Q}{C_{Q_i}} \quad \text{, dimensionless} \qquad (3\text{-}64)$$

where

C_{Q_T} = total torque coefficient of the tandem rotor system, dimensionless

C_{Q_o} = profile torque coefficient for the single rotor, dimensionless

C_{Q_s} = total torque coefficient of each single rotor, dimensionless

ΔC_Q = increase in tandem torque coefficient over the single rotor torque coefficient, dimensionless

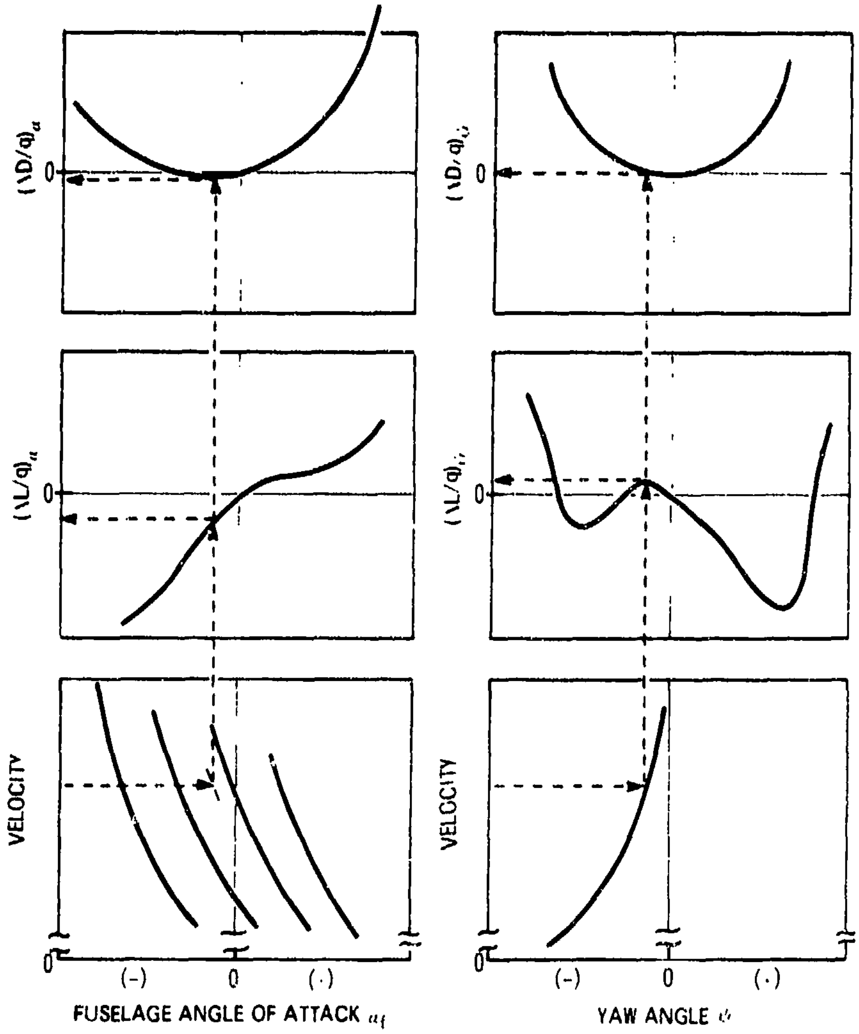
C_{Q_i} = induced torque coefficient of the single rotor, dimensionless

The theoretical ideal induced power hp_i required by a tandem-rotor system to generate a given thrust then can be expressed in terms of simple momentum theory (Eq. 3-5) as

$$hp_i = \frac{Tv}{550} K = \frac{KT^{3/2}}{550\sqrt{2\rho A}} \qquad (3\text{-}65)$$

In this case A is the projected disk area of the two rotors (see Fig. 3-28).

In the hovering regime numerous tests have resulted in a power correction factor as a function of rotor shaft spacing ratio s_R (see Fig. 3-28) that agrees reasonably well with theoretically derived trends (Fig. 3-29). The test data, obtained with rotors employing blades with



NOTE: TOTAL BODY DRAG = $DRAG_{\psi=0,\,\alpha=0} + [(\Delta D/q)_\alpha + (\Delta D/q)_\psi]\,q$
TOTAL BODY LIFT = $LIFT_{\psi=0,\,\alpha=0} + [(\Delta L/q)_\alpha + (\Delta L/q)_\psi]\,q$

Fig. 3-26. Airframe Effects, Effect of Fuselage Pitch and Yaw

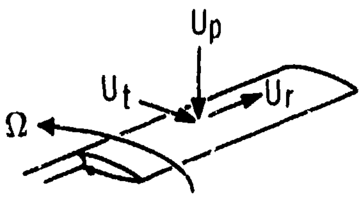


Fig. 3-27. Velocity Components

planforms and twists consistent with standard practice (not assuming "ideal" twist or taper) fall, as would be expected, between the theoretical curves for uniform and parabolic spanwise blade loading derived in Ref. 30. The estimation of vertical drag (par. 3-2.1.1.9) must include the effects of the increased downwash of the adjacent or overlapped rotors.

In forward flight, the effect of interference on induced power is considerably greater than in hover because the air inflow is nearly horizontal, reducing the effective aspect ratio of the lifting system by about one-half compared with two isolated rotors. This effect can be approximated analytically by using the analogy of two wings in tandem, with appropriate values of gap and stagger, and defining the induced downwash relationship in the system (Ref. 31). The induced power correction factor K_u that results from this approach is expressed as

$$K_u = 1 + \frac{d_f}{2} \text{, dimensionless} \tag{3-66}$$

where d_f, induced power interference parameter, is given by

$$d_f = \frac{\sqrt{1 + s_R^2} + s_R \cos\gamma}{\sqrt{1 + s_R^2}\,(1 + s_R^2 \sin^2\gamma)} \text{, d'less} \tag{3-67}$$

where

s_R = ratio of the distance between rotor shafts and rotor radius R, dimensionless, (see Fig. 3-28)

γ = wake skew angle, rad

and the wake skew angle γ is calculated approximately from

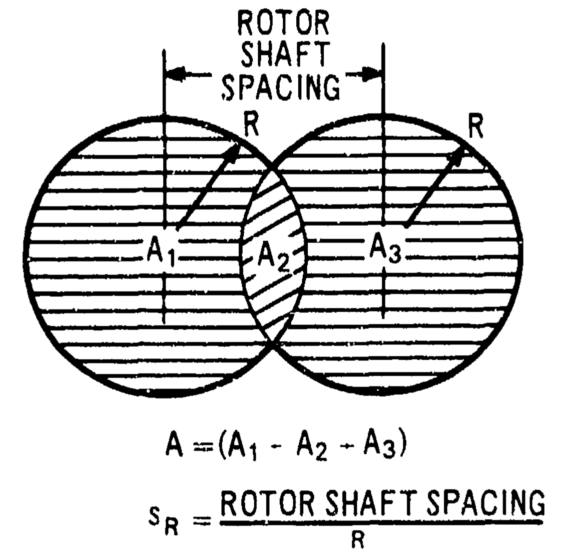


Fig. 3-28. Projected Disk Area, Overlapping Rotors

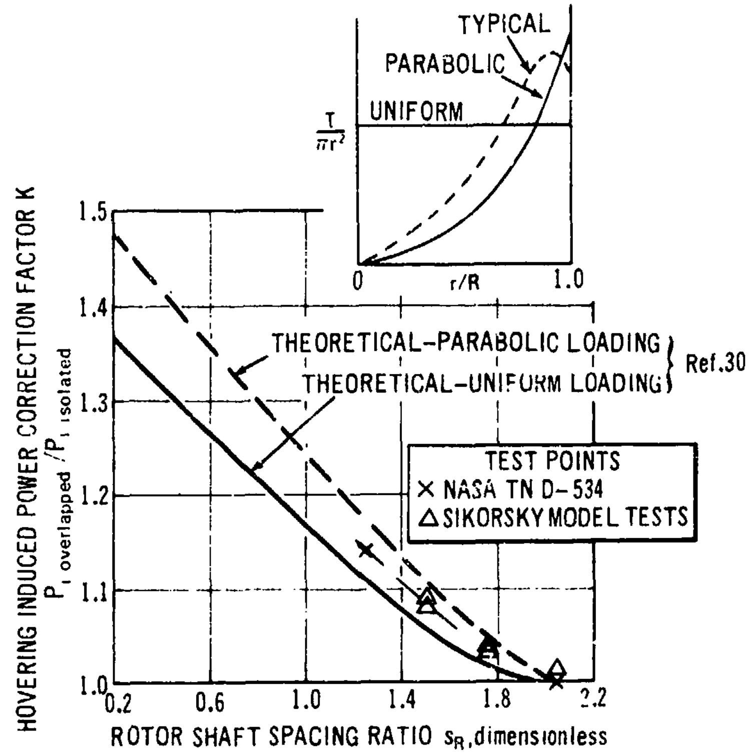


Fig. 3-29. Hovering Induced Power Correction Due to Overlap

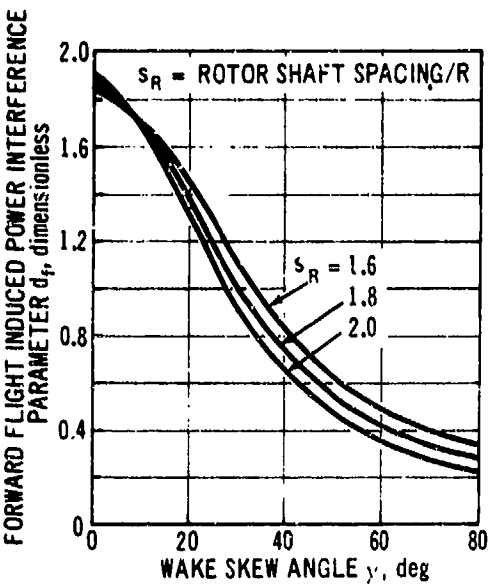


Fig. 3-30. Tandem-rotor Interference Factors

$$\gamma = \mathrm{Tan}^{-1}\left(\frac{1.57T_f}{2\rho A_f V^2}\right), \text{rad} \qquad (3\text{-}68)$$

where

T_f = thrust of forward rotor, lb
A_f = area of the forward rotor, ft^2

Eq. 3-67 is presented graphically by Fig. 3-30. Note that this method, at reasonable values of γ(0 deg $\leq \gamma \leq$ 10 deg), indicates a value for d_f of the order of 1.8, which represents a 90% increase in the induced power of the tandem system over that for the single rotor. This method provides a guide to the order of magnitude of tandem-rotor interference in forward flight. However, consideration of tip losses, which reduce the diameter of the stream tube of inflowing air, decreases the effective aspect ratio even more; hence, induced power correction factors of more than 2.0 are possible. Actual variation of K_u with forward speed is difficult to establish because it varies with aircraft geometry. Furthermore, flight techniques such as sideslip flight have been developed to compensate for the rotor interference penalty. The amount of sideslip used is determined by the best trade-off between the reduction in power losses due to rotor interference and the increase in parasite power that results from sideslip flight.

3-2.1.4 Fixed Aerodynamic Surfaces

Estimation of the characteristics of a wing or fin surface under a given set of conditions can be accomplished readily provided that information on lift and drag coefficients as a function of angle of attack is available for some given aspect ratio. At a required lift coefficient, the angle of attack, drag coefficient, and lift curve slope for the planform under consideration are found by the method of Ref. 32. This method includes the introduction of spanwise efficiency factors into the theoretical relationship between two-dimensional and three-dimensional airfoil characteristics based upon an elliptical distribution of lift over the three-dimensional (finite span) wing. Once the general nondimensional aerodynamic characteristics of the wing or similar surface are established, the lift and drag forces at any condition are calculated by the conventional relationships of fixed-wing aerodynamics,

$$L = \frac{\rho V^2}{2} SC_L \quad , \text{lb} \qquad (3\text{-}69)$$

and

$$D = \frac{\rho V^2}{2} SC_D \quad , \text{lb} \qquad (3\text{-}70)$$

where

S = planform area of the surface, ft^2

3-2.2 COMPOUND CONFIGURATIONS

3-2.2.1 Speed Capability of Helicopters

A compound helicopter is one that has a wing to relieve the rotor of its lifting requirements, partially or totally, and an auxiliary propulsion device to relieve the rotor of its propulsion requirements partially or totally. Such a helicopter can achieve a higher maximum forward speed than can a conventional helicopter, and retains hovering capability.

The maximum speed attainable by a conventional helicopter is limited by drag divergence on the advancing blade and/or by one of the several phenomena lumped together under the term "blade stall" on the retreating blade. To obtain the highest maximum aircraft speed, a rotor speed must be selected that allows the retreating blade to experience blade stall at the same time that the advancing blade experiences drag

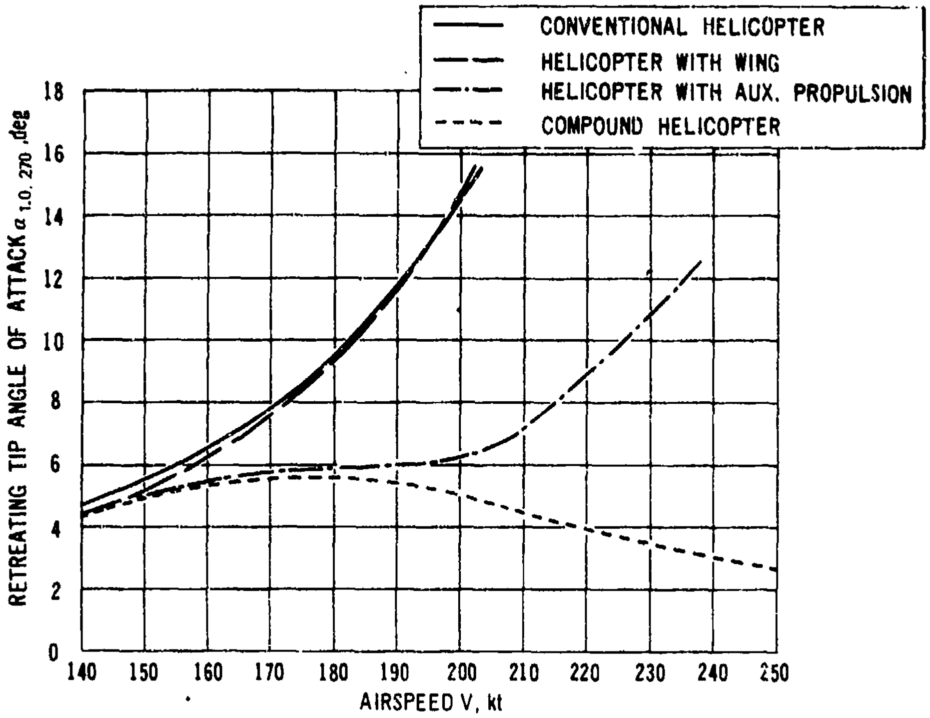


Fig. 3-31. Retreating Tip Angle of Attack

divergence. Fig. 3-31 shows the retreating tip angle of attack (which is an indicator of blade stall) as a function of forward speed for a conventional helicopter, a helicopter with a wing, a helicopter with auxiliary propulsion, and a helicopter with both a wing and auxiliary propulsion (compound helicopter). In this example, the same rotor is assumed on each of the aircraft. The rotor has a solidity of 0.1, a blade twist of -5 deg, and a hovering disk loading of 5 lb/ft^2. The rotor, and hence tip, speed is adjusted with forward speed so that the advancing tip Mach number is kept constant at 0.92. Fig. 3-31 shows that, for a retreating tip angle of attack of 12 deg, both the conventional and the winged helicopter can fly to about 190 kt, the helicopter with auxiliary propulsion can fly to about 235 kt, and the compound helicopter has no speed limitation due to blade stall. The maximum speeds of the first three types might be raised slightly by optimizing solidity and twist, but the values presented are typical of the present state-of-the-art.

3-2.2.2 Preliminary Design Considerations

Once it has been decided that a compound helicopter configuration is necessary to satisfy a given speed requirement, the goal of the preliminary design effort should be the same as for any other aircraft; i.e., to obtain the smallest physical dimensions, lightest weight, and most efficient design that can satisfy simultaneously all of the requirements specified.

For any aircraft the design requirements tend to be conflicting, but for compound helicopters the conflict is dramatic and is characterized by the philosophy that whatever helps the high-speed capability hurts the hovering capability and vice versa. For a given payload and mission time, the size of the aircraft is determined by the type of auxiliary propulsion and the rotor disk loading selected. If the mission requires a relatively short period of high-speed flight, a simple turbojet or fanjet engine may be used; this approach is attractive especially when a compound configuration evolves from an existing helicopter design. For relatively long high-speed mission legs, the complexity of a propeller usually will prove to be justified by its higher propulsive efficiency. Another configuration that might be considered during preliminary design is a pressure-jet rotor that operates in autorotation in high-speed flight while the driving gases are ducted to a nozzle for forward propulsion.

If, for the sake of simplicity, a separate jet engine is selected for auxiliary propulsion, the selection of the rotor power plant rating and the rotor disk loading is based upon the hovering requirement and the basic trade-off is the same as for a conventional helicopter: i.e., the larger the power plant, the higher the disk loading can be, thus producing a smaller aircraft. If a propeller is chosen for auxiliary propulsion and is to be driven by the same engine as the rotor, the engine rating usually is set by the high-speed requirement and the rotor disk loading selected is the highest at which the hover requirement can be met. In some cases the disk loading is not determined in this manner but is limited to some smaller value by a maximum limitation upon the rotor downwash velocity or by consideration of good autorotational characteristics.

3-2.2.3 Selection of Configuration Parameters

3-2.2.3.1 Wing

The wing is required for high-speed operation, but it produces performance penalties in hover. Therefore, the definition of its parameters is an exercise in "least-worst compromise" rather than in "optimization". These wing parameters, discussed below, include:

1. Area
2. Span
3. Incidence
4. Incorporation of flaps
5. Location.

The wing area will depend, to a large extent, upon the predicted performance, vibration, and stress characteristics of the rotor. From a performance standpoint, the rotor should carry as much load as possible at high speed; but from vibration and stress standpoints, the rotor should be unloaded completely. Thus, the design criterion for wing lift is some fraction of the aircraft gross weight at high speed, and the value of this fraction is dependent upon the degree of confidence in the rotor design. Once the design value of wing lift is selected, primarily upon the basis of experience with similar aircraft, the wing area can be calculated.

Another type of criterion that might define the wing area is a specified maneuver, such as a 2-g turn at 150 kt. If the combination of the rotor, as defined by the specified hover capability, and the wing, as defined by the high-speed requirement, is not sufficient to produce 2 g's, either the rotor blade area or the wing area must be increased. In some cases it will prove advantageous to make the change to the wing rather than to the rotor; but, to minimize the wing structural weight and the download in hover, the wing area should be no larger than is necessary to satisfy the most critical requirement. Based upon theory and some test data, the wing download in hover is equal approximately to one-half the disk loading times the wing area exposed to the rotor wake.

The selection of the wing span involves a trade-off study. Large span has the advantage of less induced drag in forward flight, but has the disadvantages of higher structural weight and more download in hover due to greater rotor wake velocities at the outer sections of the wing. The selection of the wing span must be based upon a judgment as to the relative importance of the conflicting requirements of speed, hover performance, and empty weight.

The selection of wing incidence with respect to the rotor shaft will be influenced by the desire to operate the rotor at low angles of attack at high speeds in order to minimize the oscillatory rotor loads. Thus, the rotor disk ideally should be horizontal at high speeds and the wing incidence, adjusted for the rotor-induced downwash angle, will be equal to the angle of attack at which the wing develops the required lift coefficient.

For most compound helicopter designs, the wing thickness, taper ratio, and airfoil section are chosen on the basis of the same considerations as are used in designing a fixed-wing aircraft to be flown at low subsonic speed. These factors are structural efficiency, low drag, and a high maximum lift coefficient.

Flaps on the wing of a compound helicopter might be considered for several reasons, e.g., to minimize the basic wing area if the required area is dictated by a maneuver requirement. In such a case the additional complexity of a flapped wing and its increased drag during the maneuver would have to be weighed against the predicted decreases in hover download and, possibly, wing weight. Another possible application of a wing flap is to decrease the hover download by deflecting the flap down 90 deg, thereby decreasing the wing area exposed to the rotor wake. An additional use for a flap is to decrease wing lift during entries into autorotation by upward deflection of the flap. This requirement applies primarily to configurations that do not use propellers for auxiliary propulsion; the phenomenon that generates this requirement is discussed in par. 3-2.2.4.3.

Items that should be considered in choosing the vertical location of the wing generally are the same as for a fixed-wing aircraft. These include crashworthiness, fueling convenience, crew visibility, landing gear arrangement, external store loading, and field-of-fire restriction. In addition, the designer of the compound helicopter must provide adequate clearance between the wing and the rotor during maneuvers and between the wing and the ground during slope landings. In choosing the fore-and-aft location of the wing, it should be recognized that the wing can be used to increase longitudinal stability by locating the aerodynamic center of the wing to the rear of the most aft operational center of gravity (CG) position. Sweepback should be considered if the structural design dictates that the wing spars should join the fuselage forward of the most aft CG position.

3-2.2.3.2 Rotor

The broad requirements for the rotor are: in hover, to operate efficiently; at moderate speeds, to provide its share of the required maneuvering capability; and, at high speeds, to produce as much lift as possible without producing excessive vibration or loads. Once the disk loading is chosen, either on the basis of using all of the installed power to hover or on the basis of maximum allowable downwash velocity, several other rotor parameters must be defined:

1. Tip speed
2. Solidity
3. Twist
4. Airfoil section.

The rotor tip speed will be established by the combination of the maximum forward speed and the maximum allowable advancing tip Mach number. Current practice places this Mach number at 0.85 to 0.95, to avoid drag divergence, depending upon the tip airfoil section used. Fig. 3-32 shows maximum tip speed as a function of forward speed for several Mach numbers. Whether the resulting rotor speed should be used throughout the flight envelope is dependent upon additional design considerations. The weights of the rotor and the drive system can be minimized by designing for a relatively high hover tip speed (700-800 fps). If the maximum tip speed dictated by the high-forward-speed requirement approaches these values, then the same tip speed should be used throughout the flight envelope; but if the tip speed for high-speed flight is considerably below this range, a two-speed design should be considered in order to take advantage of the potential weight saving. Factors that must be considered when making this decision are the potential problems involved in operating the rotor over a wide rpm range; noise due to high tip speeds in hover; and, in the case of a compound with a propeller, the desire to operate the propeller at low tip speeds in hover and at high tip speeds in forward flight (the opposite of the requirement for the rotor). The latter problem can be eliminated by declutching the propeller from the drive system in hover and low-speed flight. The power thus saved in hover should be enough to justify the weight of the clutch and perhaps even its development cost.

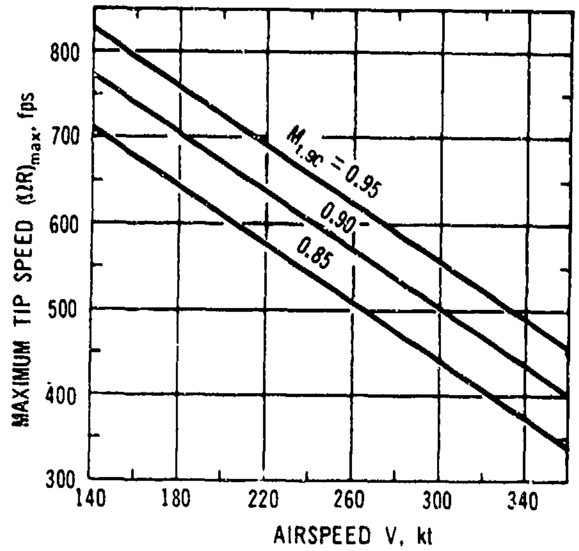


Fig. 3-32. Tip Speed Limitations for Operation at Constant Mach Number

The rotor solidity is determined by one or the other of two requirements, hover performance or maneuver capability. For a given disk loading and tip speed, the solidity helps to define the hover Figure of Merit. It has been shown that the Figure of Merit is also a function of C_T/σ, twist, and tip Mach number (par. 3-2.1.1.7). For steady turns, the rotor must supply whatever lift the wing cannot. The solidity that meets this requirement is

$$\sigma_{opt_{max}} = \frac{nW_g - L_{w_{max}}}{\rho(\Omega R)^2 A\left(\frac{C_T}{\sigma}\right)_{max}}, \text{ d'less} \qquad (3\text{-}71)$$

where

n = load factor for turn, dimensionless
W_g = helicopter gross weight, lb
$L_{w_{max}}$ = maximum lift of wing, lb

The usable value of $(C_T/\sigma)_{max}$ for a new rotor design is not defined easily, but a compilation of wind tunnel and flight test data leads to the empirical plot of $(C_T/\sigma)_{max}$ as a function of advance ratio μ, of Fig. 3-33, which represents the boundary above which the rotor profile power rises rapidly, indicating significant areas of blade stall. The solidity corresponding to the best hover Figure of Merit and the solidity that satisfies the

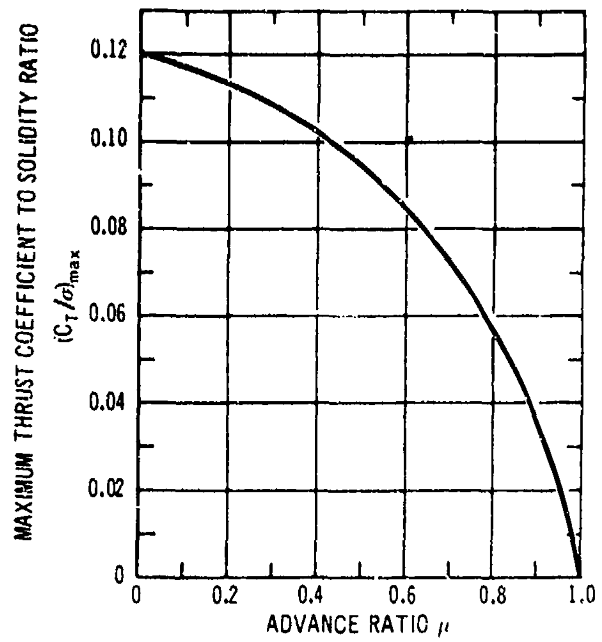


Fig. 3-33. Maximum Thrust Coefficient/Solidity for Steady Flight

maneuver requirement should be compared and the highest value chosen for the design.

High blade twist is helpful in increasing the hover performance, but it can produce large oscillatory blade loads at high forward speeds. The trade-off is difficult to assess during preliminary design so a moderate value of about -5 to -8 deg usually is chosen as a compromise.

In order to operate at high speed with high advancing tip Mach numbers, the airfoil section of the blades section should be thin. In order to be efficient in hover and to have good maneuvering capability, the blades should be thick. The trends of critical drag divergence Mach number M_{DD} and of section maximum lift coefficient $c_{l_{max}}$ with thickness ratio are shown on Fig. 3-34. A partially satisfactory theoretical solution to these conflicting requirements is to use a blade that is tapered in thickness from the root to the tip where the highest Mach numbers are experienced. Whether or not this actually can be done is dependent somewhat upon the method of blade construction (par. 4-9.1). Forward camber, sometimes called "droop snoot", may be used to make thin sections act like thicker sections with respect to maximum lift coefficient by giving the critical upper nose a gentler contour that keeps the airflow from separating prematurely, yet does not affect the critical Mach number materially.

3-2.2.3.3 Auxiliary Propulsion

If a jet engine is selected for propulsion, the choice of its location is relatively flexible, with constraints imposed by balance considerations; the necessity to avoid placing critical components, ground personnel, or flammable ground objects in the hot exhaust; and noise levels in the cockpit and passenger/troop cabin. The selection of the best location for a propeller is more difficult and involves finding a location where the propeller will not be in the way of the rotor in flight and landing conditions, of the ground during slope landings, of people during normal and emergency situations, of fire from flexible weapons, and of debris or jettisoned stores. At the same time, the complexity of the drive train, the noise in the cockpit and passenger/troop cabin, and the structural dynamic response should be held to a minimum.

The basic advantage of a propeller over a jet engine is that the propeller moves more air. For the same reason large propellers are preferred to small propellers; the larger the diameter, the higher the propulsive efficiency. However, an increase in propeller size increases its weight and the weight of the associated drive train, and adds to the difficulty of integration into the design. Thus, another trade-off decision must be made in choosing the optimum propeller diameter based upon high-speed capability versus empty weight. Other propeller parameters that must be selected are rpm and total activity factor. Propeller selection is discussed in par. 3-3.3.2.

A propeller usually responds more rapidly to pilot demands for changes in thrust than does a jet engine. However, this is a secondary advantage of propellers because rapidity of horizontal thrust control normally is not a critical consideration for compound helicopters.

3-2.2.3.4 Torque Balancing and Directional Control

Compound helicopters using rotor arrangements with natural torque balancing—such as those with tandem, coaxial, synchropter, or jet-powered rotors—achieve directional control in the same manner as do conventional helicopters with these rotor configurations.

In the case of the single-rotor, shaft-driven, compound helicopter, however, there is a possibility of using the auxiliary propulsion device instead of a tail rotor for antitorque and directional control. When this device is acting as a primary control, its reliability is a

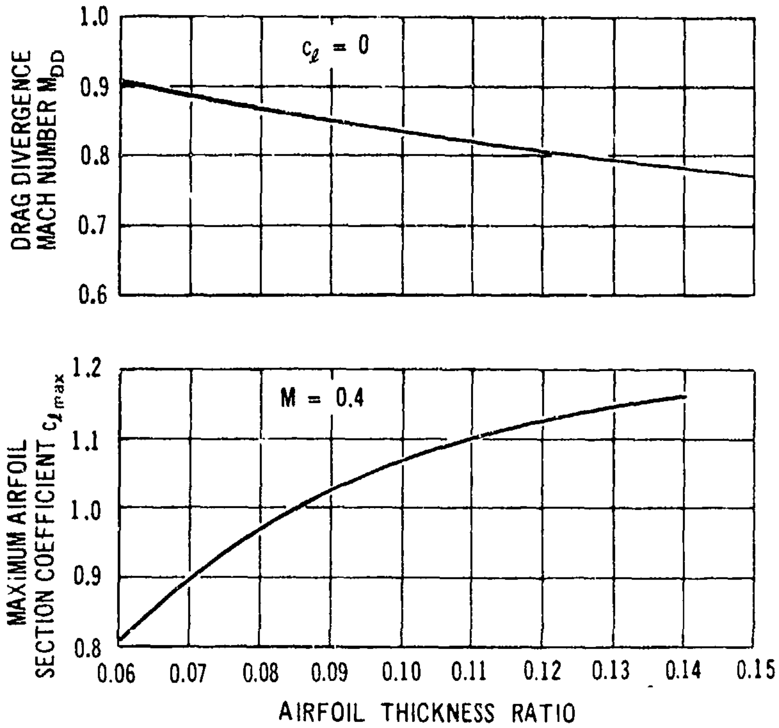


Fig. 3-34. Effect of Airfoil Thickness Ratio on Limit Airfoil Section Characteristics

major consideration which precludes the use of a separate jet engine for this purpose.

If a single propeller is used, the possibility of employing it either as a swiveling tail rotor or in conjunction with flow-turning vanes should be considered; both of these systems have been flown on prototype aircraft. When the two functions are combined, the swiveling tail rotor is lighter than the combination of a separate propeller and tail rotor—but at the cost of mechanical complexity in the swiveling mechanism and in the control system, which must make the transition from rudder pedal control in hover to a propeller pitch control in forward flight. (In high-speed flight, an airplane rudder can be used for directional control.) The pusher propeller with flow-turning vanes also combines the two functions but involves the same control complexity. In addition, this device becomes ineffective at some rearward speed that may be below that required to hover over a spot in a tail wind.

Another possibility for using the auxiliary propulsion device for antitorque and directional control exists if two wing-mounted propellers with differential thrust are used. This scheme however, while feasible, can require substantially more power in hover than the tail rotor it eliminates. The power would be essentially the same if the product of the total disk area and the distance between the propellers was the same as the product of the disk area and the moment arm of the tail rotor. For most practicable configurations, it will be found that this product for wing-mounted propellers is one-third to one-quarter that for a tail rotor and, consequently, the power required for antitorque is three to four times that used by a tail rotor. However, for compound helicopters having a large power plant because of the high-speed requirement, but limited to a low disk loading by downwash considerations, the high power required by this antitorque scheme may not produce a significant penalty.

If a conventional tail rotor is to be used, the tradeoffs are the same as for a conventional helicopter; i.e., the larger the tail rotor, the less power it requires but the more it weighs and the more awkward the design becomes. Because the tail rotor experiences the same forward speed as the main rotor in forward flight, it is logical to use the same tip speed unless the tail rotor blade tip section has a different critical Mach number than that of the main rotor. The critical thrust condition for the tail rotor will come either in balancing main rotor torque during a full-power vertical climb at sea level or in hover under a high-altitude, hot-day condition. Both situations should be analyzed and the one requiring the highest solidity should be preferred.

3-2.2.4 Compound Helicopter Performance

3-2.2.4.1 Wing/Rotor Lift Sharing

It is useful, in developing methods for calculating compound helicopter performance, to think of the aircraft as a biplane having an upper wing with an adjustable flap corresponding to the rotor and its collective pitch control. If the upper wing of a biplane has a larger span than the bottom wing, it pays to carry most of the lift on the upper wing to take advantage of its lower span loading and, thus, its lower induced drag. The same idea applies to a compound helicopter if the rotor diameter is greater than the wing span. This is illustrated by examining the following basic equation for the drag D of the compound:

$$D = D_p + D_{o_w} + D_{o_R} + \frac{L_R^2}{2\rho\pi R^2 V^2} + \frac{L_w^2}{2\rho\pi \left(\frac{b}{2}\right)^2 V^2}, \text{ lb} \qquad (3\text{-}72)$$

where

b = wing span, ft

Subscripts

w = wing

R = rotor

If it is assumed that the first three terms are independent of lift, then the variation of drag with lift is dependent only upon the last two terms. The drag then is minimum when

$$L_R = \frac{W_g}{1 + (b/2R)^2}, \text{ lb} \qquad (3\text{-}73)$$

For the highest performance, the rotor should be operated up to the above stated value of lift, or to the limit as defined by excessive oscillating loads. The oscillatory load trend corresponds roughly to the product of collective pitch and the cube of the advance ratio. The point at which these loads become limiting is a function of the structural and dynamic characteristics of the rotor and fuselage and may not be known fully until the aircraft flies.

3-2.2.4.2 Calculating Procedure for Power Required

The calculation of the power required for a compound helicopter in level flight is more complicated than that for a conventional helicopter but still is relatively straightforward. The following procedure can be performed by hand calculation or programmed on a computer:

1. Obtain by wind tunnel testing, or by estimation, plots of effective lift and drag areas and pitching moment volume (L/q ft^2, D/q ft^2, and M/q ft^3, respectively) versus the angle of attack of the fuselage reference line α_{fus} for the complete aircraft without rotors.

2. For the selected gross weight and forward speed, assume a series of values for fuselage angle of attack. Based upon small-angle assumptions, which are valid for compound helicopters in level flight, the required rotor thrust, or lift L_R, is:

$$L_R = W_g - \frac{L_w}{q} q, \text{ lb} \qquad (3\text{-}74)$$

3. Calculate the rotor tip path plane angle of attack α_R as

$$\alpha_R = \alpha_{fus} + 57.3\frac{v}{V} + a_{1_s} + i_s, \text{ deg} \qquad (3\text{-}75)$$

where

a_{1_s} = longitudinal tilt of rotor tip path plane relative to rotor shaft, deg

i_s = angle of rotor shaft relative to a reference perpendicular to the fuselage reference line, deg

and, for $T/(2\rho A) \ll V^2/2$ Eq. 3-35 results in

$$v = \frac{L_R}{2\rho A V}, \text{ fps} \qquad (3\text{-}76)$$

(The rotor-induced downwash at the wing is assumed to be equal to the momentum value of induced velocity at the rotor disk.)

and

$$a_{1_s} = \frac{-(M/q)q - W_g(\Delta s_{sh})}{\partial M_R/\partial a_{1_s}}, \text{ deg} \qquad (3\text{-}77)$$

where

Δs_{sh} = distance between helicopter CG and rotor shaft, parallel to fuselage reference line, ft

M_R = rotor pitching moment, lb-ft

4. Use rotor performance charts for the correct advance ratio, and, knowing rotor thrust (lift) and rotor angle of attack, find collective pitch and plot it as a function of fuselage angle of attack.

5. Select even increments of collective pitch and tabulate the corresponding values of rotor thrust, rotor angle of attack, and fuselage angle of attack.

6. Use rotor charts to find rotor shaft power and H force for the selected values of collective pitch and rotor thrust.

7. Compute total power associated with the rotor as the sum of main and tail rotor requirements and transmission and drive system losses.

8. Compute the required propulsion thrust T_p as

$$T_p = \frac{D}{q} q + H + L_R \frac{\alpha_R}{57.3} \text{ , lb} \quad (3\text{-}78)$$

where

H = rotor H-force perpendicular to control axis, lb

9. For propeller-powered compound helicopters, the propeller power hp_p is

$$hp_P = \frac{T_P V}{550\eta_P} \quad (3\text{-}79)$$

where

η_p = propeller efficiency factor, dimensionless

Subscript

P = propeller

The upper plot of Fig. 3-35 shows the results of using the foregoing procedure to compute the power required for a typical compound helicopter. The advantage of operating at high collective pitch angles is evident. The bottom plot of the figure shows the product of collective pitch and the cube of the advance ratio (this parameter is roughly proportional to the oscillatory loads) and indicates the disadvantage of operating at high collective pitch. Fig. 3-36 shows component power requirements for operation at collective pitch values of 4 and 10 deg. The primary effect of operating at a higher collective pitch angle is a more significant decrease in the power associated with the wing as compared to the increase in power associated with the rotor.

3-2.2.4.3 Autorotation of a Compound Helicopter

Autorotation of the rotor of a compound helicopter requires the same conditions as for other helicopters; i.e., the lift vectors on the blades must be tilted far enough forward to overcome the drag of the blades. A fully unloaded rotor cannot autorotate because it has no lift vectors to tilt forward. For this reason if the wing is large enough to support the entire weight of the helicopter during entries into autorotation at high speeds, it is difficult to load the rotor enough to enable it to autorotate. One solution is for the pilot to perform steady turns at a load factor high enough so that both the wing and the rotor are required to produce lift. Another solution is to reduce the wing lift mechanically with spoilers, with an incidence reduction, or with an up-deflected flap. For compounds with propellers, it is not necessary to autorotate the rotor initially if the propeller pitch can be reduced sufficiently to operate the propeller as a windmill; this procedure allows power to be extracted from the airstream at high speeds so as to keep the entire power train operating until the speed is low enough to allow the rotor to be used in normal autorotation.

3-2.3 WIND TUNNEL TESTING

3-2.3.1 Objectives

The objectives of wind tunnel testing are one or more of the following:

1. To support a specific design project
2. To perform research into fundamental phenomena
3. To investigate advanced concepts.

This paragraph deals with the preliminary design aspects of wind tunnel testing that are included in the first of these objectives.

3-2.3.2 Support of a Project

In the discussion of wind tunnel testing that follows, it should be remembered that test planning should consider the reliability of the resulting data. Dimensional similarity, particularly in terms of Reynolds number and Mach number, is desired, but the relationship between model size and tunnel size also must be considered because of the significant influence of wind tunnel wall effects upon helicopter test data.

3-2.3.2.1 Airframe Drag and Stability Studies

Models with rotor hubs, but without rotor blades, are tested during preliminary design to produce early estimates of drag and stability and to provide an opportunity to improve these parameters before the design is frozen. Results from such tests are extremely sensitive to Reynolds number and it is unlikely that absolute values of drag will be obtained by testing small-scale models. Drag studies conducted on small-scale models usually are approached on the basis of incremental variations due to configuration changes.

Two sources of drag, discussed for illustrative purposes, are the rotor hub and the aft end of a typical cargo compartment. The latter often is designed with

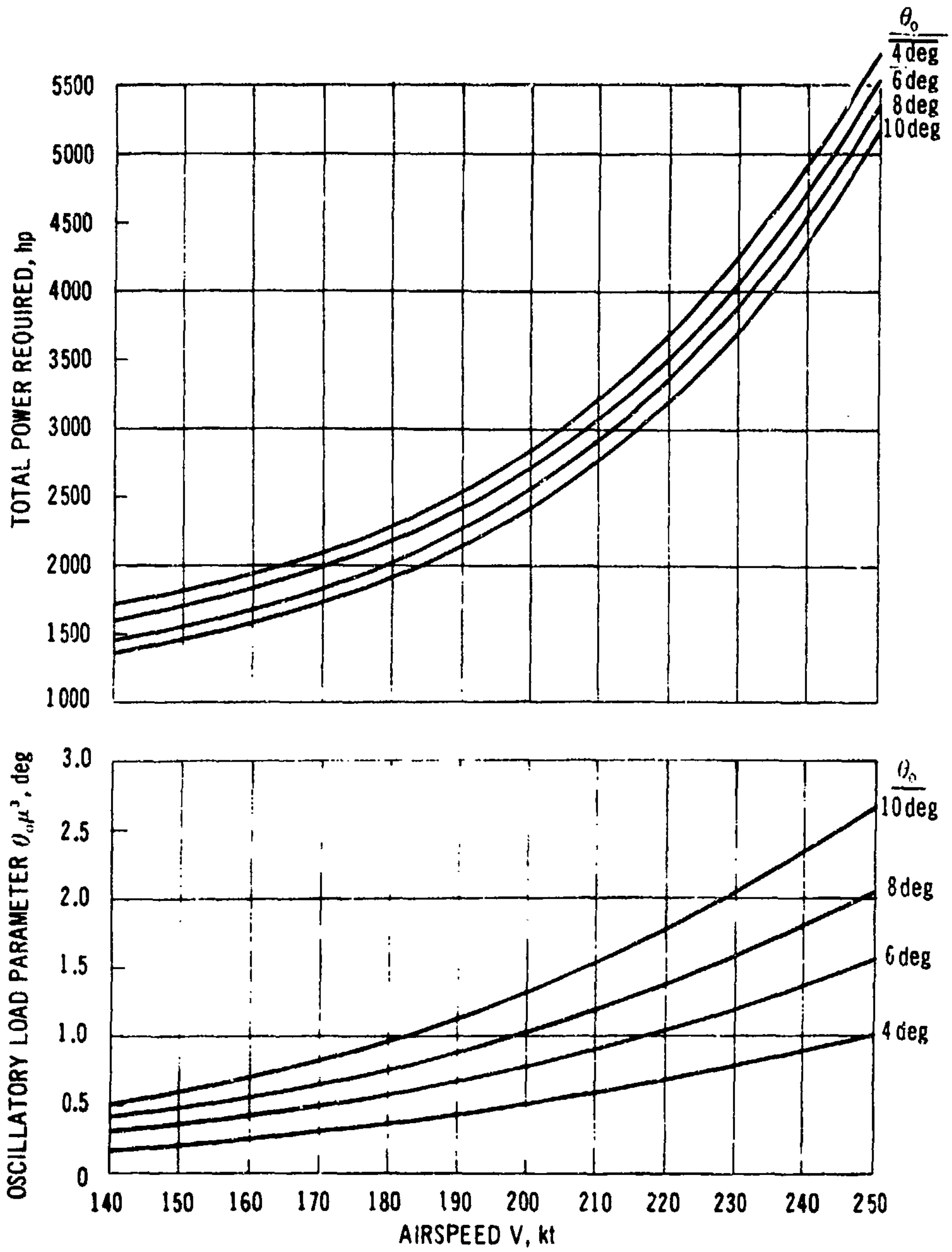


Fig. 3-35. Effect of Collective Pitch on Power and Oscillatory Loads

a very large closure angle because of the tactical advantage of rear ramp loading. The rotor hub is a bluff body that, although difficult to fair as a unit, can benefit from streamlining of individual components. During wind tunnel testing the hub should be rotated to obtain proper values of hub/pylon drag.

The hub not only produces drag of its own, but, in many cases, induces flow separation on the portion of the fuselage behind it. The degree of separation often can be reduced through the use of some form of a vortex generator or boundary layer control specifically tailored through wind tunnel tests. In a similar manner, the problem of separation behind a blunt cargo compartment can be attacked through changes in fuselage lines or the use of strakes or vortex generators. Wind tunnel testing of high-performance helicopters inevitably involves evaluating the reduction in drag resulting from a retractable versus fixed landing gear.

3-2.3.2.2 Powered Models

The second phase in a complete series of wind tunnel tests is the addition of a powered rotor to the airframe model. The resulting model is useful for determining rotor-airframe interference effects and stability derivatives but has not proven to be particularly useful in determining the performance of the rotor itself because of significant aerodynamic and dynamic scale effects. In extreme flight conditions involving significant blade stall or compressibility effects, the Reynolds numbers and the oscillating aeroelastic twist—which is a function of blade structural and centrifugal stiffness in bending and in torsion—play very important roles; matching these characteristics with small-scale rotors so far has not been practicable. In the case of interference effects, however, useful results can be obtained with a rotor that is similar geometrically, but not necessarily dynamically, to the full-scale rotor. The interference effects may be significant especially in producing pitching moments during the transition from hover to forward flight and in producing rotor-fuselage interference drag in all flight conditions.

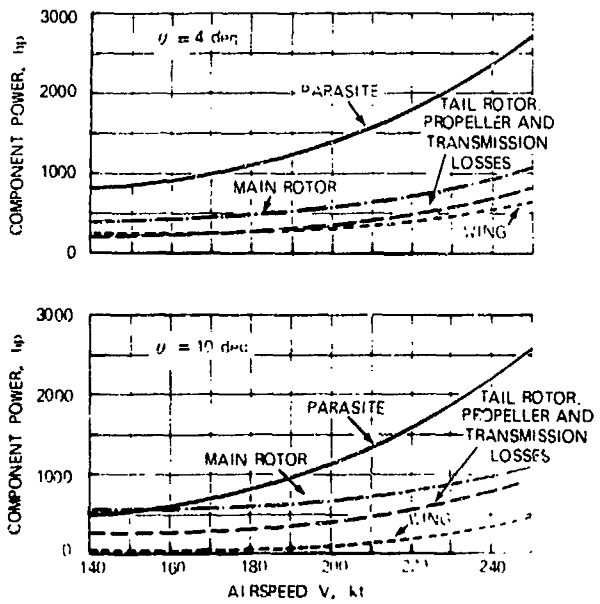


Fig. 3-36. Component Power at Two Values of Collective Pitch

3-2.3.2.3 Dynamic Models

An investigation of oscillatory loads and of the dynamics of the rotor and its control system can be made with a rotor model that is scaled both geometrically and dynamically in terms of the natural frequencies of its flapping, inplane, and torsional blade modes and of the control modes associated with control system flexibility. Unfortunately, the knowledge of these frequencies comes near the end of the design effort, and it is likely that the actual rotor will be flying before the model can be designed, built, and tested. Nevertheless, the availability of a dynamic wind tunnel model during the flight test portion of the project can be an important factor in rapid and safe investigation of critical phenomena as they arise (Ref. 33).

3-2.3.2.4 Special Models

An example of a special wind tunnel test that might be required to support other phases of the design effort is the measurement of engine inlet pressure recovery and flow distribution at the face of the compressor. For this type of test, it is necessary to install a pump in a model to induce the correct airflow into the inlet. A second type of test involves measuring pressures on the windshield and other critical components through use of pressure tubes connected to a manometer board or to a scanning pressure transducer. It may be desirable to measure bending moments in such components as wings and horizontal stabilizers to allow the designers to use measured values, rather than theoretical (which normally tend to be conservative). The resulting design will thus have minimum structural weight. Also, it may be necessary to conduct external store jettison tests with the wind tunnel model to establish the flight envelope in which dropped stores fall clear of the aircraft, and to determine whether a positive ejection mechanism—such as a spring or an explosive device—is required. Wind tunnel jettison tests require models of stores that must have the correct aerodynamic shape, weight, and CG location. The model must be such that the ratio of aerodynamic forces to inertia forces is correct for the model scale and the wind tunnel speed being used to simulate flight conditions. Results are recorded with movie cameras taking front and side views. The model need not have rotors, but it helps in interpreting film results if the tip path planes of the main rotor(s), tail rotor, and propeller are defined with wire hoops.

3-2.3.2.5 Two-dimensional Airfoil Tests

For a rotor design that pioneers in the use of a new airfoil section, it is desirable to obtain the airfoil characteristics for use in the rotor analysis, in a two-dimensional tunnel. These characteristics should include lift, drag, and pitching moments throughout the angle of attack, Mach number, and Reynolds number ranges in which the blade elements operate. Ref. 34 contains a complete description of a typical steady-state, two-dimensional wind tunnel test. A variation of this type of test involves testing the airfoil while it is subjected to oscillating plunge and pitch motions at various mean angle of attack settings. The results then can be used to evaluate the nonsteady characteristics of the airfoil as represented by the hysteresis loops in lift, drag, and pitching moment. Another type of two-dimensional testing used to evaluate three-dimensional effects is the testing of the airfoil at the various values of sweep that can occur on the blade and that have been found to affect the maximum lift coefficient significantly. Three-dimensional effects can be obtained by testing a blade tip at various sweep angles. All of the steady, nonsteady, and three-dimensional airfoil characteristics can be used in the sophisticated rotor analysis programs that account for blade element sweep, unsteady tip vortex induced flows, and blade flapping and torsional flexibility.

3-2.3.2.6 Testing of Actual Hardware

Wind tunnel testing of component hardware often is required to support the design and analysis effort further. For example, the drag of antennas and external stores and the aerodynamic loads opposing the motion of rotating weapon turrets and gun-sights are difficult to predict analytically or with small-scale models, but are evaluated easily in a wind tunnel.

The testing of a complete aircraft in a full-scale wind tunnel generally is not included in the early project planning. During the flight test phase, however, it may become desirable to investigate one or more specific problems in the areas of rotor dynamics, stability, or performance, and the designer may determine that these problems can be investigated with either less danger or more precision in the wind tunnel than in flight (Ref. 35). Full-scale tests currently are limited by tunnel capability to speeds of less than 190 kt and to rotor diameters of less than 65 ft. The aircraft should be equipped with flight test instrumentation for measuring rotor loads, a real-time monitoring system, and a quick-acting remote control system.

Mounting the helicopter on the wind tunnel balance system changes the dynamic environment from that of free flight in two significant ways. First, the balance system has its own set of modes of oscillation that, if excited at their natural frequencies by oscillating rotor loads, can confuse the dynamic data being measured at the rotor and, in some cases, can lead to structural failures. To avoid such problems, the natural frequencies of the balance system should be predetermined by shake tests and rotor speeds then selected so as to avoid them; or the balance natural frequencies should be changed by modifications to the mass, damping, or structural stiffness. The second significant change in the dynamic environment arises from the ability of the wind tunnel supports to provide essentially immediate reactions to aircraft loads and moments. This is in contrast to the aircraft in free flight, which produces reactions proportional only to aircraft inertia. The consequence of this difference is that rotor forces—which in free flight would cause the aircraft to move and to relieve the forces—are not relieved in the wind tunnel but may build up to catastrophic magnitudes. For this reason it is important to be able to monitor critical rotor loads and to relieve them quickly with the control system.

3-2.3.3 Model Design Considerations

3-2.3.3.1 Models Without Rotors

Models without rotors, which are used for studies of drag and stability qualities, should be designed similarly to comparable fixed-wing models. The model should be as large as possible and the test tunnel speed as high as possible to enable testing at high Reynolds numbers. This is important especially when determining the drag of bluff body components such as rotor hubs, landing gear, or weapons. Consideration must be given to compressibility effects, which frequently can become significant due to high local Mach numbers and can result in considerable error in the translation of small-scale results to full scale.

The use of transition strips to establish the turbulent boundary layer at the forward part of each component is recommended. Because the determination of hub drag is an important objective of this type of testing, the hub should be built as accurately as possible. Blade stubs with a span equal to the chord should be installed to duplicate the hub-blade junction. If practicable, the hub should be rotated with a motor at a speed corresponding to the full-scale advance ratio at the test tunnel speed. Major components should be made removable and the model should be designed for testing both right side up and upside down to evaluate both hub drag and landing gear drag away from the influence of the balance supports.

3-2.3.3.2 Models With Rotors

3-2.3.3.2.1 Rotor Design

In this type of model, the rotor diameter should be as large as possible to allow testing as closely as possible to full-scale Reynolds numbers. If the primary focus of interest is the high-speed regime, the rotor diameter can be up to 75% of the tunnel test section width. If low-speed problems are to be investigated, the rotor diameter should not exceed 25% of the test section height so as to minimize recirculation of the rotor wake from the tunnel floor. The model rotor tip speed should be the same as that proposed for the full-scale helicopter in order to preserve true Mach numbers and induced velocity effects. When it is not possible to operate the tunnel at such a speed, the design rotor speed should be reduced in order to maintain the true advance ratio, because this is a primary rotor parameter while Mach number and induced effects are secondary parameters.

The scaling of mass and structural parameters for dynamic models should attempt to match natural frequencies and ratios of aerodynamic-to-inertia parameters so as to allow investigation of flutter, mechanical vibration and stability, performance, and flying qualities (Ref. 33).

3-2.3.3.2.2 Rotor Control Systems

Depending upon the objectives of the model test, the rotor control system may be either nonexistent or nearly as complete as that of the actual aircraft. The simplest models have flapping or teetering rotors, no cyclic pitch, and collective pitch that is adjustable only during a model change. Data points taken with this type of model always will have the tip path plane tilted with respect to the shaft, except for specific combinations of shaft angle of attack and tip speed ratio for each collective pitch setting. For some types of tests these out-of-trim data are valid and useful, but for other tests only trim conditions are of interest. The incorporation of a remote control system for both cyclic and collective pitch allows each datum point to represent a trim condition at the desired rotor thrust and hub moment; in other words, the testing time can be reduced drastically at the cost of model complexity. For models without cyclic pitch, the flapping of the tip path plane must be monitored closely to prevent tail boom strikes at high collective pitch and tip speed ratios.

3-2.3.3.2.3 Rotor Power Supply

For some specialized tests, the rotor can be tested in autorotation and therefore no power supply is required. The more usual test, however, requires that the rotor be powered. The model power plant will be scaled down from the actual power plant by a scale factor dependent on the specific similarity requirements of the test. For pressure-jet rotors, compressed air is used from the shop supply or from a special compressor, depending upon the quantity of air required. Successful shaft-driven model tests have been conducted using electrical and hydraulic power. Many wind tunnels now use pneumatic motors for scale models.

The electrical system of powering models usually consists of a variable-frequency motor-generator, set outside the test section, and a synchronous motor in the model. Most suitable synchronous motors are water-cooled so as to obtain high power in a small envelope. The wires and cooling tubes to the motor, along with any instrumentation wiring, should be attached to the model in such a way as to introduce minimum forces and moments due to air loads or changes in angles of attack. In addition, all wiring should be shielded to avoid interference. Fig. 3-37 shows how this may be done.

A hydraulic system that has been used satisfactorily consists of a constant speed-constant volume pump with a controllable bypass valve outside the test section and a hydraulic motor in the model. The bypass valve is used to control the amount of fluid going to the model and, hence, its speed. To prevent hydraulic pressure from producing excessive balance tares, the tubing running from the pump to the model support should be straight with solid elbow fittings as shown in Fig. 3-38. It still will be necessary, however, to evaluate the balance—reading tares as a function of hydraulic pressure for making final corrections to the test results.

3-2.3.3.2.4 Instrumentation

The sophistication of model instrumentation will be dictated by the objectives of the test program. Simple tests might use only the wind tunnel balance system, whereas more complicated tests might require instrumentation on the order of that used for flight tests, e.g., strain gages and various types of potentiometers to measure rotor and control system loads and positions. The recording of measurements in the rotating system requires either slip rings or a radio transmitter mounted on the hub, both systems having been used satisfactorily.

For many types of tests, it is necessary to differentiate between the thrust of the rotor and the lift on the rest of the model. Rotor thrust can be obtained by measuring:

1. The tension in the rotor shaft
2. The average bending moment in the rotor hub or blade root
3. The average blade coning.

All of the methods require measurements in the rotating system, and the last two methods rely upon a calibration that may be difficult to perform in a wind tunnel. (It should be noted that measuring rotor thrust always is a difficult task whether in a wind tunnel or in flight.) As an alternative approach, the possibility of attaching the entire rotor system to the rest of the model with a separate strain gage balance system should be considered. This attaching method has the advantage of being outside the rotating system (thus not requiring slip rings), and it can be used to measure not only thrust but shaft torque, rotor pitching and rolling moments, and longitudinal and lateral rotor forces as well. Such a balance system should be designed so as to introduce no new modes of vibration with natural frequencies in the range of the test rotor speeds.

For model rotors in which dynamic phenomena are primary considerations, it is important that critical measurements be monitored continuously. Some tests require not only several critical displays but several observers to monitor them. No compromise should be made concerning the number of critical loads to be

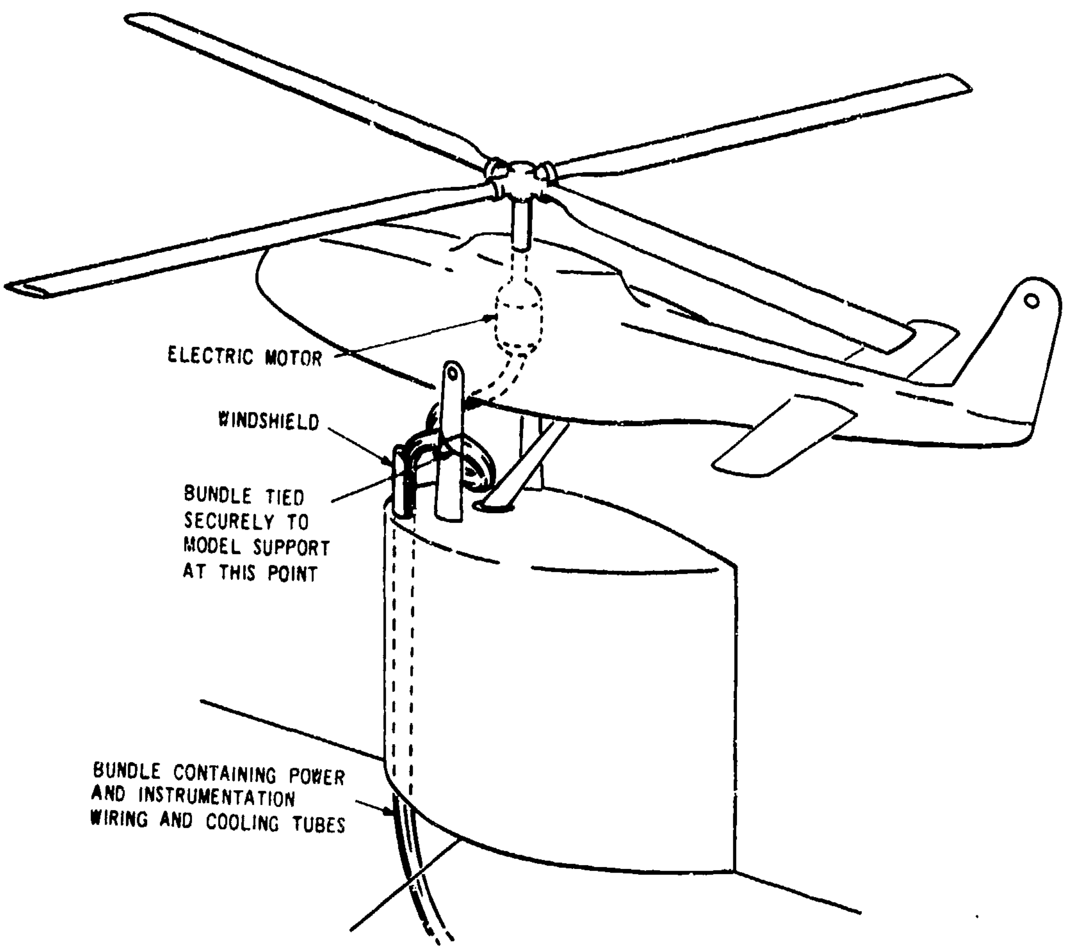


Fig. 3-37. Installation of Electrically Powered Model

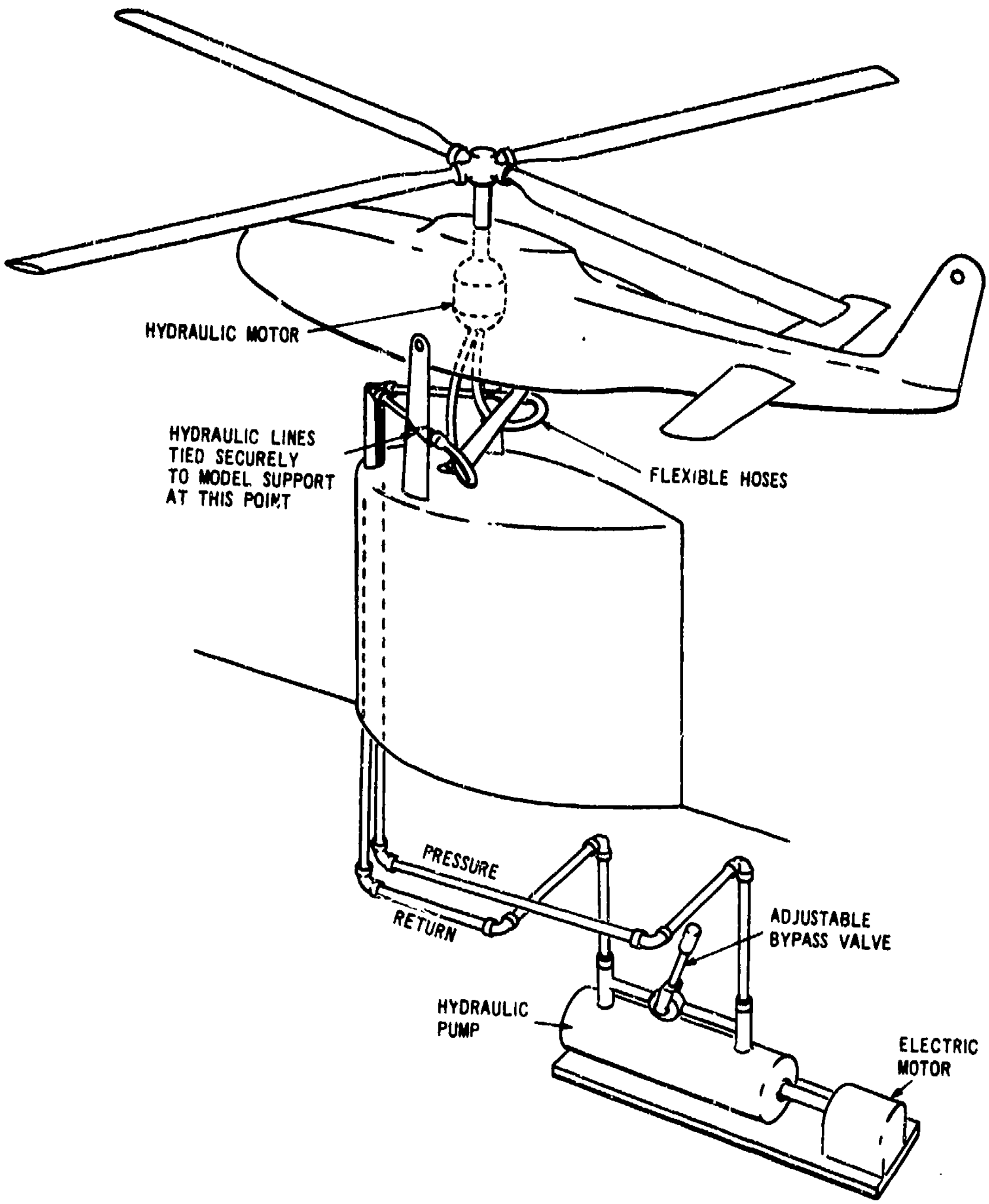


Fig. 3-38. Installation of Hydraulically Powered Model

displayed on an oscilloscope, with the limits clearly marked. A maximum of information, clearly displayed in a recognized fashion, and alert personnel are required for satisfactory testing. Any dynamic measurements that are important to decisions that must be made during the test should be recorded on an instant-developing oscillograph or on a Brush recorder. All data should be recorded on permanent oscillographs or tape recorders for subsequent data reduction. It is highly desirable in critical tests to have continuous motion picture coverage. A means must be provided for identifying the various types of data being recorded with tunnel run number and, if possible, with a standard time signal.

Rotor speed can be determined in several ways. In rotors driven by a variable-frequency electrical system, the frequency meter on the motor-generator set can be calibrated to read rotor speed directly. In more elaborate models, a tachometer generator can be belt- or gear-driven by the rotor shaft with a conventional aircraft tachometer at the operator's station. For very precise measurement, an Event-Per-Unit-Time Counter can be used to count the electrical impulses generated by one or more magnets fastened to the rotor shaft as they pass an induction coil on the nonrotating portion of the model. The same electrical impulse can be recorded as an indication of rotor azimuth position on any system that is being used to record loads or dynamic data.

3-2.3.4 Test Procedures

The test procedures for rotorless helicopter models are the same as those for models of fixed-wing aircraft: angle of attack runs at zero yaw and yaw runs at representative angles of attack—all at the highest practicable tunnel speed that results in the best available correlation of Reynolds number and Mach number to full-scale conditions. Configuration changes should include stabilizer incidence changes and removal of of individual components. In some cases, flow visualization using tufts or oil sublimation will be useful in defining sources of drag.

A model with a rotor introduces four additional test parameters: advance ratio, collective pitch, longitudinal cyclic pitch, and lateral cyclic pitch. For data points representing trim conditions, the collective pitch should be such as to produce the desired rotor thrust and the cyclic pitch should be such as to produce the desired pitching and rolling moments about the CG. Once the trim point has been established, stability derivatives can be determined with respect to angle of attack, forward speed, rotor speed, cyclic pitch, collective pitch, and side slip angle.

Tests of dynamic or full-scale models often involve the determination of stability boundaries for certain modes of oscillation. The procedure for this type of investigation is to establish the trim conditions and then to measure the time the model requires to damp to half-amplitude following a control system doublet. The boundary then is approached by increasing rotor thrust or tunnel speed by a small increment and repeating the doublet. A running plot of the inverse of the time to damp to half-amplitude is maintained. As soon as the plot can be extrapolated to zero, the stability boundary is assumed to be established and the test is concluded without actually subjecting the model to the instability. Determination of stability boundaries is one of the most dangerous aspects of wind tunnel testing and should be attempted only after a maximum effort has been made by analysis to predict these instabilities.

3-2.3.5 Data Reduction

3-2.3.5.1 Wind Tunnel Wall Corrections

Wind tunnel wall corrections for rotors are a function of the wake skew angle and the dimensions of the tunnel test section with respect to the rotor diameter. For cases in which the rotor wake skew angle is near 90 deg (the wake is going straight back from the rotor), the corrections are the same as for a circular wing of the same span. For cases in which the skew angle is near 0 deg (as in hover), the recirculation inside the test section probably will produce unstable test conditions that will make quantitative measurements useless. For skew angles between the two extremes, wall correction methods are given in Ref. 36 for a variety of test section dimensions and shapes.

3-2.3.5.2 Presentation of Results

The preferred form for the presentation of results of wind tunnel tests of helicopter models is somewhat different than for fixed-wing models. The wing is the principal source of aerodynamic forces upon an airplane so it is customary to nondimensionalize measured forces and moments based upon the wing dimensions. This procedure allows a quick comparison with other airplanes and the form is the same as that used in the analysis of performance and flying qualities. On a helicopter, however, the rotor is the principal source of forces and moments, and these factors are related closely to the forces and moments on the rest of the aircraft. For this reason rotor characteristics are nondimensionalized based upon the rotor dimensions, but the forces and moments of the remainder of the aircraft are presented best in the form of D/q and M/q as a function of angle of attack and sideslip angle where these quantities refer to full-scale forces and moments. Note that D/q is the equivalent flat plate area, which is the normal unit of drag used in helicopter analysis. Fig. 3-39 shows a typical set of test data for a model without rotor.

Rotor data—either from a rotor-alone test or from a complete model-with-rotor test—should be presented in the same format as is used in analysis. The independent variables that specify the test conditions are tunnel speed V, tip speed ΩR, angle of attack of the shaft α_s, collective pitch, longitudinal cyclic pitch, and lat-

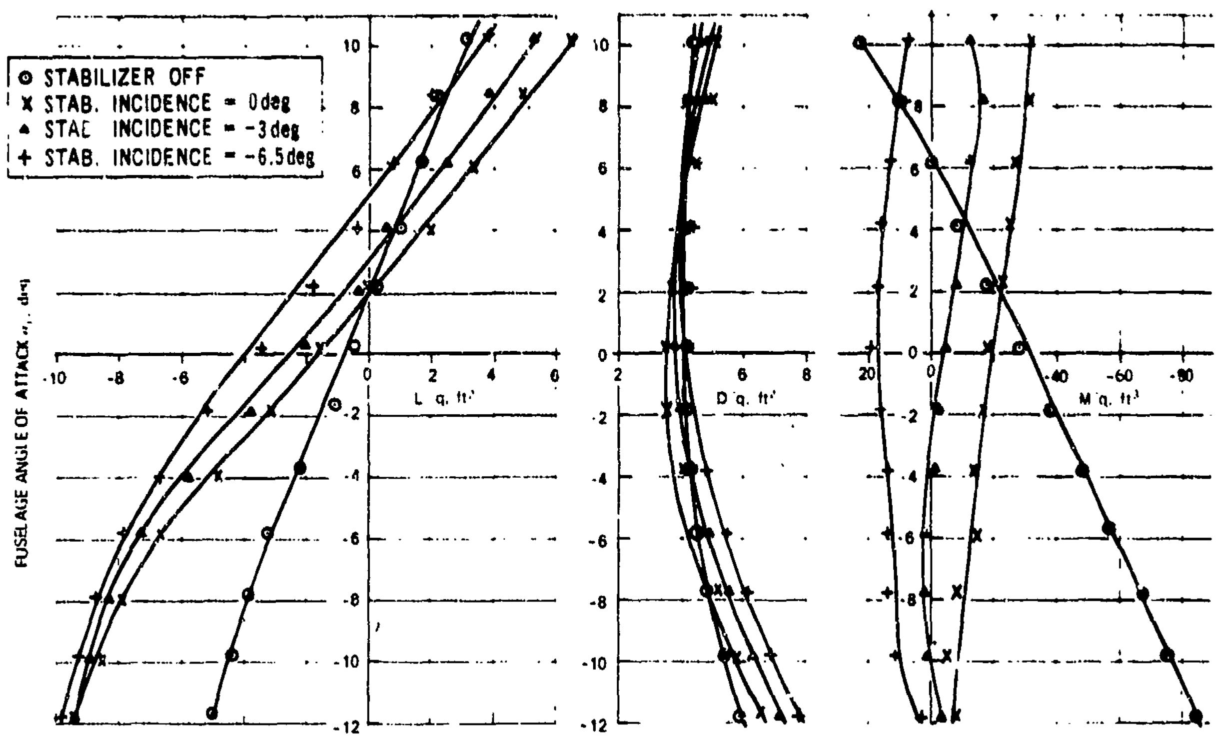


Fig. 3-39. Typical Results for Model Without Rotor

eral cyclic pitch. The corresponding dependent variables of most interest are C_{LR}/σ, C_{XR}/σ, C_P/σ, C_{P_0}/σ, and α_C and their variations. A typical plot of wind tunnel data for a rotor is shown in Fig. 3-40 as taken from Ref. 37.

3-2.3.6 Other Related Tests

3-2.3.6.1 Rotor Whirl Tower

Whirl towers are used:

1. To measure the hover performance of existing rotors
2. To verify the structural integrity and dynamic stability of new or modified rotors
3. To track blades to a tolerance
4. To do specialized research into such factors as airfoil characteristics, wake structure, noise, and ground effect.

In order to allow testing in out-of-ground-effect conditions, a conventionally mounted rotor should be at least one diameter above the ground. If the rotor can be mounted upside down, a tower height of 40% of the diameter will be sufficient to make the ground effect negligible. Upside-down mounting is not always practicable with actual helicopter rotors, but it usually is feasible for model rotors designed from the beginning for whirl tower use. Depending upon the primary purpose of the whirl tower, it may be equipped to measure rotor thrust and power, blade loads, control system loads and positions, blade track, and wake characteristics. In some stability investigations, it is useful to gimbal-mount the rotor support on a frame that can simulate the actual pitch and roll inertias of the aircraft. Such tests should be conducted in near-zero wind conditions (under 3 kt), especially if performance characteristics are desired.

3-2.3.6.2 Download Test Facilities

Because every pound of fuselage or wing download decreases the payload capability by a pound in hover, it is important to be able to determine this penalty accurately during the preliminary design phase. Analytical procedures using calculated wake characteristics and estimated drag coefficients for the aircraft components in the wake are useful as first estimates, but these should be verified by using the model designed for the wind tunnel and a whirl tower using a geometrically scaled rotor (Fig. 3-41). For structural simplicity, and to avoid ground effect, the rotor can be mounted with its axis horizontal and the model mounted beside it with a separate support and balance system. To insure that the wake characteristics are the same as at full scale, the rotor model should be capable of developing the same disk loading and tip speed as the actual rotor. The effect of the ground upon download can be determined by mounting a ground plane next to the model.

3-3 PROPULSION

3-3.1 PROPULSION SYSTEM ANALYSIS

3-3.1.1 Propulsion Techniques

Two types of helicopter gas turbine engine propulsion techniques will be discussed in this paragraph:

1. Shaft drive system. This system uses a turboshaft engine(s), either a free power turbine or a coupled version, to supply the shaft horsepower necessary for driving the helicopter rotor(s)/propeller(s) through a shafting and gearing system. The coupled version normally is not used due to the superior performance characteristics of the free turbine in a helicopter environment, such as better fuel consumption at part loads and better power control under fluctuating load conditions. In the free turbine, gas generator speed is independent of helicopter rotor speed.

2. Reaction drive system. This system uses a turbofan or turbojet engine to provide reaction force. One method uses a series of ducts to carry the exhaust gases to the helicopter rotor, where they are expelled through nozzles in the trailing edge and/or tips to drive the rotor. Another method uses engines mounted at the ends of the helicopter rotor blades to drive the rotor.

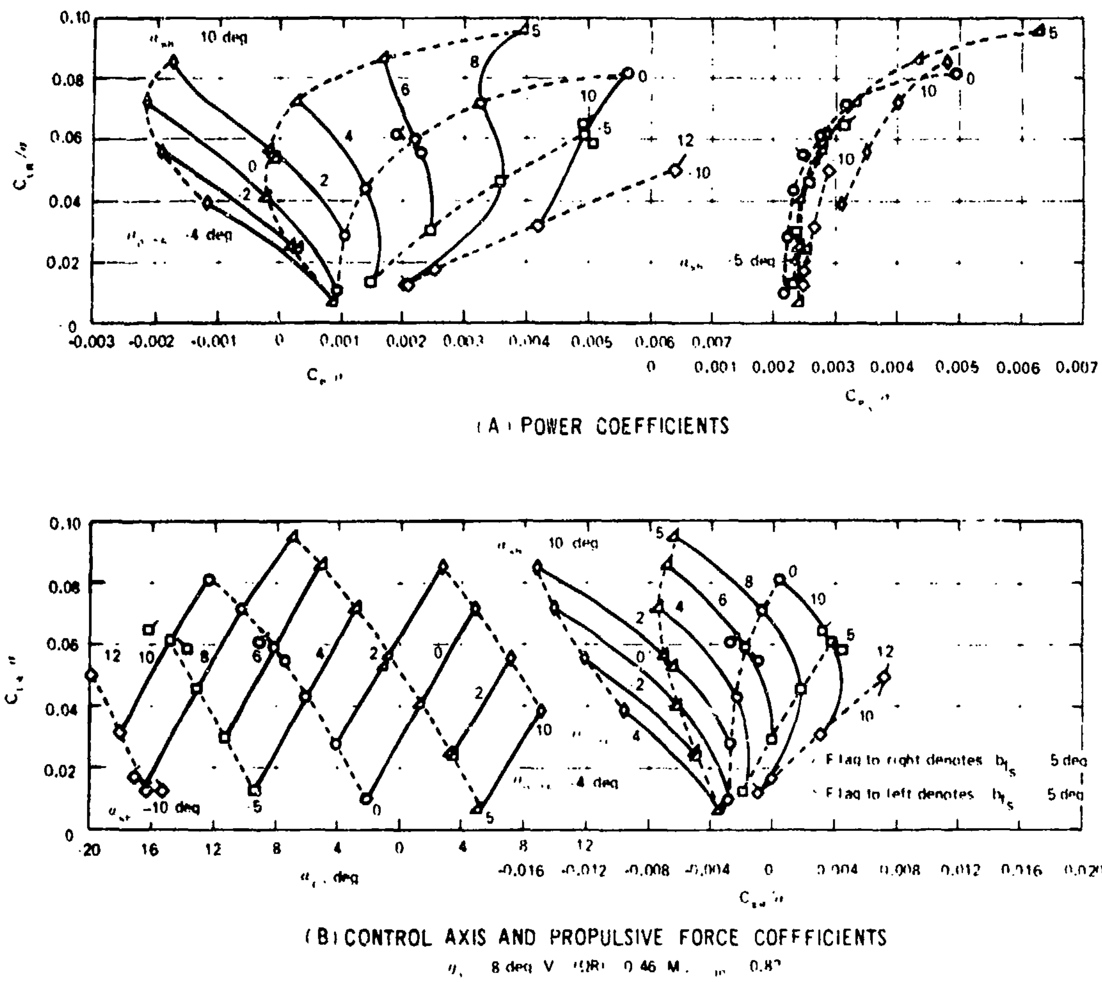


Fig. 3-40. Typical Rotor Results from a Wind Tunnel Test

Primary emphasis in this paragraph will be on the free power turbine turboshaft engine. It is believed that this emphasis properly reflects the projected helicopter design trends for the foreseeable future. Furthermore, the thermodynamic concepts enunciated generally will be applicable to the reaction systems as well.

3-3.1.2 Gas Turbine Engine Characteristics

The gas turbine engine cycle variables of primary interest to the helicopter designer are specific fuel consumption *SFC* and specific power SHP/W_a, where *SHP* is the shaft horsepower and W_a the weight flow of air. *SFC* exerts a primary influence upon helicopter range and endurance. The major effects of SHP/W_a are upon engine size and weight. While cycle pressure ratio, design philosophy, and engine configuration also can affect the final engine size and weight significantly, the largest single influence is airflow rate W_a. For a given power rating, this is established by the cycle value of *SHP*.

Before reviewing the cycle analysis for gas turbine engines, it may be helpful to provide an overview of gas turbine engine technology. Figs. 3-42 and 3-43 show the historical improvements in engine *SFC* and weight along with some estimates about the future. The major reasons for performance improvements have been increases in cycle pressure ratio and turbine inlet temperature.

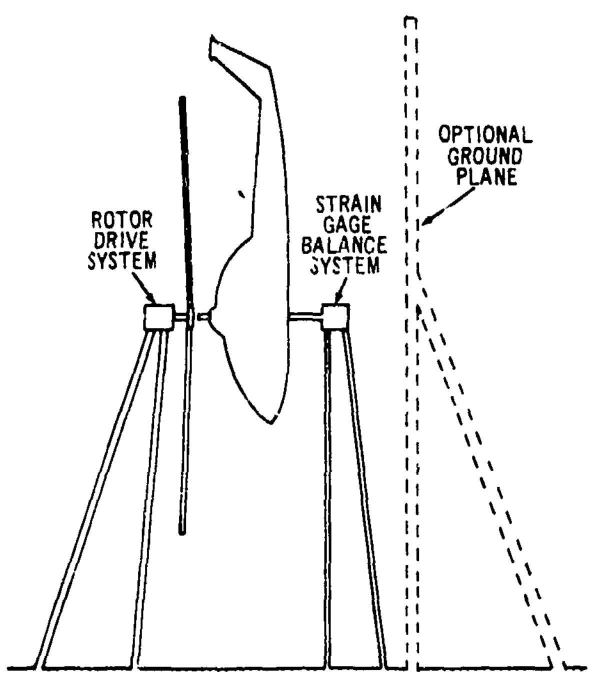


Fig. 3-41. Arrangement of Download Facility

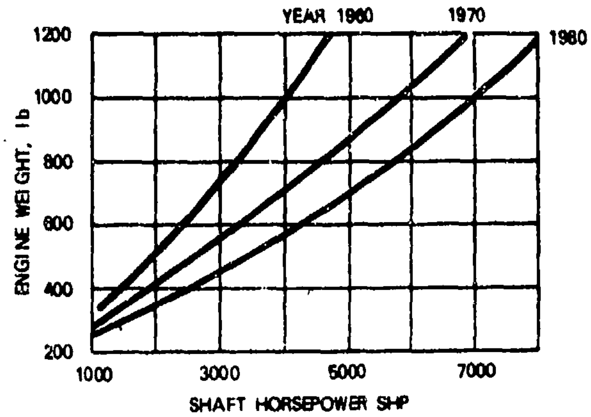


Fig. 3-42. Engine Weight vs Shaft Horsepower

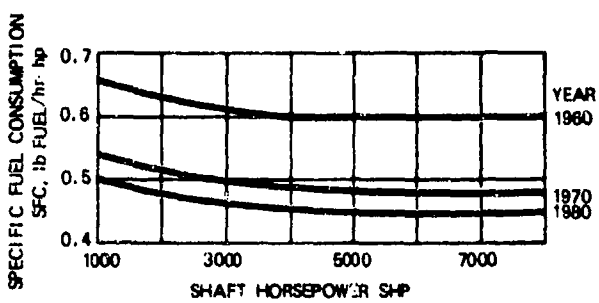


Fig. 3-43. Engine Specific Fuel Consumption vs Shaft Horsepower

3-3.1.2.1 Cycle Description

The analytical model for the gas turbine engine is the Brayton cycle shown in Fig. 3-44. A complete analysis of the cycle is included in AMCP 706-285. The discussion here is limited to a review of the critical parameters and of their effects upon the performance characteristics of the engine. This discussion will assume an air cycle; i.e., the working fluid is assumed to be air with constant values of specific heat and molecular weight throughout the cycle. While this assumption does not reflect accurately the real conditions in a gas turbine engine, it does permit a straightforward development of the fundamental engine characteristics in closed analytical form. The refinement of accounting for variable gas properties, while essential to the engine designer, does not contribute to an understanding of the turbine engine interrelations.

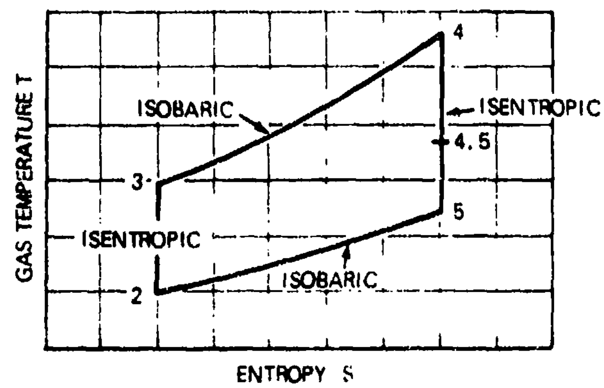


Fig. 3-44. Gas Turbine Engine Cycle (Generalized)

As shown in Fig. 3-44, the ideal turbine engine cycle is made up of two isentropic (constant entropy) and two isobaric (constant pressure) processes. In this ideal case, air is compressed without losses from State 2 to 3, heating (fuel addition) takes place at constant pressure from 3 to 4, followed by isentropic expansion through the turbine from 4 to 5. The cycle is completed, conceptually, by cooling the exhaust gases from 4 to 2. This, of course, takes place in the atmosphere, external to the engine.

For the free power turbine engine cycle, the power required to drive the compressor is provided in the expansion from 4 to 4.5. The useful or output power is represented by the expansion from 4.5 to 5. This model assumes that the gas generator turbine is located upstream of the output turbine (power turbine). While this is not a necessary constraint for free power turbine engines, it is normal. The output power is the difference between the total expansion power developed (4 to 5) and the compressor power required (3 to 2).

3-3.1.2.2 Specific Fuel Consumption (SFC)

The gas turbine engine, like any other heat engine, is constrained to less than 100% thermal efficiency by two factors. First, even if process losses (pressure losses, inefficiencies in compression/expansion, radiation heat losses, etc.) could be prevented, the second law of thermodynamics demands that only a portion of the thermal energy (fuel) added to the cycle can be converted into useful work. For the ideal case of no process losses and constant gas properties, the *SFC*, expressed as lb fuel/hp-hr, is only a function of cycle pressure ratio. If a second set of constraints relating to individual process efficiencies is considered, it is found that an optimum pressure ratio exists for each combination of efficiency and maximum cycle temperature (turbine inlet temperature). This optimum pressure ratio yields minimum *SFC*, and increases beyond optimum give corresponding increases in *SFC*. Typical results are shown in Fig. 3-45, along with a comparison with the ideal case.

3-3.1.2.3 Specific Power

The other dependent cycle variable of interest to the helicopter designer is specific power, i.e., the power produced in the cycle by each lb/sec of airflow. This cycle variable is a major index of engine weight for a given power capability because it establishes the engine envelope.

For the ideal air cycle, specific power, unlike *SFC*, is a function of both cycle pressure ratio and turbine inlet temperature T_4. From the definition of *SFC* it follows that

$$\frac{SHP}{W_a} = \frac{W_f/W_a}{SFC} \sim \frac{c_p(T_4 - T_3)}{SFC} \qquad (3\text{-}80)$$

where

SHP = shaft horsepower
W_a = weight flow of air, lb/sec
W_f = engine fuel flow, lb/hr
SFC = specific fuel consumption, lb/hp-hr
c_p = specific heat at constant pressure, Btu/lb-°R
T = total temperature, °R (T_3 at turbine station 3 etc.)

This relationship is plotted in Fig. 3-46. Eq. 3-80 is valid for both the ideal and real cases because it is an identity by definition.

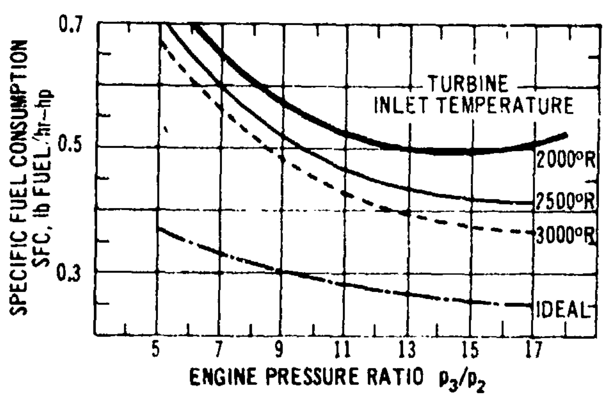


Fig. 3-45. Effect of Gas Turbine Cycle Parameters Upon Specific Fuel Consumption

3-3.1.3 Process Analysis

In the paragraphs that follow, the gas turbine engine is broken down for analysis into its primary elements/processes:

1. Inlet
2. Compressor
3. Combustor
4. Turbine
5. Exhaust (duct).

For each element, a simple statement of the first law of thermodynamics (AMCP 706-285) is applied, together with appropriate constraints, to yield the analytical basis for detailed calculations.

Results for each major engine component are developed subsequently. Note that the results from the application of the first law are energy balance statements. Other thermodynamic relationships are introduced as required to translate the fundamental equations into statements concerning such factors as component pressure ratio and temperature ratio.

For each equation developed, a parenthetical note is added when the result applies only to either the ideal or the real case. When the result is applicable to both, no note is appended.

The individual processes discussed subsequently are diagrammed for convenience in Fig. 3-47 for both the ideal and real cases.

3-3.1.3.1 Inlet

The inlet converts the kinetic energy associated with the free stream or forward flight velocity into stagnation pressure and temperature. In a real engine the stagnation state (velocity = 0) is never reached, because an infinite cross section would be required to pass the required flow rate. The concept is useful, however, even for low-velocity helicopters because it provides a convenient method of accounting for induction system losses.

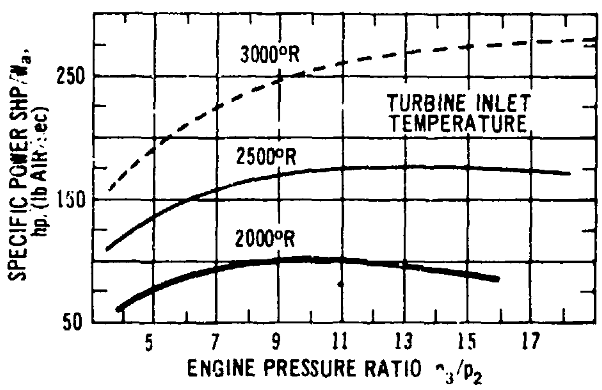


Fig. 3-46. Effect of Gas Turbine Cycle Parameters Upon Specific Power

The fundamental constraint for the energy equation is that the gross energy level is unchanged, with energy conversion taking place between velocity and static enthalpy.

$$h_2 = h_0 + \frac{V^2}{2gJ} \quad \text{, Btu/lb} \qquad (3\text{-}81)$$

where

g = acceleration due to gravity, fps²
h = enthalpy, Btu/lb
J = mechanical equivalent of heat, ft-lb/Btu (1 ft-lb = 778 Btu)

Subscripts

0 = freestream conditions at velocity V
1, 2 etc. = conditions at cycle stage 1, 2 etc. (see Fig. 3-47)

For the ideal case of a reversible process, the temperature ratio and pressure ratio relationships are

$$\frac{T_2}{T_0} = 1 + \left(\frac{k-1}{2}\right)M_0^2 \quad \text{, dimensionless} \qquad (3\text{-}82)$$

$$\frac{p_2}{p_0} = \left(\frac{T_2}{T_0}\right)^{\left(\frac{k}{k-1}\right)} \quad \text{, dimensionless} \qquad (3\text{-}83)$$

where

k = ratio of specific heats c_p/c_v, dimensionless
M = Mach number, dimensionless
p = total pressure, lb/ft²

A number of efficiency or loss definitions can be used to describe the real process. Perhaps the most straightforward is inlet pressure loss ratio, defined as Pressure Loss $= (p_2' - p_2)/p_2'$, where the prime value p_2' is the ideal. Typical values range from 1-3% and can be assumed constant over the flight and power ranges.

3-3.1.3.2 Compressor

The basic assumption for the compression process is that it takes place adiabatically (no heat added or rejected). Of course, some heat loss to the environment does occur. Compressor case temperatures of 1000°F and higher will be commonplace with high (> 15:1) pressure ratios, and it is clear that these temperatures result in heat loss to the atmosphere. However, while this heat loss—expressed in Btu/hr—may be substantial, it is negligible when expressed in terms relevant for the energy equation (Btu/hr)/lb air. Similar remarks concerning external heat loss are relevant for both the combustor and the turbine expansion processes discussed later.

With this assumption, the energy relation simplifies to:

$$W_{sh} = -(h_3 - h_2) \quad , \text{Btu/lb} \qquad (3\text{-}84)$$

where

W_{sh} = shaft work, Btu/lb

The negative sign indicates that work is done on the system. For the ideal reversible case, the pressure ratio to temperature ratio relationship is:

$$\frac{p_3'}{p_2} = \left(\frac{T_3}{T_2}\right)^{\left(\frac{k}{k-1}\right)} \quad , \text{dimensionless} \qquad (3\text{-}85)$$

In the process, a compressor polytropic efficiency η_c is introduced as

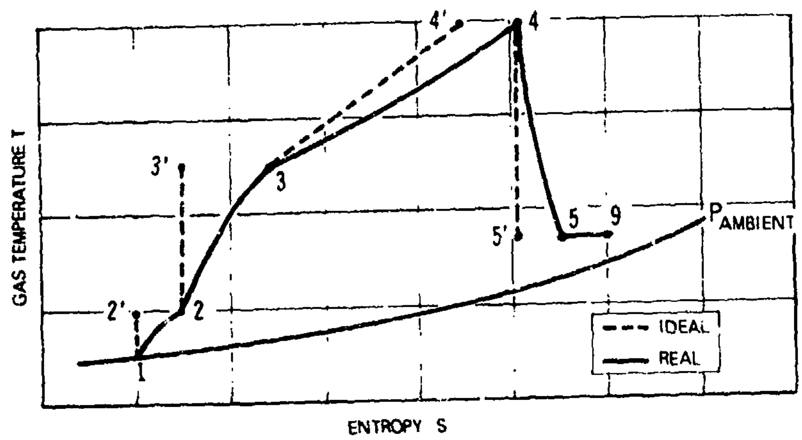
$$\frac{p_3}{p_2} = \left(\frac{T_3}{T_2}\right)^{\left(\frac{\eta_c k}{k-1}\right)} \quad , \text{dimensionless} \qquad (3\text{-}86)$$

A representative range of compressor polytropic efficiency is 80-90%.

3-3.1.3.3 Combustor

For the combustion process $W_{sh} = 0$ and the energy added in the combustor in the form of fuel Q (Btu) equals the change in enthalpy Δh. While this equation establishes the basic energy relationship, a further expansion is required to translate this into more significant terms involving fuel-to-air ratio, heating value of the fuel, compressor exit temperature, and turbine inlet temperature. The combustion process is a complex one and is discussed in detail in AMCP 706-285. The ideal process assumes that all of the energy released from the fuel is captured in the air streams without loss. It also assumes complete combustion and hence release of all of the energy available. Because neither of these assumptions is valid, a combustion efficiency η_b is introduced to account for deviations from the ideal energy addition. A representative range of combustion efficiency is 98-99%.

In addition, stagnation pressure drop takes place during combustion and has two components: the friction pressure drop and the momentum pressure loss.



KEY: 1–2 INLET DIFFUSER
2–3 COMPRESSOR
3–4 COMBUSTOR
4–5 TURBINE
5–9 EXHAUST DIFFUSER

(Ref. SAE ARP 681B)

Fig. 3-47. Gas Turbine Engine Cycle Process—Ideal and Real

3-3.1.3.4 Turbine

This expansion process also is assumed adiabatic, as in the compressor, resulting in the energy equation

$$W_{sh} = -(h_5 - h_4) = c_p(T_5 - T_4) \quad \text{, Btu/lb} \quad (3\text{-}87)$$

As with the compressor, a turbine polytropic efficiency η_t is introduced such that

$$\frac{p_4}{p_5} = \left(\frac{p_4}{p_5'}\right)^{1/\eta_t}$$

$$= \left(\frac{T_4}{T_5}\right)^{\left(\frac{k}{k-1}\right)^{1/\eta_t}} \quad \text{, dimensionless} \quad (3\text{-}88)$$

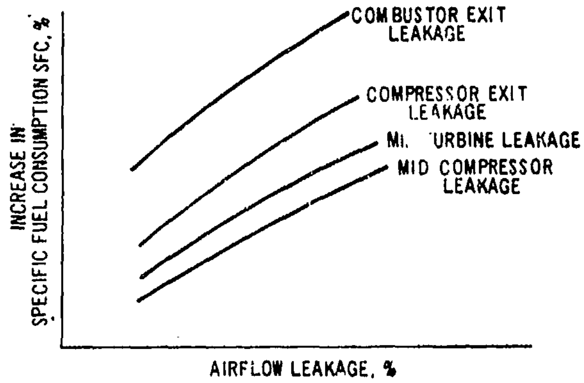


Fig. 3-48. Effects of Leakage Upon Specific Fuel Consumption

3-3.1.3.5 Exhaust Duct

This element is similar to the inlet in terms of the energy relationship. Again it is assumed that both Q and W_{sh} are zero, giving an energy relationship of the same form as for the induction system (Eq. 3-81).

In the case of a turboshaft engine, the exhaust system normally is designed to diffuse the relatively high turbine exit Mach number to a low value in the it plane. In this case, the ideal and real processes also are of the same form as for the ram compression process of the induction system (Eqs. 3-82 and 3-83).

A turbojet or turbofan engine that might find application as a turbojet or turbofan tip drive power plant would use an accelerating exhaust nozzle rather than a diffusing duct.

3-3.1.3.6 Other Considerations

The preceding paragraphs developed the fundamental descriptive relationships for the individual components, as well as the overall dependency of *SFC* and *SHP/W_a* upon cycle/component characteristics. These results now require consideration in conjunction with the effects of:

1. Air leakage from the engine
2. Turbine cooling air
3. Reynolds number effects
4. Gas property variations
5. Fuel heating valves.

1. Leakage. The performance penalties associated with air leakage from the engine are severe. While every effort is made to minimize leakage, design constraints never permit the attainment of a leak-free system. The tendency toward higher operating pressures and temperature levels in order to attain minimum *SFC* and maximum SHP/W_a compounds this difficulty by raising static force levels and temperature gradients throughout the engine.

The effects of leakage are shown in Fig. 3-48 for a typical engine cycle. It can be seen that the degradation in *SFC* varies depending upon the location of the leak. Fundamentally, the losses due to leakage increase as the leak point moves aft in the compressor (more work is done on the air and then lost to the cycle). As the leak point moves aft from the combustor exit through the turbine, two independent effects take place. First, the fuel that has been burned with the leakage air also is lost. Second, as the leak point moves aft through the turbine, less and less expansion work potential is lost and the *SFC* and SHP/W_a degradation decreases. If the leak occurs in the exhaust duct, no important effects are apparent.

An important distinction must be made between these effects and those to be expected from an engine that suddenly "springs a leak". In the latter case, rematch effects between turbine and compressor will degrade *SFC* and SHP/W_a even more because turbine geometry optimization, presumed for each point of Fig. 3-48, no longer will apply.

2. Cooling. The demand for better engine performance, lower *SFC*, and higher SHP/W_a leads to higher pressure ratios and turbine inlet temperatures. Because currently available materials will not permit operation at temperatures equal to required gas temperatures, turbine cooling is used. Cooling air from the compressor is circulated through the turbine elements to maintain material temperatures within acceptable limits.

Turbine cooling air can be thought of as a special form of leakage—where the leaked air is lost to the cycle temporarily, throttled to a lower pressure, and then reintroduced into a downstream turbine. The losses associated with turbine cooling air then become a function of where the air is extracted from the compressor (midpoint, exit, etc.) and where it is introduced into the turbine.

3. Reynolds number effects. The discussion to this point has concentrated upon thermodynamic considerations. It should be recognized, however, that a thermodynamic approach, while giving point-to-point relationships (compressor work $= -\Delta h$), usually cannot address the details of how the process takes place. In particular, the design of compressors, turbines, combustors, inlets, and exhaust systems is largely the work of the engine aerodynamicist. It is this discipline that configures the detail blading and sets tip speeds, incidence angles, and flow paths to meet the thermodynamic cycle goals. As in any aerodynamic device, the flow regimes and loss characteristics are sensitive to the absolute value of Reynolds number. The significance for an aircraft engine is that the operating Reynolds number in the engine decreases significantly as the aircraft operating altitude increases. Associated with the lower Reynolds number at increased altitude is degradation of compressor efficiency and compressor flow capacity, and possibly a combustor pressure drop. Although the Reynolds number in the turbine also decreases, efficiency and flow characteristics usually are not affected because the initial sea level Reynolds number typically is very high.

The subject of Reynolds number effects has been treated extensively in the technical literature. Definitive treatment in this handbook obviously is impossible. The subject is mentioned here primarily to caution helicopter designers that sea level data will not necessarily be applicable at altitude.

4. Gas properties. As noted in par. 3-3.1.2.1, the assumption of constant gas properties leads to straightforward and quickly available analytical results. While these give valuable insight into the major operating parameters of the cycle, they will not be sufficiently accurate for generating engine performance data. In particular, the specific heats vary through the engine as a function of local gas temperatures and composition. The composition changes from air in the compressor to a combination of products of combustion and air at the combustor exit and through the turbine. Modern, high-speed, digital computers handle these variables with ease, usually by storing the applicable thermodynamic data. This note on gas properties is intended primarily as a caution against using standard air data.

5. Fuel heating value. The heating value of a given fuel is determined in calorimeter tests conducted under "standard" conditions; the reactants (fuel and air) are brought to 77°F, ignited, and the products cooled to 77°F. The heat released in this cooling process is the heating value (HV). To use accurately the HV thus determined, the combustion equation must reflect that neither the reactants nor the products are at 77°F in the gas turbine engine.

3-3.1.4 Partial Power Cycle Analysis

The cruise power requirement for helicopters normally is well below the power required for hover. Hence, the *SFC* point of primary interest to the helicopter designer may not be given by design point cycle results. Fig. 3-49 shows the typical variation of *SFC* with *SHP* as a percentage of maximum *SHP* for a gas turbine engine. The typical power range for cruise also is indicated.

The degradation in *SFC* at partial power is caused primarily by the reduction in engine pressure ratio. Component efficiency variations with power modulations are not a primary influence on *SFC* degradation, at least for the free power turbine engine. Careful attention to detail component design and matching actually can result in component efficiency improvement at partial power.

The fundamental characteristics of the partial power operating mode for both free power turbine and coupled turbine designs are developed in the paragraphs that follow.

For this analysis, it is assumed that the power turbine nozzle remains choked throughout the power

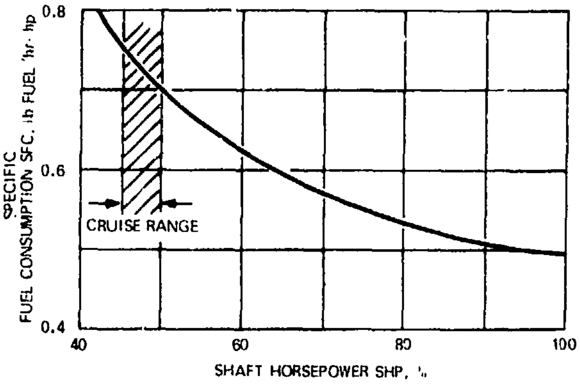


Fig. 3-49. Variation of Specific Fuel Consumption With Shaft Horsepower (Typical)

range, and that component efficiencies remain constant. Both assumptions are excellent approximations of actual engine conditions. Further partial power predictions based upon this model are appropriate to two-rotor compressor designs. In this case, extension of the basic reasoning permits accurate prediction of the individual compressor operating lines.

With these assumptions, it can be shown that partial power turbine inlet temperature ratio and engine pressure ratio are related by Eq. 3-86. An alternative expression in terms of compressor temperature ratio is

$$\left[\frac{T_4/T_2}{(T_3/T_2)-1}\right]_{par} = \left[\frac{T_4/T_2}{(T_3/T_2)-1}\right]_{des}, \text{ d'less} \quad (3\text{-}89)$$

where

Subscript

des = design

par = partial power or load

The *SFC* partial power operating line thus can be plotted on the design point cycle chart once the design point is selected, as shown in Fig. 3-50 (baseline data reproduced from Fig. 3-45).

Partial power (as a percentage of design power) also can be superimposed upon the data of Fig. 3-50.

$$\frac{SHP}{SHP_{des}} = \%\text{SHP}$$

$$= \left[\frac{SHP/W_a}{(SHP/W_a)_{des}}\right]\frac{W_a}{(W_a)_{des}}, \text{ d'less} \quad (3\text{-}90)$$

The first term is a function of T_4/T_2 and T_3/T_2 and component efficiencies, and therefore can be calculated

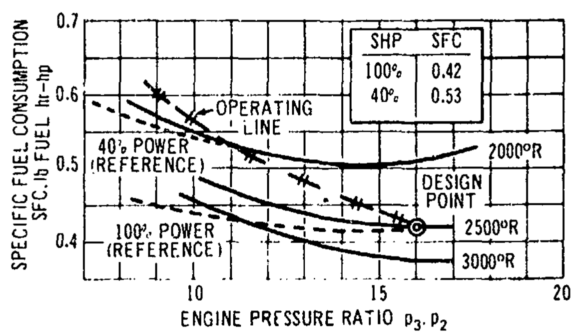


Fig. 3-50. Variation of Cycle Parameters With Partial Power Operation (Typical)

for each point along the *SFC* operating line. As a consequence of the assumption of a choked power turbine nozzle, the referred flow into the gas generator turbine remains at a fixed value. This leads to:

$$\frac{W_a}{(W_a)_{des}} = \frac{P}{P_{des}}\left[\frac{(T_4/T_3)_{des}}{T_4/T_3}\right]^{1/2}, \text{ d'less} \quad (3\text{-}91)$$

Hence, each point along the operating line of Fig. 3-50 will yield a unique value of partial power percent. This also is presented on Fig. 3-50 when the percent power data are shown as a family of lines. The intersection points with $(T_4/T_2)/[(T_3/T_2) - 1]$ = constant give the power and *SFC* characteristic shown in the inset. The technique outlined for integrating design point with partial power/*SFC* cycle calculations should prove valuable in itself. Perhaps more important, however, is the conclusion that partial power engine performance at helicopter cruise power essentially is established by the engine design point selection. Some fine tuning of partial power *SFC* is possible through techniques such as free power turbine speed selection and component efficiency matching. However, there are fundamental constraints operating to force reduction in the primary cycle variables (pressure ratio and peak temperature) and hence increase in *SFC* at partial power settings.

The approximations of *SFC* characteristics will demonstrate a distinct advantage for the free power turbine engine over the coupled type, given equivalent design points for both. While this trend is considered valid in general, a particular engine design could show the advantage to be negligible after other engine requirements are imposed upon the design selection process.

3-3.1.5 Reaction Systems

Drive configurations using reaction principles are possible and have been built. Two such reaction systems are:

1. Hot (or warm) gas system, in which pressurized gas is carried through the rotor and exhausted rearward at the rotor tip. The thrust developed by this high-velocity exit stream provides the required rotor drive power.

2. Tip drive system, in which a turbojet or turbofan engine is mounted at the rotor tip. The jet or fan thrust supplies the necessary rotor drive power.

The selection of an optimized reaction system over a conventional design cannot be made upon the basis of the system thermodynamic characteristics; instead, mission performance and economics are the major criteria guiding such decisions. The paragraphs that follow are restricted to development of the thermodynamic model and governing equations for reaction systems. No attempt is made to indicate where such systems might prove superior to conventional designs.

3-3.1.5.1 Hot Gas Cycle

The hot gas cycle, like the tip jet, eliminates the need for a transmission system. Because conventional transmission system weight represents 8-12% of helicopter gross weight, this would appear to be a significant advantage. However, a penalty in fuel consumption is incurred. Also, in addition to mission fuel weight, other subsystem weights are affected, particularly that of the rotor, which generally becomes heavier.

With conventional power transmission systems, power is transferred from the engine to the rotor at an efficiency of 95-97%. For the ideal hot gas system, this power transfer efficiency is dependent upon the kinetic energy "losses" of the exit stream. In the real case, friction and heat losses through the rotor duct also must be considered.

1. Ideal Gas Flow. The helicopter rotor acts as both a compressor and a turbine. As the gas flows radially outward, it is compressed in proportion to the square of the rotor tip speed. When the gas expands through the tip jet nozzle, a propulsion force is produced that is proportional to the exit gas velocity (relative to the nozzle). Because this force acts at a radius about the center of rotation, power is produced that is proportional to the product of exit gas velocity and rotor tip speed. This drive power balances the sum of compression power required and the usual rotor drag power.

Analogous to the concept of propulsion efficiency, a transfer efficiency for the system is defined as the ratio of useful work produced to the total energy expended. Here the kinetic energy "wasted" in the exhaust must be accounted for. With this definition, the transfer efficiency η_{tr} is given by

$$\eta_{tr} = \frac{\text{useful work}}{\text{useful work} + \text{exit kinetic energy}}\text{, d'less} \quad (3\text{-}92)$$

or, per unit mass

$$\eta_{tr} = \frac{V_eV_t - V_t^2}{(V_eV_t - V_t^2) + [(V_e - V_t)^2/2]}$$

$$= \frac{2}{1 + (V_e/V_t)}\text{, dimensionless} \quad (3\text{-}93)$$

where

V_e = exit velocity, fps
V_t = rotor tip speed, fps

The term $(V_e - V_t)$ is the absolute value of exit velocity (stationary observer). This formulation of η_{tr} is identical to that of propulsion efficiency for turbojets and turbofans, where V_t replaces the usual flight velocity term V. The exit velocity V_e is dependent upon the amount of compression work done in the rotor as well as the gas power produced by the turbine engine cycle. A relationship for η_{tr} in terms of rotor tip speed and the cycle power produced, while more cumbersome, is considered appropriate for this paragraph on cycle analysis. From the relationship (per unit mass)

Work Produced = Cycle Work + Compression Work

$$V_e^2 = V_0^2 + V_t^2 \quad (3\text{-}94)$$

the transfer efficiency η_{tr} can be expressed as:

$$\eta_{tr} = \frac{V_t}{V_0}\sqrt{\left(1 + \frac{V_t}{V_0}\right)^2 - \left(\frac{V_t}{V_0}\right)^2}\text{, d'less} \quad (3\text{-}95)$$

where V_0 is defined on the basis of cycle performance, such that the kinetic energy or head that would be developed by the gas if it were expanded isentropically to ambient pressure would equal $V_0^2/(2g)$ (ft-lb/lb).

2. Real Gas Flow. In real gas flow, friction and heat transfer losses occur through the rotor. In this case, a loss factor LF is defined as:

$$LF = \frac{(V_e')^2 - V_e^2}{V_t^2}\text{, dimensionless} \quad (3\text{-}96)$$

where

V_e' = no-loss exit velocity, fps
V_e = actual exit velocity, fps

With this definition, the transfer efficiency η_{tr} becomes

$$\eta_{tr} = 2\left[\left(\frac{V_t}{V_0}\right)\sqrt{1 + (1 - LF)\left(\frac{V_t}{V_0}\right)^2} - \left(\frac{V_t}{V_0}\right)^2\right], \text{ dimensionless} \quad (3\text{-}97)$$

Fig. 3-51 shows results for both the ideal and real flow cases.

3-3.1.5.2 Tip Jet

The Figure of Merit for the tip jet, defined as the ratio of useful power produced to total energy expended, takes the familiar form of the jet transfer efficiency $\eta_{tr(jet)}$

$$\eta_{tr(jet)} = \frac{2}{1 + (V_e/V_t)}, \text{ dimensionless} \quad (3\text{-}98)$$

The similarity with the hot gas cycle propulsion efficiency formulation is apparent. For the tip jet case, however, further expansion of this basic relationship is not required because there is no intervening duct between the gas generator and the propulsion nozzle.

The thermodynamic nature of the tip jet as a rotor drive system is relatively straightforward. However, unique problems can arise in the mechanical design of such a system because of the steady g-field imposed. This new environment demands particular care in the design of fuel and lubrication systems. The designers of bearings and mounts also must pay special attention to the gyroscopic forces attendant to the rotor rotation.

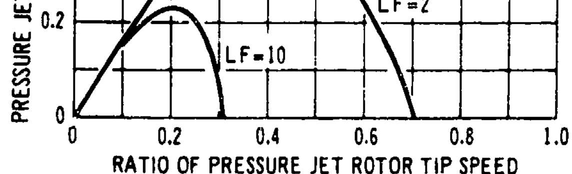


Fig. 3-51. Pressure Jet Transfer Efficiency as a Function of Cycle Losses

3-3.2 ENGINE CHARACTERISTICS AND SELECTION

3-3.2.1 Basic Considerations

The selection of the helicopter engine must be based upon the system approach. All characteristics influencing the performance of the engine as installed in the aircraft must be considered and the influence of each characteristic weighed in the final selection.

Basic trade-offs—among performance, development time, cost, maintainability, durability, and reliability—are made during the development of an engine, and subsequently become important aspects in engine selection. The airframe manufacturer cannot evaluate performance on the basis of the engine alone, as the end result he seeks is the total helicopter system performance. Engine configuration (size, weight, and arrangement), interface requirements, and other airframe-limiting factors must be considered in the overall evaluation. Engine costs and development time (if applicable) can be evaluated relative to program scope; however, maintainability, reliability, and durability, although important, lack uniform criteria.

Two engine types are in use in helicopters: the reciprocating, or piston, engine and the turboshaft engine. The current trend is toward the exclusive use of the turboshaft engine because it is lighter, smaller, and more reliable than the piston engine; however, the piston engine is more efficient (lower *SFC*) at present and has a longer history of Army usage. Current gas turbine engines in the 2000- to 3000-hp class produce approximately 4 hp/lb of engine weight at maximum power, whereas piston engines produce less than 1 hp/lb. This weight advantage permits more payload on the gas turbine engine-powered helicopter. The smaller relative size of the gas turbine engine reduces aerodynamic drag, permits more convenient location on, or within, the helicopter and allows greater payload.

Reliability of an engine is dependent to a degree upon the state of development of the individual engine. The experience of the Army with turbine engine-powered aircraft clearly indicates that the rotary motion of the gas turbine engine shows a significant inherent improvement over the reciprocating motion of the piston engine insofar as reliability is concerned. Gas turbine engines used in helicopter applications have a time-between-overhaul (TBO) on the order of 1200 hr whereas reciprocating engines have a TBO on the order of 500 hr.

Fuel consumption may be higher for the gas turbine engine. The *SFC* usually is 0.5-0.7 lb/hr-hp, whereas the *SFC* for a comparable piston engine would be approximately 0.4. However, the *SFC* of the gas turbine engine is continuing to decrease as further development permits the use of higher compressor pressure ratios and more efficient components.

Because of the lower maintenance costs of the gas turbine engine (lifetime system costs of parts and labor) and the lower cost of turbine fuel compared to aviation gasoline, the lifetime cost of the gas turbine engine is lower than that of the piston engine (Ref. 38).

Three other advantages of the gas turbine engine over the piston engine are:

1. Improved ability to start at low temperatures
2. Ability to produce full power immediately (no warmup)
3. Ability to burn a wide range of fuels.

Because of the advantages of the gas turbine engine, from the overall system viewpoint, and the trend toward the exclusive use of that engine in Army helicopters, only the gas turbine engine is considered in the following discussion.

3-3.2.1.1 Gas Turbine Engine

Fig. 3-52 is a photograph of a cutaway of a typical turboshaft engine. Beginning at the left of the engine the major components are the gearbox, the compressor (consisting of five axial stages and one centrifugal stage), and an annular combustion chamber surrounding the four-stage axial turbine.

Ref. 39 lists the characteristics of gas turbine and piston engines used in military helicopters. It includes turboshaft engines with power levels from 317 hp to 4800 hp, but *SFC* ranges only from 0.48 to 0.69.

3-3.2.1.2 Coupled and Free Power Turbine Engines

Two basic configurations of turboshaft engines are possible: the single-spool type, called the "coupled engine"; or a type with a separate power turbine, usually called the free power turbine engine (Fig. 3-53). In Fig. 3-53(A) the gas turbine engine has a single spool and the excess power of the turbine (over that required to drive the compressor) is available to power the helicopter. In the engine shown schematically as Fig. 3-53(B) the gas generator turbine extracts only enough power to drive the compressor. The remainder of the energy is extracted by the power turbine with an independent (free) shaft to power the helicopter. As discussed in par. 3-3.1, there is no difference in the thermodynamics of the two engines and, at design load, there is no difference in *SFC*. However, as the load is decreased from the design condition, the *SFC* of the single-spool (coupled) engine increases more rapidly than does that of the free power turbine design. The primary advantage of the free power turbine design over the single-spool or coupled engine is the increased operational flexibility, because the gas generator and the power turbine can be operated at different speeds. Thus, the gas generator speed can be controlled—to tailor the air weight flow W_a, to obtain better component efficiencies, and obtain more favorable pressure ratios and turbine inlet temperatures—independent of the rotor speed requirements. Because of the improved efficiency of the free power turbine design, most helicopter turboshaft engines are of this type.

3-3.2.1.3 Cycle Modifications

In the selection of design parameters for a helicopter engine for a given set of mission profiles, extensive studies are made of the Brayton cycle (idealized gas turbine cycle). These studies involve various combinations of compressor pressure ratios and turbine inlet temperatures, with realistic values of component efficiency, pressure loss, and leakage.

Two other modifications are considered for special missions—the use of regeneration and reheat. Both techniques are shown schematically in Fig. 3-54. Regeneration involves the use of a heat exchanger to extract some of the energy (heat) left in the gases as they leave the turbine and to transfer this energy to the air leaving the compressor (Fig. 3-54(A)). The recovery of this energy reduces the amount of fuel required to heat the air to the turbine inlet temperature, thus reducing the *SFC* and increasing the efficiency.

The regenerator has a particularly beneficial effect upon the efficiency at partial power. In fact, gas turbine engines with regenerators have been fabricated that gave improved efficiencies over diesel engines. The disadvantages of the regenerator are its size, weight, and cost. Unless the flight range of the helicopter is long, it usually is not possible to justify a regenerator by the weight of fuel saved. Consequently, it would appear that only for special missions will regenerators be employed in helicopter engines.

The reheat cycle shown in Fig. 3-54(B) incorporates a combustion chamber between two turbines. Because the primary combustor operates with a lean fuel/air ratio, sufficient oxygen is available to react with additional fuel. Reheat increases the work output of the power turbine because the turbine output is proportional directly to the inlet temperature. For a typical case of an engine with a pressure ratio of 8:1 and a turbine inlet temperature of 1600°F, a reheat combus-

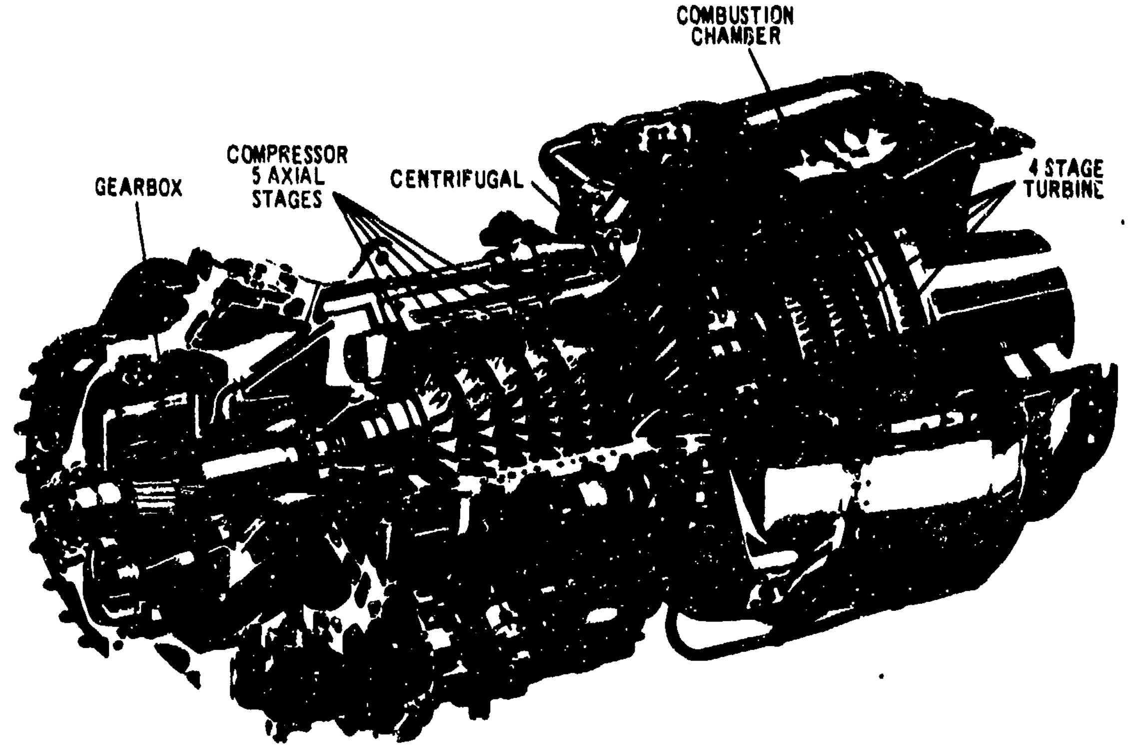


Fig. 3-52. Cutaway of a Typical Turboshaft Engine (T53L-13)

tor would increase the output of the engine from 88.5 Btu/lb-of-air to 114 Btu/lb-of-air, or 29%. The energy is added at a lower pressure so that, thermodynamically, the efficiency of the reheat cycle—compared to the basic cycle—must decrease.

Reheat combustors can be made reasonably small and lightweight; it is primarily the complexity of the control that makes reheat undesirable. If power turbine cooling is required as a result of reheat, further complications are encountered because the power turbine normally would not require cooling. However, for special applications, the reheat cycle could show system advantages.

3-3.2.2 Selection of Design Point Cycle Parameters

From an examination of Figs. 3-44 through 3-47 which are presented for the simple Brayton cycle (no regeneration or reheat), it is obvious that high compressor pressure ratios and T_4's are required for high specific output and low *SFC*. Assuming constant specific heats ($c_p = 0.24$, $c_v = 0.17$), no pressure drop, and no change in mass flow (perfect gas, isentropic process), the output and thermal efficiency of a basic cycle turboshaft engine can be written in terms of the pressure and temperature ratios, specific heats, and component efficiencies.

To obtain the optimum design pressure ratio for a given value of T_4 for the cycle (with the simplifying assumptions), these equations may be differentiated and set equal to zero. It is assumed that the component efficiencies are not functions of pressure ratio. (As pressure ratio is increased in actual engines, however, the component efficiencies do decrease.) For maximum output

$$\left(\frac{p_3}{p_2}\right)_{opt\ out}^{\left(\frac{k-1}{k}\right)} = \left[\eta_c \eta_t \frac{T_4}{T_0}\right]^{1/2}, \text{ d'less} \qquad (3\text{-}99)$$

where

T_4 = turbine inlet temperature = 2540°F (3000°R)

η_c = compressor polytropic efficiency = 0.83

η_t = turbine polytropic efficiency = 0.87

the optimum compressor pressure ratio p_3/p_2 is shown to be

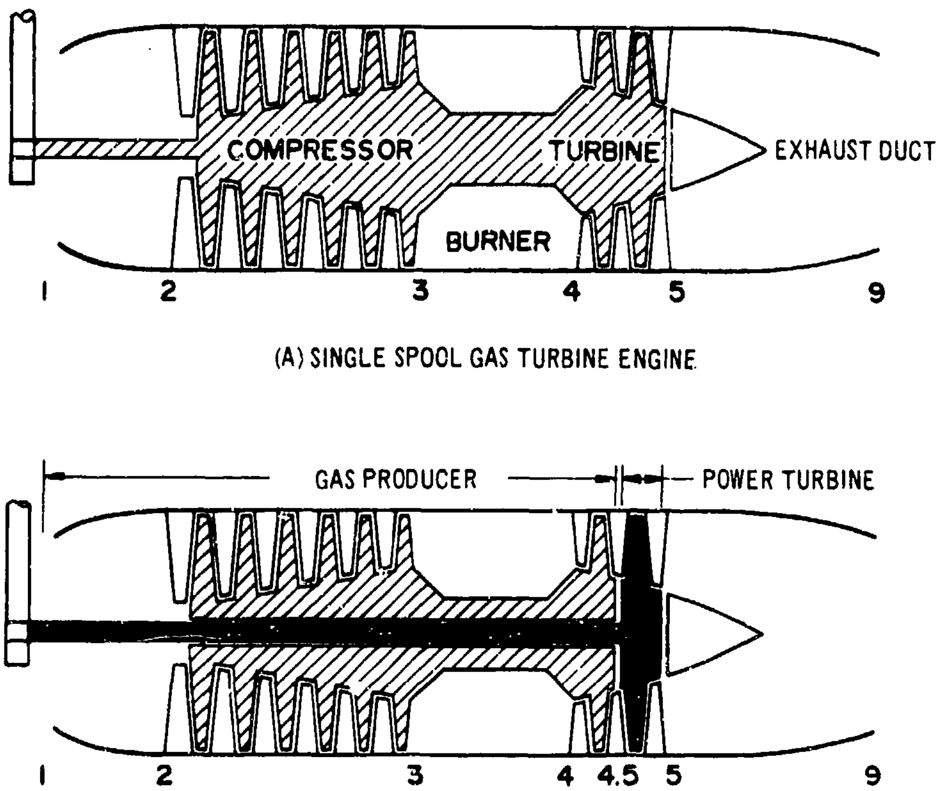
$$\left(\frac{p_3}{p_2}\right)_{opt\ out} = \left[0.83(0.87)\ \frac{3000}{519}\right]^{\frac{1.41}{2(1.41-1)}}$$

$$= 11.7 \qquad (3\text{-}100)$$

For maximum thermal efficiency (minimum *SFC*) a more complex relationship results (Ref. 40). Evaluation of the expression for the same T_4 and component efficiencies results in a value of $p_3/p_2 = 29.8$.

Because of the simplifying assumptions introduced in deriving the equations, the results obtained are only approximate. The equations are useful, however, and are sufficiently correct for obtaining general performance characteristics of the engine; i.e., while the optimum pressure ratio in the actual engine is not 11.7 for maximum output and the optimum pressure ratio for maximum thermal efficiency is not 29.8, the ratios are of the order of 12 and 30, respectively. These values indicate the importance of a high value for pressure ratio; they also indicate that an engine does not give both the highest specific output and the least *SFC* at the same pressure ratio. It should be noted that the curves of Fig. 3-46 (specific power versus pressure ratio) are relatively flat and that there is little, if any, loss in specific power as the cycle pressure ratio is increased beyond about 12.

In the previous example the T_4 was selected as 2540°F. Much engineering effort must be devoted to the selection of the T_4 for a given engine. All values of T_4 above approximately 1800°F require the use of air bled from the compressor for cooling of the turbine at least in its early stages. This is accomplished by bleeding air early in the compression cycle and passing it



(A) SINGLE SPOOL GAS TURBINE ENGINE

(B) FREE TURBINE GAS TURBINE ENGINE

Fig. 3-53. Helicopter Turboshaft Engines

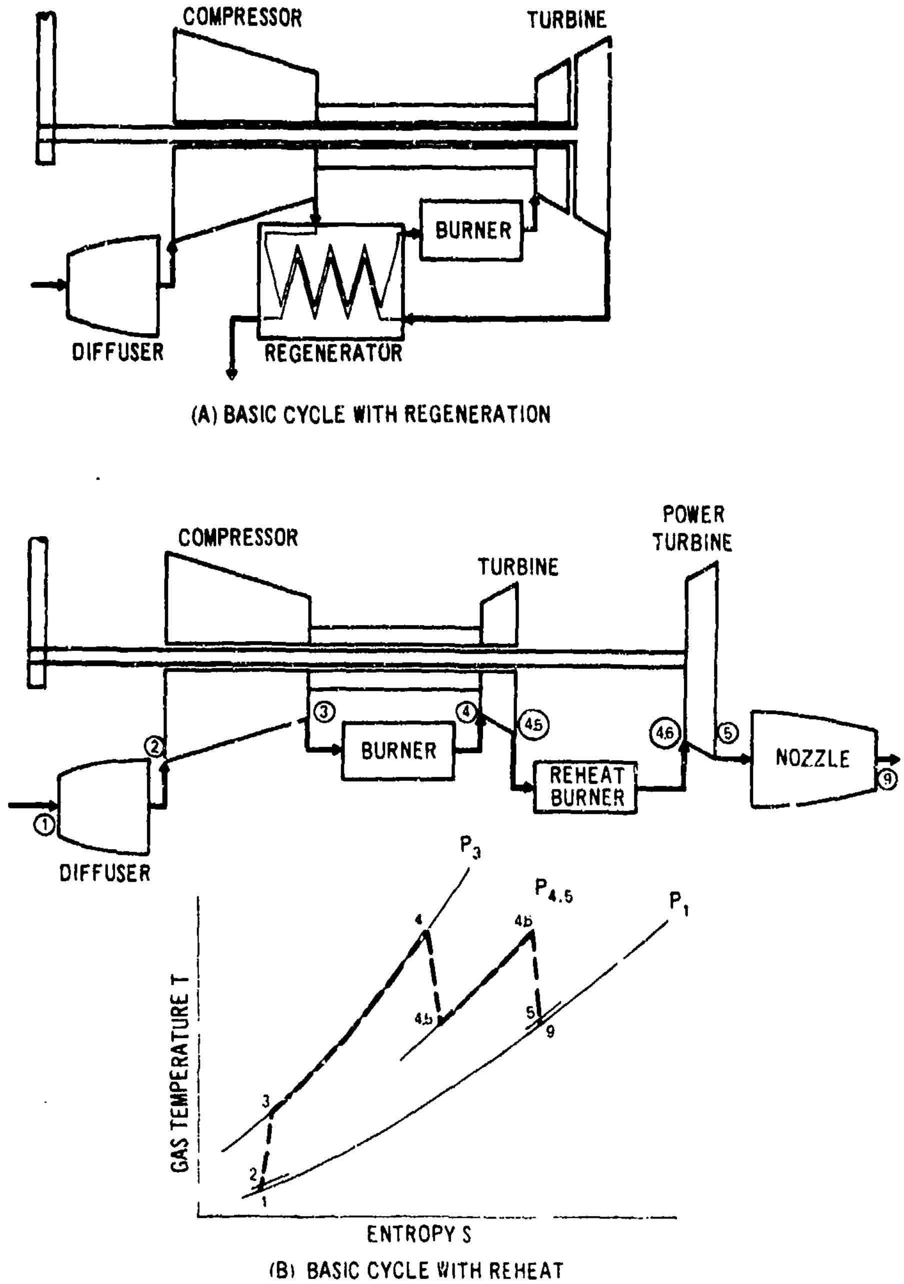


Fig. 3-54. Modifications to the Basic Gas Turbine Engine Cycle

through the turbine blades, from which it exits downstream into the turbine gas flow.

An estimate of cooling air requirements for blade cooling is given in Fig. 3-55. It should be noted that as the air is bled from the compressor for turbine cooling, it is lost for turbine work (at least for the cooled stages) and consequently there is a significant loss in engine output. Therefore, careful studies must be made to select the proper balance between the increase in work associated with increased T_4 and the loss in work associated with the air bled for cooling.

Developmental engines have been operated at 3000°F with cooled blades, but the T_4 of existing engines is well below this value—in the range of 1900-2250°F. Turbine

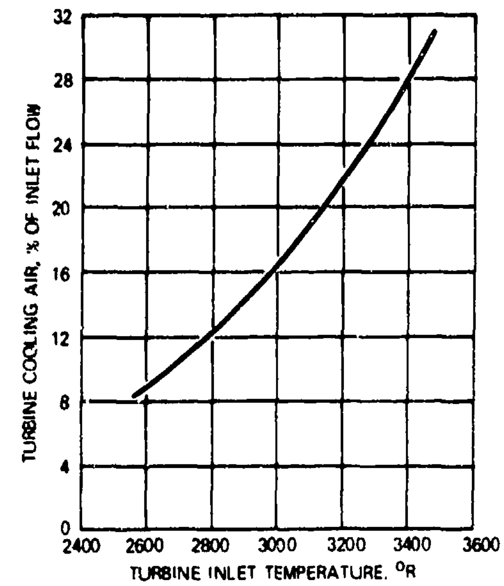


Fig. 3-55. Effect of Turbine Inlet Temperature on Turbine Cooling Air Requirements

inlet temperatures and pressure ratios are expected to continue to increase in the future.

Currently, turbine engine compressors can be built with pressure ratios of about 25 with good efficiency. Compressors with pressure ratios greater than approximately 12 must have variable stators and/or two or more rotors operating at different speeds to obtain good operating characteristics. This complication, combined with the extremely short blades associated with the small airflows through the current sizes of helicopter engines, tends to keep the pressure ratios of helicopter engines less than 15. Several of the current designs of helicopter engines have compressors using several axial stages with a final centrifugal stage to assist in overcoming the problems associated with the operation of the shortest blades at the discharge end of the compressor.

Ref. 39 shows that current military helicopter engines have compressor pressure ratios of about 6 to 8, with a maximum value of 14. Researchers are looking into means of increasing the pressure ratio of the compressor by increasing the pressure ratio per stage of both axial and centrifugal compressors while maintaining good efficiency. Success in this research would mean that small, lightweight, efficient, high-pressure-ratio compressors could be built. As the size of the engine increases, so does the ease of increasing the pressure ratio. Thus, for all engines, but especially the large models, the compressor pressure ratio is expected to increase in the future, leading to attendant increases in specific power and efficiency.

It should be noted that the shaft output of an engine can be increased by increasing the weight flow of air W_a through the engine, because the output is proportional directly to W_a. The *SFC* of the engine is not significantly changed by changing the airflow rate.

As stated earlier, the choice of the desired engine parameters and then an engine for a given helicopter, must be based upon the system approach. Apparent basic design improvements are often offset by disadvantages. Examples are:

1. As compressor pressure ratios are increased, overall compressor efficiency decreases and the compressor becomes heavier; furthermore, the number of gas producer (compressor power) turbine stages increases, or each stage becomes more heavily loaded, with a decrease in efficiency.

2. As T_4 is raised, either more exotic turbine materials must be used or cooling air must be bled from the compressor and ducted to the turbine, with the resulting complications and losses associated with that process.

Therefore, the advantages of a higher compressor pressure ratio or a higher T_4 must be balanced against the design, development, and production problems. Required power can also be achieved by increasing engine size so as to increase the airflow through the engine. Increases in overall efficiency will only be attained by improvements in compressor and/or turbine detail design and/or component efficiency.

3-3.2.2.1 Partial Power Operation

After a preliminary choice has been made for the design values of compressor pressure ratio p_3/p_2 and T_4, the operation of the engine must be investigated. Because the major portion of the operating life of a helicopter engine is at less than full power, considerable effort must be directed toward selecting the partial power design point parameter such that output and efficiency will not degrade excessively as the value of T_4 and engine gas producer speed decrease as the power is reduced. If such factors as weight, size, and component efficiency could be held constant, it would be desirable to select a pressure ratio greater than the optimum for the maximum power condition.

3-3.2.2.2 Analytical Design

The gas turbine engine manufacturers have developed computer programs that design the components of an engine for selected values of airflow, pressure ratio, turbine inlet temperature, etc. These programs are combinations of theory and experience since it is not possible to handle many of the flow and structural mechanics problems theoretically. In the analysis of the compressor, for example, a typical program would produce as an output such elements as the number of stages, the rotational speed, the blade shapes, blade spacings (solidity), compressor weight, and compressor efficiency; allowance would be made for boundary layers on the blade and casing surfaces, and radial equilibrium would be treated. A program might be written for the design point and a separate program for partial power operation, or the programs might be combined to produce, as outputs, the basic hardware design and the compressor operating parameters for the complete range of operating conditions. It is possible to design the program so as to display the operating parameters as they are being calculated so that input variables may be changed and recalculations made and displayed.

After computer designs of all components are prepared, they are integrated into a complete engine program and the overall engine performance is calculated. Most of the secondary effects—variation of component efficiency with flow rate and speed—are included in these computer programs so that they are much more accurate than the simplified analyses discussed earlier.

Obviously, these computer programs are extremely valuable in reducing design time and costs and in selecting a most likely combination of components from the infinite combinations possible. Unfortunately, there are limitations in these programs because of the complexity of factors such as fluid mechanics and structural mechanics in gas turbine design. Some of the incompletely understood problems are boundary layer separation, three-dimensional flow around compressor blades, transonic flow, mixing, mixing in the presence of combustion, combustion, blade vibration characteristics for complex shapes, and blade vibration characteristics in the presence of a high velocity gas flow. Thus, the components and engines still require considerable development after the hardware has been fabricated.

3-3.2.3 Control System

The control system of a helicopter engine enables the engine to operate in the many modes required. Ideally, the system controls the engine so that either minimum *SFC* or maximum acceleration is achieved, depending upon the requirements of the pilot. The general control requirements for the helicopter engine are (Ref. 41):

1. The engine must start throughout a wide range of environmental conditions. The range of conditions may be specified as unassisted or assisted starts.

2. The control system must provide transient operation (more or less power) with a minimum response time while avoiding stall, overtemperature, and burner blowout.

3. The engine must accelerate or decelerate smoothly to new power levels and must operate stably at a selected operating point.

4. The control system must keep the engine operating at or below the maximum turbine inlet temperature, the maximum rotative speed, and the maximum output power.

Table 3-1 lists the definable regimes of operation, the effects of the independent variables upon performance, and the possible control parameters.

3-3.2.3.1 Simple Control System

The simpler and, in general, smaller helicopter engines incorporate a simple control system. A schematic of a typical control system for a free turbine engine is presented as Fig. 3-56. In this control scheme only five variables are sensed: compressor discharge pressure p_3, gas producer speed N_g, power turbine speed N_p, throttle position, and fuel pump discharge pressure p_f. The system shown is a pneumatic-mechanical system where the pneumatic air is supplied by the compressor. The rotor speeds are sensed mechanically by two flyweight governors driven through gears at speeds proportional to N_g and N_p, respectively.

Basically, fuel flow is a function of p_3 with modulations obtained by opening bleed valves through the action of the governors sensing N_g and N_p. The fuel control system is based upon controlling the engine power output by sensing N_g. With the throttle in ground idle, N_g is controlled by the N_g regulator. With the throttle fully open, N_p will be held constant at 100% of maximum and N_g is established by power turbine governor action upon the gas producer fuel control. The control system is designed for a reduced fuel flow during the initial starting cycle. N_g must be 15% of maximum for fuel flow to begin. As p_3 increases, a bellows acts to increase the fuel flow in accordance with an acceleration schedule; this continues until at 52% of maximum N_g the flyweight governor acts to reduce the flow, stabilizing to an equilibrium of 62.6% of maximum N_g with the throttle in ground idle. At this condition, N_p will be less than 100%. When the throttle is moved to full open with minimum collective pitch, the fuel flow will be increased to the acceleration schedule and N_g and N_p will increase. N_p will increase until it reaches 100%, at which time the power turbine governor will act to hold the fuel flow constant. At this condition N_g will be between 78% and 82%. If the throttle is open fully and the collective pitch is increased, N_p will droop. The decrease in flyweight force in the power turbine governor changes the bleed valve position so as to cause the fuel flow to increase to restore N_p to 100%. The increase in fuel flow causes N_g and the T_4 to increase, which increases the energy flow to the power turbine, restoring N_p to 100%. As power demand increases, the fuel flow continues to increase until N_g reaches a maximum of 100%; with N_p at 100%, this is the maximum output of the engine.

Obviously, this discussion is simplified. There are compensatory devices such as an overspeed device that shuts off fuel flow at an N_p of 112%, an overtemperature device, and a throttle setting so that the pilot can operate the engine with N_p at other than 100%.

3-3.2.3.2 Variable-geometry Control System

There are three variables that a more complicated engine control system must control directly: fuel flow, variable vanes in the compressor, and variable vanes in the turbine.

Fuel flow is regulated to provide power changes, to control the gas generator speed, to limit the power turbine speed, to limit the output power, to prevent overtemperature during transient operation, to avoid compressor stall, and to prevent burner blowout.

The variable compressor vanes are positioned to provide good compressor efficiency and compressor stall margin throughout the power operating range.

The variable turbine vanes are controlled to maintain rated gas generator turbine inlet gas temperature T_4 over most of the output power range. Special engine shutdown features must be provided for engine protection.

In a typical operating mode for a helicopter engine incorporating variable compressor and turbine vanes, the engine is started automatically and T_4 increases as the throttle is advanced from idle. The maximum value of T_4 is reached early in throttle travel. After reaching the maximum value, the temperature is held constant by varying the geometry. Increasing fuel flow increases the gas generator speed while holding maximum T_4 minimizes fuel consumption. Power modulation is achieved by gas generator speed governing with fuel flow. The speed setting of the gas generator governor is a function primarily of throttle position and compressor inlet air temperature.

The compressor variable vanes are scheduled as a function of gas generator speed and compressor inlet air temperature. The schedule is set to maintain good compressor efficiency and stall margin during both steady-state and transient operation.

The turbine variable vane schedule is a function of throttle position. Because throttle position leads gas

TABLE 3-1
CONTROL PARAMETERS

(A) DEFINABLE REGIMES OF OPERATION:

1. STARTING	6. MAX POWER
2. ACCELERATION	7. IDLE
3. DECELERATION	
4. SHUTDOWN	8. MAX ROTOR SPEED
5. CRUISE POWER	9. MAX GEAR TORQUE

(B) EFFECT OF INDEPENDENT VARIABLES ON PERFORMANCE:

INDEPENDENT VARIABLE	WHEN	EFFECT ON OPERATION
1. FUEL FLOW	INCREASING	INCREASES TURBINE TEMPERATURE, POWER, rpm
2. ROTOR BLADE PITCH ANGLE	INCREASING	REDUCES rpm
3. COMPRESSOR VARIABLE GEOMETRY	UNLOADING	INCREASES STALL MARGIN AT LOW SPEEDS
4. TURBINE VARIABLE GEOMETRY	OPENING	IMPROVES SURGE MARGIN CHANGES SPEED RELATIONSHIP

(C) POSSIBLE CONTROL PARAMETERS:

1. THROTTLE POSITION	7. P_2	11. T_2
2. GAS GENERATOR, rpm	8. P_3	12. T_3
3. POWER TURBINE, rpm		
4. ROTOR BLADE ANGLE	9. P_4	13. T_4
5. GEARBOX TORQUE	10. P_5	14. T_5
6. FLIGHT CONDITION		

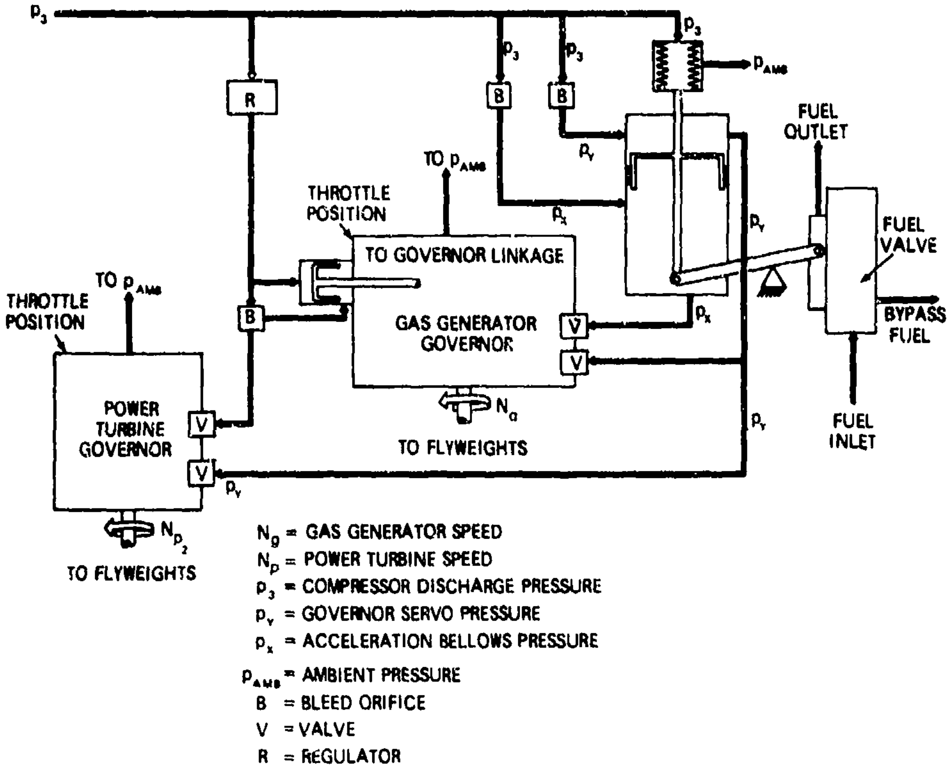


Fig. 3-56. Schematic Diagram of a Simple Gas Turbine Engine Control (Ref. 42)

generator speed on transients, the vanes are positioned to aid in providing minimum transient response time.

A safety feature usually designed into the control system provides automatic shutdown of the fuel valve in the event of gas generator or power turbine overspeed or excessive turbine temperature.

3-3.3 ROTOR AND PROPELLER ANALYSIS

3-3.3.1 Rotor Configurations

3-3.3.1.1 Rotor Types

The primary distinguishing features of rotary-wing aircraft are the rotor system and the drive system associated with the configuration. In shaft-driven systems, the transmissions use gear and bearing design techniques that are quite similar to those used in other fields, although the degree of refinement in design has been carried to a very advanced state. The rotors used in helicopters, however, find almost no counterpart elsewhere and are unique in their operating conditions. Many distinct types of rotors have been used successfully, and to see why these have found application, it is necessary to outline the range of operating conditions for which a rotor must be designed. The rotor must provide sufficient thrust and hence lift for sea level hovering and an additional margin of thrust for vertical climb and for hovering at higher altitudes. In forward flight the rotor provides the propulsive thrust as well as the lift to sustain the aircraft. The rotor also provides the required aircraft control forces for roll and pitch, as well as acceleration fore, aft, laterally, and vertically.

As the rotor moves through the air in forward flight, it experiences changes in airflow that give rise to periodic fluctuations of aerodynamic forces. Fig. 3-57 depicts the rotor motion in forward flight. At position A a blade element is advancing into the helicopter airflow and its velocity relative to the air is $(\Omega r + V)$. At position C the blade element is retreating from the helicopter airflow and its velocity relative to the air is $(\Omega r - V)$. At Points B and D the velocity is Ωr normal to the blade radial axis but has a radial flow velocity varying from $+V$ to $-V$. Thus, the rotor blade elements experience a gross fluctuation in velocity normal to the radial axis of $\pm V$ during every revolution. In addition, the blade experiences varying inflow/outflow velocities during each revolution. Corresponding to

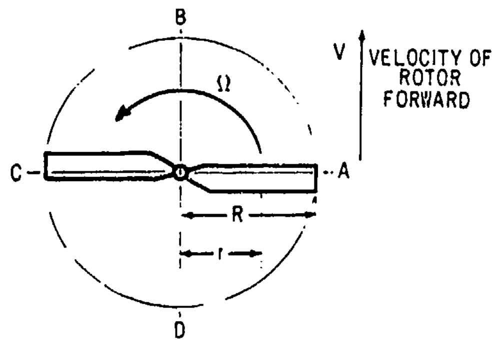


Fig. 3-57. Plan View of Rotor in Flight

this total velocity variation, the blades experience varying lift and drag forces during every revolution.

To accommodate the wide range of requirements, rotor systems have evolved into a few fundamental types, namely:

1. Fully articulated rotors, in which the blades are attached to the hub by means of hinges that allow the blades to move freely up and down (flap) in a vertical plane about the horizontal hinge, and by hinges that allow the blades to rock back and forth in the plane of rotation (lead-lag motion) about a vertical hinge. In effect, the blades are mounted to the hub structure upon universal joints.

2. Semirigid or floating-hub rotors, in which the blades are connected rigidly to each other through a hub structure and the rotor is allowed to tilt or rock with respect to the rotor drive shaft. When a two-bladed rotor is mounted in this fashion, it usually has a single teetering hinge and the blade motion resembles that of a child's teeter-totter.

3. Fully rigid or hingeless rotors, in which the blades are connected rigidly to the hub structure and the hub in turn is connected rigidly to the rotor mast. No flapping or lag hinges are used in these rotors.

4. Flex-hinge or strap-hinge rotors, in which a degree of rigidity for the blade connection is achieved that lies between that of a freely hinged blade and the high value of stiffness found with rigid connections.

In all of these rotor systems some type of blade pitch change hinge is incorporated. Such hinges allow the blades to pivot about an axis that runs parallel to the blade span. The pitch change hinge often is referred to as the blade-feathering hinge and is usually located so as to pass near the quarter chord of the blade.

The numbers of blades that have been used on rotors include as few as one counterweighted blade and as many as eight blades. One-bladed configurations tend to have somewhat higher vibration levels, while the multibladed configurations tend to have reduced vibration levels, provided the blades are all in balance. The most common number of blades lies in the range of two to five, and two blades usually are used for simplicity. The trend in larger modern helicopters seems to be moving toward increased numbers of blades.

3-3.3.1.2 Rotor Hub Geometry

To illustrate better the actual configurations of rotor hubs, some typical hub geometries are considered in the paragraphs that follow.

There are two general types of fully articulated rotors. In one the flapping and lag hinges are separated, and in the other these hinge axes intersect. The first type is shown schematically in Fig. 3-58. In this rotor design, the flapping hinge passes through the center of rotation. In Fig. 3-59 a similar hub arrangement is shown except that the flapping hinge is located outboard of the center of rotation. Such a configuration is known as an offset hinge hub. Fig. 3-60 is a photograph of an early hub of the type shown in Fig. 3-58. The second category of fully articulated rotors, those with coincident flap and lag hinges, is shown schematically in Fig. 3-61. This hub is also of the offset type. A photograph of a typical hub of this type is shown in Fig. 3-62.

The two-bladed semirigid (teetering hinge) rotor configuration is popular for smaller rotors. The simplest form is shown schematically in Fig. 3-63. A photograph of the same configuration is shown in Fig. 3-64.

A good example of the floating hub is that used by Doman (Fig. 3-65). The rotor unit itself is rigid or hingeless except for the necessary feathering or pitch change bearings. The entire rotor assembly is mounted to the vertical mast by means of a universal joint, and the power to the rotor is passed from the drive shaft through the joint.

Fig. 3-66 illustrates a hingeless rotor as exemplified by the rotor on the Bölkow BO 105 helicopter, which incorporates feathering hinges for blade pitch change.

3-3.3.1.3 Rotor Control Systems

Control of the helicopter is achieved primarily through the control of the rotor system. Although the helicopter, in principle, can move in three mutually perpendicular directions and can rotate about three axes, in practice some of the controls are normally coupled together so that four independent pilot controls are provided. These are:

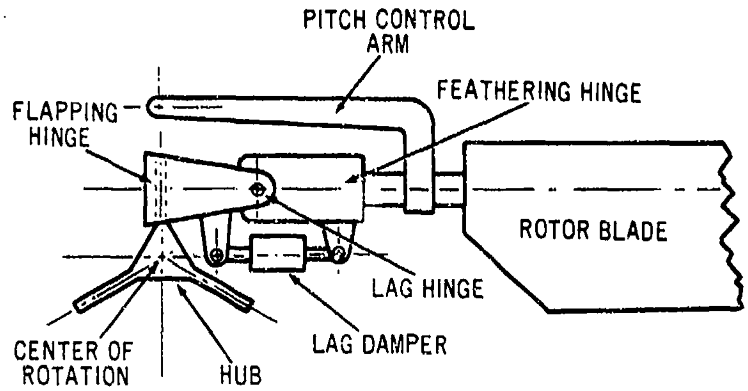


Fig. 3-58. Fully Articulated Rotor With Central Flapping Hinge

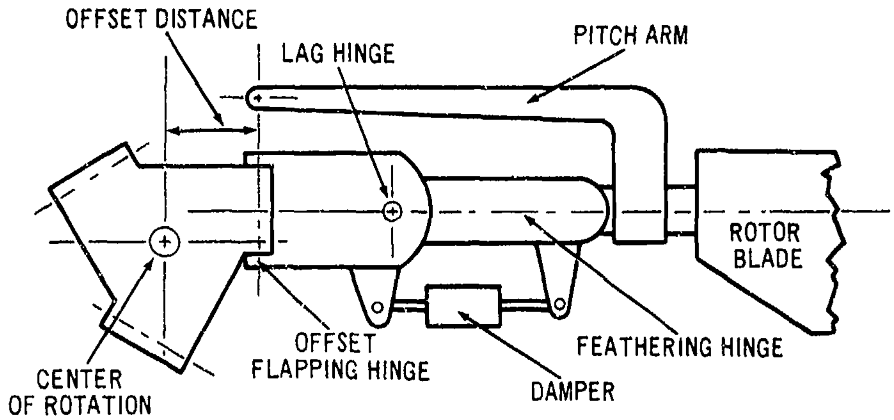


Fig. 3-59. Fully Articulated Rotor With Offset Flapping Hinge

1. Vertical control. This control allows the vertical position of the aircraft to be selected at will. The usual technique used is to change the thrust of the rotor by changing the pitch (angle of attack) of the rotor blades. An increase in pitch leads to an increase in thrust if constant rotor speed is maintained. Some designs have tried to keep the blade pitch fixed and to vary the thrust by varying engine speed, but none of these designs have advanced to full production because of the limited response available with fixed pitch systems.

2. Yaw or directional control. This control allows the pilot to select a prescribed heading by providing for rotation of the aircraft about a vertical axis that is fixed relative to the aircraft. The most common schemes for obtaining yaw control are use of a tail rotor in single-main-rotor machines, the differential tilting of rotors in helicopters with tandem or laterally disposed rotors, and use of differential torque in helicopters with coaxial rotors. In forward flight aerodynamic surfaces also can be used, but in low-speed flight they are ineffective.

3. Longitudinal control. In this control, pitching and fore and aft translation are coupled together. The rotor produces both a horizontal force component and a moment tending to pitch the fuselage. As the aircraft tilts in the desired direction, the rotor tilts in that direction also, and an additional force is exerted in the desired direction. In a helicopter having a single rotor, the control forces and moments are generated by that rotor. In a tandem configuration the large displacement between rotors allows the use of both differential thrust and individual rotor tilt to achieve pitching of the aircraft for control.

4. Lateral control. This control allows the pilot to move the aircraft horizontally in a direction perpen-

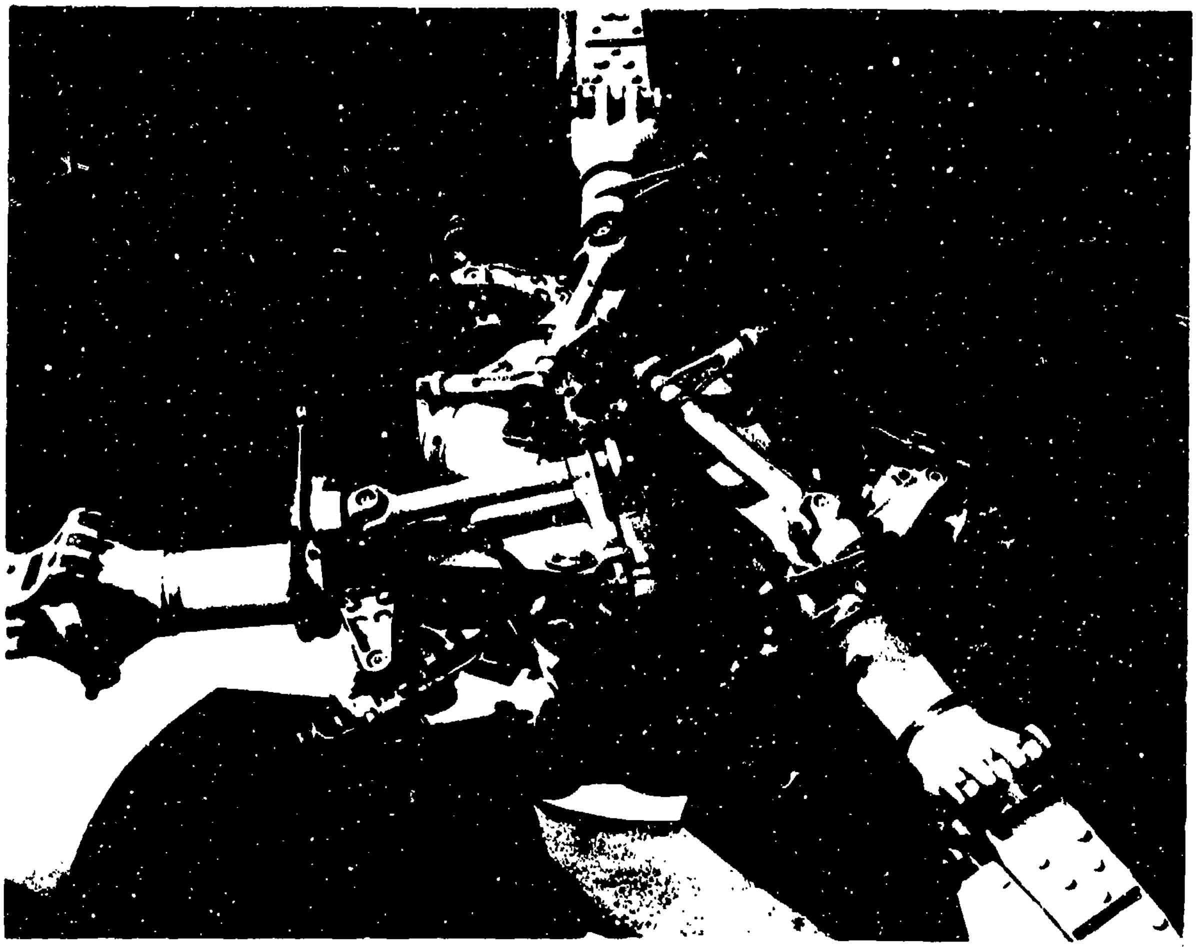


Fig. 3-60. Photograph of Fully Articulated Rotor With Separated Hinges

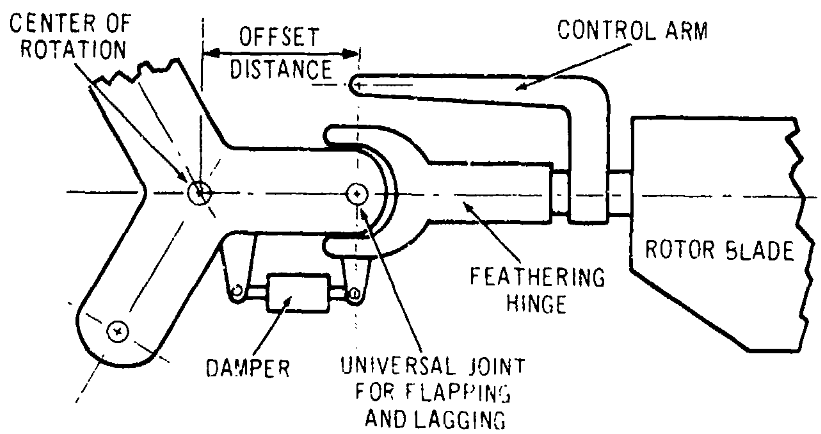


Fig. 3-61. Schematic of Fully Articulated Hub With Coincident Hinges

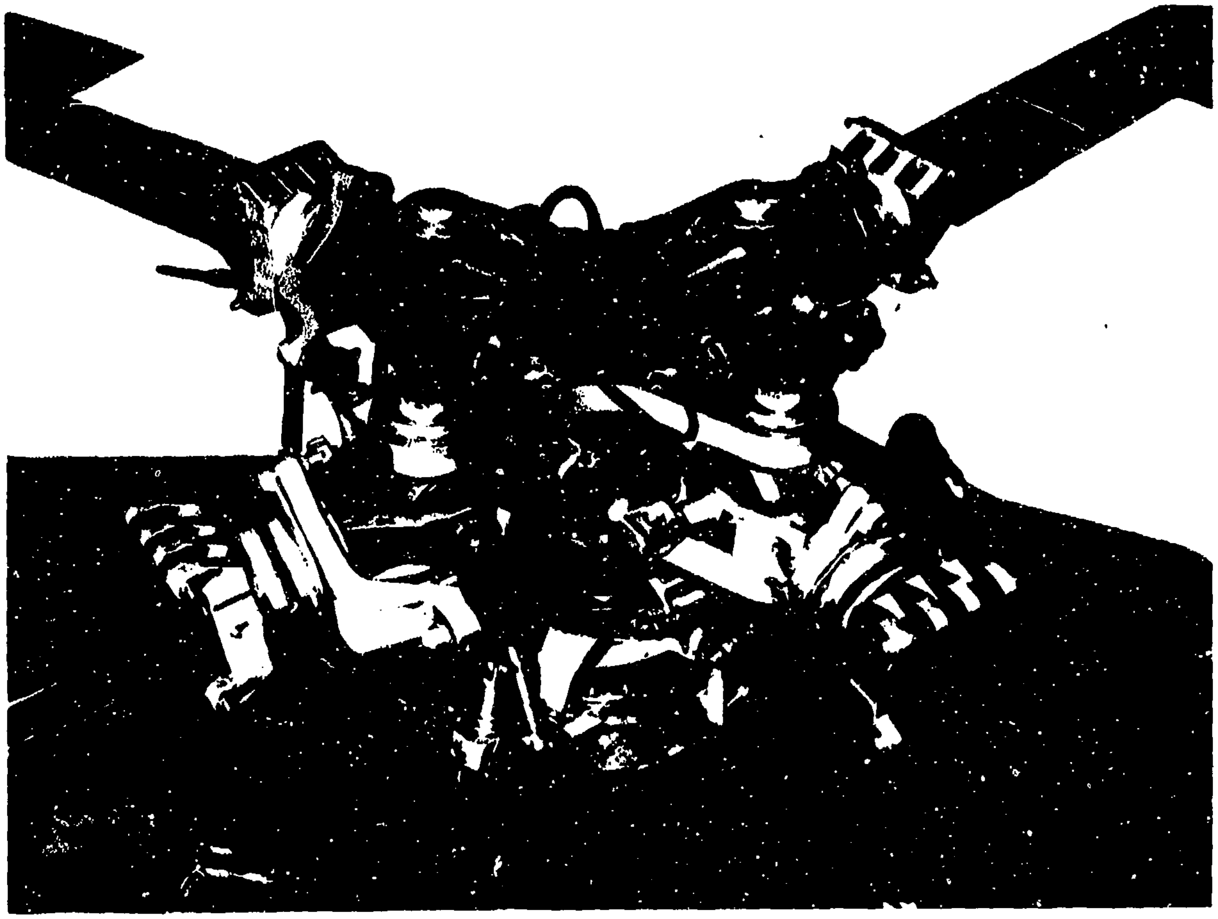


Fig. 3-62. Photograph of Fully Articulated Rotor With Coincident Hinges

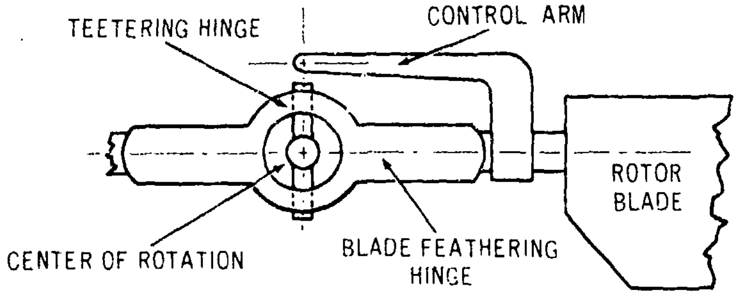


Fig. 3-63. Schematic of Two-bladed Semirigid Rotor

dicular to the fore and aft motion. It is similar to the longitudinal control in that rolling action and lateral translation are coupled together to achieve the desired aircraft response. The techniques used to achieve lateral control are the same as discussed for longitudinal control except that the laterally disposed configuration is analogous in this case to the tandem longitudinal control.

Lateral and longitudinal control through the rotor usually are achieved by producing moments about the rotor hub, by tilting the rotor thrust vector, or by a combination of the two. If the rotor of a helicopter is tilted at an angle with respect to the fuselage, a change in the resultant thrust vector results and forces and moments are exerted upon the fuselage. The action of this type of control is illustrated in Fig. 3-67. If the rotor tilts, a horizontal force component H exists that can accelerate the aircraft, and a moment is created

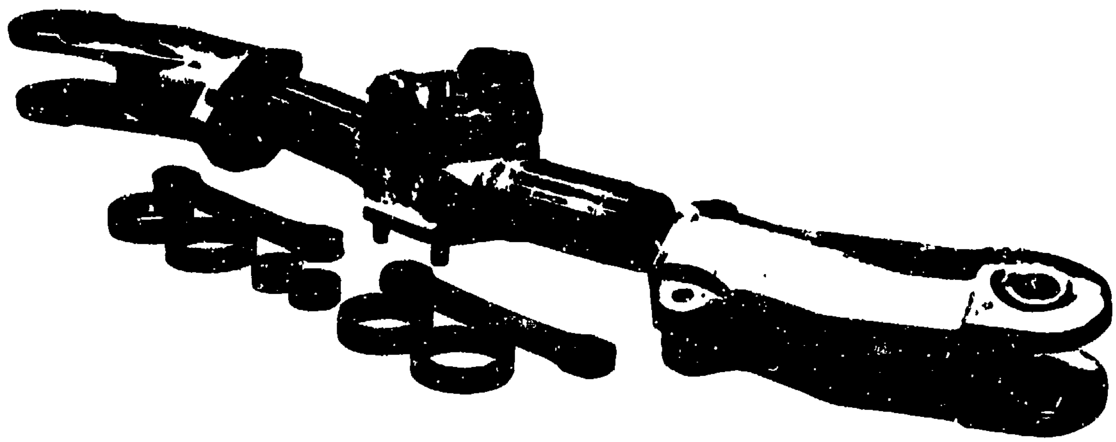


Fig. 3-64. Photograph of Two-bladed Semirigid Rotor

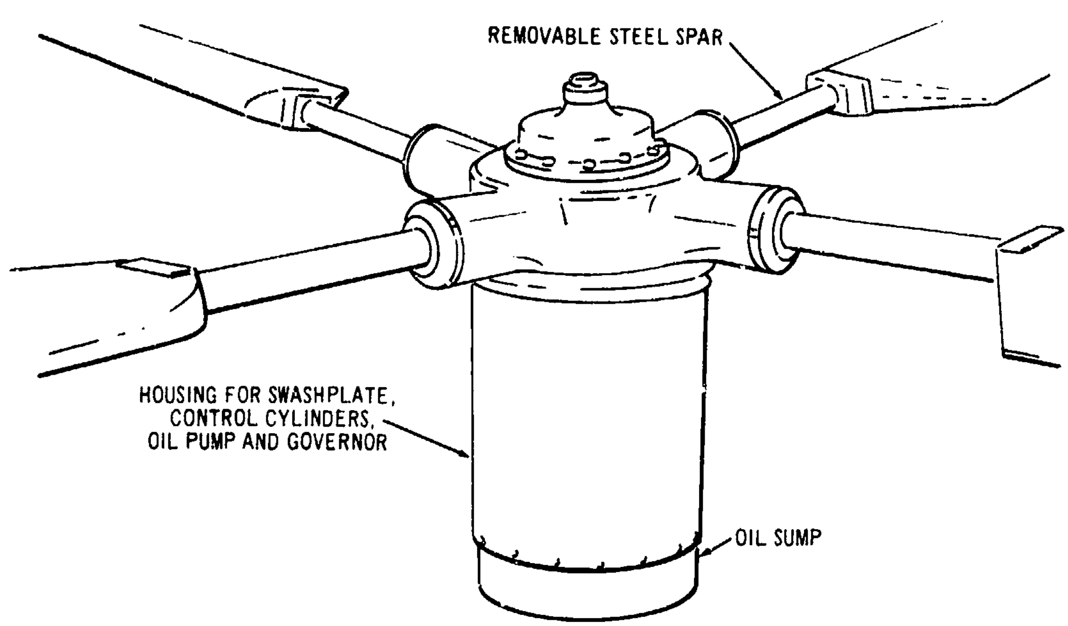


Fig. 3-65. Floating Hub Rotor

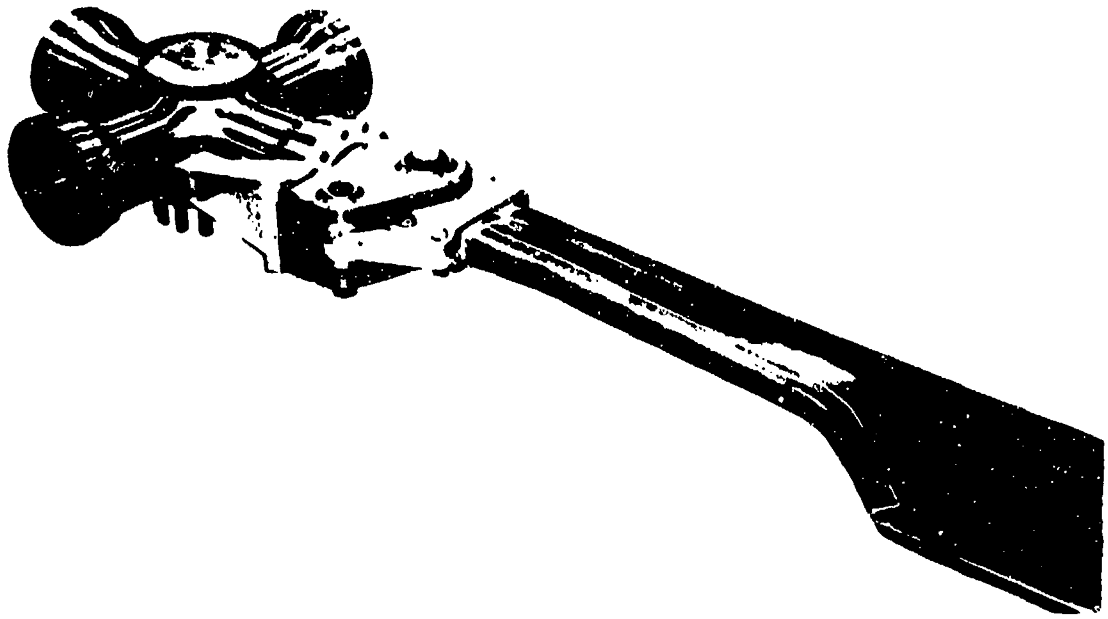


Fig. 3-66. Modern Hingeless Rotor

about the aircraft CG that can cause the fuselage to tilt in response to the rotor tilt.

Two general means exist for obtaining the necessary effective rotor tilt. If the rotor is of the semirigid or floating hub type, it is possible actually to tilt the hub body in the direction desired. Aerodynamic forces will cause the blades to follow the hub and the tip path plane will tend to line up parallel to the hub. This type of control is called direct control and has found use only in small autogyros and jet-driven rotors in special cases.

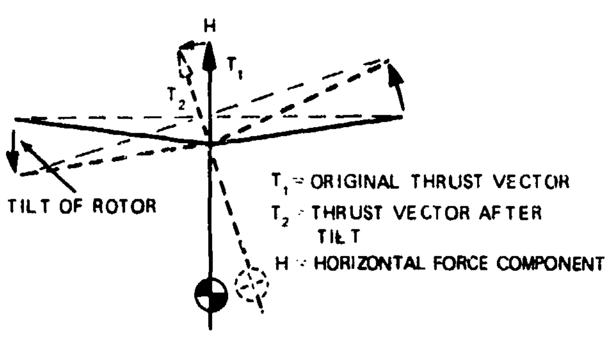


Fig. 3-67. Tilt of Rotor Thrust Vector

The more common technique is to vary the pitch or angle of attack of each blade cyclically once per revolution. This varying pitch technique is called cyclic pitch control. As the pitch is increased periodically in a given azimuth region, increased lift forces are generated that cause moments to be exerted upon the blades. If the blades are articulated, they flap in response to these moments. Because the resultant flapping occurs once per revolution, an effective tilting of the whole rotor tip path plane occurs and the resultant thrust vector tilts in response. If the rotor is rigid, the lift moments on the blades resulting from cyclic pitch cause a precession of the rotor to a new tilted position. Because the rotor is connected rigidly to the transmission and fuselage, the precessional actions must carry the fuselage as well as the rotor to the new position. It should be noted that in almost all modern rotors, blade pitch can be controlled directly by the pilot.

3-3.3.1.4 Elements of Rotor Motion

From an aerodynamic and dynamic standpoint, the ideal rotor is a rigid one with an infinitely stiff structure. In such a system the applied air loads and dynamic loads cause no deflections of the system and the problems of dynamic response disappear. However, as has been found repeatedly by designers who have worked with rigid or even semirigid rotor systems, the components are not infinitely stiff; instead, varying degrees of flexibility are inherent in their design. This flexibility results in components that respond dynamically to the various loads imposed upon the b...des.

To illustrate this point, consider the flapping motion (or flapping degree of freedom) of one blade attached to a hub structure that is attached rigidly to the rotor mast. A given blade can be attached to the hub by several means:

1. A flapping hinge at the centerline
2. An offset flapping hinge
3. A flapping hinge with a spring to restrain the blade flapping about the hinge
4. A spring strap retention or flexural hinge replacing the bearings
5. A stiff flexural hinge
6. Direct attachment to the hub without any hinges or separate flexures (rigid or hingeless rotor).

The listing of attachment methods merely indicates a relatively continuous spectrum of increasing stiffness of blade mounting. Because the blade is the same regardless of the attachment method, maximum resistance to motion will come when the blade is mounted directly to the hub and the least resistance will come when the simple flapping hinge is used. The other methods provide varying degrees of stiffness. The blade attached directly to the hub will bend under the load and tends to act as an equivalent flexure hinge itself. Consequently, the demarcation between the flexural mountings and the rigid rotor becomes indistinct. None of the configurations actually is rigid for this would require infinite stiffness; none of the configurations is completely without stiffness because centrifugal force acts as a powerful force to govern the flapping motion of the blades.

Each of these mounting configurations will have associated with it a natural frequency of flapping motion that will be dependent upon the stiffness of the mounting method. Thus, the primary difference among the various means of blade attachment lies in the variation over a limited range of the dynamic response of the rotor to the loads imposed upon it. Because the centrifugal restoring moment is so powerful, differences in blade mounting techniques produce relatively small effects upon the flapping response except in the case of small rotors, where high stiffnesses may be achieved.

This discussion, therefore, deals primarily with the motions and operation of the fully articulated rotor. This type is the most widely used, and it demonstrates most of the phenomena necessary for an understanding of rotor operation.

Fig. 3-68 depicts a simplified rotor as viewed from above. The rotor rotates in a counterclockwise direction with the downwind position as the zero degree point of reference.

To obtain a clear picture of the fundamentals of blade motion, consider a rotor made up of blade elements that consist of a single square paddle with a chord length c, a span length $\Delta r = 1$, and a mass $m_s = W_s/g$. It is assumed that the paddle supporting arm is hinged at the center of rotation and that the supporting arm introduces no inertial or aerodynamic forces. It also is assumed that the hinge axis lies in the plane of rotation and is perpendicular to the blade span axis.

For the purposes of the equation of motion, consider this simple blade element as rotating about a vertical axis in a vacuum, with no aerodynamic forces acting upon it. In Fig. 3-69, which shows the rotor from the rear, β_s is the flapping angle of the blade element. Summing moments about point O, we find

$$I_s\ddot{\beta}_s = -W_s r\cos\beta_s - \frac{W_s}{g}(\Omega^2 r) r \sin\beta_s \text{ , lb-ft} \quad (3\text{-}101)$$

where

I_s = blade element mass moment of inertia, slug-ft^2

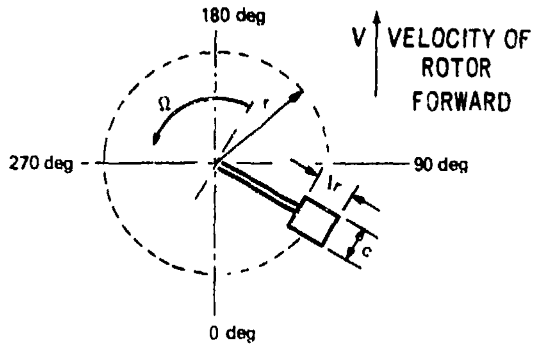


Fig. 3-68. Rotation of Blade Increment, Plan View

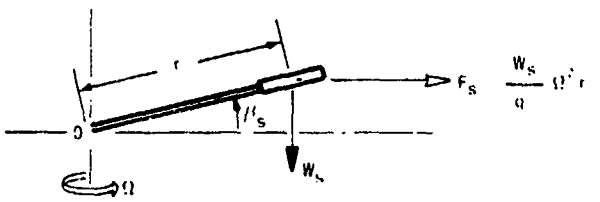


Fig. 3-69. Rotation of Blade Increment, Rear View

$\ddot{\beta}$ = flapping angular acceleration, rad/sec^2

W_s = weight of blade element, lb

For the very small angles of motion involved in ordinary rotor flapping motion, $\sin \beta_s \approx \beta_s$, and $\cos \beta_s \approx 1$; hence

$$I_s \ddot{\beta}_s + \frac{W_s}{g} \Omega^2 r^2 \beta_s = -W_s r \quad \text{, lb-ft} \qquad (3\text{-}102)$$

Because for the paddle element $I_s = (W_s/g)r^2$,

$$\ddot{\beta}_s + \Omega^2 \beta_s = -\frac{g}{r} \quad \text{, rad/sec}^2 \qquad (3\text{-}103)$$

This equation can be compared to the equation of a simple vertical spring mass system where

m = mass, slug

k = spring constant, lb/ft

x = displacement of the mass, ft

$\ddot{x}$ = linear acceleration of the mass, ft/sec^2

The equation of motion for this system is

$$m\ddot{x} + kx = mg \quad \text{, lb} \qquad (3\text{-}104)$$

or

$$\ddot{x} + \left(\frac{k}{m}\right) x = g \quad \text{, ft/sec}^2 \qquad (3\text{-}105)$$

The most important portion of this equation is the homogeneous portion given by

$$\ddot{x} + \left(\frac{k}{m}\right) x = 0 \quad \text{, ft/sec}^2 \qquad (3\text{-}106)$$

The solution leads to the natural frequency ω_n of the vibrating mass which is

$$\omega_n = \left(\frac{k}{m}\right)^{1/2} \quad \text{, sec}^{-1} \qquad (3\text{-}107)$$

By direct analogy the natural vibrating frequency ω_n of the flapping paddle blade can be obtained from the coefficient of β_s in the equation. Hence, for the blade case

$$\omega_n = \Omega \quad \text{, rad/sec} \qquad (3\text{-}108)$$

or the natural frequency of the paddle blade is equal exactly to the rotational speed of the rotor.

It can be shown readily that this result is true for any articulated rotor freely hinged for flapping at the centerline of rotation. This means that the rotor blade in such a system requires essentially no force to cause it to flap once per revolution. Because once-per-revolution flapping is actually a tilting of the rotor disk, it follows that little control force is required to tilt the rotor thrust vector in the direction desired for aircraft control.

The foregoing analysis of flapping motion is a greatly simplified summary of the operation of an actual flapping rotor. As discussed previously, all other rotor configurations tend to increase the flapping stiffness above this fundamental value. The added stiffness that can be achieved with most rotors, however, falls in a range such that the natural frequency of the lowest flapping or flap bending mode tends to be just slightly above rotor speed at the operating point. Typically, for a uniform blade mounted without hinges to rigid hub, the lowest flapping frequency is only approximately 20% above the frequency of the same blade hinged at the centerline of rotation.

3-3.3.1.5 Characteristics of Rotors

Each of the various rotor configurations presented has certain unique characteristics that tend to recommend it for particular operating conditions.

3-3.3.1.5.1 Fully Articulated Rotors

This type of rotor allows a great amount of flexibility in design. For example, the number of blades may range from three up to eight or more, and to rotor diameters in excess of 100 ft. The individual blade flapping hinges insure that the moments in the blade structure reduce to zero at the hinge; thus, bending moments in both the blade root attachments and the hub structure are reduced by the presence of the hinge. The individual hinges allow the blades to flap freely in response to changing gross weight, maneuver, or gust load conditions. In the case of pullup maneuvers, as the thrust of the rotor increases, the blades will cone up to a new position, yet the bending moments at the flapping hinges will remain essentially zero.

In addition, it is possible to use a wide range of hinge configurations to accomplish a specific design objective. For example, offset of the flapping hinge from the center of rotation, as illustrated by some of the preceding figures, provides for a large increase in control power exerted by the rotor. The moments exerted on the fuselage by the tilt of the rotor thrust vector are augmented by the moments created by the centrifugal force components acting at the offset hinges. It is possible to tilt the hinges so that blade flapping and pitch change are coupled together to achieve certain desired results. It is also possible to incorporate pitch cone coupling which reduces blade collective pitch whenever the total thrust of the rotor tends to increase.

The primary advantage of the fully articulated rotor system is its versatility. It can be used with a wide range of sizes, gross weights, and numbers of blades, and a variety of hinge configurations.

Although many advantages accrue with the use of the fully articulated rotor, it has certain inherent drawbacks. When a rotor blade flaps, its center of mass moves radially in the centrifugal field created by the rotation of the rotor. This radial motion gives rise to Coriolis accelerations that lead to large forces in the inplane direction of the rotor blades. These flapping-induced inplane forces in turn can lead to high bending moments in the chordwise direction near the rotor hub. To alleviate these high moments, the lag hinges are located outboard from the center of rotation and allow the blades to oscillate through a small angle around the hinge. Although these lag hinges reduce the bending moments induced by the Coriolis accelerations, another serious problem arises when blade chordwise freedom is allowed. It is possible for the lead and lag motions of the individual blades to couple with motions of the fuselage and landing gear that can lead to a destructive instability known as "ground resonance" (par. 5-2.5). To prevent destructive ground resonant oscillations, it is necessary to add lag dampers around the lag hinges to limit the lead-lag oscillations by absorbing their energy.

The combination of flapping hinges, lag hinges, lag dampers, and the associated bearings, spindles, housings, retainers, plus the lubrication provisions, leads to considerable complexity in the rotor hub of a fully articulated rotor. In addition, because of the arrangement of the hinges and the blade motions associated with the hinges, the pitch control mechanisms for these rotors become more complex in order to prevent unwanted coupling between blade motion and pitch change. Thus, complexity is the major drawback of the fully articulated rotors.

3-3.3.1.5.2 Two-bladed Teetering Semirigid Rotors

The two-bladed semirigid rotor is used widely because of its inherent simplicity and relatively low cost. In the configuration most commonly used, the rotor hub structure is mounted upon a set of bearings whose axis is normal generally to the blade span axis. The two blades are mounted on the hub by means of pitch change bearings. Because the blades are mounted directly to the hub, they can flap in unison in a manner similar to a teeter-totter, and hence this type often is referred to as a teetering rotor. The teetering action of the blades allows them to perform first harmonic or once-per-revolution flapping in a completely satisfactory manner.

Because forward flight air loads are complex, rotor blades are loaded at many higher frequencies that are multiples of rotor speed. In the articulated rotor, the blades can flap in response to these higher frequencies. In the teetering rotor, the blades cannot flap individually; and as a consequence, these higher frequency loads introduce bending moments into the blade root and hub structure.

Two-bladed teetering rotors usually are designed so that the blades have a fixed coning angle. The coning angle of a rotor is the average angle between the blades and a plane perpendicular to the axis of rotation. The coning angle of a hovering rotor is shown in Fig. 3-70. The lift and centrifugal forces tend to move the blade upward to a position of equilibrium. If a hinge is incorporated for each blade, each blade will turn through the angle β_0. This average angle is called the coning angle because in hover the rotor blades tend to form a cone. The blades of a two-bladed teetering rotor are attached to the hub with a builtin value of coning angle β_0. This average coning angle counteracts the steady lift moments of the hub quite effectively for a particular design thrust and rotor speed. If, however, the thrust of the rotor is changed drastically due, for example, to a sudden takeoff or large overload, the bending moments in the hub can increase greatly. This is because the two blades are attached rigidly to the hub and hence cannot cone further upward to alleviate the increased loading conditions.

Because the rotor assembly is mounted upon a single teetering axis, it is possible to generate oscillating mo-

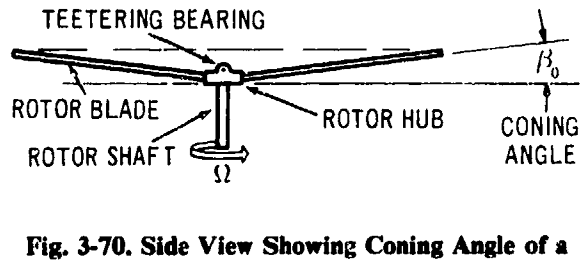


Fig. 3-70. Side View Showing Coning Angle of a Hovering Rotor

ments in the drive shaft as the blades flap in response to forward flight loads. The effects of such oscillating moments in both the transmission system and the rotor hub bearings must be considered.

The flapping bearings of a two-bladed teetering rotor essentially are unloaded because the centrifugal force of one blade passes through the hub structure and directly to the opposite blade. Another advantage of such a rotor is that it can be designed so as to preclude the occurrence of ground resonance instabilities through use of blades that have a great deal of stiffness in the chordwise direction. The added stiffness requires added weight, but the result is a very durable and rugged rotor system.

The size and weight ranges of the teetering two-bladed rotor have been relatively limited. Typically, rotor sizes have ranged from 15 ft up to more than 50 ft in diameter and aircraft gross weights from a few hundred pounds to somewhat over 10,000 lb. As these rotors get larger, their dynamic vibration and weight problems tend to increase.

3-3.3.1.5.3 Modified Two-bladed Rotors

Several variations of the standard two-bladed teetering rotor have been studied and developed. In most versions an attempt is made to allow the coning angle of the rotor to change under load. One method is to incorporate a central coning bearing assembly that is separate from the teetering bearings. Thus, as the load changes, the blades cone to a new equilibrium position and the blades flap about the teetering hinge. Other types use a common teetering-coning hinge that accomplishes the same result. Another variation was used on the 135-ft-diameter, two-bladed XH-17 helicopter rotor, which was driven by pressure jets at the tips of the blades. In this configuration, each blade was mounted onto the hub by means of flexural straps that acted as hinges and allowed the blades to cone or flap as necessary under load.

Most of these variations of the simple teetering rotor result in a reduction in the moments induced due to blade loading changes and, as a result, allow these rotors to be used over a much larger range of sizes and gross weights. However, some of the dynamic and structural weight problems associated with chordwise actions still remain.

3-3.3.1.5.4 Rigid or Hingeless Rotors

The primary feature of the rigid or hingeless rotor is the absence of both the flapping and the lag hinges. Blade feathering bearings still are used, however, to permit the collective blade pitch change necessary to vary rotor thrust and obtain aircraft control.

Elimination of the flap and lag hinges leads to a considerable simplification in the design of the rotor hub because the many bearings, spindles, housings, and seals associated with these bearings also are eliminated. Because the blades can carry moments directly into the hub and then directly into the rotor mast, hingeless rotors can develop a large amount of control power. The added moments can be used to achieve more rapid control response from the helicopter than normally would be associated with an articulated rotor. The elimination of the hinges also results in a simplification of the rotor control system, because many of the links and elements found in an articulated rotor are eliminated or reduced in size.

Hingeless rotors have some inherent drawbacks, however. Because the blades are connected directly to the hub and the hub is connected directly to the rotor mast, any changes in thrust loading or air loading will produce bending moments in the blades and hub structures. Also, a mean coning angle is built into the hub attachment, as with the teetering, two-bladed rotor. Thus, a sudden pullup can lead to high stresses in the blade roots and hub. In addition, all the harmonic loadings above the first will tend to cause high blade and hub stresses, and operation at gross weights or rotor speeds other than the design values will increase the average moments in the hub region. The net result of these increased loads is that fatigue stresses can become very high, and therefore the useful life of the hub may be relatively short. To reduce the stresses induced by these loadings, it often is necessary to increase the stiffness and weight of the blade roots and hubs, and the resultant rotor weight penalties can become very large. The alternative to increasing hub stiffness and weight has been to reduce stiffness in the flapping direction. The flexure thus provided will alleviate high stresses and still result in a relatively simple configuration, but the control power will be reduced somewhat.

Ground resonance instabilities can be avoided in hingeless rotors in a manner similar to that used in teetering rotors. A high degree of chordwise stiffness is required such that the lowest natural chordwise frequency is above the rotor speed under operational conditions. Rotors without lag hinges have been operated with chordwise frequencies below operating rotor speed, but such an approach must be pursued only with caution and a comprehensive test program.

The floating hub rotor combines features of the hingeless rotor hub system with the tilting feature of the teetering rotor. In this design, the hingeless rotor is allowed to tilt in any direction on the rotor mast in the manner of the first-harmonic flapping of an articulated rotor. Thus, the floating hub rotor retains the simplicity of the hingeless rotor and the transient first harmonic stresses on the hub are alleviated. Because a builtin coning angle is used and because the blades cannot flap in response to higher harmonic loadings, the floating hub rotor can develop high moments and stresses like the hingeless rotor. The floating hub rotor also must be designed carefully so as to prevent the occurrence of ground resonance instability; and because the rotor hub assembly is free to tilt and perform first harmonic flapping readily, its control power is reduced to that found in ordinary articulated rotors with a small offset in the flapping hinge. But an increase in control power can be achieved by incorporating spring restraint in the tilting hub.

3-3.3.1.6 Other Rotor System Considerations

The means employed to counteract the driving torque necessary to turn the rotor always has been a governing characteristic in helicopter design. To provide an efficient lifting system with a minimum amount of installed power plant weight and cost and a minimum fuel consumption rate, the rotor diameter must be made very large in comparison to an ordinary propeller. It also is necessary to design the rotor so that blade tip velocities remain somewhat below sonic speed. These characteristics of large diameter and limited tip speed result in the large, slow-turning rotors common to today's rotary-wing aircraft.

Since torque Q can be expressed as

$$Q = \frac{P}{\Omega}\ , \text{lb-ft} \qquad (3\text{-}109)$$

it is evident that the slow turning rotors result in high values of torque. As shaft torques increase, transmission size, weight, complexity, and cost increase as well. To alleviate the effects of severe torque loadings, tradeoffs must be made among the various conflicting design requirements so as to obtain a good overall balance in the final configuration.

Because of the severe torque loading considerations, many seemingly unrelated rotor configurations have been suggested, studied, and developed, and a few have reached production status. Configurations employing shaft power include the following:

1. Single main rotor:
   a. Single rotor with antitorque tail rotor in rear of aircraft
   b. Single rotor with laterally disposed antitorque rotor
2. More than one main rotor:
   a. Two rotors coaxially mounted and rotating in opposite directions
   b. Two rotors displaced laterally to the fuselage and rotating in opposite directions
   c. Two rotors mounted in tandem and rotating in opposite directions
   d. Two rotors, rotating in opposite directions, whose blades intermesh with each other by varying amounts. Such rotors can be arranged either laterally or in tandem.
   e. Various numbers of rotors greater than 2, including 3-, 4-, and 6-rotor configurations.

Use of jet-driven rotors stems from the desire to eliminate the torque problem at its source. If the blades are driven by the reaction forces of jets located within the rotating system, e.g., on the blades, the only shaft torques passed through to the fuselage are those due to bearing friction and power takeoffs for auxiliaries. With jet drives, both the torque problem and massive transmissions are eliminated. There are, however, many other difficulties associated with the optimum use of jet drives; and although many jet drives have been proposed and attempted, no purely jet-driven rotor configuration has had sustained production. The most important drawback to their use has been a relatively high fuel consumption rate that has limited them to short-range, special-purpose missions. New technology in turbines and materials, however, may allow reconsideration of these designs in the future. Some typical jet-driven rotor configurations include:

1. Tip-mounted engines:
   a. Ram jet engines
   b. Pulse jet engines
   c. Gas turbines
   d. Rockets
2. Pressure jet systems:
   a. Compressed air to tip jet
   b. Compressed air to tip afterburner
   c. Turbofan exhaust to tip jet
   d. Turbojet exhaust to tip jet.

The use of engines mounted at the blade tips is beset with problems such as those resulting from centrifugal force and control loads. The pressure jet configurations suffer from blade design limitations and duct losses. Maximum engine installation simplicity is achieved through the use of the "hot cycle" systems of Items 2c and 2d.

3-3.3.2 Propeller Selection

3-3.3.2.1 Propeller Types

Throughout the history of propeller-driven aircraft, the most widely used propeller configuration has been the conventional free-air, variable-pitch type, usually incorporating from two to four rigidly retained blades. For special applications, several other propeller concepts show promise. These include the shrouded propeller, the variable-camber propeller, the variable-diameter propeller, the multibladed prop-fan with and without shrouds, and the rotor-prop. Each of these concepts offers specific characteristics to meet particular aircraft requirements.

Adding a shroud to a conventional propeller permits performance to be maintained with a reduction in diameter and affords substantial noise attenuation. For larger, high-speed aircraft, particularly those with VTOL capability, the variable-camber concept provides the best potential for improving overall performance by matching the propeller to both takeoff and high-speed level flight conditions. This is accomplished by pairing conventional blades such that in takeoff they function mutually as a slotted flap airfoil and in cruise as two independent, low-camber airfoils. Full-scale tests of this concept have shown reduced noise in addition to confirming the cambering effect of the paired blades under static conditions. Although at a much earlier stage of development than the variable-camber propeller, the variable-diameter propeller offers another way to improve the matching of takeoff and cruise performance. However, because of the increased airfoil thickness and limitations on blade twist distribution necessary to permit blade telescoping for reduced diameter, the high-speed efficiency of this concept inherently is below that of the variable-camber concept. Moreover, no noise reduction at takeoff can be anticipated.

It is well known that propeller noise can be reduced by operating at a low tip speed, provided that the blade area is sufficient to prevent stall. A low-tip-speed, multibladed prop-fan offers the potential of reduced noise and small diameter at performance levels comparable to a conventional propeller. Further advances in performance, noise reduction, and compactness may be obtained from a shrouded prop-fan that combines the favorable characteristics of the multiblades and the shroud.

For large VTOL craft incorporating a tilt-wing or tilt-rotor for transition from hover to level flight, the rotor-prop has certain attractions. The hinged retention permits large diameters with narrow, low-activity-factor blades at reasonable weights for high performance in hover and transition. Acceptable cruise performance can be obtained by large reductions in rotational speeds between takeoff and high speed flight. A disadvantage is that the success of this concept is dependent upon the availability of engines with large rpm spreads at a small loss in *SFC* between takeoff and high-speed conditions. Moreover, the high advance ratios in cruise and at V_{max} tend to reduce efficiency. This reduced efficiency may be offset by using very low activity factor blades, but at the expense of increased propeller weight.

3-3.3.2.2 Helicopter Application

Performance requirements for propellers for compound helicopters differ considerably from those normally associated with conventional propeller-driven aircraft. In the latter case, the propeller provides the thrust requirement for all flight regimes, including takeoff, climb, cruise, and V_{max}. Because of this broad operating spectrum, the aerodynamic design of the propeller becomes a comprehensive analysis effort. The final propeller configuration generally is that which meets the primary performance requirements at the expense of some compromise in off-design conditions.

The helicopter application does not involve this wide operating spectrum. With a compound helicopter, the propeller normally is used only for cruise, and accordingly it is optimized for maximum efficiency at the design cruise conditions. A second design condition results when the propeller is driven by the same engine(s) that drives the main rotor. Thus, during hover, where no forward thrust is required, the propeller is designed to absorb minimum power because each horsepower lost to the propeller results in a loss of 8-10 lb of thrust. Therefore, in some cases the propeller cruise performance might be compromised so as to attain minimum power in hover. Other considerations influencing propeller aerodynamic design include inflight reversing for air braking and the use of the propeller as an antitorque rotor by swiveling it about a vertical axis. Finally, the downwash of the main rotor on the propeller during hover and climb must be considered.

In summary, although the performance spectrum is narrower, the selection of propellers for helicopters involves several unique considerations not encountered in conventional aircraft propeller installations.

3-3.3.2.3 Analytical Procedures

Successful propeller design involves two prime considerations: attainment of the required aerodynamic performance and structural integrity. Thus, the success of the design effort is dependent upon the availability of reliable aerodynamic and structural design criteria. Accordingly, propeller manufacturers are devoting considerable effort to the development of the analytical design methods discussed briefly here.

The most reliable and widely used propeller performance calculation method is based upon an advanced form of the blade element theory. In this theory, the aerodynamic forces acting upon a series of radial elements are calculated and then integrated over the blade radius to establish the total forces. The vortex theory and Goldstein's solution (Ref. 43) for the radial distribution of circulation for a finite number of blades have now been applied to the blade element theory. Thus, an analytical method has been evolved that permits the accurate calculation of the efficiency of any arbitrary propeller configuration, operating at any imposed condition.

Over the years this method has been refined, along with the associated two-dimensional airfoil data, to the point where comparisons with experimental data on both full-scale and model propellers indicate an accuracy of better than $\pm 2\%$ in the vicinity of the design point and only slightly lower accuracy for off-design points. The method has been programmed on a digital computer from which up to 200 efficiency points per minute can be calculated. Included as options to this basic program are subroutines for calculating aerodynamic twisting moments on the blades and computing total and azimuthal distributions of air loads with inclined and nonsymmetrical inflows.

More recently, a noise subroutine has been added as an option based upon an extension of the work of Ref. 44. This program uses the propeller air loading distribution computed by the method outlined above and calculates from it both far and near field sound pressure levels in terms of decibels at any prescribed location for any propeller geometry and operating condition. If requested, the program also computes perceived noise levels (PNL) and effective perceived noise levels (EPNL) corrected for tone and time duration.

For tilt-wing VTOL aircraft, in which the propeller is in transition from hover to horizontal flight, a new method, based upon rotor theory, has been developed to compute the six moment and force components generated by the propeller during transition. The method can handle cyclic propellers as an option.

For shrouded propellers and shrouded prop-fans, the blade element calculation method is based upon the one-dimensional, inviscid, incompressible momentum theory from the work of Ref. 45. In order to incorporate shroud drag, an empirical correction has been evolved that is dependent upon the shroud exit area ratio and the free-stream Mach number.

Because an adequate theory has not been derived for calculation of reverse thrust and windmilling drag, an empirical method has been developed from test data to predict propeller performance for these off-design operating conditions with acceptable accuracy.

The main function of the propeller—to produce thrust at a minimum expense in power—must be accomplished with hardware that affords maximum reliability at minimum weight and cost. Thus, it is essential that structural design criteria and material development be commensurate in refinement with the aerodynamic criteria.

The design of a propeller system begins with blade definition involving both aerodynamic and structural considerations. The blade definition generates the steady and vibratory loads to which the propeller must be designed and consequently affects the design of the barrel, actuator, control, and gearbox. To aid the designer, many theoretical analyses and associated computer programs for obtaining optimum structures have been derived and developed. These programs cover both steady and vibratory stresses as well as many secondary structural aspects. A detailed discussion of these structural design tools and their application to hardware design is presented in AMCP 706-202.

3-3.3.2.4 Propeller Selection Procedure

3-3.3.2.4.1 Basic Considerations

In the propeller preliminary design phase, only aerodynamic sizing is carried out in detail. The other design considerations, particularly blade structure, usually are estimated on the basis of past experience and preliminary analysis merely to assure the feasibility of the aerodynamic selection.

The computerized strip analysis methods developed from the refined theory form the fundamental criteria used by the industry to design propellers for all applications. Moreover, their derivatives and the empirical methods reviewed are used to compute most of the aerodynamic data and to support structural design.

Recognizing the need for a more convenient propeller performance analysis method for use in preliminary design, the propeller industry has published generalized performance calculation manuals based upon the computer programs and propeller performance theory. These manuals, widely used throughout the aircraft industry, present static and inflight performance data covering a complete range of potential operating conditions for a family of propellers, with variations of the major geometric parameters. In addition to performance data, the manuals include methods for estimating propeller weight and far-field noise at zero airspeed.

One recommended set of such manuals (Refs. 46, 47, and 48) covers conventional propellers, shrouded propellers, and variable camber propellers, respectively.

The preliminary design procedure detailed subsequently is based upon the data included in Ref. 46 for conventional propellers. The discussion covers the concept of the process, definition of the basic performance and prime blade geometric parameters, use of generalized performance plots and other aerodynamic data, and a step-by-step procedure for propeller selection. All of the necessary aerodynamic data are presented in the series of charts included herein. To facilitate the demonstration of this method, a sample propeller selection problem for a hypothetical, representative helicopter is set up and the selection process is undertaken in detail.

3-3.3.2.4.2 Definitions

On a nondimensional basis, propellers are defined by number of blades, blade activity factor AF, and integrated design lift coefficient C_{L_i}

Activity factor AF was defined early in propeller technology as a power absorption factor. It is defined rigorously as:

$$AF = \left(\frac{10}{D}\right)^5 \int_0^R br^3\, dr \quad \text{, dimensionless} \qquad (3\text{-}110)$$

where

D = propeller diameter, ft
R = propeller radius, ft
b = elemental width, ft
r = elemental radius, ft

In nondimensional terms,

$$AF = \frac{10^5}{16} \int_0^{1.0} \left(\frac{b}{D}\right)\left(\frac{r}{R}\right)^3 d\left(\frac{r}{R}\right) \qquad (3\text{-}111)$$

Activity factor today is used as a weighted measure of the width distribution of the blade, e.g., a blade with a purely rectangular planform derives half of its activity factor from the outer 15% of the blade. The numerical weighting results in convenient values for the AF, normally between 50 and 150.

With the advent of laminar flow airfoil sections with unlimited section design lift coefficients independent of thickness ratio, the term integrated design lift coefficient C_{L_i} was defined as:

$$C_{L_i} = \frac{64}{D^4} \int_0^R c_{l_{des}} r^3\, dr \quad \text{, dimensionless} \qquad (3\text{-}112)$$

where

$c_{l_{des}}$ = blade element design lift coefficient, dimensionless

whence

$$C_{L_i} = 4 \int_0^{1.0} c_{l_{des}} \left(\frac{r}{R}\right)^3 d\left(\frac{r}{R}\right) \quad \text{, d'less} \qquad (3\text{-}113)$$

with this term being weighted similarly to activity factor.

Other parameters—such as those that define the aerodynamic shape of a blade, the thickness ratio, and the twist distributions—have some effect upon blade performance. However, their effects are equivalent to small changes in the activity factor and C_{L_i}, and therefore they are not used as prime variables in this method. These parameters, along with airfoil section choice, are included in the later optimum blade process.

The nondimensional coefficients advance ratio J, power coefficient C_P, and thrust coefficient C_T, used to determine propeller performance, are defined by

$$J = \frac{V}{nD} \quad \text{, dimensionless} \qquad (3\text{-}114)$$

$$C_P = \frac{P}{\rho n^3 D^5} = \frac{SHP(\rho_0/\rho)}{2003\left(\frac{N}{10^3}\right)^3\left(\frac{D}{10}\right)^5} \quad \text{, d'less} \qquad (3\text{-}115)$$

$$C_T = \frac{T}{\rho n^2 D^4} = \frac{T(\rho_0/\rho)}{6610\left(\frac{N}{10^3}\right)^2\left(\frac{D}{10}\right)^4} \quad \text{, d'less} \qquad (3\text{-}116)$$

where

V = true airspeed, fps
n = propeller speed, rps
N = propeller speed, rpm

P = engine power, ft-lb/sec
SHP = engine output shaft horsepower
T = propeller thrust, lb

The propeller efficiency factor η_P is defined as

$$\eta_P = \frac{TV}{P} = \frac{C_T J}{C_P}\text{, dimensionless} \quad (3\text{-}117)$$

and thrust is now stated as

$$T = \frac{550\ \mathrm{SHP}\ \eta_P}{V}\text{, lb} \quad (3\text{-}118)$$

3-3.3.2.4.3 Performance Calculation Method

A generalized performance calculation method for propellers operating at normal flight speeds is described. The form selected was governed primarily by the consideration of ease of usage and the elimination of the principal deficiency of existing empirical methods, i.e., the deterioration of accuracy at extreme operating conditions and blade geometries. Accordingly, the method incorporated a series of performance charts with each chart accurately defining performance for a specific propeller geometric configuration over the complete range of potential operating conditions.

It is good design practice on compound helicopters to select a propeller that does not involve compressibility. Fig. 3-71 presents the criterion for no compressibility loss in the form of the maximum allowable integrated design lift coefficient C_{L_i} as a function of aircraft Mach number and NDf_c, where f_c is the ratio of the speed of sound at sea level standard day to the speed of sound at the specific operation condition.

An ideal performance chart is provided for three-bladed propellers (Fig. 3-72). This example chart represents the performance of propellers with minimum induced losses and zero profile losses for a finite number of blades. An actual propeller design never can achieve optimum performance; but by careful tailoring for the design condition, it is possible to approach the optimum. Interpolation among charts that present a systematic variation of each major shape parameter will define performance for any desired propeller configuration. The performance charts provided for the generalized performance method, therefore, depict the variation of the power coefficient C_P with advance ratio J and efficiency η_P. Example data are presented for three-bladed propellers with given values of blade integrated design lift coefficients C_{L_i} and activity factor AF 0.3 (Fig. 3-73). Comparable charts for two additional values of C_{L_i} and three additional values of AF are included in Ref. 46, together with a comparable set of charts for four-bladed propellers. Selection of design parameters for a given specification requirement can be accomplished by crossplotting appropriate values from these charts. The data are limited to conditions where there is no compressibility, i.e., the sections are operating below their critical Mach numbers.

The performance charts are based upon a constant velocity through the propeller disk because blocking effects, i.e., the actual velocity through the disk, vary greatly with the aircraft geometry that is in close proximity to the propeller installation. These blocking effects have an influence upon the absolute values of performance; however, they usually do not change the comparative performance values of propellers of different designs. Therefore, the elimination of blocking effects from the procedure will not detract from the usefulness of the method for preliminary design propeller selection. In general, blocking has the same effect upon the optimum efficiency as upon the actual efficiency.

An empirical method, based upon actual propeller tests, is used to calculate the minimum shaft horsepower required to drive the propeller at full speed at static conditions and also for the calculation of reverse thrust. This method is referred to as the generalized thrust and torque chart method and uses the charts of Figs. 3-74 through 3-82. Although minimum horsepower required is sensitive to variables such as blade thickness, twist, camber, and width distribution, experience has shown that the empirical method suffices for preliminary design application. In the final blade optimization procedure, the minimum power figure is calculated by a refined strip analysis method. However, the reverse thrust performance still must be calculated with the empirical method because the strip analysis theory does not apply.

Performance requirements for compound helicopters vary. The airframe manufacturer usually will specify a maximum propeller diameter that will fit into his design envelope. Because a given engine is specified, the power available to the propeller and the thrust requirements also are known. At times the propeller rotational speed is specified; at other times it is undetermined because gearboxes must be designed to be mated to the main rotor. At this stage, consideration must be given to weight and noise as well as to performance. As the diameter becomes smaller, the activity factor also becomes lower, and the propeller system consequently will be lighter. A reduction in propeller tip speed generally will reduce the system noise level.

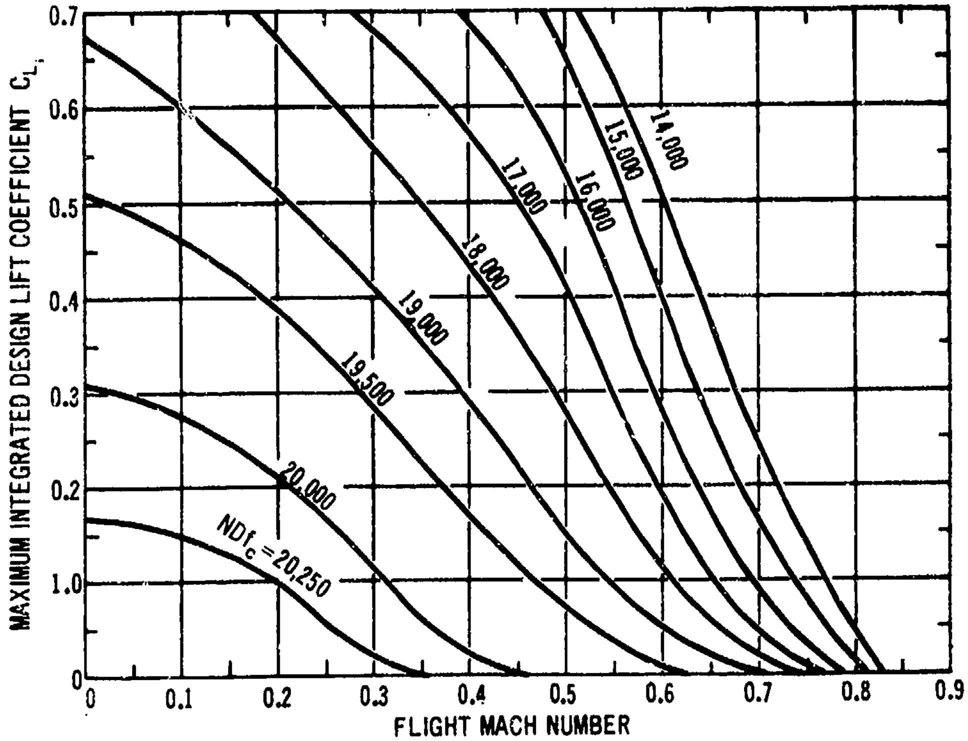


Fig. 3-71. Maximum Integrated Design Lift Coefficient To Avoid Compressibility Losses

A generalized weight formula that has proved to be fairly accurate has been developed over the years. The total system weight W in pounds is given by

$$W = K_1 \left[\left(\frac{D}{10}\right)^{1.85} \left(\frac{B}{4}\right)^{0.7} \left(\frac{AF}{100}\right)^{0.6} \times \left(\frac{ND}{20,000}\right)^{0.5} (M + 1.0)^{0.5} \left(\frac{SHP/D^2}{10}\right)^{0.12} \right] + K_2 (\text{Torque})^{0.24} \quad , \text{lb} \qquad (3\text{-}119)$$

where the portion containing K_1 is the propeller weight and the portion containing K_2 is the gearbox weight. The K factors are a function of propeller type as well as of materials. For compound helicopter applications, two propeller types can be considered. A conventional propeller—including barrel and blades, spinner, pitch actuator, pitch lock, deicing capability, feathering capability, control, and a gearbox—is employed in installations where the power input comes directly from the engine, which either is separate or is decoupled from the main rotor drive shaft. A shaft-driven propeller is used in installations where the propeller is slaved to the main rotor drive shaft. The latter propeller does not include pitch lock, feathering capability, deicing, or a control, and its structure is lighter. A gearbox may or may not be included.

3-3.4 OTHER PROPULSIVE DEVICES

As noted earlier, the pure helicopter is limited to forward flight speeds below 200 kt because of compressibility effects on the advancing rotor blade and stall effects on the retreating blade. Beyond that speed, the main rotor becomes ineffective aerodynamically. Faster forward speeds may be achieved by compounding.

The discussion of auxiliary propulsive devices focuses on the speed range above 150 kt. The addition of auxiliary propulsive devices to the basic helicopter design introduces several new factors to be considered in the system design. First, the propulsive device is selected from among four principal types: propeller, shrouded propeller, turbofan, and turbojet. Second,

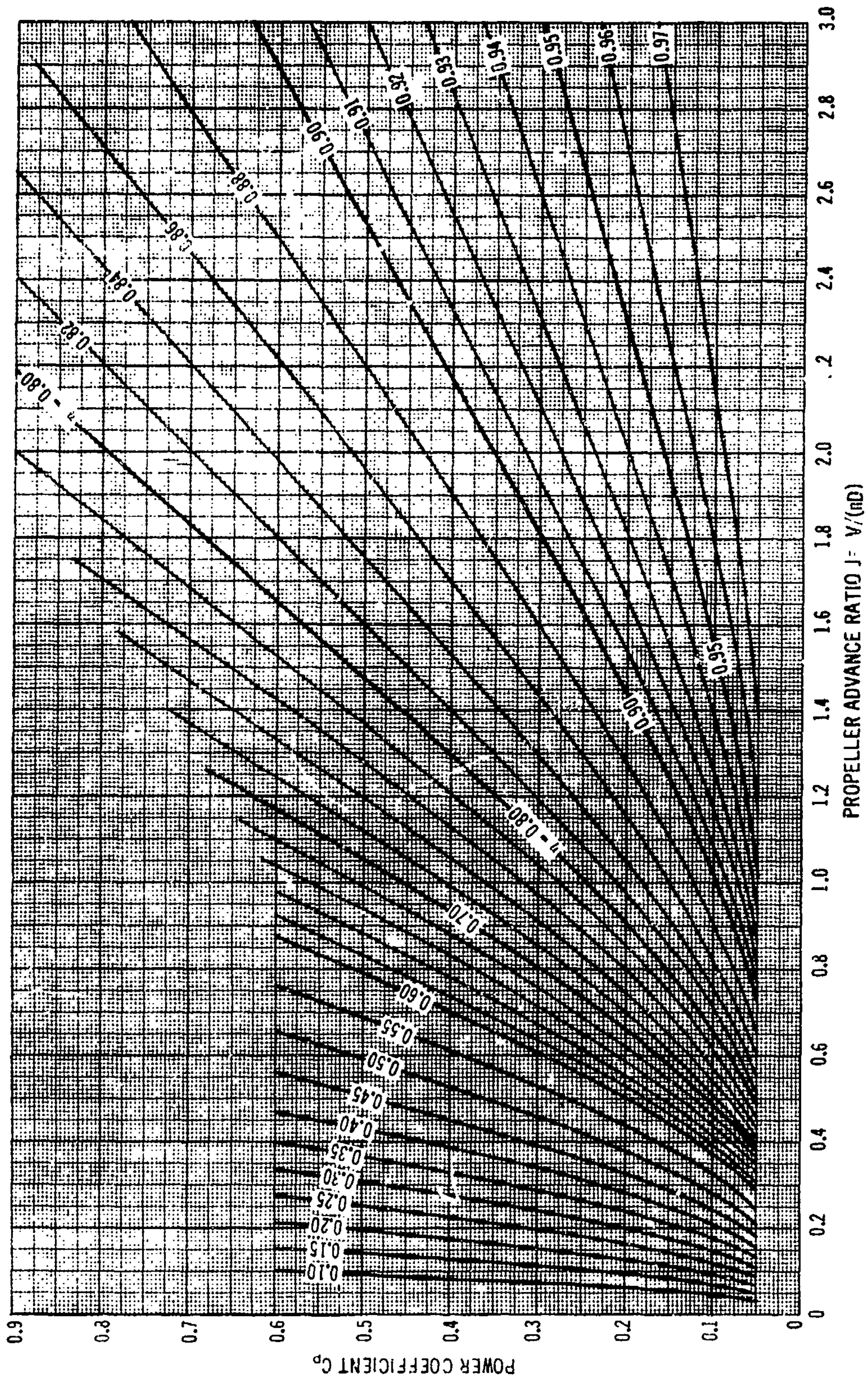


Fig. 3-72. Optimum Efficiency Chart for a Three-bladed Propeller

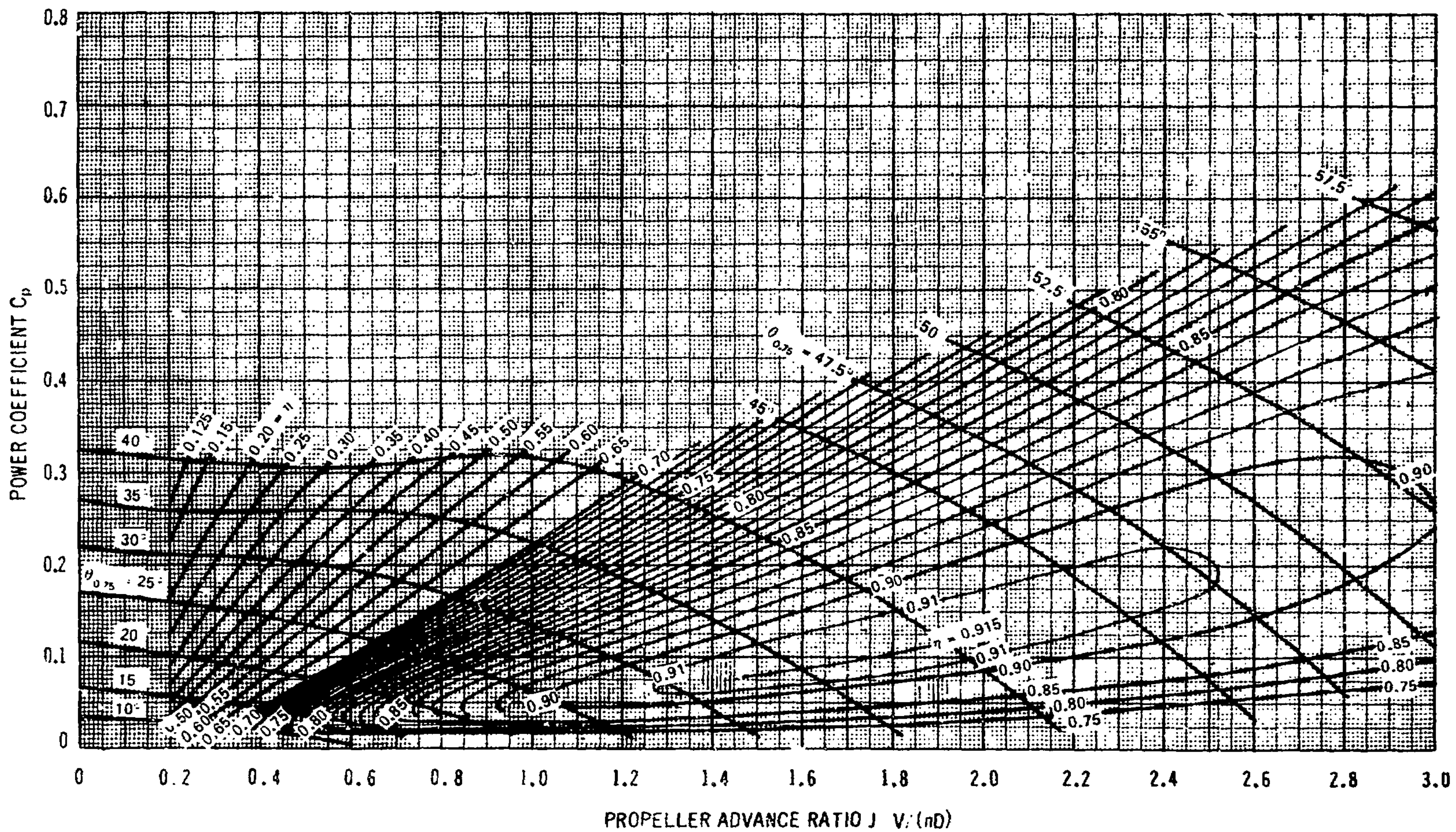


Fig. 3-73. Hamilton Standard Propeller Efficiency Chart for a Three-bladed, 100 Activity Factor, 0.3 Integrated Design Lift Coefficient Propeller

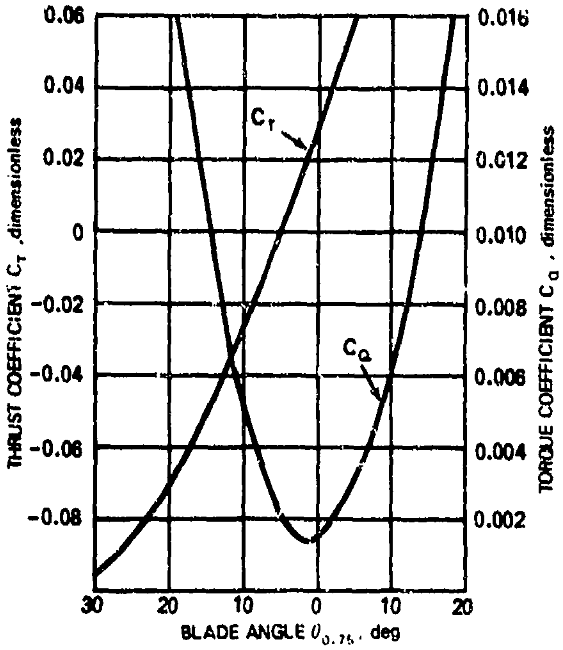


Fig. 3-74. Advance Ratio Equal Zero Portion of Generalized Torque and Thrust Charts

these devices introduce the possibility of using the devices for antitorque and directional (yaw) control as well as propulsion. Finally, the design of the main control system must accommodate the interfacing of the auxiliary propulsion device with the main rotor during transition from low to high flight speeds, and must provide the additional controls needed for operation of the propulsive devices alone.

Considering the ideal propulsive efficiency η_p of the different types of propulsive devices

$$\eta_p = \frac{TV}{P} = \frac{1}{1 + \frac{\xi}{2}}, \text{ dimensionless} \quad (3\text{-}120)$$

where

T = propulsive device thrust, lb
V = free stream velocity relative to the propulsive device (flight speed), fps
P = power transferred from the engine to the propulsive device, ft-lb/sec
V_j = effective exhaust jet velocity far behind the propulsive device, fps

$\xi = (V_j - V)/V$, dimensionless

the merit of the devices decreases in the order listed—propeller, shrouded propeller, turbofan, turbojet—in the speed range of interest because the exhaust velocities increase in that order.

The requirement to provide much of the total power aboard the vehicle in the form of shaft power for driving the main rotor under hover conditions, combined with the more favorable propulsive efficiency of propellers or shrouded propellers, favors the selection of these devices for auxiliary propulsion. Depending upon the application, however, other factors may lead to selection of one of the other types. For instance, weight, complexity, and reliability considerations may reduce the relative merit of propellers and shrouded propellers.

3-3.4.1 Propeller Propulsion

Propellers designed for primary propulsion in compound helicopters are similar in most respects to those for conventional aircraft. However, the propeller or rotor for antitorque control more closely parallels the main rotor in its operation. But even that parallel has

$$C_{Q_{eff}} = \left[C_Q \times (3/B)^{0.83} \times Q_{AF}\right] - \Delta C_Q'$$

$$Q_{C_{eff}} = \left[Q_C \times (3/B)^{0.83} \times Q_{AF}\right] - \Delta Q_C'$$

$$Q_C = \frac{Q}{\rho V^2 D^3} \quad (J \geq 1)$$

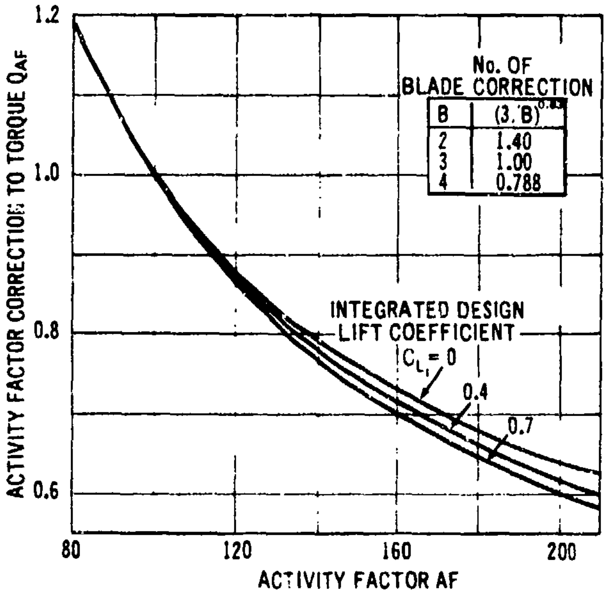


Fig. 3-75. Effect of Activity Factor on Torque Coefficient

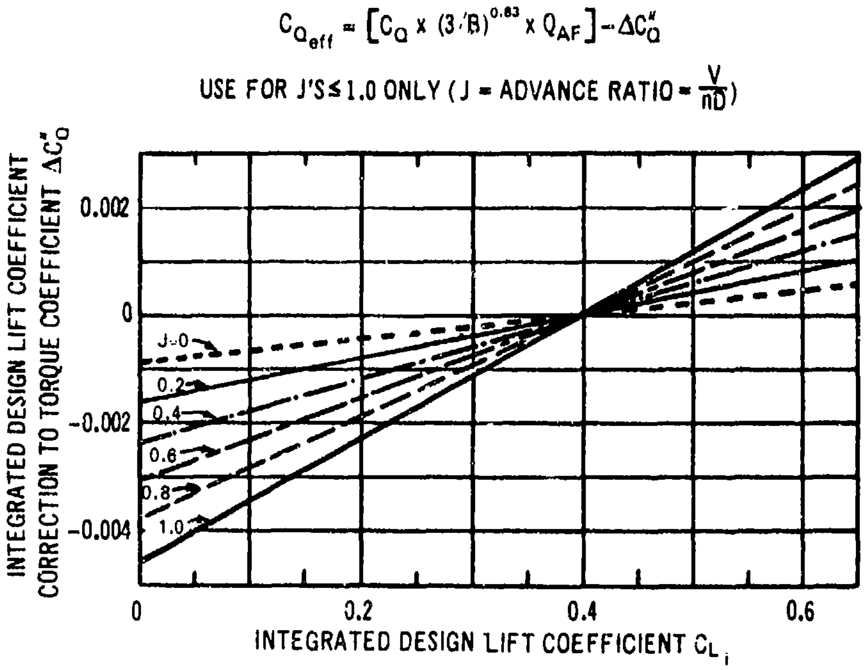


Fig. 3-76. Effect of Integrated Design Lift Coefficient on Torque Coefficient, J ≤ 1

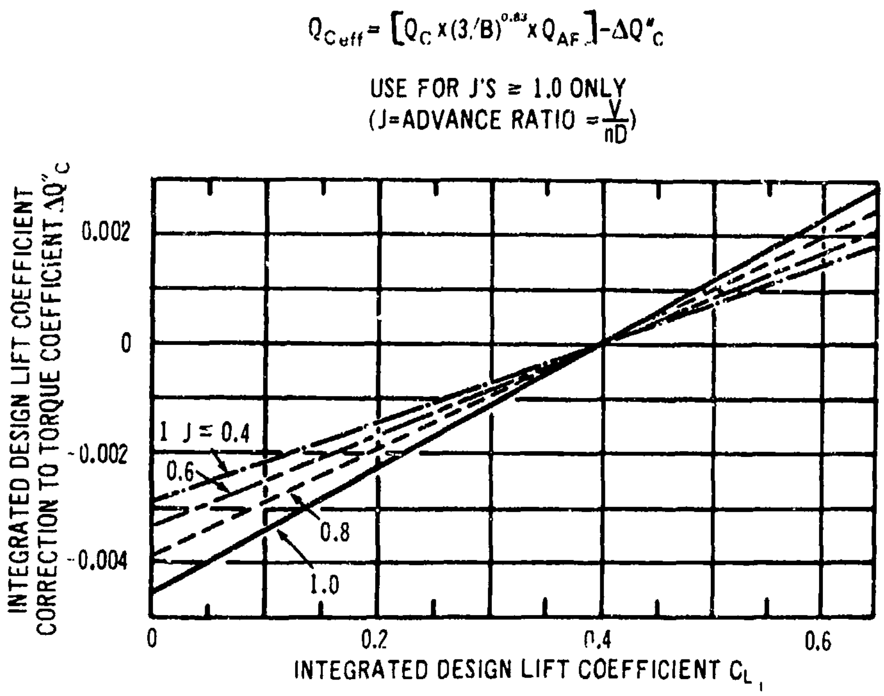


Fig. 3-77. Effect of Integrated Design Lift Coefficient on Torque Coefficient, J ≥ 1

severe limitations because of the complex range of flow fields within which the tail rotor may operate.

3-3.4.1.1 Primary Propulsion

Having decided to employ propeller propulsion for primary thrust in high-speed flight, the system designer must consider several factors. For example, the number of propellers to be used and their locations on the helicopter must be determined, and the drive train, its controls, and the propeller aerodynamic controls must be designed. In the selection of positions for the propeller(s), the most advantageous location from an aerodynamic viewpoint probably is forward of the main rotor slipstream because this allows a relatively smooth and predictable inflow. Another advantage of the forward location is the benefit accrued from the flow of the propeller slipstream over the wing surface. Analyses of such flows may be found in Refs. 49 and 50. Such factors as operational considerations and complexity and weight of the systems also may influence the location of the propeller(s). For example, if only one propeller is used for primary propulsion, it may be advantageous to combine the drive trains for the propulsive propeller and a separate tail rotor at the rear of the airframe.

The propeller(s) used for auxiliary propulsion may provide antitorque control and directional (yaw) control in the following ways:

1. Use of differential (cyclic) pitch control for a single propeller
2. Use of differential thrust for twin propeller configurations
3. Provision of turning vanes in the propeller slipstream
4. Swiveling the propeller about a vertical axis.

The power required for this function may be higher than that needed for a comparable tail rotor under hovering or low-speed flight conditions because the moment arm length is smaller in practicable configurations. In the case of a single propeller configuration, combining the functions of propulsion and control requires a significant compromise in the design of the propeller that would degrade cruise performance. Under high-speed flight conditions, fixed aerodynamic surfaces normally are provided to unload the tail rotor, thus reducing the power required.

3-3.4.1.2 Tail Rotor

The tail rotor of a conventional helicopter with a single rotor generally consumes 8-10% of the engine power under hovering conditions and somewhat less (3-4%) under forward flight conditions. Depending upon the design of fixed aerodynamic surfaces that may be incorporated, the tail rotor may be unloaded substantially at higher flight speeds. The aerodynamic flow

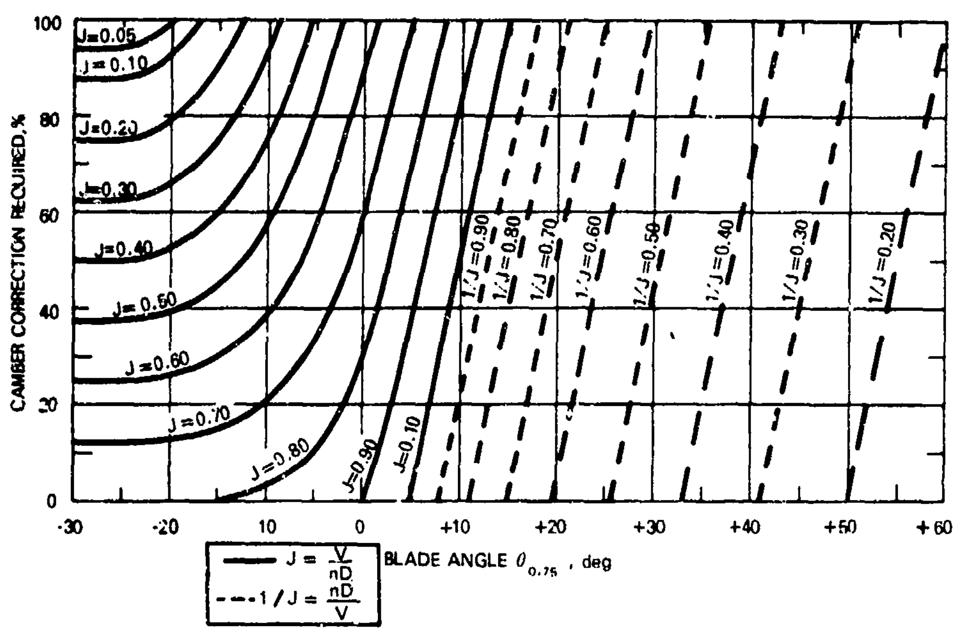


Fig. 3-78. Variation of Percentage of Camber Correction Required for Torque and Thrust With Advance Ratio and Blade Angle

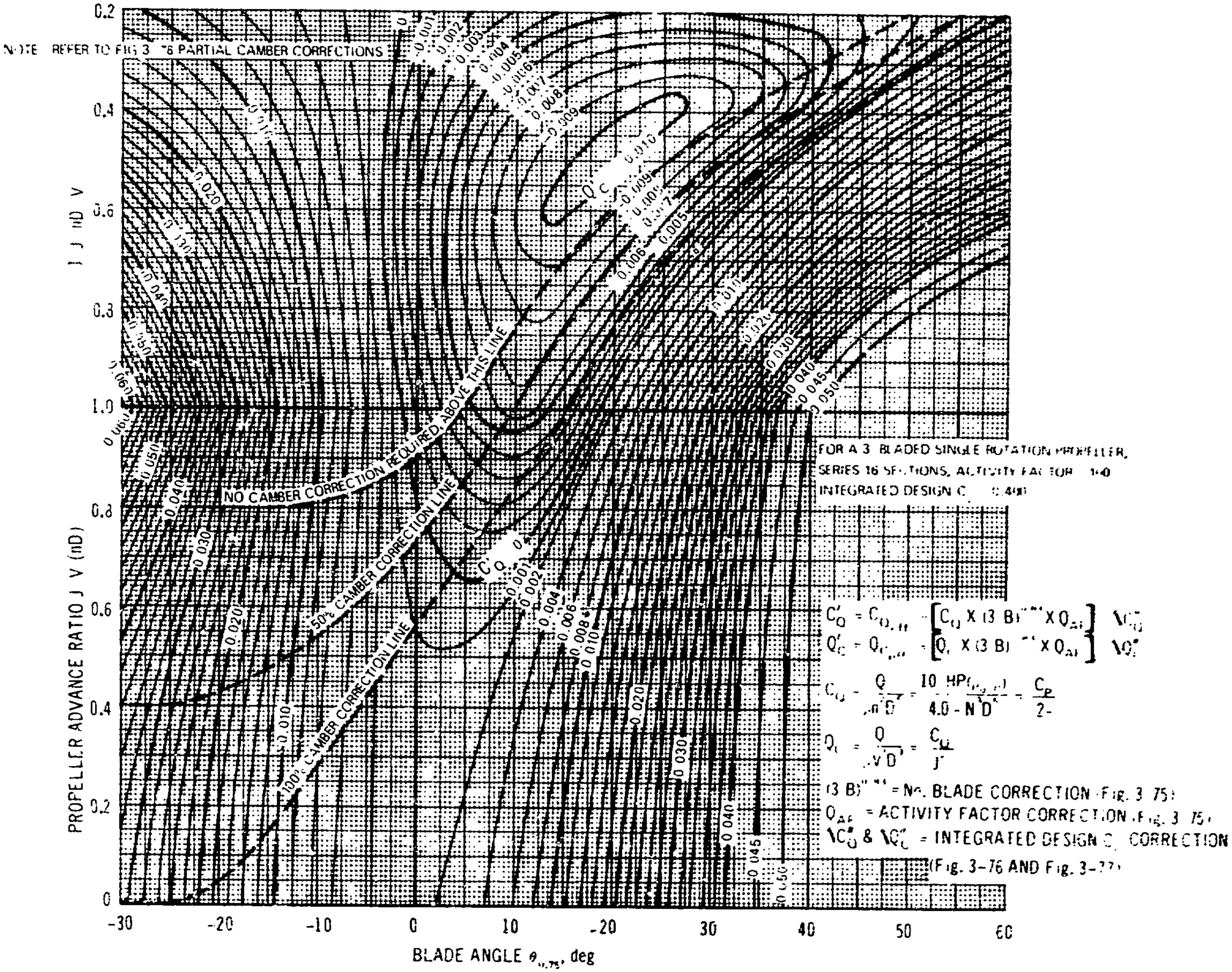


Fig. 3-79. Variation of Advance Ratio With Blade Angle at Constant Effective Torque Coefficient

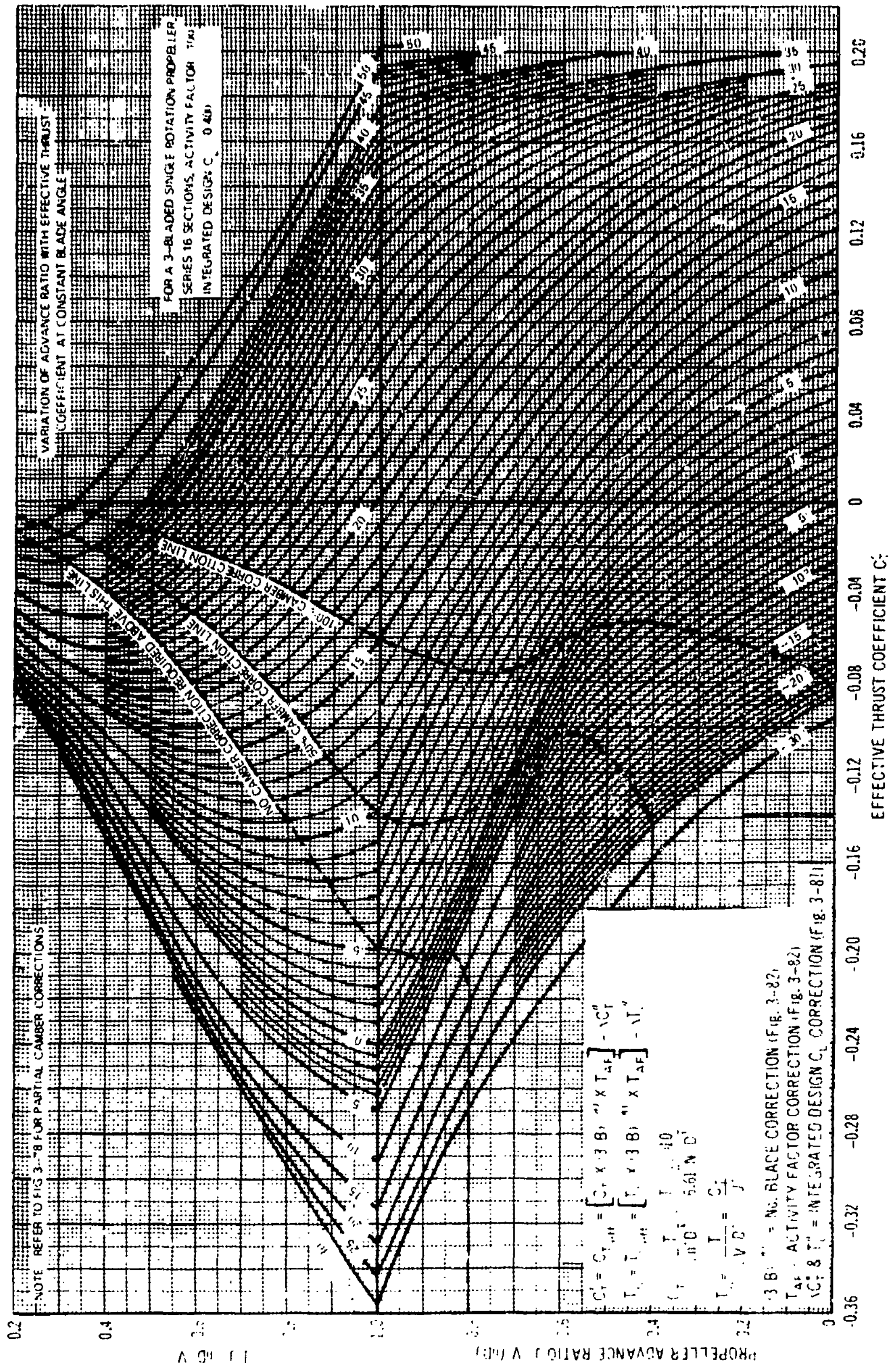


Fig. 3-80. Variation of Advance Ratio With Effective Thrust Coefficient at Constant Blade Angle

$$C_T' = C_{T_{eff}} = \left[C_T \times (3/B)^{0.83} \times T_{AF}\right] - \Delta C_T''$$

$$T_C' = T_{C_{eff}} = \left[T_C \times (3/B)^{0.83} \times T_{AF}\right] - \Delta T_C''$$

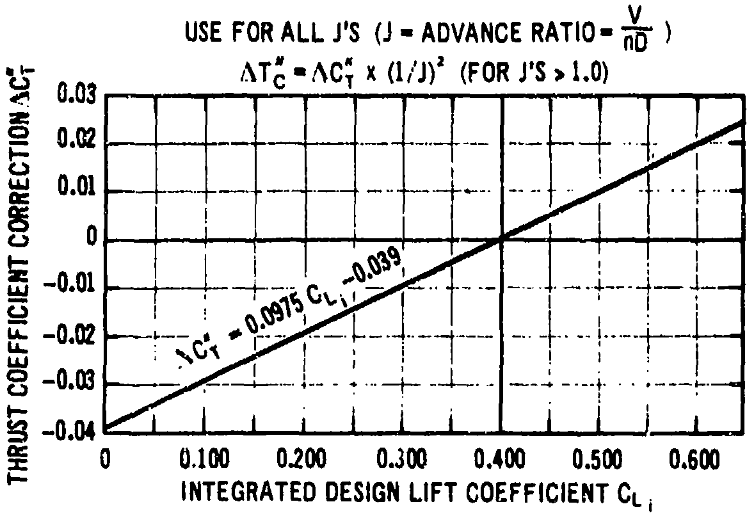


Fig. 3-81. Effect of Integrated Design Lift Coefficient on Thrust Coefficient

field experienced by the tail rotor includes the following conditions:

1. Static operation in hover

2. Inplane component of freestream velocity in forward or rearward flight, as for the main rotor

3. Positive or negative freestream velocity along the rotor axis in sideward flight or yawing motions, including operation in the "vortex ring" state

4. Significant interaction with the induced flow field of the main rotor under all conditions.

Thus, the design of the tail rotor involves semiempirical compromises. Control normally is effected by pitch changes. A detailed discussion of the factors involved in the design of conventional tail rotors may be found in Refs. 51 and 52.

The potential merit of employing a swiveling tail rotor to provide auxiliary propulsive thrust in forward flight has been studied in flight tests (Ref. 53). In this case, the tail rotor had a fixed pitch and could pivot about a vertical axis under the control of the helicopter rudder pedals. Adequate control power was demonstrated, and the transition from hover to forward flight was smooth. This design obviously has the advantage of simplicity through avoidance of the necessity for pitch controls and other elements associated with separate propellers for antitorque control and forward flight. On the other hand, the design requirements for a tail rotor and a propeller for forward flight speeds above 150 kt differ significantly, and a single propeller designed to handle both propulsive and antitorque functions is inefficient in both. The propeller for auxiliary propulsion, for example, requires a distribution of twist far different from that desired and normally employed in tail rotor design. Also, the higher flight speeds could involve the transmittal of a large fraction of the total power aboard the vehicle to the thrust-producing propeller. In this case, the complexity of swiveling elements in the drive train might be undesirable.

3-3.4.2 Shrouded Propeller Propulsion

A shrouded (ducted) propeller for forward flight propulsion offers certain advantages over conventional free propellers, particularly under low-speed or static conditions. However, the improvement in efficiency diminishes at higher speeds because of the drag of the shroud. Although the concept is not new and the state-of-the-art in design is suitable for preliminary design purposes, more data are needed for refined analysis, including off-design performance at various angles of attack.

There are two important design considerations for a shrouded propeller:

1. The provision of a shroud around a propeller is beneficial only when the device operates at sufficiently high values of the thrust coefficient $C_{T_{SP}}$ given by

$$C_{T_{SP}} = \frac{T_{SP}}{\rho A_p V^2/2}\text{, dimensionless} \qquad (3\text{-}121)$$

where

$C_{T_{SP}}$ = total thrust coefficient for the shrouded propeller, dimensionless
A_p = area of the propeller disk, ft^2
V = free stream velocity, fps
T_{SP} = total thrust of the shrouded propeller, lb

2. The duct should be designed to provide an increase of velocity at the plane of the propeller (converging inlet).

The first consideration arises from taking into account the drag of the duct, which is part of the total propulsive device. It implies that the device will function best either under static conditions ($V = 0$) or at high disk loadings T_P/A_P. There is little point in shrouding a lightly loaded propeller. The requirement to operate at higher disk loadings for forward flight leads to inherently lower propulsive efficiencies by comparison with lightly loaded, unshrouded propellers.

The total thrust of a ducted propeller includes contributions from both the duct and the propeller. According to Ref. 54, a partial thrust coefficient for the thrust C_{T_P} of the propeller alone should be introduced.

$$C_{T_P} = \frac{\Delta p}{\rho V^2/2}\text{, dimensionless} \qquad (3\text{-}122)$$

$$C'_T = C_{T_{eff}} = \left[C_T \times (3/8)^{0.83} \times T_{AF}\right] - \Delta C''_T$$

$$T'_C = T_{C_{eff}} = \left[T_C \times (3/8)^{0.83} \times T_{AF}\right] - \Delta T''_C$$

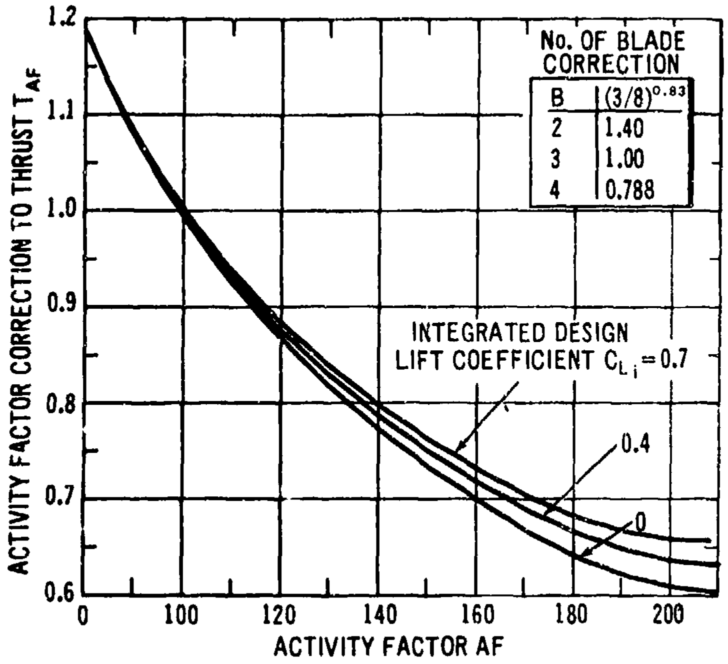


Fig. 3-82. Effect of Activity Factor on Thrust Coefficient

where

Δp = pressure rise across the propeller, psf

From Bernoulli's equation, applied to the flows upstream and downstream from the propeller, $\Delta p = \rho(V_j^2 - V^2)/2$. Using approximations of the simple momentum theory of propellers, in the absence of the duct the ratio of the velocity at the plane of the propeller V_P to the free stream value V would be

$$\frac{V_P}{V} = \frac{V + V_j}{2V} = 1 + \xi/2 \quad \text{, dimensionless} \qquad (3\text{-}123)$$

where

$\xi = (V_j - V)/V$, dimensionless

The thrust coefficient C_{T_P} for the unshrouded propeller becomes

$$C_{T_P} = 2\xi\left(1 + \frac{\xi}{2}\right) \quad \text{, dimensionless} \qquad (3\text{-}124)$$

The presence of the shroud will alter the velocity at the plane of the propeller. Under this condition an incremental velocity ratio δ is defined so that

$$\frac{V_P}{V} = 1 + \frac{\xi}{2} + \delta \quad \text{, dimensionless} \qquad (3\text{-}125)$$

Depending upon the design of the shroud, δ may be positive or negative. For the ducts of interest here, δ should be positive (converging ducts). The thrust coefficient $C_{T_{SP}}$ for a shrouded propeller then becomes

$$C_{T_{SP}} = 2\xi\left(1 + \frac{\xi}{2} + \delta\right) \quad \text{, dimensionless} \qquad (3\text{-}126)$$

The corresponding ideal propulsive efficiency for the shrouded propeller $\eta_{P_{SP}}$ from Eqs. 3-120 and 3-124, is

$$\eta_{P_{SP}} = \frac{1}{1 + \xi/2} = \frac{2}{1 + \sqrt{1 + C_{T_P}}} \quad \text{dimensionless} \qquad (3\text{-}127)$$

which is dependent upon the thrust coefficient of the propeller alone. Comparing the expressions for $C_{T_{SP}}$ and C_{T_P} as in Eqs. 3-126 and 3-124, it may be seen that, for δ positive (converging duct) and assuming equal thrust coefficients, ξ will be lower for the ducted propeller and so $\eta_{P_{SP}}$ will be higher. This comparison should not be interpreted to imply that the values of C_{T_P} for free propellers and $C_{T_{SP}}$ for ducted propellers designed for the same application would be chosen to be equal.

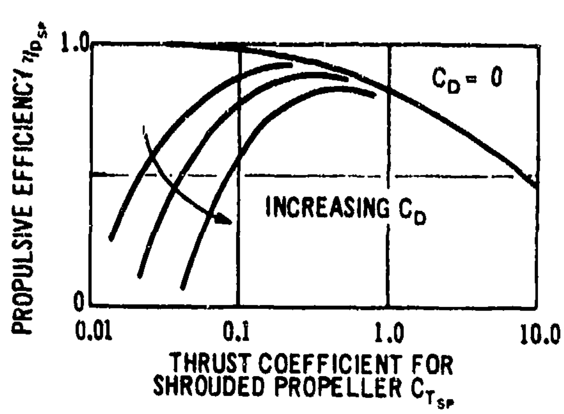


Fig. 3-83. Variations of Shrouded Propeller Performance as Influenced by Shroud Drag

In accounting for the drag of the duct by defining a suitable drag coefficient C_D, the overall propulsive efficiency of the shrouded propeller would vary, as illustrated in Fig. 3-83. As shown, depending upon the design of the duct and the resulting C_D value, the propulsive efficiency $\eta_{P_{SP}}$ decays rapidly at the lower thrust coefficient values.

The design of a ducted propeller involves the aerodynamics of both the duct and the propeller. The propeller design follows well-developed methods for axial flow fans, as described in Refs. 55, 56, and 57, given the flow field interference effects of the duct. The aerodynamic design of the duct may be treated by the method of singularities, i.e., by replacing the duct by a suitable distribution of ring vortices (Ref. 54). In determining the total circulation of the ring airfoil, it is necessary in this procedure to account for the interference effects of the propeller upon the flow about the ring airfoil. The influence of the propeller may be characterized by a distribution of vortex rings over the surface of the propeller, in addition to a cylindrical vortex tube representing the downstream jet flow. Characterization of a given geometry by this method involves lengthy computations leading to the solution of the resulting integral equation, as in the conventional aerodynamic theory of wings. For preliminary design purposes, some simplifications may be acceptable, such as characterization of the shroud by a cylindrical distribution of ring vortices, or even a single, suitably positioned vortex

(Ref. 58). One of the features of better duct design, apart from the requirement to provide a positive δ, is a relatively large inlet lip radius of curvature, on the order of 5-10% of the inlet duct radius (Ref. 54).

Several aspects of ducted propellers require consideration in the selection of such devices for auxiliary propulsion. A comparison of the ducted propeller with a free propeller must account for the fact that the ducted propeller may be loaded to its tip, and the effective advance ratio may not vary to the point where pitch controls are needed in the ducted propeller. Turning vanes may be provided in the ducted propeller to recover the rotational energy of the slipstream. The disk loading of a ducted propeller (100-500 psf) tends to be substantially higher than that for unducted propellers (10-100 psf); this leads to higher speeds, lower torques, and thus lighter weights at a given power level. These merits are reduced by the weight and drag of the duct and the resulting lower propulsive efficiency.

The ducted propeller has been considered for antitorque control in the form of a propeller-in-fan configuration for the SA.341 (Ref. 59), and has been used in the past on other nonproduction vehicles.

Figs. 3-84 through 3-87 illustrate wind tunnel results for the ducted propellers used in the Bell Aerospace X-22A (Ref. 60). Fig. 3-84 shows the dimensions of the duct; typical performance data for different propeller pitch angles β are presented in Fig. 3-85; and some test results involving flow separation from the shroud, including the effects of model scale, are presented in Figs. 3-86 and 3-87.

3-3.4.3 Turbofan and Turbojet Propulsion

Turbofan and turbojet engines, because of their relatively high exhaust velocities, are lower in propulsive efficiency than propeller or ducted propeller devices. Nonetheless, there are factors that may lead to their selection for some auxiliary propulsive purposes.

The basic arrangement of a turbofan engine is shown in Fig. 3-88. The turbojet engine configuration may be visualized simply by removing the fan, its turbine, and the external ducting. The compressor C, the burner, and the turbine T_1 form the basic gas generator unit. In the turbofan configuration, part of the energy generated by the gas leaving the gas generator drives the fan turbine T_2, and the remainder of the energy available

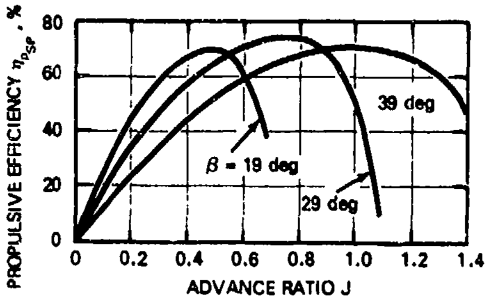


Fig. 3-85. Performance Data for the Ducted Propeller Overall Efficiency at Different Propeller Pitch Angles

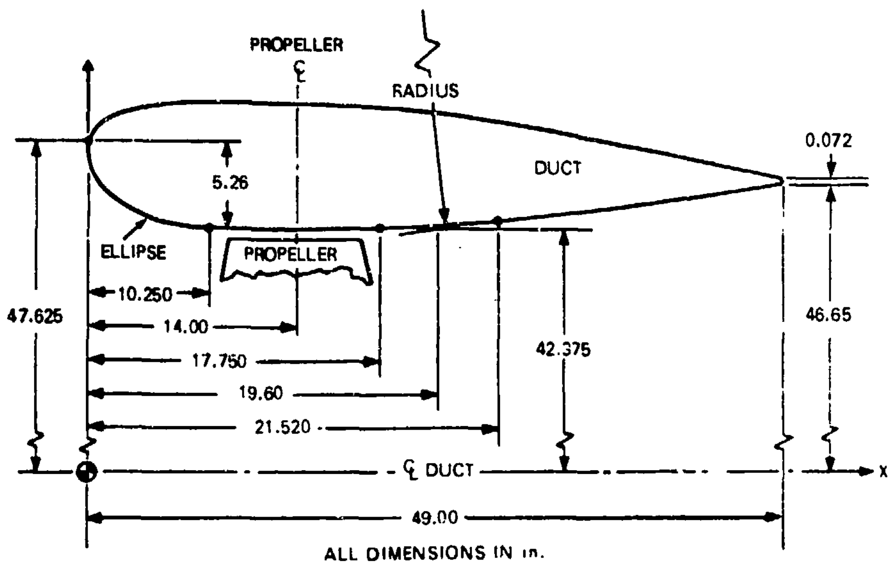


Fig. 3-84. Dimensions of the Experimental Duct for Ducted Propeller Tests

produces thrust in the exhaust nozzle A_7. In the turbojet, all of the available energy in the gas generator exhaust is employed directly for thrust production in the exhaust nozzle. The bypass ratio BPR for the turbofan engine is defined as the ratio of the mass air flow rate through the fan to that through the gas generator.

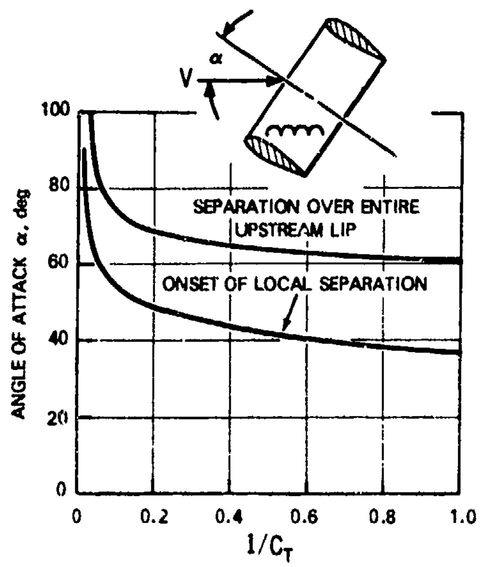


Fig. 3-86. Flow Separation on the Lower Part of the Entrance Lip of the Propeller Duct

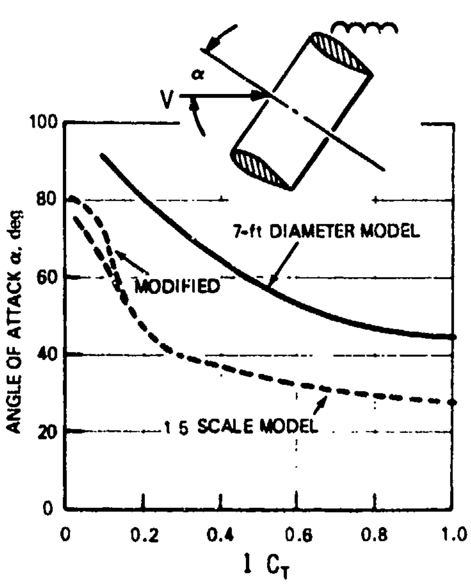


Fig. 3-87. Flow Separation on the Upper Part of the Entrance Lip of the Propeller Duct

For turbofan engines for the auxiliary propulsive application, high bypass ratios (above 3) are necessary if these engines are to be reasonably competitive.

Figs. 3-89 and 3-90 illustrate the performance parameters relative thrust and thrust specific fuel consumption (TSFC) as a function of airspeed for a range of bypass ratios, and compare these parameters with a turboprop.

Because of the high propulsive power required at higher flight speeds, integration of the propulsion and rotor power systems must be considered. The convertible fan/shaft (CF/S) engine concept discussed in Ref. 61 accomplishes this objective. The CF/S concept involves an engine that can provide shaft power under low-speed flight conditions and propulsive thrust by means of turbofan configuration at higher flight speeds. Advantages of integration of the fan into the engine—as opposed to a separate fan unit—include lighter weight, lower drag, benefits of precompression for the flow entering the compressor of the gas generator, and simpler installation. Such an installation is illustrated in Fig. 3-91.

Turbojet engines probably would not be selected for compound helicopter designs requiring sustained auxiliary propulsion primarily because of their relatively poor propulsive efficiency (high TSFC) at such flight speeds. On the other hand, if mission requirements dictate only short periods of auxiliary propulsion for high-speed dashes, turbojet engines may appear more favorable because of their relatively lighter weight by comparison with turbofan engines of comparable thrust. Early investigations of vehicle dynamics with auxiliary propulsion frequently were performed using small turbojet engines as auxiliary propulsive units because of their simplicity and the economy of adding the units on existing helicopters. The subject of auxiliary propulsion unit selection and installation configuration is treated extensively in Refs. 62 through 65.

3-3.4.4 Augmentation Systems

One way to avoid sizing a helicopter power plant for continuous operation at the extremes of the required power specifications when continuous full-power operation is not actually needed is to provide a system for temporary augmentation of the power available. For example, a requirement for 6000-ft elevation, 95°F day, and takeoff power with one engine out could be met in this manner. Augmentation methods (Refs. 66 and 67) include:

1. Precompressor injection of liquids. This procedure is intended to lower the temperature of the air entering the compressor and thereby to reduce the re-

quired work of compression. Evaporation of the liquid within the compressor produces effects similar to intercooling between stages. However, the use of water/alcohol injection is not acceptable to the U.S. Army due to contamination problems in field use

2. Supercharging the compressor by using a separate precompressor stage driven by a separate, external turbine, or by the power turbine through a clutching arrangement

3. Intercooling the compressor by using stored coolants

4. Augmenting the compressor flow by various methods (variable geometry and bleed arrangement)

5. Overspeed and overtemperature operation of the engine

6. Interburning between turbine stages

7. Use of separate gas generators to increase the power turbine mass flow rate

8. Use of separate auxiliary engines

9. Bypassing the regenerator in engines with regeneration.

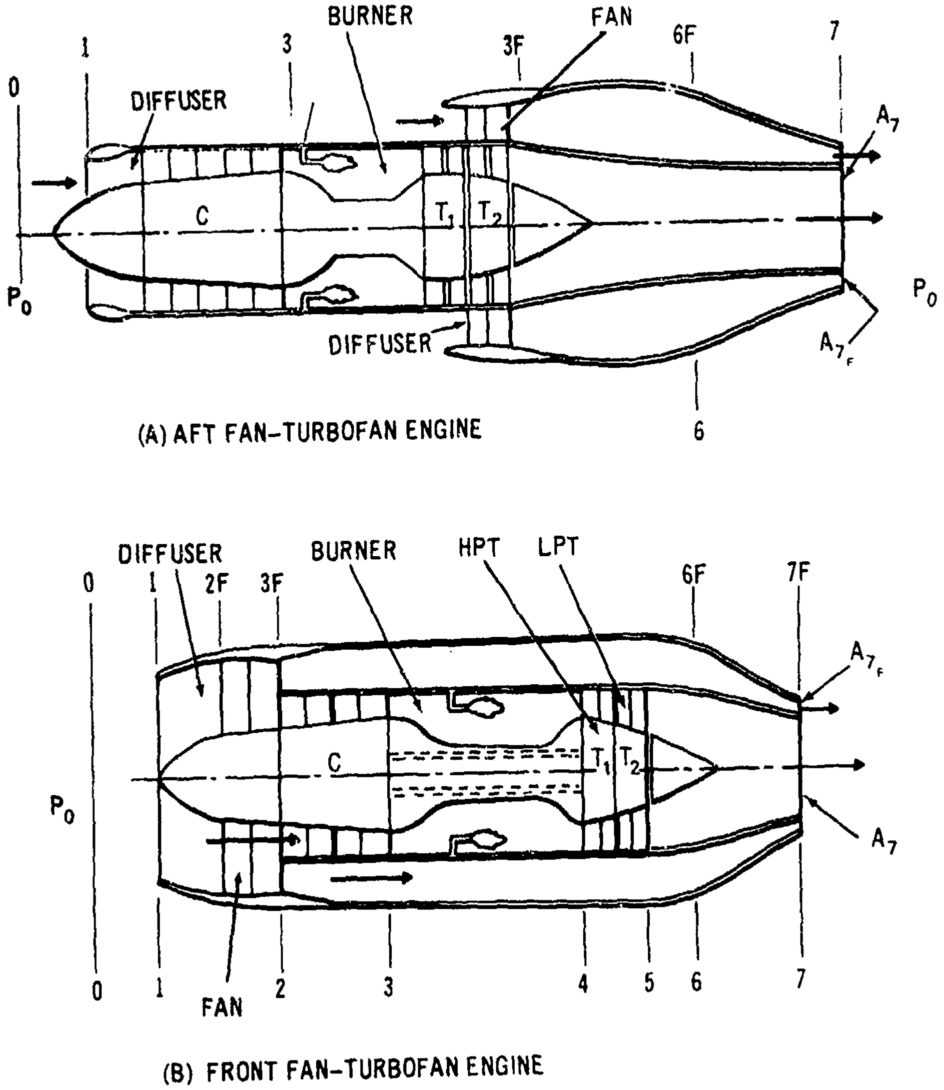


Fig. 3-88. Schematic Diagrams Illustrating Component Arrangements in Turbofan Engines (AMCP 706-285)

The results of use of water/alcohol injection ahead of the compressor, in a specific application are shown in Fig. 3-92. In computing the augmentation, it was assumed that the evaporation of the water/alcohol mixture occurred by a wet compression process within the compressor. However, actual systems suffer from:

1. Adverse effects of high relative humidity in the ambient atmosphere

2. Variation of the sites of the evaporation process within the compressor due to varying water/alcohol to air ratios and engine operating conditions, along with residence times within the compressor that are too short for complete evaporation

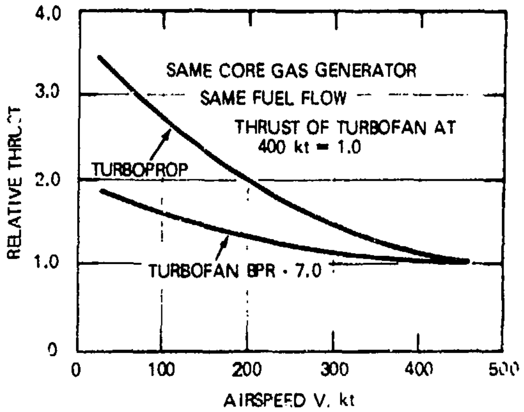


Fig. 3-89. Comparison of Performance for Turboprop and Turbofan Engines

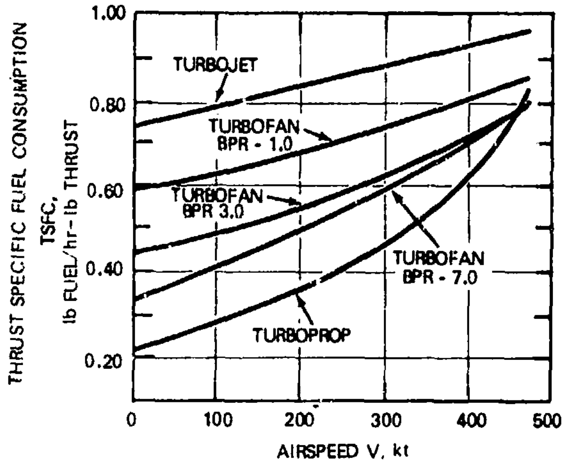


Fig. 3-90. Thrust Specific Fuel Consumption (TSFC) for Turboprop, Turbofan, and Turbojet Engines as a Function of Flight Speed

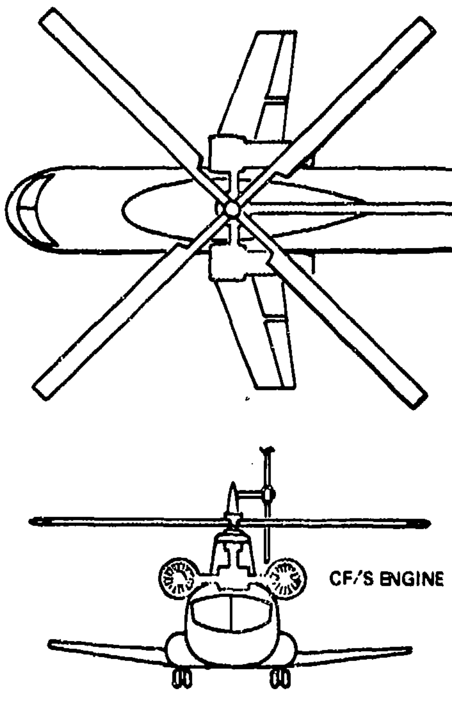


Fig. 3-91. Sample Configuration of a Convertible Fan/Shaft Engine

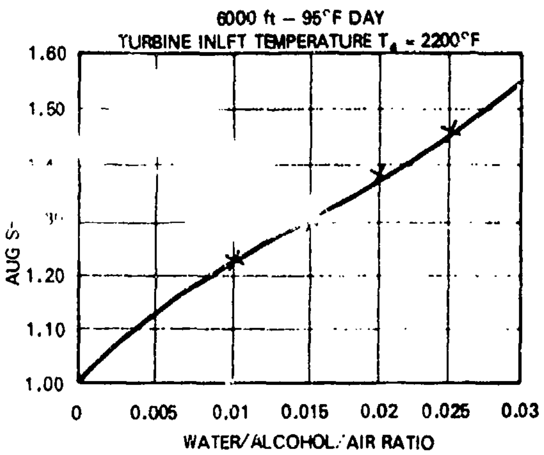


Fig. 3-92. Ideal Augmentation Provided by Water/Alcohol Injection for a Turboshaft Engine (Ref. 67)

3. The complexity of the additional water supply system and storage requirements (which are sensitive to the required duration of the augmentation) and attendant engine controls.

3-4 HELICOPTER PRELIMINARY DESIGN STUDY

3-4.1 PARAMETRIC ANALYSIS

Parametric analysis is required during the helicopter preliminary design study because of the many design parameters to be specified. A number of these parameters are interrelated in such a manner that a variation in one parameter may result in the simultaneous change of one or more related parameters. The study of the relationships among design parameters and of the effects of variations of these parameters upon helicopter mission performance is the proper domain of parametric analysis.

The complicated nature of helicopters, as well as the importance of many variables that are not quantified easily, has precluded the overall optimization of helicopter configurations by classical techniques. However, classical optimization techniques may be applied to design problems where the pertinent relationships are well known and defined. Also, the availability of large-capacity computing equipment makes possible further application of classical techniques to helicopter design. These techniques are discussed in par. 3-4.1.1. A schematic diagram of a typical preliminary design study is shown in Fig. 3-93 (Ref. 68).

Parametric analysis is a continuing task during the preliminary design study. In the early stages, the parametric analysis furnishes preliminary estimates of design parameters based upon the state-of-the-art. As the design becomes more clearly defined by the design synthesis and related tasks, the scope of the parametric analysis may be narrowed and more detail furnished on selected parameters and relationships.

The principal use for data from the parametric analysis is in design synthesis. During the preliminary design study there is a continuous interchange of information between the design synthesis and parametric analysis. The parametric analysis will provide information regarding trade-offs between major design variables, including data on the effects of the variation of particular design variables on helicopter or subsystem performance. The principal relationships used in the preparation of these data are discussed in par. 3-4.1.2.

The actual conduct of the parametric analysis study will depend upon the way in which the mission requirements are specified in the requirement document. Some simple examples of the manner in which studies are conducted are given in par. 3-4.1.3.

3-4.1.1 Optimization in Helicopter Design

The objective of the preliminary design study is the specification of the best configuration to accomplish a given set of mission objectives. Implicit in this objective are the following two steps:

1. Define the helicopter configurations that will meet the mission requirements.

2. Select the best helicopter from the candidate configurations.

In the language of classical optimization theory, Step 1 is the definition of constraints. Step 2 requires the selection of an objective function and the maximization of this function within the domain bounded by the constraints. The application of optimization to helicopter design is discussed in the paragraphs that follow.

3-4.1.1.1 Objective Functions

Selection of the best helicopter configuration to satisfy a given set of mission objectives starts with the definition of what is meant by "best", or, the selection of an objective function. Some possible optimization criteria are:

1. Minimum cost
2. Minimum size
3. Minimum gross weight
4. Maximum performance (payload, range, endurance, etc.)
5. Maximum performance per unit cost.

(Actually, because optimization usually is referred to as a process of maximization, the objective functions for those criteria listed as minimum should be replaced by their negatives for computational purposes.)

The selection of an objective function is dependent upon its suitability for the purposes of the particular design study and upon the ease (or even possibility) of computing values for the function from information normally available during the preliminary design study.

For instance, consider minimum cost. In order to optimize a design according to this criterion, it is necessary to know the effect upon cost of variations in design parameters such as disk loading and power loading. It is necessary to consider not only development and manufacturing costs, but also the costs of operation, maintenance, provisions, and spares over the life cycle of the helicopter. The development of refined cost analysis methods may make it easier to optimize on a cost basis in the future than has been possible in the past.

In contrast to the general nonavailability of suitable cost relationships, a considerable volume of helicopter

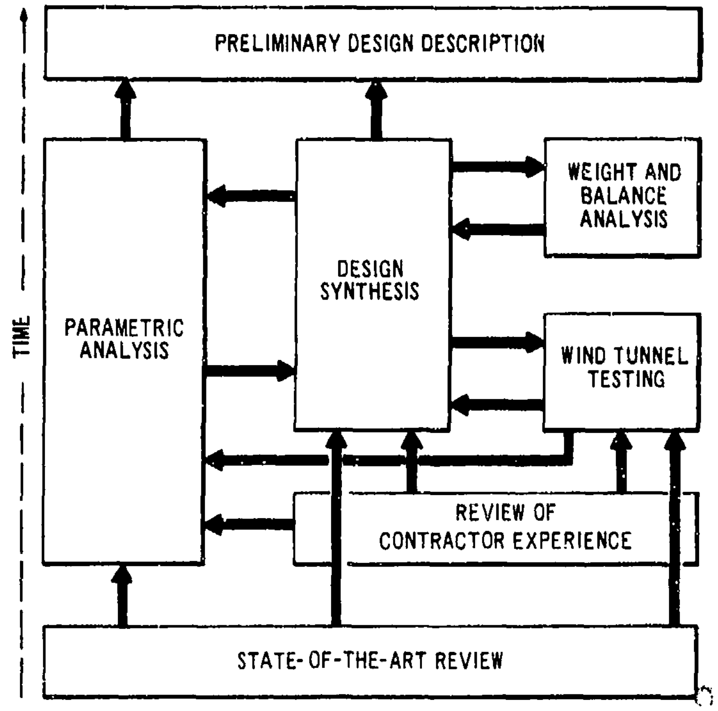


Fig. 3-93. Helicopter Preliminary Design Study

weight data has been accumulated. Minimum gross weight is, therefore, a practical choice for an optimization criterion.

Maximum performance per unit of cost more commonly is called maximum cost-effectiveness, and requirements for cost-effectiveness analysis are a part of all new military design studies. However, due to the aforementioned limitations on cost data, cost-effectiveness usually is not treated at the configuration optimization level. For a discussion of cost-effectiveness in helicopter design, see par. 2-4.

3-4.1.1.2 Constraints

For any objective function chosen, a given set of problem variables $v_1, v_2, \ldots, v_n$ should result in a unique value for the function. The variables are design parameters such as disk loading, power loading, and tip speed. In general, the larger the number of variables carried through the analysis, the more nearly optimum the solution, but also the more complicated the computational procedure. Whenever possible, the analysis should be limited to a few important independent variables; values for less important variables should be drawn from experience, engineering judgment, or suboptimization (par. 3-4.1.1.4).

In general, the variables $v_1, v_2, \ldots, v_n$, may not take on values outside some finite range determined by the mission objectives. The constraints bounding acceptable configurations involving interrelated variables usually will be in the form of an inequality such as $f'(v_1, v_2, \ldots, v_n) > a$ or $f'(v_1, v_2, \ldots, v_n) < b$, where a and b are numerical values. An example of a mission-specified constraint is illustrated in Fig. 3-94. In the figure, which presents power loading against disk loading for a given hover ceiling, the combination of power loading and disk loading must lie in the shaded region to satisfy the specified hover requirement.

In addition to the mission-specified constraints, some constraints arise from physical limitations. For instance, the rotor usually is constrained to operate at a tip speed Mach number below that value at which significantly increased drag and vibration occur due to compressibility effects. Minimum acceptable levels of rotor blade noise may result in an even lower value for maximum Mach number at the blade tip.

Other constraints, such as a minimum gross weight objective or a minimum installed horsepower requirement, may be specified directly in the requirement document.

Actual examples of constraints that arise in helicopter design are discussed in pars. 3-4.1.2 and 3-4.1.3.

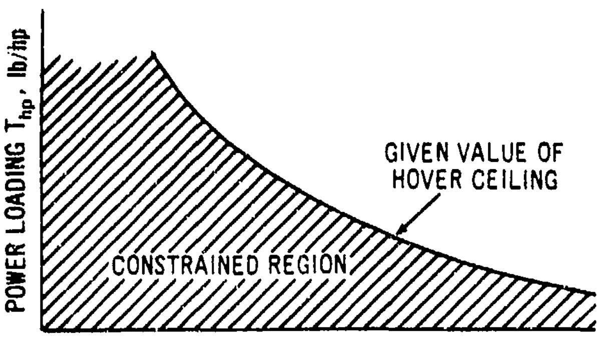


Fig. 3-94. Mission-specified Constraint

3-4.1.1.3 Optimization Techniques

Mathematical programming techniques such as linear, nonlinear, quadratic, and dynamic programming have been applied successfully to optimization problems in many diverse fields. These techniques are used to maximize a specialized objective function (usually linear) of many variables, subject to many specialized constraints (linear, quadratic, etc.). These techniques usually are not appropriate to overall helicopter design optimization because of the large number of imposed restrictions. However, the number of independent variables usually can be kept small by exercising good engineering judgment and suboptimization. Maximum seeking methods, including graphical methods, then can be used to advantage.

3-4.1.1.3.1 Maximum Seeking Methods

The independent variables in helicopter design normally are continuous. The number of possible combinations of parameters, therefore, is infinite, and all combinations of parameters cannot be tested in order to pick the best. Thus, a sequential technique is needed to vary the independent variables according to some pre-established plan so as to approach a maximum in the shortest possible time.

A detailed discussion of maximum seeking methods is beyond the scope of this handbook. For a complete study of these methods, see Refs. 69 and 70 or any recent book on operations research.

3-4.1.1.3.2 Graphical Techniques

The most commonly used techniques in engineering optimization are graphical. Graphical methods are straightforward and easily understood, and the engineer can use them independently of computers and programmers. In addition, graphical presentations not only indicate optimum sets of parameters, but also indicate the sensitivity of the objective function to small changes in the independent variables. These sensitivities become more important as the design process proceeds, and practical limitations are placed upon some variables.

Graphical techniques also have several important disadvantages. When using graphs, it is possible to carry only a few major parameters through the design process from beginning to end, and other important variables must be assigned values based upon experience and judgment. If it is found necessary at some later time to change one of these assigned values, a considerable time delay may result while the entire procedure (or a large part of it) is carried out for the new value(s). In addition, as more is learned about the fundamental relationships in helicopter design, more detail will be needed in the design process. Graphical techniques have a limited potential for extension to account for the advancing state-of-the-art in helicopter design.

3-4.1.1.4 Suboptimization

Optimization rigorously applied requires that each independent variable be specified completely in determining the optimum and that a change in any of the independent variables results in a value of the objective function no larger than the optimum. In practice, this rigorous application of optimization cannot be used in helicopter design for two reasons. First, there are many independent variables, resulting in an unmanageable optimization problem. Second, any variable must appear in the objective function in order to be used in optimization. For common objective functions such as gross weight or cost, the relevant estimating relationships have been derived on the basis of a few major variables; thus, other variables are superfluous for optimization.

The term "suboptimization" is used here to denote optimization on a limited scale, whereby a few variables are assigned values that maximize some intermediate objective function that experience has shown to be important for helicopter performance. For instance, main rotor solidity may be chosen to maximize rotor Figure of Merit, a measure of the nondimensional induced power loss in a rotor when converting torque into hovering thrust.

3-4.1.2 Basic Relationships for Parametric Analysis

Parametric analysis uses relationships from power plant analysis, aerodynamic analysis, weight analysis, and perhaps cost analysis to define those helicopter designs that will meet the mission specifications, and from them to pick the best design relative to some optimization criterion. In most cases, the relationships are approximate for two reasons. First, mathematical models are merely approximations of reality. Second, many of the relationships are empirical or at least use simplifying assumptions.

The mathematical forms or empirical fairings that may be used to specify a relationship between design variables are not unique. The evolution in helicopter analysis brings increasing precision, and simplifying assumptions are removed with regularity, usually at the expense of computational economy.

A particular relationship between design variables must be based upon a consideration of design study operational factors, among which are:

1. Need for precision at the particular design stage
2. Availability of needed data
3. Acceptability of any time delay involved in computations
4. Cost of computations.

The paragraphs that follow present some of the fundamental relationships used in parametric analysis. In each case the form of the relationship has been chosen to allow its use without the necessity for elaborate computations. The basic methods of parametric analysis remain valid regardless of the means used to compute the required relationships.

3-4.1.2.1 Engine Performance Relationships

The engine performance relationships pertinent to parametric analysis are the temperature dependence and altitude dependence of engine power output, and the relationship of fuel consumption to engine power output.

3-4.1.2.1.1 Altitude Dependence of Power Output

The power output of a gas turbine is proportional directly to the mass of air flowing through it per unit of time. Owing to the reduction of density of the ambient air for a given volume flow, this mass decreases with altitude. On the other hand, under standard conditions gas turbine thermal efficiency increases with altitude due to decreasing ambient air temperature. Parametric analysis makes use of curves of percent maximum continuous power (at sea level) plotted versus altitude. These may be derived from test data for a single engine or faired from data for a number of representative engines. To illustrate the effect of altitude on engine performance, Fig. 3-95 was plotted from a theoretical analysis of the Brayton cycle, considering

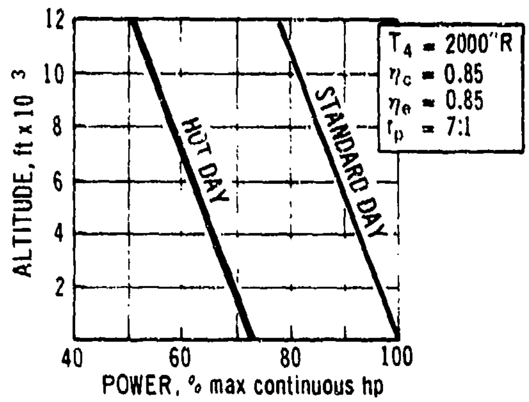


Fig. 3-95. Altitude Dependence of Turbine Engine Power

air as a perfect gas. The curve will be found to agree reasonably well with data for actual engines with the same efficiencies η_c, η_e, pressure ratio r_p, and turbine inlet temperature T_4. Fig. 3-95 also presents a curve of the "hot day" performance of the engine, with a hot day defined by increasing the standard temperature by 57 deg F at all altitudes.

3-4.1.2.1.2 Temperature Dependence of Available Power

Two factors combine to decrease the power output of a gas turbine engine as ambient air temperature increases. Most important is the fact that engine thermal efficiency decreases with increasing temperature. In addition, increasing temperature at constant atmospheric pressure results in a decrease in air density that in turn causes a decrease in the mass of air flowing through the turbine per unit of time. For use in parametric analysis, empirical data for the temperature dependence of power output generally are satisfactory. Fig. 3-96, computed from theoretical considerations, illustrates the variation of available power with ambient temperature keeping T_4 constant.

3-4.1.2.1.3 Specific Fuel Consumption

The specific fuel consumption of a gas turbine engine is shown in Fig. 3-97. This figure (Ref. 71) represents a fairing of test data. Although the data upon which the curve is based are not current, the curve represents the normal trend of specific fuel consumption with varying power setting.

3-4.1.2.2 Helicopter Power Requirements

The power requirements of a helicopter can be divided into the following categories:

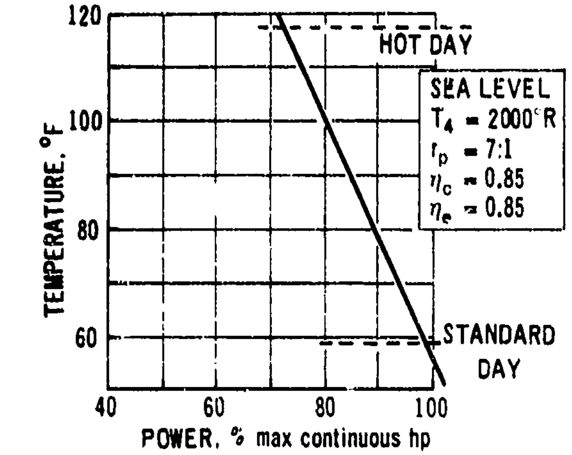


Fig. 3-96. Temperature Dependence of Turbine Engine Power

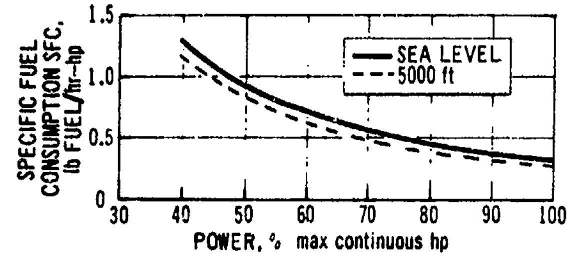


Fig. 3-97. Gas Turbine Specific Fuel Consumption

1. Main rotor power
2. Control power (tail rotor power)
3. Accessory power
4. Auxiliary propulsive power
5. Gear and transmission losses.

All of the requirements do not apply to all helicopter configurations, but each power requirement must be estimated carefully when it is appropriate to the configuration under study.

3-4.1.2.2.1 Main Rotor Power

The main rotor power requirement may be divided further into the following categories:

1. Main-rotor induced power hp_i
2. Main-rotor profile power hp_o
3. Parasite power hp_p.

The total main-rotor shaft power rhp requirement is given by

$$rhp = hp_i + hp_o + hp_p \qquad (3\text{-}128)$$

The dependence of these power requirements upon helicopter design variables is described in par. 3-2, and is simplified and summarized in the paragraphs that follow.

3-4.1.2.2.1.1 Main-rotor Induced Power

The main-rotor induced power is the power required to overcome the induced drag of the main rotor. The induced power requirements in hover and in forward flight are:

1. Hover. In hover, the induced power is given by the expression

$$hp_i = \frac{T}{550}\sqrt{\frac{w}{2\rho B^2}} \qquad (3\text{-}129)$$

where

w = rotor disk loading $T/(\pi R^2)$, lb/ft^2

B = tip loss factor, dimensionless

The expression for induced power given in Eq. 3-129 is derived from momentum theory and is based upon a uniform distribution of flow through the actuator disk. In the more realistic case of triangular flow distribution (Ref. 71)

$$hp_i = \frac{1.13T}{550}\sqrt{\frac{w}{2\rho B^2}} \qquad (3\text{-}130)$$

The factor B, accounting for the loss of lift at the blade tips due to three-dimensional flow, can be computed from Eq. 3-18 for rotors of low solidity.

2. Forward flight. In forward flight, the induced power is less than in hover because of the lift of the rotor due to forward velocity. Induced power can be expressed as

$$hp_i = \frac{1.13T}{550}\sqrt{\left(\frac{w}{2\rho B^2}\right)}\, K_u \qquad (3\text{-}131)$$

An estimate of the induced power correction factor K_u can be obtained from Wald's Equation (Eq. 3-32). When Eq. 3-32 is modified to express forward velocity V in knots, and the tip path plane assumed to be parallel to the helicopter velocity vector, $\alpha_{tpp} = 0$, the factor K_u is simply the ratio of the induced velocities v/v_0. Therefore

$$K_u^4 + K_u^2 \left(\frac{1.69V}{v_0}\right)^2 = 1 \quad \text{, dimensionless} \qquad (3\text{-}132)$$

Using the expression for v_0

$$v_0 = \sqrt{w/(2\rho B^2)} \quad \text{, fps} \qquad (3\text{-}133)$$

values of K_u can be computed. A plot of K_u at sea level for various values of disk loading is given in Fig. 3-98.

3-4.1.2.2.1.2 Main-rotor Profile Power

Profile power is the power required to overcome the profile drag (skin friction and pressure drag) of the main rotor. Profile power in forward flight may be computed from Eq. 3-42.

In hover, the advance ratio μ vanishes, giving a lower value for hp_o. For the computation of the mean drag coefficient $\bar{C}_D$, see par. 3-2.1.2.1.2.

In terms of thrust and blade loading w/σ, Eq. 3-42 can be rewritten as

$$hp_o = \frac{T}{4400} \frac{\rho(\Omega R)^3 \bar{C}_D}{w/\sigma} (1 + 4.65\mu^2) \qquad (3\text{-}134)$$

where

σ = rotor solidity, $bc/\pi R$, dimensionless

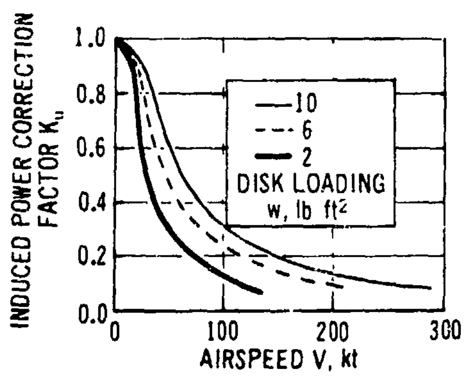


Fig. 3-98. Induced Power Correction

3-4.1.2.2.1.3 Parasite Power

Parasite power hp_p is the power required to overcome the parasite drag of the helicopter. In the absence of auxiliary propulsion, this power must be supplied by the main rotor.

Parasite drag in aeronautical theory usually is taken to be exactly opposite in direction to the velocity vector. However, in helicopter design a simplification results if the parasite drag is limited only to the drag opposing horizontal motion. The vertical component of parasite drag is considered as an increase in downloading (vertical drag).

The standard expression for parasite power (par. 3-2.1.2.1.3) is given by Eq. 3-47. In hover or vertical climb, where the forward velocity vanishes, there is no parasite drag.

3-4.1.2.2.1.4 Power Requirements for Tandem-rotor Configurations

The profile power and parasite power requirements for tandem-rotor configurations are given by the same expressions used for single rotors. The induced power, however, requires a special treatment. The geometry of a tandem-rotor configuration is shown in Fig. 3-99.

The tandem-rotor effective disk area A_e is computed on the basis of effective blade length BR, and also accounts for any overlapped areas. The induced power hp_i in hover is given by the expression

$$hp_i = \frac{T}{550}\sqrt{\frac{T}{2\rho A_e}}\, K \qquad (3\text{-}135)$$

The tandem-rotor interference factor K is discussed in par. 3-2.1.3.

In forward flight the value of hp_i given by Eq. 3-135 must be multiplied by a factor K_u as in the case of a single-rotor configuration.

$$hp_i = \frac{T}{550}\sqrt{\left(\frac{T}{2\rho A_e}\right)}\, K K_u \qquad (3\text{-}136)$$

The factor K_u is computed from Eq. 3-137, derived in Ref. 71 from momentum theory,

$$K_u^4 + \left(\frac{A_v}{A_e}\right)^2 \left(\frac{1.69V}{v_0}\right)^2 K_u^2 = 1. \quad \text{, d'less} \qquad (3\text{-}137)$$

where

$v_0 = [T/(2\rho A_v)]^{1/2}$

A_v = vertical area equal to single rotor plus vertical gap area (Fig. 3-99)

3-4.1.2.2.2 Control Power

Control power is the power required by the tail rotor in single-rotor configurations to counteract the torque of the main rotor and to provide directional control.

The main rotor torque Q is given by the equation

$$Q = \frac{550\ rhp}{\Omega}\ \text{, lb-ft} \tag{3-138}$$

This torque must be balanced by a counter torque produced by a tail rotor thrust T_{TR} at the end of a tail rotor moment arm s_{TR}.

$$Q = T_{TR}s_{TR}\ \text{, lb-ft} \tag{3-139}$$

Combining Eqs. 3-138 and 3-139, the expression for tail rotor thrust is obtained.

$$T_{TR} = \frac{550\ rhp}{\Omega s_{TR}}\ \text{, lb} \tag{3-140}$$

Tail rotor power hp_{TR} is composed of induced power and profile power, which may be computed from Eqs. 3-131 and 3-134, respectively, if the main rotor variables are replaced by the corresponding tail rotor variables. The tail rotor thrust must be sufficient to counteract the main rotor torque and also to provide adequate yaw control and maneuvering ability.

3-4.1.2.2.3 Accessory Power

Accessory power hp_a is the power required by accessories such as generators, air conditioners, and winches. Accessory power is computed by summing the power requirements of all the accessories, including the losses in the power takeoffs from the power plant or drive system.

3-4.1.2.2.4 Auxiliary Propulsive Power

Auxiliary propulsive power hp_{aux} is the power required by a propeller or other auxiliary propulsion device. Auxiliary propulsion is employed with or without fixed aerodynamic surfaces to permit higher forward speeds than can be obtained when all propulsive requirements are met by the main rotor.

Auxiliary propulsion may be supplied by a jet engine separate from the main engine(s). In this case it is necessary for the auxiliary engine only to provide sufficient thrust to overcome the parasite drag of the helicopter and the profile drag of the wing (if a wing is used). However, use of a separate engine is not the usual procedure in helicopters because of the low propulsive efficiency of a jet engine at low speed (par.

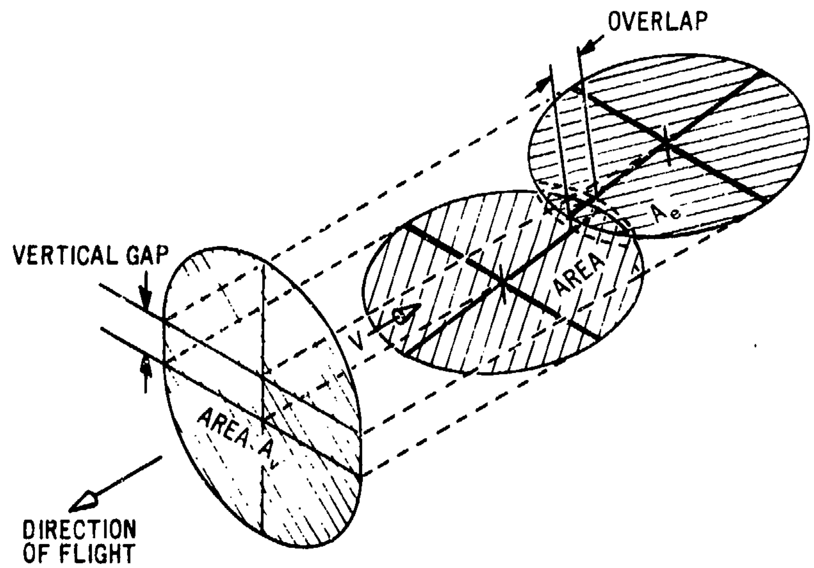


Fig. 3-99. Tandem-rotor Geometry

3-3.3.4). When such a configuration is used, it usually is to allow inexpensive modification of an existing design. The more normal practice is to use a propeller driven by the main engines. Power requirements for the auxiliary propeller are discussed in par. 3-2.2.4.2.

3-4.1.2.2.5 Gear and Transmission Power Losses

Gear and transmission power losses hp_g comprise that power which is lost in friction associated with the drive train. These losses usually are estimated from past experience with similar components as a percentage of the total horsepower requirement for the helicopter.

3-4.1.2.2.6 Total Helicopter Power Requirement

The total helicopter power requirement BHP is given by

$$BHP = rhp + hp_{TR} + hp_{aux} + hp_a + hp_g \qquad (3\text{-}141)$$

The power requirements for hover, forward flight, and climb are discussed in the paragraphs that follow.

3-4.1.2.2.6.1 Hover Power Required Out-of-ground Effect (OGE)

For most helicopter designs, the most stringent power requirement is to support hover out-of-ground effect (OGE) at the design altitude and temperature condition.

It is possible that the propeller (if used) can be disengaged in hover so that hp_{aux} vanishes. However, the clutch installation may result in unacceptable cost and weight penalties. In such cases the propeller is kept in motion with the pitch set so as to produce zero net thrust. The value of hp_{aux} then is just the power required to overcome the propeller profile drag.

The main rotor power in hover rhp_{hov} is given by

$$rhp_{hov} = hp_i + hp_o \qquad (3\text{-}142)$$

The induced power and profile power are given by Eqs. 3-129 and 3-134 (with $\mu = 0$), respectively. The thrust T to be used is the sum of gross weight and hover download (vertical drag). The computation of hover download is discussed in par. 3-2.1.1.9.

The factor hp_{TR} in Eq. 3-141 can be computed from the considerations in par. 3-4.1.2.2.2, or may be estimated as a fraction K_t of BHP, i.e.,

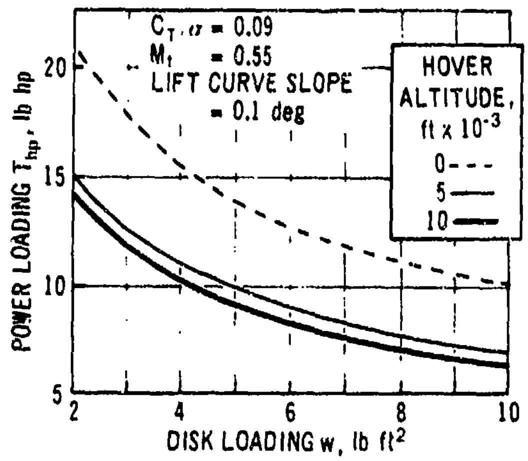


Fig. 3-100. Hover Power Loading at Altitude

$$hp_{TR} = K_t(BHP) \qquad (3\text{-}143)$$

The factor hp_g usually is computed as a fraction K_g of BHP from the relationship

$$hp_g = K_g(BHP) \qquad (3\text{-}144)$$

where the values of K_t and K_g are estimated from statistics of past helicopter designs.

A useful presentation of power required in hover is shown in Fig. 3-100, where the power loading T_{hp} is given by

$$T_{hp} = \frac{W_g}{BHP}\ , \text{lb/hp} \qquad (3\text{-}145)$$

where

W_g = gross weight, lb

The curves of horsepower required to hover at altitude, shown in Fig. 3-100, can be converted to requirements for installed sea level maximum continuous horsepower by applying a factor to account for the loss of engine power output at altitude. The altitude correction factors are given by Fig. 3-95. A temperature adjustment factor can be read from Fig. 3-96 if the hover conditions are required to be met at temperatures other than standard. The resulting curves are shown in Fig. 3-101.

Implicit in the curves of Figs. 3-100 and 3-101 are values of ΩR, σ, B, C_D, and hover download D_v. The values of C_D, B, and σ are dependent upon the rotor

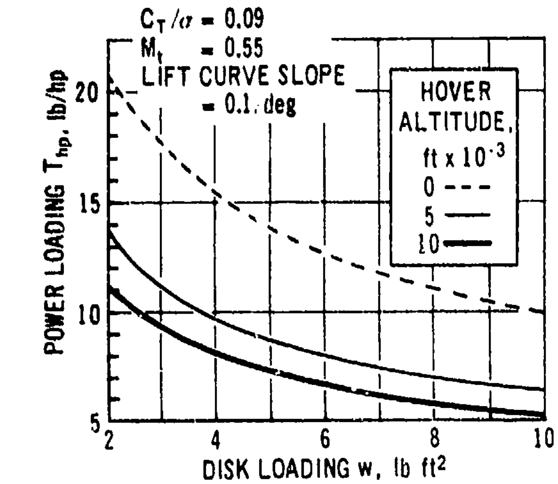


Fig. 3-101. Sea Level Power Loading

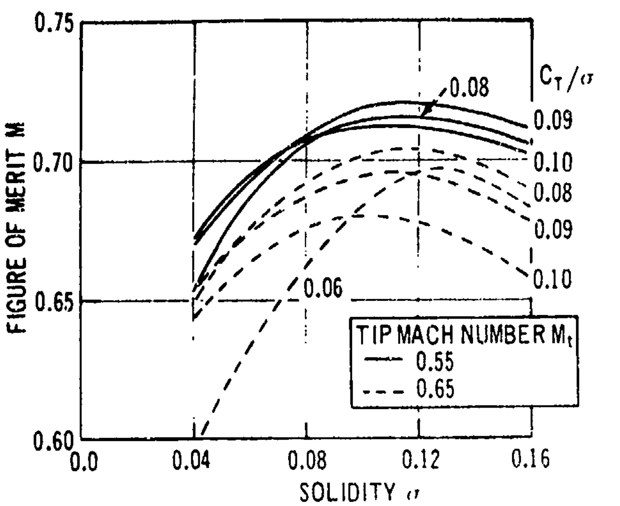


Fig. 3-102. Rotor Figure of Merit

design. The rotor solidity σ may be chosen to maximize the rotor Figure of Merit using curves similar to those in par. 3-2.1.1.7. A plot of rotor Figure of Merit against solidity for various values of the ratio C_T/σ for two tip Mach numbers is shown in Fig. 3-102. These curves are computed by the following procedure:

1. For a selected value of C_T/σ, tabulate values of maximum Figure of Merit for selected values of solidity.

2. Multiply the maximum Figure of Merit by the Figure of Merit Ratio FMR for the appropriate values of σ, C_T/σ, and M_t, and plot.

Fig. 3-102 shows that, for a tip Mach number M_t of 0.55, the Figure of Merit is maximized over a wide range of values of σ when a value of 0.09 is assigned to C_T/σ. The optimum value of C_T/σ is lower for a tip Mach number of 0.65. A plot of maximum Figure of Merit (from the curves of Fig. 3-102) versus the corresponding values of C_T/σ is shown for two tip Mach numbers in Fig. 3-103. It can be seen that such a figure can be used to select an optimum value of C_T/σ for given values of the other rotor parameters.

The use of optimum values of C_T/σ will result in a considerable simplification of the expression for main rotor hover profile power. The value of solidity will be given by

$$\sigma = \frac{C_T}{(C_T/\sigma)_{opt}} \text{, dimensionless} \quad (3\text{-}146)$$

Upon applying the definitions of thrust coefficient C_T and disk loading w, and rearranging,

$$w/\sigma = \rho(\Omega R)^2 (C_T/\sigma)_{opt} \text{, lb/ft}^2 \quad (3\text{-}147)$$

Inserting this expression for w/σ into Eq. 3-134 and combining with Eq. 3-130 gives the total rhp to hover OGE rhp_{HOGE}

$$rhp_{HOGE} = \frac{1.13T}{550}\sqrt{\frac{w}{2\rho B^2}} + \frac{T\Omega R\bar{C}_D}{4400(C_T/\sigma)_{opt}} \quad (3\text{-}148)$$

The value of rotor tip speed ΩR usually is governed

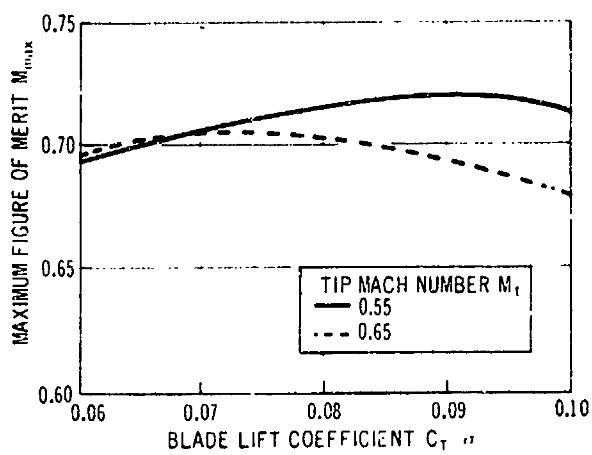


Fig. 3-103. Maximum Rotor Figure of Merit

by consideration of retreating tip blade stall or advancing tip compressibility in forward flight.

3-4.1.2.2.6.2 Hover Power Required In-ground Effect (IGE)

The power required to hover in-ground effect (IGE) is determined by applying a ground effect correction to the OGE hover power requirement. The total *rhp* to hover IGE is then given by rhp_{HIGE} where

$$rhp_{HIGE} = \left[\Lambda\left(\Delta C_{Q_\infty} + \frac{\sigma\delta_0}{8}\right)\right]\frac{\rho\pi R^2(\Omega R)^3}{550} \qquad (3\text{-}149)$$

An estimate of Λ (par. 3-2.1.1.8), valid only for $Z > R$, is given by

$$\Lambda = \sqrt[3]{\frac{Z}{2R}}\text{, dimensionless} \qquad (3\text{-}150)$$

For computations at lower wheel heights, see par. 3-2.1.1.8.

3-4.1.2.2.6.3 Power Required in Forward Flight

The expression for main rotor power in forward flight rhp_{ff} is given by combining Eqs. 3-131, 3-134, and 3-47

$$rhp_{ff} = \frac{1.13T}{550}\left[\frac{w}{2\rho B^2}\right]^{1/2} K_u + \left(\frac{T}{4400}\right)\left[\frac{\rho(\Omega R)^3\bar{C}_D}{w/\sigma}\right](1 + 4.65\mu^2) + \frac{f_T\rho V^3}{1100} \qquad (3\text{-}151)$$

If auxiliary propulsion is provided, hp_{aux} will be subtracted from this expression.

A typical plot of inverse power loading required in forward flight versus forward velocity is shown in Fig. 3-104. The dip in the curve is due to the fact that the induced power term of Eq. 3-151 predominates in low-speed flight while the parasite power term predominates at high speed. The additional power requirement due to blade stall and compressibility effects at high speed is not shown in Fig. 3-104 because the problems associated with these effects preclude normal flight at such speeds.

The curves of Fig. 3-104 presume a constant altitude, a constant rotor tip speed, and a constant value of f_T/W_g. Any of these parameters may be varied to generate sets of parametric curves if required for analysis.

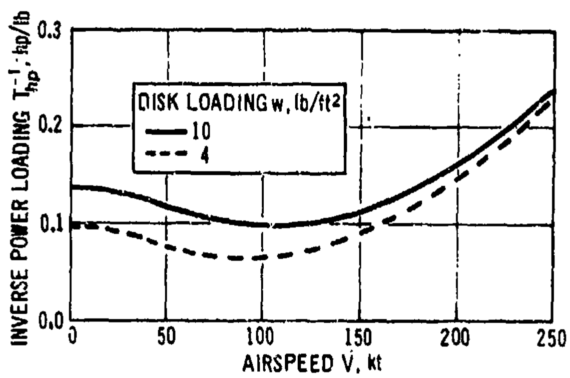


Fig. 3-104. Power Required in Forward Flight

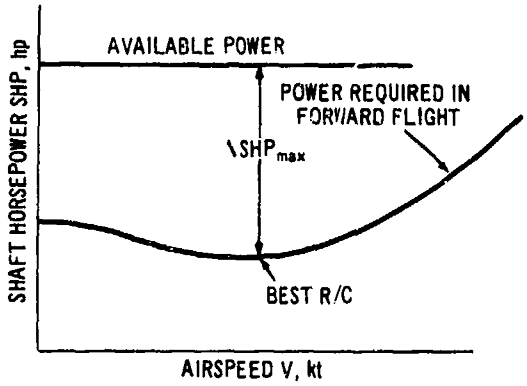


Fig. 3-105. Climb Power Determination

3-4.1.2.2.6.4 Power Required to Climb

The expression for helicopter rate of climb *R/C* (par. 3-4.2.6) is

$$R/C = \frac{\Delta SHP \times 33{,}000 \times K_p}{W_g}\text{, ft/min.} \qquad (3\text{-}152)$$

The factor ΔSHP is the additional power (over that required to maintain altitude) available to climb, and K_p is the climb efficiency factor. Obviously a helicopter *R/C* will be at maximum when the forward velocity is that which minimizes the power requirement in level forward flight, as shown in Fig. 3-105. If vertical climb is required, ΔSHP must be read at the ordinate for zero forward velocity. In this case, the installed power must be sufficient to produce the required rate of climb as given by Eq. 3-152.

3-4.1.2.3 Fuel Requirements

3-4.1.2.3.1 Fuel Flow in Hover

The computation of fuel flow in hover is accomplished by the use of Fig. 3-106. Chart A is a cross plot of Fig. 3-101 for a fixed gross weight and shows the design power loading appropriate to the normal rated power NRP required for hovering at the design hover altitude. At a selected altitude, determine the power loading required. Calculate the %NRP as shown, and enter Chart B at that value. Read the specific fuel consumption SFC from the appropriate altitude curve (interpolating as required).

Fig. 3-106 also shows the construction of Chart C, which introduces the fuel weight ratio rate dR_F/dt, where R_F is the ratio of the fuel weight to the gross weight (fuel weight/W_g). Values of dR_F/dt are calculated for various altitudes, and hence unique values of T_{hp} and SFC, and plotted as shown.

3-4.1.2.3.2 Fuel Flow in Forward Flight

The fuel flow in forward flight is of prime importance in the selection of a helicopter configuration that is required to have a specified range or radius of action. The computation of this parameter is illustrated in Fig. 3-107. Chart A is a typical plot of reciprocal power loading $T_{h_r}^{-1}$ as a function of airspeed for a fixed disk loading and altitude (as described in Fig. 3-104). The chart also shows the available reciprocal power loading at normal rated power NRP. The procedure followed to obtain SFC from Chart B is the same as for hover flight and as illustrated in Fig. 3-107.

Chart C is constructed in the same way as that on Fig. 3-106, and shows the minimum value for dR_F/dt, which defines the flight speed conditions for maximum endurance. Fig. 3-107 introduces, in Chart D, the fuel rate ratio per unit range dR_F/dRg. This is calculated as shown in the figure. The minimum value of dR_F/dRg indicates the conditions pertinent to cruise for maximum range. In practice, normal cruise is conducted at a slightly higher airspeed than shown for reasons discussed extensively in par. 3-4.2.4.

3-4.1.2.3.3 Fuel Flow in Climb

The fuel weight ratio rate during climb is given by

$$\left(\frac{dR_F}{dt}\right)_{cl} = \frac{SFC_{cl}}{T_{hp_{des}}}, \text{hr}^{-1} \quad (3\text{-}153)$$

where it is assumed that normal rated power NRP is used in climbing. In this case, the value of SFC_{cl} is the SFC at NRP. The ratio of fuel required to climb from altitude h_0 (ft) to altitude h_1 (ft) at a rate of climb R/C (fpm) to mean gross weight $R_{F_{cl}}$ is given by

$$R_{F_{cl}} = \frac{h_1 - h_0}{60R/C}\left(\frac{dR_F}{dt}\right)_{cl}, \text{dimensionless} \quad (3\text{-}154)$$

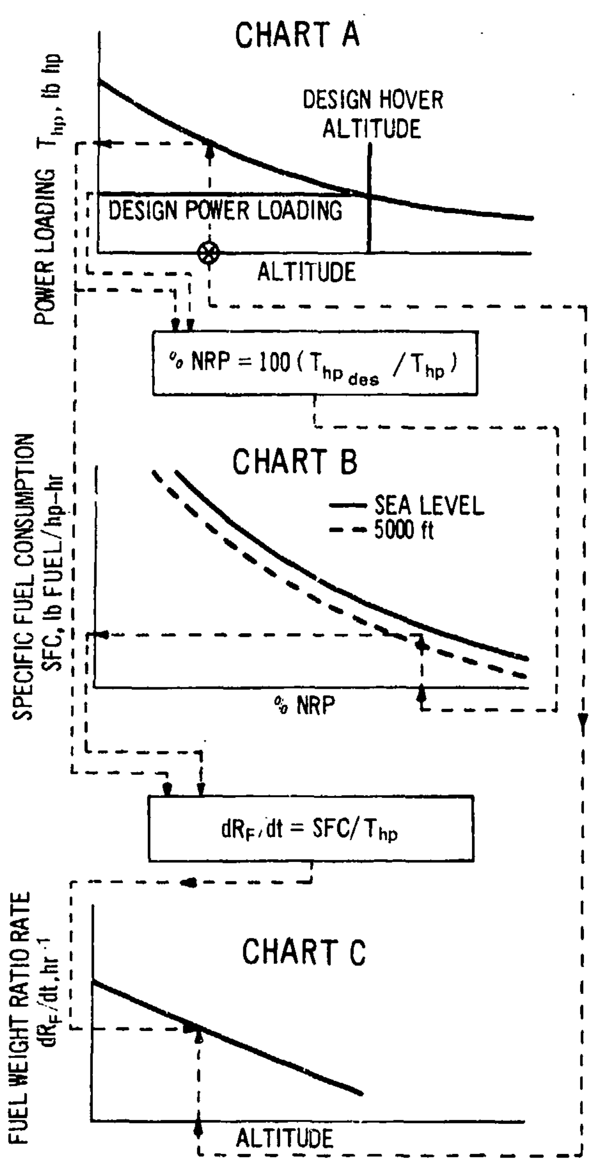


Fig. 3-106. Fuel Flow in Hover

3-4.1.2.4 Helicopter Weight-estimating Relationships

In addition to the equations for required power, the parametric analysis requires a set of equations for estimating helicopter gross weight as a function of major design parameters.

Weight-estimating relationships usually are based upon statistical data from previous helicopter designs. For more detail on weight estimation and for guidance regarding appropriate parametric relationships, see par. 10-2.4.

For configuration selection, an equation expressing the ratio of empty weight (less fuel tank) to gross weight as a function of major design parameters is required. The parameters upon which the empty weight ratio is dependent should be the ones that will be used in configuration selection techniques. For the techniques discussed in par. 3-4.1.3, the empty weight ratio should be a function of gross weight, tip speed, disk loading, power loading, and number of engines. Additional parameters, such as equivalent drag area, can be included in the configuration selection process at the expense of additional computational effort.

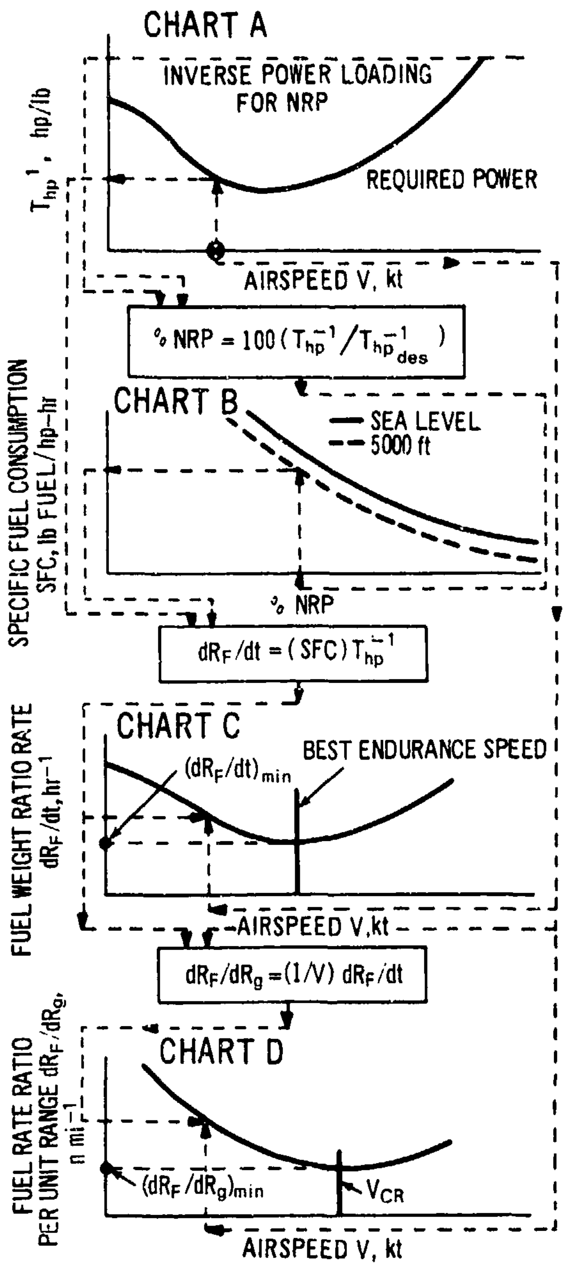


Fig. 3-107. Fuel Required in Forward Flight

3-4.1.3 Techniques of Configuration Selection

The domain of feasible helicopter configurations is defined by balancing carefully the conflicting requirements of the aerodynamic analysis and the weight analysis. A definite link between the aerodynamic and weight requirements is the fuel weight ratio R_F.

The design mission requirements normally will include either a range requirement or an endurance requirement. Either of these, when combined with other mission requirements, will result in a minimum required R_F. Parametric analysis will define this minimum R_F as a function of selected major design parameters. Meanwhile, the weight analysis will define a maximum available R_F as a function of the same set of parameters. A feasible helicopter configuration is one for which the minimum required R_F is no less than the maximum available R_F.

The set of feasible configurations established by the R_F method may be analyzed further to determine which configuration optimizes some selected objective function.

The R_F method for helicopter design is discussed in Ref. 71, where minimum gross weight is the criterion used for optimization.

3-4.1.3.1 Feasible Configurations

3-4.1.3.1.1 Installed Power and Tip Speed Selection

The R_F method for configuration selection discussed here requires a value of installed power loading to be selected in advance. This value usually is selected on the basis of the most rigid hover requirement. To determine the most rigid requirement, power loading charts such as those shown in Fig. 3-100 must be prepared for each specified hover requirement (in terms of altitude, ambient air temperature, IGE, or OGE) and a composite curve of power loading versus disk loading is then prepared. Each point on the composite power loading curve will be the minimum power loading shown on any of the individual power loading curves for the particular disk loading.

In some cases the climb or forward flight requirement may predominate. In such a case, the power loading required for the predominating flight condition will replace the relevant portion of the composite hover power loading curve.

The rotor should be most efficient in producing thrust when operating at design maximum tip speed. The rotor is constrained to operate at a speed no greater than that for which retreating tip blade stall occurs at the maximum forward speed. An additional consideration that may limit the tip speed is the compressibility at the advancing blade tip. The resulting vibration level or incremental power requirement may effectively limit the tip speed. Because tip speed appears in both the aerodynamic and weight relationships, it may be carried over into the optimization procedure at the cost of increased computational effort.

3-4.1.3.1.2 Available Fuel Weight Ratio

The R_F method of defining feasible helicopter configurations requires the comparison of available R_F with required R_F. The available fuel weight ratio $R_{F_{av}}$ is derived from weight relationships. The weight W_F of fuel and fuel tank is given by the equation

$$W_F = W_g - W_p - W_c - \phi W_g \quad \text{, lb} \qquad (3\text{-}155)$$

where

W_g = gross weight, lb
W_p = payload, lb
W_c = crew weight, lb
ϕ = ratio of empty weight to gross weight, dimensionless

Upon dividing by gross weight,

$$R_{F_{av}} = \left(\frac{1}{1+K_F}\right) \times \left(1 - \frac{W_p + W_c}{W_g} - \phi\right) \quad \text{, d'less} \qquad (3\text{-}156)$$

where

K_F = weight of fuel tank per pound of fuel, dimensionless

and the factor $1/(1 + K_F)$ hence accounts for fuel tank weight. The empty weight to gross weight ratio ϕ is a function of gross weight, disk loading, power loading, and tip speed. Therefore, the fuel weight ratio is a function of these same variables. For selected values of power loading and tip speed, the resulting $R_{F_{av}}$ as a function of W_g will be as shown in Fig. 3-108.

3-4.1.3.1.3 Required Fuel Weight Ratio

The fuel required to perform a mission at a specified range is the sum of the fuel required to climb to cruise altitude, the cruise fuel required, and any additional fuel required for starting, maneuvering, and descent at destination. The required fuel weight ratio $R_{F_{req}}$ is expressed by the equation

$$R_{F_{req}} = \left(\frac{1}{1-K_r}\right) \times (\Delta R_{F1} + \Delta R_{F2} + \Delta R_{F3}) \quad \text{, d'less} \qquad (3\text{-}157)$$

where

K_r = fuel reserve factor, dimensionless
ΔR_{F1} = fuel weight ratio required to climb, dimensionless
ΔR_{F2} = fuel weight ratio required to cruise, dimensionless
ΔR_{F3} = fuel weight ratio required to start and maneuver, dimensionless

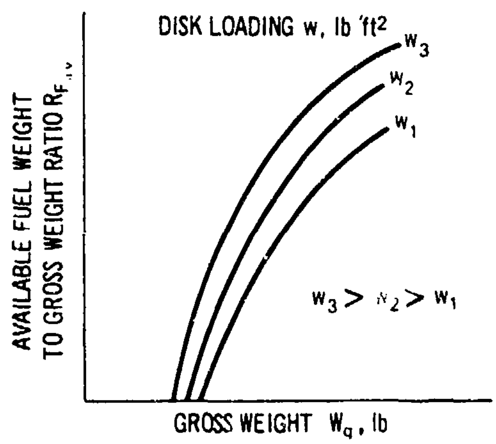


Fig. 3-108. Available Fuel Weight Ratio

The factor K_r accounts for a fuel reserve expressed as a fraction of initial fuel load. If starting and maneuvering are assumed to occur at normal rated power NRP, the fuel requirement per pound of gross weight is given by

$$\Delta R_{F3} = \frac{\Delta t\left[SFC_{NRP}\right]}{T_{hp}} \quad \text{, dimensionless} \qquad (3\text{-}158)$$

where

Δt = the time required for starting and maneuvers, hr

SFC_{NRP} = the specific fuel consumption at full power, lb/hp-hr

The term ΔR_{F1} is given by Eq. 3-154, where it is designated $R_{F_{cl}}$. If a return trip is specified in the mission requirement, the values of ΔR_{F1} and ΔR_{F3} must be doubled. The value of ΔR_{F2} is given for a simple constant altitude cruise mission of range Rg (n mi) by

$$\Delta R_{F2} = \left(\frac{dR_F}{dRg}\right) Rg \quad \text{, dimensionless} \qquad (3\text{-}159)$$

where the value of dR_F/dRg is computed as shown in Fig. 3-107. For more complicated mission profiles and related analyses refer to pars. 3-4.2.3 through 3-4.2.7.

The value of dR_F/dRg, and hence of the required fuel weight ratio $R_{F_{req}}$, is a function of gross weight, power loading, tip speed, and f_T/W_g. The selection of values of power loading and tip speed is discussed in par. 3-4.1.3.1.1. The parameter f_T/W_g may be carried as a variable, or a value may be assigned based upon wind tunnel tests or upon a statistically based estimation procedure. When specific values of all other parameters have been assigned, a plot of required fuel weight ratio for several values of disk loading can be obtained. Such a plot is shown in Fig. 3-109.

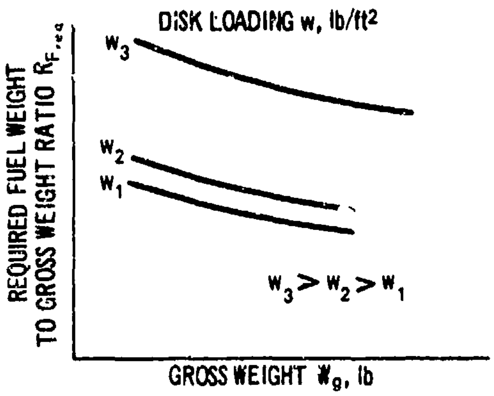


Fig. 3-109. Required Fuel Weight Ratio

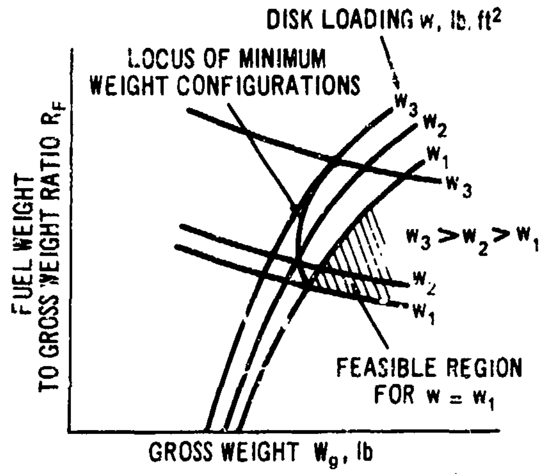


Fig. 3-110. Configuration Selection

3-4.1.3.1.4 Fuel Weight Ratio Method

The required fuel weight ratio of Fig. 3-109 and the available fuel weight ratio of Fig. 3-108 are shown plotted together in Fig. 3-110. The condition for a feasible configuration is

$$R_F \geq R_{F_{req}} \quad \text{, dimensionless} \qquad (3\text{-}160)$$

The shaded region on Fig. 3-110 is the feasible region for disk loading w_1. The point of intersection of the $R_{F_{av}}$ and $R_{F_{req}}$ curves for each disk loading defines the minimum gross weight configuration for that disk loading. The locus of minimum weight configurations also is shown. If the objective for optimization is minimum gross weight, the optimum configuration (gross weight and disk loading) can be read directly from Fig. 3-110.

The basic R_F method can be extended by including in the analysis additional parameters such as tip speed ΩR or f_T/W_g. The resulting graph then would consist of a number of sets of R_F curves, each set generating a locus of minimum gross weight configurations. The choice of an optimum design is dependent upon the optimization criterion selected. Optimization is discussed in more detail in paragraphs that follow.

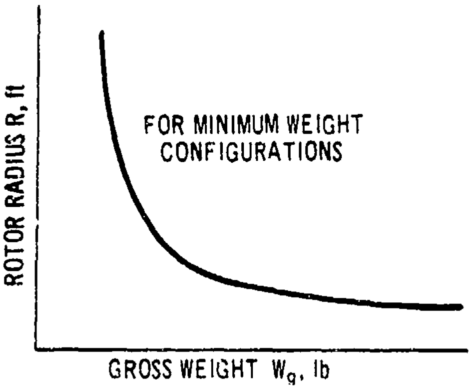


Fig. 3-111. Main Rotor Radius

3-4.1.3.2 Optimization

3-4.1.3.2.1 Minimum Weight and Minimum Size Configurations

The relative sizes of the minimum gross weight configurations from Fig. 3-110 can be seen in Fig. 3-111. This figure is a plot of main rotor radius as a function of gross weight for the minimum weight configurations shown in Fig. 3-110, used to determine the effects of size or weight limitations or requirements that also might be applicable to the design. The rotor blade radius R is given by the equation

$$R = \sqrt{\frac{W_g}{\pi w}}\ \text{, ft} \qquad (3\text{-}161)$$

Fig. 3-111 in conjunction with Eq. 3-161 shows that the disk loading w for a minimum weight configuration increases much more rapidly than does the gross weight and hence the rotor radius decreases. Therefore, the minimum size configuration for a given mission profile generally is the heaviest allowable configuration.

3-4.1.3.2.2 Minimum Cost Configurations

A replot of the feasible locus of minimum weight configurations (from Fig. 3-110) is shown in Fig. 3-112, superimposed upon constant cost lines. The constant cost lines in this figure represent trends only and should not be considered precise in shape or value. The minimum cost configuration is the configuration defined by the point of tangency between the lowest cost curve and the locus of minimum weight configurations.

3-4.1.3.2.3 Cost Effectiveness

A preferred optimization criterion is maximum cost effectiveness (effectiveness per unit cost) for some appropriate measure of effectiveness. As an illustration of this method, radius of action will be used as the measure of effectiveness. The curves of Fig. 3-109 are valid only for a given radius of action. Hence, the feasible loci of minimum weight configurations in Figs. 3-110 and 3-112 also are valid only for that single radius of action. Loci of minimum weight configurations for several radii of action Rg_1, Rg_2, Rg_3, (dashed lines) are shown in Fig. 3-113 superimposed upon constant cost lines. Also shown are lines of constant radius of action per unit cost (solid lines). These cost-effectiveness lines surround a "peak" because the value of cost-effectiveness is increasing on the concave sides of the curves. The maximum cost-effectiveness configuration is found at the top of the peak.

The cost-effectiveness method also can be used with a number of other effectiveness measures such as endurance or payload.

3-4.2 MISSION PERFORMANCE

This paragraph presents the methods used to calculate helicopter mission performance capabilities for a specific configuration. A typical helicopter mission profile is depicted in Fig. 3-114. Noted on the profile are the mission elements that are analyzed when determining the mission performance capability of a configuration. To calculate the performance, it first is necessary to determine the aircraft maximum gross weight at takeoff. The takeoff capability is dependent upon

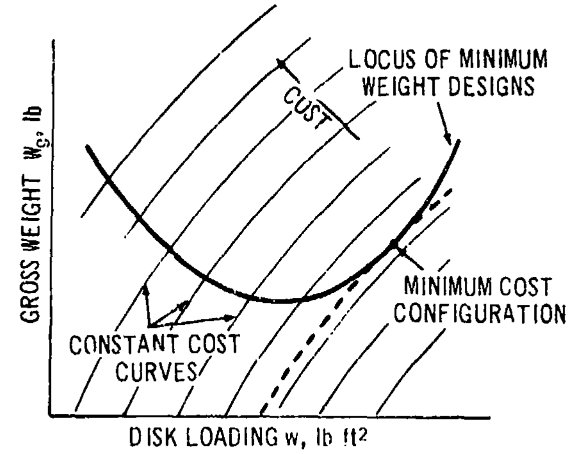


Fig. 3-112. Minimum Cost Configurations

ambient conditions and the takeoff criteria. For helicopters, the takeoff criteria usually are based upon either hover out-of-ground-effect or hover in-ground-effect capability. Once the maximum takeoff gross weight capability has been determined and the elements of the mission specified, the mission fuel requirements can be calculated. Fuel allowances are calculated for engine and rotor startup and pretakeoff checks (usually termed "warmup"), hover, climb to cruise altitude, cruise over the specified range or for the specified endurance, and hover time at landing. From a given maximum takeoff gross weight, which includes the fuel requirements, and with the weight of the aircraft, associated equipment, and crew defined, the payload capability for the mission can be determined.

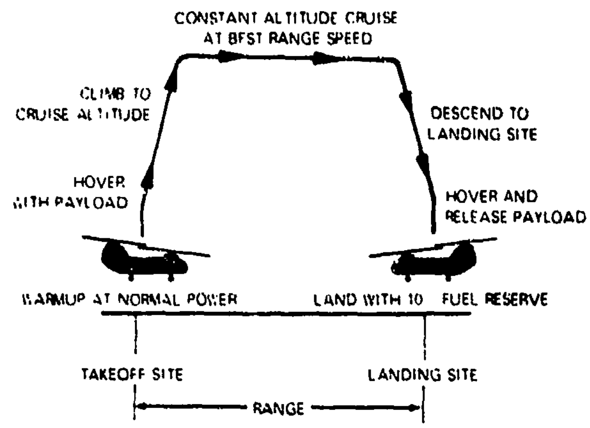


Fig. 3-114. Typical Mission Profile

3-4.2.1 Hover Ceiling

3-4.2.1.1 General

Hover ceiling is defined as the maximum altitude at which a given helicopter can remain aloft stationary with respect to the ground under zero wind velocity conditions. Helicopter capabilities are affected markedly by ambient temperature and proximity to the ground.

Fig. 3-115 represents a typical format used to show helicopter hover ceiling capability when hovering at a wheel (skid) height greater than approximately two rotor diameters above the ground, which is termed hovering OGE, and when hovering close to the ground, which is termed hovering IGE. The upper portion of the chart shows OGE performance capabilities over a range of pressure altitudes and ambient temperature conditions. By entering the chart at specified altitude and temperature conditions, the OGE gross weight capability can be read at the corresponding point on the gross weight scale. By following a path to the appropriate wheel height parallel to the guidelines shown on the bottom of the chart, the IGE hover gross weight capability can be determined. This format generally is used when presenting hover capabilities in the pilot's operating manual, and in technical documents such as detail specifications and proposals.

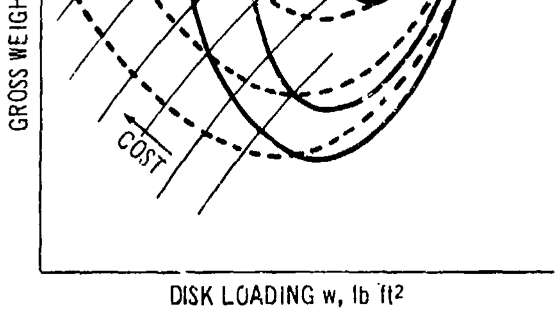


Fig. 3-113. Cost-effectiveness Optimization

The reduction in gross weight capability that is displayed on the hover chart at high altitude and high temperature is characteristic of turbine-powered helicopters. The decreased capability is a result principally of the engine-power-available characteristics. Power required for constant gross weight is relatively insensitive to altitude and temperature variations.

Hover ceiling capabilities generally are shown for a gross weight range extending from the minimum operating weight to the maximum gross weight approved for the aircraft and for a range of altitudes extending from sea level to the maximum operating altitude. Transmission torque limitations on power available, and altitude limitations on specific components, may restrict the performance capabilities of the aircraft. On flight handbook and detail specification charts, ambient temperatures for the approved or design operating range generally are shown. For specific technical investigations or system analysis studies, the temperature and altitude ranges generally are narrower and may be limited to the temperature or altitude of interest.

The variation in gross weight capability as a function of wheel or skid height shows that the proximity of the ground influences, by its effect upon the downwash field, the performance of a rotor. As the helicopter approaches the ground in hover, the blade pitch required to produce a given thrust is reduced, with a resultant decrease in induced power required. When this condition exists, the helicopter is hovering IGE. As the height above the ground is increased, the benefit of ground effect decreases, resulting in an increase in induced power. A point finally is reached at which the height above the ground can be increased further with no resultant increase in hover power required. At this point, the aircraft is hovering OGE. The significant parameter determining the benefit derived from ground effect is the ratio of rotor height above the ground to rotor diameter Z/R. Based upon measured helicopter characteristics, there is little evidence of ground effect above $Z/R = 2.0$.

When studying specific configurations, the distance between the rotor hub and the wheel or skid is accounted for and performance usually is shown as a function of wheel or skid height. The minimum recommended wheel height is determined from flying qualities and controllability characteristics when operating near the ground (2-ft skid height).

3-4.2.1.2 Method of Analysis

The analysis method flow diagram in Fig. 3-116 depicts the steps used when calculating hover ceiling capability. Chart A of Fig. 3-116 depicts the installed shaft horsepower characteristics as a function of ambient temperature for various altitude conditions. Installed shaft horsepower is developed from the basic uninstalled engine power available charactersitics with adjustments that account for losses from elements of the airframe inlet and exhaust system; e.g., inlet geometry and screens, particle separators, and infrared suppressors. The adjustments also include an allowance for bleed air extracted from the compressor stages for anti-ice protection of engine inlets when operating in cold ambient conditions or for cabin and cockpit air-conditioning systems when operating in hot ambient conditons. The engine power rating used for calculating hovering capability is selected based upon the duration of hover specified for the mission. Maximum power is used when the hover portion of the mission is less than 5 or 10 min in duration. Intermediate power is used for hover of up to 30 min and maximum continuous power

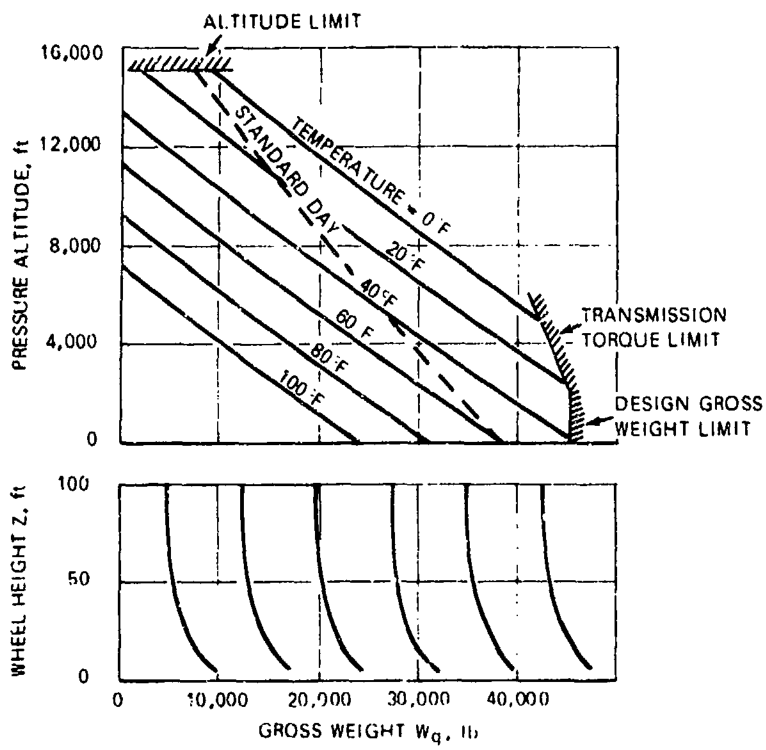


Fig. 3-115. Hover Ceiling

is used for missions requiring hover periods in excess of 30 min. Chart B of Fig. 3-116 depicts the aircraft power required characteristics for hover OGE as a function of the weight coefficient C_W, defined as $W_t/[\rho A(\Omega R)^2]$. The power required may be determined theoretically or by testing of models or full-scale configurations. When the power required is derived theoretically, it is developed for the basic rotor (see Fig. 3-117) and adjustments are made to account for vertical drag on the fuselage due to rotor downwash; tail rotor losses (for single-rotor helicopters) or rotor interference losses (for an overlapped tandem-rotor helicopter); transmission system losses; accessory power required for pumps, blowers, and electrical generators; and additional power must be added when the helicopter is to be operated at critical combinations of thrust and high tip Mach numbers where significant compressibility losses are incurred. These adjustments between basic isolated rotor performance and actual aircraft performance are illustrated clearly in Fig. 3-117.

To calculate hover ceiling capability, enter Chart A of Fig. 3-116 at the ambient conditions desired. The example is shown for ambient conditions of 5000 ft pressure altitude and a temperature of 95°F. At those ambient conditions, the installed shaft horsepower can be determined and the nondimensional power coefficient, C_P, can be calculated. Using the ambient temperature, the Mach number of the blade tip M_0 can be calculated by dividing the tip speed ΩR by the speed of sound a at that temperature. Chart B is then entered at the power coefficient and Mach number to determine the weight coefficient out-of-ground-effect $C_{W_{OGE}}$, which is converted to gross weight capability and plotted on the hover ceiling chart (Chart D) for the selected ambient temperature. Similarly, Chart C is used to determine the gross weight capability when hovering IGE at the example wheel height of 30 ft, and is the basis for the IGE hover ceiling chart (Chart E).

3-4.2.2 Payload

3-4.2.2.1 General

Helicopter mission performance criteria generally require transportation of a specified payload over a specified distance. The payload may be in the form of troops, troop supplies and ammunition, weapons, or vehicles. Many of these items can be loaded efficiently into the cabin of the helicopter. In many instances, however, the dimensions of the load exceed those of the cabin. Because of the ability of the helicopter to hover over a fixed point, these loads may be suspended and transported externally to the cabin. In some cases it simply is more efficient to acquire and deposit the cargo externally. Because the externally suspended cargo imposes a drag penalty upon the configuration, speed capability and range capability are reduced. Fig. 3-118 depicts typical characteristics for internal and external payload range capabilities of a helicopter.

As for any aircraft operating at a constant takeoff gross weight, when fuel is added to increase the range capability, a commensurate reduction in payload is necessary. When the internal fuel capacity is reached, large reductions in payload produce relatively small increases in range capability. The range capability can be extended significantly, however, by adding internally or externally mounted auxiliary fuel tanks, as shown by the dashed line on Fig. 3-118. Limitations upon the payload capability can result from space restrictions or from floor loading limitations for internal cargo, and from the design limitations of the cargo hook in the case of external loads.

Payload capability for a helicopter is defined as the takeoff gross weight less the sum of the mission fuel required, the fixed useful load, and the aircraft empty weight. Aircraft empty weight and fixed useful load are inherent characteristics of any specific configuration. Takeoff gross weight is dependent upon the takeoff criteria specified. Required mission fuel is dependent upon the mission profile specified and normally includes a 5% increase over that which is calculated by using the engine manufacturer's specification. These items are discussed in more detail in the paragraphs that follow.

3-4.2.2.2 Definition of Weights

The takeoff and landing gross weights of the helicopter are comprised of the elements listed in par. 10-2.6.

3-4.2.2.2.1 Takeoff Gross Weight

Examples of takeoff criteria normally used to define mission takeoff gross weight are hover OGE, hover IGE at a specified wheel height, vertical takeoff at a specified vertical rate of climb, and structural limitations. For gross weights in excess of those that allow hover IGE capability, a rolling takeoff may be specified. In some cases, the takeoff gross weight may be established by the en route cruise capability with one engine inoperative or by performance requirements specified for the helicopter at a given landing site. The structural limitation on takeoff gross weight may be any one of the design limitations, such as basic structural design gross weight or alternate design gross weight, depending upon the mission.

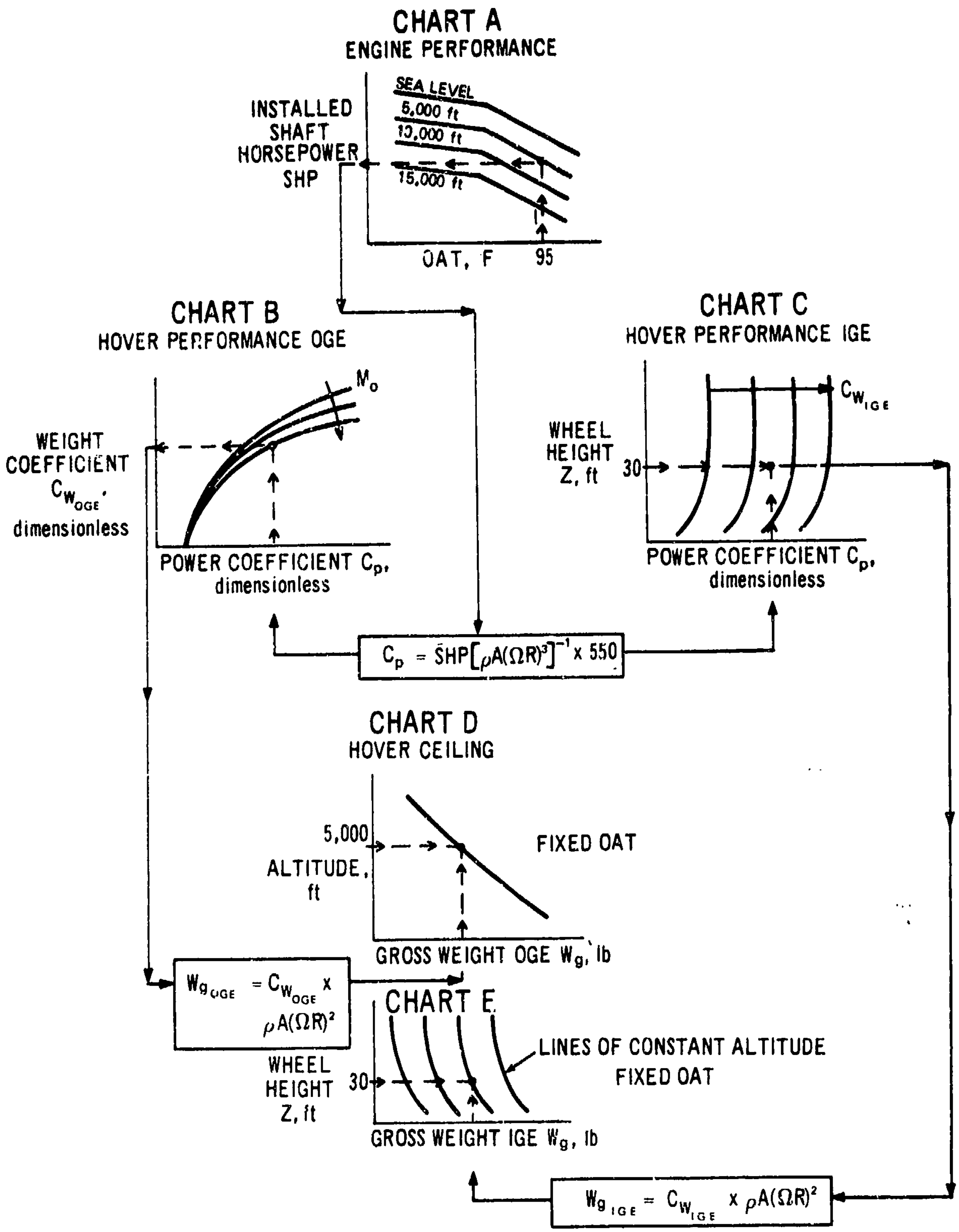


Fig. 3-116. Hover Ceiling Calculation Procedure

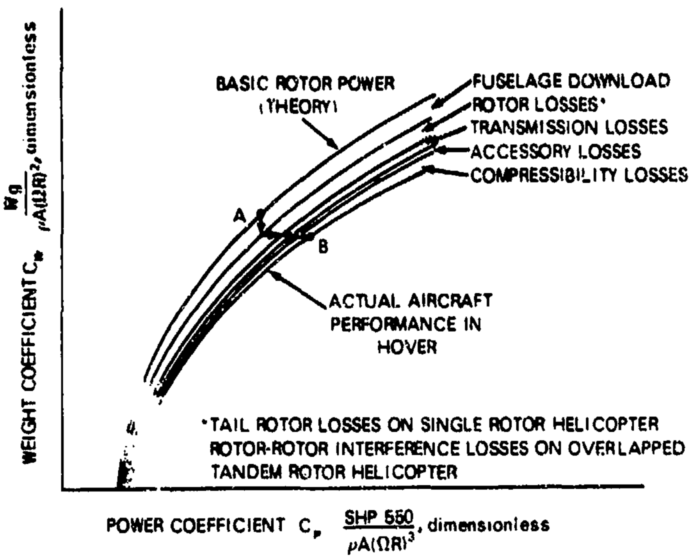


Fig. 3-117. Aircraft Hover Power Development

3-4.2.2.2.2 Empty Weight

The items that make up the empty weight of a helicopter are given in par. 10-2.6. They comprise a completely assembled, ready-to-fly helicopter, and include the fluids required for operation of various systems, e.g., transmission oil and hydraulic fluid. Not included in empty weight are the trapped fluids for engine lubrication or the trapped or unusable fuel in the fuel system. Special items to be considered as part of the empty weight of a given helicopter will be incorporated in the applicable model specification. For special missions requiring maximum useful load, it may be desirable to reduce the empty weight by stripping, i.e., removing nonessential items. The extent of the stripping is dependent upon the objective of the mission and upon whether or not the stripping can be accomplished in the field.

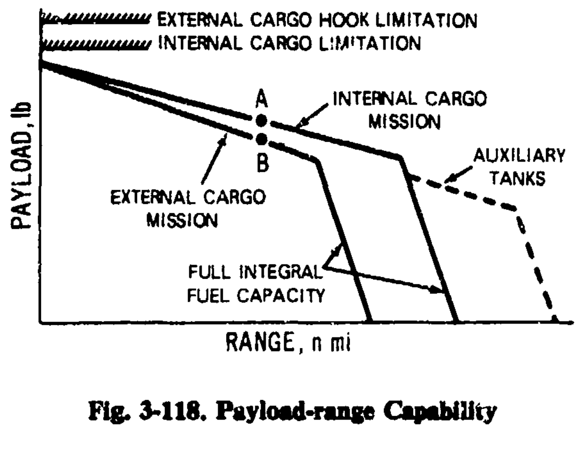


Fig. 3-118. Payload-range Capability

3-4.2.2.2.3 Basic Weight

The basic weight comprises the empty weight, all fixed operating equipment, mission armament, auxiliary fuel tanks, ballast, and trapped and unusable fuel and oil. The basic weight of a given aircraft will vary with structural modifications and changes in fixed operating equipment. The term "basic weight", when qualified with a word indicating the type of mission involved—basic weight for combat, basic weight for ferry—may be used in conjunction with directives specifying equipment for those missions.

In preliminary design, the weight of internal auxiliary fuel tanks (bladder) is estimated based upon 0.3 lb/gal of auxiliary fuel, while external auxiliary tank weight is estimated based upon 0.5 lb/gal of auxiliary fuel.

3-4.2.2.2.4 Fixed Useful Load

Fixed useful load is comprised of items such as oil, crew, crewmen's baggage, and emergency and other specified auxiliary equipment.

3-4.2.2.2.5 Operating Weight

Operating weight is the sum of the empty weight and the fixed useful load.

3-4.2.2.2.6 Useful Load

Useful load includes all items from the fixed useful load list plus fuel, cargo, ammunition, bombs, drop loads, passengers, and external auxiliary fuel tanks (if they are to be disposed of during flight).

3-4.2.3 Mission Profile

3-4.2.3.1 Mission Profile Definition

Fuel calculations for determining the payload-range capabilities of helicopters are dependent upon the mission profile specified. Innumerable variations in mission profile can be developed by rearrangement of a few basic elements. The typical mission profile is depicted schematically in Fig. 3-114.

The engine fuel consumption rate *shall* be increased 5% over that quoted by the engine manufacturer to compensate for variations among service aircraft and operating techniques. The fuel grade is generally that which is considered standard for the engine(s) installed, and the fuel weight is based upon the associated fuel density. Typical helicopter turbine engine fuels include JP-4 grade at 6.5 lb/gal and JP-5 grade at 6.8 lb/gal. Deviations from these expressed normal conditions are cited in MIL-M-7700 and MIL-C-5011.

3-4.2.3.2 Mission Profile Elements

The following paragraphs detail considerations affecting weight and fuel consumption for each phase of the typical mission profile depicted in Fig. 3-114. To simplify data presentation the concept of referred or generalized values is introduced at this time by use of generalizing coefficients δ and θ, where δ is the ambient pressure ratio $p/p_{std_{sl}}$ and θ the ambient temperature ratio $T/T_{std_{sl}}$.

3-4.2.3.2.1 Warmup

Engine start and aircraft checkout commonly are termed "warmup". A normal fuel allowance for warmup is 5 min at normal power and is calculated by use of a chart similar to Fig. 3-119 prepared from the manufacturer's performance data for the engine installed. The method is as follows:

1. For the specified ambient condition, calculate the generalized shaft horse power $SHP/(\delta\sqrt{\theta})$ using maximum continuous power.

2. Enter the chart for the calculated value of $SHP/(\delta\sqrt{\theta})$ and obtain generalized fuel flow $W_f/(\delta\sqrt{\theta})$ using the 5% increase curve.

3. Calculate the fuel required for warmup using the following equation and the appropriate warmup time Δt in hours:

$$\text{Fuel Required} = \left(\frac{W_f}{\delta\sqrt{\theta}}\right)(\delta\sqrt{\theta})(\Delta t) \quad \text{, lb} \qquad (3\text{-}162)$$

where

W_f = fuel flow, lb/hr

δ = ambient pressure ratio, $p_{amb}/p_{std_{sl}}$, dimensionless

θ = ambient temperature ratio, $T_{amb}/T_{std_{sl}}$ dimensionless

Δt = time increment, hr

$\delta\sqrt{\theta}$ = generalizing coefficient, dimensionless

3-4.2.3.2.2 Hover at Takeoff

In many cases, the hover time at takeoff is negligible and is not included in the mission profile, particularly when payloads are carried internally. However, the hover time required for external load acquisition can be significant, and in this case the fuel requirements must be considered in determining the total mission fuel. By using the hover performance and engine fuel flow characteristics shown in Fig. 3-120, fuel required for hover is obtained as follows:

1. For the specified ambient condition (OGE) and the takeoff gross weight, calculate the dimensionless weight coefficient $C_W = W_g/(\rho A \Omega^2 R^2)$.

2. Enter Chart A of Fig. 3-120 and read the dimensionless power coefficient C_P required to hover at takeoff.

3. Convert C_P to generalized engine power required $SHP/(\delta\sqrt{\theta})$ and enter Chart B to determine the generalized engine fuel flow $W_f/(\delta\sqrt{\theta})$.

4. Calculate the fuel burned during hover at takeoff by using Eq. 3-162 with the appropriate hover time.

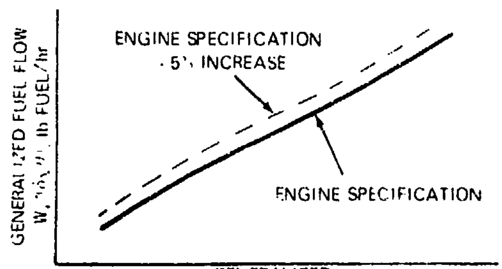


Fig. 3-119. Engine Fuel Flow Characteristics (Typical)

3-4.2.3.2.3 Climb to Cruise Altitude

Time, distance, and fuel required to climb to cruise altitude are accounted for when determining absolute operational capabilities of a helicopter, but usually are ignored when simple, comparative analyses are conducted. Often, when climb fuel allowance is accounted for, no distance credit is taken. Fig. 3-121 is a typical chart showing time, distance, and fuel required to climb. The development of this chart is presented in par. 3-4.2.6.4.

3-4.2.3.2.4 Cruise at Constant Altitude

Cruise generally is conducted at best range speed but conditions may require maximum speed as limited by the normal power of the engines. In some cases, maximum speed may be limited by the structural capabilities of the helicopter or, for external load missions, the speed may be limited by the aerodynamic instability of the external load. This is depicted in the helicopter speed capability chart of Fig. 3-122.

After establishing the aircraft takeoff gross weight, the mission profile (see Fig. 3-114), the desired range, and the range-fuel data presented in Figs. 3-123 and 3-124, the fuel for cruise can be determined. A detailed description of the development of Figs. 3-123 and 3-124 is presented in par. 3-4.2.4. The specific range represents the nautical miles traveled per pound of fuel

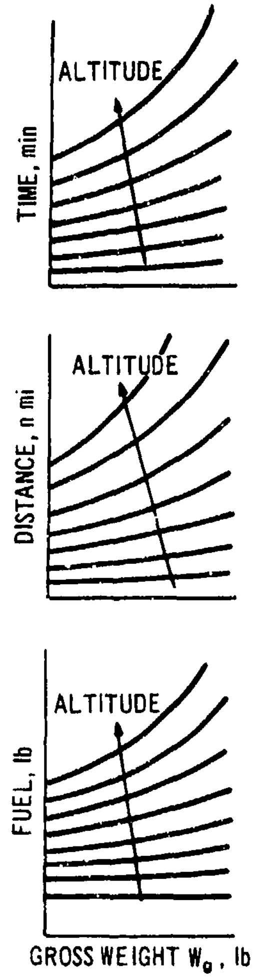


Fig. 3-121. Time, Distance, and Fuel To Climb from Sea Level

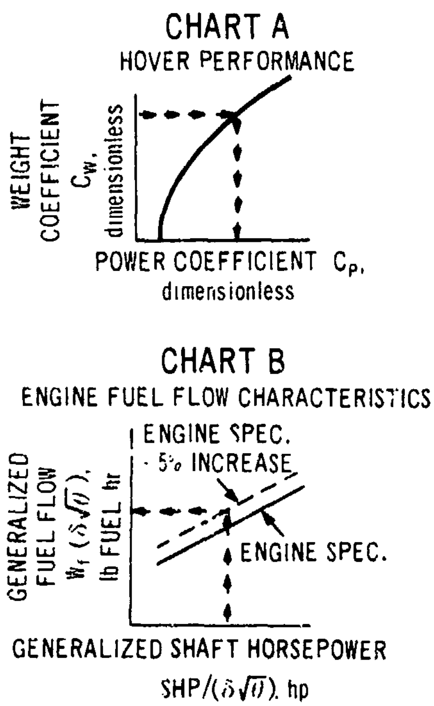


Fig. 3-120. Hover Fuel Calculation

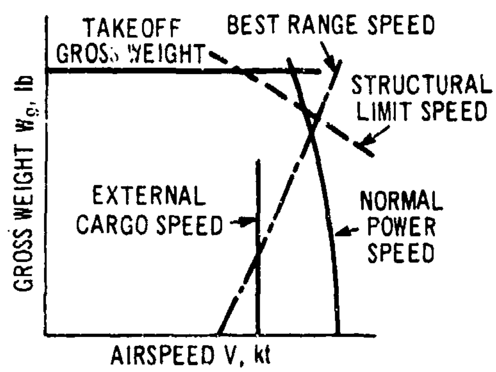


Fig. 3-122. Helicopter Speed Capability

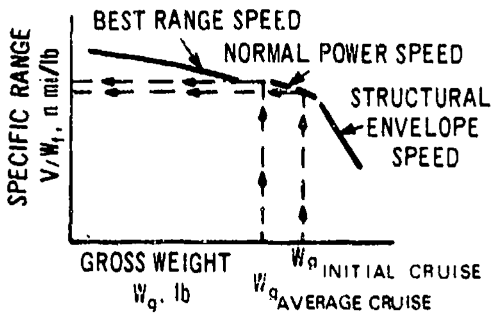


Fig. 3-123. Specific Range Performance

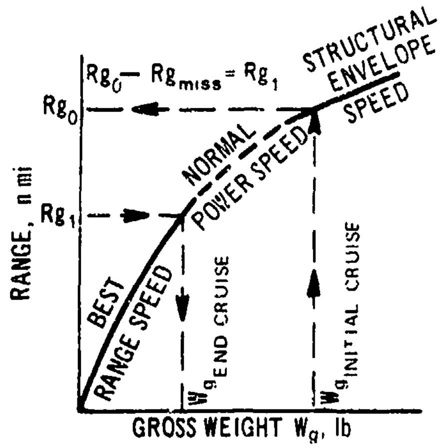


Fig. 3-124. Range Index Curve

burned. The weight of fuel burned in a mission of range Rg (n mi) in time t (hr) is f_w (lb). Thus, the specific range Rg/f_w can also be expressed as $(Rg/t)/(f_w/t)$ or V/W_f (n mi/lb) where V is the true airspeed in kt and W_f the fuel flow in lb/hr. In order to use the range-fuel data of Figs. 3-123 and 3-124, the initial cruise gross weight must be determined. For the particular mission profile being discussed, the helicopter must warm up, hover, and climb before initiating the cruise portion of the mission. Therefore, the fuel burned during hover, warmup, and climb is subtracted from the takeoff gross weight to obtain the initial cruise gross weight. By using the specific range performance of Fig. 3-123, the cruising fuel required is determined as follows:

1. Enter Fig. 3-123 at initial cruise gross weight and read the associated specific range.

2. Calculate the approximate cruise fuel required for the mission by dividing the desired range by the specific range; i.e., Fuel (lb) = Range (n mi)/Specific Range (n mi/lb).

3. From the initial cruise gross weight subtract one-half of the fuel requirement obtained in the steps just discussed to obtain an approximate average cruise gross weight.

4. Re-enter Fig. 3-123 at the average cruise gross weight and read a new specific range.

5. Recalculate the fuel required as in step 2 and determine a new average cruise gross weight.

The iteration process is required because the average gross weight over the cruise portion of the mission is less than the initial cruise gross weight by one-half of the fuel burned. Because the specific range performance increases as gross weight decreases, a fuel requirement based upon the initial gross weight will be greater than necessary.

Normally, only one or two iterations are required to converge average mission gross weights to a point where the fuel requirement change is insignificant. In a few cases, however, a third iteration may be necessary before this condition is reached. An alternative, more direct method for determining cruise fuel is to use a range index curve (see Fig. 3-124). This curve also is entered at the initial cruise gross weight, but a range Rg_0 is read. From Rg_0 the desired mission range Rg_{miss} is subtracted yielding Rg_1. Reenter the range index curve at Rg_1 and obtain the end cruise gross weight. The difference between initial cruise and end cruise gross weights is the cruise fuel required.

3-4.2.3.2.5 Descent to Landing Site

The fuel used during the descent from cruise altitude usually is considered negligible and no distance credit is allowed.

3-4.2.3.2.6 Hover at Landing Site

As with hover at takeoff, hover time at landing usually is negligible for internal load missions. With external cargo, the hover time is significant due to the time involved in depositing the cargo. The method of obtaining the fuel requirement is identical to that described for hover at takeoff with the exception that the gross weight at the end of the cruise is used to establish the power required.

3-4.2.3.2.7 Reserve Allowance

Reserve fuel allowance often is specified as 10% of the total initial fuel at engine start. For very short missions this results in a very small reserve allowance; therefore, 20- or 30-min cruise at best endurance speed or best range speed is used instead. Employing the elements of the mission profile of Fig. 3-114, the total fuel, less reserve, required for the mission is the sum of the fuel required for warmup, hover at takeoff, climb to altitude, cruise, and hover at landing. For a 10% fuel reserve, the total fuel and reserve fuel are expressed by the following equations:

$$\left.\begin{aligned}&\text{Total Fuel} = (\text{Total Fuel Less Reserve})/0.9\\&\text{and}\\&\text{Reserve Fuel} = 0.11\ (\text{Total Fuel Less Reserve})\end{aligned}\right\}\quad(3\text{-}163)$$

A typical summary payload-range calculation chart is presented in Fig. 3-125, and includes a step-by-step procedure for calculation of the fuel required for each of the mission elements and the associated payload for a given range, using the specific range performance method. This chart consists of the flight condition, the step number to indicate the correct order for calculation, the item being determined, and the appropriate unit of each item. The column to the right is to be filled in with the values associated with the specified payload-range mission. When used in conjunction with the previously discussed mission elements, the chart provides an orderly and accurate approach for obtaining helicopter payload-range capabilities.

3-4.2.3.3 External Load Mission

When the payload is to be carried in the form of external cargo, the payload-range capability of the helicopter may be reduced significantly (see Fig. 3-116). The reason for this reduction is the increased drag produced by the load. For a fixed airspeed V the additional power ΔSHP required is proportional directly to the increased drag ΔD. This increase in power required results in a higher engine fuel flow rate; therefore, the specific range V/W_f is lower for external load missions than for internal load missions. In addition, the safe airspeed for transport of an external load may be lower than the speed for best range, causing a further reduction in specific range (Fig. 3-126).

For fixed conditions of mission profile, takeoff gross weight W_g, empty weight W_e, and range Rg, the fuel required to perform the mission with an external load will be greater than that required with an internal load. As a result, the payload will be smaller. This fact is illustrated by Points A and B on Fig. 3-118.

3-4.2.3.4 Effect of Altitude on Payload-range Capability

The typical effect of increasing cruise altitude on payload-range capability is depicted in Fig. 3-127. At a high takeoff gross weight W_{g_2}, the payload-range capability is better for cruising at sea level than for cruising at altitude. But for a low takeoff gross weight W_{g_1}, cruising at altitude provides the best payload-range capability. This trend reversal is a result of the nature of the range-fuel performance shown by the specific range performance curve in Fig. 3-128.

The development of specific range performance is explained in detail in par. 3-4.2.4 and is used here only to clarify the effect of altitude upon payload-range capability. At a high takeoff gross weight W_{g_2}, the specific range is much better at sea level than at 5000 ft; i.e., for a fixed range, the fuel burned will be less and the payload will be greater for the sea level cruise or, if the fuel is held constant, the range for sea level cruise will be greater. For a low takeoff gross weight, the specific range parameter shows the reverse effect; i.e., cruising at altitude provides better specific range capability than cruising at sea level.

For short-range missions, the time and fuel required to climb to the best cruising altitude usually negate the benefit of the better cruise performance attained. However, for ferry missions over long distances, best-range capability is obtained by climbing to maintain the optimum cruise altitude as fuel is burned off, as indicated by the dashed curve of Fig. 3-128.

3-4.2.4 Range

Range capability of a helicopter is a product of the fuel consumption rate of the configuration at the specified cruise speed. The cruise speed specified is dependent upon the objectives of the mission. The paragraphs that follow describe how range capability varies as a function of speed, cruise altitude, gross weight, and wind strength and direction.

3-4.2.4.1 Specific Range Parameter Computation

In order to compute the specific range parameter, which is a measure of the range capability of the configuration, the following basic information must be available:

1. Engine fuel-flow characteristics as illustrated in Chart A, Fig. 3-129

2. Power-required characteristics as illustrated in Charts B_1 and B_2, Fig. 3-129.

Chart A of Fig. 3-129 is typical of turboshaft engine fuel-flow characteristics. Over a substantial portion of the useful range of the curve, the fuel-flow characteristics may be approximated by the linear equation $W_f/(\delta\sqrt{\theta}) = \alpha + \beta SHP/(\delta\sqrt{\theta})$, where α and β are constants representing the y-axis intercept and slope, respectively, of the relationship between generalized fuel flow rate $W_f/(\delta\sqrt{\theta})$ and generalized shaft horsepower $SHP/(\delta\sqrt{\theta})$ This is a convenient form for the computation that follows. (The actual engine characteristics are obtained from the manufacturer's detail specification.)

Charts B_1 and B_2 of Fig. 3-129 represent two common methods of presenting helicopter performance data. Chart B_1 is the dimensionless C_P, C_T, μ, and hovering tip Mach number M_0 method; Chart B_2 is the generalized shaft horsepower $SHP/(\delta\sqrt{\theta})$, generalized gross weight W_g/δ, generalized true airspeed $V/\sqrt{\theta}$, and generalized rotor speed $N/\sqrt{\theta}$ method. The equivalence of these two methods can be shown as follows: the thrust coefficient C_T is defined by Eq. 3-11.

MISSION FUEL REQUIREMENTS

CONDITION	STEP NO.	ITEM	UNITS	PAYLOAD-RANGE MISSION
PRE-TAKEOFF	1	TAKEOFF GROSS WEIGHT (TOGW)	lb	
	2	WARMUP FUEL	lb	
	3	HOVER FUEL AT TOGW	lb	
CLIMB TO CRUISE	4	INITIAL CLIMB GW	lb	
	5	RATE OF CLIMB	fpm	
	6	TIME TO CLIMB TO ALTITUDE	min	
	7	SPEED FOR BEST CLIMB	kt	
	8	DISTANCE COVERED IN CLIMB	n mi	
	9	FUEL USED IN CLIMB	lb	
CRUISE OUTBOUND	10	INITIAL CRUISE GW	lb	
	11	INITIAL SPECIFIC RANGE	n mi lb	
	12	CRUISE DISTANCE-RANGE	n mi	
	13	APPROXIMATE CRUISE FUEL	lb	
	14	AVERAGE CRUISE GROSS WEIGHT	lb	
	15	AVERAGE SPECIFIC RANGE	n mi lb	
	16	ACTUAL CRUISE FUEL	lb	
DESCENT	17	INITIAL DESCENT GW	lb	
	18	RATE OF DESCENT	fpm	
	19	TIME TO DESCEND TO ALTITUDE	min	
	20	FUEL USED FOR DESCENT	lb	
OPERATION AT LANDING SITE	21	LANDING GW	lb	
	22	HOVER FUEL AT LANDING GW	lb	

PAYLOAD DETERMINATION

DETERMINE OPERATING WT	A	WEIGHT EMPTY	lb	
	B	FIXED USEFUL LOAD	lb	
	C	OPERATING WT	lb	
DETERMINE MISSION FUEL & RESERVE FUEL	D	WARMUP FUEL	lb	
	E	HOVER FUEL	lb	
	F	CLIMB FUEL	lb	
	G	CRUISE FUEL	lb	
	H	DESCENT FUEL	lb	
	I	HOVER FUEL	lb	
	J	MISSION FUEL LESS RESERVE	lb	
	K	RESERVE FUEL	lb	
	L	TOTAL MISSION FUEL	lb	

PAYLOAD = TAKEOFF GROSS WEIGHT − OPERATING WEIGHT − TOTAL MISSION FUEL

Fig. 3-125. Payload-range Calculation Chart

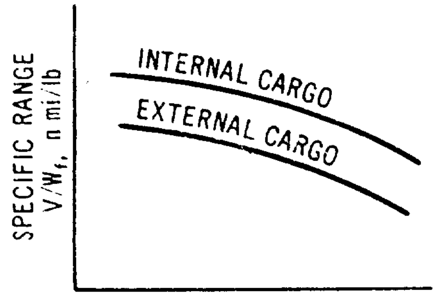


Fig. 3-126. Specific Range Performance

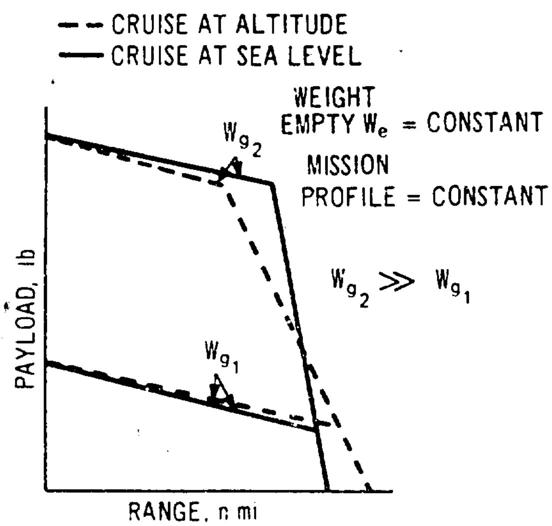


Fig. 3-127. Payload-range Capability

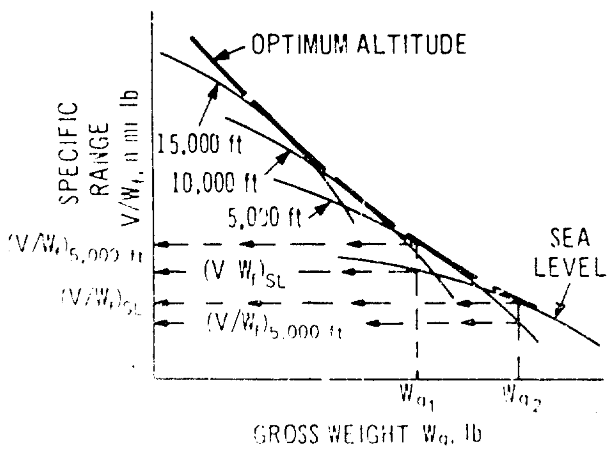


Fig. 3-128. Optimum Specific Range Performance

By using the standard definition of $\sigma = \rho/\rho_0$, the following equation is obtained:

$$C_T = \frac{W_g}{\sigma\rho_0 A(\Omega R)^2} \text{, dimensionless} \quad (3\text{-}164)$$

Also, $\sigma = \delta/\theta$, and tip speed ΩR can be expressed as a function of rotor speed N (rpm) by $(\pi R/30)N$. Thus,

$$C_T = \frac{W_g/\delta}{\rho_0 A\left(\frac{\pi R}{30}\right)^2\left(\frac{N}{\sqrt{\theta}}\right)^2}$$

$$= \left(\frac{900}{\rho_0 A\pi^2R^2}\right)\frac{W_g/\delta}{(N/\sqrt{\theta})^2} \text{, d'less} \quad (3\text{-}165)$$

Because the grouped terms are constant for a specific configuration, it can be concluded that generalized data presented at constant W_g/δ and constant $N/\sqrt{\theta}$ is equivalent to presenting data at constant C_T as in the dimensionless method.

In a similar manner it can be shown that the dimensionless power coefficient C_P is equivalent to

$$C_P = \frac{550(30)^3}{\rho_0 A\pi^3R^3}\left[\frac{SHP/(\delta\sqrt{\theta})}{(N/\sqrt{\theta})^3}\right] \text{, d'less} \quad (3\text{-}166)$$

Again the grouped terms are constant for a specific configuration, and data presented in the generalized method are equivalent to those in the dimensionless method.

The final comparison of the two methods is made with respect to forward speed and advancing tip Mach number M_t. The advance ratio μ is defined as the ratio of forward speed V to rotor tip speed ΩR and therefore can be expressed as

$$\mu = \frac{30}{\pi R}\left(\frac{V/\sqrt{\theta}}{N/\sqrt{\theta}}\right) \text{, dimensionless} \quad (3\text{-}167)$$

Thus, data generalized in terms of $V/\sqrt{\theta}$ and $N/\sqrt{\theta}$ can be related directly to advance ratio for a specific configuration. The advancing tip Mach number $M_{1,0,90}$ is given by:

$$M_{1,0,90} = M_0(1+\mu) \text{, dimensionless} \quad (3\text{-}168)$$

where

M_0 = rotor tip Mach number hovering, $(\Omega R)/a$, dimensionless

$M_{x,\psi}$ = rotor blade Mach number at radius ratio (r/R) x and azimuth angle ψ, dimensionless

Substituting for M_0 and using the relationship $a = a_0\sqrt{\theta}$, where a_0 is the speed of sound at standard sea level temperature, the following expression is obtained:

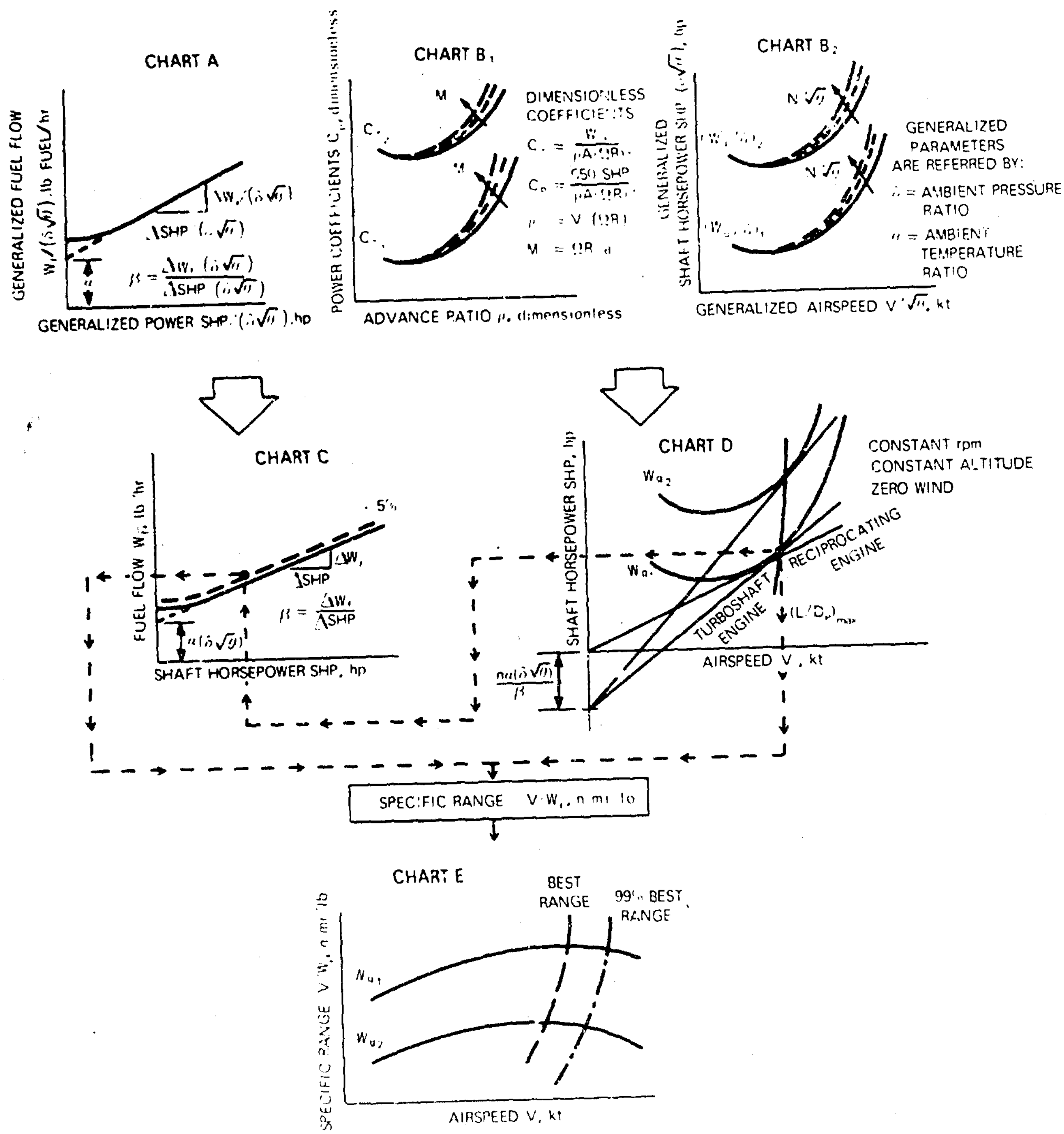


Fig. 3-129. Elements Required for Specific Range Computation

$$M_{1.0,90} = \frac{\Omega R}{a}\left(1 + \frac{V}{\Omega R}\right)$$

$$= \frac{\pi R}{30 a_0}\left(\frac{N}{\sqrt{\theta}}\right)\left(1 + \frac{30}{\pi R}\left[\frac{V/\sqrt{\theta}}{N/\sqrt{\theta}}\right]\right) \quad (3\text{-}169)$$

Again it is shown that the advancing tip Mach number, which is experienced at a given value $V/\sqrt{\theta}$ and $N/\sqrt{\theta}$, can be expressed in terms of the corresponding μ and M_0 in the dimensionless system.

The previous discussion shows that the two methods can be used interchangeably and will give equal performance values for a given combination of flight speed, gross weight, and rotor speed, provided that the rotational tip Mach number M_0 is held constant. To obtain model or flight test power polars for a series of gross weight conditions to be used at a specified ambient temperature condition, it is necessary to hold a constant test tip Mach number M_0.

Even though the two methods are shown to be equivalent, the generalized method is becoming increasingly popular for two reasons:

1. The generalized method uses quantities that are of a familiar magnitude.
2. A significant number of performance comparisons are made at sea level standard conditions where δ and θ have values of unity, and thus the performance figures can be used directly in dimensional form. The dimensionless method requires elaborate calculations even for the standard day condition.

Helicopter performance data are obtained from the following sources:

1. Flight test
2. Wind tunnel model test
3. Theoretically estimated performance.

By applying the appropriate factors representing the specified ambient conditions, the dimensional forms of engine characteristics and aircraft performance (obtained from either the dimensionless or the generalized method) are obtained as illustrated in Charts C and D of Fig. 3-129. The engine specific fuel flow *shall* be increased by 5% to satisfy Military Specification requirements.

By combining the information on Charts C and D (Fig. 3-129), the specific range V/W_f can be determined as a function of airspeed as illustrated by Chart E. The maximum value of specific range occurs at a speed designated as best-range speed. This speed corresponds to the point of maximum lift-to-drag ratio L/D_e, as illustrated in Chart D. D_e is the helicopter total effective drag (including consideration of a finite fuel flow α for a turboshaft engine at zero power) and is discussed in further detail in the next paragraph. Specific range *shall* be computed at an airspeed commensurate with the 99% best-specific-range value providing that airspeed does not exceed any airspeed limitations. The selected airspeed is the higher of the two speeds at which specific range is equal to 99% of the maximum available at a specific gross weight. This practice allows the helicopter to cruise at speeds 5-7% faster while sacrificing only 1% in range capability. Thus, a significant increase in mission speed is realized with an associated reduction in mission time and hence improved helicopter productivity. Furthermore, as fuel is used and gross weight decreases, the best-range airspeed increases. By starting cruise at the higher speed the helicopter remains at a better-than-99%-specific-range airspeed without constant changes in control settings.

The total helicopter effective drag D_e is derived in the following manner. For conventional fixed-wing aircraft with reciprocating or jet engine power plants, the total power requirement can be expressed as drag D times the aircraft velocity V, in knots, i.e.,

$$SHP = \frac{1.69VD}{550} \quad (3\text{-}170)$$

and the resulting value of SHP can be used to determine the fuel-flow rate. This definition also would apply to a helicopter with a reciprocating engine power plant. However, for a helicopter with turboshaft engine(s) the expression must be modified slightly because turboshaft engine fuel-flow characteristics vary slightly from the linear relationships that apply to reciprocating and jet engines. Chart C, Fig. 3-129, illustrates a typical turboshaft engine fuel-flow curve. Except at extremely low powers, the fuel-flow power relationship can be approximated by $W_f = \alpha(\delta\sqrt{\theta}) + \beta(SHP)$. The portion of the power spectrum for which this approximation is valid is normally between 30% and 100% of normal rated power. Assuming the fuel used prior to the production of useful SHP is proportional to the number of engines, the total fuel-flow rate can be expressed as

$$W_f = \beta\left[\frac{n\alpha}{\beta}(\delta\sqrt{\theta}) + SHP\right], \text{ lb/hr} \quad (3\text{-}171)$$

where

n = number of engines

The term $(n\alpha/\beta)(\delta\sqrt{\theta})$ represents an effective additional power when viewed relative to the engine fuel flow versus power slope (β), as illustrated in Chart D of Fig. 3-129. Extending the effective drag definition,

$$SHP + \frac{n\alpha}{\beta}(\delta\sqrt{\theta}) = \frac{1.69 VD_e}{550} = \frac{VD_e}{325} \quad (3\text{-}172)$$

an effective drag D_e is obtained that accounts for the basic fuel consumption of a turboshaft engine when producing zero power. Chart D graphically illustrates the effect of turboshaft fuel-flow characteristics upon cruise speeds. As a consequence of these characteristics, maximum available range with turboshaft engines occurs at higher cruise speeds than is the case with reciprocating engines.

3-4.2.4.2 Variation With Gross Weight

The method presented in the previous paragraph establishes the specific range parameter as a function of airspeed only. To determine the optimum helicopter range capability, it is necessary to determine the speed schedule that is flown as the gross weight decreases due to fuel burnoff. Charts B and C, Fig. 3-130, illustrate typical speed schedules.

At high gross weights, the helicopter speed may be limited by the structural flight envelope. At intermediate weights, the speed generally is limited by the maximum continuous power (normal power) available, corrected for ram effects. Noted on Chart B is the incremental power derived from ram effects at high cruise speeds. Due to the comparatively low forward flight speeds and the typical ram recovery characteristics of helicopter engine inlets, only a small increase in power available can be expected due to ram effects. Therefore, power correction for ram generally is ignored and static engine power often is used. At mid and low gross weight conditions, the helicopter is able to fly at 99% of best-range speed.

Charts D and E, Fig. 3-130, depict the effect of speed upon the specific range parameter using the engine fuel-flow characteristics of Chart A in the same manner as with Fig. 3-129. Chart D illustrates the fact that the normal power and structural envelope restrictions limit operation at high gross weights to speeds below the desired 99% best-range speed. Severe speed restrictions would be required, however, before an appreciable impact upon the aircraft range capability would occur. As shown, a 5-7% reduction in flight speed causes the aircraft to fly near the 100% best-range speed (better range); a further 5-7% airspeed reduction places the cruise speed at about 99% of best specific range speed and on the low-speed side of the optimum range speed, with no sacrifice in range. If the cruise speeds are restricted further, the range performance is reduced significantly. Chart E further illustrates the specific range available when flying within the constraints shown in Charts C and D.

3-4.2.4.3 Range Index Method

From the specified range cruise schedule presented in par. 3-4.2.4.2, the helicopter range performance can be computed as follows. During flight, a decrease in helicopter gross weight ΔW_g is equal to the amount of fuel consumed—Δ fuel $= \Delta W_g$. To determine the range capability for an initial weight W_{g_0} and final weight W_{g_1},

$$Rg = \int_{W_{g_0}}^{W_{g_1}} -\frac{V}{W_f}\, dW_g \quad \text{, n mi} \qquad (3\text{-}173)$$

This expression normally is evaluated by numerical integration to determine range capability. A useful method for solving range problems, which is helpful especially if a great number of missions are to be analyzed, is the range index curve. Fig. 3-131 illustrates the development of this curve.

Chart A of Fig. 3-131 shows the cruise speed capability associated with the specific range and range index curves of Charts B and C. This is a convenient presentation format and is used when calculating mission cruise time. From the incremental range equation, the index curves may be developed by cumulative integration of the expression. The trends shown in Chart C are indicative of the performance capabilities generally obtained. The integration normally is carried out over gross weights ranging from minimum operating weight to the aircraft maximum gross weight. The convex nature of the index lines indicates that the increment in available range increases with decreasing W_g for a constant amount of fuel burnoff.

Chart C also illustrates the use of a range index curve. The range capability is obtained by entering Chart B at the initial cruise gross weight W_{g_0} to obtain the Rg_0 index, and by entering at the final gross weight W_{g_1} (established by amount of fuel c[illegible]med) to determine the Rg_1 index. The mission ran[illegible] hen is given by $Rg_{miss} = Rg_0 - Rg_1$. Thus, the range can be determined for any amount of fuel used up to and including full fuel. The special case of a ferry mission is discussed in par. 3-4.2.4.6.

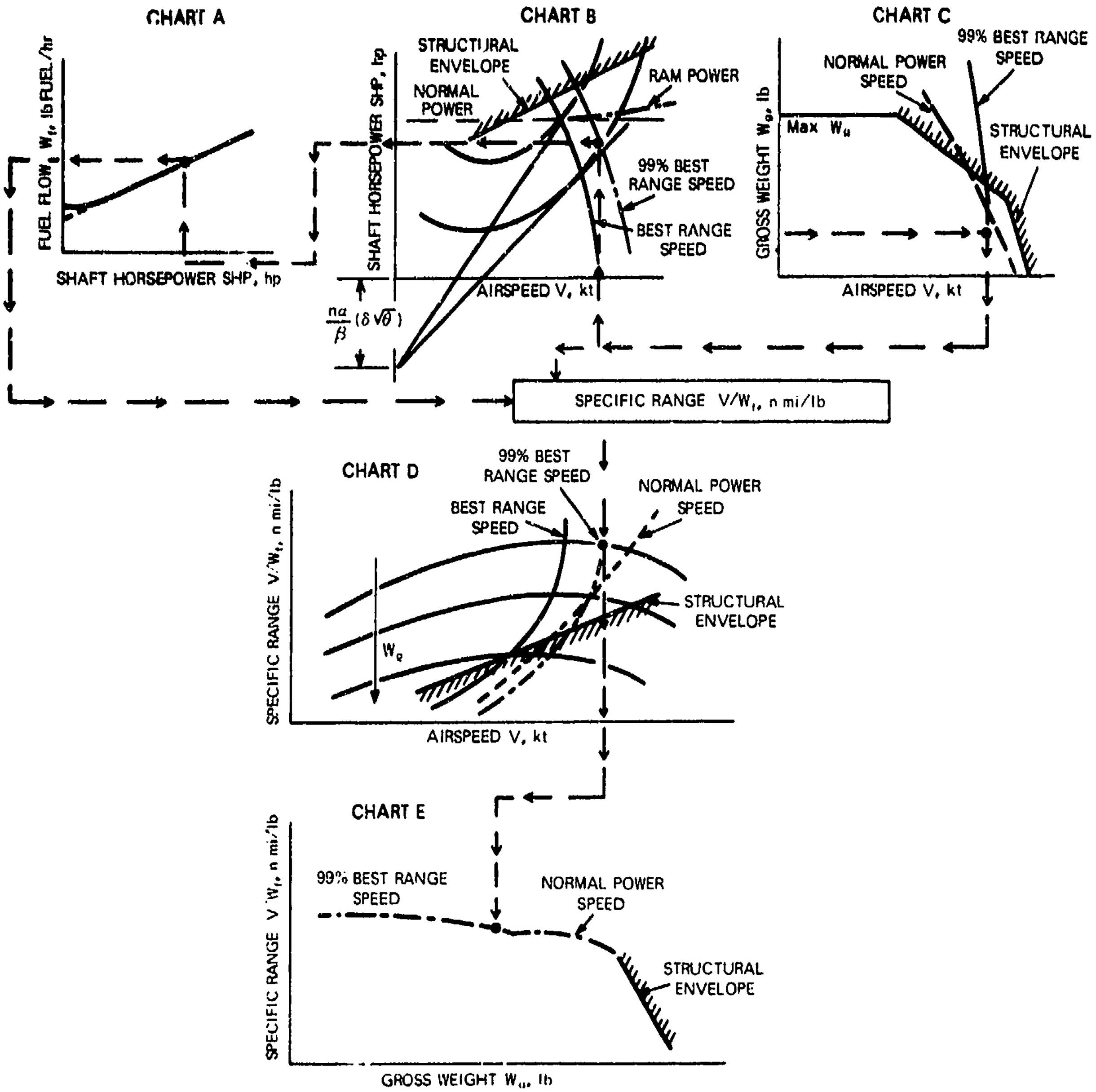


Fig. 3-130. Specific-range/Gross-weight Schedule

3-4.2.4.4 Effect of Altitude and Gross Weight on Range

Charts A_1 through A_3, Fig. 3-132, illustrate the effect of altitude upon cruise speed for a nominal gross weight. At low altitudes, the helicopter is able to fly at best-range speeds. As altitude increases, the normal power and structural envelope speeds decrease, causing limitations upon the maximum allowable speeds. Chart B, Fig. 3-132, shows typical variations of cruise speeds as a function of altitude and gross weight. Chart C illustrates the resulting variation in specific range with altitude. At each gross weight, an altitude condition exists at which the optimum specific range is available. This altitude corresponds to a condition where the helicopter is operating at the optimum L/D_e for the particular gross weight.

The increase in available specific range with increas-

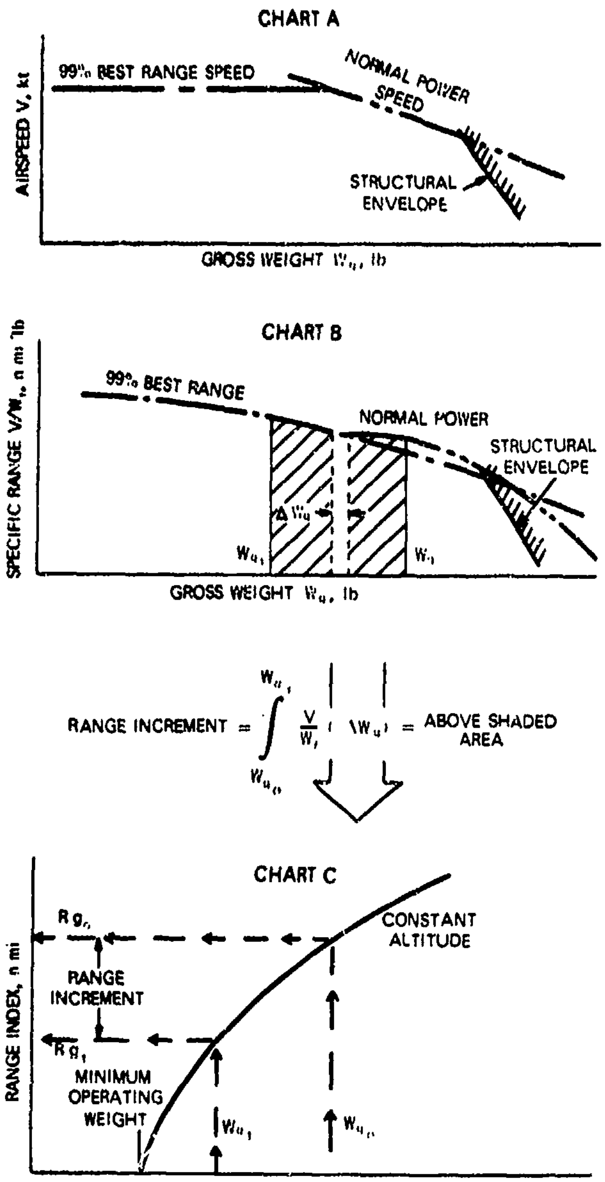


Fig. 3-131. Range Index Method

ing altitude for lower gross weights is attributable to the improvement in turboshaft engine efficiency that occurs with increasing altitude. At high gross weights, the increased power required due to altitude overcomes the improved engine characteristics. The locus of maximum specific range points in Chart C represents an optimum altitude schedule for obtaining the maximum available range at each gross weight. Chart D depicts the effects of gross weight and altitude upon the range index curves developed previously. This chart indicates that for low gross weights the maximum increment of range for a given fuel weight is available at high altitudes; conversely, the maximum increment is available at low altitudes for high gross weights. This is a result of the altitude trends shown in Chart C. At all altitudes in Chart D, the lower weights exhibit superior range increments for a given fuel weight.

3-4.2.4.5 Effect of Wind on Range Capability

Prevailing winds affect both range capability and optimum cruise speed. To compute the specific range parameter with the effect of wind incorporated, it is necessary only to add tail wind or to subtract head wind from the true airspeed to find the resultant ground speed. The specific range expression becomes:

$$\text{Specific Range} = \left(\frac{V \pm \text{Wind}}{W_f}\right), \text{n mi/lb} \quad (3\text{-}174)$$

Chart A, Fig. 3-133, illustrates the effect of wind on the optimum airspeed required to maximize range capability. Chart B illustrates the variation of specific range and airspeed for optimum range; the optimum airspeed decreases for tail winds and increases for head winds. Charts C and D present the variation of range with gross weight and winds. From Chart D, wind effects are shown to have a significant impact on the trends of the range index curve and can result in substantial adjustments in range capability when considering large quantities of cruise fuel.

3-4.2.4.6 Ferry Range Capability and Range Extension

To determine maximum range capability for a given configuration, the helicopter is considered as carrying no payload. Limitations that affect the ferry range capability are fuel capacity and takeoff criteria. Substantially increased range capability can be achieved by adding auxiliary fuel tanks. With their use, the empty weight of the configuration will increase by about 0.5 lb/gal of auxiliary fuel for external tanks and 0.3 lb/gal for internal bladder tanks. Examples of takeoff criteria used when studying ferry mission capability are hover requirements OGE or IGE, takeoff distance required to clear a specified obstacle, or use of a rolling takeoff from a runway. In some cases, the takeoff gross weight may be limited by the design maximum gross weight.

The range capability of multiengine helicopters can be extended further by shutting down one engine after the gross weight is reduced sufficiently as a result of fuel consumption and maintaining cruise on the remaining operating engine(s). This improvement in

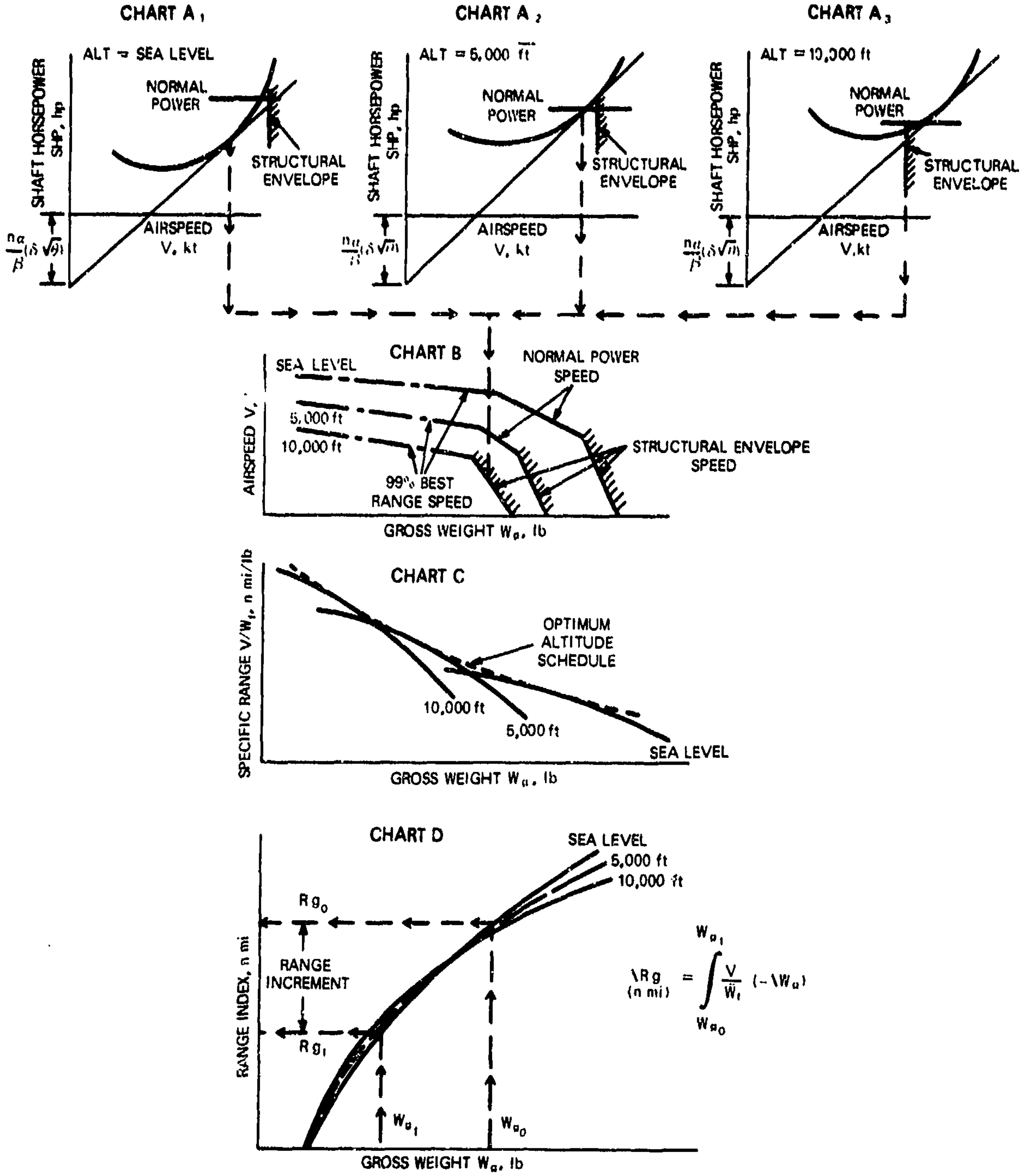


Fig. 3-132. Effect of Altitude and Gross Weight on Range

range derives principally from the SFC characteristics of the turbine engine. At low gross weights, the reduced power required for best-range speed precludes operating the engines at the optimum SFC level. By shutting down one engine, the remaining engine(s) operate at higher powers and nearer their maximum efficiency, thereby extending the range over that achievable when all engines are operating. This technique has been used to establish world distance records for helicopters. Under normal circumstances, however, it is not common practice to shut down one engine on ferry missions.

A helicopter ferry mission requires the use of operating conditions that will optimize range capability. Thus, the aircraft starts at its maximum allowable gross weight and cruises until its weight reaches the empty weight plus required fuel reserves and fixed useful load. To gain the most advantageous flight conditions, the aircraft performs a cruise climb at the optimum altitude-gross weight schedule of par. 3-4.2.4.4. Chart A of Fig. 3-134 illustrates a typical altitude schedule for a ferry mission. The initial portion of the mission is performed at low altitude (best for high gross weight range

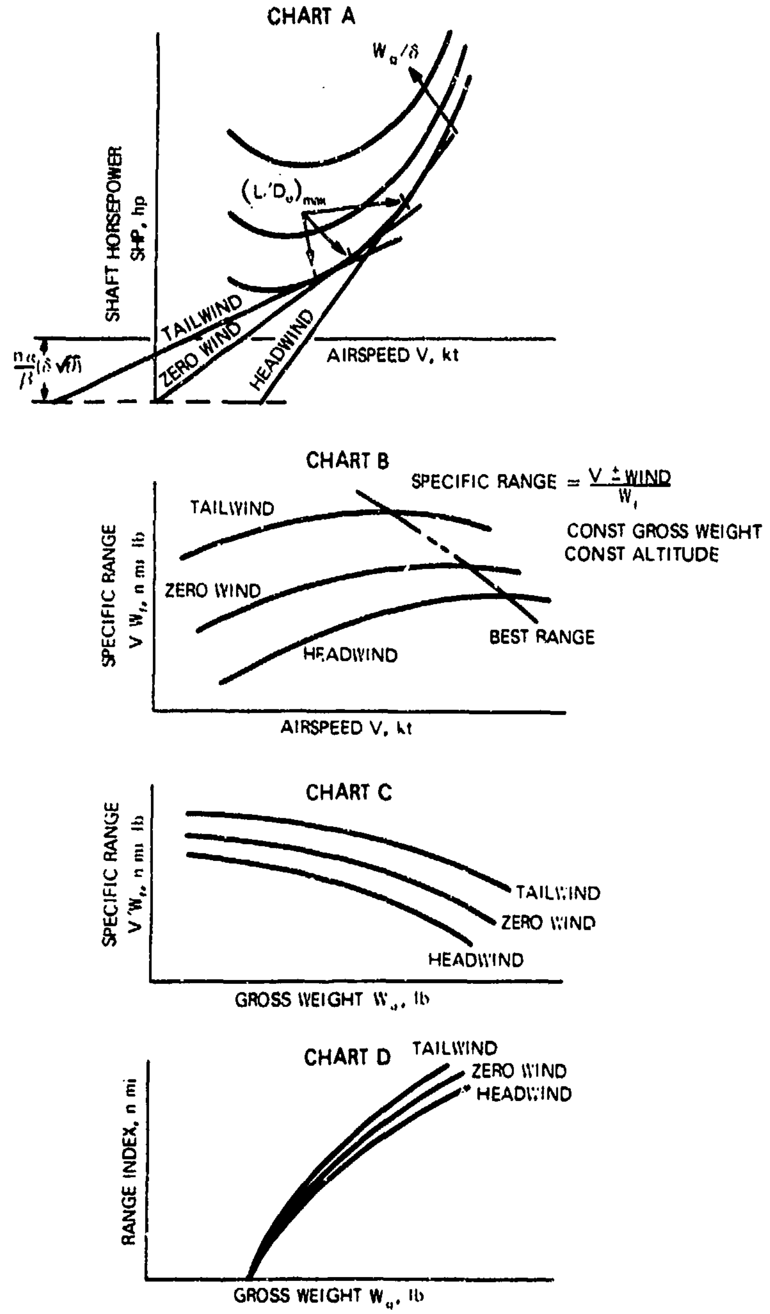


Fig. 3-133. Effect of Wind on Range Capability

performance). Altitude is increased as fuel is burned off and gross weight decreases. The associated airspeed schedule is shown in Chart B, Fig. 3-134, and it normally is limited by the structural envelope or by normal power at high weights, and by 99% best-range speed at lower gross weights. Chart C, Fig. 3-134, depicts the gross weight-range schedule for a ferry mission and illustrates the benefit of flying an optimum altitude schedule as opposed to a constant altitude. In using an optimum altitude cruise-climb, however, the power required to perform the climb must be considered. Chart D illustrates the incremental climb power, which can be computed using a rate of change of potential energy relationship. The K_p term in the expression is a climb efficiency term. A more complete explanation of the calculation of climb power is presented in par. 3-4.2.6. Climb effects can be integrated into the specific range computation by modifying the expression to:

$$\text{Specific Range} = \frac{V}{W_{f_{lev}} + \beta(\Delta SHP)_{cl}}\ , \text{n mi/lb} \quad (3\text{-}175)$$

However, it can be obtained with sufficient accuracy by taking the operating conditions at the midpoint of the mission, computing the shaft horsepower required for climb ΔSHP_{cl}, determining the associated incremental fuel-flow rate ΔW_f, and calculating the climb fuel as the product of ΔW_f and t (the total mission time). This fuel component then is unavailable for cruise, and the ferry range is computed using the remaining fuel and level flight specific range values. Chart E illustrates the effect of cruising at optimum altitude as opposed to constant altitude. The climb power correction represents a small reduction in the overall range capability.

3-4.2.4.7 Range Parameter in Generalized Form

As an alternative method for preliminary determination of performance, helicopter range performance can be computed in the generalized form without first computing dimensional quantities. Fig. 3-135 illustrates the development of a generalized specific range parameter.

The procedure used to compute the generalized range parameter is similar to that of par. 3-4.2.4.6, differing only in the generalized nature of the final data. Charts A through C, Fig. 3-135, depict the development of the range parameter as a function of generalized airspeed and generalized gross weight. The locus of best range and 99% best range speeds is illustrated in Chart C. Normal power and structural envelope limits are not shown because they do not collapse typically to one generalized limit but are dependent upon altitude and temperature. The effects of normal power and structural envelope limitations normally need to be considered only in the event of high gross weights or extreme ambient temperature conditions. Range performance can be computed at any desired altitude by using Chart D and the appropriate pressure ratio δ for the specific altitude. Chart D is shown for constant $N/\sqrt{\theta}$. However, the variation of specific range with $N/\sqrt{\theta}$ is relatively minor compared with the effects of gross weight and altitude.

The generalized method is not as accurate as the dimensional approach, but in most cases it has sufficient accuracy for preliminary design range calculations. The generalized format also lends itself to explaining how increased range capability is obtained by performing a cruise climb.

Fig. 3-136 illustrates generalized power polars at constant generalized rotor speed. Superimposed upon Chart A, Fig. 3-136, are lines that determine $(L/D_e)_{max}$ at each gross weight. The locus of $(L/D_e)_{max}$ points forms a cruise schedule from which the maximum range at each generalized gross weight can be obtained. Chart B illustrates that there is a point on the cruise schedule at which the optimum L/D_e is available. The generalized weight and airspeed corresponding to this point provide the optimum specific range value that can be achieved. For range missions initiated at gross weights above the W_g/δ for $(L/D_e)_{opt}$ maximum specific range is obtained by flying at sea level and following the cruise schedule cited. When sufficient fuel is burned to decrease the gross weight to the W_g/δ for $(L/D_e)_{opt}$, the aircraft then begins a slow climb to maintain the W_g/δ value as the gross weight decreases further. This procedure allows the aircraft to cruise at optimum L/D_e and, therefore, to obtain optimum range performance capability. For range missions initiated at gross weights below the W_g/δ for $(L/D_e)_{opt}$, the maximum range is obtained by climbing initially to the altitude that gives W_g/δ equal to the desired value and then continuing a slow climb to maintain this value as fuel is burned off.

3-4.2.4.8 Breguet Range Equation

Breguet developed an expression for the maximum range Rg_{max} of a fixed-wing aircraft with a reciprocating engine power plant. The result is of the form (Ref. 72):

$$Rg_{max} = 325\left(\frac{L}{D}\right)_{max}\frac{\eta_p}{c}\ln\left(\frac{W_{g_0}}{W_{g_1}}\right)\ , \text{n mi} \quad (3\text{-}176)$$

where

c = shaft engine specific fuel consumption, lb-fuel/hp-hr

W_{g_0}, W_{g_1} = initial and final gross weight, respectively, lb

η_p = propulsive efficiency, dimensionless

This expression assumes that the aircraft is cruising at $(L/D)_{max}$ throughout the mission by being flown at constant angle of attack to maintain C_L and C_D constant while velocity is allowed to vary.

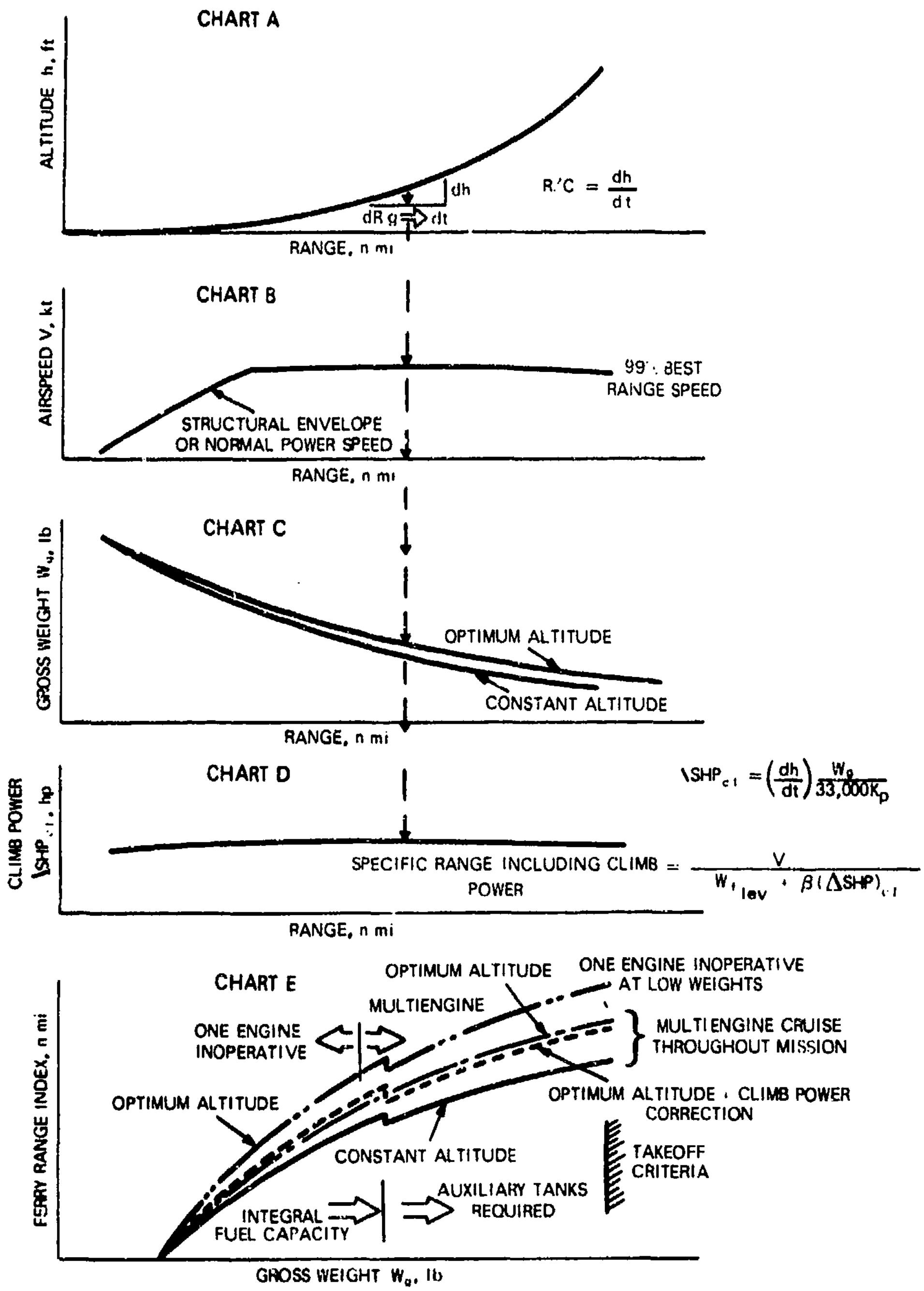


Fig. 3-134. Ferry Range Mission

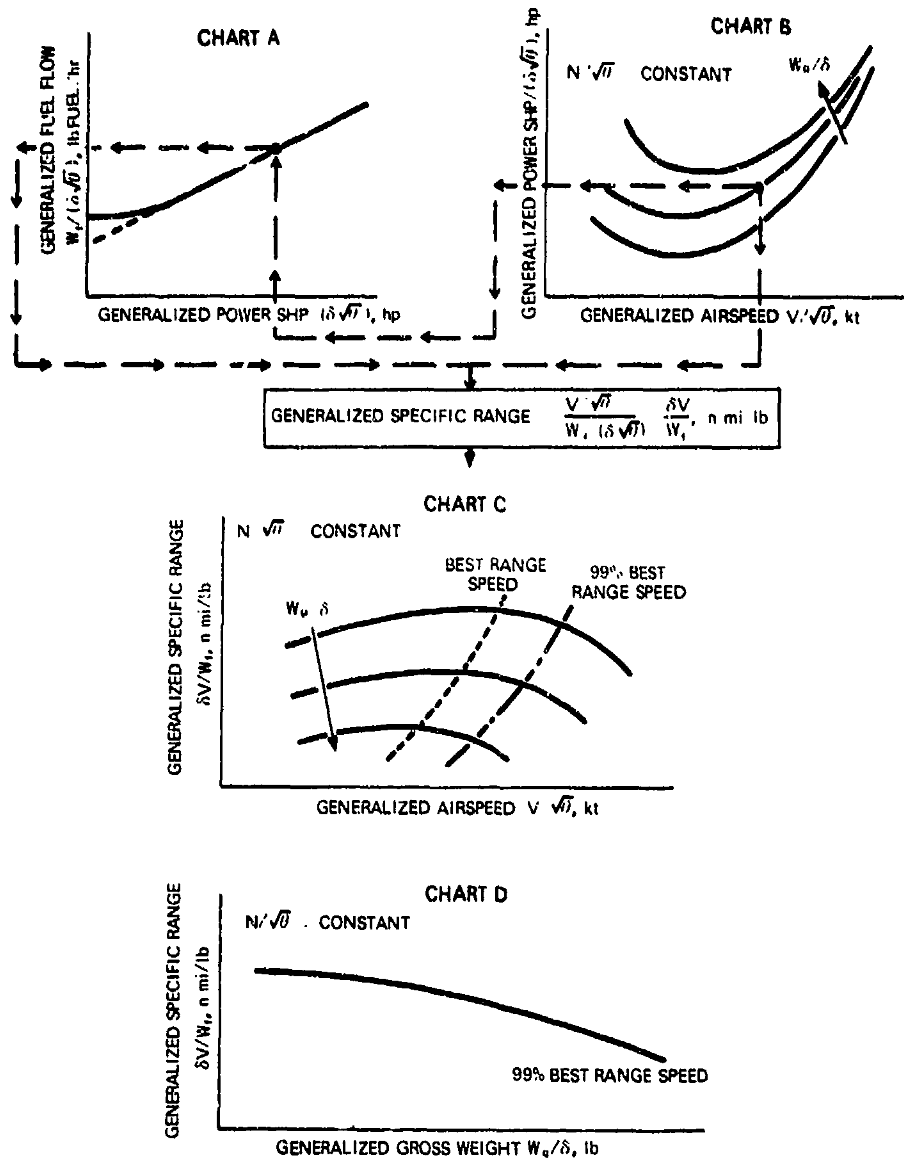


Fig. 3-135. Generalized Range Parameter

A similar expression was determined by Breguet for a fixed-wing aircraft with a turbojet power plant:

$$R_{g_{max}} = \frac{2}{c'}\left(\frac{C_L^{1/2}}{C_D}\right)_{max}\left(\frac{295}{\sigma S}\right)^{1/2} \times \left(\sqrt{W_{g_0}} - \sqrt{W_{g_1}}\right), \text{n mi} \quad (3\text{-}177)$$

where

c' = jet engine specific fuel consumption (function of altitude), lb-fuel/lb-thrust-hr
S = wing area, ft^2
σ = ambient density ratio, dimensionless

This expression is valid strictly only when cruise is at constant altitude; however, Perkins and Hage (Ref. 72) determined that it gives results within 3% of the optimum range obtained by flying an optimum altitude schedule.

The Breguet approach can be extended to helicopters with turboshaft power plants by making a few modifications. The incremental range is given by Eq. 3-173. Using Eq. 3-171, the incremental range equation becomes

$$dRg = \frac{-VdW_g}{\beta\left[SHP + \frac{n\alpha}{\beta}\delta\sqrt{\theta}\right]}\ , \text{n mi} \qquad (3\text{-}178)$$

Now, introducing effective drag D_e (Eq. 3-172), and assuming lift equal to weight, the incremental range expression becomes

$$dRg = \frac{-325}{\beta}\left(\frac{L}{D_e}\right)\frac{dW_g}{W_g}\ , \text{n mi} \qquad (3\text{-}179)$$

which integrates to

$$Rg = \frac{325}{\beta}\left(\frac{L}{D_e}\right)\ln\left(\frac{W_{g_0}}{W_{g_1}}\right)\ , \text{n mi} \qquad (3\text{-}180)$$

The range thus is determined to be proportional to L/D_e and the range again can be maximized by performing a cruise-climb to maintain L/D_e at its optimum value as weight is reduced by fuel burnoff.

3-4.2.5 Endurance

The endurance of a helicopter is defined as the length of time for which it can remain airborne while using a specified quantity of fuel. Search operations at slow speeds (80-120 kt) or loiter operations are examples of special mission requirements that require maximum endurance capability. The maximum endurance capability is achieved by operating the helicopter at the speed for which minimum power is required. In the discussion that follows, the method of analysis is described and the effects of gross weight and altitude upon endurance are discussed. In addition, an alternative analysis method that uses the Breguet endurance equation is presented.

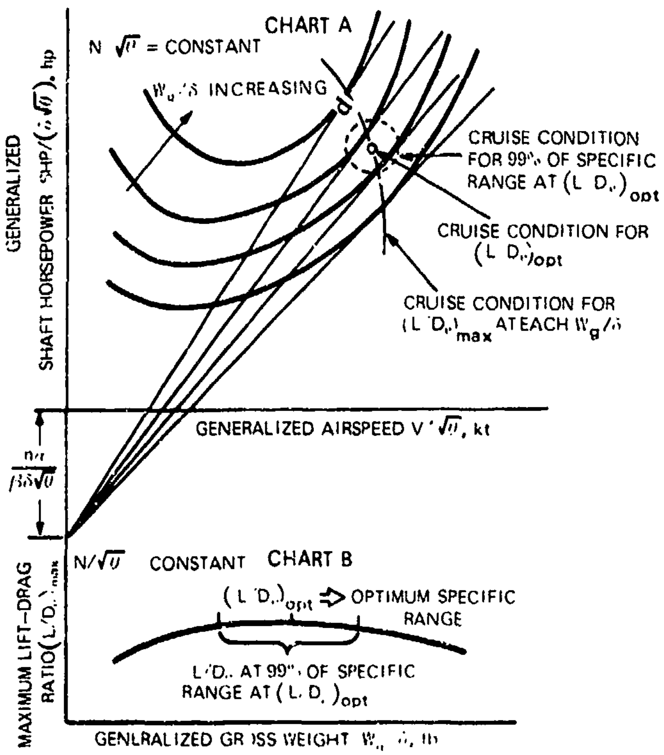


Fig. 3-136. Generalized Helicopter Performance

3-4.2.5.1 Analysis Method

The endurance capability can be determined if the following elements are available:

1. Fuel flow versus power characteristics (generalized) of the installed engine(s)

2. Airspeed versus power polars (generalized or dimensionless) for the helicopter.

Charts A, B_1, and B_2, Fig. 3-137, illustrate typical examples of these elements.

The engine fuel flow characteristics are obtained from the manufacturer's detail specification, and the aircraft performance can be obtained from one or more of the following sources: flight test data, wind tunnel model test data, or theoretically estimated data. The engine fuel flow characteristics *shall* be 5% above the manufacturer's specification to comply with Military Specification requirements (MIL-C-5011). The dimensional charts for helicopter performance (Charts C and D, Fig. 3-137) can be obtained by applying the appropriate factors representing the specified ambient conditions to either dimensionless or generalized data because the two methods yield equivalent results (par. 3-4.2.4).

The endurance capability of an aircraft is solely a function of the rate at which fuel is consumed during flight. To maximize endurance it is necessary to minimize fuel consumption W_f. The engine power requirements therefore must be kept at a minimum, and this requires that the aircraft be flown at the airspeed for minimum power as illustrated in Chart D of Fig. 3-137. From Chart D the power requirements at the speed for maximum endurance can be obtained; entering Chart C at the required power, the fuel consumption of the engine(s) can be determined. The minimum fuel flow thus is established as a function of gross weight, and Chart E illustrates the typical trend of increasing fuel flow requirements with increased gross weight. The incremental endurance Δt is defined by

$$\Delta t = -\frac{\Delta W_g}{W_f} \text{ , hr} \qquad (3\text{-}181)$$

where the fuel burned is equal to the change in aircraft weight.

The total endurance of an aircraft starting at an initial gross weight W_{g_0} and ending at a final gross weight W_{g_1} is given by

$$t = \int_{W_{g_0}}^{W_{g_1}} -\frac{dW_g}{W_f} \text{ , hr} \qquad (3\text{-}182)$$

This expression normally is evaluated by numerical integration to determine endurance capability.

Another approach to this solution, one which is particularly valuable if a number of endurance requirements are to be analyzed, is the endurance index curve shown in Chart F, Fig. 3-137. The index curve is obtained by cumulative integration of the expression $-\Delta W_g/W_f$. The trends presented are typical of results obtained for helicopters with turboshaft engines. If the integration is performed from the maximum aircraft weight to the minimum operating weight, the curve can be used for all gross weights of interest. Fig. 3-137 also presents an example of the use of the index curve. The incremental endurance is obtained by entering the endurance index curves of Chart F at the initial gross weight W_{g_0} to determine the t_0 index, and at the final gross weight W_{g_1} (dictated by amount of fuel consumed) to determine the t_1 index. The endurance increment then is given by $t_0 - t_1$.

3-4.2.5.2 Effect of Gross Weight and Altitude on Endurance

The effects of gross weight and altitude on endurance capability are shown in Fig. 3-138. For constant altitude conditions, the endurance increment increases with decreasing gross weight for a constant increment in fuel consumption. Fig. 3-138 further illustrates that higher altitudes offer slightly better endurance capabilities especially at high gross weights. The range benefit resulting from increased altitude is attributable to the improvement in the efficiency of the turboshaft engine with altitude as well as to the decreased fuselage and blade profile drag.

The effect of variation in minimum power required due to small changes in rotor speed is relatively minor compared to the effects of gross weight and altitude, and generally can be ignored.

Prevailing winds, which are an important consideration when computing range performance, have no effect upon endurance capability because endurance is dependent solely on fuel consumption rate.

3-4.2.5.3 Breguet Endurance Equation

The approach used by Breguet to develop an expression for range capability also can be used to obtain an endurance expression. For a fixed-wing aircraft with a

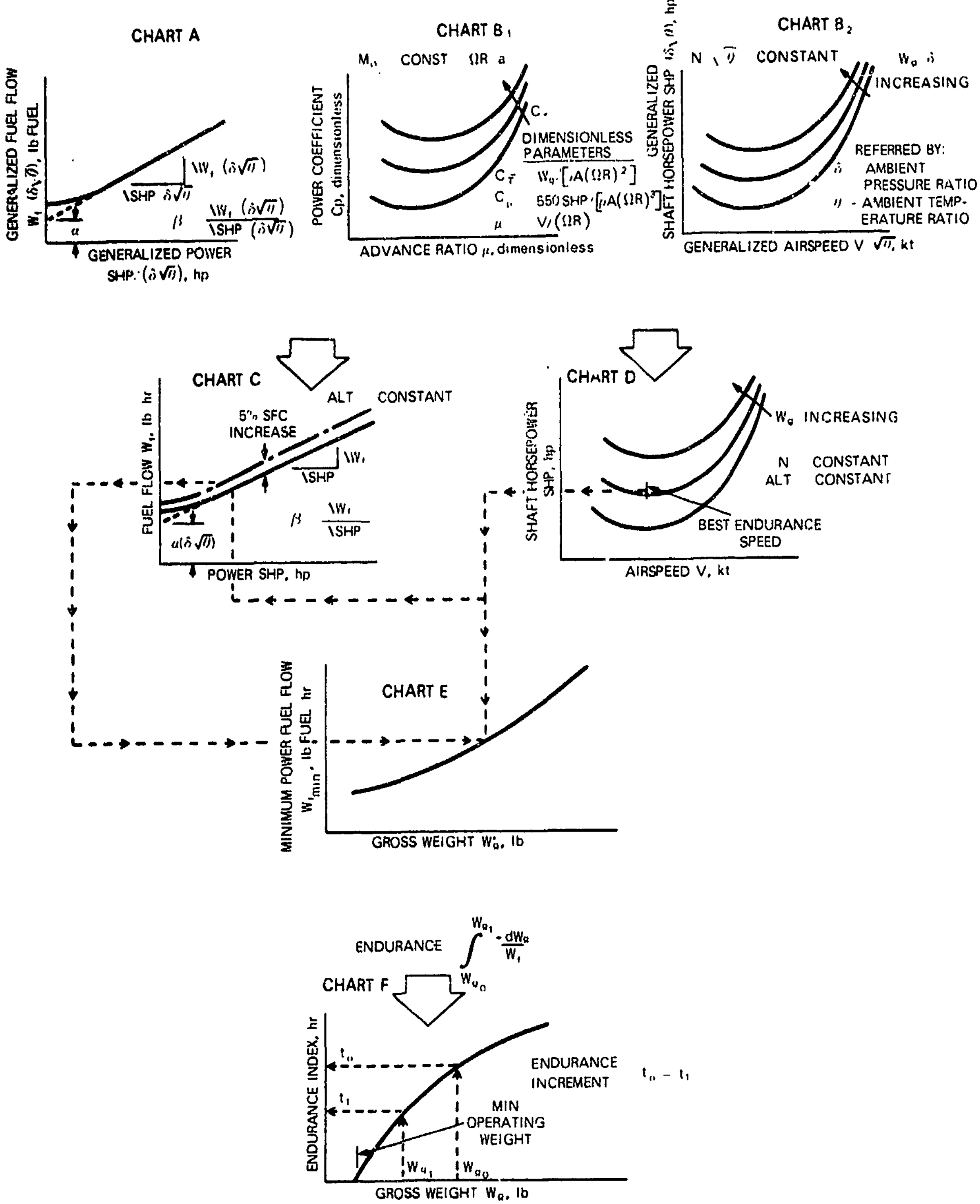


Fig. 3-137. Elements Required To Compute Endurance

reciprocating engine power plant, the endurance expression is (Ref. 72)

$$t_{max} = 37.9\frac{\eta_p}{c}\left(\frac{C_L^{3/2}}{C_D}\right)_{max}\left(\frac{\sigma S}{W_{g_0}}\right)^{1/2} \times$$

$$\left[\left(\frac{W_{g_0}}{W_{g_1}}\right)^{1/2} - 1\right] \quad \text{, hr} \qquad (3\text{-}183)$$

To achieve maximum endurance, the quantity $C_L^{3/2}/C_D$ must be maximized. Due to σ in Eq. 3-183, the maximum endurance is available at low altitudes for equivalent values of $C_L^{3/2}/C_D$

For a fixed-wing aircraft with a turbojet power plant, a similar expression is obtained:

$$t_{max} = \frac{1}{c'}\left(\frac{L}{D}\right)\ln\left(\frac{W_{g_0}}{W_{g_1}}\right) \quad \text{, hr} \qquad (3\text{-}184)$$

This expression is strictly valid only for constant altitude because SFC is a function of altitude. Optimum endurance is obtained when the aircraft maintains a flight velocity that maximizes $(L/D)/c'$.

Extending the Breguet technique to helicopters with turboshaft power plants, a similar expression can be obtained. The incremental endurance is given by Eq. 3-182. Thus

$$dt = \frac{-dW_g}{\beta\left[SHP + \frac{n\alpha}{\beta}(\delta\sqrt{\theta})\right]} \quad \text{, hr} \qquad (3\text{-}185)$$

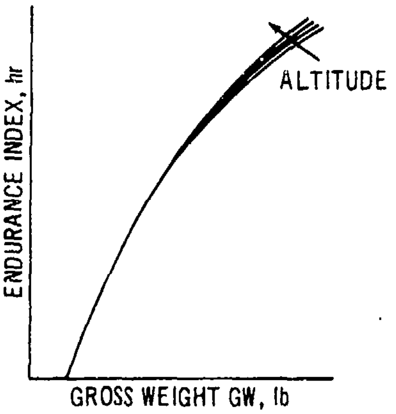


Fig. 3-138. Effect of Gross Weight and Altitude on Endurance

and from Eq. 3-179

$$dt = \frac{dRg}{V} = -\frac{325}{\beta V}\left(\frac{L}{D_e}\right)\frac{dW_g}{W_g} \quad \text{, hr} \qquad (3\text{-}186)$$

To integrate Eq. 3-186, it is helpful to make use of another system of dimensionless quantities that includes a free stream dynamic pressure term ($q = \rho(1.69V)^2/2$) similar to fixed-wing convention. The helicopter coefficients for lift L_c and effective drag D_{e_c} are given by

$$\left.\begin{aligned} L = W_g &= \frac{1}{2}\rho(1.69V)^2(2R)^2\sigma L_c \quad \text{, lb} \\ \therefore V &= \frac{1}{1.69}\left[\frac{2W_g}{\rho(2R)^2\sigma L_c}\right]^{1/2} \quad \text{, kt} \end{aligned}\right\} (3\text{-}187)$$

$$\left.\begin{aligned} D_e &= \frac{1}{2}\rho(1.69V)^2(2R)^2\sigma D_{e_c} \quad \text{, lb} \\ \therefore L/D_e &= L_c/D_{e_c} \quad \text{, dimensionless} \end{aligned}\right\} (3\text{-}188)$$

where

$1.69V$ = velocity, fps (V is velocity in kt)
$2R$ = rotor diameter, ft
σ = rotor solidity, dimensionless

A development of this dimensionless system can be found in Ref. 73. Eq. 3-186 then rearranges to

$$dt = -\frac{550}{\beta}\left[\frac{\rho(2R)^2\sigma}{2}\right]^{1/2} \times$$

$$\left(\frac{L_c^{3/2}}{D_{e_c}}\right)\frac{dW_g}{W_g^{3/2}} \quad \text{, hr} \qquad (3\text{-}189)$$

and integrating over the limits of W_{g_0} to W_{g_1} yields

$$t = \frac{37.9}{\beta}\left[\frac{\rho}{\rho_o}(2R)^2\sigma\right]^{1/2}\left(\frac{L_c^{3/2}}{D_{e_c}}\right) \times$$

$$\left[\sqrt{\frac{1}{W_{g_1}}} - \sqrt{\frac{1}{W_{g_0}}}\right] \quad \text{, hr} \qquad (3\text{-}190)$$

For typical helicopters with turboshaft power plants,

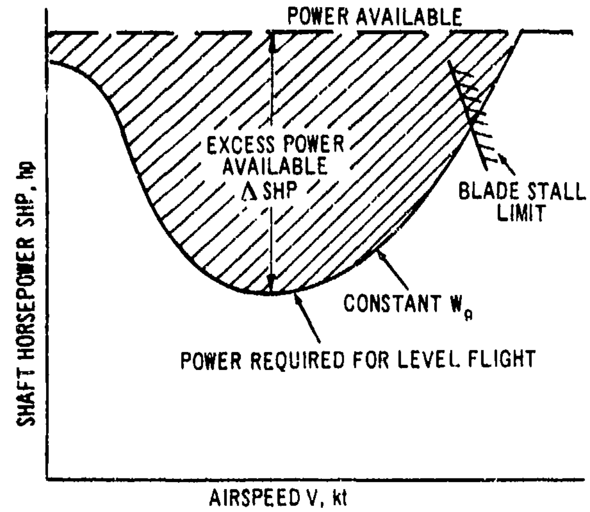


Fig. 3-139. Level Flight Power Required

the product generally is constant with altitude, showing a slight increase in endurance capability with increased altitude.

3-4.2.6 Rate of Climb—Maximum and Vertical

3-4.2.6.1 Basic Considerations

Rate of climb capability is derived from the power available beyond that required for level flight or hover. The variation with forward speed in power required for level flight and power available is shown in Fig. 3-139. The excess power available is represented by the cross-hatched area. Typical rate of climb characteristics for a helicopter are shown in Fig. 3-140.

Methods for determining the climb efficiency term K_p, the rate of climb capability at low forward speeds and at maximum forward speeds, and the vertical rate of climb are described in the paragraphs that follow.

3-4.2.6.2 Forward Climb

Fig. 3-141 illustrates typical formats used to present forward climb performance. The charts define specific climb capabilities at fixed gross weights and ambient conditions for a given engine power level. Chart A presents R/C performance variation with altitude at various gross weights and standard day temperatures at altitude. Chart B provides R/C data at a given gross weight, showing the effect of nonstandard day temperatures at altitude.

A format frequently used to present climb capability in Operator's Manuals is shown in Chart C. Climb capability is determined by entering at a specified weight and pressure altitude to determine the rate of climb at standard temperature conditions on the baseline, and then following the guide line to obtain an approximate correction for operations at nonstandard temperatures. This format provides only an approximate correction for operations at nonstandard temperatures because the guide lines represent a mean of the variation of engine power available and aircraft power required with temperature over a range of altitudes.

Forward rate of climb R/C is calculated with the aid of the power relationship of Eq. 3-191.

$$R/C = \frac{33{,}000(\Delta SHP)K_p}{W_g}, \text{fpm} \qquad (3\text{-}191)$$

where

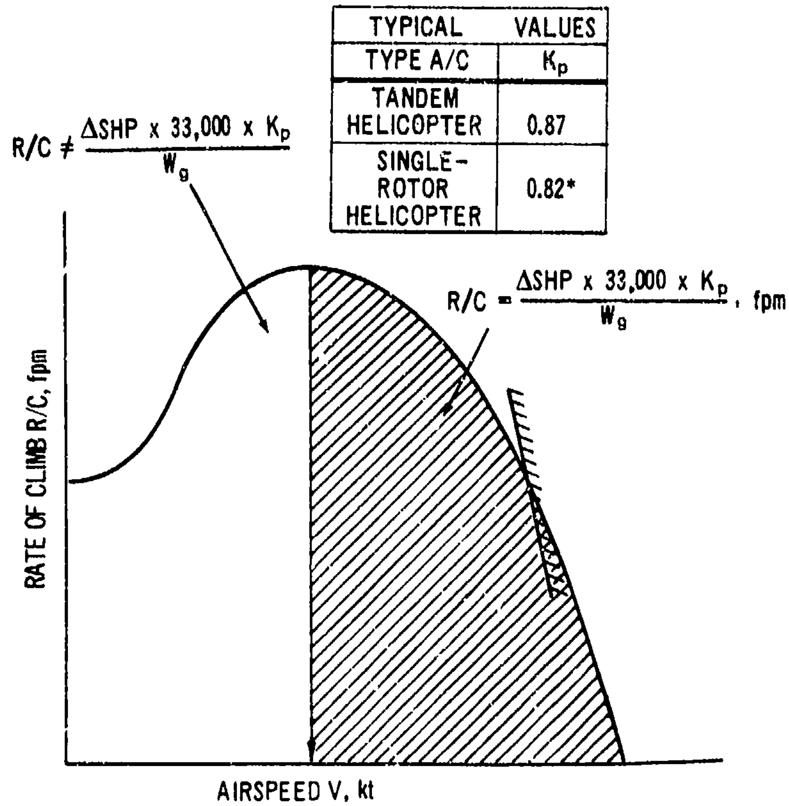
ΔSHP = excess power available above level flight power required at a fixed airspeed as shown in Fig. 3-139, hp

K_p = climb efficiency factor, dimensionless

Maximum rate of climb for a given weight occurs at the speed at which power required for level flight is at a minimum. This speed increases with increasing weight and altitude. A typical plot of minimum level flight power required, obtained from a "crossplot" of existing power polars, as shown in Fig. 3-142, is used as the basis for determining the excess power available used in Eq. 3-191.

Flight test experience on existing helicopters has shown that a climb efficiency factor K_p must be applied to allow for transmission efficiency, rotor efficiency (induced velocity and power variations in climb), and increases in fuselage download. The increase in download results from the added vertical component of the flight path velocity and the increase in rotor induced velocity.

The climb efficiency factor can be obtained from flight test data simply by relating the actual climb performance to that determined from the theoretical power relationship ($\Delta SHP \sim W_g(R/C)$). If flight test data are not available, the following method is used to define the value of K_p at a given airspeed. The rotor performance map shown in Chart A, Fig. 3-143, is generated either from model test results (including rotor and fuselage) or from theoretical trim and power analyses that provide for both rotor and fuselage effects (fuselage drag, download, and rotor/fuselage interference). The lines of constant power define the locus of



*ADDITIONAL 5% FOR TAIL ROTOR POWER

Fig. 3-140. Climb Capability

possible lift and propulsive force F_p combinations for a particular helicopter configuration. The level flight trim condition shown in Chart A is defined for a propulsive force of zero for various gross weights and powers. In level flight, the climb angle γ is zero and, therefore, the lift equals the gross weight.

Chart B, Fig. 3-143, illustrates the vector diagram used to calculate rate of climb using excess net propulsive force ΔF_p at constant gross weight. The associated force relationship is shown in Chart C. An arc of constant gross weight is constructed across the lines of constant power. A resultant vector representing gross weight is drawn from the origin to each operating point on the power curves, and is resolved into an excess propulsive force ΔF_p and a net lift $W_g \cos \gamma$. The excess propulsive force defines the climb angle as shown in Eq. 3-192.

$$\gamma = \mathrm{Sin}^{-1} (\Delta F_p / W_g) \quad \text{, deg} \qquad (3\text{-}192)$$

Climb velocity is determined with the aid of Eq. 3-193 using the climb angle γ derived in Eq. 3-192. The airspeed for which the rotor map is presented represents the flight path speed $V_{flt\ path}$

$$V_{cl} = V_{flt\ path} \sin \gamma \quad \text{, fps} \qquad (3\text{-}193)$$

where V_{cl} and $V_{flt\ path}$ are in ft/sec units.

The increased power required to achieve the calculated climb velocity is the difference between the trim point and the operating point defined on the power map at constant gross weight. This increased power is combined with the climb velocity in Eq. 3-194 to calculate the value of K_p

$$K_p = \frac{V_{cl} W_g}{(\Delta SHP) 550} \quad \text{, dimensionless} \qquad (3\text{-}194)$$

The climb efficiency factor is developed at airspeeds defined by minimum level flight power required. Because rotor efficiency in a climb remains essentially constant at speeds above minimum power, K_p varies with rate of climb and is difficult to determine for a general condition. Therefore, to insure an accurate evaluation of climb performance in the low-speed region, the use of power maps in lieu of the power equation is recommended.

Fig. 3-144 illustrates the calculation technique used with Eq. 3-191 to determine maximum rate of climb. The engine power available at a desired ambient condition is derived from data provided by the engine manufacturer (see Chart A). Chart B then is used to determine the associated minimum level flight power required at a given weight.

The excess power—the difference between power required and power available—is used in Eq. 3-191 with the previously established K_p to determine the climb rate. This process is repeated at different weights and

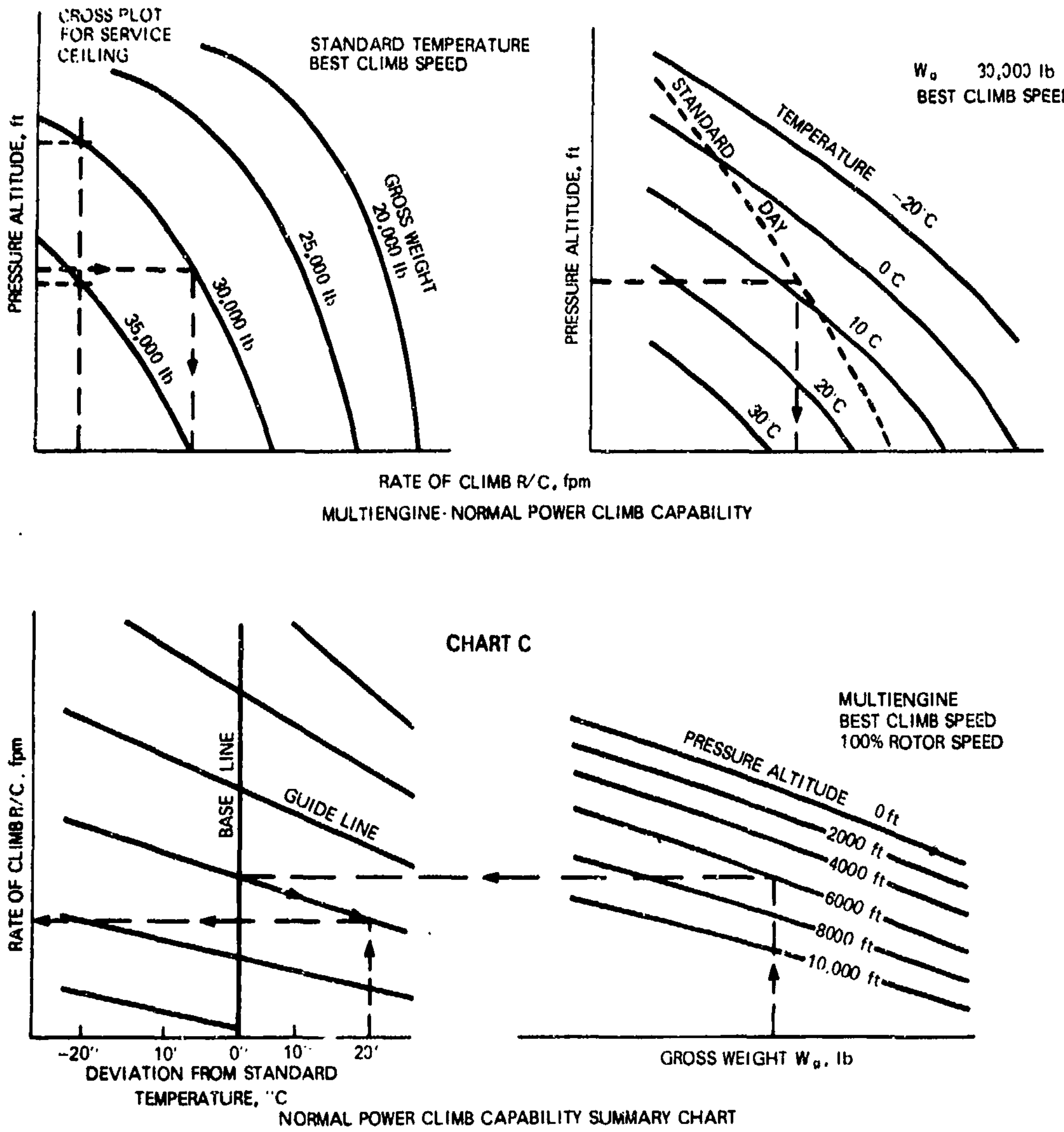


Fig. 3-141. Forward Climb Capability—Graphical Format

ambient conditions in order to obtain the series of climb curves shown in Chart C.

3-4.2.6.3 Service and Combat Ceilings

Service ceiling is defined as the maximum altitude at which the aircraft exhibits a 100 fpm rate of climb capability at a given temperature, while combat ceiling requires a climb capability of 500 fpm. These ceilings normally are defined at best climb speed, using normal or intermediate engine power ratings. Fig. 3-145 illustrates the standard presentational format for multiengine service and combat ceilings, and one engine inoperative (OEI) service ceiling. A ceiling is derived by entering the appropriate chart at the desired gross weight and temperature; e.g., for a typical helicopter at a gross weight of 36,000 lb and operating at an ambient temperature of 20°C, the multiengine service ceiling is 8000 ft, the combat ceiling is 7000 ft, and the OEI service ceiling is 4700 ft.

The ceiling capability chart is determined from a cross-plot of the climb capability/altitude chart (Fig. 3-141) at the climb rate associated with service or combat ceiling. An alternative calculation method, which can be used in the absence of a climb capability plot, is depicted in Fig. 3-146. First, the values of minimum power required for level flight (Chart A) are adjusted to reflect the additional climb power required for the 100 and 500 fpm climb rates, using the power relationship of Eq. 3-191. These values of total power required in climb are plotted in Chart C for a given altitude and temperature. Then, the resulting data are combined with the given power available at the ambient condition of interest to calculate ceiling capability as follows. Enter Chart B at a specific pressure altitude to determine the engine power available at the ambient temperature. Trace right to Chart C at fixed power and intersect the power required curve for the desired rate of climb. From this intersection, trace down to determine the gross weight. The gross weight and pressure altitude then are combined to form the ceiling capability (Chart D). OEI service ceiling is calculated in a similar manner.

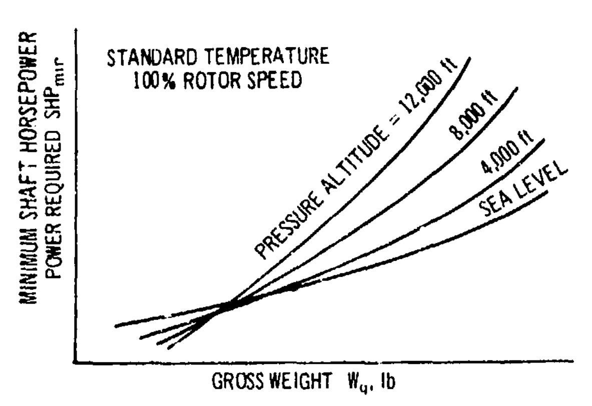


Fig. 3-142. Minimum Level Flight Power Required

3-4.2.6.4 Time, Fuel, and Distance to Climb

A typical graphical presentation of the time required, fuel used, and horizontal distance covered in a climb was shown in Fig. 3-121. These curves provide a direct reading of climb at a given altitude as a function of gross weight.

The determination of time to climb Δt is accomplished through the use of the following relationship:

$$\Delta t = 1/2 \left[\frac{\Delta h}{(R/C)_0} + \frac{\Delta h}{(R/C)_1}\right], \text{min} \qquad (3\text{-}195)$$

where

Δh = change in altitude, ft
$(R/C)_0$ = initial rate of climb, fpm
$(R/C)_1$ = final rate of climb, fpm

Fuel consumed and horizontal distance traveled during the climb are calculated from the time to climb and the average fuel flow rate and true airspeed applicable during the climb. For climbs of long duration, several smaller increments of altitude change may be used, with the results for time, fuel, and distance each being summed to obtain the final result.

3-4.2.6.5 Vertical Climb

Fig. 3-147 illustrates the manner in which vertical climb capability is presented in performance documents. Chart A presents sea level vertical rates of climb at various temperature conditions as a function of gross weight. Vertical climb capability for a series of gross weights at varying altitudes is provided by Chart B. Chart C presents OGE hover capability with a vertical climb correction chart in a manner normally incorporated in an aircraft Operator's Manual. Use of this format is initiated by determining the OGE hover capability at the ambient condition of interest from the upper portion of the chart. The correction for a given R/C then is applied by following the guideline to obtain the maximum gross weight at which the desired vertical climb capability exists.

Fig. 3-148 presents a typical breakdown of hover power required and illustrates the relationship of the component power requirements to the excess power used to calculate vertical climb.

The vertical climb capability, neglecting download, is expressed in terms of the total velocity through the rotor disk and the equivalent rotor-induced velocity

$$R/C = 60(U - \nu_{cl}) \quad \text{, fpm} \qquad (3\text{-}196)$$

where

U = total velocity through the rotor, fps

ν_{cl} = rotor-induced velocity in the climb, fps

The total velocity U is defined for a specified thrust and engine power available as

$$U = \frac{(rhp_{av} - rhp_0)550}{T} \quad \text{, fps} \qquad (3\text{-}197)$$

where

rhp_{av} = SHP_{av} − (transmission losses + tail rotor horsepower)

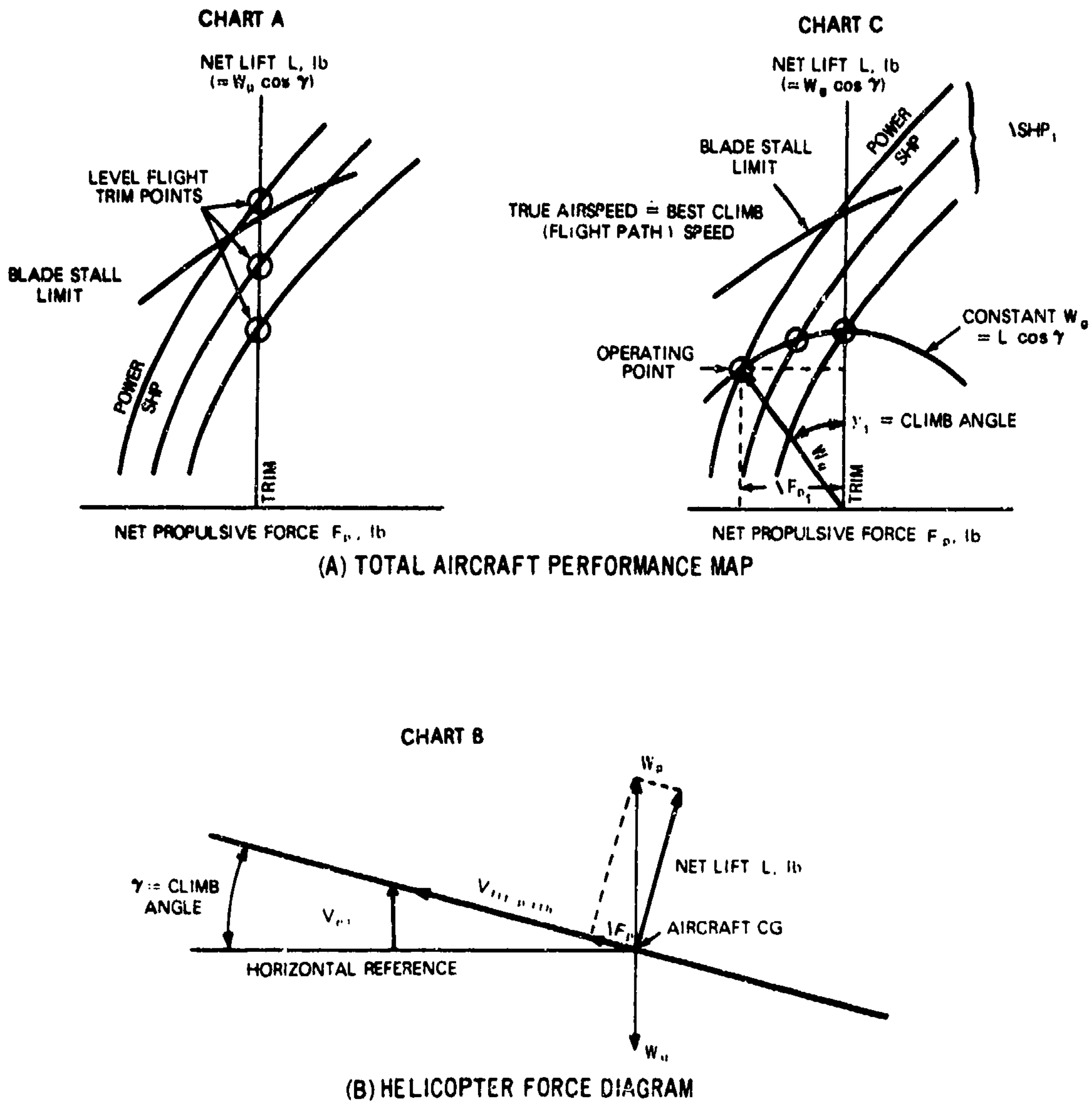


Fig. 3-143. Application of Rotor/Power Maps to Climb Performance

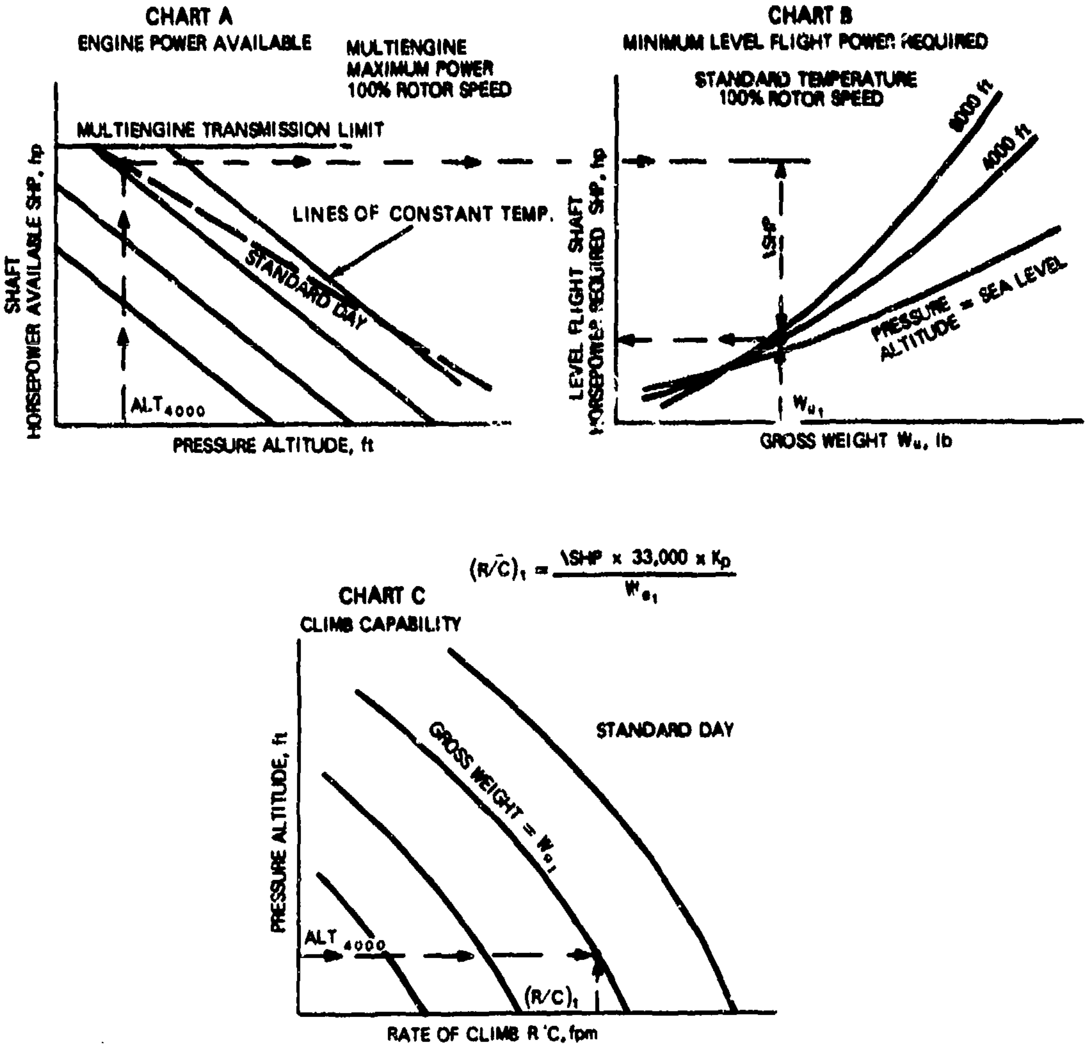


Fig. 3-144. Maximum Forward Rate of Climb Determination

rhp_o = rotor profile horsepower (theoretical)

T = gross weight + hover download, lb

The equivalent rotor-induced velocity in climb is related to both the total velocity and the hover-induced velocity as follows:

$$v_{cl} = \frac{v_{hov}^2}{U}\text{ , fps} \tag{3-198}$$

The rotor-induced velocity in hover can be expressed as

$$v_{hov} = \frac{550\, rhp_{i_{hov}}}{T}\text{ , fps} \tag{3-199}$$

where

$rhp_{i_{hov}}$ = rhp_{req} − (rhp_o + tail rotor horsepower)

rhp_{req} = hover horsepower required at the specified thrust

This analysis assumes that the rotor profile power in vertical climb is equivalent to that in hover. This assumption is valid for a rotor employing blades of nonlinear twist and has sufficient accuracy for application to a typical blade with linear twist.

The computation of vertical climb capability as discussed is based upon a zero fuselage download condition. Fuselage vertical drag in a vertical climb is dependent upon the sum of the climb velocity and the rotor-induced velocity. An increase in fuselage download during a vertical climb requires an increase in rotor thrust over that required to hover. Eq. 3-200 presents a method of calculating vertical drag in climb $D_{V_{cl}}$:

$$D_{V_{cl}} = \frac{\rho}{2} C_{D_V} S_V (V_{cl} + 2v_{cl})^2 \quad \text{, lb} \qquad (3\text{-}200)$$

where

C_{D_V} = mean drag coefficient of fuselage in vertical flow, dimensionless

S_V = planform area of fuselage exposed to the vertical flow, ft^2

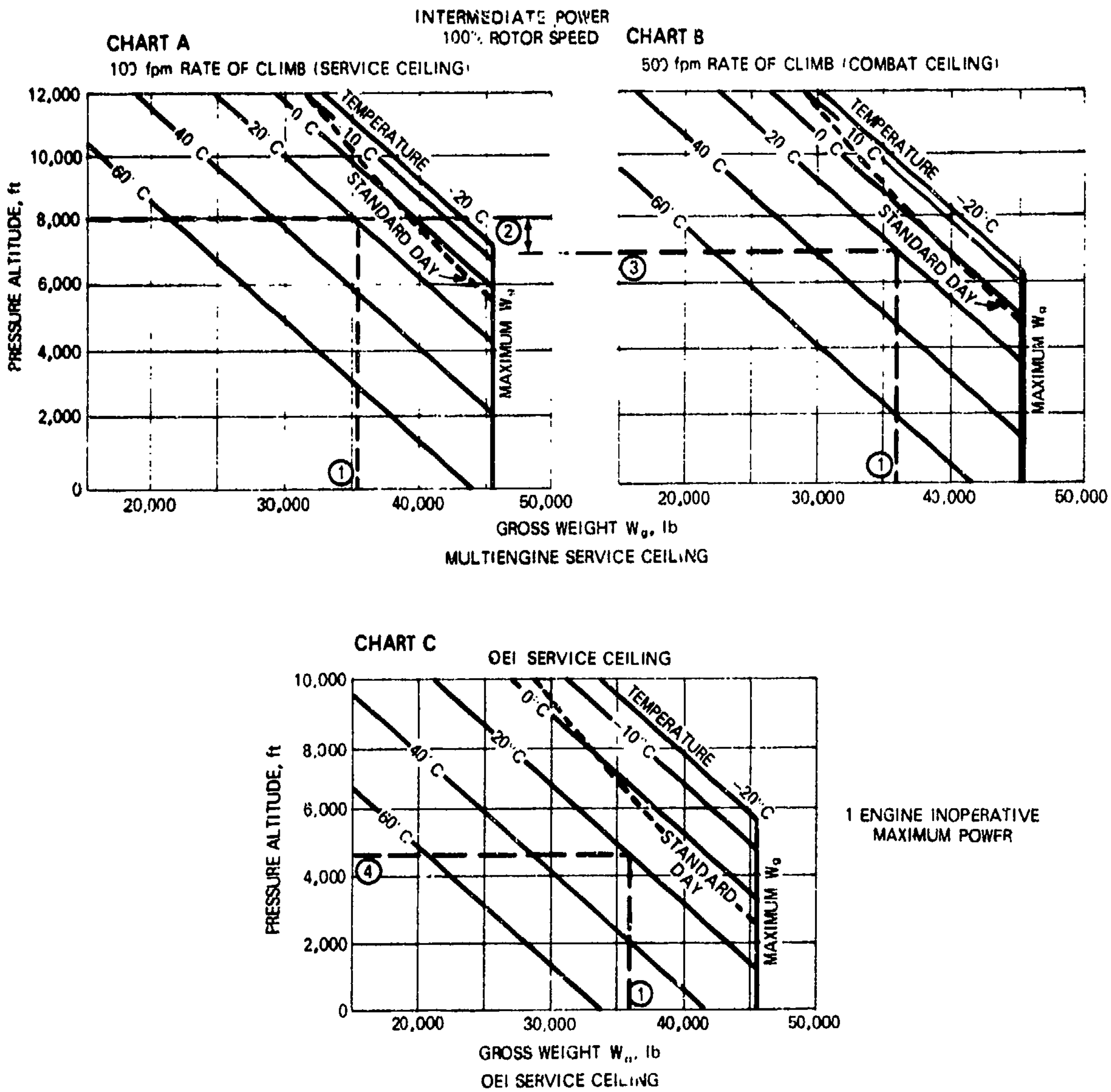


Fig. 3-145. Service and Combat Ceiling Capability

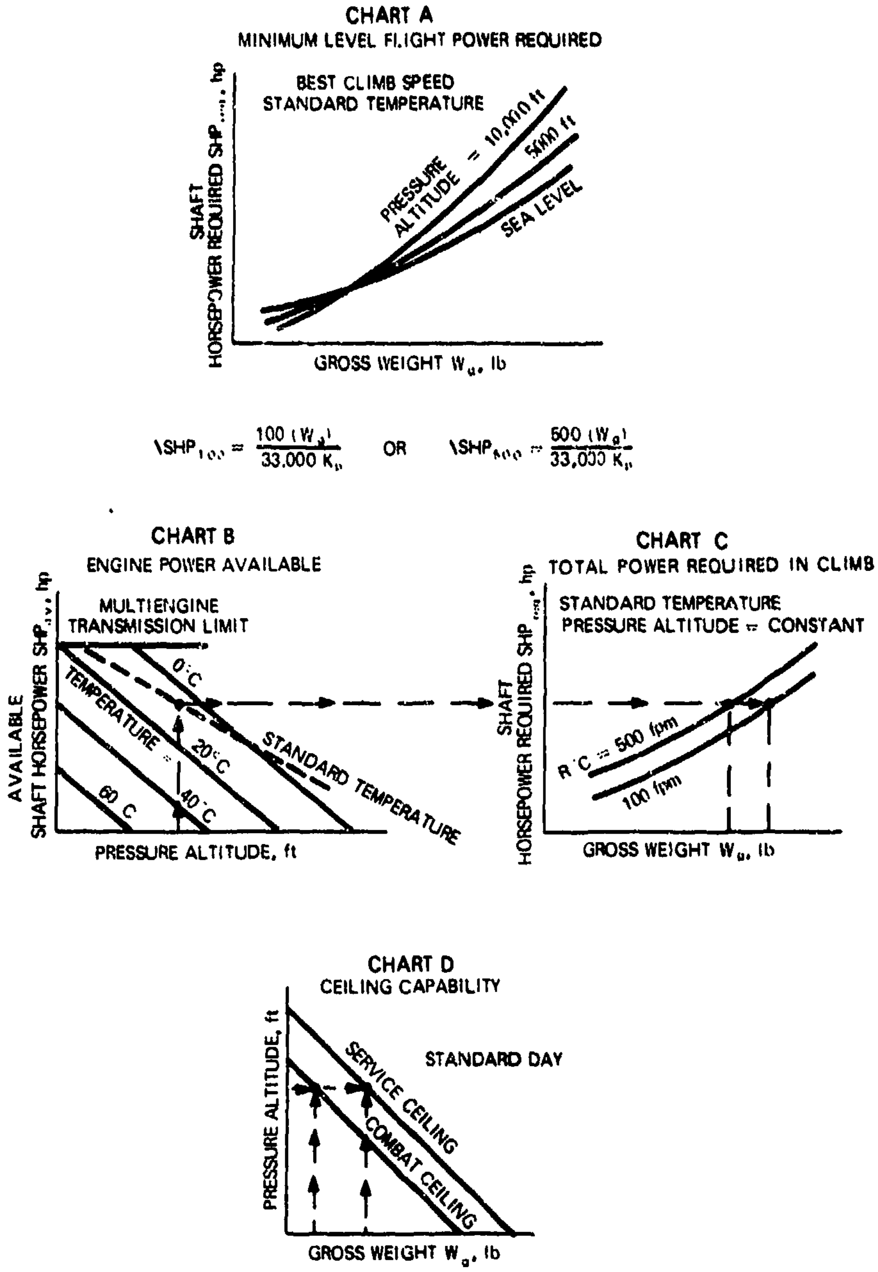


Fig. 3-146. Ceiling Capability Calculation

A more rigorous calculation of vertical climb download would relate the individual fuselage section drag coefficients and areas to their positions relative to the rotor, and would allow for downwash velocities less than $2v$, i.e., not an infinite distance downstream (par. 3-2.1.1.9).

3-4.2.7 Takeoff and Landing

3-4.2.7.1 General

Typical takeoff profiles are shown in Fig. 3-149. The vertical takeoff profile can be achieved by a helicopter only if power available is in excess of that required for hover out-of-ground effect (HOGE). The rolling takeoff profile is used when the helicopter has less power available than that required to hover in-ground effect (HIGE).

A practical consideration that usually dictates the takeoff profile used in service is the emergency run-on landing capability following an engine failure. Unless the helicopter is capable (or very nearly so) of hovering with one engine inoperative OEI, it cannot return with safety vertically to the takeoff point following an engine failure. A typical safe takeoff height/velocity corridor is shown in Fig. 3-150. For a helicopter with sufficient power to HOGE, the maximum safe hover height at zero velocity is typically about 0.5 to 0.6 times the rotor diameter. Therefore, takeoff capabilities are estimated realistically based upon a height-velocity profile such as that shown in Fig. 3-150.

When a helicopter is operated in accordance with its height-velocity limitations, the takeoff is accomplished by accelerating in level flight from hover to the airspeed at which the altitude above the ground can be increased. By climbing at the minimum safe speed, the aircraft can climb at the maximum climb angle until it clears a specified obstacle height. Minimum takeoff distance capability is derived from the power available in excess of that required for level, unaccelerated flight. The excess power required for acceleration and climb is depicted in Fig. 3-151.

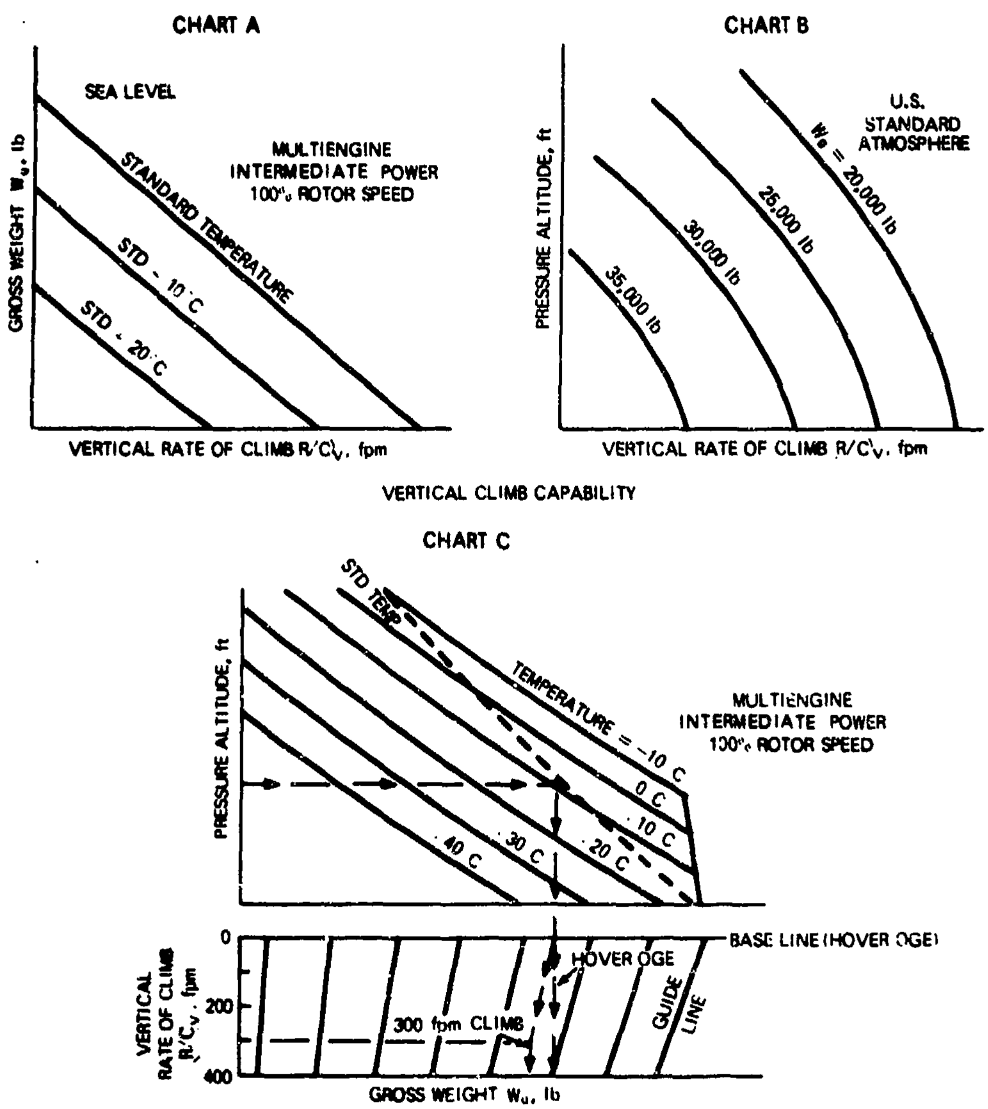


Fig. 3-147. Vertical Climb Performance

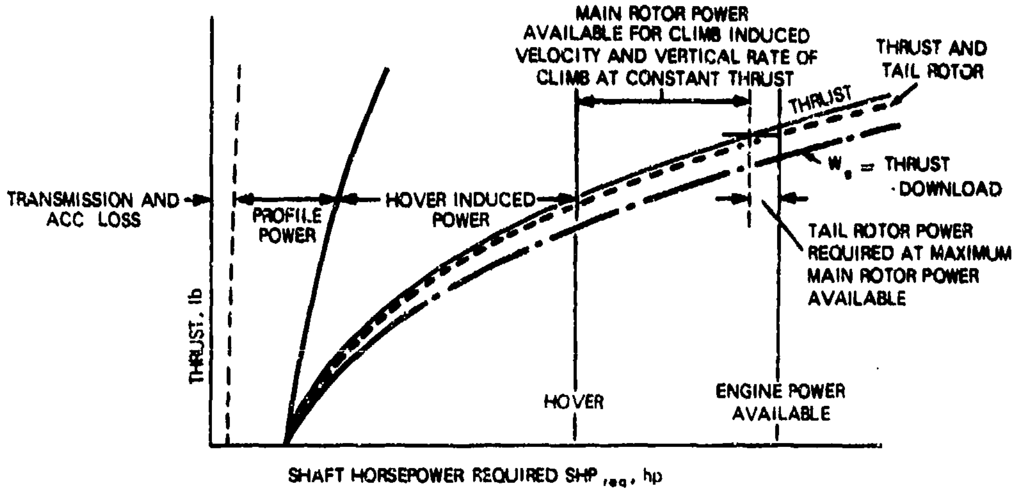


Fig. 3-148. Determination of Vertical Climb Capability

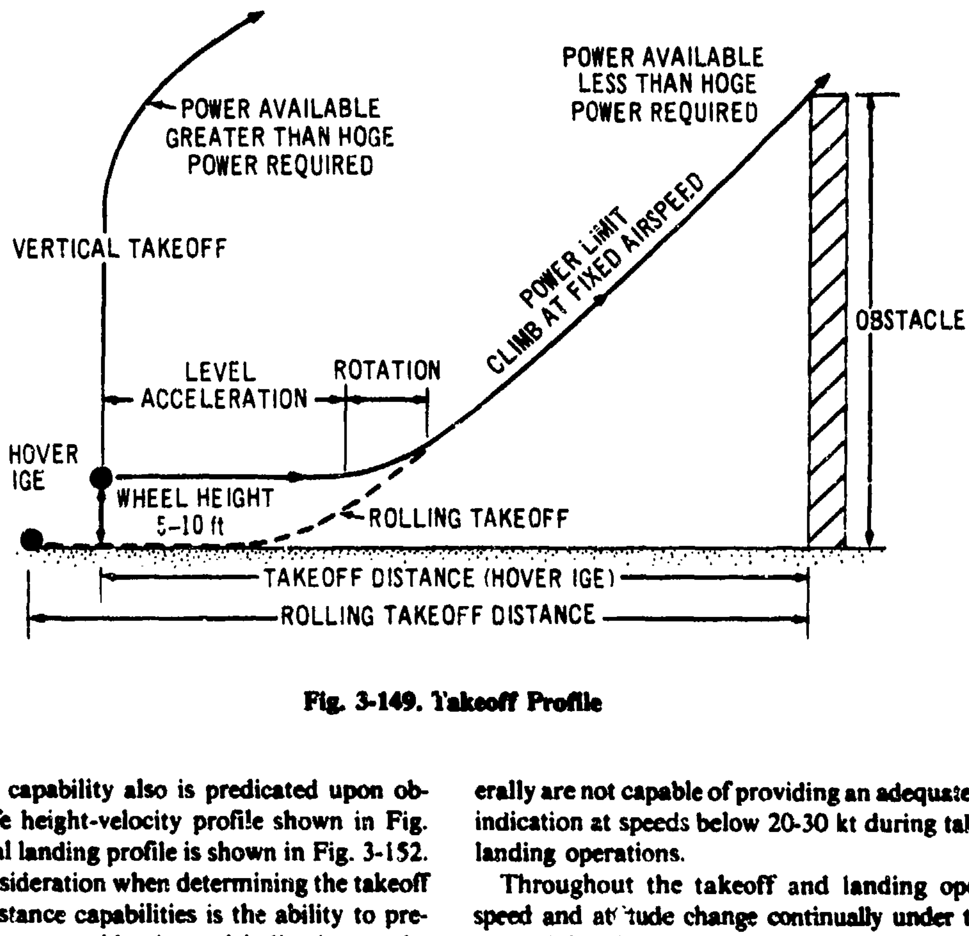


Fig. 3-149. Takeoff Profile

The landing capability also is predicated upon observing the safe height-velocity profile shown in Fig. 3-150. A typical landing profile is shown in Fig. 3-152.

Another consideration when determining the takeoff and landing distance capabilities is the ability to present a readable, repeatable airspeed indication to the pilot during the takeoff and landing maneuvers. Helicopter airspeed measurement systems in use today generally are not capable of providing an adequate airspeed indication at speeds below 20-30 kt during takeoff and landing operations.

Throughout the takeoff and landing operations, speed and attitude change continually under the influence of the pilot's control inputs. Consideration of this nonuniform motion rarely is necessary when determining performance capabilities of the helicopter. The takeoff distance calculations that follow can be made with sufficient accuracy by considering only the lift, drag, and propulsive forces while ignoring the inertia forces associated with the nonuniform motions that occur during the maneuver.

3-4.2.7.2 Takeoff Performance

The takeoff maneuver consists of three separate parts: acceleration, rotation, and climb. The paragraphs that follow describe the method for analyzing each phase of the maneuver. A series of helicopter performance charts, including fuselage characteristics, is used to derive the basic data.

3-4.2.7.2.1 Acceleration

Chart A, Fig. 3-153, illustrates a performance chart and the method of obtaining the propulsive force F_p and power required at a fixed lift (constant vertical thrust) for evaluating level acceleration. Chart B, which is derived from Chart A, presents a summary of net propulsive force ΔF_p for the airspeed range of interest during the acceleration. To provide an accurate appraisal of capability, this diagram reflects IGE operation.

The power available and the aircraft attitude limitations shown in Chart B, Fig. 3-153, define the maximum propulsive force available for accelerating the aircraft. This propulsive force is divided by the gross weight to define the variation of horizontal acceleration a_H capability with true airspeed as indicated in Chart A, Fig. 3-154.

Integration of this curve, by taking increments in airspeed and average acceleration a_a, defines the increase in true airspeed with time presented in Chart B, Fig. 3-160. Strip integration of the airspeed/time variation results in the total distance associated with the acceleration phase of the takeoff maneuver shown in Chart C. To define the performance characteristics with accuracy, the strip integration must use very small increments of velocity and time. This integration is accomplished most efficiently by a computer.

3-4.2.7.2.2 Rotation

When the target airspeed for climb is reached, the helicopter is rotated from a nose-low attitude to the trim attitude for climb, and a fixed airspeed is maintained. This attitude change may be as much as 20 deg for takeoffs using moderate power. During the rota-

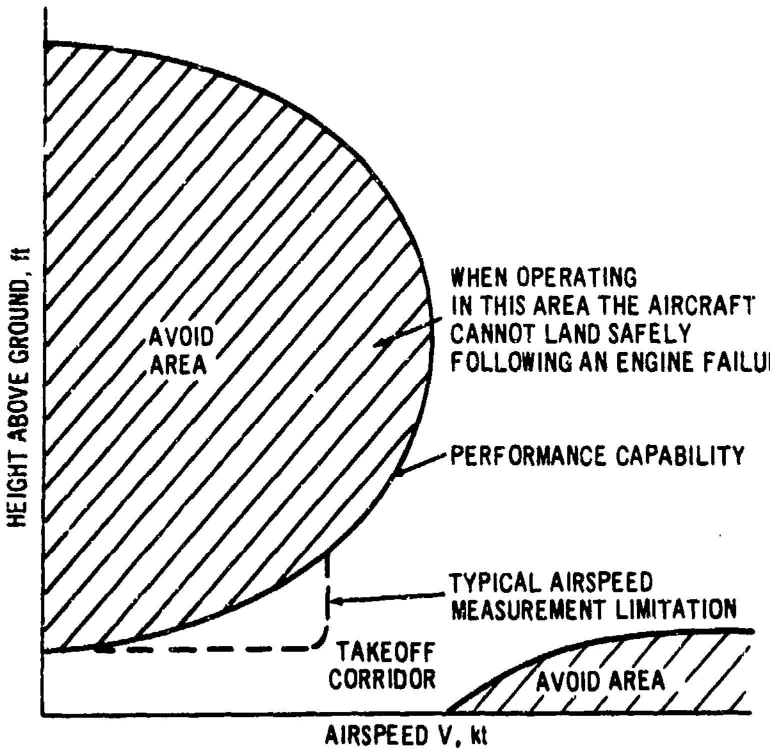


Fig. 3-150. Height-velocity Profile

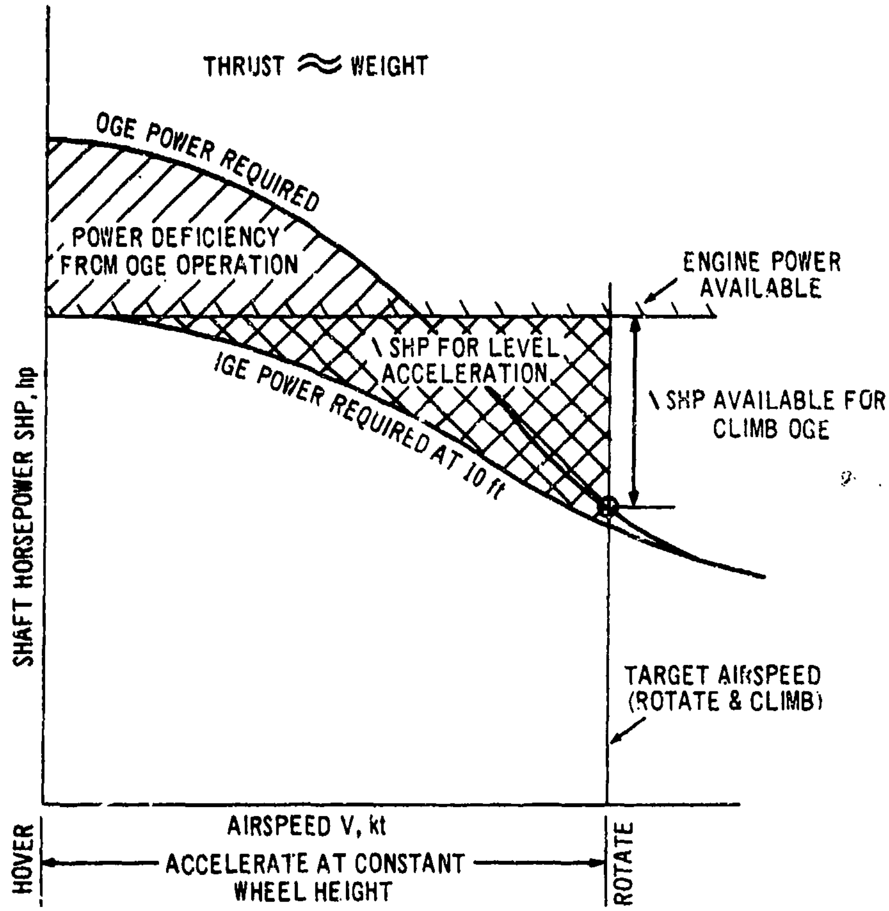


Fig. 3-151. Power Requirements for Takeoff

tion, the aircraft continues to accelerate slightly in the horizontal plane while accelerating vertically to the power-limited rate of climb speed. A rigorous analysis of the rotation would include specific aircraft pitch rate capability $d\theta/dt$ and evaluation of the horizontal and vertical acceleration at constant power and some average pitch attitude.

A reasonable approximation of the distance traveled during rotation is made by assuming a typical pitch rate (8-10 deg/sec) and calculating the time required to rotate. The power-limited climb attitude is defined from power charts at the desired airspeed and rate of climb. Eq. 3-201 is used to derive the time interval for rotation Δt_{rot}:

$$(\Delta t)_{rot} = \frac{\theta_{cl} - \theta_{acc}}{(\Delta\theta/\Delta t)_{nom}} \text{ , sec} \qquad (3\text{-}201)$$

where

$(\Delta\theta/\Delta t)_{nom}$ = nominal pitch rate, deg/sec

θ_{cl} = trim climb attitude, deg

θ_{acc} = acceleration attitude, deg

If airspeed and altitude are assumed constant during the rotation, the distance required for the rotation is equal to the product of the rotational airspeed and the rotational time from Eq. 3-201.

3-4.2.7.2.3 *Climb*

Following the rotation, the aircraft initiates the climb phase at a constant speed. The power map of Fig. 3-155, Chart A, is used to derive the rate of climb capability at the fixed airspeed. A vector representing gross weight is drawn to the various operating points along the constant-power lines, and the resulting excess net propulsive force is converted to a steady rate of climb using Eqs. 3-192 and 3-193. A summary of rate of climb capability at fixed gross weight and several airspeeds and power levels is presented in Chart B, Fig. 3-155. Because the aircraft is climbing away from the ground at some forward speed, ground effect is not included. This chart is used to derive the time and distance required to travel from level acceleration height to obstacle height.

Fig. 3-156 presents a series of charts that detail the climb distance analysis. Chart A of Fig. 3-156 is derived from Chart B of Fig. 3-155 at a given engine

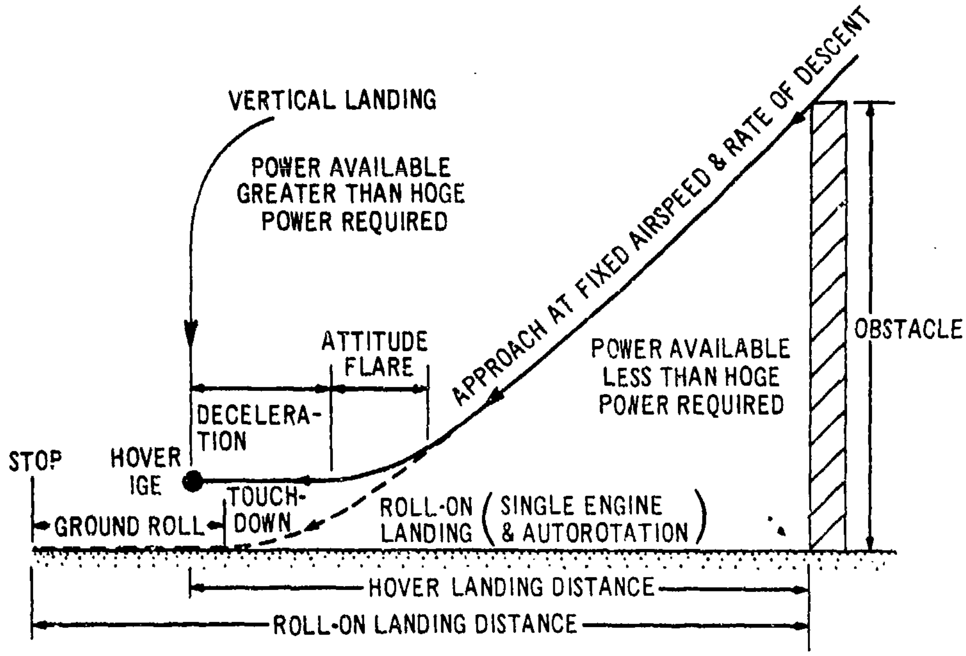


Fig. 3-152. Landing Profile

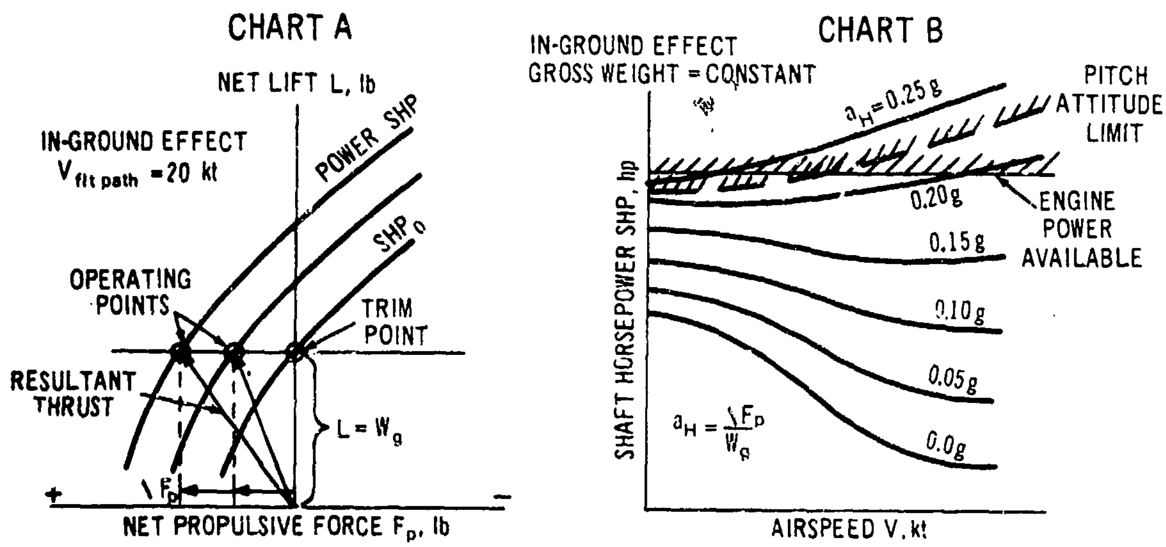


Fig. 3-153. Development of Takeoff Level Acceleration Capability

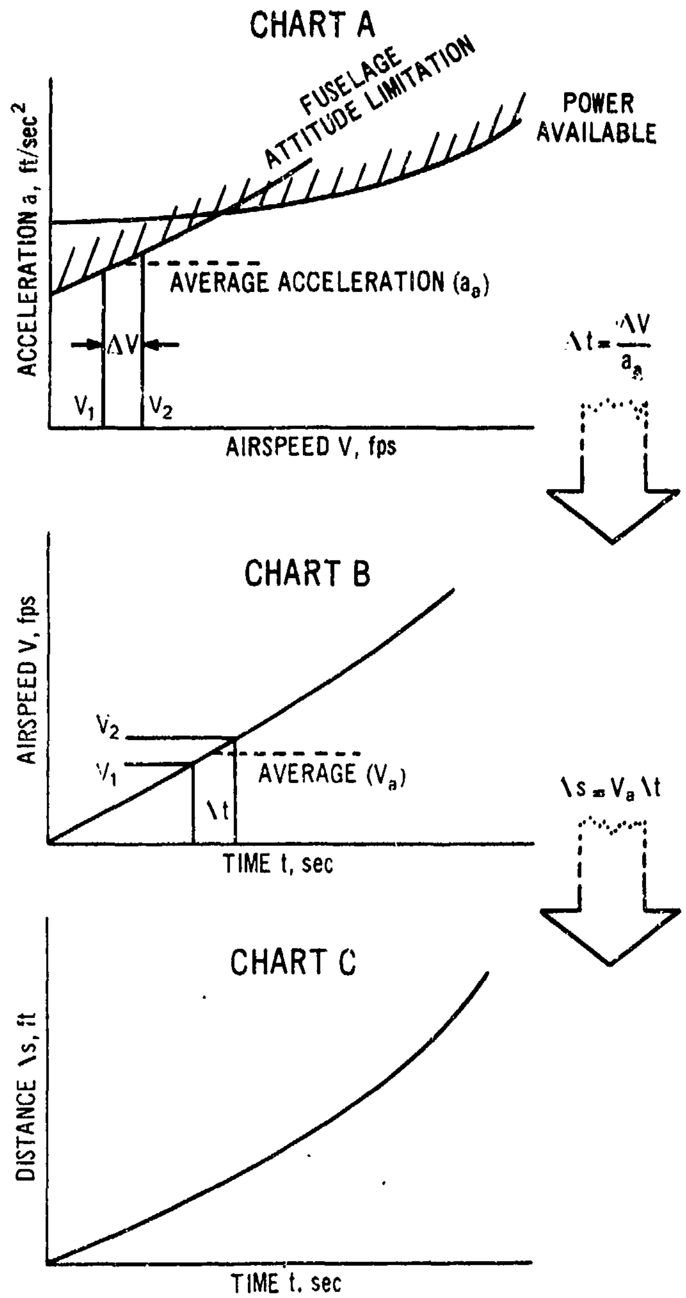


Fig. 3-154. Takeoff Performance — Acceleration Phase

power. Chart B, Fig. 3-156, is defined for several flight path airspeeds using Chart A and Eq. 3-202

$$\Delta h = V_{cl}(\Delta t) \quad , \text{ft} \qquad (3\text{-}202)$$

A given obstacle height is used to determine the time to climb at a selected airspeed. Chart C, Fig. 3-156, is constructed using the horizontal component of the flight path speed.

$$\Delta s = V_{flt\ path}(\cos \gamma)\, \Delta t \quad , \text{ft} \qquad (3\text{-}203)$$

where γ is $\sin^{-1}(V_{cl}/V_{flt\ path})$.

The climb angle γ defined above provides a measure of the steepness of the climbout. The maximum climb angle, as limited by power available, will provide the minimum takeoff distance. Considerations of low-speed operation in the height-velocity diagram and low-speed airspeed measurement system accuracy may limit the climb angle to less than maximum obtainable with available power and thereby lengthen the takeoff distance.

3-4.2.7.3 Landing Performance

The landing maneuver consists of three phases: approach, attitude flare, and deceleration. This is illustrated in Fig. 3-152. During the initial descent to the landing area, the airspeed and the rate of descent are held constant. As the ground is approached, an attitude flare is initiated (pitch attitude rotation) by increasing pitch attitude above the flight trim value. During the initial flare, the rate of descent is reduced to zero for a hover landing condition or to an acceptable descent rate for a roll-on landing. The final approach is made at zero or a small rate of descent with pitch attitude sufficiently above trim to supply a decelerating thrust. When there is not sufficient power to perform an IGE hover, a roll-on landing at some forward airspeed is required. The geometry of the particular helicopter will dictate landing pitch attitude limitations based upon tail rotor, landing gear, or fuselage clearances.

3-4.2.7.3.1 Approach

Multiengine landing capability is developed from power charts in a manner similar to that used in determining takeoff capability. For the landing situation, negative values of net propulsive force are used to define the rate of descent at a fixed gross weight and several airspeeds. Fig. 3-157 illustrates the development of the power requirement for the initial descent at fixed airspeed. The diagram is constructed similarly to Fig. 3-155 and the resulting negative propulsive force is converted to a steady descent rate using the relationships of Eqs. 3-192 and -193, with the climb angle γ replaced by the descent angle γ_D. Chart B, Fig. 3-157, is derived from Chart A for constant gross weight at several flight path airspeeds $V_{flt\ path}$ and negative propulsive forces $-\Delta F_p$. The rate of descent normally is limited to the zero power (autorotation) condition. If the rate of descent is increased beyond this condition, the rotor speed will increase above the trim power-on setting.

Fig. 3-158 illustrates the analysis of the initial approach phase of landing. An average value for the approach rate of descent is chosen from Chart A, Fig. 3-158. Because the landing is not power-limited, several combinations of approach airspeed and rate of descent are possible. Approach speeds of 30-40 kt and rates of descent of 1000-1500 fpm are conditions commonly used for both dual- and single-engine landings. Chart B is derived using several values for approach rate of

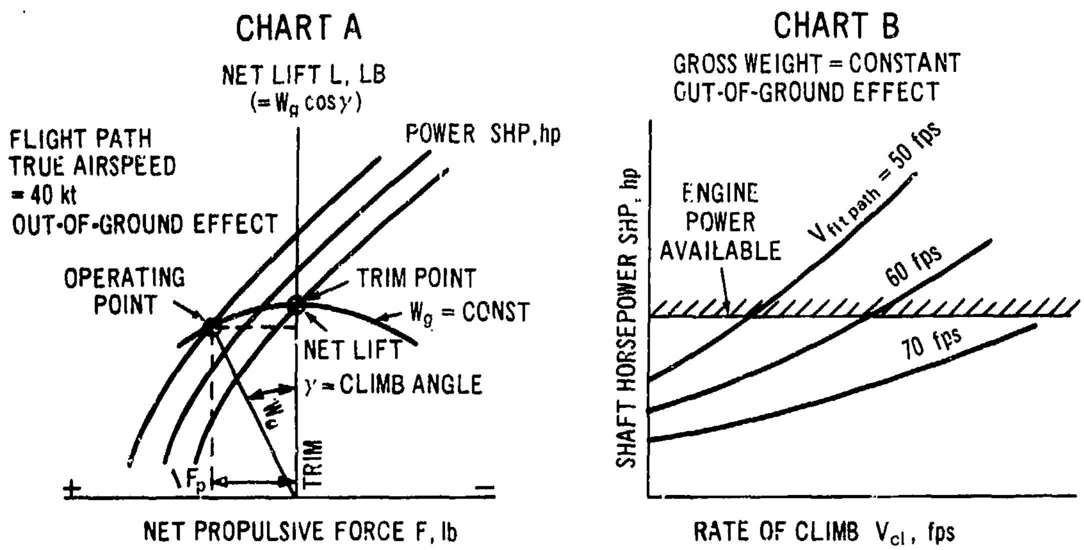


Fig. 3-155. Development of Takeoff Performance Climb Capability

descent and various descent times as shown on the chart.

The distance traveled during the initial descent as shown in Chart C, Fig. 3-158, is defined by the horizontal component of the flight path speed in the same manner as climb distance is calculated (Eq. 3-203) but with descent angle $\gamma_D = \text{Sin}^{-1}(V_D/V_{fl\,path})$ in lieu of climb angle.

3-4.2.7.3.2 Rotation

During the rotation phase, the fuselage is rotated nose-up to provide horizontal deceleration and to reduce the rate of descent. Deceleration is achieved by converting the negative propulsive force associated with the rate of descent to a decelerative force. Rigorous treatment of this transient condition would employ power charts to define the propulsive force and net lift at each fuselage angle during rotation. An approximation of the height lost and distance traveled during fuselage rotation is derived using a typical pitch attitude rate (8-10 deg/sec). The time is calculated as

$$(\Delta t)_{fl} = \frac{(\theta_{fl} - \theta_D)}{(\Delta\theta/\Delta t)_{nom}}, \text{ sec} \quad (3\text{-}204)$$

where

θ_{fl} = body attitude for level deceleration, deg

θ_D = body attitude during approach, deg

The average rate of descent during rotation and the flare time increment are used to define the height loss during the initial flare,

$$(\Delta h)_{fl} = \left(\frac{V_{D_{init}} + V_{D_{fin}}}{2}\right)(\Delta t)_{fl}, \text{ ft} \quad (3\text{-}205)$$

The horizontal distance traveled is equal to the product of the approach airspeed and the time required to flare, from Eq. 3-204.

3-4.2.7.3.3 Deceleration

The final attitude flare is accomplished at a fixed rate of descent while the helicopter is being decelerated to the horizontal touchdown speed. Fig. 3-159 illustrates the use of power charts to determine the power requirements and decelerative forces. The vector diagram of Chart A, Fig. 3-159, is constructed in a manner similar to the takeoff condition diagram; however, the net propulsive force is negative. Chart B defines the power requirements during the deceleration for a fixed gross weight. At the higher airspeeds, autorotation (zero power) limits the deceleration; maximum attitude limits deceleration at the low airspeeds.

The deceleration capability shown in Chart A, Fig. 3-160, is derived from Chart B of Fig. 3-159 for the specific gross weight. Chart B, Fig. 3-160, presents a time history of approach velocity while Chart C, Fig. 3-160, defines the distance traveled during the deceleration. The total landing distance over a specified obstacle height is the sum of the three component distances of the landing maneuver.

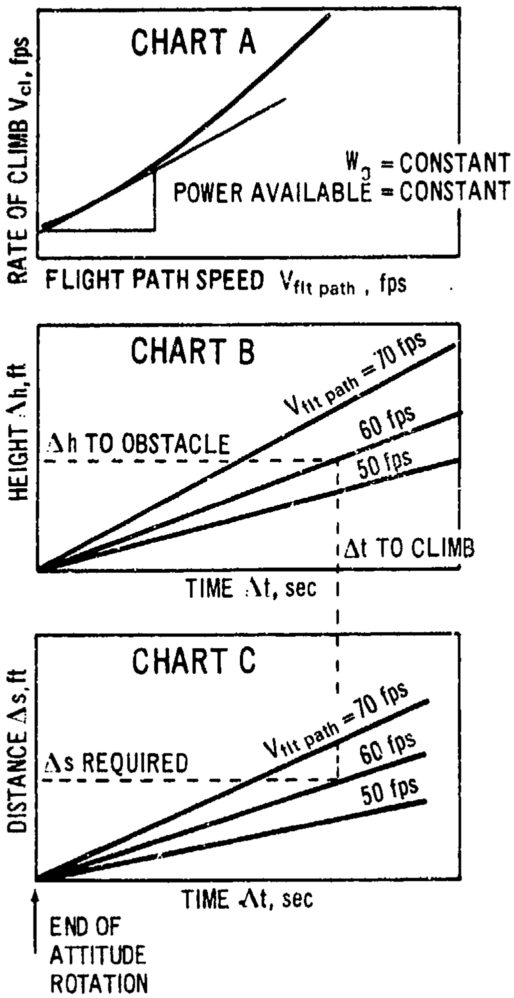


Fig. 3-156. Takeoff Performance — Climb Phase

3-4.3 AIRSPEED-ALTITUDE LIMITS

3-4.3.1 General

Airspeed-altitude limitations of helicopters can be categorized as those relating to power, structural integrity, stability, maneuverability, and comfort. The relative importance of each factor is dependent upon the configuration and the operational application. For example, a helicopter carrying bulky external loads can be speed-constrained by those loads. Mission requirements may include extreme maneuver ability, which means substantial control margins and maneuvering capabilities, and the airspeeds and/or altitudes at which these are not available become limitations upon the helicopter flight envelope.

Helicopter speed capability tends to decrease with altitude as retreating blade stall produces excessive blade and control system vibratory loads. The addition of a wing permits speed capability to be maintained to higher altitudes. Adding auxiliary propulsion allows higher speed at all altitudes provided adequate power is available (Fig. 3-161).

The paragraphs that follow discuss the various helicopter airspeed-altitude constraints and their sensitivity to design parameters and mission requirements.

3-4.3.2 Definitions

Speeds commonly defined are:

1. IAS. Indicated airspeed is equal to the pitot static airspeed indicator reading as installed in the helicopter without correction for airspeed indicator system errors but including the sea level standard adiabatic compressible flow correction. (This latter correction is included in the calibration of the airspeed instrument dials.)

2. CAS. Calibrated airspeed is equal to the airspeed indicator reading corrected for position and instrument error. (As a result of the sea level adiabatic compressible flow correction to the airspeed instrument dial, CAS is equal to the true airspeed (TAS) in standard atmosphere at sea level.)

3. EAS. Equivalent airspeed is equal to the airspeed indicator reading corrected for position error, instrument error, and for adiabatic compressible flow for the particular altitude. (EAS is equal to CAS at sea level in standard atmosphere.)

4. TAS. True airspeed of the helicopter relative to undisturbed air.

The true airspeed to equivalent airspeed relationship TAS $= \mathrm{EAS}(\rho_0/\rho)^{1/2}$ is shown graphically in Fig. 3-162. Indicated airspeed often is used in describing helicopter performance because flight handbooks are in the pilot's frame of reference. The discussion following, however, limits itself to use of true airspeed, a practice consistent with the requirements of MIL-C-5011 for definition of standard aircraft characteristics.

Altitude also can be defined in several ways. Absolute, or tapeline, altitude is the measured distance above a ground reference, as would be recorded by a radar altimeter. Pressure altitude is the altitude above mean sea level in a standard atmosphere at which a given barometric pressure is to be found. Pressure alti-

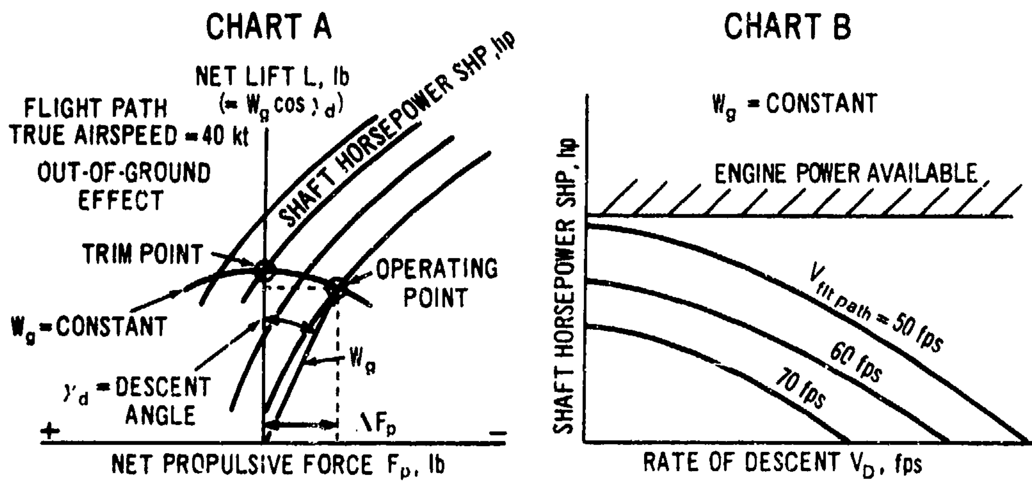


Fig. 3-157. Development of Approach Rate of Descent

tude is recorded by a standard barometric altimeter that is set to a standard sea level pressure of 29.92 in. of mercury. Density altitude is pressure altitude adjusted for temperature deviations from standard; e.g., 6000 ft pressure altitude at 95°F is equivalent to 9600 ft density altitude (same density as 9600 ft pressure altitude at standard day temperature). Density altitude often is a convenient way to normalize density-dependent performance factors. In this discussion, however, altitude refers to pressure altitude under standard atmospheric conditions.

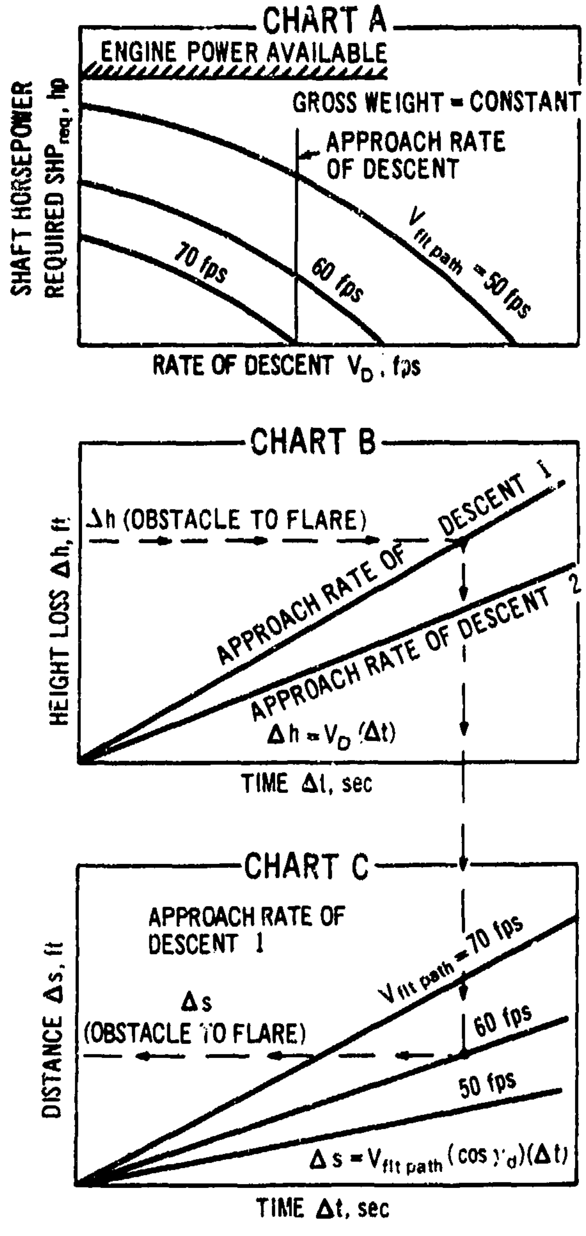


Fig. 3-158. Landing Performance — Approach Phase

3-4.3.3 Power Constraints

The fundamental limitation upon flight speed is the power available. Power-limited speed is simply the speed at which required power equals available power. Available power may be constrained by engine output or by transmission rating. The engine rating usable for each speed category generally is defined by the aircraft detail specification or, in its absence, by MIL-C-5011.

Power-limited speed usually is maximized at some nominal altitude and varies only slightly with altitude at standard temperature. However, temperature variation has significant impact upon power available because the power output of turbine engines is very sensitive to ambient temperature.

The minimum flight speed of a helicopter also can be power-limited. At zero airspeed, altitude is constrained by the OGE hover ceiling. The power-limited speed envelope is shaped like an inverted power-required curve.

Constraints also are imposed by requirements for safe recovery in the event that an engine becomes inoperative. This criterion will normally define minimum and maximum speed limitations. The altitude-velocity diagram defines the airspeed altitude envelope within which the aircraft can operate and still recover safely from an engine power loss (Fig. 3-163).

The low-speed/altitude boundary is determined by the ability to autorotate and land safely in the case of power loss. Recoverability from partial or complete power loss also may define the high-speed boundary. At speeds above some limiting value, a sudden reduction in power may produce an unacceptable loss of rotor rpm or altitude before speed can be reduced to the new power limit. This characteristic is a dynamic one that is dependent upon rotor inertia and pilot response, and thus is not generalized easily. However, as helicopter speeds approach 200 kt, it may become an important constraint upon allowable speed at low altitude.

Obviously, one engine inoperative (OEI) criteria are influenced directly by the number of engines installed. Introduction of transmission rating limitations upon available power is a method often used to insure adequate power margin.

3-4.3.4 Structural Constraint

Structural limitations upon helicopter flight speeds can be broken down into those relating to one or more of four areas: the airframe, the rotor blades, the rotor control system, and the rotor shaft.

Airframe structural integrity for level flight is keyed to dynamic pressure. The surface pressures, both positive and negative, generated by the slipstream must be within design limits. Helicopters assume various pitch and sideslip orientations depending upon CG location and control input, and the pressures must be acceptable within a given range of this angular orientation. Because surface pressures are directly proportional to free stream dynamic pressure, speed is very critical. Within appropriate margins for trim orientation and gust conditions, the airframe structural true airspeed envelope will increase with altitude and temperature as air density decreases (following a constant equivalent airspeed line).

Rotor blade stresses represent another important structural constraint. The degree of blade structural damage that may occur is dependent upon both steady and vibratory stresses generated. The higher the steady stress, the lower the allowable vibratory stress for the same damage criteria. Steady stress is a function primarily of gross weight and centrifugal force, and is relatively insensitive to speed and altitude. Vibratory stress, on the other hand, arises because of variations in loading as the blade rotates azimuthally, and is speed-sensitive. As speed increases, the variation in the loading on the advancing and retreating blades increases, causing higher vibratory loads and corresponding stresses. Both high angle of attack and stall on the retreating blade, and possibly negative angle of attack and compressibility on the advancing blade, contribute to increased vibratory stresses. For a given rotor system and rpm, stress-limited speed tends to vary inversely with gross weight. Both main and tail rotor blade stresses, of course, must be considered.

Rotor system control loads represent one of the most significant constraints upon helicopter speed. As with blade stresses, control loads have both a steady and a vibratory content. Steady loads are a function of blade loading and control system geometry, while vibratory loads result from the azimuthal variation in blade loads.

Because it is configured to govern blade pitch angle, the control system is very sensitive to blade bending and aerodynamic pitching moments. Thus, control loads build up rapidly with blade deflection when the retreating blade tip begins to stall, and an aerodynamic nose-down pitching moment is generated. The onset of retreating blade stall can be related to the angle of attack of the blade tip, which in turn is established by local dynamic pressure and required blade lift. For a given gross weight, therefore, onset of stall is proportional to the square of the net tip speed and air density. Thus, for a given level of acceptable control load, limit speed will vary approximately as:

$$V_{lim} \approx \Omega R - \sqrt{\frac{k}{\rho W_g}}\ , \text{fps} \qquad (3\text{-}206)$$

where

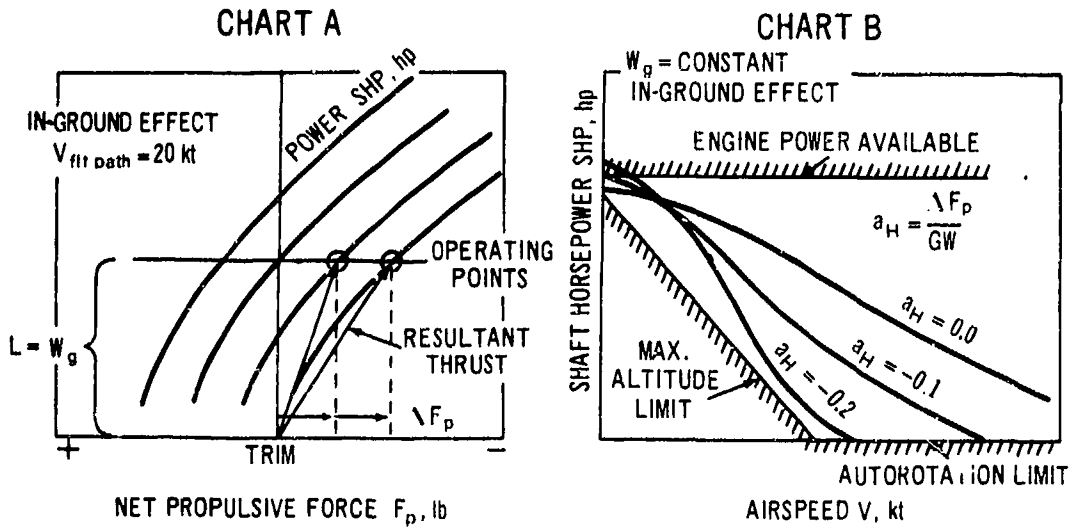


Fig. 3-159. Determination of Landing Deceleration Capability

k = retreating blade stall factor dependent upon blade geometry and loading, lb²/ft²

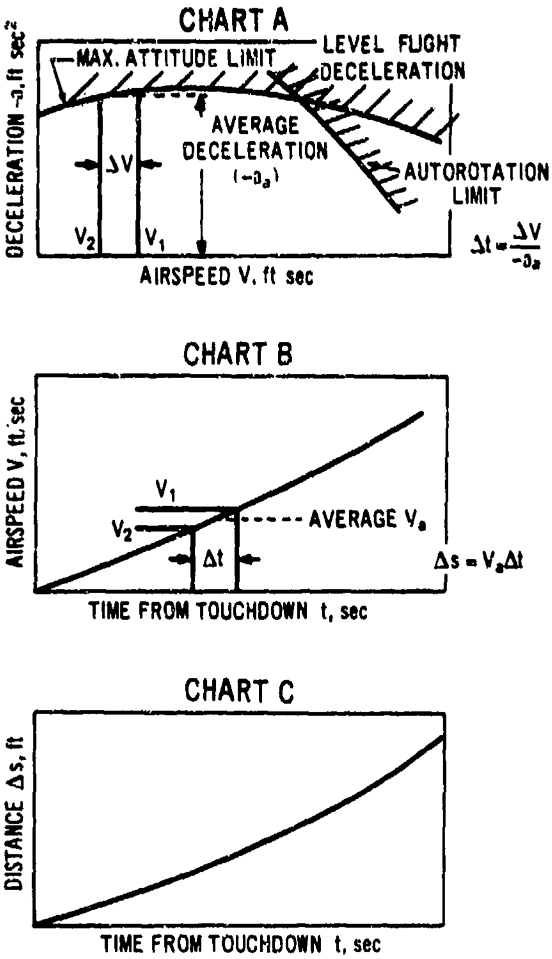


Fig. 3-160. Landing Performance — Deceleration Phase

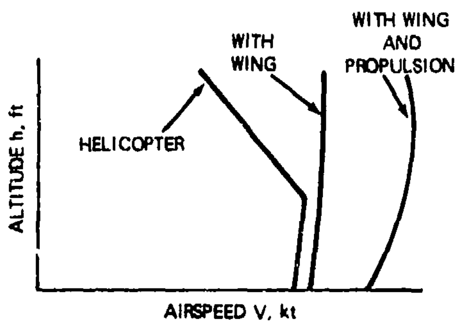


Fig. 3-161. Typical Altitude-speed Limits

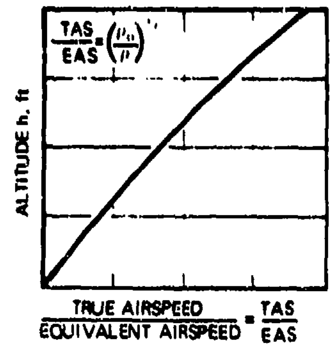


Fig. 3-162. Relationship Between True and Equivalent Airspeed

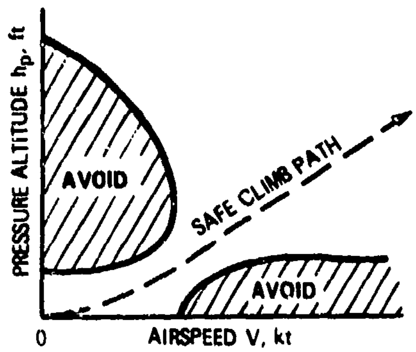


Fig. 3-163. Typical Altitude-velocity Diagram

For conventional helicopters, the empirical constant k generally is in the order of 50 lb²/ft².

A large number of rotor modifications that can help alleviate retreating blade stall and attendant control loads suggest themselves. The maximum blade lift coefficient can be increased with cambered airfoil sections or boundary layer control with appropriate consideration for the associated aerodynamic pitching moment. Spanwise loading can be optimized with high twist, second harmonic feathering control, and lateral CG offset obtained with offset or semirigid flapping hinges. And, of course, the rotor can be unloaded either by increasing blade area or by incorporating a wing and/or auxiliary propulsion.

Rotor shaft stresses in bending are caused by the hub moment generated by the rotor to trim the helicopter in an acceptable attitude. The required hub moment is a function of gross weight, speed, and CG location. If a requirement to keep the airframe reasonably level at extreme CG excursions demands very high hub mo-

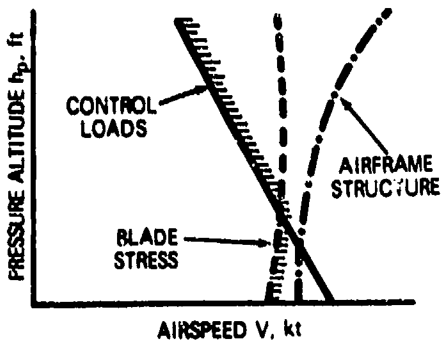


Fig. 3-164. Typical Structural Altitude-speed Constraints

ments, limiting shaft stresses may constrain speed or altitude.

The general character of individual structural altitude/airspeed limits now can be summarized. Airframe constraints are proportional to dynamic pressure so that allowable true airspeed increases with altitude (Fig. 3-164). Blade stresses are dependent chiefly upon gross weight and speed, and are independent of altitude. Control loads are a function of both gross weight and altitude, and generally represent the most limiting structural constraint. Shaft stresses generally are not limiting.

It must be remembered that the sensitivity of blade stresses and control loads to gross weight is more properly a sensitivity to blade lift, so that speed increases can be realized through the use of wings and/or auxiliary propulsion to unload the rotor and expand the envelope.

3-4.3.5 Stability Constraints

Stability is the tendency to return to equilibrium following a disturbance. A flight condition resulting in any uncontrollable instability is unacceptable, and, in fact, a generous margin must be provided for level flight conditions to accommodate necessary maneuvering and potential gust loadings.

Stability constraints upon airspeed consist of those introduced by the rotor, the blades, the airframe, and an external load. Generalization as to the relative importance of these, and definition of specific trends with regard to altitude or gross weight, is found in Chapter 6. The significance of stability limits cannot be overemphasized, however, as the consequences of flying inadvertently into an unstable regime can be catastrophic.

3-4.3.6 Maneuverability and Controllability

Controllability is a less-specific property than stability, and relates to speed, accuracy, and the ease with which the helicopter can be made to maneuver from the steady-state flight condition.

The most obvious controllability limitation is that established by maximum available control travel. A helicopter obtains forward propulsion by superimposing a longitudinal cyclic blade angle variation on top of the collective pitch required to provide lift. The location of the CG imposes the most significant variant in control requirements. The aerodynamic trim of the airframe in forward flight imposes variations at higher speeds. Climb or descent attitude and power condition introduce additional variables to be considered. Speed often is restricted by aft locations of the CG, and therefore can be limited by inadequate blade angle or pilot control travel. A control margin for necessary maneuvering from the steady-state trim flight condition must be kept available and, in general, it should be at least 10% throughout the operational flight envelope.

Collective pitch constraints also can limit speed and altitude capabilities at lower speeds because the required blade angle increases with inflow in climb and forward flight and inversely with air density at constant gross weight.

At low speeds, tail rotor adequacy must be considered. The tail rotor must counteract main rotor torque and also must provide sufficient additional yawing moment for precise hovering control and maneuvering. Because the latter usually involves transient rather than steady-state operation, power is not the limiting factor. Instead, the sensitivity of tail rotor response (the slope of thrust with rudder input) and the absolute pitch-limited thrust of the tail rotor are constraining. Tail rotor thrust/pitch sensitivity is related to blade loading, and decreases markedly as the blades begin to stall. A reasonable ground rule is that sensitivity should not decrease to less than half the steady trim value. At high speed, a vertical fin helps trim main rotor torque, and tail rotor thrust/pitch sensitivity is enhanced greatly. Thus, tail rotor adequacy considerations do not affect the high-speed envelope except to the degree that tail rotor stresses become significant. The altitude-airspeed constraints imposed by main rotor control limits and tail rotor adequacy, then, look something like Fig. 3-165.

The altitude-airspeed envelope also can be restricted by considerations of crew and passenger comfort. Comfort factors include atmospheric oxygen content, vibration, noise, and attitude.

The most obvious altitude constraint is the oxygen level in the ambient atmosphere. Without pressurization or special oxygen apparatus, personnel should not be exposed to extended operation at pressure altitudes above 10,000 ft, where the oxygen pressure is two-thirds that at sea level.

Several of the structural and stability constraints also may cause undesirable vibration. Fuselage vibration induced by the rotor, engines, and transmission generally increases with gross weight and forward speed (Fig. 3-166). The unique response characteristics of the airframe significantly affect the local impact of vibration, and precise tuning often can reduce the vibratory impact at specific flight conditions. Continuous flight obviously permits less vibration than short spurts at maximum speed. The need to operate sensitive equipment such as gunsights or motion sensors also may limit the acceptable level of vibration.

The nose-down attitude of the aircraft cannot exceed that acceptable to the crew and/or passengers. As

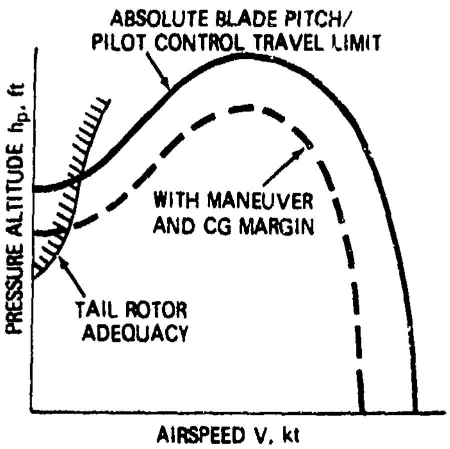


Fig. 3-165. Altitude Constraints Imposed by Main Rotor Control Limits and Tail Rotor Adequacy

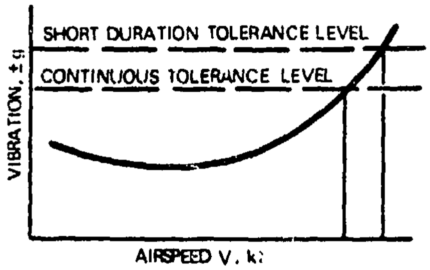


Fig. 3-166. Typical Vibration Characteristics

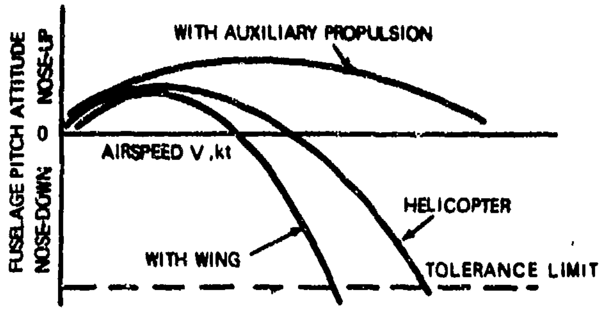


Fig. 3-167. Fuselage Pitch Attitude

speed increases, the helicopter nose must be dropped to provide necessary rotor propulsive force without excessive blade flapping. A forward CG condition aggravates the problem. Often, high-speed helicopters incorporate built-in forward rotor shaft incidence as compensation. Use of a wing to unload rotor lift also can aggravate the attitude problem, because the rotor must be tilted even further forward to provide propulsive force with a smaller thrust vector. Use of auxiliary propulsion, on the other hand, eliminates the need for the rotor to tilt forward and allows the aircraft to be trimmed level (Fig. 3-167).

Noise, both external and internal, imposes a final potential constraint upon speed. As speed increases, the retreating blade may stall and the advancing blade tip is exposed to higher Mach numbers. The resulting pressure impulses are observed as noise and may be unacceptable. Crew or passenger tolerance to the noise transmitted internally by the rotor, engines, or drive system also may restrict allowable speed for sustained flight.

3-5 SPECIAL CONSIDERATIONS

3-5.1 AUTOROTATIONAL CHARACTERISTICS

3-5.1.1 Basic Mechanism of Autorotation

One of the important features of a helicopter, in contrast to most other types of aircraft, is the ability it affords to make a landing following a complete power failure with a reasonable expectation of no injuries and no structural damage. Both the military and the FAA recognize this capability by requiring that it be demonstrated even in multiengine helicopters. Autorotation not only is used following power failure, but also is generally the fastest means of descent in case of other unexpected events such as fire, severe vibration, or control system malfunction.

The basic mechanism of autorotation and hover is illustrated in Fig. 3-168, which shows the lift and drag vectors acting upon a blade element in hover, in a slow vertical descent, and in a fast vertical descent. The lift vector is held constant for the three flight modes by decreasing the pitch of the blade as the rate of descent is increased. It may be seen that at some rate of descent the lift vector L, which always is perpendicular to the local resultant velocity, will be tilted forward enough to balance the drag vector D. At this point, no power is required to keep the blade element rotating and it is in autorotation. In the case of a complete rotor, the lift and drag forces on every blade element will not necessarily be balanced, but the integrated torque due to the forward tilt of the lift vectors on all of the elements will balance the integrated torque due to the drag on all of the elements. This balancing of the torque applies to autorotation in forward flight as well as in a vertical descent.

For a given combination of disk loading and collective pitch, vertical autorotation is a stable flight condition that is defined by a unique combination of rotor speed and rate of descent. This means that if the rate of descent increases, both the blade lift and its forward tilt also will increase with a resulting increase in rotor speed. The latter produces another increase in lift and a corresponding decrease in the rate of descent back to its original value. If the rotor speed increases without an accompanying increase in the rate of descent, the lift vector will tilt back and the drag vector will increase, thus causing the rotor speed to return toward its initial value. Similarly, in forward flight autorotation is stable at a given combination of forward speed, disk loading, and collective pitch and is defined by a unique combination of rotor angle of attack with respect to the flight path, glide path angle with respect to the horizon, and rotor speed.

3-5.1.2 Entry into Autorotation

The entry into autorotation is the maneuver occurring between the instant of power plant failure and the point at which steady autorotation is achieved. In normal flight, the helicopter rotor requires power to keep it rotating. If the source of power suddenly fails, the rotor initially will require the same power but, being unable to obtain it from the power plant, will obtain it from its own kinetic energy by slowing down. During this process, it can approach a stalled condition that, if actually encountered, would bring the rotor to an uncontrollable flight condition. An equation for the rate at which the rotor slows down if the pilot takes no action following the power failure can be derived by assuming that the decelerating torque is proportional to the square of the rotor speed. The resultant equation is a function of the original torque Q_0 and rotor speed Ω_0:

$$\dot{\Omega} = -\frac{Q_0(\Omega/\Omega_0)^2}{I}, \text{ rad/sec}^2 \qquad (3\text{-}207)$$

Eq. 3-207 can be rewritten as a nonlinear integral equation in terms of the equivalent time t_e during which the total kinetic energy could be converted into power at the initial level; e.g., by increasing collective pitch as the rotor speed decreases:

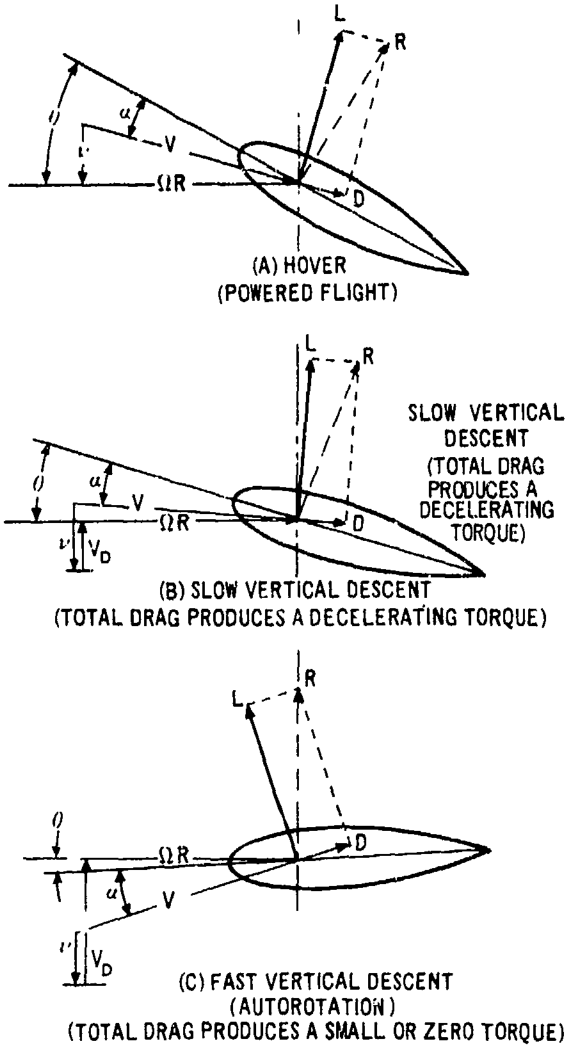


Fig. 3-168. Basic Mechanism of Autorotation

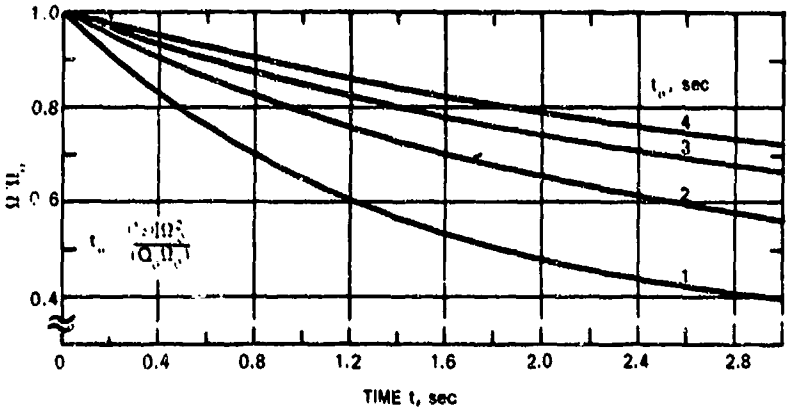


Fig. 3-169. Rotor Speed Decay Following Power Failure

$$\frac{\Omega}{\Omega_0} = \frac{1}{2t_e}\int_0^t \left(\frac{\Omega}{\Omega_0}\right)^2 dt \quad \text{, dimensionless} \qquad (3\text{-}208)$$

where

$$t_e = \frac{I\Omega_0/2}{Q_0} = \frac{I\Omega_0^2/2}{(Q_0\Omega_0)} \quad \text{, sec} \qquad (3\text{-}209)$$

The denominator of the second form of the expression for t_e is the rotor power. For most helicopters and flight conditions, the equivalent time is between 1 and 4 sec. Eq. 3-208 has been evaluated for several values of equivalent time and the results plotted on Fig. 3-169. It may be seen from this figure that the rotor speed can decay by as much as 35% in the first second after power loss.

To prevent a dangerously low rotor speed, the pilot must initiate prompt control action following power failure. In hover and low-speed flight, the proper pilot action is to reduce the collective pitch in order to reduce the power required, and quickly to obtain the rate of descent necessary for steady autorotation. At high speeds, the decrease in collective pitch may be delayed if, instead, an immediate cyclic nose-up flare is performed. This flare simulates a rate of descent by establishing an attitude such that the air is coming upward through the rotor. Another reason for this type of entry into autorotation at high speed is that collective pitch reduction causes the rotor to flap down in front, thus delaying achievement of the condition of upward flow through the rotor. Sometimes this nose-down flapping, in conjunction with the accompanying reduction in coning, is large enough to cause a blade strike on the fore-body.

In order to give the pilot an opportunity to react with sufficient rapidity, he must be given an adequate indication of engine failure. This generally is not a problem in a reciprocating-engine helicopter because the change in noise level is readily apparent. With turbine-powered helicopters, however, the prime source of noise in the cockpit may be the transmission, which produces essentially the same noise level whether under power or not. For this reason an engine failure alarm system separate from the normal flight and engine instruments may be required. The signals that can be used to trigger the alarm include a rotor speed below some preset level; high rotor speed deceleration rates; zero or negative torque between the engine and the transmission; or a sudden drop in engine temperature, pressure, or fuel flow. It may be necessary to use more than one of these signals to avoid nuisance alarms.

During entry into autorotation from hover, the pilot may elect either to make a nearly vertical descent all the way to the ground or to make a transition to forward-flight autorotation. Because the rate of descent in autorotation is proportional roughly to the power required by the rotor, the minimum rate of descent occurs at the forward speed corresponding to the minimum point on the power-required curve. Vertical descents are practicable only on helicopters with low disk loadings; such helicopters also normally require lower power for hovering. When entering vertical autorotation, a special problem may occur as the helicopter accelerates toward its stable rate of descent and passes through the critical part of the "vortex ring" state. The vortex ring state exists between the hover mode and the autorotation mode for vertical descent.

In this condition, the rotor carries with it a mass of rotating air, like a smoke ring, which is rotating down through the middle of the rotor and up on the outside. This mass of air becomes very unstable in the critical part of the vortex ring state and breaks down in large and random fluctuations that act upon the rotor as sharp-edged gusts, producing a situation in which the helicopter is difficult to control. Fortunately, the vortex ring state either can be passed through quickly by decreasing rotor power or alleviated by descending with a moderate amount of forward speed.

During autorotation at forward speed, the rotor speed sometimes is difficult to raise from the low value it decreases to during the entry if the minimum collective pitch setting is not low enough to maintain the rotor speed at the desired value. In these cases, the rotor speed can be increased by making steady turns; the increased load factor and corresponding increased angle of attack increase the flow of air up through the rotor and, thus, the energy that can be extracted from the airstream. This extra energy allows autorotation at a higher rotor speed at the expense of a higher rate of descent.

3-5.1.3 Calculation of the Rate of Descent in Autorotation

3-5.1.3.1 Vertical Autorotation

The equation for the rate of descent V_D in steady vertical autorotation may be derived by the method in Ref. 16 by setting the idealized rotor equation for torque coefficient equal to zero:

$$C_Q = \frac{\sigma\bar{C}_D}{8} - \frac{C_T^{3/2}}{\sqrt{2}}(\bar{V}_D - \bar{v})$$

$$= 0 \quad \text{, dimensionless} \qquad (3\text{-}210)$$

In this equation the descent and induced velocities have been nondimensionalized by dividing by the induced velocity in hover v_0. Eq. 3-210 can be rewritten as

$$\frac{\sigma\bar{C}_D/8}{C_T^{3/2}/\sqrt{2}} = \bar{V}_D - \bar{v} \quad \text{, dimensionless} \qquad (3\text{-}211)$$

In the autorotative condition, the nondimensionalized rate of descent $\bar{V}_D$, and the nondimensionalized net velocity through the rotor $\bar{V}_D - \bar{v}$, are related as shown on Fig. 3-170. Thus $\bar{V}_D$ can be found for the value of $\bar{V}_D - \bar{v}$ determined in Eq. 3-211 and V_D can be found in turn by multiplication by v_0 from Eq. 3-4. Note that from Fig. 3-170, an average value of $\bar{V}_D$ is 2.0. Thus the rate of descent in vertical autorotation is approximately twice the corresponding value of the induced velocity in hover. For sea level conditions (ρ = 0.002378 slug/ft^3), this gives approximately:

$$V_D \approx 29\sqrt{w} \quad \text{, fps} \qquad (3\text{-}212)$$

(This equation, incidentally, also gives a good approximation for the rate of descent of a parachute.)

3-5.1.3.2 Forward Flight Autorotation

In forward flight autorotation, the rate of descent can be calculated either by an approximate method when quick results are required or by a more rigorous method when greater accuracy or the effect of configuration changes are desired. The approximate method is based upon the concept that the power available is supplied by the rate of loss of potential energy and the assumption that the power required is the same as for level flight at the same speed. The equation for rate of descent using the approximate procedure is

$$V_D = \frac{550hp_{req}}{W_g} \quad \text{, fps} \qquad (3\text{-}213)$$

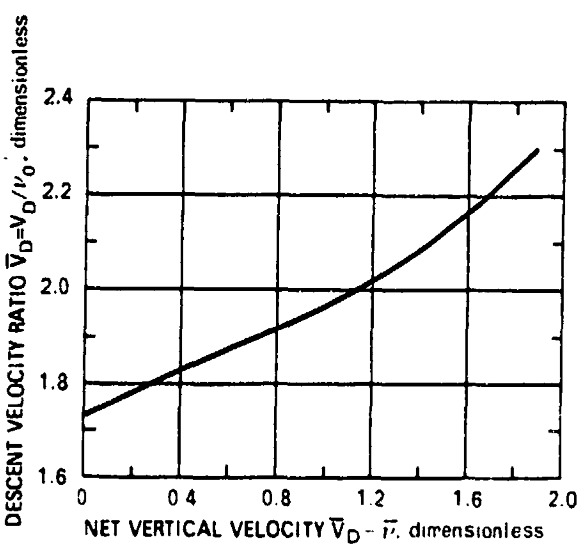


Fig. 3-170. Nondimensional Velocities in Vertical Autorotation

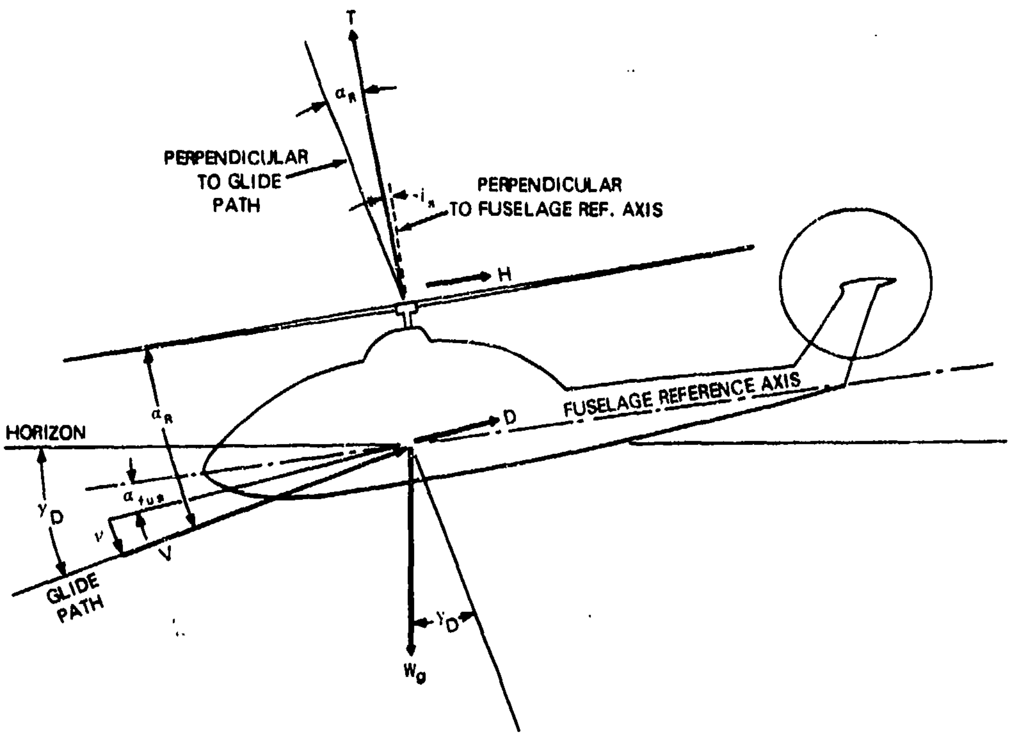


Fig. 3-171. Forces on Helicopter in Autorotation

This equation cannot be considered more accurate than $\pm 15\%$ because the power required in autorotation is not the same as for forward flight due to the difference in local blade element angle of attack distributions and fuselage drag contributions for the two flight conditions.

A more accurate method is based upon the use of rotor charts (par. 3-2.1.2.2.5) and the equation of forces along the glide path acting upon the helicopter as shown in Fig. 3-171. Substantiation of this method is determined through actual flight test by the "sawtooth rate of descent" procedures discussed in AMCP 706-203. The rate of descent is

$$V_D = V \sin \gamma_D \quad \text{, fps} \qquad (3\text{-}214)$$

Using small-angle assumptions, $\sin \gamma_D$ is estimated by

$$\sin \gamma_D = \frac{T\alpha_R + D + H}{W_g} \quad \text{, dimensionless} \qquad (3\text{-}215)$$

The calculation procedure for a given gross weight, speed along the flight path, tip speed, and density involves the following steps:

1. Estimate the torque that the rotor must supply in autorotation to drive the tail rotor, transmission, and accessories. Compute the corresponding value of negative C_Q/σ.

2. For the value of C_T/σ corresponding to the particular gross weight, tip speed, and density and the value of C_Q/σ determined by Step 1, use a suitable rotor chart for the proper tip speed ratio to find the collective pitch θ_0

3. For values of C_T/σ and θ_0 determine the angle of attack of the rotor α_R and the H-force from the rotor charts.

4. Find the fuselage angle of attack α_{fus} as

$$\alpha_{fus} = \alpha_R - i_s - \frac{\nu}{V} \qquad (3\text{-}216)$$

5. Find the fuselage drag as

$$D = q\left(\frac{D}{q}\right) \quad \text{, lb} \qquad (3\text{-}217)$$

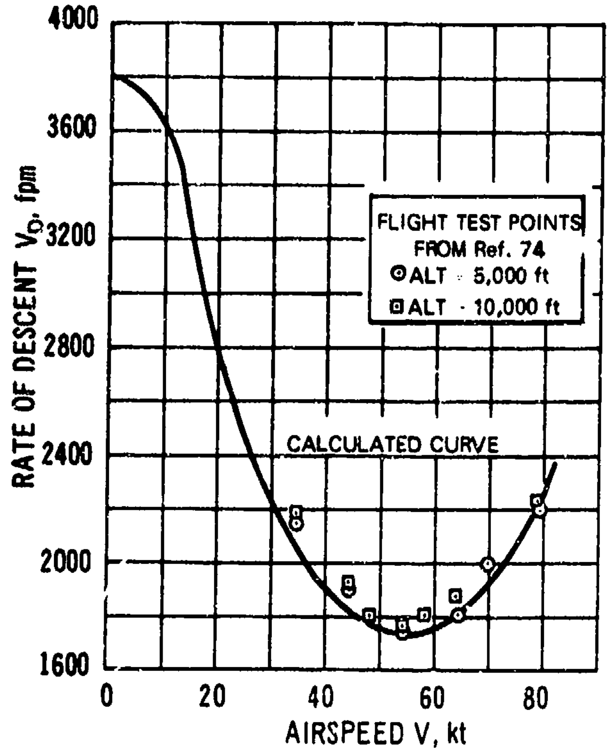


Fig. 3-172. Rate of Descent in Autorotation

where D/q is a function of α_{fus} as measured in a wind tunnel or obtained by estimation.

6. Calculate the rate of descent and the glide angle from Eqs. 3-214 and 3-215.

The results of a sample calculation using this method are shown on Fig. 3-172 for the UH-1, along with points determined in flight test (Ref. 74).

3-5.1.4 Glide Distance

The radius of the zone within which the pilot must select his landing spot following a power failure is equal to the horizontal projection of his maximum glide distance from the altitude at which he established a steady-state autorotation. This altitude is either the altitude at which power failure occurred or the altitude to which he can zoom (speed reduction climb) following power failure. The zoom maneuver is possible when the power failure occurs at a forward speed V_0 higher than the autorotational speed V_1. The equation for the altitude gained during the zoom is derived from the equation for the change of energy.

$$W_g(h_1 - h_0) = \frac{W_g}{2g}(V_0^2 - V_1^2) + \frac{I}{2}(\Omega_0^2 - \Omega_1^2) - 550\int_0^t (hp)dt, \text{ ft-lb} \quad (3\text{-}218)$$

The horsepower hp in Eq. 3-218 is assumed to be the average between the two conditions, and the time t is the time required to decelerate from V_0 to V_1 using the component of gravity along the climb path. The altitude gained is

$$h_1 - h_0 = \frac{\frac{W_g}{2g}(V_0^2 - V_1^2) + \frac{I}{2}(\Omega_0^2 - \Omega_1^2)}{W_g} - \frac{550\left[\frac{(hp_0 + hp_1)}{2}\right]\left[\frac{V_0 - V_1}{g \sin \gamma_{cl}}\right]}{W_g}, \text{ ft} \quad (3\text{-}219)$$

where

$$\gamma_{cl} = \text{Cos}^{-1}\frac{(C_T/\sigma)_{max}}{(C_T/\sigma)_{nor}}, \text{deg} \quad (3\text{-}220)$$

and

$$\Omega_1^2 = \Omega_0^2\left[\frac{(C_T/\sigma)_{nor}}{(C_T/\sigma)_{max}}\right], (\text{rad/sec})^2 \quad (3\text{-}221)$$

In Eqs. 3-220 and 3-221, the value of $(C_T/\sigma)_{max}$ may be estimated using the lower boundary of Fig. 3-173, which has been compiled from a study of wind tunnel and flight test results and represents the boundary above which the rotor profile power rises rapidly, indicating significant areas of blade stall. The equation for the maximum glide distance then is

$$d_{max} = h_1 \frac{V_1}{V_D}, \text{ ft} \quad (3\text{-}222)$$

3-5.1.5 Flare

The objective of the flare maneuver is to make a transition from steady-state autorotation, with moderate forward and vertical velocities, to a touchdown, with small or no forward and vertical velocities. An idealized flare maneuver is illustrated in Fig. 3-174 and starts with a cyclic flare at constant collective pitch, in which the increased rotor thrust and its aft tilt component are used to decrease both the vertical and horizontal velocity components. At the end of this cyclic flare, the aircraft should be near the ground with its vertical velocity $V_D = 0$ or within the design sink speed of the landing gear $V_{s_{des}}$ and with the horizontal velocity corresponding to autorotation at the angle of attack to which the helicopter has been pitched. The flare angle of attack is the highest angle from which the helicopter can be rotated nose-down to a level attitude in the time during which the rotor energy can be used to develop hovering thrust. The final rotation and collective flare are used to bring both velocity components as close to zero as possible.

The maximum allowable touchdown speed is dependent upon vehicle configuration and landing gear capability. A maximum touchdown speed of 15 kt is specified in MIL-H-8501. A means of estimating this speed during design is given by the following procedure:

1. Calculate the maximum allowable angle of attack α_{max} at the end of the cyclic flare as a function of the maximum nose-down pitch rate $\dot{\theta}_{max}$ and the time t_{hov} available for the maneuver

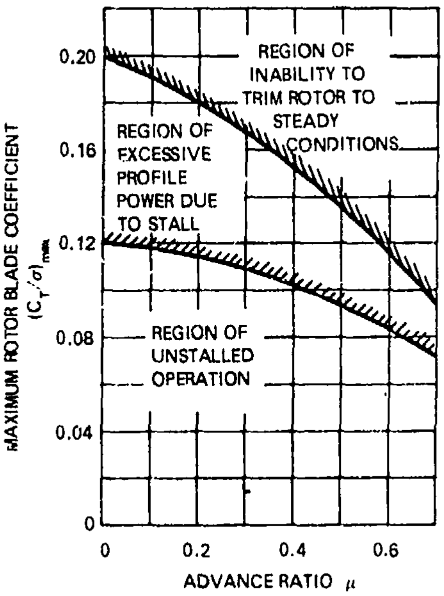


Fig. 3-173. Maximum Rotor Capability

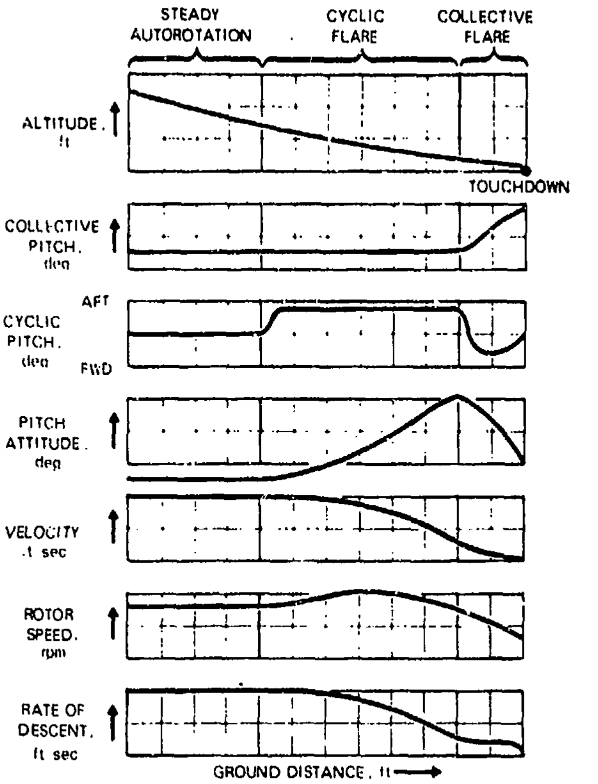


Fig. 3-174. Idealized Flare Maneuver

$$\alpha_{max} = \dot{\theta}_{max} t_{hov} \quad , \text{deg} \tag{3-223}$$

where

$$t_{hov} = \frac{\frac{I}{2}(\Omega^2_{nor} - \Omega^2_{min})}{550hp_H} \quad , \text{sec} \tag{3-224}$$

and

$$\Omega^2_{min} = \frac{W_g}{(C_T/\sigma)_{max}\rho\sigma\pi R^4} \quad , (\text{rad/sec})^2 \tag{3-225}$$

For this maneuver, the value of $(C_T/\sigma)_{max}$ may be found from the upper boundary on Fig. 3-173, which represents the limits of conditions in which the rotor can be controlled even though much of it is stalled. In some cases, the maximum allowable angle will not be as defined by Eq. 3-223, but will be limited to some smaller value by the pilot's loss of visual contact with the ground at high angles.

2. From Fig. 3-175, find the value of the minimum advance ratio at which autorotation can be sustained at the maximum flare angle while still developing a vertical component of rotor thrust equal to the gross weight.

3. Find the minimum touchdown velocity in kt as

$$V_{min} = \frac{\mu_{min}(\Omega R)_{nor} - (g\alpha_{max} t_{hov})/2}{1.69} \text{, kt} \quad (3\text{-}226)$$

where $(g\alpha_{max}\, t_{hov})/2$ is the decrease of forward speed during the rotation.

Although the rate of descent in steady autorotation does not enter into the calculation of the touchdown velocity in a direct manner, it does affect the pilot's chances of achieving the idealized flare that has been assumed. The cyclic flare is a precision maneuver in which the pilot, using only his cyclic and collective controls, must solve simultaneously the equations of motion for vertical forces, horizontal forces, and pitching moments so as to end the flare within narrow limits of height above the ground, rate of descent, forward speed, and angle of attack. The higher the rate of descent at the start of the cyclic flare, the less time he has to correct prior control inputs in order to perform the maneuver satisfactorily and to make his actual touchdown within the limitations of the landing gear design sink speed. The flare maneuver can result in large blade flapping angles, and the designer must provide sufficient clearance between the aft fuselage and the rotor to prevent blade strikes. No simple analytical procedures are available for computing how much flapping a pilot will induce while making an autorotational flare, so the designer must be guided by his judgment and experience.

3-5.1.6 Height-Velocity Curve

3-5.1.6.1 Single-engine Helicopters

The ability of the helicopter to demonstrate an actual flare and landing at a touchdown speed within the design capability of the landing gear is restricted by the initial combination of altitude and forward velocity whose boundaries define the height-velocity curve or "deadman's curve". A typical height-velocity curve is shown in Fig. 3-163.

A first approximation of the height-velocity curve can be generated during preliminary design by a combination of empirical and analytical considerations as outlined in Ref. 75. This method makes use of the generalized nondimensional height-velocity curve shown on Fig. 3-176, which has been generated from experimental data obtained using several single-engine helicopters as flown by skilled test pilots. For this reason the method is considered to produce a minimum unsafe area, but not necessarily an operational envelope for everyday flying. The lower boundary was obtained with no delay in collective pitch reduction and the upper boundary with a one-second delay.

In order to establish the diagram for a given helicopter, three heights and one airspeed must be found: low hover height h_{lo}, high hover height h_{hi}, critical hover height h_{crit}, and critical airspeed V_{crit}.

The value of h_{lo} can be calculated by assuming that the pilot is alert and immediately can establish a rate of descent equal to the landing gear design sink speed $V_{s_{des}}$ and that he can maintain that rate of descent

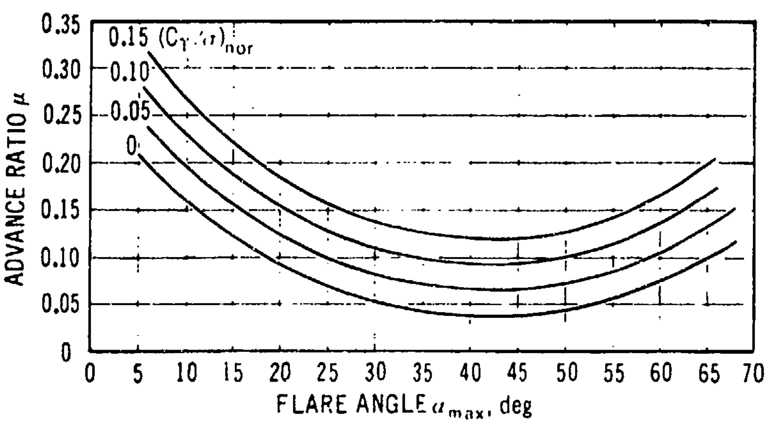


Fig. 3-175. Conditions for Autorotation at End of Cyclic Flare

during the time that the available kinetic energy of the rotor can provide power equivalent to that required for hover IGE without exceeding a value of blade section lift coefficient c_l of 1.2. The latter is a figure that is achievable by at least two out of three helicopters, according to Ref. 75. The resultant equation is

$$h_{lo} = \frac{V_{s_{des}}\left[I\Omega_0^2\left(1 - 2.24\sqrt{C_T/\sigma}\right)\right]}{1100hp_{HOGE}\Lambda}, \text{ ft} \quad (3\text{-}227)$$

where Λ is the ground effect parameter and is shown as a function of the rotor height-to-diameter ratio on Fig. 3-15.

Analysis of the experimental height-velocity data produced the relationship between the high hover height h_{hi} and V_{crit}^2 shown on Fig. 3-177 and also identified V_{crit} as a function of the speed for minimum power V_E and C_L/σ as shown on Fig. 3-178.

The final required altitude is h_{crit}, which, according to Ref. 75, can be considered to be 95 ft for all single-engine helicopters until further flight test programs are carried out in this field.

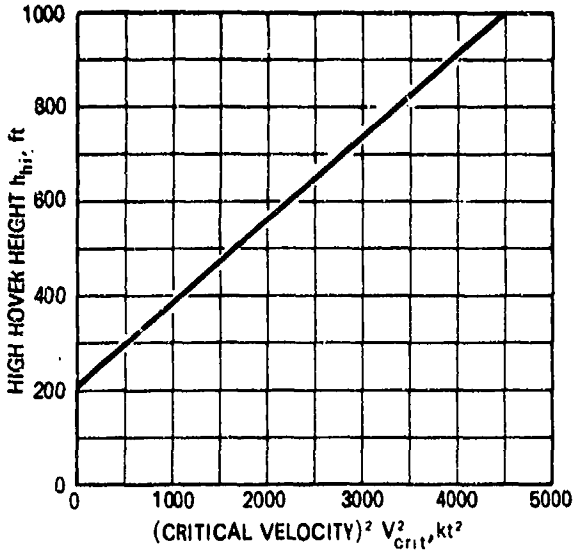


Fig. 3-177. High Hover Height

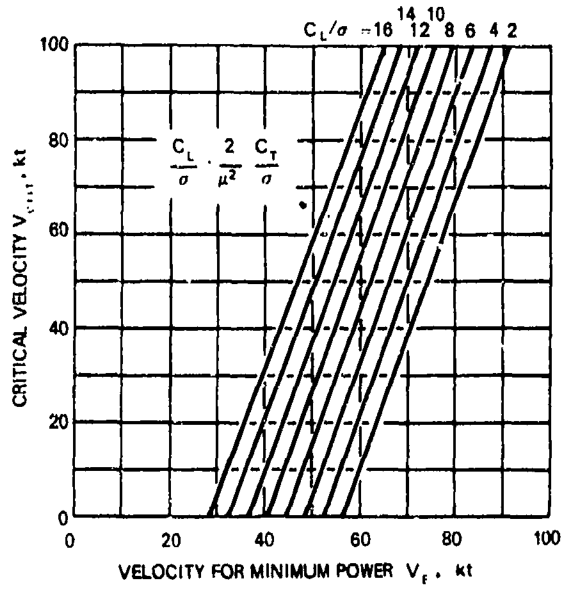


Fig. 3-178. Critical Velocity

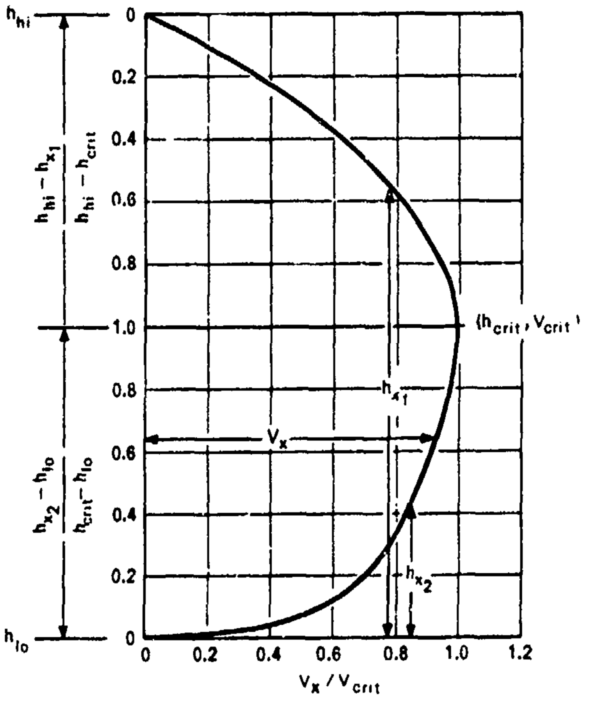


Fig. 3-176. Generalized Nondimensional Height-velocity Curve for Single-engine Helicopters

The method has been used to calculate the height-velocity curve for the UH-1 and the results are plotted on Fig. 3-179 along with flight test points representing both safe and critical landings from Ref. 74. It may be seen that, for a maneuver that is highly dependent upon pilot technique, the calculated curve is reasonably close to the measured points. As previously mentioned, this method can be used only to define a minimum unsafe

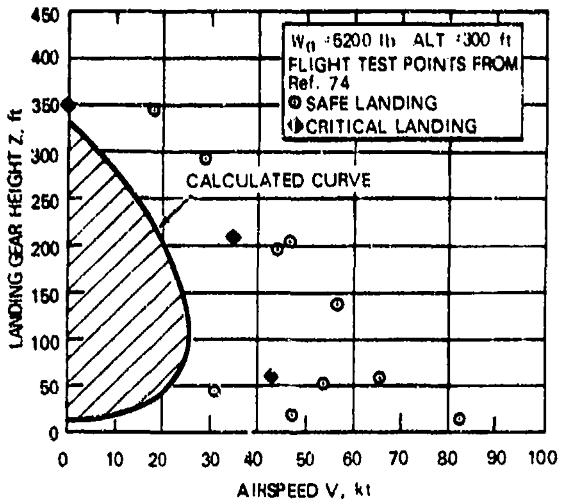


Fig. 3-179. Height-velocity Curve for UH-1

area. The "avoid" area is unconservatively small and can be considered only a first approximation of an operational limitation.

The second, or high-speed, portion of the height-velocity curve shown on Fig. 3-163 is simply a warning that a power failure at high speed and close to the ground is a dangerous situation. No analytical method has been developed for predicting this portion of the curve and some presentations omit it entirely. Two considerations regarding the high-speed portion of the height-velocity curve are worth noting. First, the pilot can be assumed to be alert and able to react quickly to a power failure. Second, for most helicopters, when the rotor slows down at constant collective pitch, the increase in tip speed ratio causes the rotor to flap back so that a pitchup is started automatically. This pitchup tends to keep rotor speed from decaying and at high speeds results in the start of a zoom maneuver.

3-5.1.6.2 Multiengine Helicopters

A power failure that results only in a partial loss of power is obviously less of a problem than a complete power failure. The height-velocity curve for this case can be developed in the same way as for the single-engine helicopter. The equation for the low hover height is

$$h_{lo} = \frac{V_{s_{des}}\left[I\Omega_0^2\left(1 - 2.24\sqrt{C_T/\sigma}\right)\right]}{1100\left(hp_{HOGE} - hp_{av}\right)\Lambda}\text{, ft} \quad (3\text{-}228)$$

Ref. 76 recommends using for the critical speed V_{crit} a value that is one-half the speed at which the remaining power can maintain a rate of descent equal to the landing gear design sink speed $V_{s_{des}}$, or one-half the forward speed at which

$$\frac{550(hp_{req} - hp_{av})}{W_g} = V_{s_{des}}\text{, fps} \quad (3\text{-}229)$$

Based upon the experimental data given for the CH-47B in Ref. 77, the corresponding critical height h_{crit} can be assumed to be 50 ft instead of the 95 ft used for single-engine helicopters. The high hover height h_{hi} can be found as a function of V_{crit} from Fig. 3-177. Fig. 3-180 shows the height-velocity diagram for the CH-47B determined by the procedure outlined, along with flight test points from Ref. 77.

3-5.2 MANEUVERING PERFORMANCE

This paragraph describes the methods used to compute helicopter performance in maneuvering flight. The design parameters that significantly affect maneuvering capability are identified, and their relative importance is discussed in general terms. Equations by which the designer may compute performance in ac-

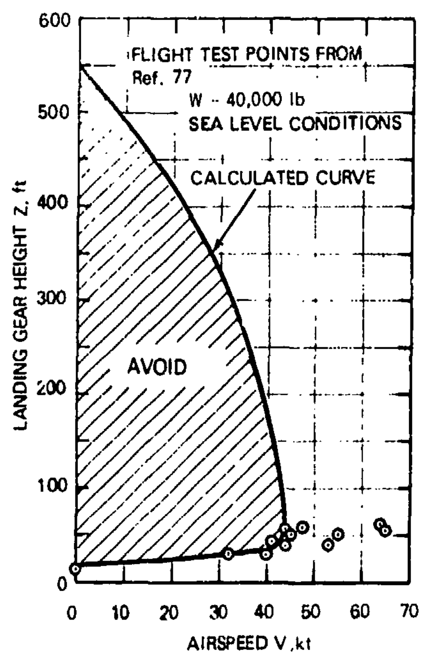


Fig. 3-180. Height-velocity Curve for CH-47B

celerated flight are presented and the limitations of current theories are mentioned.

The theory of maneuvering performance of helicopters is in its infancy. There is no definitive literature on the subject, so an unpublished theory based upon energy and momentum theory is presented. It has been found useful in comparing the performance of a wide variety of helicopter designs.

The limitations of the performance of a rotor in maneuvering flight are dependent heavily on its design. A viable theory that would account for all the variations of rotor systems would have to include dynamic, aeroelastic, and structural effects, as well as aerodynamic effects. Although the technologies associated with each of these effects include theoretical approaches to the problem of predicting maneuvering flight limitations, the currently available methods must be applied separately because of the complexity of combining them. The most common limitations are aerodynamic (stall), dynamic (vibration), or aeroelastic (flutter). Only the aerodynamic limitations of rotor performance are defined quantitatively here; the others are discussed in qualitative terms.

3-5.2.1 Power Required in Accelerated Flight

The power required of the helicopter in equilibrium level flight must be determined before its performance in accelerated flight can be computed. Methods for this determination are discussed in par. 3-2.

If power available exceeds that required for unaccelerated level flight, this excess power may be used:

1. To change energy states
   a. By climbing (or descending if power is deficient)
   b. By accelerating (or decelerating if power is deficient)
2. To induce normal acceleration
   a. At constant energy (called sustained load factor capability)
   b. With loss of energy (called transient load factor[1] capability).

3-5.2.2 Changing Energy States

The capability of the aircraft to change energy states is limited by the power available because power is simply the rate of change of energy. Excess power may be used to increase altitude (potential energy). The rate of climb may be approximated by Eq. 3-191. The efficiency constant K_p is typically about 0.85 for climbing flight and 0.80 for descending flight. Autorotational performance may be computed from Eq. 3-191 by setting the power available ΔSHP to zero.

Excess power also may be used to accelerate the vehicle at constant altitude. The time required to accelerate from V_1 to V_2 is given by

$$t = \int_{V_1}^{V_2} \frac{(W_g/g)V}{550(hp_{av} - hp_{req})}\,dV \quad \text{, sec} \qquad (3\text{-}230)$$

In deriving Eq. 3-230, it was assumed that all the excess power is used to generate a force that accelerates the vehicle horizontally. Although this force approaches infinity as velocity approaches zero, the equation is so arranged that the integrand goes to zero in hovering flight.

Decelerative performance may be computed similarly. The power required by the rotor during a cyclic flare decreases because the flow state approaches that of autorotation. The amount of this reduction in the rotor power must not be negative, however, because it would cause rotor overspeed. Therefore, all the power required at a given speed is assumed to generate a force F_d that decelerates the vehicle by means of the equation

$$F_d = \frac{550hp_{req}}{V} \quad \text{, lb} \qquad (3\text{-}231)$$

The time to decelerate from V_2 to V_1 is given by

$$t = \int_{V_2}^{V_1} \frac{W_g/g}{F_d}\,dV \quad \text{, sec} \qquad (3\text{-}232)$$

which is the same as Eq. 3-230 if hp_{av} is set to zero.

3-5.2.3 Normal Acceleration Capability

Excess power may be used to increase lift above the 1-g-flight value in order to turn the aircraft in either the vertical (symmetrical pullup) or the horizontal (conventional turn) plane. The normal acceleration produced by a given amount of excess power is nearly independent of flight path orientation, so the accelerated flight performance equations can be derived from considerations of level turning flight.

[1] Under this definition, a constant-speed descending turn performed at constant load factor uses rotor transient load factor capability.

Increasing the angle of attack of an airfoil increases its lift, but also increases its drag; therefore, in accelerating flight more power is required. This increased power is absorbed by the rotor when both collective and cyclic pitch controls are used to attain load factors that require full power. A maneuver (turn) of this kind demonstrates the aircraft sustained load factor capability. It is apparent that sustained load factor performance is dependent highly upon the amount of excess power available.

When still higher normal acceleration is demanded, the aircraft can lose either altitude (potential energy) or airspeed (kinetic energy) in exchange for the required total power. Turns of this kind make use of the rotor transient load factor capability, which may be limited by blade stall, rotor instabilities, or vibration. The power required for transient load factor maneuvers is quite high, however, and energy is lost rapidly. Furthermore, the capability of the rotor to absorb power diminishes because it approaches the autorotative flow state at high angles of attack (Refs. 78 and 79). Therefore, helicopters always will be limited in their capability to decelerate in high load factor maneuvers when compared with fixed-wing aircraft which are not subject to this reduction in capability to absorb power. Compound helicopters with propellers geared to the rotor are not restricted in the same way as helicopters, but in this regard are like fixed-wing aircraft.

Sustained load factor performance is a function not only of the excess power available hp_{ex}, but also of the induced velocity and the change in rotor profile power with angle of attack. If all available power is used to produce normal acceleration, the following equation expresses the normal load factor n:

$$n = \sqrt{1 + \frac{550hp_{ex}}{W_g v_{eq}}} \text{, dimensionless} \quad (3\text{-}233)$$

where

$$v_{eq} = v \left[1 + \frac{49}{4} \frac{\delta_2}{\sigma a^2} (1 + 4.6\mu^2)\right] \text{, fps} \quad (3\text{-}234)$$

where v is obtained from Eq. 3-35. The value of v_{eq} may be approximated as $2v$, if the data necessary to use Eq. 3-234 are not available.

The transient load factor capability of a rotor is calculated by dividing the maximum attainable lift coefficient by the 1-g lift coefficient for the same flight condition. Thrust and airspeed limitations upon helicopter performance are reflected in the cockpit. These placarded maximum rotor loads and speeds can be converted into rotor lift coefficients and advance ratios. Fig. 3-181 shows a typical example of such a relationship (Ref. 80). The rotor blade lift coefficient t_c shown is dimensionless and is equal to $2C_T/\sigma$. The values of t_c that result from the placarded airspeed limits are indicated by the flagged, open triangles; the solid triangles are data points for transient load factor maneuvering flight. The dashed line was computed from the empirical relationship shown and represents the values of t_c at which power divergence occurs for a teetering rotor. The values applicable to other types of rotors or using different airfoil sections may vary significantly from these.

Although the t_c versus μ characteristics of rotor systems vary widely, an aerodynamically limited value of t_c at different advance ratios can be computed if the variation of maximum lift coefficient with Mach number for the blade section used is known. The analytical model for this computation is illustrated in Fig. 3-182. Throughout the retreating blade region, the blade sections are assumed to be at the maximum lift coefficient out to the blade station at which the resulting lift moments of the advancing and retreating blades are equal and opposite, and the lift is assumed to be zero beyond that station. Fig. 3-183 shows the results of such an analysis for two relationships of maximum lift coefficient to Mach number. The solid triangles indicate data from maneuvering flight tests. The circles show data from full-scale wind tunnel tests of the same rotor system.

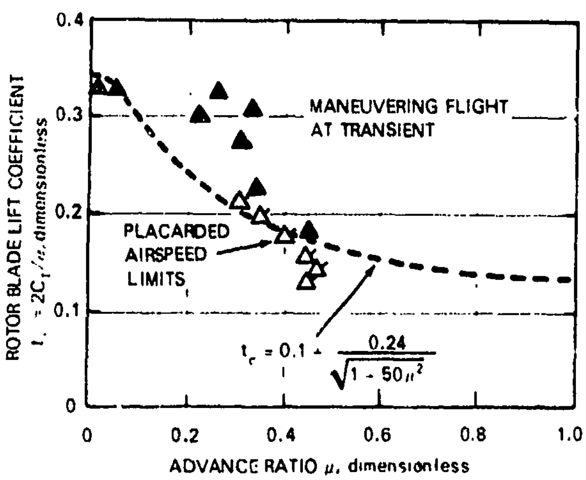


Fig. 3-181. Typical Rotor Thrust Capability (Ref. 80)

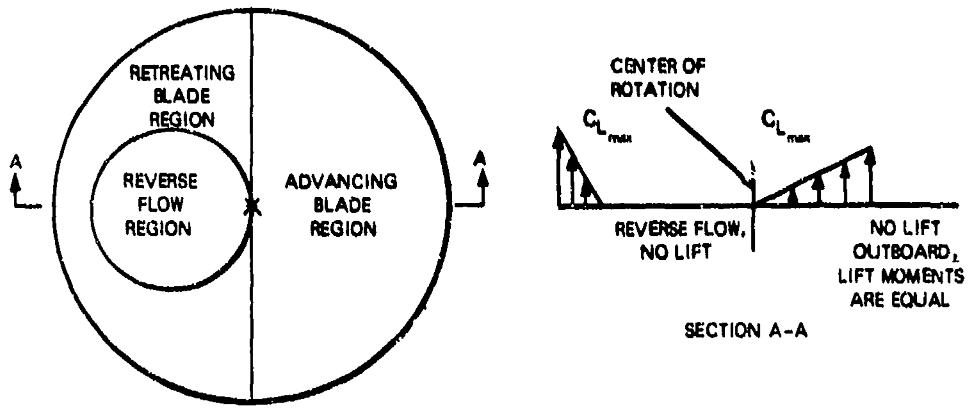


Fig. 3-182. Analytical Model for Maximum Rotor Thrust

This relatively simple approach to the aerodynamic limitation of rotor thrust appears to be reasonably accurate for at least one rotor system and it predicts the correct trend with increasing airspeed. However, the adequacy of the method to account for the effects other than aerodynamic which were discussed previously has not been demonstrated for all types of rotors. Therefore, empirical methods based upon test data for rotors comparable to that being proposed may be equally effective for preliminary design purposes.

It is apparent from Fig. 3-183 that the transient load factor capability of rotor systems decreases rapidly with increasing speed. Significant improvements in transient load factor capability can be made by careful attention to the blade section properties at different radial locations along the rotor blade.

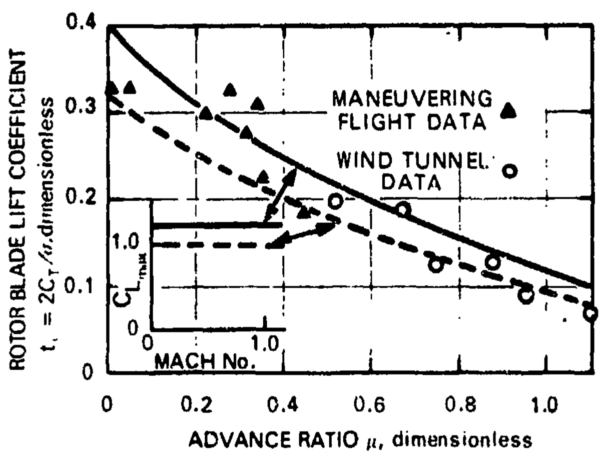


Fig. 3-183. Analytical Aerodynamic Maximum Thrust

3-5.2.4 Effect of Wings

The effect of wings upon load factor capability is to increase the change of lift that corresponds to a change in the aircraft angle of attack. The equivalent induced velocity w_{eq} for a wing may be determined by including the δ_2 component of drag in the same way as for the rotor.

$$w_{eq} = w\left[1 + \frac{\pi(AR)e}{a_w^2}\delta_2\right] \text{, fps} \quad (3\text{-}235)$$

where

$$w = \frac{L_w}{2\rho e V A_{ref}} \quad (3\text{-}236)$$

where

a_w = wing lift curve slope, rad^{-1}
A_{ref} = $\pi b^2/4$, ft^2
AR = wing aspect ratio, b^2/S, dimensionless
b = wing span, ft
e = wing span efficiency factor, dimensionless
L_w = wing lift, lb
S = wing area, ft^2

Eq. 3-235 is similar to Eq. 3-234 because the same momentum theory approach was used for both the fixed-wing and the rotor equations. The sustained normal load factor n for a fixed-wing aircraft can be expressed in a form similar to Eq. 3-233 for helicopters:

$$n = \sqrt{1 + \frac{550hp_{ex}}{W_g\left[w_{eq} - \frac{k}{(\rho/\rho_0)^3 V^5}\right]}} \quad (3\text{-}237)$$

where

k = empirical constant, $(fps)^6$

The additional term $k/[(\rho/\rho_0)^3 V^5]$ accounts for the rapid increase in wing drag at speeds near stall. It may be neglected for calculations of turning performance at speeds higher than about 130% of wing stall speed.

The energy expressions for pure helicopters and for fixed-wing aircraft may be combined for the computation of the sustained turn performance of winged helicopters. The equations presented are based upon the assumptions that the wing has fixed incidence, that it is positioned below (in the downwash of) the primary lifting rotor, and that its lift does not affect rotor thrust significantly. The equations may be modified to account for other configurations.

The fundamental technique for computing constant-energy turning performance is to use all of the excess power to produce lift; i.e., to overcome the additional induced drag that accompanies additional lift. Equations must be developed for rotor thrust and drag and for wing lift and drag, as well as for the changes in these quantities in accelerated flight. The following relationships assume that downwash angles are small, that momentum theory is valid for determining rotor induced velocity, and that the value of induced velocity at the rotor disk changes with wing angle of attack by v/V radians:

$$L_w = a_w A_w q\left(\alpha_w + \alpha_{fus} - \frac{v}{V}\right) \quad , \text{lb} \quad (3\text{-}238)$$

where

α_w = wing angle of incidence, rad
A_w = wing area, ft²

$$T = 2\rho A V v \quad , \text{lb} \quad (3\text{-}239)$$

$$nW_g = T + L_w \quad , \text{lb} \quad (3\text{-}240)$$

The change in wing lift with angle of attack is given by

$$\frac{dL_w}{d\alpha_{fus}} = \frac{\partial L_w}{\partial \alpha_{fus}} + \frac{\partial L_w}{\partial \alpha_i}\frac{\partial \alpha_i}{\partial T}\frac{\partial T}{\partial \alpha_{fus}} \quad , \text{lb/rad} \quad (3\text{-}241)$$

where

α_i = induced incremental angle of attack, rad

Differentiating Eqs. 3-238 and 3-239

$$\frac{dL_w}{d\alpha_{fus}} = a_w A_w q\left(1 - \frac{\partial T/\partial \alpha_{fus}}{4Aq}\right) \quad , \text{lb/rad} \quad (3\text{-}242)$$

where

$$\frac{\partial T}{\partial \alpha_{fus}} = \frac{\sigma a \mu}{2}\left[\rho A (\Omega R)^2\right] \quad , \text{lb/rad} \quad (3\text{-}243)$$

This last expression (Eq. 3-243) for rotor thrust change with angle of attack at constant collective pitch is derived from linear rotor theory. It represents the product of the dynamic pressure at the tip of the blade, the blade area, the slope of the blade section lift curve, and the advance ratio of the rotor. If collective pitch is reduced in high load factor maneuvers, a greater change in angle of attack will be required for the same load factor than if the collective pitch is fixed. This is one way the division of lift between the wing and the rotor in maneuvering flight may be controlled.

The change of normal acceleration with angle of attack may be determined from Eqs. 3-240, 3-242, and 3-243:

$$\frac{dn}{d\alpha_{fus}} = \frac{1}{W_g}\left[\frac{\partial T}{\partial \alpha_{fus}} + \left(a_w A_w q\right) \times \left(1 - \frac{\partial T/\partial \alpha_{fus}}{4Aq}\right)\right] \quad , \text{d'less} \quad (3\text{-}244)$$

The first term of Eq. 3-244 represents the increase in thrust and the last term represents the decrement of wing lift caused by increased rotor-induced velocity. From Eq. 3-243 it can be shown that the wing contribution of Eq. 3-244 reduces to $a_w A_w q\{1 - [\sigma a/(4\mu)]\}$. The term in braces may be thought of as the load factor effectiveness of the wing, which is zero if $\mu = \sigma a/4$ and becomes increasingly positive at higher speeds. Because a is typically about 2π, if $\sigma = 0.065$, the zero-lift value of μ is 0.102, which corresponds to a speed of about 45 kt. At this low speed the small-angle assumption is imprecise enough that an exact solution of Eq. 3-238 demands the use of the more precise term, $\text{Tan}^{-1}(v/V)$. However, the wing load factor effectiveness term shows how the wing helps in maneuvering flight. The load factor effectiveness of the wing would be 0.75 at 180 kt and only 0.5 at 135 kt. Thus, it is apparent that a wing is effective in maneuvering flight only at high forward speeds. However, rotors designed for high forward speeds have higher solidities than the 0.065 of the example given previously; a value of about 0.11 is typical. This greater solidity of the rotor reduces the wing load factor effectiveness; at 180 kt it would be reduced from 0.75 to 0.575.

From Eqs. 3-235 through 3-244, an equation for an effective induced velocity of the wing and rotor combined v_{Rw} may be derived and sustained load factor performance computed. If f_w represents the fraction of the vertical force supported by the wings, then the induced power may be computed from

$$\begin{aligned} P_i &= L_w w_{eq} + T v_{eq} \\ &= n W_g \left[f_w w_{eq} + (1 - f_w) v_{eq} \right] \\ &= n W_g v_{Rw}, \text{ ft-lb/sec} \end{aligned} \quad (3\text{-}245)$$

The constant-energy turn performance of a winged helicopter (see Eqs. 3-233 and 3-237), is given by

$$n = \sqrt{1 + \frac{550 hp_{ex}}{W_g v_{Rw}}}, \text{ dimensionless} \quad (3\text{-}246)$$

where v_{Rw} is computed for level flight in equilibrium, so that $n = 1$.

Transient load factor maneuvers generally are accompanied by a greater change in the aircraft angle of attack than are constant-energy maneuvers. In transient load factor maneuvers, the wing produces more lift than the given equations indicate. Although further work in this area is necessary, the methodology is straightforward and the equations presented in this paragraph should be sufficient to extend the theory.

3-5.3 ENGINE(S) OFF/INOPERATIVE CONDITIONS

General performance considerations and trim requirements are discussed to highlight the helicopter one-engine off/out operational characteristics. Methods for determination of helicopter performance capabilities with one-engine off or inoperative also are described.

3-5.3.1 Autorotation

A unique feature of the helicopter is its ability to effect an autorotative landing in the event of total power failure. The installation of more than one engine improves the reliability of the aircraft in that power failure of one engine does not make such a landing necessary. Redundancy and improved performance are realized through multiple engine installations.

Methods for analysis of autorotational performance are discussed in par. 3-5.1. Both vertical and forward flight autorotation are analyzed, with the special case of partial power loss mentioned briefly.

A rapid assessment of the safety margin, system adequacy, or gross weight capability of a helicopter to effect an autorotative landing can be made by establishing an autorotative index AI criterion. Such an index can be simply the ratio of the rotor rotational kinetic energy to the power required to hover, at a given gross weight;

$$AI = \frac{I\Omega^2}{1100 hp_{HOGE}}, \text{ sec} \quad (3\text{-}247)$$

where

I = rotor inertia, slug-ft^2

The value of such an index is dependent upon the availability of flight test data, which is the only reliable basis to use as a reference for comparison. Once this reference is established, the index can be used on other systems under study and their merits judged relative to known characteristics of similar aircraft. Under this criterion an AI of no less than 1.7 appears to be acceptable operationally for single-rotor aircraft.

3-5.3.2 Performance With One Engine Inoperative (OEI)

The requirement to maintain a 100-fpm rate of climb with one engine inoperative (OEI) normally is not critical for helicopters with three or more engines. However, the analysis that follows is valid for the loss of one or more engines.

The speed for minimum power and therefore for maximum power loading is used to derive maximum performance, according to these criteria. Fig. 3-184 shows a typical level-flight power required curve and the intersection of OEI power available. Because the power required (and available) to hover is about twice that for the best rate of climb speed, the loss of one engine in a twin-engine aircraft still leaves sufficient power to continue flight. The difference between the power available and the power required is the climb

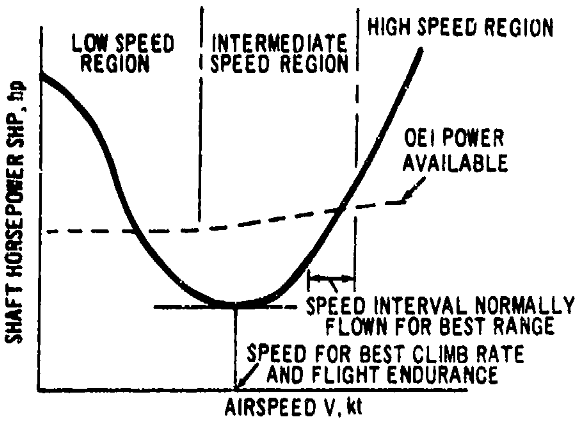


Fig. 3-184. Power Required Characteristic in Steady-state Forward Flight

power available, and the rate of climb R/C for any condition can be defined by Eq. 3-191. Climb performance with one engine operative is calculated by the same methods as when all engines are operative, making appropriate adjustments in all parameters for the reduced value of ΔSHP.

3-5.3.3 OEI Range and Endurance

A multiengine helicopter can cruise efficiently with one engine inoperative provided sufficient power is available for the particular flight conditions. A turbine engine manifests a significant improvement in fuel flow per horsepower (SFC) at high power levels, thus making it attractive to shut down one such engine as a means of increasing either range or endurance. Engine-power on/idle/off management is desirable in a multiengine aircraft for optimizing range or endurance. While the multiengine reliability then is dependent upon the ability to restart the inoperative engine, other flight characteristics in the intermediate speed region are not affected adversely as in a fixed-wing multiengine aircraft, where thrust asymmetry will require large control forces for trim (par. 3-5.3.5). As illustrated in Fig. 3-184, the speeds for both best endurance and best range occur in the aircraft intermediate speed region.

When the OEI capability becomes critical for continuing flight, rotor and engine speed must be optimized to provide a sufficient margin between total power available and power required and to minimize fuel flow rate. Fig. 3-185 shows the variation of SFC at a fixed engine rpm. The fuel saving possible by shutting down one engine is shown clearly by the slope of this curve. While the exact shape of the SFC curve will vary from engine to engine and the operating powers will change with gross weight, speed, and atmospheric conditions, the trend remains as illustrated. Using Fig. 3-186 to optimize the operating rpm for a given set of conditions, the range of rpm that will provide sufficient power to sustain flight can be found. Fig. 3-187 illustrates the effect upon fuel consumption characteristics of varying engine speed. For typical flight conditions, the maximum range for a twin-engine helicopter can be increased by about 15% by shutting down one engine. Similarly, endurance is increased by about 25%. The increase in range or endurance resulting from shutting down an engine normally will be smaller in the case of a helicopter with three or more engines unless the available multiengine power greatly exceeds the power required, in which case large gains in endurance and range may be possible. Fig. 3-188 shows typical trends of best range speed, maximum SFC, and corresponding rotor speed management with gross weight. This type of information is of great value to the pilot, in case of loss of one engine, when maximum fuel saving is paramount.

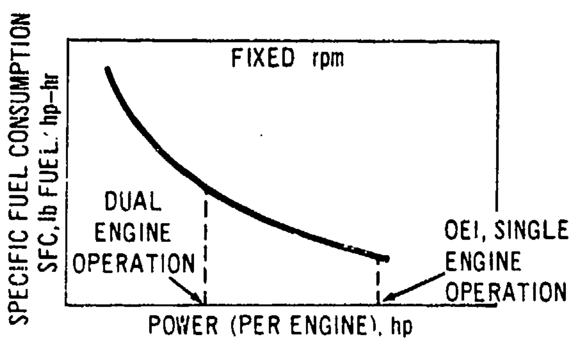


Fig. 3-185. Specific Fuel Consumption vs Engine Power

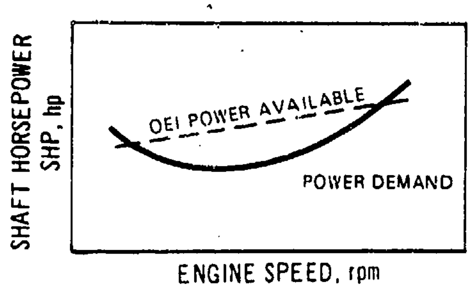


Fig. 3-186. Effect of Engine/Rotor Rpm on Power

3-5.3.4 Takeoff and Landing

The takeoff and landing regimes of flight are the portions of the flight profile where most accidents occur. As helicopter development has led to improved payload capabilities, various regulations, both commercial and military, have evolved to insure a reasonable level of safety during these maneuvers.

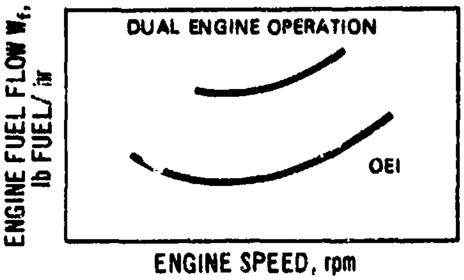


Fig. 3-187. Variation of Fuel Flow With Engine Rpm

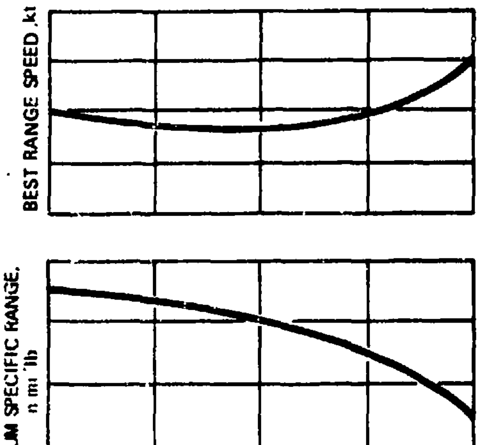


Fig. 3-188. Maximum Range Management Variables

Takeoff procedures can be classified into three basic groups:

1. Vertical
2. Oblique
3. Horizontal.

Basic flight paths for these procedures are shown in Fig. 3-189.

Takeoff flight path profiles generally are predicated upon the occurrence of an engine malfunction during takeoff and therefore rejecting or continuing the flight with the remaining engine(s) if possible. For multiengine helicopters the critical decision point (CDP) is defined as a combination of speed and altitude at which, if an engine suddenly becomes inoperative, the takeoff can be rejected and a safe landing effected within the heliport (prepared surface) boundaries. As an alternative, CDP allows the flight to be continued with a clearance past the edge of the heliport of no less than H_1 (Fig. 3-189), with the minimum height of the flight path yielding an obstacle clearance of no less than H_2 and maintaining an obstacle clearance of no less than H_3 during the single-engine climbout. The magnitudes of H_1, H_2, and H_3 are dependent upon individual mission requirements and should be provided by the procuring activity, or selected on an objective basis and approved by the procuring activity.

In Fig. 3-189 the distance s_1 defines the prepared landing surface, or field size, and is predicated upon the rejected takeoff distance plus aircraft length, although the OEI landing distance also must be considered in this determination. The distances s_2 and s_3 are the criteria for establishing the takeoff procedures. Although the oblique procedure yields greater operating weights than does the vertical, the minimum climbout speed is limited by the ability of the airspeed system accurately and repeatedly to indicate the desired speed. Present airspeed systems are not reliable below speeds of 20 kt. The use of the vertical procedure eliminates the requirement for the airspeed system by using a visual ground reference to maintain near-vertical ascents. The effects of operating weight and field size requirements upon takeoff procedure are shown in Fig. 3-190.

For all takeoff procedures the flight path must not go into the "avoid" region of the height-velocity (H-V) limitation curve (par. 3-5.1) for the operating weights under consideration.

Considerations for landing are similar to those for takeoff, with the CDP having its equivalent in the landing decision point (LDP). LDP can be defined as the point from which the aircraft can clear the front of the prepared surface by H_1 and make a safe landing if an engine malfunctions at or past that point, or can accel-

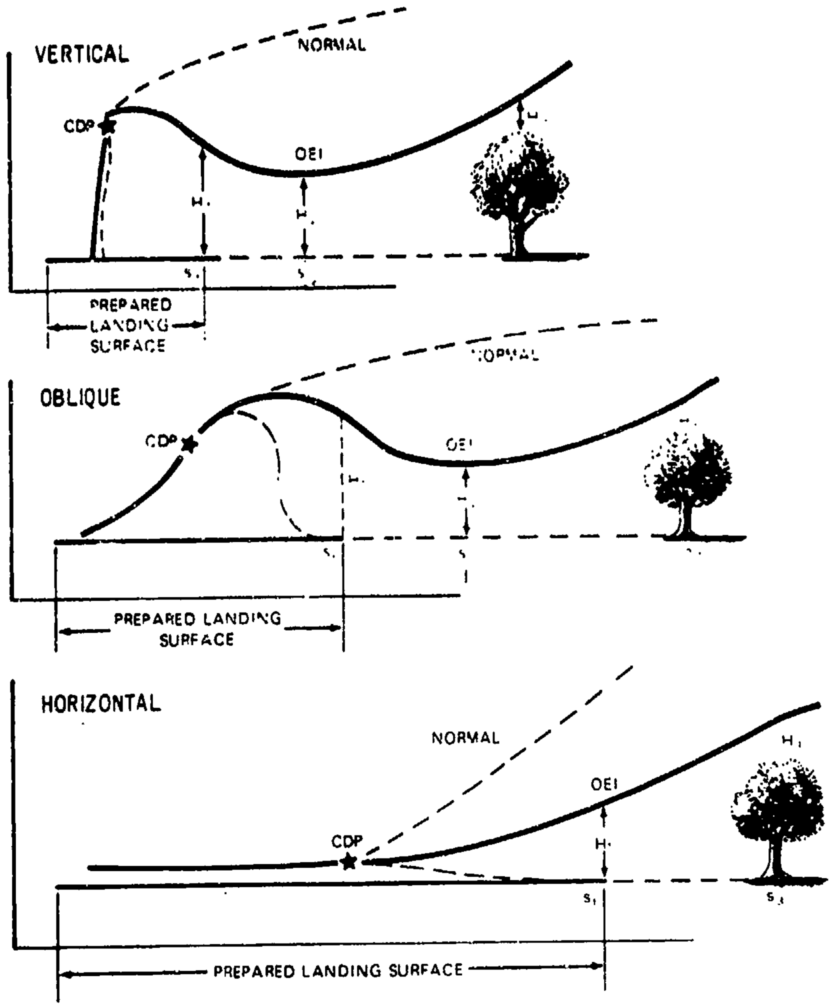


Fig. 3-189. Flight Path Profiles for Takeoff Procedures

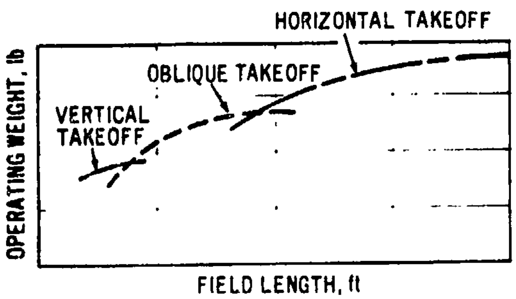


Fig. 3-190. Influence of Takeoff Procedure on Operating Weight and Field Size

erate and continue the flight OEI and maintain an obstacle clearance of H_2 (Fig. 3-191). The values for H_1 and H_2 should be obtained from pertinent military requirements.

The prepared surface size requirements established for landing are based upon the point-to-point distance measured from the point where the aircraft passes through height H_1 to a full stop plus the helicopter length. A safety factor generally is applied to this distance to account for variables such as threshold clearance and runway surface condition.

The field size requirement for a given helicopter is established as the greater of the lengths required for the rejected takeoff and landing maneuver. Field size requirements for landings generally will follow the same trend as shown in Fig. 3-190.

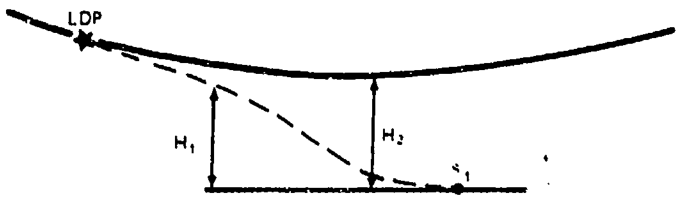


Fig. 3-191. Landing Profile

3-5.3.4.1 Calculation of Flight Paths

The calculation of takeoff and landing profiles involves the analysis of rotor aerodynamics in accelerating low-speed flight ($V < 60$ kt). Because blade element theory or numerical methods are not adaptable readily to low-speed flight regimes, energy methods are employed. The energy method outlined in par. 3-2 yields an acceptable base for estimating flight profiles, although some of the simplifying assumptions must be modified. During takeoff and landing maneuvers the rotor experiences large tip path plane angular changes, and the assumption that rotor thrust is vertical no longer is valid ($\alpha_{tpp} \neq 0$). Therefore, the rotor-induced velocity as defined by Eq. 3-35 of par. 3-2.1 is used directly. Its form should not be simplified by small angle assumptions because rotor angle has a powerful effect upon induced power in low-speed flight. The value of ν/ν_0 can be solved by an iterative process. First, however, Eq. 3-32 of par. 3-2.1 is divided by its first derivative with respect to ν/ν_0. The iterative process would take the form

$$\frac{\nu}{\nu_0} = 1 \quad \text{, first estimate}$$

$$\left(\frac{\nu}{\nu_0}\right)_p = \frac{\nu}{\nu_0} - \frac{\left[\left(\frac{\nu}{\nu_0}\right)^4 - 2\left(\frac{\nu}{\nu_0}\right)^3\left(\frac{V}{\nu_0}\right)\sin\alpha + \left(\frac{\nu}{\nu_0}\right)^2\left(\frac{V}{\nu_0}\right)^2 - 1\right]}{\left[4\left(\frac{\nu}{\nu_0}\right)^3 - 6\left(\frac{\nu}{\nu_0}\right)^2\left(\frac{V}{\nu_0}\right)\sin\alpha + 2\left(\frac{\nu}{\nu_0}\right)\left(\frac{V}{\nu_0}\right)^2\right]} \qquad (3\text{-}248)$$

$$\frac{\nu}{\nu_0} = \left(\frac{\nu}{\nu_0}\right)_p \quad \text{, dimensionless}$$

The resubstitution of $(\nu/\nu_0)_p$ for ν/ν_0 can be terminated when a specified tolerance with ν/ν_0 is reached; generally, three or four passes are sufficient.

The induced power then can be expressed nondimensionally for any flight condition as

$$C_{P_i} = \frac{C_T^{3/2}}{\sqrt{2}B}\left(\frac{\nu}{\nu_0}\right) \quad \text{, dimensionless} \qquad (3\text{-}249)$$

In the evaluation of takeoff and landing performance, acceleration OGE and IGE must be taken into account. Ground effect data covering a wide range of thrust levels and forward speeds must be available for accurate analysis of ground clearance and vertical touchdown speeds. The incorporation of ground effect into the consideration of power required modifies Eq. 3-249 to

$$C_{P_i} = \frac{C_T^{3/2}}{\sqrt{2}B}\left(\frac{\nu}{\nu_0}\right)\Lambda \quad \text{, dimensionless} \qquad (3\text{-}250)$$

The method outlined in par. 3-2 illustrates the procedures for evaluating power requirements for a known operating weight. In a takeoff or landing analysis the helicopter generally starts from a steady-state flight condition and engine power is applied. Thus, the relationship that thrust equals gross weight plus vertical drag is no longer valid and must be expressed as

$$F_V = T\cos\alpha_{tpp} - (W_g + D_V) \quad \text{, lb} \qquad (3\text{-}251)$$

$$F_H = T\sin\alpha_{tpp} - D_H \quad \text{, lb} \qquad (3\text{-}252)$$

The total thrust vector may be assumed to be perpendicular to the tip path plane axis in low-speed flight. The flight path then can be computed from the equations of motion

$$\Delta s_V = V_V \Delta t + \frac{1}{2} a_V (\Delta t)^2 \quad , \text{ft} \quad (3\text{-}253)$$

$$\Delta s_H = V_H \Delta t + \frac{1}{2} a_H (\Delta t)^2 \quad , \text{ft} \quad (3\text{-}254)$$

where

$$a_V = \frac{F_V g}{W_g}, \qquad a_H = \frac{F_V g}{W_g}, \text{fps} \qquad (3\text{-}255)$$

The rotor thrust for a given power input and rpm change can best be found from the total power equation by an iterative process

$$(SHP)\eta + \frac{I}{2dt}(\Omega_1^2 - \Omega_2^2) = hp_i + hp_o + hp_p + hp_{cl} \quad (3\text{-}256)$$

The thrust-power relationships for the induced power hp_i and the profile power hp_o are given in par. 3-2. The thrust relationship to the climb power hp_{cl} may be derived from Eq. 3-191. Due to the complexities of solving directly for thrust, a reasonable approximation of actual flight profiles can be obtained by selecting finite time intervals and using the average SHP, ΩR, V, and V_V as constants across the selected time interval. This, in effect, yields a step function integral of the flight path; the smaller the time intervals, the more accurate the computed flight path.

As can be seen from this analysis, the helicopter trajectory during the takeoff and landing maneuvers is dependent strictly upon the cyclic input (tip path plane angle) and the rate of power application. At the point of assumed engine malfunction, the control inputs—cyclic and collective—become the controlling factors for attaining desired climbout speeds and using available rotor kinetic energy for minimizing the distance to the minimum point on the continued flight path, or for minimizing the vertical impact speed for rejected takeoffs and landings. High-speed computers are required for practicable evaluation of flight paths.

3-5.3.4.2 Procedure for Calculating Takeoff and Landing Problems

Although no single procedure covers all possible takeoff and landing problems, a generalized method of establishing takeoff and landing operational capabilities is outlined herein. A flow chart for a routine analytical takeoff model is shown in Fig. 3-192 with typical trade-offs and acceptable solutions covered in Figs. 3-193, 3-194, and 3-195. The flow chart for a landing routine analytical model is shown in Fig. 3-196, with typical trade-offs and acceptable solutions covered in Figs. 3-197, 3-198, and 3-199.

The discussion contained herein generalizes actual situations. The analysis of a specific situation will require that the effects of the independent variables upon the particular operational requirements be explored in much more detail.

3-5.3.4.3 Weight-altitude-temperature Characteristics

The procedures previously outlined indicate the basic methods for establishing the maximum operating weight for a prescribed takeoff procedure. The CDP and the twin-engine climbout torque (TEQ) are allowed to be a fallout so as to establish the maximum weight capabilities for a given altitude and temperature. A weight-altitude-temperature (WAT) curve based upon variations of CDP and TEQ with altitude and temperature becomes operationally unfeasible. Also, these variations do not safely account for the conditions when the aircraft is not loaded to the maximum allowable operating weight for which the takeoff procedure was established.

To establish an operationally feasible takeoff procedure that can be used for any gross weight-altitude-temperature combination, the procedure that follows is used:

The CDP, TEQ, and maximum allowable operating weight are established for a given altitude, temperature, and defined takeoff procedure. The result is referred to as the base procedure CDP. For operational practicability, this CDP is held fixed for all WAT combinations. The TEQ is used to the extent of determining the vertical velocity at the base procedure CDP. This vertical velocity is then held fixed for all WAT combinations.

The steps for determining each altitude and temperature other than the base point are:

1. Establish weight-TEQ relationship for the fixed velocity at the decision point. This appears as Curve (A) in Fig. 3-200.

2. For three TEQs, establish operating weight capability for safe rejected takeoff from design decision point. This is Curve (B) in Fig. 3-200.

3. Compute continued flight regime at three TEQs to obtain operating weight for safe continued flight, as shown in Curve (C) of Fig. 3-200. The operating weight will be the lower of the weights defined by the intersection of Curves (A) and (B) or (A) and (C). These three

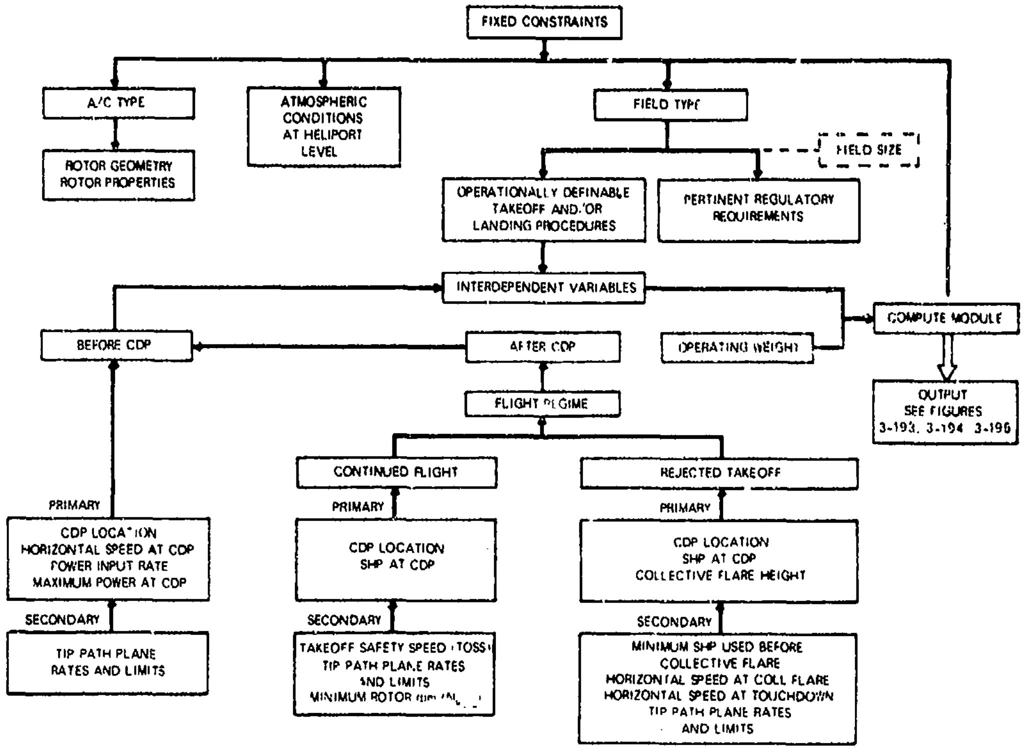


Fig. 3-192. Continued Flight and Rejected Takeoff Diagram

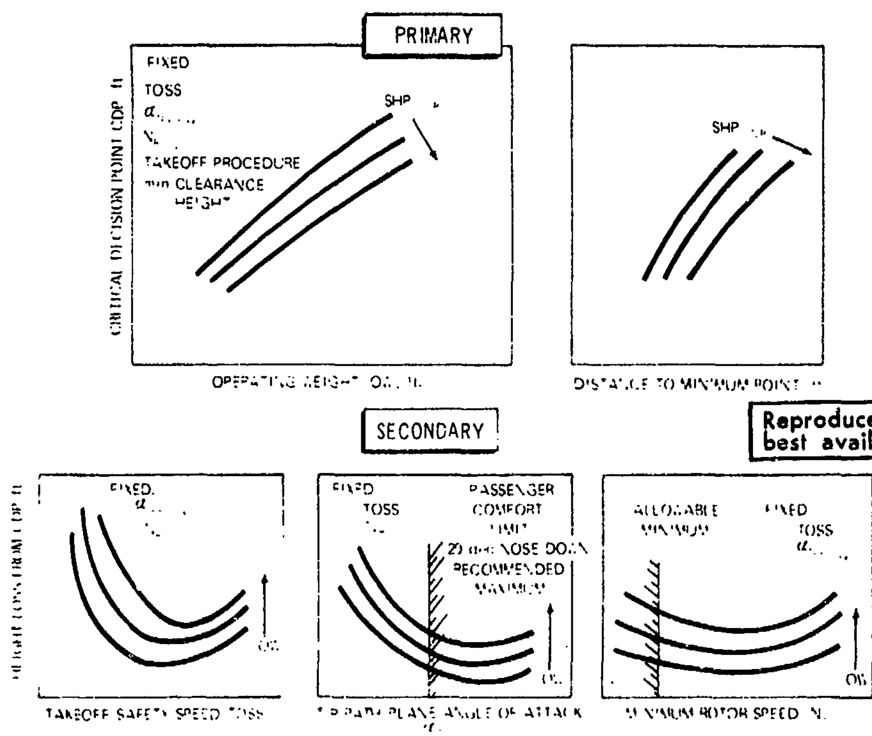
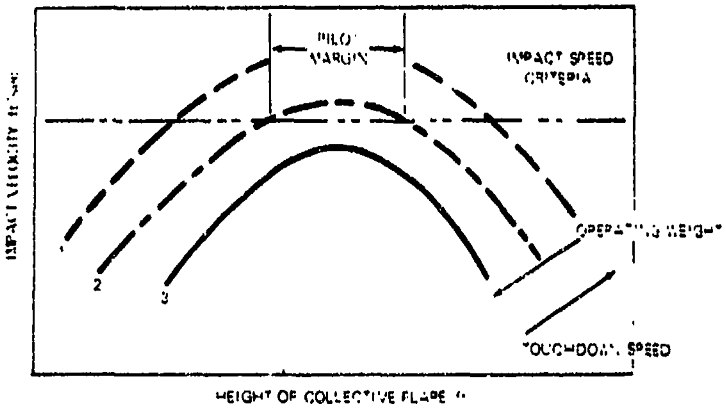


Fig. 3-193. Outputs—Continued Flight After Engine Malfunction



PRIMARY ITERATION FOR OUTPUT CONDITION:

FIRST TRIAL: HORIZONTAL TOUCHDOWN SPEED ft, HORIZONTAL SPEED AT COLLECTIVE FLARE POINT ft, OEI POWER

SECONDARY ITERATIONS FOR OUTPUT CONDITION:

CURVE 1: IF VERTICAL OPERATION:
a. INCREASE WEIGHT
b. INCREASE CDP

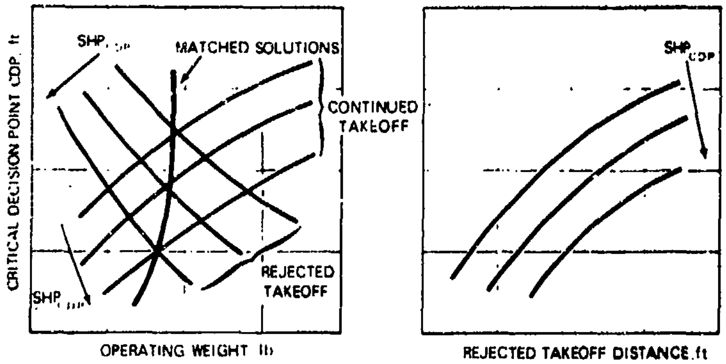
IF NONVERTICAL OPERATION (FWD SPEED AT CDP):
a. REDUCE POWER LEVEL USED DURING DESCENT TO COLLECTIVE FLARE (THIS WILL MINIMIZE HORIZONTAL DISTANCE REQUIRED)
b. INCREASE WEIGHT
c. INCREASE CDP

CURVE 2: DESIRED SOLUTION
PILOT MARGIN SHOULD BE ± 3-5 ft FROM PEAK

CURVE 3: IF VERTICAL OPERATION:
a. DECREASE WEIGHT
b. DECREASE CDP

IF NONVERTICAL POPERATION
a. INCREASE SPEED AT COLLECTIVE FLARE POINT
b. INCREASE SPEED AT TOUCHDOWN POINT
c. ADJUST a OR b ABOVE.

Fig. 3-194. Outputs—Rejected Takeoff After Engine Malfunction



1. THE CDP IS THAT POINT FROM WHICH THE AIRCRAFT CAN CONTINUE ITS FLIGHT AFTER ONE ENGINE MALFUNCTIONS AND MAINTAIN A MINIMUM HEIGHT OF 'X' ft IN RELATION TO THE TAKEOFF SURFACE OR REJECT THE TAKEOFF AND CONTACT THE GROUND WITH A VERTICAL IMPACT SPEED OF NO GREATER THAN 'Y' ft sec.
2. THE INTERSECTION OF THE CURVES ESTABLISHED FROM CONTINUED FLIGHT RUNS AND REJECTED TAKEOFF RUNS YIELDS THE OPERATING WEIGHT-CDP-SHP_{CDP} COMBINATIONS WHICH SATISFIES BOTH CONDITIONS.
3. THE FIELD LENGTH REQUIREMENTS ARE BASED ON THE REJECTED TAKEOFF DISTANCE. BECAUSE THE COMPUTED DISTANCE IS THE POINT-TO-POINT DISTANCE OF THE MAIN WHEEL THE AIRCRAFT LENGTH MUST BE ADDED TO THE COMPUTED DISTANCE.
4. THE MINIMUM REQUIRED FIELD LENGTH IS EQUAL TO THE GREATER OF THE TWO LENGTHS DEFINED BY THE REJECTED TAKEOFF AND THAT REQUIRED FOR LANDING. (Fig 3-199)

Fig. 3-195. Determination of Matched CDPs for the Continued Flight and Rejected Takeoff Cases

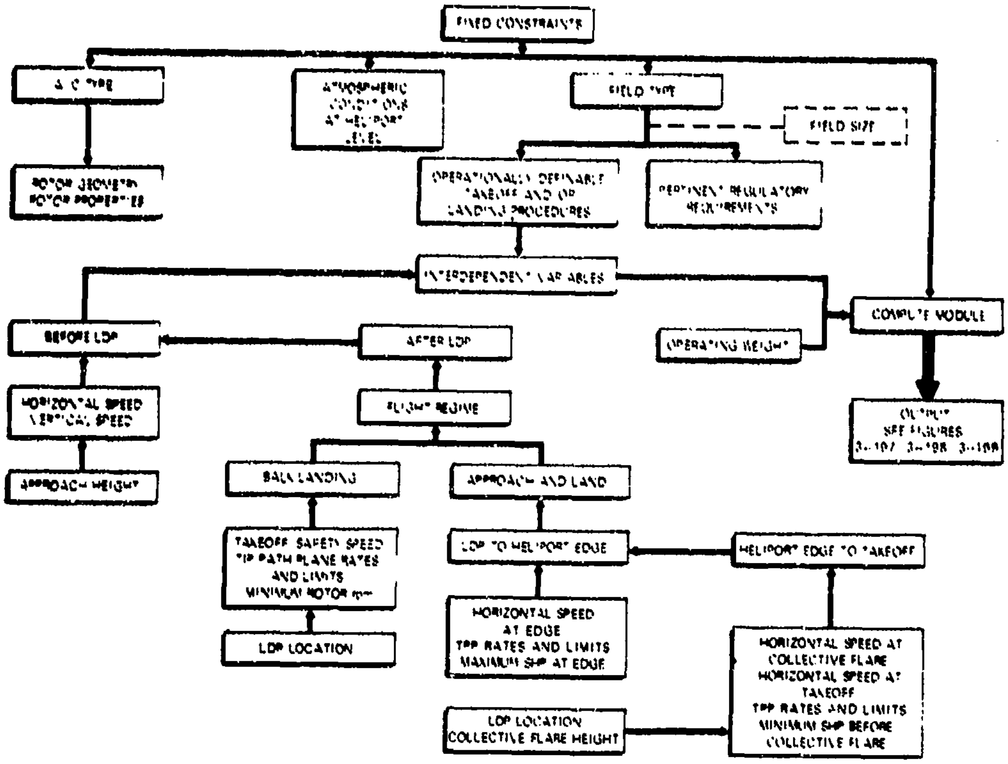
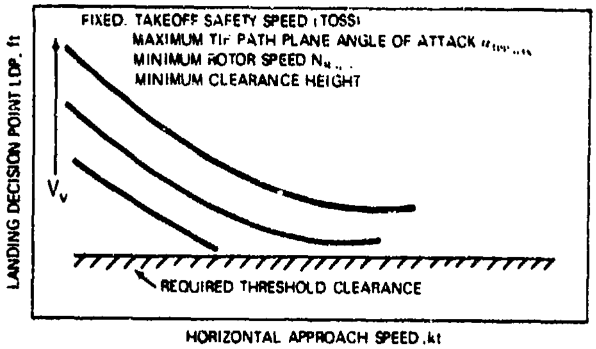


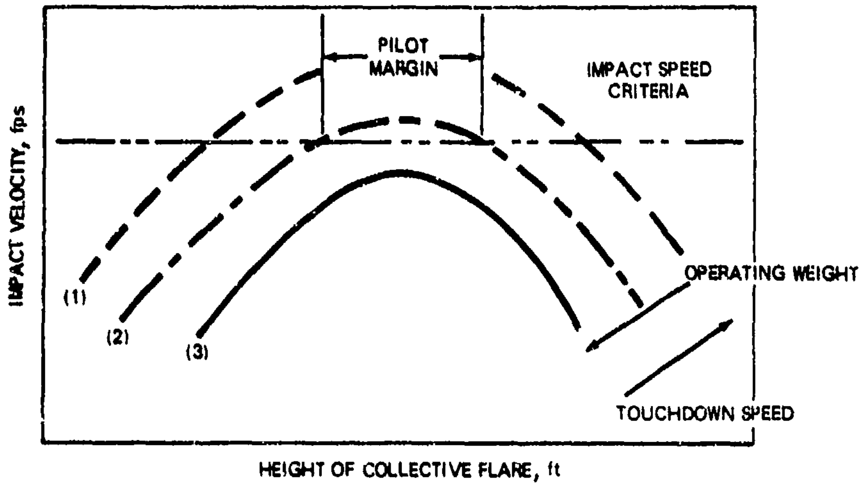
Fig. 3-196. Balked Landing and Landing Diagram



OUTPUT BALK LANDING

1. LANDINGS ARE GENERALLY RUN AT THE SOLUTION WEIGHTS FOR THE CONTINUED FLIGHT AND REJECTED TAKEOFF CONDITIONS.
2. THE PROCEDURES ESTABLISHED AFTER ONE ENGINE MALFUNCTIONS FOR THE CONTINUED FLIGHT REGIME ARE MAINTAINED FOR THE BALK LANDING, i.e. TOSS, $\alpha_{tpp_{max}}$, AND $N_{R_{min}}$.
3. THE PRIMARY ITERATIONS INVOLVE LOCATING THE LDP AS A FUNCTION OF THE HORIZONTAL AND VERTICAL APPROACH SPEED AND N_R.
4. A TRADE-OFF WITH OPERATING WEIGHT CAN BE MADE USING THE SAME PROCEDURE AS CONTINUED FLIGHT BY SUBSTITUTING LDP FOR CDP AND APPROACH VERTICAL VELOCITY FOR SHP_{tip}. THE PLOT WILL BE SIMILAR TO THAT SHOWN IN Fig. 3-194.

Fig. 3-197. Landing Decision Point (LDP) Characteristic With Vertical and Horizontal Approach Speed



OUTPUT LANDINGS

THE OUTPUT FOR LANDINGS IS TREATED IN THE SAME MANNER AS THAT FOR REJECTED TAKEOFF.

SECONDARY ITERATIONS

DIAGNOSTICS:

a AIRCRAFT LANDED SHORT OF PROSPECTIVE LANDING SITE.

FIX: INCREASE SPEED AT EDGE OF HELIPORT.

b AIRCRAFT LANDED BEYOND PROSPECTIVE LANDING SITE.

FIX: DECREASE SPEED AT EDGE OF HELIPORT.
DECREASE OPERATING WEIGHT.

RESULTS YIELD CURVE:

(1) a REDUCE POWER LEVEL USED DURING CYCLIC FLARE.
b INCREASE VERTICAL APPROACH VELOCITY (REQUIRES LDP INCREASE).
c INCREASE WEIGHT.

(2) DESIRED SOLUTION. PILOT MARGIN SHOULD BE ± 5 ft FROM PEAK.

(3) a INCREASE SPEED AT COLLECTIVE FLARE POINT.
b INCREASE SPEED AT TOUCHDOWN POINT.
c DECREASE VERTICAL APPROACH VELOCITY (REQUIRES LDP DECREASE).
d DECREASE WEIGHT.
e INCREASE HORIZONTAL APPROACH VELOCITY (REQUIRES LDP DECREASE).

Fig. 3-198. Primary Iteration for Impact Velocity With Height-collective Flare

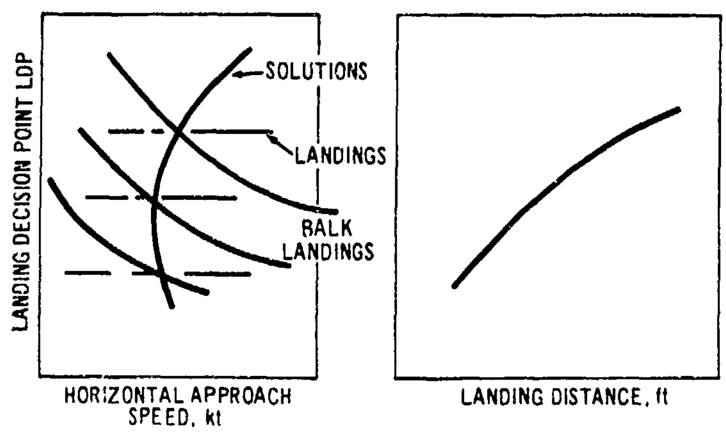


Fig. 3-199. Determination of Matched LDPs for the Landing and Balked Landing Cases

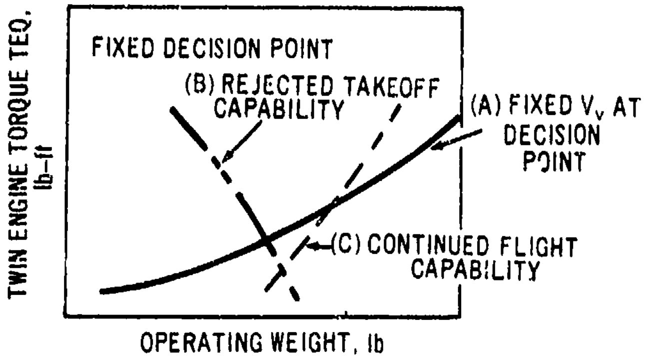
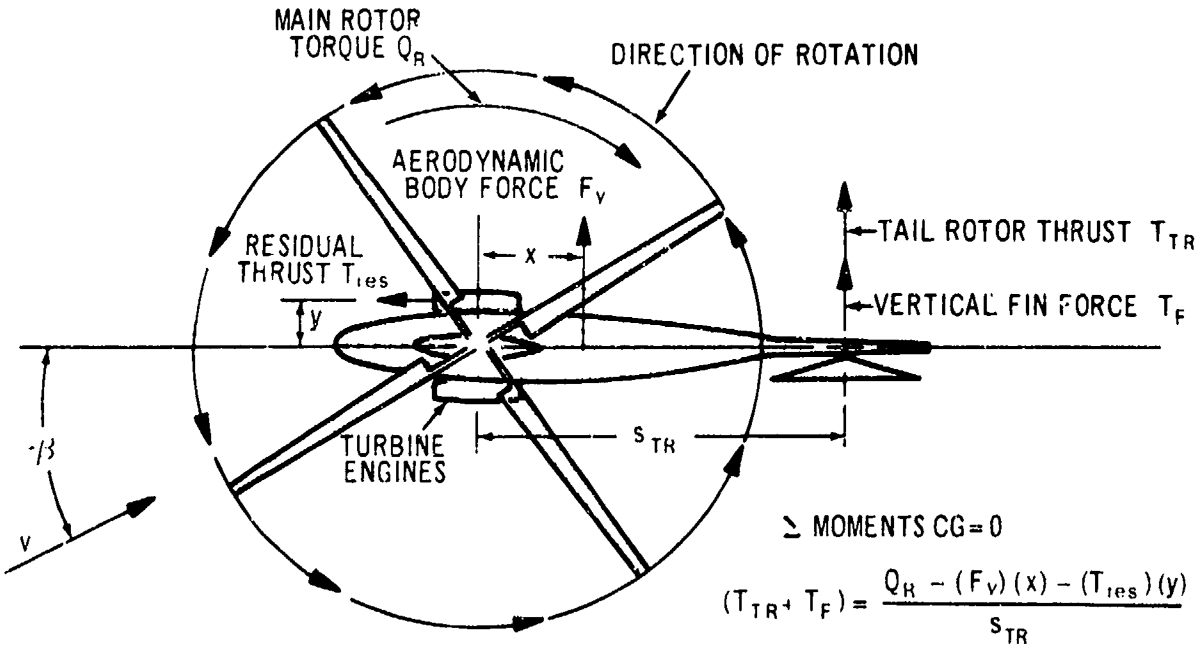


Fig. 3-200. Weight / Twin-engine Torque (TEQ) Relationship

curves will have a common TEQ-operating weight intersection for the base procedure point.

3-5.3.5 Aircraft Trim Characteristics

For the conventional single-rotor helicopter, the loss of one engine will require a change in yaw control when operating in the region where the remaining engine(s) cannot provide the required torque. In this case, the aircraft will be slowed to the power condition at which trim requirements can be satisfied. Fig. 3-201 shows the moments acting upon the aircraft while in sideslip trimmed flight. Note that primary control is derived from the antitorque device(s) at the aft section of the fuselage. The size of the vertical fin and/or its camber, the adequacy of control range on the tail rotor, the control range in the movable rudder (if available), and the roll attitude contribution of each become the primary design attributes for study under this flight situation. Efficient use of energy in driving the free turbine, with very low exit velocity as a consequence, makes the residual thrust small and, therefore, not a critical factor in the trim equation. Also, most engine installations are located very close to the transmission and the yawing moment produced by residual engine thrust is minimal.



$$(T_{TR} + T_F) = \frac{Q_R - (F_V)(x) - (T_{res})(y)}{S_{TR}}$$

Fig. 3-201. Moments Acting on Aircraft While in Sideslip Trimmed Flight

3-6 LIST OF SYMBOLS

A = area, ft^2
= rotor disk area, ft^2
AF = propeller activity factor, dimensionless
AI = autorotative index, sec
AR = aspect ratio, dimensionless
A_e = effective disk area of tandem rotor, ft^2
A_{1_s} = coefficient of cos ψ in Fourier series that expresses blade pitch angle θ relative to the rotor shaft, hence lateral cyclic pitch, deg or rad
a = slope of lift coefficient versus angle of attack curve, per rad or per deg
= speed of sound, fps
= numerical value (function of variables $V_1, V_2, \ldots, V_n$)
= linear acceleration, fps^2
a_0 = speed of sound at standard sea-level temperature, dimensionless
a_n = coefficient of cos $n\psi$ term in Fourier series for flapping angle β as function of azimuth angle ψ, deg or rad
a_{1_s} = coefficient of cos ψ in Fourier series for flapping angle β relative to shaft, hence longitudinal tilt of tip path plane with respect to the rotor shaft, positive up in front, deg or rad
B = tip loss factor, dimensionless
BHP = total helicopter horsepower required
BPR = bypass ratio, the ratio of the mass rate of airflow through the fan to that through the core gas generator, dimensionless
BR = effective blade length, ft
B_{1_s} = coefficient of sin ψ in Fourier series that expresses blade pitch angle θ relative to the rotor shaft, hence longitudinal cyclic pitch, deg or rad
b = numerical value, (function of variables $V_1, V_2, \ldots, V_n$)
= propeller blade section elemental width, ft
= number of blades
= wing span, ft
b_n = coefficient of sin $n\psi$ term in Fourier series for flapping angle β as a function of azimuth angle ψ, deg or rad
b_{1_s} = coefficient of sin ψ in Fourier series for flapping angle β relative to the rotor shaft, hence lateral tilt of tip path plane, positive up on retreating side, deg or rad
C_D = drag coefficient, dimensionless
$\overline{C}_D$ = mean drag coefficient, dimensionless
C_L = lift coefficient, dimensionless
$\overline{C}_L$ = mean lift coefficient, dimensionless
C_{L_i} = integrated design lift coefficient (propeller blade), dimensionless
C_{LR} = rotor lift coefficient, dimensionless
C_P = power coefficient, dimensionless
C_Q = torque coefficient, dimensionless
C_{Q_∞} = torque coefficient OGE, dimensionless
C_T = thrust coefficient, dimensionless
C_{T_∞} = thrust coefficient OGE, dimensionless
C_{T_I} = thrust coefficient corrected for reduced inflow due to ground effect, dimensionless
C_W = weight coefficient, dimensionless
C_{XR} = rotor horizontal force coefficient, dimensionless
c = blade chord, ft
= shaft engine specific fuel consumption, lb-fuel/hp-hr
c' = jet engine specific fuel consumption, lb-fuel/lb-thrust-hr
c_d = section drag coefficient, dimensionless
c_l = section lift coefficient, dimensionless
c_p = specific heat at constant pressure, Btu/lb-°R

c_v = specific heat at constant volume, Btu/lb-°R
D = drag, lb
= diameter, ft
D_e = helicopter total effective drag, lb
D_e = helicopter effective drag coefficient, dimensionless
D_V = hover download, lb
d = glide distance, ft
d_f = induced power interference parameter, dimensionless
e = wing span efficiency factor, dimensionless
F = force, lb
= thrust-coefficient, dimensionless
f = equivalent flat plate drag area, ft^2
= thrust coefficient, dimensionless
f_c = ratio of speed of sound at standard day sea level to speed of sound at specified operating conditions, dimensionless
f_W = weight of fuel, lb
f_w = fraction of vertical force supported by wing, dimensionless
g = acceleration due to gravity, ft/sec^2
H = rotor H-force perpendicular to rotor shaft, lb
= clearance height, ft
HV = heating valve of fuel, Btu/lb
h = altitude, ft
= enthalpy, Btu/lb
hp = horsepower, hp (1 hp = 550 ft-lb/sec)
hp_a = accessory power, hp
hp_{aux} = auxiliary propulsive power, hp
hp_g = gear and transmission power losses, hp
I = mass moment of inertia of blade about flapping hinge, slug-ft^2
= rotor moment of inertia, slug-ft^2
I_p = polar area moment of inertia, ft^4
i_s = angle of rotor shaft relative to a reference perpendicular to the fuselage reference line, deg
J = mechanical equivalent of heat, ft-lb/Btu (1 ft-lb = 778 Btu)
= advance ratio, dimensionless
K = tandem rotor interference factor, dimensionless
K_F = weight of fuel tank per pound of fuel, dimensionless
K_g = gear and transmission loss ratio, dimensionless
K_p = climb efficiency factor, dimensionless
K_r = fuel reserve factor, dimensionless
K_t = control power ratio, dimensionless
K_u = induced power correction factor (forward flight), dimensionless
K_1 = factor related to propeller weight, dimensionless
K_2 = factor related to gearbox weight, dimensionless
KE = kinetic energy, ft-lb
k = retreating blade stall factor dependent upon blade geometry and loading, lb^2/ft^2
= empirical constant, (fps)^6
= specific heat ratio c_p/c_v, dimensionless
= spring constant, lb/ft
L = lift, lb
L_c = helicopter lift coefficient, dimensionless
LF = loss factor, dimensionless
M = Mach number, dimensionless
= rotor Figure of Merit, dimensionless
= moment, lb-ft
M_t = tip Mach number, dimensionless
M_0 = rotor tip Mach number (hovering), dimensionless
$M_{x,\psi}$ = rotor blade Mach number at radius ratio (r/R) x and azimuth angle ψ, dimensionless
m = mass, slug
N = rotational speed, rpm
N_g = gas producer rotational speed, rpm
N_p = power turbine rotational speed, rpm
NRP = normal rated power, hp
$N/\sqrt{\theta}$ = generalized rotor speed, rpm

n = load factor, dimensionless
= number of engines
= propeller speed, rev/sec

n, n_1, n_2 = numerical constants, dimensionless

P = power, ft-lb/sec

P_w = power transferred from the engine to the propeller, ft-lb/sec

p = total pressure, lb/ft^2

p' = total pressure (ideal cycle), lb/ft^2

Q = rotor torque, lb-ft
= energy added in the combustor in the form of fuel, Btu/sec

q = dynamic pressure, lb/ft^2

R = rotor or propeller radius, ft
= resultant force vector on blade element, lb

R/C = rate of climb, fpm

R_F = fuel weight to gross weight ratio, dimensionless

Rg = range, n mi

r = elemental radius, ft

rhp = main rotor shaft horsepower

r_p = compression ratio, dimensionless

S = reference area, ft^2
= wing area, ft^2

SFC = specific fuel consumption, lb/hp-hr

SHP = shaft horsepower

$SHP/(\delta\sqrt{\theta})$ = generalized shaft horsepower

s = distance, ft

s_R = rotor shaft spacing ratio, dimensionless

T = thrust, lb
= rotor or propeller thrust, lb
= total temperature, °R

T_{hp} = power loading, lb/hp

t = time, sec or hr
= endurance, hr

t_c = rotor blade lift coefficient, dimensionless

t_e = time required to dissipate rotor kinetic energy in hover down to rotor speed corresponding to stall at initial rotor thrust, sec

U = total velocity through rotor disk, fps

V = free stream velocity, fps

V = helicopter speed, fps
= true airspeed of forward flight, kt

V' = resultant velocity, function of v and V in forward flight, fps

V_D = vertical rate of descent, fps

$\bar{V}_D$ = nondimensionalized vertical rate of descent, dimensionless

V_E = airspeed for minimum power, kt

V_e = duct exit velocity, fps

V'_e = no loss duct exit velocity, fps

V_j = effective exhaust jet velocity far behind the propulsive device, fps

V_L = local velocity at blade element, fps

V_p = axial velocity at the plane of the propeller, fps

V_s = sink speed at time of impact, fps

V_t = rotor tip speed, fps

$V/\sqrt{\theta}$ = generalized airspeed, kt

V_0 = forward speed at time of power failure, fps

$V_1, V_2, \ldots, V_n$ = selected variables

W = weight, lb

W_a = weight flow of air, lb/sec

W_c = crew weight, lb

W_e = weight empty, lb

W_F = weight of fuel and fuel tank, lb

W_f = engine fuel flow, lb-fuel/hr

$W_f/(\delta\sqrt{\theta})$ = generalized engine fuel flow, lb-fuel/hr

W_g = gross weight of helicopter, lb

W_g/δ = generalized gross weight, dimensionless

W_p = payload, lb

W_{sh} = shaft work, Btu/lb

w = rotor disk loading, lb/ft^2

w_{eq} = equivalent induced velocity for a wing, fps

x = generalized radius ratio r/R, dimensionless
= displacement along longitudinal axis, ft

y = lateral displacement, ft

Z = height from rotor centroid to ground, ft

α = angle of attack, rad or deg
= turboshaft fuel flow intercept at zero power, lb-fuel/hr

α_C = rotor control plane angle of attack, deg
α_i = induced incremental angle of attack = v/V, rad
α_R = rotor tip plane path angle of attack, deg
α_{tpp} = tip path plane tilt, deg
α_w = wing angle of incidence, rad
$\alpha_{x,\psi}$ = rotor blade element angle of attack, at radius ratio (r/R) x and azimuth angle ψ, rad or deg
β = flapping angle, rad or deg
= propeller pitch angle, deg
= helicopter yaw angle, deg
= turboshaft fuel flow slope versus shaft horsepower, lb-fuel/hr-hp
β_0 = rotor coning angle, rad or deg
Γ = vortex strength, ft^2/sec
γ = flight path angle with respect to horizon, deg
= Lock number, dimensionless
= wake skew angle, rad
ΔR_{F1} = fuel weight ratio required to climb, dimensionless
ΔR_{F2} = fuel weight ratio required in cruise, dimensionless
ΔR_{F3} = fuel weight ratio required to start and maneuver, dimensionless
δ = altitude pressure ratio, dimensionless
= incremental velocity ratio (change in V_p/V) due to the presence of a propeller duct, dimensionless
δ_0, δ_1, and δ_2 = coefficients of equation $c_d = \delta_0 + \delta_1\alpha + \delta_2\alpha^2$, dimensionless
$\delta V/W_f$ = generalized specific range, n mi/lb-fuel
η = efficiency, dimensionless
= mechanical efficiency of the helicopter, dimensionless
η_C = Cheeseman forward flight efficiency factor, dimensionless
η_e = expansion efficiency, dimensionless
η_p = propulsive efficiency, dimensionless
$\eta_{tr(jet)}$ = jet transfer efficiency, dimensionless
θ = helicopter pitch attitude, deg or rad
= altitude temperature ratio, dimensionless
= rotor blade section pitch angle, deg or rad
θ_x = blade pitch angle at radius ratio (r/R) x, deg
θ_0 = collective pitch
θ_1 = total blade twist root to tip, deg
Λ = ground effect correction factor, dimensionless
λ = inflow ratio, dimensionless
μ = advance ratio, dimensionless
ν = rotor-induced velocity, fps
$\bar{\nu}$ = generalized induced velocity, ν/ν_0, dimensionless
ν_0 = rotor induced velocity in hover OGE, fps
ν_2 = rotor-induced velocity infinitely downstream, fps
ξ = $(V_j - V)/V$, dimensionless
ρ = density of air, slug/ft^3
ρ_0 = NACA standard air density at sea level, 0.002378 slug/ft^3
σ = rotor solidity, dimensionless
= ratio of ambient air density to standard sea level density, dimensionless
τ_p = pressure ratio, dimensionless
ϕ = empty weight to gross weight ratio, dimensionless
= rotor inflow angle, rad
ψ = azimuth angle of rotor blade from its downwind position, rad or deg
= yaw angle, deg
Ω = rotor angular speed, rad/sec
ΩR = rotor tip speed, fps
ω_n = natural frequency of vibration, rad/sec

Subscripts

A = aerodynamic
a = average
acc = acceleration
amb = ambient
av = available
b = combustion
c = compressor
ch = chart
cl = climb
cr = cruise

crit = critical
D = descent
DD = drag divergence
d = deceleration
des = design
E = elastic
e = effective
eq = equivalent
ex = excess
F = vertical fin
f = fuel
ff = forward flight
fl = flare
flt path = flight path
fus = fuselage
H = horizontal
HIGE = hover in-ground effect
HOGE = hover out-of-ground effect
hi = high
hov = hover
I = inertia
IGE = in-ground effect
i = induced
init = initial
lev = level flight
lo = low
max = maximum
man = maneuver
min = minimum
miss = mission
nom = nominal
nor = normal
NRP = normal rated power
o = profile
OGE = out-of-ground effect
opt = optimum
P = propeller
p = pressure
= parasite
= perpendicular
= propulsive
par = partial load or power
R = rotor
R/D = rate of descent
r = radial
ref = reference
req = required
res = residual
rot = rotation
s = single rotor
= single blade element
sh = shaft
sl = sea level
SP = shrouded propeller
st = stall
std = standard
T = total
t = turbine
TR = tail rotor
= tangential
tr = transfer
V = vertical
W = weight
w = wing
x = with respect to longitudinal (*x*) axis
y = with respect to lateral (*y*) axis
z = with respect to vertical (*z*) axis
0 = standard or initial condition
1 = intermediate or, in the case of single-step operation, final condition. Also, engine station 1 (induction system inlet)
2 = final condition for a two-step operation, e.g., flow-through a rotor. Also, engine station 2 (compressor inlet)
3, 4, etc. = subsequent stations in turbine engine, or equivalent thermodynamic cycle. See Fig. 3-53 for designations.

Prefixes

d = differential of
f = function of
Δ = increment of

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CHAPTER 4

STRUCTURAL DESIGN

4-1 INTRODUCTION

This chapter is concerned with the definition of structural design criteria for helicopters, the determination of basic loads based upon these criteria, and the demonstration, by preliminary structural analysis, of the adequacy of the structure to withstand these loads. The design criteria included are based primarily upon existing Military Specifications. In some areas acceptable criteria are not provided by these specifications and, therefore, experience dictates the requirement. Some instances of the latter are fatigue spectra and fail-safe criteria.

Design criteria are developed for the following conditions:

1. Flight and takeoff
2. Landing
3. Ground
4. Controls
5. Special loadings
6. Miscellaneous.

These criteria, together with the design parameters and characteristics of a given model helicopter, are used to calculate basic loads. Methods for the determination of loads applicable to components such as rotor systems and landing gear are described in detail. For those components for which the load determination procedure is explained adequately in available publications, appropriate references are cited.

It is a practical impossibility to analyze structural adequacy for every loading condition a helicopter might encounter. However, experience has shown that there are only a few critical loading conditions. During the preliminary design process, these critical conditions must be identified and the applicable loads calculated. If the helicopter can withstand these critical loads, it will have an adequate margin of safety for all other loads normally encountered.

Army helicopters are classified by mission utilization (See Chapter 1); preliminary load development is accomplished with the end use of the vehicle as guidance.

A design parameter fundamental to basic helicopter loads is design limit flight speed V_{DL}. Other speeds (par. 4-2), used primarily for developing fatigue spectra, are percentages of the design limit flight speed or of the maximum level flight speed V_H.

Three values of gross weight to be used in preliminary design are minimum design gross weight, basic structural design gross weight, and maximum alternate gross weight. These criteria are defined and their application discussed in par. 4-3. Other helicopter weights, applicable to specific loading conditions, also are discussed.

The determination of flight and takeoff loading requirements (par. 4-4) is based upon several flight conditions that must be examined at maximum and minimum design rotor speeds, both power-on and power-off. Some of the design conditions are design limit speed, symmetrical dives and pullouts, vertical takeoff, rolling pullout, and yaw. Additional maneuvers and mission conditions of significance in the determination of component fatigue or service lives also are discussed.

Landing load criteria reflect the fact that Army helicopter environments include unprepared landing areas, adverse weather and terrain, and relatively inexperienced personnel. Par. 4-5 lists preliminary design requirements for level and asymmetric landings, with and without forward speed, for various helicopter sizes, weights, and configurations. Reserve energy requirements and crash load factors also are specified.

Ground handling can impose critical loads upon the aircraft structure. Par. 4-6 describes preliminary design requirements for taxiing, jacking and mooring, and towing and transport.

Loading conditions resulting from acceleration and braking of the rotor are discussed in par. 4-7. The most critical conditions usually occur when maximum engine power is applied to the system at low rotor speed, causing peak torques in the transmission and large bending moments on the rotor blades. The rotor-braking load to be considered involves bringing the rotor to rest from a low rotor speed.

Preliminary design for hard points required for the attachment of external stores involves load analyses to insure the structural integrity of hard-point fittings and their supporting structure, store and pylon fairings, and adjacent structures affected by the presence of external stores. Requirements for use of test data and models for analyzing hard-point loads are given in par. 4-8.

Substantiation of the structural adequacy of rotor, drive, and control systems is described in par. 4-9. The determination of applicable loads, both steady and alternating, is discussed, together with appropriate methods for preliminary structural analysis. Control system load origins, the criteria for structural design, and methods of localizing loads are given in this paragraph. Structural consideration must be given to ultimate loads, both in flight and during ground handling, and to periodic (fatigue) loads.

The structural substantiation of the airframe is treated in par. 4-10. For purposes of this discussion airframe structure includes the fuselage, wing and empennage, landing gear, and mission equipment installations. Limit loads are to be multiplied by the ultimate factor of safety (1.5), and a positive margin against failure is to be maintained. Yielding is not permitted at limit load and failure is not permitted at ultimate load. One exception to this requirement is in the design of certain landing gear such as skid gear; experience has shown that yielding of the energy-absorbing skid or spring device may be permitted for the limit condition specified.

Because it contains a large complement of rotating machinery and has a rotor that constantly generates a cyclic load input, a helicopter operates in a severe fatigue environment. A large number of dynamic components are designed primarily so that they will provide adequate fatigue strength. A design requirement is a minimum fatigue or service life, generally 3600 hr, for parts that cannot be designed fail-safe. Methods for determination of service lives are discussed in par. 4-11. Both preliminary estimates and the final determination, which is based upon flight load surveys and laboratory tests to failure, are treated.

4-2 DESIGN FLIGHT SPEEDS

At least two design flight speeds *shall* be used in preliminary structural design. The first is V_H, the design maximum level flight speed in forward, rearward, or sideward flight. MIL-S-8698 defines V_H in forward flight as the maximum speed attainable at the basic design gross weight in level flight using intermediate (formerly military or 30 min) power, or as may be limited by blade stall or compressibility effects. For preliminary design, V_H *shall* be not less than the maximum level flight speed attainable in forward, sideward, and rearward at each applicable gross weight using intermediate power. Minimum values of 35 kt are required for V_H in sideward or rearward flight with specific values being established by the procuring activity for each model helicopter. These determinations depend upon individual mission requirements.

The second design flight speed is V_{DL}, design limit flight speed. This speed is defined as a given percentage above V_H. The ratio V_{DL}/V_H *shall* be not less than 1.25 for attack, 1.20 for utility, and 1.15 for observation, training, cargo, or heavy-lift helicopters.

The design flight speeds are used throughout the preliminary design process as a basis for evaluating critical loading conditions. Specific applications of the design flight speeds appear throughout this chapter. Additional flight speeds also are specified for individual loading conditions. Examples of these are:

1. Autorotational dive speed V_{D_a}: the design maximum flight speed in autorotation
2. The flight speed for minimum rate of descent in autorotation $V_{V_s\,min}$
3. The flight speed for maximum rate of climb $V_{R/C\,max}$: the forward flight speed at which power required for level flight is minimum, hence at which power available for climb is maximum
4. Cruise speed V_{cr}: the highest forward flight speed at which specific range (n mi/lb fuel) is 99% of maximum
5. Endurance speed V_{end}: the flight speed at which fuel consumption rate (lb/hr) in level forward flight is minimum.

Values of these flight speeds may be stated in the helicopter detail specification. If they are not so specified, they should be calculated (see par. 3-4).

The operating limit flight speed is the "never-exceed" speed V_{NE}. The value of this "red-line" airspeed should be equal to V_{DL}. However, vibration or stress limits upon helicopter flight speeds may be found during the flight test program. In such cases V_{NE} may have to be less than V_{DL}. During preliminary design, V_{NE} *shall* be assumed equal to V_{DL}.

Helicopter speeds pertinent to landing conditions are discussed in par. 4-5.3.

4-3 GROSS WEIGHTS

Three gross weights W_g are significant to the preliminary design of helicopters: minimum design, basic structural design W_{g_b}, and maximum alternate design gross weights W_{g_a}.

The minimum design gross weight is the lowest gross weight considered practicable for flight. This gross weight for all models *shall* consist of all items of the helicopter basic weight (see Chapter 10) plus:

1. 5% of usable fuel
2. Minimum quantity of oil consistent with Item 1
3. Minimum crew (170 lb).

This gross weight represents a helicopter returning from a mission with all disposable payload items expended. Maximum load factor and angular acceleration are obtained at the extremities of the helicopter during flight maneuvers at this gross weight.

The basic structural design gross weight for all helicopter models *shall* be the takeoff gross weight with full internal fuel and with full internal and external load items required for the performance of the primary mission, as defined in the helicopter detail specification. Inclusion of full internal fuel in this definition of basic structural design gross weight is a departure from MIL-S-8698 that has been found necessary to provide adequate strength in Army helicopters. Design gross weight, as defined in Chapter 10, remains the basis for performance calculations.

The maximum alternate design gross weight *shall* be as prescribed in the helicopter detail specification. In any case, the maximum alternate design gross weight *shall* not be greater than the maximum gross weight at which the helicopter can take off from an unprepared field of 800-ft length and clear a 50-ft obstacle in not less than an additional 200 ft. Such takeoff *shall* be at sea level on a standard day and *shall* use the lesser of intermediate power or maximum transmission torque. The load factors n_a applicable to this gross weight *shall* be the load factors n_b specified for the basic structural design gross weight multiplied by the ratio of the basic structural design gross weight to the maximum alternate design gross weight, $n_a = n_b(W_{g_b}/W_{g_a})$. These load factors *shall* not be less than 2.0. (Load factor n is the ratio of a given load to the weight with which it is associated.) The maximum alternate design gross weight *shall* be used for landing and ground-handling conditions to the extent specified.

In the development of design loads for conditions for which these design gross weights are applicable, the range of center of gravity (CG) locations resulting from all practicable arrangements of variable and removable items for which provision is required *shall* be considered. Gross weights to be used for design *shall* be all critical gross weights between the minimum and the maximum alternate design gross weights.

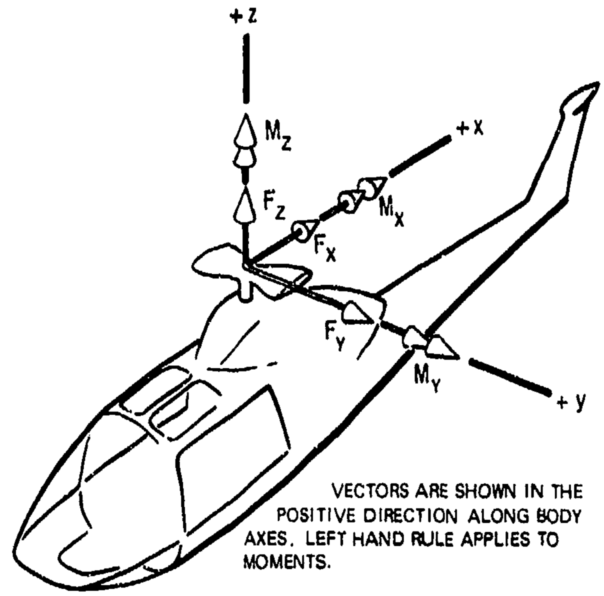


Fig. 4-1. Positive Sign and Vector Conventions for Forces Acting on the Helicopter

4-4 FLIGHT AND TAKEOFF LOADING CONDITIONS

4-4.1 FLIGHT ENVELOPES AND MISSION PROFILES

Forces and moments acting upon the helicopter during flight can be represented by forces and moments acting along and about three mutually perpendicular body axes. Sign and vector conventions for the forces and moments acting along these principal axes are as depicted in Fig. 4-1. Note that this coordinate system is the opposite of that used to define the motions of the helicopter (Chapter 6).

At any instant in time, all aerodynamic and inertia forces *shall* be in equilibrium, and the following conditions *shall* be satisfied:

1. Summation of forces along each of three mutually perpendicular reference axes equals zero
2. Summation of moments about each of three mutually perpendicular reference axes equals zero.

Rotor thrust *shall* be in equilibrium with the aerodynamic and inertia forces acting upon the aircraft. For the case of trimmed 1-g flight, the rotor thrust T is equal to the resultant of the weight W_g and the aerodynamic drag D, and has a line of action in opposition to the resultant F_R of these two forces, as shown in Fig.

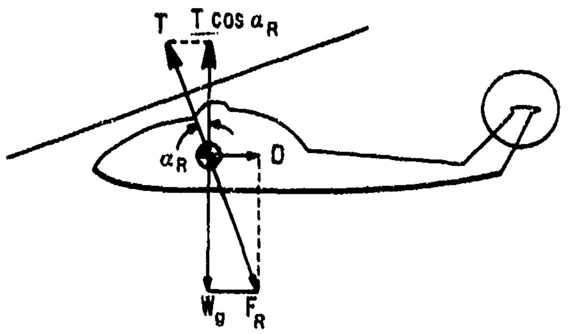


Fig. 4-2. Resolution of Weight and Drag in Trimmed Flight

4-2. (For simplicity, the drag force is shown as acting through the CG of the helicopter.)

For flight conditions in which there is no angular acceleration, inertia and gravity forces *shall* be distributed in the same manner as weights, with their resultant acting through the aircraft CG. Inertia and gravity forces, for convenience in analysis, are combined as the product of a normal load factor n_z and the gross weight W_g

$$F_z = n_z W_g \quad \text{, lb} \qquad (4\text{-}1)$$

where

F_z = normal force component, lb

The equation defining the relationship between load factor n_z and linear acceleration a_z for zero pitch attitude is:

$$n_z = 1 + \frac{a_z}{g} \text{ , dimensionless} \qquad (4\text{-}2)$$

where

g = acceleration due to gravity, 32.2 ft/sec^2

Additional forces and moments are created during those flight conditions in which the helicopter is being maneuvered by the pilot, or in which external forces such as gusts cause the aircraft to be accelerated in either a linear or an angular fashion. These forces and moments produce linear and angular accelerations that result in balancing inertia forces and moments. The aircraft, when considered as a free body, remains in equilibrium.

As an example of the stated principles, fore or aft movement of the control stick changes the direction of thrust of the rotor, resulting in an unbalanced moment about the y-axis. This moment causes the aircraft to pitch about the y-axis, and the resulting angular acceleration when multiplied by inertia results in a moment counterbalancing the applied moment, as shown in Fig. 4-3.

For angular accelerations, a system similar to that for linear accelerations is used, except that the force F_z acting upon any element in the aircraft is also a function of the distance x' of that element from the CG, and is defined by the equation

$$F_z = w\left(n_z + \frac{x'\ddot{\theta}}{g}\right) \text{ , lb} \qquad (4\text{-}3)$$

where

w = weight of element, lb

$\ddot{\theta}$ = angular acceleration, rad/sec^2

Any unbalance between the rotor thrust and the aircraft gross weight will cause a corresponding acceleration in the vertical direction a_z and load factor n_z (Eq. 4-2) such that

$$T \cos \alpha_R = n_z W_g \quad \text{, lb} \qquad (4\text{-}4)$$

where

T = rotor thrust, lb

α_R = angle between thrust vector and vertical, rad

The same rationale can be applied to any flight condition, e.g., a rolling or yawing maneuver.

The primary modification for a compound helicopter from a structural standpoint is that force-generating systems other than the rotor are introduced. These additional systems are generally wings, either with or without auxiliary lift and control devices such as flaps

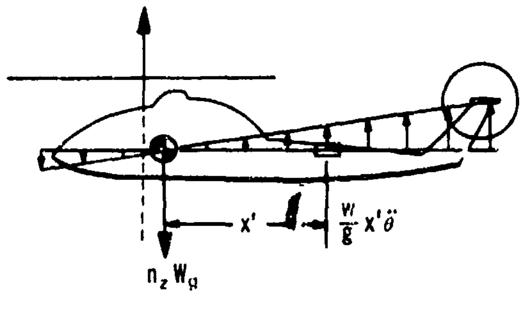


Fig. 4-3. Moments Resulting From Control Stick Movement

and ailerons, and thrust producers such as propellers or auxiliary jet engines.

4-4.1.1 Basic Flight Loading Conditions

During its life cycle a helicopter is subjected to a wide variety of loads, many for short durations of time. Some loads are generated by pilot input and some originate from natural sources such as gusts and rough landing surfaces. A helicopter lives in a severe fatigue environment because of its large complement of rotating machinery and its rotor that constantly generates a cyclic load input. The dynamic components—including drive shafts, transmissions, rotor hubs, and blades—are sized primarily from a consideration of fatigue strength. Although many portions of the airframe are designed from the standpoint of ultimate strength, a thorough fatigue investigation is necessary in those areas most affected by dynamic or oscillatory loads arising from sources such as rotors or weapons.

Critical flight loading conditions normally considered in the design of a pure helicopter are defined in MIL-S-8698. These conditions are:

1. Maximum speed (design limit speed V_{DL})
2. Symmetrical dive and pullout at design limit speed V_{DL} and at 0.6 V_H, approximately the speed of maximum load factor capability
3. Vertical takeoff (jump takeoff)
4. Rolling pullout
5. Yaw
6. Autorotational maneuvers.

These flight conditions must be examined at maximum and minimum design rotor speeds. In addition, a pushover maneuver to the minimum design load factor, which usually is 0.0, *shall* be investigated.

Design of a compound helicopter requires investigation of conditions other than those defined in MIL-S-8698. Other conditions to be examined will be associated with the lift or propulsion forces generated by the wing stabilizer or auxiliary lift and propulsion devices, and will, in general, follow classical aircraft design methods. Flight conditions requiring investigation shall be applicable loads defined in MIL-A-8860, MIL-A-8861, MIL-A-8865, and MIL-A-8866. Applicability will be determined by the individual aircraft configuration.

Considerations other than those defined in these Military Specifications may be necessary for particular situations or specialized aircraft missions in order to insure adequate strength in the aircraft.

Load factor capability can be derived directly in terms of rotor thrust by the method contained in Ref. 1. This method defines load factor capability of a rotor in terms of the maximum mean blade lift coefficient $\bar{C}_{L_{max}}$. The expression for load factor capability $n_{z_{max}}$ is:

$$n_{z_{max}} = \frac{\bar{C}_{L_{max}}}{\bar{C}_{L_t}} \left(\frac{B^3 + \frac{3}{2}B\mu_N^2 - \frac{4}{3}\mu_N^3}{B^3 + \frac{3}{2}B\mu_t^2 - \frac{4}{3}\mu_t^3} \right) \times \left(\frac{\Omega_N}{\Omega_t} \right)^2 \left(\frac{\cos a_{0_N}}{\cos a_{0_t}} \right)^3 \quad (4\text{-}5)$$

where

a_0 = rotor coning angle, rad or deg
B = rotor blade tip loss factor, dimensionless
μ = advance ratio, $(V \cos \alpha_R)/(\Omega R)$, dimensionless
Ω = rotor speed, rad/sec
V = forward flight speed, fps
R = rotor radius, ft
α_R = rotor angle of attack, rad or deg

and subscripts

t = trimmed 1-g flight
N = value at maximum load factor, $n_{z_{max}}$

The formula is conservative because the maximum mean rotor lift coefficient is an ideal theoretical value, usually not achievable with actual rotor blades. Modifications can be made to account for factors such as blades tapering in thickness and blade root cutout areas.

Lift capability of auxiliary lifting devices—such as wings, horizontal tails, and ailerons—is determined by classical methods defined in texts on aerodynamics and loads. Airloads theoretically should be based upon the maximum wing lift force coefficient representing the highest angle of attack at stall for which the wing is analyzed, but never should be less than those corresponding to the design limit load factor. The force coefficient should be multiplied by an appropriate factor to compensate for transient lift forces above those actually achievable by the wing at steady-state stall. A factor of 1.25 is recommended to cover momentary loads during gusts and sudden maneuvers. An alternate factor may be substituted when dynamic airfoil characteristics are available and can be used in the derivation.

For compound helicopters and other aircraft configured with both wings and rotors, a rational combination of the lift capabilities of both devices is required. The present state-of-the-art is such that only a small amount of data is available on compound helicopters. The effective lift capabilities and the interactions between wings and rotors are not well defined.

It is conservative to add the lift capability of the wings directly to that of the rotor, neglecting any lag in response time between the rotor and the wings. This approach would be conservative for any type of control system used in a compound helicopter—from an aircraft that is flown purely in the helicopter mode without auxiliary controls such as ailerons, elevators, and rudders to an aircraft that has a wide array of aerodynamic control devices on wings and empennage. Time lag in response between forces generated by the wing and those generated by the rotor will vary widely, depending upon configuration. The rotor response, in general, will tend to lead the wing response when no auxiliary control devices are installed on the wing because wing lift is dependent upon a change in angle of attack of the entire aircraft. This criterion for structural design of compound helicopters is treated further in par. 4-4.2.

Because of the uncertainties of estimating interaction among body, rotor, and wing, it is difficult to compute the drag forces acting upon the aircraft accurately by analytical methods. As a result, preliminary design calculations are best developed from aerodynamic drag data obtained in the wind tunnel. Values of lift, drag, and pitching moment for all attitudes and angles of attack can be obtained for the complete aircraft, vehicle-less-rotor, vehicle-less-wing, and vehicle-less-wing-and-rotor. These data can be used directly in calculating basic air loads for structural design purposes. Verification of wind tunnel model data can be obtained later on full-scale aircraft during the flight test program.

4-4.1.2 Mission Profile and Fatigue Analysis

Because the rotor system and dynamic components of a helicopter are sized principally from fatigue considerations, fatigue analysis is a basic requirement of helicopter design. The design *shall* insure that the helicopter is capable of achieving the operational design flight requirements, is safe to operate, is free from fatigue failure throughout its design life, and has adequate reliability with a minimum of maintenance. At the same time, it is important to avoid penalizing the aircraft with excessive weight so that it cannot perform its mission efficiently.

Fundamental to achieving these aims is a clear definition of the missions the aircraft is to perform. Because most helicopters will be used for a number of different purposes, it is necessary to define a variety of missions that a specific helicopter will perform during its design life. An estimate of the percentage of useful life that will be spent performing each mission also must be made.

A large number of variable parameters enters into the determination of fatigue life. In addition to defining aircraft missions in quantitative terms, it is necessary to make estimates of a large number of factors and to take these into consideration during generation of a fatigue loading spectrum. Some of the more important of these variables are:

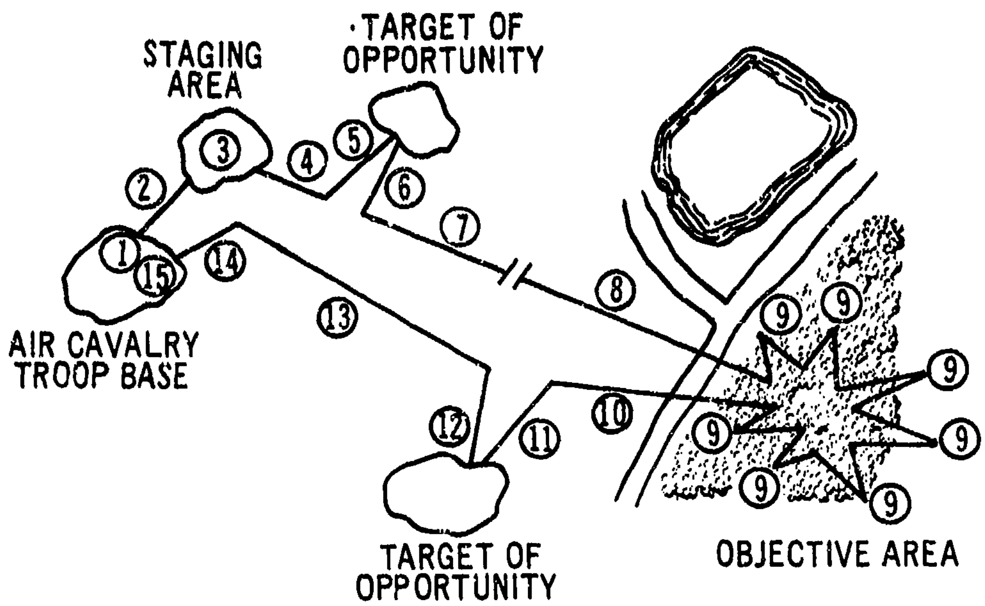
1. Aircraft weight at takeoff, during mission, and at landing
2. Flight altitudes
3. Number of takeoffs and landings per mission
4. Loads due to external stores and/or cargo slings
5. Airspeeds
6. CG range
7. Rotor speeds
8. Gust effects
9. Magnitude, number, and duration of pilot-initiated load factors during maneuvers
10. Sinking speed during landing
11. Autorotation.

Two typical mission profiles for an armed helicopter are shown in Figs. 4-4 and 4-5. Such mission scenarios will be defined by the procuring activity and used by the aircraft developer as the basis for fatigue analysis. Information contained in the mission profiles will be expanded into a spectrum of flight and ground conditions as shown in Table 4-1.

Time for each event and the number and duration of events occurring during the aircraft life are determined from engineering estimates based upon the class of helicopter, field experience with that class of aircraft, and estimated performance of the vehicle itself. These estimates normally are made by the aircraft designer and approved by the procuring activity. Percentage of flight time spent in each flight condition then can be computed from the equation in Note 2 of Table 4-1.

Inherent in the profile is a definition of aircraft life, expressed in flight hours. This value usually will be defined by the procuring activity prior to initiation of aircraft development.

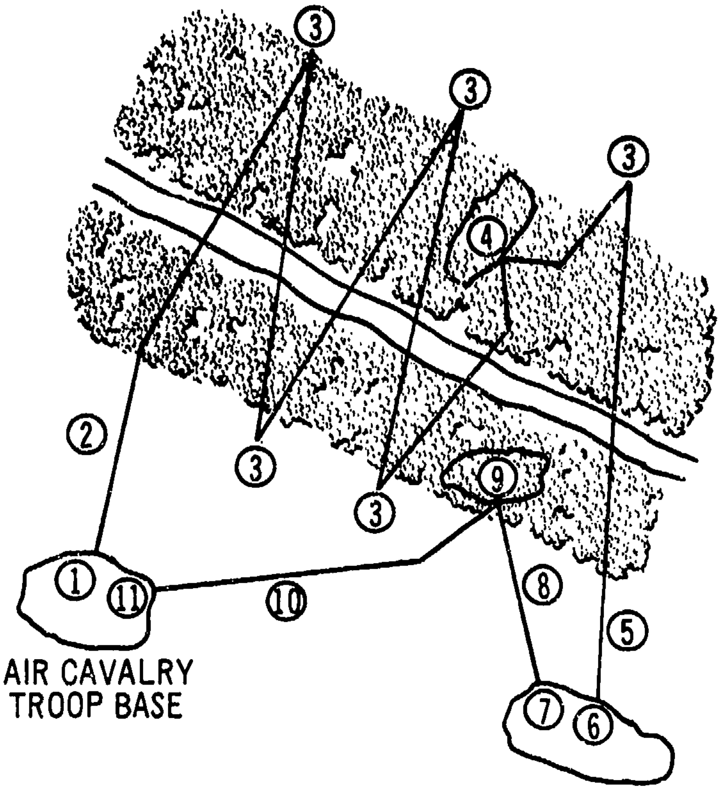
Because a helicopter is flown at a variety of airspeeds, weight and CG distributions, rotor speeds, and altitudes, the flight condition spectrum must be expanded further, as indicated in Table 4-1, to consider



AVERAGE MISSION DURATION	45 min
PERCENTAGE OF TOTAL SERVICE TIME	30%
AVERAGE WEIGHT AT TAKEOFF	14,000 lb
AVERAGE WEIGHT AT MISSION COMPLETION	11,000 lb
TYPICAL FLIGHT ALTITUDE	500 ft
AVERAGE NUMBER OF TAKEOFFS AND LANDINGS	1.0

PHASE	% MISSION TIME	VELOCITY RANGE, kt
1 TAKEOFF AND HOVER	2.0	0 – 40
2 FLIGHT TO STAGING AREA	6.0	40 – 150
3 RENDEZVOUS	10.0	0 – 50
4 ESCORT AT CRUISE SPEED	10.0	100 – 150
5 INVESTIGATE AND NEUTRALIZE GROUND TARGET OF OPPORTUNITY	5.0	150 – 200
6 DASH TO RETURN TO ESCORT	3.0	180 – 200
7 ESCORT AT CRUISE SPEED	10.0	100 – 150
8 DASH TO SECURE OBJECTIVE AREA	5.0	180 – 200
9 SECURE OBJECTIVE AREA, PROTECT LANDING AND SUPPORT LANDED TROOPS	14.0	40 – 150
10 ESCORT AT CRUISE SPEED	10.0	100 – 150
11 INVESTIGATE AND NEUTRALIZE GROUND TARGET OF OPPORTUNITY	5.0	150 – 200
12 DASH TO RETURN TO ESCORT	3.0	180 – 200
13 ESCORT AT CRUISE SPEED	10.0	100 – 150
14 RETURN TO BASE	5.0	40 – 150
15 HOVER AND LANDING	2.0	0 – 40

Fig. 4-4. Escort Mission Profile



AVERAGE MISSION DURATION	60 min
PERCENTAGE OF TOTAL SERVICE TIME	25%
AVERAGE WEIGHT AT TAKEOFF: INITIAL	14,000 lb
SECOND	12,000 lb
AVERAGE WEIGHT AT MISSION COMPLETION	10,000 lb
TYPICAL FLIGHT ALTITUDE	1,000 ft
AVERAGE NUMBER OF TAKEOFFS AND LANDINGS	2.0

PHASE	% MISSION TIME	VELOCITY RANGE, kt
1 TAKEOFF AND HOVER	2.0	0 - 40
2 FLIGHT TO RECONNAISSANCE AREA AT CRUISE SPEED	10.0	40 - 150
3 RECONNAISSANCE	35.0	40 - 150
4 INVESTIGATE AND NEUTRALIZE GROUND TARGET OF OPPORTUNITY	6.0	150 - 200
5 FLIGHT TO GROUND TROOP HEADQUARTERS AT CRUISE SPEED	10.0	40 - 150
6 LANDING AND HOVER	2.0	0 - 40
7 TAKEOFF AND HOVER	2.0	0 - 40
8 FLIGHT TO TARGET AT CRUISE SPEED	10.0	40 - 150
9 ENGAGE AND NEUTRALIZE TARGET	6.0	40 - 150
10 RETURN TO BASE AT CRUISE SPEED	15.0	40 - 150
11 HOVER AND LAND	2.0	0 - 40

Fig. 4-5. Reconnaissance Mission Profile

TABLE 4-1
FLIGHT CONDITION SPECTRUM FOR FATIGUE ANALYSIS

FLIGHT CATEGORY	FLIGHT CONDITION	TIME PER EVENT	NO. OF EVENTS DURING AIRCRAFT LIFE	% FLIGHT TIME	EXPANSION PARAMETERS: WEIGHT, CG DISTRIBUTION	EXPANSION PARAMETERS: ROTOR SPEED DISTRIBUTION	EXPANSION PARAMETERS: ALTITUDE DISTRIBUTION
STEADY FLIGHT, POWER ON	HOVER AND AIR TAXI	NOTE 1	NOTE 1	NOTE 2			
	LOW SPEED CRUISE						
	MEDIUM SPEED CRUISE						
	HIGH SPEED CRUISE						
	MAX LEVEL FLIGHT SPEED						
	DIVE SPEED						
TRANSITION	LOW SPEED TRANSITION	NOTE 1	NOTE 1	NOTE 2			
	HIGH SPEED TRANSITION						
	TRANSITION TO AUTOROTATION						
	TRANSITION FROM AUTOROTATION						
MANEUVERS POWER ON	CYLIC AND COLLECTIVE PULL-UPS	NOTE 1	NOTE 1	NOTE 2			
	ROLLING PULL-OUT						
	RIGHT TURNS						
	LEFT TURNS						
	STEEP CLIMBING TURNS RIGHT						
	STEEP CLIMBING TURNS LEFT						
	TURN-ON-THE-SPOT						
	DYNAMIC YAW (RUDDER REVERSALS)						
	LONGITUDINAL CONTROL REVERSAL						
	VERTICAL CONTROL REVERSAL						
	QUICK-STOP						
	JUMP TAKEOFF						
	FLARE						
POWER OFF	AUTOROTATION STEADY DESCENT	NOTE 1	NOTE 1	NOTE 2			
	AUTOROTATION RIGHT TURN						
	AUTOROTATION LEFT TURN						
	AUTOROTATION PULL-UP						
	AUTOROTATION RUDDER REVERSAL						
	AUTOROTATION LATERAL CONTROL REVERSAL						
	AUTOROTATION VERTICAL CONTROL REVERSAL						
	AUTOROTATION FLARE						

NOTES:
1. VALUES DETERMINED BY ENGINEERING ANALYSIS OF MISSION PROFILES
2. % FLIGHT TIME = TIME PER EVENT X NUMBER OF EVENTS DURING AIRCRAFT LIFE X 100

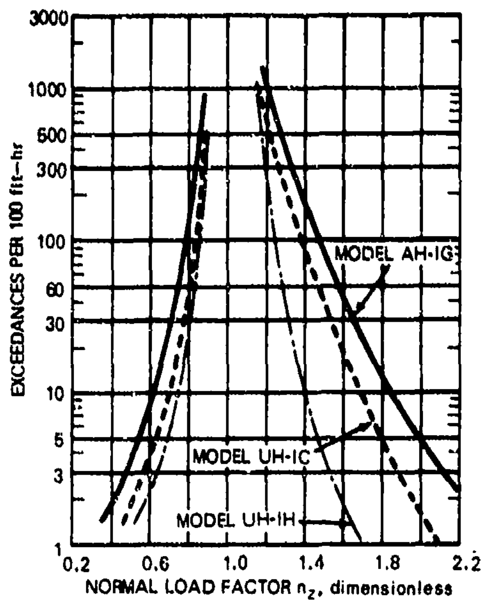


Fig. 4-6. Frequency of Occurrence of Load Factors (Ref. 2)

these parameters because fatigue life is affected greatly by each.

For armed helicopters, estimates of the amount of firing of guns and other weapons also are required in order that fatigue lives of structural components affected by these loads can be determined.

In addition to the spectrum of flight conditions, severity distributions of load factors due to pilot-initiated maneuvers and gust encounters are required. Fig. 4-6 (Ref. 2) is representative of the severity of load factors due to maneuvers and gusts during the life of an attack helicopter and a utility helicopter, while Fig. 4-7 (Ref. 3) represents such load factors for a fixed-wing aircraft. Fig. 4-7 presents cumulative frequency of occurrence, which is the probability of occurrence of a given or greater gust velocity.

Data presented in Fig. 4-6 are suitable for use during preliminary design and analysis of attack and utility helicopters, while Fig. 4-7 is suitable for all helicopter classes (pars. 4-4.2 and 4-4.3). Fatigue damage due to landings and ground conditions (pars. 4-5 and 4-6) also must be included in the computation of fatigue life of the helicopter. Final determination of fatigue life of the airframe and dynamic components (par. 4-11) can be made only after laboratory fatigue testing of specimens and determination of the magnitude of loads during the flight load survey.

4-4.2 LIMIT LOAD FACTORS

4-4.2.1 Symmetrical Flight

The nature of the forces and moments acting upon a helicopter is discussed in par. 4-4.1 and a discussion of load factors as they relate to the rotor hub is included in par. 4-10.2. A clear concept of limit loads is an important factor in the efficient design and safe operation of a helicopter. The discussion that follows provides more information on limit load factors as they apply to various types of maneuvers and flight conditions.

The vertical accelerations that establish the maneuver limit loads can be related to the maximum thrust capability of the rotor. The maximum load factor is the ratio of the maximum possible thrust that can be developed by the rotor to the gross weight. The maximum thrust is determined by assuming the maximum lift coefficient at all blade sections. This, according to Ref. 4, may be represented by:

$$\left(\frac{C_T}{\sigma}\right)_{max} = \frac{T_{max}}{\rho b c_e R(\Omega R)^2} \qquad (4\text{-}6)$$

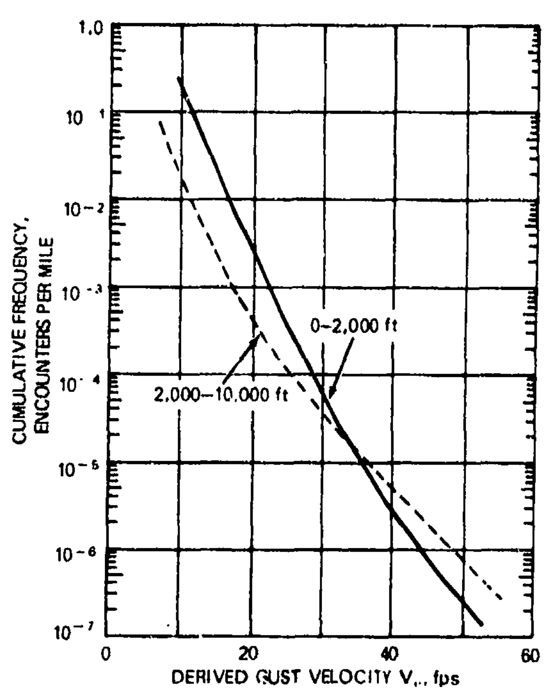


Fig. 4-7. Derived Gust Velocity Encounter Distribution (Ref. 3)

where

C_T = thrust coefficient, $T/[\rho\pi R^2(\Omega R)^2]$, dimensionless
ΩR = rotor tip speed, fps
ρ = air density, slug/ft^3
σ = rotor solidity, $bc_e/(\pi R)$, dimensionless
b = number of blades
c_e = blade chord (effective), ft

The lift coefficient for the basic structural design gross weight, in the same manner, may be represented by:

$$\left(\frac{C_T}{\sigma}\right)_{design} = \frac{T_{design}}{\rho b c_e R(\Omega R)^2} \quad (4\text{-}7)$$

The maximum possible load factor $n_{z_{max}}$ or maximum g load obtainable at basic structural design gross weight then would be obtained from Eqs. 4-6 and 4-7 as an alternate to Eq. 4-5:

$$n_{z_{max}} = \frac{(C_T/\sigma)_{max}}{(C_T/\sigma)_{design}} \quad (4\text{-}8)$$

This maximum attainable load factor can be computed for a given rotor system and vehicle combination. However, MIL-S-8698 has established limit load factors to be used in the design and qualification of three different classes of helicopters. They are shown in Table 4-2. The requirements of one of these classes probably will be applicable to any new helicopter design. On the other hand, the Request for Proposal (RFP) may establish requirements that are different. In addition, more up-to-date test data on similar vehicles may be available that justifies different limit load factors. These data also should be considered.

The determination of realistic load factors has been a continuing effort by industry and various Governmental agencies. Structural design criteria that are peculiar to the varied missions performed by Army helicopters are necessary to assure procurement of vehicles that will provide satisfactory service throughout their useful lives. A substantial amount of flight load data now has been acquired on Army helicopters in operational environments. These data should be reviewed as a part of the preliminary design effort on a new helicopter.

TABLE 4-2
HELICOPTER DESIGN LIMIT LOAD FACTORS

	LIMIT LOAD FACTORS	MISSION CLASSIFICATION
CLASS I	+3.5, -0.5	AH, OH, TH, UH
CLASS II	+3.0, -0.5	CH, PAYLOAD < 5000 lb
CLASS III	+2.5, -0.5	CH, PAYLOAD > 5000 lb

Ref. 1 summarizes some early flight test data on load factors obtained in maneuvers. The maximum load factor measured in the test program covered by the report was 2.68 g obtained by a combined cyclic and collective pullup during autorotation at 50 mph with motion of the collective control delayed approximately 1.5 sec. Other high load factors obtained during the tests were:

1. Cyclic pitch pullup, level flight, 85 mph; 2.38 g
2. Cyclic pitch pullup, autorotation, 80 mph; 2.55 g
3. Cyclic pitch pullup, 5-deg dive in autorotation, 50 mph; 2.52 g.

Fig. 12b of Ref. 5 summarizes data obtained on turbine-powered helicopters with gross weights less than 10,000 lb. These data showed a maximum maneuver load factor of 2.5 g and fewer than 50 accumulated occurrences above 2.0 g for each 1000 hr in operation.

Ref. 6 summarizes load factor data obtained on three light observation helicopters and two large load-lifting helicopters. Maximum maneuver loads for the light helicopters generally did not exceed 2.5 g, and the number of occurrences above 2.0 g also was fewer than 50 per 1000 hr. The maximum maneuver load factors for the heavy-lift helicopters were all below 2.0 g. Similar data for AH and UH helicopters are shown in Fig. 4-6 (Ref. 2).

In contrast to the case of positive acceleration, the limitations on negative load factor provided by reaching $\overline{C_{L_{max}}}$ are of little practical interest. Control moments are reduced during negative accelerations and all data show that -0.5 g is an adequate negative acceleration for consideration in design. Inherent blade motion limitation is one of the factors affecting the magnitude of negative acceleration. The most common sources for negative load factors are gust loads (par. 4-4.3).

4-4.2.1.1 Control of Limit Load Factors

As flight speed increases, a given rotor angle-of-attack change produces a larger thrust increment so that large load factors may be reached without large attitude changes. The maximum loads thus may be obtained by variation in rate of control application, magnitude of control movement, and airspeed.

Artificial limitation of load factors may be accomplished by:

1. Rotor speed regulation
2. Dampers in controls
3. Force gradients in controls.

Except for rotor speed regulation, these artificial methods of limitation have not been popular for helicopter design and operation.

Limit load factors for configurations such as compounds and helicopters with unconventional rotors must be established on an individual basis. However, the general approach for these unconventional vehicles is the same in that the load factors used must be justified thoroughly by test and analytical data.

4-4.2.1.2 V-n Diagram

Typical V-n diagrams are shown in Fig. 4-8. The load factors shown are the limit load factors for hypothetical helicopter designs. The upper limit of 3.5 g and the lower limit of −0.5 g in the low airspeed and normal airspeed region were established by MIL-S-8698. The limits at high airspeed were determined by rotor blade stall and blade tip Mach limits. The load factor n_z shown in this diagram is:

$$n_z = \cos\theta \pm \frac{a_z}{g} \tag{4-9}$$

where

θ = pitch angle of helicopter body axis, deg

4-4.2.1.3 Maneuvers (Symmetrical Flight)

Various maneuvers that are classified as symmetrical flight maneuvers include the following:

1. Hover. Hover is normally at or near zero airspeed and the load factor is one.

2. Takeoff and climb. Vertical takeoff and vertical climb are inherent capabilities of the helicopter. A jump takeoff maneuver consists of rapid application of collective pitch on the ground while the rotor is turning at maximum power-on rotational speed. The maximum vertical load factor from this maneuver seldom exceeds 1.6. A maximum rate of climb from takeoff is a demonstration maneuver and results in relatively modest accelerations and associated load factors of 1.5 and less.

3. Level flight. The resultant thrust load during level flight is reacted by the weight of the vehicle, the inertial load due to longitudinal acceleration, and the drag load from airspeed. As airspeed and longitudinal acceleration increase, the longitudinal load increases accordingly, causing an increase in the required thrust. A diagram of these level flight forces is shown in Fig. 4-9. Higher load factors are more likely in the high-speed regime of level flight because the relatively large thrust increment from the rotor facilitates large load factors with modest flight attitude changes. Larger control capability, however, is available in the airspeed range around 0.6 V_H.

4. One-g dive. A 1-g dive maneuver consists of operating the helicopter in a dive attitude with rotor thrust approximately equal to gross weight (load factor = 1.0). This maneuver often is used (per MIL-S-8698) to demonstrate maximum speed followed by maximum load factor during the pullup from the dive.

5. Pullup. The pullup demonstrates maximum load factor for symmetrical flight. The design conditions of MIL-S-8698 are at airspeeds of V_{DL} and 0.6 V_H for maximum rotor speed and minimum rotor speed. Fig. 4-10 shows a diagram of forces encountered in the maneuver. The loads originate from gravity, aerodynamic pitching moment, inertia, and the centrifugal force caused by the angular velocity of the helicopter following an arc. Longitudinal loads also are encountered at the rotor hub point of attachment because of the force associated with the moment and because of the drag and the longitudinal component of the gravity load. These longitudinal loads must be considered for this maneuver as well as for other maneuvers to which they are applicable. Lateral loads also are applicable in symmetrical flight maneuvers because of lateral movements, directional controls, and tail rotor thrust. Even though these longitudinal and lateral loads have their associated limit load factors, they are not discussed in as much detail because their magnitudes are small compared with the vertical thrust loads and longitudinal moments.

4-4.2.1.4 Rotor Speed and Power Ranges

There is a significant difference in load factor consideration between power-on and power-off operations. The allowable range of power-on rotor speed is relatively small, being limited on the high side by engine limits and on the low side by helicopter rotor blade stall and associated vibration and comfort levels. Power-off operation, on the other hand, has a comparatively large allowable range of rotor speeds. The maximum allowable rotor speed in autorotation is considerably higher than the maximum speed allowable power-on.

This maximum power-off rotor speed limit is essentially the design limit of the rotor (with some reduction for a factor of safety). The minimum power-off rotor rpm may be set at the lowest level from which, as

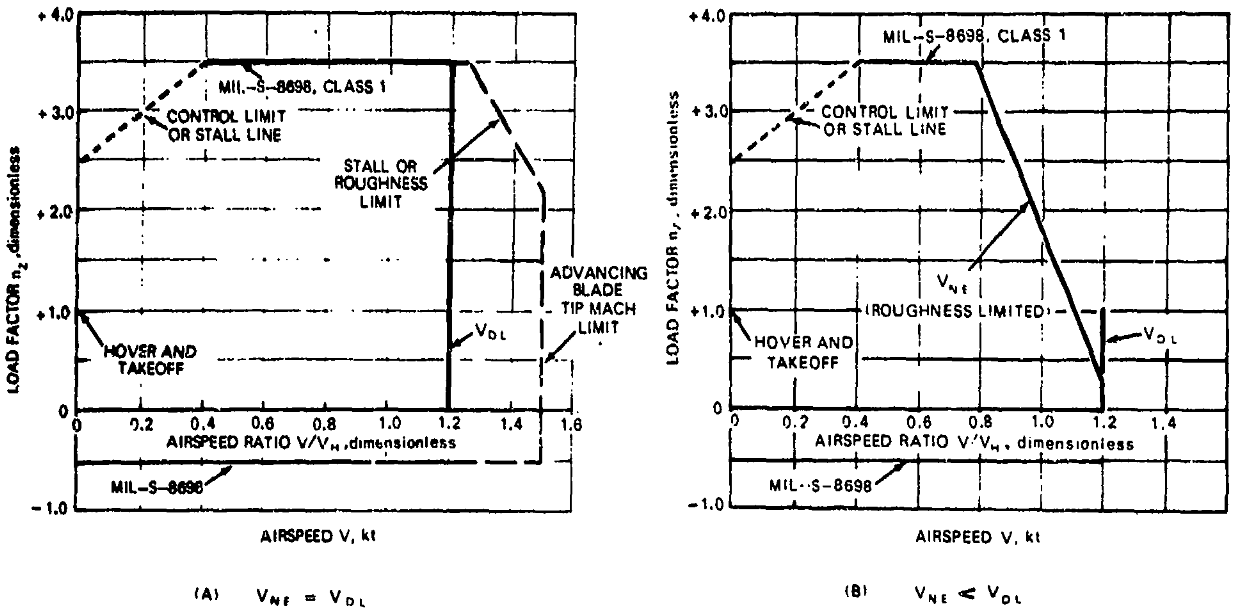


Fig. 4-8. Typical V-n Diagrams Showing Flight Maneuvers

shown by flight test experience, rotor speed recovery can be accomplished in a safe length of time. As shown in Eq. 4-6, the maximum thrust capability T_{max} of the rotor varies directly as $(\Omega R)^2$. The maximum possible thrust capability of the rotor is, therefore, at the maximum rotor speed, which is a power-off condition.

This relationship, which ultimately results in the determination of the maximum possible limit load factor, is illustrated in Fig. 4-11, which is a graph of maximum possible vertical load factor versus rotor speed for a hypothetical rotor system. This graph shows that the power-on operating range is below the maximum normal load capability of the rotor. Thus, demonstration maneuvers involving load factors should be conducted under partial power and autorotational flight conditions where the rotor can be operated at a higher speed.

4-4.2.1.5 Load Factors for Other Than Normal Gross Weights

If a helicopter is operated at a weight greater or less than the normal design gross weight, it must be assumed that a different maneuvering load margin is available. For this reason it is customary to specify a higher-than-normal gross weight—normally called an alternate design gross weight (par. 4-3)—in the detail specification. The load factor for this configuration should not be less than 2.0 (MIL-S-8698).

The configuration that involves a lighter-than-normal gross weight also should be considered. Certain missions may call for off-loading a portion of the fuel and cargo. This creates a condition where the relationship between rotor thrust capability and actual gross weight results in a higher limit load factor. The principal requirement for application of this higher load factor would be for certain equipment items.

4-4.2.1.6 Fatigue Analysis

The effect of a particular maneuver on fatigue life must be determined by inflight measurement of the stresses on the components for which the fatigue life information is needed. Certain components may accumulate damaging fatigue cycles during high-speed flight with a relatively modest load factor increase. On the other hand, higher load factors in other flight regimes may result in no fatigue damage. It is essential, therefore, that information on both the load factor and the resulting stress is known.

In addition, it is important to have an accurate measurement or estimate of the maneuver time spectrum. A typical maneuver time history showing load factor versus time is illustrated in Fig. 4-12. This type of maneuver time history may be used to determine the approximate number of load cycles at each load factor for the particular flight condition. The number of such maneuvers in a given time can be estimated from data such as that in Ref. 6. A total cycle count for load factors and associated stresses then can be accumulated to establish the effect of the maneuver upon the fatigue life of a component.

This procedure would be repeated for all other maneuvers that have been determined to be critical for fatigue life of components. The preliminary design task should include definition of the possible maneuver spectrum for the helicopter being designed. It is not possible to make a definitive fatigue analysis without flight test data from the actual helicopter, but background data may be used in a preliminary analysis that is of value and may assist in achievement of a near-optimum basic design.

4-4.2.1.7 Static Analysis

The static analysis of the helicopter should be based upon one or several critical maneuvers that result in the maximum possible limit load factors. These maneuvers also should be included in the structural demonstration to verify the maximum limit load capability or design of the rotor and helicopter system.

4-4.2.2 Asymmetrical Flight

Combined load factors (vertical, lateral, and longitudinal) and simultaneous rotational accelerations combine to establish the maximum load factors during asymmetrical flight operations. Power-on, power-off, and rotor speed variations also are applicable to asymmetrical flight maneuvers (par. 4-4.2.1.4). Asymmetrical flight maneuvers include rolling pullups, sideslips and yaw, and sideward flight.

A rolling pullup maneuver consists of a pullup while the helicopter is in a rolling flight attitude. The roll or bank attitude results in a load factor $n_z = 1/\cos\phi$, where ϕ = bank angle measured between the rotor shaft centerline and vertical. This load factor then is increased in proportion to the pullup and turn rates with the additional normal loads applied as shown in

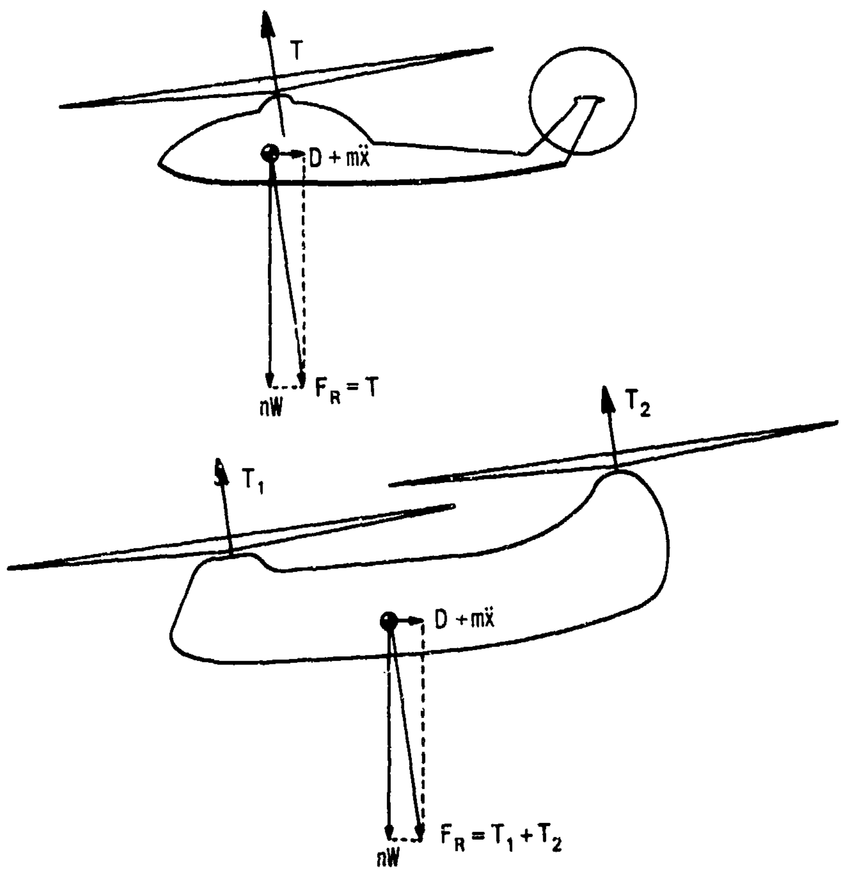


Fig. 4-9. Level Flight Forces

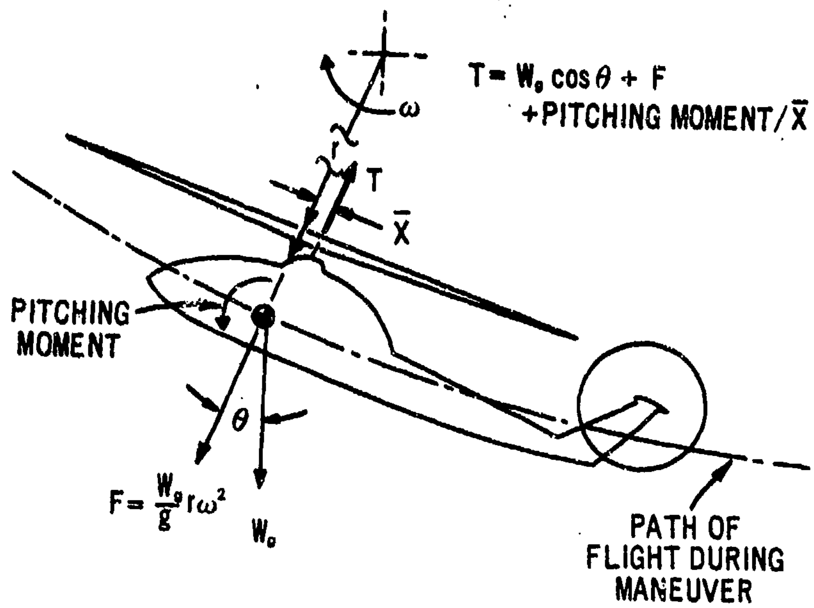


Fig. 4-10. Pullup Maneuver

Fig. 4-10. The primary load factors encountered in the rolling pullup maneuver are in the normal vertical direction. However, the longitudinal and lateral loads also are significant, as is the lateral moment required for the roll rate, and all *shall* be considered. The rolling pullup maneuver is one of the principal design maneuvers of MIL-S-8698.

The sideslips, yaw, and sideward flight maneuvers provide the highest lateral load factors. Limit load factors for the tail boom and directional control system also are established by these maneuvers.

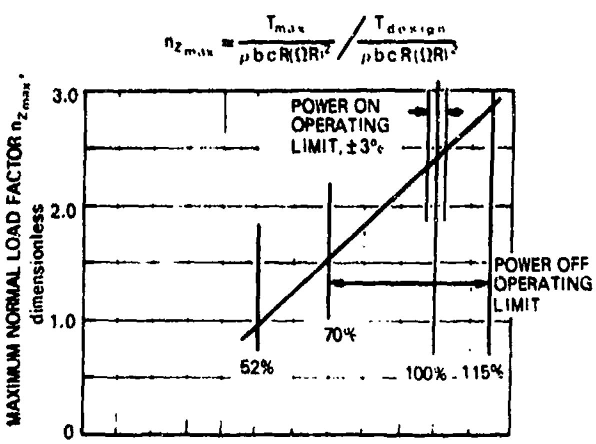


Fig. 4-11. Maximum Load Factor vs Rotor Speed (Hypothetical But Typical Rotor)

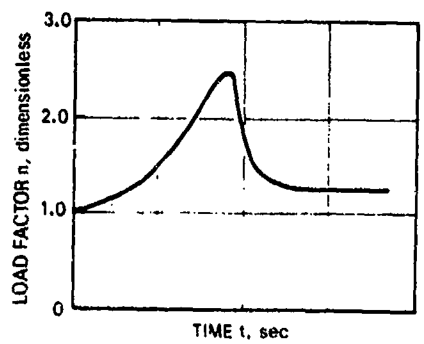


Fig. 4-12. Typical Maneuver—Time Spectrum

4-4.3 GUSTS

It has been established that the reaction of a helicopter to gusts is less severe than is the reaction of a fixed-wing aircraft. Ref. 7 summarizes test results from operating a helicopter and a fixed-wing aircraft side-by-side and shows much larger load factors from gusts for the fixed-wing aircraft. Other tests have given the same results.

However, load factors on helicopters due to gusts are not insignificant. An incremental load factor of 0.9 due to a gust was experienced by a commercial helicopter during a test in the Chicago area (Ref. 8).

The trend toward higher airspeeds for helicopters makes gust criteria more significant because gust load factors normally increase with airspeed. A gust load criterion is given in MIL-S-8698, including definition of a gust alleviation factor. This factor, according to the specification, varies as a function of disk loading and becomes equal to unity at 6 psf and above. Studies (Ref. 9), however, have found gust alleviation factors in accordance with this specification to be too conservative and to result in load factors that are excessive. It is apparent that gust load factors should be established carefully. The best approach during the preliminary design stage is to review the available data, especially data on similar vehicles, and to establish and then to justify the gust load factors to be used for the new design.

Because gust load factors have been a relatively important aspect of fixed-wing aircraft design, a large amount of data on gusts has been accumulated. One such example is Ref. 10, which contains information such as gust length versus gust velocity and cumulative frequency of occurrence per mile of flight. Additional data on cumulative frequency of occurrence are given in Fig. 4-7 (Ref. 3).

4-4.3.1 Conditions Requiring Gust Load Factors

The factors that enter into the determination of gust load factors include:

1. RFP and applicable specifications
2. Fatigue considerations
3. Airspeed range
4. Configuration
5. Exposed surfaces.

The RFP and the referenced specifications often establish specific requirements for gust load factor considerations. MIL-S-8698, for example, cites a specific gust velocity to be applied, together with the schedule for gust alleviation factor discussed previously. The gust velocity specified is $50(\sigma^{1/2})$ fps where $\sigma = \rho/\rho_o$, the ratio of the density of air at the altitude under consideration to the density of air at sea level. Therefore, the gust velocity to be considered at sea level is 50 fps. FAR Part 29 specifies a gust velocity of 30 fps for design.

Fatigue considerations require gust load factor data, especially if the critical design loads for the helicopter are set by gust conditions, or if the expected operational environment is one in which gusty air is normal.

The operating airspeed is an important factor in the determination of gust loads. Ref. 7 shows lower gust load factors at lower speeds from both calculated data and test data. This has been confirmed in other tests and investigations (Ref. 9). Fig. 4-13 is a V-n diagram showing the relative load factors due to gusts as a function of airspeed. This is consistent with the gust load criteria for fixed-wing aircraft (MIL-A-8861), where the incremental load factor due to gusts is proportional directly to equivalent airspeed. A comparable relationship is not given in MIL-S-8698.

Rotorcraft other than the pure helicopter, which has freedom for blade flapping, *shall* be given special consideration in regard to gust load factors. Blades stiff or rigid in the flapwise direction, along with auxiliary wings, would tend to make such a rotorcraft behave more like a fixed-wing aircraft. Supporting data on gust effects are mandatory for this kind of aircraft.

Gust load factor data also are required for ground conditions when the rotor is stopped or in transition between the running and stopped positions. The behavior of rotor blades in the transitional period under gusty conditions is an important consideration in the design of the rotor hub. The structural criteria for the up stops and droop stops of the blades often are based upon gust factors. Gust loads on fixed surfaces also *shall* be considered and are a function of the gust velocity, the surface area of exposure, and the effective inertia of the vehicle. The dynamic response of the helicopter to the specified gust loads also *shall* be examined.

4-4.3.2 Gust Influence

The response of a lifting rotor to gusts is difficult to predict analytically, even when using computers, because of the transient nature of the disturbance and the large number of variables involved. A rigorous mathematical analysis should include transient blade flapping, blade flexibility, induced velocity changes, and vertical motion of the vehicle. For preliminary design, however, both computer data and flight data (such as Ref. 9) are useful and should be used to the maximum extent practicable.

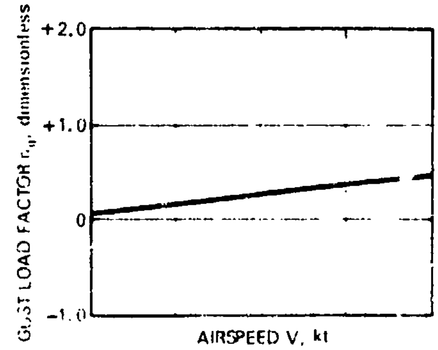


Fig. 4-13. Gust Load Factor vs Helicopter Forward Velocity

Gust velocity causes a change in velocity through the rotor, resulting in a change in rotor thrust. This change in thrust is related primarily to a change in angle of attack of the rotor α_R. The incremental load factor from the thrust change may be calculated by an approach that considers the gust to produce only an angle-of-attack change and that neglects any alleviation factors. This may be expressed in the following manner (Ref. 7):

$$\frac{\Delta n}{U} = \frac{dC_T}{d\alpha_R} \frac{(\Omega R)^2}{V} \rho \frac{\pi R^2}{W_g} \quad (4\text{-}10)$$

where

Δn = load factor increment
U = gust speed, fps
V = flight speed, fps

Because the change in angle of attack is influenced by vehicle inertia, gust wave shape, and other physical conditions, a gust alleviation factor is established by computation or by test data. The most accurate factor is based upon test data from similar vehicles.

When considering gust alleviation factors for helicopters, data on fixed-wing aircraft sometimes have been used for comparative purposes. Helicopters, however, differ from fixed-wing aircraft in that the velocity at the helicopter blade-tip leading edge is higher in relation to the gust velocity than is the velocity at the wing leading edge. Another factor is the flexibility and load attenuation provided by the flapping hinges and flexible blades normally used on helicopters.

Information on the atmospheric gusts pertinent to the helicopter is of fundamental importance to preliminary design. Much of this information is included in Ref. 10, and those data should be used for estimating gust load factors, especially in cases where the alleviation factors are computed.

4-4.3.3 Gust Loads During Maneuvers

The superposition of gust loads and maneuver loads occurs infrequently, and generally the combination of these loads does not exceed the helicopter limit load. The limit maneuver load factor is determined by the maximum capability of the rotor (par. 4-4.2), and exceeding this load factor is not possible even with the addition of a gust load. Maneuvering under gusty air conditions is not normally a helicopter demonstration requirement.

4-4.3.4 Fatigue Loadings

The influence of gust loadings upon the fatigue life of helicopter components depends upon the relative sensitivity of the particular vehicle to gust conditions. Fig. 4-7 shows a cumulative frequency of occurrence per mile of flight versus gust velocity. From this figure, at 2,000 to 10,000 ft, one occurrence of 50 fps or more per 1.3 million mi of flight is indicated. Gusts of 30 fps or higher will occur only once per 27,000 mi. If these data are appropriate for a given helicopter, gust loading would have a small effect upon fatigue life because 30-50 fps gusts normally would result in a load factor less than limit load factor when a reasonable alleviation factor was applied.

4-5 LANDING CONDITIONS

4-5.1 DESIGN LIMIT LANDING REQUIREMENTS

Army operating environments involve unprepared areas and terrain types that must be considered in the selection of a landing gear type and in subsequent design iterations. Consideration of both level and asymmetric landings—with and without forward speed—at varying helicopter sizes, weights, and configurations is essential. The parameters of the landing gear design or configuration sensitive to the anticipated operational factors are likewise of interest.

4-5.1.1 Symmetrical Landings

Symmetrical landings can be accomplished in more than one manner—from hover; during a power-on approach with forward velocity, possibly including drift; fully autorotational with or without forward speed, and possibly with drift; or, in rare cases, under emergency conditions involving abnormal descent velocity. These rare, emergency landings usually are attributable to battle damage, vehicle malfunction, adverse weather or terrain conditions, or pilot error.

Power-on landings normally result in relatively low descent velocities at ground contact, usually not exceeding 5 fps. Fully autorotational landings rarely are performed with large multiengine helicopters but are frequent with the smaller single-engine models, to the point that they *shall* be considered as normal landings. These normal autorotational landings usually do not exceed 6.5 fps descent velocity at ground contact. In battle zone operations, a descent velocity of 8 fps may be considered a normal sinking speed at ground contact. Under emergency conditions, or in unusual situations as noted, descent velocities of 15 fps or more occasionally may be encountered (Ref. 11).

The sinking speeds at ground contact in all of these operating conditions are affected significantly by such vehicle design parameters as autorotational minimum descent rate; vehicle size, configuration, and weight; density altitude (Ref. 12); and many other factors. It may be necessary to increase the minimum design sink rates in order to allow for unusual or abnormal vehicle parameters.

4-5.1.1.1 Design Sinking Velocity

Minimum values for the design limit sinking speeds at ground contact are given by MIL-S-8698 as:

1. Minimum flying weight and basic structural design gross weight—8 fps together with 0.67 W_g rotor lift throughout the impact
2. Alternate design gross weight—6 fps together with 0.67 W_g rotor lift throughout the impact.

Due to the inadequacy of these criteria to account for the severe usage of Army helicopters under combat conditions, the design sink speed *shall* be a minimum of 10 fps in lieu of 8 fps for all new designs. The horizontal speeds with which the design limit sinking speed *shall* be combined *shall* include all values between zero and 120% of the speed corresponding to minimum power required for level flight at the landing gross weight.

Factors to be considered for any particular combination of design configuration and operational environment are:

1. Steady-state descent velocity in autorotation (or with all engines at flight idle for multiengine helicopters)
2. The forward velocity associated with the minimum steady-state autorotational descent velocity
3. Maximum operational density altitude to be designed for as a normal anticipated operating requirement
4. Disk loading
5. Vehicle size and landing gear configuration as indicated by Fig. 4-14, where Δ = effective additional drop height
6. Vehicle low-speed stability and control
7. Pilot location relative to CG or relative to ground contact (Fig. 4-15). As a general rule, the likelihood of higher descent velocities at ground contact increases as the pilot's distance ahead of the CG and ahead of or above the initial ground contact point is increased.
8. The requirement, for certain types of helicopters, that evasive maneuvers be performed immediately before and during the landing flare. The design descent velocities must be increased in such cases to reflect these special tactical requirements.

4-5.1.1.2 Landing Attitude Requirements

Landing conditions that require structural load consideration are the following variations of attitudes and loadings:

1. Tricycle gear, both with and without a 0.25 minimum drag factor on all contacting wheels (Fig. 4-16):
   a. 3-point
   b. Nose first, near level
   c. Aft first, near level[1]
   d. Aft first, maximum nose-up (as limited by tail-bumper)[1]
   e. Tail-bumper or tail wheel first (if applicable)
   f. 50% maximum vertical load for Conditions 1b through 1e applied in any horizontal direction one gear at a time, in combination with the vertical loads. Free-swivelling nose gear *shall* be assumed to line up with the obstruction load (lockable or steerable nose gear *shall* be treated as nonswivelling).
   g. Conditions 1a through 1f with spin-up and spring-back loads, if more critical.
2. Skid gear, both with and without a 0.5 drag factor (Fig. 4-17):
   a. Perfectly level
   b. Nose first, near level
   c. Aft first, near level
   d. Aft first, maximum nose-up
   e. Tail-bumper first
   f. Level landing on symmetrical obstructions located at forward contact points, and, alternatively, midway between the skid attach points.
3. Tail-wheel gear, both with and without a 0.25 minimum drag factor on all contacting wheels (Fig. 4-18):
   a. Level (main first)[1]

[1] If the aft (or main) gear is not located more than 15 deg aft (and below) the CG with reference to the normal level attitude (fuselage reference line), no pitching relief to the impact loads is to be considered. However, if the aft (or main) gear is located more than 15 deg aft (and below) the CG, pitching relief may be considered (Ref. 13), provided the additional (pitching) momentum is considered in regard to possible critical secondary impact of the forward gear and provided the effect of the increased height of the CG above ground at initial contact is accounted for as the gear position is moved aft.

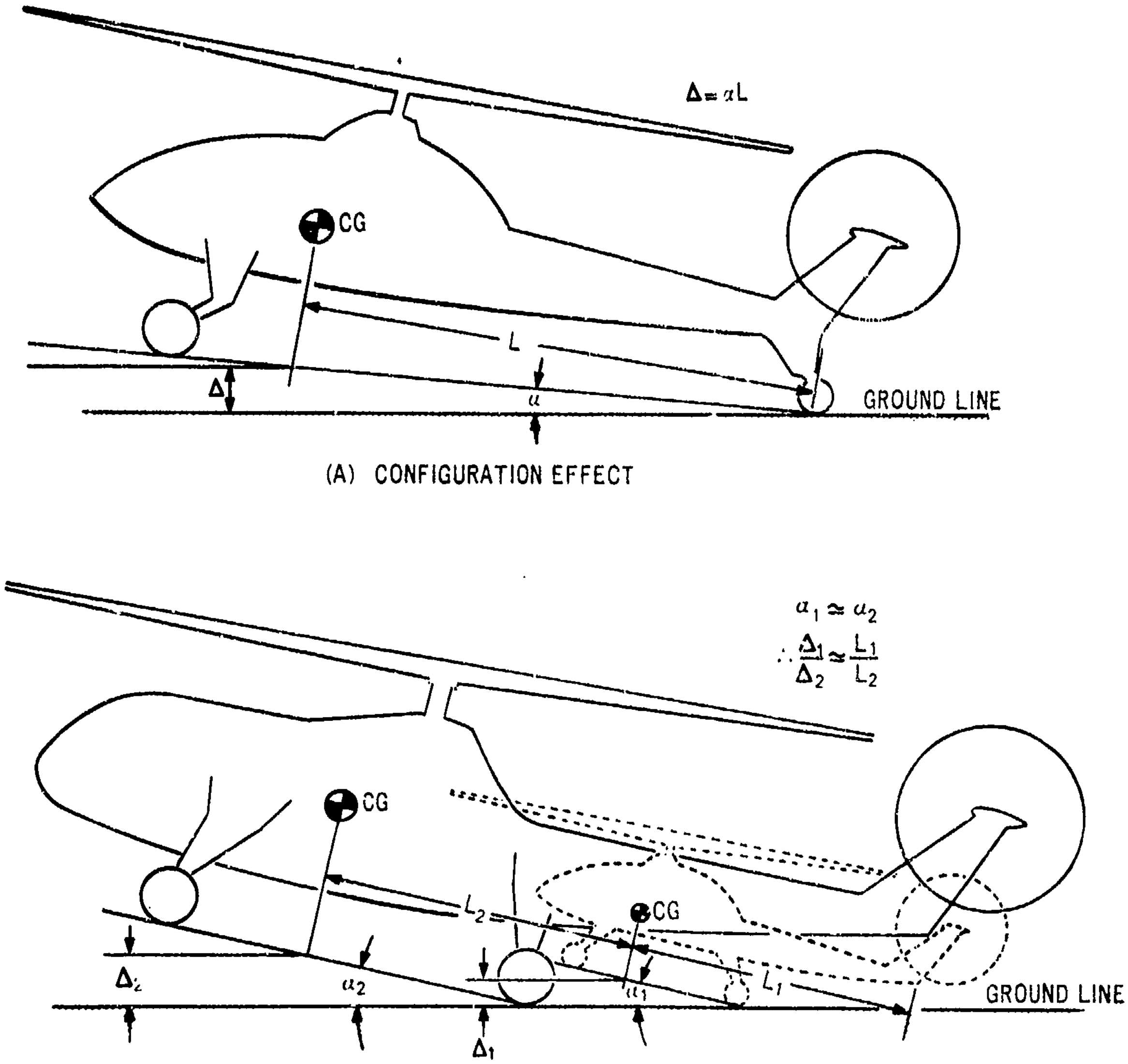


Fig. 4-14. Considerations Affecting Design Limit Sinking Speed

b. 3-point
c. Tail first
d. 50% maximum vertical load for Conditions 3a and 3c applied in any horizontal direction one gear at a time, in combination with the vertical loads. Free-swivelling tail gear *shall* be assumed to line up with the obstruction load (lockable or steerable tail gear *shall* be treated as nonswivelling).
e. Conditions 3a through 3c with spin-up and spring-back loads, if more critical.

4. Quadricycle gear, both with and without a 0.25 minimum drag factor on all contacting wheels (Fig. 4-19):

a. 4-point
b. Nose first, near level
c. Aft first, near level[1] (footnote p. 4-18)
d. Aft first, max nose-up[1] (footnote p. 4-18)
e. Tail-bumper first (if applicable)
f. Uneven terrain, near level attitude; either sufficient terrain unevenness *shall* be assumed so as to result in a three-point loading, or

alternatively, a vertical unevenness equal to 2% of wheelbase, crest to trough, *shall* be assumed at the most critical wave length and skew angle for the particular gear configuration. For this condition, 75% of the ground load factor at the CG for Condition 4a, but not less than 2.0 g, may be used.

g. 50% maximum vertical load for Conditions 4b through 4f applied in any horizontal direction one gear at a time, in combination with the vertical loads. Free-swivelling forward gear *shall* be assumed to line up with the obstruction load (lockable or steerable forward gears *shall* be treated as nonswivelling).

h. Conditions 4a through 4d with spin-up and spring-back loads, if more critical.

4-5.1.1.3 Weight and CG Factors

The CGs to be considered in conjunction with any landing weight shall include the most adverse combination of vertical, longitudinal, and lateral limits for the

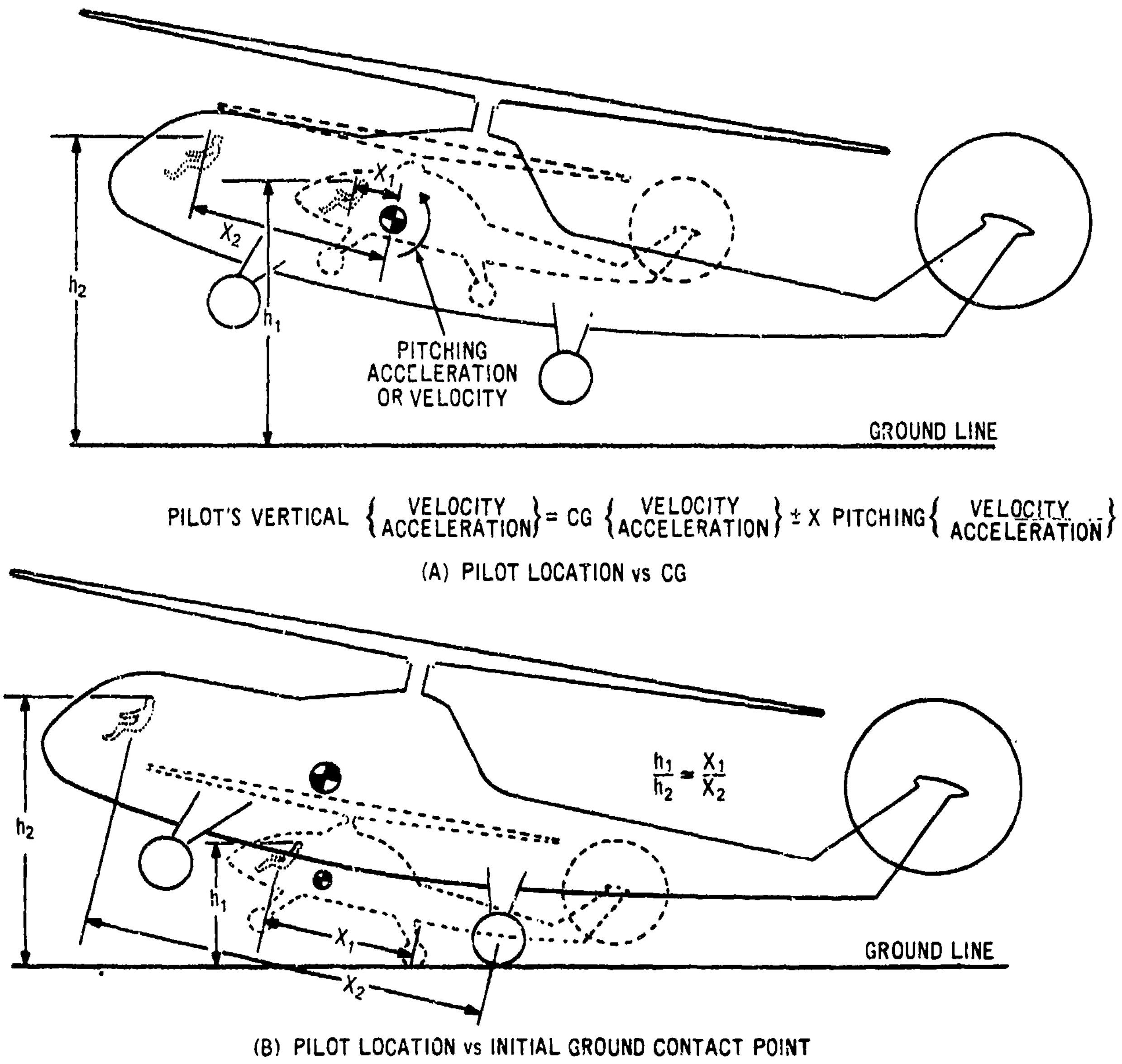


Fig. 4-15. Effects of Pilot's Location Upon Sinking Speed at Touchdown

particular helicopter, between and including minimum flying weight and alternate design gross weight.

As a minimum, the weights to be considered in regard to landing requirements are basic structural design gross weight, minimum design gross weight, and alternate design gross weight.

The mass moments of inertia to be considered at any weight and CG should reflect the most adverse distribution within the weight and CG limits chosen for investigation. This consideration should include external as well as internal weight items, provided they essentially are attached rigidly to the vehicle.

4-5.1.1.4 Evaluation of Vehicle Designs

Factors affecting the maximum sinking velocities at ground contact for optimizing helicopter designs include:

1. Disk loading. Higher disk loading means higher induced power losses in the rotor, and thus higher descent velocities in autorotation, other factors being equal.

2. Blade loading. Very little variability of this factor is noted in modern helicopter designs (Refs. 13 and 14).

3. Vehicle aerodynamic cleanness. By reducing drag, sinking velocities in autorotation are reduced in several ways:

   a. the L/D improvement results in higher forward velocity at a given sinking velocity
   b. the glide angle thus is flattened
   c. the optimum minimum sinking velocity is reduced slightly because of the reduction in induced power loss at the increased forward speed

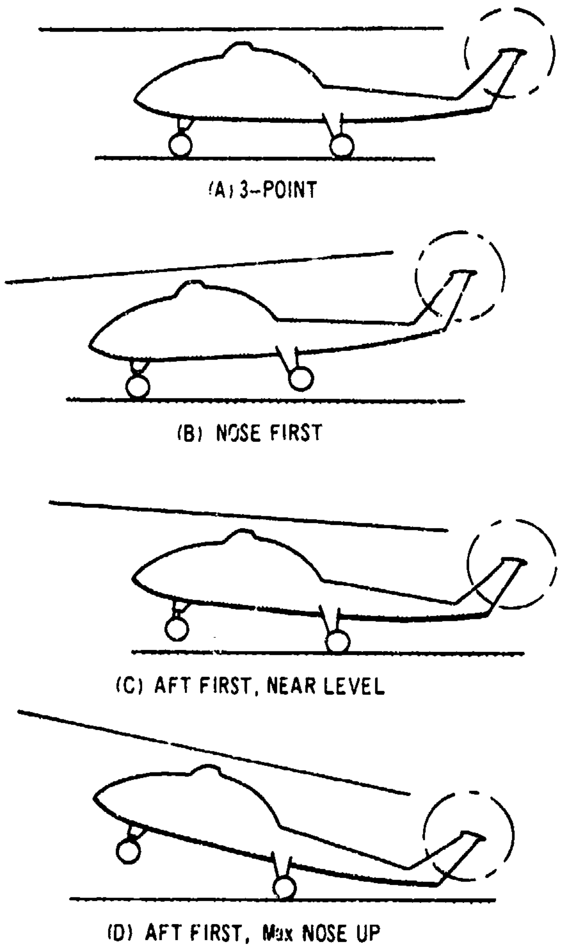


Fig. 4-16. Landing Attitudes, Tricycle Gear

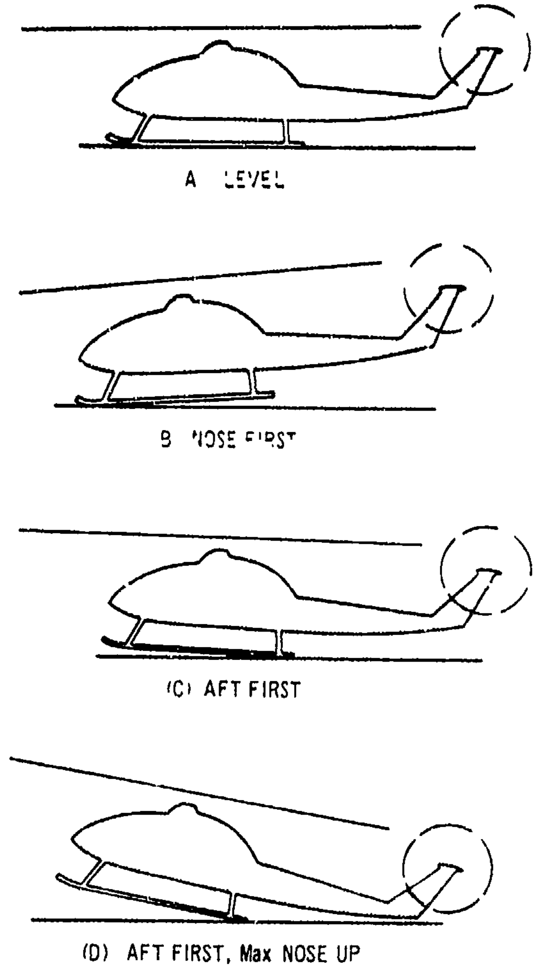


Fig. 4-17. Landing Attitudes, Skid Gear

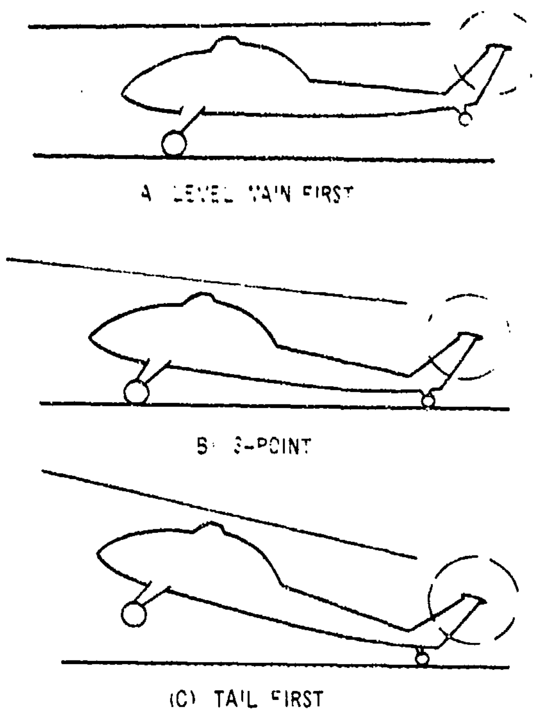


Fig. 4-18. Landing Attitudes, Tail Wheel Gear

d. the better glide path permits more time for pilot reaction to winds and terrain variables

e. the higher forward speed provides more vehicle kinetic energy, which can be converted to rotor kinetic energy during the final flare.

4. Control power and vehicle response. The ability to maneuver precisely during the approach, flare, and final touchdown has an important effect upon descent velocity at ground contact, although this effect is difficult to define in an absolute sense.

5. Pilot visibility and location relative to the vehicle CG and to the point of initial ground contact (Fig. 4-15). The poorer the visibility in the forward-downward-sideward directions, the poorer will be the pilot's ability to minimize the descent velocity at ground contact and also the greater the chance of pilot error in judging the flare maneuver relative to the terrain. Similar effects result as the pilot's location is moved forward and upward relative to the initial ground contact point (as may occur in larger vehicles). Also, as the pilot is positioned farther forward of the CG, pitching accelerations and velocities at his station become larger in comparison to the accelerations and velocities at the CG, thus reducing "pilot feel" during the crucial final portion of the landing maneuver.

4-5.1.1.5 Preliminary Substantiation Requirements

Where a particular design or mission requirement indicates adverse factors of the types in par. 4-5.1.1.4, the values of sinking velocities to be used for preliminary design must be substantiated in a rational manner. This may be accomplished either by analytical comparison relating back to the (minimum) conventional criteria, or by actual landing test demonstrations under similar conditions—comparing the more unconventional design to a similar, operationally proven, design. In most cases, analytical comparisons should suffice for preliminary design purposes.

4-5.1.2 Asymmetrical Landings

In general, the limit descent velocities, at the vehicle CG, to be used with asymmetrical landing conditions for preliminary design purposes, are not different significantly from the values to be used with the level landing conditions of par. 4-5.1.1. The most prominent

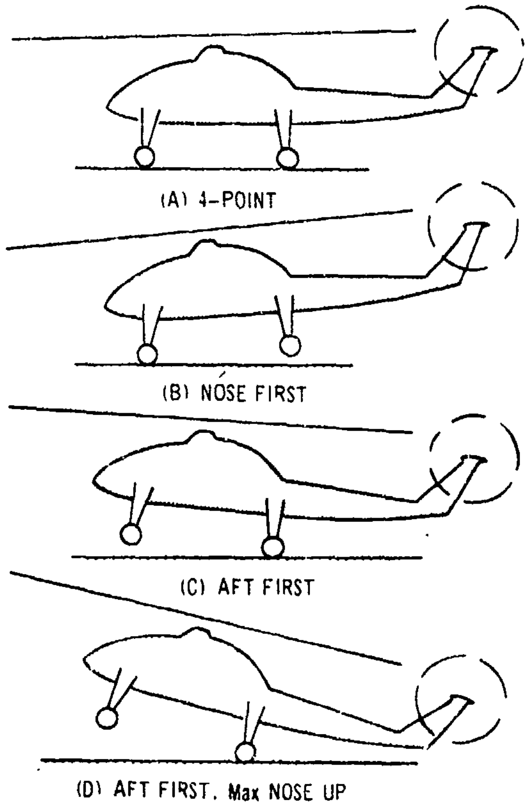


Fig. 4-19. Landing Attitudes, Quadricycle Gear

factors causing asymmetrical landing impacts in normal operation (Fig. 4-20) are:

1. Variable wind direction or gusty wind conditions
2. Adverse terrain or obstructions
3. Poor visibility
4. Evasive tactics prior to and during the landing maneuver
5. CG extremes or other vehicle factors
6. Inexpert piloting
7. Drift
8. Yaw attitude or velocity
9. Roll attitude or velocity
10. Slope landings.

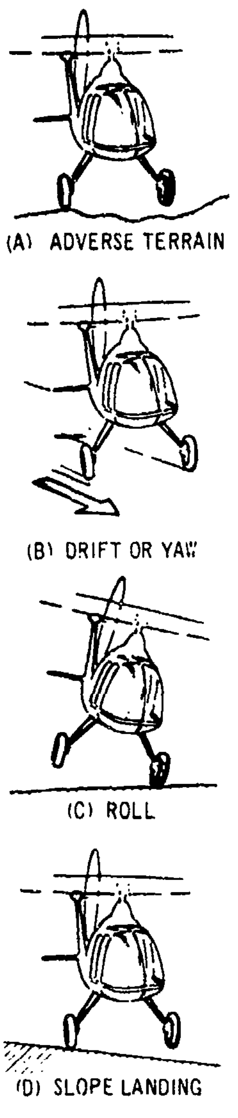


Fig. 4-20. Most Common Asymmetrical Landing Attitudes

Because the sinking speeds at ground contact are affected by many different vehicle design parameters (Refs. 12 and 15), it may be necessary to increase the 10 fps minimum design limit sink rate (par. 4-5.1.1.1) to account for unusual or abnormal vehicle and operational parameters. The horizontal landing speeds for asymmetrical landings *shall* be the same as for level landings (par. 4-5.1.1.1).

4-5.1.2.1 Landing Attitude Requirements

Asymmetrical landings require design consideration of the following variations of attitudes and conditions using, as a minimum, the same limit sinking velocities as for level landings (par. 4-5.1.1.1):

1. Tricycle gear, both with and without a 0.25 drag factor on all contacting wheels (Fig. 4-21):
   a. One aft gear first, near level[2]
   b. One aft gear first, maximum nose-up[2]
   c. Each of Conditions 1a and 1b with spin-up and spring-back loads, if more critical
   d. Drift landings, aft gear first, near level, vertical reaction at each gear equal to 50% of maximum vertical reaction from level and near-level symmetrical landings (par. 4-5.1.1.2, Conditions 1a and 1c), inward side load on one gear equal to 80% of applicable vertical load and outward side load on other gear equal to 60% of applicable vertical load simultaneously, zero drag load
   e. Landing with roll velocity: each of Conditions 1a through 1c, with 0.25 rad/sec roll velocity at ground contact in the most adverse combination with 75% limit sinking velocity
   f. Landing with roll displacement: each of Conditions 1a through 1c, with 5-deg roll attitude in the most adverse combination with 75% limit sinking velocity

[2] If the aft (or main) gear is not located more than 15 deg aft (and below) the CG with reference to the normal level attitude (fuselage reference line), no pitching relief to the impact loads is to be considered. However, if the aft (or main) gear is located more than 15 deg aft (and below) the CG, pitching relief may be considered (Ref. 16), provided the additional pitching momentum is considered in regard to possible critical secondary impact of the forward gear and provided the effect of the increased height of the CG above ground at initial contact is accounted for as the gear position is moved aft.

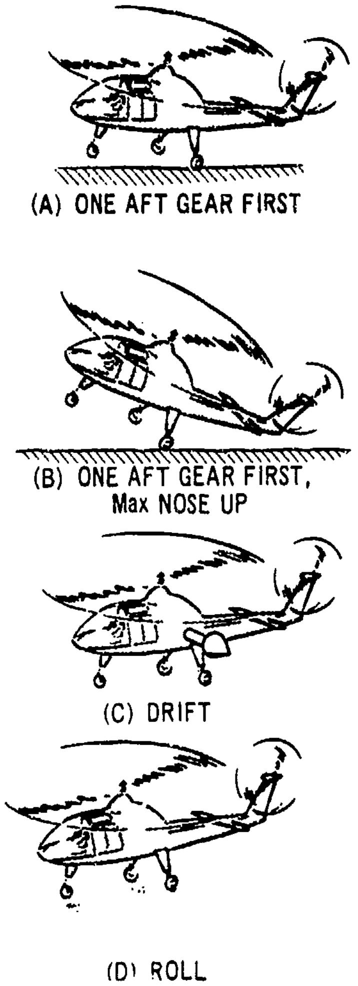


Fig. 4-21. Asymmetrical Attitudes, Tricycle Gear

g. 50% maximum vertical gear load for Conditions 1a and 1c applied in any horizontal direction one gear at a time, in combination with the vertical loads. Free-swivelling nose gear *shall* be assumed to line up with the obstruction load (lockable or steerable nose gear *shall* be treated as nonswivelling if more critical).

h. Slope landing: a 15-deg ground slope in the most adverse direction in combination with an 8 fps descent velocity

i. Tail-bumper-first attitude with 0.5 friction factor acting up to ±30 deg from aft, whichever is more critical.

2. Skid gear, both with and without 0.5 drag factor (Fig. 4-22):

a. One skid first, near level laterally; nose first, near level longitudinally; level longitudinally; aft first, near level longitudinally; and aft first, maximum nose-up

b. One skid first, without drag but with 0.5 friction factor acting inboard on one skid and outboard on the other. Assume same attitudes as for Condition 2a.

c. Level landing on asymmetrical obstruction located at one skid forward contact point

d. Tail-bumper-first attitude, with 0.5 friction coefficient acting up to ±30 deg from aft, whichever is more critical

e. Level landing with yaw velocity: a yaw velocity at ground contact sufficient to develop asymmetrical 0.5 friction factors as a transverse couple applied at the most critical locations on the skids and alternatively as a longitudinal couple, zero symmetrical drag

f. Level landing with roll velocity: Condition 2a, with 0.25 rad/sec roll velocity at ground contact in most adverse combination with 75% limit sinking velocity

g. Landing with roll displacement: Condition

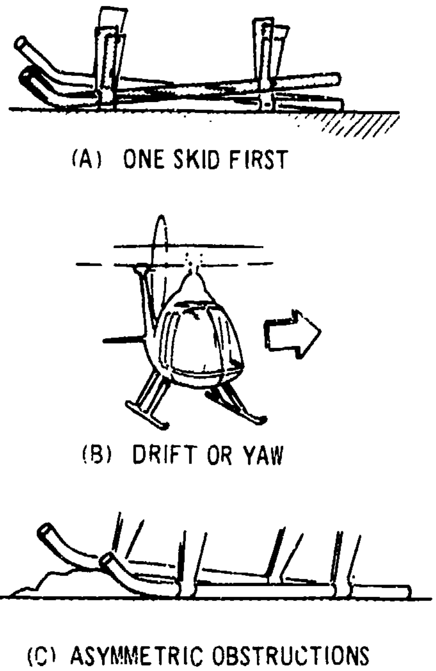


Fig. 4-22. Asymmetrical Attitudes, Skid Gear

2a, with 5-deg roll attitude in the most adverse combination with 75% limit sinking velocity

h. Slope landing: a 15-deg slope in the most adverse direction in combination with an 8 fps sinking velocity.

3. Tail-wheel gear, both with and without a 0.25 drag factor on all contacting wheels (Fig. 4-23):

a. One main gear first, level attitude[2] (footnote p. 4-23)
b. One main gear first, near 3-point attitude[2]
c. Each of Conditions 3a and 3b with spin-up and spring-back loads, if more critical
d. Drift landings: main gear first, level, vertical reaction at each gear equal to 50% of maximum vertical reaction from level and 3-point symmetrical landings (par. 4-5.1.1.2, Conditions 3a and 3b), inward side load one gear equal to 80% of applicable vertical load and outward side load on other gear equal to 60% of applicable vertical load simultaneously, zero drag load
e. Landing with roll velocity: each of Conditions 3a through 3c with 0.25 rad/sec roll velocity at ground contact in the most adverse combination with 75% limit sinking velocity
f. Landing with roll displacement: each of Conditions 3a through 3c with 5-deg roll attitude in the most adverse combination with 75% limit sinking velocity
g. 50% maximum vertical gear load for Conditions 3a through 3c, applied in any critical horizontal direction one gear at a time, in combination with the other gear vertical loads. Free-swivelling tail gear *shall* be assumed to line up with the obstruction load (lockable or steerable tail gear *shall* be treated as nonswivelling, if more critical).
h. Slope landing: a 15-deg slope in the most adverse direction together with a sinking velocity of 8 fps

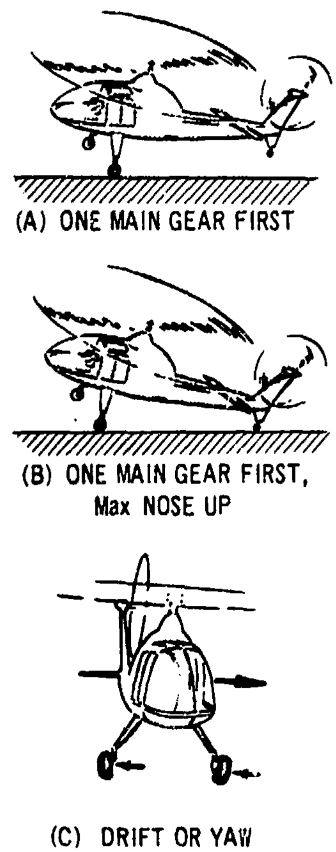


Fig. 4-23. Asymmetrical Attitudes, Tail Wheel Gear

4. Quadricycle gear, both with and without 0.25 drag factor on all contacting wheels (Fig. 4-24):

a. One side first, near level laterally; nose first, near level longitudinally; level longitudinally; aft first, near level longitudinally; maximum nose-up[2] (footnote p. 4-23)
b. Condition 4a with spin-up and spring-back loads, if more critical
c. Drift landings: one, or both, aft gear first, near level, vertical reaction at each gear equal to 50% of maximum vertical reaction from 4-point and near level symmetrical landings (par. 4-5.1.1.2, Conditions 4a through 4c), inward side load on gear on one side equal to 80% of applicable vertical load and outward side load on gear on other side equal to 60% of applicable vertical load simultaneously, zero drag load
d. Landing with roll velocity: Conditions 4a and 4b with 0.25 rad/sec roll velocity at ground contact in most adverse combination with 75% limit sinking velocity
e. Landing with roll displacement: Conditions 4a and 4b with 5-deg roll attitude in the most adverse combination with 75% limit sinking velocity
f. 50% maximum vertical load for Conditions 4a and 4b applied in any horizontal direction one gear at a time, in combination with the vertical loads. Free-swivelling nose gear *shall*

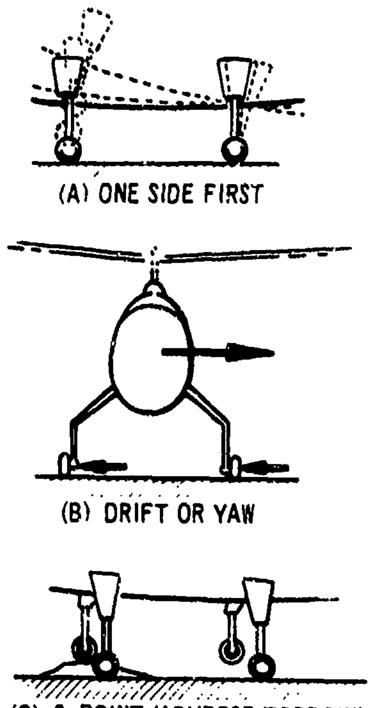


Fig. 4-24. Asymmetrical Attitudes, Quadricycle Gear

be assumed to line up with the obstruction load (lockable or steerable nose gears *shall* be treated as nonswivelling).

g. Slope landing: a 15-deg slope in the most adverse direction, with a sinking velocity of 8 fps

h. Tail-bumper-first attitude: with 0.5 friction factor acting up to ±30 deg from aft, whichever is more critical.

4-5.1.2.2 Weight and CG Factors

The weights and CGs to be considered for asymmetrical landings should be the same for symmetrical landings (par. 4-5.1.1.3). It is important that the most critical vehicle mass moments of inertia in roll and pitch also be considered in this regard.

4-5.1.2.3 Evaluation of Vehicle Designs

Factors affecting the local sinking velocities and attitudes at ground contact during asymmetrical landings for optimization of helicopter design are noted in the following paragraphs (these are in addition to the factors of par. 4-5.1.1.4):

1. Conformability (Fig. 4-25): the ability of the landing gear configuration to conform to adverse terrain conditions (especially in regard to quadricycle-type gear)

2. Yaw stability and control: the degree to which the vehicle can be maintained in optimum horizontal alignment with the direction of motion during the landing maneuver under normal expected variations of wind and terrain conditions

3. Roll stability and control: the capability of maintaining optimum roll alignment with respect to the ground under the normal expected variations of wind and terrain

4. Slope landing capability: heavily dependent upon the overall helicopter configuration, including type and configuration of landing gear, control power, size, etc.

4-5.2 RESERVE ENERGY REQUIREMENTS

4-5.2.1 Reserve Energy Descent Velocities

The reserve energy requirements for helicopter landing impacts are important to both the safety and the continued operational availability of the vehicles under the anticipated military operating environment (Ref. 11). As noted in par. 4-5.1.1 and in Ref. 17, it is important not only to specify the design limit descent velocity, under which no landing gear damage is incurred, but also to specify a more severe descent velocity under which limited damage to the landing gear and/or airframe would be acceptable operationally. The value of this latter descent velocity has been specified in MIL-S-8698 as $\sqrt{1.5}$ times the design limit descent velocity, resulting in a minimum value of approximately 10 fps. This value does not reflect the much wider variability of the helicopter operating environment (Ref. 18), especially with regard to terrain, weather conditions, rate of descent during landing approach, and the more severe demands upon pilot skill. Thus, the criteria for the design reserve energy descent velocities at ground contact for Army helicopters are as follows:

1. $\sqrt{1.5}$ × (design limit sinking velocity) = 12.24 fps. Under this severity of impact, minor, quickly repairable or replaceable damage to the landing gear components only is to be permitted. No damage to the airframe that would prevent continued safe vehicle operation is permitted.

2. 2.0 × (design limit sinking velocity) = 20 fps. Under this severity of impact, major landing gear damage is permissible, provided that complete collapse or sudden catastrophic failure does not result and that

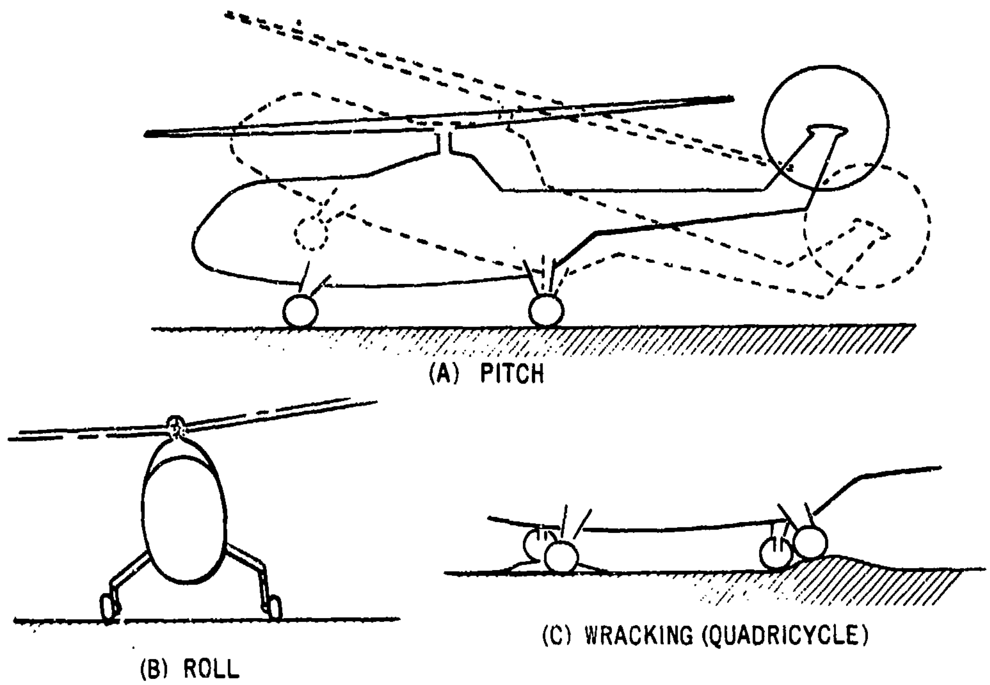


Fig. 4-25. Considerations Affecting Conformability of Landing Gear to Terrain

only minor, field repairable damage to the airframe is likely to be incurred.

4-5.2.2 Reserve Energy Design Considerations

As stated in par. 4-5.2.1 and Ref. 19, it is essential that means be provided in helicopter landing gear design to absorb additional impact energy while limiting the magnitude of the loads imposed upon the vehicle. Characteristics that help in achieving maximum reserve energy capability include the effective dissipation of the initial impact energy so as to minimize bounce and the severity of secondary impact, and effective load compensation for "hydraulic lock" (Ref. 20) of air-oil shock struts or for the elastic "spring" effect of underdamped landing gear designs. A yielding "structural fuse" (e.g., honeycomb-filled cylinder in landing gear system with yield load above normal landing gear limit load) has been found to be most effective in limiting vehicle damage for the unusually high descent velocities occasionally encountered in service.

As an important side benefit, effective energy dissipation—or deadening—and load-limiting "structural fuse" have been found by computer studies, extreme drop tests, and actual service experience to reduce substantially the likelihood of blade-to-boom interference ("boom-chops") during the landing impact.

Formerly it was thought that reserve energy impact capability was dependent largely upon reserve strength (which adds cost and weight penalties), but now it has been proven that relatively low landing load factors are acceptable, and even desirable, provided adequate provision is made in the landing gear design for energy dissipation and load compensation. This is true particularly of vehicles that are to be operated routinely for pilot training or in the battle zone environment. As shown in Ref. 17, little or no weight and/or cost penalty need result from the provision of relatively severe reserve energy capability in a landing gear design, provided proper optimization of the desired characteristics is included during the preliminary design stage of a vehicle. For example, substantial experience now is available on helicopters with landing gears having reserve energy descent velocity capabilities on the order of 15 fps, even though the design limit ground load factor was on the order of 2.0 to 2.5. These landing gears also are among the lightest in the industry, exploding the myth of an excessive weight penalty for an adequate reserve energy capability.

While structural yielding can be utilized efficiently in achieving adequate reserve energy capability at little or no overall weight penalty, there no doubt are alternative concepts that would be effective for achieving the specified objectives.

4-5.2.3 Other Considerations

Because the reserve energy descent velocities specified inherently take into account abnormally severe impact conditions, detail consideration of such additional parameters as density altitude, gross weight, wind and terrain, etc., is not necessary. The basic structural design gross weight should be used for preliminary design purposes, along with only the most probable vehicle attitudes and ground frictional considerations during the severe impact. It also should be kept in mind that these extreme values of vertical velocity at ground contact are generally the result of an excessive descent velocity during approach, an incompleted or insufficient flare maneuver, or excessive hover height in relation to available rotor energy after completion of the landing flare. Loss of engine power near the ground, or battle damage, also occasionally are contributing factors. Except for the landing from a hover, appreciable forward velocity may exist, along with some drift component. Ref. 11 indicates that the forward velocity at impact generally is no greater than that for best approach speed-power-off, i.e., best glide angle. Therefore, for design purposes the reserve energy descent velocity *shall* be combined with a horizontal velocity equal to 120% of the speed for minimum power required. This combination of velocities should be considered throughout the attitude range from 15-deg nose-down to the maximum nose-up attitude attained during a maximum horizontal deceleration maneuver.

4-5.2.4 Autorotational Capability Indices (Landings)

At least two or more autorotational indices have been found useful in judging the comparative capability of helicopters to make successful autorotational landings (Ref. 11). In general, these methods are used to determine an "autorotation constant" K, which is a ratio of the useful rotor kinetic energy KE either to the power required to hover or to the autorotational sink rate energy (or power) as in the following equations:

$$\left.\begin{aligned} K_1 &= \frac{\text{rotor kinetic energy}}{\text{hover power}} \\ K_2 &= \frac{\text{rotor kinetic energy}}{\text{sink rate energy}} \\ KE &= \frac{I_p}{2}\left(\Omega_A^2 - \Omega_{min}^2\right)\text{, ft-lb} \end{aligned}\right\} \quad (4\text{-}11)$$

where

I_p = polar mass moment of inertia of rotor, slug-ft^2

Ω_A = rotor speed for autorotation, rad/sec

Ω_{min} = minimum rotor speed at which rotor thrust = helicopter weight, rad/sec

Hover power required and autorotational sink rates are taken from performance data normally available for the particular helicopter configuration being considered. The value K_1 may be interpreted physically as a comparative "stand-off time", while value K_2 may be interpreted physically as a comparative capability of arresting the sink rate. The effects of relative landing gear impact capability and of relative vehicle kinetic energy due to forward speed, as well as of relative control power, also must be taken into consideration along with the autorotational constants noted.

Ref. 21 uses the results of three height-velocity flight test programs to formulate a semi-empirical procedure for showing the effects of density altitude and helicopter gross weight on the shape of the height-velocity (H-V) diagram for autorotational landings.

4-5.3 CRASH LOADS

The helicopter *shall* be designed for protection of the occupants during a crash. In the paragraphs that follow, the application of crashworthy structural design features for maximum protection of occupants is discussed. Included is a discussion of the aircraft crash environment, fuselage structural design considerations, and controlled deformation of structure (primary, secondary, and crew seats). Retention of equipment such as transmissions, rotor masts, seats, and occupants in the cabin and cockpit regions is discussed. The retention of other equipment and stores as applicable to occupant protection from "missiles" within the decelerating fuselage also is discussed.

4-5.3.1 Crash Environment

The crash environments of aircraft range from the insignificant hard landing to the nonsurvivable landing. When an aircraft crashes, motion continues until the kinetic energy has been attenuated, primarily through the application of force through distance. The decelerative force is a function of the kinetic energy of the aircraft and the distance through which it moves during deceleration.

In the crash of an aircraft with a purely vertical velocity component, this movement is permitted by deformation of both the terrain upon which the aircraft crashes and of the structure of the aircraft. If the aircraft crashes on soft soil, considerable soil deformation will occur and the decelerative load will be less because of the distance traveled against the force compacting or moving the soil, and against the crushing strength of the fuselage. If the aircraft crashes on a rigid surface such as concrete, the deformation distance essentially will be supplied entirely by the crushing fuselage, resulting in a higher load factor.

If the aircraft crashes with a high longitudinal component of velocity, the longitudinal decelerative loading can be a function of many things. These include friction, plowing and gouging of earth, aircraft longitudinal crush strength (in the case of barrier impact), or local crush strength of the fuselage impacting local barriers such as trees, posts, and rocks.

Combinations involving longitudinal, vertical, and lateral components of velocity include longitudinal, vertical, and lateral decelerative loads. Relatively high vertical decelerative loads also can be applied through the process of rapidly changing the direction of the longitudinal velocity component of the aircraft structure, as when an aircraft with high longitudinal velocity impacts a relatively rigid surface at even a very slight angle. Consequently, consideration must be given to the existence of high vertical decelerative loads in accidents consisting of primarily longitudinal impact velocity components, as well as in accidents having high vertical velocity components.

The crash environment for helicopters provides a high potential for rollover because of the vertical location of the CG and because of the turning rotor. Rotor strikes on trees or other obstacles tend to flip the aircraft on its side. Because more than half of the significant survivable accidents of rotary-wing aircraft now involve rollover, lateral retention and strength and ceiling support strength in the occupied regions are of extreme significance.

Another environmental hazard can be created by the main rotor of the helicopter. When the aircraft crashes, the rotor blades deflect downward which can have serious consequences, in addition to striking the ground and rolling the aircraft. For example, the torsional and bending load transmitted to the rotor mast and transmission upon contact of the blade(s) with the ground tends to tear the transmission free and displace it into the occupied sections of the aircraft. In addition, vertical deflection of a rotor at impact may permit the blades to pass through parts of the fuselage, creating an intrusion hazard for the occupants in those regions.

During a typical crash the aircraft structure progressively collapses at its crush strength. The total deformation distance of the structure is a function of the kinetic energy of the aircraft and the structural depth and strength. The decelerative loads transmitted to the occupied sections of the aircraft thus are reduced from those experienced by the contact point of the aircraft. Additional deformation distance reduces the loads experienced by the occupants and occupied portions of the aircraft.

Survivable accidents (defined by evaluation of present-day aircraft structure and existing accident records) are explained in Ref. 11. Deceleration-versus-time pulses and, thus, the defined impact velocity changes representative of the 95th percentile survivable crash environments, have been determined for the various directions and are presented in Table 4-3. Terminology is defined in Fig. 4-26. Helicopters should be designed to protect all occupants in crashes having these characteristics. However, the peak *g* levels listed are a result of the structural strength of existing aircraft and are not necessarily desirable design levels. Helicopters *shall* be designed to protect the occupants in crashes which produce the design velocity changes listed. Because of variations in the impacted surface, aircraft orientation, and effect upon mission performance, environmental conditions have been reduced to design criteria for various critical portions of the aircraft. These criteria are presented in the discussion that follows.

4-5.3.2 Structural Design

The aircraft, once involved in a crash, is expendable, and preference is given to occupant protection. Therefore, the helicopter should be designed to provide the maximum degree of protection possible to the occupants. The structure surrounding occupied areas *shall* be the strongest in the aircraft and *shall* remain reasonably intact. If the protective shell collapses around the occupants during a crash, then efforts to improve their chances for survival by improvement of occupant restraint systems or by reduction of post-crash hazards are futile. The structure *shall* be designed to crush and deform in a controlled, predictable manner so that forces and decelerations imposed upon occupants are minimized while still maintaining the protective shell. This means that analysis for crashworthiness must consider the large deflections of structural members and joints, with loading in the plastic strain range.

Airframe structure should be designed first for normal flight loads, landing loads, and ground handling loads; but the requirements of crashworthiness should be kept in mind. A dynamic analysis should be made to determine impact forces and accelerations, particularly those decelerative forces transmitted to the structure that supports personnel seats. This analysis is necessary to determine the degree of structural deformation required to permit personnel survival and to determine methods to prevent complete failure of that structure that surrounds personnel. In order to permit plastic deformation of the structure, the use of a safety factor of 1.0 with a margin of safety of 0.0 based upon yield strength is desirable for the crash load condition (Ref. 11). In areas where gross structural deformations are anticipated, joints should be designed and analyzed to reduce the probability of failure under large angular deflections and linear displacements and to provide maximum capability for energy absorption.

Crushed structure that remains intact and in place can help to shield remaining structures and personnel from damage in subsequent impacts. During a crash, loads are rapidly increasing, not instantaneous. In most cases, a minimum of 10 msec is required for loads to reach maximum values. Under such conditions, inertial effects may be of importance although strain rate effects in materials probably are insignificant. Material ductility is required to insure that crushing, twisting, and buckling of the structural shell can occur without rupture.

The structure that contacts the impact surface first is usually the first to deform. The localized deformation continues either until the kinetic energy of the aircraft

TABLE 4-3
SUMMARY OF DESIGN PULSES FOR HELICOPTERS

IMPACT DIRECTION	VELOCITY CHANGE, fps	PEAK ACCEL, g	AVERAGE ACCEL, g	PULSE DURATION t, sec
LONGITUDINAL (COCKPIT)	50	30	15	0.104
LONGITUDINAL (PASSENGER COMPARTMENT)	50	24	12	0.130
VERTICAL	42	48	24	0.054
LATERAL	25	16	8	0.097

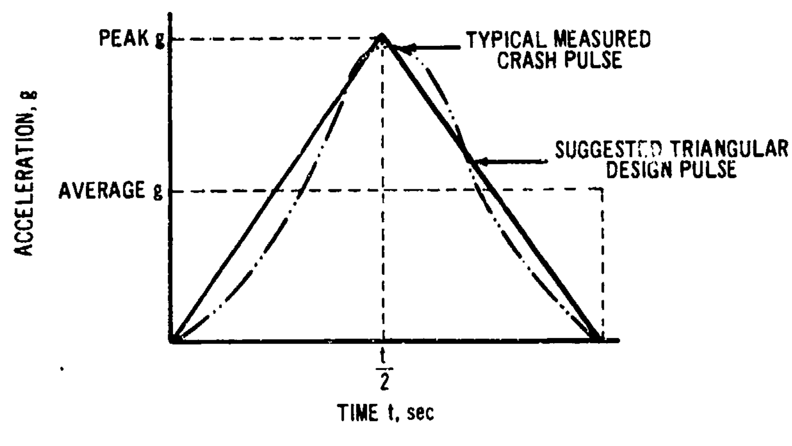


Fig. 4-26. Typical Impact Pulse for Helicopters

is absorbed or until the localized deformation capability is exhausted. Additional structure then can be involved in the deformation. For example, a circular-shaped fuselage, when loaded radially, deforms in a manner that produces a continually increasing contact surface. This characteristic can be used easily to provide desirable force versus deformation and occupant protective shell strength in an aircraft fuselage.

Occupant survival considerations include:

1. Increasing the energy-absorption capacity of the structure surrounding the occupiable areas. Energy-absorbing capability of the structure beneath the floor is extremely important because helicopter crashes typically involve a relatively high vertical deceleration component.

2. Designing the structure that makes initial contact with the ground so as to minimize gouging and scooping of soil, hence reducing the deceleration and reducing the longitudinal forces

3. Designing for safe breakaway of components such as the tail rotor boom and external stores during a crash to effect a reduction in the mass of the aircraft, hence reducing the strength required in the cockpit and cabin structure for energy absorption

4. Reinforcement of cockpit and cabin structure to permit withstanding of crash forces without significant reduction in occupiable volume

5. Designing tiedowns of equipment located in occupied areas for retention during a crash.

4-5.3.2.1 Lateral Impact

The helicopter *shall* be designed to withstand lateral impact on a rigid surface (such as concrete) with velocity changes of 30 fps without serious injury to the occupants. The helicopter must provide the required degree of protection for lateral crashes in which the fuselage is rotated ± 20 deg about its roll axis and ± 20 deg about its yaw axis.

4-5.3.2.2 Vertical Impact

The helicopter *shall* be designed to withstand vertical crashes onto a rigid surface with impact velocity changes of 42 fps without serious injury to the occupants. The helicopter *shall* be designed to provide the required degree of protection for vertical crashes in which the fuselage is rotated ± 15 deg about its pitch axis and ± 30 deg about its roll axis.

The design *shall* include provisions for reducing the decelerative loads imposed upon occupants. A recommended provision is energy-absorbing passenger and crew seats that protect occupants in crashes at energy levels that are survivable from the standpoint of general cockpit and cabin collapse.

The threat of general cockpit and cabin collapse under vertical impact may be reduced by:

1. Transfer of mass from the top of the fuselage to the cockpit and/or cabin floor

2. Localized strengthening at locations of large concentrations of mass attached to upper structure

3. Design of subfloor, sidewall, cockpit, and cabin structure that increases elastic energy absorption or provides for plastic energy absorption at loads less than the general collapse load to maintain primary cabin integrity

4. Use of energy-absorbing landing gear to reduce the severity of cockpit and cabin decelerations for minor impacts.

4-5.3.2.3 Longitudinal Impact

The helicopter *shall* be designed to withstand a longitudinal impact into a rigid wall or barrier with an impact velocity change of 20 fps without serious injury to the occupants, including crew members.

The helicopter also *shall* be designed to withstand longitudinal crashes onto terrain surfaces with 50 fps velocity changes without serious injury to the occupants. The underbelly of the aircraft *shall* be designed to minimize gouging and plowing of soil to minimize the decelerative loading and limit it to the design values. In order to accomplish this, a ductile material having an elongation of at least 10% should be used. In addition, it is recommended that belly skins on aircraft weighing up to 3000 lb should be capable of sustaining running loads of 1500 lb/in.; over 3000 lb but under 6000 lb, 2400 lb/in.; and over 6000 lb, 3000 lb/in. over at least the forward 20% of the basic fuselage length.

4-5.3.2.4 Combined Vertical and Forward Resultant Velocity

The helicopter *shall* be designed to withstand crashes with a resultant impact velocity change of 50 fps without serious injury to occupants. The sink velocity component applied simultaneously with the longitudinal velocity component *shall* not exceed 42 fps.

4-5.3.2.5 Rollover

The helicopter *shall* be capable of impacting sod-type terrain at a 5-deg impact angle up to a horizontal velocity of 100 fps without overturning. In addition, with the helicopter inverted or on its side, the structure *shall* be able to withstand the loads resulting from the basic structural design gross weight and the following load factors:

1. 4.0 perpendicular to a waterline
2. 4.0 parallel to a waterline
3. 2.0 laterally.

4-5.3.2.6 Landing Gear

The landing gear *shall* be capable of totally decelerating a fully loaded helicopter (basic structural design gross weight) from an impact velocity of 20 fps with only minor airframe damage and without transmitting excessive forces to the fuselage or to the occupants (par. 4-5.2.1). The gear *shall* not penetrate into the occupied section of the helicopter.

4-5.3.2.7 Overhead Masses

Massive components located overhead, such as the transmission and rotor mast and other items that might cause injury to personnel, *shall* be designed to withstand the following separately applied loads: lateral, $\pm$ 18 g; longitudinal, $\pm$ 20 g; and vertical, + 20 g and − 10 g. For simultaneously applied loads, each of the cited loads *shall* be applied in turn at its maximum value while the remaining two are reduced to one-half of their maximum values.

4-5.3.3 Seat and Restraint System Design

To perform their intended retention functions, the seat, attachments, and supporting structure *shall* possess sufficient strength to reduce the occupant velocity to zero relative to the helicopter structure to which the seat is attached. In addition, both the restraint system and the seat *shall* possess physical characteristics that tend to reduce rather than to amplify the decelerative load transmitted to the occupant from the helicopter structure. Because the harness and the seat provide the interface between the occupant and the helicopter and will be in contact with the occupant for long periods of time, comfort must not be reduced significantly by efforts to increase crashworthiness. Crew seats *shall* be designed in accordance with MIL-S-58095.

During a potentially survivable crash, the load that the supporting structure must carry can be reduced through deformation of the seat structure, by load-limiting devices, or through a combination of both. The objective of intentionally load-limiting seat systems is to use the space between the seat and the floor for relative displacement of the seat and occupant with respect to the airframe, thus decreasing the load transmitted to the occupant. The intent is to maintain tolerable loads upon the occupant throughout the crash pulse. Factors that affect the final design of a seat and restraint system include human tolerance limits, the design input pulse, the occupant weight, the weight of the seat, the cushion, and the restraint system.

Additional factors that affect the design of load-limiting seats include the weight of the movable part of the seat and the available stroke distance. The occupant weights to be considered in design of crew seats range from 211 to 146 lb, representing the 95th to 5th percentile Army aviator, respectively. Crew seat strength *shall* be based on the 75th percentile soldier at a weight of 235 lb; however, design of the energy-absorbing system must include consideration for the full range of occupant weights. Dynamic analyses of the cushion, occupant, seat, and restraint system must be accomplished to establish an optimum load-limiting system within the available stroke length. Stroke length should be maximized to achieve the maximum protection for the occupant.

The seats *shall* be capable of maintaining their structural integrity and attachment to the airframe during floor angular and linear deflections resulting from a crash. Attachments, therefore, *shall* be capable of undergoing angular and linear displacements while maintaining design shear, tensile, and compressive loading requirements. Floor attachment joints *shall* be capable of allowing a universal $\pm$ 10 deg angular rotation without failure. Attachments of seat members to seat support structures *shall* be capable of permitting the linear misalignment of the floor attachment points without imposing excessive loads on seat members.

The load-carrying capacity of components that are deformed beyond their elastic limit *shall* be considered in determining the ultimate strength of the seat. The use of ductile materials is desirable. Materials having an elongation of 10% or greater are recommended for use on all critical structural members on seat and restraint systems.

Detailed information and requirements on seat and restraint system design may be found in Ref. 11 and MIL-S-58095.

4-5.3.4 Other Equipment and Stores

All equipment and stores that are carried openly in the crew or troop/passenger compartments and are of sufficient or critical size, mass and location to constitute a hazard to personnel when torn free in a crash *shall* be provided with restraint devices and/or be anchored securely to structure capable of restraining the equipment in a survivable crash. Minimum design load factors for such items *shall* be in accordance with par. 4-5.3.2.7.

It is a relatively simple task to restrain small items of ancillary equipment to withstand the specified static loads without significant weight penalties. For larger items, however, weight penalties might be incurred. In addition, and perhaps more important, the available supporting structure may not be capable of withstanding the loads anticipated. For this reason the use of load limiters for heavier equipment or stores may be an option to the static strength requirements. If load limiters are employed, they *shall* conform to the seat retention principles described in Ref. 11. Also, if the stroking of such load limiters allows the equipment or stores to enter the occupant strike envelope, the equipment *shall* be padded in accordance with Ref. 11.

4-6 TAXI AND GROUND-HANDLING LOADING CONDITIONS

Taxiing and ground-crew handling can impose critical loads on the helicopter basic structure, landing gear, and ground-handling equipment. This paragraph presents the taxiing and ground-handling criteria that *shall* be considered during the preliminary design of certain portions of the helicopter, based on MIL-S-8698 and MIL-A-8862.

Load factors and weights that are consistent with realistic operating conditions *shall* be used. Thus, it may be necessary to go beyond the minimum arbitrary load factors specified in MIL-S-8698.

4-6.1 GROUND MANEUVERING

Ground maneuvering conditions may occur while the helicopter is at its maximum weight; therefore, loads *shall* be computed using the maximum alternate design gross weight. Ground maneuvering loads are caused by various braking conditions, by turning or pivoting, and by operation over uneven surfaces. While calculating the critical loadings, it should be kept in mind that the purpose is to determine whether the maximum load in a local portion of the structure is more severe or more critical in magnitude and/or direction than that which results from normal landing conditions.

4-6.1.1 Braking Conditions

Only three-wheel landing gear configurations, either nose wheel or tail wheel, are considered in the paragraphs that follow. For quadricycle gear configurations, criteria comparable to those given *shall* be applicable.

4-6.1.1.1 Two-point Braked Roll

For either a nose-wheel or a tail-wheel configuration, the requirements of MIL-A-8862 *shall* apply for the two-point braked roll except that the vertical load factor at the CG *shall* be 1.2 for all gross weights. This loading condition is shown in Fig. 4-27.

4-6.1.1.2 Three-point Braked Roll

The requirements of MIL-A-8862 *shall* apply to the three-point braked roll of helicopters with nose wheel landing gear, except that the vertical load factor at the CG *shall* be 1.2 for all gross weights. This loading condition is shown in Fig. 4-28.

4-6.1.1.3 Unsymmetrical Braking

For nose-wheel helicopters, the unsymmetrical braking requirements of MIL-A-8862 *shall* apply.

4-6.1.1.4 Reverse Braking

For both nose-wheel and tail-wheel helicopters, the reverse braking requirements of MIL-A-8862 *shall* apply.

4-6.1.1.5 Wheel, Brakes, and Tire Heating

In the selection of wheels, brakes, and tires, the requirements of MIL-W-5013, MIL-T-5041, and MIL-B-8584 are applicable. The heat generated during braking *shall* not result in stresses that will cause explosion or failure of these components during and subsequent to prolonged and repeated brake application.

4-6.1.2 Turning

The turning requirements of MIL-A-8862 *shall* apply to both nose-wheel and tail-wheel helicopters. The following formulas based upon loads and dimensions as defined by Fig. 4-29, may be used to determine the loads. For the gear on the outside of the turn

$$F_{V_{M_1}} = 0.5\frac{Wb}{d} + n_s\frac{We}{t}, \text{ lb} \qquad (4\text{-}12)$$

$$F_{S_{M_1}} = n_s F_{V_{M_1}}, \text{ lb} \qquad (4\text{-}13)$$

$$F_{V_A} = \frac{Wa}{d}, \text{ lb} \qquad (4\text{-}14)$$

$$F_{S_A} = n_s F_{V_A}, \text{ lb} \qquad (4\text{-}15)$$

In Eqs. 4-12 through 4-15 the value of the lateral load factor n_s *shall* be the overturning value, $n_s = 0.5bt/(de)$, although n_s need not be greater than 0.5.

For the gear on the inside of the turn

$$F_{V_{M_2}} = 0.5\frac{Wb}{d} - n_s\frac{We}{t}\ , \text{lb} \qquad (4\text{-}16)$$

$$F_{S_{M_2}} = n_s F_{V_{M_2}}\ , \text{lb} \qquad (4\text{-}17)$$

In Eqs. 4-16 and 4-17 the value of the lateral load factor n_s *shall* be the value giving maximum vertical and side load on the wheel on the inside of the turn, $n_s = 0.25bt/(de)$, although n_s need not be greater than 0.5.

For auxiliary gear, the values of Eqs. 4-12 through 4-15 *shall* be used.

4-6.1.3 Pivoting

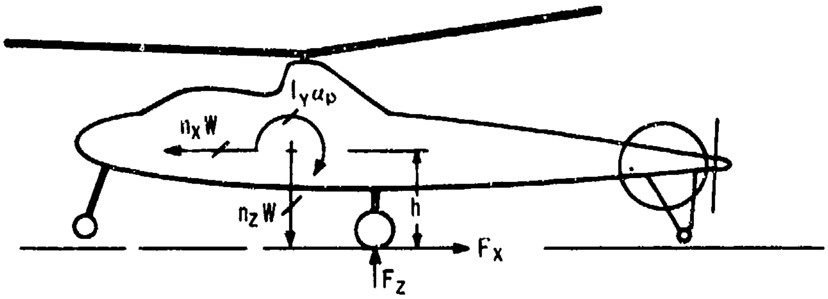
The pivoting requirements of MIL-A-8862 *shall* apply to both nose-wheel and tail-wheel helicopters.

4-6.1.4 Taxiing

The taxiing requirements of MIL-A-8862 *shall* apply to both nose-wheel and tail-wheel helicopters.

4-6.1.5 Special Tail-gear Conditions

The special tail gear conditions of MIL-A-8862 *shall* apply to tail-gear helicopters. These special conditions include a tail-gear obstruction condition.

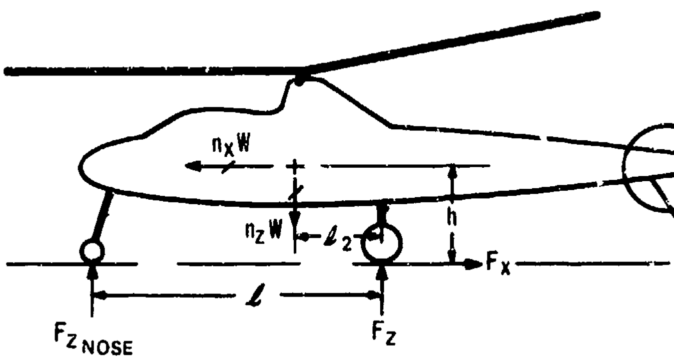


$$\alpha_p = \frac{F_X h}{I_Y}$$

$$n_X W = F_X = 0.8F_Z$$

$$n_Z W = F_Z = 1.2W$$

Fig. 4-27. Braked Roll, Two-point



$$n_X W = F_X = 0.8F_Z$$

$$F_{Z_{NOSE}} = \frac{F_X h}{\ell} + \frac{n_Z W \ell_2}{\ell}$$

$$n_Z W = F_Z + F_{Z_{NOSE}} = 1.2W$$

Fig. 4-28. Braked Roll, Three-point

4-6.2 JACKING AND MOORING CONDITIONS

As a part of the preliminary design of the helicopter, the load investigations should include loading due to jacking and mooring. This paragraph presents the criteria for these conditions and a discussion of applications.

4-6.2.1 Jacking Loads

For maintenance such as repairing or changing landing gear components and for weighing and balancing the helicopter, jack points are provided. The jack-point fittings and their backup structure must be analyzed for the forces imposed. Jacking loads for helicopters *shall* be in accordance with MIL-A-8862 except that the maximum alternate design gross weight *shall* apply. Jacking conditions generally do not present overall structural design problems but may produce loads that are critical locally.

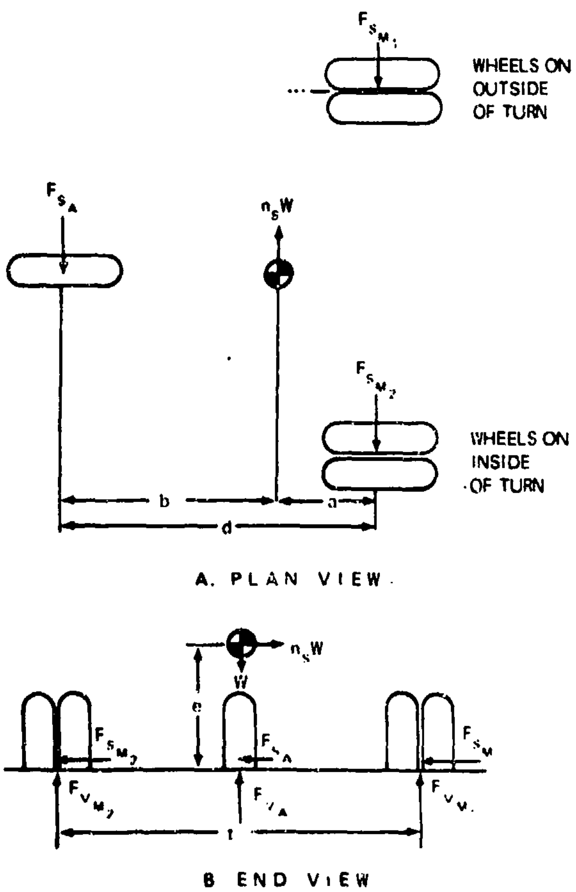


Fig. 4-29. Turning

4-6.2.2 Mooring Loads

With the helicopter secured in the static attitude and with rotor blades secured and control surfaces locked, a 70-kt wind *shall* be imposed from any horizontal direction. Under those conditions a helicopter of conventional configuration generally will not develop loads that exceed the friction resistance of the landing gear, even with the helicopter in an empty weight configuration.

Eq. 4-18 is used to compute the horizontal wind load F.

$$F = C_D \left(\frac{\rho V^2}{2}\right) A \quad \text{, lb} \tag{4-18}$$

where

A = presented area, ft^2
C_D = drag coefficient, dimensionless
V = wind speed, fps

With the addition of wings to the helicopter configuration, a significant lift force may be developed. The mooring system must react the resultant of the lift and drag (wind) loads. The wing lift L would be

$$L = C_L \left(\frac{\rho V^2}{2}\right) S \quad \text{, lb} \tag{4-19}$$

where

C_L = lift coefficient, dimensionless
S = wing area (planform), ft^2

The lift coefficient C_L in this case is based upon the angle of attack that the wing presents in the static attitude.

4-6.3 TOWING AND TRANSPORT LOADS

This paragraph considers the loads developed during towing and transport. While towing is traditional and requires no amplification, transport does. Transport refers to delivery by land vehicle, by airplane, or by helicopter (including sling carriage).

4-6.3.1 Towing Loads

The towing requirements of MIL-A-8862 *shall* apply to all helicopters, except that the applicable weight *shall* be the maximum alternate design gross weight.

4-6.3.2 Transport Loads

For purposes of design for transport, a limit load factor of 2.67 is considered appropriate for application to the CG within a 30-deg vertical cone at the hoisting (sling) attachment or on the support points as mounted in truck, train, or aircraft. Gross weights must be appropriate to the loading condition; for the sling condition, full internal fuel *shall* be included, while for vehicle transit, neither fuel nor payload need be included. For truck, train, and aircraft delivery, the helicopter sometimes is disassembled, supported in cradles, and boxed. Transportability considerations are discussed in par. 13-3.

4-7 MISCELLANEOUS LOADING CONDITIONS

4-7.1 ROTOR ACCELERATION

Rotor acceleration is the result of an unbalance of torque acting upon the drive system.

The case considered herein will be limited to acceleration from ground idle to normal operating rotor speed. During rapid accelerations, the rotor and drive systems can be subjected to high transient torque loads that can overstress the system momentarily. The primary considerations are initial rotor speed, response of the engine to pilot command, and inertia of the rotor/drive system. In most applications, integrated engine control systems or automatic engine start equipment are incorporated to assist the pilot in avoiding an overtorque condition during startup and normal missions.

Generally, the most critical conditions will occur when maximum engine power is applied to the system at low rotor speed, causing high peak torques in the mast and transmission and high inplane bending moments on the rotor blades.

The peak loads developed during a maximum power acceleration are functions of the particular engine and rotor/drive system. The loads acting upon a rotor with articulated blades are not necessarily distributed equally to each of the rotor blades. However, the loads upon a rotor that does not incorporate drag hinges are distributed equally. The distribution of loads upon articulated and rigid rotors is considered in par. 4-7.1.3.

The factors influencing peak engine torque developed in a maximum acceleration are:

1. Rotor speed at ground idle
2. Engine acceleration time
3. Total rotor/drive system inertia
4. Ambient temperature.

Figs. 4-30(A) and (B) show the relationship of horsepower and torque, respectively, to rotor speed for a typical free-turbine engine. Line G-A-F in both figures depicts a typical rapid acceleration from ground idle to flight idle. The engine will accelerate from Point G to Point A (gas generator topping) with little rotor speed change. As rotor speed increases (Point A to Point F), the engine maintains topping conditions until the governed rotor speed is reached. In the governed range, the engine governor schedules power, through fuel flow, to match engine conditions to flight idle power required.

From Fig. 4-30 the effect on the baseline acceleration case (Line G-A) of the principal system and environmental factors can be shown as follows, using the relationship between inertia I, torque Q, and angular acceleration $\Delta N/\Delta t$:

1. Rotor speed at ground idle. Ground idle speed at a lower power is shown by Point LG. A rapid acceleration from this condition is shown by Line LG-B, which is parallel to the base Line G-A but intercepts the topping torque line at a higher peak value.

2. Engine acceleration time. The engine (gas generator) accelerates to peak torque along Line G-A with little rotor speed change. If the engine were to be accelerated more rapidly (Line G-C), less rotor speed change would occur and peak torque would be higher; if the engine were to be accelerated more slowly (Line G-D), the peak torque during the engine acceleration time would be lower.

3. Total rotor/drive system inertia. An increase in inertia would lower the rotor speed change and increase the peak torque (Line G-C), while a decrease in inertia would decrease peak torque (Line G-D).

4. Ambient temperature. A decrease in ambient temperature normally will increase the rotor speed at ground idle, increase the engine acceleration rate, and increase the maximum torque available.

The procedure for estimating the loading condition resulting from acceleration of the rotor system involves the determination of ground idle speed, peak engine torque, and rotor blade loads.

4-7.1.1 Determination of Ground Idle Rotor Speed

The ground idle speed is determined by equating engine power available to the aerodynamic rotor loading at the minimum pitch or thrust position. The airframe manufacturer usually has some latitude in selection of the engine power—hence, rotor speed—at ground idle. Selection of rotor speed for ground idle involves insuring that the rotor system is stable with regard to blade flapping and ground resonance. Other

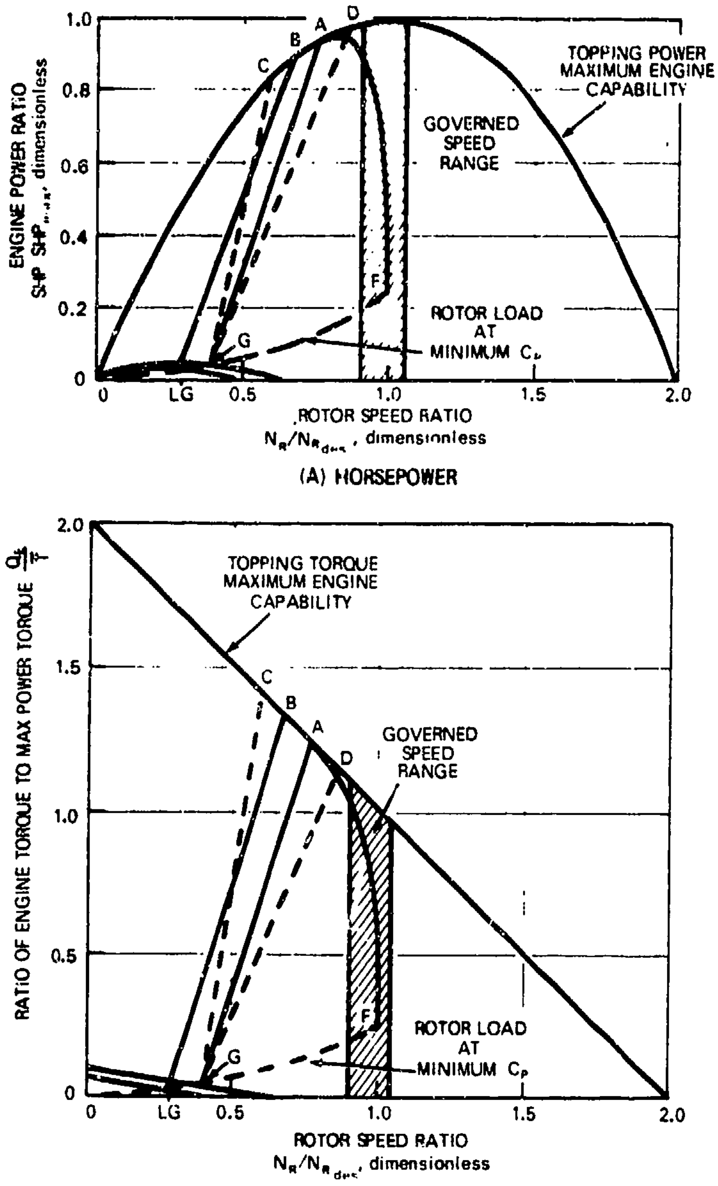


Fig. 4-30. Horsepower and Torque, Respectively, vs Rotor Speed at Topping and Ground Idle Powers for a Typical Free-turbine Engine

factors influencing the selection of ground idle speed are noise and downwash velocity.

Generally, the ground idle rotor speed of helicopters is 40-50% of normal operating speed. For conservatism of acceleration loads, ground idle speed may be assumed as 40% (the lower value will lead to a more critical peak torque). The following approximation may be used to estimate rotor speed when power is specified:

$$\frac{N_o}{N^*} = \sqrt[3]{\frac{\text{ground idle power}}{\text{aerodynamic rotor load}}} \qquad (4\text{-}20)$$

where

N_o = ground idle rotor speed, rpm
N^* = flight rotor speed, rpm

Aerodynamic load in this equation is the power absorbed by the rotor at minimum collective pitch and 100% flight rotor speed N^*.

4-7.1.2 Determination of Peak Engine Torque

In order to determine the peak transient torque developed in a rapid engine acceleration, an estimate of rotor speed change ΔN during engine acceleration must be made. From the basic relationship for angular acceleration, the equation describing the expression for ΔN is derived:

$$\Delta N = Q_{ave} \frac{\Delta t}{I_{total}} \left(\frac{30}{\pi}\right) \text{, rpm} \qquad (4\text{-}21)$$

where

I_{total} = total rotor/drive system mass moment of inertia, slug-ft^2

Q_{ave} = average torque acting on the rotor/drive system, lb-ft

Δt = engine acceleration time, sec

The engine acceleration time is listed in engine model specifications. The maximum allowable time for acceleration from ground idle to maximum power is 10 sec (MIL-E-8593). The engine manufacturer usually adjusts the engine to produce a sea level standard day acceleration of 5 sec or less.

The average net torque available for acceleration of the rotor is the average engine torque minus the average aerodynamic rotor torque. The aerodynamic torque Q_A can be estimated by:

$$Q_A = (QCP)\left(\frac{N}{N^*}\right)^2 \bar{T} \text{, lb-ft} \qquad (4\text{-}22)$$

where

QCP = ratio of aerodynamic rotor torque at minimum collective pitch and 100% rotor speed to $\bar{T}$, dimensionless

$\bar{T}$ = total (one or more engines) engine torque at maximum rated power, lb-ft

The engine output torque during a rapid acceleration can be estimated by a function of time to the nth power t^n where n varies from 4 to 6. Thus, the engine torque Q_E between ground idle Q_{GI} and its peak value Q_P can be approximated by:

$$Q_E \approx Q_{GI} + Q_P \left(\frac{t}{\Delta t}\right)^n \text{, lb-ft} \qquad (4\text{-}23)$$

With this information, the rotor speed change during the engine acceleration can be estimated and, by use of the typical torque/speed relationship given in Fig. 4-30(B), the peak torque Q_P can be approximated.

The distribution of the peak transient torque in a drive system that connects two or more rotors is proportional to the inertia of each rotor/drive and to the aerodynamic load of each rotor. Thus, the total engine torque $Q_{R/D}$ transmitted through each rotor drive shaft is defined by

$$Q_{R/D} = (Q_E - Q_A)\left(\frac{I_{R/D}}{I_{total}}\right) + Q_A\left(\frac{QCP_{R/D}}{QCP_{total}}\right) \text{, lb-ft} \qquad (4\text{-}24)$$

where

$I_{R/D}$ = single rotor/drive system mass moment of inertia, slug-ft^2

$QCP_{R/D}$ = QCP of single rotor/drive system, dimensionless

During rapid accelerations, the torque distribution usually is dependent more upon inertia ratio than upon aerodynamic loads.

For rapid engine accelerations, the peak torque Q_P transmitted by the rotor shaft is estimated by use of Eq. 4-24, with Q_E replaced by Q_P.

Because the maximum available torque of turboshaft engines increases as ambient temperature decreases, more power is available and engines accelerate faster at lower temperatures. Therefore, transient torques during accelerations also will increase with decreasing ambient temperature, as discussed briefly in par. 4-7.1. The temperature at which maximum power occurs can be determined from the engine specification. This temperature usually is dependent upon the engine control and is not necessarily the lowest ambient temperature for which the engine is qualified.

To determine the peak transient torque loads at the temperature at which maximum power is produced, the pertinent parameters must be recalculated. Engine acceleration time Δt can be approximated as a function of the absolute temperature ratio as follows:

$$\Delta t_{T_0} = \theta \Delta t_{std} \text{, sec} \qquad (4\text{-}25)$$

where

Δt_{T_a} = acceleration time at any ambient temperature T_a, sec

Δt_{STD} = acceleration time at standard atmospheric temperature, (59°F or 519°R), sec

θ = ratio of absolute temperatures, $T_a/519$, dimensionless

4-7.1.3 Rotor Blade Loads

When high transient torques are applied to a rotor at low rotor speeds, and where centrifugal stiffening effects are small, the rotor blades will be subjected to high inplane bending. To estimate the blade loads, the torque acting at the rotor hub must be determined. The equation used for calculating rotor hub torque Q_{hub} is

$$Q_{hub} = (Q_E - Q_A)\left(\frac{I_{rotor}}{I_{total}}\right) + Q_A\left(\frac{QCP_{R/D}}{QCP_{total}}\right), \text{lb-ft} \quad (4\text{-}26)$$

Through replacement of Q_E by Q_P in Eq. 4-26, the peak torques to which the rotor hub will be subjected during a rapid engine acceleration can be estimated.

Criteria for distribution of loads on articulated or rigid rotors are provided in MIL-S-8698. For the purpose of analyzing rotor acceleration loads, criteria from this specification are interpreted as follows:

1. Rigid rotors. For rotor blades without drag hinges, Q_{hub} will be distributed equally to each blade.

2. Articulated rotors. The inertia load will be distributed to each blade but the aerodynamic load will be distributed to any two blades of a three-bladed rotor or any three blades of a four-bladed rotor or any four blades of a five-bladed rotor. The duration of the unbalanced aerodynamic load should be considered as the time per revolution divided by the number of blades. At the start of acceleration from ground idle, centrifugal effects can be so low as to allow the blades to strike the drag stops. In this case, the momentum change caused by that collision must be considered and distributed similarly to the aerodynamic load.

4-7.2 ROTOR BRAKING LOADS

Generally, rotor brakes are designed to decelerate the rotor quickly during rotor shutdown. Also, a rotor brake often is required for use on parked aircraft to prevent wind-caused rotor rotation or rotor rotation while operating the engine at its idle speed. For these conditions, the torque rating required of the brake will be low, usually less than 10% of the maximum torque experienced during rotor acceleration to flight speed.

The rotor-braking load to be considered involves bringing the rotor to rest from a low rotor speed (ground idle or lower). Normally, the brake will decelerate the rotor in 15-30 sec by the application of a constant pressure to the braking surface. Higher pressure may be required when the rotor is to be held during engine startup. Under certain conditions, application of the brake at low rotor speed will cause the transmission to stop almost instantaneously, while the rotor blades will continue their motion until they strike the lead stops. On rigid rotors the blade motion will continue until the kinetic energy of the rotor is absorbed by blade chordwise bending. In either case, the inertia loads of the rotor must be considered.

The maximum torque transmitted by the brake is a function of initial speed, length of application, and frequency of application. The brake manufacturer's specification usually provides sufficient information from which to determine these torques.

4-8 SPECIAL LOADING CONDITIONS

4-8.1 HARD-POINT LOADS

4-8.1.1 External Store Installations

When hard points for the attachment of external stores are required, the stores to be considered will be specified by the procuring activity. Normally, a variation of loading conditions and stores must be considered, including but not necessarily limited to auxiliary fuel tanks, spray tanks, smoke dispensers, bombs, gun pods, mine and flare dispensers, rocket launchers, and missiles. The contractor *shall* insure the structural integrity of:

1. Hard-point fittings and their supporting structure

2. Store and pylon fairings

3. Control surfaces and adjacent helicopter structure affected by the presence or operation of the stores.

4-8.1.1.1 Design Criteria

All external store installations, whether latent (fuel tanks, droppable munitions) or active (guns, rocket launchers, power-ejected munitions), must be substantiated for static and dynamic loads, as applicable. The following loads must be considered:

1. Flight loads: (par. 4-4.1)

2. Landing loads: (pars. 4-5.1 and 4-5.2)

3. Taxi and ground-handling loads: (par. 4-6)
4. Ejection loads (power-ejected stores)
5. Fatigue loads due to gun firing, rocket launching, etc.
6. Crash loads: (par. 4-5.3).

Ejection loads applicable to power ejected stores should be provided in the system specification. Fatigue loads applicable to external stores installations are discussed in par. 4-8.1.1.7.

If crash loads on external stores are not specified, the crash load factors used *shall* be the same as the crash load factors specified as applicable to other mass items (transmission, engine, etc.) which would endanger the occupants of the helicopter in the event of a survivable crash.

4-8.1.1.2 Weight

As a rule, static loads for hard points must be developed for the maximum weight of the stores to be supported. However, if the sequence in which the disposable load is expended or the combination of store loading with rotor capability or maneuver rate are such that the loads on the hard points could become critical with partial loading, then these combinations must be considered in the development of the static loads.

4-8.1.1.3 Center of Gravity

If provisions are made for boresighting, if partial store loads can be carried, e.g., rocket pods not fully loaded, or if flexible weapons are to be mounted on the hard points, static loads must be developed for the extreme travel of the CG of the store.

4-8.1.1.4 Methods of Analysis

It is usual for the geometry of the fitting installation and the structural arrangement of the store pylon to result in a statically indeterminate structure. There are several generally accepted methods of determining the loads on such structures, and any of them may be used, including Refs. 22, 23, and 24. If the store is suspended by a rack, the suspension loads *shall* be determined in accordance with the appendix of MIL-A-8591. The load factor diagrams of this specification are not applicable to helicopters.

4-8.1.1.5 Extent of Substantiation

The local support structure for external stores must be substantiated for the loads, shears, bending moments, and torsions resulting from the loads on the external stores under all conditions considered in par. 4-8.1.1.1. The local support structure for the stores is those pylons, frames, fittings, skins and other helicopter structural members for which stores loads are critical, to the point(s) at which other loading conditions are critical.

For crash conditions, the supporting structure for the hard-point installation must be substantiated up to the point that the loads are resisted by shear forces in a beam, bulkhead web, or skin; there is no requirement to show a balanced structure beyond that point.

4-8.1.1.6 Aerodynamic Load Determination

The following aerodynamic loads must be determined on a preliminary basis for the full range of potential flight attitudes:

1. Lift, drag, and associated bending moments on the stores
2. Pressure distribution in the region of the stores
3. Aerodynamic buffeting on control surfaces in the wake of the stores.

Analytical methods, described in par. 3-2.1, or wind tunnel testing, described in par. 3-2.3, may be employed for load determination. Wind tunnel data may be derived from prior tests of comparable configurations or from preliminary tests of the proposed configuration. These aerodynamic loads must be added to the inertia loads to determine critical static loads, to determine fairing and closure loads, or to estimate the fatigue lives of structures subject to oscillatory loads.

4-8.1.1.7 Dynamic Loads

When external stores are installed, dynamic loads in the helicopter may be generated from one or more of the following sources:

1. The response of the stores to rotor-induced vibration
2. The reactions to store activation (gun firing, rocket launching, flare dispensing, etc.)
3. The control-surface loads generated by aerodynamic disturbances induced by the stores
4. The blast overpressures of store activation impinging upon adjacent structure.

All external store installations *shall* be substantiated for the dynamic loads that arise from any or all of these sources. Adequate consideration must be given to all known factors that influence these loads, as described in the discussion that follows.

Vibratory loads arising from response of the stores to rotor-induced vibration are influenced by the following variables that must be known or assumed in order to determine the resulting dynamic loads:

1. The weight and the CG of the store, and their

variation with the release of expendable load in part or whole

2. The mass moments of inertia of the store about its three principal axes

3. The stiffness of the internal structure of the store, particularly in the area contacted by the sway-brace pads

4. The natural frequency of the hard-point supporting structure

5. Variations in the gross weight and CG of the helicopter and in rotor speed.

Reaction forces are influenced by factors such as recoil loads, total impulse, and rate of fire of weapons, and by the natural frequency and damping characteristics of the hard-point supporting structure. These factors are discussed in detail in pars. 4-8.1.2 and 4-8.1.3.

The shapes of the stores, the speeds and maneuvers of the helicopter, and the shapes and locations of the control surfaces are the principal factors influencing the loads generated by aerodynamic disturbances. Wind tunnel data are the best means of confirming the characteristics of the airflow in the area of interest because analytical predictions cannot account reliably for the effects of flow interferences.

Loads resulting from the impingement of blast overpressures upon adjacent structures are influenced by the pressure and velocity of the shock wave, the location of adjacent structure, and the natural frequency and damping characteristics of the structural elements. These factors are discussed in detail in par. 4-8.1.4.

4-8.1.1.8 Flight Load Determination

Flight vibratory loads *shall* be determined finally by a flight load survey (including weapon firing where applicable) as described in Chapter 8, AMCP 706-203. For preliminary design, vibratory loads must be determined or established by rational means and the effects investigated by means of appropriate analysis.

A dynamic model of the airframe structure that includes at least the first three flexible modes in the vertical and lateral directions must be computed from the estimated weight and stiffness distributions. A maximum of 2% of critical damping must be used in the structural model. The mass and inertia of the store and the spring rate and damping of the support system also must be included in the dynamic model. Unless intentional support system damping is provided, a maximum of 2% of critical damping in the support system must be in the analysis. The model must be subjected to oscillatory loads simulating the loads imposed upon the airframe by the rotor. The responses of the store and the oscillatory loads in the support system must be calculated. For stores with variable mass—such as mine dispensers, fuel tanks, or rocket launchers—the mass that produces the coincidence between the predominant main rotor harmonic frequencies and the natural frequency of the support system must be computed and reported.

Substantiation of local support structure for external stores for vibratory (fatigue) loads is required to the same extent that substantiation is required for crash loads (par. 4-8.1.1.5).

4-8.1.2 Reaction Forces

In order to analyze the effects of jettisoning stores, some consideration of helicopter stability and control parameters must be included. In general, an asymmetric jettison will create an offset CG and a rolling moment that will tend to aggravate any potential clearance problem between the store and elements of the helicopter. Therefore, the jettison impulse should be sufficient to provide a safe clearance margin, but should be minimized as much as possible in order to relieve the rolling moment input and the support loads. This specific impulse may be determined in the following manner:

1. Determine the force developed against the store by the jettison mechanism at any instant during the jettison sequence.

2. Plot this force versus the time of the jettison sequence and integrate the curve thus developed.

For developmental power-jettison mechanisms, an estimate based upon tests can be used and the expected range of variations can be stated conservatively.

Reaction loads to the hard points may be developed from the known or assumed power-jettison force. However, helicopter inertia relief must be ignored in calculating these reaction loads to include the possibility of symmetric jettison.

4-8.1.3 Firing Frequency Dynamics

The installation of repeated-fire weapons on helicopters has in many cases been a trial-and-error process, with design modifications following test results until the system was acceptable. Some weapons—such as the 40 mm XM129 Grenade Launcher and the 7.62 mm XM134 Machine Gun—have required little if any dynamic analysis. For larger weapons—such as the 20 mm M61 Cannon—the recoil adapters were developed with consideration of the firing rate and the dynamic characteristics of the support system. Dynamic analysis usually can be used effectively in tailoring the recoil system, in selecting the firing rate of the weapon system, and in estimating the design fatigue loads.

Generally, because of the complex structure of the support system, reaction forces can be determined accurately only by firing tests; however, an analysis using the impulse of the projectile, the firing rate, and the calculated dynamic characteristics of the support system can give reasonable design parameters. The reaction forces usually are significantly lower for a helicopter support system than for a hard stand. The only exception is when the firing rate or one of its first few harmonics coincides with a lightly damped natural frequency in the support system. The support structure should be designed to avoid resonance of its fundamental modes with the firing rate and, if possible, with the first two harmonics of the firing rate. If gun design parameters require that the weapon fire at a fundamental frequency of the support system, provisions for damping the support system response should be considered.

4-8.1.4 Blast Overpressures

If active weapons are close enough to the helicopter for blast overpressures to impinge upon adjacent structural elements, the affected areas must be substantiated for this additional loading. This substantiation may be by test or by a combination of test and analysis. The aim of the analytical procedure is to determine the loads and stresses in the structural elements that result from their dynamic responses to the blast loading. An analytical approach usable for design purposes is described in Ref. 25. This reference also presents an excellent bibliography of additional work in the field.

In order to determine the desired dynamic response characteristics, two important parameters must be known or determined: (1) the free-field blast pressure, and (2) the time of arrival of the blast wave as it sweeps across the structure under consideration. For guns, including those with muzzle brakes, blast diffusers, or blast suppressors, the analytical procedure described in Volume I of Ref. 25 may be used or the data may be determined experimentally. For rocket launchers, missiles, or power-ejected munitions, the data must be determined experimentally because no adequate analytical procedure exists. The RFP may provide such available test data as are pertinent.

With these two parameters defined, the dynamic responses of specific structural elements may be determined. For the cases that follow, the analytical procedures described in Volume II of Ref. 25 may be employed and the results corroborated during qualification testing:

1. Elastic response of a rectangular membrane
2. Response of a simply supported circular plate
3. Large-deflection, out-of-plane response of a pin-jointed framework
4. Elastic-plastic response of a rectangular plate
5. Dynamic response of a beam with coupled bending and torsion
6. Coupled bending torsional response of rotor blades subjected to blast loading.

Alternatively, the dynamic responses of specific structural elements may be determined experimentally by testing representative sections of the structure. The results of either approach must be corroborated during qualification testing.

The effects of repeated exposure to the structural-response loads must be considered when locating items of equipment whose operating characteristics might be affected by such exposure. Such equipment must be far enough from the interior side of the exposed structural element or its mounting structure to preclude any operating failure or deficiency due to such exposure.

4-8.2 EXTERNAL CARGO

The helciopter is unique in its usefulness for carrying external cargo. Many shapes and sizes of external cargo, both rigidly attached (but removable) and sling loaded, have been lifted and transported. Most external cargo transported by helicopters is handled by an external sling attached to the airframe structure. A variation in application of the external sling is the use of a hoist to lift loads into the cabin. Also, the lifting and aerial towing of downed aircraft have proven to be effective applications for the external cargo sling. With the development of helicopters such as the HLH which carry all cargo externally, rigid attachments will become more common.

4-8.2.1 Rigidly Attached External Cargo

This discussion of rigidly attached external cargo is limited to the design requirements imposed upon the helicopter structure. However, these requirements are based upon the possibility that the rigidly attached container will be "man-rated", i.e., a "people-pod". The helicopter attachments must, of course, withstand maximum flight maneuver and landing loads, but the most important considerations are the crash load conditions. During preliminary design consideration also must be given to such things as container location and protection from ground impact or other obstructions for crash protection. The crash condition should be handled as an impact/energy problem, as discussed in par. 4-5.3, instead of by the use of arbitrary load factors. When handled in this manner, the helicopter structure must deform a predetermined amount to absorb the crash energy. In determining the magnitude of the loads for estimates of the degree of collapse required of the airframe, impact velocities must be assumed. These velocities *shall* be those associated with a 95% survivable crash as defined in Ref. 11. A more complete discussion of the crash loads is provided in par. 4-5.3.

4-8.2.2 Sling-loaded External Cargo

The combined loads for designing the sling support fitting will consist of the static weight of the external load and the aerodynamic loads adjusted for inertia and safety factors. Also to be considered is sling-load bounce, as discussed in par. 4-10.6.

4-8.2.2.1 Static Loads

The magnitude of the dead weight sling load required normally is established in the system specification. However, for those cases where the exact load is not specified, and for consideration of growth possibilities, the maximum sling load weight should be equal at least to the difference between the basic structural design gross weight and the minimum design gross weight (par. 4-3). If an alternate design gross weight is specified for the helicopter, the sling load fittings *shall* be capable of accommodating a proportionately higher load.

4-8.2.2.2 Aerodynamic Loads

Aerodynamic loads may be an important consideration for sling-loaded external cargo. Their significance is a function of the shape, volume, and density of the external loads being considered. For high-density cargo the aerodynamic loads usually can be ignored. Vertical loads are experienced from rotor downdraft, and drag and lift loads are encountered in forward flight. Additional drag loads sometimes are imposed artificially to provide flight stability to the slung cargo. The allowable magnitude of the drag loads must be determined from prior test experience, aerodynamic analysis, and wind tunnel test data. The maximum external drag load, once determined, establishes an operational limitation on the aircraft as indicated in Fig. 4-31. This, in effect, limits the amount of drag that must be reacted by the helicopter fitting.

4-8.2.2.3 Combined Loads

The maximum combined sling loads to be applied to the helicopter fitting are determined by vectorially adding the dead weight and the aerodynamic loads. The resultant load then is increased by a dynamic magnification factor of 2.0 to obtain the limit load. The line of action of this load relative to the helicopter fitting should be derived from the flight attitude normal for the helicopter, when operating without sling load, at the flight speed for which the aerodynamic load acting on the sling load is calculated. Alternatively, the line of action can be considered to act within a cone of action of 30 deg from the vertical. The relationship of the helicopter to the vertical depends upon the flight attitude required at the maximum speed with sling load.

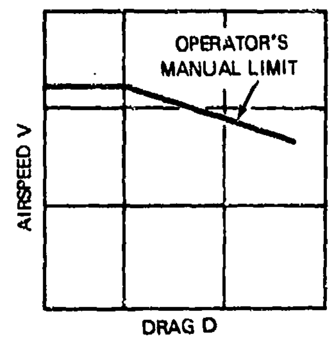


Fig. 4-31. Drag Limits for Helicopter Fittings

4-8.2.2.4 Asymmetrical Loading

Several characteristics of sling-loaded cargo operating conditions make the consideration of asymmetrical loading necessary for design of the helicopter interface fitting. All sling loads are subject to pendulum oscillation and to yaw or a minor amount of directional instability. In order to account for these conditions, the fitting loading direction should be assumed to vary proportionally to the control capability of the helicopter.

4-8.2.3 Lifting and Aerial Towing

Lifting and aerial towing of downed aircraft recently have become important uses for the external sling. The loading applied to the helicopter interface fitting will not be altered beyond that discussed in the previous paragraphs during the preliminary design stage for the towing application.

The primary differences in the loading considerations are those brought about by the aerodynamic characteristics of the towed vehicle. Specifically, these are lift, drag, and flight stability—all of which are considerations for the particular lifting and towing operation.

If the lift characteristics of the towed vehicle introduce a problem, its aerodynamic surfaces can be modified by the temporary installation of spoilers.

Flight stability during the towing operation can be improved by the attachment of a drogue chute. These modifications will increase the drag, thus reducing the maximum permissible flight speed.

4-8.3 FLOORING AND WORK PLATFORMS

Structural design criteria for floors and work platforms depend to a great extent upon the ultimate use of the particular helicopter being developed. This paragraph lists design criteria for distributed and concentrated design loads for the floor and platform areas as well as local loads to be applied at the tiedown fittings. Also discussed are requirements such as interchangeability and replaceability, and the need for ruggedness and durability.

4-8.3.1 Flooring Design Criteria

Lightweight design is a major consideration for helicopter flooring. The lightest weight materials that are capable of meeting the structural requirements, and are economically feasible, *shall* be used. Specific criteria are described in the paragraphs that follow. The type of loading—crew or cargo—will determine the strength requirements.

The personnel restraint and seat load paths normally pass through the floor structure and into the basic helicopter structure. The floor itself, therefore, does not carry the entire seat restraint load, even though the fittings may be located at the floor surface and give partial support to the floor. The ultimate load requirements for these personnel restraint fittings are the applicable crash loads (par. 4-5.3).

The fittings for cargo tiedown also are attached to the basic helicopter structure, directing the tiedown load paths through the floor in the same manner as the personnel restraint fittings.

The same fittings often are used alternately for personnel restraint and cargo tiedown, depending upon the operational mission. A well-executed preliminary design develops a maximum degree of multiple usage for the floor fittings. The design loads applicable to the tiedown fittings are due to flight maneuvers and to the appropriate crash conditions, to prevent shifting of cargo so that occupants of the helicopter are not endangered.

One of the objectives in helicopter floor design is to resolve most of the floor loads into a pattern where the concentrated load-carrying fittings, such as tiedown and seat attachment fittings, also provide the major support for the floor. An approach that has proven to be successful for the preliminary design of helicopter floors consists of surveying the various load arrangements anticipated for the vehicle and then laying out a pattern of a minimum number of fitting attachment points that fits any and all helicopter personnel, equipment, and cargo loading conditions. This would include provisions for cargo load tiedown, seat attachment, personnel restraint, special loading requirements, and any attachment included for special mission requirements defined in the RFP system specification. The fittings should be located in a standard pattern (Ref. 26). When the most efficient fitting arrangement has been established, additional structure as required is provided for the floor.

Cargo loading aids such as tracks and floor pulleys can introduce concentrated loads. Thus, particular attention should be given to distributing the load among the fittings already provided. The loading aids and associated criteria will depend upon the cargo loads for which provision is required; such information normally will be included in the RFP system specification.

Design limit floor pressures *shall* be $75n_z$ psf for crew floors and 300 n_z psf for cargo areas. The load factors n_z *shall* be 3.5 plus increments due to angular accelerations from maneuvers, or the load factor resulting from hard landings up to 15 fps, whichever is greater. In addition, the following criteria *shall* apply.

1. The cargo floor *shall* have a local loading capability for a 50-psi limit load applied to a single 0.5 ft^2 area (8 in. x 9 in. to 3 in. x 24 in.)within any 6 ft^2 area.

2. A cargo floor panel, 18 in. square and supported at two edges, *shall* withstand the impact from a sturdy pine box, uniformly loaded so that the box and its contents weigh 200 lb, dropped from a height of 15 in. above the panel (distance between panel and lowest corner of box) so that one corner of the box strikes the center of the panel. The local deformation in the floor caused by this impact *shall* not exceed 0.3 in. A line between the corner of the box and the contact *shall* be vertical upon impact. The corner radius of the box *shall* not exceed 0.5 in.

3. The cargo floor and loading ramps *shall* withstand, without undue surface wear or evidence of fatigue cracking, the effects of 1000 complete trips in a fixed path of a 1000-lb load applied by a steel wheel 8 in. in diameter and 2.5 in. wide.

4. The cargo floor *shall* have the capability to withstand track and/or tire pressures consistent with the applicable vehicle loading requirement. The applicable single-axle weight loads will be as specified by the procuring activity. The cargo floor and loading ramps *shall* be capable of withstanding the loads resulting

from application of a limit load factor of 2.0 to the loads imposed during loading of vehicles having the maximum applicable axle loads.

4-8.3.2 Work Platform Design Criteria

Work platforms are used for supporting maintenance personnel, tools, and, in some cases, certain helicopter parts; therefore, they are subject to concentrated loads. For preliminary design purposes, work platforms *shall* be capable of supporting a 200-lb man with all his weight on one foot at a load factor of 1.5 without permanent deformation. If the platform is sufficiently large to support more than one person, the load criteria *shall* be increased proportionately. In all cases, allowance for support of tools and the appropriate helciopter parts also *shall* be made.

4-8.3.3 Interchangeability and Replaceability

Floors of helicopters are subject to heavy usage and, therefore, the surface may need periodic replacement. In case a panel of the floor is damaged and requires replacement, it should be replaceable at the organizational maintenance level without the aid of special tools and with a minimum expenditure of time.

As interchangeability of floor parts, fittings, and panels is advantageous, attention should be paid to standard spacings and standard design configurations (Ref. 26).

See par. 11-2 for a more detailed discussion of interchangeability/replaceability.

4-8.3.4 Durability

Floors and work platforms of helicopters must be durable in order to withstand heavy service and rough handling. The materials used in these areas must be resistant to corrosion and deterioration, and any porous material must be treated to reduce moisture absorption. Good drainage should be provided, especially for work platforms where snow or ice might accumulate and create a safety hazard. The preferred surface for a work platform is corrugated metal that is self-draining and inherently skid-resistant.

4-8.4 DOORS AND HATCHES

Doors and hatches must be easy to operate and rugged, and particular care is required to minimize their malfunctioning in the field environment. The application of human engineering principles is an important loading consideration, and MIL-STD-1472 *shall* be used as a guide for personnel loadings during the design of these components.

Doors and hatches include the following closures for all openings provided in the helicopter for entry, egress, and access:

1. Access doors
2. Hinged or sliding canopies
3. Sliding doors
4. Passenger doors
5. Crew doors
6. Cargo compartment doors
7. Emergency doors
8. Escape hatches.

This paragraph discusses the structural design criteria for the doors and hatches including distributed load, concentrated load, and deflection criteria. Requirements for quick removal and replacement during normal operation, and for emergency jettison and/or personnel egress, are specified.

4-8.4.1 Design Criteria

A series of mock-ups and component investigations during the preliminary design development is recommended for achieving optimum door and hatch configurations. Operational design criteria to be considered during this preliminary design stage include:

1. Doors and hatches *shall* withstand the airloads resulting from flight at maximum speed V_{DL} and sideslip angles.

2. Doors *shall* be rugged and resistant to rough handling or lack of maintenance during service operation.

3. Doors *shall* provide easy access to seats and cargo loading areas.

4. Doors and hatches *shall* provide easy egress in case of emergency.

5. Sealing from rain, snow, and external environment *shall* be provided.

6. Mechanisms *shall* operate under all applicable environmental and weather conditions.

7. Handles *shall* be easy to operate but must resist being twisted or broken off inadvertently.

8. All doors that may restrict the egress of personnel *shall* be easily removable and the opening clean and clear in times of emergency.

9. Doors and hatches *shall* withstand applicable wind gusts in either the open or closed position.

10. Doors *shall* be so arranged and located that the

aerodynamic loads in forward flight will not force them open.

11. Production tolerances and structural deflections resulting from limit loads *shall* not interfere with satisfactory operation of the doors and hatches.

4-8.4.1.1 Concentrated Loads

All concentrated loads associated with the use and operation of doors and hatches terminate in the latches and hinges. The sources of these loads are:

1. Open canopy or open vent door during approach or taxi operation
2. Gusts
3. Outward push from personnel
4. Handles (normal and emergency)
5. Seal loads
6. Air loads
7. Rough handling.

If a sliding or hinged canopy is used, it should be designed to withstand an air load from taxi operations of up to 60 kt. The closing mechanism should provide for closing of the canopy in not more than 10 sec with one motion by one hand. All sides *shall* be locked. The loads for operating this opening and closing mechanism should be based upon the criteria of MIL-STD-1472. Additional and more detailed design criteria for an overhead canopy may be obtained from Ref. 27. These same criteria apply to a vent door if such a door is provided.

All doors that are subject to damage by ground gusts *shall* be provided with a means to absorb the energy resulting from a 40-kt ground gust occurring during opening or closing. These doors and access doors or panels *shall* be provided with a positive hold-open feature that will withstand gust loads to 65 kt when the door or panel is in the open position and unattended.

Due to possible inadvertent loading by personnel, doors into personnel compartments should be capable of withstanding a load of 300 lb without opening. This load is assumed to be applied upon a 10 in.^{2} area at any point on the surface of the door. Doors into other compartments also may be subjected to similar loading conditions. The magnitude of the load applied depends upon the use and application of each door under consideration. The 300-lb load *shall* be applied in all cases if a lower requirement cannot be justified.

The normal door handle load for operation should not exceed 10 lb and emergency handles should be designed to operate at 10-30 lb. These emergency handles *shall* withstand an emergency load of 300 lb. The handles *shall* be designed to break at 300+ lb, and they should not be stronger than the internal parts that they operate. This precludes continued attempts by trapped personnel to open an emergency door after the mechanism has failed, and causes them to seek alternate exits. Additional criteria for emergency handle loads and design may be obtained from Ref. 28.

Seal loads may originate from either static seals or pressurized seals. Pressurized seals have a greater potential for creating concentrated loads on the hinges and latches of the doors. Pressurization normally is supplied by the engine compressor system or by an auxiliary power system. The seal must be analyzed for maximum loads, and these loads should not cause an unlatching tendency.

The air loads on doors and hatches for helicopters probably are minimal when compared to the many personnel-oriented loads. The air loads, however, should be investigated, including the application of the appropriate gust criteria as outlined in par. 4-4.3.

Consideration of rough handling of doors is recommended during the preliminary design stage. Experience from past field operation is most helpful in estimating loads. A checklist of rough handling conditions with estimated resultant loads is one method of examining these possible problems. The list could include items such as:

1. Using an opened door as a step
2. Failing to latch the door in a strong wind
3. Testing the strength of the handle as a curiosity
4. Wind loads from other helicopters being run up or taxied nearby or flown close overhead.

4-8.4.1.2 Distributed Loads

Cargo doors *shall* be capable of being opened at all flight speeds up to 80 kt EAS. It must be possible to perform the function of opening the doors while in flight at all speeds from 0 kt (hover) to 80 kt EAS; if the doors are opened at low speed, flight at speeds greater than 80 kt EAS should be possible. The highest speed practicable without adverse effect upon the design is desired.

Personnel compartment doors, and particularly the doors to troop compartments, *shall* be capable of withstanding the loads resulting from flight in the doors-open configuration at all flight speeds up to a minimum 110% of cruise speed (the higher speed at which specific range is 99% of maximum). The highest speed in the doors-open configuration that is practicable without adverse effect upon the design is desired.

4-8.4.1.3 Deflection Criteria

The most important deflection criterion is the assurance that deflection from any and all sources *shall* not prevent the door or hatch from performing its task as a satisfactory closure. Also, the door or hatch *shall* be free to open and close under all operating conditions and especially under emergency conditions. This means that adequate clearance *shall* be provided where deflections are anticipated. In those cases where deflections are present, they *shall* not tend to disengage the latching system but, instead, either should increase the engagement without interference or should not affect it.

The following sources of deflections that could affect the performance of doors and hatches are presented and discussed briefly as a checklist for preliminary design consideration:

1. Flight and landing gear loads. Deflections from flight loads and landing gear loads are considered normal and *shall* not be very large.

2. Seal pressurization. Deflection from seal pressurization must be checked and the seal *shall* be located so that any deflection resulting from its operation is in a direction to improve latch engagement.

3. Manufacturing. Permanent deflections sometimes are encountered in the manufacturing process as a result of deforming parts to meet the established contour.

4. Overloads. Permanent deflections may be encountered from overloads due to hard landings, or other abnormal operating conditions. Effort should be made during the preliminary design stage to establish basic door and hatch configurations that are inherently forgiving under unusual deflection conditions.

4-8.4.1.4 Production Tolerance

The basic preliminary design of the doors and hatches should include the consideration of practicable production tolerances. Manufacturing representatives should be consulted to determine the tolerances that can be held reasonably when door and frame structures are made in separate jigs. There must be sufficient clearance between the door and frame so that tolerance buildup cannot cause interference between the door, or any attachment to the door, and the frame. The minimum clearance never should be less than 0.20 in.

Flushness and gap tolerance requirements between the doors and the helicopter skin become a matter for performance consideration and should be allowed for in the basic door configuration established during the preliminary design stage. The exact limits and methods for achieving these limits must be worked out during the detail design and are discussed in Chapter 11, AMCP 706-202.

4-8.4.2 Removal and Replacement

Fast removal and replacement of doors and hatches should be possible. Full consideration of this requirement during the preliminary design is necessary for the establishment of satisfactory configurations.

4-8.4.3 Emergency Jettison and/or Personnel Egress

Most personnel compartment doors are equipped with jettison devices. These devices are periodically checked as an operational and maintenance procedure.

MIL-STD-1472 states that the simplest possible escape mode consistent with safety and effectiveness *shall* be provided. Ref. 28 states that emergency hatches and doors should be designed to be opened quickly, operated easily, and, wherever possible, jettisonable. These, and other criteria pertinent to emergency evacuation of personnel, are discussed in further detail in par. 13-2.2.1.

4-8.4.4 Size of Openings

The size of the openings for helicopters should be as small as possible, yet large enough to permit efficient performance by personnel and to provide adequate convenience and safety. All openings that personnel may contact must have well-rounded and smooth edges with no obstructions.

4-8.4.4.1 Access Openings

Access openings that provide for adjusting and handling interior items *shall* be sufficiently large to permit the required operations and, if possible, to provide an adequate view of the components being worked on. The dimensions of the openings *shall* be no less than those shown in MIL-STD-1472 for the appropriate access requirements for arms, hands, or fingers. Allowance should be made for maintenance personnel wearing cold weather clothing.

4-8.4.4.2 Entry Openings

The size of entries often is established by the RFP or by the size of the equipment and cargo to be transported. Such openings should be sized to provide adequate clearances and to permit convenient and expeditious loading and unloading of the specified equipment and cargo. Those openings that are provided primarily for personnel entry *shall* be sized to accommodate the 95th percentile man (MIL-STD-1472) with adequate clearance for comfort and convenience.

4-8.4.4.3 Egress Openings

The primary concern for size of egress openings is related to emergency exit. The emergencies to be considered include aerial escape, ground evacuation, and ditching. These requirements are discussed in par. 13-2.2.1.

4-8.5 STEPS AND HANDHOLDS

In order to achieve the optimum compatibility between the equipment and human performance, conveniences such as steps and handholds must be provided to make possible the best use of the equipment. Well-placed steps and handholds will enable maintenance personnel to accomplish necessary maintenance quickly, safely, and effectively. This applies especially to helicopters because a large proportion of their high-maintenance components and equipment is located relatively high above the ground when the vehicle is parked.

There are virtually no specifications available that provide detailed guides for the provision and design of steps and handholds for helicopters. The most appropriate source of general guidance is MIL-STD-1472.

4-8.5.1 Steps

After the need for a step is established, the requirements of size, strength, and other considerations next must be established. Sizing should be provided so as to accommodate the 2nd to 98th percentile anthropometry summarized in MIL-STD-1472.

4-8.5.1.1 Size and Clearance Requirements

There are three general types of steps, all of which are associated with different size and clearance requirements. The type—internal recessed, external fixed, or removable—selected for a particular installation will depend upon its weight, cost, and other usage and system considerations.

4-8.5.1.2 Strength Requirements

The strength requirements of the steps and their mounting will be established by the RFP and/or the proposed application of the preliminary design. Unless otherwise specified, the weight of the personnel for which the steps and associated fittings are designed should be assumed to be the 95th percentile, as shown in MIL-STD-1472.

A limit load factor of 2.0 must be assumed, together with the standard ultimate factor of 1.5. In cases where a lower limit load can be justified, a reduction of the 2-g limit is acceptable. The structural design of all steps *shall* allow for use by two men each applying his weight on one foot, unless it can be shown that two cannot use the same step at the same time.

4-8.5.1.3 Other Considerations

Other considerations for the preliminary design stage of steps for helicopters are concerned primarily with the safety aspects. For all steps, attention must be given to the provision of an effective nonskid surface. This can be accomplished by several methods, including the application of nonskid material per MIL-W-5050 and MIL-W-5044. In order to retain the nonskid condition provided in the design, consideration must be given to the possible environmental conditions, and provision should be made for features such as drainage and adaptability to easy clearing of ice and snow or other environmentally caused obstructions.

If removable or hinged steps are used, it is mandatory that a means be provided for assuring their return to the proper flight operational position prior to takeoff. In cases where the step itself is a protuberance, it should be identified by noticeable color coding or the equivalent.

4-8.5.2 Handholds

The need for and location of a handhold are best established by experience. Thus, use of a mock-up or experience on a similar installation is essential.

4-8.5.2.1 Size and Clearance Requirements for Handholds

There are a number of shapes applicable for use as handholds, among which are the towel bar, T-bar, J-bar, and recessed and knob shapes. Recommendations for dimensions and clearances for these various shapes are given in MIL-STD-1472.

The size of the handhold should be in accordance with the purpose for which it is provided. It may be designed for grasping by two hands, by one hand, or by two fingers. Two-handed handholds are not common for helicopters, but both the one-hand and two-finger configurations are useful.

4-8.5.2.2 Strength Requirements

The strength requirements of each handhold should be determined on an individual basis, with consideration of circumstances such as the possible mechanical force advantage. The first consideration is the requirements of the RFP; then other information from experience and testing should be considered. MIL-STD-1472 may be used as a general guide for typical arm and hand forces.

For preliminary design purposes, unless more accurate information is available, the strength of the handhold should be based upon the weight of the 95th percentile U.S. Army personnel (MIL-STD-1472). The handhold should be designed for a limit load factor of 1.0 with the normal ultimate load factor of 1.5.

Local circumstances may alter these load requirements. If a two-finger handhold is being provided, the design load may be reduced accordingly, or, if the handhold is positioned so that an additional dynamic load could be applied, the strength of the fitting must be increased.

4-8.5.2.3 Other Considerations

Often, the most convenient handhold is a portion of the structure that already is available. The utilization of available surfaces and projections as handholds should be encouraged, but the structure supporting these surfaces and projections must allow for the additional loads.

4-9 STRUCTURAL SUBSTANTIATION, ROTOR, DRIVE, AND CONTROL SYSTEMS

Substantiation of the structural adequacy of a helicopter and of its compliance with the criteria outlined in the preceding paragraphs is accomplished by analysis and/or test. The required analyses consist of the calculation of stresses resulting from the application of appropriate loads to the various components of the structure and, by comparing these stresses with those allowable, the determination of individual margins of safety. The development of the basic loads and the substantiation of the rotor, drive, and control systems in accordance with pertinent criteria is discussed in the paragraphs that follow. The substantiation of the airframe structure, including installation and support of all helicopter subsystems is discussed in par. 4-10.

4-9.1 ROTOR LOADS—STEADY AND UNSTEADY

4-9.1.1 Airloads

In producing the desired lift and propulsive forces, the rotor blades encounter time-varying dynamic and aerodynamic loads that cause steady and vibratory stresses, aircraft vibration, and noise and that can lead to blade instabilities. It must be assumed that the airload and dynamic load distributions are in equilibrium.

When a helicopter is in forward flight, the airload distribution on a rotor blade varies substantially both along the blade radius and around the azimuth. Fig. 4-32 shows a typical variation of the normal force on a blade as it rotates. It is evident that the various peaks and valleys in the spanwise airload distribution produce significant harmonic content, causing blade stress and aircraft vibration. The sources of the blade airloads are numerous and complex. For ease of discussion, they are grouped into four areas:

1. Impressed blade pitch
2. Airflow
3. Rotor position and motion
4. Airfoil characteristics.

4-9.1.1.1 Impressed Blade Pitch

There are three basic sources of impressed blade pitch: blade twist, control inputs, and pitch coupling.

1. Blade twist. Negative twist, in which the outer end of the blade has a smaller pitch angle than the root end, is used to provide a more uniform lift distribution along the blade. The redistribution of load caused by twist generally produces higher vibratory stresses for higher values of negative twist. For a possible explanation of this phenomenon, see Fig. 4-33. The primary contributor to the vibratory bending stresses is the dissymmetry between the loads on the advancing and retreating rotor blades in forward flight. Blades with high negative twist carry negative lift outboard and positive lift inboard on the advancing side, and carry positive outboard lift on the retreating side of the rotor disk, producing significant harmonic excitation of the blade. The loading distribution for an untwisted (0-deg twist) blade is quite different; the loading on the advancing side is similar to that on the retreating side, thus reducing the dissymmetry and the resultant vibratory blade stresses.

2. Control inputs. Main rotor controls generally include both collective pitch and cyclic pitch, producing a steady pitch angle and a one-per-rev variation in pitch angle on the blade, respectively. Tail rotor control generally is limited to collective pitch. Both pitch inputs lead to airloads in all harmonics due to interaction with other parameters.

3. Pitch coupling. A third source of impressed blade pitch is the coupling between blade pitch and other degrees of freedom of the rotor. The principal coupling terms are pitch-flap δ_3, and pitch-lag α_1, in which blade pitch varies with blade flapping or coning, and with blade lagging or hunting, respectively. Pitch coupling also exists with flap bending/torsion and chord bending/torsion degrees of freedom of the blades.

4-9.1.1.2 Airflow

Two prime factors influencing blade airloads are forward velocity V and rotor tip speed ΩR. These are related by the advance ratio $\mu = V/(\Omega R)$, which denotes the size of the reverse flow region of the rotor disk. This region forms a circle centered at azimuth angle $\psi = 270$ deg with a diameter of μR (see Fig. 4-34). Reverse flow can be significant in that air flowing over the blade from the trailing edge can produce large pitching moments about the blade pitch axis. For normal helicopter advance ratios, the dynamic pressure in the reverse flow region is low, and this effect is not significant. However, for compound configurations in which rotor speed will be lowered to give advance ratios approaching 1, reverse flow effects will be significant.

For most flight conditions, the dynamic pressure is created primarily by the tangential velocity of air over a blade. Analyses generally have included the vertical component of velocity, which will be discussed later, but have neglected the radial component. Recent research has indicated that radial flow may be significant for some conditions. For the primary velocity component at a blade section, tangential velocity U_T, the magnitude at a point A on the blade, at radius r from the hub, is given by

$$U_T = \Omega r + V \sin \psi$$
$$= \Omega R\left(\frac{r}{R} + \mu \sin \psi\right), \text{ fps} \qquad (4\text{-}27)$$

where

ψ = blade azimuth angle, deg

Because dynamic pressure is proportional to U_T^2, there are one- and two-per-rev variations in dynamic pressure due to the $\mu \sin \psi$ term. For a given tip speed these harmonic components become larger at higher advance ratios.

The variations in tangential velocity around the azimuth also affect airloads by varying the effective angle of attack along the blade. By neglecting radial

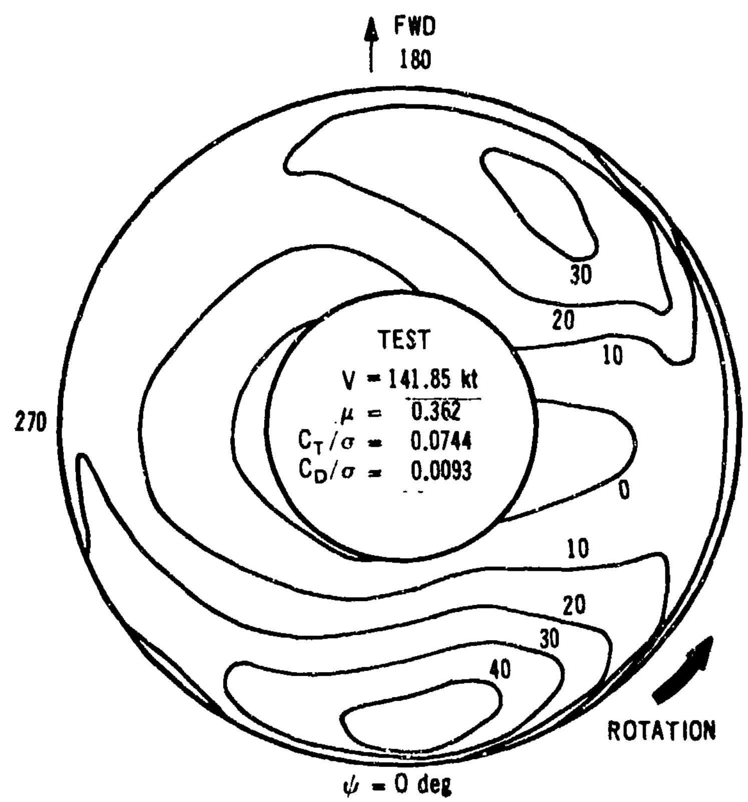


Fig. 4-32. Contour Map Over the Rotor Disk of Test Airloads, lb/in.

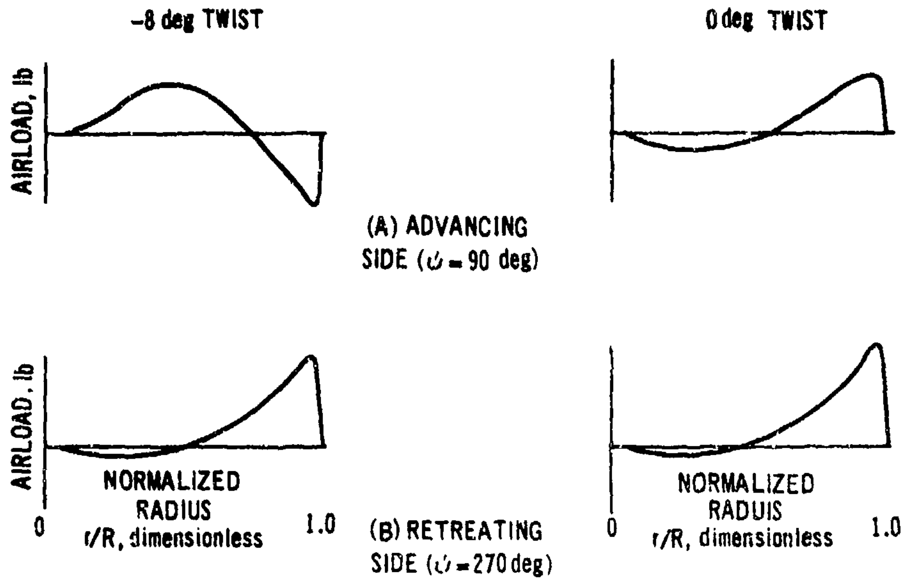


Fig. 4-33. Effect of Blade Twist on Loading Distribution

flow effects, this change in angle of attack can be determined by considering tangential airflow and the flow vertically up (inflow) through the rotor disk. The net airflow is not parallel to the rotor disk but is in a direction given by the vector sum of the inflow U_P and the tangential velocity U_T. This has the effect of decreasing the angle of attack by an angle $\phi = \mathrm{Tan}^{-1}(U_P/U_T)$. Because both U_P and U_T vary azimuthally and radially, the angle ϕ also will be a function of azimuth and will have many significant harmonics, the number depending in part upon the rotor inflow model used.

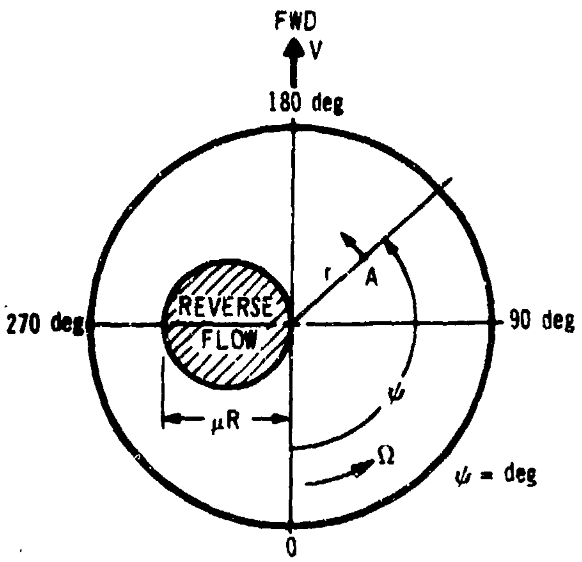


Fig. 4-34. Rotor Plan View (Looking Down)

For most design applications, uniform inflow over the rotor disk is assumed. The value of the induced inflow is given by classical momentum theory (Ref. 4). In reality, the inflow over the rotor disk is not uniform; the strength of the vortices in the wake of each blade determines the induced inflow distribution. The location of the tip vortex for each blade of a rotor at low and high speeds is shown in Fig. 4-35. The presence of the wake elements near the rotor disk affects inflow especially in the immediate vicinity of the vortex. Computer programs for calculating the nonuniform inflow over a rotor disk have been developed (Refs. 29 and 30). These have assumed that the wake is as shown (see Fig. 4-35); i.e., the wake does not distort due to vortex interaction. Research now is under way to define the rotor inflow when wake distortions are taken into account (Ref. 31). The wakes move due to vortex interaction as shown (see Fig. 4-36). Distortion effects may be important for compound configurations, in which the wake remains very close to the rotor disk due to the nearly zero rotor shaft tilt when the rotor is not required to supply propulsive force.

In addition to the effect of rotor wake upon inflow, there are other effects that generally are not included in routine design analyses but that can affect the rotor inflow. The presence of an airframe and propellers will alter main rotor inflow (Ref. 32). In the case of the tail rotor, the influence of the rotor wake and flow over the main rotor hub, as well as blockage due to the tail pylon, can be significant factors (Refs. 33 and 34).

4-9.1.1.3 Rotor Position and Motion

Both the U_P and U_T terms are affected both by the actual blade location in space and by the motion of the blade. The forward velocity of the aircraft must be corrected by the sum of the rotor shaft inclination to the flight direction and the local blade flapping angle to arrive at the local U_P and U_T components of forward velocity. In addition, the out-of-plane or flapping velocity of a blade element affects the U_P term, while lag velocity affects U_T. The flapping velocity term is a significant term for providing aerodynamic damping of the blade.

It would require a lengthy derivation to define how all of the elements discussed specifically enter into the angle-of-attack calculation and the dynamic pressure calculation. This is discussed in Refs. 35, 36, and 37. A typical angle-of-attack distribution for a rotor at high forward speed is shown in Fig. 4-37. A typical local velocity (Mach number) distribution from which the dynamic pressure is calculated is shown in Fig. 4-38. These two elements, along with the airfoil characteristics, must be known to define the airloads on a rotor blade.

4-9.1.1.4 Airfoil Characteristics

In the earliest rotor performance analyses, designers assumed that an airfoil could be represented simply as a lift-curve slope and an angle of attack. This model neglected stall, but as long as the vehicle was installed-power limited (as early helicopters were) and operated at low flights speeds, little or none of the rotor disk encountered stall. A more sophisticated model avoided stall by not permitting the blade section lift coefficient to exceed some fixed value. (In these models, drag and section pitching moment customarily were ignored.)

In fixed-wing practice the primary importance of stall is the definite upper limit to lift. However, stall on a helicopter is accompanied by so many other effects of near-equal importance that the simple models previously mentioned are inadequate for prediction of rotor airloads and stresses. Therefore, two-dimensional aerodynamic data were introduced into the analyses. These data, based upon two-dimensional wind tunnel data, were used because the aspect ratio of a helicopter rotor blade normally is large enough that finite aspect ratio corrections can be neglected. Such effects are represented instead by an empirical "tip loss" and/or "root loss" in the lift (and sometimes pitching moment) and by the addition of a drag coefficient increment for surface roughness and similar effects.

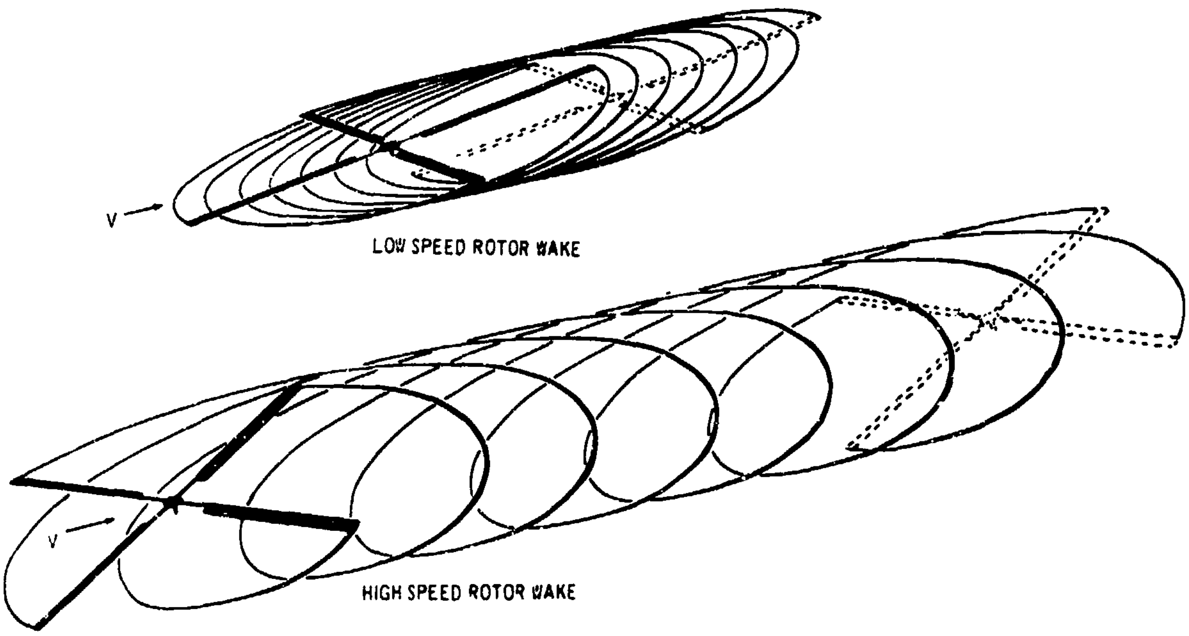


Fig. 4-35. Nondistorted Rotor Wake Geometry (Wake Skew Neglected)

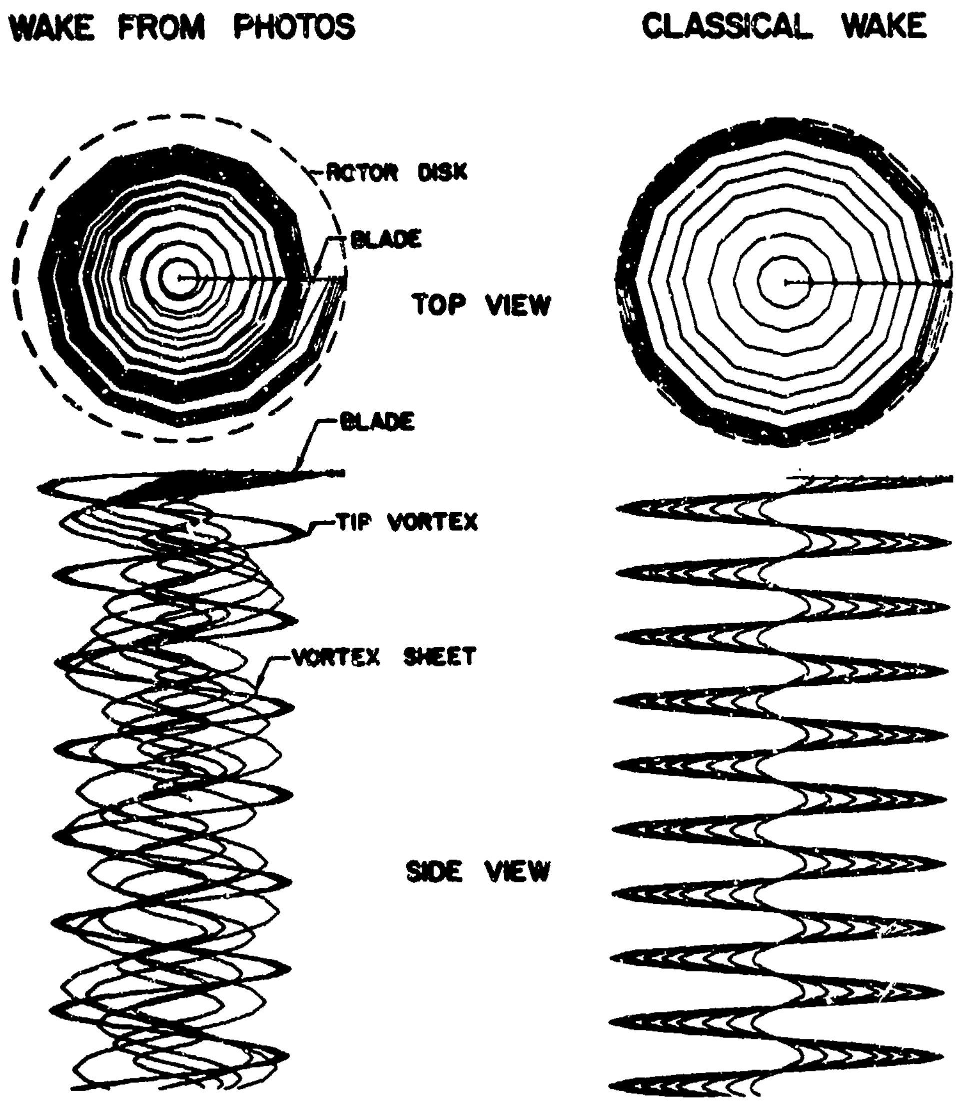


Fig. 4-36. Wake Trajectories for One Blade (Hovering Flight, OGE)

With the inclusion of two-dimensional aerodynamic data, it is instructive to observe the effects of lift, drag, and pitching moment on rotor loads, particularly in the vicinity of stall. Lift, as commonly defined, is perpendicular to the relative wind. Thus, the primary component of lift is in the flapwise direction, which is out of the plane of rotation. However, a small inplane component exists and may tend either to retard blade rotation (thus requiring added power) or to aid it (as in the case of autorotation). Its magnitude may be similar to that of the airfoil drag force. Conversely, the primary result of airfoil drag is excitation of blade chordwise bending, while a small component of drag is out of the plane of rotation.

It is obvious that both of these forces are capable of providing a moment about the blade feathering axis. With a blade deflected above or below the feathering axis, moments tending to increase or decrease blade pitch may be generated by those airloads that are eccentric to the feathering or control axis. When the elastic axis of the blade is ahead of or behind the feathering axis, the airloads also can cause moments that tend to increase or decrease pitch. These effects may tend partially to cancel each other; by appropriate blade design these moments may be arranged to be compensating. The blade aerodynamic pitching moment then becomes the primary excitation in blade torsion.

Because of variations in Mach number, Reynolds number, and angle of attack, each spanwise portion of the rotor blade stalls at a different azimuthal location. For some conditions, the lift stall is sharp and well defined; for others, it is simply a region where lift ceases increasing with angle of attack and remains constant. However, steady-state stall always is accompanied by a rapid increase in both drag and pitching moment components of aerodynamic excitation. Thus, if a segment of a blade penetrates the stall region momentarily as it proceeds around the azimuth, local impulsive loading will occur and may excite various vibratory modes. As Mach number increases, it becomes more important to include finite aspect ratio effects in the determination of applicable airfoil characteristics. This is particularly true of pitching moment coefficient C_M for its effect upon blade torsion.

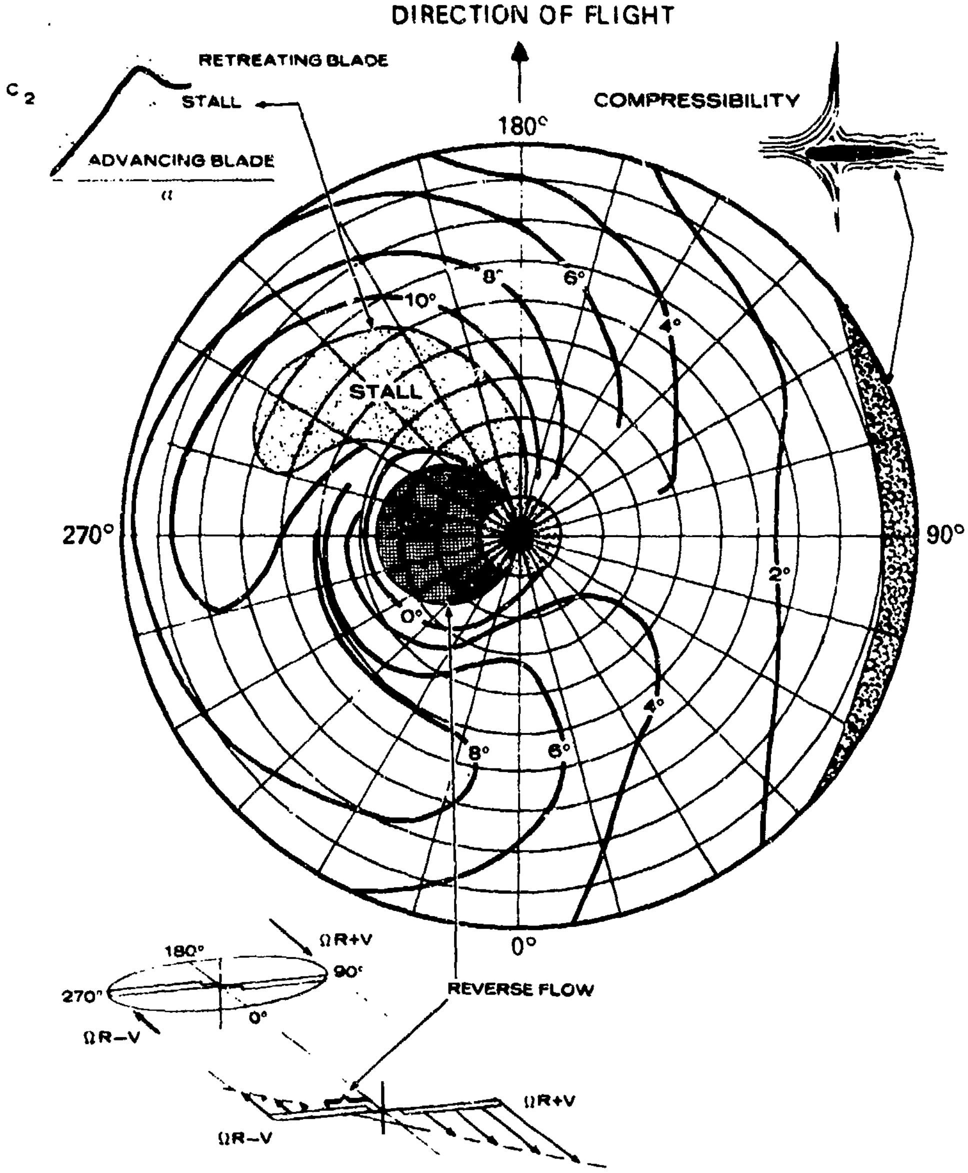


Fig. 4-37. Angle-of-attack Distribution at High Forward Speed

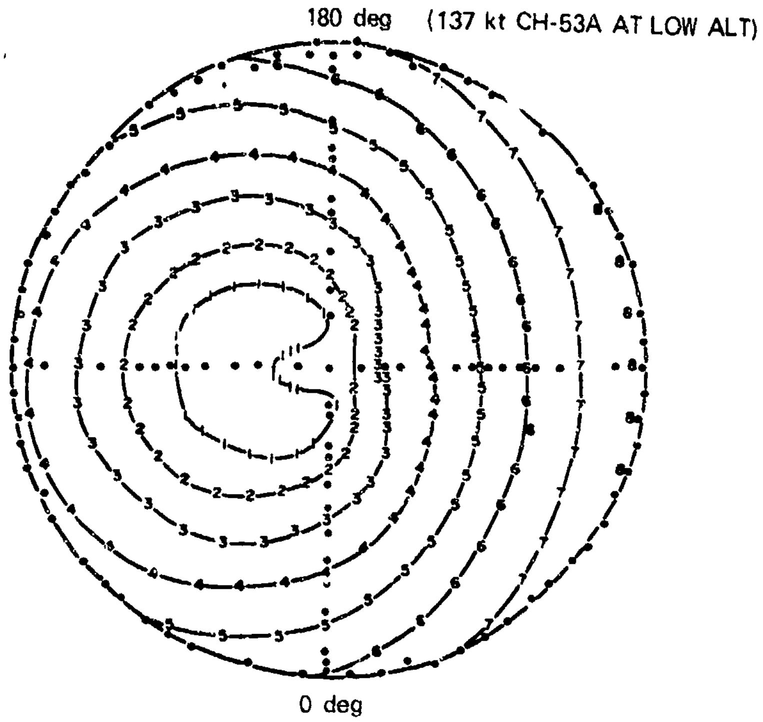


Fig. 4-38. Mach Number Distribution at High Forward Speed

Because stall characteristics vary widely with airfoil section, it might be expected that each analytical investigation would use only the airfoil data applicable to the particular design. This is not always true. Many helicopters have been built with only a few different airfoil sections, and data on these sections are available readily. Wind tunnel data on a more exotic section contemplated for a new design often are unavailable, or very expensive to obtain; but because the flow field itself is difficult to define, the data from a closely related "standard" section often can be used without significant loss of accuracy. An even more common practice during preliminary analysis is to neglect the effect upon section characteristics of production/operational departures of airfoil contour such as erosion strips, blunt trailing edges, and tabs, from the theoretical section contour.

In most of the more advanced computerized analyses, the lift, drag, and blade-pitching moment coefficients are obtained from a table look-up procedure based upon both angle of attack and Mach number. Such tables are generated either from wind tunnel data or from the collections of such data in the literature and the classic source for such data is Ref. 38. In either case, the table look-up procedure should handle adequately the nonlinearities encountered. While Mach number usually is defined better than angle of attack, the uncertainties in describing the flow field make data at the more extreme Mach numbers questionable because of wall effects, transonic flow, and other complications in gathering the initial airfoil data in the wind tunnel. Reynolds number effects often are lumped with Mach number because, during a constant chord airfoil test in a given atmosphere, Reynolds number and Mach number vary together.

Recent analytical improvements in the treatment of airfoil properties include attempts to include radial flow and unsteady aerodynamic effects. Radial flow effects are the helicopter analog of swept-wing theory, and arise from considering the skewed flow on the airfoil. Skewed flow implies that the airfoil section as seen by the airstream should be modified to account for the fact that the flow does not run perpendicular to the span, and that the total velocity and dynamic pressure also are modified. Further work is required in this area (see Chapter 3). Unsteady aerodynamic effects arise from the fact that the actual rotor blade angle of attack changes too rapidly to be represented properly by steady-state aerodynamics. Most existing unsteady data were obtained for simple harmonic motion of an airfoil (Refs. 39 and 40), while rotor blades execute multiharmonic motion. Thus, work on the application of unsteady aerodynamic data to helicopter rotors requires careful consideration of the proper means of making the data and the environment compatible.

4-9.1.1.5 Transient Loads

Two of the more important perturbations encountered by helicopter rotors are gust loads and flight maneuver loads. Both are amenable to simplified descriptions and are treated in detail only rarely. Here, both simplified and detailed models will be discussed for completeness.

The actions of a gust in producing a load transient on a rotor begin with the change in angle of attack that a shift in the flow field logically will cause. This changes the lift, drag, and moments appearing on the rotor blade. The blade reacts to these changes and in so doing feeds a force and moment change into the airframe. The airframe reacts to these forces and to any added forces the gust produces (such as a change in the lift of the body and wing). Thus, a gust produces a load transient that occurs as an aerodynamic impulse, followed by dynamic responses that tend to alleviate the initial shock.

The aerodynamic impulse is a function of the amplitude and wave form of the atmospheric disturbance, but is not easily predictable because a given angle of attack change $\Delta\alpha$ or ma_ not cause a corresponding linear change in C_L, C_D, and C_M. Stall effects and other nonlinearities complicate the picture.

In addition to this uncertainty in the effect of a given $\Delta\alpha$, it should be realized that the $\Delta\alpha$ is itself an unknown because a gust is a near-random atmospheric disturbance. To further complicate matters, the gust alleviation produced by aircraft dynamic responses is a complex phenomenon that varies with the vehicle under study. In the light of such complexities, simplifications commonly are adopted.

Discrete gust models are used for design purposes in examining the limit blade response or extreme load. This involves modeling only the limit gust of interest. Because of their direct effect upon blade angle of attack, vertical gusts are much more critical and are the only ones normally considered. Experience has shown that severe step functions do not occur in the atmosphere; that dynamic response alleviation often is quite significant; and that a better empirical approach can account for both effects (par. 4-4.3). Such an approach might use ramp and/or trigonometric functions. A typical example is a 50-fps sine-squared function with a ramp length of 90 ft (see Ref. 9).

The gust having been described, the means of applying it to the rotor system are considered. In the simplest rotor models the gust is converted into an instantaneous effective change in the inflow ratio $\Delta\lambda$, and this $\Delta\lambda$ is applied sumultaneously to the entire rotor disk. For a more sophisticated description, a gust penetration model is used. This method requires that the gust be applied to each blade element/time combination as a local change in the flow field rather than as a change in the average downwash. The effect of the gust is accounted for in the analysis as a change in angle of attack, which in turn can be accounted for in complex blade analyses such as are discussed in par. 4-9.1.2.

The load transient produced by a maneuver is in some ways analogous to the gust load transients previously discussed. Pilot action initiates an angle of attack change in the rotor system, producing forces and moments fed from the rotor to the aircraft, which then responds dynamically. However, the maneuver loads are a closed loop; the dynamic response of the aircraft causes further changes in the aerodynamic loading; and the process repeats. While the gust situation involves excitation primarily in one direction, with dynamic response alleviating the loading without truly changing the qualitative character, maneuver loads are multidirectional and the dynamic responses are of much more significance. This easily is seen in, for example, a roll. To obtain a large roll acceleration, the lift must be asymmetrical. At high roll rates, gyroscopic precession is significant. Similarly, a 30 deg/sec roll rate on a large helicopter (rotor diameter of about 70 ft) implies a vertical velocity component at the tip of about 20 fps, which increases the angle of attack on one side and decreases it on the other.

There are two common means of modeling maneuver loads. In the first case, only the primary effect of a maneuver—the increase in load factor on the aircraft and thus the lift required from the rotor disk—is considered. This also may include consideration of redistribution of airloads to reflect a changed hub moment.

However, this method is inadequate for prediction of the details of the blade loading and the resultant stresses. A more exact approach involves applying, at the rotor hub, the acceleration and rotation time histories actually felt there, along with the corresponding time history of control positions. This has the effect of fully describing the maneuver and has been found to provide insight into the source of the stress changes caused in maneuvering flight. This may be done in a closed loop—assuming that the forces and moments that the rotor produces are consistent with the accelerations of the hub by which one describes the maneuver —or it may be done open loop, ignoring the need for fuselage dynamics to obtain a consistent set of dynamic conditions. As long as the maneuver is well known (such as a banked turn, roll, and dive pullout) and the model has been shown to be consistent for earlier cases, open loop operation probably is adequate. Further descriptions of this type of analysis can be found in Ref. 41.

4-9.1.2 Analytic Approach to Rotor Load Prediction

Accurate analytic prediction of the loads generated by a rotor requires a sophisticated mathematical description of the dynamic characteristics of the rotor system, together with accurate definition of aerodynamic forcing functions. The dynamic complexity of rotors and the aerodynamic environment in which they operate makes this no simple task. By way of example, a system comprising only the rotor blades and the rotor hub will be discussed. This neglects airframe/rotor system dynamic coupling.

The primary objective is the prediction of the responses of the rotor blades to the aerodynamic forces. The responses having been established, the rotor loads can be calculated. To do this, the differential equations of motion of the dynamic system are set up first.

A common method of representing the blade dynamics is to assume that the blade displacements q_B can be represented by the sum of a series of modal coordinates (Fig. 4-39). These modal coordinates, by separation of variables, can be represented by the product of the mode shape variables $\phi_i(x)$ (functions of blade radius only), and by a function that defines the magnitude of the particular modal participation ξ_i (a function of time only). The modal functions generally are referred to as normal or principal modes and have the important mathematical property of being orthogonal; i.e., motion in each normal mode can exist independently. Thus, the number of normal modes chosen is equivalent to the number of degrees of freedom describing the motion of each blade. This method can be applied to represent the flapwise, chordwise, and torsional blade displacements. Clearly, the greater the number of modes chosen to define the deflections, the more accurate the representation. The number used is dependent upon the order of the highest forcing frequency. In practice, a total of approximately 10-15 modes generally is sufficient to define the motions of each blade.

The mode shapes and corresponding frequencies can be obtained by any standard method of obtaining the free vibration characteristics of a continuous system, e.g., the method of Myklestad (Refs. 41 and 42). This modal representation is applicable to blades with any general form of root boundary conditions, from fully articulated through rigid.

The fact that the normal modes are orthogonal does not preclude the inclusion of dynamic coupling terms in the description of the rotor system. This is achieved by simply including the coupling effects as applied forces. Based upon the desired degree of accuracy, the physical description of the blades may include the effects of any or all of the following: blade twist, chordwise mass unbalance, counterweights, noncoincidence of elastic axis and quarter chord, and radially varying blade properties.

To obtain the dynamic system differential equations of motion, the absolute translational and rotational coordinates of motion of any blade element are obtained. This is achieved conveniently through a series of coordinate transformations, which allow definition of the blade velocity vector that is used to specify the blade kinetic energy. Potential and dissipation energies resulting, for example, from blade root springs and dampers are obtained readily through use of the relative coordinates of motion. The analytic forms of these energy functions having been obtained, the method of Lagrange may be used to establish the differential equations. Lagrange's method states that

$$\frac{d(\partial T/\partial \dot{q}_s)}{dt} - \partial T/\partial q_s + \partial V/\partial q_s + \partial D/\partial q_s = Q_s \quad (4\text{-}28)$$

where

T = kinetic energy, ft-lb
V = potential energy, ft-lb
D = dissipation energy, ft-lb
q_s = coordinates, or degrees of freedom, defining system motion, ft

The Q_s are applied generalized forces that will be discussed in a subsequent paragraph. For accurate representation, significant nonlinear terms generally are retained in the equations. In addition to these nonlinearities, the equations contain time-dependent coefficient terms—the dependency arising from the rotational nature of the system.

The aerodynamic characteristics of a given airfoil cannot be calculated accurately. Linear analytic expressions defining the airfoil characteristics can be used in certain stability studies, but such a practice generally is not acceptable for accurate prediction of rotor loads. There are many reasons for this, not the least being the very complex nature of the airflow around the airfoil, particularly in regions of stall or high Mach number operation. For this reason aerodynamic inputs rely upon the results of large numbers of wind tunnel tests (see par. 4-9.1.1.4).

The aerodynamic forces experienced by the rotor system, in addition to being functions of the aerodynamic characteristics of the blades, are functions of the rotor system displacements and velocities. For this reason there is a high degree of coupling between dynamic system motions and the aerodynamic environment.

In producing thrust, a rotor generates induced velocities across the rotor disk (Ref. 31). These velocities are important as they affect the blade angle-of-attack distributions and, hence, blade load characteristics. They are the result of the shed and trailing wakes generated by the rotor blades, and various degrees of sophistication may be employed in predicting their effects. The simplest method assumes a uniform distribution of induced velocities across the disk, whereas the most rigorous methods calculate the free wake geometry and the resultant induced velocity distribution across the disk. The degree of accuracy required in the rotor loads prediction determines which method is employed.

The generalized forces Q_s, which as stated earlier may be the result of dynamic coupling or aerodynamic effects, may be obtained by employing the principle of virtual work. This in its simplest form states that

$$Q_s = \sum_i P_i \frac{\partial \{\mathbf{X}_i\}}{\partial q_s}\ , \text{lb} \tag{4-29}$$

where

P_i = forces applied at any station i, lb

$\mathbf{X}_i$ = absolute position vectors of the points i, ft

Combining the dynamic system equations and the generalized forcing functions leads to a complex, nonlinear set of second order differential equations with time-dependent coefficients. These can be solved only by numerical techniques and high-speed computers. Briefly, the method employed is to compute the system

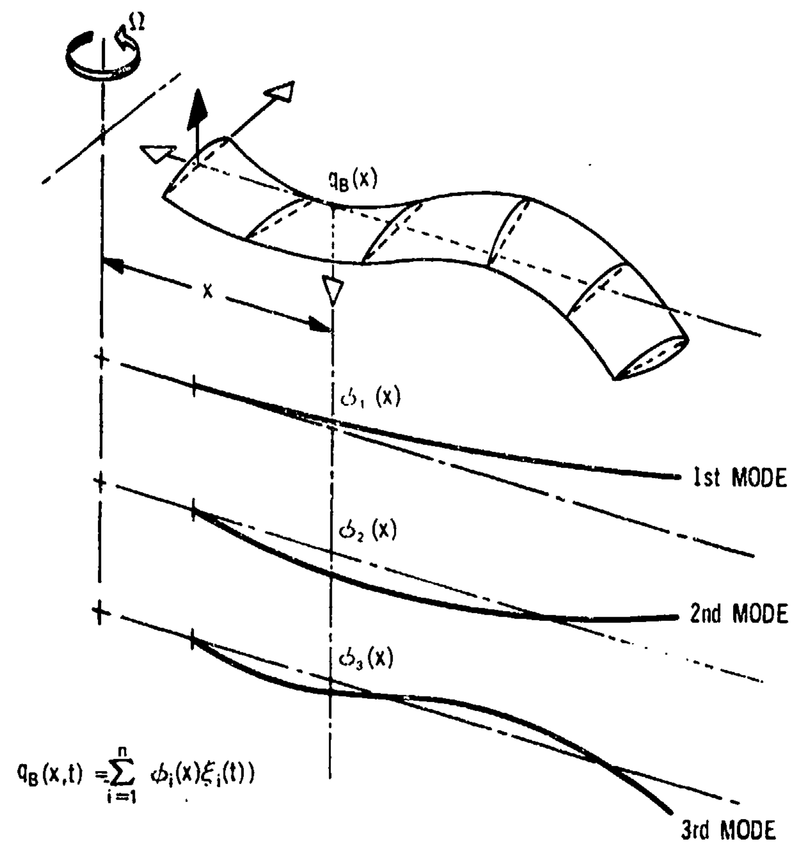


Fig. 4-39. Modal Representation of Blade Displacements

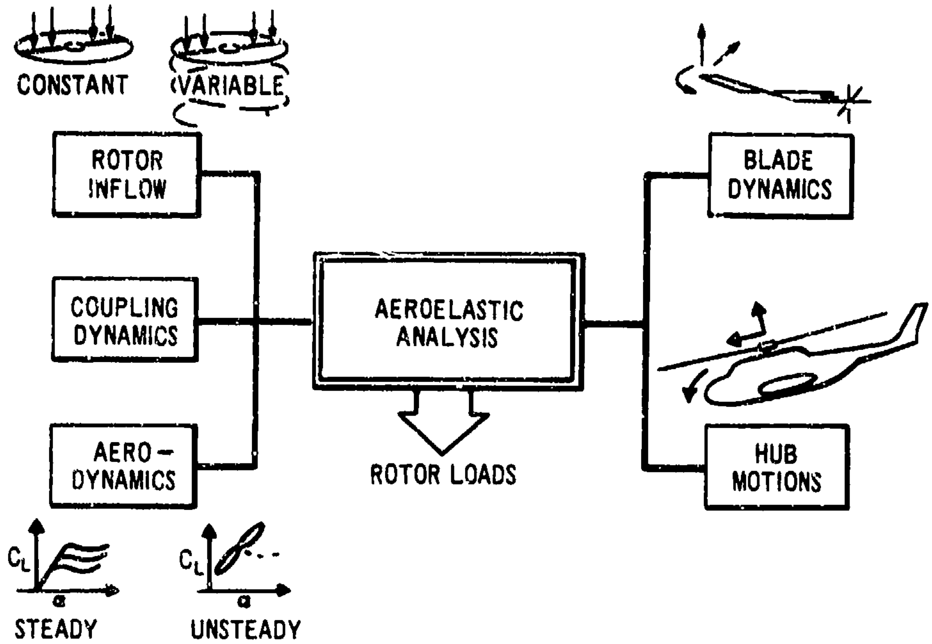


Fig. 4-40. Elements in Analysis

response characteristics at each instant of time as the blades travel azimuthally. In this way the transient blade deflections and the moments and stresses consistent with these deflections are obtained. At the same time, the shears and moments at the blade root are calculated.

For steady flight all blades track in the same manner; therefore, hub forces and moments may be obtained from a summation over all blades of the root shears and moments derived from a single blade analysis. However, under certain circumstances, e.g., certain conditions of turbulence, a multibladed analysis is necessary.

Fig. 4-40 shows in block form how the elements discussed may be structured to form the basic analysis.

It has not been possible here to discuss all of the important aspects of rotor load predictions. For example, the effects of airframe dynamics or airframe/rotor interference have not been included; but it has been shown that the problem is complex and requires sophisticated treatments to obtain accurate forecasts of the rotor response and load characteristics. More detailed information is available in Refs. 35, 36, 37, and 43.

Unfortunately, few simplified procedures give meaningful results. There is, however, a simplified method for estimating flapwise bending moment characteristics of rotor blades (Ref. 44) that merits mention. This report shows that the flapwise bending moments are basically linear functions of several independent rotor parameters and can be computed by transfer function, superposition techniques. Transfer coefficients that relate rotor parameters to harmonics of moment are presented in the form of design charts for a wide range of parameters and operating conditions. Detailed procedures for using these design charts in conjunction with performance charts (Ref. 45) are described and illustrated with sample calculations. The method is shown to provide quantitatively accurate results at advance ratios from 0.25 to 0.4.

4-9.1.3 Preliminary Design Considerations

The basic requirements for rotor blade analysis are contained in MIL-S-8698. In addition, checks of centrifugal stress, static bending stress and deflection, and resonant frequencies are used in the early stages to size the blade structure.

Before the blade can be analyzed structurally, the various axes of the blade must be located. Because symmetrical airfoils usually are used, the blade chord is taken as the centerline of symmetry of all sections of the blade structure. This leaves the axes perpendicular to the mean chord—the flexural (or neutral) axis for chordwise bending, the shear center, the feathering axis—to be located, and the center of tensile restraint and the mass centroid (or CG) of the entire blade.

Symmetrical airfoils with practicable construction, operating at low Mach numbers and below stall, have their aerodynamic centers at the quarter-chord point (Fig. 4-41). This usually is used as the location of the feathering axis and the mass centroid to avoid stall and torsional divergence. The shear center also is located at the quarter-chord because the summation of all torsional shears should be zero; any other location would produce a torsional shear moment. To keep the blade straight in flight, the intersection of the flapping and lead-lag hinges also is placed at the quarter chord; consequently, the flexural axis should be at the same location. The blade structure must be proportioned so as to place these axes in approximately the correct location. If they are not, an iterative step is needed at this stage, altering the distributions of mass and/or stiffness to correct the deficiency.

Although symmetrical airfoils normally are used, it may with some blade sections be desirable to locate the mass centroid and the shear center forward of the aerodynamic center for greater blade stability.

Because the blade is subjected to various combinations of flapwise and chordwise bending, centrifugal tension, flapwise and chordwise shear, and torsional moments, it is necessary to determine the spar cross-sectional area, flapwise and chordwise moments of inertia, mass distribution, mean area enclosed by the spar outer and inner boundaries, spar mean line perimeter, and wall thickness at several locations (Fig. 4-42). These properties, in addition to the inherent properties of the spar material—such as its tensile and shear moduli of elasticity—are sufficient to determine the flapwise and chordwise bending stresses, shear stresses, torsional rigidity, torsional stresses, spar weight, and natural frequencies.

Approximately six stations along the blade [illegible] lected, and the spar section is laid out at each [illegible]. The properties of each section are calc[illegible] [illegible]dividual curves of cross-sectional area and fl[illegible]wise and chordwise moments of inertia are plotted against blade radius. Because a plot of blade mass distribution also is required for use in the calculation of blade weight, natural frequencies, and stresses, an estimate of the weights of the nonstructural blade components at each station is included, based upon experience and trend studies.

Centrifugal stresses created in the spar by blade rotation then are checked at several stations along the blade spar. Slight modifications are made, as required, to the spar wall thickness to adjust for overstressed or understressed conditions. The centrifugal stress should be selected on the basis of repeated application, (start-stop cycle) and with consideration for the superposition of vibratory stresses in flight. A maximum value of 20% of allowable tensile stress can be used.

With the blade approximately sized in this manner, the blade is considered to be cantilevered horizontally at the root, and static bending stresses and deflections are computed. Modifications may be made to the spanwise taper to adjust the tip deflection to a value suitable for adequate operating clearance of other airframe components (par. 13-1) and to provide a positive mar-

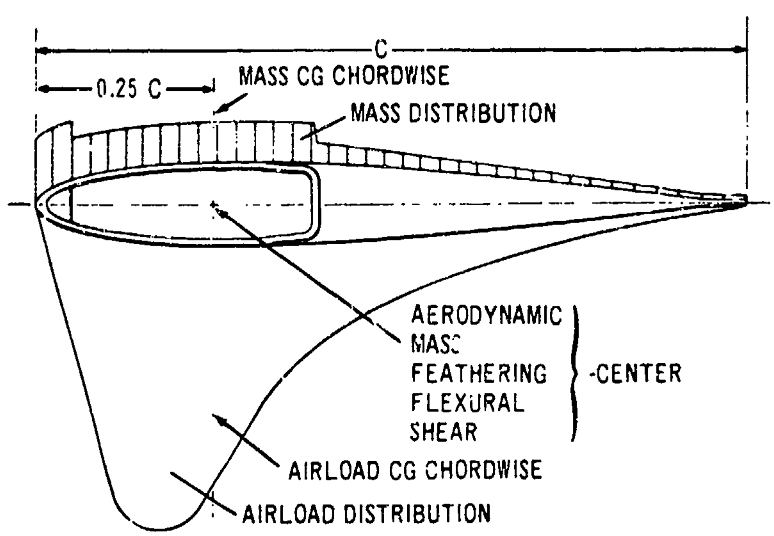


Fig. 4-41. Typical Blade Cross-sectional Loading

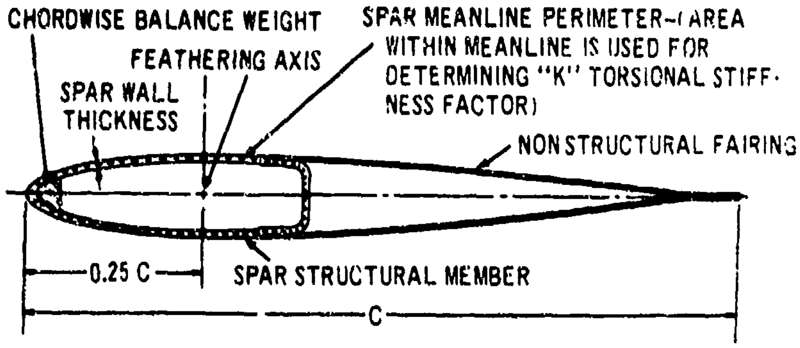


Fig. 4-42. Representative Blade Structure

gin of safety with a limit load factor of 4.67 for ground handling.

Buckling of the spar may occur at a stress below the allowable static stress if the compressive stress due to bending exceeds a critical value. The subsequent analysis treats the spar wall as a curved plate and defines a critical compressive stress as a function of the dimensions and modulus of elasticity of the plate.

The dimensions used in the buckling analysis are taken from scale drawings of the blade spar sections. By ignoring sharply curved front and rear extremities of the spar, a circular arc is fitted to the upper or lower spar wall shape, using a three-point fit. The radius of curvature r_c and wall thickness t, both in in., are used in selecting the proper curve for determination of the dimensionless constant K_c (Fig. 4-43). The chordal width W of the spar, also in in., is used to determine the abscissa $W^2/(r_c t)$ on the graph. The selection of the appropriate value for W is very important inasmuch as its contribution to the critical stress is of the second order (W^2). Appropriate values of t and W, combined with modulus of elasticity E, in psi, and K_c are used in the calculation of critical buckling stress $F_{c_{cr}}$ in the equation:

$$F_{c_{cr}} = K_c \frac{Et^2}{W^2} \text{ , psi} \tag{4-30}$$

$F_{c_{cr}}$ may then be plotted against rotor radius. It is a limiting value of axial compressive stress for the blade spar. Bending stress may exceed this value by the amount of the tensile stress superimposed by centrifugal force. Thus, the conditions for which buckling becomes a critical design consideration are most likely to be those wherein the rotor is stopped, or nearly stopped.

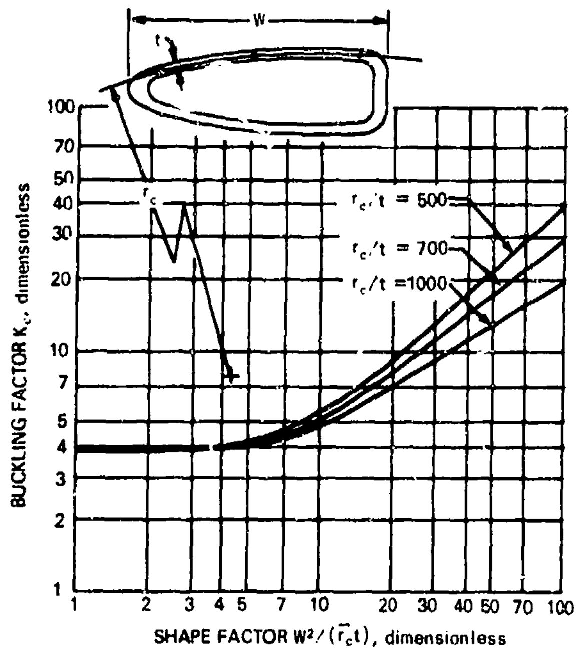


Fig. 4-43. Determination of Constant K_c to Calculate Critical Buckling Stress (Eq. 4-39)

The blade now is sized to approximately the proportions required by dynamic loading considerations. In subsequent analytical steps, severe dimensional changes involving complete iteration of previous work are unlikely. The calculation of resonant frequencies for flapwise and chordwise bending are discussed in par. 5-3.

Small adjustments now are made in the section properties to correct for dimensional changes that have been made during these preliminary analyses. Torsional stiffness is calculated for each section by standard methods, e.g., Ref. 46. The properties now are ready for more extensive stress calculation, and can be plotted against rotor radius for convenience.

Rotor blade stress and fatigue analysis are discussed in detail in par. 4-11.

4-9.2 HUB LOADS

The hub is the central structure of both main and tail rotors. On any kind of helicopter the hub transmits flight loads to the airframe, and on shaft-driven helicopters it is the structure through which the drive torque is distributed to the blades.

4-9.2.1 Characteristics of Rotor Hubs

The blades of all rotors are attached to a mechanism that permits their pitch or incidence to be varied. This mechanism is usually the most outboard portion of the hub. (In some rotors this function is accomplished by twisting the blade itself.) Inboard, the nonpitching part of the hub may have hinges that permit the blades to move in the flapping and lead-lag planes.

Rotor blades will seek an equilibrium position in the vertical plane by moving around the flapping hinge, real or simulated. The main loads determining the equilibrium position are the thrust, or vertical airload, and the centrifugal force. Because the centrifugal force is 10-20 times as large as the vertical airload, the blade axis will have a slight angle with respect to the plane of rotation. This angle is known as coning angle.

If a load that varies once per revolution is applied to the blade, the latter will respond with a once-per-revolution motion about its flapping hinge. The result is a tilt of the plane in which the blade is moving, with the angle of tilt equal to the flapping angle of the individual blades. Most helicopters are controlled by tilting the plane of rotation of all blades.

The thrust vector will tilt with the plane of rotation and will have a component in the direction of tilt. It is this component that makes the helicopter move in the direction of rotor tilt. In level flight the vertical component of the thrust vector is equal to the weight of the helicopter.

A number of different types of rotor hubs have been developed and are in use. The characteristics of the fundamental concepts employed for rotor control are discussed in par. 3-3.3. The advantages and disadvantages of the various types are reviewed in that discussion.

4-9.2.2 Analysis of Rotor Loads

The determination of loads on an individual rotor blade is discussed in par. 4-9.1. The combination at the hub of the periodically varying loads on the rotor blades and the resultant loads that are transmitted through the hub to the airframe are discussed in par. 5-2.

The physical characteristics of helicopter rotors, as described by the distribution of mass and elastic properties and the aerodynamic environment in which they operate, are far from linear with respect to either radial location or azimuthal orientation. The typical digital computer analysis, therefore, treats the rotor by considering various aerodynamic and dynamic effects at discrete points as a function of radial location and azimuth position. By increasing both the number of radial locations and the number of azimuth positions, the solution for the nonlinear physical system may be obtained to within an acceptable degree of accuracy.

For preliminary design, the principal uses of computer analyses are the screening of proposed structural configurations and the initial dynamic tuning to avoid resonance with known excitation frequencies. The current techniques include the computation of aerodynamic forces along the blades based upon steady-state aerodynamic coefficients. These forces are applied to rotating beams that are restrained to the mast by appropriate hub kinematics so as to represent the boundary conditions for rigid, semirigid, gimbaled, or articulated rotors, as appropriate. The resulting steady and vibratory blade moments then are calculated for use in the stress calculations. The resultant shear loads at the hub are combined to form the forces that will cause the fuselage to vibrate. Much design effort is spent to reduce the oscillatory loads in the rotor and to minimize the vibration level at high forward velocities. Rotor hubs generally are fatigue-critical components, and their substantiation must include fatigue life determination (par. 4-11).

Nondimensional coefficients that are convenient for expressing thrust capability are the blade-loading coefficient t_c

$$t_c = \frac{2T}{bcR\rho(\Omega R)^2} \text{, dimensionless} \qquad (4\text{-}31)$$

and the mean blade lift coefficient $\bar{C}_L$

$$\bar{C}_L = \frac{7T}{bcR\rho(\Omega R)^2} \text{, dimensionless} \qquad (4\text{-}32)$$

where

T = rotor thrust, lb
b = number of blades
c = blade chord, ft
R = rotor radius, ft
Ω = rotor speed, rad/sec
ρ = air density, slug/ft^3

These coefficients are related

$$\bar{C}_L = 3.5 t_c \tag{4-33}$$

The rotor-thrust coefficient C_T

$$C_T = \frac{T}{\rho \pi R^2 (\Omega R)^2} \tag{4-34}$$

is not as convenient an expression of thrust capability because it includes the rotor solidity σ (the ratio of blade area to disk area)

$$7C_T = \sigma \bar{C}_L \text{ , or}$$
$$2C_T = \sigma t_c \tag{4-35}$$

Until recently, most rotor blades used symmetrical sections, such as NACA 0012. With these blades, the loads in maneuvers consistently were less than t_c = 0.35, or $\bar{C}_L$ = 1.23. Newer blade designs, using camber in the forward part of the section, have greater lift capacity; blade-loading coefficients of 0.38 can be attained. Airfoil sections now being developed should make it possible to reach t_c = 0.40.

As the forward speed of the helicopter increases, rotor thrust capability is reduced. This is because the blade is subjected to conditions of airspeed and inflow angle that are different for each azimuth position. The blade reaches conditions of limit lift capability, or stall, in some azimuth positions earlier than in others. Vibration and control problems associated with this occurrence will prevent the rotor from reaching its maximum thrust capabilities under these conditions.

4-9.2.2.1 Flapwise Loads

Current structural design requirements dictate that a rotor be substantiated for a static flapwise load envelope as shown in Fig. 4-44. The conditions represented by the envelope can be substantiated by demonstrating that the rotor in question satisfies the load conditions at the corners of the envelope. Such an envelope, however, is unrealistic because thrust capability is not constant over the rotor-speed range. If the load factor is correct for the normal operating speed, it is too conservative at lower speeds.

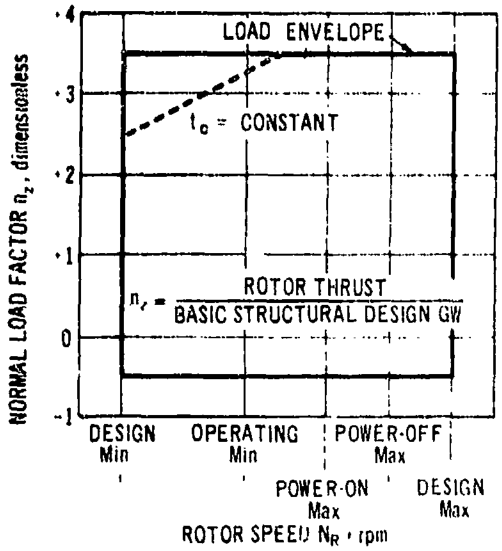


Fig. 4-44. Typical Flapwise-load Envelope

4-9.2.2.2 Inplane Loads

In powered flight, drive torque is transmitted from the drive shaft or mast through the hub to the blades. Along each blade, this torque is balanced by the inplane components of the airload. In forward flight and maneuvers, the position of this basic equilibrium point oscillates as the airload varies; and the oscillation may be reinforced if its frequency is near the natural frequency of the lead-lag motion.

In articulated rotors, the loads cause oscillatory lead-lag motions that are damped by a mechanism in the hub. In rotors that are stiff inplane, the inplane loads are only lightly damped. The loads transmitted by the blades to the hub may be balanced by opposing loads on the other blades. Because it is difficult to calculate limit loads in lightly damped structures that are excited at frequencies near resonance, the design limit loads for inplane-stiff rotors usually are derived from flight-test data on similar configurations. The limit root inplane moment $M_{c\,lim}$ can be expressed conveniently in terms of drive torque M_T and the number of blades b. For powered flight

$$M_{c\,lim} = \frac{M_T}{b}(1 \pm K) \text{ , lb-ft} \tag{4-36}$$

and for power-off conditions

$$M_{c\,lim} = \frac{M_T}{b}(0 \pm K) \quad \text{, lb-ft} \qquad (4\text{-}37)$$

In these equations K is an empirically determined factor.

Flight loads measured on various chordwise-stiff two-bladed and three-bladed rotors during maneuvers have shown peak moments corresponding to $K = 3.0$ for small helicopters and to $K = 4.5$ for medium-size helicopters. Because a large number of parameters influence the magnitude of the peak inplane bending moments reached, K factors cannot be specified for classes of helicopters. The choice of the proper factor for the limit chordwise moment is left to the discretion of the designer. Moments for new designs can be derived by using K factors determined from flight load data for similar helicopters.

4-9.2.3 Design Loads for Fatigue

It is important that allowable levels for oscillatory or fatigue loads be established during preliminary design. The frequencies of these loads range from multiples of the rotor speed to once or less per flight. Because the number of cycles accumulates rapidly at the higher frequencies, fatigue damage cannot be permitted for the load levels that occur at the frequencies; and, conversely, high loads should not occur at frequencies high enough to accumulate a critical amount of fatigue damage. Therefore, the loading spectrum of the helicopter (par. 4-4.1) is an important criterion in the design of its rotor hub. Fatigue analysis is discussed in detail in par. 4-11.

4-9.2.3.1 Endurance Load Level

Because it is impracticable during preliminary design to analyze each component for a complete loading spectrum, a common practice is to use the oscillatory loads that occur in high-speed level flight as a yardstick for fatigue analysis and to apply factors to these loads to account for the material, the load spectrum, and the desired life of the components (Ref. 47). The level flight loadings can be calculated if the necessary parameters (weight, power, and drag) and the rotor dimensions and speed are known. Whenever feasible, the endurance limit for the hub components is established at or above these load levels so that no fatigue damage accumulates in high-speed level flight. Oscillatory loads resulting from maneuvers and other conditions in the loading spectrum for which fatigue damage can be permitted to accumulate can be related to the high-speed level flight loads by appropriate factors. Suitable values for the appropriate factors generally can be found by statistical analysis of flight load data for previous designs.

4-9.2.3.2 Maneuver Loads

Although the mean loads and the load amplitudes under the transient conditions of a given maneuver can be calculated, the results are far less accurate than are those for the level flight conditions. One reason for this is that the load level reached in actual maneuvers is dependent strongly upon pilot handling and transient response. Also, many factors, both known and unknown, that cannot be considered in the analysis have an influence on the result. Therefore, the general approach of analyzing for fatigue based upon steady flight conditions and applying appropriate factors to cover maneuvers has proved to be more practicable during preliminary design. This is in addition to a static analysis that uses a limit-load envelope that includes any peak loads that would be experienced in maneuvers.

4-9.2.3.3 Ground-air Load Cycle

A type of repetitive loading that usually is covered by static analysis, but that may require separate treatment because of the possibility of fatigue damage, is the ground-to-air load cycle. For certain parts of the rotor, especially the blade-retention system and the flapwise flexures, the stress cycle from standstill through runup to flight is considerable. If the helicopter is used for many flights of short duration, fatigue damage may accumulate in a relatively moderate number of load cycles, e.g., less than 10^4, of high stress level. The material S-N curves, plots of failure load versus number of cycles to failure, are not well defined at numbers of cycles less than 10^4 because fatigue tests are not run regularly at such high stress levels. However, if the designer is aware that a considerable number of high stress cycles may occur, he has the option to safeguard the structure by the introduction of an additional factor in the preliminary static analysis.

4-9.2.4 Miscellaneous Loading Conditions

A number of conditions may produce loads on the hub that need to be investigated separately. These conditions are discussed briefly in the paragraphs that follow.

4-9.2.4.1 Starting and Shutdown Loads

Upon starting the rotor, the drive torque is balanced only by the inertial loads along the blades. The blades of articulated rotors lag and hit the lag stops that are necessary to limit this motion. Drive torque distributed over the blades, therefore, provides a loading condition for the stop and the adjacent structure. In the same way, shutdown loads arise from the application of a rotor brake. The rated torque of the brake, distributed over the blades, is the loading condition for the lead stop and its adjacent structure. For a rotor that is stiff inplane, the starting or shutdown torque is usually minor in comparison to the rated torque at the rotor operating speed.

The criteria upon which these loads are based are given in pars. 4-7.1 and 4-7.2.

4-9.2.4.2 Wind Loads

A rotor can be vulnerable to damage from wind loads when the helicopter is parked, or when the rotor is turning at less than its operating speed. These loads upon stationary blades, tethered or free, can be calculated readily. During runup and shutdown, however, the rotor may pass through rotor speeds where flapping due to the wind load becomes divergent, resulting in very high loads in the hub. Establishing applicable loads in this condition is left to the designer, using a rational basis appropriate for the rotor system under consideration.

4-9.2.4.3 Stop-banging Loads

Freely flapping rotor blades are supported at rest by a lower flapping stop, or "droop stop", which may be either a rigid part of the hub or a mechanism that moves out of the way to provide flapping freedom in flight. As a rule, gimbal-mounted and seesaw rotors do not have droop stops; instead, flapping freedom of the entire hub is limited by a stop. In either case, the flapping stops experience loads whenever the motions of blades or the rotor exceed the available travel, and no load requirement exists for this condition. However, the flapping stop and rotor structure of articulated rotors *shall* be substantiated for the 1.0-g static weight moment, multiplied by a limit load factor of 4.67 (par. 4-9.1.3). In the case of seesaw and gimbal-mounted rotors, a root moment should be added to account for the acceleration loads of the other blade or blades.

Although arbitrary, the limit load factor of 4.67 will cover many cases on the ground where the blades strike the flapping stops. These cases may include ground handling (transport, taxiing, hoisting), turning the rotor at low speed in a strong wind, and also the case where the helicopter is parked with the rotor untethered while another helicopter hovers in close proximity or overhead.

It should be noted that the loads incurred when the flapping stops are hit in flight are of an altogether different order of magnitude. This condition must be avoided, and the design *shall* be such that, should it happen inadvertently, no components of primary importance to flight safety are damaged.

4-9.2.4.4 Folding Loads

If the hub has provisions for folding the blades, the structure should be substantiated for the loads experienced throughout the range of motion from folded to unfolded. There is no specification criterion applicable to this kind of operation. The gravity loads should be multiplied by a moderate factor (2.0) to cover ground handling, and a minimum wind velocity of 45 kt should be included to cover the likelihood that the operation may be performed in wind, and to be consistent with par. 4-9.2.4.2.

4-9.2.4.5 Mooring and Tiedown Loads

To protect rotor blades from damage when the helicopter is parked, they should be moored or tethered. The mooring or tiedown loads *shall* be determined in a rational manner, based upon the criteria given in par. 4-6.2.2.

4-9.2.4.6 Hoisting Loads

If the rotor hub has provisions for hoisting the helicopter (either special purpose hoisting lugs or otherwise) the loads *shall* be based upon the basic structural design gross weight minus crew and payload and *shall* include a load factor of 2.0 (par. 4-6.3). The helicopter may be hoisted either by appropriate ground equipment or by another helicopter. In the latter case the applicable load should include a component of drag upon the helicopter during transport.

4-9.3 MECHANICAL DRIVE SYSTEM LOADS

Drive system loads occur primarily as a result of the transmission of engine power to the rotors. The initial prime mover load, starting at the engine as torque, produces various dynamic and static loads as the input horsepower is transmitted through the drive train to the rotor system(s). The following loads generally are present on helicopter drive systems and must be considered during the predesign phase of the aircraft:

1. Steady torques
2. Oscillating torques
3. Axial or thrust loads
4. Radial loads
5. Tangential loads
6. Bending loads
7. Rotating beam loading
8. Structural deflection loads

9. Misalignment loads
10. Torque reaction loads
11. Mast bending and shear loads
12. Aircraft load factor
13. Gust loads
14. Crash loads
15. Dynamic loads from resonances
16. Dynamic loads from geometric effects
17. Gear tooth contact loads
18. Frictional loads
19. Loads that produce brinnelling
20. Bearing loads, bearing cage loads, and bearing preloading.

These loads combine to develop stresses, both steady and oscillatory, that must be accounted for adequately through the proper selection of materials and allowable stresses. Values of allowable stress specified in MIL-HDBK-5 may be used for sizing helicopter drive system components. When data for proposed materials are not provided by MIL-HDBK-5, the values used must be validated on a rational basis.

The magnitudes of the applicable drive system loads are dependent upon the mission spectra projected for the life of the vehicle. A growth factor should be considered in the preliminary design. Designing the drive system initially for moderate design stresses assures minimum development risk while also providing for subsequent growth. The difference between the initial design stress and the maximum allowable operating stress represents the initial growth potential. This growth potential should provide for an increase in engine power capability that usually will be specified in the RFP.

4-9.3.1 Load Spectra

The drive system designer must know the power required for the typical design mission spectrum and aircraft gross weights. Rotational speeds must be specified for the various power requirements because design stresses are based upon torque loadings, and fatigue considerations are related to numbers of cycles when loads are above the material endurance limits. Also, drive system rotational speeds must be defined, particularly for long shaft sections. The section properties of the shafting must be selected so that no critical speed resonances occur within $\pm 10\%$ of any operating speed of any section of the shafting (see MIL-T-5955 and par. 5-5).

Horsepower is related to torque Q in lb-in. and rotational speed N in rpm by the formula

$$hp = \frac{QN}{63,000} \tag{4-38}$$

A fundamental principle is that torque Q is proportional inversely to rotational speed. The influence of the torque value varies in magnitude among the different types of components. For example, transmission gear stage weight varies approximately as the 0.7 power of torque, while shafting weight varies approximately as the 0.38 power of torque. This explains the potential of achieving lighter weight drive systems with higher speed components. Drive system state-of-the-art is improving continuously to allow higher rotational speeds through the use of improved materials, higher component accuracy, and better understanding of dynamic loadings of bearings, gears, and other critical components.

The load spectrum must include all possible transient and overload conditions so that design limit loads may be established. Duration of transient loads also is important for determining fatigue damage cycles and possible life limits to fatigue-loaded parts.

The load and rpm spectra for a typical small helicopter may appear like those shown in Tables 4-4, 4-5, and 4-6.

The maximum continuous (normal power) engine rating must be matched to the airframe power requirements for the mission spectrum. The input power must be distributed properly in a manner such as is shown in Table 4-6. It should be noted that in one flight condition—autorotation—the rotor is driving, resulting in side gear tooth loading and possibly in reversed thrust loads on some bearings. Although these loads are not high, this condition still must be provided for in the gear tooth development and in the bearing configuration selection.

4-9.3.2 Cubic Mean Load

The sizing of bearings and the computation of bearing life, as well as a preliminary determination of gear tooth compressive fatigue (Hertz) stress, are dependent upon the cubic mean load. This is because the compressive fatigue loading is related to both bearing life and gear tooth surface durability approximately inversely as the cube of the load.

TABLE 4-4
RPM RANGE—TYPICAL SMALL HELICOPTER

POWER-ON	ENGINE	TAIL DRIVE SHAFT	TAIL ROTOR	MAIN ROTOR
MINIMUM rpm	6,000 (100% N_p)	2,170	2,974	495
MAXIMUM rpm	6,240 (104% N_p)	2,257	3,093	515
POWER-OFF				
DESIGN MINIMUM rpm	...	666	2,283	380
DESIGN MAXIMUM rpm	..	2,486	3,407	567

TABLE 4-5
ENGINE HORSEPOWER—TYPICAL SMALL HELICOPTER

TAKEOFF POWER (5-min RATING)	300 hp AT 6,000 rpm OR 312 hp AT 6,240 rpm
MAXIMUM CONTINUOUS POWER	270 hp AT 6,000 rpm OR 281 hp AT 6,240 rpm
10-sec TRANSIENT (MGT* LIMITS PER ENGINE SPECS)	340 hp AT 6 000 rpm

*MGT - MEASURED GAS TEMPERATURE

The cubic mean load $\overline{P}$ may be defined by the following equation:

$$\overline{P} = P_{max} \sqrt[3]{\sum^{n} t_i \left(\frac{P_i}{P_{max}} \right)^3} \text{, hp} \qquad (4\text{-}39)$$

where

$$\sum^{n} t_i = 1$$

P_i = power required at load spectrum condition i, hp
P_{max} = max rated power, hp
t_i = ratio of time at load spectrum condition i to total load spectrum time, dimensionless

By use of the small helicopter example of Table 4-6, the input cubic mean power would be approximately 270

TABLE 4-6
LOAD SPECTRUM—TYPICAL SMALL HELICOPTER

CONDITION	MAIN TRANSMISSION		TR GEARBOX, hp	% TIME
	BEFORE TR* DRIVE, hp	AFTER TR* DRIVE, hp		
CRUISE	260	237	23	84
HOVER	270	244	26	5
AUTOROTATION	2	17	15	7
CLIMB	300	265	35	3.6
TRANSIENT	340	280	80	0.4
MANEUVER			150	0.01

*TR = TAIL ROTOR

hp for the main transmission and 26 hp for the tail rotor gearbox. These loads are converted into torque according to Eq. 4-38 and used for gear tooth size and bearing life determinations.

Gear teeth must be designed for a balance between surface durability and tooth bending fatigue loading such that the gear teeth will, based on initial design loads, have an infinite life. Properly designed gear teeth will have a surface pitting as their principal life-limiting failure mode. Gear tooth strength parameters and criteria are discussed further in Chapter 4, AMCP 706-202 and Refs. 48-51.

4-9.3.3 Steady Loads

Most drive system loads are fatigue-related dynamic loads. However, low cycle and static loads also must be considered. For example, crash loads on the mounting system and ultimate loads on drive shafting should be established to account for brush, tree, grass, and water strikes by the tail rotor. Mounting loads, in addition to torque-related loads, should include crash load factors in accordance with par. 4-5.3.

Tail rotors add thrust and bending loads to the tail rotor drive shaft that must be combined with the dynamic loads in determining endurance limits. Main rotors may be supported so that the main transmission components are unaffected by rotor loads. If the rotor loads are reacted by the drive system, associated deflection effects must be considered, along with the normal load reaction design. Also, the integrity of the main rotor supporting structure is a critical parameter and may require higher design loads than internal gearbox components so that autorotation is still possible in event of an internal gearbox failure. Steady loads from the use of rotor brakes also must be reacted through the transmission housing. Par. 7-6.5 describes predesign considerations for rotor brakes.

Control system bell cranks and levers often are supported from the drive system housings. Loads from the control system should be reacted adequately with a minimum of deflection and with an adequate margin of safety.

Some gearboxes have provisions for internal gearbox jam-up protection by means of a shear-section structural fuse that will open to permit maintenance of rotor speed. Careful consideration must be given to the design criteria for such components to insure that malfunctions do not occur under any normal operational conditions of the aircraft.

4-9.3.4 Fatigue Loads

Historically, most drive system failures of a serious nature are caused by fatigue fractures. The following list indicates some of the reasons for past service-related fatigue failures:

1. Consistent operation above specification torque limits (high stress/low cycle loads)
2. Stress raisers (notches) caused from tool marks and scratches in critical areas
3. Undetected damage allowed to remain on the aircraft; e.g., bullet or fragment damage, buckled drive shafts from tail rotor strikes, and corrosion
4. Inadequate fatigue testing to establish realistic component life
5. Higher loads in critical areas than indicated by stress analysis
6. Metallurgical defects, resulting in reduced endurance limits
7. Quality control problems, e.g., section sizes less than minimum tolerance limits
8. Assembly errors
9. Reduced strength caused from overheating by a previous malfunction; i.e., spinning bearing races, jammed bearings from loss of oil and overheating, and excessive deflections
10. Poor process control; e.g., grinding-burn cracks at gear tooth roots, loss of case hardness as a result of improper grind clean-up, hydrogen embrittlement from plating, and chemical attack of critical surfaces
11. Misalignment during mounting of accessories and drive shafts, causing excessively high alternating bending moments.

It can be seen that human factors are critical and must be accounted for in order to attain adequate fatigue lives or endurance limits.

Torsional oscillations and alternating bending moments in drive shafts must be verified by and correlated with flight test strain gage data before new helicopters become operational. This is necessary because some reversed loading cannot be fully analyzed—such as torsional oscillations caused by engine fuel-governor servo instabilities, combined loading from multiple load paths, internal deflections, and rotor feedback oscillations during various maneuvers and flight attitudes.

4-9.3.5 Load Analysis for Typical Helicopter

The following subsystems, typical of single rotor-tail rotor helicopter of small- to utility-size, are sufficiently different to require specialized load analysis treatment:

1. Main and tail rotor gearboxes
2. Free-wheeling unit (clutch)
3. Spiral bevel gear stage
4. Planetary gear stage
5. Spur or helical gear stage
6. Main housing
7. Drive shafts and couplings.

4-9.3.5.1 Main and Tail Rotor Gearboxes

Tail rotor gearboxes and most main rotor gearboxes not only transmit torque with a change in speed and direction, but also support and react rotor loads. These rotor loads must be considered during the predesign of the gearbox and output gear shaft.

As described in par. 4-9.3.1, the rotor cubic mean loads must be determined for the aircraft mission spectra. A typical thrust spectrum for a small helicopter tail rotor gearbox is shown in Table 4-7.

A cubic mean tail rotor thrust, derived from Table 4-7 and using Eq. 4-39, must be assumed to act in conjunction with a mean transverse load. These thrust and transverse loads act in conjunction with the design torque spectrum.

The output-shaft bending-fatigue loading also is affected by cyclic torques and loads from the rotor. For the small helicopter example, a $\pm$150-lb cyclic load is applied perpendicular to the shaft centerline in conjunction with a cyclic torque of $\pm$450 lb-in.

Additionally, the shaft *shall* have a calculated critical speed margin of $\pm$10% of any operating speed of any shaft section, as required by MIL-T-5955. Statically, the shaft should not take a permanent set from normal operating mission or handling loads. Small helicopter military missions, for example, frequently involve tail rotor strikes of various objects such as birds or tall grass, and the helicopter must be capable of mission completion without excessive tail rotor vibrations. As a result, shaft stiffnesses that are higher than

TABLE 4-7
TAIL ROTOR THRUST SPECTRUM—TYPICAL SMALL HELICOPTER

TAIL ROTOR THRUST SPECTRUM · LOH		
CONDITION	THRUST, lb	TIME, %
EXTREME MANEUVER (POWER OFF)	+ 390	0.3
MANEUVER	+ 230	4.4
INTERMEDIATE	+ 112	77.4
CRUISE	+ 70	17.9

normal may be required to account for the loads encountered from typical Army missions.

Limit loads, which are the maximum operational static loads, for which permanent deformation is not permissible, are higher than loads causing fatigue stresses. For the small helicopter example, the limit loads are:

1. Thrust: 430 lb
2. Angle of Thrust: 15 deg
3. Input Torque: 3260 lb-in.
4. Control System: 585 lb.

Typical small helicopter tail rotor gearbox loading diagrams are shown in Fig. 4-45. It can be seen that a complex set of loads, both steady and cyclic, must be reacted through the gear shafts to the housing via bearings and finally to the attaching aircraft structure. Gear shaft B, the tail rotor drive shaft, is the most highly loaded element, and is frequently the only life-limited part of the gearbox. A description of the loads from input torque T_i and output torque T_o, control loads P_1 and P_2, and tail rotor reaction loads R_T and R_S is shown in Table 4-8. A main transmission that supports the main rotor would have similar load characteristics to consider in its design.

All the shaft loads described in Table 4-8 must be transmitted to the housing through the bearings as a combination of axial and radial loads. Housing and bearing deflections must be controlled to prevent additional loading from excessive deflections at the gear mesh.

Gear tooth contact loads vary during the tooth mesh from sliding to rolling action. These loads result in tooth bending, compressive fatigue (Hertz), and the scoring or wear-producing pressure-velocity (P-V) factors. Values of P-V may be high enough to result in excessive wear or scoring unless adequate lubrication is provided. Deflections or geometrical errors causing less than optimum tooth patterns may cause gear tooth contact loads to become unmanageably high. Excessive wear, causing short MTBFs and TBOs, will result in poor mission effectiveness.

Field failures of the drive train have resulted in the past from inadequate load analysis. Rotating beam bending loads are especially serious if quality control problems also are present; e.g., stress raisers from poor machining, less-than-minimum sections, or heat treatment deficiencies.

Because all loads must pass through the gearbox housing and attachment lugs into the aircraft structure, redundant load paths should permit adequate structural integrity in the event of a loss of one load path, such as a mounting lug. The gearbox housing should be designed to be strong enough to react crash loads without either causing catastrophic results during flight or injuring the crew during crash conditions.

It can be seen from the previous discussion that the loads applied to a tail rotor gearbox, or any gearbox that reacts rotor loads, combine into a complex pattern that is difficult to analyze. Although a complete analysis normally would not be performed during the preliminary design phase, the principal static and dynamic loads should be analyzed sufficiently to insure structural integrity within the selected size, weight, and performance of the drive unit.

4-9.3.5.2 Free-wheeling Unit

Free-wheeling units are one-way clutches that transmit torque in one direction, but that free-wheel in the opposite direction. During autorotational flight they allow rotor speed to be maintained without the rotor airloads being required to drive the engine or other drive train components.

Other helicopter applications may require continuous overrunning when more than one rotational speed is used by the drive system. For example, the XV-3 tilting prop-rotor helicopter used a two-speed transmission that required continuous operation of separate free-wheeling units in either the helicopter or the high rotor speed flight mode. Multiple-engine installations also require one-way clutches so that free-wheeling can occur in the event of an engine shutdown. One-way clutches usually are located in the high rotational speed areas, where torques are lower, for best weight and cost-effectiveness.

Both operating modes, drive and free-wheel, must be considered during a load analysis because each mode contributes different loads to the drive system.

1. Drive mode. The operating, or torque-transmitting, mode transfers tangential forces by a wedging action whereby the inner race and outer races are connected by sprags or rollers. Sprag-type clutches should be positioned and separated so that multiple load paths

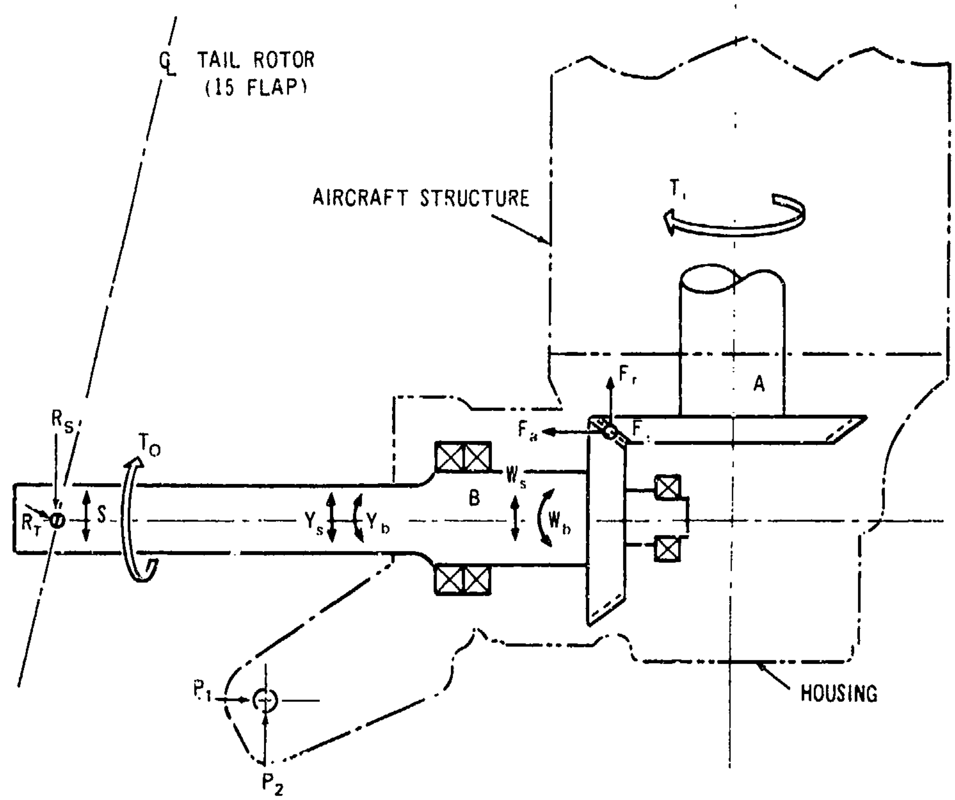


Fig. 4-45. Typical Tail Rotor Gearbox Load Diagram

transmit torque equally at a precise wedging angle between the races. The angle is selected so that at maximum or limit torques the sprags will not go over center or slip.

In this mode, the following should be analyzed:

a. Surface compressive fatigue (Hertz stress) of the races and rollers
b. Hoop stresses of the races
c. Position of sprags or rollers at maximum load.

Extremely hard sprags or rollers and races are required to prevent brinelling and compressive fatigue failures. Minimum case hardnesses of R_c 60 and adequate support by proper core structures are required for the free-wheel components.

Both inner and outer races must be sufficiently rigid to prevent excessive deflections under the high hoop stress loads. Cases of free-wheeling unit failure have occurred when the sprags went over center because of excessive race strains. Bearing support and geometric accuracy must permit equal loading over the length of the sprags or rollers. Excessive end loading may cause brinelling and affect the overrunning free-wheeling mode.

2. Free-wheel mode. Successful one-way clutch operation is dependent upon a low coefficient of friction between the races and sprag or roller units to minimize power loss and lubrication requirements. Sprags and rollers are controlled by springs so that contact with the races should never be lost due to centrifugal forces. This feature also insures engagement when sudden torque is applied in the drive direction and minimizes impact loads upon engagement. Normal forces during the free-wheel mode must be low because the differential speeds between inner and outer races are high. The heat generated by friction in the free-wheeling mode must be stabilized to allow continuous or high-altitude autorotational clutch operations. Successful one-way clutches require a lubrication system and surface finishes adequate to maintain an oil film. Further information on overrunning clutches is included in par. 7-6.

TABLE 4-8
TAIL ROTOR GEARBOX LOADS

LOAD	TYPES	DESCRIPTION
F_t	TANGENTIAL, GEAR TOOTH	STEADY ± OSCILLATING
F_r	RADIAL, GEAR TOOTH	RADIAL COMPONENT OF TANGENTIAL
F_a	AXIAL, GEAR TOOTH	AXIAL COMPONENT OF TANGENTIAL
W_s	ROTATING BEAM SHEAR	FATIGUE LOADS
W_b	ROTATING BEAM BENDING	FATIGUE LOADS
Y_s	MAST SHEAR	STEADY + OSCILLATING
Y_b	MAST BENDING	STEADY + OSCILLATING
T_o	TORQUE REACTION LOADS (FROM TAIL ROTOR)	STEADY + CYCLIC
T_i	TORQUE REACTION LOADS (INPUT DRIVE)	
P_1 & P_2	CONTROL REACTION LOADS	STEADY (MAY CAUSE DEFLECTION LOADS
R_T	TAIL ROTOR THRUST	CUBIC MEAN AVERAGE STEADY LOAD
R_S	DRAG LOAD FROM ROTOR AND FLAT PLANE AREA	STEADY (INCLUDING GUST LOADS)
S	DYNAMIC LOADS VIBRATION FROM IMBALANCE AIR LOADS RESONANCES FROM CRITICAL SPEEDS AND HARMONICS MISALIGNMENT OF ROTOR MOUNTING	

4-9.3.5.3 Main Transmission Bevel Gear Stages

Par. 4-9.3.5.1 discusses tail rotor gearbox loads using a spiral bevel gear example for a small helicopter. This paragraph deals with predesign concepts involving main transmission loads. Main transmissions typically differ from tail rotor gearsets in the following areas:

1. Wider range of speeds and loads
2. Wider tooth faces and higher tooth contact ratios
3. Larger reduction ratios
4. More complex lubrication systems
5. More critical tooth contact patterns
6. Combined loadings (two gears on one shaft)
7. Ring gear mounting requirements
8. Bearing arrangements.

Whenever drive train direction changes are required in aerospace applications, spiral bevel gears most likely will be used. One common bevel gear system being used in aerospace applications has gear tooth geometry arranged to achieve a balance in bending and compressive fatigue allowable loads and in gear tooth reaction loads (radial and axial). The gear tooth geometry in many high-load applications considers tooth contact ratios adequate to minimize gear noise. Gear tooth geometry also can be adjusted to produce lower net loads for efficient bearing sizes and arrangements. Bearing radial and thrust loads are determined by a vectorial summation of the tangential, radial, and thrust loads produced by bevel gear geometry. When more than one gear is supported by a bearing complex, all the vectors from each gear system must be combined.

Trends indicate that input gear stages will be required to operate at higher and higher speeds in the future. The current state-of-the-art includes gear tooth peripheral speeds to 30,000 fpm. Dynamic loads at higher speeds must be considered and analyzed in the preliminary design phase to insure feasibility. Damping techniques and/or a higher degree of tooth accuracy also should be considered to minimize dynamic loads at the higher speeds.

The trends are to employ gears of finer pitch with larger contact ratios (more than two teeth in contact) in order to provide quieter gears and higher allowable loads. Wide tooth faces, approaching 33% of the cone distance, require greater design and manufacturing accuracy to optimize tooth contact patterns at maximum continuous power ratings. Pinions and gears normally are straddle-mounted and supported so that deflections are minimal and consistent; however, overhung mounting arrangements can provide adequate stiffness. Tooth developments and gear cutting equipment must be able to assure constant contact pattern control in production. A gear tooth with a large contact pattern area is required to assure a load distribution compatible with calculated bending and compressive fatigue margins. With these controls, and with high quality gear materials that are carburized and ground properly, Gleason bending stress allowables in the range 35,000-40,000 psi and compressive fatigue (Hertz) values to 260,000 psi can be expected during the 1970s.

Gear wear and scoring are functions of the load-carrying ability of the lubricant used. Latest state-of-the-art developments indicate that scoring and wear are related to the Ryder scoring index of the oils. The Ryder scoring index is determined by a standard test procedure and reflects the load carrying capability of a lubricant. Continuing improvements in the synthetic lubricants will result in continuing reductions in gear wear rates.

Bolt mounting of ring gears to gear shafts should include an adequate number of bolts so that the frictional forces (from clamp-up) can carry the maximum torque to eliminate potential fretting or erosion problems. Hole clearances and bolt head seating must be such as to preclude the possibility of only one or two fasteners carrying the torque load. There have been several cases where bolt heads have popped as a result of bending loads on the bolts. Best design practice allows only tensile loads in the mounting bolts, with sufficient tension to permit the static friction forces to be higher than the highest expected transient load. The static friction coefficient selected for the load analysis should consider oil-lubricated members and not dry coefficients of frictions. Some ring gears have been mounted so that splines carry the torque loads. However, fretting corrosion from the working of relatively flexible ring gears has been the cause of service problems in past designs. Further information about this type of installation is contained in Chapter 7.

Bearings should be arranged for the most efficient management of the gear load paths. Gears and their bearings should be designed as a unit to obtain the optimum package. Manufacturer's recommendations must be reviewed, understood, and qualified by the transmission designer. It also is important that the bearing manufacturer understand the application as fully as possible prior to making recommendations. The bearing manufacturers can be relied upon to verify the most efficient arrangements and bearing geometry for minimal deflections and optimum bearing life. Anti-Friction Bearing Manufacturers Association (AFBMA) life calculations also may be used. Present practice is to use computer programs to determine the optimum bearing design.

4-9.3.5.4 Planetary Gear Stages

Planetary gears are used commonly on the higher torque output stages because the torque loading can be shared by multiple planet pinions. If six planets are used, for example, gear tooth loads can approach one-sixth those experienced with a single mesh stage Much finer gear diametral pitches are possible, along with smaller gear diameters. Generally, planetary stages are more compact and potentially lighter, but at the expense of more complexity by virtue of the larger number of gears and bearings required.

The arrangement of planetary drives most commonly used is the fixed ring type with the sun gear driving and the planet carrier being the driven or output member. This arrangement typically produces a reduction ratio above 3.0 to 1. Other arrangements of planetary drives have been used in helicopters to produce reversed direction drive by fixing the planet carrier. Driving of the ring gear with a fixed sun is used when reduction ratios less than 2.0 to 1 are desired.

A load diagram of a standard helicopter planetary stage having a driving sun gear, a driven planet, and a fixed ring gear is shown in Fig. 4-46.

The sun gear tangential forces T_S and the separating forces S_S of all meshes, assuming equal load sharing, will provide a balanced load condition on the sun gear. The tangential load T_P on the planet gear at its two mesh points produces a resultant output load W_P at each planet equal to the sum of the applied tangential loads. The separating loads S_P are equal and opposite, resulting in a zero force at the output necessary for maintenance of planet gear tooth alignment and positioning under load. A couple, resulting from twisting deflection, occurs as a function of rigidity and the amount of offset between the planet pinion line of ac-

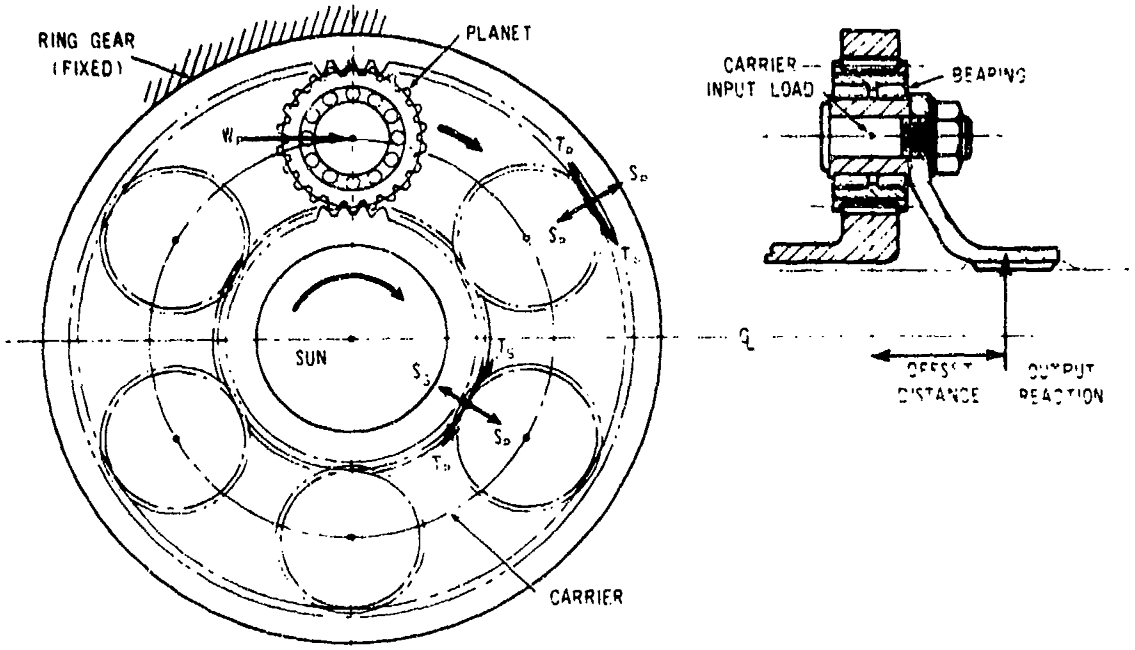


Fig. 4-46. Planetary System Load Diagram

tion and the torque reaction centerline of the output member.

Although planetary systems are the lightest and most compact units for high-torque gear stages, their design must provide for minimum component misalignment in order to attain a near uniform distribution of load on all planets. Development programs sometimes are required to optimize gear tooth involute profile modifications in highly loaded systems. Tooth scoring tendencies resulting from inaccuracies and deflections of the multiple gear meshes usually are controlled by proper balance of involute profile modifications. A small lead modification also may be developed to assure equal loading across the tooth faces.

The floating sun gear requirement may present a development problem on the spline drive if the angular misalignment, spline length, and lubrication are not correct. Fretting of the splines may result in short-life components.

Cal .30 and .50 hits and loss of lubrication are more critical with planetary systems than others, as evidenced by battle statistics and test programs with current U.S. Army helicopters. However, this problem has been alleviated by providing for a redundant oil supply in the original gearbox design. The ability to return to base without transmission jamming is an important consideration for future predesign evaluations.

4-9.3.5.5 Spur and Helical Gear Systems

The present state-of-the-art in helicopter and engine spur and helical gear systems is documented in Refs. 52 and 53. Also, AGMA and Aerospace Gearing Committee standards are well developed and should be used in predesign gear tooth sizing and load determinations.

Straight and helical spur gears have been used for primary transmission drives, while straight spur gears usually are used in tail rotor and accessory drives. The use of spur gears that produce only radial and tangential forces, and require no provision for gear thrust loads, simplifies the design and assembly-disassembly procedures. Bearings may have a degree of axial freedom, thus requiring no preloading or shimming. The lack of thrust with spur gears permits mounting with roller bearings, which provide higher capacity per pound of weight. The weight of the gear, if mounted on a vertical shaft, normally can be reacted by the end faces of the rollers. Ball bearings also can be used in lieu of a roller bearing if desired. In lightly loaded systems, such as accessory drives, ball bearings more commonly are used. A trade-off exists, however, because spur gears usually are noisy compared to single or double helical gears with their higher tooth contact ratios.

When loads are insignificant, such as in tachometer and lube pump drives, lightweight spur gears of lower quality than the primary gear stages have been used in an effort to reduce cost. However, service experience has shown that premature transmission removal and overhaul has resulted because of excessive accessory gear tooth wear. This wear eventually may activate the chip detection system or cause iron content levels in the oil analysis to exceed prescribed limits. To be cost-effective, the accessory drive train should have accuracy and hardness adequate to achieve a life at least equal to, and preferably longer than, the primary train.

Helical gears are used commonly for high-horsepower applications. The helical gear permits higher tooth contact ratios, resulting in a generally smoother and quieter drive. The helical gear produces axial loads that are proportional to the helix angle, and these loads must be reacted by the bearing arrangement and housing.

With the advent of practicable electron beam (EB) welding techniques, two helical gears may be matched in a herringbone arrangement to eliminate or cancel the axial loads that are characteristic of helical gear meshes. This method of manufacture dispenses with the center tool relief groove otherwise required for a one-piece herringbone gear. The herringbone arrangement also can be achieved with a two piece design. Bearings for herringbone gears can be of lower capacity and simplified to accept only radial loads. This type of mesh requires axial freedom of one member to the other to permit equalization of gear tooth loads on the double width gear, requiring the bearing of one gear to provide axial location and to react the weight of the gear mesh.

A trade-off of efficiency exists whenever spur, helical, or double helical gears are used as idler gears, i.e., for positioning or control of direction of rotation. When an idler gear is located in a drive train, it is driven by one gear while it drives another gear. This results in a load on the idler axis equal to the sum of the loads of both meshes. For example, if the idler were in line with the adjacent gears, the bearing loads on the idler would be double those of an equivalent single mesh drive. Fig. 4-46 shows the same additive loading condition for planetary pinions.

4-9.3.5.6 Main Housing Loads

Helicopter transmission housings usually are constructed from aluminum or magnesium because of their low densities and because a considerable portion of the housing is sized to a minimum practicable wall thickness. The castings usually require minimum wall thicknesses of approximately 0.188 in. to assure sound structure compatible with foundry practices. Because the minimum wall thickness is defined by these process requirements, housings usually are designed conservatively for static strength. Fatigue loads on housings also should be considered, particularly in the mounting lug areas. Good design practices are required to minimize deflections resulting from gear and bearing loads.

Bearing reaction load paths are analyzed to assure adequate backup support structure. Experience has shown that some transmissions have had to be derated because excessive housing deflections prevented development of gear tooth contact patterns suitable for maximum capacity. Efficient location of casting structure will permit low weight/hp transmissions.

The amplitudes and directions of the loads resulting from the vectorial addition of tangential, axial, and radial loads must be known to assure proper placement of stiffening ribs. The load paths then can be directed to the transmission support structure in the most efficient manner. Large, unsupported flat plate areas should be avoided to minimize gear deflections.

Gearbox mounting loads related to maximum design torques must be controlled to preclude failures. The attachment fittings should be adequate to maintain support integrity in event of internal transmission failures. Extra margins are needed for those designs requiring transmission pylon support for rotor loads. The strength calculations for all mounting lugs and attachments must consider rotor-induced fatigue loads.

Crash loads must be specified and controlled so that the transmission pylon does not tear loose and endanger the crew. Impact loads must be considered whenever the soft supports reach the end of their travel. Simulated aircraft tests should be specified in the system specification to assure crash integrity.

Transmission housings may be used to react some of the steady and vibratory rotor control loads. Determination of the control load applied to the transmission is necessary for sizing of housing support areas. These loads, if possible, should be based upon test data for similar aircraft. Again, excessive deflections may affect control stability and/or gear performance.

4-9.3.5.7 Drive Shaft and Coupling Loads

The paragraphs that follow treat gear shaft loads that have not been discussed in par. 4-9.3.5.1. Reference also should be made to the detail discussions of pars. 7-5 and 7-6.4.

Gear shaft loading, including aerodynamically induced loading, becomes perhaps the most complex loading condition of the drive system. The tail rotor output gear shaft, as shown in Fig. 4-45, is an example. The loads are three-dimensional; are analyzed in the x-, y-, and z-directions; and then are combined vectorially to obtain shear and bending moments. These combined loads ultimately are absorbed as radial and axial loads through the bearings. A typical loading diagram, a shear diagram, and a bending moment diagram for the y-direction are shown in Fig. 4-47. They are for the output shaft shown in Fig. 4-45, and it can be seen that the most critical section occurs at the sharp change of diameter at the outboard face of the driven bevel gear. Similar diagrams are developed for the x- and z-direction, and loads then are combined vectorially to determine locations and magnitudes of the maximum combined loads. These loads then are used to determine adequate strength at the critical sections.

Loads for the gear shaft should distinguish fatigue and static loading conditions. Failures normally occur from fatigue loading and primarily from excessive rotating beam stresses. Drive shafts can carry steady and cyclic torque loading, depending upon the configuration, and the cyclic loading may be as high as the steady torque.

Drive shafts that connect gearboxes and the engine(s) usually are designed to operate below or between two shaft critical speeds. Therefore, multiple drive shaft bearing supports usually are not required. For shafts operating below the primary harmonic frequency, short unsupported shaft sections are suitable. Bearing supports must be designed to semifloat so as to prevent airframe deflections from inducing high loads on the drive shaft and/or bearings. If the shaft is designed to operate above the first critical speed, damping of the shaft is required during runup of the aircraft.

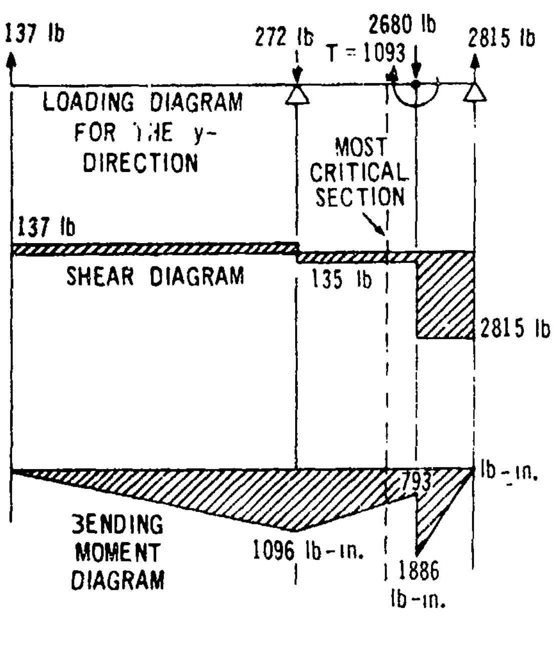


Fig. 4-47. Loading Diagram for y-Direction

Helicopters using tail rotors also must use long drive shafts. These shafts initially are sized to carry the design torque. Wind-up deflections can be large with these high length-to-diameter-ratio shafts and, with reversed loading also present, a dynamic analysis is desirable to assure operation outside the torsional natural frequencies.

Combat experience on a typical scout helicopter mission has shown the need for an extra safety margin in the shafting design due to tail rotor blade strikes on various objects and water. The high peak torsional loads transmitted to the drive shaft from blade impact loading have buckled or wound up some drive shafts, causing a shortening or bending of the shaft. This results in a potential failure of the shaft or mating components for which this type of loading was not initially considered.

Stresses on drive shaft couplings are similar to the torsional stresses on drive shafts, except that couplings also must absorb the reversed bending loads caused from misalignments. These misalignments can be quite large when soft-mounted engines or transmissions are involved. It is essential that the maximum operational misalignment be known so that proper coupling types can be selected. The maximum operational misalignments must include the initial installation misalignments, i.e., engine-to-transmission centerline offsets or nonparallelism tolerances, as well as maximum values of inflight deflections. These additive deflections must not be above the limitations of the coupling design. When couplings are short-coupled on relatively short drive shafts, the designer must take care to align and minimize deflections properly. Types of couplings and their respective capabilities are discussed in par. 7-6.

4-9.4 CONTROL SYSTEM SUBSTANTIATION

This paragraph describes the substantiation of the helicopter control systems during the preliminary design. Methods to be used for the determination of the loads are presented for cyclic, collective, directional and miscellaneous control systems, and the requirements for analysis and evaluation of the strength and reliability of these systems are outlined. The impact of system failure also is discussed.

4-9.4.1 Control System Description

Helicopter cockpit controls typically consist of a floor-mounted cyclic pitch control stick, a collective pitch control stick, and directional control pedals. In helicopter configurations having a single main rotor and a tail rotor, vertical control is obtained by the collective control system, lateral and longitudinal control by the cyclic control system, and directional control by the tail rotor control system. The collective control system consists of a series of push-pull rods, bell cranks, and brackets that carry the pilot-applied control force from the collective stick (which is located on the left side of the pilot's seat convenient to the pilot's left hand) to the root of each main rotor blade. Vertical control is obtained by feathering each main rotor blade by an equal amount.

The cyclic control system, in order of force transmission, consists of a floor-mounted cyclic stick (operated by the pilot's right hand); a series of push-pull rods, bell cranks, and brackets to the swashplate on the drive shaft; and push-pull rods from the swashplate to the rotating rocker arms that are connected to the roots of the main rotor blades. Lateral and longitudinal control is obtained by tilting the swashplate, causing cyclic feathering of the main rotor blades and thus tilting the main rotor disk plane in the direction of the desired motion. A typical control system is shown schematically in Fig. 4-48.

In the case of tandem-rotor helicopters, the cockpit controls are unchanged, but the necessary control moments are produced in a different manner. Longitudinal control is produced by differential collective pitch of the fore and aft rotors, while lateral control is obtained by cyclic feathering of the blades of the two rotors and tilting of both thrust vectors in the same direction simultaneously.

A typical system of upper controls is shown in Fig. 4-49 primarily to identify terminology. There can be much variation in the configuration details of upper control systems, but they are similar in concept.

The stationary swashplate, which encircles the rotor shaft, is nonrotating and has three degrees of freedom: vertical translation, lateral tilt, and longitudinal tilt.

A rotating swashplate ring rests atop the stationary swashplate ring and is separated by a thrust bearing. The rotating swashplate is driven by a linkage to the rotor shaft and rotates with the rotor system. The rotating ring has the same degrees of freedom as its stationary counterpart and follows all motions imparted to the stationary ring on a one-for-one basis.

The linkage that drives the rotating ring is referred to as the drive scissors. It normally is affixed to the rotor shaft by a spline and is attached to the swashplate edge with a ball and socket connection. Two hinges accommodate variations in swashplate height and tilt.

Attached to the circumference of the rotating swashplate are the pitch links, one for each rotor blade. The upper end of the pitch link is attached to the control horn of the blade. The pitch links impart a pitch motion to the rotor blades, controlling the angle of attack of the blade and the magnitude and direction of rotor thrust (Fig. 4-50).

The position of the swashplate determines the height of the pitch link relative to the blade and, consequently, the pitch angle of the blade. Raising or lowering the swashplate parallel to the rotor shaft centerline raises or lowers all pitch links and changes the angle of attack of all blades simultaneously. This is known as collective control input. Tilting the swashplate results in a pitch link height that varies on a sinusoidal basis with a cycle of one rotor revolution. Changing the angle of attack of the blades in this manner is called a cyclic pitch change. As illustrated in Fig. 4-51, a cyclic pitch change alters the direction of the rotor thrust vector while a collective pitch change alters the magnitude of the vertical component of rotor thrust. To obtain acceptable feel and force levels at the cockpit controls may require power boost and/or isolation of the lower control system from rotor-induced loads. Conventional mechanical control systems are categorized by the degree of mechanical influence in the system. Each of the system types illustrated in Fig. 4-48 is discussed in the following paragraphs:

1. TYPE I. Mechanical flight control system. A reversible control system wherein the cockpit controls are linked mechanically by a series of rods and bell cranks directly to the control horn of the rotor blade. Such systems commonly include bungee capsules to react control loads that are too high for pilot reaction. A diagram of a bungee capsule is shown in Fig. 4-52.

2. TYPE II. Power-boosted flight control system. A reversible control system wherein the pilot effort, which is exerted through a set of mechanical linkages, is boosted by a power source at some point in the linkages.

3. TYPE III. Power-operated flight control system. An irreversible control system wherein the pilot, through a set of mechanical linkages, actuates a power control that in turn moves a linkage attached to the control horn of the rotor blade. Irreversibility results from the isolation of each end of the system from the loads applied at the opposite end.

The helicopter control system also may incorporate a stability augmentation system (SAS). A typical SAS

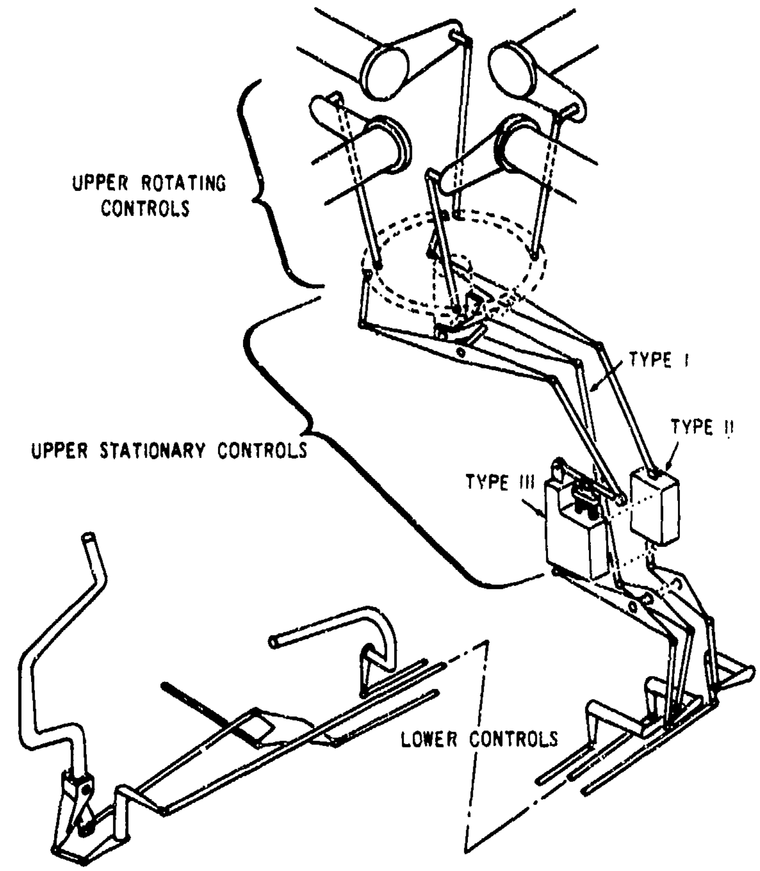


Fig. 4-48. Typical Main Rotor Control System Schematic

consists of a sensing gyro, an electronic control package, and an extensible link in the mechanical portion of the control system for each helicopter axis to be stabilized. The SAS opposes, with limited authority, any transient motions of the helicopter about the stabilized axis. The extensible links are installed in push-pull rod locations between the cockpit and the hydraulic boost actuators.

Directional control of the single-rotor configuration is obtained by feathering each tail rotor blade by an equal amount. The pilot applies the control force to foot pedals, and the force is transmitted through cables or push-pull rods to the tail rotor gearbox. A linkage carries the force along or through the output shaft to the tail rotor. A typical tail rotor control is shown schematically in Fig. 4-53.

In the tandem-rotor helicopter, directional control is produced by lateral tilt of the rotor thrust vectors in opposite directions. Thus, the motions of the directional control pedals must be mixed with the lateral motion of the cyclic stick to control the positions of the two swashplates.

4-9.4.2 Description of Load Sources and Reactions

The primary control loads on a helicopter are aerodynamic and dynamic, and they originate in the systems in which blade angle of attack is being controlled. Main and tail rotor blade pitching moments are fed back into the control system through the pitch links.

While performing their primary function of positioning the blades, the pitch links also must react the blade torsional moments. The magnitude of this load on the pitch link is a function of the blade pitching inertia, the blade torsional moment, and the normal distance from the blade pitch axis to the pitch link. In the unstalled flight regime, the harmonic content of the alternating portion of the pitch link load is primarily one/rev as shown in Fig. 4-54.

Blade moments are caused by inplane movement of the blade center of pressure (CP) away from the feathering axis. These movements may be caused by aerodynamic CP shifts or by inplane deflections of the blade due to Coriolis forces and aerodynamic drag. Also, blade flapping will cause feedback loads in systems where δ_3 (mechanical coupling between the blade pitch setting θ and flapping angle β) is provided. As the blade flaps in such systems, blade pitch angle also must change and the pitching inertia of the blade must be overcome. Upon the inception of moment stall, a higher frequency loading occurs over a portion of the rotor cycle. This load, resulting from a torsional blade oscillation, occurs when the blade is located at approximately 270 deg azimuth. A typical stalled pitch link waveform is shown in Fig. 4-54.

The periodically varying pitching moments about the blade feathering axis feed back into the control system (see par. 6-3). In the case of main rotors, the reaction divides between the collective and cyclic systems. Tail rotors have no cyclic system and hence their collective system must react the entire feedback load.

By considering the parts of various control systems, including the pilot and the controlled element, the sources and reactions of loads are determined. The path of the pitch link loads is through the body of the pitch links into the attachment lugs of the rotating swashplate ring. The rotating ring also is loaded by the drive scissors. The steady force required to drive the rotating controls acts tangentially to the rotating swashplate.

Except for friction losses, all loads are transferred through the swashplate bearing to the stationary controls below. Loads on the stationary swashplate ring as well as on the individual control linkages can be cal-

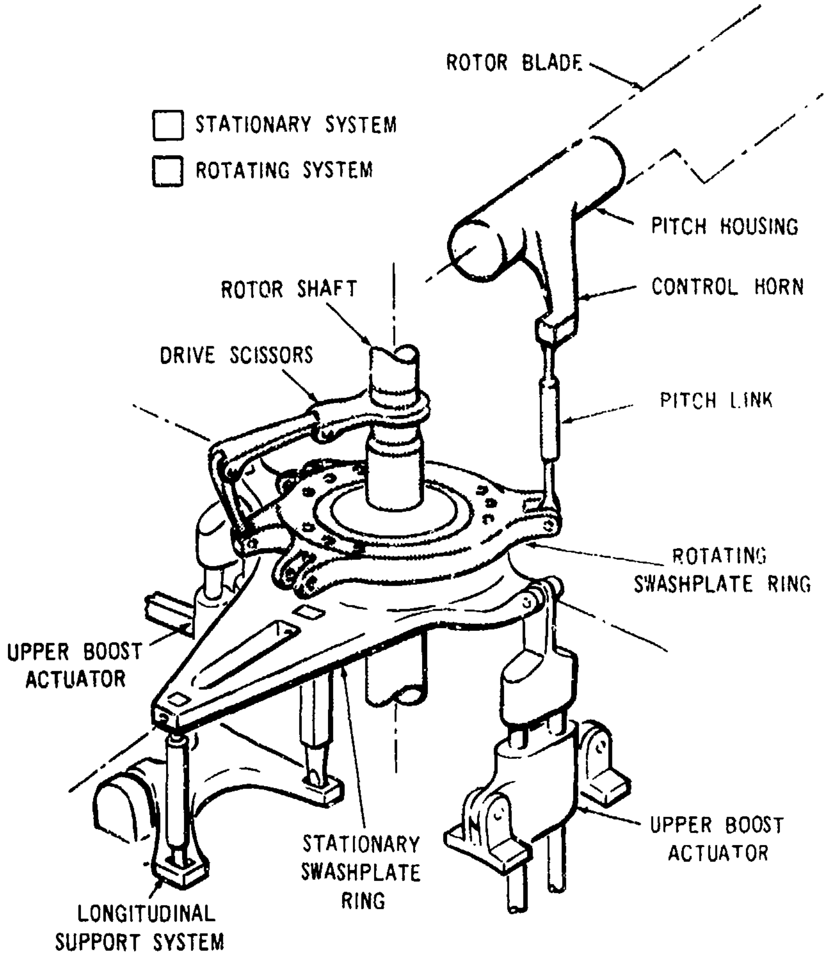


Fig. 4-49. Typical Upper Control System on Aft Rotor of a Tandem Helicopter

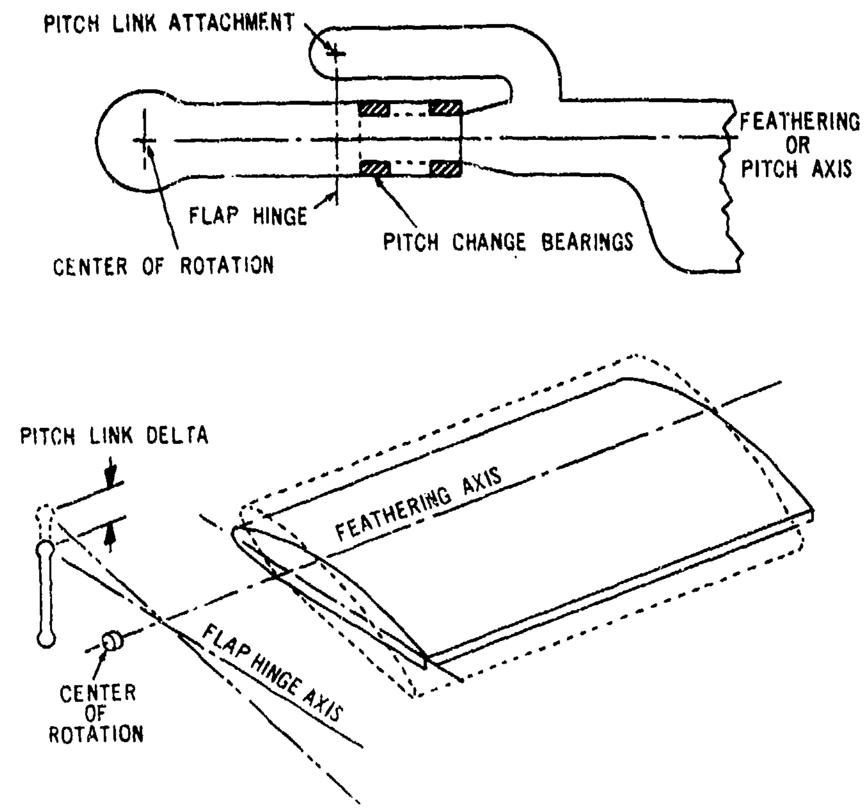


Fig. 4-50. Blade Pitch Motion for Fully Articulated Rotor

culated as a function of pitch link loads. Swashplate thrust loads are obtained by summing the pitch link loads with the proper phasing. Swashplate moment in the fixed system may be obtained by calculating moments due to the pitch link loads about the lateral and longitudinal axes. The loads reacted by the individual control linkages also are determined in this manner.

The loads generated by the drive scissors and carried through the swashplate bearing by friction are relatively small. Because they are in the plane of the swashplate, as opposed to pitch link loads that act 90 deg out-of-plane, they need not be a primary consideration in these calculations, but some provision must be made to react them to prevent rotation of the fixed controls.

The harmonics of stationary swashplate loads are a function of the number of blades in a rotor system and the harmonics of the pitch link load. Alternating thrust on the stationary swashplate can result only from pitch link load frequencies of integer multiples of the number of blades. The alternating thrust will be of the same frequency as the pitch link load frequency. Alternating moment on the stationary swashplate results from pitch link frequencies of one greater or one less than integer multiples of the number of blades and will be the same frequency as the integer multiple of the number of rotor blades.

4-9.4.2.1 Trim Actuators

Trim actuators are introduced into a system to give the pilot force-feel and to allow centering of the stick to trim out CG and airspeed effects. These actuators utilize a spring(s) to provide the required trim force. Because the pilot must work against the trim spring, some of his effort will be reacted by the trim unit. The spring forces must be sufficiently light to permit hovering control. The maximum load that the trim device will be required to react is equal to the maximum spring deflection multiplied by the spring rate.

4-9.4.2.2 Rate Restrictors

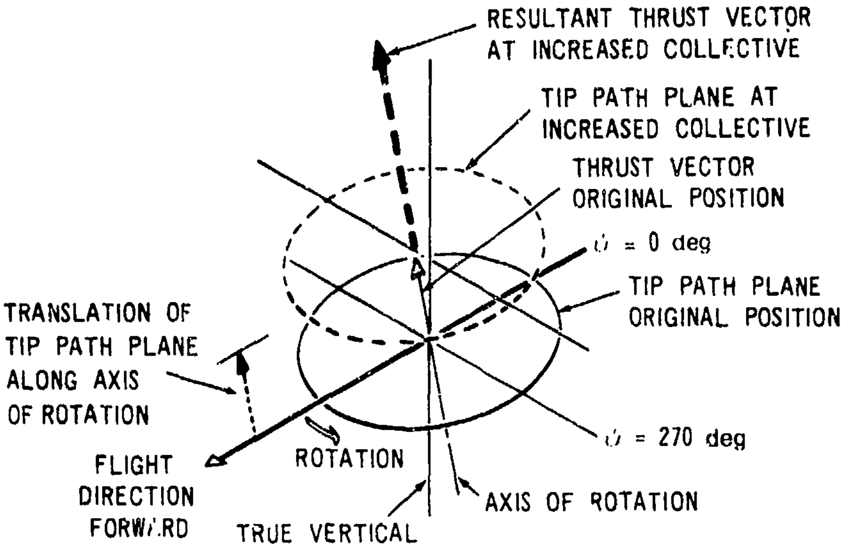
In systems in which rapid pilot stick movement can result in excessive loads, restrictors sometimes are installed to limit (by hydraulic means) the rate at which the pilot can move the controls. Load absorbed by this type of mechanism is a function of the velocity of stick motion.

4-9.4.2.3 Hydraulic Boost

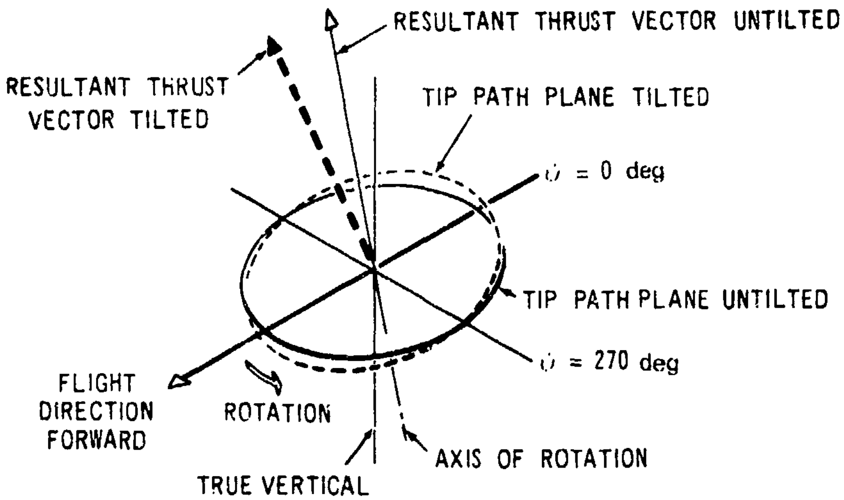
When designing a boosted control system (Type II), pilot effort must be added to the hydraulic effort to obtain the highest load in the system between the blade and the actuator support. Irreversible features, which react vibratory load from the controlled components, may be added to the boost system. In this case, vibratory feedback loads are not reacted by the pilot. However, some or all of the steady loads may be fed back to the pilot.

4-9.4.2.4 Power Systems

A power system (Type III, hydraulic or electrical) is one that applies and reacts all of the load to the controlled component. The pilot moves a servo-mechanism, valve, or rheostat. Redundancies are necessary in the hydraulic or electrical portion of these systems



(A) COLLECTIVE PITCH CHANGE



(B) CYCLIC PITCH CHANGE

Fig. 4-51. Rotor Pitch Change

to allow for possible system failures. Division of the load between primary and backup systems must be considered during the design of powered control systems.

4-9.4.2.5 Stability Augmentation

Stability augmentation systems are designed to have a particular "authority" to control the system. In the case of a system containing extensible links for output, these links must be capable of reacting loads on the same basis as the push-pull rods they replace.

4-9.4.2.6 Vibration Absorbers

It is possible to install dynamic absorbers in unisolated (Type I and Type II) control systems to provide isolation of vibratory forces being fed back from the rotor system. A typical isolator is a spring-mass system having a natural frequency equal to the frequency of the vibration to be isolated (usually *b*/rev, where *b* is the number of blades). The spring may consist of a strip of spring steel of proper length, with the appropriate mass fastened to the free end to provide the required frequency. When this isolator is mounted at a pivot point in the control system where it is excited by the undesired frequency, it will vibrate in response and hence will "absorb" the troublesome vibration. Such an isolator is effective only against vibrations very close to its own natural frequency. Therefore, its usefulness is questionable if the rotor speed, and hence the vibration frequency *b*/rev, varies significantly ($> \pm 1.5\%$).

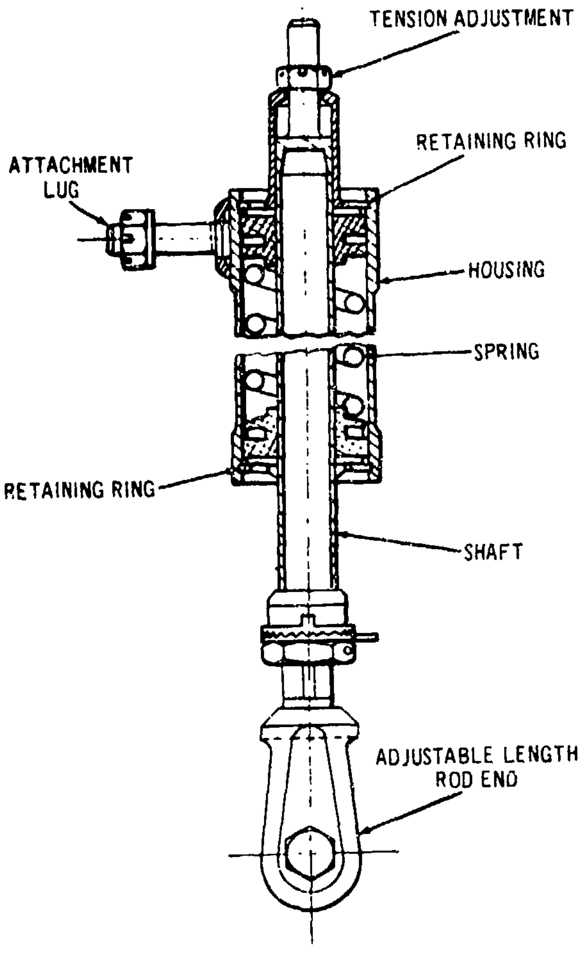


Fig. 4-52. Typical Bungee Capsule

4-9.4.2.7 System Stops

Stops are located at various places in the system to prevent overtravel and subsequent jamming, or interference, of moving parts. One set of stops usually is chosen to serve as the primary point at which system rigging takes place. The stops react control system loads only under specified conditions in which maximum control displacements are required.

4-9.4.3 Determination of Loads

Limit pilot effort loads given in Table 4-9 *shall* apply in control system design. These loads *shall* be distributed through the system to the point of irreversibility and are to be applied with the pertinent cockpit control in any position within its limits of travel. For the design of dual-control systems, 75% of the pilot-applied load *shall* be applied, in the same direction or in opposition, simultaneously at each control station. When duplicate or redundant control circuits are employed, the control system loads *shall* be applied to each system separately with the other system disconnected.

Beyond the point of reversibility, either the boost-plus-pilot-effort loads, power unit loads, or feedback loads *shall* apply for design, depending upon the type of system and the magnitude of these loads. However, the loads specified or calculated for a control system are not necessarily the final criteria for preliminary design. The combined stiffness of the blade and control system required to avoid rotor instabilities, including flutter and weave, should be considered. Preliminary sizing often must be based upon required stiffness rather than on load.

Various techniques are used for the prediction of control system loads. The loads in the upper control system can be related directly to the pitch link loads. The pitch link loads in turn can be related to the loads on and the responding motions of the rotor blades. Control load calculations, therefore, are related to the blade load determinations discussed in par. 4-9.1. Such methods are suitable at least for the prediction of pitch link loads in unstalled level flight.

The complex analytical methods required to compute preliminary control system design loads are handled best through computer studies. Fig. 4-55 presents an outline of a computer method that is related directly to blade airloads. Details of control load analysis are beyond the scope of this handbook because of the many detail differences in systems.

The procedure for determining blade control loads begins with the computation of flapwise response. The resulting flap motions then are used in the chordwise response determination. (Coriolis loads are a result of flap motions.) Lastly, these results are inputs to the program used to determine torsional response. In turn, the pitch link loads about the feathering axis are computed.

Portions of the control system that rotate must be oriented in the proper position when distributing load. As an example, a swashplate is a type of cam that will

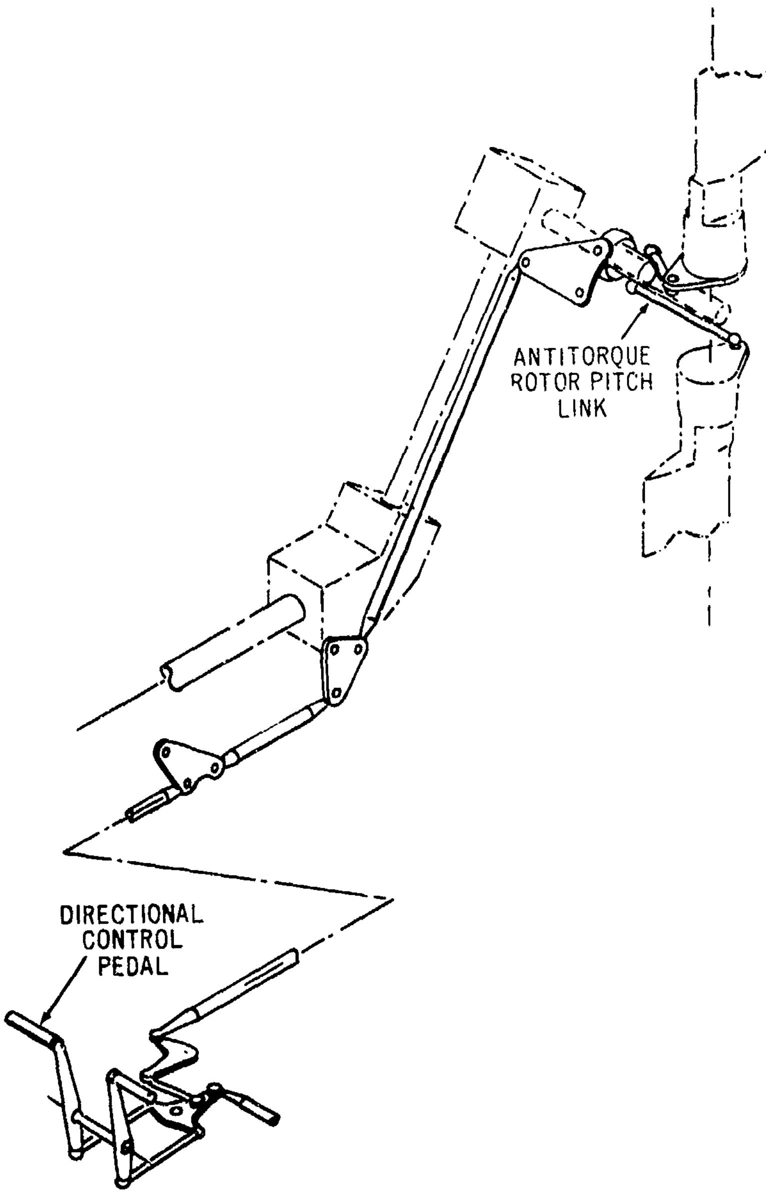


Fig. 4-53. Typical Tail Rotor Control System Schematic

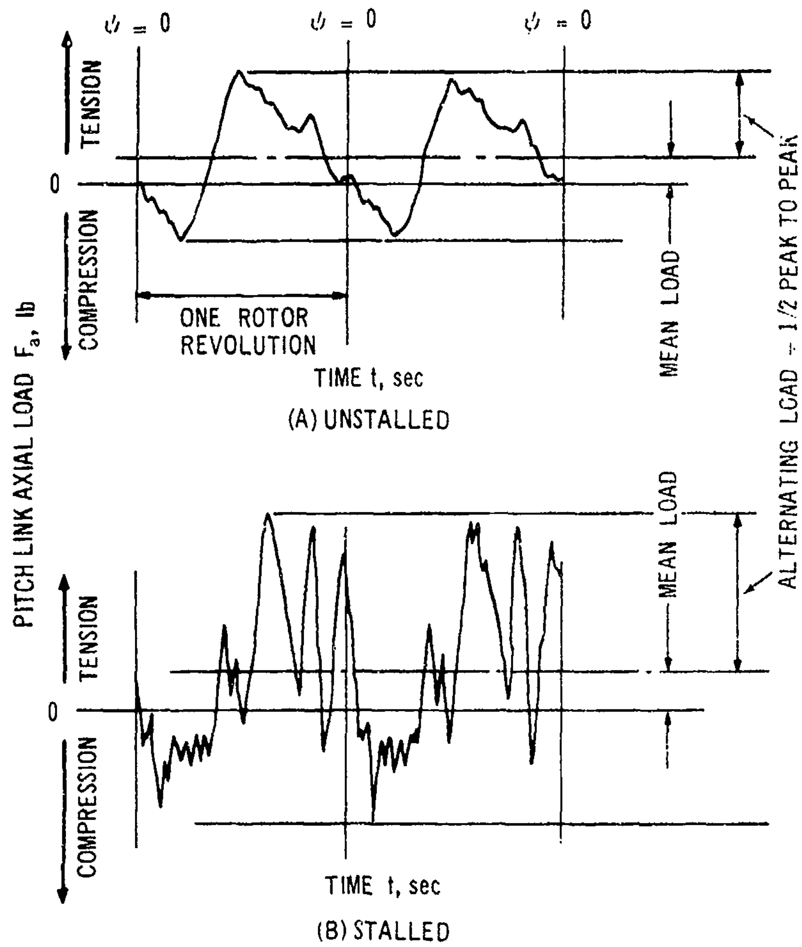


Fig. 4-54. Pitch Link Load Waveform

distribute load sinusoidally. At certain azimuthal positions, no load is transmitted. A rational orientation of the entire system is required as load distribution progresses.

Main rotor control loads usually are mixed at some point in the system, i.e., cyclic and collective loads as in Table 4-9 are mixed above the swashplate to obtain the loads to be applied to the blade pitch control horn. In the single main rotor/tail rotor configuration, there is no need for a mixer in the directional control system. In tandem-rotor systems, where directional control is obtained by differential lateral cyclic control of the fore and aft rotors, loads from the cockpit cyclic, collective, and directional controls must be combined.

Other load-producing phenomena that warrant identification are Mach instability and droop stop pounding. The first of these normally is not considered in detail during preliminary design of a control system. The second is considered only as a ground load condition (par. 4-9.2.4.3).

Mach instability results from blade tip speeds approaching the speed of sound. As Mach 1 is approached, the airflow across the blade is distorted, changing the relationship between the center of lift and the pitch axis and producing a torsional moment of one-half/rev frequency. Accurate predictions of the critical Mach number and the magnitude of pitch link loads during operation at that Mach number are not within the present state-of-the-art. Some production helicopters currently are operating at Mach numbers as high as 0.96 with no apparent problems, while Mach instability has been osberved on other helicopters operating in the same Mach number region. This apparent discrepancy may be a result of differences in airfoil design.

Droop stops are provided to limit the downward travel of a horizontally hinged rotor blade and to prevent blade-fuselage contact during ground operations When the hub arm hits the droop stop, the resulting condition is referred to as droop stop pounding. If the CG of the blade is not coincident with the blade pitch axis, the inertia of the blade at impact will cause a pitching moment that must be reacted by the pitch link.

Definition of those maneuvers that generate high fatigue loads in the control system is difficult. Experience has shown that the maneuvers themselves normally do not produce fatigue loads of significant magnitude. For example, a maneuver conducted at one altitude sometimes will produce much higher loads than an identical maneuver at another altitude. Maneuvers are critical when they prematurely induce the load-producing phenomena discussed previously.

4-9.4.4 Ground Condition Load Criteria

Criteria pertinent to ground loading of controls are very limited. MIL-S-8698 specifies wind loads on unsecured helicopters. Even these criteria must be interpreted to define the combination of steady and gust winds necessary to derive realistic design loads.

There are various ground load situations that can generate control system loads. The most obvious is a wind-induced torque moment on the blade, and MIL-S-8698 provides the criteria for this situation. The wind should be considered to act upon the blade at the angle that results in the greatest pitching moment.

Control loads resulting from the weight of the blade can be magnified by the influence of external forces, such as the rotor downwash of an adjacent helicopter. To allow for this type of situation, a load factor of 4.67 is recommended. Experience has shown this to be a reasonable value for a droop-stop-pounding limit load. Because of the varying flat plate area or lift-to-weight ratios that exist in blades of different construction, other load factors may be equally rational. An applicable limit load factor must be established rationally for each blade design and assessment must be made of the impact upon control loads.

4-9.4.5 Miscellaneous Controls

Miscellaneous controls—such as those for the engine throttle, landing gear retraction and extension, rotor brake, fuel shutoff, and parking brake—*shall* be designed for limit pilot loads ranging from 150 lb (cranks, wheels, or levers) to 133 in.-lb (twist grips) for the desired range of control forces as listed in MIL-STD-1472.

4-9.4.6 System Failure Effects

The impact of system failures upon design loads should be considered during the preliminary design of control systems. If difficulty in predicting operating loads is experienced due to system complexity or lack of experience with a given mechanism, conservative factors should be introduced into the loading analysis. Another technique that can be useful in satisfying system requirements is to incorporate redundancy.

TABLE 4-9
PILOT-APPLIED LOADS (LIMIT)

HELICOPTER CONTROL	COCKPIT CONTROL	FORCE, lb	POINT OF APPLICATION	DIRECTION
LATERAL	STICK	100	TOP OF STICK GRIP	PERPENDICULAR TO A STRAIGHT LINE JOINING THE TOP OF THE STICK GRIP AND THE PIVOT POINT, PARALLEL TO LATERAL AXIS.
LONGITUDINAL	STICK	200	TOP OF STICK GRIP	PERPENDICULAR TO A STRAIGHT LINE JOINING THE TOP OF THE STICK GRIP AND THE PIVOT POINT, PARALLEL TO THE LONGITUDINAL PLANE OF SYMMETRY.
DIRECTIONAL	RUDDER PEDAL	300	POINT OF CONTACT OF FOOT WITH PEDAL	PARALLEL TO THE PROJECTION ON THE PLANE OF SYMMETRY OF A LINE CONNECTING THE POINT OF APPLICATION AND THE PILOT'S HIP JOINT, WITH THE PILOT'S SEAT IN ITS MEAN POSITION.
VERTICAL	COLLECTIVE STICK	150	TOP OF STICK GRIP	± 30 deg TO THE PLANE OF MOVEMENT OF THE CONTROL.

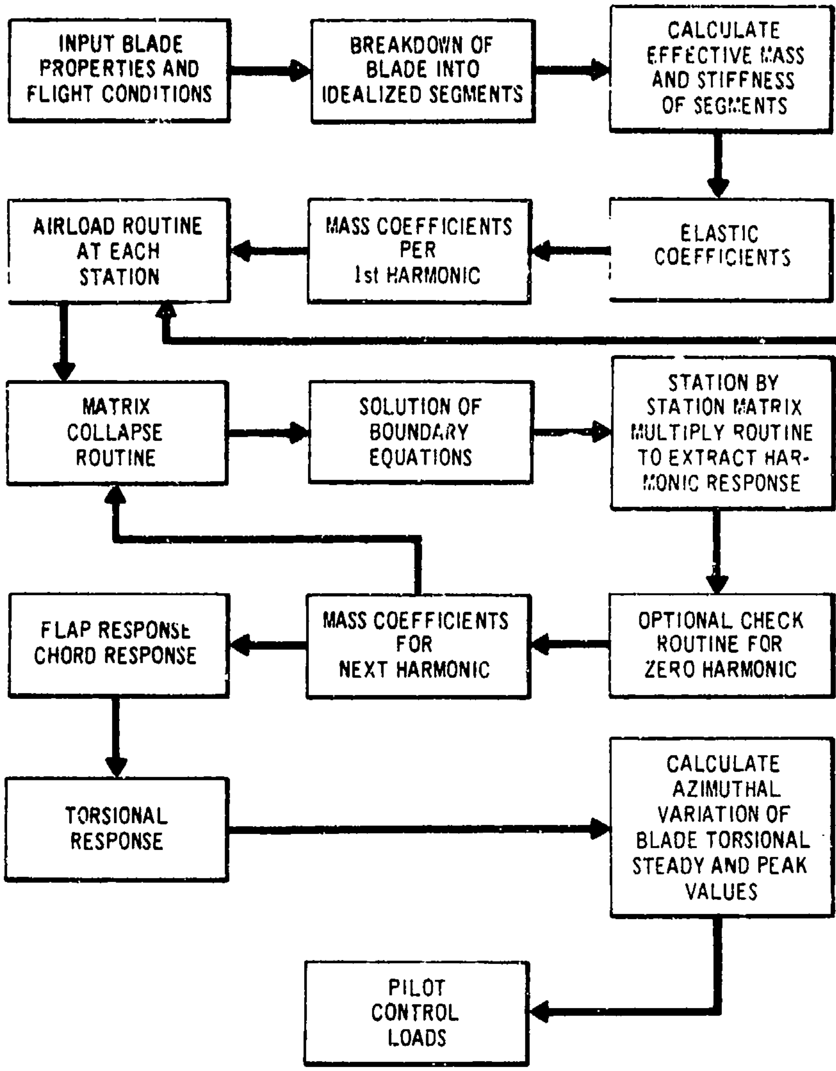


Fig. 4-55. Block Diagram, Control Load Determination

It is not sufficient to design against traditional system failures such as static, fatigue, or jamming. Lack of system stiffness also will promote failure because the controls are only a part of the rotor dynamic system. Rotor weave and similar instabilities are caused by "spongy" control of the blades.

Dual hydraulic systems are required when a control cannot be pilot-actuated following a single hydraulic system failure.

Fail-safe methods should be used where practicable. In this case, failure of a critical member shifts the entire load to another remaining member; therefore, the full design load and required stiffness criteria also are applicable to the redundant part. At the time of a failure, or at the next convenient inspection period, failure of an individual part should be recognizable so that it can be replaced.

4-9.4.7 Control System Substantiation

This paragraph discusses the initial sizing and substantiation of control components during preliminary design. It is important, for purposes of this discussion, to identify preliminary design as the period before parts have been fabricated and hence neither bench test nor flight test data are available. The analyst thus must rely upon past experience, structural criteria, and predicted loads.

The sizing of control components is a matter of equating strength to load. The design loads are generated from structural criteria, the resultant stresses are determined by analysis, and the design stresses then are compared with allowable stresses. An iterative process continues until a favorable balance between design and allowable stress is achieved. The maximum anticipated static loads (limit loads) are evaluated against the minimum expected yield strength to verify that no structural yielding can occur. In addition, the maximum anticipated static load is multiplied by a safety factor, and the structure is evaluated against minimum expected ultimate strength to verify that no failure can occur. A structural evaluation of the effects of alternating (fatigue) loads also is conducted. In the case of fatigue loads, both the magnitude of stress and the number of cycles of stress are important. Each cycle of alternating stress above the endurance stress level is damaging to the structure. After exposure to many cycles of fatigue stress, it may be necessary to retire components from service to preclude fatigue failure. The prediction of the fatigue life is a more complex task than the evaluation of a single maximum static load.

The control load criteria and basic loads are discussed in pars. 4-9.4.3 through 4-9.4.5. Analysis methods are found in textbooks and in contractor design manuals. Static strength allowables for metals are found in MIL-HDBK-5 (nonmetals are not common in control systems).

Fatigue design allowables are not documented widely, primarily because of the many factors that affect material fatigue strength. Fatigue life determination is discussed in detail in par. 4-11.

Preliminary design of fatigue-critical control components does not eliminate the necessity of a bench test program to determine actual fatigue strength; the variables affecting component fatigue life demand bench testing. However, if an expected level of fatigue life is to be achieved in test, a thorough preliminary design evaluation is required. Close attention to the characteristics of fatigue during preliminary design will minimize the necessity for subsequent redesign.

4-9.4.7.1 Lower Controls

The lower controls are a system of mechanical linkages from the pilot to the control system isolation point (par. 4-9.4.1). Criteria relating to pilot-applied loads normally dictate the size of the lower controls. The design loads (Table 4-9) are based upon the physical limits of a pilot and are much larger than the forces normally required to operate the system. They seldom are encountered and hence are considered as static loads. Possible control system loads resulting from ground conditions, such as high winds or during maintenance functions, also should be considered during preliminary design.

4-9.4.7.1.1 Static Design Load

The initial step in sizing an element in the control system is the identification of the applicable critical load(s). Each load or combination of loads required by the criteria is applied at the applicable cockpit control(s) and reacted at the system isolation point. The loads and reactions upon each element of the control system are determined from statics (equilibrium of forces) and control system geometry. The most adverse (highest load) position within the possible travel is assumed.

The critical element load for each possible failure mode is identified from the described load study and a free-body diagram then is drawn. This process is repeated for each element in the control system to determine the maximum loads for each loading condition. While push-pull rods usually are critical in tension or compression, bell cranks and bolts may be critical in bending. Therefore, loadings for each possible failure mode must be established.

4-9.4.7.1.2 Analysis

Stress, deflection, or stability for each of the possible failure modes must be determined by analysis. For example, for a push-pull rod, tension stresses are determined in the section through the bolt hole in the rod end, at the minor diameter of the rod end threads, and at the major diameter of the threads in the barrel of the rod. The shear-bearing failure mode of the rod end also is evaluated. The rod then is analyzed for the maximum compression load, particularly for column instability.

Structural failure normally is associated with a condition where the applied stress exceeds the strength of the material, leading to breakage. Another classical mode of failure is structural collapse or instability Such failures cannot be predicted by knowing the stress in the member but rather depend upon geometric factors and material stiffness. Expressions for the critical or column failure load for columns with different end conditions and nonconstant cross sections, may be found in Refs. 22 and 46. Derivations of these expressions can be found in Ref. 54.

Structural analysis also includes consideration of deformation. Evaluation of control system deformations may be required to assure adequate clearance for moving parts and to optimize the stiffness of the controls. The interaction of control system stiffness and rotor blade performance is discussed subsequently. Appropriate expressions for deformations due to axial load and transverse loads as well as solutions to beam deflection problems for numerous end conditions and loading combinations are tabulated in Ref. 46.

Another consideration during preliminary design is structural resonance. Structural collapse can occur if a structure is subjected to vibratory loads near the natural frequency of the structure; thus the natural frequency of control components should be well separated from rotor and drive system frequencies. Ref. 55 is one of many texts that present the fundamental equations pertinent to the investigation of structural resonance.

4-9.4.7.1.3 Material Allowables

MIL-HDBK-5 contains the standards of strength and stiffness properties of metals commonly used in control systems and also contains allowables for fasteners and, to a degree, the strength characteristics of structural elements such as tubes.

Test coupons of a given metal exhibit small variations in strength and stiffness properties. The statistical significance of this scatter is included in MIL-HDBK-5 data. Strength information is classified statistically while stiffness data are average values; MIL-HDBK-5 "A" or "S" values are recommended. An "A" value is a strength level expected from 99% of the material with a confidence level of 95%, while an "S" value is a material specification minimum. In the event that MIL-HDBK-5 or other approved sources do not contain the required allowable stress, the values used for analysis *shall* be justified or validated by the designer.

4-9.4.7.1.4 Margin of Safety

The margin of safety MS is the numerical expression of the balance between design stress and allowable stress. Simply stated,

$$MS = \frac{\text{Allowable Stress}}{\text{Stress due to Design Load}} - 1 \qquad (4\text{-}40)$$

When the margin of safety is zero, or has a small positive value, a desirable balance between allowable and design stresses has been achieved. If the design is good fundamentally, small positive margins imply a minimum-weight structure. Zero margins rarely are achieved because of material and design constraints.

Margins of safety are determined for both design limit and design ultimate loads and stresses relative to allowable yield and ultimate stresses, respectively, at all critical locations for all control components. These margins *shall* be tabulated for ease of review and evaluation.

4-9.4.7.2 Rotating and Stationary Upper Controls

The rotating and stationary upper controls usually are sized by alternating loads, i.e., by fatigue considerations. The effects upon the control system of peak static loads during the extreme, but seldom encountered, maneuvers and during ground conditions must be evaluated. However, control components normally are adequate for ultimate design loads if they have been sized to withstand the periodic (alternating) loads occurring during each rotor revolution for the numbers of cycles accumulated during the design fatigue life.

The criteria for determining the magnitude of the design alternating loads are not covered specifically in the applicable specification (MIL-S-8698). The intent of the specification is to minimize the possibility of fatigue failure and to provide a minimum of 1000 hr of service life. However, preliminary design objectives for current Army helicopters well exceed the 1000-hr minimum life requirement.

Control loads experienced by a helicopter can be regulated by such factors as speed, gross weight, and maneuvers. However, design loads related to rotor stall may be used to initiate preliminary design because stalled conditions are a practicable limit.

4-9.4.7.2.1 Alternating Design Loads

The detailed fatigue analysis of a control component requires an understanding of the magnitude and frequency of the alternating loads in relation to the intended use of the helicopter. For example, the magnitude of the alternating pitch link load at a given density altitude typically varies as a function of gross weight and airspeed.

Fig. 4-56 shows typical pitch link loads at a common density altitude for two gross weights as a function of airspeed. Considering only the loads shown in Fig. 4-56, the required pitch link fatigue strength is dependent upon the amount of time the helicopter spends at each weight and at the various airspeeds involved.

The curves in Fig. 4-56 are for steady-state level flight conditions; the load shown is the unstalled alternating load of Fig. 4-54(A). Operating the helicopter at different density altitudes will result in different pitch link loads at each gross weight.

The introduction of control displacements to accomplish helicopter maneuvers will cause alternating pitch link loads to increase above the steady-state value during all or portions of the maneuver. The magnitude of this increase will be related to the type, duration, and severity of the maneuver.

It is apparent that the analytical determination of pitch link loads for all altitudes, gross weights, rotor speeds, and airspeeds in level flight and in maneuvers is impracticable. Measured flight loads covering these variables ultimately will be required for the final determination of component fatigue life, but for preliminary design a single alternating load is preferable and is available. The state-of-the-art is such that pitch link loads for steady-state flight conditions can be predicted adequately even into the blade stall region (par. 4-9.4.3).

Fig. 4-57 displays the significant features of a predicted alternating pitch link load as a function of airspeed. The blade stall region is identified by sharply increasing pitch link loads, together with an increase in frequency due to the introduction of higher harmonics of rotor speed, as seen in Fig. 4-54(B).

The helicopter level flight structural envelope normally is limited to airspeeds at or only slightly beyond the inception of blade stall. This is because of the rapid rotor load increase with small increments of speed beyond this point. Maneuvers conducted within the flight envelope usually result in loads higher than the level flight values, due to the fact that, during the time that the maneuver is producing positive accelerations, the helicopter is effectively at a higher gross weight.

The design load for the pitch link, and from it the load in other upper control components, may be based upon the load predicted at the inception of blade stall. The predicted load, increased by an experience factor, becomes the design load. The factor will vary with:

1. Configuration (tandem or single)
2. Utilization (maneuver spectrum)
3. Component material (steel, aluminum, or composite)
4. Desired growth potential (possible new airfoils).

No single factor can be recommended for all helicopters. Experience with helicopters of similar configuration is the best guide. It is doubtful that any factor less than 1.2 would prove adequate.

Because the pitch link alternating load is fundamentally a one-per-rev load, the number of cycles of alternating load accumulates rapidly. For example, if the helicopter operates at 300 rotor rpm, the pitch link experiences 18,000 load cycles per flight hour, or 18×10^6 cycles in 1000 flight hours. Therefore, to provide an acceptable fatigue life, it is imperative that the level flight pitch link loads be below the endurance limit for the part. Hence the pitch link should be sized for infinite life, i.e., design load $\leq$ endurance limit, at the design pitch link load.

4-9.4.7.2.2 *Static Strength*

In addition to the requirement to provide adequate fatigue strength, the capability of the rotating and stationary upper controls for the extreme maneuver and ground-handling load conditions must be evaluated. The design limit load for the extreme maneuver may be derived by multiplying the maximum level flight pitch link load (steady plus alternating) by a factor approximately equal to the design load factor for the helicopter. The rotating and stationary controls should be evaluated for these loads in the same way that the lower controls are evaluated for pilot-applied loads (par. 4-9.4.7.1).

The hydraulic force required of the boost actuators is determined from the loads during maximum maneu-

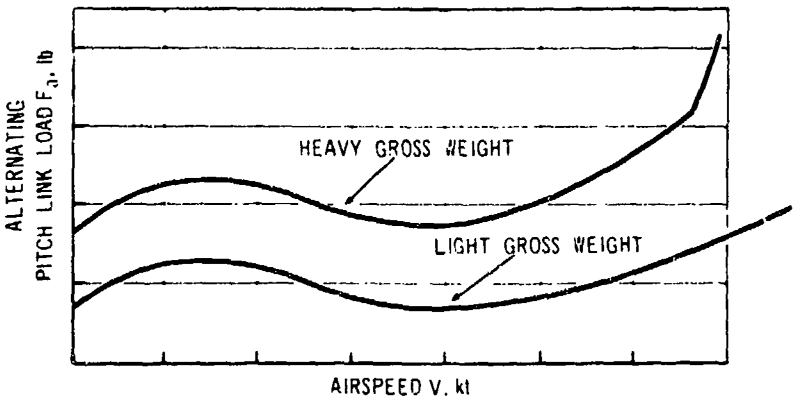


Fig. 4-56. Alternating Pitch Link Load vs Airspeed

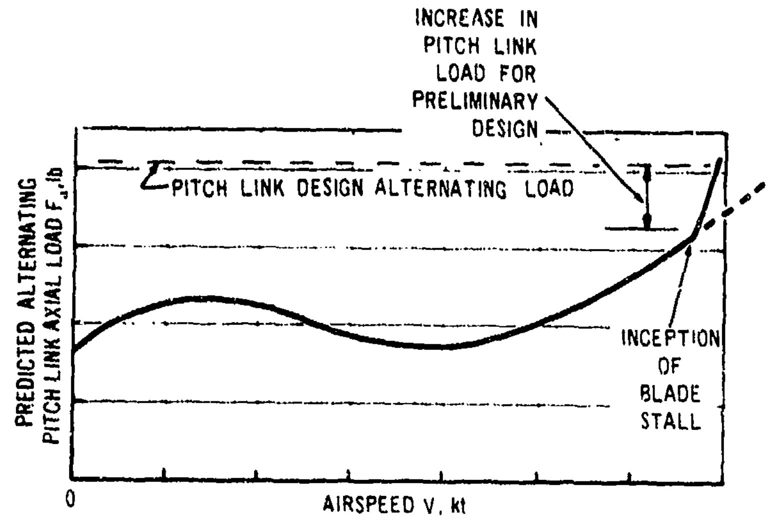


Fig. 4-57. Predicted Alternating Pitch Link Load vs Airspeed

ver conditions. The hydraulic actuator system must have sufficient pressure and piston cross-sectional area to provide control capability (to change blade pitch) under these maximum load conditions. Boost systems powered by other than hydraulic means also must function in the presence of these maximum loads.

4-9.4.7.2.3 *Stiffness Requirements*

The proper functioning of the control system is dependent not only upon strength, but also upon stiffness. The fundamental equation of dynamic motion for a simple spring-mass system as shown in Fig. 4-58 is

$$I\ddot{\theta} + K\theta = 0 \qquad (4\text{-}41)$$

where

I = mass moment of inertia, slug-ft^2
θ = pitch angle, rad
$\ddot{\theta}$ = angular acceleration, rad/sec^2
K = spring rate, ft-lb/rad

Eq. 4-41 and Fig. 4-58 represent a simplification of the actual case, with both damping terms and forcing functions omitted. Also, K represents the total torsional spring rate of the system, which comprises the spring rates of the rotor blade and the control system considered in series; it is the amount of torsion required to produce one radian of angular deflection, and includes the deflection in the control system necessary to achieve the blade deflection. The dynamic response of the system is a function of $(K/I)^{0.5}$. Hence, K, including the contribution of the control system, must be evaluated and controlled to optimize rotor blade performance. The requirements for control system stiffness should be established early in the program and provided for in the preliminary design phase.

4-9.4.7.3 Structural Concepts

The majority of helicopter control components in existence today are safe-life designs. The safe-life method predicts the fatigue life of a component analytically, using component fatigue strength as determined by test, fatigue loads as measured in flight, and the anticipated utilization of the vehicle in service. Fatigue life determination is discussed in par. 4-11.

A fail-safe structure is one in which an obvious indication of an impending structural failure is provided and can be detected in sufficient time to preclude a catastrophic situation from developing. Fail-safety is a structural concept that will receive increased attention in the next generation of helicopter control systems.

Some examples of fail-safe design are:

1. Redundant load paths. Either of two load paths can carry the required flight loads safely for the duration of the inspection interval. Regularly scheduled inspections must be adequate to detect failure and actually must be conducted for the method to achieve its purpose.

2. Standby load paths. An alternate load path begins to carry load only after a structural failure in the primary load path. Once again, the routine inspection must be capable of detecting the failure and must be enforced for the method to be effective.

3. **Partial failure indicator.** A positive indication of partial failure is given in sufficient time to avoid a catastrophic situation. For example, laminated structure may be used, and initial failure of a single laminate could release a dye as an indication of failure. In another system, a structural crack may be detected by means of an internal pressure change. A hollow structure is pressurized or evacuated and a crack in the structure permits pressure equalization with the outside air.

Fail-safe structural concepts are currently in development and should be considered in the preliminary design of control system components.

4-10 AIRFRAME STRUCTURAL SUBSTANTIATION

The structural design criteria applicable to Army helicopters are defined in the preceding paragraphs, and the preliminary design substantiation requirements for several subsystems are discussed. In the cases of the rotor and mechanical drive subsystems, the substantiation of the structural adequacy of the components cannot be readily separated from the development of the applicable loads. However, in the cases of the airframe subsystems, the sizing of components and the substantiation of the ability of the subsystems to withstand the critical design loads are independent of the determination and development of these loads. In the paragraphs that follow, the substantiation of the adequacy of these subsystems and components during the preliminary design phase is discussed. The airframe is further subdivided into fuselage, wing and empennage, landing gear, and internal and external equipment installation subsystems for purposes of this discussion.

4-10.1 FUSELAGE STRUCTURE

4-10.1.1 Fuselage System Description

The fuselage is defined as that structure which supports the useful load, supports and connects the dynamic components, and provides an interface between the aerodynamic and ground environments. The fuselage components include the cabin and cockpit, rotor pylon, engine compartment, cargo and baggage compartments, and tail boom.

The cockpit and cabin contain shear and bending structure that supports crew, passengers, fuel tank, and equipment (useful load); and interconnecting shear, axial load, torsional, and bending structure for the support of dynamic components. This portion of the fuselage also has secondary structure and fairings to provide airstream protection for the occupants and an external contour as required for aerodynamic efficiency.

Seating provisions must include anchoring structure capable of retaining the occupants during crash conditions.

A rotor pylon is a structure, mounted upon the fuselage, that supports a main rotor(s) at sufficient height to give the rotor clearance over the tail boom and tail rotor (if applicable), fuselage, and personnel on the ground. Shear and bending are the primary loadings for a pylon structure.

The engine compartment structure must have firewalls and provisions to support the engine and mechanical drive components. Firewalls generally are flat panels that, if not stiffened for efficient shear capacity, must be stiffened to prevent panel vibration at engine or drive system operating frequencies. Such vibration may result both in fatigue failure of the panel and increased noise levels in adjacent compartments.

Cargo and baggage compartments must have tie-down provisions to retain the contents. Compartments

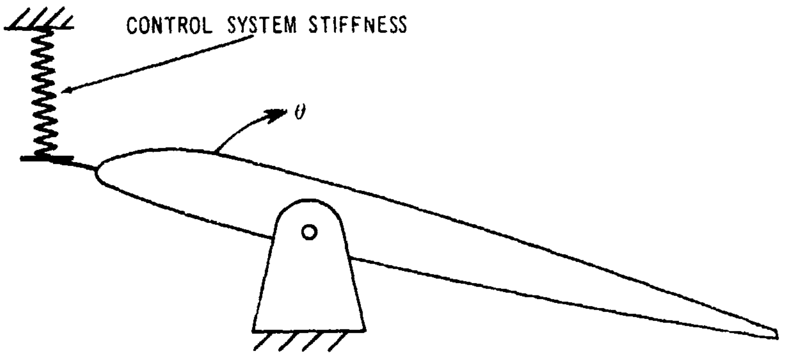


Fig. 4-58. Spring Mass System

used for cargo or baggage should not contain unprotected control linkages, wiring, or flammable fluid lines because linkages could be fouled and jammed, and unprotected wiring and/or lines are a fire hazard.

A tail boom is a structure that attaches a tail rotor and its mechanical drive to the fuselage. Loading is primarily in shear, bending, and torsion. Auxiliary aerodynamic surfaces, such as a vertical fin and a horizontal stabilizer, often are mounted on the tail boom. Care should be exercised in the design of the tail boom to provide adequate stiffness so that the tail boom, with all of its supported masses, will not have natural frequencies close to operating frequencies.

4-10.1.2 Load Sources

The major sources of fuselage load are the rotors, which produce large static and vibratory loads, and the landing gear, which produces high static loads. The landing gear loads actually are dynamic in nature, but because peak landing loads are applied for a relatively short period they are evaluated against static strength properties.

Next in importance are mass-item inertia loads due to flight and landing manuvering accelerations. Airloads due to the dynamic pressure of high-speed flight also are important. Such airloads produce the critical design loads for windscreen, fins, stabilizers, wings, and fairings.

Mechanical drive systems produce loads at the fuselage attachments for speed-change or direction-change gearboxes. In many cases the gearbox supports a shaft that powers a rotor or propeller; and, therefore, the rotor loads are distributed to the fuselage through the gearbox.

Control system loads on the fuselage occur at locations where mechanical advantage and directional changes take place. Anchor points for boost or power cylinders require particular design attention due to the high vibratory loads fed back from rotor controls.

Finally, loading produced by equipment, cargo, and payload occurs during accelerated maneuvers. Handling loads during ground positioning and jacking also must be considered during the preliminary design.

Although secondary structural elements may not be highly loaded under flight or landing conditions, they may have a significant loading during manufacture and this must be considered.

If other factors are equal, the lightest fuselage will be the smallest one that adequately can enclose the required personnel and/or cargo, fuel, fixed equipment, and power plant/drive system. This is due to the heavy dependence of the weight of primary structure upon the length of the load paths and of the weight of secondary structure upon surface area. Fuselage loads also can be affected considerably by drive system loads and by the arrangement of the rotors relative to the fuselage. In the last few years, fail-safe structural considerations have become another important preliminary design consideration for the fuselage, with regard not only to improved flight safety, but also to greater resistance to battle damage and increased assurance of successful mission accomplishment.

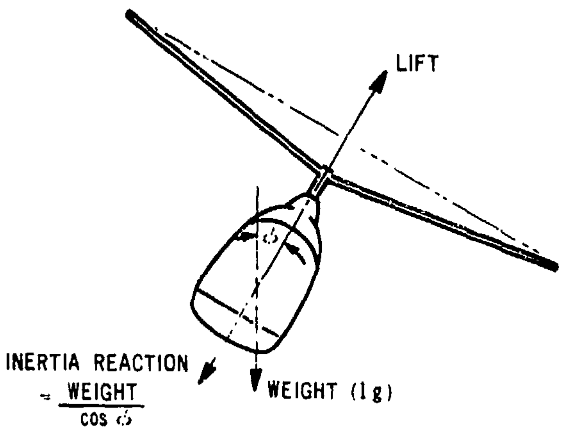
The basic fuselage structure is loaded, in general, by a complex system of external loads, e.g., rotors and landing gear, that are reacted by the inertia of the vehicle and its contents. The internal member loads generally are distributed among redundant load paths in a manner dependent upon the relative stiffnesses of the various load paths. For analysis purposes, this internal load distribution is accomplished either by a comprehensive computer program such as that of Ref. 56, or mathematical model—modified by experience and simplified by engineering judgment—that is amenable to solution with a desk calculator. After the determination of the critical external and internal loads for the fuselage, the structural sizing of the individual members is accomplished in a standard manner by structural analysis.

An additional factor involved in helicopter fuselage design is structural stiffness. Stiffness requirements are determined largely by the rotor configuration, and they cannot be overlooked inasmuch as the fuselage is a part of the dynamic system. A requirement for additional stiffness may add weight if the increase is accomplished in a nonoptimum manner (Ref. 57).

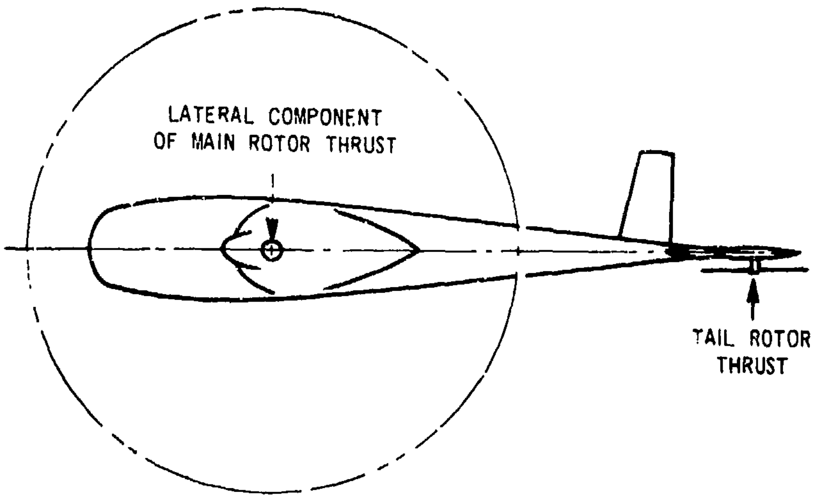
4-10.1.2.1 Flight Maneuver Loads

Flight maneuver loads as they affect the fuselage are associated with either steady-state or transient motions about or along the three coordinate axes of the helicopter. Examples of maneuvers involving relatively steady-state motions are banked turns, hovering turns, sideward flight, yawed flight, and spiral dives. Examples of maneuvers involving transient motions are flares, pull-ups, pushovers, entries into or recoveries from steady-state maneuvers, rolling or yawed pullouts, and control reversals. In general, steady-state maneuvers such as those in Fig. 4-59 are performed with near-constant linear and angular velocities along or about the three vehicle axes, and thus normally do not involve significant angular accelerations. Similarly, the transient maneuvers shown in Fig. 4-60 require changing angular and/or linear velocities, and normally involve angular accelerations. The major linear (along the axes) accelerations occur in the vertical (lift) direction, with smaller values of acceleration along the other two axes. The various accelerations of the fuselage that are of concern for preliminary structural design are derived from the forces and moments applied at the main rotor(s) and tail rotor (if applicable) during these maneuvers. These include lift, drag, and side loads as well as roll, pitch, and torque moments from the main rotor, along with similar types of (yaw) forces from the tail rotor. In addition, there are aerodynamic loads from the tail surfaces (par. 4-10.2).

Par. 4-4.2 specifies the values of vertical acceleration to be used for helicopter structural design at basic structural design gross weight and maximum alternate design gross weight conditions. It is possible in most helicopters to exceed these accelerations at light gross weights; and in some helicopters, it may be possible to exceed these specified accelerations at the basic structural design gross weight. Because the ability to develop sufficient rotor lift to exceed the design load factors may be of operational or military value (allowing increased maneuverability, better speed and/or altitude capability, or increased ferry range), as well as being advantageous from the standpoint of safety, it may be desirable to base the preliminary structural design upon the actual maximum lifting capability of the rotor(s). However, there is some design load factor for any particular type of helicopter beyond which little operational or military advantage can be realized. This is because the increased structural weight offsets any other gains.



(A) BANKED TURN



(B) HOVERING TURNS

Fig. 4-59. Typical Steady-state Maneuvers

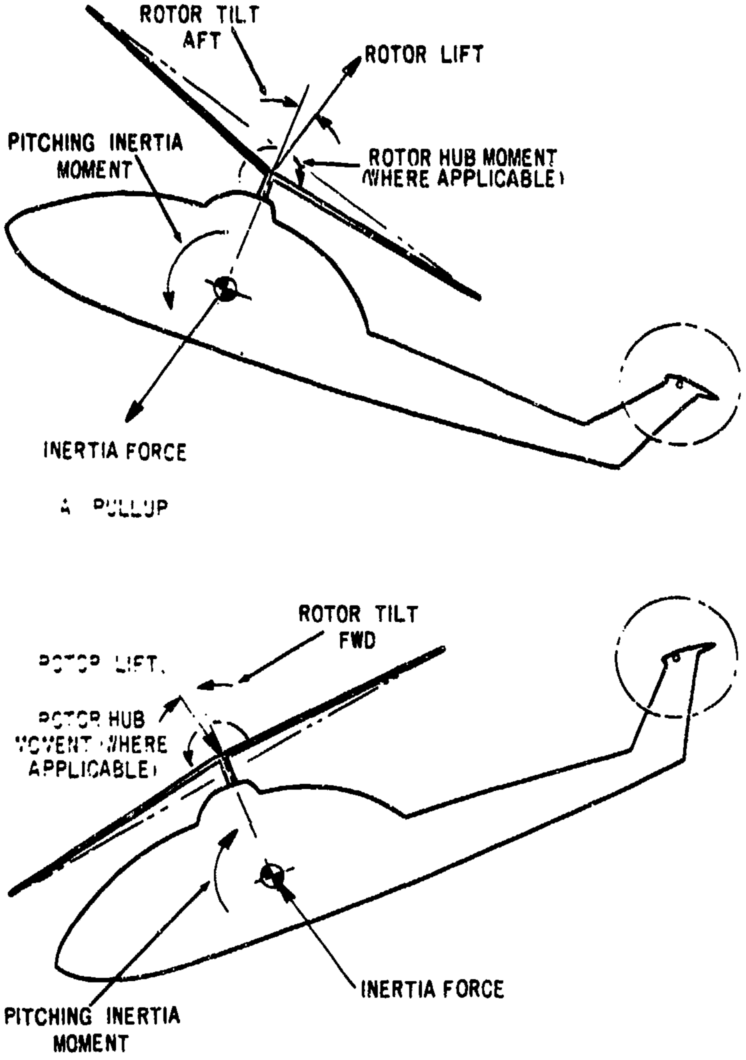


Fig. 4-60. Typical Transient Maneuvers

Fig. 4-61 shows the relationship of load factor n_z to pitching velocity $\dot{\theta}$ and airspeed V for quasi-steady-state conditions, i.e., a pullup into a climb with no reduction in forward speed. The relationship between bank angle ϕ and load factor n_z for turns at constant altitude, as shown in Fig. 4-59(A), is presented in Fig. 4-62.

4-10.1.2.2 Gust Loads

Gust load criteria are given in par. 4-4.3. The critical effects upon the fuselage structure normally result from the gust loads applied to the horizontal or vertical surfaces. Generally, the gust loads may be expected to be critical only for the aft portions of the fuselage structure, such as the tail boom. Depending upon the configuration, inertia loads due to pitching or yawing accelerations resulting from gust loads can be quite high in these areas. Gust loadings on auxiliary lifting surfaces, if provided, also must be investigated for simultaneous loads imposed upon the fuselage.

4-10.1.2.3 Landing Loads

Landing loads (pars. 4-5.1 and 4-5.2) should be applied to the fuselage structure wherever the specified

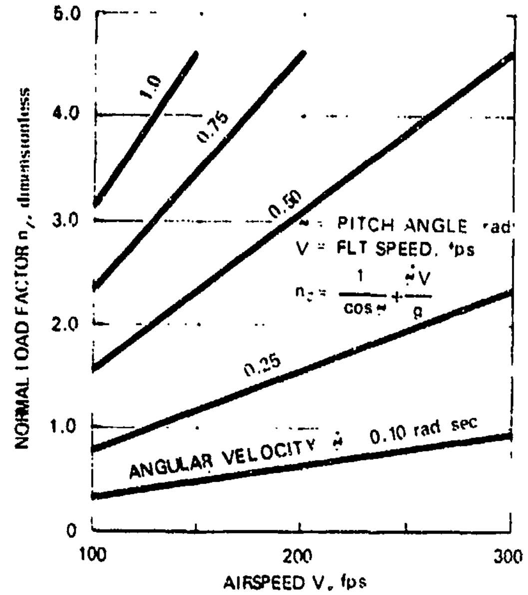


Fig. 4-61. Load Factor vs Airspeed (For Various Pitch Velocities

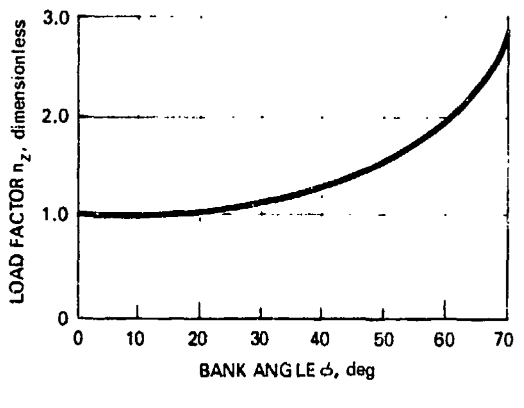


Fig. 4-62. Load Factor vs Bank Angle

conditions are likely to be the critical loadings. Permanent deformation or other damage detrimental to the fuselage prime structure *shall* not occur prior to obvious damage to the landing gear installation.

In applying the design landing loads to the fuselage, it is important to consider the simultaneously acting main rotor, tail rotor, and driving torque loads to the extent that the resulting combination may be critical. From a preliminary design standpoint, the landing loads usually are critical for at least that portion of the fuselage prime structure in the vicinity of the various landing gear attachments. In addition, landing conditions often can produce the most severe pitching and rolling accelerations that will be applied to the fuselage. Asymmetrical landing conditions may be critical for the main central portion of the fuselage because of the wracking, or torsional, loading. Landing impact loads and attendant vehicle accelerations normally are obtained for preliminary design purposes from computer programs developed for this purpose (par. 4-10.3). The values obtained later are confirmed by actual landing gear drop tests.

It is important for the designer to keep in mind that flight safety, flight design loads, and strength considerations are of higher priority for structural design purposes than are landing or crash loads to which the helicopter may be exposed only rarely. Thus, any weight penalty due to providing structure adequate for the landing and crash loads must be considered as a trade-off against such requirements as performance, fuel, payload, armor, or armament. It is essential that, during the preliminary design phase, all possible means be investigated to avoid imposing unnecessary weight penalties due to nonflight loads. For example, additional landing gear stroke or the provision of yielding deformation may be utilized to attenuate landing loads.

4-10.1.2.4 Crash Loads

Crash load requirements are outlined in par. 4-5.3. The provision of crashworthiness in the fuselage structure is not only a design load problem but, even more important, a load-limiting problem. In general, information from Ref. 11 should be used for fuselage preliminary design from the crash load viewpoint. Ref. 17 also presents design objectives and methods for accomplishing and evaluating crashworthiness provisions, along with a discussion of the weight penalty versus safety improvement trade-off and guidelines for trading off load factor versus deformation distance under load during survivable crash impact.

Generally speaking, vertical crash accelerations at the floor line of the fuselage cannot be attenuated to the full extent required for minimization of injury to the vehicle occupants. This is because of the added crushing or yielding distance under load that would have to be provided below the floor level in the fuselage. However, from a design viewpoint there are several possibilities: (1) providing additional deformation under load within the seats, (2) suspending the seats to the airframe in such a manner that the entire seat displaces vertically the desired distance under the desired deceleration loads, and (3) provision of equivalent deformation under load by the combined seat/substructure.

Under probable helicopter crash impact conditions, the provision of energy absorption features in the horizontal direction is of relatively less value, particularly where horizontal velocity components may be high. Frictional plowing effects of the terrain inherently give excellent energy absorption provided the fuselage structure is designed to fail uniformly and not become a "scoop". Energy absorption provisions within the fuselage are desirable for "solid wall" impacts and for sideward impacts. Rollover protection of the occupied cabin/cockpit area also should be provided (par. 4-5.3).

The criteria for crew and passenger restraint systems are given in par. 4-5.3. Wherever possible, without significant weight penalty, additional deceleration capability in the forward direction is desired. However, the provision of adequate restraint capability in the forward and sideward directions is effective only if the occupant restraint system (harness) completely controls the horizontal motions of the occupant (and prevents his sliding out from under the harness) while he is subjected to the anticipated horizontal forces, regardless of vertical deformations of his seat and/or its support while attenuating anticipated simultaneous vertical forces. Thus, shoulder and leg/pelvic restraint adequate for maximum occupant survivability under severe crash impacts should be provided.

It is highly desirable that the control pedals not fail under the maximum loads that may be applied by the crew's feet during crash impact. Local items of mass in the cockpit/cabin area should be supported to the anticipated load factor corresponding to their particular location. Instrument clusters or control consoles that may be a secondary impact hazard to the occupants should be designed either to break away completely in a noncritical direction or to provide additional local energy absorption in case they are struck by a crew member. Delethalizing of the control stick also is desirable.

All provision for deformation under load, including the occupant restraint harness and seat structure, should be of a damped or plastic nature if possible, rather than of an elastic or spring nature. Undamped elastic deformation can result in dynamic overshoot to even higher peak loads than otherwise would occur with a well-damped elastic or plastically deforming rigid restraint system (Ref. 17).

Finally, it is desirable that sufficient support strength be provided to prevent the rotor(s) and transmission(s) from separating from the fuselage during a survivable impact. Where the rotor(s)/fuselage configuration is such that the blades could hit the occupied cockpit/cabin area even if the rotor did not tear loose from its supports, additional cabin protection is required so that the occupied areas are not violated. Frangible rotor blade tip design also should be considered in order to minimize helicopter damage from blade strike.

There is a vehicle "size effect" upon crash survivability because more deformation distance is inherently available in larger vehicles for attenuation of the deceleration g's, provided that equivalent use is made of the available deformation distance within the lower fuselage. Thus, it is possible to obtain significantly lower crash impact load factors for larger helicopters.

4-10.1.2.5 Other Loadings

Other fuselage loadings include:

1. Cargo and passenger floor loadings. In general, crew and passenger seat support loadings will be critical for the crash load factor requirements of the model specification. Design floor loadings are given in par. 4-8.3. Higher values of floor loading may be specified in the detail model specification, and lateral and horizontal values also are normally specified. Where wheeled or tracked vehicles are to be carried, their special localized loadings also must be considered. Provision may be required in the flooring design for unusually dense or localized types of cargo.

2. Drive system reaction loads. The fuselage structure supporting the drive system components should not fail at loads below those corresponding to the failure loads for the drive system itself. Thus, the fuselage support structure should be designed to remain safe for continued flight and landing after any transient overload within the failure limits of the drive system. Usually, these system failure limits will be determined by the torsional strengths of the drive shafts or their interconnections.

3. Control load reactions. It is essential that the most severe combinations of the design limit control system loads be used as the design limit fuselage support structure loads. Combined or opposing loads should be applied in the case of dual controls, and loads from hydraulic or other power actuators also must be considered (par. 4-9.4). The critical control system reaction loads should be applied to the fuselage structure in combination with the other design limit loads for the fuselage.

4. Fuel tank support structure. Inasmuch as post-crash fire remains one of the major helicopter accident hazards, it is essential that the fuselage structure in the vicinity of the fuel tanks be designed not only to support the fuel tanks up to the specified rupture limits, but also to avoid types of failure that would be likely to tear or puncture the tank under severe impact conditions. Consideration also should be given to location of

fuel cells so as to allow them to break away from the aircraft at specified loadings.

5. Induced or secondary loads. Structural deflections, resonant amplifications, and stiffness requirements are possible sources of secondary, or induced, loads in primary fuselage structure. These loads must be identified and evaluated, and appropriate values must be added to other critical loads.

4-10.1.2.6 Load Paths

It is essential that at least one load path be provided to resist each critical loading that may be imposed based upon any of the specified loadings. For static stability there must be at least one independent constraint for each independent loading degree of freedom. It is desirable that more than one load path be provided for the critical loadings so as to produce the degree of structural redundancy needed for a fail-safe structure and to resist catastrophic failures from battle damage, fatigue damage, or other undetected failure.

As a rule, the more direct the load path between the externally applied loads and the internal resisting inertial forces, the lighter will be the fuselage structure for a given overall design strength level. It also can be shown by elastic-energy structural weight analogy (Ref. 58) that the structure with the shortest load paths generally will be the stiffest structure.

The greater the extent to which the major supported loads (fuel, payload, and engines) are located closely below and around the main lifting rotor(s), the shorter the prime load paths can be. The fuselage load paths that are determined during the preliminary design phase largely will determine the weight of the primary fuselage structure.

4-10.1.3 Determination of Loads

Essentially there are four steps in determining loads for the fuselage:

1. Weight distribution (1g)
2. Unit shear, moment, and torsion distributions
3. Specific maneuver and landing condition load distributions (shear, moment, and torsion)
4. Critical condition selection, and superposition of system loads.

The applicability of the unit method to preliminary analysis may be questioned because detailed information normally is not available. However, satisfactory preliminary distributions are obtained by using preliminary weights and configurations in spite of a limited number of inputs. It is recommended that the data for fuselage load determination be calculated and plotted by computer. As detail weights and configurations are determined, the computer inputs are changed easily and the results updated promptly.

4-10.1.3.1 Weight Distribution

A graphic presentation of weight versus station is prepared for the extreme CG conditions from the weight statements. Items that have specific attachments can be distributed as concentrated loads. Items such as fuel, cargo, and structural weight must be distributed as running loads. An example of weight distribution is presented in Fig. 4-63. The illustration is for a tail boom, but the same procedure is applicable to the fuselage. It is necessary to investigate all loadings, including the minimum flying weight condition, because critical equipment loadings may occur during accelerated flight maneuvers under such conditions.

4-10.1.3.2 Unit Shear, Moment, and Torsion Distribution

External loads from the rotors and landing gear cause each item supported by the fuselage to undergo an acceleration during maneuvers or landings. In developing the unit shear, moment, and torsion distributions, assumptions are made as to how each load source will load the fuselage. For a unit case (flight maneuvers), a unit load factor to the CG is applied separately along each axis. Unit angular accelerations also are

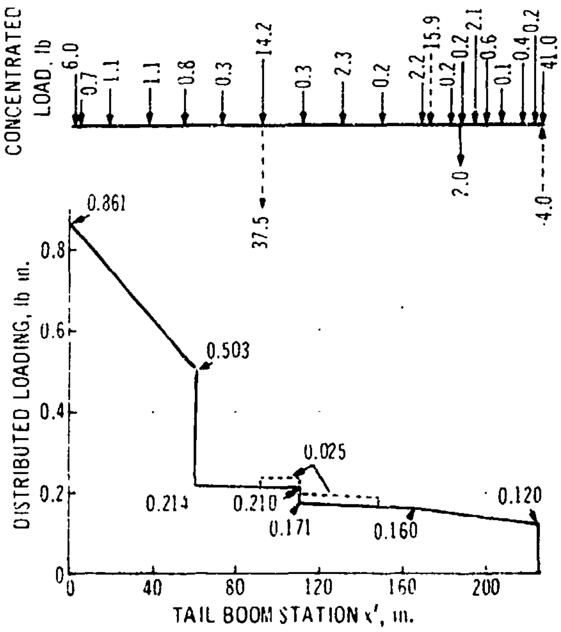


Fig. 4-63. Typical Tail Boom Weight Distribution

applied separately about each axis. The result will be a series of curves depicting the reaction by inertia loads of unit linear and angular accelerations applied to the helicopter CG. Because not all the items are located on the helicopter centerline, a load factor along a given axis also may produce moments about the other axes.

Positive sign conventions should be defined, as shown on Fig. 4-1, for a single main rotor/tail rotor configuration.

Curves then are computed for each unit case by summing the shears, moments, and torques beginning at fuselage station zero and progressing toward the aft end. The vertical shear V_z due to a unit vertical load factor ($n_z = 1$) is determined by integrating the combined static load applicable for the chosen loading condition and CG location. The curve starts with zero shear at the most forward part of the helicopter fuselage and increases to a maximum at the aft end. It is evident that $V_x = V_y = V_z$ where $n_x = 1$ for V_x, $n_y = 1$ for V_y, and $n_z = 1$ for V_z.

The unit moment curve M_y ($n_z = 1$) is obtained by integrating the shear curve V_z ($n_z = 1$) starting at the most forward part of the fuselage. It also is evident that, because the unit shears are equal, $M_z = M_y$ where $n_y = 1$ for M_z, and $n_z = 1$ for M_y.

The M_x ($n_z = 1$) curve is obtained by a summation of the individual moments starting at the front of the fuselage. Applying the same procedure to a load factor of $n_x = 1$, it is seen that $M_z = -M_x$ where $n_x = 1$ for M_z, and $n_z = 1$ for M_x. These two unit loads may be small and, if so determined, may be neglected in determining the final fuselage loading.

Now, considering a unit loading of $n_x = 1$, many of the mass items might be displaced vertically from the assumed structural shear center, giving rise to a unit loading M_y where $n_x = 1$. A similar analysis of the loads due to a side load results in a unit loading M_z where $n_y = 1$.

Unit angular accelerations α equal to 1.0 rad/sec^2 produce inertia load factors at any point on the fuselage that are given by:

$$n_z = \frac{1.0}{386}\left[(y-\bar{y})\alpha_x - (x-\bar{x})\alpha_y\right] \qquad (4\text{-}42)$$

$$n_x = \frac{1.0}{386}\left[(z-\bar{z})\alpha_y - (y-\bar{y})\alpha_z\right] \qquad (4\text{-}43)$$

$$n_y = \frac{1.0}{386}\left[(x-\bar{x})\alpha_z - (z-\bar{z})\alpha_x\right] \qquad (4\text{-}44)$$

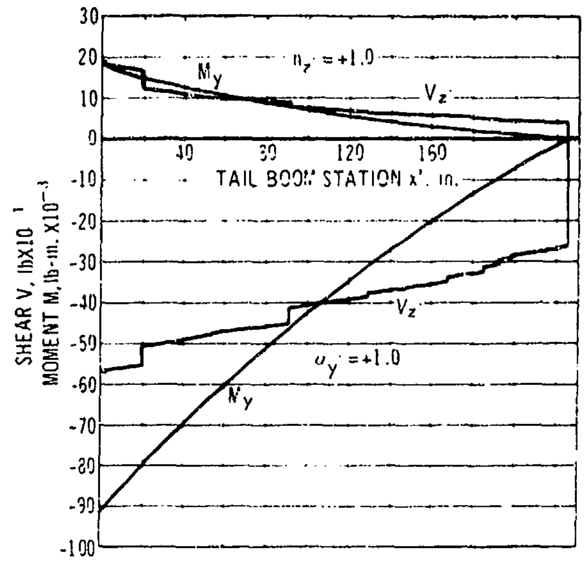


Fig. 4-64. Unit Loading for Tail Boom

where x, y and z represent the coordinates of the point on the fuselage; $\bar{x}$, $\bar{y}$, and $\bar{z}$ represent the coordinates of the helicopter CG; and all coordinates are measured in inches. Those load factors that depend upon location on the fuselage are treated in the manner described previously for the constant unit load factors, resulting in more shear and moment curves.

In addition to inertia load factors resulting from the angular accelerations, certain concentrated masses are large enough that their local moments of inertia are of importance. Examples of these items are landing gear, engine, rigid rotors, structure, and armament. It should be remembered that fuel and oil have only small local moments of inertia, depending upon viscosity and tank shape, and do not act as solids when an angular acceleration is applied. The local inertia moments M of these individual masses due to angular acceleration α are given by $M_i = -I\alpha_i$ with the appropriate axis indicated by the subscript i. These moments are transferred to the fuselage in the manner described previously for the unit load factors. Combining these moments with the distributed loadings due to angular acceleration gives the final unit shear and moment curves for unit angular accelerations. Fig. 4-64 provides an example of a unit loading curve based on the example weight distribution of Fig. 4-63.

4-10.1.3.3 Specific Maneuver and Landing Condition Load Curves

Although many operational loading conditions are specified, only a few are important enough to be considered during preliminary design. In certain instances, loading conditions can be combined conservatively for design purposes.

A computer program is used to determine the actual limit loads due to the static loading; i.e., the shears and moments due to the unit load factors and angular accelerations are multiplied by the design load factors and angular accelerations.

The external forces and moments must balance the internal inertia loads and moments:

$$
\begin{aligned}
M_x &= I_x\alpha_x - I_{xz}\alpha_z - I_{xy}\alpha_y \\
M_y &= I_y\alpha_y - I_{yx}\alpha_x - I_{zx}\alpha_z \\
M_z &= I_z\alpha_z - I_{zy}\alpha_y - I_{xy}\alpha_x
\end{aligned}
\tag{4-45}
$$

where

M = moment, lb-in.
I_i = moment of inertia, slug-in.^{2}
I_{ij} = product of inertia, slug-in.^{2}

Subscripts indicate axes of interest.

Loads contributing to the moments are main and tail rotor thrust, drag, torque and side forces, airloads on aerodynamic surfaces, landing gear, armament recoil, etc.

One more set of equations is required in order to determine the actual loading curves For flight conditions,

$$
\begin{aligned}
n_x &= \frac{-D}{W_g} \\
n_y &= \frac{H+T_t}{W_g} \\
n_z &= \frac{-L}{W_g}
\end{aligned}
\tag{4-46}
$$

where

L = lift, lb
D = drag, lb
H = side force, lb
T = main rotor thrust, lb
T_t = tail rotor thrust, lb
W_g = helicopter gross weight, lb

For landing conditions, the critical condition is a power-off landing (the main rotor torque is zero). Sink rate is not high during power-on approaches and, therefore, the power-off case is used for design. For landing,

$$T = \tfrac{2}{3} W_g \tag{4-47}$$

Therefore, summing vertical forces

$$n_z W_g = n_j W_g + \tfrac{2}{3} W_g \tag{4-48}$$

or

$$n_z = n_j + \tfrac{2}{3} \tag{4-49}$$

where n_j is determined by drop test (par. 4-5). The special dynamic landing, where a forward-velocity landing is made with the ground reaction not directed through the CG and the resulting pitching moment is reacted by inertia, must be analyzed to determine the angular accelerations.

The limit internal shear loads are combined with the applicable external loads to create shear V and moment M curves, as shown in the example curves in Fig. 4-65.

The combined shear curve then will be described at any station *sta* as

$$V_{sta} = \sum_{0}^{sta} V_{int} + \sum_{0}^{sta} F_{ext} \quad \text{, lb} \tag{4-50}$$

Th

$$
\begin{aligned}
M_{sta} = &\sum_{0}^{sta} V_{int}(sta - x) \\
&+ \sum_{0}^{sta} F_{ext}(sta - x) \quad \text{, lb-in.}
\end{aligned}
\tag{4-51}
$$

where

x = station at which shear or external force is applied, in.

4-10.1.3.4 Critical Condition Selection and Superposition of System Loads

The design flight, landing, crash, and other loading conditions applicable to helicopters are presented in

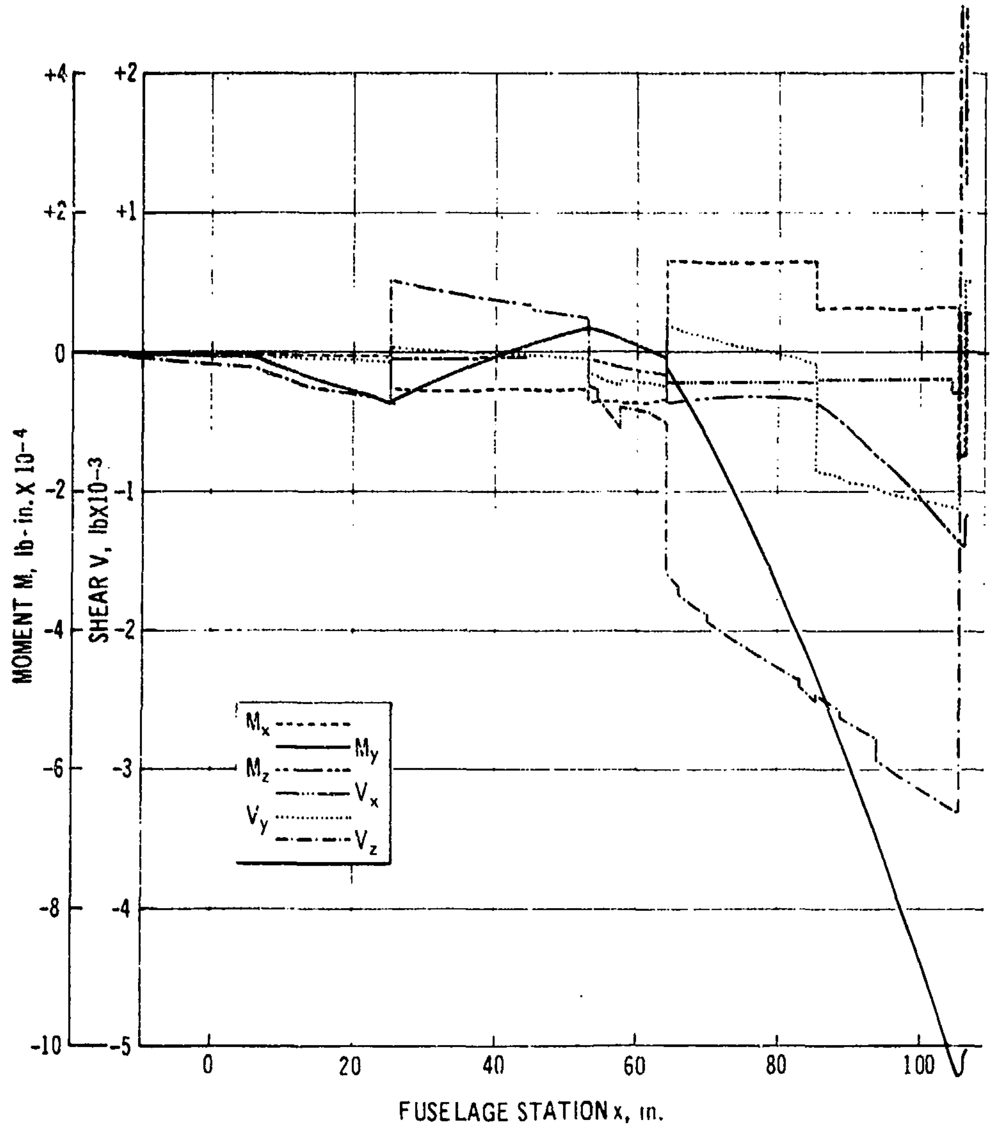


Fig. 4-65. Typical Fuselage Shear and Moment Curves

pars. 4-4 through 4-8. From these conditions, critical loading conditions for specific portions of the fuselage structure can be selected. The CG load factors and external loads can be derived for these conditions, and the loading curves developed as explained in par. 4-10.1.3.3.

Identification of critical conditions is difficult, being dependent both upon mission applications and upon the fuselage configurations being considered, e.g., tandem-rotor and single-rotor with tail rotor.

A tandem arrangement of two main rotors, with the useful load supported between them, allows a larger CG range and, therefore, peak bending moments must be considered over the center section of the fuselage. Pullups and level landings produce the maximum vertical bending moments, which are critical for the bending stringers.

The maximum yaw condition, or unsymmetrical landing, will be critical for the torsion-carrying structure (skin). The maximum yaw condition can be combined with $n_z = 2$ to obtain preliminary design loads for the shear and torsion-carrying skin panels. Frames and bulkheads are located where controls, gearboxes, landing gear, equipment, and useful load items are attached or supported. This distributes concentrated system loads into the skin and, together with secondary frames, reduces skin panel sizes. Rotor pylons carry both the tension due to lift and the bending and shear due to drag and side forces down to the fuselage proper. Crash loads are applied to large items of mass that, if torn loose, would endanger the occupants.

For single-rotor helicopters, pullup and level landing moments determine the design of the forward fuselage downbending stringers, frames, and bulkheads located under the useful load. The maximum yaw condition contributes critical side bending and torsion to the tail boom. The dynamic landing (forward velocity or run-on) presents the maximum vertical bending for the tail boom. A nosedown landing will result in the highest shear and upbending in the forward fuselage. The maximum roll condition or unsymmetrical landing gives maximum pylon side bending.

4-10.1.4 Fail-safe Aspects

Fail-safe or damage tolerant design is achieved by providing multiple active load paths, inactive backup structure, or a design that inhibits propagation of failures. Important to all fail-safe methods is the inspection feature; the design must make it highly improbable that a failure, once initiated, will go undetected. Fuselage components such as rotor pylons, tail boom attachments, and transmission attachments all are candidates for fail-safe design.

In the multiple active load path design, the critical loads should be applied with one path failed, and adequate stiffness also must be incorporated to meet all pertinent dynamic requirements with one load path failed. It has been popular to design structure so that after one load path has failed, the remaining structure can support a specified percentage of limit load, but this is not necessarily adequate. Instead, the structure must continue to take the applied fatigue loads until the failure is detected and repaired. Therefore, anticipated fatigue loads should be calculated for the structure and adequate strength provided so that no further failure will occur prior to the next regular inspection.

Often a failure of one load path will change the stiffness of the structure so that a change in the dynamic environment will signal the pilot that damage has occurred. A higher level of vibration or noise may result even though system stiffness still is adequate.

A second inactive load path implies complete failure of one load path with the secondary member taking all the load following failure of the primary structure. The same criteria obviously apply regarding ability to continue operation until a repair is made.

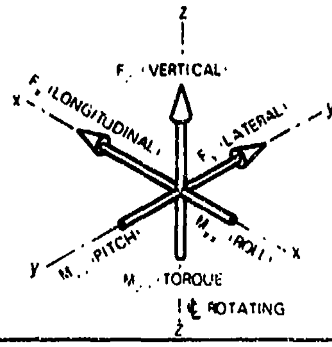
4-10.1.5 Fatigue Considerations

Fatigue considerations usually are less significant in the preliminary design of the fuselage than for dynamic components such as rotor or drive systems. Often, when the vibratory loads transmitted to the fuselage are reduced to the levels necessary for pilot comfort, the fatigue stresses are at a level where good design, materials, and fabrication quality will result in an adequate fatigue life. Also, except at the rotor pylon areas and major attachment areas of the primary structure, the normal fuselage structure is sufficiently redundant to provide a reasonable amount of fail-safety should a fatigue crack develop. Steady stress levels also are lower in the fuselage structure than in rotor or drive system components because most of the structure is critical for elastic stability rather than for material strength. However, local stresses in the vicinity of major disassembly attachments and main rotor pylon structure—as well as in structure immediately adjacent to tail rotor, tail surfaces, and tail rotor or main gearbox attachments—may be more critical for material strength rather than for elastic stability and, thus, require more detail consideration of probable fatigue stress levels.

Fatigue load considerations for the fuselage (Ref. 59) involve oscillatory loads from main and tail rotors, including effects of maneuvers and centrifugal unbalance (especially from the tail rotor); aerodynamic excitation of surfaces in the rotor wake; ground to air cycles; and weapon firing. Practical considerations should allow for amounts of main and tail rotor unbalance and for out-of-track conditions in service considerably in excess of normal drawing and manufacturing limits. Fuselage fatigue loads are affected greatly by increased vibratory load responses or amplification due to airframe resonances lying at, or close to, known rotor transmitted frequencies (Fig. 4-66); such as 1-per-rev and N-per-rev of both the main and tail rotor (N equals the number of blades per rotor).

Occasionally, it may be impracticable to avoid operating at near-resonant conditions for all gross weight conditions. If such is the case, some damping or detuning system must be employed in the cockpit or cabin area to achieve the vibration levels required for crew comfort. In this case, the actual vibratory loads being transmitted into the fuselage structure from the rotor(s) may be considerably higher than the overall internal fuselage vibratory response would indicate and still may require extensive investigation.

In general, the vibratory loads transmitted from the rotor are reduced significantly as the number of blades in the rotor is increased, assuming equivalent avoid-



LOAD VECTOR	FATIGUE LOADS TO STRUCTURE			
	ROTATING		NONROTATING	
	FREQUENCY	COMBINING FACTOR	FREQUENCY	COMBINING FACTOR
(1) F_z, M_{zz}	iNP	$\sum^{i} 100$ AT iNP	iNP	$\sum^{i} 100$ AT iNP
(2) F_x, F_y, M_{xx}, M_{yy}	$(iN-1)P$ AND $(iN+1)P$	$\sum^{i} 100$ AT $(iN-1)P$ AND $\sum^{i} 100$ AT $(iN+1)P$	iNP	$\sum^{i} 50$ AT $(iN-1)P$ + 50 AT $(iN+1)P$

WHERE N NUMBER OF BLADES
P SPEED OF ROTATION
i AN INTEGER

(1) FOR THESE VECTORS ALL HARMONICS OTHER THAN iNP ARE REACTIONLESS (i.e. CANCELLING) IN BOTH ROTATING AND NONROTATING STRUCTURE

(2) FOR THESE VECTORS ALL HARMONICS OTHER THAN $iNP \pm 1$ ARE REACTIONLESS IN BOTH ROTATING AND NONROTATING STRUCTURE. ALSO, IN THE TRANSMISSION FROM ROTATING TO NONROTATING STRUCTURE, EVEN 50% OF BOTH $iNP \pm 1$ HARMONICS ARE LOST (CANCELLED)

Fig. 4-66. Rotor Vibratory Loads Transmitted to Structure

ance of blade resonant frequencies. Also, a greater number of blades per rotor increases the spread between 1-per-rev and *N*-per-rev, giving the designer greater leeway to avoid resonant response to the rotor-induced loads.

Improvement of the fatigue quality of the fuselage is possible without weight penalty through use of adhesive bonding, shot peening, or coining in high-stress concentration areas of the structure; specification of adequate materials and parts inspections; adequate clamping or preloading; or use of cast or machined parts with fillets adequate to reduce or avoid stress concentrations. Fiber-reinforced plastics, which may be used for certain portions of the fuselage structure, also tend to exhibit excellent fatigue strengths in relation to their static strength levels.

4-10.1.6 Internal Loads

Current helicopter requirements generally are best satisfied with fully enclosed, well-streamlined monocoque or semimonocoque shell construction, with the skin and/or stiffening members carrying the major internal loads. Structures of this type inherently are redundant in structural function. The determination of the internal load distribution in redundant or discontinuous monocoque or semimonocoque structures requires consideration of the way the loads are transmitted into and out of a structural member. This is true especially at a discontinuity or at an attachment fitting, and in the case of division of the internal loads among the redundant members.

By St. Venant's principle, conventional, simple, standard beam, column, tension, shear, and torque box formulas of structural analysis textbooks should be applicable except at local areas of significant change in the structural section or at points of load application. Where discontinuities exist in the structure, additional analysis incorporating elasticity considerations such as "shear lag" (Ref. 60) also should be used. Where the load is distributed among structural members, a redundancy of the internal loads is indicated, and special elasticity methods such as "least work" or elastic energy should be used.

4-10.1.7 Preliminary Sizing

Once the basic external and inertial loads for the fuselage have been determined, and the primary load paths have been established tentatively by preliminary design layout drawings, the internal load distribution can be determined for the initial member sizes, skin gages, etc. Emphasis should be placed upon optimizing the structural design, i.e., maximizing the strength versus weight and cost. This optimization also must consider stiffness in order to minimize vibratory response of the fuselage.

Ref. 57 presents methods for optimum structural design, using a "structural index" to scale similar structures of different sizes and loading intensities and compare them on the basis of equivalent optimum allowable stress and primary structural weight. The elastic energy structural weight analogy of Ref. 58 also allows prediction of theoretically "ideal" (minimum) structural weights based upon the design loading and stiffness requirements for the structure, the strength-to-density ratio of the material, the design geometry, and the type of stress. This weight prediction technique, although limited in practical application, is useful particularly for preliminary design as a potential medium for weight efficiency comparison prior to finalization of the hardware details. Ref. 61 details the use of elastic-energy principles for theoretical strength/weight optimization of a complex structure; the lightest, i.e., most optimum, structure results when the strain energy densities of all elements of the structure are equal.

Computer programs, such as that of Ref. 56, can be used to analyze complex redundant structures such as a complete semimonocoque fuselage. These programs take into account a multiplicity of external load conditions, structural redundancy, resistance to inertial loads, and internal loads such as those imposed by drive systems or controls. At the present time, structural sizing input iterations to the computer program are made by the designer and structural analyst. In the future, complete structural synthesis may be performed by the computer.

4-10.1.8 Substantiation

Substantiation of the fuselage structure for preliminary design purposes should consist of:

1. A thorough basic load analysis
2. Selection of the loading conditions most likely to be critical for the primary structure
3. A comprehensive internal load analysis for the selected critical external loading conditions.

These steps are followed by a member and skin sizing type of preliminary stress analysis. Experience with similar fuselage designs may be drawn upon for substantiation where such data are available. Otherwise, sufficient analytical investigation must be accomplished to assure that the critical and allowable stresses in all of the major members have been obtained. The substantiation may draw upon information such as published test data and preliminary specimen test data. Any unusual or unconventional structural features always should be investigated in considerable detail to reduce later program risk. Margins of safety at all critical sections *shall* be tabulated for review and evaluation.

4-10.2 WING AND EMPENNAGE SUBSTANTIATION

A wing installed on a helicopter is subject to a velocity range from low-speed rearward and sideward flight to maximum forward airspeed. The wing may contain fuel tanks, external store mounts, and engine and landing gear mounts. The helicopter wing design will differ from the conventional fixed-wing arrangement in that different angles of incidence will appear on the left-hand and right-hand wings because of the asymmetrical induced velocities of the rotor. Ailerons and trim tabs may be necessary for some helicopters with articulated rotors, but may not be required on rigid-rotor designs.

For the general treatment of wing loads, the best method is to build up the total loading by superimposing the applicable individual loading distributions as discussed in the fuselage loads section. For convenience, all such distributions should be given with respect to a common load reference-axis system.

In general, the horizontal stabilizer may be treated in a manner similar to the wing. Its functions basically are to stabilize the helicopter about the y-axis and to provide aerodynamic damping of the pitching motion of the vehicle. On a true helicopter, the horizontal stabilizer area is usually rather small, acting merely as a weather vane. On a compound helicopter, however, its function becomes more like that of the stabilizer on a fixed-wing aircraft, although no elevator is present. This is because at high velocity the rotor only contributes a stabilizing moment correction to the equilibrium, while the stabilizer contributes its lift and damping forces. Therefore, the contribution of the roll rate and roll accelerations dependent distribution is more pronounced.

If a propeller and tail rotor are in the immediate proximity of the stabilizer, their inflow velocities must be added to the stabilizer loading. Also, a distribution of rotor downwash must be considered, dependent upon rotor lift and forward velocity. The effects of rotor downwash and propeller and tail rotor inflow on the stabilizer load distribution must be investigated to establish design loads. For gust and maneuver response, it should be noted that the local inertial load factors are quite different from those at the aircraft CG because of the rotational accelerations.

If a pusher propeller is installed, then the horizontal stabilizer may support the tail rotor. The eccentric placement of the tail rotor mass can affect the dynamic response characteristics of the entire vehicle.

The structural analysis of the vertical stabilizer is similar to that required for the wing and horizontal stabilizer. The vertical stabilizer damps and stabilizes the motion about the z-axis. On most conventional helicopters it supports the tail rotor and partially blanks off the tail rotor inflow. On some configurations, a tail landing gear may be mounted on the vertical stabilizer.

Any empennage loading that constitutes a torsional excitation should have a driving frequency remote from the fuselage torsional natural frequency. Because fluctuation of airloads due to rotor velocity will be governed by the rotor frequency and its harmonics, these frequencies must not be coincident with any of the fuselage or lifting surface natural frequencies. Because natural frequencies are functions of both total mass and mass distribution, it is necessary to investigate various weight configurations.

4-10.2.1 Basic Considerations

For any loading condition, the helicopter first must be in steady-state or dynamic equilibrium. Each particular condition obtained from the mission analysis or the design envelope, therefore, will be related to a performance analysis or landing load analysis of the total vehicle. For each particular wing or empennage loading condition, the equations describing the equilibrium state must include all applicable inertial, aerodynamic, propulsion, and landing gear force and moment components.

The inertial components are dependent upon the weight distribution of the configuration pertinent to the loading case under consideration (for a given structural stiffness). The aerodynamic components are dependent upon pertinent velocity, altitude, temperature, and attitude (for given vehicle geometry). The propulsion components follow from the performance analysis, and the landing gear loads are determined from forward velocity, sink speed, total effective lift, runway roughness, runway slope, and gross weight—again pertinent to the load case under consideration.

Although single conditions can be hand-calculated easily, computer methods are preferred. Also, the many loading conditions of the fatigue spectrum that must be analyzed make it desirable to conduct all loading analyses on a computer program capable of accepting the input for any loading condition. It is beyond the scope of this handbook to present such a program in its entirety, because many program steps are tailored to the particular vehicle, but the essential steps are:

1. Lay out a geometric grid for the definition of the loading distributions and make sure to represent concentrated loads and moments on points such as jack pads, tank pylons, armament mountings, and landing gear fulcrum. Include allowance for gear extension and brake moments.

2. Lay out a geometric grid for the definition of internal loading (shear, moment) at significant locations. Provide for proper representation of steps in shears and moments by defining one station immediately before the applied load, and one immediately beyond the applied load.

For actual design it is convenient to prepare the geometric data once in matrix form for the entire vehicle, and then to store the data. The third step is to compile the loading input for the individual load cases and arrange the output in a convenient tabulation of shears and moments. These data are grouped as:

1. Applied distributed and concentrated loading grid:
   - a. Weight or mass distribution
     - (1) Empty weight
     - (2) Fuel and payload
   - b. Airload distribution factors
     - (1) Zero lift distribution
     - (2) Lift distribution due to angle of attack (pitch angle), etc.
   - c. Dynamic increment distribution factors
     - (1) Dynamic landing

(2) Dynamic taxi
(3) Gust
d. Unit concentrated loads
(1) Main landing gear left
(2) Main landing gear right, etc., including pads, lugs, etc.

2. Load case data:
a. Weight configuration
b. Inertial load factors
(1) Maneuver load factor
(2) Dynamic landing factor
(3) Dynamic taxi factor, etc.
c. Aerodynamic factors
(1) Pitch, roll, and yaw angles
(2) Pitch, roll, and yaw rates
d. Velocities
(1) Flight speed
(2) Sink speed.

The same method may be followed for the empennage, using its particular geometry and distributions, while the load case data may be applicable for the entire vehicle. In fact, the described procedure was developed to analyze a complete airframe, including the fuselage. Entries such as dynamic landing and taxi distribution factors were based upon a separate modal analysis normalized on a reference input load. This reference load was contained in the program equations, such that the input multipliers for dynamic landing factor 2b(2) and dynamic taxi factor 2b(3), for example, were calculated from the landing gear load inputs.

The analytical methods discussed in this paragraph are essentially the same as those used on conventional fixed-wing aircraft. Dynamic loads, however, must be treated more thoroughly because of the risk of resonance at harmonics of the rotary component frequencies (see Chapter 5). One cannot define an independent set of loadings for the wing and empennage. Instead, they are an integral part of the total vehicle, and will respond in a particular way for a particular loading condition.

The dynamic load treatment for landing and taxi conditions in this paragraph is an example of a possible method of presentation when dynamic inputs are obtained from a separate analog or digital program. However, it would not be too cumbersome to substitute any other presentation preferred by the structural dynamics engineer (see Chapter 5).

4-10.2.2 Sign Convention

Because the left and right wings on a compound helicopter are different, and because the fatigue loading analysis demands treatment of a complete vehicle equilibrium condition, it is necessary for economic use of the computer to define shear and moments with the help of a free-body diagram. Such definition has the added advantage that the actual reactions of the major structural components upon one another are shown on the plotted diagram

A step-by-step development of the equations for loading analysis is not presented. For digital computation, such equations would be developed using FORTRAN notation rather than algebraic symbols because of better flexibility in notation of subscripts.

It is convenient to describe the weight distribution as a basic helicopter distribution, equal for all configurations, plus an incremental payload weight distribution for the individual mission configuration.

Selection of a geometric model grid for mass distribution as shown on Figs. 4-67 and 4-68 permits arrangement of the weight data in matrix form $\{WT\}$ as:

$$\{WT\} = \{W\} + \{WCON\}\{S\} \quad (4\text{-}52)$$

where

$\{W\}$ = matrix of empty weight, lb
$\{WCON\}$ = matrix describing the incremental payload weight distribution, lb
$\{S\}$ = selection matrix for incremental weight, dimensionless.

The inertial force at any grid point is obtained by multiplication by the corresponding load factor.

The distributed inertial forces are determined by these operations. By adding the aerodynamic forces and the externally applied forces, the shear forces at any location on the structure may be calculated by using an applicable geometric matrix.

The moment arms from each force to the location on the structure where the moment is calculated are determined for inertial forces, aerodynamic forces, and externally applied forces, respectively, and moment equations are derived.

The geometric grid models on Figs. 4-67 and 4-68 show, for clarity, only one of the three possible force and moment components at each load station. Monitoring stations as a rule appear in pairs to account properly for steps in the shear and moment curves.

4-10.2.3 Inertial Loads

Static loading conditions are independent of time, and inertial loads are due only to the acceleration due

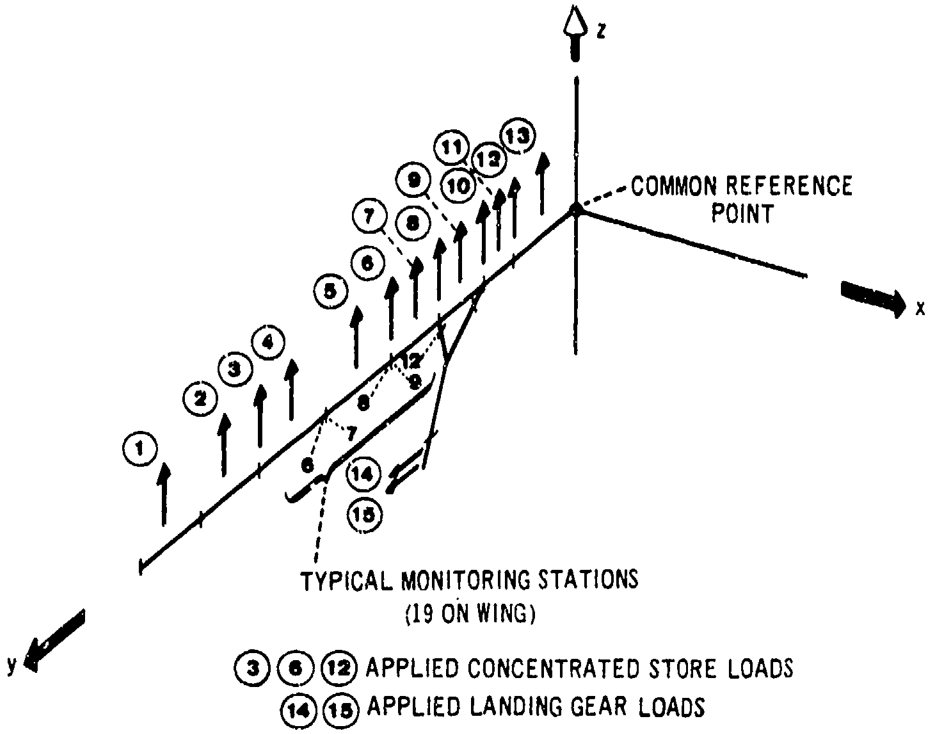


Fig. 4-67. Left Wing Load Grid, Showing Load Stations and Monitoring Stations

to gravity or to the sustained steady maneuvering load factor. Transient maneuvering, takeoff, landing, and gust loads, however, are time-dependent, and the local inertial forces become dependent upon the local acceleration of the structure. This acceleration is simply the second time derivative of the local deflection or, in other words, is dependent upon the mode shape of the dynamic response of the structure to the forcing function (see Chapter 5). Such mode shapes are different for different designs and loading conditions because of their dependence upon mass distribution, structural damping, structural stiffness, and the applied forcing function as they appear in the general equilibrium equations

$$m\ddot{x} + c\dot{x} + kx = F(t)$$
$$I\ddot{\theta} + C\dot{\theta} + K\theta = M(t) \tag{4-53}$$

where

m = mass, lb-sec²/in.
c = damping constant, lb-sec/in.
k = spring constant, lb/in.
I = mass moment of inertia, lb-in.-sec²
C = rotational damping coefficient, lb-sec-in./rad
K = rotational spring constant, in.-lb/rad
x = horizontal displacement, in.
θ = pitch angle, rad
$F(t)$ = time-dependent external force, lb
$M(t)$ = time-dependent external moment, lb-in.

These equations must be specific to the entire system, and with the actual restraint for the airborne vehicle (free-free), and with applicable gear restraint (gear stiffness) entered.

For each dynamic loading condition the accelerations $\ddot{x}$ and $\ddot{\theta}$ can be obtained at any lumped mass station and, by normalizing on a unit input in $F(t)$ and $M(t)$, the results may be tabulated in a matrix. These dynamic loads are especially important on the wing and empennage because of the relative flexibility of these structures with respect to the fuselage.

A survey of taxi and landing runs for different configurations of the helicopter at varying velocities and sinking speeds can be made on an analog computer. From studies on a dynamic structural model, the prominent mode shapes of the dynamic response can be established, and incremental dynamic load factors at the mass reference points can be calculated. An example of the qualitative effects of a dynamic load factor increment upon the shear and bending of the wing is shown in Fig. 4-69.

For a unit gearload F the incremental load factor at the reference point (see Fig. 4-70) is found from:

$$\frac{n_i}{F} = \frac{\lambda_i \lambda_F \gamma}{\sum_0^k W_i (x_i)^2} \tag{4-54}$$

where

λ_i = corresponding mode shape ordinate normalized on $\lambda_k = 1$

λ_F = normalized mode shape ordinate at the force station

γ = dynamic response factor x_d/x_s for a ramp force excitation with zero damping (see Fig. 4-71).

subscript i = station along the span $0 \rightarrow k$ where the weight W_i is concentrated

4-10.2.4 Aerodynamic Load Distribution

The aerodynamic load distribution—derived from the general expressions for aerodynamic forces and moments as functions of dynamic pressure, angle of attack, and appropriate coefficients—also can be expressed in matrix form.

The coordinates of the center of pressure of the force components for this particular distribution may be arranged in matrix form, and a simple integration procedure will give shears and moments at any desired location on the structure.

Similar distributions can be written for zero lift distribution. lift due to roll and yaw velocities and acceler-

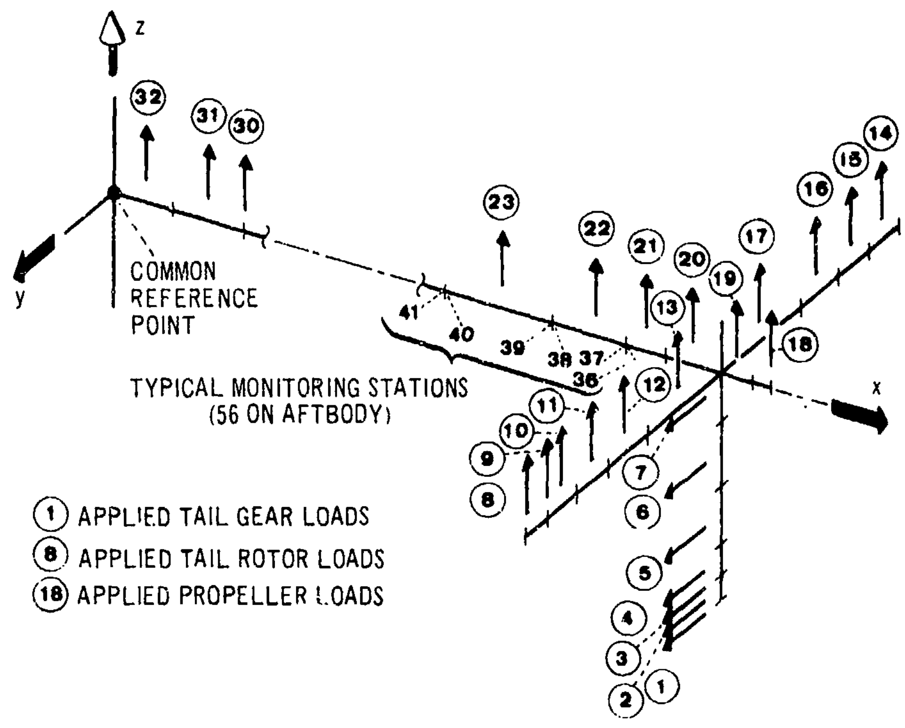


Fig. 4-68. Empennage and Aft Body Load Grid, Showing Load Stations and Monitoring Stations

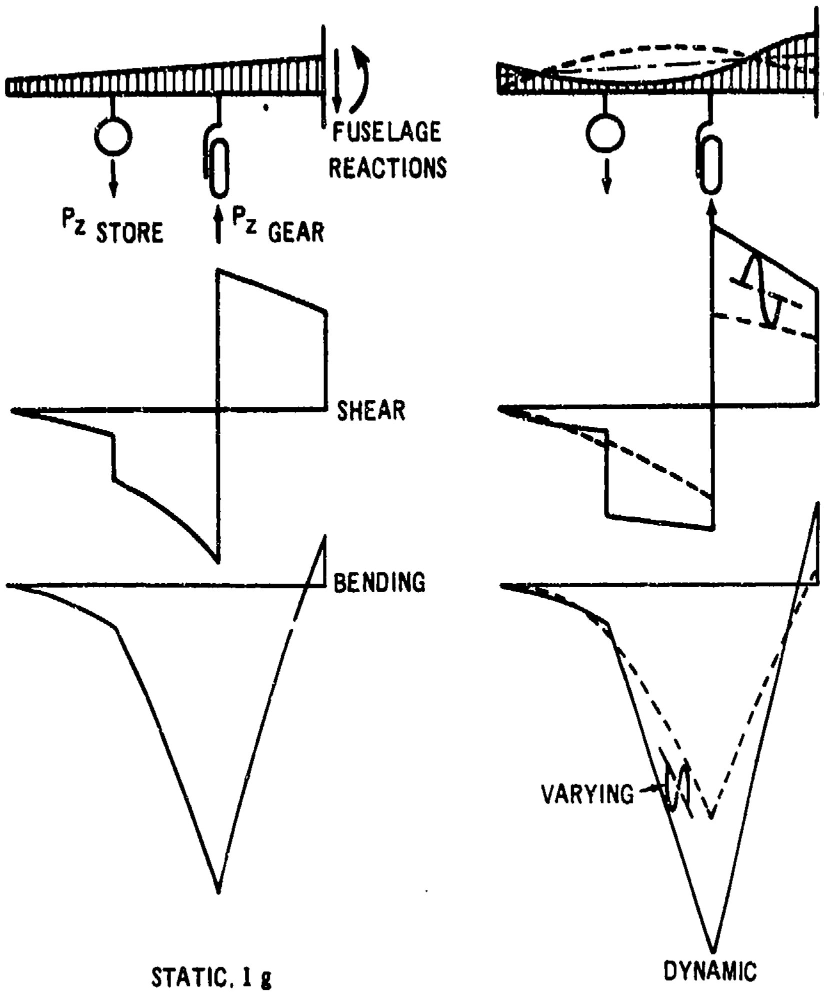


Fig. 4-69. Effects of Dynamic Load Factor Increment

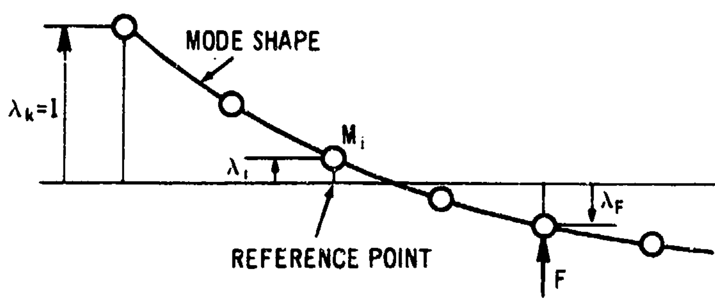


Fig. 4-70. Dynamic Increment Definition

ations, and any other aerodynamic parameters that need to be considered. Fig. 4-72 shows a typical span-wise lift distribution, which can be related to the load-point grid of Figs. 4-67 and 4-68.

The impact upon the structure from blast pressure due to weapon firings is dependent upon the particular characteristics of the weapon system and upon the proximity of such a system to the structure. Therefore, the designer must obtain the relevant data, e.g., that shown on Fig. 4-73, from the supplier of any such ordnance. Also, the duration of this pressure must be stated. As with the aerodynamic distribution, these loads must be entered on the corresponding load station grid points.

4-10.2.5 Gust Loading

Gust intensity usually is given in terms of a vertical gust velocity U_{de}, and its intensity as a probability func-

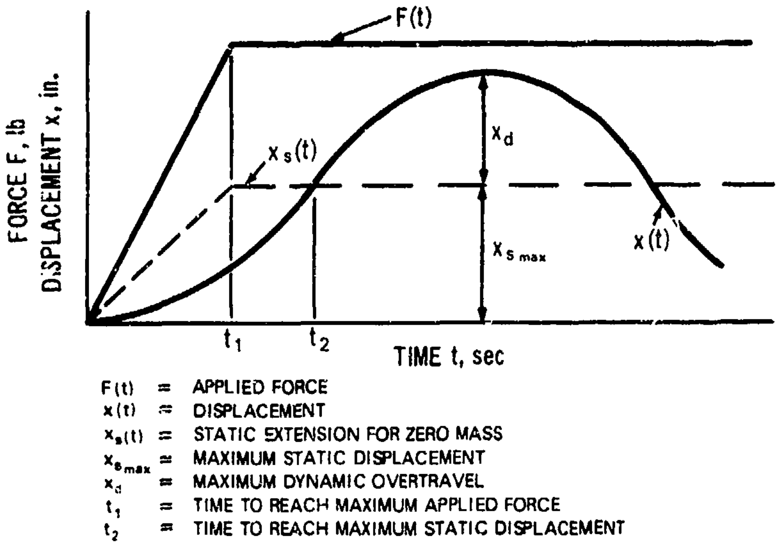


Fig. 4-71. Ramp Force Excitation

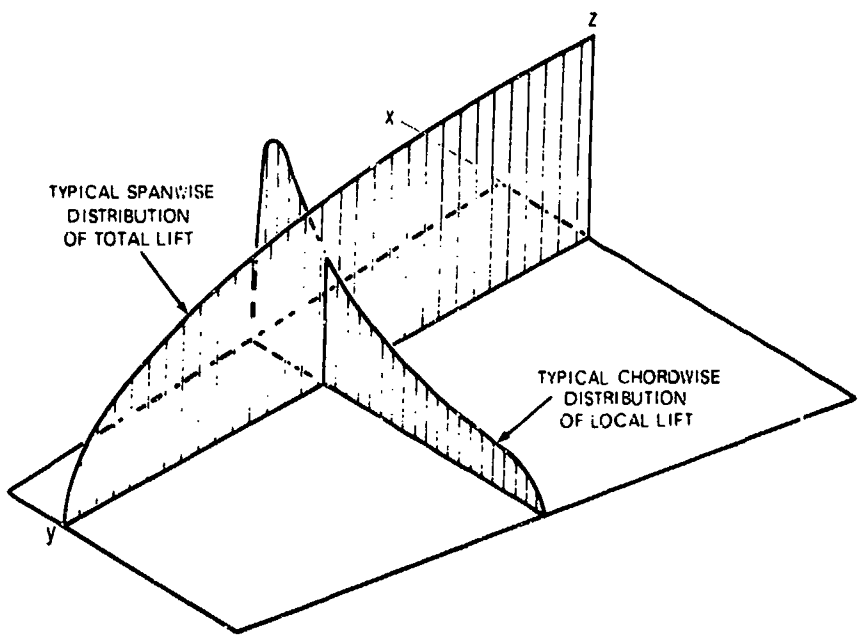


Fig. 4-72. Typical Spanwise and Chordwise Lift Distribution

tion of miles flown. The probability distribution is used to determine gust fatigue loading spectra and may be used to determine design limit loads (par. 4-4.3).

For fixed-wing aircraft, an empirical gust alleviation factor is defined, and the analysis is based upon an incremental change in angle of attack.

For a compound helicopter, however, such a relatively simple presentation is not valid because of the entirely different aerodynamic environment of rotor and wing and the difference in structural stiffness between the rotor and the vehicle. It is not correct to treat the rotor separately and the vehicle as a fixed-wing aircraft and then add up the results. The only correct presentation is to find the dynamic response of the total vehicle to a time history of $V_e + U_{de}$ (equivalent flight speed plus design gust velocity, also equivalent airspeed). Methods of analysis on this subject are mostly proprietary and have not been treated in the literature. Also, because dynamic response is peculiar to the particular design configuration, general treatment of the subject is inconclusive as far as actual hardware design is concerned. Recent trends in gust intensity definition in terms of the power spectral density of gusts are described in FAA ADS-53.

One aspect, not fully explored for compound helicopter design, is the effect of wing stall at low velocity, where the value of $\Delta\alpha = U_{de}/V_e$ may result in significant lift loss and cause significant negative vertical accelerations. Because such lift loss subsequently affects the rotor inflow velocity, a time history of the total event must be considered in order to evaluate the associated rotor response.

As shown in Fig. 4-74, depending upon which value of α is pertinent at the time that the gust occurs, the incremental $\Delta\alpha$ may result in the wing angle of attack α entering the stall region.

Where the original lift force equilibrium may be represented by point (A) on Fig. 4-75, the gust may change this to point (B) or (C), while the rotor may experience its own increment (R). The net lift resulting from these changes will differ from the original 1-g lift and, therefore, will cause an incremental load factor on the vehicle.

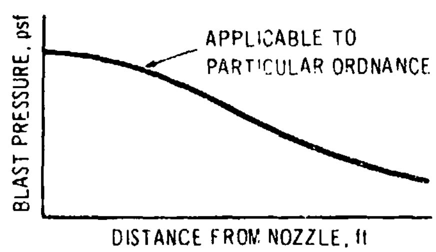


Fig. 4-73. Typical Ordnance Blast Pressure

4-10.2.6 Ground-handling Loads

Ground-handling load criteria are specified in par. 4-6. The requirement for equilibrium is, of course, that the resultant of all reactions opposes the action at the specified tow lug. The location of ground and inertial reactions for the same tow force might be quite different, causing a different load path with respect to the vehicle, as shown in Fig. 4-76. Externally applied loads resulting from these conditions are entered—together with the landing gear, propeller and tail rotor loads—into the external load matrices.

4-10.2.7 Mooring, Jacking, Hoist, and Sling Loads

Mooring and jacking criteria are given in par. 4-6.2. The mooring loads are based upon exposure of the unattended aircraft to severe gusty winds from any direction. The manufacturer usually will specify recommended mooring practice. In applying the criteria, the applicable landing gear deflections should be considered in the determination of exposed areas and moment arms. If parking on the soil of an unprepared field is specified, deflection due to soil impression (and "digging") should be considered.

Jacking provisions usually are made on all landing gear units to facilitate tire and brake service. The aircraft is usually emptied for maintenance, and jacking provisions on the aircraft structure thus allow for a specified empty weight. To insure safety for the working crews, lateral and vertical load factors also are specified. It is mandatory that only approved jacking equipment be used.

Hoists and slings usually will be applicable to the fuselage only, but they also may be applied to an inboard wing store mount. Therefore, it is essential that the rated loading of such mounts includes appropriate dynamic factors resulting from maneuvers during airlifts by other helicopters. Examples are gusts encountered within the specified operational weather limits, and the inherent dynamic effects of the sling or hoist system (cable, rope, belt webbing, hoist speed, and braking characteristics). Also, any pendulum action or wind and drag forces should be within the stability boundaries of the aircraft.

4-10.2.8 Structural Design Requirements

The criteria for the structural design of the wing and empennage include avoidance of buckling due to air

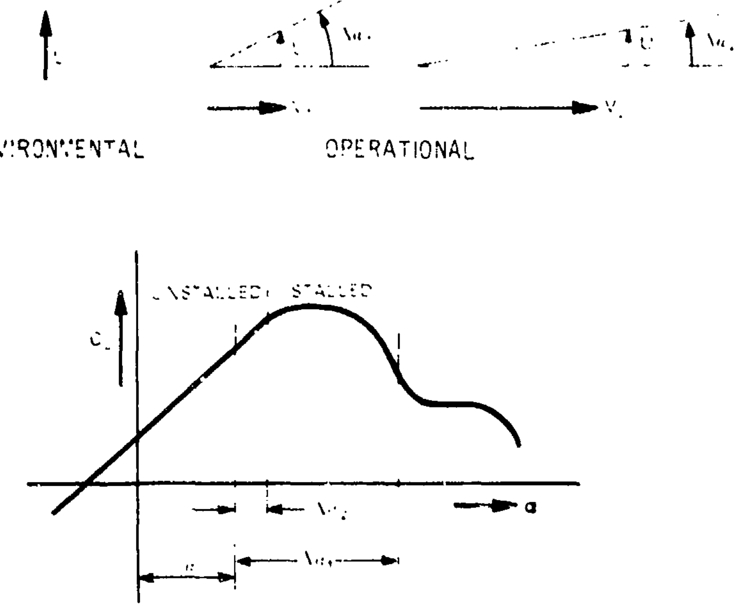


Fig. 4-74. Effect of Velocity on Incremental Lift Coefficient in Gust Encounter

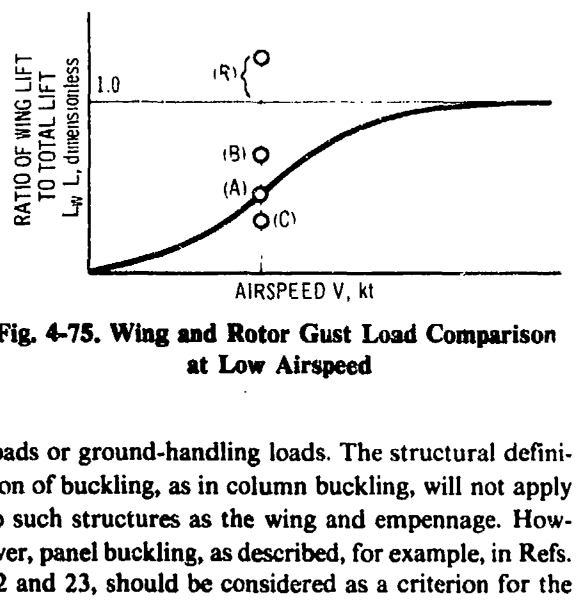


Fig. 4-75. Wing and Rotor Gust Load Comparison at Low Airspeed

loads or ground-handling loads. The structural definition of buckling, as in column buckling, will not apply to such structures as the wing and empennage. However, panel buckling, as described, for example, in Refs. 22 and 23, should be considered as a criterion for the properties of a skin-stringer combination where the buckling load per unit width can be derived from the bending moment divided by the effective height between upper and lower surfaces. This analysis will provide the critical compression stress of the panel configuration.

During ground-handling, only hard points in the structure—such as jack pads, mooring lugs, or towing lugs on the gear—may be used for concentrated load application. The dissipation of such loads into the structure should be such as to avoid structural crippling.

The true theoretical airfoil shape will be affected by structural deformation due to local pressures, and combined torsion and bending of the wing. Because such deflections will affect airfoil performance, the aerodynamic criteria must provide a tolerance for the magnitude of these local panel deflections, usually specified for level flight at a given airspeed and for the vehicle at rest on the landing gear. Therefore, a specific stiffness of the skin panels is required, dictating a minimum skin gage at different locations on the aircraft. Fig. 4-77 shows the general effect of panel dimensions on the critical shearflow, with the stiffening effect of the panel radius appearing as an additional term. The factors K_1 and K_2 should be obtained from an approved stress analysis manual. Skin gages also may be limited by the size and type of fastener used (see MIL-HDBK-5). Thus, either appearance or aerodynamic cleanliness requirements may dictate a heavier skin gage than would have been required for strength or stiffness only. Manufacturing practice, usually described in material processing specifications by each manufacturer, is another limiting factor in the choice of skin gages.

Depending upon the proximity of the tail surfaces to the engine exhaust and upon the acoustical output of the engine, it might be necessary to design the local

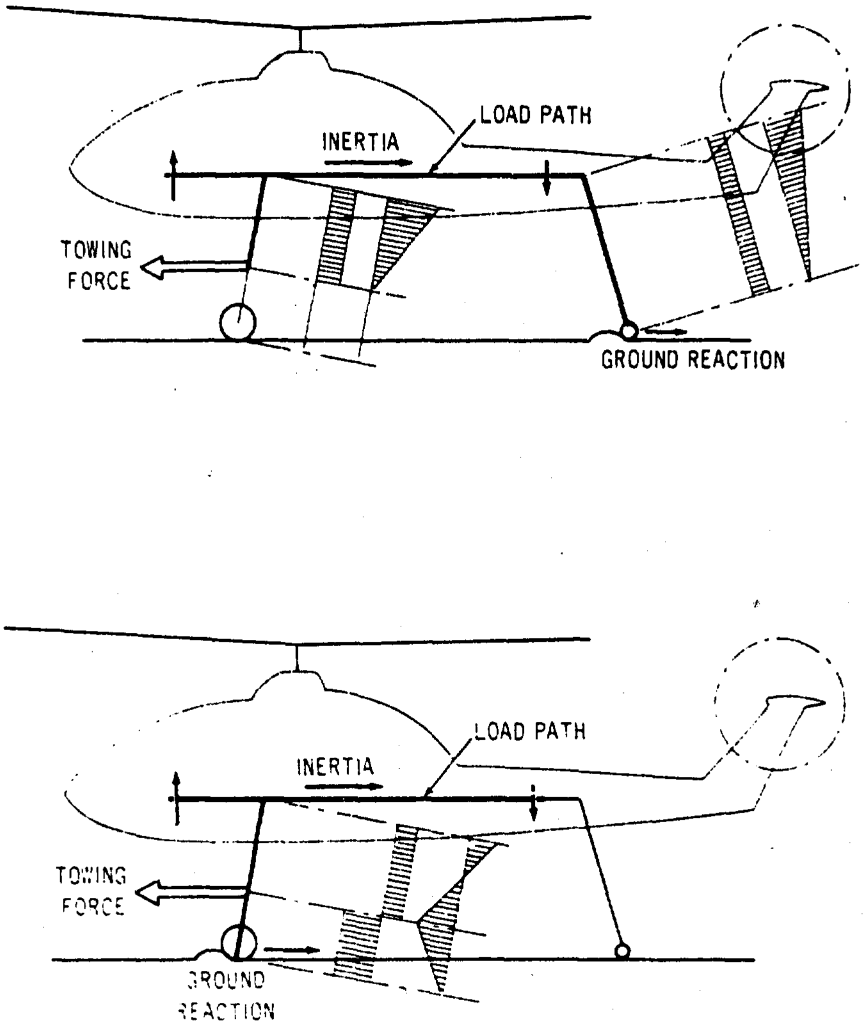
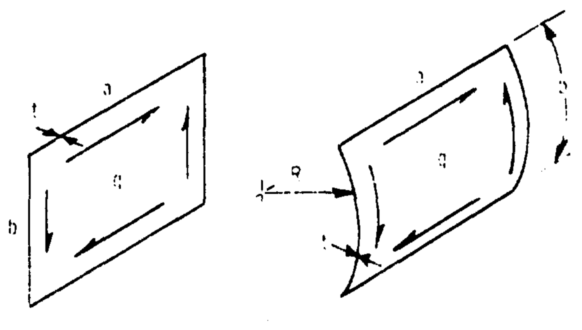


Fig. 4-76. Comparison of Towing Conditions With Different Restraints



$q_{cr} = (K_1 E_t (\frac{t}{b})^2 + (K_2 E_t (\frac{t}{R}))$

$K_1 = f(b/t, b/a, \text{EDGE RESTRAINT})$

$K_2 = f(t/R)$

E_t = TANGENT MODULUS, psi

Fig. 4-77. Allowable Shear Buckling Shearflow

structure to withstand sonic fatigue. This entails the selection of panel parameters for which the response to the random noise input produces stress levels below the endurance limit of the structure. Careful consideration must be given to edge design, support symmetry, fasteners, local deflection possibilities, stiffener restraints, etc., because this type of loading behaves like a continuous frequency sweep, passing through all the natural frequencies of the structure and accumulating millions of cycles. A more detailed discussion of this subject is contained in Ref. 59.

4-10.2.9 Combined Loading Conditions

Whatever the detail structure of the wing may be, in its simplest form it may be represented schematically as shown in Fig. 4-78.

The first approach of the substantiation effort is to determine the spanwise loading distribution according to conventional cantilever beam analysis, obtaining shear and moment diagrams and showing the three internal force components and three moment components at each spanwise station on a convenient reference axis. This operation must be repeated for all significant loading conditions in order to find the actual combined loading for each condition. For instance, combined shear and torsion for two conditions may compare as presented in Fig. 4-79, and combined bending and axial load may be compared on the basis of Fig. 4-80.

4-10.2.10 Stress Analysis

It thus is possible to represent all combined loading conditions in an envelope as shown in Fig. 4-81, giving all the loading details required for stress analysis.

The cantilever beam representation gives all of the equilibrium forces and moments required for wing and empennage design, as well as the reactions at the fuselage mating plane to be entered as actions on the fuselage. Margins of safety for both limit and ultimate loads *shall* be determined and tabulated for all the primary structural members of the wing and empennage, with emphasis upon the fittings for attachment to the main fuselage structure. The analytical methods and techniques to be used are well described in such standard texts as Refs. 22 and 23.

4-10.3 LANDING GEAR SUBSTANTIATION

Landing gear loads are dependent upon descent velocity, type of energy absorber, and landing gear configuration. Descent velocity criteria are specified in par. 4-5.1. For the purpose of this presentation, load factor determination is discussed; then specific loads are explained with special emphasis on type of gear (wheel, skid, and float). The landing gear then can be sized and substantiated for the critical landing and ground handling loads.

4-10.3.1 Load Factor Determination

The landing load factor ultimately is determined by drop tests, but for preliminary design purposes an appropriate value can be assumed. Generally, the landing load factor is chosen so that it is equal to that expected during a pullup maneuver. This assures that the structure is a balanced design. In any case, it is recommended that $n_{z_{lim}}$ not be less than 2.67 because even a minor vertical crash will give an ultimate load factor of at least 4.0.

Having chosen the load factor n_z, the energy absorption deflection—including all components such as tires, shock struts, elastic or yielding springs, and float bags—can be determined. The energy to be absorbed upon landing is kinetic energy KE plus potential energy PE, or

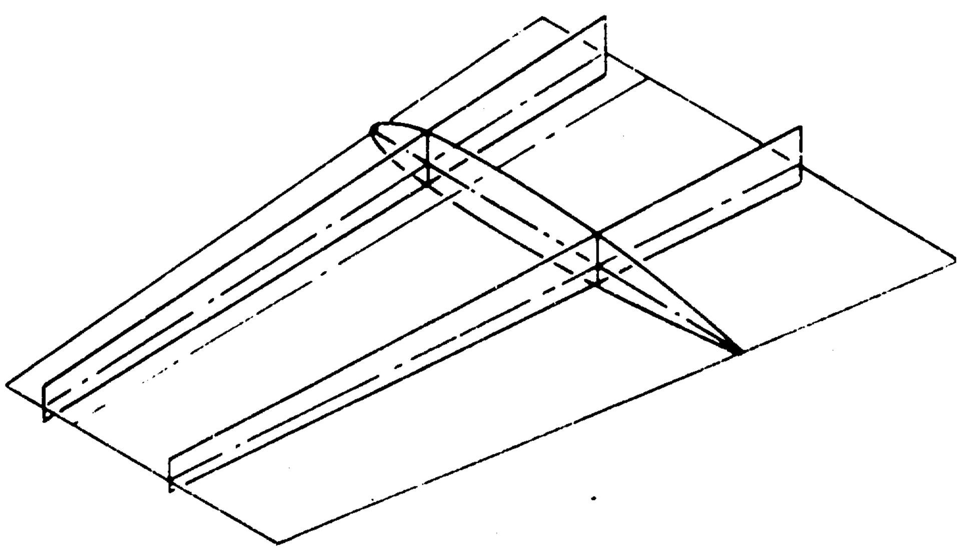
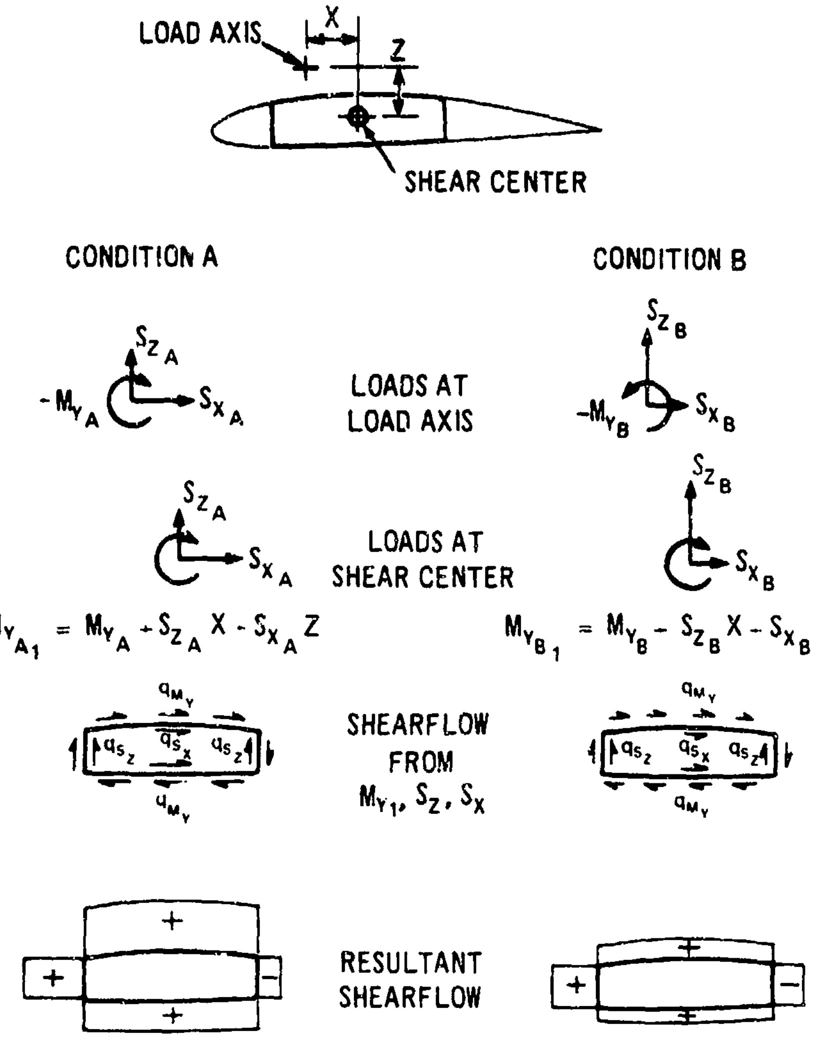


Fig. 4-78. Schematic Beam-rib Structure



$$q = q_{M_Y} + q_S + q_{S_X} = \frac{M_Y}{2A} + \frac{S_Z Q_{XX}}{I_{XX}} + \frac{S_X Q_{ZZ}}{I_{ZZ}}$$

Fig. 4-79. Comparison of Combined Shear and Torsion for Two Loading Conditions

$$KE + PE = \frac{W_g V_s^2}{2g} + W_g(1-L)\delta \quad \text{, ft-lb} \quad (4\text{-}55)$$

Absorbed energy AE is

$$AE = KR_{gr}\delta = K(n_z W_g - LW_g)\delta \quad \text{, ft-lb} \quad (4\text{-}56)$$

where

V_s = descent velocity (par. 4-5.1), fps
n_z = load factor, dimensionless
L = rotor lift ratio = rotor lift/gross weight (= 0.67 from par. 4-5.1), dimensionless
R_{gr} = ground reaction, lb
K = energy absorber efficiency, d'less
W_g = gross weight, lb
δ = absorber deflection required during a limit landing, ft

Equating Eqs. 4-55 and 4-56

$$\frac{V_s^2}{2g} + (1-L)\delta = K(n_z - L)\delta \quad , \text{ft} \qquad (4\text{-}57)$$

and solving for δ

$$\delta = \frac{V_s^2}{2g}\left[\frac{1}{K(n_z - L) - (1 - L)}\right] \quad , \text{ft} \qquad (4\text{-}58)$$

As written, Eqs. 4-56 through 4-58 are strictly accurate only for landing gear systems containing only one energy absorbing component or device. For example, with an elastic spring absorber such as used with skid landing gear, the appropriate value of $K = 0.5$. However, when an oleo strut is being used together with a tire, care must be exercised in the selection of the proper value of K. The respective deflections of the oleo strut and the tire must be considered, and a composite value of K determined, based upon the total deflection δ being the sum of tire deflection and oleo deflection. The landing gear then is designed to provide this value of total deflection during a limit landing.

The limit landing gear load or ground reaction R_{gr} is, by definition,

$$R_{gr} = W_g[n_z - L] \quad , \text{lb} \qquad (4\text{-}59)$$

This load is divided among components of the land

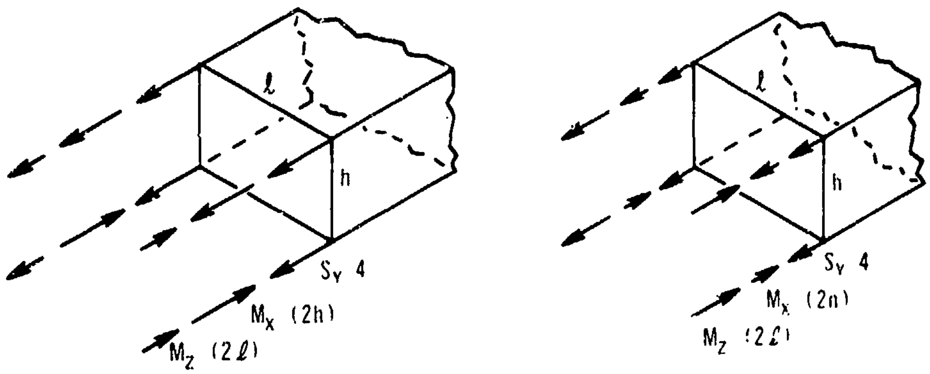


Fig. 4-80. Combined Bending and Axial Load for Two Loading Conditions

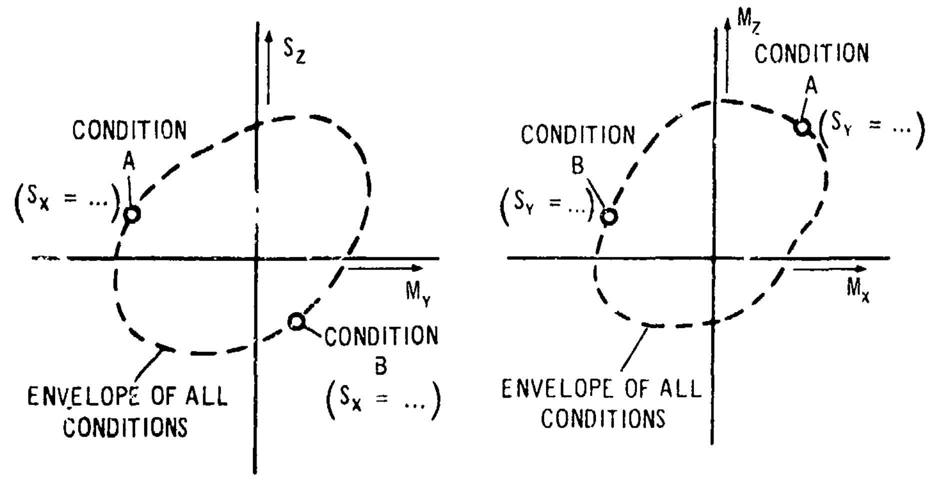


Fig. 4-81. Combined Loading Condition Envelopes at a Particular Location

ing gear (nose, tail, and main) for design purposes as presented in the paragraphs that follow.

4-10.3.2 Specific Landing Gear Load Determination

Criteria for the specific conditions that *shall* be considered are detailed in par. 4-5.1.1 (level landings) and par. 4-5.1.2 (asymmetric landings). For the applicable type of landing gear the loads for each condition must be determined by rational or conservative methods. Critical loads then are used in the design of the gear and attachments and applied to the fuselage.

4-10.3.2.1 Wheel Gear Loads

Loads on the tire, oleo strut, and support structure are in accordance with par. 4-5.1.

As most of the landing conditions for wheel gear are explicit and the determination of loads is straightforward, no further treatment is necessary here. The conditions to be considered, with due regard to critical weight and CG position, are:

1. Spin-up and spring-back (with appropriate vertical reaction)
2. Maximum vertical reaction level landings without and with forward speed (with appropriate fore and aft load)
3. Asymmetric landings with drift or horizontal forces at one wheel reacted by helicopter linear and rotational inertia about the CG
4. Taxiing (braking, turning, and obstruction)
5. Ground handling (towing)
6. Landing gear extension and retraction
7. Special conditions.

The helicopter rotor and landing gear combination must be designed for freedom from ground resonance. With a configuration having lead-lag hinges and oleo landing gear, it is probable that an instability of the helicopter on its gear will take place unless sufficient damping is provided in either the rotor system or the oleo strut (par. 5-2.5).

4-10.3.2.2 Skid Gear Loads

The analysis that follows outlines a procedure for determining loads for the dynamic landing condition for helicopters equipped with skid landing gear. The helicopter is assumed to have two horizontally mounted "spring tubes" as energy-absorbing devices. With the skid runners in contact with the ground, the system may be represented schematically by a mass supported by four springs, as illustrated in Fig. 4-82.

The critical loading condition is maximum forward CG at design gross weight. The method follows:

1. Determine:

W_g = gross weight of helicopter, lb
I = pitching moment of inertia, lb-in.-sec^2
a, b, c, and d = dimensions identified in Fig. 4-82, in.
k_1, k_2, k_3, and k_4 = spring rates identified in Fig. 4-82, lb/in.

2. Evaluate:

$$
\begin{aligned}
K_1 &= k_2 + k_4 \quad, \text{lb/in.}\\
K_2 &= k_2 c + k_4 d \quad, \text{lb}\\
K_3 &= k_1 + k_3 \quad, \text{lb/in.}\\
K_4 &= k_1 b - k_3 a \quad, \text{lb}\\
K_5 &= k_1 b^2 + k_2 c^2 + k_3 a^2 + k_4 d^2 \quad, \text{lb-in.}
\end{aligned}
\tag{4-60}
$$

3. Determine values of P_x, P_z, and M_θ for static condition:

$$
\begin{aligned}
P_x &= M_\theta = 0 \quad, \text{lb}\\
P_z &= W_g(1-L) = W_g/3 \quad, \text{lb}
\end{aligned}
\tag{4-61}
$$

where

L = ratio of rotor lift to gross weight = 0.67 (par. 4-5.1.1.1)

4. Evaluate the static deflections:

$$
\begin{aligned}
x_{ST} &= \frac{K_2 K_4 P_z}{K_1 K_4^2 + K_2^2 K_3 - K_1 K_3 K_5} \quad, \text{in.}\\
z_{ST} &= \frac{(K_2^2 - K_1 K_5) P_z}{K_1 K_4^2 + K_2^2 K_3 - K_1 K_3 K_5} \quad, \text{in.}\\
\theta_{ST} &= \frac{K_1 K_4 P_z}{K_1 K_4^2 + K_2^2 K_3 - K_1 K_3 K_5} \quad, \text{rad}
\end{aligned}
\tag{4-62}
$$

5. Determine frequencies:

a. If K_1 does not equal K_3, solve the following equation for the three roots ω_1^2, ω_2^2, and ω_3^2:

$$\omega^6 - \left(\frac{K_1 + K_3}{W_g/g} + \frac{K_5}{I}\right)\omega^4 + \left[\frac{K_1 K_3}{(W_g/g)^2} + \frac{(K_1 + K_3)K_5 - K_2^2 - K_4^2}{IW_g/g}\right]\omega^2 + \frac{K_1 K_4^2 + K_2^2 K_3 - K_1 K_3 K_5}{I(W_g/g)^2} = 0 \qquad (4\text{-}63)$$

where

g = acceleration due to gravity, in./sec²

b. If $K_1 = K_3$, determine the frequencies for the three natural modes:

$$\omega_1^2 = \frac{K_1}{W_g/g}$$

$$\omega_2^2 = \frac{1}{2}\left(\frac{K_1}{W_g/g} + \frac{K_5}{I}\right) + \frac{1}{2}\sqrt{\left(\frac{K_1}{W_g/g} - \frac{K_5}{I}\right)^2 + 4\left(\frac{K_2^2 + K_4^2}{IW_g/g}\right)} \qquad (4\text{-}64)$$

$$\omega_3^2 = \frac{1}{2}\left(\frac{K_1}{W_g/g} + \frac{K_5}{I}\right) - \frac{1}{2}\sqrt{\left(\frac{K_1}{W_g/g} - \frac{K_5}{I}\right)^2 + 4\left(\frac{K_2^2 + K_4^2}{IW_g/g}\right)}$$

6. Evaluate Q_1, Q_2, Q_3, R_1, R_2, and R_3 from

$$Q_i = \frac{K_2}{K_4}\left(\frac{\omega_i^2 W_g - K_3 g}{K_1 g - \omega_i^2 W_g}\right) \qquad (4\text{-}65)$$

$$R_i = \left(\frac{\omega_i^2 W_g - K_3 g}{K_4 g}\right)$$

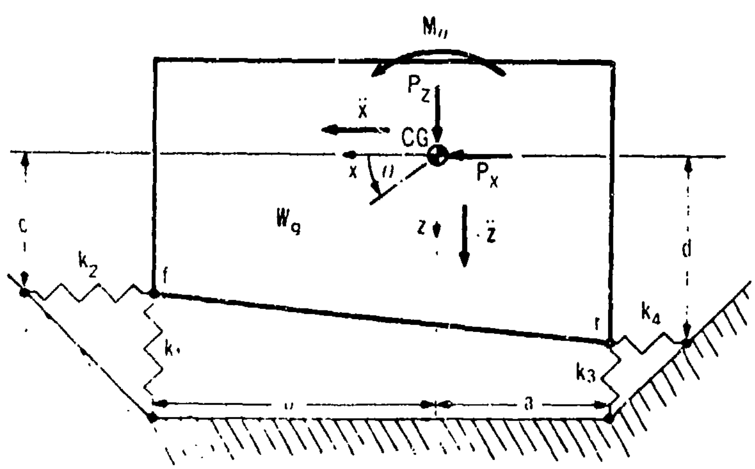


Fig. 4-82. Schematic of Spring Substitution for Skid Gear

where

Q_i = generalized displacement x_i/z_i, dimensionless

R_i = generalized displacement θ_i/z_i, in.$^{-1}$

If $K_1 = K_3$, use

$$Q_1 = \frac{K_2}{K_4}, R_1 = 0$$

for (4-66)

$$\omega_1^2 = \frac{K_1 g}{W_g} = \frac{K_3 g}{W_g}$$

and for $i = 2$ and $i = 3$, use

$$Q_i = -\frac{K_2}{K_4} \quad (4\text{-}67)$$

with R_i from Eq. 4-65

7. Determine initial conditions at $t = 0$ (time skids contact ground):

$$\begin{aligned} x &= x_o & \dot{x} &= \dot{x}_o \\ z &= z_o & \dot{z} &= \dot{z}_o = V_s \\ \theta &= \theta_o & \dot{\theta} &= \dot{\theta}_o = 0 \end{aligned} \quad (4\text{-}68)$$

$\dot{x}_o$ is varied until the maximum horizontal ground reaction is equal to one-half the vertical ground reaction.

8. Set up the following equations:

$$\begin{aligned} &Q_1 z_1 \sin\gamma_1 + Q_2 z_2 \sin\gamma_2 + Q_3 z_3 \sin\gamma_3 = x_o - x_{ST} \\ &z_1 \sin\gamma_1 + z_2 \sin\gamma_2 + z_3 \sin\gamma_3 = z_o - z_{ST} \\ &R_1 z_1 \sin\gamma_1 + R_2 z_2 \sin\gamma_2 + R_3 z_3 \sin\gamma_3 = \theta_o - \theta_{ST} \end{aligned} \quad (4\text{-}69)$$

Solve for $z_1 \sin\gamma_1$, $z_2 \sin\gamma_2$, and $z_3 \sin\gamma_3$.

9. Set up the following equations:

$$\begin{aligned} &\omega_1 Q_1 z_1 \cos\gamma_1 + \omega_2 Q_2 z_2 \cos\gamma_2 + \omega_3 Q_3 z_3 \cos\gamma_3 = \dot{x}_o \\ &\omega_1 z_1 \cos\gamma_1 + \omega_2 z_2 \cos\gamma_2 + \omega_3 z_3 \cos\gamma_3 = \dot{z}_o \\ &\omega_1 R_1 z_1 \cos\gamma_1 + \omega_2 R_2 z_2 \cos\gamma_2 + \omega_3 R_3 z_3 \cos\gamma_3 = \dot{\theta}_o \end{aligned} \quad (4\text{-}70)$$

Solve for $z_1 \cos\gamma_1$, $z_2 \cos\gamma_2$, and $z_3 \cos\gamma_3$.

10. For $i = 1, 2,$ and 3, evaluate:

$$\gamma_i = \mathrm{Tan}^{-1}\left(\frac{z_i \sin\gamma_i}{z_i \cos\gamma_i}\right) \quad x_i = Q_i z_i$$

$$z_i = \frac{z_i \sin\gamma_i}{\sin\left[\mathrm{Tan}^{-1}\left(\frac{z_i \sin\gamma_i}{z_i \cos\gamma_i}\right)\right]} \quad \theta_i = R_i z_i \quad (4\text{-}71)$$

11. Substitute the solution to Eqs. 4-69, 4-70, and 4-71 into the following equations:

$$\begin{aligned} x &= x_1 \sin(\omega_1 t + \gamma_1) + x_2 \sin(\omega_2 t + \gamma_2) + x_3 \sin(\omega_3 t + \gamma_3) + x_{ST} \\ z &= z_1 \sin(\omega_1 t + \gamma_1) + z_2 \sin(\omega_2 t + \gamma_2) + z_3 \sin(\omega_3 t + \gamma_3) + z_{ST} \\ \theta &= \theta_1 \sin(\omega_1 t + \gamma_1) + \theta_2 \sin(\omega_2 t + \gamma_2) + \theta_3 \sin(\omega_3 t + \gamma_3) + \theta_{ST} \end{aligned} \quad (4\text{-}72)$$

Fig. 4-83 presents representative curves as obtained from Step 11 and the equations:

$$\frac{W_g}{g}\ddot{x} + (k_2 + k_4)x - (k_2c + k_4d)\theta = P_x$$

$$\frac{W_g}{g}\ddot{z} + (k_1 + k_3)z + (k_1b - k_3a)\theta = P_z \quad (4\text{-}73)$$

$$I\ddot{\theta} + (k_1b^2 + k_2c^2 + k_3a^2 + k_4d^2)\theta - (k_2c + k_4d)x + (k_1b - k_3a)z = M_\theta$$

where

M_θ = $LW_g \times$ (distance to CG) throughout landing.

4-10.3.2.3 Float Gear Loads

Float gear can be analyzed in a manner similar to skid gear with changes in the forward, aft, and side components. Unless specified otherwise by the procuring activity, the criteria of FAR Part 27 are applicable for float landing gear. The dynamic condition is less severe than that for skid gear.

4-10.3.3 Reserve Energy Loads

A reserve energy condition is used for ultimate design of the landing gear. The criteria for this condition are given in par. 4-5.2. The increased load due to the reserve energy condition can be expressed in terms of the ratio of the ground reactions at design limit sink speed and at the reserve energy sink speed. This ratio can be calculated, using Eqs. 4-57 through 4-59, with appropriate values of sink speed V_s, absorber efficiency K, and rotor lift ratio L for the design limit and reserve energy conditions.

4-10.3.4 Preliminary Structural Sizing

The preliminary structural sizing of the landing gear normally can be accomplished using standard detail methods of structural analysis, in combination with the design limit and reserve energy loads as well as the various other design loads, and including ground-handling loads. The methods of Ref. 57 should be used whenever possible to assure an optimized structural configuration. Very-high-strength materials normally can be specified for the landing gear components inasmuch as flight safety is not involved and the critical landing gear loads rarely are encountered.

Energy absorption sizing for the normal limit contact velocities usually relates to sizing the oleo strut (if applicable) and the flexible series member, such as tire or beam. Sizing for the reserve energy requirements in accordance with par. 4-5.2 involves the plastically deforming element, or structural load-fuse. Also, there may be size and/or configurational effects that require stiffness or damping of the oleo struts in roll or pitch (such as to prevent ground resonance or for better terrain conformability) that is different from the damping required for the vertical impact. Thus, some method of

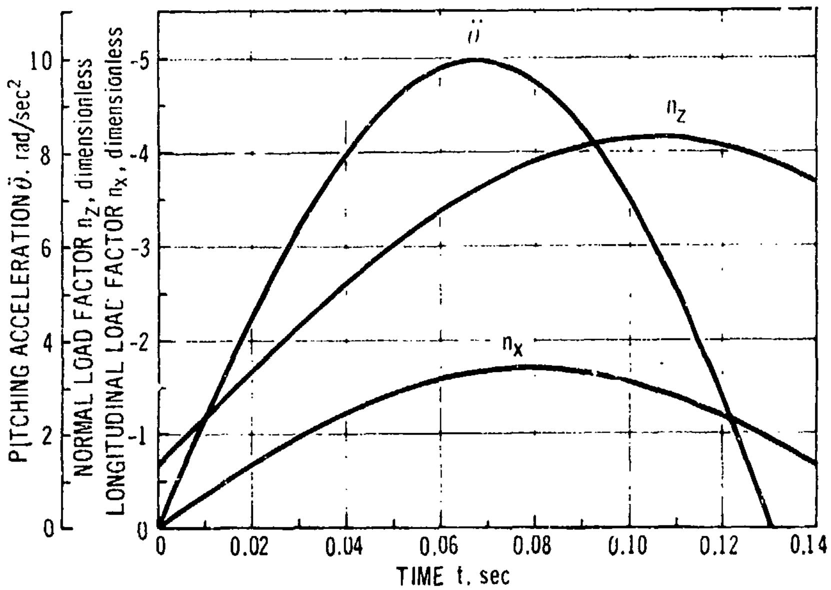


Fig. 4-83. Time History of Level Landing With Forward Velocity

mechanically or hydraulically separating the symmetrical and asymmetrical energy absorption characteristics of the landing gear may be necessary.

4-10.3.5 Substantiation of Landing Gear Design

Substantiation of helicopter landing gear designs should consist of a thorough discussion of any unique or unconventional features, their technical or other advantages, and the possible technical risks involved. Impact absorption characteristics, both limit and reserve energy, should be justified based upon combinations of theoretical analyses, computer dynamic simulation studies, and comparison with existing drop test data. The ground loads due to the limit and reserve energy impacts—as well as the other miscellaneous design load requirements of MIL-S-8698 and MIL-A-8862—should be investigated to a reasonable degree of confidence prior to structural sizing of the landing gear components. Special loading conditions may be necessary for particular landing gear configurations (par. 4-5.1). The miscellaneous ground-handling requirements of Ref. 62 also should be investigated from the design load standpoint. Landing gear motion, stiffness, and damping may need to be investigated with regard to assuring freedom from the likelihood of ground resonance (see Ref. 63 for detailed methods of substantiation).

After the design loads, impact requirements, and ground resonance suitability requirements have been investigated, structural sizing of the various landing gear components can be accomplished using standard structural analysis procedures. Margins of safety for limit and reserve energy conditions *shall* be calculated and tabulated for principal landing gear components.

4-10.4 SUBSTANTIATION OF MISSION EQUIPMENT INSTALLATIONS

Normally, the mission equipment requirements for a helicopter have been established by the time the design is subjected to its preliminary structural analysis. However, because this equipment is not a part of the basic helicopter, there may be a tendency to overlook provisions for such equipment in the initial preliminary design. This procedure is unacceptable because the efficiency in adapting the helicopter to the use of mission equipment often determines system effectiveness. In fact, the design integration of mission equipment provisions into the basic helicopter may be the most important task in the early design stages. For example, a nonoptimized location of a weapon pod may result in a serious stability and control problem. Also, upon subjecting a mission equipment mounting location to preliminary structural analysis, it may become apparent that the internal structure of the helicopter must be revised to provide efficient load paths.

Therefore, the preliminary structural analysis for mission equipment *shall* encompass more than merely a stress analysis. It *shall* include considerations of location that might affect personnel accommodations, handling characteristics, and safety concepts. Essential considerations include CG locations, stiffness, maneuver loads, aerodynamic loads, recoil loads, location relative to crew, and ease of maintenance of mission equipment.

4-10.4.1 Equipment Loads

The basic loads associated with helicopter equipment *shall*, where possible, be based on the criteria that have been established in the foregoing paragraphs of this chapter. It is recognized, however, that such equipment involves a wide variety of components and associated operating conditions; therefore, some operating conditions will not have been covered in the previously established criteria. In those cases where the exact criteria have not been discussed, the logic behind the criteria that were established and discussed should be used in establishing the new criteria needed. Additional equipment criteria and requirements also may be established in the RFP.

Equipment loads resulting from maneuver and crash load conditions are discussed here, with emphasis on the need to restrain equipment within the cabin area during a crash and to restrain equipment installed outside from penetrating into the cabin area. Belly-mounted and pod-mounted armament loads are included.

4-10.4.1.1 Equipment Types

A representative listing of equipment items has been divided into three broad categories:

1. Equipment installed within the cabin
2. Equipment installed internally, but outside of the cabin area
3. Externally installed equipment.

This listing should be adequate for establishing the methods and constraints for determination of the required basic loads.

4-10.4.1.1.1 Cabin Equipment

Equipment installed in the cabin includes, but is not limited to:

1. Environmental control system components
2. Fire extinguishers
3. Pilot operated sights or observation equipment
4. Oxygen equipment
5. Electronic components
6. Personnel armor
7. Instruments
8. Other mission equipment.

4-10.4.1.1.2

Equipment installed outside the cabin but inside the helicopter includes, but is not limited to:

1. Environmental control system components
2. Pneumatic system components
3. Hydraulic system components
4. Ballast
5. Engine and engine accessories
6. Battery and electrical components
7. Power supplies for instrumentation and electronics
8. Electronic components
9. Fuel tanks and oil tanks
10. Coolers and radiators
11. Mission equipment.

4-10.4.1.1.3 Externally Installed Equipment

Externally installed equipment includes, but is not limited to:

1. Belly-mounted armament
2. Pod-mounted armament
3. Auxiliary fuel tanks
4. Mission equipment.

4-10.4.1.2 Load Types

All loads, including those within the equipment, must be accounted for and considered in the design of a helicopter The primary consideration in this discussion, however, is the externally applied loads. These loads include inertial loads from flight maneuvers, air loads, vibratory loads, and crash loads.

4-10.4.1.2.1 Flight Maneuver Loads

In the determination of inertial loads from flight maneuvers, the three required inputs for the computation are:

1. The design flight load factors and angular accelerations
2. Weights and moments of inertia of the equipment
3. Location of the equipment relative to the helicopter CG.

Limit load factors are discussed in par. 4-4.2. Equipment and equipment mounting *shall* be designed for the limit load factor of 3.5 that is required by MIL-S-8698 for Class I service, unless a different factor is justified.

The weight of mission equipment normally will be specified in the RFP. An accurate weight must be established for each component, including those components not specified by the RFP. Also, the location of the equipment must be determined. In those cases where disposable loads such as fuel or ammunition are carried, the locations for varying load conditions must be examined to establish the most critical loading. Inertial moments due to angular accelerations can be important, and an item of equipment located at a significant distance from the helicopter CG may be subjected to significant load factors as a result of these accelerations.

In addition, the location of the equipment CG in relation to its mounting points may be an important consideration, with significant increases in support loads resulting from offsets or eccentricities in the support system.

Other flight-related loads that sometimes may exceed the flight maneuver loads are landing, taxiing, and ground-handling loads. The load factors associated with these operating conditions also must be considered. The landing criteria are discussed in par. 4-5, taxi and ground-handling loads in par. 4-6.

4-10.4.1.2.2 Air Loads

Externally mounted equipment is subjected to air loads caused by airspeed, rotor downwash, and gusts, and in some cases by tail rotor or propeller thrust. Significant reductions in air loads may be obtained by the use of proper airfoil shapes and streamlined fairings on the exposed equipment—good design practice dictates such an approach. However, the air loads never can be eliminated and must be included in any design analysis.

The air loads are best determined from test data. If testing is impracticable, applicable loads must be developed by analytical methods. The loads from rotor downwash and tail rotor or propeller thrust may be determined by application of an estimated air velocity, a drag coefficient, and the reference area of the equipment in the air stream. An important consideration is the proximity of the equipment installation to the rotors and propellers. The determination of these loads may initiate additional design iterations, resulting in a more efficient configuration.

The gust criteria have been discussed in par. 4-4.3. The resultant air velocity associated with the limit gust should be applied so that the most critical and largest area is exposed.

4-10.4.1.2.3 Vibratory Loads

Vibratory loads on equipment are caused by vibrations from several sources, among which are:

1. Self-imposed vibration
2. Helicopter structural vibration
3. Rotor and propeller blade downwash frequency
4. Aerodynamic buffeting
5. Armament firing frequency.

Additional vibratory loads that normally are low in frequency and are experienced only occasionally are blast loads from armament and reaction loads from jettison or launching operations.

Equipment vibratory loads often are the critical loads that establish the structural configuration. Once the strength for these loads is adequate, all other loads —including flight maneuver and crash loads—become secondary in importance. This is applicable especially to certain armament installations where the firing frequency loads are high. Another factor is the large reduction in allowable cyclic stress or endurance limit caused by stress raisers, which do not reduce significantly the allowable stress for an ultimate load such as a crash load.

The self-imposed equipment vibratory loads originate from equipment containing rotating or oscillating components such as fans. These vibratory loads should be isolated from the helicopter where possible, but their consideration must be a part of the preliminary design basic load analysis.

All equipment will be subjected to helicopter structural vibration, and allowance for the resulting loads cannot be neglected during the preliminary design stage. This is particularly true of installations for sensitive equipment such as flight instruments and electronic assemblies. The vibratory loads on these components must be determined and used in the design of appropriate vibration mounts. The most direct approach for establishing these loads often is review of test data from similar helicopters. The possibility of fatigue failure of flight instrument and avionic equipment installations as a result of the vibratory loads to which they are exposed *shall* be considered.

Vibratory loads from rotor and propeller blade downwash impulses must be evaluated for all externally installed equipment. The frequency is a direct function of the rotational speed and number of blades. The magnitude of the load is determined by test or by rational analytical means, as discussed in par. 4-10.4.1.2.2.

The determination of firing frequency loads from armament installations and gun pods is discussed in par. 4-8.1.3. Loads from blast overpressure and reaction loads from jettison or launching operations are discussed in pars. 4-8.1.2 and 4-8.1.4.

4-10.4.1.3 Determination of Loads

The determination of the basic loads associated with equipment and equipment installations often is limited during preliminary design to the most critical loading condition. All applicable loading conditions are considered, the most critical combination selected, and the loads determined. The critical loads will be useful for design of some of the equipment, as well as of fittings and other structural components of the helicopter.

4-10.4.1.3.1 Loads on Cabin Equipment

Crash loads are the critical design loads for equipment that is stored in the cabin. All equipment must be restrained so that clearance is maintained between it and all personnel during any crash within the limits of human survivability. A properly designed cabin will maintain its shape during a crash, and it is essential that pieces of equipment do not become loose missiles that can injure the occupants. In addition, under the prescribed crash load factors the structural fittings supporting the equipment must not deform to the extent that the occupants can be injured.

Location of equipment is an important factor for crash consideration. Because the maximum impact velocity is normally in the vertical direction for helicopters, equipment stored on the floor or in the lower portion of the cabin is less likely to become lethal. This, however, does not make the separation of a component in this area from its attachment an acceptable condition. It is characteristic to have the cabin experience at least one bounce after the major crash impact. Therefore, it is mandatory that the equipment be retained in its attached position to prevent personnei injury during the negative acceleration cycle. The requirement, then, is to provide fittings on the equipment and the helicopter that will withstand without separation all load factors prescribed for the crash condition.

Several special equipment items that are located in the cabin warrant specific attention, namely:

1. Components that could be stored loose—small arms, flight handbook, briefcase, etc.

2. Shock-mounted equipment

3. Movable components such as armament sights that require temporary storage.

Loose equipment must be stored in a location that reduces its likelihood of causing injury under crash conditions, and it must be tied down with straps or equivalent devices capable of withstanding the prescribed load factors. Equipment such as the flight handbook, for example, is less dangerous if located in the lower forward portion of the cabin. Small arms or briefcases should not remain loose in the cabin; enclosed compartments should be provided that are strong enough to restrain these items during a crash.

Shock mounts are installed on sensitive equipment to isolate helicopter vibratory loads. Because these loads seldom exceed 1 g, particular attention should be paid to the effect of basic crash load factors to these components. Avionic and flight instrument installation also *shall* be evaluated to assure their ability to withstand the vibratory loads without failure due to fatigue.

Movable components such as variable position armament sighting equipment also must remain intact and in position during a crash. The crash loads should apply to the unit and its mounting in its stored position, provided it is used only during specific operational conditions where a crash is extremely unlikely. The provision of a structurally sound stored position is mandatory for the protection of members of the crew who normally use the device.

4-10.4.1.3.2 Loads For Equipment Installed Internally but Outside of Cabin

Crash loads are not necessarily the critical design loads for a number of equipment items installed inside the helicopter but outside of the cabin. The determination of critical loads depends upon the location of the specific item of equipment. An engine located just above the cabin, for example, must be restrained from penetrating the cabin and, therefore, the crash load criteria may be critical. On the other hand, if the engine is located in a low position remote from the cabin, such that the occupants of the cabin would not be endangered when the engine broke loose from its mounts under crash conditions, other loads such as flight maneuver and vibrational loads would be the critical design conditions. Both fuel and oil tanks require special consideration for crash conditions because of their post-crash fire hazard. However, both types of tanks have specific requirements for pressure tests when installed in the helicopter. A more complete discussion of the engine installation, and of fuel system and oil system requirements, is included in pars. 8-2, 8-4, and 8-5.

All equipment items should be reviewed to determine the likelihood of cabin penetration in the event of a crash. This applies especially to high-density items such as batteries, ballast, and certain electronic units. The walls between compartments of a helicopter normally do not provide a very significant barrier against loose equipment moving at a velocity of 10-40 fps.

Mountings for system components such as pumps, starters, generators, pneumatic assemblies, and environmental control system items normally are designed by stiffness and vibration requirements. Even though some justification can be made for certain items of equipment being permitted to separate from their normal positions during a crash, in the most acceptable configuration all parts will remain together as a single assembly during all operations including a crash. Crash loads, therefore, have maximum priority as the critical load requirement unless vibratory or maneuver load conditions exceed them. Clear justification for any departure from this approach *shall* be mandatory.

4-10.4.1.3.3 Loads on Externally Attached Equipment

The basic loads for externally attached equipment often are furnished as part of the RFP because the external items tend to be mission-essential equipment. The primary items classified as externally mounted mission equipment are hooks and slings for external cargo, weapon pods, and similar items such as antennas, tanks, and special lights. The specific loads that must be provided for in the support and accommodation of externally mounted mission equipment are lift, drag, downwash, gust, inertia, jettison, buffeting, recoil, and drogue.

The flight maneuver loads applied to weapon pods for preliminary structural analysis should be based upon the maximum capability of the helicopter. The loads must include components due to pitch, roll, and yaw velocities, and accelerations as determined by pod location in addition to the load factor at the CG. In establishing aerodynamic loads, it is preferable to use quantitative data from similar vehicles or from wind tunnel tests. Loads must be based upon speeds up to V_{DL}.

Gust loads can be an especially important consideration for large-area components for which the gust load could be a significant factor in the sizing of the fittings.

The crash condition should be considered as an impact energy problem in which much of the impact load is absorbed by deformation of the basic helicopter. The criteria of par. 4-5.3 will provide a high probability of occupant survival under a crash impact velocity of 42 fps in the vertical direction. For consistency with this capability, any externally mounted piece of equipment that might injure an occupant of the helicopter if it should break loose from its mounting *shall* have the strength to withstand deceleration of at least 20 g without separation.

Certain externally mounted mission equipment may be permitted to break loose from the helicopter if there is assurance that no injury will result. In cases where fuel is involved, the requirements of the most recent criteria for crashworthy fuel cells should apply.

At the preliminary structural analysis stage, it normally is considered sufficient to size the fittings for mission equipment on the basis of maximum maneuver and maximum static loads. In many cases, however, it is advisable to consider fatigue loading conditions on certain externally mounted mission equipment. This is applicable especially for equipment such as weapon pods, where recoil load may be of such magnitude that low-cycle fatigue becomes a consideration. Some modern weapons accumulate cycles at a very high rate and, if the loads are known to be high, the possibilities of fatigue must be considered during the preliminary analysis period. The effect of blast upon the surrounding skin, as well as load impulses, also must be considered.

There are a number of other areas concerning fatigue of externally mounted mission equipment that may be used for reviewing the fittings for possible fatigue problems. They include:

1. Mount stiffness. The stiffness of the fitting that supports the mission equipment assembly should be estimated analytically. This stiffness then can be used in computing the resonant frequency of the assembly/fitting combination. The computed frequency should be at least 25%, and preferably 50%, above the recoil frequency or main rotor frequencies.

2. Aerodynamic excitation. Data on aerodynamic buffeting and any information on tendencies of flutter in mission equipment installation should be examined carefully.

3. Sling load bounce. The combined stiffness of the external sling and the helicopter sling attachment fitting must be such that the sling load does not tend to bounce in resonance with the rotor or rotor blade frequencies. This has been a problem in previous helicopter/sling load configurations, and available data should be reviewed for the purpose of avoiding the problem. If such loads are inevitable, the fitting design must allow for them at the preliminary design stage.

Jettison loads, landing loads, and emergency release loads for the external cargo sling probably will not determine the size of the external equipment mounting fittings, but they should be examined as a final check.

The jettisoning of an assembly normally is an emergency action. The reaction loads may be as high as the limit load but should not exceed this amount. Any fitting that may be subjected to a jettison load should be checked for this condition during the preliminary design stage.

Normal landing conditions should not affect the preliminary sizing of externally mounted mission equipment fittings. However, these fittings should be checked for the reserve energy landing condition (par. 4-5.2) if applicable. Landing with side or drag loads produces large angular accelerations that may result in critical loads for certain installations.

4-10.4.2 Load Paths

A most important consideration in the preliminary design of a helicopter is the provision of load paths that will result in maximum stiffness. This is true especially in the case of externally mounted mission equipment.

When an inherently stiff support is used, more flexibility in helicopter utilization is offered for missions where variation in the weight of externally mounted mission equipment is desirable. Adequate stiffness reduces the critical nature of the installation because resonant frequencies are more likely to be above the operating range of the rotor.

This support stiffness can be achieved best by the use of efficient load paths. These load paths should lead into at least two primary load-carrying fuselage structural members with a minimum amount of eccentricity. When inspection capability can be provided, a fail-safe configuration is highly desirable for externally mounted mission equipment such as those weapon pods that operate with relatively high shock and recoil loads.

There are configurations, however, where one load path to a primary structural member is adequate. In all cases, the length of the load path to basic structure should be as short as possible.

The provision of good load paths is a primary function of preliminary design analysis. In the case of cargo loads, there is a large variation in loadings and load locations. Therefore, special study of load paths in the cargo compartments is required.

4-10.4.3 Other Considerations

The preliminary structural analysis becomes an iterative process during the design stage because there are considerations other than loads and fundamental structural design. These include:

1. Functional requirements. The functional operation of a unit of mission equipment may dictate certain aspects of its installation. An external litter, for example, is most useful if located near a door or window. Performance of certain avionic antennas is highly dependent upon location. Special lights also require more than normal consideration.

2. Performance. Helicopter performance never can be minimized, regardless of how vital is the mission equipment being installed. Allowance must be made for the mounting of fairings for reduction of aerodynamic loads. In addition, equipment performance as a function of its specific location on the helicopter must be studied. Certain flight control and electronic components must be located in specific areas where the environment is satisfactory or where they are accessible to the crew. Historical data on similar helicopter installations are valuable for such considerations.

3. Weight. Past results have indicated that inadequate weight control sometimes is exercised on mission equipment provisions. Each fitting and load path must be evaluated on an individual basis with the goal of reducing weight. The evaluation of the specific fitting, of course, must be preceded by evaluation of all other factors that have affected its weight.

4. CG location/weight and balance. Effort should be made to locate weight items in such a manner that a minimum CG shift is effected and that the weight items are as near as possible to the lifting rotor(s). From a crash survival standpoint, the CG should be as near to the ground as possible to minimize rollover.

5. Safety. Safety aspects must be considered during the development of installation provisions for mission equipment. Hazards that may be associated with each unit of mission equipment must be identified and allowed for in the installation design. This study may result in modification of mounting locations and in the installation of additional equipment. Litter installations must be placed so as to provide the maximum comfort and safety to the occupants and to other passengers or crew members. The litters must be fastened securely to a structure of sufficient strength to withstand crash loads without breakaway. Also, the litters should be oriented so that the safety straps securing the patient will be acting in their most efficient direction in case of a crash.

6. Human factors. Human factors must be considered in all preliminary design, but especially with regard to internal cargo loading. The number of alternative loading possibilities is quite large, and the placement of fittings and other provisions must assure the most appropriate location and distribution of the loads.

7. Boresighting. Installations involving weapon pods probably will have requirements for boresighting, which could result in the need for multiple fittings instead of a single fitting.

8. Vibration. The magnitude of the vibrations initiated by mission equipment is an important consideration for flight comfort and, consequently, for mission effectiveness. Careful consideration of the possible vibration spectrum should be a part of the preliminary structural analysis. Historical flight test data and flight evaluation information are good sources for such information; analytical estimates also are useful.

9. Ground and landing clearance. The location of externally mounted mission equipment sometimes is restricted by its proximity to the ground while the helicopter is parked. The preliminary structural analysis should include consideration of maximum deflections for both normal and hard landings. The deflections of the landing gear, the mounting fitting, and the mission equipment assembly should be considered in the analysis.

10. Spare parts and maintenance. All maintenance and support requirements should be sufficiently simple to assure reliable operation and service at the organizational level. The mounting provisions and fittings must lend themselves to ease of inspection and must be accessible for service, repairs, and replacement. The fittings should be protected from possible adverse environmental conditions that might reduce their structural integrity.

11. Yaw stability. Yaw stability considerations are applicable primarily to mission equipment such as weapon pods, where recoil loads from firing and launching could be relatively high. The locations of these assemblies relative to stabilizer surfaces also may be a factor in the yaw stability considerations.

12. Asymmetrical loads. External asymmetrical loads could be experienced when some externally mounted mission equipment is jettisoned. The effect from such loads should be examined.

4-10.4.4 Preliminary Sizing

After the basic loads have been established and accumulated and the tentative locations of the mission equipment installation fittings and other provisions have been determined by preliminary layout, the initial sizing and structural analysis can be made. In most cases, the fitting design at this stage will be based upon limit load and ultimate load values.

The preliminary structural analysis should include several considerations pertaining to the materials from which the fittings will be made. If the fitting is to be made from a casting, for example, there must be an allowance for the casting factors. If a forging is planned, a check should be made to assure that the design being evaluated can be manufactured with an acceptable metallic grain direction.

4-10.4.5 Substantiation

The preliminary sizing of the externally mounted mission equipment fittings can be substantiated by calculating their weights and comparing the results with previously established weight allocations. This procedure provides a quick first-order substantiation. In addition, a thorough review of the load paths must be made to assure that there is a path for each load, and that the path is direct and of minimum length. If these requirements are fulfilled and the preliminary structural analysis indicates adequate strength, the fitting designs are adequate for preliminary design purposes. The margins of safety for critical loading conditions for the fittings for attachment of major items of mission essential equipment *shall* be calculated and tabulated.

4-11 FATIGUE LIFE DETERMINATION

4-11.1 GENERAL

Many helicopter components are subject to alternating loading at fairly high frequencies. The primary source of these loads is the harmonic variation of the aerodynamic loading of a rotary wing in translational flight. The critical loading on these components is not a static load, to be compared with the yield strength, but rather is a fatigue load.

Fatigue strength of a given component can be defined in terms of an endurance limit, or it can be stated in terms of a fatigue life. The endurance limit is the maximum value of alternating stress to which the component can be subjected for an infinite number of cycles without failure. Fatigue life is that number of stress cycles that can be sustained prior to failure. For a given frequency of alternating load, fatigue life is stated in hours rather than cycles.

Following determination that a given component is loaded critically by alternating loads and hence subject to fatigue analysis, the loading must be defined in detail, and the fatigue properties of the component must be determined. A determination then can be made of the fatigue life of the component. Each of these phases of the analysis is discussed in the paragraphs that follow.

During preliminary design it normally is not practicable to determine component fatigue lives because the fatigue properties cannot be defined without component tests and the fatigue loadings cannot be determined without flight tests. It is appropriate, however, to identify those fatigue-critical components (par. 4-11.2) for which the design objective will be infinite life and those for which a finite life will be established. The preliminary stress analysis report *shall* include substantiation of acceptable life characteristics for all rotor system components as a minimum. In these substantiations the properties of the components *shall* be estimated and the critical alternating loads *shall* be developed by rational and conservative methods. The methods for determination of component properties and of applicable fatigue loads *shall* be included in the report.

Although final determination of fatigue lives is a part of the airworthiness qualification of a new helicopter (AMCP 706-203), the methods for these determinations also are discussed in the paragraphs that follow. These methods are applicable for both preliminary design and final qualification. In the latter case, of course, test data will be used and the spectra of applied loads will be more complete.

4-11.2 FATIGUE-CRITICAL COMPONENTS

All components for which a fatigue life determination is required are defined as fatigue-critical components. All rotor system components—blades, grips, hubs, control horns—and all control system components, rotating and nonrotating, between the rotors and a point of load isolation *shall* be fatigue-critical components. All drive shafts *shall* be fatigue-critical components—along with all drive system support structures including engine mounts and main transmission and antitorque rotor gearbox mounts and supports.

A fatigue life determination *shall* be made for all fatigue-critical components. If this determination indicates infinite life, the component no longer need be classified as fatigue-critical for the specific mission loading. A new life determination *shall* be made in case

TABLE 4-10
SUMMARY OF TYPICAL MISSIONS FOR UH CLASS HELICOPTER

	MISSION DESCRIPTION
A	PERSONNEL TRANSPORT
B	FIRE SUPPRESSION
C	INTERNAL CARGO
D	EXTERNAL CARGO
E	MEDICAL EVACUATION
F	TRAINING
G	FERRY
H	SHORT-HAUL VIP TRANSPORT

of any change of fatigue loading upon the component. Such changes may result from changes to the mission gross weight, to a mission profile, or to the mission frequency (anticipated utilization).

4-11.3 FATIGUE LOADINGS

As discussed in par. 4-4, fatigue loadings vary with flight conditions, with the most severe loadings resulting from maneuvers. The frequency of occurrence of a given loading, or the number of occurrences per 100 hr of helicopter operation, is dependent upon the mission profiles of the individual missions assigned, together with the frequency of performance of each mission. The frequency of occurrence distribution is obtained by considering carefully all factors that affect the frequency of performance of each of the applicable flight conditions.

In formulating the frequency of occurrence distribution, the following factors should be considered:

1. Helicopter missions as defined in the applicable specification (see Table 4-10 for a summary of these missions for the example being considered)

2. Frequency of performance of each assigned mission

3. Frequency of occurrence of individual flight conditions through analysis of all missions, primary and alternate. Effects of field environment based upon pertinent operational experience and anticipated deployment should be included.

4. Density altitudes at which each leg of the applicable mission profiles will be flown

5. Practicable gross weight operating ranges, including consideration of usable fuel load and expendable payload.

Using the detail specification as a beginning point, mission profiles and other information about prospective use of the helicopter should be analyzed to establish rough percentages of time in the various modes of flight. The potential helicopter missions *shall* be evaluated for altitudes, and this factor considered in the detailed breakdowns. Once these percentages have been established, each group of maneuvers (in ground effect, level flight, transition, etc.) can be considered individually and a detailed breakdown of flight time calculated for each particular maneuver. Next, each of the individual maneuvers should be considered separately and the specific airspeed breakdown within the maneuver developed. Finally, a careful analysis produces the gross weight and rotor speed breakdowns. Table 4-11 is a sample mission analysis for one of the missions to be considered. Fig. 4-84 presents the mission profile for the mission analyzed in Table 4-11. The same type of profile would be constructed for each of the other assigned missions, and a frequency of occurrence analysis similar to Table 4-11 would be made for each mission.

4-11.3.1 Determination of Composite Maneuver Spectrum

Because flight loads are dependent upon the density altitude at which the flight condition occurs, it is necessary to establish a frequency of occurrence distribution as a function of altitude. (It is acceptable to establish altitude ranges and to determine the frequency of occurrence within each such range.) Table 4-12 is an example of such a distribution. Each of the assigned missions is listed, together with the relative frequency of performance of each in the selected range of density altitude h_d (0-4000 ft h_d in Table 4-12). The relative frequency of each mission is presented as percentage of utilization. Similar distributions for the other two altitude ranges (4000-8000 ft and 8000-12,000 ft) are not shown separately, but their effect is included in Table 4-13, which gives a composite flight condition frequency of occurrence distribution for the helicopter. The effects of each of the eight assigned missions shown in Table 4-10 are included, and the relative frequency of operation in each of the three altitude ranges is shown.

The missions listed in Table 4-10 are considered typical of those that might be considered applicable for a UH class helicopter. Comparable lists can be prepared for other classes of helicopters, based upon mission assignments included in the RFP or in the helicopter

TABLE 4-11
FREQUENCY OF OCCURRENCE FOR MISSION A (PERSONNEL TRANSPORT)

MISSION TIME: 120 min	TIME PER MANEUVER EVENT, sec	BASE TO FIELD NO. OF EVENTS	BASE TO FIELD TOTAL NO. OF min	RETURN TO BASE NO. OF EVENTS	RETURN TO BASE TOTAL NO. OF min	MISSION % TIME
1. HOVER IGE		1	2.00	1	1.16	2.63
2. HOVER OGE		1	0.12	1	0.12	0.20
3. LOITER A/S			27.61			23.00
4. LEVEL FLIGHT 0.6 V_{NE}			1.54		1.54	2.58
5. LEVEL FLIGHT 0.7 V_{NE}			0.30		0.30	0.50
6. CRUISE 0.8 V_{NE}			9.40		7.05	13.72
7. CRUISE 0.9 V_{NE}			18.81		15.57	28.65
8. HIGH-SPEED V_{NE}			4.54		4.54	7.56
9. FLAT PITCH FLIGHT IDLE		1	2.00		0.40	2.00
10 NORMAL START	120	1	2.00			1.67
11. NORMAL SHUTDOWN	60				[illegible].00	0.83
12. IGE TURNS	6	1	[illegible]			0.08
13. IGE CONTROL REVERSALS	1		0.0[illegible]	1	0.02	0.03
14. IGE SIDEWARD FLIGHT	5					
15. IGE REARWARD FLIGHT	3					
16. VTO TO 40 ft AND ACCEL.	6	1	0.10	1	0.10	0.17
17. NORMAL TAKEOFF AND ACCEL.		1	0.30			0.25
18. SLIDE TAKEOFF AND ACCEL.						
19. TE SLIDE-ON LANDING						
20. TE APPR.AND LANDING	[illegible]			1	0.33	0.28
21. SE APPR.AND LANDING	[illegible]					
22. SE APPR.WITH TE RE[illegible]						
23. TE CLIMB		5	4.62	1	0.76	4.46
24. SE CLIMB						
25. ACCEL.CLIM[illegible]S TO CRUISE	24	4	1.60			1.33
26. OGE TURNS	8	12	1.60			1.33
27. OGE CONTROL RE[illegible]SALS	1	2	0.03			0.03
28. CYCLIC PULL-UPS	2	1	0.03			0.03
29. DECEL.TO DESCENT A/S	30	3	1.50	1	0.50	1.67
30. TE DESCENT		5	2.52	1	1.67	3.50
31. SE DESCENT						
32. TE TO SE TRS IN CLIMB	2					
33. TE TO SE TRS IN CRUISE	2					
34. SE TO TE TRANSITION	4					
35. SLING LOAD LIFTOFF						
36. SLING LOAD LANDING						
37. MIN PWR.APPR.-PWR.REC IGE	24			1	0.40	0.33
38. FIRE SUPPRESSION PUSHOVER	2	6	0.20			0.17
39. FIRE SUPPRESSION DIVE	14	6	1.40			1.17
40. FIRE SUPPRESSION PULL-UP	10	6	1.00			0.83
41. FIRE SUPPRESSION HIGH "g" TURNS	6	6	1.20			1.00
TOTAL						100.00

EXAMPLE

NOTES.

TAKEOFF GROSS WEIGHT	- 10,000 lb	V_{NE}	VELOCITY NEVER TO BE EXCEEDED
LANDING GROSS WEIGHT	- 8,850 lb	IGE	IN GROUND EFFECT
APPROXIMATE MISSION RADIUS	- 60 mi	OGE	OUT OF GROUND EFFECT
CRUISE ALTITUDE	- 2,000 ft ABOVE TERRAIN	A/S	AIRSPEED
TAKEOFF ALTITUDE RANGE	- 0 TO 4000 ft	TRS	TRANSITION
TE	- TWIN ENGINE	REC	RECOVERY
SE	- SINGLE ENGINE	VTO	VERTICAL TAKEOFF

detail specification together with secondary missions such as ferry and training.

In any case the procedure used in developing the composite maneuver spectrum is the same:

1. Develop a typical profile for each applicable mission.
2. List the flight conditions and maneuvers required during completion of the profiled missions.
3. Estimate the time per condition or event.
4. Estimate the number of occurrences of each condition and event per mission.
5. Accumulate times for each condition and/or event and convert to percentage of mission time.
6. Estimate relative frequency of performance of assigned missions as percentage of total time.
7. Estimate relative frequency of performance of assigned missions at specified altitude or range of altitudes as percentage of total time at altitude.
8. Accumulate percentage of total time at specified altitude or range of altitudes, using percentage of total time for each type of mission (Item 6) as a weighting factor.
9. Accumulate percentage of total time for each occurrence or event at specified altitude or range of altitudes, using percentage time at altitude for each mission (Item 7) as a weighting factor.
10. Accumulate composite percentage of total time for each occurrence or event, using percentage total time at each specified altitude or range of altitudes (Item 8) as a weighting factor.

This procedure was employed to develop the example spectrum given by Tables 4-11, 4-12, and 4-13. This composite spectrum must be distributed further by mission gross weight and rotor speed to develop the fatigue loading spectrum that will be used in fatigue life determinations.

4-11.3.2 Determination of Gross Weight and Rotor Speed Distributions

The frequency of occurrence distributions of mission gross weight result from analysis of each of the applicable mission profiles to determine takeoff gross weight, fuel consumption rate, mission duration, and amount and rate of depletion of expendable payload.

For the example missions of Table 4-10, this analysis is presented in Fig. 4-85. This figure shows the variation of gross weight for each of the eight missions in the form of the cumulative percentage of total mission time at or below a given gross weight. By use of the percentage total time for each mission as the weighting factor, a single, composite relationship of cumulative time at or below a given gross weight can be developed.

For the example case, three gross weight ranges—7500-8200 lb, 8200-9500 lb, and 9500-10,500 lb—were selected. From the composite variation of mission gross weight with time, the percentage total time for occurrence of, or operation in, each gross weight range was determined. These percentages are shown in Table 4-14. As with the effect of altitude, acceptable alternative procedures would use either specific values or more—and hence narrower—ranges of values of operating gross weight.

The final parameter to be considered in the development of the fatigue loading spectrum is the rotor speed. The frequency of occurrence of rotor speeds less than, equal to, and greater than the design value must be estimated. The anticipated use of the helicopter, operational experience with other helicopters with similar mission assignments, and the characteristics of the rotor speed control system (engine governor and control) must be considered when making this estimate. In the case of the utility class helicopter being discussed, an estimate was made for the distribution of rotor speeds above and below the design value during level flight conditions. It was assumed that maneuvers would be

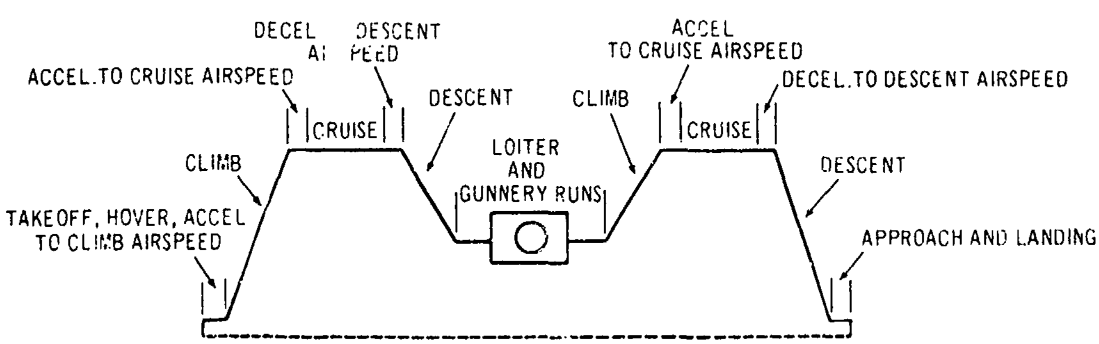


Fig. 4-84. Schematic Mission Profile—Personnel Transport Mission

TABLE 4-12
FREQUENCY OF OCCURRENCE—ALL MISSIONS

MANEUVER	A	B	C	D	E	F	G	H
% UTILIZATION	31.00	37.00	14.00	2.00	5.00	2.00	1.00	8.00
	% TIME							
1. LOITER A/S	23.00	30.00	1.00	0.50	0.49	5.44	0	0.42
2. LEVEL FLIGHT 0.6 V_{NE}	2.58	0.58	2.00	1.00	0.98	3.71	0	1.12
3. LEVEL FLIGHT 0.7 V_{NE}	0.50	2.00	3.50	2.43	2.73	6.67	8.33	1.96
4. CRUISE 0.8 V_{NE}	13.72	7.61	9.60	10.43	7.84	8.68	60.00	11.19
5. CRUISE 0.9 V_{NE}	28.65	29.70	36.43	31.67	17.15	5.21	12.44	34.13
6. HIGH-SPEED V_{NE}	7.56	2.49	11.38	2.43	39.08	3.33	7.00	3.08
7. IGE HOVER	2.63	2.10	8.51	7.45	1.73	4.73	1.45	1.40
8. OGE HOVER	0.20	0.03	0.50	2.48	0.22	0.17	0	0
9. FLAT PITCH/FLIGHT IDLE	2.00	2.00	0	0	12.26	2.00	0	11.29
10. NORMAL START	1.67	1.33	6.15	4.76	1.95	3.33	1.11	5.60
11. NORMAL SHUTDOWN	0.83	0.66	3.08	2.38	0.98	[illegible]	0.56	2.81
12. IGE TURNS	0.08	0.07	0.31	0.71	0.59	[illegible]	0.06	1.12
13. IGE CONTROL REVERSALS	0.03	0.03	0.08	0.26	[illegible]	0.55	[illegible]	0.20
14. IGE SIDEWARD FLIGHT	0	0	0.13	0.40	[illegible]	0.28	0	0.70
15. IGE REARWARD FLIGHT	0	0	0.08	0	[illegible]	[illegible]	0	0.14
16. VTO TO 40 ft AND ACCEL.	0.17	0.07	0.31	[illegible]	[illegible]	0.17	0	0.28
17. NORMAL TAKEOFF AND ACCEL.	0.25	0.20	[illegible]	0.71	[illegible]	3.46	0.17	0.84
18. SLIDE TAKEOFF AND ACCEL.	0	0	[illegible]	0	0	0.04	0	0
19. TE SLIDE-ON LANDING	0	0	[illegible]	0	0	0.05	0	0
20. TE APPR. AND LANDING	0.28	[illegible]	[illegible]	[illegible]	0.98	3.28	0.18	1.87
21. SE APPR. AND LANDING	0	0	0	0	0	0.16	0	0
22. SE APPR. WITH TE REC IGE	0	0	0	0	0	0.40	0	0
23. TE CLIMB	4.46	3.59	4.74	8.21	3.50	5.37	3.43	6.33
24. SE CLIMB	0	0	0	0	0	1.79	0	0
25. ACCEL. CLIMB A/S TO CRUISE	1.33	1.87	1.23	2.86	1.17	2.05	0.28	2.24
26. OGE TURNS	1.33	3.53	1.03	3.79	0.52	3.44	0.29	1.87
27. OGE CONTROL REVERSALS	0.03	0.02	0.05	0.21	0.12	0.55	0.01	0.20
28. CYCLIC PULL-UPS	0.03	0.02	0	0.07	0	0.55	0.02	0
29. DECEL TO DESCENT A/S	1.67	2.33	1.54	3.57	1.47	2.50	0.22	2.80
30. TE DESCENT	3.50	5.01	5.18	8.37	3.92	6.72	4.44	8.41
31. SE DESCENT	0	0	0	0	0	2.55	0	0
32. TE TO SE TRANSITION IN CLIMB	0	0	0	0	0	2.55	0	0
33. TE TO SE TRANSITION IN CRUISE	0	0	0	0	0	0.05	0	0
34. SE TO TE TRANSITION	0	0	0	0	0	0.22	0	0
35. SLING LOAD LIFT OFF	0	0	0	0.48	0	0.66	0	0
36. SLING LOAD LANDING	0	0	0	0.48	0	0.66	0	0
37. MIN PWR. APPR.-PWR. REC IGE	0.33	0.27	1.23	2.86	1.17	0.67	0	0
38. FIRE SUPPRESSION PUSHOVER	0.17	0.27	0	0	0	1.11	0	0
39. FIRE SUPPRESSION DIVE	1.17	1.87	0	0	0	7.78	0	0
40. FIRE SUPPRESSION PULL-UP	0.83	1.33	0	0	0	5.55	0	0
41. FIRE SUPPRESSION HIGH "g" TURNS	1.00	0.80	0	0	0	2.50	0	0

NOTE: V_{NE} FOR MISSION D IS THE V_{NE} FOR SLING LOAD AND IS NOT THE SAME V_{NE} USED FOR OTHER MISSIONS.

TABLE 4-13
COMPOSITE MANEUVER SPECTRUM

SE = SINGLE ENGINE
TE = TWIN ENGINE

MANEUVER	DENSITY ALTITUDE (ft X 10⁻³) 0 TO 4	4 TO 8	8 TO 12	
	% TIME			
	40.0	50.0	10.0	COMPOSITE
1. LOITER A S	18.55	17.76	17.51	18.05
2. LEVEL FLIGHT 0.6 V_{NE}	1.53	2.88	2.71	2.32
3. LEVEL FLIGHT 0.7 V_{NE}	1.94	1.93	1.78	1.92
4. CRUISE 0.8 V_{NE}	10.68	6.01	4.02	7.68
5. CRUISE 0.9 V_{NE}	29.42	17.71	12.23	21.85
6. HIGH-SPEED $V_{...}$	7.24	23.07	31.11	17.54
7. IGE HOVER	3.24	**	**	**
8. OGE HOVER	0.21			
9. FLAT PITCH FLIGHT IDLE	2.92			
10. NORMAL START	2.59			
11. NORMAL SHUTDOWN	1.29			
12. IGE TURNS	0.26			
13. IGE CONTROL REVERSALS	0.07			
14. IGE SIDEWARD FLIGHT	0.10			
15. IGE REARWARD FLIGHT	0.03			
16. VTO TO 40 ft AND ACCEL	[illegible]			
17. NORMAL TAKEOFF AND ACCEL.	0.37			
18. SLIDE TAKEOFF AND ACCEL.	[illegible]			
19. TE SLIDE-ON LANDING	0.11			
20. TE APPROACH AND LANDING	0.48			
21. SE APPROACH AND LANDING	0.01			
22. SE APPR. WITH TE REC. IGE	0.01			
23. TE CLIMB	4.36			
24. SE CLIMB	0.04			
25. ACCEL.CLIMB A/S TO CRUISE	1.62			
26. OGE TURNS	2.19			
27. OGE CONTROL REVERSALS	0.06			
28. CYCLIC PULL-UPS	0.03			
29. DECEL.TO DESCENT A/S	2.02			
30. TE DESCENT	4.84			
31. SE DESCENT	0.05			
32. TE TO SE TRANSITION IN CLIMB	0.01			
33. TE TO SE TRANSITION IN CRUISE	0.01			
34. SE TO TE TRANSITION	0.01			
35. SLING LOAD LIFTOFF	0.02			
36. SLING LOAD LANDING	0.02			
37. MIN.PWR. APPR.-PWR. REC.IGE	0.50			
38. FIRE SUPPRESSION PUSHOVER	0.17			
39. FIRE SUPPRESSION DIVE	1.21			
40. FIRE SUPPRESSION PULL-UP	0.86			
41. FIRE SUPPRESSION HIGH "g" TURNS	0.66			
TOTAL 100%				

*THIS COLUMN IS THE COMPOSITE SPECTRUM TO BE USED IN LIFE DETERMINATION.
** THE VALUES OF No. 7 THROUGH No. 41 ARE IDENTICAL TO THOSE LISTED FOR 0 TO 4000 ft.

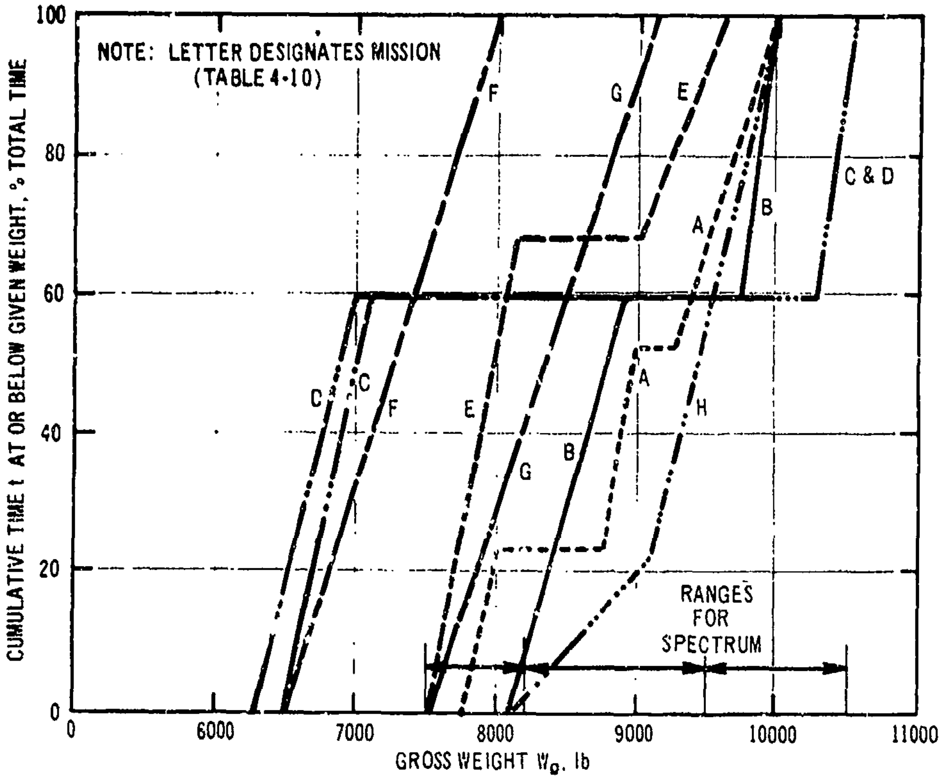


Fig. 4-85. Mission Gross Weight Variation With Time

TABLE 4-14
GROSS WEIGHT, ALTITUDE, AND ROTOR SPEED DISTRIBUTIONS

		LEVEL FLIGHT				MANEUVERS
GROSS WEIGHT RANGE	DENSITY ALTITUDE RANGE	% AT W_g	% AT ALTITUDE	% OF rpm LOW	% OF rpm HIGH	W_g (MID-RANGE rpm)
7500 TO 8200 lb	0 TO 4000 ft	20	40	10	90	20
8200 TO 9500 lb	4000 TO 8000 ft	35	50	10	90	35
9500 TO 10,500 lb	8000 TO 12,000 ft	45	10	10	90	45

performed with the rotor speed maintained at the design value.

The gross weight, altitude, and rotor speed distributions are shown in Table 4-14 as percentages of total time. These values then are applied to the flight spectrum for use in fatigue life calculations.

Determination of the loads on each fatigue-critical component ultimately will be accomplished during the flight load survey (AMCP 706-203). During preliminary design it is necessary to use calculated loads. It also is appropriate to select a minimum number of flight conditions as representative of the more complete maneuver occurrence tables required for test purposes. Experience with helicopters performing similar missions is usually a sufficient basis upon which to select maneuver and flight conditions that will be critical for the individual fatigue-critical components.

The fatigue loading spectrum *shall* be prepared by the contractor and approved by the procuring activity. This spectrum must be developed carefully. It must

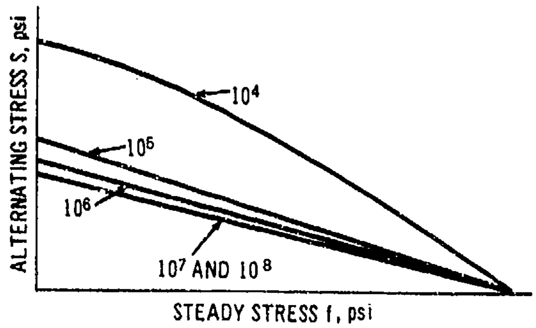


Fig. 4-86. Goodman Diagram

represent the proposed utilization of the helicopter realistically with respect both to frequency of performance of the assigned missions and to the maneuvers and flight conditions required during the performance of each of the missions.

4-11.4 FATIGUE PROPERTIES

Fatigue properties for use in preliminary design are based mainly on small specimen test results. Small specimen fatigue data are compiled into a constant-life fatigue diagram (also known as a Goodman diagram) (Fig. 4-86). This diagram presents mean results for small specimens for fixed numbers of cycles of vibratory stress to failure, defining the permissible vibratory stresses for all applied steady stresses from zero to the ultimate tensile stress of the material. Compressive steady stresses normally are not applicable to rotor system design, and the effects of compressive stress upon the constant-life diagrams may be ignored.

An S-N diagram is constructued from the constant-life fatigue diagram for each steady stress under consideration. A vertical line drawn at a given value of steady stress on the Goodman diagram intersects each constant-life line at the value of vibratory stress corresponding to that number of cycles (Fig. 4-87). Cross-plotting in this fashion produces the mean S-N diagram for the steady-stress levels of interest for smooth, small specimens. Each point on these curves represents a combination of steady stress, vibratory stress, and number of cycles for which 50% of the specimens will survive and 50% will fracture.

For design purposes, it is necessary to reduce the vibratory stress level to eliminate most of the failures. Therefore, it is necessary to describe the nature of the scatter experienced in fatigue testing of the material, i.e., to determine or select a type of statistical distribution of the test points. In many instances, a normal distribution adequately describes the scatter, sometimes a log-normal or a Weibull distribution is applicable. Numerous tests of homogeneous material usually will describe a smooth curve. The acceptable level of risk must be determined and the mean curve reduced by an amount that will exclude all but the number of test points representing that risk. As an example, it is common practice to use a normal distribution and to compute the standard deviation σ of the failing vibratory stress at a constant number of cycles and steady stress. This deviation usually is between 10% and 15% of the mean value of vibratory stress at failure. The mean curve then is reduced by three standard deviations, which excludes all but about one point in 800. The limiting value of structural reliability is then 799/800 or 0.9987 for a single component, which is an adequate level of risk for most applications. If the lowest test point remains below the adjusted curve, i.e., lies more than three σ below the mean curve, consideration must be given to passing the adjusted curve through the lowest test point. When the data being used are component test data, it becomes more important that the adjusted curve lie below the lowest test point.

A full-size component is more likely to fail than a small test specimen because it has a larger volume of stressed material. In essence, the full-size component can be thought of as a large number of small specimens bunched together; the failure of any one will initiate the failure of the entire component. Appropriate factors for reducing the S-N curves to account for size effect can be determined by comparing full-size and small specimen fatigue test results, or from statistical studies such as Ref. 64.

The smoothness of the surface, the degree to which it may have been work-hardened, the presence of shot-peening, and the amount of protection against corrosion, erosion, or nicks and scratches in service all influence the allowable vibratory stress.

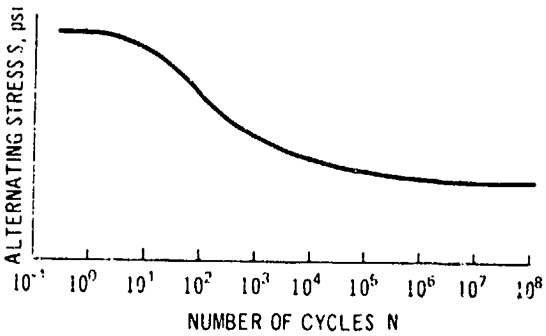


Fig. 4-87. Typical S-N Curve

The working S-N curve—which incorporates reductions for scatter, size, and surface—probably will have ordinates between 25% and 50% of those of the mean curve based on small specimens. The complete S-N diagram (for any combination of steady stress, scatter, size, and surface) starts at 1/4 cycle, which defines a failure in terms of the application of the component. As an example, a statically loaded member, in an application in which deformation is unimportant, does not fail until it is fractured. In contrast, a machine member whose exact size and shape are important, or which is to be loaded repeatedly, never must be allowed to yield, and the occurrence of an induced stress in excess of the elastic limit signifies a failure. For rotor blade analysis, the value of the ordinate at 1/4 cycle should not be higher than yield stress $F_{ty}/1.15$ or ultimate stress $F_{tu}/1.50$, whichever is less. The tangent at 1/4 cycle approaches the horizontal because the second 1/4 cycle cannot do more fatigue damage than the first 1/4 cycle did. The curve will be concave downward at the left end, and concave upward on the right. The point of inflection is usually between 10^2 and 10^6 cycles, and is influenced by the manner in which the working curve has been reduced from the mean curve. Beyond 10^7 cycles, the curve flattens so much that for many materials it can be considered to have a constant ordinate, which is called the "endurance limit". For those materials whose S-N curves have a noticeable slope beyond 10^7 cycles, the ordinate at 10^8 cycles can be considered as an endurance limit for preliminary design purposes.

The curves for aluminum alloys, for example, have a gradual slope that continues past 10^8 cycles (see Fig. 4-88) whereas those for steel alloys are steeper initially but flatten at about 10^7 cycles. In Fig. 4-88 the S-N curves have been made nondimensional by plotting the ratio of failure stress S to endurance limit stress E.

The significance of the shapes of the S-N curves can be seen in the following example. If an aluminum part is sized so that the stresses produced by loads that occur 10^8 times during its useful life can be supported without failure, then loads of twice that magnitude can be carried for 2.5×10^5 cycles. In contrast, a steel part sized for loads that occur 10^8 times during its life will take loads of twice that magnitude only for 6×10^4 cycles; thus, if the parts are sized so that the most frequent loading produces stresses at the endurance level, an aluminum part will withstand more cycles at load levels above its endurance limit than will a steel part.

An aluminum part designed for infinite life based on the stress levels of high-speed level flight in a medium-sized utility helicopter will absorb the damaging loads caused by maneuvers and still have a useful component life (at least 1000 hr). For a similar part in steel, however, the same loads must be multiplied by a factor of the order of 1.4 to compensate for the more rapid accumulation of fatigue damage in steel at stress levels above endurance. Comparable factors may be used to adjust for a flight spectrum in which a larger proportion of the time is spent in maneuvers, as for an attack helicopter. The magnitude of these factors can be found by a statistical analysis of flight load data for previous designs. Mission profile and fatigue analysis are discussed in par. 4-4.1.2 and par. 4-11.3.

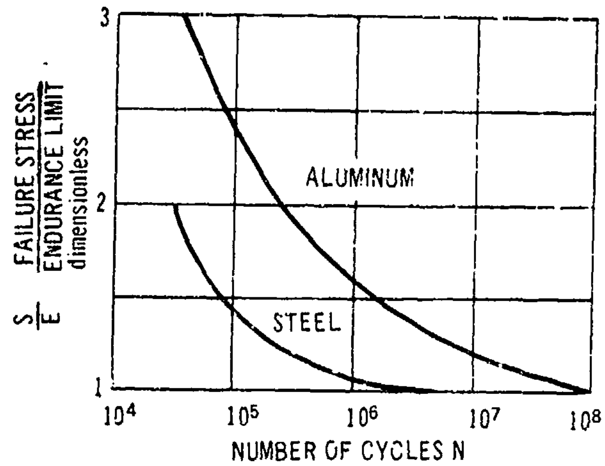


Fig. 4-88. S-N Curve Shapes for Steel and Aluminum

The fatigue properties, including endurance limit, for critical helicopter components ultimately must be determined by test either of actual components or of appropriate material specimens. Component fatigue testing is discussed in Chapter 7, AMCP 706-203. The S-N curves derived from actual component test data *shall* be reduced for scatter using a minimum of three standard deviations. If sufficient specimens are not tested to failure at the same vibratory stress level to permit definition of a standard deviation, the test curve *shall* be reduced by a minimum of 25% to obtain a working curve to be used in fatigue life determination. Analytical substantiation of infinite life, based upon small specimen endurance limit data, is discussed in par. 4-11.8.

4-11.5 FATIGUE LIFE

An engineering approach to fatigue life prediction is the "cumulative damage hypothesis" or Miner's Rule (Ref. 65). It is explained by reference to Fig. 4-89.

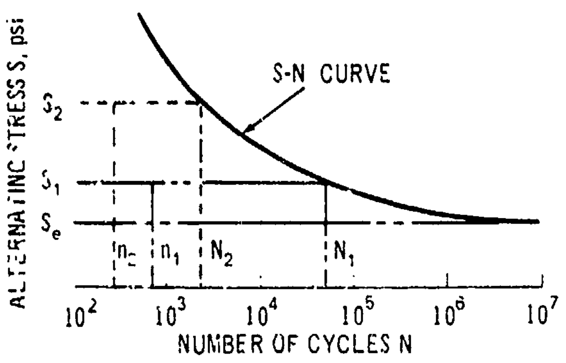


Fig. 4-89. Example S-N Curve

The theory is that component failure will occur when N_1 cycles of a constant alternating stress equal to S_1 have been applied. If the part in question is exposed to alternating stresses of unequal magnitude, the fatigue damage at each different stress level is dependent upon the number of cycles of stress applied versus the capability of the part at that stress level. Hence, if n_1 cycles of stress equal to S_1 are applied, the fatigue damage is equal to n_1/N_1. Likewise, if n_2 cycles of stress equal to S_2 are applied, the fatigue damage is equal to n_2/N_2. Part failure is anticipated when the summation of $n_1/N_1 + n_2/N_2 + n_3/N_3 + \ldots$ equals unity. Hence, Miner's hypothesis states that fatigue failures occur when

$$\sum \frac{N_i}{M_i} = 1.0 \qquad (4\text{-}74)$$

where

N_i = number of cycles of operation at a specified stress level

M_i = abscissa of the working S-N curve at the specified stress level

The way in which the calculation customarily is organized is shown in a more realistic example. The operating conditions are listed in Table 4-15. For each condition, the steady and vibratory stresses at a critical location on the blade spar are given, along with the number of elapsed cycles of loading at these stresses per 100 hr of operation of the helicopter.

The effect of repeatedly starting and stopping the rotor is to add a small number of cycles of relatively high stresses, which have been found to result in fatigue damage to some components. The most obvious stress cycle is due to applying and removing steady flight stress once per flight. Additional stress cycles, due to accelerating and decelerating a complex dynamic system, are predicted best on the basis of stress surveys of similar helicopters.

For the example shown, a simplified, schematic stress history of rotor start and stop also is shown (see Fig. 4-90). The portions of the stress history are broken down into cycles (a, b, c, and d) of various steady and vibratory stress levels. Not all of the loading conditions shown will apply to all components.

Cycle (a) is due to the response of the blades to the starting impulse, Cycle (b) represents running through a resonant frequency, Cycle (c) is the basic application and removal of steady flight stress, and Cycle (d) represents the blades hitting the damper stops. These stress cycles are shown in Table 4-15 under the heading "Starting and Stopping". For this portion of the table, the lower steady stress level requires the use of a different S-N curve than is applicable for the normal rotor speed portion of the loading spectrum.

The abscissa of the S-N curve for each stress is taken from the appropriate S-N diagram and entered under "allowable cycles". The ratio of elapsed cycles to allowable cycles is the "damage", or fraction of fatigue life used up. In this example only Cycle (c) results in any fatigue damage. The conditions tabulated are for 100 hr of typical operation. The calculated fatigue life is obtained from the following proportion, in which the fatigue life is the length of hours for which damage = 1.00:

$$\frac{\text{calculated fatigue life (hours)}}{100 \text{ hr}} = \frac{1.00 \text{ damage}}{\text{damage in } 100 \text{ hr}} \qquad (4\text{-}75)$$

An example calculation is shown in Table 4-15.

Two important points can be noted from the example. First, the conditions that will occur for large amounts of time must produce stresses below the endurance limit. Examples of this are cruise and steady hover for this mission. Second, the life calculation is dominated by the high-load-factor condition, which accounts for roughly 40% of the fatigue damage. To improve fatigue life, it is necessary to reduce the load factor, reduce the stresses associated with the load factor, or reduce the number of cycles at this condition. Because the design load factor and the amount of time (number of cycles) at that condition are determined by the assigned missions and their frequency of performance, apparently the only approach available to the designer is the reduction of the vibratory stress level, which requires a redesign. However, prior to undertaking to redesign either the system or the component, it

TABLE 4-15
FATIGUE LIFE CALCULATION

OPERATING CONDITION (200 rpm EXCEPT AS NOTED)	STEADY STRESS, Ksi	VIBRATORY STRESS, ± Ksi	ELAPSED TIME, hr	ELAPSED CYCLES, N	ALLOWABLE CYCLES, M	DAMAGE, N/M
STARTING AND STOPPING:						
(a) INITIAL ACCEL.	0	5.0	—	8×10^1	~	—
(b) RUNUP	10	5.0	—	16×10^1	~	—
(c) STEADY FLT. STRESS	10	10.0	—	8×10^1	1.0×10^7	0.000008
(d) STOPPING	0	5.0	—	8×10^1	~	—
WARMUP AND TAKEOFF	10	7.5	2.50	3×10^4	~	—
HOVER-STEADY	20	5.8	21.40	2.6×10^5	~	—
HOVER-TURNS AND REVERSALS	20	8.0	4.60	4.5×10^4	1.8×10^8	0.000250
CRUISE	20	7.5	64.30	7.7×10^5	~	—
MANEUVERS AND GUSTS:						
L.F. = 1.13	20	7.6	5.00	5.0×10^4	~	—
1.38	20	8.0	1.25	1.5×10^4	1.8×10^8	0.000080
1.88	20	10.0	0.42	5.0×10^3	4.5×10^6	0.001100
2.13	20	12.2	0.25	3.0×10^3	5.0×10^5	0.006000
2.50	20	13.3	0.16	1.9×10^3	2.3×10^5	0.008300
2.75	20	14.5	0.10	1.2×10^3	1.2×10^5	0.010000
FLARE, 220 rpm	24	9.0	0.02	2.6×10^2	5.0×10^6	0.000050
TOTAL DAMAGE IN 100 hr = 0.025788						0.025788
CALCULATED FATIGUE LIFE $= \dfrac{100}{0.0258} = 3880$ hr						

usually is appropriate to re-examine the measured flight load data. The data included in Table 4-15 are reduced from the flight data with the assumption that the maximum values of steady and oscillatory load occur throughout the entire maneuver, for the number of cycles actually recorded during the maneuver (Chapter 8, AMCP 706-203). This method of data reduction is preferred because it is known to be conservative. By reanalyzing the flight data to reflect more accurately the actual number of cycles at each level of damaging stress, a new fatigue life can be calculated. If this less conservative fatigue life also is unacceptably low, the part must be redesigned or flight restrictions imposed to preclude operations that will result in early fatigue failure.

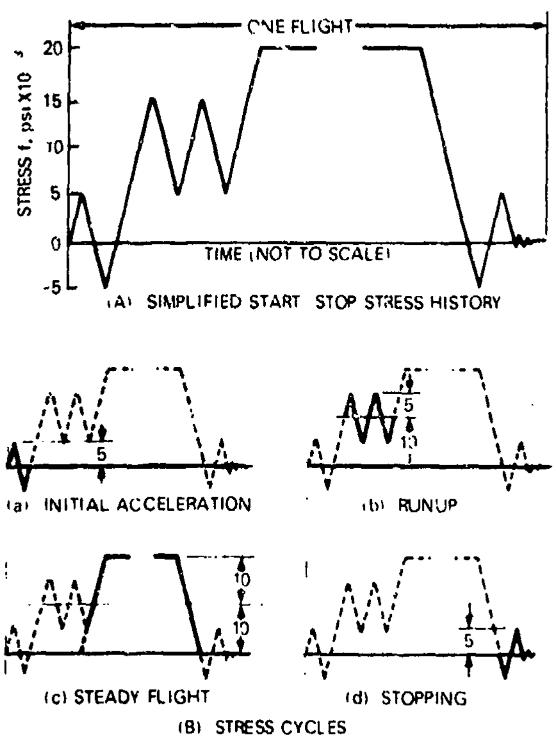


Fig. 4-90. Ground-air-ground Cycles

4-11.6 SERVICE LIFE CALCULATION

Service life, or retirement time, for fatigue-critical components for which a finite fatigue life is calculated often is specified as a period less than the calculated life in order to reduce further the probability of failure in service. For example, for those components for which the calculated fatigue life is 4000 hr or less, the service life may be taken as 75% of the calculated life, while for calculated lives greater than 4000 hr, the service life may be taken as 2000 hr plus 37.5% of the calculated life. The service life for each fatigue critical component *shall* be approved by the procuring activity, based upon the fatigue life determined by the contractor.

4-11.7 SPECTRUM TESTING

The fatigue life determinations discussed in par. 4-11.5 rely upon the linear cumulative damage hypothesis and upon the development of S-N curves applicable to the individual component. These S-N curves are developed by subjecting samples, whether simple material specimens or actual structural components, to appropriate combinations of steady and alternating stresses. The steady stress must be essentially the same for all test points defining a given S-N curve and the alternating stress is a constant amplitude, or value, for each sample tested.

An alternate type of fatigue test applies varying amplitudes of alternating stress, combined with the appropriate steady stress level. This spectrum testing attempts to reproduce the loadings to which the component will be subjected in service. Fatigue life determination then is based upon the number of times the specimen withstands the repetition of the complete load spectrum prior to failure.

Two different types of loading spectra are possible. The block loading spectrum consists of grouping separately the loadings representative of a large number of flights. This type of testing would be most appropriate to components subjected to relatively simple loading (amplitude of alternating stress not greatly dependent upon severity of maneuver) but with a large variation in steady stress level. For example, this type of testing was applied to the tension-torsion wire pack for the AH-56A. The unit block represented approximately 5% of the anticipated life of the unit. The spectrum consisted of repetition of the alternating loadings encountered at normal rotor speed when the steady stress (due primarily to centrifugal force) was at its highest level. The number of cycles of start-stop loading corresponding to the same period of time, or number of flights, then were grouped together at the end of the block.

The flight-by-flight spectrum consists of the variations of both steady and alternating stresses representative of operational experience. Higher loads encountered less frequently than once per flight are introduced periodically in order to include their effects in terms of both amplitude and frequency. The flight-by-flight spectrum is applicable to those components whose loadings and/or load paths are complex. This type of loading produces at various points in the specimen the full range of amplitudes and combinations of bending and torsional stresses that represent the operational environment. Components such as rotor blade retentions may be tested in this manner.

If the test loading spectra are based upon flight test data for the helicopter, with loading frequencies based upon the proposed utilization, spectrum testing can approach a laboratory reproduction of operational experience. Successful completion of a given number of blocks or flights prior to failure then would be directly representative of component fatigue life. However, application of an appropriate life reduction factor to account for scatter is required. Adequate methods for determination of this factor are not available in the literature. Therefore, the factor and the component fatigue life must be determined and substantiated case by case.

When the test loading spectra are prepared from calculated loads, the determination of fatigue life becomes more difficult. If, when flight test data become available, the test spectrum is found to be generally similar to that based upon flight test data, a modification of the life reduction factor must be established to permit determination of component life with statistical reliability comparable to that for a spectrum based upon a flight load survey. However, if the test spectrum is not comparable directly to that based upon flight test data, further analysis of the data is required.

For the case where the applied load spectrum is not comparable directly to the flight load survey spectrum, the fatigue life determination is made by linear cumulative damage calculations (Ref. 66). The number of test cycles applied first is reduced by the applicable test life reduction factor. An S-N curve then is developed for the component by determining the value of stress concentration factor, or notch factor K_T, which, when applied to the small specimen S-N curve for the component material, will predict exactly the reduced life using cumulative damage calculations. This curve then is used in conjunction with the loading spectrum defined by the flight load survey to determine the fatigue life of the component. This determination would use the method described in par. 4-11.5.

If, at a later date, the fatigue load spectrum for a component changes—as a result either of altered mission profile or frequency of occurrence, or of helicopter weight growth or flight envelope change—the effect upon fatigue life must be determined. When the component S-N curve has been determined by test, the fatigue life for the new spectrum can be determined readily by the method described in par. 4-11.5. If the component was subjected originally to spectrum tests, the new fatigue life determination requires either new tests with an amended spectrum or the derivation of an applicable S-N curve by the means described by Ref. 66. Experience with derived S-N curves is limited and the reliability of fatigue life determination based upon them is not widely accepted. For this reason spectrum testing of fatigue-critical components *shall* not be used without prior approval of the procuring activity.

4-11.8 INFINITE FATIGUE LIFE

The fatigue life of a component is infinite if all alternating stresses are below the endurance limit. For components such as control system parts for which stiffness is a primary design criterion, infinite life may be achieved without compromise. Infinite life may be demonstrated without test in certain cases.

The Goodman diagram can be used to demonstrate infinite life. Fig. 4-86 includes an endurance limit diagram in the form of the 10^7-10^8 cycle line. This diagram was prepared from small sample data and, therefore, represents the mean endurance level for the given material. By application of a factor for data scatter, an ideal endurance level diagram can be obtained. Reduction of this limit by a stress concentration factor K_T appropriate for the configuration of the component and the imposed loads results in an endurance level appropriate for the component under evaluation, considering design configuration, and manufacturing processes, including surface treatment (Fig. 4-91). Reduction of the resulting component endurance limit by 67% provides a working curve acceptable for fatigue life calculation.

If the maximum measured fatigue stress falls below this operating curve (Point A), infinite life is demonstrated, and fatigue testing is not required. When the maximum measured fatigue stress lies above the operating curve (Point B), the part *shall* be subjected to fatigue testing. If the S-N curve developed as a result of this testing demonstrates an endurance limit that is higher than the maximum measured vibratory stress, infinite life shall have been demonstrated.

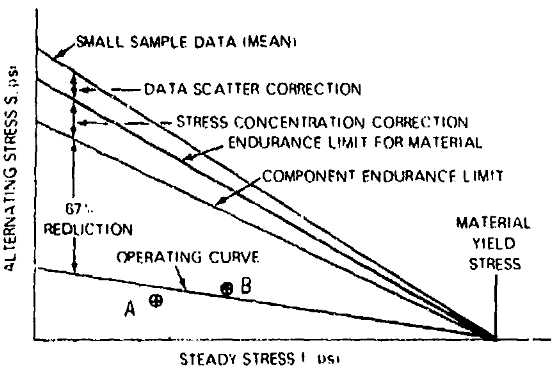


Fig. 4-91. Analytical Demonstration of Infinite Fatigue Life

4-12 LIST OF SYMBOLS

A = cross-sectional area, in.^{2}
AE = absorbed energy, ft-lb
a = linear acceleration, ft/sec^2
= panel length, in.
a_0 = rotor coning angle, rad or deg
B = rotor blade tip loss factor, dimensionless
b = number of blades
= panel width, in.
C = rotational damping constant, lb-sec-in./rad
C_D = drag coefficient, dimensionless
C_L = lift coefficient, dimensionless
$\bar{C}_L$ = mean blade lift coefficient, dimensionless
$C_{L_{max}}$ = maximum mean rotor lift coefficient, dimensionless
C_{L_t} = trim one g mean rotor lift coefficient, dimensionless
C_T = rotor thrust coefficient, dimensionless
c = blade section chord, ft
= damping constant, lb-sec/in.
c_e = blade chord (effective), ft
D = drag force, lb
= dissipative energy, ft-lb
E = endurance limit, psi
= Young's modulus of elasticity, psi
F = force, lb
= horizontal wind load, lb
$F_{c_{cr}}$ = compressive stress critical for buckling, psi
F_R = resultant force, lb
F_x, F_y, F_z = force components along coordinate axes, lb
F_{tu} = ultimate stress, psi
F_{ty} = yield stress, psi
f = stress level, psi
g = acceleration due to gravity, 32.2 ft/sec^2 or 386.4 in./sec^2
H = side force, lb
h = altitude, ft
I = mass moment of inertia, lb-in.-sec^2 or slug-ft^2
= pitching moment of inertia, lb-in.-sec^2 or slug-ft^2

I_p = polar mass moment of inertia of rotor, slug-ft^2

I_{RD} = single rotor/drive system mass moment of inertia, slug-ft^2

I_{total} = total rotor/drive system mass moment of inertia, slug-ft^2

K = empirical factor, dimensionless
= energy absorber efficiency, dimensionless
= rotational spring constant, lb-in./rad or lb-ft/rad

K_c = constant, dimensionless

K_T = stress concentration factor (notch), dimensionless

KE = kinetic energy, ft-lb

k = spring constant, lb/in.

L = lift, lb
= rotor lift ratio, dimensionless

M = moment, lb-ft or lb-in.
= abscissa of S-N curve

M_c = inplane moment, lb-ft

M_T = drive torque, lb-ft

M_θ = pitching moment, lb-in.

MS = margin of safety, dimensionless

m = mass, general, lb-sec^2/in. or lb-sec^2/ft

N = allowable number of cycles from S-N data
= rotor speed or engine output speed, rpm
= number of blades in the rotor

N^* = flight rotor speed, rpm

N_i = number of cycles of operation

N_o = ground idle rotor speed, rpm

n = exponent, dimensionless
= load factor, dimensionless
= number of cycles applied at a specific stress level in a loading spectrum

n_j = drop test load factor, dimensionless

n_s = lateral load factor, dimensionless

P = applied load, lb
= power, hp

$\bar{P}$ = cubic mean load, hp

P_i = forces applied at station i, lb

PE = potential energy, ft-lb

Q = torque, lb-ft or lb-in.

Q_{av} = average torque acting on the rotor/drive system, lb-ft

Q_i = generalized displacement, dimensionless

$Q_{R/D}$ = torque of single rotor/drive system, lb-ft

Q_s = generalized forces, lb

QCP = ratio of aerodynamic rotor torque at minimum collective pitch and 100% rotor speed to $\bar{T}$, dimensionless

$QCP_{R/D}$ = QCP of single rotor/drive system, dimensionless

q = shear flow, lb/in.

q_B = blade displacement, in.

q_s = coordinates, or degrees of freedom, defining system motion, in. or ft

R = rotor radius, ft
= curved panel radius, in.
= reaction force, lb

R_i = generalized displacement, in.$^{-1}$

r = distance from elemental mass to center of rotation, ft

r_c = radius of curvature of spar, in.

S = wing area (planform), ft^2
= failure stress, psi

$\{S\}$ = selection matrix for incremental weight, dimensionless

S_x, etc. = shear load at monitoring station, lb

sta = station

T = kinetic energy, ft-lb
= rotor thrust, lb

T_a = ambient temperature, °R or °F

$\bar{T}$ = total engine torque at maximum rated power, lb-ft

T_t = tail rotor thrust, lb

t = thickness, in.
= time, sec

t_i = ratio of time at load spectrum condition i to total load spectrum time, dimensionless

Δt = engine acceleration time, sec

t_c = blade loading coefficient, dimensionless

U = gust speed, fps

U_{de} = design limit gust velocity, fps EAS

U_p = inflow velocity, fps

U_T = tangential velocity, fps

V = forward flight speed, kt or fps
= potential energy, ft-lb
= shear, lb
= wind speed, fps

V_{DL} = design limit flight speed, kt

V_{Da} = autorotational dive speed, kt

V_H = design maximum level flight speed in forward, rearward, or sideward flight, kt
V_{NE} = operating limit flight speed ("never-exceed" speed), kt
$V_{V_s\,min}$ = flight speed for minimum rate of descent in autorotation, kt
V_{cr} = cruise speed (highest forward flight speed at which specific range is 99% of maximum), kt
$V_{R/C\,max}$ = flight speed for maximum rate of climb, kt
V_e = equivalent flight speed, kt
= derived gust velocity, fps
V_{end} = endurance speed (flight speed at which fuel consumption rate in level forward flight is minimum, kt
V_{max} = maximum velocity, kt
V_s = descent velocity, fps
W = weight, lb
= chordal width of spar, in.
$\{W\}$ = matrix of empty weight, lb
$\{WT\}$ = weight data, in matrix form
$\{WCON\}$ = matrix describing the incremental weight for specific configuration, lb
W_g = gross weight, lb
w = weight of an element, lb
X = arm of rotor thrust vector about CG
x = coordinate in longitudinal direction, in.
x = location of helicopter CG, in.
x' = distance from any element to the aircraft CG, ft
$\dot{x}$ = velocity, general, in./sec
$\ddot{x}$ = acceleration, general, in./sec^2
Y = coordinate in transverse direction, in.
Y = location of helicopter CG, in.
z = coordinate in vertical direction, in.
z = location of helicopter CG, in.
α = angular acceleration, rad/sec^2
= angle of attack, rad
α_R = rotor angle of attack, rad
α_i = induced angle of attack, rad
β = blade flapping angle, rad
β_o = coning angle, rad
γ = dynamic response factor for given excitation, dimensionless
Δ = increment
δ = required deflection during a limit landing, ft
θ = pitch angle, rad or deg
= blade pitch angle, rad
= pitch angle of helciopter body axis, deg
= ratio of absolute temperatures, $T_u/519$, dimensionless
$\dot{\theta}$ = rotational velocity, rad/sec
$\ddot{\theta}$ = rotational or angular acceleration, rad/sec^2
λ = normalized mode shape ordinate, dimensionless
= inflow ratio, dimensionless
μ = advance ratio, dimensionless
ρ = air density, slug/ft^3 or lb-sec^2/in.^{4}
σ = rotor solidity, $bc_e/(\pi R)$, dimensionless
ϕ = roll, or bank, displacement, deg or rad
= blade mode shape variable, in.
X_i = absolute position vectors of the points i, in. or ft
ψ = blade azimuth angle, deg
Ω = rotor speed, rad/sec
Ω_{min} = minimum rotor speed at which rotor thrust = helicopter weight, rad/sec
ΩR = rotor tip speed, rad/sec
ω = response frequency, rad/sec
= vibrating frequency, rad/sec
ω_N = natural frequency, rad/sec

Subscripts

A = aerodynamic
= autorotation
a = maximum alternate design condition
b = basic structural design condition
ave = average
cr = critical
E = engine
ext = external
GI = ground idle
gr = ground
i = station
= integer
int = internal
i,j = axes of interest
lim = limit
max = maximum

min = minimum
N = value at maximum load factor, $n_{z_{max}}$
o = initial condition
P = peak
R/D = rotor drive
ST = static deflection
STD = standard conditions
s = side
sta = station
t = trim condition
ult = ultimate
x,y,z = coordinate axes

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CHAPTER 5

DYNAMICS

5-1 INTRODUCTION

The dynamic characteristics of rotary-wing aircraft are unique in the field of aviation. Dynamic considerations in helicopters involve blade design, control systems, and pilot and passenger comfort, and directly affect service life and maintenance of the aircraft. Satisfactory dynamic analysis and design will mean a smooth, dependable aircraft. Inadequate dynamic analysis may result in a catastrophic failure in a development program, or in a helicopter that must enter service with a restricted flight envelope.

Consideration of dynamics in this chapter includes the related areas of vibration, nature of exciting forces, response, resonant systems, and self-excited vibrations or instabilities.

Vibrations of mechanical systems fall into three categories:

1. Forced
2. Free
3. Self-excited.

In the case of forced vibrations the dynamic system will continue to oscillate as long as an external alternating force is applied. When the force is removed, the amplitude of the vibration will decay at a rate that is dependent upon the damping or rate of energy dissipation in the system. If the frequency of the exciting force approaches the same value as a natural frequency of the dynamic mechanical system, the amplitude of the motion of the mechanical system will become large and will be limited by the damping within the system. Such a condition is known as resonance.

A free vibration is one in which a dynamic mechanical system is displaced from its rest or equilibrium position and is released; then the system proceeds to oscillate with an amplitude that also decreases at a rate that is dependent upon the amount of damping present.

A self-excited vibration of a mechanical system more properly is termed an instability. Instability in a dynamic system is characterized by growth of the amplitude of a vibration with time, even in the absence of any external exciting force. The rate of increase of the oscillation is dependent directly upon the detailed characteristics of the mechanical system. Such instabilities may lead to large amplitudes of motion and failure of the machine part. In some cases, the rate of amplitude increase may be so great as to be categorized as "explosive". On the other hand, some instabilities may have slow growth rates and can be controlled. Some may reach a definite amplitude and grow no further; these are "limit cycle" instabilities. In some cases, addition of external damping may be sufficient to prevent the occurrence of the instability or to limit its amplitude.

For helicopters, the most prevalent vibrations fall into the free and forced vibration categories. Typically, most rotor blade bending and transmission shaft vibrations are of the forced vibration category. The bouncing of a blade on the droop stops would be typical of a free vibration. Ground resonance—a coupled oscillation between rotor blade lag motion and fuselage motion while on the ground—is an example of an instability and one that can be explosive.

Flutter is the self-excited, undamped, simple harmonic vibration of an aerodynamic surface and its associated structure in one or more of its natural modes. It is caused by the combining of aerodynamic, inertial, and elastic effects in such a manner as to extract energy from the airstream. At the critical flutter speed, the amplitude of oscillation following an initial disturbance will be maintained. At a higher speed, the amplitude will increase.

Divergence is the static instability of an aerodynamic surface which occurs when the torsional rigidity of the structure is exceeded by aerodynamic twisting moments. If the elastic axis of a wing is aft of the aerodynamic center, the torsional moment about the elastic axis due to the lift at the aerodynamic center tends to increase the angle of attack; this further increases the lift and, therefore, the torsional moment. For speeds below some critical speed (the divergence speed), the additional increments of twist and moment become smaller so that at each speed below the divergence speed an equilibrium position finally is attained (i.e., the process of moment increasing angle and thereby increasing moment, etc., is convergent). Above this critical speed the process is divergent. Because the lift force on a rotor blade section is opposed largely by a component of the centrifugal force, the chordwise separation between the aerodynamic center and the center of gravity has a significant effect on torsional divergence.

The various types of vibration phenomena are presented in greater detail in subsequent paragraphs of this chapter.

The primary reason for the emphasis on dynamics lies in the nature of main rotor and drive system operation and the importance of these systems to the whole machine. Both of these systems can provide vibratory force or motion inputs into the airframe. The airframe in turn can respond to these inputs in a variety of ways. For instance, if the frequency of the vibratory force is close to the natural frequencies of the fuselage, resonance may occur with resultant unpleasant vibration levels in the cabin. It also is possible that coupling between the rotor and the fuselage may exist, causing feedback from the fuselage into the rotor with adverse results.

The rotor is the primary object of concern in any study of the dynamics of rotary-wing aircraft. It is the source of most of the vibratory forces, and is a complicated dynamic system. Not only does it rotate, but its individual components also describe various motions while they rotate about the drive shaft. Blade flapping, lagging, and pitch change occur periodically in response to airloads and pilot control action. Rotor blade flexing takes place in response to the airloads. The forces that are generated within the rotor by these actions usually are multiples of the rotor rotational frequency and are harmonic in nature. The forces enter into the fuselage in a unique fashion. Alternating thrust forces or torques will enter the fuselage at the same frequency as the force that occurs in the rotating rotor. However, the rotor tends to act as a filter, and will cancel out all harmonics that are less than the number of blades on the rotor.

In the case of inplane forces in the rotor and the change necessary when moving from rotating to stationary coordinates, the designer may expect these forces to be fed into the stationary fuselage. If there should be a simple mass imbalance in the rotor, an unbalanced centrifugal force would act radially outward from the center of the rotor. Viewed from the rotating system, this unbalance would be seen as a force of constant amplitude. Viewed from the stationary coordinates, the unbalanced force of constant amplitude would rotate once per revolution. In the case of alternating inplane forces, a similar action would take place.

The forces that are transmitted to the fuselage will be equal in general to the frequency in the rotating system $\pm\Omega$, where Ω is the rotor speed. It also is possible that, for certain numbers of blades on a rotor and certain inplane frequencies, cancellations of inplane forces can result (Ref. 1). Because of these cancellations and the frequency changes that can occur with rotors, it often is difficult to pinpoint the exact source of any vibration frequency that may be found in the fuselage of a helicopter.

The rotor itself as a dynamic device displays other important characteristics. As indicated in par. 3-3.3, a fully articulated blade tends to flap as a rigid beam at a natural frequency close to rotor rotational speed. Because of the centrifugal twisting moments on a blade section, the flapping frequency of a torsionally rigid blade without pitch control restraint also tends to be close to rotor speed. The motion of the rotor blade in the lag direction is dependent directly upon the details of the mounting provisions at the hub, but for a typical fully articulated blade the lag frequency is approximately one-third of the rotor speed. As detailed in par. 5-3 various coupling actions can occur among the flapping, lagging, and pitching motions of the blades.

Rotor blades are long, narrow structures and are quite flexible. Consequently, they respond to time-varying airloadings in a dynamic fashion. Typically, the blades bend or twist in certain characteristic normal modes. Each of these modes of motion has a corresponding natural frequency. If the frequency of the applied airloads becomes close to any of these natural frequencies, the blades will respond by bending or twisting in the mode affected by the airload. Such amplified bending can lead to high blade stresses or high forces in the control system. The topic of blade response to aerodynamic forcing function is presented in greater detail in pars. 4-9.1 and 4-11.1, along with a discussion of blade motion and structural dynamic response.

In the remaining paragraphs of this chapter, various areas of dynamics are covered in greater detail. Airframe dynamics are discussed in par. 5-2. Par. 5-3, on rotor dynamics, emphasizes the potential instabilities that can take place. Other paragraphs deal with lifting surface dynamics and the torsional characteristics of helicopter drive systems. Because of its comprehensive nature, a thorough review of Ref. 2 should be considered as a supplement to this chapter.

5-2 AIRFRAME DYNAMICS AND VIBRATION

5-2.1 GENERAL

Mechanical vibration is a term that describes the oscillatory motion resulting from fluctuating forces acting upon a dynamic system, that is, a system possessing mass and elasticity. A helicopter is subject to vibrations induced by the rotors through the shafts or through aerodynamic vortex impingement upon the structure, and by the engines, gearboxes, or other portions of the dynamic drive train through the component mounting structure. Generally, main rotor(s) excitation through the shaft is the most significant source pertaining to aircraft comfort. The resulting oscillations are measured in terms of frequency, direction, and amplitude.

The frequency of a regularly repeated motion is defined as the number of complete cycles per unit time. The simplest kind of vibration is that which exhibits a single frequency. Simple vibration is not encountered in helicopters; these aircraft vibrate at two, three, or even four frequencies. These frequencies usually are multiple harmonics of the rotor excitation frequency. For example, a three-bladed helicopter usually vibrates at $1P$, $3P$, $6P$, and $9P$, where P is the rotational speed of the rotor in revolutions per unit time. In general, the more blades a helicopter has, the less likely it is that the harmonic frequencies will be significant; e.g., a five-bladed rotor usually exhibits only $1P$ and $5P$.

Some vibrations are nonperiodic in that a plot of displacement versus time shows no regularity. Nonperiodic vibrations typically are caused by gusts, control inputs, projectile hits, and some types of weapon firing. The types of vibration are shown in Fig. 5-1.

Vibration also may be classified as translational and rotational. From a human comfort and performance standpoint, rotational motion usually is insignificant, while vertical, lateral, and sometimes longitudinal translational vibrations are important.

The parameters used in describing the intensity of vibratory motion are the amplitudes of the excursions. Amplitude usually is measured by the displacement of the oscillation about its mean position. This amplitude can be expressed by either single amplitude or double amplitude (see Fig. 5-2).

The amplitude of vibration for the simple harmonic motion illustrated also can be expressed in terms of velocity or acceleration. In particular, the velocity and acceleration amplitudes of a simple harmonic motion are related to displacement amplitude by the following expressions:

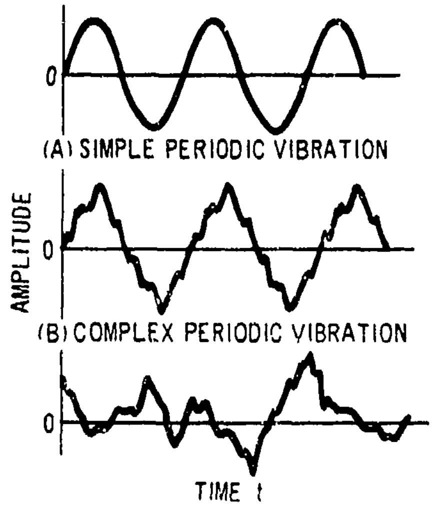

Fig. 5-1. Three Types of Vibration

$$\text{velocity: } \dot{x} = \omega x \qquad (5\text{-}1)$$

$$\text{acceleration: } \ddot{x} = \omega^2 x \qquad (5\text{-}2)$$

where

ω = frequency of oscillation, rad/sec
x = displacement, ft

The interaction between each of these variables affects the efficiency with which pilots and crew members can carry out their tasks and, therefore, must be taken into account in the design of rotary-wing vehicles.

During preliminary design of any helicopter, it is necessary to analyze the vibration characteristics of the airframe under steady-state conditions in order to assure compliance with the vibration requirements. The vibration levels experienced by the structure, aircraft components, crew, passengers, and cargo will depend upon the attributes of the forces and the structure.

In their normal environment, humans are not required to counteract the effects of vibration. Thus, when asking a crew to perform flight tasks in a vibratory environment such as a helicopter, care must be taken to insure that the vibratory effects will not be so severe as to degrade crew performance. Specific effects of excessive vibration on man are:

1. Motion sickness
2. Interference with orientation and coordination
3. Discomfort, and finally pain and damage to tissues

4. Interference with senses of touch, vision, etc., and with the performance of skilled tasks

5. Immediate or short-term phenomena such as fatigue, loss of sleep, psychosomatic or neuropsychiatric symptoms

6. Long-term cumulative impairment of brain function, circulation, etc.

The direct biological effects of vibration are recognized easily. More difficult to assess, but even more important in their implications, are the vague indirect effects of vibration upon human behavior and the ability to work. Recent research efforts have been devoted toward developing tools with which to measure the workload involved in flying rotary-wing vehicles. Successful development of such tools will permit objective measurement of effort expended in flying a vehicle, and thus will throw light on the effects of vibratory environment on performance and pilot efficiency.

5-2.2 VIBRATION CONTROL ANALYSIS AND DESIGN TECHNIQUES

Considerable attention must be devoted to the dynamic design of a helicopter if a vibration environment within the existing specification is to be expected. The paragraphs that follow outline the various analytical and test methods that can be used by the helicopter design engineer to help insure an acceptable vibration environment. It should be recognized that use of these tools only can increase the probability and confidence that an aircraft will exhibit an acceptable vibration environment in service, because it virtually is impossible (technically and economically) to conduct a design program that would guarantee design goals.

Controlling n-per-revolution vibrations of the airframe involves design considerations in both the rotor system and the fuselage. Basically, design effort should be directed toward minimizing vibratory loads generated by the rotor system and minimizing the response of the fuselage to the vibratory loads.

The first step in determining vibratory loads consists of defining the airload distribution on the rotor blades. Several methods are available:

1. Blade applied load analysis:
   a. Uniform inflow
   b. Variable inflow
2. Wind tunnel test data.

With the airload distributions defined, the blade response and finally the vibratory rotor forces can be assessed. Available methods are:

1. Blade response analysis:
   a. Rigid blade
   b. Flexible blade
   c. Uncoupled or coupled edgewise, flatwise, and torsional
   d. With or without main rotor head impedance
2. Use of rotor force measurements from similar aircraft.

With the rotor vibratory forces defined, the sensitivity of the aircraft to these forces should be determined. Some of the available methods are:

1. Fuselage analysis:
   a. Rigid airframe
   b. Flexible airframe
   c. Uncoupled or coupled airframe modes
2. Shake testing
3. Use of flight test data from similar aircraft.

As stated previously, the use of these tools does not guarantee an aircraft with an acceptable vibrational environment. It is, therefore, most beneficial to make provisions for vibration control in the design of the aircraft. Typical examples of vibration control devices are:

1. Rotor blade tuning

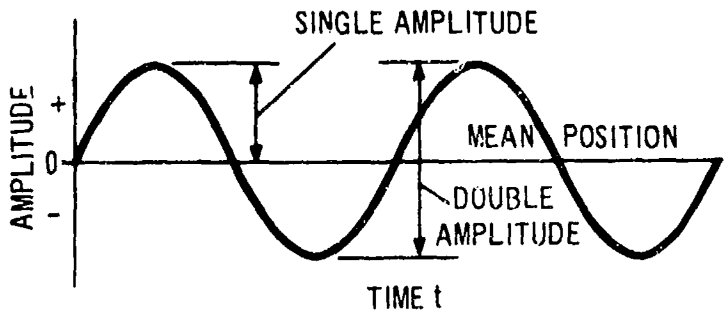

Fig. 5-2. Simple Harmonic Motion

2. Rotor vibration absorbers
3. Cockpit or cabin vibration absorbers
4. Transmission support structural tuning
5. Airframe structural tuning
6. Airframe component structural tuning
7. Crew seat and seat cushion design.

Further information on the details of blade applied load analysis and blade response analysis can be found in par. 4-9. Techniques of fuselage analysis are discussed in the paragraphs that follow. Design criteria pertinent to helicopter vibration levels also are reviewed.

5-2.2.1 Vibration Analysis

The characteristics of the forces that affect the response of the structure are the frequency content, the amplitudes and relative phases at each frequency, and the distribution over the airframe. The major forces of excitation in helicopters are at discrete frequencies and are applied at specific points rather than distributed over an area (see par. 5-2.4). Thus, in order to predict precisely the vibration levels to be expected in a helicopter, it is necessary to know the points at which the applied forces act and the amplitude of the component at each frequency. The amplitudes of these forces may be known only approximately during preliminary design. However, knowing the frequencies and points of application, it is possible to determine the sensitivity of the airframe to each of these components and thus to determine whether any corrective action is necessary.

The vibrations of the airframe may be represented as a superposition of the responses of each of its normal modes to each of the applied forces (see par. 5-2.2.2). This approach is useful especially during preliminary design, because the natural frequencies and mode shapes alone will give a good indication of whether or not the structure can be expected to be overly sensitive to any particular component or force.

5-2.2.2 Equations of Motion

The equations of motion of the airframe, using present state-of-the-art techniques, are formulated by considering the structure to consist of a number of lumped masses interconnected by spring and damping elements. The number of masses and number of degrees of freedom used will depend upon the detailed information available and the analyst's judgment as to the most rational representation of the structure. As a minimum, however, the degrees of freedom (coordinates) should include all the points of interest (such as the pilot seat and vibration-sensitive instruments), the points of application of forces (such as the main and tail rotor hubs on a helicopter), and sufficient additional points to represent adequately the shapes of all of the normal modes to be considered.

The equations for such a system can be written most conveniently in matrix form (Ref. 3). If the number of coordinates is N, then the mass matrix $[M]$, the damping matrix $[C]$, and the stiffness matrix $[K]$ are square, symmetrical, and of order N. The displacements y_t and the applied forces f_t are represented by $N \times 1$ column matrices (vectors), where each element is a function of time and represents the displacement of, and the force at, a particular coordinate. The equation, then, is written

$$[M]\ddot{y}_t + [C]\dot{y}_t + [K]y_t = f_t \tag{5-3}$$

For the steady-state condition at a frequency ω, the column matrices f_t and y_t may be written

$$\left.\begin{aligned} f_t &= fe^{i\omega t}, \text{ lb} \\ y_t &= ye^{i\omega t}, \text{ ft} \end{aligned}\right\} \tag{5-4}$$

where f and y are column matrices representing the amplitudes and are not functions of time t.

The separate elements are, in general, complex, representing the relative phasing between the force components and the response of each of the coordinates. For example, if an element of f is real, then the real part of an element of y is the component in-phase with the force and the imaginary part is that component that is out-of-phase by 90 deg with the force. This relationship is shown in Figs. 5-3 and 5-4.

Substituting the steady-state conditions of Eq. 5-4 into Eq. 5-3 results in the following equation of motion

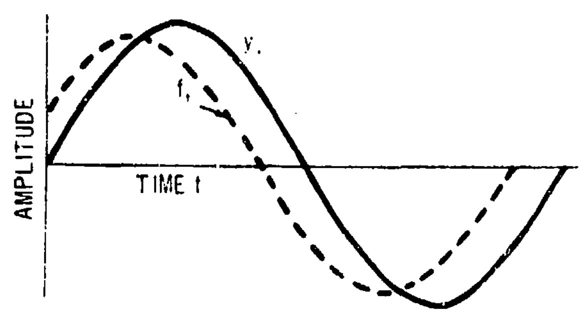

Fig. 5-3. Time History of an Element of f_t and y_t in Steady State

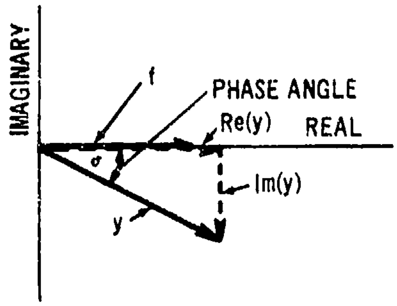

Fig. 5-4. Complex Representation of Condition of Fig. 5-3

$$[-\omega^2 M + i\omega C + K]y = f \tag{5-5}$$

The type of damping often associated with structures is of the form such that $C = (g/\omega) K$, i.e., the damping force is proportional to the deflection (spring force) but in phase with the velocity (Refs. 4 and 5). Typical values for the structural damping coefficient g for helicopter structures range from 0.02 to 0.08. The paragraph that follows shows that this small value of g has little effect upon the response except in the vicinity of a resonance. It generally is inadvisable to operate close enough to a resonance for the effect of damping to be significant. Thus, for the purpose of this analysis, the damping term may be ignored and the equation of motion will be written as follows:

$$[-\omega^2 M + K]y = f \tag{5-6}$$

5-2.2.3 Normal Mode Solution

If Eq. 5-6 is considered with no force acting ($f = 0$), the solution for y would be trivial (zero) except in the special cases where the determinant of $-\omega^2 M + K$ is zero. This will occur at special values of $\omega = \omega_n$, the "natural frequencies" of the system. If there are N degrees of freedom, there ordinarily will be N distinct natural frequencies. Corresponding to each ω_n, there is a solution $y = \phi_n$, the "normal mode". The normal modes are the relative displacements of the points that have been chosen as the degrees of freedom of the structure. These quantities satisfy the equation

$$K\phi_n = \omega_n^2 M\phi_n \tag{5-7}$$

obtained from Eq. 5-6 with $f = 0$, $\omega = \omega_n$, and $y = \phi_n$. Numerical methods for calculating the natural frequencies and normal modes are discussed in par. 5-2.2.4.

The normal modes ϕ_n are orthogonal with respect to the mass matrix $[M]$, i.e., $\phi_n^T M\phi_m = 0$, when $m \neq n$ and $\phi_n^T M\phi_n = m_n$, the "generalized mass" or "modal mass" of the nth mode, when $m = n$. This property allows separation of the response of the structure into the responses of each mode treated separately as a single degree-of-freedom system.

Consider y to be a linear combination of the normal modes as follows:

$$y = \sum_{n=1}^{N} q_n\phi_n \tag{5-8}$$

where q_n is the amplitude of the nth mode. Substitute this into Eq. 5-6 and use Eq. 5-7 to get

$$\sum_{n=1}^{N} q_n(\omega_n^2 - \omega^2)M\phi_n = f \tag{5-9}$$

Premultiply by ϕ_m^T, and use the orthogonality relationship to eliminate all terms of the summation except for $n = m$, resulting in

$$q_m(\omega_m^2 - \omega^2)\phi_m^T M\phi_m = \phi_m^T f \tag{5-10}$$

$$q_n = \frac{\phi_n^T f}{\omega_n^2 m_n\left[1 - \left(\dfrac{\omega}{\omega_n}\right)^2\right]} \tag{5-11}$$

Thus, the excitation of each mode may be calculated independently of the others (Eq. 5-11) and the response of the structure then can be formed by summing the separate responses of each mode (Eq. 5-8).

As seen from Eq. 5-11, the excitation of a mode by a force applied at a point is dependent upon the relative amplitude of the mode at that point $\phi_n^T f$, the generalized mass of the mode m_n, and the proximity of the frequency of the force ω to the natural frequency of the mode ω_n.

As an illustration of the effect of the mode shapes upon the response of a system, consider Fig. 5-5, which is an illustration of a typical normal mode of a helicopter fuselage. Consider the vibration of Point A due to unit forces of the same frequency at Points B, C, and D. Based upon the relative amplitudes of the mode shape shown, it is apparent that, in this mode, Point A responds most to a force at Point D, second most to a force at Point C, and least to a force at Point B. The minimum excitation of a mode will occur when the force is near a node (zero amplitude) and the maximum will occur when the force is applied near the point of maximum amplitude. Similarly, for the same excitation, points near a node will respond least and points near an antinode will respond most.

The excitation of the mode is dependent strongly upon the forcing frequency and the natural frequency of the mode in question. How the amplitude response of a mode varies with the forcing frequency is shown in Fig. 5-6. This is a plot of q_n versus ω/ω_n from Eq. 5-11. Also shown are the effects of the structural damping parameter g. It is apparent that when the excitation force is not near a resonance (natural frequency), the value of the damping coefficient is immaterial. It also is undesirable to operate the vehicle in the frequency range in which the effect of damping is significant. This partly is because of the relatively high amplitudes of vibration to be expected. Another undesirable feature is the steep slope of the response curve (Fig. 5-6). This means that small changes in frequency of excitation or natural frequency of the helicopter can make large differences in the vibration level. Such changes could result from operation over the usually small rotor speed range or normal changes in payload. Thus, in addition to the high vibration levels to be expected, the vibration will be difficult to predict and will change significantly over normal operating conditions.

In addition to the considerations of shape and frequency, one other parameter that affects the vibration level is the generalized mass $m_n = \phi_n^T M \phi_n$. As shown in Eq. 5-11, modes having a small generalized mass will tend to be excited more severely. Modes having small generalized masses, however, usually will have signifi-

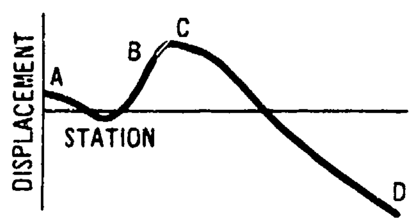

Fig. 5-5. Typical Normal Mode of a Helicopter in Plane of Symmetry

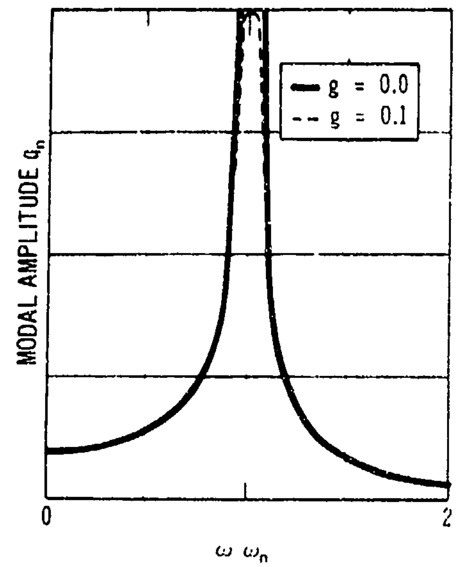

Fig. 5-6. Relative Modal Amplitude vs Forcing Frequency

cant deflections only over a small region of the structure.

Based upon the previous discussion, the following conclusions can be stated:

1. Points in the airframe near the high deflection regions of an excitable mode will be subject to greater vibration than points near nodes.

2. Modes having high deflection near sources of forcing will be subject to greater excitation than those having nodes near the forcing source.

3. Modes with natural frequencies within about 10% of a forcing frequency will tend to be excited significantly.

4. Modes with natural frequencies within about 20% of a forcing frequency may be excited significantly and the response may be quite variable under normal variations in rotor speed or payload.

5. Modes having small generalized masses will tend to be excited easily, but exciting forces and responses may be limited to local areas on the structure.

5-2.2.4 Methods of Calculating Normal Modes and Natural Frequencies

The previous discussion described certain characteristics of the vibration response of an airframe in terms of the normal modes of the structure. There are a number of rational methods available for computation of normal modes and natural frequencies. The proper one to select will depend upon the data available (which will vary considerably during preliminary design), and the type of structure under consideration. Because the numerical procedure for most of the methods is laborious, the selection of a particular rational method may depend upon which of these methods has been computerized.

The information available during preliminary design often is insufficient to allow the reliable computation of modes such as torsion or coupled bending-torsion of the fuselage. The modes most amenable to computation at this stage are those of uncoupled bending in the vertical or lateral directions. These also are the modes that usually are the most susceptible to excitation. Thus, it often is appropriate to represent the fuselage as a beam in transverse bending. In the methods available. the structure is divided into a number of segments—typically 10 to 20 in the early design stages. A lumped mass and an average effective bending rigidity *EI* are estimated for each segment. Some available methods that are suited especially to this representation are given subsequently.

The mode that can be expected to be closest in frequency to the major applied forces usually will be other than the one with the lowest bending frequency. Thus, the method used must be capable of obtaining normal modes above the first. In addition, it is recognized that the boundary conditions for a vehicle in flight are those of a free-free beam. The method used *shall* handle this condition properly.

In addition to the beam representation, the more general finite element approach is discussed in par. 5-2.2.4.3. This method may be suitable when sufficient structural detail is available.

5-2.2.4.1 Stodola Method and Matrix Iteration

The Stodola Method is used for beam bending (see Refs. 5 and 6). The general approach is:

1. Assume a trial mode shape (deflection).

2. Compute the shape of the inertial loading by multiplying the mass by the assumed mode shape.

3. Compute the shear by numerically integrating this loading.

4. Compute the bending moment by integrating the shear.

5. Divide the bending moment by the local *EI* to obtain the radius of curvature.

6. Integrate two more times to obtain the deflection and compare with Step 1.

When Steps 6 and 1 are proportional to each other, the ratio is the natural frequency squared and the deflection is the normal mode. When they are not proportional, then the result of Step 6 is used in Step 1 and the procedure is repeated until convergence. It can be shown that this iteration will converge on the lowest frequency mode contained in the trial. It is important to note that during the integrations of Steps 3, 4, and 6 it is necessary to take the boundary conditions into account. It may be necessary to carry along unknown initial conditions until a later step when they can be evaluated. The details of these computations are covered in the references given.

When modes above the first are desired, it is necessary to remove all components of the lower modes from the trial mode shape. When solving a free-free system, it also is necessary to remove the two rigid-body modes representing uniform translation and rotation about the CG. These procedures are treated adequately in the references.

The method of matrix iteration using influence coefficients is equivalent analytically to the Stodola Method but is more suited to automatic computation. This method is discussed in Refs. 4 and 7 where the free-free condition is treated specifically.

An alternate method of computing the higher modes by matrix iteration, which is particularly suitable for automatic computation, is presented in Ref. 7.

5-2.2.4.2 Myklestad Method

Another important method is that attributed to Myklestad (see Refs. 4 and 5). In this procedure, which is a modification of the Holzer Method, the frequency is varied and the shape of the beam is computed at each frequency. When all the boundary conditions are satisfied, the frequency used is the natural frequency and the deflection is the corresponding normal mode shape. The result often is shown as a plot of an applied force versus forcing frequency. Whenever the curve crosses the axis (the force is zero), a natural frequency exists.

The main advantage of this method is that no special treatment is required for special boundary conditions such as free-free, and it is unnecessary to obtain the lower modes before calculating the higher ones. Care must be exercised, however, to avoid skipping one or more modes.

5-2.2.4.3 Finite Element Analysis

The essence of the finite element technique consists of the formation of the stiffness matrix of a structure by superimposing the effects of small standard elements such as plates and rods. These elements are joined at points (nodes) and the stiffness elements relating to these points are obtained. Once the stiffness matrix is obtained and a mass matrix is calculated, matrix iteration (as in Ref. 5) may be used for the free-body modes. It will be necessary to extend this procedure, however, to take into consideration all six rigid-body modes.

This technique usually involves the use of a large, complex computer program and requires the knowledge of considerable structural detail. When the necessary information is known, such methods may provide the most reliable results.

5-2.2.4.4 Other Methods

There are other methods that should be mentioned. The Rayleigh and Rayleigh-Ritz Methods (Refs. 5 and 7) can be used to obtain the lowest frequency bending mode. This is not appropriate here because higher modes than the first generally are excited in helicopters.

The method of Associated Matrices (see Refs. 8 and 9) allows representation of different parts of the structure in different ways (e.g., beam, lumped parameter) and permits the assumed elastic axis to bend and follow the shape of the fuselage. This method is useful especially where a simple straight beam is an intuitively poor representation.

When the analysis is being made of a symmetrical structure (such as the uncoupled vertical bending of a pair of wings), the free-body motions can be separated into cantilever- and pin-free motions of half the structure. This method is treated in Refs. 5 and 7.

Often, it is necessary to consider a flexible portion of the airframe that is attached to a stiff and heavy portion, such as a tail boom that in turn is attached to the main fuselage. Under such a condition, an approximation of the tail boom vibration characteristics may be obtained by treating it as a cantilever beam. Methods such as this represent the lowest level of sophistication, but can be useful during preliminary design to provide an early warning of a potential vibration problem.

5-2.3 VIBRATION REDUCTION

There are several potential methods of reducing vibration levels. If possible, the most desirable method would be to reduce the applied vibratory force. This consideration is, however, outside the scope of this paragraph. Par. 5-2.4.1 refers to a device that absorbs vibratory forces at the hub.

The remaining methods, in one way or another, involve modification of the structure. These may be divided into the categories of airframe structural modification, vibration absorbers, and vibration isolators. Each is discussed in the paragraphs that follow.

5-2.3.1 Airframe Modification

When excessive vibration is predicted because an airframe natural frequency is close to a forcing frequency, it may be possible to change the frequency through structural changes. Adding mass will lower the frequency; removing mass will increase it. The greater the deflection of the point at which the mass is changed, the more effective the change will be. It is undesirable, of course, to add mass solely for this purpose. It also is unlikely that there will be excess mass that may be removed to increase the frequency. It may be possible, however, to shift one or more mass items from low deflection to high deflection regions to lower the natural frequency or to do the reverse and raise the frequency. Of course, other constraints such as helicopter CG and the utility of the items involved must be considered. Such changes may involve large masses and therefore may be quite impracticable, except possibly in the early design stages when major rearrangements of equipment are still feasible.

It also is possible to change the natural frequencies by changing the stiffness. The most effective areas are where the curvature is the greatest. Increasing the stiffness will increase the frequency. Such a change probably will tend to be more effective and more practicable than a mass change. The effect of the accompanying mass change must be considered if it is significant.

The extent of the modification required can be determined by selecting points at which changes may be made and repeating the original analysis with a relatively small change at each of the points in question. For small changes, the change in the square of the frequency will be proportional to the change in stiffness and inversely proportional to the change in mass at each point. By use of the solutions obtained, it now is possible to determine the magnitudes of the changes necessary to obtain the required change in frequency.

5-2.3.2 Vibration Absorbers

When it is not feasible to change the character of the vibration response of the fuselage through structural changes, it may be possible to make local improvements through the judicious use of dynamic vibration absorbers (see Refs. 10 and 11).

The main disadvantage of these devices is that their added weight ordinarily serves no other purpose and thus reduces the effective payload of the vehicle. An application in which an available mass (in this case, a battery) was used as an absorber mass is given in Ref. 12.

The narrow band width of an undamped absorber is not as severe a disadvantage in helicopters as in other vehicles because the excitation frequency varies relatively little. However, even small variations may have significant effects, especially when the absorber mass is small. A variable tuning absorber for use in helicopters has been developed and is described in Ref. 13.

When an absorber has a small mass (the only feasible circumstance in a helicopter), it causes a small change in the natural frequencies but produces a significant change in the deflection shape by reducing the amplitude of motion at the point of application. Thus, it is most effective when the point in question has a large deflection in the excited mode.

The dynamic vibration absorber basically is a simple spring mass system tuned to the frequency of excitation. When it is undamped, it acts to make the deflection of the point of attachment equal to zero. Damping and off-tuning tend to deteriorate the effectiveness of the absorber.

In general, the use of absorbers is not recommended except when no other method has been successful.

5-2.3.3 Vibration Isolation

Another potential means of reducing vibration in the fuselage is to isolate the major source of vibration. In a helicopter, this is the main rotor. This concept has been in the research stage for many years. Two recent approaches to the problem using active and passive devices are given in Refs. 14 and 15. In each case, the rotor and tramsmission (and possibly the engine) are separated from the rest of the airframe by devices that reduce the transmissibility of the predominant frequencies. The analytical investigation of vibration isolation is discussed in the paragraphs that follow.

5-2.3.3.1 Mathematical Methods

Because of excessive static deflections, no attempt is made to isolate vertical shaking forces by the use of spring elements. In fact, the vertical stiffness usually is given as large a value as possible. The commonly used rotor isolation systems are reducible to the equivalent of a system in which the isolated mass is mounted in a gimbal in such a manner that only relative angular motions of the fuselage and the isolated mass are permitted. The relative angular motion of the isolated mass with respect to the fuselage is resisted by an equivalent angular spring. On the assumption that the conditions required for feasibility have been met, the problem may be simplified further by solving the roll condition and the pitching condition as separate coplanar problems. A mathematical model of such a system is shown in Fig. 5-7. In this model all the masses to be isolated—such as the transmission, the main rotor shaft, the main rotor and the control mechanisms—are lumped at their resultant CG and designated as a point mass m_2 having a mass polar moment of inertia about its centroidal axis of I_2. The remainder of the helicopter and its cargo are lumped at the resultant CG and designated as a point mass m_1 having a mass polar moment of inertia of I_1.

The procedure in designing a rotor isolation system may be simple or complex, depending upon whether shaking forces or shaking moments are produced at the rotor hub. Teetering rotors produce only shear forces and moments about the axis of rotation, whereas rigid rotors may produce both flapping moments and shears at the rotor hub. Articulated rotors also may produce moments due to offset. In each case, the system may be described by the equations of motion for the system of Fig. 5-7 as follows:

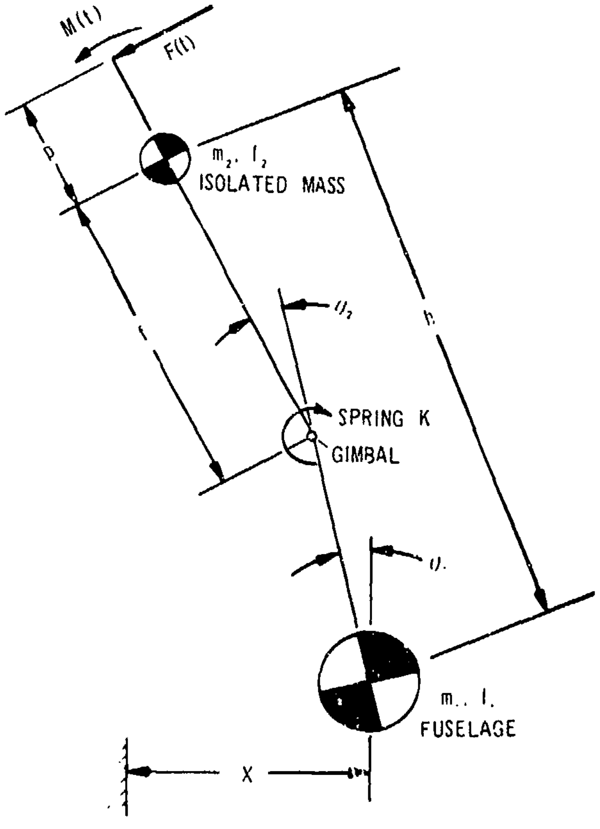

Fig. 5-7. Mathematical Model of an Isolation System

$$\begin{bmatrix} -(m_1+m_2)\omega^2 & m_2 h\omega^2 & m_2 f\omega^2 \\ m_2 h\omega^2 & (I_1+I_2+m_2h^2)\omega^2 & (I_2+m_2hf)\omega^2 \\ m_2 f\omega^2 & (I_2+m_2hf)\omega^2 & (I_2+m_2f^2)\omega^2+K \end{bmatrix} \times \begin{bmatrix} x \\ \theta_1 \\ \theta_2 \end{bmatrix} = \{Q\} \quad (5\text{-}12)$$

where

$$\{Q\} = \begin{bmatrix} -F \\ (h+p)F \\ -(f+p)F \end{bmatrix} \text{ for a shear force of amplitude } F$$

and

$$\{Q\} = \begin{bmatrix} 0 \\ M \\ M \end{bmatrix} \text{ for a moment of amplitude } M$$

Responses may be found easily by the use of Cramer's rule. Thus, for shear excitation at the rotor hub

$$\frac{\theta_1}{F} = \left\{ m_1\omega^2(I_2 - m_2fp)(f-h) + K[p(m_1+m_2)+m_1h] \right\} \times \Big[K\omega^2[m_1m_2h^2 + (m_1+m_2)(I_1+I_2)] - \omega^4\{m_1m_2[I_1f^2 + I_2(f-h)^2] + I_1I_2(m_1+m_2)\}\Big]^{-1}, \text{rad/lb} \quad (5\text{-}13)$$

If the numerator and the denominator of this expression both are divided by the stiffness K, which then is made infinite, a response is obtained for the case of the isolate, the mass being attached rigidly to the fuselage (no isolation). As a means of measuring the effectiveness of an isolation system, the normalized response is defined as the ratio of the given response to the response that would occur if there were no isolation system. Thus, the normalized response for shear excitation θ_{1N_F} is

$$\theta_{1N_F} = \{m_1\omega^2(I_2 - m_2fp)(f-h) + K[p(m_1+m_2)+m_1h]\} \times \{m_1m_2h^2 + (m_1+m_2)(I_1+I_2)\} \times \Big[\{K[m_1m_2h^2 + (m_1+m_2)(I_1+I_2)] - \omega^2(m_1m_2[I_1f^2 + I_2(f-h)^2] + I_1I_2(m_1+m_2))\} \times \{p(m_1+m_2)+m_1h\}\Big]^{-1}, \text{d'less} \quad (5\text{-}14)$$

By similar logic a normalized response θ_{1N_M} may be developed for exciting moments applied at the rotor head. Thus

$$\theta_{1N_M} = [K(m_1+m_2) - m_1m_2\omega^2(f^2 - hf)] \times \left[\frac{m_1m_2h^2}{(m_1+m_2)} + I_1 + I_2\right] \times \Big[K[m_1m_2h^2 + (m_1+m_2)(I_1+I_2)] - \omega^2\{m_1m_2[I_1f^2 + I_2(h-f)^2] + I_1I_2(m_1+m_2)\}\Big]^{-1}, \text{d'less} \quad (5\text{-}15)$$

The optimum attenuation of the rotor forces occurs when the normalized response is a minimum. It will be noted that if the numerator of the response function is set equal to zero, a value of f defining the gimbal location may be calculated so that the angular motion of the fuselage is zero. However, in practice, other considerations, such as reducing the relative motions between the transmission and its various input and output drive shafts, govern the selection of the gimbal point. It then becomes a matter of selecting a stiffness within the constraints of system natural frequency and static deflection characteristics. If the designer is successful in locating the natural frequencies of the isolated package at one-third to one-half of the forcing frequency, good isolation results. A typical normalized response for a small helicopter is shown in Fig. 5-8.

In this discussion the analysis has been simplified greatly by the assumption that both the rotor and the fuselage may be treated as rigid bodies. The assumption is valid if the fuselage and blade vibration modes are separated sufficiently from the forcing frequency. However, as helicopters become larger, this premise becomes less tenable; also, static deflections of the selected isolator springs become larger than can be tolerated. In these cases the whole fuselage and rotor must be considered as a multi-degree-of-freedom system. The mode frequencies may be adjusted or degrees of freedom may be added at the transmission, which may prove to be desirable. It is necessary to consider each configuration on its own merits.

5-2.3.3.2 Recent Developments

A method that may be used to solve the isolation problem for an intermediate-to-large helicopter is called the DAVI (Dynamic Antiresonant Vibration Isolator) (Ref. 15). The essential elements of the system are indicated in Fig. 5-9. In this system the isolated mass is attached to the fuselage through a spring element in parallel with a weighted lever; the mass and its

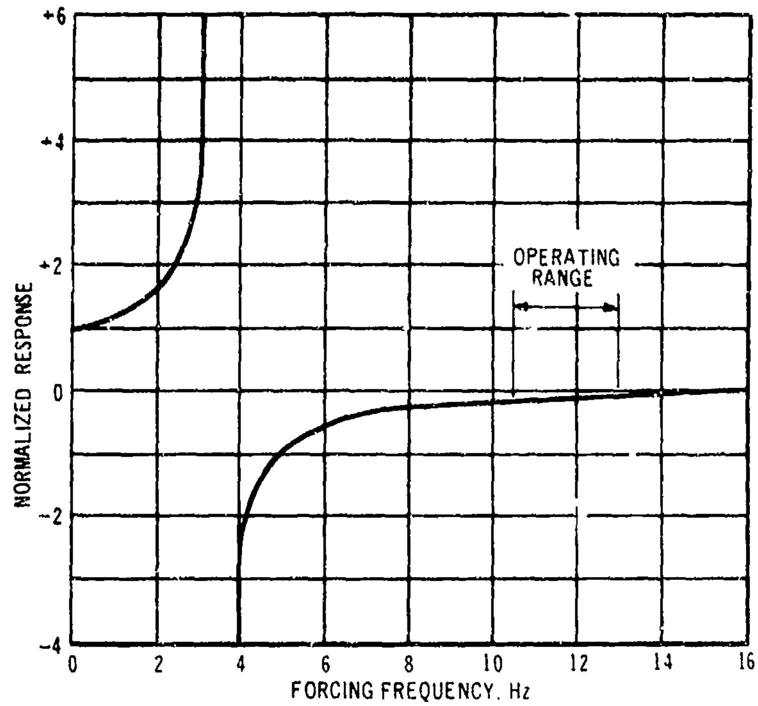

Fig. 5-8. Normalized Response for a Typical Isolation System

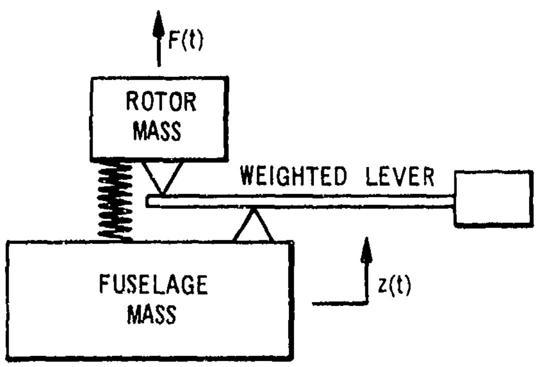

Fig. 5-9. Principal Elements of the DAVI System

mechanical advantage are such that at the selected forcing frequency the inertial force is equal and opposite to the spring force. Thus, theoretically, there is no resultant force acting upon the fuselage. The proposed design merits consideration in future designs as well as for retrofit.

Another recent development particularly useful in helicopters with rotors having low *n*/rev frequencies (*n* equals number of blades) is the nodalized beam concept described in Ref. 16. This concept advocates suspending the entire helicopter airframe from a support beam. By attaching the airframe only to the nodal points of the beam, the fuselage is isolated completely.

5-2.3.3.3 Fail-safe Design

The design and inspection of isolation systems can be made such as to insure against complete failure. Depending upon the rotor system used, the effects of a reduction or increase in stiffness or cross-stiffness due to the failure of an elastic element may result in a destabilization of the system, such as ground resonance. These effects should be considered in the overall design of the rotor control and isolation system. The hazards due to such a failure of an elastic element may be minimized by distributing the required stiffness over several elastic elements in order to minimize the change in stiffness. Sometimes means can be provided to insure that the element functions even if failure occurs; for example, the elastomer is preloaded in compression. In general, the designer will provide means to limit the motion of the isolated package by using mechanical stops, cables, or auxiliary nonlinear springs to restrict the motion in the event of a failure of the elastic elements.

5-2.4 EXCITATION SOURCES

The dominant vibratory forces in a helicopter are those forces transmitted through the rotor hub or by rotor/wing aerodynamic interference. These forces are largely aerodynamic in origin. Other aerodynamic and mechanical sources of excitation exist but generally are of minor importance compared to the rotor forces.

5-2.4.1 Rotor Forces

The steady-state forces acting upon a rotor blade are at frequencies that are harmonics of the rotor speed. If the blades all are identical aerodynamically and inertially, then the only vibratory forces or moments transmitted to the fuselage will be at harmonics that are multiples of the number of blades. Thus, a two-bladed rotor will transmit only multiples of two cycles per revolution forces, and a three-bladed rotor only multiples of three (see Refs. 1 and 17).

Consider first the vertical forces and shaft torque. The vertical shear force F of each blade can be expressed as

$$F = f_0 + f_1 \cos\psi + g_1 \sin\psi + f_2 \cos 2\psi + g_2 \sin 2\psi + \cdots + f_m \cos m\psi + g_m \sin m\psi \tag{5-16}$$

where

ψ = azimuth angle of one rotor blade, rad

f_i = amplitude of cos component of force, lb

g_i = amplitude of sin component of force, lb

Thus, if the force on blade "A" is $F(\psi)$, then the force on blade "B" is $F(\psi + 2\pi/n)$, etc. where n is the number of blades. When those expressions are summed, there results

$$\frac{F_{total}}{n} = f_0 + f_n \cos n\psi + g_n \sin n\psi + f_{2n} \cos 2n\psi + g_{2n} \sin 2n\psi + \cdots + f_{mn} \cos mn\psi + g_{mn} \sin mn\psi \tag{5-17}$$

Because the vector representing shaft torque is in the same direction, the same form results.

Forces whose vectors rotate in the plane perpendicular to the shaft behave in a slightly different fashion when resolved into the fixed system. These forces are inplane shear, radial shear, and pitching moment. The forces into the fixed system transmitted are still only at harmonics that are multiples of the number of blades. They are produced, however, by forces in the rotating system that are one harmonic above or one harmonic below the transmitted frequency. For example, a three-bladed rotor transmits a third harmonic lateral force (or rolling moment) that is due to second and fourth harmonic inplane loads. Table 5-1 shows how the loads transmitted by the hub are dependent upon the number of blades in the rotor, as well as the frequencies of the blade input forces.

The just-described characteristics of the transmitted forces are true rigorously only if all the forces acting upon each blade are the same when the blades are at the same azimuthal position. This will be the case only when the blades are identical. When there are small discrepancies in aerodynamic or inertial characteristics, complete cancellation of terms will not occur and at least small components of other harmonics will be transmitted. This effect has been recorded in wind tunnel tests (Ref. 18) and from flight test data (Ref. 9). Except for the first harmonic force due to steady effects such as blade unbalance, the only forces that will be significant will be at harmonics of rotor speed, as indicated previously.

A study of main rotor hub excitation of helicopters is reported in Ref. 15 in which the available data from several sources were examined. The data contained considerable scatter; however, reasonable generalized criteria were obtained. These standardized excitation levels, based upon amplitude of the predominant n/rev component being unity, are: the 1/rev component, 0.10; the $2n$/rev component, 0.40; and the $3n$/rev and $4n$/rev, 0.10.

Rotor blade dynamics can have a significant effect upon the amplitude of the forces transmitted through the hub. When a blade natural frequency is close to the frequency of a force that is not cancelled at the hub, a potential problem exists due to the amplification of the forces. For example, on a four-bladed rotor, the following natural frequencies should be avoided: 4/rev out-of-plane, and 3/rev and 5/rev inplane and/or torsion.

The resonant frequencies for flapwise and chordwise bending must be computed as functions of rotor speed, using plots of blade weight distribution and moments of inertia (par. 4-9). The determination of blade resonant frequencies could proceed in a simple, closed-form calculation if mode shapes, blade root restraints, and hub motions were known accurately. Usually these only can be estimated, so that an approximate solution is the best that can be obtained without extensive iteration. The method of Ref. 19 usually produces results that are accurate to within 10%, and has been used widely for preliminary design calculations. This method assumes that the centerline of rotation is rigid in space, and that rotation is at a constant angular velocity. The mode shapes are assumed and used as the basis for frequency calculations, as in the Rayleigh Method. Blade root restraint in the hub is taken as either a frictionless pivot or a rigid joint in the flapping and lead-lag planes. Mass and stiffness distributions are taken as linear functions of radius, which involves some degree of fairing the actual distributions. Provisions are made for representation of concentrated mass at the tip of the blade. For the appropriate root restraint and mass and stiffness distributions, coefficients of rotating and nonrotating resonant frequencies are taken from graphs. The various modal frequencies then can be shown on a Southwell diagram (see Fig. 5-10) in which the frequencies are plotted against rotor speed, with lines superimposed to show multiples of rotor speed.

In examining the Southwell diagram for coincidence of exciting and resonant frequencies, it is well to keep in mind that root restraint nearly always is more flexible than assumed, so that the resonant frequencies predicted by Yntema's Method (and even by many more sophisticated techniques) are often too high. It thus is safer to have an indicated resonant frequency occur slightly below coincidence with an exciting frequency rather than slightly above it.

One device, the "bifilar absorber" described in Ref. 20, is designed to eliminate the major inplane component of excitation at the hub. This is a rotating pendular mass mounted at the main rotor hub and acting as a vibration absorber. It has the advantage of eliminating the vibratory force almost at its source.

5-2.4.2 Other Sources

All the comments of the previous paragraph apply equally well to a tail rotor, a lifting fan, a propulsive propeller, or any similar rotating aerodynamic surface.

The blade pitching moments mentioned ... 5-

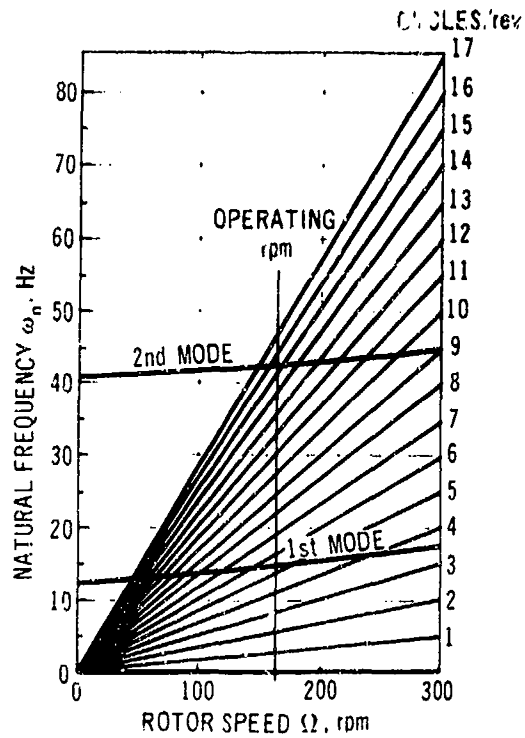

Fig. 5-10. Southwell Diagram

TABLE 5-1
INPLANE LONGITUDINAL OR LATERAL BLADE FORCES TRANSMITTED TO FUSELAGE

FREQUENCY OF VARYING LOAD ON ONE BLADE, per rev	FREQUENCY OF LOAD ON HUB, per rev		
	TWO-BLADED ROTOR	THREE-BLADED ROTOR	FOUR-BLADED ROTOR
1	STEADY AND 2	STEADY	STEADY
2	--	3	--
3	2 AND 4	--	4
4	--	3	--
5	4 AND 6	6	4
6	--	--	.

2.4.1 can be transmitted to the fuselage as vibratory excitation through the control system. This considera-tion is discussed in Ref. 21. Similar considerations apply to any control system, whether manual, augmented, or automatic.

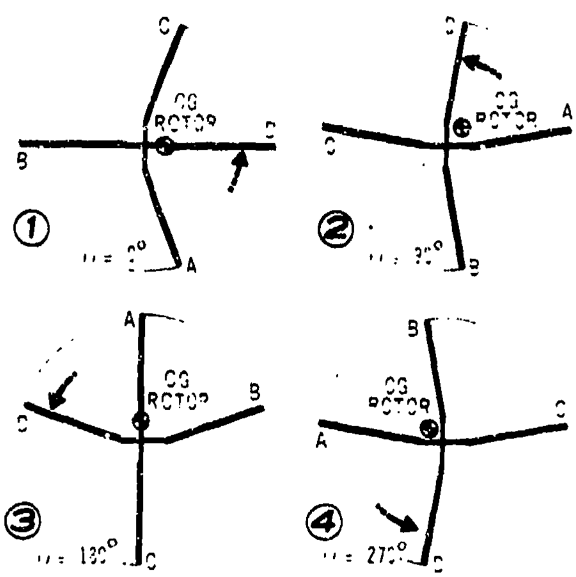

Fig. 5-11. Rotor Viewed in Fixed System Oscillating in Ground Resonance Mode

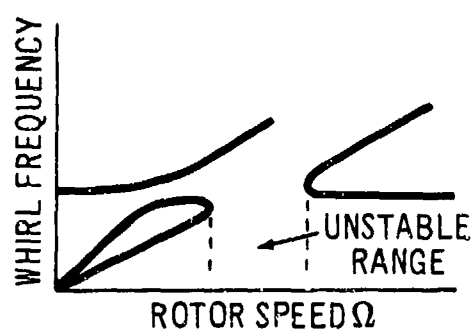

Fig. 5-12. Plot of Roots of Undamped Ground Resonance Equations

All rotating components such as engines, transmission gears, and couplings are potential sources of vibration. These often are mounted on resilient mountings that act as isolators (see par. 7-7). For a general discussion of such sources, see Ref. 22.

5-2.5 GROUND RESONANCE

Because the instability known as "ground resonance" involves the dynamics of the airframe as well as of the rotor, it is appropriate that there should be a discussion in this paragraph in addition to that of par. 5-3.3.3. Presented here is a brief summary of the classical Coleman theory (Ref. 23) and a common method of evaluating the ground resonance characteristics of a rotor-fuselage combination. It should be noted that this type of ground resonance is applicable to hinged rotors with small hinge offsets where there is no consideration of rotor aerodynamics. Mechanical instability (ground and air) analysis of hingeless rotors or hinged rotors with large flap hinge offsets must include rotor aerodynamics in order to obtain meaningful results. This type of mechanical instability is treated in Ref. 24 and discussed further in par. 5-3.3.3.

5-2.5.1 Description of the Phenomenon

Ground resonance is a self-excited mechanical instability caused by coupling of the motion of the hub in the plane of the rotor and the motion of the rotor CG due to inplane motion of the blades. This potentially destructive instability usually occurs when the helicopter is on the ground and the landing gear spring and damping rates affect the motion of the hub. The phenomenon will be discussed briefly, first for the common case of three or more hinged blades.

The inplane motion of the blades causes the CG of the rotor to be displaced from the hub. This condition, for a four-bladed rotor, is illustrated in Fig. 5-11.

This whirling of the CG of the rotor in turn produces an oscillating force on the hub. The hub can be considered to consist of effective masses, springs, and dampers. These quantities reflect the characteristics of the fuselage and, in general, will be different in the lateral and fore-and-aft directions. The rotating force applied to the hub causes it to deflect; this motion in turn applies forces to the blades, tending to cause inplane motion. When this interaction is such that the motions tend to increase, there is an instability. For a given set of parameters, the instability manifests itself as a range of rotor speeds in which an unstable oscillation will occur. The solution obtained by solving Coleman's equations for the case of no damping is shown in Fig. 5-12.

The phenomenon for a one- or two-bladed rotor is essentially the same, except that the unsymmetrical condition introduces certain complications into the equations; there are more ranges of instability associated with this condition. This is well treated in Ref. 23. Further physical descriptions of the phenomenon are given by Refs. 25 and 26.

5-2.5.2 Important Parameters

The parameters involved in the computation of the ranges of instability are discussed subsequently.

There are three quantities associated with the hub degree of freedom: the effective mass m_h, the spring constant K, and the damping constant B. In general, there will be several values of each of these parameters in each direction depending upon the mode of the fuselage under consideration. For preliminary design of a helicopter, it usually is sufficient to consider the fuselage to be a rigid body on the landing gear, which supplies all the spring and damping effects. Parameters such as the tire stiffness (vertical and lateral) and the effect of gear extension (under partially airborne conditions) must be considered.

The parameters associated with the rotor are: blade mass m; lag hinge offset e; distance b of center of mass of blade from lag hinge; radius of gyration r of blade; centering spring rate K_β, and blade damping B_β. These parameters are constants for a given design. Blades without lag hinges may be simulated by an effective lag hinge and centering spring that will give the same lowest frequency and will approximate the mode shape.

These quantities commonly are grouped into the following dimensionless parameters:

$$\left.\begin{aligned} \Lambda_1 &= \frac{e}{b\left(1+\dfrac{r^2}{b^2}\right)} \\ \Lambda_2 &= \frac{K_\beta}{I\omega_r^2} \\ \Lambda_3 &= \frac{\mu}{2\left(1+\dfrac{r^2}{b^2}\right)} \end{aligned}\right\} \qquad (5\text{-}18)$$

where

I = blade mass moment of inertia about lag hinge, slug-ft^2

ω_r = hub natural frequency (reference), rad/sec

μ = mass ratio $nm_b/(m_f + nm_b)$

In addition, the two dimensionless parameters

$$\left.\begin{aligned} \lambda_\beta &= \frac{B_\beta}{I\omega_r} \\ \lambda_f &= \frac{B_f}{M\omega_r} \end{aligned}\right\} \qquad (5\text{-}19)$$

are used to represent the blade and hub damping, respectively. The rotor angular velocity Ω and the angular whirling velocity ω_f measured in the fixed system are written nondimensionally in terms of ω_r.

5-2.5.3 Method of Analysis

The first step in an analysis should be to calculate any undamped instability ranges. The blade parameters are fixed, but ordinarily there will be several sets of hub parameters to be analyzed. There should be an analysis for each hub natural frequency up to the vicinity of the maximum operating rotor speed. In addition, the effect of any parameter such as gross weight or landing gear strut extension that will affect the hub natural frequencies should be investigated.

The method given in Ref. 23 allows the computation of undamped instability ranges. Charts showing the ranges for values of Λ_1, Λ_2, and Λ_3 are given for $s = 0, 1, \infty$, for rotors with two or more blades, where s is the ratio of the hub spring rates in the two directions. Thus the charts touch the entire range from rigid to free in one direction and include the case of isotropic supports ($s = 1$). The data obtained from such a study will give a good indication of any potential instability ranges within the operating conditions of the helicopter.

If an instability range is indicated and the configuration is such that blade and hub damping are feasible, then it may be possible to eliminate this unstable range through the use of dampers. As shown in Refs. 23 and 25 for the case of rotors with three or more blades, a specified product of the two damping ratios is required. This clearly implies that damping of both the hub and blade is required to eliminate the instability. Ref. 25 gives an approximate form for this product as

$$\lambda_\beta \lambda_f > \frac{\Lambda_3}{\omega - 1} \qquad (5\text{-}20)$$

$$\omega = \frac{1+\sqrt{\Lambda_1 + \Lambda_2 - \Lambda_1\Lambda_2}}{1 - \Lambda_1} \qquad (5\text{-}21)$$

A more detailed solution, corresponding to the case of infinite stiffness in one direction, has been presented in convenient chart form in Ref. 26.

For an articulated rotor with three or more blades, the usual means of avoiding the instability is to use appropriate blade and landing gear dampers as discussed previously. For two-bladed rotors without hinges, inplane natural frequencies normally are above the critical range for ground resonance. Therefore such resonance is avoided (Ref. 26). Because two-bladed rotors with lag hinges are susceptible to shaft-critical instabilities over a wide speed range, they are very seldom considered.

5-2.6 DESIGN CRITERIA AND CONSIDERATIONS

With respect to criteria, information for designers in terms of allowable vibration levels is available from a great many sources. The only available specification that has vibration levels aimed at preventing crew discomfort is MIL-H-8501. However, experience has shown that the vibration levels specified, if strictly observed, would not yield a satisfactory design. Pilot and passenger experience has shown that levels considerably below the 0.15-g requirement of MIL-H-8501 are required for low frequency comfort (Ref. 27). More realistic criteria considering both helicopter reliability and crew comfort, based upon data presented in Refs. 27 and 28, are being incorporated into new military specifications superseding the MIL-H-8501 vibration requirements. These new vibration characteristics requirements are as follows:

1. Controls. Vibration levels in any direction at all controls *shall* not exceed an acceleration of 0.10 g for frequencies up to 5 Hz, a velocity of 1.4 in./sec for frequencies between 5 and 32 Hz, and a double amplitude of 0.008 in. for frequencies above 32 Hz. This requirement *shall* apply at all steady speeds within the helicopter design envelope, during slow and rapid transitions from one speed to another, and during transitions from one steady acceleration to another.

2. Personnel stations. At the pilot, copilot, and passenger seat structure (near the floor), at the pilot and copilot heel rest positions (on the floor), and at litter stations, at all steady speeds between 30 kt rearward and cruise velocity V_c, vibratory acceleration *shall* not exceed 0.05 g at or below the fundamental main rotor passage frequency (n/rev) and vibratory velocity *shall* not exceed 0.6 in./sec at frequencies greater than n/rev. At steady speeds from V_c to design limit velocity V_H the maximum vibratory acceleration at the personnel stations *shall* not exceed 0.2 g up to 20 Hz, and vibratory velocity *shall* not exceed 0.7 in./sec for frequencies greater than 20 Hz. During slow and rapid linear acceleration or deceleration between any speeds within the flight envelope, vibration levels at the personnel stations *shall* not exceed a velocity of 1.0 in./sec for frequencies up to 50 Hz, and a double amplitude of 0.003 in. for all frequencies above 50 Hz.

In the low-frequency region (below 10 Hz), the vibration requirements are complicated by the presence of human body responses. There is a vertical resonance near 5 Hz, while a resonance near 2 Hz is the dominant mode in horizontal vibration of a seated man. The resonant frequencies and maximum transmissibilities are affected by many factors including muscle tension, body position and posture, weight of clothing and equipment, and the complexity of the excitation. Airframe vibration levels for acceptable comfort in the cockpit and cabin will vary at low frequencies due to the important resonances.

It should be noted that these new requirements do not represent final comprehensive vibration criteria relating to pilot and crew comfort and performances. For example, the cumulative degradation of comfort caused by vibrations at more than one frequency has been disregarded. Likewise, the combined effects of 8-hr exposure to noise, extreme temperature, and vibration upon pilot task performance have not been explored fully.

Refs. 29 and 30 describe the work completed in the field of visual disturbances. The designer should review and utilize these data where possible.

5-3 ROTOR SYSTEM INSTABILITIES

5-3.1 GENERAL

Rotor system instabilities have been characterized by a variety of names such as flutter, low-frequency flutter, stall flutter, wake flutter, whirl flutter, ground resonance, air resonance, vertical bounce, weaving, shuffling, pilot-induced oscillation (PIO), and others. A more systematic way of classifying rotor system instabilities is according to the essential mode contents of the unstable oscillations. In the subsequent discussion, unless otherwise stated, the vibration modes and the frequencies will be described in a hub-fixed rotating reference system. If we assume a rigidly supported rotor hub and swashplate, and ignore coupling effects, there are three types of structural modes:

1. Blade pitching or torsion modes with angular displacements about a longitudinal blade axis

2. Blade flapping or flap-bending modes with out-of-plane blade displacements

3. Blade lead-lag modes with inplane blade displacements.

The actual modes are coupled aerodynamically, elastically, and inertially; they include some blade pitching or torsion and some blade out-of-plane and/or inplane motion.

With respect to the principal mode contents, the designer can distinguish among three types of instabilities:

1. Blade flutter
2. Flapping instability
3. Lead-lag instability.

Blade flutter, including wake and stall flutter, essentially involves a blade pitching or torsional mode that may or may not be coupled with out-of-plane or inplane blade modes. The flutter frequency is equal to or lower than the corresponding natural frequency of the purely structural torsional mode. Flapping or flap-bending instability involves an out-of-plane blade mode that may or may not be coupled with blade pitch, torsion, or lead-lag motions. The instability occurs at high rotor advance ratios and, depending upon aerodynamic coupling effects, may have a frequency substantially different from the natural frequency of the corresponding, purely structural out-of-plane mode. Lead-lag instability involves an inplane blade mode and may or may not be coupled with blade torsional or out-of-plane motions. This type of instability is easiest to identify because, even in the case of coupling with either torsion or out-of-plane blade motions, it occurs with a frequency approximately equal to the natural frequency of the corresponding structural inplane mode.

In addition to the oscillatory types of instabilities, there are aperiodic divergences. Frequently, the analytically easier-to-determine aperiodic divergence boundary is only slightly different from the oscillatory stability boundary and, therefore, can be used as an approximation of the latter.

If the previous assumption of a rigidly supported rotor hub and swashplate is removed, there are several ways in which single-blade modes can be coupled. The flutter modes of the single blades are coupled through swashplate-tilting and swashplate-vertical motions. The flapping and flap-bending modes of the single blades are coupled through hub-tilting and hub-vertical motions. The lead-lag modes of the single blades are coupled through hub-lateral and rotational motions. Finally, couplings between the individual blades can occur if blade flapping deflection or flapping moment signals are fed back into collective or cyclic pitch controls.

All coupled blade modes are characterized by the fact that each blade performs the same oscillation as every other blade except for a difference in phase. There are four types of multiblade, coupled, modes:

1. Collective modes, where the phase angles of all blade oscillations are identical

2. Differential collective modes, which are possible only for even-bladed rotors, where the oscillations of subsequent blades are in counter phase

3. Regressing modes, where the location of maximum blade deflection rotates in a rotor-fixed reference system opposite to the direction of rotor rotation

4. Advancing modes, where the location of maximum blade deflection rotates in a rotor-fixed reference system in the direction of rotor rotation.

If we count positions of the blades in the direction of rotation from 0 to the $n-1$, and if we assume that each blade oscillates with a circular frequency ω, then Fig. 5-13 represents the phase relationship for the first regressing mode. The deflection η_k of Blade No. 0 as a function of ωt is shown in the curve. Blade No. 1 leads in phase by $2\pi/n$; Blade No. $(n-1)$ lags in phase by $2\pi/n$. Fig. 5-13 shows Blade No. 0 in its maximum deflection. After elapse of a time $2\pi/(\omega n)$, Blade No. $(n-1)$ will reach maximum deflection. The location of maximum blade deflection, therefore, rotates backward with respect to rotor rotation. Fig. 5-14 represents the phase relationship for the first advancing mode. Blade No. 1 lags in phase by $2\pi/n$, and the location of maximum blade deflection rotates forward with respect to the sense of rotor rotation.

The second regressing and advancing modes are characterized by Blade No. 1 leading and lagging respectively Blade No. 0 in phase angle by $4\pi/n$, the third regressing and advancing modes by a leading and lagging phase angle respectively of $6\pi/n$, etc. The nth blade is identical to the 0th blade and, with respect to

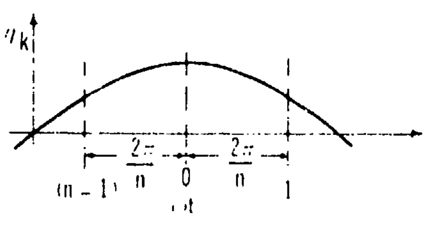

Fig. 5-13. Phase Relations of Blades $(n-1)$, 0, 1 in Regressing Mode

the 0th blade, must have a phase lead or lag equal to a multiple of 2π.

For even-bladed rotors, the phase difference of the oscillations of subsequent blades can be $\pm\pi$. In this case, the regressing and advancing modes are identical and such a mode suitably is called a differential collective mode, represented in Fig. 5-15.

For an n-bladed rotor it is necessary to introduce n multiblade generalized coordinates. For a two-bladed rotor, the two multiblade coordinates describe the mode contents of the collective and differential collective modes. For a three-bladed rotor, the three multiblade coordinates describe the mode contents of the collective and the first regressing and the first advancing modes. For a four-bladed rotor the four multiblade coordinates refer to the collective, the differential collective, and the first regressing and the first advancing modes. For a five-bladed rotor, the five multiblade coordinates describe the mode contents of the collective, the first and second regressing, and the first and second advancing modes, etc. The differential collective and the higher-than-first regressing and advancing modes are self-contained and do not feed forces or moments into the airframe. However, due to their aerodynamic coupling with the other, nonself-contained modes, they influence the dynamic stability limits.

Much of the published dynamic stability work is limited to single-blade treatments. The earliest and best known multiblade analysis concerns mechanical instability, or ground resonance, when the first regressing

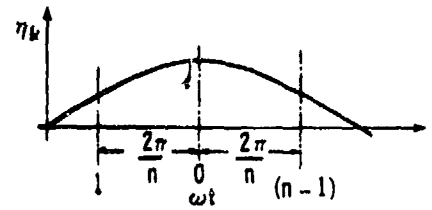

Fig. 5-14. Phase Relations of Blades (n − 1), 0, 1 in Advancing Mode

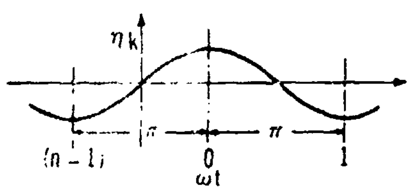

Fig. 5-15. Phase Relations of Blades (n − 1), 0, 1 in Differential Collective Mode

lead-lag mode becomes unstable. The phenomenon called air resonance is similar, except for the inclusion of aerodynamic effects, whereby instability also can occur in the first advancing lead-lag mode. Whirl flutter is a rotor-airframe instability involving advancing or regressing flapping or flap-bending modes. Vertical bounce involves an unstable collective flapping or flap-bending mode, while PIO can be produced by pilot coupling with either advancing or regressing flapping or lead-lag modes. Finally, it has been established that coupling of blade modes resulting in unstable characteristics is possible not only through hub and swashplate motions or feedback controls, but also through purely aerodynamic blade coupling whereby the tip vortices of one blade affect the lift, drag, and torsional moments of the trailing blade.

5-3.2 SINGLE-BLADE INSTABILITY ANALYSES

The single-blade analysis assumes that the hub and the swashplate are supported rigidly. Usually such an analysis will provide only a crude approximation of an instability, the actual stability limit being lowered or raised by the various coupling mechanisms between blades. The fact that a single-blade analysis does not uncover any instability should not preclude further probing or studying of multiblade, coupled rotor-airframe instabilities, which will be discussed later.

5-3.2.1 Single-blade Flutter and Torsional Divergence

Most flutter work has been done for zero rotor advance ratio. In this case there are three types of flutter: classical, wake, and stall.

A classical flutter analysis (Ref. 31) is similar to that for a fixed wing except that the variation of relative airspeed over the blade length is considered in a type of strip analysis and that additional terms from Coriolis accelerations are included. Reduced frequencies $b\omega/V$ for helicopter blades are in the order of 0.1 to 0.3, using a representative velocity V at $0.7R$ or $0.8R$, where b is the half-chord of the blade in ft, ω is the angular frequency of the blades in rad/sec, V is the relative flow velocity in ft/sec, and R is the rotor radius in ft. According to classical flutter theory, oscillatory angle of attack changes produce a lift that is smaller than for the equilibrium condition and that lags in phase. Lift deficiency and phase lag are functions of the reduced frequency $b\omega/V$ (see Fig. 5-16). The pitching inertia of the blade is increased by an aerodynamic term, and there is an aerodynamic pitch damping term. Often it is pos-

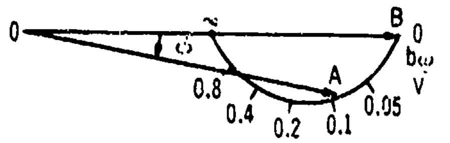

Fig. 5-16. Lift Deficiency OA/OB and Phase Lag ϕ vs Reduced Frequency $b\omega/V$

sible to omit the phase lag of the lift, the aerodynamic pitching inertia, and the aerodynamic pitch damping; this is called the "quasi-static" approach. In a further simplification, all damping terms are omitted—including the vertical damping. One then obtains a system that is conservative over the entire speed range up to the flutter speed and that now is characterized by a coalescence of the torsional and an out-of-plane mode. In many cases the approximation for this frequency coalescence method is quite good.

The principal parameter determining the flutter speed is the offset of the chordwise CG from the aerodynamic center (a.c.). If this offset is zero, flutter usually is not possible. For a given aft location of the CG with respect to the a.c., increasing torsional stiffness tends to increase the flutter speed. However, the situation is more complex when pitch-flap coupling is used that can lead to flutter in a higher flap-bending mode. Because most helicopter blades are designed so that chordwise CG and a.c. are close together, flutter rarely has been a problem at zero advance ratio.

At low blade angles of attack and at low lift, wake flutter may develop. This is caused by a rotor blade operating close to its own wake or to that of the preceding blade (Ref. 32). Flutter of this type, if it occurs, disappears with increasing lift and increasing advance ratio.

At high blade angles of attack, stall flutter may develop. This is caused by a hysteresis loop of pitching moment versus angle of attack feeding energy into the pitching oscillation. Data on oscillating airfoil characteristics in the vicinity of stall are scarce. Ref. 33 includes such data for a NACA 0012 airfoil. Fig. 5-17 taken from Ref. 34 shows a flutter boundary measured with a single-blade rotor versus blade pitch angle for a NACA 23012 airfoil when the chordwise CG is at 37% and the elastic axis at 26%. The flutter speed drops rapidly with increasing blade pitch angle. Classical flutter theory evaluated in Ref. 31 results in this case in a reduced flutter speed of $V/(b\omega_t) = 5$ and a flutter frequency ratio of $\omega/\omega_t = 0.47$, where V is taken at $0.8R$ and ω_t is the natural frequency of the blade in torsion. The experimental flutter frequency ω is about $0.7\ \omega_t$ at low blade pitch and increases to ω_t at high blade pitch. Fig. 5-17 should be considered merely as an example and not as a typical case. The unsteady airfoil moment characteristics depend in a complex manner upon Mach number, Reynolds number, mean angle of attack, amplitude of angle of attack, and reduced frequency. The available data for NACA 0012 and 23010 oscillating airfoils show maximum unstable pitching moment characteristics for mean angles of attack of 15 to 20 deg, reduced frequencies of 0.2 to 0.4, and Mach numbers of 0.3 to 0.4. Stable pitching moments occur at both lower and higher values of mean angles of attack, of reduced frequency, and of Mach number.

Correlation of analytical and experimental results on single-blade flutter usually is not very good, even at zero advance ratio. For forward flight conditions, with their time-variable relative velocities, no valid flutter analysis is available at present. However, a frozen azimuth type of flutter analysis can be useful in order to avoid large blade loads, because such high loads invariably occur in conditions for which the frozen azimuth analysis predicts flutter. The two most critical azimuth angles (measured counterclockwise from the rear of the rotor disk) are 90 deg, where the relative velocity is maximum, and 270 deg, where the angle of attack is maximum and where conditions of reversed flow exist over a portion or all of the blade. Fig. 5-18, taken from Ref. 31, shows the torsional amplitude versus chordwise CG location, computed from a sophisticated single-blade program, for an articulated blade operating at advance ratio $\mu = 0.3$ at low pitch. If the 90-deg azimuth condition is frozen, the blade flutters at an aft chordwise CG position of somewhat above 0.10. While actual flutter does not occur because of the transient nature of this condition, the frozen azimuth flutter limit does indicate a region of rapidly increasing torsional amplitude. Exceeding this limit would lead to excessive blade loads or deflections.

Similarly, the 270-deg frozen azimuth flutter limit provides a useful indication of the upper advance ratio boundary that would lead to rapidly increasing torsional loads and deflections. For advance ratios of one or more, when the entire retreating blade experiences reversed flow and when the aft chordwise CG location with respect to the reversed flow a.c. is about 0.5, the 270-deg frozen azimuth flutter limit is quite close to the corresponding torsional divergence limit, which is easier to compute. Fig. 5-19, taken from Ref. 35, shows for a nonarticulated rotor the computed torsional stress amplitude versus advance ratio μ together with the 270-deg frozen azimuth torsional divergence limit for an advancing blade tip Mach number of 0.85. Again it

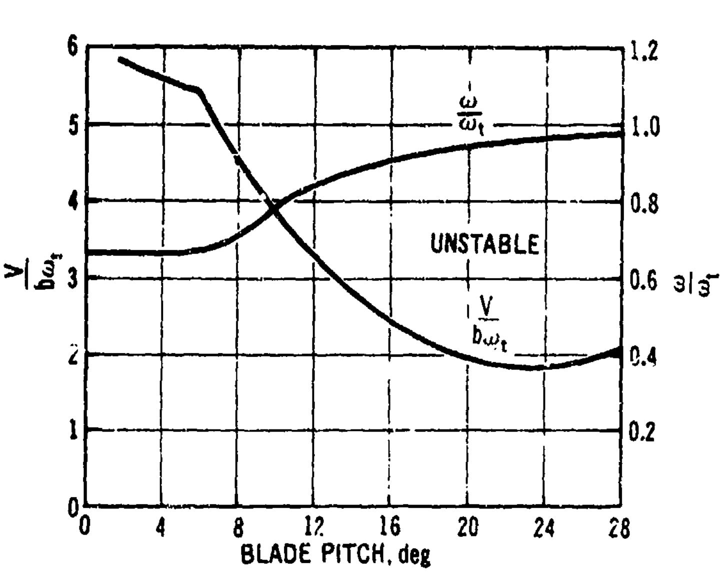

Fig. 5-17. Experimental Reduced Flutter Speed $V/(b\omega_t)$ and Flutter Frequency ω/ω_t for Single Blade With 37% Chordwise CG Location

is seen that the torsional frozen azimuth divergence boundary is in a region of rapidly increasing torsional stress amplitudes. The torsional divergence boundary at 270-deg azimuth is an important limitation to high advance ratio operation. Fig. 5-20, also taken from Ref. 35, shows this limitation in the form of such a boundary intersecting the boundary of 0.85 Mach number at the advancing blade tip.

5-3.2.2 Single-blade Flapping Instability

In a vertical or low advance ratio flight condition, blade flapping instability is possible in the case of a power failure without a sufficiently rapid reduction in

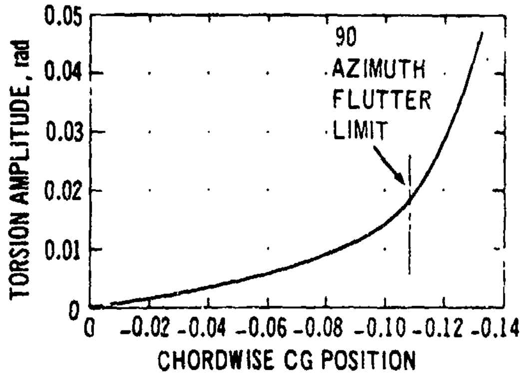

Fig. 5-18. Torsional Amplitude vs Chordwise CG Position for μ = 0.3, Articulated Blade

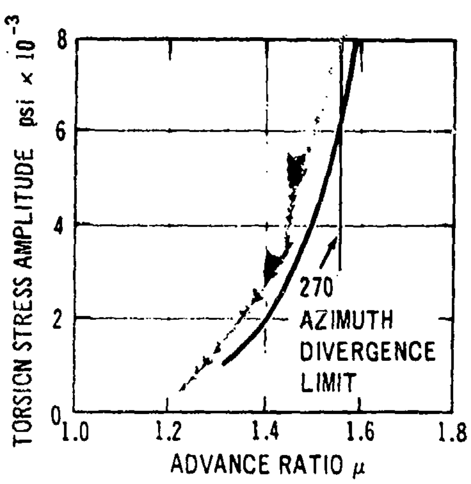

Fig. 5-19. Torsional Stress Amplitude vs Advance Ratio, Nonarticulated Blade

blade pitch to values within the autorotational range. In this case, rotor speed is lost and helicopter sinking speed is increased until the up-flow velocity through the rotor produces an up-flapping aerodynamic moment on the blade that no longer can be balanced by the down-flapping moment of the centrifugal force. At high advance ratio this situation also can occur without power failure—if a frozen azimuth angle of 180 deg is considered. For a rigid blade hinged at the rotor center and on the assumption that the quasi-steady aerodynamic analysis of Ref. 36 is valid, flapping divergence at 180-deg azimuth will occur for

$$\gamma\left(\frac{B^3\mu}{6}+\frac{B^4\theta_\beta}{8}\right) > 1 \qquad (5\text{-}22)$$

This limit is shown in Fig. 5-21 for a tip loss factor of B = 0.97 and for values of pitch-flap coupling ratio θ_β of 0 and −0.5. Because of the transient nature of the frozen azimuth unstable condition, actual instability of the blade occurs at substantially higher values of rotor advance ratio μ and Lock number γ, and is of an oscillatory type. It is interesting, however, that the frozen azimuth divergence limit usually agrees with an entirely different limit computed in Ref. 37 and also shown in Fig. 5-21 in dashed lines. The Ref. 37 limit was obtained by calculating, for a condition of blade tip Mach No. 0.85, the maximum flapping angle following a sudden gust input of 30 fps at zero azimuth and limiting this maximum flapping angle to 15 deg. The beneficial effect of negative pitch-flap coupling ratio θ_β (positive δ_3) is evident for both sets of limit curves. This result is for a rigid blade only, and blade flexibility in torsion and bending has a very significant effect, as will be shown later.

By use of the linearized flapping equations as given in Ref. 36, it is possible to define a flapping stability margin with the help of Floquet's theory of linear differential equations with periodic coefficients. The transient solution to such a system of equations can be written in the form

$$\{x\} = [A(t)]\,[\alpha_k \exp(\lambda_k + i\omega_k)] \qquad (5\text{-}23)$$

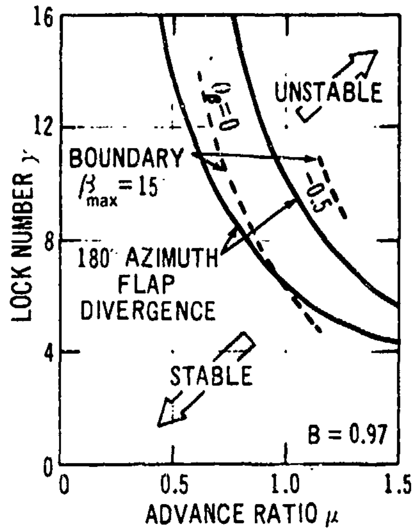

Fig. 5-21. Frozen Azimuth Flapping Instability Boundary

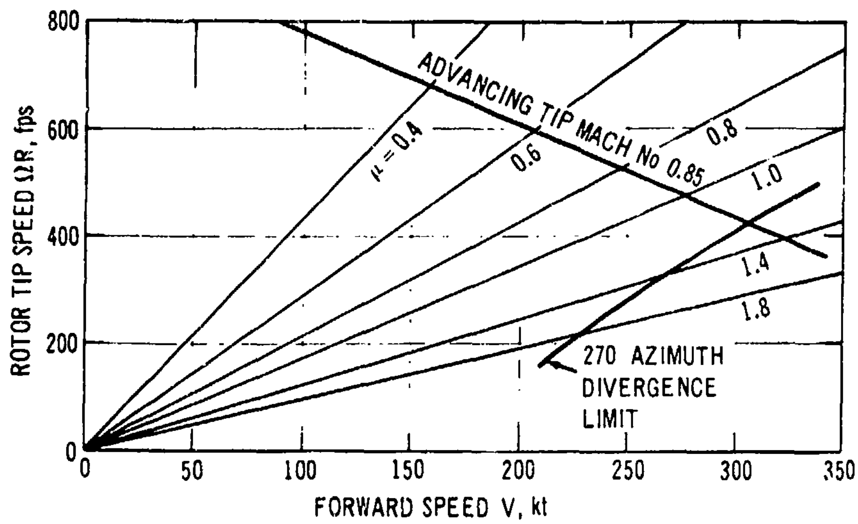

Fig. 5-20. Boundary from Advancing Tip Mach No. 0.85 and 270-deg Azimuth Divergence Limit for Torsional Blade Stiffness Parameter $2GJ/(2\pi\rho c^2 R^2)$ = 11,000(fps)^2

where

$\{x\}$ = column matrix of state variables

$[A(t)]$ = matrix of periodic time functions

α_k = constants derived from initial conditions

$\exp(\lambda_k + i\omega_k)$ = complex eigenvalues of the transition matrix $[Q]$ defined by

$$\{x(T)\} = [Q]\{x(0)\} \tag{5-24}$$

where

$T = 2\pi/\Omega$ is the period, sec.

The eigenvalue of $[Q]$ with the largest absolute value corresponds to the least stable mode. If the absolute value of an eigenvalue is larger than one, the corresponding mode is unstable. The real part λ also can be used as a stability measure, with negative λ indicating stability and positive λ instability. A practicable way of computing the transitional matrix $[Q]$—also applicable for multiblade stability problems—is given in Ref. 38. Fig. 5-22 taken from Ref. 38 shows lines of constant λ in a $\gamma - \mu$ plot for an articulated blade. The dashed lines separate three regions where $\omega/\Omega = 1/2$, $\omega/\Omega = 1.0$ and where ω/Ω varies. The dash-dot curve represents a ridge line. If, at a given γ, the advance ratio μ is increased beyond this line, the damping $-\lambda$ deteriorates rapidly until the blade motion becomes unstable at $\lambda = 0$. Comparing this ridge line with the 180-deg azimuth flapping divergence limit for $\theta_\beta = 0$ of Fig. 5-21, it is seen that excellent damping of the blade transient motion is retained considerably beyond the 180-deg frozen azimuth flapping stability boundary.

A hingeless rotor blade can be represented approximately by an articulated rigid blade with an elastic hinge restraint. An example of the effect of elastic hinge restraint upon the flapping stability limit is shown in Fig. 5-23 taken from Ref. 36. The parameter P is the ratio of blade natural frequency with elastic restraint to that without restraint, and $P^2 - 1$ is proportional to the hinge spring constant. The boundary in Fig. 5-23 is for an advance ratio of $\mu = 2.4$; it shows that the instability is most severe at a blade inertia number (Lock number) of $\gamma = 8$ and that, for an elastic root restraint parameter $(P^2 - 1) > 0.7$, no instability is possible at $\mu = 2.4$.

An example of the effects of torsional flexibility, flap-bending flexibility, and pitch-flap coupling ratio θ_β on the flapping stability of an articulated blade is shown in Fig. 5-24 taken from Ref. 39. Linearized aerodynamics, including reversed flow effects but excluding compressibility effects, have been used; and time histories after a disturbance have been computed. An amplitude ratio of subsequent oscillations of 1.0 represents the stability boundary. For most cases the frequency of the unstable mode is $\Omega/2$; for some cases it is Ω. The 270-deg azimuth torsional divergence limit also is shown and agrees well with the stability limit including torsional flexibility. The natural frequencies of the torsional and flap-bending modes are $\omega_t = 10\Omega$ and $\omega_b = 2.5\Omega$, respectively. In spite of the high torsional stiffness, the flapping stability limit is reduced substantially by including torsional flexibility. Also, the effect of pitch-flap coupling θ_β is reversed due to torsional flexibility, $-\theta_\beta$ improving the stability of the torsionally rigid blade but reducing the stability limit of the torsionally flexible blade. The inclusion of flap-bending flexibility has little effect when $\theta_\beta = 0$ but further reduces the stability in the presence of pitch-flap coupling $\theta_\beta < 0$. The latter effect is caused by an increase in blade pitch with up-blade deflection in the blade tip region, because an up-tip deflection corresponds in the first flap-bending mode to a down-slope of the blade centerline at the hinge. In spite of the fact that pitch-flap coupling is beneficial for the flapping stability of a rigid blade, it actually is detrimental to the flapping stability in all practicable blade designs.

However, the results of Ref. 39, without flap-bending, have been confirmed in Ref. 40, which shows for some examples that dynamic instability for uncoupled blade torsion occurs at an advance ratio only slightly higher than that for 270-deg azimuth torsional divergence.

Ref. 41 reports on a case where an instability with a frequency of $\Omega/2$ at a much lower advance ratio of $\mu = 0.5$ was observed in flight. It appeared to be a limit-cycle phenomenon and the amplitude of the self-excited oscillation was a function of the advancing blade tip Mach number. By feeding some hypothetical nonlinear blade-pitching-moment characteristics into an available, elaborate, nonlinear single-blade computer program, the phenomenon could be reproduced analytically. It was concluded that the instability was caused by statically unstable pitching moments occurring at high subsonic Mach numbers at small or negative blade tip angles of attack. Hence the instability boundary should be increased when thinner blade tip sections are used. The observed instability shows that high subsonic advancing blade tip Mach numbers may be critical with respect to subharmonic self-excited oscillations.

Except for this Mach number effect, both the 180-deg frozen azimuth flapping divergence limit and the 270-deg frozen azimuth torsional divergence limit are useful indicators of severe rotor operational conditions.

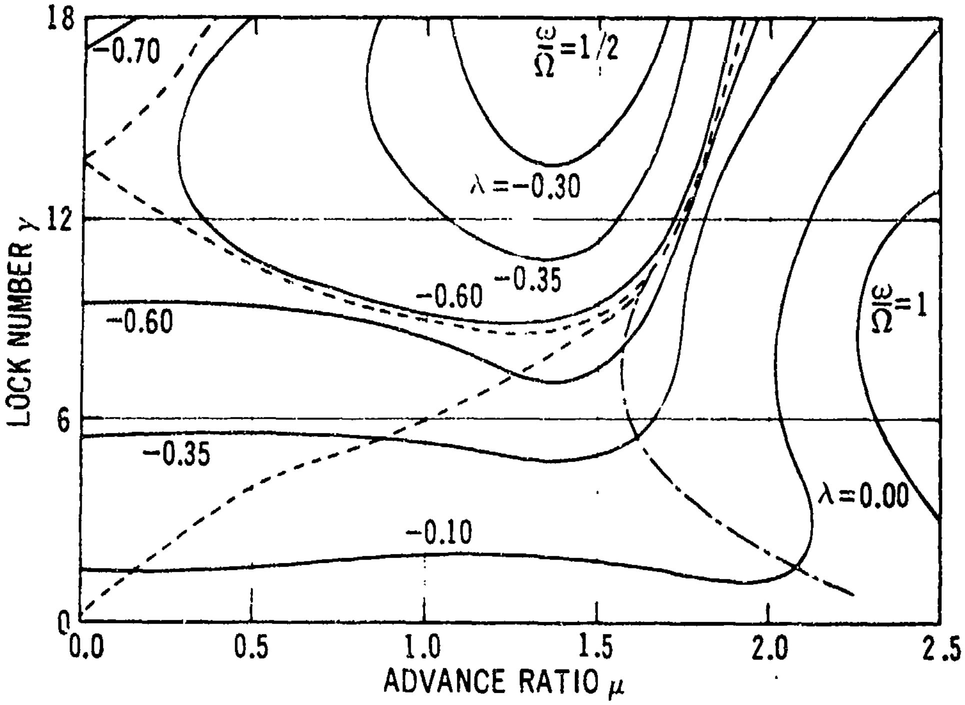

Fig. 5-22. Damping Levels of Articulated Blade, B = 1.00

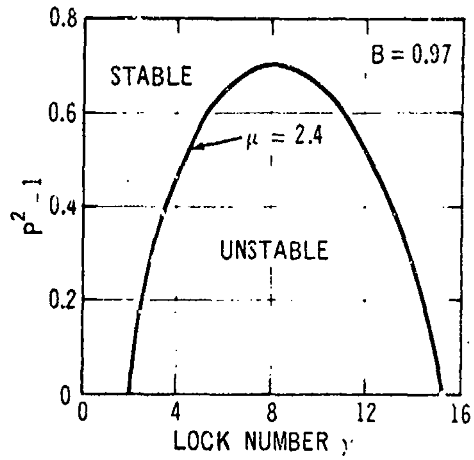

Fig. 5-23. Flapping Stability Boundary for Rigid Articulated Blade With Elastic Root Restraint

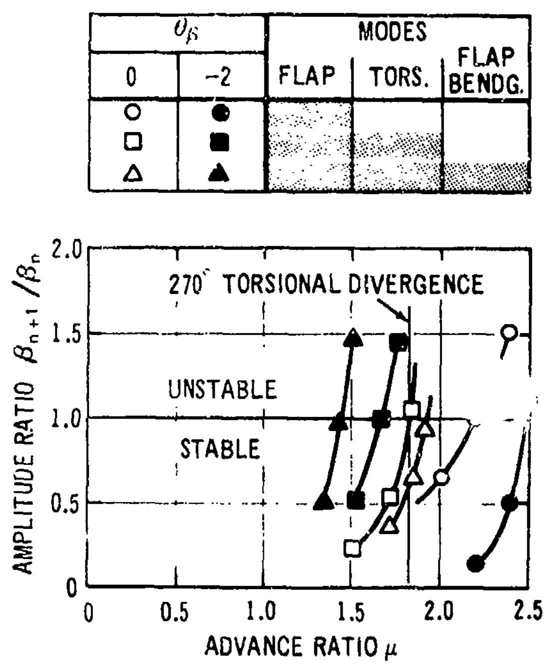

Fig. 5-24. Amplitude Ratio vs Advance Ratio for Articulated Blade, γ = 5 With and Without Torsional and Flap-bending Flexibility

While the 180-deg flapping divergence limit can be exceeded substantially without encountering actual instability, the 270-deg torsional divergence limit is a reasonable approximation for the dynamic instability that may occur earlier, particularly when using a pitch-flap coupling.

5-3.2.3 Single-blade Coupled Pitch-flap-lag Instability

The concept of single-blade lead-lag motions assumes that the shaft rotates with uniform angular velocity. Because the shaft is relatively flexible in torsion, this assumption is not valid for collective lead-lag motions, when all blades move simultaneously fore or aft. However, for cyclic lead-lag motions, for which the resulting shaft torque is zero, the concept of single-blade lead-lag motions is a good approximation if coupling between the blades from horizontal hub motions can be ignored.

Single-blade pitch-flap-lag instability basically is a nonlinear phenomenon and involves the product of flapping angle and rate of flapping. The flapping rate is proportional to the Coriolis acceleration in the plane of rotation. One can linearize the problem by stipulating a mean coning angle β_0 about which comparatively small flapping oscillations take place. The lead-lag oscillation is damped negatively, or amplified, if the lead-lag damping ratio η_ζ satisfies the condition

$$\eta_\zeta < \frac{\beta_0 \theta_\zeta \gamma}{8\bar{\omega}_\zeta} \quad (5\text{-}25)$$

where

$\bar{\omega}_\zeta$ = undamped lead-lag frequency ratio, ω_ζ/Ω, dimensionless

θ_ζ = pitch-lead coupling ratio, $\partial\theta/\partial\zeta$, dimensionless

This stability criterion was established first in somewhat different form (Ref. 42). It is applicable to hinged blades, when ω_ζ is small, and indicates that kinematics that result in an increase in pitch with lead angle should be avoided unless one is sure that adequate lead-lag damping always is present to overcompensate for the negative damping from the pitch-lead coupling. On the other hand, with a negative θ_ζ (increase in blade pitch with lag angle), considerable damping of the lead-lag mode is possible at positive coning (mean flapping) angle β_0 However, the sign of the damping is reversed with a reversal of the sign for β_0, so this method of damping the lead-lag motion is not feasible unless negative mean flapping angles can be avoided or kept small.

Pitch-lag coupling is possible not only from control linkage kinematics but also from torsional flexibility. Assume, for example, a hingeless blade with preconed feathering axis and a low lead-lag natural blade frequency. The vibration mode then has little damping, and elastic coupling effects, although small, may produce instability. Fig. 5-25 shows a rotor operating with less lift than is designed into the preconing angle. The blade is shown in a drooped and leading position. The forward inertia forces and the lift forces have pitch-up moments with respect to the feathering axis, resulting in an elastic pitch-lead coupling that may produce instability.

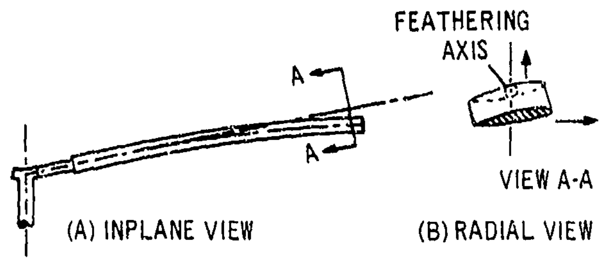

Fig. 5-25. Hingeless Blade in Drooped and Leading Position

For a lead-lag oscillation frequency ratio $\omega_\zeta > 1$, and again assuming the undamped blade flapping frequency ratio $\bar{\omega}_\beta = 0$, the instability criterion of Eq. 5-25 is replaced by

$$\eta_\zeta < \frac{\beta_0 \theta_\zeta \gamma}{8\bar{\omega}_\zeta(1-\bar{\omega}_\zeta^2)} \quad (5\text{-}26)$$

For positive β_0, instability now occurs for negative θ_ζ, or decrease of blade pitch with lead.

The evaluation of the available lead-lag damping ratio η_ζ is difficult. The aerodynamic damping depends upon the shaft power and is larger for powered flight than for autorotational flight. Because of the usual nonlinearity in the damping characteristics, the mechanical lead-lag damping also is difficult to evaluate, particularly in the presence of forced lead-lag motions that can reduce drastically the available damping for the potentially unstable natural lead-lag mode. Some of these effects are analyzed in Ref. 43.

A mild instability also can occur for $\theta_\zeta = 0$, as shown in Ref. 44. When the flapping natural frequency term $\sqrt{1+\omega_\beta^2}$ and the lead-lag natural frequency $\bar{\omega}_\zeta$ are not too different from each other, one obtains the instability criterion

$$\eta_\zeta < \frac{16\beta_0^2\bar{\omega}_\beta^2}{\bar{\omega}_\zeta\gamma} \quad (5\text{-}27)$$

The conclusion obtained in Ref. 44, namely, that instability for hingeless rotors is possible only when blade flapping frequency $\bar{\omega}_\beta \neq 0$, differs from the conclusions of Ref. 45. A simplified expression representing the potentially destabilizing flap-lag coupling is presented in Ref. 45 as

$$F_{\dot{\zeta}} C_{\dot{\beta}} = \left(\frac{\gamma\theta}{8}\right)^2 \left(\frac{3}{2} - \frac{1}{p^2}\right) \frac{1}{p^2} \quad , \text{ d'less} \qquad (5\text{-}28)$$

where, by use of a simple approximation of the inflow parameter,

$$F_{\dot{\zeta}} = \left(\frac{\gamma}{8}\right)\left(\frac{3\theta}{2}\right) - 2\beta_0$$

$$C_{\dot{\beta}} = 2\beta_0 \qquad (5\text{-}29)$$

$$\beta_0 = \left(\frac{\iota}{8}\right)\left(\frac{\theta}{2}\right)\left(\frac{1}{p^2}\right)$$

$$p^2 = \bar{\omega}_\beta{}^2 + 1$$

It can be seen that flap-lag coupling does not vanish when $\bar{\omega}_\beta = 0$ ($p = 1$). The discrepancy between the two conclusions is attributed to the difference in the derivation of the aerodynamic flap and lead moments, as discussed in Ref. 45.

Stability criteria as in Eqs. 5-25, 5-26, 5-27, and 5-28 should be considered only as first approximations for preliminary design purposes. The actual limits require more complete representations of the rotor and a more accurate evaluation of the stability criterion.

5-3.3 COUPLED ROTOR/AIRFRAME INSTABILITY ANALYSES

Most actual rotor instabilities involve essential coupling effects between blades. In the dynamic analysis, individual blade coordinates can be used to evaluate the system of coupled equations for the individual blades, as in Ref. 38. However, it is better to introduce generalized multiblade coordinates, because the mode shapes then are easier to interpret. This allows presentation of the distinction between essential and nonessential modes contributing to an instability. Generalized multiblade lead-lag coordinates first were introduced in Ref. 46 in the form of complex numbers

$$\phi_j = \frac{i}{n} \sum_{k=0}^{n-1} \eta_k e^{ij2\pi k/n} \qquad (5\text{-}30)$$

$$\phi_j^* = -\phi_{n-j}$$

where η_k is the deflection of the kth blade.

It is shown in Ref. 46 that, for the problem of ground resonance, only the equation for ϕ_1 is coupled with the rotor support equation of motion, while the equations for ϕ_j, j being other than 1 or $n - 1$, are uncoupled. The solution for ϕ_1 provides what were called the first advancing and the first regressing modes in par. 5-3.1. However, for problems other than ground resonance, forward flight rotor aerodynamics generates coupling terms between all of the multiblade mode equations.

From a point of view of easy visualization of the multiblade modes, it is preferable to use real rather than complex multiblade coordinates. The deflection of the kth blade then is given in terms of such coordinates η_i by

$$\eta_k = \eta_0 + \eta_d(-1)^k + \eta_I \cos\psi_k + \eta_{II}\sin\psi_k + \eta_{III}\cos(2\psi_k) + \eta_{IV}\sin(2\psi_k) + \cdots \qquad (5\text{-}31)$$

$$k = 0, 2, \ldots, n-1$$

where

ψ = blade azimuth angle, rad

For n blades the n generalized multiblade coordinates are taken as the first n of the quantities η_0, η_d, η_I, η_{II}, η_{III}, η_{IV}, whereby the differential collective coordinate η_d occurs only in even-bladed rotors. Inverting Eq. 5-31 for n blades one obtains

$$\eta_0 = \frac{1}{n}\sum_{k=0}^{n-1} \eta_k$$

$$\eta_d = \frac{1}{n}\sum_{k=0}^{n-1} (-1)^k \eta_k \qquad (5\text{-}32)$$

$$\eta_I = \frac{2}{n}\sum_{k=0}^{n-1} \eta_k \cos\psi_k$$

$$\eta_{II} = \frac{2}{n}\sum_{k=0}^{n-1} \eta_k \sin\psi_k \qquad (5\text{-}33)$$

$$\eta_{III} = \frac{2}{n}\sum_{k=0}^{n-1} \eta_k \cos(2\psi_k)$$
$$\eta_{IV} = \frac{2}{n}\sum_{k=0}^{n-1} \eta_k \sin(2\psi_k) \tag{5-34}$$

where

$$\psi_k = \Omega t + \frac{2\pi k}{n}, \quad k = 0, 1, \ldots, n-1 \tag{5-35}$$

The cyclic modes η_I, η_{II}, η_{III}, η_{IV}, etc., are seen as nonrotating modes. For example, if η represents flapping, η_I would be fore and aft tilting, η_{II} sidewise tilting, η_{III} warping about longitudinal and transverse axes, and η_{IV} warping about the axes ± 45 deg to the longitudinal axis. In the absence of aerodynamics, η_0 and η_d would represent uncoupled normal modes and (η_I, η_{II}), (η_{III}, η_{IV}) would represent pairs of uncoupled normal modes. If we assume for the 0th blade the harmonic motion cos ωt, the first regressing or advancing modes are given by

$$\eta_k = \cos\left(\omega t \pm \frac{2\pi k}{n}\right) \tag{5-36}$$

Inserting Eq. 5-36 into Eq. 5-33, one obtains

$$\eta_I = \cos(\Omega \pm \omega)t$$
$$\eta_{II} = \sin(\Omega \pm \omega)t \tag{5-37}$$

In terms of the cyclic coordinates η_I and η_{II} the first regressing mode is indicated by the frequency $\Omega - \omega$, the first advancing mode by the frequency $\Omega + \omega$.

Multiblade modes, as defined by Eqs. 5-31 through 5-35 can be computed for the fundamental and the higher individual blade modes in torsion, flap-bending, or lead-lag bending. In a final analysis, coupling of these multiblade modes with the main fuselage modes must be included.

5-3.3.1 Multiblade Flutter

No multiblade classical flutter analysis has been published to date. According to the concept of multiblade generalized coordinates, there would be collective flutter or torsional divergence and cyclic flutter in the advancing or regressing modes. At zero forward flight speed it would not be difficult to analyze the multiblade flutter conditions. At forward speed not even the problem of single-blade flutter has been solved satisfactorily. The frozen azimuth flutter analysis could be extended to include coupling effects among the blades and with the airframe. However, the concept of multiblade generalized coordinates would not be applicable because, in the case of fictitious frozen azimuth, the various blades would not oscillate with the same amplitude and with $2\pi k/n$ phase differences. Nevertheless multiblade flutter is not a design consideration, although the problem is of academic interest.

5-3.3.2 Multiblade Flapping Instability

Multiblade flapping instability is encountered at high rotor advance ratio, both for axial flow—as in the case of a prop-rotor—and in oblique flow, as in the case of a compound helicopter. Instability for axial flow often is called whirl flutter, although blade torsion essentially is not involved. In fact, this type of instability is possible for entirely rigid rotor blades. In whirl flutter studies, the expressions "retrograde whirl" and "progressive whirl" refer to an airframe-fixed coordinate system. A regressing mode in the previously defined sense, therefore, can be either a retrograde or a progressive mode, depending upon the sign of $\Omega - \omega$. An advancing mode, however, always is a progressive mode.

5-3.3.2.1 Axial Flow

Ref. 47 presents an exhaustive study of axial-flow multiblade flapping instability. The assumptions are that rigid blades are connected to the hub at its center by elastically restrained flapping hinges, and that the horizontal rotor shaft can perform elastically restrained angular pitching and yawing motions about a pivot, or nodal point, 0 located downstream of the rotor (see Fig. 5-26). Refs. 47 and 48 give an interpretation of this work in simplified terms. At high rotor advance ratio μ, there are substantial aerodynamic negative spring and negative damping moments for the nacelle angular motions about the nodal point, leading to static divergence or dynamic instability. The system is treated as one with four degrees of freedom: the two first cyclic multiblade flapping modes and the two nacelle modes with the angular deflections ϕ_x and ϕ_y. For a pivot ratio of $h/R = 0.31$, a rotor to nacelle effective inertia ratio of 0.27, and an axial flow rotor advance ratio $\mu = 1.0$, Fig. 5-27, taken from Ref. 47, shows stability boundaries in terms of nacelle-pitching natural frequency ω_{ϕ_x}/Ω and undamped blade-flapping frequency ω_β/Ω for a variety of nacelle yawing to nacelle pitching frequency ratios $\omega_{\phi_y}/\omega_{\phi_x}$. The areas above the boundaries represent stable conditions; the solid lines indicate oscillatory instability; the dashed line indicates static divergence. The unstable mode is a retrograde (in the airframe-fixed coordinate system) circular nacelle motion about the nodal point with relatively little blade flapping. The nacelle stiffness requirement at the stability boundary reduces with decreasing blade flapping natural frequency until intersection with the static divergence limit, when the trend is reversed. In the region $1.1 \leq \omega_\beta/\Omega \leq 1.3$, which is typical of prop-rotors, the static divergence limit is not much above the oscillatory instability limit. The static divergence limit may be used in preliminary design if whirl flutter is a primary criterion in the determination of wing stiffness. For blade flapping frequencies $\omega_\beta/\Omega < 1.1$, the unstable mode changes to a progressive (in the airframe-fixed reference system) circular motion of the nacelle about the nodal point with large blade flapping content. The nacelle stiffness requirements at the stability boundary then are increasing steeply with decreasing blade flapping frequency ω_β/Ω. While this indicates that, from a stability viewpoint, a flapping frequency of about $\omega_\beta/\Omega = 1.1$ would be best, Ref. 48 points out that to avoid excessive flapping deflections or flap-bending stresses, for the dynamic system studied, higher values for ω_β/Ω may be necessary. Also, the analysis in Ref. 47 neglects effects of preconing, of elastic blade torsion, and of elastic inplane blade bending—all of which may modify appreciably the results shown in Fig. 5-27. The pivot distance h/R is destabilizing and the differences in stability boundaries power-on and power-off are small. The inclusion of nonlinear aerodynamic terms neglected in the analysis for Fig. 5-27 leads, in case of the progressive mode, to a nondivergent limit cycle.

The analysis of Ref. 49 is for a two-bladed teetering rotor in axial flow, and the results are compared to wind tunnel test results. Pitch-flap coupling is included and has a substantial effect. Instabilities are possible in both the retrograde and progressive modes. The nacelle mode, with little blade-flapping content, can be suppressed with moderate nacelle damping. The unstable rotor cyclic flapping mode, occurring at much lower frequencies, is the predominant instability at high rotor advance ratio for configurations with high negative pitch-flap coupling ratios, and anisotropic pylon stiffness. The stability of this mode can be influenced by swashplate-nacelle displacement coupling. In Ref. 49 the problem was treated while both including and excluding nonlinearities, and agreement was found in the stability boundaries.

The whirl flutter phenomenon has been investigated quite extensively, both analytically and experimentally. Typical of these studies are Refs. 50-55, in which the effects of nacelle stiffness and damping, hub stiffness, pitch-flap coupling, and flapping hinge offset have been examined. A review of all available data shows that

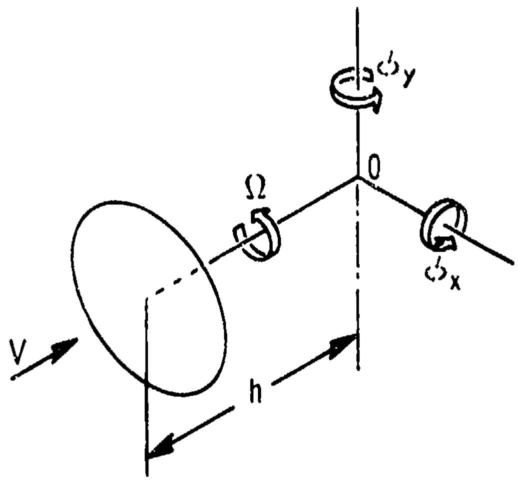

Fig. 5-26. Prop-rotor Pitching and Yawing About Pivot

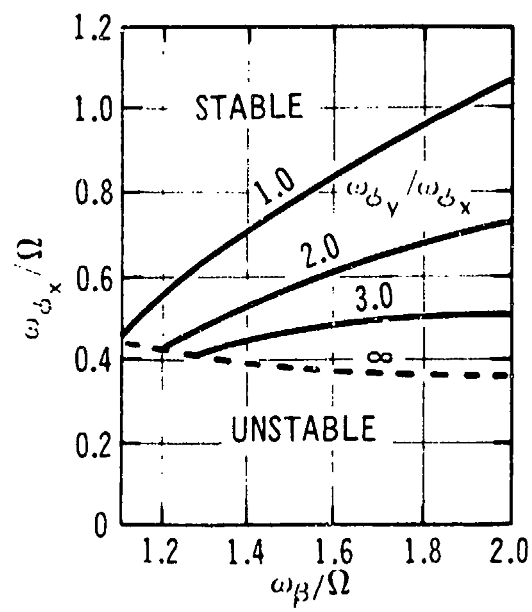

Fig. 5-27. Stability Boundaries for System of Fig. 5-26

both retrograde and progressive modes are possible with symmetrical or unsymmetrical nacelle stiffness configurations, depending on whether the pitch-flap coupling ratio is positive or negative. Also, for $\omega_{\phi_y}/\omega_{\phi_x} > 1$, a resonance type coupling destabilizes the system when $\omega_{\phi_y} \sim \Omega + \omega_\beta$. The effect of structural damping on the flutter modes that is given is Refs. 47-49 is verified, and the parameter combinations that generate bimodal instability, or simultaneous progressive and retrograde modal flutter, can be identified.

Ref. 56 describes a study similar to that of Ref. 47 except that hinge offset, flapping restraint, and pylon isotropy were varied. The results of Ref. 47 were extended to show that the flapping frequency for optimum pylon stiffness design is invariant for combinations of hinge offset and flapping restraint, thus confirming the equivalence of these two parameters. The optimum flapping frequency was found to be dependent upon pylon isotropy. Design criteria recommended to avoid whirl flutter are discussed in par. 5-4.5.

5-3.3.2.2 Oblique Flow

Multiblade flapping instability for oblique rotor flow can be investigated by means of a linearized blade flapping equation, written in a rotor-fixed reference system and ignoring reversed flow effects. The periodic forcing functions applicable to such an equation are given in Ref. 36. Introducing multiblade coordinates, as defined by Eqs. 5-31 through 5-35, and assuming feedback relations between the multiblade flapping coordinates β_I, β_{II}, and the forward and left cyclic pitch coordinates θ_I, θ_{II}, respectively, of the form,

$$\begin{aligned} \dot{\theta}_I + L\theta_I &= -K\beta_I \\ \dot{\theta}_{II} + L\theta_{II} &= -K\beta_{II} \end{aligned} \qquad (5\text{-}38)$$

where L and K are rotor tilt feedback constants, one obtains five coupled equations for a three-bladed rotor and six coupled equations for a four-bladed rotor for β_0, β_d (only for four-bladed), $\beta_I, \beta_{II}, \theta_I$, and θ_{II}.

The equations for the three-bladed rotor (except for Eq. 5-38) have periodic coefficients $\sin 3\Omega t$ and $\cos 3\Omega t$; those for a four-bladed rotor have periodic coefficients $\sin 2\Omega t$, $\cos 2\Omega t$, $\sin 4\Omega t$, and $\cos 4\Omega t$. Fig. 5-28, taken from Ref. 57, shows the root locus plots ω versus λ for constant $L = 0.1$ and varying integral feedback constant K, comparing the cases of three blades, four blades, and where the terms with periodic coefficients have been neglected. The latter case uncouples the β_d mode from the remaining modes so that there is no difference between three and four blades. Fig. 5-28 should be considered only as a trend study, because at the advance ratio $\mu = 0.8$, reversed flow effects are not negligible and have, for rigid blades, a stabilizing influence. The periodic coefficient cases have been computed with the Floquet transition matrix method of Ref. 38 discussed in par. 5-3.2.2. According to this method one obtains for the parameters of the example ($\mu = 0.8$, $\gamma = 8$, $P = 1.15$, $B = 0.97$) for a single blade two flapping modes with the same frequency and two damping values $\gamma_1 = -0.26$ and $\gamma_2 = -0.63$. The time scale has been selected so that $\omega = 1$ means a flapping frequency of once per revolution. The matrix of periodic time functions is periodic with frequency $\omega = 1$, so the values of ω_k are defined only for additive terms $\pm n$, $n = 1,2,3,\ldots$ Of these many possibilities, we select the two flapping eigenvalues $\exp(-0.26 + i)$ and $\exp(-0.63 + i)$. Combining several uncoupled blades, collective and differential collective modes have the same eigenvalues; the advancing mode has the eigenvalues $\exp(-0.26 + 2i)$; $\exp(-0.63 + 2i)$; and the regressing mode has the eigenvalues $\exp(-0.26)$, $\exp(-0.63)$. Introducing integral feedback with the gain factor K defined by Eq. 5-38 and $L = 0.1$, the blades become coupled and two more modes are added because of the feedback system of Eq. 5-38. The three-bladed rotor with feedback system, therefore, has eight modes; the four-bladed rotor, 10 modes. In Fig. 5-28 all curves start with a gain factor $K = 0.2$. Therefore an additional mode associated with each of the two lower sets of curves is not shown. A complete presentation would include branch points at $\omega = 0$ and extensions into the region representing negative values of frequency ω.

The labels "advancing", "collective", and "regressing" mode can be used only for the uncoupled or weakly coupled blade motions; with coupling the modes become mixed. Of particular interest is the mixing of the lowly damped advancing and collective modes for a three-bladed rotor, resulting in a branch point at $\omega = 1.5$ and a subsequent separation in a mode with increasing damping and one with decreasing damping, both at $\omega = 1.5$, the latter being unstable at $K = 0.8$. No such mixing of these two modes occurs in a four-bladed rotor, which also is unstable at $K = 0.8$ but in a different mode and with a different ω. Omitting the periodic terms in the equations of blade motion, as is done sometimes for the sake of simplicity, washes out the difference between three and four blades. Also, there are only one mode for a single blade and only four modes for a complete rotor including the feedback system (Eq. 5-38), as indicated by the dashed lines in Fig. 5-28. The constant coefficient system does

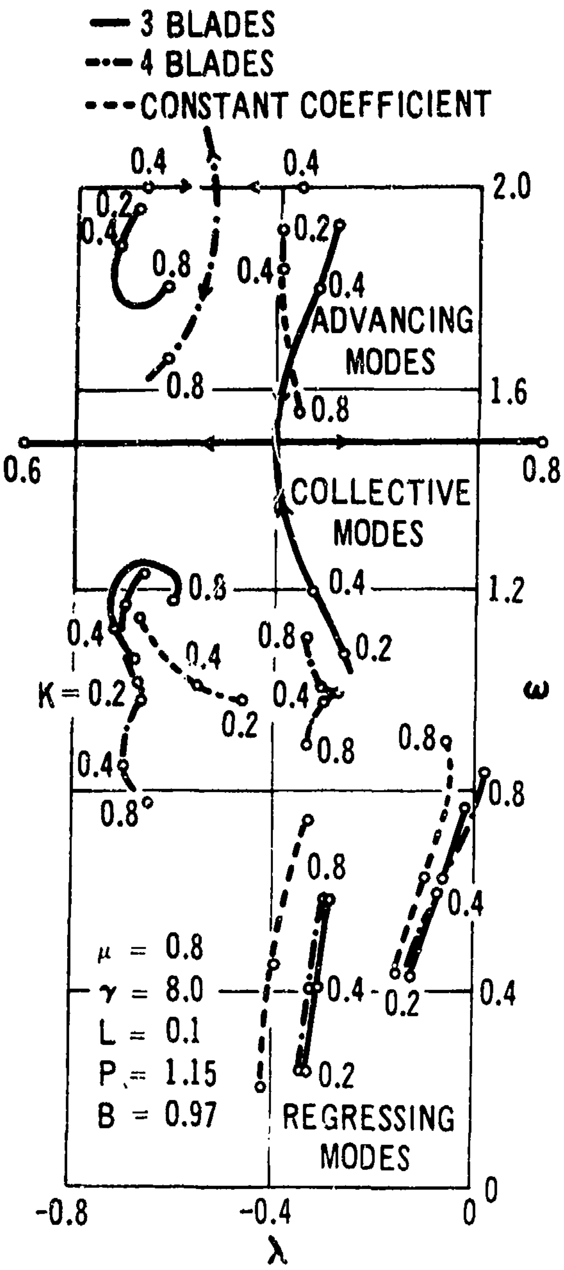

Fig. 5-28. Multiblade Root Plots for Three- and Four-bladed Rotors With Integral Tilting Moment Feedback

not result in an instability up to $K = 0.8$ and therefore is unconservative.

Fig. 5-28 is for hinged rigid blades with elastic flapping restraints, for rigid hub support, and for rigid controls. Fig. 5-24 and Refs. 39 and 40 show the substantial influences of torsional blade or control flexibility, of pitch-flap coupling, and of blade flap-bending. Other effects that must be included in a complete analysis are coupling with elastic and rigid-body airframe modes and with multiblade lead-lag modes.

5-3.3.3 Coupled Rotor-Airframe Lead-lag Instability

Ground resonance has been treated classically as self-excited blade lead-lag oscillations when the rotor center is restrained elastically in the x-direction and rigid in the y-direction (Fig. 5-29). The analysis follows Ref. 46 in the improved form of Ref. 58. Usually the lateral direction is the critical one, so that the x-direction will be sidewise. It is not difficult to extend the analysis to include the y-motion of the rotor center. Both blades and airframe are assumed to have viscous damping with the lead-lag damping ratio η_ζ and lateral airframe η_x. The blades are hinged at the distance e from the rotor center. In a reference system moving with the hub center, there is an inertia force $-dm\ddot{x}$ acting upon the blade mass element dm that has, in the lead direction, a moment with respect to the hinge of $\ddot{x}S \sin\psi_k$, S being the static mass moment of the blade about the hinge. The equation of blade motion in the rotor fixed reference system then is

$$\ddot{\zeta}_k + 2\omega_\zeta\eta_\zeta\dot{\zeta}_k + \omega_\zeta^2\zeta_k = \frac{\ddot{x}S}{I}\sin\psi_k \qquad (5\text{-}39)$$

with the undamped blade lead-lag natural frequency

$$\omega_\zeta = \Omega\sqrt{\frac{eS}{I}} \qquad (5\text{-}40)$$

where

ζ_k = lead angle of kth blade, rad

I = mass moment of inertia of blade about lag hinge, slug-ft²

The equilibrium of forces on the hub center in the x-direction is expressed by the equation

$$\ddot{x} + 2\omega_x\eta_x\dot{x} + \omega_x^2 x = \frac{S}{M}\sum_k\left[(\ddot{\zeta}_k - \Omega^2\zeta_k)\sin\psi_k + 2\Omega\dot{\zeta}_k\cos\psi_k\right] \qquad (5\text{-}41)$$

The right-hand side of Eq. 5-41 represents the forces of the blades upon the rotor center in the

x-direction—the first term from angular inertia $-dm\ddot{\zeta}_k$; the second term from the centrifugal force component in the lead direction, and the third term from the variation of centrifugal force due to ζ_k. Higher order terms are neglected because ζ_k and e are assumed to be small. M is the equivalent airframe mass at the rotor center that includes the blade masses. We now introduce the cyclic multiblade lead coordinates

$$\left.\begin{aligned} \zeta_I &= \frac{2}{n}\sum_k \zeta_k \cos\psi_k \\ \zeta_{II} &= \frac{2}{n}\sum_k \zeta_k \sin\psi_k \end{aligned}\right\} \qquad (5\text{-}42)$$

Considering for n blades, $n > 3$, one has the relations

$$\left.\begin{aligned} &\sum_k \sin\psi_k \cos\psi_k = 0 \\ &\sum_k \sin^2\psi_k = \sum_k \cos^2\psi_k = n/2 \end{aligned}\right\} \qquad (5\text{-}43)$$

The inverses of Eq. 5-42 are

$$\zeta_k = \zeta_I \cos\psi_k + \zeta_{II}\sin\psi_k, \quad k = 0, \ldots, n-1 \qquad (5\text{-}44)$$

Multiplying Eq. 5-39 first by $\sin\psi_k$ and then by $\cos\psi_k$, and summing each equation over k, one obtains with Eq. 5-43

$$\ddot{\zeta}_{II} + 2\omega_\zeta\eta_\zeta\dot{\zeta}_{II} - (\Omega^2 - \omega_\zeta^2)\zeta_{II} - 2\Omega\zeta_I - 2\omega_\zeta\eta_\zeta\Omega\zeta_I = \frac{\ddot{x}S}{I} \qquad (5\text{-}45)$$

$$\ddot{\zeta}_I + 2\omega_\zeta\eta_\zeta\dot{\zeta}_I - (\Omega^2 - \omega_\zeta^2)\zeta_I + 2\Omega\dot{\zeta}_{II} + 2\omega_\zeta\eta_\zeta\Omega\zeta_{II} = 0 \qquad (5\text{-}46)$$

Performing the summations over k on the right-hand side of Eq. 5-41, one obtains

$$\ddot{x} + 2\omega_x\eta_x\dot{x} + \omega_x^2 x = \frac{nS}{2M}\ddot{\zeta}_{II} \qquad (5\text{-}47)$$

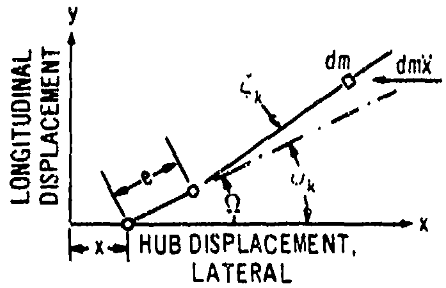

Fig. 5-29. Definition of Parameters for Ground Resonance Analysis

The homogeneous system of Eqs. 5-45 and 5-46 has a sixth order characteristic equation with three conjugate complex roots. The combination of the cyclic modes ζ_I, ζ_{II} representing a regressing blade mode can become unstable under certain conditions. Fig. 5-30 taken from Ref. 57 shows boundaries of blade and airframe damping ratios η_ζ and η_x versus nondimensional rotor speed Ω/ω_x at the limit of stability. If either η_ζ or η_x is smaller than indicated by a point on one of the curves, instability occurs. Maximum damping is required at a rotor speed $\Omega/\omega_x = 1/(1 - \omega_\zeta/\Omega)$, for which the sum of blade natural frequency ω_ζ and airframe natural frequency ω_x is equal to Ω. The plots in Fig. 5-30 refer to $(\omega_\zeta/\Omega)^2 = 0.1$ and $nS^2/(2MI) = 0.025$. Corresponding sets of curves for $(\omega_\zeta/\Omega)^2 = 0.0625$ and 0.090 in combination with $nS^2/(2MI) = 0.02, 0.04, 0.06, 0.08$, and 0.10 are given in Ref. 17. For a two-bladed rotor the same problem would result in differential equations with periodic coefficients and could be solved with the Floquet transition matrix method discussed in par. 5-3.2.2 in a simpler way than in Ref. 46.

Coupled airframe/rotor lead-lag instability also can occur in the air. In this case aerodynamic effects become important and should be included in the analysis. Aerodynamic effects provide a coupling between lead-lag and flapping motions, and also provide positive or negative damping for the airframe motion and for the blade motions. Fig. 5-31 shows, schematically, the lowest airframe frequency ω_x on the ground and in the air; and $(\Omega - \omega_\zeta)$ for hinged, hingeless inplane soft, and hingeless inplane-stiff blades plotted versus rotor speed. The shaded areas refer to potential instability regions of the rotor speed centered around the intersection of an ω_x with an $(\Omega - \omega_\zeta)$ curve. Similar curves can be shown for the $(\Omega + \omega_\zeta)$ mode. The analysis should include the instability regions around the intersection of ω_x with this mode as well. For hinged blades the regions associated with the $(\Omega -$

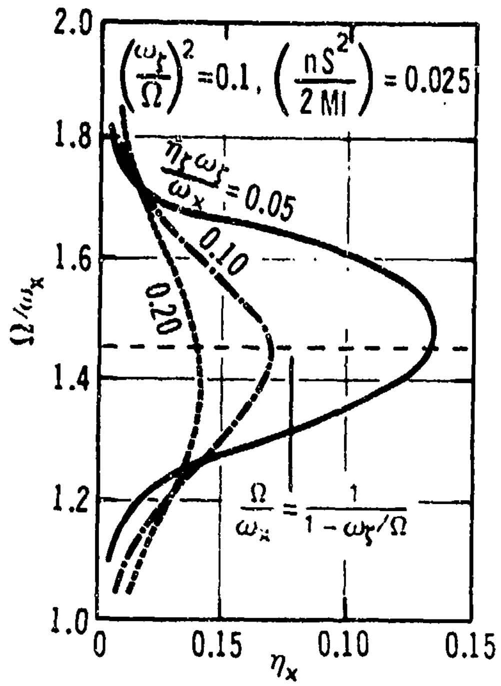

Fig. 5-30. Damping Ratios for Blade and Airframe at Ground Resonance Stability Limit

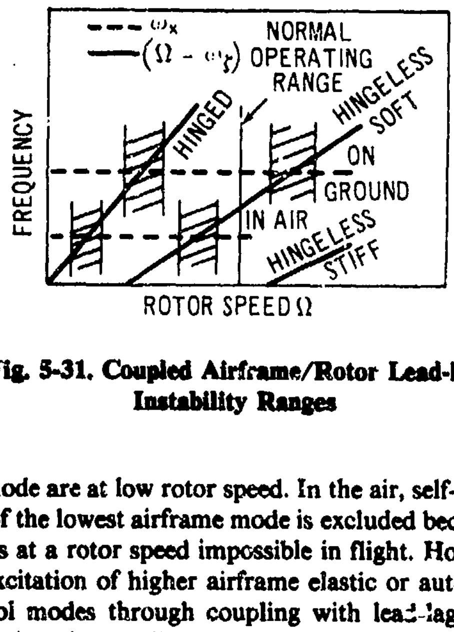

Fig. 5-31. Coupled Airframe/Rotor Lead-lag Instability Ranges

ω_ζ) mode are at low rotor speed. In the air, self-excitation of the lowest airframe mode is excluded because it occurs at a rotor speed impossible in flight. However, self-excitation of higher airframe elastic or automatic control modes through coupling with lead-lag blade modes is quite possible and must be investigated.

For a hingeless inplane-soft blade, Fig. 5-31 indicates potential unstable rotor speed regions both on the ground at a higher than normal rotor speed and in the air at a lower than normal speed. Because of the low damping ratio of a hingeless blade (in the order of 0.02), the avoidance of actual instabilities in these regions is difficult. Ref. 59 compares analytical and flight test results for such a case. Although the instability at about 85% of normal rotor speed could not be removed, the analysis predicted the measured damping ratios of the potentially unstable mode quite well. Ref. 60 reports on a similar case where self-excitation could be avoided. The aerodynamic damping of the helicopter in roll is an important parameter in suppressing the potential instability and must be evaluated carefully. Because of the low structural blade damping, it is important to consider many design details that influence the aerodynamic damping of the lead-lag mode, such as preconing, droop, sweep, and both elastic and kinematic pitch-flap and pitch-lead coupling.

As indicated in Fig. 5-31, a hingeless inplane stiff blade does not have potential instability regions at or below normal rotor speed either on the ground or in the air, this applies to those instabilities caused by coupling of airframe modes having horizontal hub motion with blade lead-lag modes.

However, unstable lead-lag modes of the kind discussed in par. 5-3.2.3 well may occur in the operating speed range of an inplane-stiff hingeless rotor, and the same attention to all the design details mentioned previously is necessary in order to avoid such instabilities. For example, Ref. 61 reports on a case in which a lead-lag instability observed on an inplane-stiff rotor could be removed by reversing the elastic pitch-lead coupling from negative to positive (see also Eq. 5-26).

Ref. 62 discusses another case of lead-lag instability of an inplane stiff rotor that occurs at high axial inflow in the prop-rotor mode of operation if the pitch-flap coupling ratio θ_β is negative. In this case, the inplane blade mode contains, at high axial inflow ratio and the associated high blade pitch setting, a rather strong out-of-plane deflection. This deflection, through the negative θ_β coupling ratio, produces the equivalent of a negative θ_ζ coupling ratio, thus destabilizing the lead-lag motion, according to Eq. 5-26. The remedy suggested in Ref. 62 is the use of a positive value of pitch-flap coupling ratio θ_β; the result is to lower the flapping divergence limit and to lower the reversed-flow, 270-deg azimuth, torsional divergence limit, but the desired effect of reducing flapping at high axial inflow ratio is achieved, as with negative θ_β.

5-3.4 EFFECT OF ROTOR CONFIGURATION ON INSTABILITIES

While some of the types of instability can occur in any rotor configuration, there are others that are characteristic of particular rotor configurations. The main rotor configurations and their associated potential instabilities are discussed in the following paragraphs.

5-3.4.1 Fully Hinged Rotor

If the feathering axis is aligned with the blade axis, as is customary, the effects of inplane or out-of-plane bending are minimized. In rotors with fully hinged blades, ground and air resonance must be avoided by providing adequate lead-lag damping. Because of the large forced lead-lag amplitudes in forward flight, linear blade dampers will cause excessive blade loads. Therefore, damping force limiters in the form of friction devices or relief valves in hydraulic dampers are necessary. The associated nonlinearity of the damping force makes the protection against ground or air resonance amplitude-dependent, and it is necessary to assess the largest amplitude for which the protection is desired. If this assessment is in error and larger amplitudes occur in operation, the helicopter is endangered.

In hinged rotors, kinematic pitch-flap and pitch-lead coupling ratios vary with rotor power and thrust, and particular care is necessary to insure that instability cannot occur under any operational condition. In this respect, critical conditions are those of large forced lead-lag oscillations that leave small amounts of damping for superimposed, potentially self-excited lead-lag oscillations. Because the blade dampers are essential for suppression of self-excitation, their reliability is very important. The same is true of the oleo landing gear struts.

High advance ratio flapping instability of hinged blades occurs earlier than for hingeless blades, and torsional or control flexibility—in combination with negative pitch-flap coupling—is particularly effective in lowering the stability limit. Nonlinear effects (from Coriolis accelerations when flapping about a coned position) are unlikely to produce instability in fully hinged-blade rotors. Other instabilities that have been observed on fully hinged-blade rotors, such as the high Mach number/low tip angle-of-attack instability or the vortex-wake-induced instability of rotors with many blades, could occur on other rotor configurations.

5-3.4.2 Semihinged Rotor

The configuration in which the blades have only flapping hinges but no lead-lag hinges has not been used often, but may have some appeal in the future. If the feathering hinge is aligned with the blade axis, the potentially unstable Coriolis acceleration coupling is avoided. In order to prevent large inplane moments from the forced lead-lag response in forward flight, the blades should be inplane soft, with the major flexibility inboard of the feathering hinge. Otherwise, a destabilizing pitch-lead coupling will result. The ground and air resonance problem of this configuration is a difficult one and requires, in addition to some damping of the inplane blade mode, careful attention to the airframe modes that might couple with cyclic inplane blade modes.

5-3.4.3 Seesaw Rotor

The widely used two-bladed seesaw, or teetering, rotor has some advantages with respect to instabilities over rotors with three or more blades. The lowest inplane blade mode is rather high because the rotor center is an antinode. Therefore, it is practicable to place the lowest inplane mode frequency above the frequency of rotor revolution, thus avoiding the equivalent of mechanical instability. At high advance ratio there is no 180-deg frozen azimuth flapping divergence, because the flapping moments on the fore and aft blades cancel each other. The two-bladed seesaw rotor thus is capable of operation at considerably higher advance ratios than the hinged rotor with three or more blades. There are, however, a number of possible coupling effects that may produce an instability (called weaving) under certain operational conditions. Careful attention to the detail design is necessary in order to avoid this instability.

5-3.4.4 Floating Hub Rotor

The floating hub configuration is the extension of the seesaw principle to three or more blades. It has been used in various ways and may have a future application, particularly with respect to tip-jet drive or to prop-rotors. The main advantage of the floating hub configuration is that, due to the alignment of the hub with the tip path plane, the forced inplane deflections and moments of the blades are kept small. It requires considerably more inplane blade stiffness than does the seesaw rotor in order to allow the lowest inplane natural frequency to be higher than the frequency of rotor revolution. Floating hub rotors have been built both with inplane soft and with inplane stiff blades. In the latter case, care *shall* be taken with both the kinematics and the elastic characteristics so that negative pitch-lead coupling, which can cause self-excited lead-lag oscillations, does not occur. For inplane soft blades, positive pitch-lead coupling *shall* be avoided and both ground and air resonance *shall* be prevented. The floating hub rotor can be destabilized if hub tilting motions are reduced by cyclic pitch feedback, which, mechanically, is easy to do. The phasing between hub tilting and cyclic pitch feedback is of great importance in avoiding early instability (Ref. 57). The phasing most likely to be selected without the benefit of a stability analysis is a forward cyclic pitch with aft tilting and a left cyclic pitch with right tilting. This phasing, however, results in a low stability limit.

5-3.4.5 Hingeless Rotor

The hingeless rotor is attractive for its high control power and for low maintenance requirements, due to the use of fewer bearings. The use of inplane stiff blades facilitates the avoidance of ground and air resonance but imposes a significant penalty in rotor weight. Inplane soft blades therefore would be desirable, but they require mechanical or aerodynamic blade damping, as well as judicious selection of natural frequencies of those airframe modes that can produce ground or air resonance. Because of the numerous elastic, inertial, and aerodynamic coupling possibilities between the blade modes of a hingeless rotor, the dynamic analysis and design are particularly demanding. Feedback elements may be required to obtain satisfactory control characteristics, or to overcome a pitch-up tendency. However, if incorporated, the feedback elements interfere with the dynamic rotor stability and add to the numerous parameters whose effect must be studied in a hingeless rotor stability analysis.

5-3.5 SUBSTANTIATION CRITERIA AND METHODS

Design, construction, and materials for a helicopter, including attached aerodynamic surfaces and main and tail rotors, *shall* be such that there *shall* be no flutter, divergence, or other instability, throughout the design envelope. This design envelope *shall* include the complete design range of airspeeds, maneuvers, altitudes, thermal conditions, ground conditions, weights, and external stores or other loading configurations for the helicopter. Instabilities include all those of mechanical, aerodynamic, or aeroelastic origin. Also, so far as is practicable the helicopter *shall* be designed to be free of flutter, divergence, or other instability following the failure of a single structural element.

Specifically, a helicopter *shall* be free of dynamic instabilities or divergence at operating conditions up to 1.15 V_{DL} at design maximum rotor speed Ω_{DL}, 1.15 Ω_{DL} at V_{DL}, and maximum load factor attainable at V_{DL} and Ω_{DL} for all design ranges of altitude, maneuvers, and loading conditions. In addition, the helicopter *shall* be free of mechanical instability (ground resonance) under conditions of 0% and 95% airborne, one blade damper inoperative, and the combination of a flat tire(s) on a single strut and the shock absorber pressure at zero (flat strut). Blade dampers (except friction dampers) *shall* provide, during ground rev-up, torques at least proportional to the angular lead-lag velocity. In the case of friction dampers, the effective damping in any mode critical from the point of view of mechanical instability *shall* be at least 30% of critical damping.

The requirements of MIL-S-8698 with respect to rotor system instabilities provide a rough guide to the problem of substantiating the freedom from instabilities for rotor systems, but many important questions remain unanswered. There is, for example, the problem of nonlinearity, which affects most types of rotor system instabilities. In many cases the nonlinearities are such that the rotor system oscillations remain within certain limits after moderate disturbances, and thus the instability appears to be mild and is characterized as a limit-cycle phenomenon. However, occasional large disturbances may lead to a severe instability, and possible destruction, of the rotor system. If essential nonlinearities are involved, as is the case for blade friction dampers or for blade hydraulic dampers with relief valves, it is necessary to specify the magnitude of disturbance that will not lead to self-excited oscillations. For example, a single-wheel landing with impact equal to landing gear ultimate load *shall* not lead to mechanical instability.

Another difficult problem not solved as yet is a useful specification for the minimum damping of vibration modes within the limit speed envelope. A damping coefficient of $g > 0.03$ as specified in MIL-A-8870 for fixed-wing aircraft flutter and vibration modes corresponds to a minimum-damping-over-critical-damping ratio of 0.015. This is a very small damping ratio and the presently available analytical methods are not capable of predicting damping ratios of rotor system vibration modes with sufficient accuracy to make a requirement for a minimum damping ratio of 0.015 meaningful. In addition to the uncertainties of estimating aerodynamic damping—e.g., for the lightly damped chordwise blade modes—structural damping of blade and airframe modes that contributes substantially to the damping ratio is equally or even more uncertain. Experimental efforts to determine structural damping are hampered by energy dissipation through the support structure, by the difficulty of simulating in a vibration test the exact free flight modes, and by the usually prevalent severe nonlinearity of the structural damping phenomenon. For all of these reasons, MIL-A-8870 and MIL-S-8698 can serve merely as broad statements of intent rather than as unambiguous rules that, when followed, will prevent serious troubles from rotor system instabilities.

Because no detailed specification requirements exist, a careful search for rotor system instabilities is indicated whenever advances in the state of the art are attempted either by widening the flight envelope of well-known rotor systems or by introducing new design features such as, for example, mechanical or electronic feedback systems, new blade hinging arrangements, or hingeless blade-hub attachments. The three complementary methods of studying potential rotor system instabilities are analysis, wind tunnel tests, and ground and flight tests.

With respect to analytical methods, one can distinguish roughly between partial and total system analyses. The first method consists of devising, for each of the potential instability types, a simplified mathematical model that nevertheless will bring out the essential features of the particular instability and will provide guidance for the selection of appropriate parameters for the dynamic design. In a partial system analysis, modes with frequencies substantially different from the frequency of the unstable mode are omitted and nonlinearities disregarded unless recognized as essential. Quasi-steady aerodynamics is applied if the reduced frequency of the potential instability is sufficiently low. For moderate advance ratios, terms with periodic coefficients are neglected, so that the actual system is replaced by a simple linear constant-coefficient system of equations. Obviously, the essential feature of the instability must be represented properly in the analysis; it would be erroneous, for example, to use a single-blade analysis for an essentially multiblade or coupled airframe/rotor instability. The method of partial system analysis, although it can provide only approximations of the actual stability boundaries, is the backbone of the dynamic design.

The increasing capability of modern computer equipment has created the desire for a total system analysis that will cover all potential instabilities in addition to providing dynamic loads, vibrations, and handling characteristics. Helicopter manufacturers have developed extensive computer programs that take into account nonlinear aerodynamic effects and that include the basic rotor and airframe elastic modes as well as the rigid body modes treated in control dynamics. These total system representations can be exercised to simulate steady and unsteady flight conditions, and to provide information on handling characteristics, rotor and airframe loads, and dynamic stability boundaries. Although such total system representations are useful in anticipating the dynamic characteristics of a helicopter after its design has been frozen, they are no substitutes for the partial system representations that provide the necessary visibility for the different types of instabilities and that reveal the essential parameters involved in the mechanism of these dynamic instabilities. Such insight would be all but impossible to extract from one of the extremely complex and expensive-to-operate total system representations.

There usually will be sufficient doubt as to the validity of some of the numerous inputs into the analysis to justify dynamic stability tests with a wind tunnel model. The unavoidable restraints of a wind tunnel model, compared with free flight, usually will cause considerable modification in the stability boundaries, whereby these boundaries can be either increased or decreased. Some types of instability may be removed entirely and others created by the restraints. The principal value of testing wind tunnel models for dynamic stability boundaries is not the direct substantiation of the true stability characteristics of the flight vehicle but the checking and refining of the analytical tools used to predict these characteristics.

Because of the imperfections of both analysis and wind tunnel model testing, well-planned ground and flight prototype tests must be performed to correlate measured responses to either impulsive or frequency excitation with analytical predictions. Obviously, it would be unwise to attempt to determine dynamic stability boundaries by flight testing. These boundaries should be well outside the flight envelope and should be inaccessible within the helicopter limit speed and maneuver loadings. The only purpose of the prototype flight testing should be to obtain another check on the validity of the analytical tools with which the various stability margins have been determined.

The procedure of substantiating freedom from instabilities in helicopters rarely has been applied in the past. The analytical tools have been developed only recently and still are imperfect. Dynamic wind tunnel model tests, which are of little value unless designed to perfect an available theory, only rarely have been used in the development of prototypes. Historically, the sequence of events often was reversed; instabilities were discovered first in flight, and such a discovery stimulated a search for analytical tools to expla the instabilities and to provide the remedies. The development of methods of dynamic analysis should tend to make such an empirical approach unnecessary in the future. To secure adequate analysis, a vigorous analysis and test program *shall* be included in any helicopter development program.

5-4 LIFTING SURFACE DYNAMICS

5-4.1 GENERAL

This paragraph presents simplified design criteria to be used as a guide for the prevention of flutter, divergence, and control reversal of fixed surfaces. Also discussed are the basic elements involved in the propeller whirl mode phenomenon and the design criteria to be used as a guide for its prevention. Guidance is provided as to acceptable practice for the design of nonstructural mass balance weights and their attachments. The criteria developed include wing torsional rigidity; aileron, elevator, and rudder mass balance; and control tab and balance weight attachment criteria.

5-4.1.1 Definitions

The definition of a number of terms, other than those previously defined, that will be used throughout the discussion of lifting surface dynamics follows:

1. Control surface reversal. Reversal in the direction of the net normal force induced by the deflected control surface due to aerodynamic moments twisting the elastic "fixed" surface. This phenomenon can be illustrated best by considering the case of aileron reversal. Normally, the lift over the wing with down aileron is increased by the aileron deflection while the lift over the wing with up aileron is decreased by the aileron deflection; thus a rolling moment results. However, because the center of pressure for the lift due to the deflected aileron usually is aft of the elastic axis, deflecting the aileron downward tends to reduce the wing angle of attack and thereby to reduce the increment of lift. For the wing with up aileron, the torsional moment due to up aileron tends to increase the wing angle of attack. Thus it can be seen that the rolling moment for an elastic wing is less than that for a rigid wing. Because the wing torsional rigidity is constant while the twisting moment due to aileron deflection increases with the square of the velocity, it is obvious that at some critical speed the rolling moment due to aileron deflection will be opposite to that normally expected at speeds below this critical speed. The critical speed, so defined, is the aileron reversal speed.

Although it theoretically is possible for a rudder/fin and/or an elevator/horizontal stabilizer to suffer control reversal—i.e., side force or vertical tail force reversal with control displacement—no such case is known to have occurred.

2. Propeller and rotor whirl flutter/propeller and rotor divergence. Aerodynamic propeller or rotor normal forces and moments provide the energy source for the phenomenon commonly referred to as whirl flutter or rotor weaving. The structural support, propeller or rotor aerodynamic forces and moments, and gyroscopically coupled freedoms can combine to produce two aeroelastic instabilities. These are (1) static divergence and (2) a divergent spiraling motion of the propeller (rotor) hub in an elliptical motion (circular, if structural and inertial properties are symmetrical) rotating in a direction opposite to the regular propeller (rotor) rotation (termed regressive motion). Large propellers that are flexible relative to the support structure develop additional disk tilt modes and usually are more whirl critical than conventional stiff propellers. These large flexible systems also are critical in an advancing mode and are thrust/drag critical. The conventional propeller operating in either a tractor or pusher configuration is not thrust, drag, or power sensitive due to the absence of disk warpage.

3. Static balance. Complete static balance of a movable control surface is obtained when the CG of the control surface lies on the hinge line, i.e., the resultant moment of the mass of the surface about the hinge line is zero. If the CG of a surface lies aft of the hinge surface, it is described as statically unbalanced; whereas if the CG lies forward of the hinge line, the surface is described as statically overbalanced.

4. Dynamic balance. A movable surface is dynamically balanced with respect to a given axis if an angular acceleration about that axis does not tend to cause the surface to rotate about its own hinge line. The dynamic balance coefficient K/I is a measure of the dynamic balance condition of the movable control surface, where K is the product of inertia of the surface (including balance weights) about the hinge and oscillation axes and I is the mass moment of inertia of the control surface (including balance weights) about the hinge axis. Physically the dynamic balance coefficient K/I may be interpreted to represent exciting torque/resisting torque.

5-4.1.2 Basis of Criteria

Because the flutter and static aeroelastic stability of a specific design is the result of a combination of aerodynamic, inertial, and elastic effects, any criteria that do not include all three effects are bound to have severe limitations. Presently accepted criteria are based upon studies of both civil and military aircraft conducted by Rosenbaum (Ref. 63), propeller-nacelle whirl flutter studies by Houbolt and Reed (Ref. 64), and the applicable sections of MIL-A-8870 and MIL-A-8866.

Although satisfactory and rational analytic methods have been available for a number of years (Refs. 5 and

7 for example) and are, in some cases, preferred to the simplified criteria contained herein, the application of these simplified criteria to conventional helicopter fixed surfaces is reasonably adequate to assure freedom from flutter. Experience with rational analyses that have been carried through for specific designs, sophisticated flutter model tests, and flight flutter tests is discussed within this paragraph to point out the applicability of the criteria as well as areas of limitation.

5-4.2 FLUTTER

Fixed surface flutter, exclusive of control surface interaction, can be attributed almost universally to coupled bending and torsion motions of the surface. Although single-degree-of-freedom flutter can occur, it generally is associated with stall and the hysteresis loop that occurs with lift and angle-of-attack variations. Although important in propeller and fan design, this mechanism generally is not considered in fixed-wing design, because maneuvering load factors and torsional stiffness relationships essentially preclude this phenomenon.

The basic practice to be followed relative to the flutter of fixed surfaces such as helicopter wings or tail surfaces is to provide good frequency separation between the fundamental bending modes and the first torsional frequency, along with reasonable dynamic balance. To promote good dynamic balance when external stores are added to a wing, the CG of the store should be placed on the local wing quarter chord. Should the stores be disposable in increments—i.e., fuel, rockets, or multiple stores on one station—their disposition should be such that quarter chord balance, or balance slightly forward of the quarter chord, is maintained approximately. The amount of CG variation allowable will be a direct result of the dynamic balance and frequency separation of bending and torsion for the specific applied load and, in some cases, is related to the aerodynamics of the store itself. Compartmentalization (usually two compartments), and a design that keeps the forward section of an external fuel tank full by drawing fuel from the forward section while pumping fuel forward from the aft section with a pressurization system, is a common practice.

Pylon elasticity characteristics as well as the amount of structural damping within the pylon are difficult to stipulate, but, as a rule, the mounted store should have a basic primitive frequency at least twice the torsional frequency in the vertical plane of motion. The possibility that the wing system whose stores configuration is designed in this manner will be critical is very low. This refers to first bending mode, first torsional mode flutter.

Engine-nacelle systems and landing gear installations on the wing do not fit within this general picture of external stores. These items nominally provide overbalance (forward CG) and, because they are not located far outboard on the span, there is relatively weak inertial coupling between the fundamental bending and torsional modes.

It is most important to note that the spanwise distribution of internal and external stores can promote a second bending mode, first torsional mode flutter. It is clear that one local section can be stabilized highly by the presence of a large concentrated mass; but, should a spanwise node occur just inboard or outboard of this station, the idler action of the node line provides a strong driving phase because its effect upon the surface segment across the node line from the loaded station is the same as an unbalance. Experience with the size of store and the planform being considered are required to assist in these judgments. The basic factors are the aerodynamic surface areas involved and their motions.

5-4.3 DIVERGENCE

A second interpretation of the definition of static aeroelastic divergence—other than that the aerodynamic torsional moment per unit of surface rotation is equal to the structural restoring moment per unit rotation—is that the wing torsional mode has been reduced to zero frequency. This interpretation shows the extreme hazard of approaching torsional divergence relative to potential unstable flutter—i.e., frequency coalescence between the wing-bending modes and the first wing torsional mode is certain to occur prior to divergence. Flutter stability is possible if adequate dynamic balance is provided, but at best this is highly sensitive to inertial variations of the configuration.

Because the worst possible phase relationships can exist when the bending and torsional frequencies are crossing, dynamic balance must have been achieved or a relatively explosive instability is possible.

Use of materials such as Fiberglas and composite filament material must be examined carefully with respect to torsional rigidity. Spanwise layup of the filaments or fibers is an attractive, structurally efficient arrangement. However, except for the shear stiffness of the bonding matrix, this system has no torsional rigidity. Layups at forty-five-degrees most often are used to relieve this problem. Although designers may recognize the "structural inefficiency" of such construction, the aeroelastic (static and dynamic) characteristics of such designs also must be monitored carefully.

5-4.4 CONTROL SURFACE BALANCE AND STIFFNESS CRITERIA

5-4.4.1 Mass Balance of Control Surface

All fixed surface control surfaces (aileron, elevators, or rudders), unless they are irreversible, *shall* contain sufficient dynamic balance to prevent flutter in all possible critical modes (Dynamic balance requirement, MIL-A-8870). Such controls rarely are applied to helicopters, but the criteria that follow apply when such surfaces are employed.

For low-speed helicopters, the adequacy of dynamic balance may be established either by flutter analyses or by demonstration that the dynamic balance coefficient K/I of the surface meets the following condition

$$\frac{K}{I} < 0.20\left[6 - \left(\frac{V_{DL}}{130}\right)^2\right] \tag{5-48}$$

where

K = product of inertia of the control surface with respect to the hinge axis of the control surface and an axis in the plane of the control surface, normal to the hinge axis, whose origin usually is on the centerline of the fuselage, slug-ft^2. Nodal lines determined from vibration tests of prototype helicopters or from analyses *shall* be used as axes when available.

I = mass moment of inertia of the control surface about its hinge axis, slug-ft^2

V_{DL} = design limit airspeed, kt

The use of balance weights to prevent flutter, as outlined in this paragraph, is to prevent the coupling of a mode involving bending of the fixed surface and rotation of the control surface. In the case of a wing system, the movements of the node line due to wing fuel loading variations or wing stores usually require that distributed balance be used for the aileron. For rudder and elevator systems it usually is not required that fully distributed balance be used, primarily due to the small change in character of the empennage modes with wing and/or fuselage inertia loading variations. Because the use of irreversible flight controls has not been applied in the design of low-subsonic-speed aircraft, no discussion of this subject will be presented. Irreversible flight control system requirements are covered extensively in MIL-A-8870.

5-4.4.2 Balance Weights

5-4.4.2.1 Location of Balance Weights

Balance weights in control surfaces *shall* be located so that the flutter safety of both the control and the fixed surfaces is assured. Insofar as is practicable, balance weights *shall* be located in the regions where the deflections of critical mode shapes are a maximum. Balance weights *shall* be distributed if possible and, for high-speed helicopters, each third of the span of each main reversible control surface *shall* be balanced statically. However, fewer than three concentrated weights may be used providing that the torsional natural frequency of the surface with balance weights installed is at least twice the critical flutter frequency. Balance weights *shall* not be located externally with respect to the planes of the control surfaces (Balance weight requirement, MIL-A-8870).

5-4.4.2.2 Rigidity and Strength of Balance Weight Attachments

The natural frequencies of vibration of the balance weights as installed *shall* be at least twice the highest frequency of the flutter mode for which the balance weight is required to be effective. Also, the installation *shall* be shown to be sound structurally. An inertial load corresponding to a load factor of 100 and, separately, repeated inertial loads of 500,000 cycles of both positive and negative load factors of 60 *shall* act upon each control surface balance weight in directions normal to the plane of the control surface. Inertial loads corresponding to a load factor of $\pm$ 30 *shall* act upon each control surface balance weight in the other two mutually perpendicular directions (see Loads on balance weight attachment requirement, MIL-A-8866).

5-4.5 PROPELLER-NACELLE WHIRL FLUTTER

Turbine-powered helicopters, such as turboprop or turbine/shaft-driven propeller/rotor configurations, may result in whirl flutter becoming a practical concern rather than of academic interest only. Consideration of large flexible rotor/propeller configurations is too complex for simple criteria to be applied relative to whirl flutter. Refs. 2, 65, and 66 represent good examples of the current state of the art relative to this subject. For the case of relatively small rigid propeller systems (13.5 ft or smaller), some guidance for the design of such systems can be given from the standpoint of divergence and whirl flutter. When the continuous system is approximated by a single natural pitch mode and a single natural yaw mode coupled by the gyroscopics of the spinning propeller, the system behaves like a gyroscopic pendulum. Neglecting the flexibility of the propeller blades eliminates the possibility of mechanical instability of such a system. The aerodynamic forces acting upon the propeller are the sources of the potential instabilities of this system, either divergence or whirl flutter (Refs. 63 and 67).

Static divergence of a pusher-propeller configuration theoretically is impossible. For the tractor-propeller configuration, the designer should assume that the wing/fuselage is fixed and that the angular stiffness of the propeller due to a force acting at the propeller centerline is computed. This propeller force *shall* be applied in the direction that produces the largest rotation. This stiffness K_α should have a value (lb/rad) equal at least to two times the propeller normal force N_p (lb) computed at V_{DL}. Should wing flexibility or nacelle configuration add substantially larger loads and moments than those obtained by consideration of the propeller alone, a detailed wing-nacelle propeller divergence analysis *shall* be made.

Propeller whirl considerations, in both pusher and tractor configurations, require that the angular stiffness computed at the propeller due to both pitching K_θ and yawing K_ψ moments be examined. As a simplified design criterion, it is recommended that the computed rms stiffness, $K_{rms} = \sqrt{(K_\theta^2 + K_\psi^2)/2}$ be multiplied by an arbitrarily assumed damping factor of 0.02. This must have a value (in.-lb/rad) equal to or in excess of the value of the aerodynamic moment of the propeller M_p (lb-in.) at 1.2 V_{DL}.

$$0.02\,K_{rms} \geq (M_p)_{1.2V_{DL}} \qquad (5\text{-}49)$$

The critical relationship between the stiffness and damping levels of the structure and/or mounting and the stability of the installation is shown in Ref. 63. Experience with models indicates that the fully coupled helicopter system will be stable if the recommended criterion is utilized. This must be verified by analysis and tests of the complete system.

These criteria do not apply to propellers with flapping hinges even though they are not subject to mechanical instability. This is due to the added degrees of freedom provided by the disk warpage, which has been assumed to be zero. Clearly defined fail-safe criteria also must be established for this configuration (see, for example, FAR Part 25).

5-5 DRIVE SYSTEM DYNAMICS

5-5.1 SCOPE

This paragraph presents a design philosophy and design criteria for attaining the freedom from critical speeds or flywheel resonances required by MIL-S-8698. Methods are shown for predicting the natural frequencies of the drive system and for optimizing the system in the early stages of design. It is not the purpose of this paragraph to provide a means of structural substantiation for any part of the drive system because this substantiation must be accomplished by analysis of the effects of the loads developed during the flight test program. However, calculations based upon the methods shown here will provide an excellent basis for interpreting the test results.

5-5.2 DESIGN PHILOSOPHY AND DESIGN CRITERIA

The ideal design would require complete freedom from vibration. The necessary deviations from the ideal *shall* assure that the residual vibration is not damaging structurally and does not interfere with the operation or control of the helicopter. Flight safety and the comfort of the passengers and crew must be assured. The criteria that follow are considered necessary for the accomplishment of these objectives during preliminary design:

1. In all branches of the drive system torsional resonance with any multiple of the product of the rotational speed and the number of blades for any of the system rotors *shall* be avoided by a minimum margin of $\pm$40% of the pertinent rotational speed. The critical speed(s) of any sections of the drive system *shall* be separated from any operating speed of the section by a minimum margin of 10% of the operating speed (see MIL-T-5955).

2. Natural frequencies of torsional vibration of any branch of the drive system *shall* not be less than 3 Hz unless means are provided to assure that there can be no inadvertent cycling of the blade pitch controls by the pilot. This is important particularly in aircraft of less than 6000 lb maximum gross weight where antitorque rotor drive shafts are known to have failed because of resonance built up by the pilot's cycling of the directional control pedals. In the case of large helicopters, time lags in the control system usually assure a slow

rate of change of blade pitch angle, thus preventing cycling at the natural frequency. However, the designer should make sure that there is no problem in this respect.

3. The torsional natural frequency of the system *shall* provide maximum compatibility with the engine control system. It is necessary to consider this at an early stage of the design because changes required to stabilize the power control system decrease the gain of the governor with a consequent increase in engine response time.

4. The failure of any part of the drive system *shall* not result in resonance of the remaining part of the drive system.

5-5.3 MATHEMATICAL METHODS

5-5.3.1 The Mathematical Model

Fig. 5-32 illustrates the mechanical drive system of a typical single-rotor helicopter. The complete system consists of main rotor, antitorque (tail) rotor, transmission, oil cooler fan, engine power turbine, and all connecting shafting.

Critical speeds of the drive system are calculated by methods given in standard texts such as Refs. 10 and 11. These critical, or flywheel resonance, speeds are the rotational speeds that coincide with the natural frequencies of the shafts in bending.

In order to investigate the torsional frequency characteristics, consider the system shown in Fig. 5-32 to be replaced by the dynamically equivalent system of Fig. 5-33. This is comprised of lumped-mass polar moments of inertia I reduced to a common speed and concentrated at the plane of the main rotor, transmission, engine, oil cooler fan, and tail rotor, respectively. The masses are connected by shafts of stiffness K reduced to the reference speed but having no mass. Mass polar moments of inertia of the rotating parts of the transmission are considered as one lump and referred to the reference speed. The mass moments of inertia of all other gears are considered to be negligible because the five primary masses shown in Fig. 5-33 are the dominant masses in the system.

This simplified representation of the drive system is valid only if the assumption that each rotor can be replaced by an equivalent lumped-mass moment of inertia remains sound. This substitution is not realistic for those rotor systems whose chordwise, or inplane, stiffness is relatively low. The effects of rotor blade flexibility upon the characteristics of the system, and upon their analysis, are discussed in par. 5-5.3.4.

Means of determining the equivalent mass moments of inertia and stiffnesses of Fig. 5-33 are shown in Refs. 68 and 69.

5-5.3.2 Solution by Matrix Methods

The equations for the system shown in Fig. 5-33 may be developed easily by operating on the kinetic and

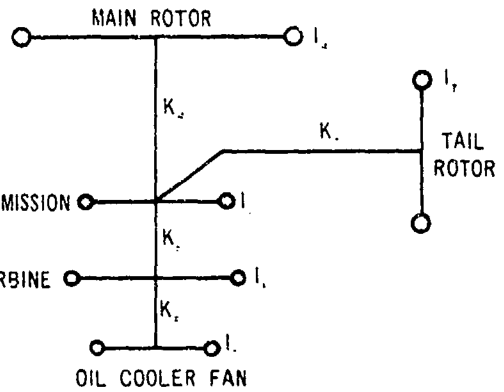

Fig. 5-33. Diagram of Sample Drive System

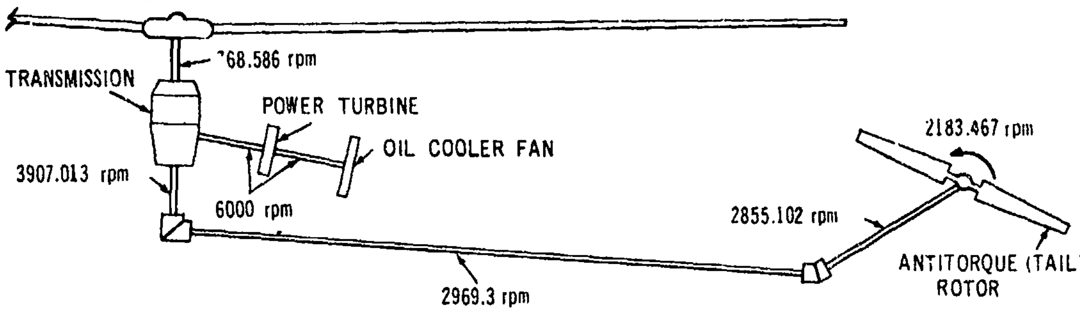

Fig. 5-32. Sample Helicopter Drive System

potential energy functions T and V, respectively, with Lagrange's equation

$$\frac{d}{dt}\left(\frac{\partial T}{\partial \dot{q}}\right) + \frac{\partial V}{\partial q} = 0 \qquad (5\text{-}50)$$

The following equations result:

$$\left\{ s^2 \begin{bmatrix} I_R & 0 & 0 & 0 & 0 \\ 0 & I_G & 0 & 0 & 0 \\ 0 & 0 & I_T & 0 & 0 \\ 0 & 0 & 0 & I_E & 0 \\ 0 & 0 & 0 & 0 & I_F \end{bmatrix} + \begin{bmatrix} K_R & K_R & 0 & 0 & 0 \\ K_R & K_R + K_T + K_E & K_T & K_E & 0 \\ 0 & K_T & K_T & 0 & 0 \\ 0 & K_E & 0 & K_E + K_F & K_F \\ 0 & 0 & 0 & K_F & K_F \end{bmatrix} \right\} \begin{bmatrix} \theta_R \\ \theta_G \\ \theta_T \\ \theta_E \\ \theta_F \end{bmatrix} = |0| \qquad (5\text{-}51)$$

where

s = operator representing time derivative, sec^{-1}

θ = angular deflection of component identified by subscript, rad

It is observed readily that this is a statically coupled system for which the stiffness matrix may be constructed immediately by simply overlapping the 2×2 stiffness matrices for adjacent stiffness elements.

The eigenvalues of this equation give roots that are the natural frequencies of the system and the eigenvectors describe the mode shapes for these natural frequencies.

5-5.3.3 Porter's Method

This matrix equation can be solved by any of a number of methods (Refs. 4, 10, and 11). Porter's method has certain merits because it can be used without high-speed computer facilities (although it can be computerized to advantage). It is well-supported by basic theory.

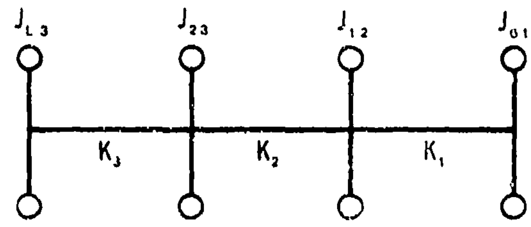

Fig. 5-34. Diagram of a Multiple-mode Torsional System

An exposition of its derivation and usage may be found in Refs. 4 and 69.

To explain the method and its uses, reference is made to Fig. 5-34 in which are shown consecutive lumped mass elements J_{O1}, J_{12}, J_{23}, and J_{L3}, connected by stiffness elements K_1, K_2, and K_3. The notation J_{mn} is used to indicate that the lumped mass is concentrated between the stiffness elements K_m and K_n.

If a mass is substituted at the left end of the stiffness element K_1 to be dynamically equivalent to the mass occurring at the right side of the stiffness element K_1, it must, during the vibration, produce the same twisting moment at the left end of the stiffness element as does the mass at the right side acting through the stiffness element K_1. We will call the substituted mass the impedance, J'_{L1}, due to the mass concentrated at the right end of the stiffness element. The subscript Ln is used to indicate the left end of the stiffness element K_n, while the subscript On is used to indicate the right end. Defining the impedance from right to left we may say:

The impedance at the right end of K_1 is $J'_{O1} = J_{O1}$.

The impedance at the left end of element K_1 is

$$J'_{L1} = \frac{J'_{O1}}{1 - \dfrac{J'_{O1}\,\omega^2}{K_1}} \qquad (5\text{-}52)$$

The impedance at the right end of element K_2 is $J'_{O2} = J'_{L1} + J_{12}$.

Impedances also may be calculated from left to right according to the definitions that follow.

The impedance at the left end of element K_3 is $J''_{L3} = J_{L3}$.

The impedance at the right end of the element K_3 is

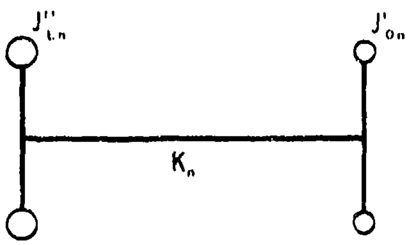

Fig. 5-35. Generalized Diagram of Individual Mode from Fig. 5-34

$$J''_{O3} = \frac{J''_{L3}}{1 - \dfrac{J''_{L3}\,\omega^2}{K_3}} \tag{5-53}$$

The impedance at the left end of element K_2 is $J''_{L2} = J''_{O3} + J_{23}$.

The prime mark indicates that the impedances are calculated from the right while the double prime mark indicates that the impedances are calculated from the left.

By these procedures the impedances at any point in the system may be calculated. For any natural frequency the impedance at a point calculated from the left must balance the impedance at the same point calculated from the right.

Thus

$$\begin{aligned} J''_{On} + J'_{On} &= 0 \\ J''_{Ln} + J'_{Ln} &= 0 \end{aligned} \tag{5-54}$$

Now if we consider the impedances J''_{Ln} and J'_{On} assembled at each end of the stiffness element K_n, we have a two-mass system comprised of the impedances or equivalent masses J''_{Ln} and J'_{On} connected by the torsional stiffness element K_n as shown in Fig. 5-35.

The standard formula for the natural frequency ω of this two-mass system is

$$\omega = \sqrt{\frac{K_n(J''_{Ln} + J'_{On})}{J''_{Ln}\,J'_{On}}} \tag{5-55}$$

Multidegree-of-freedom torsional systems may be analyzed by assuming a trial value of the natural frequency, selecting a stiffness element, calculating the impedance at each end, and solving for ω. This value of ω will be exactly equal to the trial value if the trial value has been correctly chosen. If not, this result is used as a new trial and the process is repeated. Convergence usually occurs within three or four iterations if the correct stiffness element has been selected. This will be made somewhat clearer in the sample calculations in par. 5-5.3.3.3.

5-5.3.3.1 Mode Shapes Calculated by Porter's Method

When the solution for the natural frequency has converged, the final iteration contains all of the information required for the calculation of the mode shape. Referring to Fig. 5-35, let the angular motion at the right end of the stiffness element be θ_{On} and let the angular motion at the left end be θ_{Ln}. Then

$$\frac{\theta_{Ln}}{\theta_{On}} = \frac{J'_{On}}{J'_{Ln}} = \frac{J''_{On}}{J''_{Ln}} \tag{5-56}$$

It should be noted that the impedance ratio must be consistent. The two impedances must be calculated from the same direction in the system, right or left.

To calculate the mode shape, a unit deflection is assumed at an arbitrary point in the system and the deflections at the other points are calculated successively using Eq. 5-56. Because the amplitudes have been calculated for a system that has been reduced to a common reference speed, the true amplitudes at points in the real system vary from the calculated values directly as the gear ratios.

5-5.3.3.2 Parametric Studies

A useful tool in the early design stage is the parametric variation of the mass and stiffness characteristics for study of their effect on the natural frequencies of the various modes. The procedure consists of selecting a number of frequencies and calculating, for each one selected, the mass required at a selected point or the stiffness required in a selected element to make the selected frequency a natural frequency. The calculated values then are plotted as curves of mass (inertia) versus natural frequency or stiffness versus natural frequency. Figs. 5-36 and 5-37 are examples. By using logarithmic coordinates for the plots, a much greater range of parameters can be covered with fewer selected points. It is usual to select frequencies in increments of 1 Hz up to 10 Hz, in increments of 2 Hz from 10 Hz

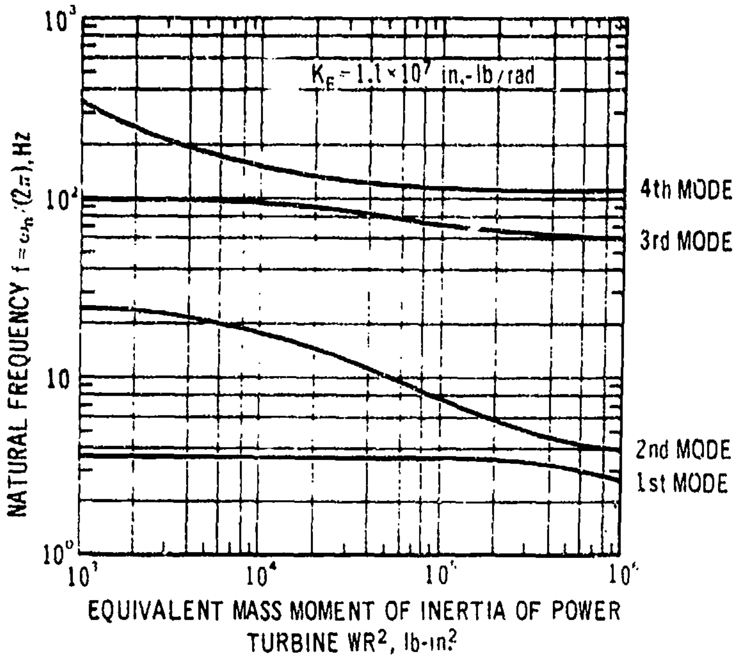

Fig. 5-36. Variation of Natural Frequency With Inertia of Power Turbine—Example

to 20 Hz, and in increments of 10 Hz above 20 Hz. In regions of discontinuities, additional frequencies may be selected.

To calculate the stiffness curve (Fig. 5-37), the system is reduced, by the methods of par. 5-5.3.3, to a two-mass system that includes the selected stiffness element. Then Eq. 5-55 is solved for the value K_n corresponding to each selected frequency.

To calculate the mass curve (Fig. 5-36), use is made of the fact that at a natural frequency the right and left impedances must match at any chosen point in the system (Eq. 5-54). Thus for a selected mass such as J_{12} in Fig. 5-34

$$J''_{O2} + J_{12} + J'_{L1} = 0 \tag{5-57}$$

Equations of the type

$$J_{mn} + J''_{On} + J'_{Lm} = 0 \tag{5-58}$$

may be solved for the values of J_{mn} required to make each selected frequency a natural frequency.

An interesting case occurs when the required value of J_{mn} is negative. The physical significance of this result is that in order to make the selected frequency a natural frequency, a device such as a tuned mass having a negative impedance is required at the location of J_{mn}.

If a tuned mass is required, a parametric study leading to an optimum combination of stiffness and mass for the tuned mass may be made by adding another degree of freedom to the system at this point.

Fig. 5-36 shows the effect of variation of the mass of the power turbine upon the several modes of the system shown in Figs. 5-32 and 5-33. Fig. 5-37 shows the effect of variation of the stiffness of the engine-to-transmission input shaft. These figures are examples of the type of information that can be obtained from a parametric study.

5-5.3.3.3 Sample Calculations

To demonstrate the use of these methods, the arithmetical steps in the calculation of the first and second natural frequencies of the system shown in Fig. 5-33 are shown in Table 5-2. Including notation to conform to that of par. 5-5.3.3, the characteristics of the system are as follows:

1. Mass Polar Moments of Inertia
   a. Main rotor $I_R = J_{O1} = 4.9 \times 10^3$ slug-in.^{2}

b. Tail rotor I_T = J_{Oa} = 1.87×10^3 slug-in.^{2}
c. Transmission I_G = J_{12} = 22.4 slug-in.^{2}
d. Power turbine I_E = J_{23} = 4.66×10^{-2} slug-in.^{2}
e. Cooling fan I_F = J_{L3} = 4.7 slug-in.^{2}

2. Shaft Stiffnesses
a. Main rotor shaft K_R = K_1 = 582,300 in.-lb/rad
b. Tail rotor shaft K_T = K_a = 117,000 in.-lb/rad
c. Turbine shaft K_E = K_2 = 11,000,000 in.-lb/rad
d. Cooling fan shaft K_F = K_3 = 798,300 in.-lb/rad

An approximation for the first natural frequency is made by fixing the left end and solving the equation

$$\omega_1^2 = \frac{K_a}{J_{Oa}} \tag{5-59}$$

A good approximation for the second frequency is to solve the equation

$$\omega_2^2 = \frac{K_1}{J_{12} + J_{23} + J_{L3}} \tag{5-60}$$

The solution will be made in the K_a stiffness element for the first frequency and in the K_1 stiffness element for the second frequency.

The mode shape calculation follows easily from the last line of each natural frequency solution. Assuming a unit angular amplitude at the main rotor, we have θ_{O1} = 1 (assumed) at the main rotor

$$\theta_{L1} = \theta_{La} = \theta_{O2} = \theta_{O1} \frac{J'_{O1}}{J'_{L1}} = \frac{4900}{-1496.6}$$

$= -3.274$ at the transmission

$$\theta_{Oa} = \theta_{La} \frac{J''_{La}}{J''_{Oa}} = -3.274 \frac{-994.1}{-187}$$

$= -17.40$ at the tail rotor

(5-61)

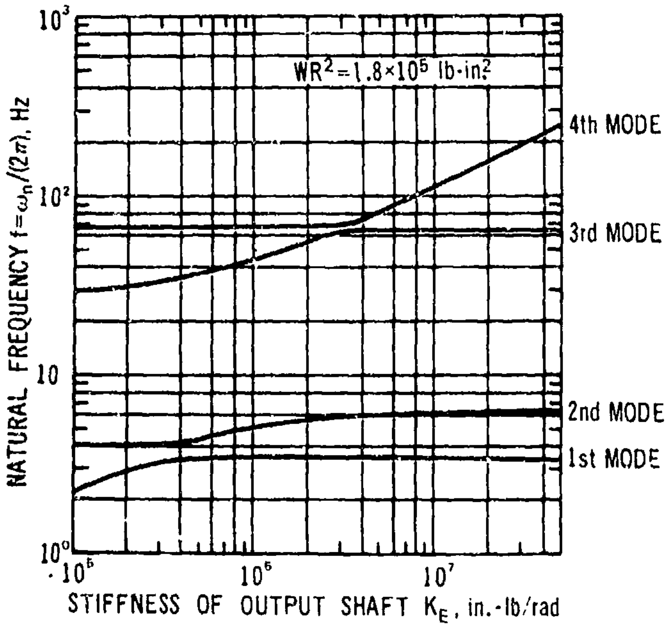

Fig. 5-37. Variation of Natural Frequency With Stiffness of Turbine Output Shaft—Example

$$\theta_{L2} = \theta_{O3} = \theta_{O2}\frac{J''_{O2}}{J''_{L2}} = -3.274\,\frac{480.1}{469.7}$$

$= -3.347$ at the turbine

$$\theta_{L3} = \theta_{O3}\frac{J''_{O3}}{J''_{L3}} = -3.347\,\frac{4.714}{4.7}$$

$= -3.357$ at the cooling fan

TABLE 5-2
SAMPLE CALCULATION, TORSIONAL NATURAL FREQUENCIES

1st MODE

1	2	3	4	5	6	7	8	9	10	11	12
ω^2	$1-\frac{J_{01}{}^{2}}{K_1}$	$J'_{L1}=\frac{J_{01}}{(2)}$	$1-\frac{J_{L3}\omega^2}{K_3}$	$J''_{O3}=\frac{J_{L3}}{(4)}$	$(5)+J_{23}$	$1-\frac{(6)\omega^2}{K_2}$	$J''_{O2}=\frac{(6)}{(7)}$	$J''_{L3}=(3)+(8)+J_{12}$	$(9)+J_{01}$	$(9)\times J_{01}$	$\omega^2=K_1\frac{(10)}{(11)}$
505.80	−3.256	−1504.9	0.9970	4.714	469.7	0.9784	480.1	−1002.4	−815.4	−187449	508.95
508.95	−3.283	−1492.5	0.9970	4.714	469.7	0.9783	480.1	−990.0	−803.0	−185122	507.48
507.95	−3.274	−1496.6	0.9970	4.714	469.7	0.9783	480.1	−994.1	−807.1	−185900	507.98

$\omega_1 = 22.54$ rad/sec

$f_1 = 22.54/(2\pi) = 3.59$ Hz

2nd MODE

1	2	3	4	5	6	7	8	9	10	11	12
ω^2	$1-\frac{J_{01}\omega^2}{K_1}$	$J'_{L1}=\frac{J_{01}}{(2)}$	$1-\frac{J_{L3}\omega^2}{K_3}$	$J''_{O3}=\frac{J_{L3}}{(4)}$	$(5)+J_{23}$	$1-\frac{(6)\omega^2}{K_2}$	$J''_{O2}=\frac{(6)}{(7)}$	$J''_{L1}=(3)+(8)+J_{12}$	$(9)+J_{01}$	$(9)\times J_{01}$	$\omega^2=K_1\frac{(10)}{(11)}$
1600.0	−1.5573	−120.08	0.99058	4.745	469.75	0.9317	504.18	406.50	5306.5	1991850	1551.3
1551.3	−1.4794	−126.40	0.99087	4.743	469.74	0.9337	503.09	399.09	5299.1	1955541	1577.9
1568.5	−1.5069	−124.10	0.99067	4.744	469.74	0.9330	503.47	401.77	5301.8	1968673	1568.2

$\omega_2 = 39.60$ rad/sec

$f_2 = 39.6/(2\pi) = 6.3$ CP

NOTE: ( ) REFERS TO COLUMN WHERE NUMERICAL VALUE IS TO BE FOUND

The required mass ratios, or their reciprocals, are contained in Columns 2, 4, and 7 of Table 5-2. In calculating the mode shape for the second natural frequency, the appropriate values are taken from the table.

$$\theta_{O1} = 1 \text{ at the main rotor}$$

$$\theta_{L1} = \theta_{La} = \theta_{L2} = \theta_{O1} \frac{J''_{O1}}{J''_{L1}} = \frac{-4900}{401.77}$$

$$= -12.196 \text{ at the transmission}$$

$$\theta_{Oa} = \theta_{La} \frac{J'_{La}}{J'_{Oa}} = \frac{-12.196}{-1.5069} \tag{5-62}$$

$$= 8.094 \text{ at the tail rotor}$$

$$\theta_{L2} = \theta_{O3} = \theta_{O2} \frac{J''_{O2}}{J''_{L2}} = \frac{-12.196}{0.9330}$$

$$= -13.072 \text{ at the turbine}$$

$$\theta_{L3} = \theta_{O3} \frac{J''_{O3}}{J''_{L3}} = \frac{-13.072}{0.99067}$$

$$= -13.194 \text{ at the cooling fan}$$

These calculated mode shapes are of sufficient accuracy to permit an evaluation of the stress due to torsional vibration in any part of the system, if the amplitude of applied moment or of angular deflection is known or can be measured at any point in the system.

5-5.3.4 Effect of Rotor Characteristics

The effect of rotor characteristics, and particularly of rotor blade flexibility, upon the torsional natural frequencies of the drive system depends upon the blade natural frequencies, and upon the degree to which the rotor participates in the mode in question. For instance, in the previous example, the rotor has no effect upon the third mode frequency, which is a function of the cooling fan mass, the power turbine mass, and the stiffness of the fan drive; nor does the rotor have any effect on the fourth mode frequency, which depends principally on the turbine mass, the lumped transmission mass, and the stiffness of the turbine-to-transmission drive. Also, any natural frequency of the drive system that is less than the first (lowest) rotor blade natural frequency will be unaffected by the rotor blade modes. Investigation of blade effects upon a particular mode can be accomplished readily by overlapping the blade matrices with the drive system matrices at the coordinate that describes the angular motion of the rotor hub. The effect of rotor blade modes must be included in the final analysis of drive system torsional characteristics and their compatibility with the engine control system.

5-5.3.5 Compatibility Between the Drive System and the Engine and Engine Control System

When a turbine engine is connected to a low-frequency propulsion system, such as occurs in most helicopter installations, the resulting combination may be unstable. The engine output shaft speed is sensed by the power turbine governor. The power control adjusts the fuel flow to maintain the selected turbine speed. The governor also can sense superimposed oscillatory motions at the power turbine, such as θ_{L2} in the sample calculations of par. 5-5.3.3.3. At certain frequencies inherent in the engine, the gains and dynamics of the engine and its control may cause sufficient phase shift to reinforce the oscillations of the drive system in one of its natural modes.

Careful analysis of the engine, power control, and drive system is necessary early in the design stage. This requires close cooperation between the designer of the drive system and the engine manufacturer. Failure to consider this problem early enough may result in compromising of engine response time and droop performance by reducing the governor gains, or in introducing time lags into the system.

Investigations of problems of this type usually are conducted by combining the transfer functions for the engine and power control system (obtained from the engine manufacturer) with the transfer functions for the drive system. The two systems are linked by the torque acting upon the power turbine and the turbine speed, which is fed back to the control system. Solutions for the problem are found easily on an analog computer, although sometimes it is desirable to plot the roots of the characteristic equation in the complex plane to study the effects of the variation of the drive system lumped parameters. Other methods of study are available and may be useful. A digital computer may be used to obtain time-histories suitable for evaluation of drive system stability. This method makes use of mathematical models of the engine, the fuel control, and the rotor and drive system. The Bode analysis and plot provide a means of evaluating both the low-frequency phase margin and the torsional gain margin (magnitude). The phase margin indicates the stability of the basic system-governing loop while the gain margin indicates the allowable increase in the loop gain of the system before divergence occurs. Because a high gain is desirable to obtain rapid response of the engine to load demands, it is important to provide a drive system that has been optimized to the point that a minimum of gain adjustment is required.

A common error in this type of analysis is to ignore the effects of the tail rotor for the purpose of simplifying the problem. If this is done without a thorough understanding of the nature of the mechanical drive system, the results of the analysis may be erroneous. In fact, there have been cases in which the natural frequency of the tail rotor was the casue of an instability. Also, the tail rotor may have considerable effect upon that natural frequency which is dependent principally upon the stiffness of the main rotor shaft. Referring to Column 3 of the second mode natural frequency calculation of Table 5-2, the tail rotor contributes an impedance at the transmission of −124.1 slug-in.^{2} This reduces the effective impedance acting upon the main rotor shaft by about 20%, which in this case is enough to make the difference between stability and instability.

For articulated rotors with lag hinge dampers it also may be necessary to include the effect of the hydraulic fluid acting as a spring in order to simulate the torque oscillations actually found with such a system.

High frequencies such as the third and fourth modes shown in Figs. 5-36 and 5-37 have no effect upon the low-frequency stability, and these degrees of freedom may be eliminated from the problem by lumping their masses with the transmission mass. This leaves only a three-mass system comprised of the main rotor and main rotor shaft, the transmission and turbine lumped masses, and the tail rotor drive shaft and tail rotor. A simplified block diagram for the system of Fig. 5-33 is shown in Fig. 5-38. The moments of inertia lumped at the transmission are designated as $I_M = I_G + I_E + I_P$.

The analysis of the compatibility of the engine control system with the dynamic characteristics of the coupled airframe-engine system is discussed in greater detail in par. 8-7.

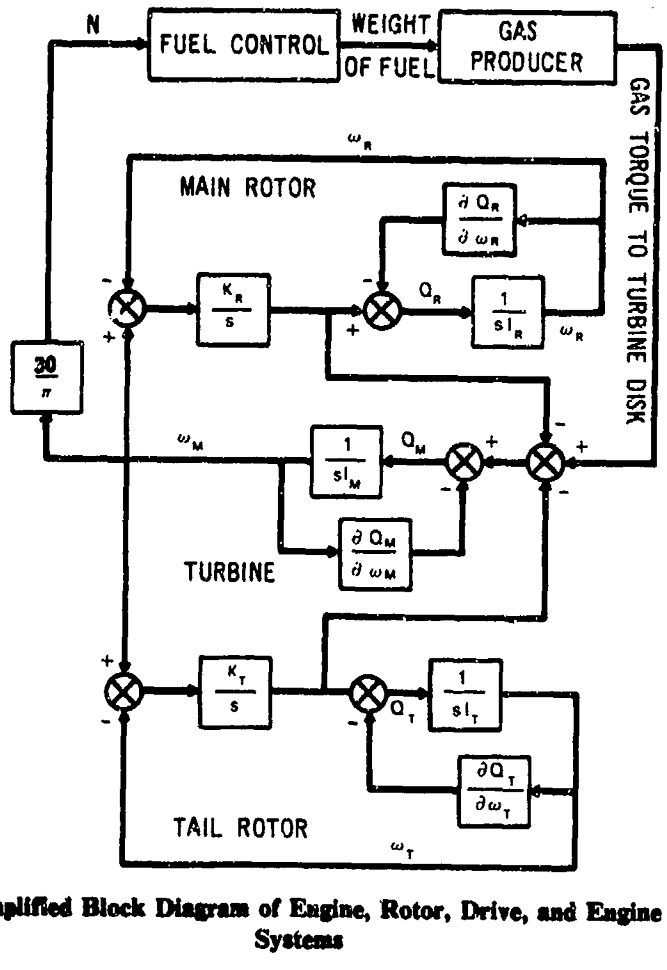

Fig. 5-38. Simplified Block Diagram of Engine, Rotor, Drive, and Engine Control Systems

5-6 LIST OF SYMBOLS

$A.F.$ = aerodynamic force, lb
$[A(t)]$ = matrix of periodic time functions
a = moment arm of $A.F.$ with respect to flap-hinge, ft
B = tip-loss factor, dimensionless
B_f = hub effective damping, lb-sec/ft
B_g = blade lag damping, lb-ft-sec/rad
b = distance from lag hinge to center of mass of blade, ft
= half-chord of blade, ft
C = damping, lb-sec/ft
$C.F.$ = centrifugal force, lb
$C_{\dot\beta} = 2\beta_0$
c = blade chord, ft
EI = bending rigidity, lb-in.^{2}
e = base of natural (Naperian) logarithms, 2.71828
= vertical (lag) hinge offset from rotor center, ft
F = shear amplitude of forcing function $f(t) = Fe^{i\omega t}$, lb
$F_{\dot\zeta} = \left(\frac{\gamma}{8}\right)\left(\frac{3\theta}{2}\right) - 2\beta_0$
f = complex column matrix of applied force amplitudes, lb
= distance from CG of m_2 to gimbal axis, ft
f_i = amplitude of cos component of force, lb
f_t = column matrix of applied force time history, lb
GJ = blade torsional stiffness, lb-ft^2
g = structural damping coefficient, dimensionless
g_i = amplitude of sin component of force, lb
h = rotor to nacelle pivot distance, ft
= distance from CG of m_1 to CG of m_2, ft
I = blade mass moment of inertia about vertical, or lag, hinge, slug-ft^2
= mass moment of inertia of the control surface about its hinge axis, slug-ft^2
= lumped mass polar moment of inertia, slug-in.^{2}
I_1 = mass polar moment of inertia of fuselage, slug-ft^2
I_2 = mass polar moment of inertia of isolated part about its centroidal axis, slug-ft^2
i = imaginary unit, $\sqrt{-1}$
J = mass polar moment of inertia (effective impedance), slug-in.^{2}
J_{ij}, J'_{ij} = lumped inertias, or impedances slug-in.^{2}
K = hub effective spring rate, lb/ft
= stiffness, lb/ft or in.-lb/rad
= product of inertia of control surface with respect to the hinge axis and to an axis in the plane of the control surface, normal to the hinge axis, slug-ft^2
K_m = stiffness of the mth stiffness element, in.-lb/rad
K_n = stiffness of the nth stiffness element, in.-lb/rad
K_{rms} = root-mean-square angular stiffness of propeller, in.-lb/rad
K_α = angular stiffness of propeller, lb/rad
K_β = centering spring constant, lb-ft/rad
K_θ, K_ψ = angular stiffness of propeller, pitch and yaw modes respectively, in.-lb/rad
k = blade number, $k = 0,1,\ldots,n-1$ counted in direction of rotation
L, K = tilting feedback constants, sec^{-1}
M = equivalent airframe mass at rotor center, including blade masses, slug
= mass, slug
= moment, lb-ft
M_p = propeller aerodynamic moment, lb-in.
m_f = effective mass of hub, slug
m_b = mass of blade, slug
m_n = generalized mass of nth mode, slug
$m\theta_I$, $m\theta_{II}$ = periodic time functions proportional to blade flapping moments from θ_I, θ_{II}, dimensionless
m_1 = mass of fuselage, slug
m_2 = mass of isolated part, slug

N = number of degrees of freedom, dimensionless
N_p = propeller normal force, lb
n = number of blades per rotor
P = ratio of blade natural frequency with elastic restraint to that without restraint
p = distance from CG of m_2 to plane of rotor hub, ft
= $\left(\bar{\omega}_\beta^2 + 1\right)^{1/2}$, dimensionless
Q = torque, lb-ft
= response to shear force of amplitude F or moment of amplitude M
$[Q]$ = transition matrix
q = generalized coordinate
q_n = modal amplitude of nth mode, ft
R = rotor radius, ft
r = radius of gyration of blade about CG, ft
S = blade mass moment about vertical hinge, slug-ft
s = ratio of effective hub springs in two directions
= operator signifying the time derivative, sec^{-1}
T = period of rotor revolution, $2\pi/\Omega$, sec
= kinetic energy in Lagrange equation, in.-lb or ft-lb
t = time, sec
V = relative flow velocity with respect to blade, fps
= potential energy in Lagrange equation, in.-lb or ft-lb
V_c = cruise airspeed, kt
V_{DL} = design limit airspeed, kt
x = lateral hub deflection, ft
= displacement, ft
$\{x\}$ = column matrix of state variables
$\{x(T)\}$ = $[Q]\{x(0)\}$
y = complex column matrix of displacements, ft
y_t = matrix of displacement time history, ft
a_k = constants derived from initial conditions
α_o = mean blade angle of attack, rad
β = blade flapping angle, positive up, rad
β_I, β_{II} = first multiblade flapping coordinates, dimensionless
β_d = differential collective multiblade flapping coordinate (only for even-bladed rotors), dimensionless
β_0 = coning, or mean blade flapping, angle, rad
γ = blade Lock number, $R^4\rho c(dc_L/d\alpha/I)$, dimensionless
$\partial Q_R/\partial\omega_R$ = damping factor for main rotor, lb-ft-sec/rad
$\partial Q_M/\partial\omega_M$ = damping factor for turbine, transmission, and fan (lumped), lb-ft-sec/rad
$\partial Q_T/\partial\omega_T$ = damping factor for tail rotor, lb-ft-sec/rad
ζ = blade lead angle, positive forward, rad
ζ_I, ζ_{II} = first cyclic lead coordinates, .
η_I, η_{II} = first multiblade cyclic coordinates, dimensionless
η_{III}, η_{IV} = second multiblade cyclic coordinates, dimensionless
η_d = differential collective multiblade coordinate (only for even-bladed rotors), dimensionless
η_k = deflection of kth blade, dimensionless
η_x = lateral airframe damping ratio, dimensionless
η_ζ = lead-lag damping ratio, dimensionless
η_0 = collective multiblade coordinate, dimensionless
θ = torsional amplitude, rad
θ_I, θ_{II} = forward and left cyclic pitch, rad
θ_β = pitch-flap coupling ratio $\partial\theta/\partial\beta$, positive for increase in blade pitch with up flapping, dimensionless
θ_ζ = pitch-lead coupling ratio $\partial\theta/\partial\zeta$, positive for increase in blade pitch with lead, dimensionless
θ_{1N_F} = normalized response to exciting forces at the rotor head, dimensionless
θ_{1N_M} = normalized response to exciting moments at the rotor head,

dimensionless

Λ_1 = $e/[b(1 + r^2/b^2)]$, dimensionless

Λ_2 = $K_\beta/(I\omega_r^2)$, dimensionless

Λ_3 = $\mu/[2(1 + r^2/b^2)]$, dimensionless

λ_f = $B_f/(M\omega_r)$, dimensionless

λ_β = $B_\beta/(I\omega_r)$, dimensionless

$\exp(\lambda_k + i\omega_k)$ = eigenvalue of Floquet transition matrix $[Q]$ defined by $[x(T)] = [Q][x(0)]$

μ = rotor advance ratio $V/(\Omega R)$, dimensionless
= mass ratio $nm_b/(m_f + nm_b)$, dimensionless

ρ = air density, slug/ft^3

ϕ_j = complex multiblade lead-lag coordinate, as defined by Coleman, dimensionless

ϕ_j^* = conjugate complex of ϕ_j

ϕ_n = mode shape, dimensionless
= amplitude of normal mode, ft

ϕ_x, ϕ_y = nacelle angular deflections in pitch and yaw, respectively, rad

ψ = azimuth angle of rotor or blade, rad or deg

ψ_k = azimuth angle of kth blade, rad

Ω = rotor rotational frequency, rad/sec

ω = frequency, rad/sec
= forcing frequency, rad/sec

ω_b = structural flap-bending frequency at $\Omega = 0$, rad/sec

ω_f = whirling angular frequency in fixed system, rad/sec

ω_i = circular natural frequency of the ith mode, rad/sec

ω_n = natural frequency, rad/sec

ω_r = reference frequency, rad/sec

ω_t = structural blade torsional frequency, rad/sec

ω_x = natural frequency of airframe (hub) lateral mode, rad/sec

$\omega_\beta, \omega_\zeta$ = undamped blade flapping and lead-lag frequencies, rad/sec

$\bar{\omega}_\beta, \bar{\omega}_\zeta$ = undamped blade flapping and lead-lag frequency ratios, ω_β/Ω, ω_ζ/Ω, respectively, dimensionless

$\omega_{\phi_x}, \omega_{\phi_y}$ = nacelle structural pitch and yaw frequency, respectively, rad/sec

Subscripts

DL = design limit

E = pertaining to the engine or engine shaft

F = pertaining to the cooling fan or fan drive shaft

G = pertaining to the transmission

k = pertaining to the kth blade

Ln = occurring at the left end of the nth stiffness element

M = pertaining to turbine, fan, and transmission lumped

mn = occurring between the mth and nth stiffness elements

On = occurring at the right end of the nth stiffness element

R = pertaining to the main rotor or main rotor drive shaft

T = pertaining to the tail rotor or tail rotor drive shaft

Superscripts

$'$ = indicates that impedances have been accumulated from the right

$''$ = indicates that impedances have been accumulated from the left

T = transpose of matrix

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CHAPTER 6

STABILITY AND CONTROL

6-1 INTRODUCTION

Principal attention in this chapter is given to the stability and control requirements for helicopters, as specified by MIL-H-8501. However, this general specification for handling qualities is acknowledged to have significant shortcomings when all classes of helicopters of current interest are considered. This chapter, therefore, includes characteristics desired or required for specific classes where appropriate.

The use of armed helicopters was not considered during the development of the MIL-H-8501 requirements. Subsequent use of helicopters as attack vehicles has shown that stability and control characteristics acceptable for other missions are not adequate for these aircraft. Although data available from existing attack helicopters (AH) have not resulted yet in updated Military Specifications with quantitative requirements, the specific characteristics and handling qualities—both required and desired—are known. This chapter includes consideration of the effects of alternate mission configurations and loadings of these vehicles upon these qualities. The effects of both normal and abnormal operation of weapon systems during a mission also are discussed.

The normal helicopter rotor is inherently unstable. The reasons for this are discussed, together with the methods available for modifying the stability characteristics. The latter treatment includes the use of stability augmentation to achieve helicopter handling qualities similar to those of a fixed-wing aircraft.

Once it has been determined that artificial stability devices should be incorporated into the design, it is appropriate to consider the extension of the capability of this equipment to include such pilot-assist or autopilot modes as heading-hold, velocity-hold, or altitude-hold. For specific mission applications, it also may be appropriate to couple this equipment with the onboard navigation system and thereby to provide an automatic flight control system (AFCS). These optional installations also are discussed.

MIL-H-8501 recognizes the increased difficulty of controlling a helicopter in the absence of external visual reference by stipulating more stringent stability requirements for operations under instrument flight conditions than are given for helicopters operated under visual flight rules. Because, as noted in Chapter 1, this handbook deals only with helicopters to be operated under day or night visual flight rules (VFR), instrument flight rules (IFR) operation is not discussed in detail. The requirements of MIL-H-8501 are presented, and the need for them is mentioned. However, neither their adequacy nor their implementation is discussed.

For an understanding of the mathematical processes and techniques employed in the solution of the equations of motion, as used in this chapter, reference should be made to a text on the subject, such as Ref. 1.

6-2 FUNDAMENTALS OF HELICOPTER STABILITY AND CONTROL

6-2.1 DEFINITIONS

Stability and control analysis is a complex process because it deals with both linear and angular motions of the helicopter with respect to each of the three axes necessary to define the position of the vehicle in space. Stability is the tendency of the helicopter to maintain or to deviate from an established flight condition. Control is the ability of the helicopter to be maneuvered or steered from one flight condition to another. The term "flying qualities" is used to designate those characteristics that are relevant to both of these aspects.

Helicopter stability and control analyses are similar to those for other aircraft types but are complicated by the ability of the helicopter to hover as well as to fly in any direction without change of heading.

6-2.1.1 Axis Systems

Fundamental to the analysis is the establishment of a reference or coordinate system to define the position and motion of the helicopter. Right-handed systems of Cartesian coordinates are conventional.

Although the coordinate systems and associated symbols are common, a variety of reference axes may be used. The choice of a reference system is dependent upon the nature of the problem being investigated and the helicopter configuration being analyzed. The most common systems of reference axes are:

1. Gravity axes
2. Stability, or wind, axes
3. Body axes.

6-2.1.2 Gravity Axes

Gravity axes refer to a coordinate system with the origin either fixed at a point on the surface of the earth or at the helicopter center of gravity (CG) and moving with the helicopter.

In each case the z-axis is pointing to the center of the earth (positive downward), the x-axis is directed along the horizon (positive forward), and the y-axis is oriented to form a right-handed orthogonal axis system (positive toward the right).

The gravity or earth axes are useful primarily as a reference system for the gravity vector, and for linear displacements and angular orientation. The use of these axes introduces certain simplifications into the stability analyses, in that the velocity components along the respective axes are independent of helicopter rotation about the CG and are functions only of translation and climb angle γ_c. In the derivation of the equations of motion using this reference system the aerodynamic force contribution is accounted for through the sine or cosine of climb angle γ_c. Further simplifications occur for level flight ($\gamma_c = 0$). However, the use of gravity axes introduces rather cumbersome corrections to inertial terms and products of inertia in accounting for helicopter rotation.

6-2.1.3 Stability Axes

The stability axes represent a coordinate system with the origin located at the helicopter CG and with the axes oriented so that the x-axis is coincident with the velocity vector and is positive pointing into the relative wind, the z-axis is perpendicular to the relative wind and is positive downward, and the y-axis is oriented to form a right-handed orthogonal axis system (positive to the right). The use of stability axes eliminates the terms containing the velocity components v_0 and w_0 along the y- and z-axes, respectively, and thus introduces substantial simplifications into the aerodynamic terms. In this case, the only linear velocity component existing is u, which is independent of helicopter rotation (as in the case of gravity axes) and which represents perturbation of the forward velocity vector u_0. However, the moment of inertia and product of inertia terms vary for each flight condition. In general, these terms are assumed to be constant in the equations of motion. This limits the use of the stability axis system to small disturbance motions.

In addition to simplification of the aerodynamic terms, the use of stability axis systems of reference has a specific application in correlating the theoretically calculated stability results with wind tunnel results, which commonly are resolved automatically parallel with and perpendicular to the wind.

6-2.1.4 Body Axes

The body axis system refers to an orthogonal system of axes fixed at the helicopter CG, rotating and translating with the helicopter. The x-axis is aligned along a reference line (datum line) fixed to the vehicle (positive pointing forward). The z-axis is perpendicular to the x-axis, positive toward the bottom of the vehicle. The y-axis is perpendicular mutually to x and z, and positive pointing to the right.

The use of axes fixed to the vehicle insures that the inertial terms in the equations of motion are constant (independent of flight conditions); furthermore, by coinciding one of the body axes with a principal axis of inertia, certain product-of-inertia terms can be eliminated. In this axis system, the aerodynamic forces and moments are dependent upon relative velocity orientation with respect to the body as defined by the angles of attack and sideslip, α and β_s, respectively.

Body axes are useful particularly in the study of helicopter dynamics because velocities and accelerations with respect to these axes are the same as those that would be experienced by a pilot or would be measured by the instruments mounted in the helicopter.

6-2.1.5 Choice of Axes

Because it is more convenient to express aerodynamic and gravitational forces and moments with respect to body axes than to express inertial forces and moments with respect to wind or gravity axes, a body axis coordinate system commonly is selected for a generalized analysis (Ref. 2). Inasmuch as the analyses presented in the paragraphs that follow are perturbation analyses, stability axes generally are employed. Regardless of the reference axis system selected, it is important that the characteristics be understood and that the externally applied forces and moments be related properly to the coordinate system being used. The body axis and stability axis systems are shown in Fig. 6-1. The notation for displacement and orientation of the helicopter also are defined in the figure.

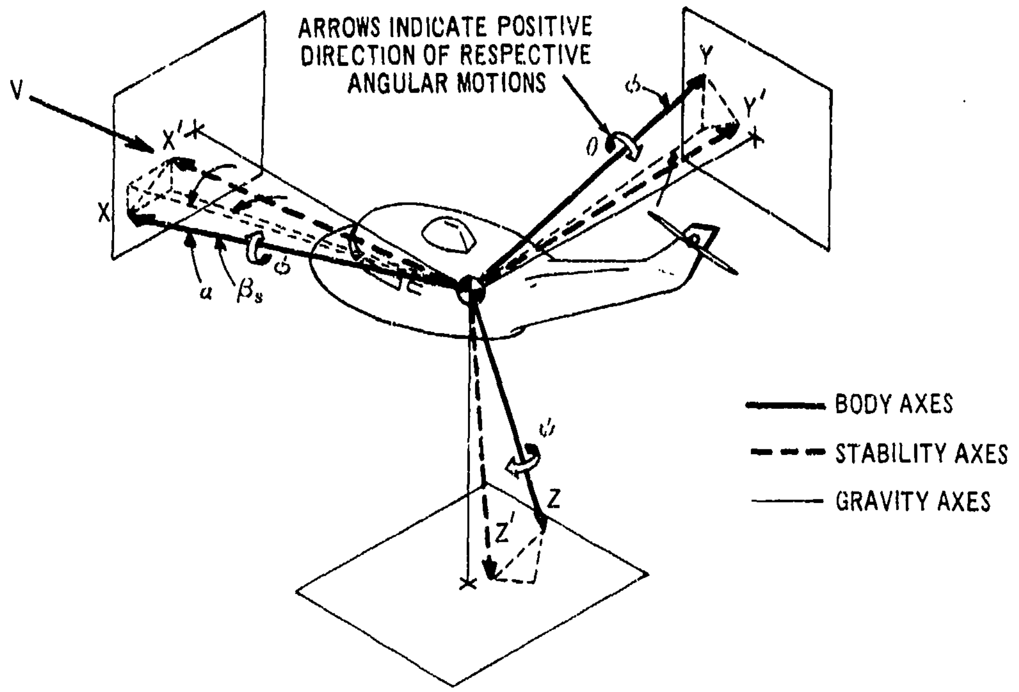


Fig. 6-1. Definition of Axis Systems

In addition to analysis of the helicopter CG motions, the dynamics of the rotor must be included in any investigation of stability and control. Complete consistency must be maintained between rotor and airframe coordinate systems when the analysis includes both.

Rotor system dynamics are described in par. 5-2, along with the instabilities that can occur within the rotor system. The analyses discussed in the paragraphs that follow use the same coordinates and reference system as is used in Chapter 5 when examining the stability and controllability of the rotor system as a part of the helicopter.

The aerodynamic forces and moments applied to the helicopter are discussed in Chapters 3, 4, and 5. In Chapter 3 the development of these forces and moments, including the periodic components of rotor blade lift and drag, is described. In Chapter 4 the application of aerodynamic loads to various parts of the helicopter structure is detailed. In Chapter 5 the analyses include treatment of the helicopter (both rotor and airframe) as an elastic rather than a rigid body.

In each of these chapters, there are minor variations in the mathematical notation. These variations stem from the origins of each of the applicable disciplines (aerodynamics, stress analysis, and dynamics) from different engineering specialties. In a similar manner, individual authors in any one of these disciplines (including stability and control) often use slightly differing notations.

The complexity of helicopter stability analysis is seen readily in Figs. 6-2 and 6-3. These figures, taken from Ref. 2, show the forces and moments acting upon a single-rotor helicopter (Fig. 6-2) and a tandem-rotor helicopter (Fig. 6-3), and demonstrate the application of a body axis reference system. They are presented here as examples that are applicable to a complete and generalized analysis. However, because such a complete system is not investigated here, no effort is made to define all of the parameters indicated.

6-2.2 EQUATIONS OF MOTION

The analysis of stability and control is primarily a study of the solutions of the pertinent equations of motion. An equation of motion is a mathematical statement of the balance between the forces or moments externally applied to a body and the inertial reactions. Ref. 2 presents the generalized equations for all six degrees of freedom (three linear and three angular). These equations were developed for application to any helicopter configuration. They can be particularized by deletion of terms inapplicable to the specific configuration and by evaluation of the pertinent parameters in accordance with the properties, dimensions, and characteristics of the configuration under study. Ref. 1 also treats the development of helicopter equations of motion.

Generalized equations are not presented here because an analysis employing all six degrees of freedom and including all applicable coupling terms is not discussed. Such an analysis is performed most often by means of an analog computer. In the paragraphs that follow, simplified analyses are discussed. At most, three degrees of freedom are treated at one time, with most analyses being concerned with a single degree of freedom. Also, the analyses performed for the most part are concerned with small perturbations from trim. The pertinent equations, therefore, are simplified by the omission of all terms having no effect upon the resulting motion(s).

Prior to analysis of the equations of motion, it is customary to reduce the coefficients to nondimensional form. In this manner the equations are simplified further. Also, the analysis of the effects of the geometric and inertial characteristics of a particular configuration upon a generalized solution is performed more readily. Several methods are in common use for the reduction of the equations to nondimensional form. A completely consistent method must be used, of course, in the development of a set of equations. Review of a given equation, or set of equations, also is meaningful only with a complete understanding of the method used; the variables may or may not have been nondimensionalized, for example. If, as is common in airplane stability analysis, the equations have been nondimensionalized completely so that the solutions represent the time-dependent functions of nondimensional variables, even the time scale is affected. Then, the particularization of the solution also involves the correction of the time scale by a time constant that is dependent upon both a characteristic dimension (wing semichord or semispan are used in airplane equations) and a reference velocity (flight speed or, in helicopter analysis, rotor tip speed). Ref. 1 provides a detailed discussion of the development of nondimensional equations.

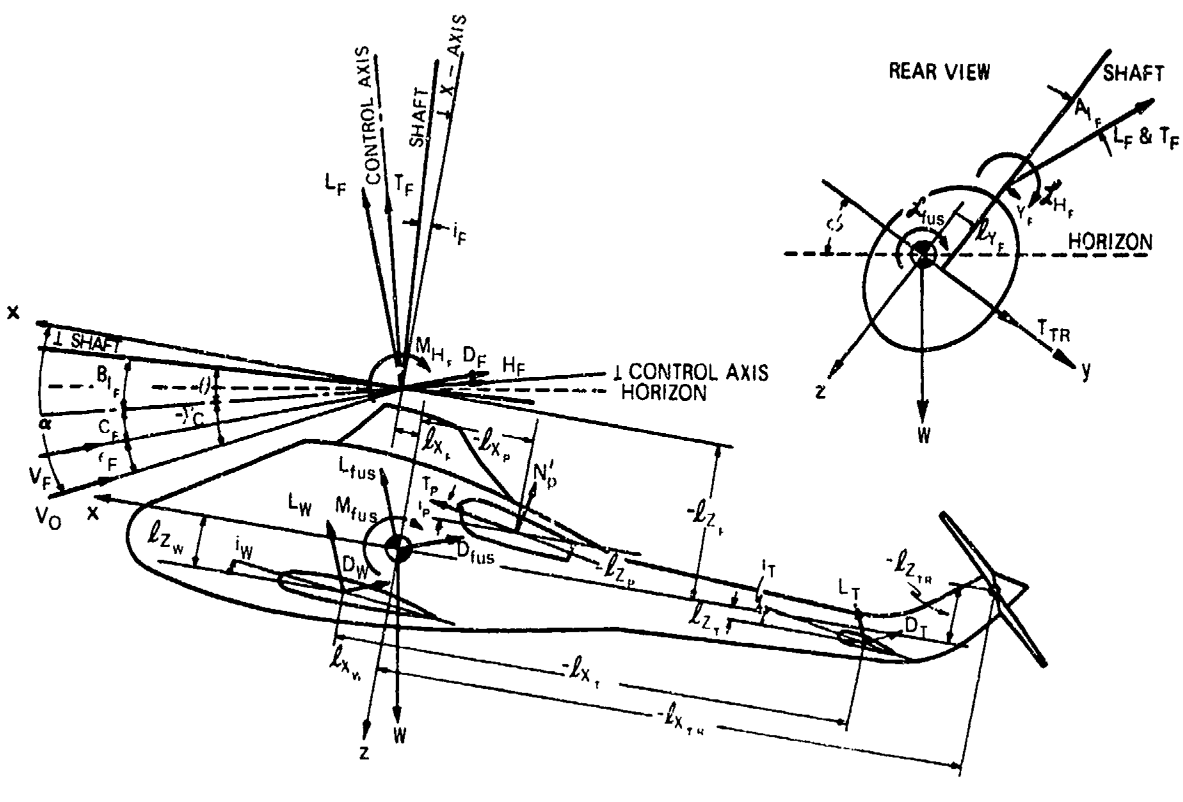


Fig. 6-2. Definition of Parameters and Sign Convention for a Compound, Single-rotor Helicopter

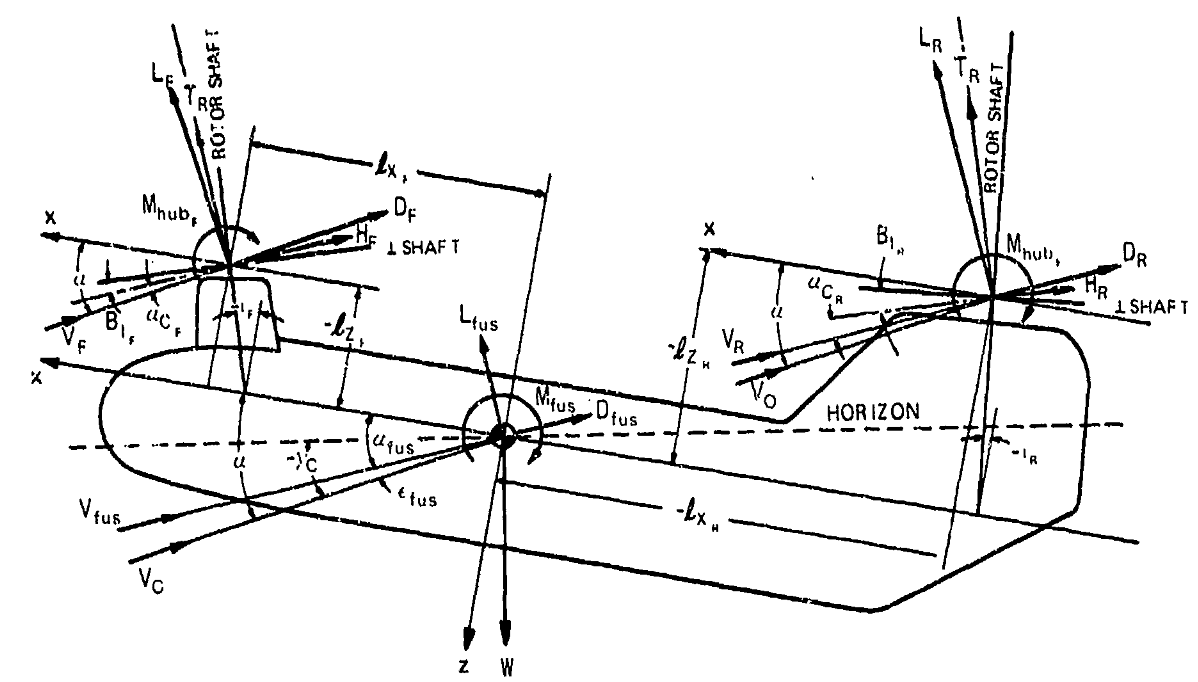


Fig. 6-3. Definition of Parameters and Sign Convention for a Tandem-rotor Helicopter

6-2.3 STABILITY DERIVATIVES

Those terms in an equation of motion that describe the relationship between the aerodynamic forces or moments acting upon the helicopter and the linear or angular displacements or velocities from which they result are called the stability derivatives. These derivatives are dependent upon the geometric and aerodynamic characteristics of the configuration under study. The geometric characteristics generally are known, but the aerodynamic characteristics often are unknown. For example, the rate of change of pitching moment with angle of attack $dM/d\alpha$ includes in its complete form the pitching moment contributions of the complete helicopter, including the rotor. Investigation of pitch stability in forward flight requires consideration of this derivative.

Evaluation of a derivative of this nature can be accomplished in several ways. For example, the helicopter can be separated into its principal components—main rotor, tail rotor, fuselage, wing, horizontal stabilizer, and landing gear—and the contribution of each determined separately. Such a determination is based upon the similarity of the respective components to items for which data are available. The contribution of each component to the moment about the CG of the helicopter, of course, includes not only the aerodynamic moment of each about its own reference center, but also both the magnitude and the direction of lift and drag forces at that reference center. The contribution of these forces to the moment about the CG is obtained as the product of the force and the appropriate moment arm to the CG.

Fundamental to the evaluation of the moment contributions of the individual components is the determination of the local flow environment of each component. Owing to the effect of rotor downwash velocity upon the airflow over the helicopter, such a determination often is difficult. As a result, reliable values for the stability derivatives of the fuselage usually can be obtained only from test data. For preliminary design studies, the data generally are obtained in a wind tunnel.

The wind tunnel determination of helicopter stability derivatives requires considerable care. The model must represent the complete helicopter. Because, as noted, it is necessary to reproduce the flow around the fuselage, the rotor must be operating and must reproduce reasonably the downwash velocity of the full-scale helicopter under the flight conditions being tested. At the same time the model rotor must be capable of isolation from the remainder of the model so that the fuselage forces and moments can be measured independently of the rotor. Even using such techniques, the reliability of the wind tunnel data is not assured because the typical problems of scale effect (Reynolds number) and tunnel wall corrections must be resolved. Furthermore, all desired derivatives cannot be measured in the wind tunnel. For example, satisfactory techniques are not available yet for the determination of angular rate derivatives, e.g., rate of change of pitching moment with pitching velocity.

Stability derivatives also can be determined from flight test data, but this procedure seldom is suitable for preliminary design purposes. The flight test method requires testing of a number of flight conditions, and the test must be planned so as to permit the isolation of one derivative based upon the difference between the results of two tests. Additional derivatives are evaluated in a similar manner. The equations of motion are written, and the solutions of the equations are measured in flight. It then is possible to calculate the derivatives compatible with the measured helicopter responses.

The rotor derivatives can be determined in a number of ways. Ref. 2 contains graphs for several of the derivatives; for the most part, these were obtained by measuring the gradients of test data for rotors that had been tested under many operating conditions. The curves presented include variations of the values of the derivatives as functions of both design characteristics and operating conditions. However, because the original testing had been performed for performance measurement, some simplifying assumptions were required in the use of the data for determination of stability derivatives.

In the paragraphs that follow, most of the derivatives have been simplified during the development of the perturbation equations of motion. In several cases the values of the rotor derivatives have been based upon theoretical expressions.

6-2.4 BASIC STABILITY AND STABILIZATION SYSTEMS

The objective of the paragraphs that follow is to present a clear understanding of helicopter stability and response characteristics. The basic problems are discussed and methods to overcome some of the undesirable features are analyzed. Because this presentation is concerned primarily with fundamentals, the mathematical models used are simplified greatly.

The blade is assumed to be rigid in bending and torsion. It has one degree of freedom—flapping—and is characterized by two independent nondimensional parameters, the Lock number γ and the flapping frequency parameter P where P = flapping frequency divided by rotor angular velocity Ω. By definition, the Lock number γ (Eq. 6-1) is a measure of the ratio of the aerodynamic forces to inertia forces:

$$\gamma = \frac{R^4 ca\rho}{I_b}\text{, dimensionless} \tag{6-1}$$

where

a = blade section lift curve slope, rad^{-1}
c = blade chord, ft
I_b = blade flapwise mass moment of inertia, slug-ft^2
ρ = density of air, slug/ft^3
R = rotor radius, ft

For full-scale rotors generally $4 < \gamma < 12$, and a low value of γ characterizes a heavy blade.

For an unrestrained blade with zero flapping hinge offset, $P = 1$. A hinge offset or elastic restraint increases the value of P. For typical rigid rotor operating at nominal rotor speed, for example, $P = 1.15$. In the case of a rigid rotor, a flapping restraint is added to the mathematical model. Its spring rate is selected such that the flapping frequency of the hinged, inelastic blade is identical with that of the first flap-bending mode of the actual blade.

The rotor blade represents a classical spring-mass-damper system that responds to its excitations with a flapping angle β, measured from a plane normal to the shaft. The flapping angle is written as

$$\beta = a_0 - a_1 \cos\psi - b_1 \sin\psi \tag{6-2}$$

where a_0 represents the coning angle, ψ describes the azimuth position of the blade, and a_1 and b_1 are components of blade flapping relative to a plane normal to the rotor shaft. For a conventional rotor, rotating counterclockwise when viewed from above, a positive a_1 value means a tip-path plane tilt to the rear and a positive b_1 denotes a tilt to the right. In these cases the rotor applies to the fuselage a nose-up pitching moment and a rolling moment to the right. For the calculation of the moments about the CG of the helicopter, two components must be taken into account: the moments generated by the tilt of the thrust vector and the moments due to flapping hinge offset and/or elastic restraint. It is assumed that the moment vector produced by the rotor thrust at all times is normal to the tip-path plane. Unless otherwise stated, the equations derived refer to a centrally arranged flapping hinge.

6-2.4.1 Pertinent Rotor Derivatives

As for fixed-wing aircraft, the following stability derivatives are of significance:

1. Static stability, or rate of change of pitching moment with angle of attack α

2. Speed stability, or rate of change of pitching moment with speed

3. Damping in pitch and roll.

For most helicopter rotors the contribution of the inplane forces to the pitching and rolling moments generally can be neglected, in which case the stability derivatives are proportional directly to the rotor derivatives defined in Eq. 6-3. However, at high forward speeds (high values of advance ratio μ) the contribution of rotor inplane forces can become sufficiently large that they can no longer be neglected.

$$\left.\begin{aligned} a_{1_\alpha} &= \frac{\partial a_1}{\partial \alpha} \quad \text{, static stability derivative, d'less} \\ a_{1_\mu} &= \frac{\partial a_1}{\partial \mu} \quad \text{, speed stability derivative, d'less} \\ a_{1_q} &= \frac{\partial a_1}{\partial q} \quad \text{, damping in pitch, sec} \\ b_{1_p} &= \frac{\partial b_1}{\partial p} \quad \text{, damping in roll, sec} \end{aligned}\right\} \quad (6\text{-}3)$$

These derivatives can be obtained from either experiments or analysis, and have a decisive influence upon the stability characteristics of helicopters.

1. Static stability. Consider a rotor in a wind tunnel. At a given tunnel speed and shaft angle of attack α the rotor is trimmed such that $a_1 = b_1 = 0$. Starting from this trimmed condition the angle of attack is increased. For conditions with essentially constant rotor speed, an increase in α results in a rearward tilt a_1 of the tip-path plane. For advance ratio μ up to approximately 0.6, Eq. 6-4 is a valid expression for static stability derivative a_{1_α}.

$$a_{1_\alpha} = 2.2\mu^2 \quad \text{, dimensionless} \qquad (6\text{-}4)$$

Inasmuch as a negative value of a derivative is required for stability, Eq. 6-4 indicates that the hovering rotor ($\mu = 0$) is neutrally stable and becomes statically unstable with forward speed. More accurately, the static instability increases in proportion with the square of the advance ratio. This leads to several undesirable characteristics, which are investigated in par. 6-2.4.4.

Eq. 6-4 holds only for driven rotors. For an autorotating rotor an increase in shaft angle of attack speeds up the rotor; by definition, this means a decrease in the advance ratio. For an autorotating rotor a_1 essentially is proportional to the advance ratio, and an increase in angle of attack results in a reduction of a_1. Thus, in forward flight an autorotating rotor is statically stable. On the other hand, this operating condition has the disadvantage that at higher advance ratios the rotor becomes very sensitive to angle-of-attack disturbances.

2. Speed stability. Let us consider again a rotor in the wind tunnel. At a given shaft angle of attack and speed the rotor is trimmed such that $a_1 = b_1 = 0$. With the controls fixed, the tunnel speed is increased. As before, a distinction must be made between powered and autorotating operating conditions. For the powered condition, an increase in tunnel speed results in a nose-up pitching moment; i.e., the rotor has positive speed stability, with its value dependent upon the blade and the specific operating conditions. For a typical helicopter near hovering, the value approximates

$$a_{1_\mu} = 0.3 \quad \text{, dimensionless} \qquad (6\text{-}5)$$

At constant shaft angle of attack, the rotor speed of an autorotating rotor changes in the same ratio as the tunnel speed. This means the advance ratio remains unchanged and, therefore, $a_{1_\mu} = 0$. In summary, a driven rotor has positive speed stability and an autorotating rotor has a speed stability that essentially is zero.

3. Pitch and roll damping. For simplicity, consider a rotor subjected to a steady pitching velocity q about the rotor center in hovering. The flapping motion is excited by the gyroscopic moment $2I_b\Omega q$ and the aerodynamic moment $2KI_b\Omega q$. The quantity

$$K = \frac{\gamma B^4}{16} \qquad (6\text{-}6)$$

includes the effect of the tip loss factor B upon the simplified expression for the aerodynamic flapping moment used in the derivation of the Lock number. The parameter K can be interpreted as the specific damping, i.e., damping divided by critical damping, of the flapping motion for the case $P = 1$ and $\mu = 0$. For a nose-up pitching motion the gyroscopic moment has its maximum at $\psi = 270$ deg and the aerodynamic moment has its maximum at $\psi = 0$. A blade with flapping restraint, i.e., $P > 1$, responds to these excitations with

$$a_1 = -\frac{q}{\Omega}\left[\frac{2K(P^2+1)}{(2K)^2+(P^2-1)^2}\right] \tag{6-7}$$

and

$$b_1 = -\frac{q}{\Omega}\left[\frac{(2K)^2-2(P^2-1)}{(2K)^2+(P^2-1)^2}\right] \tag{6-8}$$

For $P = 1$ these equations simplify, and the derivatives with respect to q of the simplified expressions become; damping in pitch a_{1_q}

$$a_{1_q} \approx -\frac{1}{K\Omega} \tag{6-9}$$

and roll-pitch coupling derivative b_{1_q}

$$b_{1_q} \approx -\frac{1}{\Omega} \tag{6-10}$$

The rotor moment generated by a_1 is proportional to the pitching velocity and opposes the pitching motion, i.e., a_{1_q} presents a pitch damping term. The lateral tilt b_1 results in a rolling moment proportional to pitching velocity q. This coupling is highly undesirable and must be kept to a minimum.

4. Pitch-roll decoupling. The most convenient method of countering the pitch-roll coupling is the introduction of an inclination of the flapping hinge. This is denoted as the angle δ_3 and its effect is to couple mechanically the pitch setting θ of each blade with its flapping angle β. To represent this effect, let the pitch variation be

$$\Delta\theta = C_1\beta \tag{6-11}$$

where

$$C_1 = -\tan\delta_3 \tag{6-12}$$

Obviously, the coupling parameter C_1 modifies the effective spring rate of the flapping motion. It can be shown that the nominal flapping frequency parameter P changes to

$$P' = \sqrt{P^2 - 2KC_1} \tag{6-13}$$

According to Eqs. 6-8 and 6-13 the pitch-roll coupling is eliminated if the δ_3 angle is selected such that

$$C_1 = \frac{P^2 - 1 - 2K^2}{2K} \tag{6-14}$$

Fig. 6-4 shows the C_1 values required for decoupling.

As there is no distinction between pitch and roll responses of a hovering rotor, the results obtained for pitch also are applicable to roll. This means, for example, that for $P = 1$

$$b_{1_p} \approx -\frac{1}{K\Omega} \tag{6-15}$$

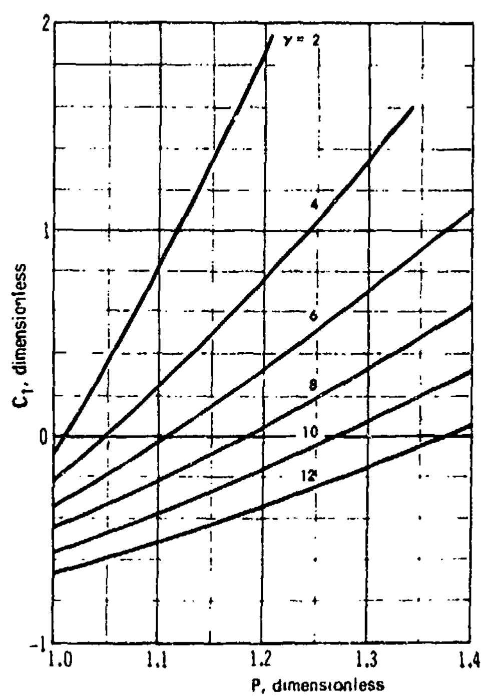


Fig. 6-4. C_1 Values Required for Pitch-roll Decoupling

$$a_{1_p} \approx \frac{1}{\Omega} \quad (6\text{-}16)$$

where p denotes the rolling velocity, positive for rolling to the right. It can be shown that for conventional advance ratios the flight velocity has only a minor effect upon the pitch damping of the rotor. The pitch-roll coupling, however, is affected markedly by the advance ratio, meaning that a compromise between the optimum values of these derivatives must be made for the flight regime of interest.

6-2.4.2 Dynamic Stability of Basic Helicopter

The longitudinal stability of a hovering single-rotor helicopter with fixed controls is investigated first. For simplicity, it is assumed that the longitudinal and lateral motions are decoupled. Because the rotor adjusts itself very quickly to changing operating conditions, it is permissible to use a quasi-static flapping equation of motion and to write

$$a_1 = a_{1_\mu}\mu + a_{1_q} q \quad (6\text{-}17)$$

The rotor derivatives a_{1_μ} and a_{1_q} are given by Eqs. 6-5 and 6-9. The tip-path plane tilt a_1 produces a pitching moment about the CG of the helicopter that can be expressed as

$$M = a_1\left(Wh + \frac{b}{2}k_\beta\right) \quad \text{, lb-ft} \quad (6\text{-}18)$$

where

b = number of blades
W = helicopter gross weight, lb
h = height of rotor hub above CG, ft
k_β = flapping stiffness of rotor blade, lb-ft/rad

The first term in the parentheses represents the contribution due to tilt of the thrust vector and the second term represents the contribution due to flapping. The spring rate k_β takes into account either an elastic blade restraint or a flapping hinge offset. It can be shown for elastic restraint that

$$k_\beta = \Omega^2 I_b(P^2 - 1) \quad \text{, lb-ft/rad} \quad (6\text{-}19)$$

and for a rotor with flapping hinge offset

$$k_\beta = eF_i$$

$$= e\Omega^2 \int_e^R m(r-e)\,dr \quad \text{, lb-ft/rad} \quad (6\text{-}20)$$

where

e = flapping hinge offset, ft
m = distributed blade mass, slug/ft
r = incremental radius, ft

In the latter case F_i is the inertial force due to the flapping acceleration $\ddot{\beta}$. This force acts at the hinge, i.e., at the distance e from the rotor center

Because in the hovering condition the vertical motion of the helicopter is not coupled with all the other degrees of freedom, the dynamic system to be discussed has, in addition to the constraint equation for blade flapping, only two additional degrees of freedom. These are pitch angle, or angle of attack α, and the horizontal translational velocity u of the CG of the aircraft as shown in Fig. 6-5.

The translational velocity of the rotor center is assumed to be identical to that of the CG, i.e., $\dot{\alpha} h \ll u$.

The two equations of motion representing the equilibrium of the horizontal forces and pitching moments read

$$W(\alpha + a_1) + \frac{W}{g}\dot{u} = 0 \quad (6\text{-}21)$$

and

$$a_1 Wh\left(1 + \frac{bk_\beta}{2Wh}\right) - I_y\ddot{\alpha} = 0 \quad (6\text{-}22)$$

where

g = acceleration due to gravity, ft/sec^2
I_y = aircraft moment of inertia in pitch, slug-ft^2

and a_1 is given by Eq. 6-17. The quantity $[1 + bk_\beta/(2Wh)]$ can be interpreted as an amplification factor for the moment derivatives of the rotor. It is worthwhile to note that for rigid rotors with elastic root restraint, this amplification factor can have a value of 10 or higher.

The equations of motion, Eqs. 6-21 and 6-22, lead to a third-order frequency equation

$$s^3 + A_2 s^2 + A_1 s + A_0 = 0 \quad (6\text{-}23)$$

where

s = Laplace operator, sec^{-1}

and

$$\left.\begin{aligned} A_2 &= ga_{1_u} - \epsilon a_{1_q} \quad , \text{sec}^{-1} \\ A_1 &= 0 \quad , \text{sec}^{-2} \\ A_0 &= g\epsilon a_{1_u} \quad , \text{sec}^{-3} \end{aligned}\right\} \quad (6\text{-}24)$$

In these equations,

$$a_{1_u} = \frac{a_{1_\mu}}{\Omega R} \quad , \text{rad/fps} \quad (6\text{-}25)$$

$$\epsilon = \frac{Wh + \frac{b}{2}k_\beta}{I_y} \quad , \text{sec}^{-2} \quad (6\text{-}26)$$

Routh's stability criteria, when applied to Eq. 6-23 in which the coefficient $A_1 = 0$, indicates that the helicopters being discussed are independent of k_β and dynamically are unstable. Generally speaking, the frequency equation, Eq. 6-23, has one negative real root and one pair of conjugate complex roots with a positive real part. The negative real root represents a decreasing aperiodic motion that generally is well damped and, therefore, of little significance. The pair of complex roots with a positive real part describes an unstable oscillation that is characterized by its period of oscillation T_0 and the time required to double amplitude T_D.

In order to give a better insight into the free oscillation of the helicopter T_0 and T_D are shown as functions of A_0 and A_2 in Figs. 6-6 and 6-7. These stability charts apply for $A_1 = 0$ and are taken from Ref. 3. Fig. 6-6 gives the period of oscillation T_0 and Fig. 6-7 the time to double amplitude T_D. A typical helicopter with $k_\beta = 0$ and $Wh/I_y = 0.5$ sec^{-2} is characterized by a frequency equation with approximately

$$A_0 = 0.1 \text{ sec}^{-3}, \quad A_2 = 0.2 \text{ sec}^{-1} \quad (6\text{-}27)$$

According to Figs. 6-6 and 6-7, for this case

$$T_0 = 16 \text{ sec}, \quad T_D = 4 \text{ sec} \quad (6\text{-}28)$$

This means the particular helicopter is unstable dynamically and has a period of oscillation of 16 sec, while the amplitude of a disturbance is doubled in 4 sec.

Next, the effect of a flapping hinge offset or elastic flapping restraint is investigated. It already has been established that the helicopter remains dynamically unstable. Numerical investigations show that, in general, the term ga_{1_u} in the expression for A_2 (Eq. 6-24) is small when compared with ϵa_{1_q} This means that the intro-

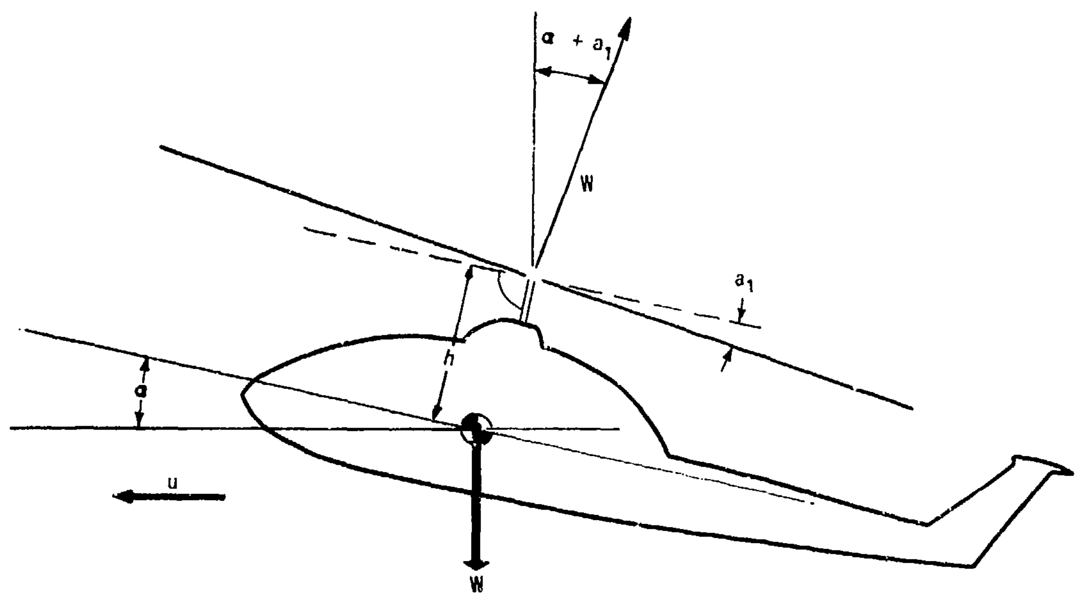


Fig. 6-5. Disturbed Hovering Condition

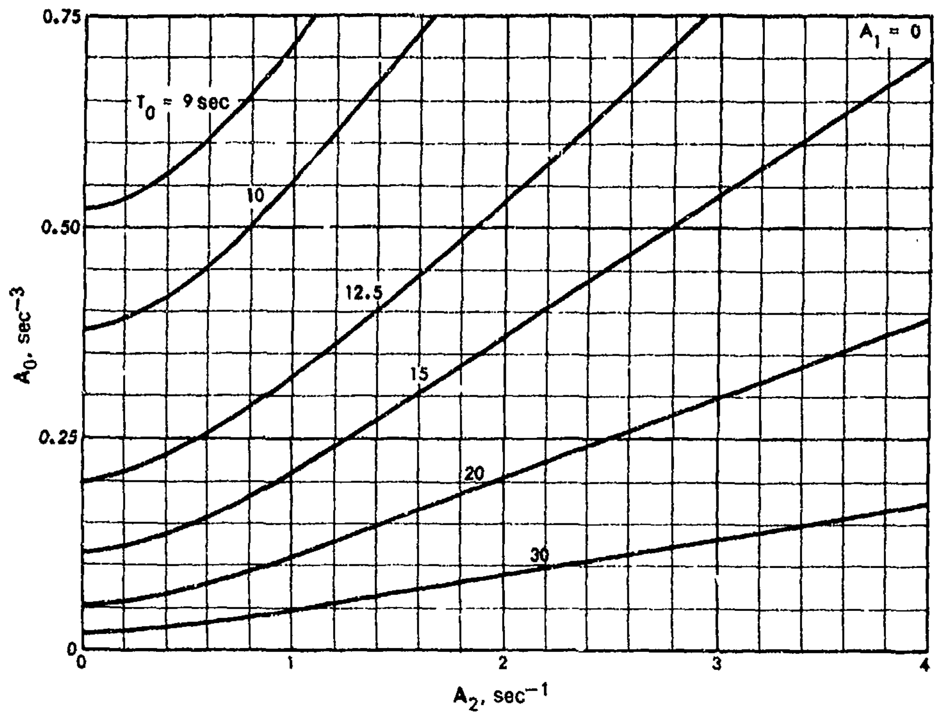


Fig. 6-6. Period of Oscillation for $A_1 = 0$

duction of a k_β effect (flapping hinge offset or restraint) multiplies the coefficients A_0 and A_2 by the same factor by which ϵ is increased. With regard to the stability charts, Figs. 6-6 and 6-7, this is represented by the operating point of the helicopter moving away from the origin of the coordinate system on a straight line that passes through both the origin and the point referring to the helicopter with $k_\beta = 0$. For $k_\beta \rightarrow \infty$ the helicopter becomes neutrally stable.

Let us assume that the helicopter previously investigated (Eq. 6-27) is equipped with an elastic root restraint that multiplies the moment derivatives and, as a result, amplifies the value of ϵ by a factor of 7 5. The coefficients of the new characteristic equation are

$$A_0 = 0.75 \text{ sec}^{-3}, \quad A_2 = 1.50 \text{ sec}^{-1} \qquad (6\text{-}29)$$

According to Figs. 6-6 and 6-7, the period of oscillation and time to double amplitude are changed to approximately

$$T_0 = 10 \text{ sec}, \quad T_D = 6 \text{ sec} \qquad (6\text{-}30)$$

Because the time to double amplitude has been increased slightly (from 4 sec to 6 sec) the dynamic stability is improved slightly. The major improvement, however, lies in the increase of the control power. Because as the initial angular acceleration due to a cyclic control application is proportional directly to ϵ, the addition of the flapping restraint in the given example multiplies the initial angular acceleration by a factor of 7.5; and the unstable motions of the helicopter rotor are controlled much more easily.

6-2.4.3 Fundamentals of Automatic Stabilization

6-2.4.3.1 Control Input Required

Let us assume that a stabilization system has been installed that applies automatic control displacements proportional to and in phase with both:

1. Angular disturbance $\Delta\alpha$ of the fuselage in pitch (attitude)
2. Pitching velocity $q = \dot{\alpha}$ (rate)

The cyclic pitch input θ_1 of the stabilization system then is expressed as

$$\theta_s = -k_1\alpha - \frac{k_2\dot{\alpha}}{\Omega} \quad \text{, rad} \tag{6-31}$$

where k_1 and k_2 are nondimensional quantities that introduce an effective static stability and pitch damping, respectively. In principle, a third automatic control application, proportional to the angular acceleration $\ddot{\alpha}$, could be considered; but the effect of such an input primarily is to change the apparent mass moment of inertia of the helicopter. Therefore, the acceleration term is of minor interest at this point.

The equations of motion for the horizontal forces and pitching moments (Eqs. 6-21 and 6-22) are not affected by the stabilization system, but Eq. 6-17 changes to

$$a_1 = a_{1_\mu}\mu + a_{1_q}q - k_1\alpha - \frac{k_2 q}{\Omega} \quad \text{, rad} \tag{6-32}$$

The equations of motion (Eqs. 6-21, 6-22, and 6-23) lead again to a third-order frequency equation similar to Eq. 6-23. However, the coefficients now read

$$A_2 = ga_{1_u} - \epsilon a_{1_q} + \frac{\epsilon k_2}{\Omega} \quad \text{, sec}^{-1}$$

$$A_1 = \epsilon k_1 \quad \text{, sec}^{-2} \tag{6-33}$$

$$A_0 = g\epsilon a_{1_u} \quad \text{, sec}^{-3}$$

From these equations some interesting conclusions can be drawn. Most important is that because a helicopter with $A_1 = 0$ is unstable dynamically, a pure k_2 input, or increase in damping, is not sufficient to achieve dynamic stability. To obtain stability, a k_1 control application, proportional to the angular disturbance $\Delta\alpha$ is required. The magnitude of the input must be selected such that

$$k_1 > \frac{ga_{1_u}}{ga_{1_u} - \epsilon a_{1_q} + \frac{\epsilon k_2}{\Omega}} \quad \text{, dimensionless} \tag{6-34}$$

However, for best results an optimum combination of the attitude and rate control inputs k_1 and k_2 is desired.

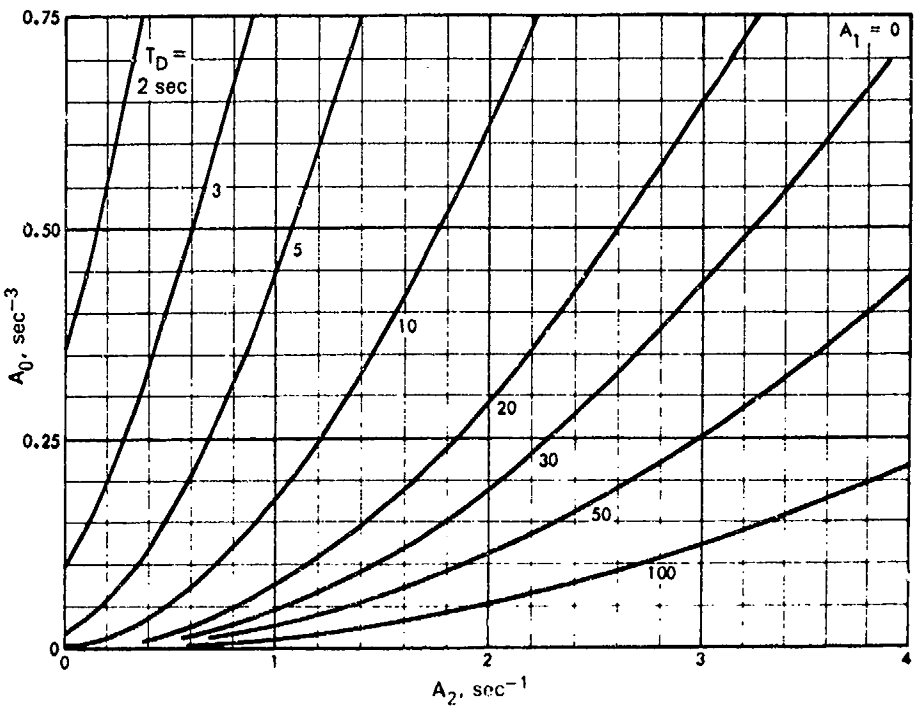


Fig. 6-7. Time to Double Amplitude for $A_1 = 0$

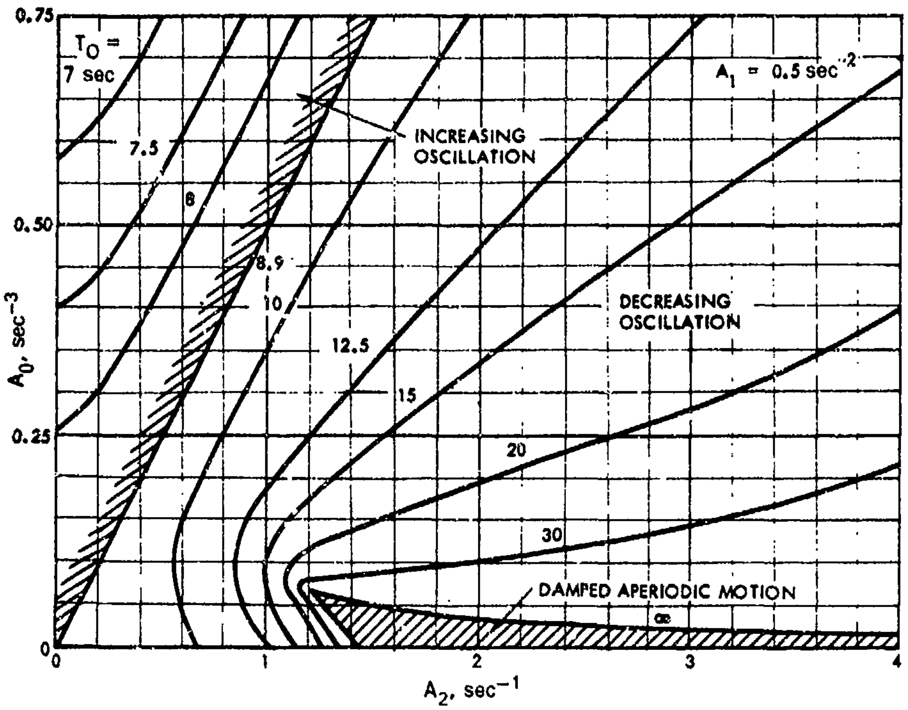


Fig. 6-8. Period of Oscillation for $A_1 = 0.5$ sec^{-2}

To illustrate this see Figs. 6-8 and 6-9, stability charts for $A_1 = 0.5$ sec^{-2}. These charts again present the period of oscillation T_0 and the times T_D, T_H in which the amplitude is doubled or halved, as the case may be. For $A_1 > 0$ the charts are divided into stable and unstable ranges. The stability boundary is defined by a straight line that passes through the origin with the slope A_1. The region above this line corresponds to an unstable helicopter and the region below it to a stable one. For the helicopter investigated, $A_1 = 0.5$ sec^{-2} corresponds to an attitude input $k_1 = A_1/\epsilon = 0.1$. If no additional damping is used, i.e., for $k_2 = 0$, the frequency equation has the coefficients

$$A_2 = 0.2 \text{ sec}^{-1}$$
$$A_1 = 0.5 \text{ sec}^{-2} \qquad (6\text{-}35)$$
$$A_0 = 0.1 \text{ sec}^{-3}$$

According to Fig. 6-9 this combination lies on the stability boundary, which means that the configuration is neutrally stable. However, the dynamic stability can be improved considerably by adding a control input proportional to the angular velocity, or a k_2 term (Eq. 6-31).

Let us assume that by proper selection of k_2 the coefficient A_2 of the frequency equation is increased from $A_2 = 0.2$ sec^{-1} to 0.7 sec^{-1}. For this case, i.e., for

$$A_2 = 0.7 \text{ sec}^{-1}$$
$$A_1 = 0.5 \text{ sec}^{-2} \qquad (6\text{-}36)$$
$$A_0 = 0.1 \text{ sec}^{-3}$$

Figs. 6-8 and 6-9 give

$$T_0 = 11 \text{ sec}, \quad T_H = 3 \text{ sec} \qquad (6\text{-}37)$$

Thus, the dynamic stability is improved significantly, as demonstrated by the change from neutral stability to a response that is well damped. For the helicopter investigated, a change of the coefficients A_2, A_1, and A_0 from the values given by Eq. 6-27 to those given by Eq. 6-36 requires a stability augmentation system with

$$k_1 = 0.1, \quad k_2 = 3.0 \tag{6-38}$$

To summarize the conditions necessary to overcome the inherent instability of a helicopter rotor, an increase in damping, or a control input proportional to and in phase with the angular velocity, is, by itself, not sufficient and merely makes the aircraft less unstable. At best (i.e., for damping coefficient $k_2 \rightarrow \infty$) neutral stability is obtained. Effective stabilization requires a proper combination of automatic control inputs proportional to the attitude and rate of change of attitude. In principle, this applies to both pitching and rolling motions.

6-2.4.3.2 Response Characteristics of Mechanical Gyratory Systems

In the past, several attempts have been made to achieve acceptable dynamic stability characteristics by the use of gyratory systems that rotate with and are coupled to the main rotor. The best known devices of this type are the Bell stabilizer bar and Hiller control rotor. Because they are discussed in detail in par. 6-4, only their basic characteristics and frequency responses are dealt with here.

These gyratory stabilization systems may be evaluated mathematically by considering one blade and one gyro arm, such as a bar, rotating together in a counterclockwise direction. The bar is 90 deg ahead of the rotor blade and pivoted to the shaft. Its angular displacement δ is coupled with the blade pitch setting θ such that the linkage ratio C_2 is given by

$$\Delta\theta = C_2\delta \tag{6-39}$$

The gyro motion is damped slightly and its natural frequency, for all practical purposes, is identical with the rotor angular velocity Ω. The gyro displacement δ is expressed as

$$\delta = \delta_s \sin\psi + \delta_c \cos\psi \tag{6-40}$$

where

ψ = azimuthal position of the blade, rad

δ_s = amplitude of $\sin\psi$ component of gyro displacement, rad

δ_c = amplitude of $\cos\psi$ component of gyro displacement, rad

Therefore, a positive value for δ_s means a tilt of the gyro tip-path plane to the rear and a positive value for δ_c

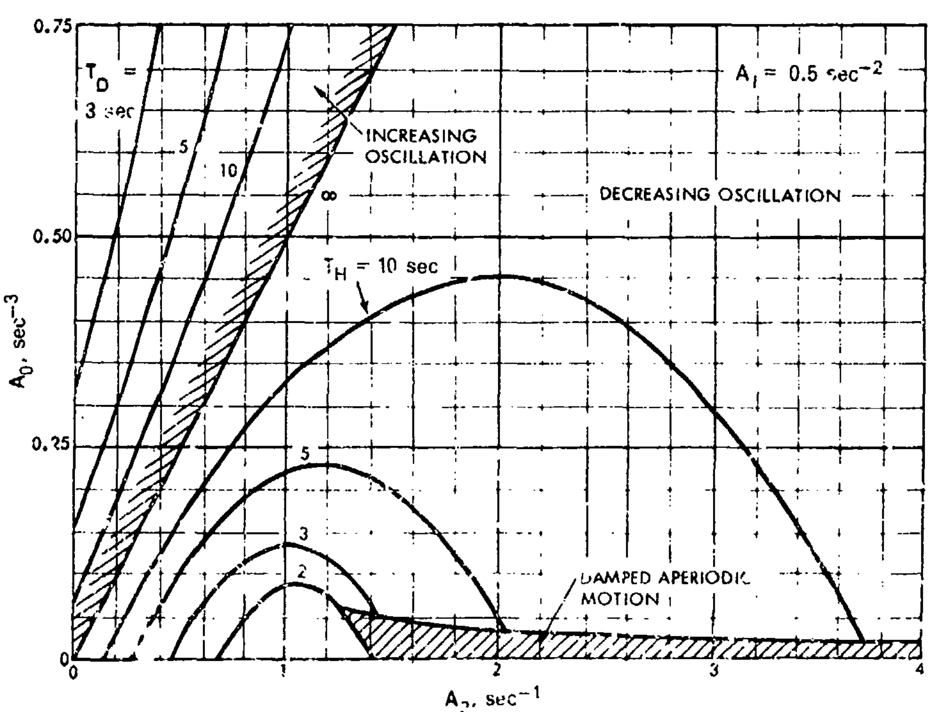


Fig. 6-9. Time to Double or Halve Amplitude for A_1 = 0.5 sec^{-2}

means a tilt to the left. If second-order coupling effects are neglected, the equation for δ_s, which determines the longitudinal cyclic pitch input, reads

$$\Omega\dot{\delta}_s + \bar{G}\Omega^2\delta_s = -\Omega\dot{\alpha} \qquad (6\text{-}41)$$

In this equation, which refers to a pitching motion of the aircraft, the right side represents the gyroscopic moment due to the pitching velocity. It is important to note that the attitude α does not appear as an excitation. The quantity $\bar{G}$ is the gyro specific damping, i.e., $\bar{G}$ = damping/critical damping. In the case of the Bell stabilizer bar, the damping is provided by a viscous damper; the motion of the Hiller control rotor is damped by the aerodynamic forces acting upon the paddles.

In order to determine the cyclic pitch imposed by the gyratory systems upon the rotor, the response of these systems to a steady-state pitching oscillation such as

$$\alpha = \alpha_0 \sin \nu t \qquad (6\text{-}42)$$

is investigated. Here α_0 denotes the amplitude and ν, the frequency of the pitching oscillation. It can be shown that the mechanical systems investigated are equivalent to a stability augmentation system with the characteristics (see Eq. 6-31)

$$k_1 = \frac{C_2\bar{\nu}^2}{\bar{\nu}^2 + \bar{G}^2} \qquad (6\text{-}43)$$

$$k_2 = \frac{C_2\bar{G}}{\bar{\nu}^2 + \bar{G}^2} \qquad (6\text{-}44)$$

In these equations the frequency ν of the pitching oscillation has been replaced by the reduced frequency $\bar{\nu}$ where

$$\bar{\nu} = \frac{\nu}{\Omega} = \frac{2\pi}{\Omega T_0} \qquad (6\text{-}45)$$

Figs. 6-10 and 6-11 show k_1/C_2 and k_2/C_2 as functions of $\bar{\nu}$ and $\bar{G}$. For a rotor angular velocity of Ω = 25 rad/sec, the reduced frequencies shown i.e., $\bar{\nu}$ = 0.005, 0.01, 0.02, and 0.03, refer to periods of oscillation of approximately T_0 = 50, 25, 13, and 8 sec, respectively.

In Fig. 6-11, $\bar{\nu}$ = 0 corresponds to a pitching motion with constant pitching velocity. From Eqs. 6-43 and 6-44 and Figs. 6-10 and 6-11 the following conclusions can be drawn:

1. Most important, the effective k_1 value k_1/C_2 is equal to unity for $\bar{G}$ = 0 and rapidly decreases with increasing gyro damping.

2. The quantity k_2 = 0 for $\bar{G}$ = 0 and has a maximum at $\bar{\nu}$ = $\bar{G}$.

3. In the range of higher damping, k_2 is practically independent of $\bar{\nu}$.

It was shown previously that effective stabilization requires a proper combination of k_1 and k_2 inputs and that k_1 is the more critical quantity. To obtain a proper attitude input k_1, the gyro damping must be kept small. On the other hand, for low $\bar{G}$ values the k_2 input into the rotor is dependent to a great extent upon the frequency ratio $\bar{\nu}$, i.e., on the period of the pitching oscillation, which cannot be selected arbitrarily. Therefore, the gyratory stabilization systems provide minimum latitude for optimization.

This condition is evident from Fig. 6-12, which refers again to the helicopter previously investigated. It is assumed that a stabilizer bar with 3% specific damping has been added. The curves of Fig. 6-12, taken from Ref. 3, show the period of oscillation T_0 and time to half amplitude T_H versus the linkage ratio C_2. As can be seen, the helicopter is approximately neutrally stable. At best, i.e., for C_2 = 0.3, the amplitude of a disturbance is halved in approximately 38 sec. The period of oscillation for this case is T_0 = 26 sec, which for an assumed angular velocity of Ω = 25 rad/sec corresponds to a reduced frequency $\bar{\nu} \approx 0.01$. According to Figs. 6-10 and 6-11 for $\bar{\nu}$ = 0.01 and $\bar{G}$ = 0.03, the gyratory system is characterized by k_1/C_2 = 0.1 and k_2/C_2 = 30. These characteristics being based upon a linkage ratio C_2 = 0.3, this gyro bar system is equivalent to a stability augmentation system with these attitude and rate gains.

$$k_1 = 0.03, \quad k_2 = 9 \qquad (6\text{-}46)$$

Comparison of Eq. 6-46 with the desirable results represented by Eq. 6-38 reveals that the attitude gain k_1 is too low. On the other hand, the rate gain k_2 is about three times larger than desired.

As is to be expected, the addition of a gyratory system reduces the inherent instability characteristics of the helicopter but the results are less than optimum. Both the Bell stabilizer bar and Hiller control rotor act primarily as damping devices for pitch and roll motions, and cannot provide sufficient attitude gain to counteract the instability.

The Hiller control rotor has the added disadvantage that its linkage ratio is fixed as $C_2 = 1$. Further, because the control rotor also is used for rotor control (the pilot applies a cyclic pitch input to the control rotor, which in turn controls the cyclic pitch of the main rotor), a compromise must be made with regard to the control rotor damping $\bar{G}$. Low damping favors the automatic stabilization function and high damping favors the control response.

In view of these shortcomings the current trend is to eliminate gyratory devices and to replace them with more sophisticated electronic systems that provide greater flexibility in selection of system gains and therefore better opportunity for system optimization.

6-2.4.4 Gust Alleviation

6-2.4.4.1 Rotor Sensitivity

The equation of motion for blade flapping (see Ref. 4) can be written as

$$\frac{2}{\gamma}\ddot{\beta} + \Omega\dot{\beta}C(\psi) + \Omega^2\beta\left[\frac{2}{\gamma}p^2 + K(\psi)\right] = \Omega^2\Sigma m(\psi) \quad (6\text{-}47)$$

where $C(\psi)$ and $K(\psi)$ represent aerodynamic damping and aerodynamic spring effects, respectively, which vary with the azimuth angle ψ. The right side of the

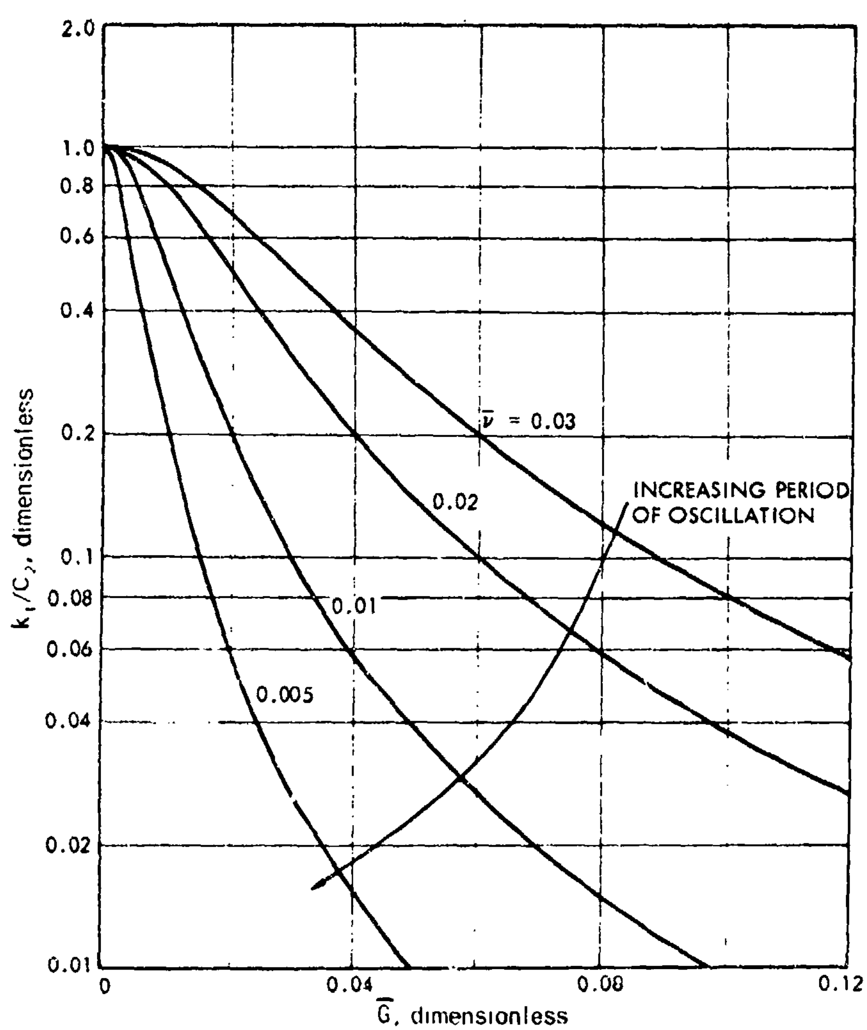


Fig. 6-10. Attitude Input of Gyratory System in Steady-state Pitching Oscillation

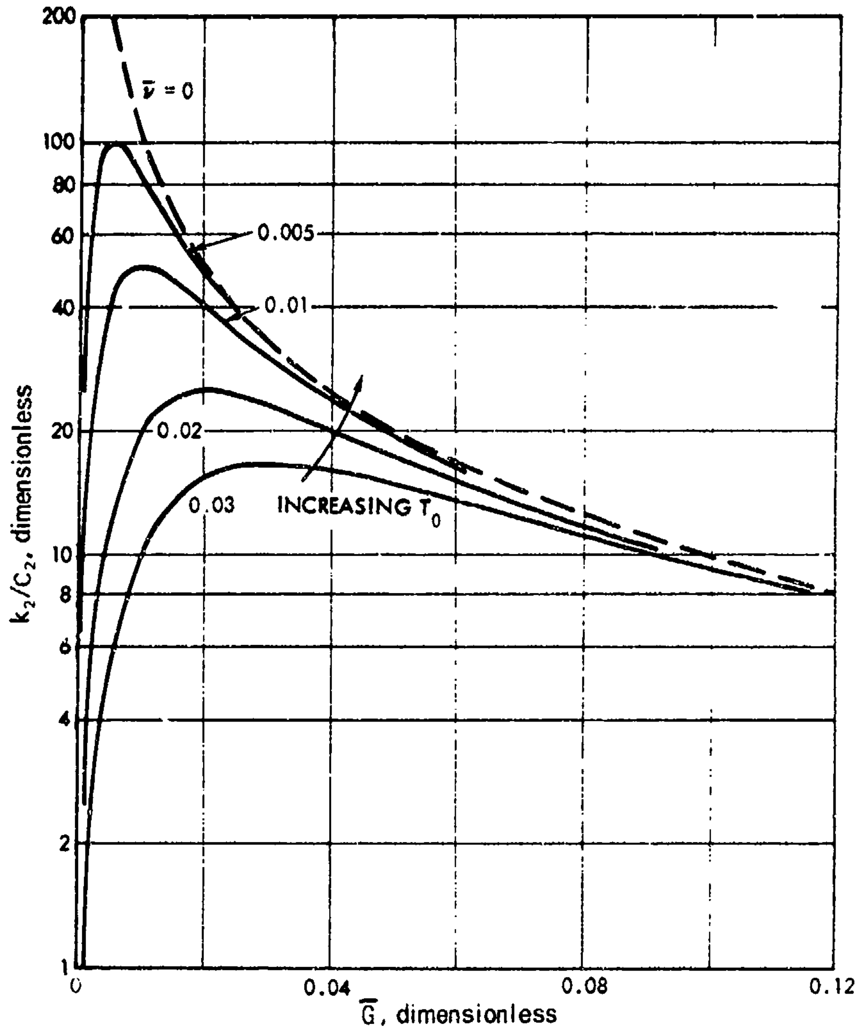


Fig. 6-11. Damping Effect of Gyratory System in Steady-state Pitching Oscillation

equation summarizes the excitations due to inflow, blade twist, and cyclic and collective pitch.

Eq. 6-47 describes a spring-mass-damper system with periodically changing damper and spring rates. As can be seen, the spring rate consists of two components. The constant term $2P^2/\gamma$ represents the restoring moments due to the centrifugal forces and the root restraint. The aerodynamic spring rate, i.e., the periodic function $K(\psi)$, is positive for the rear half and negative for the front half of the rotor disk. The amplitude of the aerodynamic spring rate increases as advance ratio μ increases. As a result, a blade operating at high advance ratios experiences transient negative spring effects at the front of the disk that upset the flapping motion. These effects finally lead to a dynamic instability of the flapping motion, which, for conventional rotors, occurs at approximately $\mu = 2$.

The periodic variation of the spring rate also results in a deterioration of the response characteristics in the stable region. Specifically, at advance ratios above approximately $\mu = 0.4$ the rotor becomes increasingly sensitive to longitudinal disturbances, i.e., to sin ψ excitations such as those produced by gusts or angle-of-attack disturbances. Thus, at higher advance ratios the rotor responds to these excitations with an excessive tip-path plane tilt that transfers undesirable moments

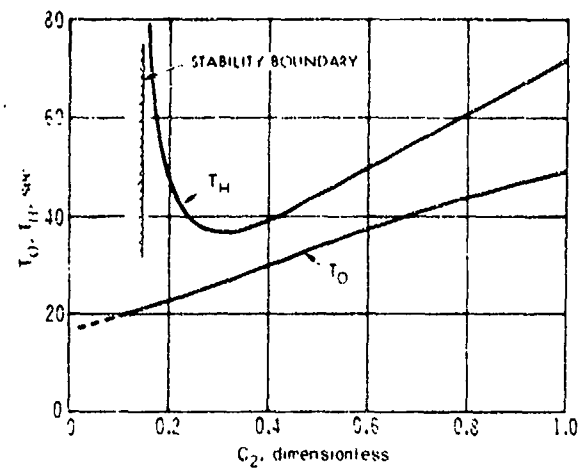


Fig. 6-12. Effect of Gyro System Linkage Ratio C_2 on Hovering Stability

to the fuselage. This especially is true for rigid rotors with elastic blade restraint.

In order to reduce the amplitude of flapping response, a mechanical or electrical servo system is required to counteract the external disturbances automatically by proper cyclic pitch input to the rotor. The next pa raphs evaluate some of the existing systems. Their drawbacks are discussed and the equations of motion for an idealized electrical system are derived.

6-2.4.4.2 Mechanical Feedback Systems

The simplest approach to reducing the flapping response of a rotor to external disturbances would be a mechanism that applies, without time lag, a longitudinal and lateral cyclic pitch proportional to the tip-path plane tilt. Such a system would provide cyclic control inputs having sine and cosine components θ_s and θ_c, respectively, of the form

$$\theta_s = -C_2 a_1 \quad \text{, rad}$$
$$\theta_c = +C_2 b_1 \quad \text{, rad} \qquad (6\text{-}48)$$

where

a_1 = cosine component of blade flapping, rad
b_1 = sine component of blade flapping, rad
C_2 = linkage ratio

Such a system has been investigated (Ref. 5). The feedback was achieved by coupling the pitch setting of each blade in a four-bladed rotor with the flapping motion of the preceding blade. Although the results obtained were not particularly impressive, they will be discussed briefly because they identify the problems that arise.

It can be shown that the feedback given by Eq. 6-48 reduces the amplitude of rotor response to one-per-rev external disturbances by the response factor RF

$$RF = \frac{1}{1 + C_2} \quad \text{, dimensionless} \qquad (6\text{-}49)$$

A conventional rotor with a centrally arranged, free-flapping hinge and no feedback system corresponds to $RF = 1$. A response factor of 0.5 indicates that the amplitude of response to a given excitation is halved. However, both theory and experiments show that for $\mu = 0$ the flapping motion becomes dynamically unstable for $C_2 = 1$ or $RF = 0.5$. Therefore, a 50% reduction in the response is the maximum that can be obtained for a hovering rotor. For $\mu = 0.8$ the stability boundary shifts to $C_2 = 0.6$. This means that at the advance ratio 0.8, a reduction of the response of less than 30% is the maximum possible. These figures reveal a characteristic that seems to be applicable to all feedback systems, i.e., with increasing advance ratio, the stability boundary of the system shifts to lower feedback values or gains; the pure feedback systems have a deteriorating effect on the original stability characteristics of the rotor.

The next logical step is to employ a subsidiary dynamic system that, ideally, is excited by the pitching and rolling moments generated by the rotor. The cyclic pitch applied is dependent upon the equation of motion of the system used. The subsidiary system can be either mechanical or electrical. Representative of the mechanical type is the Lockheed rigid rotor control system, which uses a gyro. The gyro is linked mechanically to the rotor and rotates with the rotor angular velocity Ω. In principle, each blade applies to the gyro a moment M_g proportional to its flapping angle,

$$M_g = \frac{1}{2} I_g A \Omega^2 \beta \quad \text{, lb-ft} \qquad (6\text{-}50)$$

where

I_g = gyro polar mass moment of inertia, slug-ft^2
A = feedback parameter, dimensionless

In the Lockheed system the gyro excitation is obtained indirectly by a forward sweep of the blade with reference to its feathering axis (Ref. 6) although other methods of mechanical implementation also are possible.

For an ideally tuned gyro, i.e., one in which the gyro natural frequency is identical with the rotor angular velocity Ω, the gyro equations of motion read

$$\begin{aligned} &2\Omega\dot{\delta}_s + 2\bar{G}\Omega^2\delta_s \\ &\quad + (\ddot{\delta}_c + 2\bar{G}\Omega\dot{\delta}_c) = -\frac{1}{2}bA\Omega^2 a_1 \\ &2\Omega\dot{\delta}_c + 2\bar{G}\Omega^2\delta_c \\ &\quad - (\ddot{\delta}_s + 2\bar{G}\Omega\dot{\delta}_s) = +\frac{1}{2}bA\Omega^2 b_1 \end{aligned} \tag{6-51}$$

These equations assume that all blades perform identical, first-order flapping motions; as before b denotes the number of blades and $\bar{G}$ is the gyro specific damping. The cyclic pitch applied to the rotor amounts to

$$\begin{aligned} \theta_s &= C_2\delta_s \quad \text{, rad} \\ \theta_c &= C_2\delta_c \quad \text{, rad} \end{aligned} \tag{6-52}$$

where C_2 again represents a linkage ratio.

For the steady-state conditions of a four-bladed rotor, Eq. 6-51 yields

$$\begin{aligned} \delta_s &= -\frac{A}{\bar{G}}a_1 \quad \text{, rad} \\ \delta_c &= +\frac{A}{\bar{G}}b_1 \quad \text{, rad} \end{aligned} \tag{6-53}$$

which corresponds to a response factor of

$$RF \approx \frac{\bar{G}}{\bar{G} + AC_2} \tag{6-54}$$

According to Eq. 6-54, the amplitude of the rotor response is dependent only upon the quantity $AC_2/\bar{G}$. For $\bar{G} = 0$, i.e., zero-gyro damping, the blade does not respond to steady-state, one-per-rev excitations, inasmuch as the cyclic pitch imposed by the gyro fully compensates for the external excitations of the rotor.

Eq. 6-54 suggests that the gyro specific damping must be kept low. The selection of the quantity AC_2 is controlled by stability considerations. It can be shown that for $\mu = 0$ the stability boundary for the rotor blade motion lies at

$$AC_2 = \frac{p^2\bar{\nu}_0^{\,4}}{bK} \tag{6-55}$$

where

$\bar{\nu}$ = reduced frequency of pitching oscillation, frequency ν/Ω, dimensionless

For the unrestrained gyro $\bar{\nu} = 1$. The stability boundary given by Eq. 6-55 represents a static instability that is not affected by the gyro specific damping. For forward flight, the conditions change and the location of the stability boundary depends also upon the value of the gyro damping.

In order to stay away from the stability boundary, one may select 50% of the value given by Eq. 6-55, e.g., $AC_2 = 0.5$. Let us further assume that $\bar{G} = 0.05$. According to Eq. 6-54, in this case the response to a disturbance is reduced to approximately 9% of that without a feedback system. This is a vast improvement over the results obtainable with internal feedback.

However, the gyro may encounter problems of its own. As is to be expected, the allowable feedback A decreases with increasing advance ratio. Fig. 6-13 shows the stability margin for a typical case, applicable to a four-bladed rotor with $P = 1.15$. The graph indicates that for advance ratios above $\mu = 1.2$, the feedback parameter A must be reduced greatly to avoid instability. According to Eq. 6-54 the gust alleviation effectiveness then suffers. This poses a problem for stopped- or slowed-rotor configurations that operate at extremely high advance ratios. This difficulty can be overcome by using a high-speed gyro, rotating with the angular velocity $\omega = f\Omega$. It can be shown that an increase in the speed ratio f has a stabilizing effect and allows use of a larger feedback term AC_2, which, in turn, improves the gust alleviation aspects.

However, all mechanical gyratory systems have inherent limitations. First, their equations of motion contain undesirable coupling terms that cannot be eliminated, as is illustrated by the expressions in the parentheses of Eq. 6-51 that couple pitch and roll control in transient conditions. Secondly, the mechanical gyro systems cease to function properly if the specific damping is too high or if the input signal from the blade (see Eq. 6-50) is disturbed; i.e., if the gyro is excited by moments other than those proportional to the flapping angle β.

6-2.4.4.3 Electronic Feedback Systems

For the reasons presented in par. 6-2.4.4.2, increased attention has been given to electronic systems for stabilization of helicopter rotors. The equations of motion used are identical in principle with those of an ideal

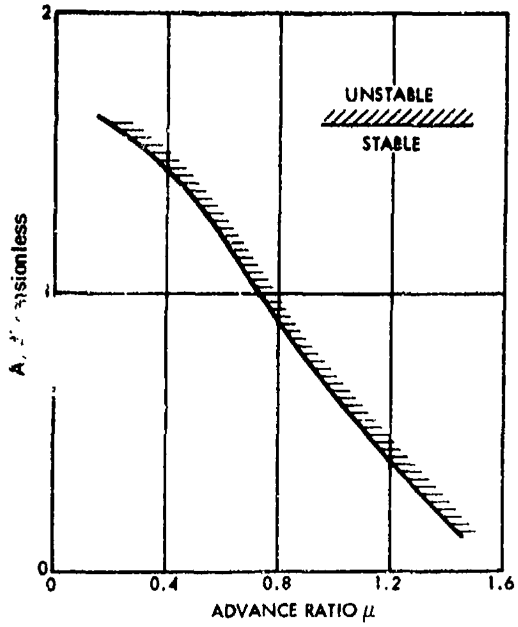


Fig. 6-13. Effect of Advance Ratio on Allowable Feedback Parameter A

gyro. The idealization refers to the elimination of undesirable excitations and to the deletion of the coupling terms in the equations of motion. The latter already have been discussed. An example of an undesirable excitation is the gyro excitation due to a pitching acceleration that results in a lateral-gyro tilt and therefore a roll input into the main rotor.

If the coupling terms in Eq. 6-51 are ignored, the expressions simplify to

$$\left.\begin{aligned} 2\dot{\delta}_s + 2\bar{G}\Omega\delta_s &= -\frac{1}{2}bA\Omega a_1 \\ 2\dot{\delta}_c + 2\bar{G}\Omega\delta_c &= +\frac{1}{2}bA\Omega b_1 \end{aligned}\right\} \quad (6\text{-}56)$$

Again, b denotes the number of blades and the feedback parameter A can be interpreted as a nondimensional actuator gain. According to Eq. 6-54 the electronic system preferably should be designed such that $\bar{G} = 0$. In this case the rotor no longer responds to steady, one-per-rev excitations; but the rotor moment derivatives, i.e., static and speed stability, and damping in pitch and roll also become essentially zero. For these reasons the addition of an autopilot becomes mandatory.

As with the mechanical systems, the allowable gain A decreases with increasing advance ratio and must be selected such that a good compromise is obtained with respect to transient response to control input, transient response to external rotor excitations, and rotor dynamic stability. Generally speaking, a decrease in the gain A improves the dynamic stability of the flapping motion but has a deteriorating effect upon the transient response of the rotor to a control input by the pilot.

If the electronic feedback system senses pitching and rolling moments, i.e., moments in the nonrotating coordinate system, Eq. 6-56 must be applied to all blades. For this case, the equations of the feedback system read

$$\left.\begin{aligned} 2\dot{\delta}_s + 2\bar{G}\Omega\delta_s &= +A\Omega\sum_{i=1}^{b}\beta_i\cos\psi_i \\ 2\dot{\delta}_c + 2\bar{G}\Omega\delta_c &= -A\Omega\sum_{i=1}^{b}\beta_i\sin\psi_i \end{aligned}\right\} \quad (6\text{-}57)$$

If the pilot applies manual control, additional excitations appear on the right side of Eq. 6-57.

The quantities δ_s and δ_c determine the swashplate tilt and cyclic pitch applied to the main rotor. However, instead of using the equation

$$\Delta\theta = C_2(\delta_s \sin\psi + \delta_c \cos\psi) \quad (6\text{-}58)$$

applicable to the mechanical gyro, a phase shift can be introduced. This means the cyclic pitch applied to the ith blade is expressed as

$$\Delta\theta_i = C_2\left[\delta_s \sin(\psi_i - \Delta) + \delta_c \cos(\psi_i - \Delta)\right] \quad (6\text{-}59)$$

The angle Δ is an additional parameter that can be used to correct rotor phasing problems for steady-state or transient forward flight conditions. It should be noted, however, that the phase angle Δ may affect the stability characteristics of the system adversely (Ref. 7).

Exploratory investigations on electronic feedback systems are reported in Refs. 7 and 8. Definitive test results are not yet available.

6-2.5 SPECIAL MISSION CONSIDERATIONS

In the preceding paragraphs consideration has been given only to the motions of the helicopter rotor, with or without the benefit of additional systems to improve the stability characteristics. However, it is the motions of the helicopter that ultimately are of interest. The rotor is of importance not only because of its stability characteristics, but also because it is the primary source of the forces and moments required for helicopter control. In the paragraphs that follow consideration is given to the effects of mission loadings or configurations upon control capabilities, characteristics, and requirements. This discussion then is followed by a more detailed examination of helicopter flight dynamics (par. 6-3).

6-2.5.1 Center of Gravity Travel

For a given helicopter, the maximum allowable CG travel from the design position will correspond to the possibility of reaching one of the following limitations at some flight condition:

1. All of the available control motion used for trim (either longitudinal or lateral)
2. Blade flapping above design limits
3. Fatigue stresses in blades, hub, and mast above design limits
4. Unacceptable longitudinal stability.

The methods for determining when these limits are reached are discussed in the paragraphs that follow.

6-2.5.1.1 Single-rotor Helicopters

The blade flapping required to balance a CG offset on a single-rotor helicopter can be calculated by examining the forces and moments acting upon the helicopter in hovering flight as shown in Fig. 6-14.

The summation of moments M about the CG is

$$\sum M = M_H - T(\bar{x} - ha_1) = 0 \qquad (6\text{-}60)$$

where M_H is the pitching moment of the rotor on the hub and the other symbols are defined in Fig. 6-14.

The pitching moment at the hub due to a single blade may be found by integrating around the azimuth the fore-and-aft component of the rotating moment due to centrifugal force CF, hinge offset e, and cyclic blade flapping a_1.

$$M_{H_{one\ blade}} = \frac{1}{2\pi}\int_0^{2\pi}(CF)ea_1\cos^2\psi\, d\psi$$

$$= \frac{1}{2}(CF)ea_1 \quad \text{, lb-ft} \qquad (6\text{-}61)$$

For a rotor with b blades

$$M_H = \frac{b}{2}(CF)ea_1 \quad \text{, lb-ft} \qquad (6\text{-}62)$$

Thus, the cyclic flapping component required to balance the CG offset in hover is

$$a_1 = \frac{\bar{x}}{h\left[1 + \dfrac{b(CF)e}{2Th}\right]} \quad \text{, rad} \qquad (6\text{-}63)$$

where the symbols not previously defined are again as shown in Fig. 6-14. For a blade with a uniform spanwise mass distribution

$$CF = \frac{3I_b\Omega^2}{2R} \quad \text{, lb} \qquad (6\text{-}64)$$

A convenient, nondimensional parameter relating blade inertia and aerodynamic characteristics is the Lock number γ (described in par. 6-2.4) as

$$\gamma = \frac{c\rho aR^4}{I_b} \qquad (6\text{-}65)$$

Combining Eqs. 6-63 and 6-64 and the definition for rotor thrust-coefficient/solidity ratio $C_T/\sigma = T/[\rho bcR(\Omega R)^2]$ with Eq. 6-65 gives the required flapping a_1 in a convenient form:

$$a_1 = \frac{\bar{x}}{h\left[1 + \dfrac{3ae}{4\gamma h(C_T/\sigma)}\right]} \quad \text{, rad} \qquad (6\text{-}66)$$

For teetering rotors or articulated rotors with zero hinge offset, e in Eq. 6-66 is zero and in that case

$$a_1 = \frac{\bar{x}}{h} \quad \text{, rad} \qquad (6\text{-}67)$$

Referring to Fig. 6-14, it is apparent that the amount of flapping necessary to balance a large forward CG offset may become large enough to permit the blade to strike the tail boom. In order to prevent this, adequate clearance *shall* be provided between the rotor blades and tail boom or other fuselage structure. In establishing this clearance, provision *shall* be made for the additional blade deflection occurring as a result of pilot control inputs, turbulence and/or gusts, and harder-than-normal landings (up to the design limits of the landing gear) while operating at maximum allowable forward CG position. For helicopters with little or no hinge offset, sufficient clearance may be achieved only with an increase in rotor mast height. This increased height also may result in increased weight.

Hingeless or spring-restrained rotors have an effective hinge offset e_{eff} that can be determined from Eq. 6-62 if the rotor stiffness M_H/a_1 is known, or from Eq. 6-68 if the flapwise natural frequency ω_n is known.

$$e_{eff} = \frac{2}{3} R \left[\left(\frac{\omega_n}{\Omega} \right)^2 - 1 \right] \text{, ft} \qquad (6\text{-}68)$$

Keeping the required CG travel to a minimum should be a design goal even on the so-called rigid rotors, with which large CG offsets are balanced by small amounts of blade flapping but at the expense of high blade, hub, and mast stresses. If these stresses are high enough to be critical in the fatigue-loading spectrum, they will require additional structural weight in the rotor components.

For maximum flexibility in tactical operations, and to permit rapid loading under combat conditions, indiscriminate loading capability is desired for Army helicopters. However, extension of the allowable range of CG travel beyond reasonable limits must be judged against potential increases in structural weight and corresponding decreases in payload.

The increments of cyclic pitch ΔB_1 and ΔA_1 required to produce the longitudinal flapping a_1 are

$$\Delta B_1 = -a_1 \text{, rad} \qquad (6\text{-}69)$$

and

$$\Delta A_1 = \frac{12e}{R\gamma} a_1 \text{, rad} \qquad (6\text{-}70)$$

Lateral CG offsets are balanced with lateral flapping b_1, and the increments of cyclic pitch required to produce this flapping are

$$\Delta A_1 = b_1 \text{, rad} \qquad (6\text{-}71)$$

and

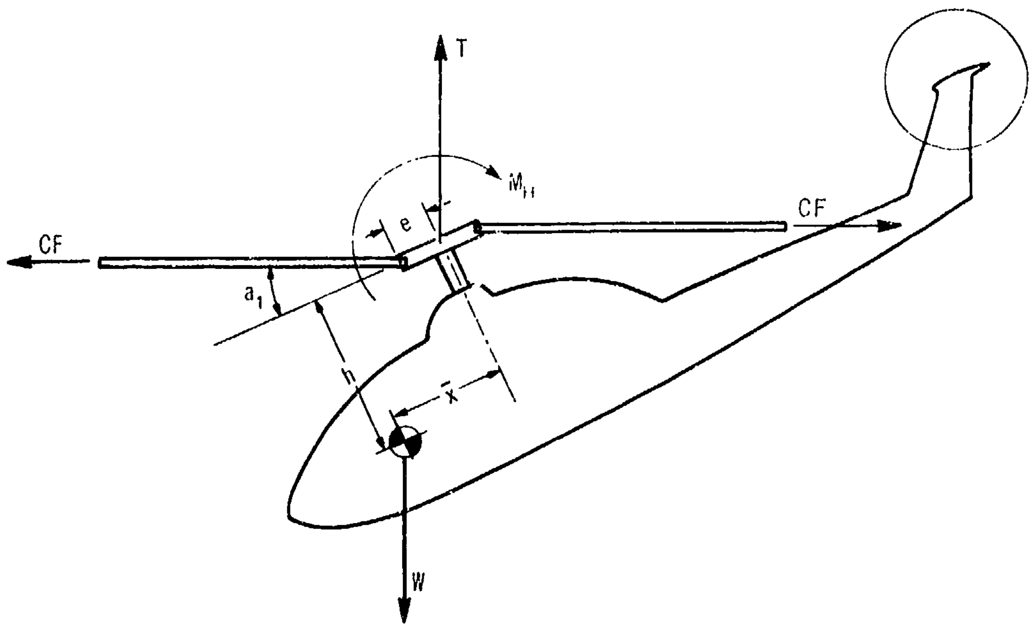


Fig. 6-14. Forces and Moments Acting on a Single-rotor Helicopter While Hovering

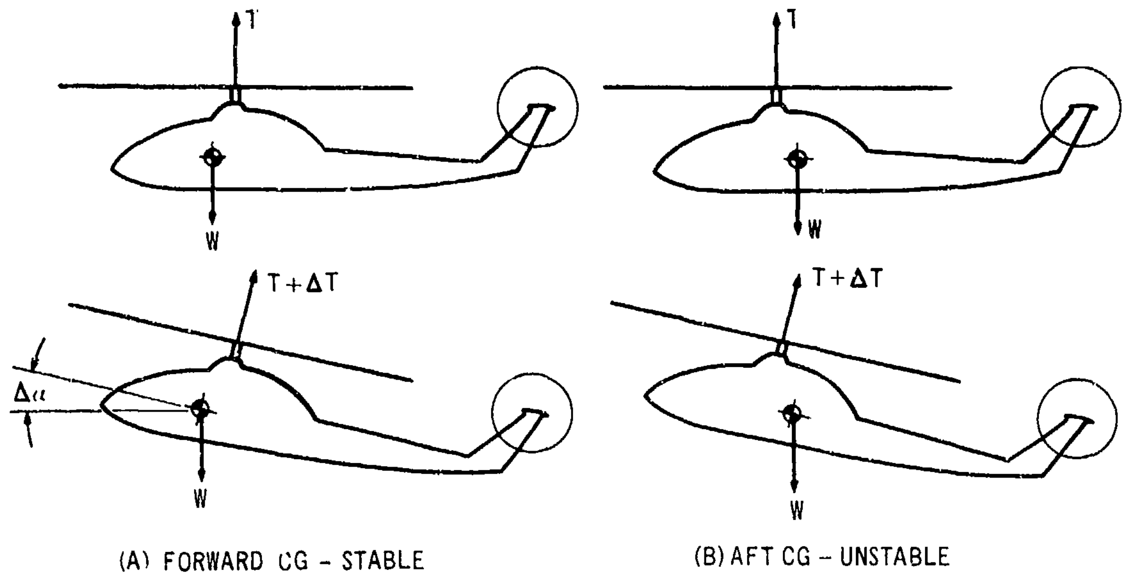


Fig. 6-15. Effect of Center of Gravity Position on Stability

$$\Delta B_1 = \frac{12e}{R\gamma} b_1 \quad \text{, rad} \qquad (6\text{-}72)$$

The increments of cyclic pitch ΔA_1 and ΔB_1 apply to forward flight as well as to hovering, and must be added to the basic cyclic pitch required to position the tip-path plane perpendicular to the mast in order to determine the total cyclic pitch required. MIL-H-8501 requires that, in all steady flight conditions, enough cyclic control to provide pitch and roll moments equal to at least 10% of the moments obtainable in hover remain above the control required for trim. The critical case, with respect to maximum longitudinal cyclic pitch, almost always is an aft CG position and maximum forward speed.

The CG position has an effect upon the longitudinal stability of the helicopter in forward flight. As shown in Fig. 6-15, if the CG is ahead of the rotor, an increase in rotor thrust due to an increase in angle of attack $\Delta\alpha$ will produce a stabilizing nose-down pitching moment about the CG.

If the CG is behind the shaft, the effect will be to produce a destabilizing, or nose-up, pitching moment. This is in addition to the inherent destabilizing moment produced by the rearward flapping of the rotor when subjected to an increased angle of attack. The nose-up moments can be balanced with a nose-down moment from a horizontal stabilizer, but, for a given helicopter in forward flight, a rearward travel of the CG eventually will result in an unstable oscillation or in a pure pitching divergence, depending upon the helicopter and the flight condition.

The stability characteristics as a function of CG position can be determined by the methods discussed in par. 6-2.4. The required degree of stability is specified in MIL-H-8501 and is summarized in Table 6-1.

In addition to the requirements of Table 6-1, MIL-H-8501 specifies that, following a longitudinal control step input, the time histories of both the normal acceleration and the pitching velocity must become concave downward within 2 sec. A helicopter that is characterized by a pure divergence probably will not meet this requirement. This helicopter also may have negative maneuvering stability, i.e., negative stick force per g, unless equipped with a bob-weight on the stick. MIL-H-8501 does not specifically require positive maneuvering stability, but this is desired for Army helicopters.

The effect of the vertical CG position generally is negligible within the possible range on existing helicopters. External loads that are attached rigidly to the helicopter are destabilizing because they lower the CG and increase the instability of the fuselage. The effects of other types of sling loads are discussed in par. 6-3.5.

6-2.5.1.2 Tandem-rotor Helicopters

A tandem-rotor helicopter compensates for lateral CG offset in the same manner as does a single-rotor helicopter, but longitudinal offset is balanced by differential thrust between the front and the rear rotors.

With a forward CG position, the front rotor must operate at a higher collective pitch than with a neutral CG position; thus, in forward flight the front rotor will have more rearward flapping, or will require more forward cyclic pitch to suppress the flapping. At high speed, a given forward CG position will correspond to the maximum allowable rearward flapping or to the maximum available trim capability. The effect of an aft CG position is similar to that on the single-rotor helicopter; the further aft the CG, the worse the longitudinal stability. For aft CG locations, the moment arm of the front rotor is greater than the moment arm of the rear rotor, and an equal change in rotor thrust due to an equal increase in angle of attack will produce a destabilizing nose-up pitching moment. The limits upon the rearward travel can be determined analytically in the same manner as for the single-rotor helicopter.

6-2.5.2 Hover Height Control

The requirements for precision in hover height control are dictated by the helicopter mission and operating environment. Specific tasks that involve precise height control are hovering sling hookups and load placement, touchdown in turbulent conditions, and personnel on- or off-loading without landing in unfriendly territory with rough terrain and tall trees. Successful accomplishment of these tasks is dependent upon the ability of the pilot to maintain close control over vertical rates and positions.

With regard to height control and precision hovering, MIL-H-8501 requires only that it *shall* be possible to keep the aircraft over a given point on the ground, for all terrain clearances and with winds of up to 3 kt, with cyclic control movements of less than ± 1.0 in. The proposed specifications described in Refs. 9 and 10 state that it should be possible to control vertical velocity within all vertical flight conditions with less than ± 0.5 in movement of the vertical control. In addition, both of these documents state that it should be possible to hover continuously in a designated wind at any height up to the disappearance of ground effect while any point on the aircraft remains within a circle of 3-ft radius. This is to be accomplished without acquiring a velocity of more than 2 fps in any horizontal direction, and without requiring undue pilot effort or skill.

Flight research has shown that the precision achievable in hover is related strongly to the height above the ground. This relationship appears to be more the result of the pilot's reduced sensitivity to visual cues at higher altitudes than to aerodynamic effects. The vertical errors found during this research were about twice as great at 50 ft as they were at 5 ft. However, the average rate of collective pitch control motion was about 50% greater at the 5-ft height, indicating that a higher workload was the price of the increased precision. Meanwhile, pilot opinion data indicated that performance and work load both were improved at 50 ft; thus, the pilots were not aware of the differences in their precision at the two heights. Therefore, the mission requirements for height above terrain must be considered when specifying hovering accuracy.

The design procedure used in developing helicopter height control systems includes obtaining desirable force characteristics, acceptable levels of sensitivity, and a comfortable physical layout. Some form of friction or breakout characteristic always has been required in the collective control to prevent unwanted inputs due to disturbances, pilot-induced oscillations (PIOs), or mass unbalance in the stick. Some recent helicopter designs have incorporated a fixed-friction device for this purpose. Early helicopters used an adjustable-friction device but this was subject to problems due both to foreign objects and to wear.

To reduce the possibility of PIO, several precautions should be taken. The natural frequency of the first

TABLE 6-1
LONGITUDINAL STABILITY REQUIREMENTS (MIL-H-8501)

PERIOD, sec	DAMPING	
	VISUAL FLIGHT (PAR 3.2.11)	INSTRUMENT FLIGHT (PAR 3.6.1.2)
0 TO 5	DAMP TO 0.5 A IN 2 CYCLES	DAMP TO 0.5 A IN 1 CYCLE
5 TO 10	AT LEAST LIGHTLY DAMPED	DAMP TO 0.5 A IN 2 CYCLES
10 TO 20	NOT DOUBLE IN 10 sec	AT LEAST LIGHTLY DAMPED
OVER 20	NO REQUIREMENT	NOT DOUBLE IN 20 sec

fuselage bending mode should be kept above the rotor rotational frequency (when the two coincide it is possible to obtain a PIO that causes the aircraft to oscillate vertically). This is discussed further in Chapters 4 and 5. In the case of crane operations, the characteristics of the cable in relation to the load weight can be important; certain combinations of cable size, material, and length create a situation where vertical PIO may be induced. The control system, including the influence of the pilot's forearm, should not have natural frequencies near the critical one-per-rev frequency.

The need for more accurate control has resulted in the use of feedbacks to the collective stick system to provide a means of controlling altitude. Because of the lack of generally available and inexpensive precision altitude measuring devices for helicopter use, little attempt is made to put a tight feedback system on the collective control. Radar altimeters have been used with success for sonar dipping (lowering a sonar transducer into the water from a hovering helicopter to search for submarines), where accuracy is required. Barometric altimeters also have been used but are not as accurate. Vertical damping can be obtained through a limited-authority inner loop in automatic stabilizing equipment (par. 6-4.3); generally, this has faster response than feedback to the collective stick, but it also requires adequate altitude sensing capability. An instantaneous vertical-speed indicator (IVSI) or integrating accelerometer also may be employed to derive a vertical rate signal, from which additional vertical damping may be obtained. Any method used must include appropriate provision for the lag in the control system. It is necessary to consider not only the collective control system, but the engine control system as well.

Inherent damping results from changes in inflow due to changes in vertical speed. The parameters affecting vertical damping are rotor tip speed, rotor thrust coefficient, solidity ratio, and lift curve slope. Because these parameters are selected for aerodynamic efficiency, there is very little the designer can do to change vertical damping. The vertical damping derivative Z_w can be evaluated for preliminary design purposes by using the following equation:

$$Z_w = \frac{-g}{\Omega R \dfrac{8C_T}{\sigma a}\left(1 + \dfrac{\sigma a}{8\sqrt{2C_T}}\right)}, \text{ sec}^{-1} \quad (6\text{-}73)$$

Therefore, vertical damping can be increased only by decreasing the quotient $RC_T/(\sigma a)$. This can be accomplished, for a given gross weight, only by increasing the rotor blade area bcR. The blade area will have been selected for optimum rotor efficiency and, therefore, as noted, an increase in inherent damping can be achieved only at the expense of performance. Empty weight and, hence, payload probably will be among the parameters adversely affected.

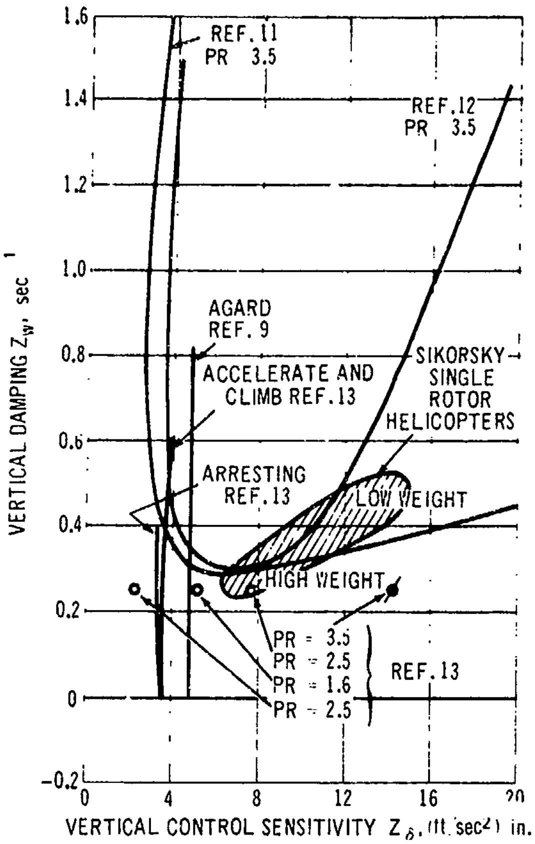


Fig. 6-16. Damping vs Control Sensitivity, Vertical Motion

In terms of vertical control sensitivity Z_δ the helicopter, because of its considerable weight variation with payload, has a wide range of sensitivity. Research has been done in this area for V/STOL vehicles, and some of the applicable criteria are shown in Fig. 6-16, along with typical helicopter values. The boundaries (Refs. 11 and 12) were developed using a moving base simulator. These boundaries have been confirmed to some extent by other investigators using the large NASA-Ames simulator and the NASA-Langley variable stability helicopter (Ref. 12).

Fig. 6-16 indicates the acceptable values of vertical control sensitivity Z_δ to be $Z_\delta \geq 4.0$ (ft/sec²)/in. (0.125 g/in.). Also the vertical damping Z_w will be acceptable only if $|Z_w| \geq 0.3$ sec^{-1}. The ideal damping/control sensitivity relationship has not been determined. The AGARD recommendations for V/STOL handling qualities (Ref. 9) indicate the optimum control sensitivity is on the order of 4.83 (ft/sec^2)/in. (0.15 g/in.) which compares favorably with Refs. 11 through 13. However, flight test data (Ref. 13) show that the simulation studies may be conservative. Nevertheless, it is recommended that the curve labeled PR = 3.5 (Ref. 12) be used as a guide for design.

Rotor pitch/cone coupling resulting from an angle of inclination δ_3 of the flapping hinge away from the normal to the pitch axis has the effect of reducing both the control sensitivity and the damping of the rotor. Therefore, if δ_3 is incorporated, its effect must be considered in setting control sensitivity. For example, to maintain the same control sensitivity with the use of pitch/cone coupling it is necessary to increase the blade pitch travel by an amount approximately equal to the product of tan δ_3 and the hover coning angle a_0. The collective pitch input gradient (deg input/in. stick displacement) also must increase proportionately because the stick travel is fixed.

Control system lags have a pronounced effect upon vertical height control. Refs. 11, 12, and 13 show that as the control system time constant increases, the aircraft becomes more difficult to fly. Increased damping is helpful in enabling the pilot to cope with the increased control system lags.

It is possible for collective control to vary from sluggish to overly sensitive if the variation in operating weight for the helicopter is large. Generally, the pitch input gradient has a fairly high value because a significant collective pitch range is required to accommodate both the high speed and the autorotation settings. This range, together with the desire to keep the collective stick trim positions within a comfortable range consistent with the requirements of MS 33575 (less than 12 in.), results in high sensitivity Z_δ at low operating weights.

For contemporary helicopters the sensitivity usually is established as described, with priority being given to considerations other than hover requirements. Eventually, as more importance is placed upon hover accuracy, the collective control system no doubt will evolve to provide some form of constant response. The technology is adequate now to improve the system; however, the justification for the added complexity is not yet sufficient to support such solutions.

While operating in ground effect (IGE), it has been noted that there is an improvement in controllability. As the helicopter moves into ground effect, the collective control becomes more of a displacement control than a rate control for small changes in trim position. The effect upon approach to a ground plane has been shown (par. 3-2) to be a reduction in power required for a given thrust, or in an increase in thrust at a given power. Therefore, within ground effect, there exists, for a given weight, a unique relationship between power and height, and a change of collective pitch results in a change of height rather than a change of thrust or of rate of climb.

6-2.5.3 Weapon Platform Requirements

Often, mission success is dependent highly upon the ability of the attackers to bring accurate fire upon a wide range of potential targets. Therefore, the performance of a helicopter as a weapon platform is an essential design criterion involving many system trade-offs. The designer must establish weapon system specifications commensurate with the helicopter upon which they will be installed and the missions the helicopter will be required to perform. Because missions vary greatly among types of helicopters, the weapon and fire control requirements also are varied. The level of sophistication of the fire control system and the characteristics of the weapons determine the requirements for platform stability.

In order to obtain specific design requirements for the helicopter as a weapon platform, it is useful to consider the design goals of a helicopter weapon system. This system should be able to detect and identify a target, rapidly hit the target with each round fired, and have the capacity to destroy or immobilize the target. Apart from the capabilities of the weapon system itself, the helicopter characteristics that enhance system effectiveness include:

1. Unobstructed visibility from the cockpit
2. Low vibratory levels in the cockpit and at the weapon station
3. Low external noise level
4. High speed
5. High maneuverability or nap-of-the-earth capability, including high rates of roll, high load factor capability, good sideslip capability, and precision control
6. Safe jettisoning clearance for weapon debris
7. Night or all-weather flight capability
8. Absence of small-amplitude random oscillations
9. Low response to gusts
10. Low response to weapon recoil forces.

The last three items represent the stability characteristics that are important during the actual firing. Current design goals are for first-round accuracy to be within 0.25 to 0.5 deg of azimuth. Because this accuracy includes the helicopter motion as well as all errors in the weapon system, the helicopter must be held at an essentially fixed heading during the aiming process, or, alternatively, the weapons must be aimed and fired with a complex fire control system.

Small-amplitude, random oscillations—primarily in yaw—are common to most helicopters but generally are overlooked in normal flight. The oscillations can be seen during flight by watching the pattern that a speck on the windshield traces out on the landscape. Heading changes of up to 5 deg are common even in calm air. These oscillations may be the result of unsteady separation at the aft end of bluff body components, such as mast fairings, engine nacelles, and cargo compartments, that affects the airflow at the tail rotor, or at the rear rotor in the case of tandems. To minimize this effect, several operational helicopters have mast fairings that act as vortex generators to establish steady flow conditions behind the mast. Pilot efforts to suppress the oscillations with pedal inputs generally are unsuccessful due to the random nature of, and the lags in, the pilot-helicopter response. Simple yaw dampers using rate gyros have resulted in order-of-magnitude decreases in both inherent oscillations and responses to gusts; such dampers should be considered for all weapon-carrying helicopters whose weapon systems are not stabilized electronically.

The response to weapon recoil forces should be sufficiently small that, during steady firing, the helicopter can be held in its trimmed attitude with control motions no larger than 0.5 in. Meeting this requirement may be a problem for a helicopter with a weapon offset horizontally from the CG, or with a turreted weapon offset below the CG.

Requirements for weapon platform stability should be compared with the basic helicopter stability requirements described in par. 6-3.4. Trade-off studies may be appropriate to establish compromises among weapon system power and accuracy, weapon system stabilization, and helicopter maneuverability and stability.

Preliminary design of a helicopter that is to serve as a weapon platform is affected by several factors other than the stability and maneuverability requirements mentioned previously. For example, provisions must be made for internal or external mounting of various weapon and fire control components. In some cases, the design and location of these components must be coordinated with requirements for alternate weapons or alternate uses of the basic helicopter.

If the helicopter is to carry a variety of ordnance, the fire control system must include a weapon selection controller and programmer. Such a unit also is required for conditions that involve launching of missiles and rockets from alternate sides of the helicopter at selective or automatic intervals or sequences. The controller provides the operator with type and quantity of ordnance. The programmer stores basic information on ordnance availability and performs weapon selection sequencing as commanded by the controller.

The requirement for safe jettisoning of weapon debris imposes design constraints that vary with the type of armament system under consideration. Armament subsystems for helicopters include flexible turreted guns, fixed guns, rockets, and missiles. The armament subsystem may or may not be integral to the helicopter. Requirements for flexibility may dictate that the armament system consist of modular "snap-on" components. Thus, the designer must coordinate the fire control requirements with the other mission requirements of the helicopter. External stores should be capable of being jettisoned singly or in multiples, provided that the warheads are unarmed. Depending upon the armament system design, missiles may be jettisoned by free fall or they may be separated explosively from their launchers.

Detail design of weapon system installations is discussed in further detail in AMCP 706-202.

6-2.5.4 VFR/IFR Flight

As noted in Chapter 1, detail requirements for operation under instrument flight rules (IFR) are beyond the scope of this handbook. Nevertheless, it is appropriate to discuss the areas in which these requirements vary from, and are more severe than, the requirements for qualification of Army helicopters for visual flight rules (VFR) day and night.

The first area in which IFR requirements differ from those of VFR is in helicopter stability and control characteristics. Experience has shown that the pilot's ability to control a helicopter under visual conditions is dependent largely upon the visual cues available to him. Deprived of these cues—especially a visible horizon—the pilot suffers a marked increase in workload in order to maintain control of the helicopter attitude. Because the communication and navigation workload also increases, the task often exceeds the capability of a single pilot.

The difficulty encountered by the pilot of a helicopter operating under IFR is not just a matter of workload; in the absence of visual cues—except as provided by instruments such as attitude, heading, angle-of-bank, rate-of-turn, altitude, and rate-of-climb indicators—it is possible for a pilot who is not concentrating upon flight control to become disoriented. In such a case his immediate, or instinctive, reaction to an external disturbance would be incorrect, probably with catastrophic results.

MIL-H-8501 specifies separate stability requirements for VFR and IFR flight (par. 6-2.5.1). In addition, different levels of damping about all three axes are stipulated. Because existing helicopters do not comply with this specification, the validity of the quantitative requirements cannot be confirmed. Nevertheless, there is no question that better stability characteristics are required for instrument flight, and that improvements in stability for most helicopters can be achieved only by the addition of some type of stability augmentation.

Stability augmentation systems are discussed in par. 6-4, as is the evolution from a system that improves stability to an automatic flight control system. The extent to which this evolution is undertaken depends upon the mission requirements for a given helicopter. Many approaches are available, including systems using ground-based navigation aids, airborne sensors (Doppler radar or inertial platforms), or combinations of these.

Minimum instrumentation and avionic requirements for Army helicopters are defined by AR 95-1. This regulation is applicable to flight operations and, therefore, is not a definition of design criteria; however, it recognizes the necessity for such flight instruments as vertical-speed and gyro-rate-turn indicators, as well as for the communication and navigation equipment required for safe and successful operation under instrument flight conditions. The only navigational equipment specified is the automatic direction finder (ADF). Otherwise, it merely is required that the airborne equipment be compatible with the ground-based navigational aids available for a given planned flight.

Based on the problems of flight attitude control, of navigational and flight path control, and of the necessary communication with air traffic control, weather stations, and other stations, Army flight regulations require that the flight crew for IFR helicopter operation *shall* include two instrument-qualified pilots.

The second area in which the IFR requirements exceed those for VFR is in weather protection. Operation under instrument conditions cannot be differentiated readily from "all-weather" operation. Known or forecast instrument flight conditions may include lightning, rain, hail, snow, or ice. Therefore, an IFR-qualified helicopter also should be all-weather qualified. However, true all-weather capability, including the ability to operate during tornadoes and hurricanes or to fly through thunderheads, can be achieved only at a prohibitive cost in helicopter empty weight, if at all. Thus, all-weather is a relative term and the design limits must be defined and must be understood by the operators.

The lack of suitable anti-icing and/or deicing equipment on most helicopters precludes flight through known or forecast icing conditions. In addition to loss of lift and increase in weight due to an accumulation of ice, uneven accumulation or irregular shedding of ice on the rotor blades may cause severe vibrations due to the resulting unbalance. The most severe danger to present helicopters, however, is ingestion of ice into the turbine engines. Ice accumulated on the windshield, fuselage, or engine nacelles may enter the air induction systems if and when it is shed in flight. If this ice enters the engine, it may cause compressor blade damage, compressor stall, or engine flameout, depending upon the severity and duration of the icing conditions.

Safe operation in conditions of heavy snow, hail, or rain also may require protective systems. Ingestion of snow or hail into the engine presents hazards similar to those resulting from ice ingestion. Extended flight in rain can necessitate protection of the rotor blades from the erosion, similar to sandblasting, caused by rain droplets.

Regular operation under instrument conditions will increase the probability of lightning strikes upon a helicopter markedly. Therefore, adequate protection against destructive lightning damage must be incorporated in helicopters to be qualified for instrument operations.

It is anticipated that pertinent requirements for the design of IFR-qualified helicopters will be added to this handbook by amendment as they are established and approved.

6-3 FLIGHT DYNAMICS

6-3.1 CONTROL POWER

Control power is a measure of the total moment or force available to the pilot for maneuvering the helicopter from a steady trimmed flight condition or for compensating for large gust disturbances. It is defined as the moment about the helicopter CG produced by a unit control displacement. Control sensitivity is the rate of change of angular displacement about the CG achieved as a result of a unit displacement of the control. The requirements of, limitations of, and methods

TABLE 6-2
MINIMUM DISPLACEMENT IN DEGREES FOLLOWING A ONE-INCH CONTROL INPUT IN HOVER, MIL-H-8501

RESPONSE	VISUAL FLIGHT, deg	INSTRUMENT FLIGHT, deg
PITCH θ IN 1 sec	$\frac{45}{\sqrt[3]{W+1000}}$	$\frac{73}{\sqrt[3]{W+1000}}$
ROLL ϕ IN 0.5 sec	$\frac{27}{\sqrt[3]{W+1000}}$	$\frac{32}{\sqrt[3]{W+1000}}$
YAW ψ IN 1 sec	$\frac{110}{\sqrt[3]{W+1000}}$	$\frac{110}{\sqrt[3]{W+1000}}$

of computing both control power and control sensitivity are discussed in the paragraphs that follow.

6-3.1.1 Requirements

6-3.1.1.1 Control Sensitivity and Damping Requirements

The combination of control sensitivity and damping governs the ability of the pilot to make precise maneuvers or to hold the helicopter steady in gusty air. Studies of variable-stability aircraft generally result in plots of control sensitivity and damping upon which boundaries can be drawn separating satisfactory from unsatisfactory flying qualities. MIL-H-8501 indirectly specifies minimum values of control sensitivity in hover by specifying the minimum angular displacement in pitch, roll, and yaw at the end of a specified time following a one-inch control step input. These requirements, which are listed in Table 6-2, are a function of gross weight and are based upon the flight testing of a variable-stability helicopter as reported in Ref. 14.

These requirements can be converted into equivalent control sensitivity by assuming that there is no lag in producing the moment and that, in hover, the helicopter can be treated as a mass-damper system. The equation of motion for pitch attitude θ is

$$M = I_y\ddot{\theta} + c_y\dot{\theta}\ \text{, lb-ft.} \tag{6-74}$$

where

M = pitching moment about CG, lb-ft

I_y = moment of inertia in pitch, slug-ft^2

c_y = damping in pitch, lb-ft /(rad/sec)

The solution to this equation is

$$\theta = \frac{M/I_y}{c_y/I_y}\left\{t + \left(\frac{1}{c_y/I_y}\right) \times \left[\exp\left(\frac{-c_y}{I_y}\right)t - 1\right]\right\}\text{, rad} \tag{6-75}$$

Eq. 6-75 can be solved for the pitch control sensitivity $(M/I_y)/\delta_{sx}$ using the displacement $\theta_{1.0}$ at the end of 1 sec

$$\frac{M/I_y}{\delta_{sx}} = \frac{(c_y/I_y)\theta_{1.0}}{\delta_{sx}\left\{1 + \left(\frac{1}{c_y/I_y}\right) \times \left[\exp\left(\frac{-c_y}{I_y}\right) - 1\right]\right\}}\text{, (rad/sec}^2\text{)/in.} \tag{6-76}$$

where

δ_{sx} = longitudinal displacement of control stick, positive aft, in.

If minimum values of θ_1 are evaluated from Table 6-2 and are substituted into Eq. 6-76, a plot of M/I_y versus c_y/I_y can be made. Such a plot is shown on Fig. 6-17 for gross weights of 1000, 10,000, and 100,000 lb for both visual and instrument flight. The corresponding equation for roll control sensitivity $(L/I_x)\,\delta_{sy}$, using the roll displacement $\phi_{0.5}$ at the end of 0.5 sec, is

$$\frac{L/I_x}{\delta_{sy}} = \frac{(c_x/I_x)\phi_{0.5}}{\delta_{sy}\left\{0.5 + \left(\frac{1}{c_x/I_x}\right) \times \left[\exp\left(\frac{-0.5c_x}{I_x}\right) - 1\right]\right\}}, \text{(rad/sec}^2\text{)/in.} \tag{6-77}$$

where

L = rolling moment about CG, lb-ft

δ_{sy} = lateral displacement of control stick, positive to right, in.

and for yaw, using the displacement $\psi_{1.0}$, at the end of 1 sec,

$$\frac{N/I_z}{\delta_p} = \frac{(c_z/I_z)\psi_{1.0}}{\delta_p\left\{1 + \left(\frac{1}{c_z/I_z}\right) \times \left[\exp\left(\frac{-c_z}{I_z}\right) - 1\right]\right\}}, \text{(rad/sec}^2\text{)/in.} \tag{6-78}$$

where

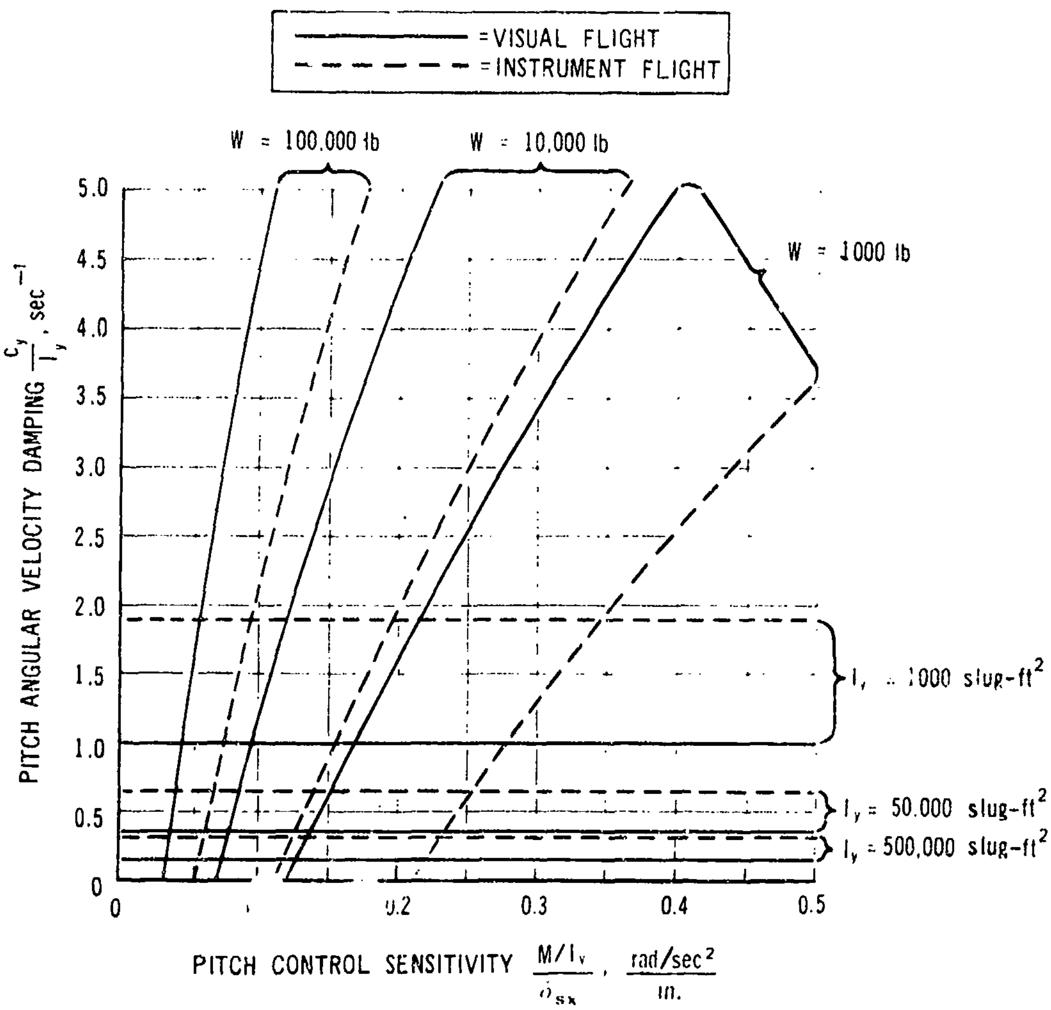


Fig. 6-17. Minimum Control Sensitivity and Damping Requirements in Pitch from MIL-H-8501

N = yawing moment about CG, lb-ft

δ_p = pedal displacement, in.

Plots for the roll and the yaw requirements are shown on Fig. 6-18 and 6-19.

Figs. 6-17 through 6-19 assume that there is no lag between the control input and the development of the moment. In practice, there are at least two lag elements that should be considered when evaluating a control system. The first is the lag of approximately one-quarter of a revolution between the cyclic pitch input and the tilt of the tip-path plane. The second is the time required for the production of the blade cyclic pitch input corresponding to a one-inch control displacement. This lag will be dependent upon breakout forces, lost motion, flexibility of the control system between the pilot's control and the blades, and flow limits on the hydraulic actuators. Some helicopters with gyro control systems will have a further lag because a step control input produces a ramp input of cyclic pitch. All of the time lags that can be accounted for between the control motion and the development of the moment must be subtracted from the appropriate time t to determine the equivalent control sensitivity.

The minimum acceptable damping about each of the three axes are specified in MIL-H-8501 and are listed in Table 6-3. These requirements have been evaluated for moments of inertia of 1000, 50,000 and 500,000 slug-ft^2 and also are plotted on Figs. 6-17 through 6-19.

The response requirements from MIL-H-8501 state that the rate of roll per inch of stick displacement should not exceed 20 deg/sec and that the maximum

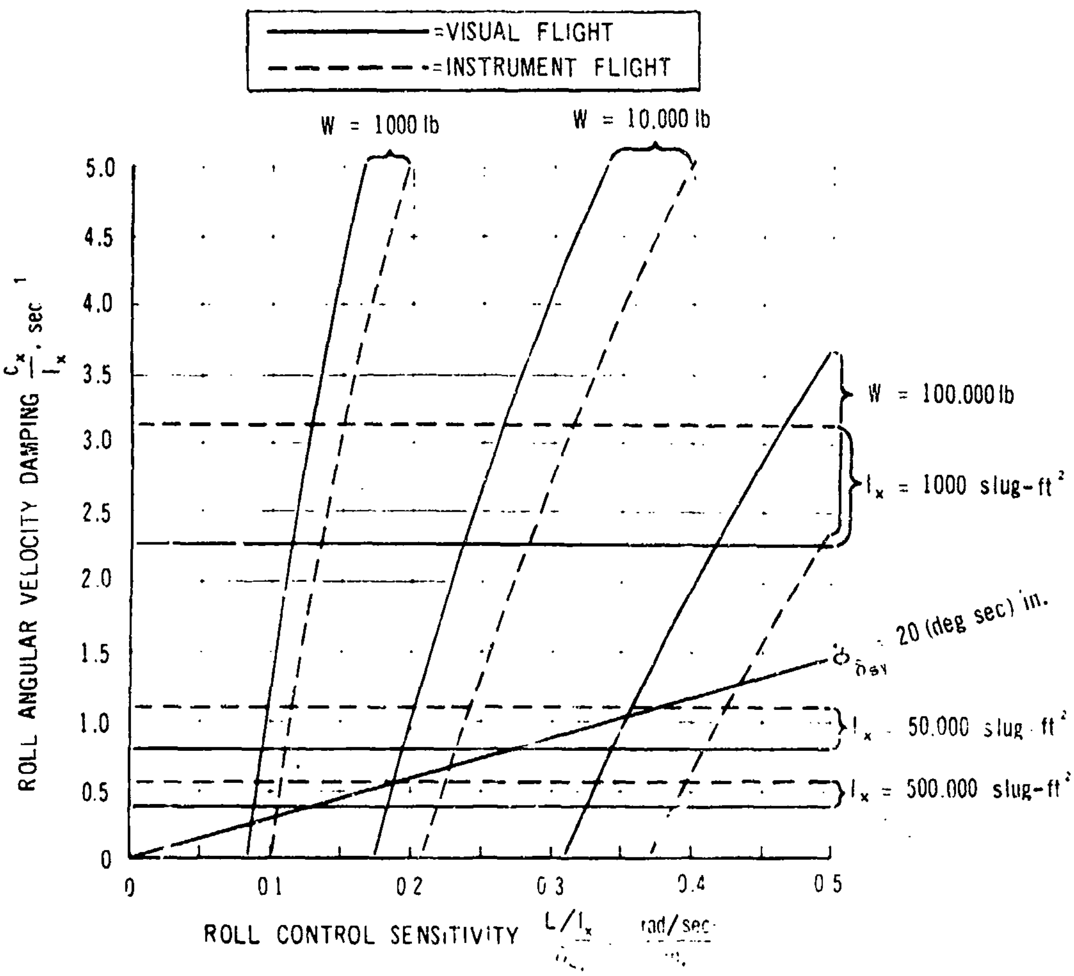


Fig. 6-18. Minimum Control Sensitivity and Damping Requirements in Roll from MIL-H-8501

TABLE 6-3
MINIMUM ACCEPTABLE DAMPING, MIL-H-8501

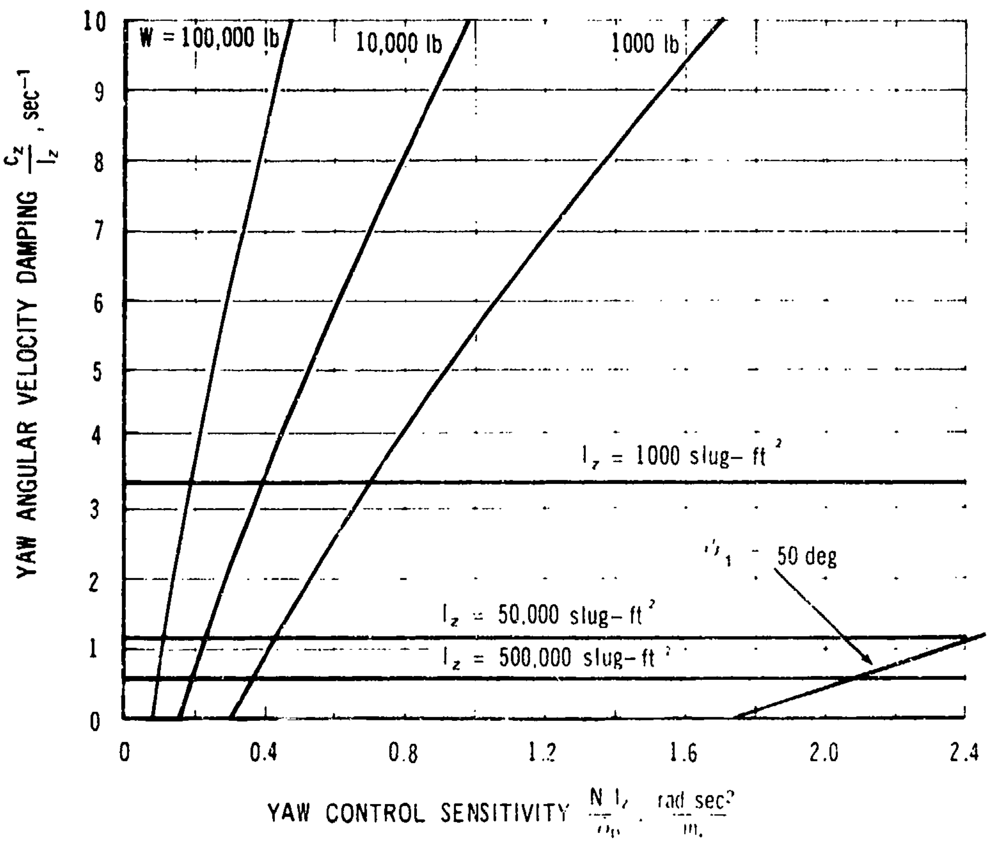
AXIS	VISUAL FLIGHT ft-lb/(rad/sec)	INSTRUMENT FLIGHT ft-lb/(rad/sec)
PITCH	$8I_y^{0.7}$	$15I_y^{0.7}$
ROLL	$18I_x^{0.7}$	$25I_x^{0.7}$
YAW	$27I_z^{0.7}$	$27I_z^{0.7}$

yaw displacement one second after a one-inch pedal input in hover should not exceed 50 deg. These requirements also have been converted into boundaries on Figs. 6-19 and 6-20, respectively.

6-3.1.1.2 Control Sensitivity and Damping Considerations

The requirements shown in Figs. 6-17 through 6-19 are the subject of some controversy, especially with regard to their dependence upon helicopter size. Recently proposed specifications have suggested relating the control sensitivity and damping requirements to the mission of the aircraft rather than to its size.

A comparison of the flight test results of two variable-stability helicopters similar in configuration but different in gross weight—the Sikorsky S-51 and the Sikorsky S-56—is shown in Fig. 6-20 based upon the data of Refs 14 and 15.



NOTE: MINIMUMS ARE THE SAME FOR BOTH VISUAL AND INSTRUMENT FLIGHT BUT FOR VISUAL FLIGHT THE DAMPING MINIMUMS ARE GOALS RATHER THAN REQUIREMENTS

Fig. 6-19. Minimum Control Sensitivity and Damping Requirements in Yaw from MIL-H-8501

HELICOPTER		W	I_y	I_x
S-51	(TYPICAL)	5,500	7,000	2,000
S-56		30,000	123,000	73,000

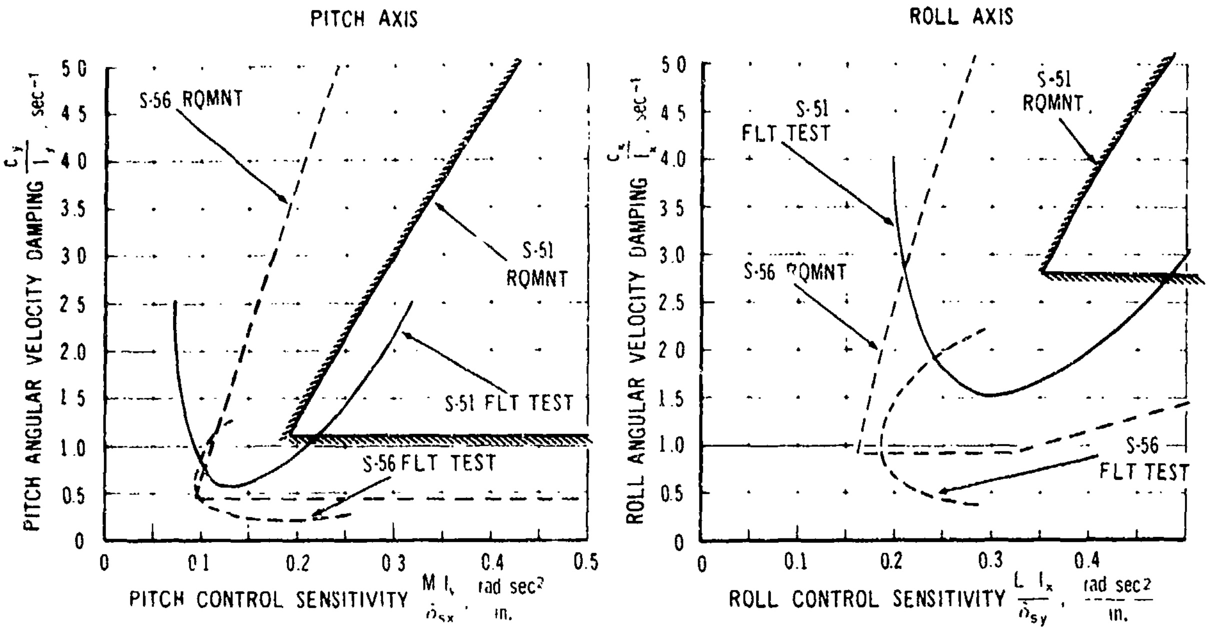


Fig. 6-20. Comparison of Flight Test Results With MIL-H-8501 Requirements for Instrument Flight

The MIL-H-8501 requirements applicable to each helicopter, based upon its respective gross weight and inertia, are presented in Fig. 6-20 for both the pitch and roll axes. Combinations of control sensitivity and damping must fall within the bounded areas in order to comply with the specification. The results of pilot evaluations of different combinations of characteristics for the two helicopters also are presented, and the format is similar to that used for the requirements. Those combinations of sensitivity and damping falling within the area defined by the indicated boundaries were approved by the evaluating test pilots. In the case of the S-51 (Ref. 14), the flight test boundaries plotted correspond to the differentiation by the pilots between "acceptable" and "minimum acceptable". For the S-56 (Ref. 15) a Cooper rating of 3.5, which is the boundary between "satisfactory" and "unsatisfactory" flying qualities, is plotted. It may be seen that, whereas the flight test boundaries do show a difference due to size (primarily in the damping level), the difference is not as great as is the difference in the requirements.

Another flight test study of a variable-stability helicopter (Ref. 16) concludes that the amount of speed stability has a significant influence on Cooper ratings near hover. This conclusion is verified by examination of the equations of motion in hover flight, which show that positive speed stability contributes to dynamic instability. This effect may explain much of the variation in pilot opinion regarding optimum combinations of control sensitivity and damping on helicopters of different sizes and configurations.

In roll and yaw, maximum boundaries of control sensitivity are established by MIL-H-8501 as shown in Figs. 6-18 and 6-19, but no similar limit is defined for pitch. The S-51 flight test results shown on Fig. 6-20, indicate however, that such a requirement should be established to prevent the pitch response from being too sensitive. A suggested requirement for design purposes is: "the pitch displacement at the end of one second following an abrupt one-inch displacement of the stick while hovering *shall* not exceed three times the minimum requirement for instrument flight".

The damping requirements of Table 6-3 are based upon the same flight tests of the variable-stability helicopter that were used to define the control response requirements. Helicopter control response can be evaluated in flight test by measuring the displacement at a specific time after a unit control input. Demonstration of compliance with the damping requirements in pitch and roll, on the other hand, cannot be accomplished easily in flight test because it requires obtaining a steady rate of pitch or roll in hover. This is difficult or impossible to achieve without either placing the helicopter into a dangerous attitude or going into translational flight. The damping in yaw, on the other hand, is found readily by measuring the yaw acceleration $\ddot{\psi}$ at the time of the step input, and the final rate of yaw $\dot{\psi}$. The damping c_z is

$$c_z = \frac{\ddot{\psi} I_z}{\dot{\psi}_{final}} \text{, lb-ft/(rad/sec)} \qquad (6\text{-}79)$$

An anomaly exists in MIL-H-8501 with respect to the yaw damping. The paragraph governing visual flight (3.3.19) states, "the yaw angular velocity damping should preferably be at least $27 I_z^{0.7}$ ft-lb per rad per sec". The paragraph governing instrument flight (3.6.1.1) states, ". . . . the following angular velocity damping requirements *shall* apply in hovering . . . directional, $27 I_z^{0.7}$ ft-lb per rad per sec". Most helicopters, both single-rotor and tandem-rotor configurations, cannot meet this requirement without artificial damping. The visual flight requirement apparently was written with this fact in mind and was stated as a goal rather than as a requirement. This distinction was lost when writing the instrument flight paragraph. In effect, it requires that all helicopters intended for instrument flight incorporate artificial yaw damping.

6-3.1.1.3 Control Power Requirements

MIL-H-8501 indirectly specifies total control power in hover by requiring minimum angular displacements at the end of a specified time following a full control input. These minimum displacements correspond to four times the minimum displacement for pitch and to three times those for roll and yaw as listed in Table 6-2 for a one-inch control step. The control power requirements are listed in Table 6-4.

For all steady-flight conditions, MIL-H-8501 further specifies that the pitch and roll control power remaining above that required for trim must be such that at least 10% of the maximum moment attainable in hover may be generated.

Minimum acceptable directional control power is specified by MIL-H-8501 by requiring that, while hovering at the maximum alternate design gross weight over a spot in a 35-kt wind, a full control input in the most critical direction will result in a yaw displacement from any heading of at least $110/(W + 1000)^{1/3}$ deg at the end of the first second.

6-3.1.1.4 Other Control Power Considerations

Control power not only is a measurement of the ability to produce angular accelerations or to balance external moments but, in conjunction with the damping, it also governs the achievable maximum angular velocities. The maximum velocities required are dependent upon the mission of the helicopter. An attack helicopter, for example, needs high rates of pitch, roll, and yaw for rapidly bringing its weapons to bear upon a target, for nap-of-the-earth flying, and for evasive maneuvers. Transport helicopters require much lower angular velocities. For single-engine helicopters, the nose-down rotation following a cyclic flare from autorotation may establish a more realistic goal for the rate of pitch than any other maneuver. In this case, it is desirable to level the helicopter during the time that stored rotor energy is available to cushion the landing. The time available from the stored rotor energy is typically about 1 sec so that a nose-down pitch rate of at least 45 deg/sec is required.

Extra control motion must be designed into those helicopters in which stability augmentation systems (SAS) can introduce control inputs without moving the pilot's controls. Such devices usually do not differentiate between the response to an external disturbance and a pilot-initiated maneuver, and tend to "wash out" the pilot's control input so that he has to move the control further than he would without the SAS. For this reason the authority of the SAS usually is limited to 15-20% of the full control input. This limited authority also is insurance that a "hardover" failure of the SAS can be overridden by the pilot.

The high control power achieved with large flapping hinge offset, spring-restrained flapping, or cantilevered blades carries with it both advantages and disadvantages. The high control power and corresponding high damping can be used to improve the helicopter flying qualities, especially when used in conjunction with relatively simple stability augmentation systems. These types of rotors can be used on compound helicopters for pitch and roll control even when the rotor is unloaded completely. On the other hand, high control power can make it difficult to match control sensitivity with reasonable stick displacement for trim at high speeds (par. 6-3.1.2). The relatively small allowable tilt of the tip-path plane with respect to the shaft, when

TABLE 6-4
MINIMUM DISPLACEMENT IN DEGREES FOLLOWING A FULL CONTROL INPUT IN HOVER, MIL-H-8501

RESPONSE	VISUAL FLIGHT, deg	INSTRUMENT FLIGHT, deg
PITCH θ IN 1 sec	$\frac{180}{\sqrt[3]{W+1000}}$	$\frac{292}{\sqrt[3]{W+1000}}$
ROLL φ IN 0.5 sec	$\frac{81}{\sqrt[3]{W+1000}}$	$\frac{96}{\sqrt[3]{W+1000}}$
YAW ψ IN 1 sec	$\frac{330}{\sqrt[3]{W+1000}}$	$\frac{330}{\sqrt[3]{W+1000}}$

compared to a teetering or an articulated rotor with little or no hinge offset, restricts the use of the rotor for producing a horizontal force for ground taxiing or running takeoffs. It also compromises the ability to make a landing on a slippery slope while keeping the rotor thrust vector vertical. Landings on nonslippery slopes can be made by using the uphill landing gear to produce a side load to balance the tilt of the rotor thrust vector during the final letdown. Another consideration of high control power is the hazard of tipping the helicopter over on the ground at low collective pitch. A suggested requirement to prevent this possibility is: "With the helicopter on the ground with minimum collective pitch, it *shall* not be possible to tip the helicopter over, to cause a blade strike on the fuselage, or to exceed the design loads in the rotor system with less than two inches of control motion or 50 lb of stick force, whichever comes first, at any rotor speed". For some helicopters this stipulation will dictate the incorporation of a device to increase the stick force gradient when collective pitch is low and the helicopter is on the ground.

No requirement for control power along the vertical axis is specified in MIL-H-8501; for a helicopter, this requirement is incorporated in the hover ceiling requirement. At the hover ceiling, the vertical control power is zero. This normally is due to a lack of shaft power available rather than to a lack of rotor thrust capability. For any less stringent condition, the vertical control power will have some positive value. For example, if a turbine-powered helicopter can hover OGE at 6000 ft at 95°F, the shaft power available at sea level on a standard day will be about 40% greater than required to hover. Therefore, the ratio of thrust capability to gross weight will be at least 1.25 for hovering OGE and appreciably higher IGE at sea level and standard temperature.

6-3.1.2 Determination of Control Characteristics

6-3.1.2.1 Pitch and Roll

Control sensitivity is governed by the moment about the CG that can be generated per inch of control displacement. Because the rotors of single-rotor and tandem-rotor helicopters are used differently to produce these control moments, the two types of helicopter will be discussed separately.

6-3.1.2.1.1 Single-rotor Helicopters

For single-rotor helicopters in hover, the pitch and roll moments are caused by blade flapping with respect to the shaft as shown in Fig. 6-21. The control sensitivity in pitch $(M/I_y)/\delta_{sx}$ is

$$\frac{M/I_y}{\delta_{sx}} = \frac{\left(Th + \frac{\partial M_H}{\partial a_1}\right)\left(-\frac{\partial B_1}{\partial \delta_{sx}}\right)}{I_y}, \text{(rad/sec}^2\text{)/in.} \quad (6\text{-}80)$$

where

a_1 = cosine component of blade flapping with respect to a plane normal to the shaft, rad

B_1 = cosine component of swashplate tilt with respect to a plane normal to the shaft, rad

h = height of rotor hub above CG, ft

M_H = pitch moment of rotor on hub, lb-ft

T = rotor thrust, lb

The rotor stiffness $\partial M_H/\partial a_1$ is discussed in par. 6-2.5.1 and may be shown by the relationships presented there to be

$$\frac{\partial M_H}{\partial a_1} = \frac{3ea\rho A_b(\Omega R)^2}{4\gamma}, \text{ lb-ft/rad} \quad (6\text{-}81)$$

where

a = blade section lift curve slope, rad^{-1}

A_b = total blade area, ft^2

e = flapping hinge offset, ft

γ = Lock number, dimensionless

The total moment M_{max} that can be produced with full stick displacement from trim in hover is

$$M_{max} = \left(Th + \frac{\partial M_H}{\partial a_1}\right)\left(-\frac{\partial B_1}{\partial \delta_{sx}}\right) \times \left(\delta_{sx_{limit}} - \delta_{sx_{trim}}\right), \text{ lb-ft} \quad (6\text{-}82)$$

In forward flight, the control moment in pitch also is a function of the CG position $\bar{x}$

$$M_{max} = \left(Th + \frac{\partial M_H}{\partial a_1} - \frac{\partial T}{\partial a_1}\bar{x}\right)\left(-\frac{\partial B_1}{\partial \delta_{sx}}\right) \times \left(\delta_{sx_{limit}} - \delta_{sx_{trim}}\right), \text{ lb-ft} \quad (6\text{-}83)$$

The stick position required for trim corresponds with the cyclic pitch required to compensate for the forward speed and to produce the flapping with respect to the shaft required to balance any CG offset or aerodynamic pitching moment from the fuselage and/or horizontal stabilizer. Writing this balance of moments in terms of the required swashplate tilt, the trim stick position is seen to be

$$\delta_{sx_{trim}} = \left[B_{1_{no\ flapping}} - \frac{\bar{x}}{h\left(1 + \frac{\partial M_H/\partial a_1}{Th}\right)} + \frac{M_A}{\partial M_H/\partial a_1}\right]\frac{\partial \delta_{sx}}{\partial B_1}, \text{ in.} \quad (6\text{-}84)$$

where

M_A = pitching moment about CG produced by aerodynamic forces on the fuselage, wing, and empennage, lb-ft

For helicopters with high rotor stiffness, such as those with a large flapping hinge offset or with cantilevered blades, a special problem is illustrated by rewriting Eq.

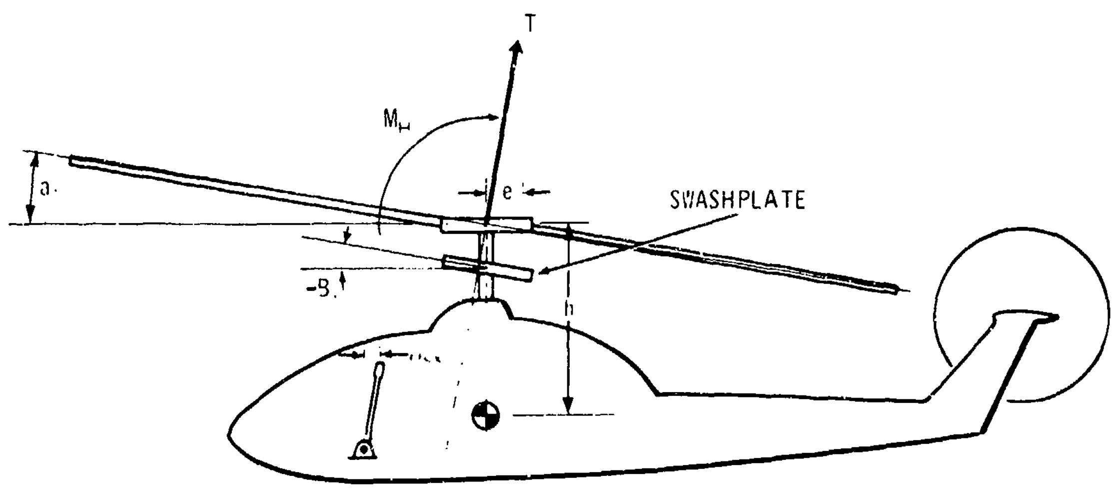


Fig. 6-21. Pitching Moments Produced in Hover by Stick Displacement on a Single-rotor Helicopter

6-84 with the stick displacement expressed as a function of the control sensitivity $(M/I_y)/\delta_{sx}$ as defined by Eq. 6-80

$$\delta_{sx_{trim}} = -\left[B_{1_{no\ flapping}} - \frac{\bar{x}}{h\left(1 + \frac{\partial M_H/\partial a_1}{Th}\right)} + \frac{M_\alpha}{\partial M_H/\partial a_1}\right]\left[\frac{Th + (\partial M_H/\partial a_1)}{(M/I_y)I_y}\right], \text{in.} \quad (6\text{-}85)$$

For high rotor stiffness, several terms are negligible and Eq. 6-85 reduces to

$$\delta_{sx_{trim}} = -\left(B_{1_{no\ flapping}}\right)\frac{\partial M_H/\partial a_1}{(M/I_y)I_y} \quad (6\text{-}86)$$

At high forward speeds, where $B_{1_{no\ flapping}}$ is high, a stiff rotor will result in a large stick displacement for a given control sensitivity and inertia. The maximum longitudinal stick displacement recommended by MS 33575 is ± 7 in. Thus, the rotor stiffness must be limited to that value which is compatible with a maximum value of control sensitivity for helicopters employing a conventional cyclic pitch control system. If a rate control system is used, the problem does not exist because the stick is centered for all steady flight conditions.

The control sensitivity in roll $(L/I_x)/\delta_{sy}$ that can be obtained from a given configuration of rotor and control system is

$$\frac{L/I_x}{\delta_{sy}} = \frac{\left(Th + \frac{\partial L_H}{\partial b_1}\right)\left(\frac{\partial A_1}{\partial \delta_{sy}}\right)}{I_x}, \text{(rad/sec}^2\text{)/in.} \quad (6\text{-}87)$$

where

A_1 = sine component of swashplate tilt with respect to a plane normal to the shaft, rad

b_1 = sine component of blade flapping with respect to a plane normal to the shaft, rad

Because the moments about the hub L_H and M_H are dependent directly upon the tilt of the thrust vector and, hence, upon cyclic flapping b_1 and a_1, respectively,

$$\frac{\partial L_H}{\partial b_1} = \frac{\partial M_H}{\partial a_1} \quad (6\text{-}88)$$

The maximum roll control moment L_{max} is

$$L_{max} = \left(Th + \frac{\partial L_H}{\partial b_1}\right)\left(\frac{\partial A_1}{\partial \delta_{sy}}\right) \times \left(\delta_{sy_{limit}} - \delta_{sy_{trim}}\right), \text{lb-ft} \quad (6\text{-}89)$$

For a helicopter with no stability augmentation, the damping in pitch and roll during hover can be considered to be due entirely to the rotor, unless the aircraft has an exceptionally large horizontal stabilizer or wing. The longitudinal flapping a_1 due to pitch velocity in hover can be expressed (Ref. 17) as

$$a_1 = \frac{-16\dot{\theta}}{\gamma\Omega}, \text{rad} \quad (6\text{-}90)$$

and thus the damping parameter is

$$\frac{c_y}{I_y} = \frac{Th + \partial M_H/\partial a_1}{I_y}\left(\frac{16}{\gamma\Omega}\right), \text{sec}^{-1} \quad (6\text{-}91)$$

Similarly,

$$\frac{c_x}{I_x} = \frac{Th + \partial L_H/\partial b_1}{I_x}\left(\frac{16}{\gamma\Omega}\right), \text{sec}^{-1} \quad (6\text{-}92)$$

The amount of cyclic pitch required to produce a given rate of roll can be determined from the equation for the maximum rate of roll $\dot{\phi}_{max}$

$$\dot{\phi}_{max} = \frac{L_{max}}{c_x}, \text{rad/sec} \quad (6\text{-}93)$$

The lateral flapping b_1 as a function of roll velocity in forward flight can be expressed (Ref. 17) as

$$b_1 = \frac{-16\dot{\phi}}{\gamma\Omega\left(1 + \frac{\mu^2}{2}\right)}, \text{rad} \quad (6\text{-}94)$$

Thus from Eqs. 6-89, 6-92, and 6-93

$$\dot{\phi}_{max} = \frac{A_1}{16}\left[\gamma\Omega\left(1 + \frac{\mu^2}{2}\right)\right], \text{rad/sec} \quad (6\text{-}95)$$

and the lateral cyclic pitch A_1 required to produce a given rate of roll $\dot{\phi}_{max}$ is

$$A_1 = \frac{16\dot{\phi}_{max}}{\gamma\Omega\left(1 + \frac{\mu^2}{2}\right)}, \text{rad} \quad (6\text{-}96)$$

6-3.1.2.1.2 Tandem-rotor Helicopters

For tandem-rotor helicopters, the equations for roll control sensitivity, control power, and damping are the same as for the single-rotor helicopter (considering both rotors). However, in pitch, control of tandem-rotor helicopters is obtained with a couple produced by differential rotor thrust as shown in Fig. 6-22.

The tandem-rotor pitch control sensitivity $[(M/I_y)/\delta_{sx}]_{tandem}$ is

$$\left(\frac{M/I_y}{\delta_{sx}}\right)_{tandem} = \frac{\left(\frac{\partial T}{\partial\theta_0}\right)\left(\frac{\partial\Delta\theta_0}{\partial\delta_{sx}}\right)x_R}{I_y}, (\text{rad/sec}^2)/\text{in.} \quad (6\text{-}97)$$

where

x_R = distance between rotors of tandem-rotor helicopter, ft
θ_0 = main rotor collective pitch, rad

and the maximum control moment $(M_{max})_{tandem}$ is

$$(M_{max})_{tandem} = \left(\frac{\partial T}{\partial\theta_0}\right)\left(\frac{\partial\Delta\theta_0}{\partial\delta_{sx}}\right) \times \left(\delta_{sx_{limit}} - \delta_{sx_{trim}}\right), \text{lb-ft} \quad (6\text{-}98)$$

where $\partial T/\partial\theta_0$ can be evaluated from the methods of par. 3-2.1. The pitch damping parameter is

$$\frac{c_y}{I_y} = \frac{2}{I_y}\left[\frac{\partial T}{\partial V_v}\left(\frac{x_R}{2}\right)^2 + \left(Th + \frac{\partial M_H}{\partial a_1}\right)\frac{16}{\gamma\Omega}\right], \text{sec}^{-1} \quad (6\text{-}99)$$

where

V_v = vertical velocity at rotor, fps

6-3.1.2.2 Yaw

6-3.1.2.2.1 Single-rotor Helicopters

For a single-rotor helicopter with a tail rotor, the control sensitivity in yaw $(N/I_z)/\delta_p$ for a given configuration is

$$\frac{N/I_z}{\delta_p} = \frac{\partial T_{TR}}{\partial\theta_{TR}}\left(\frac{\partial\theta_{TR}}{\partial\delta_p}\right)\frac{x_{TR}}{I_z}, (\text{rad/sec}^2)/\text{in.} \quad (6\text{-}100)$$

where

x_{TR} = horizontal distance between tail rotor and CG, ft
T_{TR} = tail rotor thrust, lb
θ_{TR} = tail rotor pitch, rad

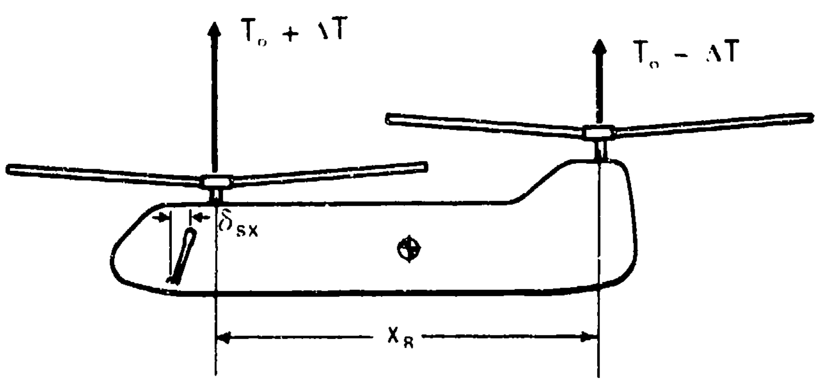


Fig. 6-22. Pitching Moments Produced in Hover by Stick Displacement on a Tandem-rotor Helicopter

and the maximum control moment N_{max} that can be provided is

$$N_{max} = \frac{\partial T_{TR}}{\partial \theta_{TR}} \left(\frac{\partial \theta_{TR}}{\partial \delta_p} \right) x_{TR} \times$$

$$\left(\delta_{p\,limit} - \delta_{p\,trim} \right) \text{, lb-ft} \quad (6\text{-}101)$$

The trim pedal position should be computed while hovering at the maximum alternate gross weight.

The yaw damping is due primarily to the tail rotor, but the contribution of the fuselage and the vertical stabilizer also should be calculated. The equation for the yaw moment N in hover as a function of yaw velocity $\dot{\psi}$ is

$$N = -\left\{ \frac{\partial T_{TR}}{\partial V_{TR}} x_{TR}^2 \dot{\psi} \right.$$

$$+ \frac{\rho}{2} \left[\int_{CG}^{nose} x d C_D (x\dot{\psi})^2 \, dx \right.$$

$$\left. \left. + \int_{CG}^{tail} x d C_D (x\dot{\psi})^2 \, dx \right] \right\} \text{, lb-ft} \quad (6\text{-}102)$$

where

C_D = drag coefficient of fuselage section, dimensionless

d = fuselage depth, ft

V_{TR} = lateral velocity at tail rotor, fps

x = horizontal distance from CG, ft

Note that the fuselage contribution is a function of $\dot{\psi}$. Therefore, Eq. 6-102 should be evaluated at the specified maximum yaw rate $\dot{\psi}_{max}$. The yaw damping c_z becomes, at $\dot{\psi}_{max}$.

$$c_z = \frac{\partial T_{TR}}{\partial V_{TR}} x_{TR}^2 + \frac{\rho}{2} \left(\dot{\psi}_{max} \right) \left[\int_{CG}^{nose} x^3 d C_D \, dx \right.$$

$$\left. + \int_{CG}^{tail} x^3 d C_D \, dx \right] \text{, lb-ft/(rad/sec)} \quad (6\text{-}103)$$

Calculations of this parameter for a number of single-rotor helicopters show that it generally is only of the order of 30-50% of the value of 27 $I_z^{0.7}$ specified as a design goal by MIL-H-5801 (par. 6-3.1.2.1).

6-3.1.2.2.2 Tandem-rotor Helicopters

Yaw control of tandem-rotor helicopters is achieved with differential lateral tip-path plane tilt to produce a couple about the CG as shown in Fig. 6-23. The tandem-rotor yaw control sensitivity is $[(N/I_z)/\delta_p]_{tandem}$ for a given configuration

$$\left(\frac{N/I_z}{\delta_p} \right)_{tandem} = \frac{T}{I_z} \left(\frac{\partial \Delta A_1}{\partial \delta_p} \right) x_R \text{, (rad/sec}^2\text{)/in.} \quad (6\text{-}104)$$

The maximum yaw control moment $(N_{max})_{tandem}$ available from a given configuration

$$(N_{max})_{tandem} = T \left(\frac{\partial \Delta A_1}{\partial \delta_p} \right) x_R \times$$

$$\left(\delta_{p\,limit} - \delta_{p\,trim} \right) \text{, lb-ft} \quad (6\text{-}105)$$

and the damping in yaw $(c_z)_{tandem}$ for the configuration at maximum yaw rate $\dot{\psi}_{max}$

$$(c_z)_{tandem} = T \left(\frac{\partial b_1}{\partial V_s} \right) \frac{x_R^2}{2} + \frac{\rho}{2} \left(\dot{\psi}_{max} \right)$$

$$\left[\int_{CG}^{nose} x^3 d C_D \, dx \right.$$

$$\left. + \int_{CG}^{tail} x^3 d C_D \, dx \right] \text{, lb-ft/(rad/sec)} \quad (6\text{-}106)$$

where

V_s = lateral velocity at the main rotor(s), fps

6-3.2 CONTROL POSITION VERSUS SPEED

6-3.2.1 Requirements

The change in longitudinal stick position required to trim a helicopter following a change in speed at a constant collective pitch and power setting is a measure of that aspect of static stability referred to as "speed stability". If the helicopter possesses speed stability, an increase in speed will produce a nose-up moment that will result in the helicopter pitching up and, therefore, slowing down. Positive speed stability requires that the pilot push the stick forward if he wishes to hold a new speed. If, on the other hand, the helicopter has negative speed stability, an increase in speed will produce a nose-down moment that will pitch the helicopter into a dive, with the speed increasing in a divergent manner. Therefore, negative speed stability requires that the pilot pull the stick aft if he wishes to hold a new speed. This type of instability can be very dangerous during flight conditions in which the pilot cannot devote full attention to the control of speed and attitude. For this reason MIL-H-8501 requires that a helicopter *shall* have positive static longitudinal control force and control position stability, with respect to speed, for all speeds and power conditions, with the exception that in the transition speed regime a moderate degree of instability is permitted. No minimum level of stability has been specified; it is sufficient that a measurable forward stick displacement and force be required to establish equilibrium at a speed above initial trim. Excessive speed stability results in large stick displacements when going from hover to maximum speed, and is objectionable to the pilot on two counts: (1) the possibility of running out of longitudinal control, and (2) the inconvenience of flying for long periods of time with the stick so far forward that the arm is straightened out uncomfortably and the elbow cannot be rested upon the knee.

A related requirement in MIL-H-8501 concerns the change in control position required to trim the helicopter following a change in power at the same speed, e.g., going from a full-power climb to autorotation. This requirement specifies that the change in longitudinal stick position during a change of power at any speed must be less than 3 in. and the change in lateral stick position must be less than 2 in. Control forces and control force gradients from trim are discussed in par. 6-3.6.

6-3.2.2 Physical Phenomena Involved

6-3.2.2.1 Single-rotor Helicopters

When the forward speed is increased from a trim value, two things happen to the rotor. First, it flaps back, tilting the rotor thrust vector and producing a hub moment. Second, depending upon the flight condition, the thrust either increases or decreases. If the

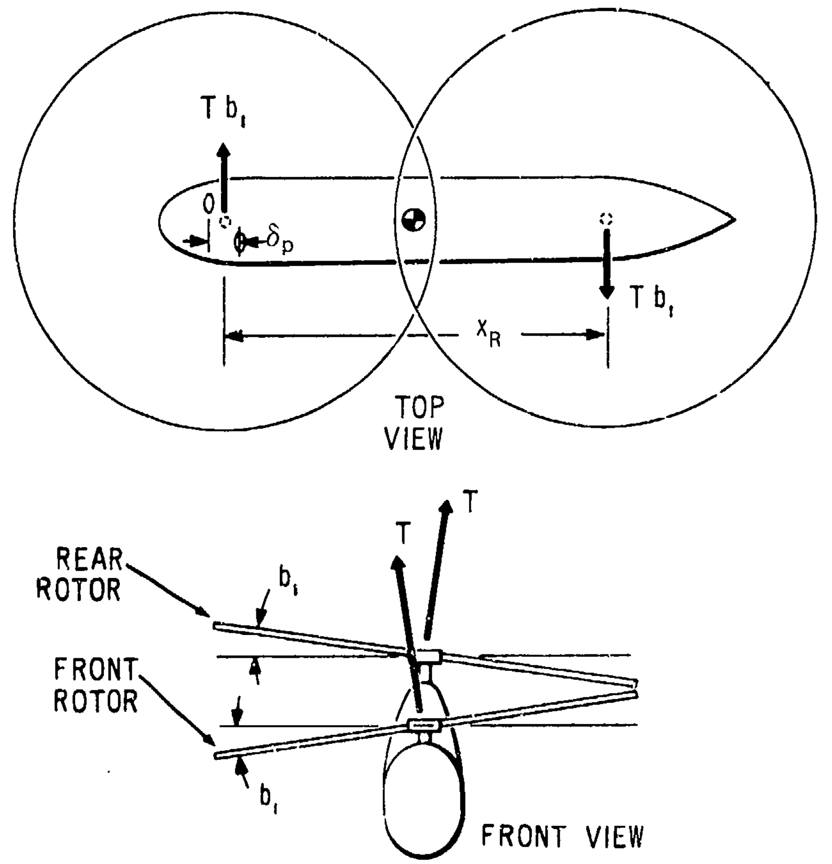


Fig. 6-23. Yawing Moments Produced in Hover by Pedal Displacements on a Tandem-rotor Helicopter

rotor is at a negative angle of attack, as is typical of helicopter flight, the rotor thrust will decrease. If the rotor is at a positive angle of attack, as in autorotation, the rotor thrust will increase. In order to achieve trim at a new flight speed at a constant collective pitch setting, the rotor thrust must be brought back to the original value by a change in the angle of attack (increased angle for powered flight and decreased for autorotation). This change in angle of attack will produce a further change in the rotor flapping, and usually a change in the aerodynamic pitching moments of the fuselage and horizontal stabilizer. The methods of determining these various effects upon the speed stability are discussed in par. 6-3.2.3.

Changes in power settings at a constant speed require corresponding changes in collective pitch, angle of attack, and tail rotor thrust. These changes in turn produce pitching and rolling moments. These moments must be compensated for by further changes in the stick position in order to retrim at the new power setting. Three separate effects contribute to the amount of stick displacement required: (1) the change in rotor flapping due to the change in collective pitch, (2) the change in rotor flapping due to the change in angle of attack, (3) the change in fuselage and stabilizer pitching moments due to the change in tail rotor thrust. The methods of determining the contribution of each of these effects are discussed in par. 6-3.2.3.

6-3.2.2.2 Tandem-rotor Helicopters

For a tandem-rotor helicopter, the change in the stick position required to trim either at a new speed or at a new power setting primarily is a function of the change in the differential thrust between the two rotors. For example, an increase in speed in normal flight produces a decrease in the thrust of each rotor. When the angle of attack is increased so as to obtain the original total thrust, the distribution of thrust between the two rotors may be different. This produces a pitching moment that will require a stick deflection to compensate. Whether the pitching moment is positive or negative depends upon the helicopter configuration, the CG, and the flight condition.

6-3.2.2.3 Synchropters

Synchropters, those helicopters having two intermeshing, laterally displaced rotors, have essentially the same speed stability characteristics as single-rotor helicopters, except in the case of changes in power setting at a constant speed. Because the shaft axes are tilted laterally, some component of rotor torque is applied to the helicopter as a pitching moment. This unwanted pitching moment must be trimmed out, either with blade flapping or with an aerodynamic moment from a horizontal stabilizer. For configurations in which the two retreating tips are outboard, increased rotor torque produces a nose-up pitching moment, requiring aft stick. When power is decreased, the stick must be moved forward.

6-3.2.3 Methods for Calculating Speed Stability

6-3.2.3.1 Single-rotor Helicopters

For a single-rotor helicopter with a conventional cyclic pitch control system, the stick position corresponds to the cyclic pitch required to trim the tip-path plane perpendicular to the shaft, plus the increment of cyclic pitch required to produce the necessary flapping with respect to the shaft. This latter increment must be sufficient to generate at the hub a moment that will balance CG offsets, aerodynamic pitching moments of the fuselage and horizontal stabilizer, and tail rotor thrust. For the helicopter shown in Fig. 6-24 the longitudinal stick position $\delta_{sx_{trim}}$ has been shown (Eq. 6-84) as

$$(\delta_{sx})_{trim} = \left[B_{1_{no\ flapping}} - \frac{\bar{x}T}{Th + \dfrac{\partial M_H}{\partial a_1}} + \frac{q(M_A/q)}{Th + \dfrac{\partial M_H}{\partial a_1}}\right]\frac{d\delta_{sx}}{dB_1}\text{, in.} \tag{6-107}$$

where

q = dynamic pressure $\rho V^2/2$, lb/ft^2

The change in stick position with a change in speed at constant thrust is

$$\frac{d\delta_{sx}}{dV} = \frac{1}{\Omega R}\left\{\left(\frac{\partial B_1}{\partial \mu}\right)_{C_T/\sigma=const} + \frac{q\left[\dfrac{1}{\mu}(M_A/q) + \dfrac{\partial(M_A/q)}{\partial\alpha}\dfrac{\partial\alpha}{\partial\mu}\right]}{Th + \dfrac{\partial M_H}{\partial a_1}}\right\} \times \left(\frac{d\delta_{sx}}{dB_1}\right)\text{, in./fps} \tag{6-108}$$

where, because of the equivalence of feathering and flapping

$$\left(\frac{\partial B_1}{\partial \mu}\right)_{C_T/\sigma = const} = \left(\frac{\partial a_1}{\partial \mu}\right)_{C_T/\sigma = const} \quad (6\text{-}109)$$

Following the techniques of Ref. 18, this derivative can be determined as a function of the collective pitch θ_0, the thrust solidity coefficient C_T/σ and the advance ratio μ. The techniques of Ref. 18 are valid for most helicopter flight conditions but should be used with caution for flight conditions involving significant blade stall or for values of advance ratio $\mu > 0.5$. The flapping derivative is

$$\left(\frac{\partial a_1}{\partial \mu}\right)_{C_T/\sigma = const} = C_1\theta_{0.75} + \frac{C_2 C_T}{\sigma} \quad (6\text{-}110)$$

where C_1 and C_2 are functions of advance ratio μ as shown on Fig. 6-25. The partial derivative of angle of attack with respect to advance ratio $\partial\alpha/\partial\mu$ can be found by equating the total derivative of the thrust solidity coefficient C_T/σ with respect to advance ratio μ to zero

$$\frac{dC_T/\sigma}{d\mu} = \frac{\partial C_T/\sigma}{\partial \mu} + \left(\frac{\partial C_T/\sigma}{\partial \alpha}\right)\frac{\partial \alpha}{\partial \mu} = 0 \quad (6\text{-}111)$$

or

$$\frac{\partial \alpha}{\partial \mu} = -\frac{\dfrac{\partial C_T/\sigma}{\partial \mu}}{\dfrac{\partial C_T/\sigma}{\partial \alpha}} \quad (6\text{-}112)$$

Using the techniques of Ref. 18, $\partial\alpha/\partial\mu$ may be written

$$\frac{\partial \alpha}{\partial \mu} = \frac{E_1\theta_{0.75}}{\mu^2} + \frac{E_2 C_T/\sigma}{\mu^3} + \frac{E_3 C_T \theta_{0.75}^2}{\mu^4} + \frac{E_4 C_T^2 \theta_{0.75}}{\sigma\mu^4} + \frac{E_5 C_T^3}{\sigma^2\mu^5} \quad (6\text{-}113)$$

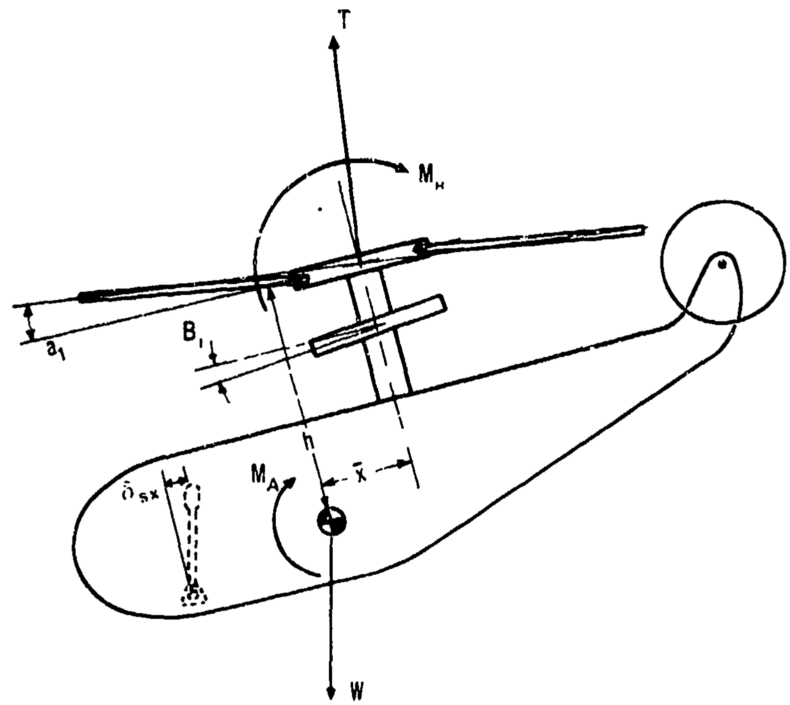


Fig. 6-24. Forces and Moments on Single-rotor Helicopter in Forward Flight

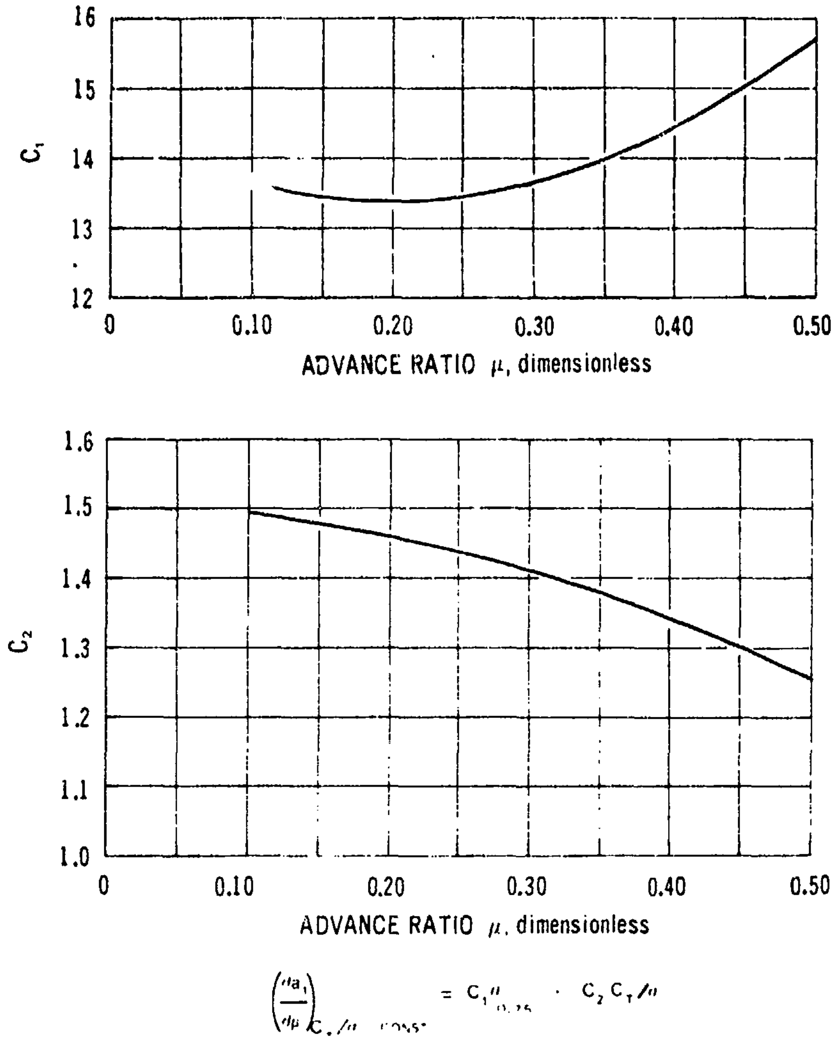


Fig. 6-25. Coefficients of Flapping With Respect to Speed Derivative

where the coefficients E_1 to E_5 are plotted on Fig. 6-26 as a function of μ and σ.

For positive speed stability, the stick must be moved forward for increasing speeds. Thus, the total derivative of Eq. 6-108 must be negative, or the terms within the brackets must be positive, because the gearing term $d\delta_{s_x}/dB_1$ is negative. At speeds above transition, the rotor contribution always is positive. The contribution of the airframe—including fuselage, empennage, and wing—can be either positive or negative depending upon the relative values of the initial moment term M_A/q and the airframe stability term $\partial(M_A/q)/\partial\alpha$. For positive speed stability, the initial moment term should be positive or nose-up. Angle-of-attack stability, i.e., a negative value for $\partial(M/q)/\partial\alpha$, actually is destabilizing with respect to speed because the increased aircraft angle of attack required to compensate for the loss in rotor thrust will produce a nose-down moment due to the increased lift of the horizontal stabilizer. Experience has shown, however, that for reasonable stabilizer sizes, this effect by itself is never large enough to cause trouble.

The change in longitudinal stick position due to a change in power setting at a constant speed and rotor thrust involves a change in collective pitch and a change in angle of attack. From Eq. 6-107 the change in stick position with respect to power control setting δ, is

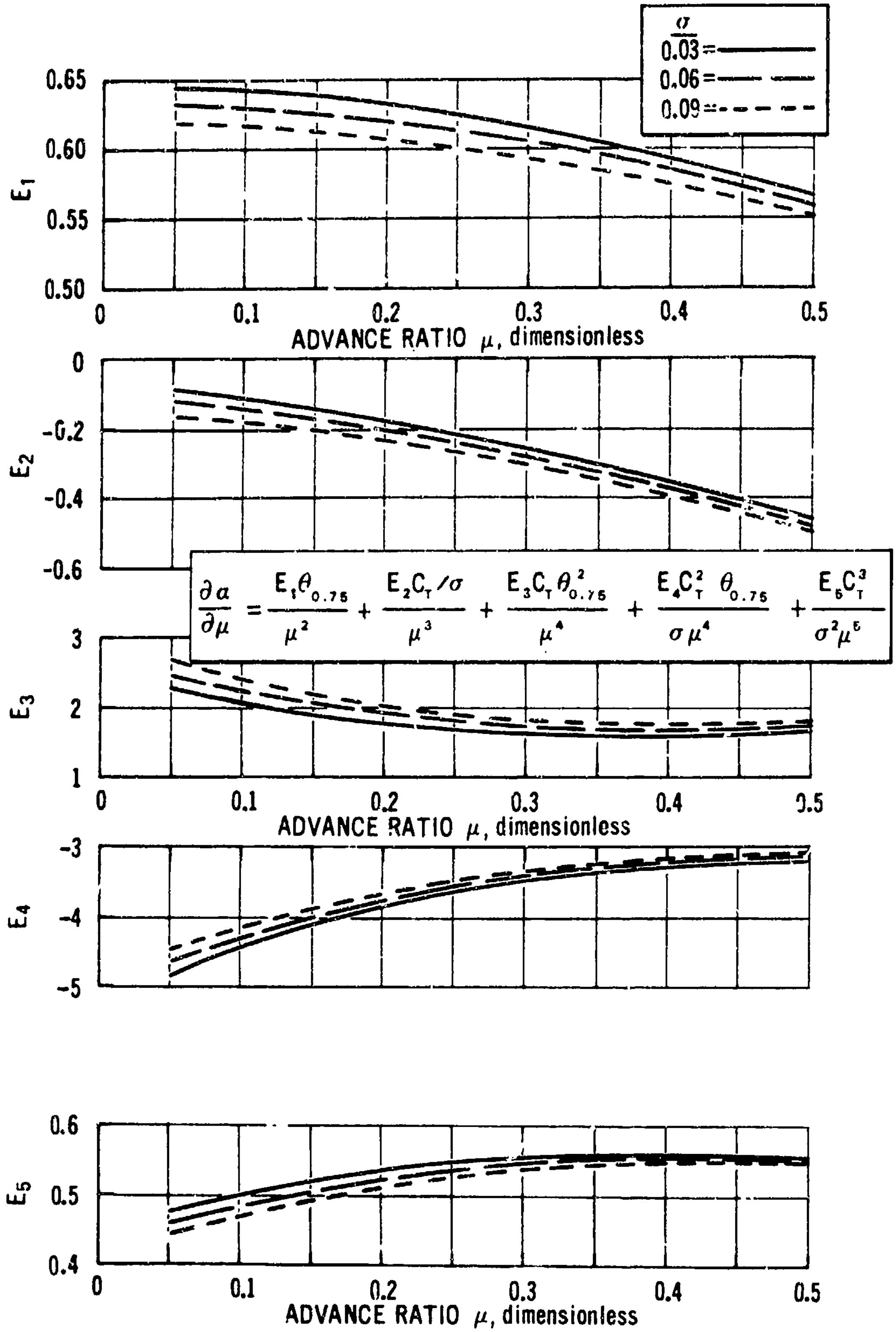


Fig. 6-26. Coefficients of Angle of Attack With Respect to Speed Derivative

$$\frac{d\delta_{sx}}{d\delta_t} = \left\{ \frac{\partial B_1}{\partial \theta_0} \Delta\theta_0 + \left[\frac{\partial B_1}{\partial \alpha} + \frac{q\left(\frac{\partial(M_A/q)}{\partial \alpha}\right)}{Th + \frac{\partial M_H}{\partial a_1}} \right] \Delta\alpha \right\} \frac{d\delta_{sx}}{dB_1} \quad (6\text{-}114)$$

Because of the equivalence of blade feathering and flapping $\partial B_1/\partial\theta_0 = \partial a_1/\partial\theta_0$, and $\partial B_1/\partial\alpha = \partial a_1/\partial\alpha$. The two flapping derivatives are shown in Fig. 6-27 as functions of advance ratio μ, with the effect of variations in the value of solidity σ also being shown.

The values of $\Delta\theta_0$ and $\Delta\alpha$ required in Eq. 6-114 must be calculated from trim considerations for the helicopter and flight conditions in question. Changes in lateral stick position are due primarily to changes in tail rotor thrust because, for the same main rotor speed and thrust, there is no significant change in lateral cyclic pitch required to trim the rotor. The change in lateral stick position due to a change in power control setting is

$$\frac{d\delta_{sy}}{d\delta_t} = -\left(\frac{\Delta T_{TR} h_{TR}}{Th + \frac{\partial L_H}{\partial b_1}} \right) \frac{d\delta_{sy}}{dA_1} \quad (6\text{-}115)$$

where

h_{TR} = height of tail rotor above the CG, ft

where

$$\frac{\partial L_H}{\partial b_1} = -\frac{\partial M_H}{\partial a_1} \quad (6\text{-}116)$$

6-3.2.3.2 Tandem-rotor Helicopters

The speed stability of a tandem-rotor helicopter can be determined from the equation for the summation of moments about the CG. For the helicopter shown in Fig. 6-28:

$$\sum M = T_1 x_1 - T_2 x_2 + a_{1_1}\left(T_1 h_1 + \frac{\partial M_{H_1}}{\partial a_{1_1}}\right) + a_{1_2}\left(T_2 h_2 + \frac{\partial M_{H_2}}{\partial a_{1_2}}\right) + q(M_A/q) = 0 \quad (6\text{-}117)$$

The speed stability is proportional to the total derivative of the moment with respect to speed V

$$\begin{aligned}\frac{d\sum M}{dV} &= [x_1 + a_{1_1} h_1]\left[\frac{\partial T_1}{\partial V} + \frac{\partial T_1}{\partial \alpha_1}\frac{\partial \alpha_1}{\partial V}\right] \\ &- [x_2 - a_{1_2} h_2]\left[\frac{\partial T_2}{\partial V} + \frac{\partial T_2}{\partial \alpha_2}\frac{\partial \alpha_2}{\partial V}\right] \\ &+ \left[T_1 h_1 + \frac{\partial M_{H_1}}{\partial a_{1_1}}\right]\left[\frac{\partial a_{1_1}}{\partial V} + \frac{\partial a_{1_1}}{\partial \alpha_1}\frac{\partial \alpha_1}{\partial V}\right] \\ &+ \left[T_2 h_2 + \frac{\partial M_{H_2}}{\partial a_{1_2}}\right]\left[\frac{\partial a_{1_2}}{\partial V} + \frac{\partial a_{1_2}}{\partial \alpha_2}\frac{\partial \alpha_2}{\partial V}\right] \\ &+ \rho V(M_A/q) + q\frac{\partial(M_A/q)}{\partial \alpha_A}\left(\frac{\partial \alpha_A}{\partial V}\right) \quad (6\text{-}118)\end{aligned}$$

where

α_A = airframe angle of attack, rad

The sign of this derivative (Eq. 6-118) determines whether the helicopter has positive or negative speed stability. Because nearly equal terms with opposite signs occur in the equation, the sign of the total can be either positive or negative depending upon the relative magnitude of the individual terms. The partial derivatives can be evaluated by the methods of Ref. 18 once the trim conditions are established from the simultaneous solution of the lift, drag, and pitching moment equations of equilibrium. Because the rear rotor and the airframe operate in the wake of the front rotor, the partial derivatives $\partial\alpha_2/\partial V$ and $\partial\alpha_A/\partial V$ must be evaluated by considering rotor downwash characteristics such as are presented in Ref. 19.

For powered flight conditions, the rotor terms of Eq. 6-118 can be identified as being either stabilizing or destabilizing as shown in Table 6-5. The fuselage terms are stabilizing for an initial nose-up pitching moment term M_A/q and destabilizing for a positive $\partial(M_A/q)/\partial\alpha_A$, i.e., an unstable fuselage. A tandem-rotor helicopter with inherent negative speed stability can be stabilized by using an auxiliary device that senses changes in speed through a "q" bellows or by means of a spring-loaded drag vane that adds a nose-up control signal into the longitudinal control system in parallel to the pilot's control.

6-3.3 SPEED STABILITY AND MANEUVERING FLIGHT

The speed stability of a helicopter normally is determined in the manner just discussed by examining the sign of the pitching moment derivative with respect to forward speed, $\partial M/\partial V$. If this derivative is positive, the helicopter is said to have a positive static speed stability. An alternative method of determining speed stability is to examine the changes in longitudinal cyclic control with speed.

To produce a static speed stability ($\partial B_1/\partial V > 0$) for fixed values of collective pitch and power control

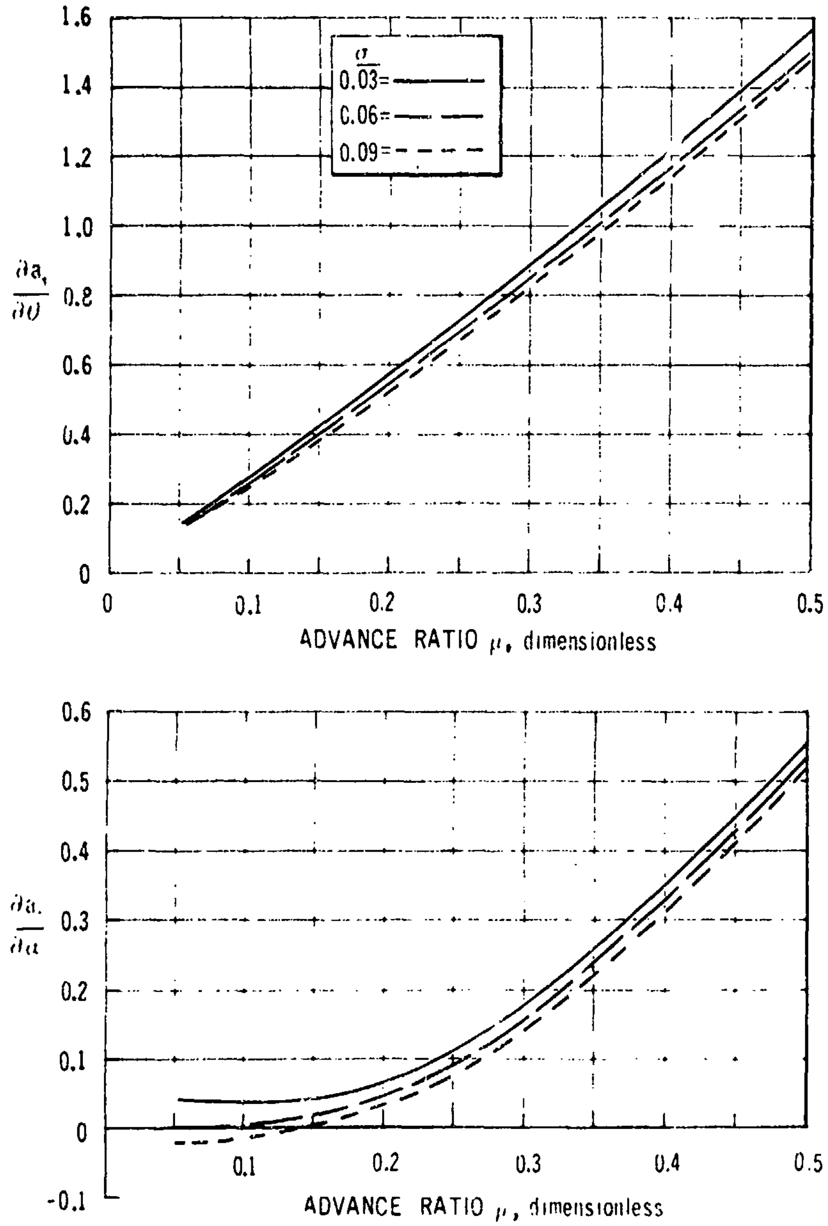


Fig. 6-27. Flapping Derivatives

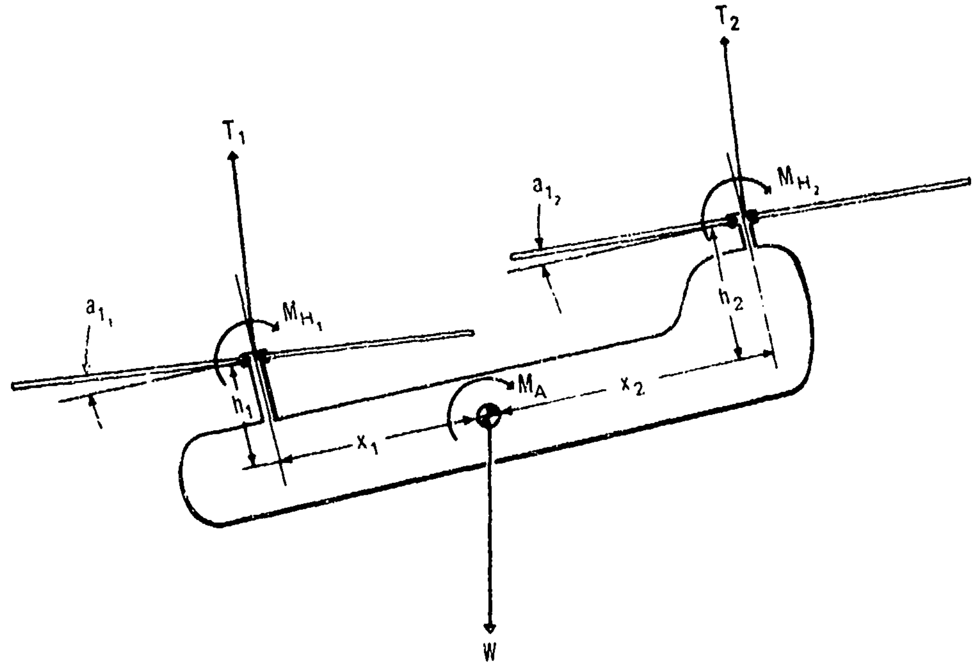


Fig. 6-28. Forces and Moments Acting on Tandem-rotor Helicopters

setting, the stick must be moved forward progressively to trim as speed increases.

Using either the pitching moment or the cyclic control derivative as the criterion, the rotor usually is stable with speed. However, the combination of a torsionally soft blade and a highly cambered blade section, or the use of a symmetrical blade whose aerodynamic center is well behind the twist axis, can result in a forward tilt of the rotor-tip path plane with an increase in speed. This results in a negative speed stability for the rotor and is unacceptable.

MIL-H-8501 requires positive speed stability, in the form of positive control position and control force stability with speed, throughout the major part of the flight regime. This stability *shall* be apparent in that a forward stick displacement and a push force *shall* be required to hold an increased value of steady forward speed, and a rearward displacement and pull force *shall* be required to hold a decreased value of speed. In the speed ranges from 15 to 50 kt forward and 10 to 30 kt rearward, mild instability is acceptable, although not desirable. The change in the unstable direction *shall* not exceed 0.5 in. for stick position.

It is clear that a high value of speed stability, in the form $\partial B_1/\partial V$, also will serve to limit helicopter forward speed, because there are real constraints upon the amount of cyclic pitch input that can be provided by the control system. Further, high values of this derivative have an adverse effect upon maneuvering capability because of the large stick displacements required. On the other hand, a very low value for speed stability is undesirable because of its importance to long-period damping.

Because the speed stability derivative $\partial B_1/\partial V$ depends primarily upon the rotor, which is symmetrical, results comparable to those found at low forward speed also are found for sideward and rearward flight. Therefore, the ability to achieve required values of sideward and rearward flight speed may be determined by the speed stability characteristics of the rotor.

6-3.4 DYNAMIC STABILITY

The dynamic stability requirements for helicopters as prescribed by MIL-H-8501 are tabulated in par. 6-2.5. Those oscillatory modes having sufficiently low frequencies—or sufficiently long periods of oscillation—to be controlled by the pilot need not be stable, i.e., positively damped.

MIL-H-8501 does not specify hovering dynamic stability requirements in the same manner as for forward flight. However, it is required that the helicopter *shall* be reasonably steady while hovering in still air (winds up to 3 kt). It *shall* be possible to keep the helicopter over a given spot on the ground, for all heights above the ground up to the disappearance of ground effect, with displacements of the cyclic controls of less than ± 1 in.

The parameters pertinent to dynamic stability characteristics are discussed in pars. 6-2.4 and 6-4.2. Because the typical rotor is unstable dynamically, both of these paragraphs discuss possible methods for achieving acceptable stability.

6-3.5 EFFECTS OF EXTERNAL STORES

It is extremely difficult at the preliminary design stage to predict with accuracy the effects of external stores upon the stability and control of a rotary-wing aircraft. Consideration must be given to weapons and other external loads (external fuel tanks) that are attached rigidly to the helicopter, as well as to sling loads. Estimates of the aerodynamic forces and moments resulting from armament system stores are subject to large errors due to aerodynamic interference effects. Experience with state-of-the-art helicopters operating at level flight speeds of up to 160 kt has shown that the effects of most store configurations are within the control capabilities of vehicles designed to the conventional criteria required by MIL-H-8501. A number of considerations and concepts, however, should be borne in mind during preliminary design, to assure that the adverse effects of external stores are minimized.

6-3.5.1 Angle of Attack

The aerodynamic environment of stores on helicopters differs from that of stores on fixed-wing aircraft principally in the much larger variations in airflow angle of attack that the helicopter must tolerate. Angle-of-attack excursions of high-performance, fixed-wing aircraft are relatively small (5-10 deg) throughout the flight envelope, even though flight modes may encompass large attitude changes. Rotary-wing aircraft, on the other hand, are characterized in flight by relatively small excursions of attitude and by large changes (40-50 deg) in angle of attack. Fig. 6-29 depicts a typical angle-of-attack envelope for a helicopter.

TABLE 6-5
STABILITY CHARACTERISTICS OF ROTOR TERMS IN TANDEM-ROTOR STABILITY

STABILIZING TERMS	DESTABILIZING TERMS
$x_2 \dfrac{\partial T_2}{\partial V}$	$x_1 \dfrac{\partial T_1}{\partial V}$
$x_1 \dfrac{\partial T_1}{\partial \alpha_1} \dfrac{\partial \alpha_1}{\partial V}$	$x_2 \dfrac{\partial T_2}{\partial \alpha_2} \dfrac{\partial \alpha_2}{\partial V}$
$a_{1_1} h_1 \dfrac{\partial T_1}{\partial \alpha_1} \dfrac{\partial \alpha_1}{\partial V}$	
$a_{1_2} h_2 \dfrac{\partial T_2}{\partial \alpha_2} \dfrac{\partial \alpha_2}{\partial V}$	
$\left[T_1 h_1 + \dfrac{\partial M_{H_1}}{\partial a_{1_1}}\right]\left[\dfrac{\partial a_{1_1}}{\partial V} + \dfrac{\partial a_{1_1}}{\partial \alpha_1} \dfrac{\partial \alpha_1}{\partial V}\right]$	
$\left[T_2 h_2 + \dfrac{\partial M_{H_2}}{\partial a_{1_2}}\right]\left[\dfrac{\partial a_{1_2}}{\partial V} + \dfrac{\partial a_{1_2}}{\partial \alpha_2} \dfrac{\partial \alpha_2}{\partial V}\right]$	

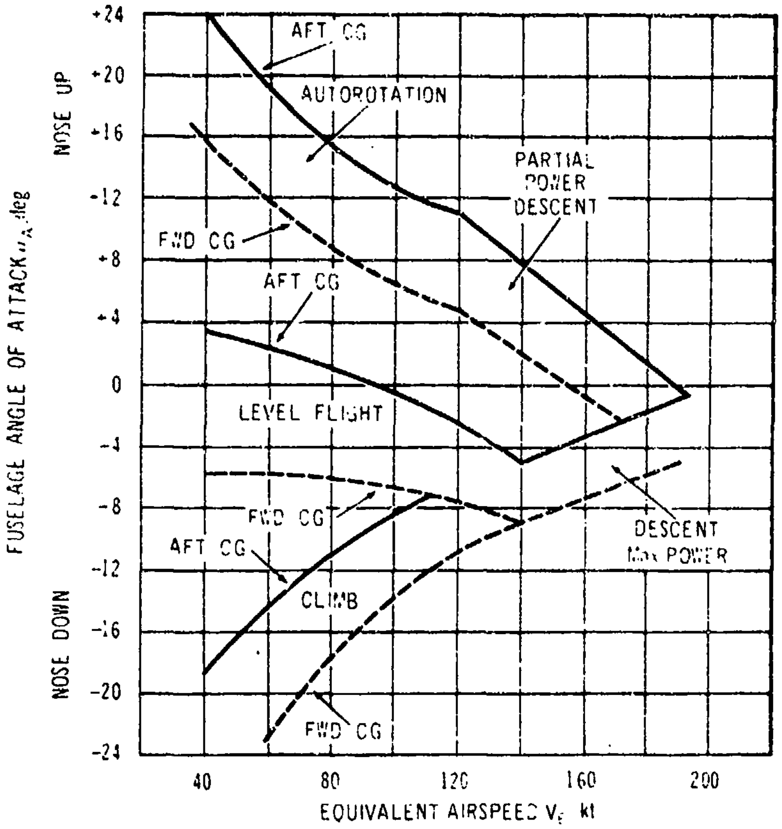


Fig. 6-29. Angle-of-attack Envelope for Armed Helicopter (Clean Configuration)

The helicopter designer must bear in mind that the vehicle configuration that results after installation of stores is subject to a wide range of airflow directions. To minimize the effects of the stores upon the airframe, it generally is desirable that they produce constant aerodynamic forces and moments throughout a wide range of angles of attack. This requirement implies, aerodynamically, that the stores should resemble a sphere. Experience with current helicopters and stores has shown that such secondary angle-of-attack considerations as the effects of the wakes of the store on elevator effectiveness seldom create problems (Refs. 20 and 21). For example, in the tests of Ref. 21 the weapon pod was so "dirty" aerodynamically that its installation resulted in a 14% reduction in helicopter range. But there were no discontinuities in static-stability gradients due to wake effects. From the test results of Refs. 20 and 21 it is concluded that wake effects have not been of major concern with the airspeed envelopes of current helicopters up to about 150 kt. A broader generalization is not justified without data from more configurations.

6-3.5.2 Attitude

Attitude, angle of attack, and boresight capability must be considered when positioning fixed forward-firing armament such as rocket pods. Fig. 6-30 is a typical map of attitudes in level and diving flight. For a typical case of a design rocket-firing flight condition of 120 kt, level flight, and forward CG, the fuselage angle of attack (Fig. 6-29) is −8 deg and its attitude (Fig. 6-30) is also −8 deg. To launch rockets along the flight path, it is necessary only that they be positioned at an angle of +8 deg with respect to the airframe reference axes (Fig. 6-31).

Most rocket attacks, however, are conducted during high-speed descending flight. A typical design flight condition would be a speed of 160 kt. At this speed, in partial-power descending flight with a forward CG, the angle of attack is −0.7 deg (Fig. 6-29) and the fuselage attitude is −16 deg (Fig. 6-30). Despite the steep nose-down attitude, the fuselage angle of attack is more positive than it is in level flight. Thus, to launch rockets along the flight path, the pods should be positioned at a positive angle of 0.7 deg to the fuselage reference system (Fig. 6-32), a change of 7.3 deg from the mounting angle for level-flight firing.

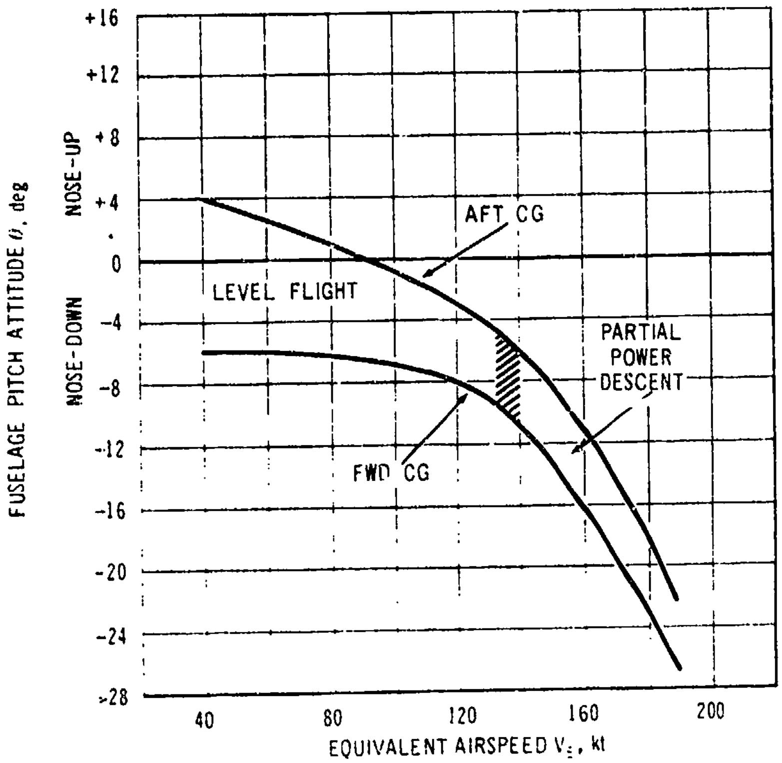


Fig. 6-30. Pitch Attitude Envelope for Armed Helicopters (Clean Configuration)

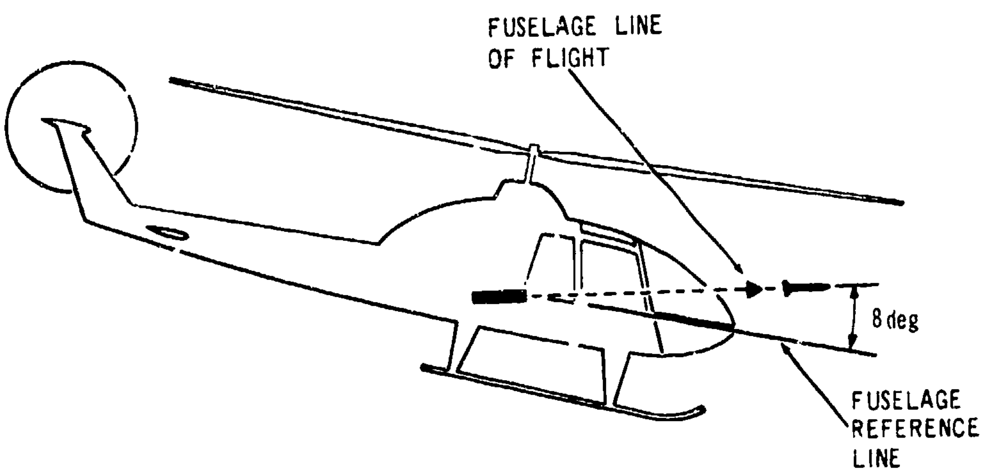


Fig. 6-31. Level Flight (120 kt)

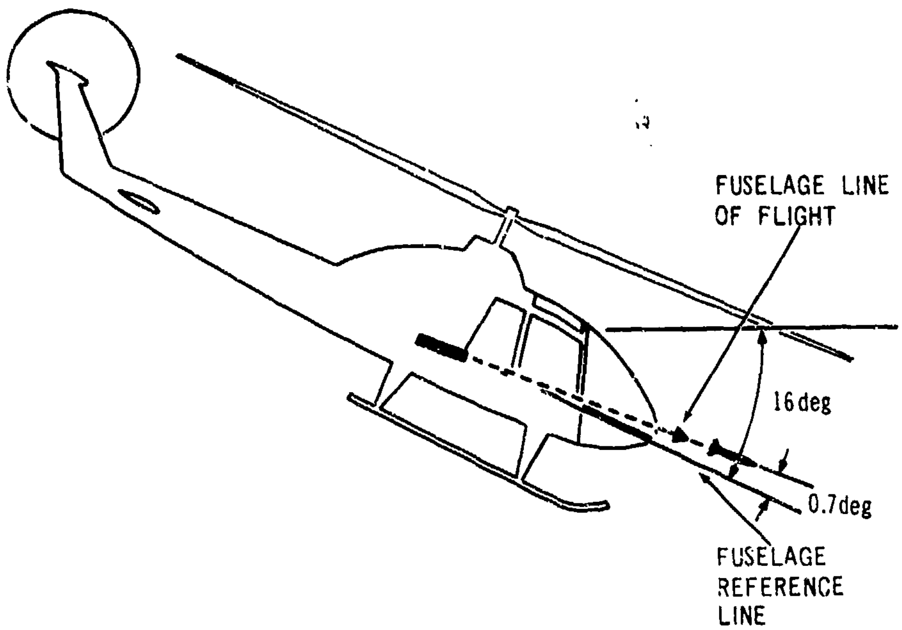


Fig. 6-32. Descending Flight (160 kt)

6-3.5.3 Aerodynamic Forces and Jettisoning of Stores or Dispensing of Payloads

The aerodynamic loading conditions affecting external stores are of concern only under specialized circumstances:

1. The jettisoning of low-density stores or stores whose shapes can generate a high ratio of lift to weight

2. The dispensing of corrosive or toxic liquid agents or lightweight ordnance that might be carried into contact with the airframe by the relative wind.

Examples of the first case are the XM157 and XM159 Rocket Pods. When empty, they resemble a set of crude annular wings. Therefore, they may generate sufficient lift to reduce markedly their separation rate after jettison. Furthermore, they seldom are stable in pitch and yaw; usually they begin tumbling after release, and they may strike the airframe several times before positive separation is attained. Therefore, if there is a requirement to jettison stores at moderate to high airspeeds, it probably will be necessary to use forced ejection. Presently available analytical determinations of the forces acting upon these types of store are inexact. The effect of Reynolds number, interference effects, rotor-downwash influence, the necessity to consider sideslip conditions, and the lack of prior knowledge of all store configurations that possibly will be used makes it difficult to predict the forces resulting from airloads even with the use of wind tunnel data.

A requirement to dispense liquid or lightweight material introduces additional design considerations. The path of the dispensed material will be nearly coincident with the relative wind about the fuselage, and because it may be necessary to keep the material away from the airframe and tail rotor because of toxicity or corrosiveness, the dispensing store must be positioned so as to preclude or minimize the possibility of such contact. While avoidance of contact throughout the complete flight envelope may not be possible, the designer must make every effort to provide adequate clearance throughout the jettison envelope. When such an envelope has been established, its limits must be provided to the pilot to assure that the helicopter is operating within the envelope at the time of jettison.

6-3.5.4 Forced Jettisons

Based upon reliability data for specific armament subsystems, consideration must be given to a rocket hang fire. The pilot must be given the data necessary to permit him to maintain control, or to retain control, following such a mishap.

Control can be upset by the effects of a forced jettison of an external store. In addition to the initial disturbance created by the recoil forces, the changes in helicopter weight and CG may require control responses about more than one axis. For some flight conditions, the changes in aerodynamic forces and moments brought about by the release of the store also may be significant. It is reasonable to assume that if any of these effects is objectionable by itself, the combined effects will be even more objectionable. For example, any recoil force resulting in an angular acceleration greater than some threshold tolerance level will be unacceptable regardless of the change in weight, balance, and/or aerodynamic force or moment that accompanies the ejection. Likewise, the jettisoning of stores must not result in a combination of weight and CG that falls outside the allowable flight envelope. Consequently, each of these effects should be evaluated separately to determine the allowable limits under conditions that otherwise are optimal.

In evaluating the combined effects of the disturbances associated with a forced jettison of stores, the most adverse conditions to be considered are those that require all of the corrective control displacement to be in the same direction. For example, a store mounted on the left wing tip may exhibit an aerodynamic download and the jettison impulse will cause the aircraft to experience a step roll-rate change (jettison impulse effect) accompanied by a roll acceleration (balance and aerodynamic effect) in the same direction. At the same time, there may be a considerable yaw moment from the change in drag. The combined effect might prove to be totally unacceptable, even though a gravity drop of the same store might be routine. For roll and yaw, this condition can occur only for asymmetric jettisons. For pitch, however, it also can occur during a symmetrical jettison if there is a longitudinal shift in the CG or a change in the aerodynamic pitching moment.

Because so many factors, including flight conditions, affect the ultimate helicopter reaction, it is difficult to predict how acceptable a particular jettison will be to the pilot when his aircraft is in an unusual attitude.

6-3.5.5 Effects of External Stores on Static and Dynamic Stability (Laterally Symmetrical Loadings)

The effects of external stores upon longitudinal and lateral-directional stability and control characteristics usually are small. The changes that have been measured can be attributed to changes in the moment of inertia of the helicopter about a particular axis, to increased drag and lift, or to the accuracy with which these parameters and characteristics can be measured in engineering flight tests. The same is true of control-response characteristics. The primary concern is control positions in stabilized flight. If sufficient aerodynamic drag is added below the helicopter CG, unstable control position gradients will result. This condition, which is true particularly of sling-load operation, may result in the definition of a limiting airspeed for a given loading condition. Generally, it is desirable to locate external stores near the CG of the helicopter and the aerodynamic center of the fuselage.

6-3.5.6 Effects of External Stores (Asymmetrical Loadings)

Special consideration must be given to asymmetric loadings of stores. Store loadings may be asymmetric by intent—a single rocket pod mounted on the right wing—or by mischance, as when simultaneous jettisoning of symmetrically installed pods is attempted and one fails to separate. All of the appropriate handling-qualities criteria of MIL-H-8501, e.g., for failure of stability augmentation systems, entry into autorotation, sideward and rearward flight, and control margins in stabilized flight, must be met for all critical asymmetrical loadings. Typical control position requirements are illustrated in Fig. 6-33.

Under normal, symmetrical store-loading conditions, control margins are adequate for normal maneuvers and emergencies. With asymmetrical loading, however, the control margins may be sufficient for hovering flight, but rapidly may approach the displacement limits in forward flight. Therefore, they may be inadequate for maneuvers and emergencies. The designer must have information about control-trim position comparable to that shown in Fig. 6-33 in order to establish flight envelopes for asymmetrical loadings based upon acceptable control margins. Furthermore, it is clear that the designer cannot merely analyze the hovering case and then assume that all forward flight conditions can be accomplished safely.

6-3.5.7 Effects of Weapon Recoil

Although not all aircraft weapons are external stores, many are detachable and can be exchanged for other weapons; in this sense they have the characteristics of external stores.

Weapon recoil can affect helicopter flight path and airspeed, and, therefore, firing accuracy as well. Depending upon the type of weapon, these adverse effects manifest themselves in one or more of the following ways:

1. They produce forces that must be counteracted.

2. They produce an angular acceleration about one or more axes.

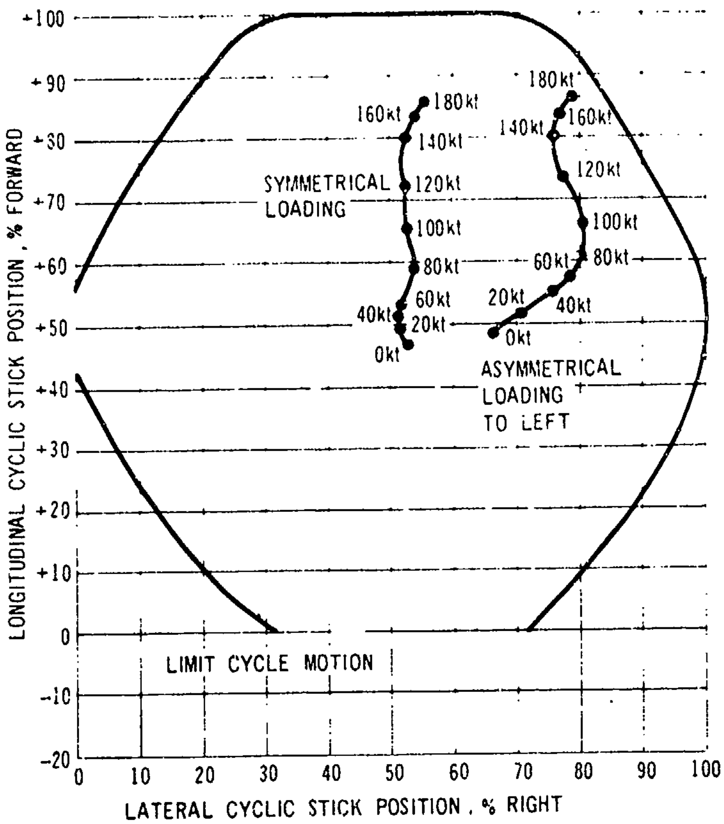


Fig. 6-33. Typical Cyclic Stick Plot

3. They can be a source of dynamic instability.

4. They can move the helicopter away from some reference path, as in a low-speed firing situation.

5. They can cause excessive vibration.

In addition to the direct effects of weapon recoil, in the form of the forces and moments applied to the airframe, consideration also must be given to secondary effects. For example, the exhaust from rockets being launched may impinge upon, or pass very close to, a horizontal stabilizer, and the resulting change in local angle of attack may result in a change of pitching moment sufficient to affect the helicopter flight path.

6-3.5.7.1 High-rate-of-fire Weapons

The most important weapon recoil parameters are the impulse per round and the rate of fire. For high-rate-of-fire weapons—such as machine guns that are capable of continuous firing—the average force can be expressed as the product of the impulse per round and the rate of fire. The moment produced by firing is dependent upon the weapon location, the direction of fire, and the recoil parameters.

The designer must establish that control of the aircraft can be maintained when the weapon is fired continuously. Control power about all axes must be adequate during every flight and firing condition in which the weapon might be used. For helicopters, the conditions that must be examined most closely are longitudinal-control margins in high-speed flight and directional-control margins under high-power conditions. Even when the margin is adequate (as it usually is in roll), recoil forces are objectionable if excessive control motions are necessary to compensate and to maintain attitude.

6-3.5.7.2 Medium- and Low-rate-of-fire Weapons

Helicopter dynamic stability can be influenced by any weapon that has a firing rate near a natural frequency of the flight mode. Generally, this consideration is important only at high speeds, where the natural frequencies are high and the dynamic stability is characterized by low damping.

6-3.5.7.3 High-impulse, Single-shot Weapons

Any weapon characterized by a large impulse-per-round is likely to be a single-shot type. The angular rate change about any axis is a function of the impulse, the weapon location, and the direction of aim. The peak angular rate change that can be tolerated may vary with speed or with tactical requirements, but usually becomes unacceptable when it exceeds 15 deg/sec. In most instances, the moments created by firing are more critical than are the forces. By restricting the location and the direction of fire of the weapon, larger recoils can be tolerated.

6-3.5.8 Sling Loads

The increasing use of helicopters for external (sling) load operations requires that consideration be given to the effects of such operations. However, there are so many variables that a generalized analysis is difficult to achieve. For example, for a fixed sling length, the pendular natural frequency is constant, yet the effect of an oscillating load will be dependent upon the mass of the load relative to the mass of the helicopter. In forward flight, the equilibrium position of a sling load is dependent upon both the mass of the load and its aerodynamic characteristics; therefore, both size and shape of the load are significant.

Coupling between airframe dynamics and sling load oscillations has been encountered in a number of cases. In addition to pendular oscillation, vertical oscillation has proven significant. The helicopter and the suspended load constitute a two-mass dynamic system, with the suspension system acting as a spring between the masses. The spring rate of a given cable is proportional directly to its length. Under certain circumstances a resonance can occur between the sling system and harmonic forces from the rotor. The helicopter may act as a rigid body or it may respond in its lowest frequency bending mode.

System parameters that result in natural frequencies approaching resonance with possible excitations must be avoided because the resultant response readily can become so severe that separation of the load is necessary for the safety of the helicopter. Pilot-induced oscillations must be considered as potential excitations.

By using multiple suspension points, the dynamic characteristics of the sling load usually can be modified sufficiently to avoid significant coupling with the dynamics of the helicopter.

6-3.6 CONTROL SYSTEMS

6-3.6.1 Control Feel

6-3.6.1.1 Force Feel

Forces at the primary cockpit controls should be such as to provide the pilot with proper force cues in any operational flight condition, e.g., a pull force on the longitudinal control results in a deceleration or in raising the nose, right sideward force on the control stick generates a right roll, push force on the right pedal results in a right yaw. Because control forces are fundamental to the ability to handle a given situation, a pilot must not be required to interpret control forces in order to apply the correct control inputs. There should be no objectionable changes in control force gradients (force per unit control displacement) under normal operating conditions, including autorotation. However, this does not rule out the possibility of reversals in control position gradients, although it would be desirable to minimize such reversals.

The stated requirements apply to all control arrangements including reversible mechanical and power-boosted system and fully powered irreversible controls. For boosted or powered irreversible systems, or in any scheme that prevents the cockpit controls from reacting to control surface loads, alternate means must be provided for producing suitable force feel. Devices such as springs, bungees, bobweights, and servos usually are acceptable for generating pilot force cues. An example of such an application is depicted schematically in Fig. 6-34, showing a typical irreversible and fully powered control system. The stick-centering and force gradient portions of the longitudinal cockpit control are shown. Link A is connected to the power train, which produces longitudinal cyclic pitch inputs, and a minimal friction force is required to move link A. Note that the control input must react against a spring force proportional to displacement at B. With this arrangement, pull forces are required for aft stick motion and push forces are needed for forward stick motion. Should the pilot desire to maintain the stick at a position other than the neutral one shown here, he can do so by pressing a trim button on the control stick. Pressing the trim button activates link C so that there are no forces due to the spring at the new stick position. The spring characteristic of this system is shown in Fig. 6-35. It should be noted that the system here includes spring preload, which provides a minimum acceptable "breakout" force and insures return of the stick to the reference position when it is displaced.

"Breakout" force is defined as the minimum force necessary, when applied at the cockpit control, to initiate motion at the corresponding control surface. As an example, the minimum longitudinal control force would be the stick force required to start increasing the longitudinal cyclic pitch at the main rotor. The breakout force is the most critical combination of the effects of such factors as friction, spring preload, detent, and mass unbalance. Limits upon the permissible range of control breakout forces are given in Table 6-6. Breakout forces are discussed further in par. 6-3.6.1.2.

Use of a bobweight to provide stick-force-per-g is quite common in fixed-wing aircraft (Fig. 6-34) but seldom is found in helicopters. The purpose of the spring K in the system shown in Fig. 6-34 is to counteract the mass unbalance of the bobweight in level flight. In accelerated flight the bobweight requires pull forces to maintain upward vertical acceleration and push forces to accelerate vertically downward.

Several specifications for maneuvering force gradient are shown in Fig. 6-36. The data of Fig. 6-36 illustrate the wide variance in maneuver criteria that can be expected depending upon speed or mission. Data based upon fixed-wing studies (adjusted to a normal load factor of 3.0) have been included from Ref. 22 to indicate the trends that might be expected if fixed-wing experience were to be used as a guideline for helicopters. The shaded area is from a high-speed helicopter program that studied maneuver forces desired by pilots. More research will be needed to identify the desirable characteristics fully and finally.

Early in the design iteration, appropriate design specifications must be identified, either by the contractor or by the procuring activity, so that maneuver criteria can be established. Mission, speed, or special re-

TABLE 6-6 CONTROL SYSTEM BREAKOUT FORCES MEASURED IN FLIGHT WITH ADJUSTABLE FRICTION OFF

CONTROL	NORMAL,* lb Min./Max	EMERGENCY, lb
PITCH	0.5/1.5	8
ROLL	0.5/1.5	7
YAW	1/7	15
COLLECTIVE	1/3	7

*FROM MIL-H-8501

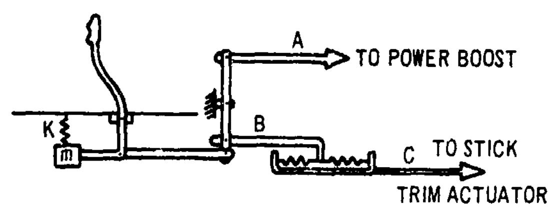


Fig. 6-34. Longitudinal Control Schematic

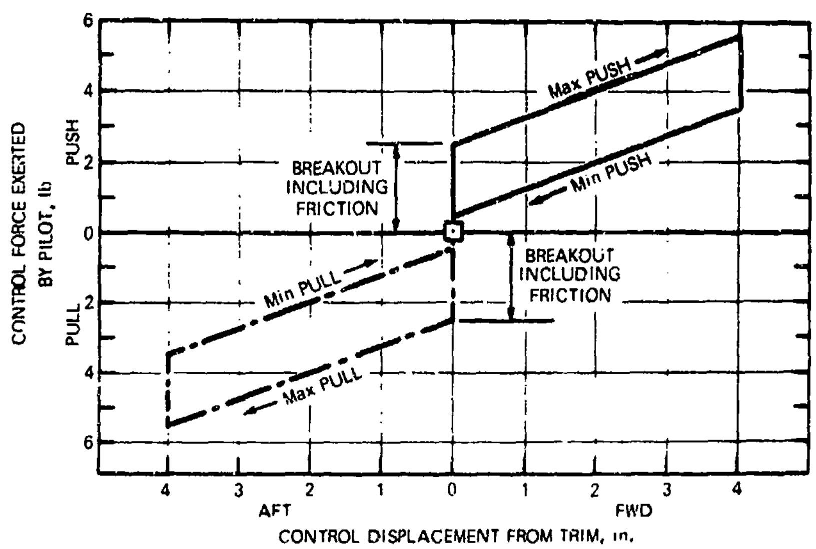


Fig. 6-35. Spring Feel System With Breakout

quirements may dictate the choice of criteria, as may the background experience and training of pilots.

Trim controls are auxiliary cockpit controls that are used by a pilot to improve his comfort and reduce his workload. They can be used to nullify control forces or to reorient a primary flight control. Examples are elevator tabs, stick-centering devices, and adjustable springs or bungees. By definition, trim controls are irreversible and must maintain their positions indefinitely unless altered or disengaged by the pilot. Trim systems are discussed further in par. 6-3.6.3.

6-3.6.1.2 Control Forces

Control forces *shall* be such that the combined forces from any source in the control system *shall* not produce undesirable handling characteristics. Longitudinal, lateral, and directional controls *shall* exhibit positive centering at any trim setting. Breakout forces and stick-centering characteristics *shall* not produce objectionable flight characteristics or permit large changes from trim conditions with controls free.

The emergency condition for which maximum acceptable breakout forces are given in Table 6-6 is a condition in which a control system failure would prevent completion of the intended mission of the helicopter, but from which a safe forced landing could be made.

The need for minimum values of control breakout forces is not necessarily obvious. It might seem that a frictionless control system would be desirable, but this is not true for several reasons. For example, if a control stick is to remain in position when released by the pilot, there must be some basic level of friction to prevent the mass unbalance of the stick and/or linkages from causing the control to fall to the side against the centering spring, thereby introducing a control input. With boosted or fully powered systems, there may be only a short mechanical linkage with little mechanical friction between a cockpit control and its power boost, and additional friction will have to be provided by other means. Also, in a servo system some friction generally is desired at the control stick to serve as a base reference for the actuator pilot value.

Similarly, excessive aircraft vibrations, or flight in gusty air, dictate a minimum level of breakout force. A frictionless control system is quite likely to result in oversensitive controls in rough air, and the mere presence of the pilot's hand on the control might introduce unwanted vibratory inputs. A minimum breakout force is required so that unwanted body motion cannot feed

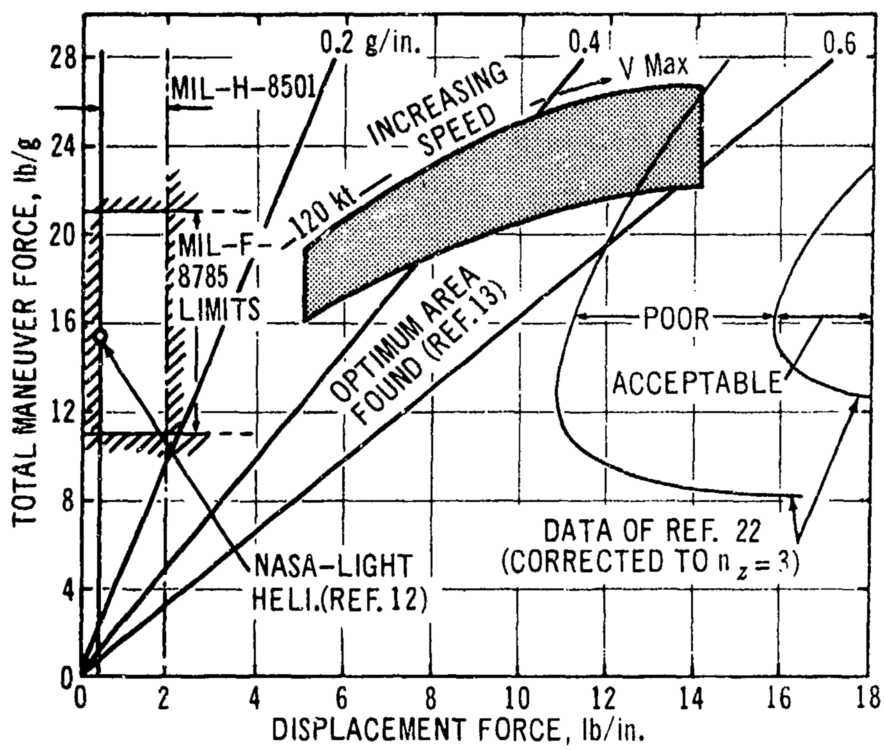


Fig. 6-36. Maneuver and Displacement Force Gradients

into the control system. In the same manner, maximum control breakout forces are specified to insure that excessive control forces do not compromise precision maneuvering or tracking. Variations in breakout forces with airspeed should be avoided, but are permissible up to the limits of Table 6-6. Collective breakout force may be measured with the adjustable friction set.

Limit control forces are listed in Table 6-7. These are the maximum permissible forces allowed in changing from any given trim and power condition to any other trim and power condition without retrimming, and they apply to the entire normal operational flight envelope. In deciding whether or not power boost is necessary, the designer must determine if the limit forces of Table 6-7 will be exceeded in normal operations. If not, the choice between a boosted and fully powered system involves trade-offs in such factors as weight, reliability, and redundancy.

When a boosted or fully powered system is used, the designer must consider the impact of its failure upon limit control forces. If a failure or malfunction results in forces greater than those listed in Table 6-7, a backup or redundant system must be provided. The overall control system must permit a safe landing within the levels of failure and reliability agreed to by the contractor and the procuring activity.

Control force sensitivity should be within the limits of Table 6-8 in the absence of other levels established by the procuring activity. These limits apply particularly to boosted or fully powered control systems, where the designer must insure acceptable, but not oversensitive, force feel for the pilot. Again, the most common means of constructing acceptable control force sensitivity is to use a spring-centering system (par. 6-3.6.1.1). Collective controls should have a minimal force sensitivity; i.e., the force felt on a collective control should vary as little as possible with control motion. Also, for any given control and trim position, design control force sensitivity for pitch, roll, and yaw controls *shall* not be less than the value equal to the breakout force per inch of control displacement (MIL-H-8501). The intent here is to prevent oversensitivity. In the event a pilot was faced with a high breakout force coupled with relatively low force sensitivity, acceleration of the control in the same direction as the control motion might occur, thus increasing pilot work load. Instead, after the breakout force has been overcome, the control system must generate forces that retard control motion.

TABLE 6-7
LIMIT CONTROL FORCES

CONTROL	NORMAL,(1) lb	EMERGENCY,(2) lb
PITCH	10	40
ROLL	7	20
YAW	30	80
COLLECTIVE	7	7

(1) BASED ON DATA FROM USAAMLTR 65-45
(2) BASED ON DATA FROM AGARD TR 408

TABLE 6-8
CONTROL FORCE SENSITIVITY

CONTROL	SENSITIVITY,* Min/Max, lb/in.
PITCH	0.5/2.5
ROLL	0.5/2.5
YAW	5/10
COLLECTIVE	MINIMAL

*BASED ON AGARD TR 408 AND MIL-H-8501

Pitch, roll, and yaw controls also *shall* show positive self-centering characteristics when displaced from any trim position. Absolute centering, or precise return to the trim position, is not necessary, although the control should return as nearly as possible to the trim position considering the breakout force. Any significant changes in attitude from trim should be easily perceptible to the pilot in terms of longitudinal stick force. For example, if a trimmed aircraft is disturbed by a gust that forces the nose down, the pilot immediately should feel a rearward force on the longitudinal stick.

The engagement of any trim or speed control device (flaps, dive brakes, etc.) should not introduce changes in the operating sense of primary flight controls, e.g., with flaps and landing gear extended and the aircraft in trim, pull forces should raise the nose and push forces should lower the nose.

For applications using a conventional control stick, breakout forces and stick force gradients must be well matched in order to provide meaningful thresholds for the pilot and to allow a harmonious relationship between force and motion (par. 6-3.6.2).

6-3.6.1.3 Control Linkage

Control free-play *shall* not be excessive, i.e., any lost motion in a control train must not result in objectionable handling characteristics or excessive pilot work load. Lost motion can occur due to slop, deadband, hysteresis, or control flexibility. As helicopters get larger, control free-play becomes more important. In the control system layout for the CH-53A depicted in Fig. 6-37, inputs from the rudder pedals must pass about 55 ft via cables or rods through 21 bell cranks or pulleys. With such a long transmission of force, there are many possibilities for building up friction or lost motion.

Wherever possible, controls should make use of rods rather than cables. Cable arrangements naturally are more susceptible to effects of flexibility, temperature changes, chafing, etc. The designer must consider carefully the weight saving of a cable configuration as opposed to the maintenance problems and the possible requirement for an expensive cable tension device.

From a practical standpoint, low cost and weight, combined with high reliability and safety, are desired in a control train, but not at the expense of handling qualities. Because of the subjective nature of the evaluations it is difficult to quantify acceptable figures on lost motion. As a guide, Table 6-9 indicates one manufacturer's set of design goals for control free-play. The guidelines are for normal operation and for the condition following simple failure in the automatic flight control system (AFCS).

6-3.6.2 Control Harmony

Control harmony is required to insure ease of control in coordinated maneuvers. The term means, basically, a harmonious relationship of pitch, roll, and yaw controls in terms of displacements, related breakout forces, force gradients, human capability, and helicopter response.

The final measure of good control harmony is the pilot's opinion of the ease of controllability. For example, a control system might be rigged with three times

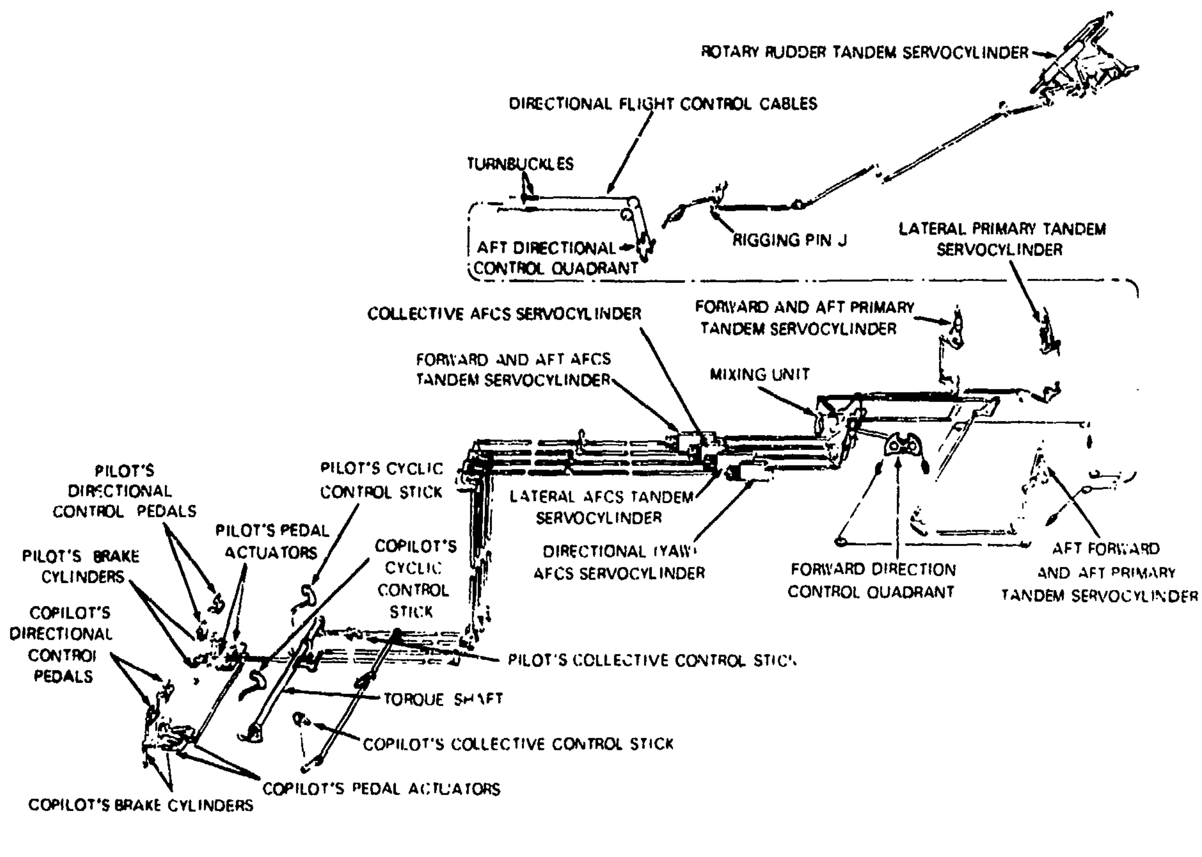


Fig. 6-37. Flight Control System Schematic

TABLE 6-9
SUGGESTED LIMITS ON FREE-PLAY OF CONTROL SYSTEMS

CHARACTERISTIC	LATERAL		LONGITUDINAL		COLLECTIVE		DIRECTIONAL	
	N.O.(2)	A.O.(3)	N.O.	A.O.	N.O.	A.O.	N.O.	A.O.
HYSTERESIS, in.	0.1	0.2	0.2	0.3	0.2	0.3	0.1	0.2
LINEARITY, %	+1.5	+3.0	+1.5	+3.0	+3.0	+5.0	+3.0	+5.0
OVERTRAVEL, %	+10	+5	+10	+5	+10	+5	+10	+5
DYNAMIC RESPONSE								
Max RATE, %/sec	100	50	100	50	100	50	100	50
PHASE–ALL MODES	N.O. 45° at 3 Hz (+10%) 65 at 3 Hz (+1.25%) 25° at 1 Hz (+10%) 45° at 1 Hz (+1.25%)				A.O. 50 at 3 Hz (+10%) 70 at 3 Hz (+1.25%) 30° at 1 Hz (+10%) 50 at 1 Hz (+1.25%)			

NOTES:
(1) FORCES AND MOTION TO BE MEASURED AT CENTER OF GRIP
(2) N.O. – NORMAL OPERATION
(3) A.O. – ALTERNATE OPERATION, i.e., AFCS SERVO OFF (1 FAILURE ONLY)

as much breakout force in the lateral control as in the longitudinal. If a pilot has to pay so much attention to the mismatch in breakout forces that precision maneuvering is compromised, the control system is said to have poor harmony. Likewise, the similarity of aircraft responses in pitch and roll, i.e., (rad/sec^2)/in., might be the measure of control harmony. As this measure also is quite subjective, there are very few data to use as a design guide. MIL-F-8785 specifies a ratio of 2:7:1 for forces in pitch, yaw, and roll (elevator, rudder, and ailerons, respectively) in maneuvering flight as a harmony criterion.

Good control harmony is important in maneuvers that require small control deflections—such as precision hover, aerial refueling, and target tracking. Design efforts should be directed toward providing a linear or smoothly varying response to control deflections and forces for all ranges of control inputs. The location and displacement of the controls from various seat locations should be such as to permit convenient and comfortable control by the pilot. High collective pitch lever and full forward stick are common control positions and should be arranged conveniently. Location of engine power control levers, visibility of instruments, and external visibility also must be considered. Helicopter cockpit dimensions are given in MS 33575. Cockpit arrangement is discussed further in par. 13-3.

The control system should be configured so as to minimize the effects of undesirable cross-coupling. Although the flying qualities MIL-H-8501 specification does not include quantitative limitations for cross-coupled responses, e.g., roll response to longitudinal displacement, or longitudinal and directional responses to collective displacement, noticeable amounts of such responses are objectionable and must be avoided. MIL-H-8501 does stipulate that, for helicopters employing power-boosted or power-operated controls, there *shall* be no lateral or directional control forces developed as a result of longitudinal control displacement, and there *shall* be no longitudinal forces developed in conjunction with lateral or directional control displacements. For unpowered controls, lateral forces of up to 20% and pedal forces of up to 75% of the associated longitudinal force are acceptable. Lateral control displacement *shall* not produce longitudinal forces in excess of 40% or pedal forces in excess of 100% of the associated lateral force. Pedal displacement *shall* not result in longitudinal forces in excess of 8% or lateral forces in excess of 6% of the associated pedal force.

With regard to coupling, the designer also must consider the effect of rotor precession angle. Because rotor blade response lags the control input by almost 90 deg of azimuth for articulated rotors, care must be taken to insure that cockpit control inputs generate corresponding rotor motions, e.g., that longitudinal stick motion yields longitudinal tilt of the rotor. If the rotor incorporates δ_3 hinge characteristics, the precession angle must be shifted correspondingly; in addition, the rotor speed control must accommodate the influence of gusts. It is recommended that mechanical linkages be used wherever possible until the reliability of alternate means is demonstrated. In the event of failure or emergency conditions, the natural cross-coupling of the helicopter at least should permit completion of a safe landing.

Special attention must be given to the coupling of airplane-type control surfaces in compound helicopters. In most cases, hover maneuver requirements have resulted in helicopter controls that are very sensitive at high forward speeds. The addition of fixed surfaces—such as vertical fins and horizontal tailplanes—tends to reduce the effects of such control sensitivity. But as the speed envelope is expanded, and wings and auxiliary propulsion are used to "unload" the rotor, it is advantageous to use airplane-type surfaces for control at higher speeds. The designer then must evaluate maneuver criteria and control sensitivity throughout the speed range to establish a control integration schedule. It is conceivable that this schedule would dictate eliminating cyclic pitch control to the rotor at very high speeds. Whatever the schedule, control integration must be smooth, with minimal changes in control sensitivity. The integration can be accomplished by any mechanical, hydraulic, or electrical means for normal operation, but the helicopter must be able to make a safe landing if a failure occurs.

6-3.6.3 Trim Systems

MIL-H-8501 requires that it *shall* be possible to trim a helicopter in pitch, roll, and sideslip in steady flight at all airspeeds, flight configurations, and power settings. The word "trim" includes the trimming of control forces to zero as well as the maintenance of attitude. All trim controls must be irreversible and must maintain their positions indefinitely unless altered by the pilot. Activation of a trim control must not introduce undesirable stick-force transients or control-force reversals. Trim controls should have design limits on rate of operation so as to permit the control to be actuated as rapidly as possible without causing oversensitivity of the primary controls in precision maneuvers. Rotational trim controls should be so rigged that their actuation will cause the aircraft to rotate in the same direction as does displacement of the primary control being trimmed.

All trim systems must include a suitable indicator to show the state of the control to the pilot. This consideration is important especially in boosted or fully powered systems where stick forces do not reflect the aerodynamic state of the aircraft. Often, stick forces are provided by a combination of springs, weights, and/or electromagnetic devices and bear little relationship to the particular flight condition or trim control setting. For example, consider the case of a trimmable horizontal tail on a helicopter. The primary function of this control is to alter the pitch attitude of the aircraft. However, use of tail position to raise the nose at high speed requires higher rotor flapping levels. If the pilot desired, he could move the longitudinal trim control, to maintain a given airspeed and rate of climb (R/C) while flying at a more pleasant attitude. However, it is conceivable that he could, by this action, demand too much increased flapping and thus unknowingly could exceed the the design flapping limit. The designer, therefore, must provide the pilot with information or warnings that will allow safe operation of the aircraft.

Trim controls range from cockpit wheels, cranks, or electrical switches on the primary controls to foot pedal adjusters. Whatever their physical arrangement, their operation must be simple, reliable, and obvious to the pilot. All trim controls should operate in the same sense throughout the flight regime. Trim control reversals during transition from powered to autorotative flight must be avoided wherever possible.

The trim control should have sufficient range and capability to nullify control forces when flight modes change from power-on to power-off.

Consideration also should be given to the functional reliability of the trim system. In the event that the trim system is less reliable than the primary system, the overall control system must have sufficient authority to override the trim system so as to permit accomplishment of the intended mission.

6-4 STABILITY AUGMENTATION

6-4.1 STABILITY REQUIREMENTS

Helicopter stability requirements are defined in MIL-H-8501. Specific requirements also are cited and discussed in prior paragraphs of this chapter. For example, the minimum acceptable damping characteristics of oscillations resulting from a single disturbance in smooth air are tabulated in par. 6-2.5.1, and the minimum acceptable levels of damping about each of the three axes are tabulated in par. 6-3.1.

It is noted in pars. 6-2.4 and 6-3.1, in particular, that the damping inherently available from a helicopter rotor generally is inadequate to permit compliance with the damping requirements given by MIL-H-8501. Therefore, compliance is possible only if some sort of stability augmentation system is employed.

Satisfactory stability characteristics alone are insufficient to provide acceptable handling qualities. It also is necessary that adequate control capability be available. MIL-H-8501 defines minimum acceptable controllability in the form of minimum values of angular displacement about each axis at the end of 1 sec following a 1-in. step displacement from hovering trim and following a full displacement from trim. The values for these requirements also are tabulated in par. 6-3.1. Separate requirements are stated for VFR and IFR helicopters.

The inadequacy of MIL-H-8501, particularly with regard to large helicopters (par. 6-3.1), has been acknowledged for a long time. However, insufficient data are available to permit amendment of its quantitative requirements with assurance that the results will be any more satisfactory. Although pilot opinion ratings may be valuable, they can provide no guidance to the designer unless an existing helicopter of comparable size (in both weight and inertia) has been evaluated in a variable-stability configuration adequate to establish meaningful rating information. Flight simulators have been employed to obtain comparable data but the correlation with flight test data has not yet been adequate to permit the use of such information to define standards for new helicopters.

Another deficiency of MIL-H-8501 is its failure to acknowledge a difference in flying quality requirements for helicopters performing different types of missions. The stability and controllability characteristics suitable for a cargo helicopter (CH), for example, are not acceptable for an attack helicopter (AH). The AH class must operate over a much larger flight envelope in both speed and load factor. And although the AH is more maneuverable, it also must be capable of being controlled very precisely when its weapon(s) have been trained upon a target and a firing run has started. Experience with [illegible] AH helicopters is being used as the basis f[illegible] the d[illegible]ment of handling quality requirements pertinent [illegible] class. Discussion of such requirements will be added to this handbook by amendment.

For Army helicopters of mission classes other than AH, the stability requirements—minimum acceptable damping of any oscillation resulting from a single disturbance in smooth air—given by MIL-H-8501 and tabulated in par. 6-2.5.1 still are considered valid. However, in lieu of the specific requirements for control sensitivity and damping given by the specification as functions of helicopter weight and inertia, it is suggested that a minimum control power margin be required. Thus, it should be possible to achieve a rate of angular displacement of 15 deg/sec about each of the three axes within at least 1.5 sec from any trimmed flight condition within the helicopter flight envelope. The required trim conditions include climbs and descents, both powered and autorotational. This requirement also replaces the MIL-H-8501 control margin requirement, which stipulates that 10% of the maximum movement available in hover be available at the limits of the flight envelope to permit control of disturbances. Specification of the control margin in terms of available rate response thus includes consideration of both damping and control characteristics in forward flight, while the MIL-H-8501 requirements are based only upon hovering values. The control response requirements of MIL-H-8501, a minimum time lag of 0.2 sec for development of an angular acceleration in the proper direction about each of the three axes, are applicable to Army helicopters of all mission classes. The requirement that the time histories of normal acceleration and angular velocity in pitch become concave downward within 2 sec following a sudden rearward displacement of the longitudinal control from trim also is applicable. Preferably, the time history of normal acceleration should be concave downward from the initiation of the maneuver, and the time history of pitch rate should be concave downward from 0.2 sec following the control displacement to the attainment of maximum rate of pitch.

Experience with existing helicopters indicates that satisfactory handling qualities—using either pilot opinion ratings or specific quantitative requirements as the basis for evaluation and judgment—can be achieved with mission classes other than observation helicopters (OH) only through the employment of artificial stability devices. Helicopters of the OH class that do not include stability augmentation equipment have been qualified as airworthy, indicating that their handling characteristics are acceptable. Although incorporation of stability augmentation equipment results in improved handling qualities even in this class, such a system has not been considered cost-effective for this class of helicopter. For new utility (UH) and CH class helicopters, artificial stabilization equipment *shall* be required. For new OH helicopters, the use of such equipment *shall* be based upon effectiveness trade-offs, unless otherwise specified by the procuring activity.

Par. 6-4.2.2 discusses the characteristics of stability augmentation systems necessary to provide given helicopter handling qualities. Subsequent paragraphs discuss the requirements pertinent to failure of stability augmentation equipment and the evolution from such devices to automatic flight control systems.

6-4.2 STABILITY AUGMENTATION SYSTEMS (SAS)

In this brief introduction to stability augmentation systems, a few selected problems are treated in simplified form. The mathematical description assumes only small perturbations from trim so that linear equations are applicable. Only longitudinal motions are considered, and they are assumed to be uncoupled from lateral-directional motions. Pilot acceptability is dependent primarily upon the characteristics of the short-period modes, although some attention also must be paid to avoidance of diverging long-period modes. The short-period pitching mode is treated by assuming constant flight speed. It is shown when and why augmentation systems are required for satisfying the minimum handling quality requirements of MIL-H-8501. Simplified equations are given for the design of augmentation systems to comply with short-period requirements. The effects of augmentation systems upon the long-period mode are discussed qualitatively.

A simplified analysis also is given for gust alleviation systems, for which only prototype experience currently is available. Helicopters with hinged blades, even with unaugmented controls, have a lower gust sensitivity than do fixed-wing aircraft. However, the potential of augmentation systems for providing further substantial decreases in gust responses of helicopters has not been realized fully as yet. Because the lowest blade natural frequency is an order of magnitude higher than the highest aircraft pitching frequency, a feedback system operating on the rotor blades has a time constant an order of magnitude shorter than a system operating on an airplane elevator or ailerons. Helicopter augmentation systems sensing rotor lift, or rotor pitching and rolling moments, and applying inputs to collective and cyclic controls, can be made to compensate for atmospheric gusts almost instantaneously. Gust alleviating augmentation systems are desirable for hinged-blade helicopters, and they are believed to be indispensable for hingeless rotors operating at high advance ratios because of the excessive gust sensitivity and pitch-up tendency of these rotors.

The effects of augmentation systems upon dynamic instabilities also are discussed. This is a complex field requiring more research, particularly with respect to hingeless rotors.

The basic helicopter equations and the simplifications used here are derived from Ref. 1. The notation also is taken from Ref. 1 wherever possible.

6-4.2.1 Pitch Control

The change in pitch attitude $\Delta\theta$ from trimmed level flight with constant flight speed is equal to the change in angle of attack from trim $\Delta\alpha$ and, using Laplace transform notation, the pitching equation of motion is

$$(s^2 - M_{\dot\theta}s - M_\alpha)\Delta\theta = M_{\delta_{sx}}\Delta\delta_{sx} \tag{6-119}$$

where

$M_{\dot\theta}$ = pitch damping derivative, $(\partial M/\partial\dot\theta)/I_y$, negative for damped motion, sec^{-1}

M_α = static stability derivative, $(\partial M/\partial\alpha)/I_y$, negative for stable behavior, sec^{-2}

$M_{\delta_{sx}}$ = control power, $(\partial M/\partial\delta_{sx})/I_y$, sec^{-2}/in.

The derivative $M_{\dot\alpha}$, representing the effect of rate of change of lift upon pitching moment, has been neglected in accordance with the assumed approximate independence of pitching and vertical motions. The characteristic equation of this second order dynamic system is of the form

$$s^2 + 2\zeta\omega s + \omega^2 = 0 \tag{6-120}$$

where

ζ = damping ratio, dimensionless

ω = frequency, rad/sec

By comparing Eqs. 6-119 and 6-120, it is seen that the natural frequency ω for zero damping and damping ratio ζ, are, respectively,

$$\omega = (-M_\alpha)^{1/2}, \qquad \zeta = -M_{\dot\theta}/(2\omega) \tag{6-121}$$

From Eq. 6-119 the pitch attitude transfer function is expressed as

$$\frac{\Delta\theta}{\Delta\delta_{sx}} = \frac{M_{\delta_{sx}}}{s^2 + 2\zeta\omega s + \omega^2} \tag{6-122}$$

and therefore the response $\Delta\bar\theta$ to a unit control step input $\Delta\delta_{sx} = 1/s$ is expressed as

$$\Delta\bar\theta = \frac{M_{\delta_{sx}}}{s(s^2 + 2\zeta\omega s + \omega^2)} \tag{6-123}$$

The asymptotic response per unit control input for the time t approaching infinity is, by definition,

$$\Delta\bar{\theta}(t)_{t\to\infty} = [s\Delta\bar{\theta}(s)]_{s\to o} = M_{\delta_{sx}}/\omega^2 \quad (6\text{-}124)$$

Inverse Laplace transformation of Eq. 6-123 into the time domain results in the time functions for $\Delta\bar{\theta}/M_{\delta_{sx}}$ shown in Fig. 6-38. Here the time is measured in periods of the undamped natural frequency $2\pi/\omega$. A desirable response is obtained for a damping ratio of $\zeta = 0.7$. The response for $\zeta = 0.3$ is undesirable because of the overshoot and the subsequent oscillatory motion; the response for $\zeta = 2.0$ is undesirable because of its sluggishness. For airplanes, the damping ratio ζ is proportional to air density over wing-loading $[e/(w/s)]^{1/2}$ While light airplanes flying at low altitudes easily can be designed with optimum damping ratios and thereby can achieve the desirable control responses of the middle curve in Fig. 6-38, high-wing-loading airplanes flying at high altitudes have very low damping ratios and need artificial pitch damping to achieve acceptable control response. MIL-F-8785 specifies for category A (precision maneuvering) and category C (takeoff and landing) flight phases values of ζ between 0.35 and 1.30.

Comparable specifications do not exist for helicopters, although similar values for ζ could be obtained with suitable augmentation systems. In fact, for helicopters, it is not even required that the asymptotic control response be a finite pitching angle $\Delta\bar{\theta}$. The reason is that, for helicopters without stability augmentation, the static stability derivative M_α is, in the low-

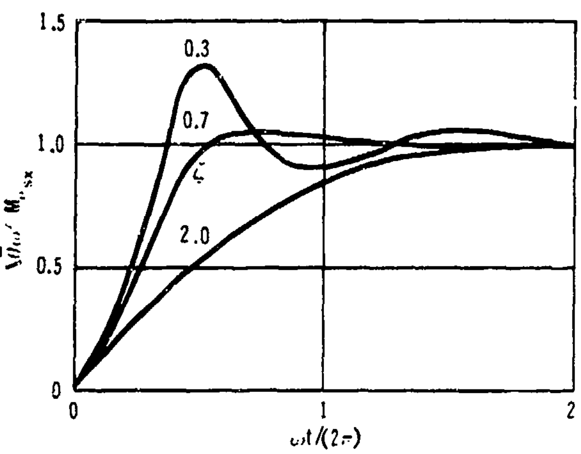


Fig. 6-38. Pitch Responses to Unit Step Control Input

flight-speed regime, quite small and often positive, indicating slight static instability. Therefore, for unaugmented helicopters operating at low flight speeds, Eq. 6-122 for the pitch response transfer function therefore, can be approximated by

$$\frac{\Delta\theta}{\Delta\delta_{sx}} = \frac{M_{\delta_{sx}}}{s^2 + s/\tau_\theta} \quad (6\text{-}125)$$

where the pitch time constant is given by

$$\tau_\theta = -1/M_{\dot{\theta}} \quad \text{, sec} \quad (6\text{-}126)$$

Eq. 6-123 for the response $\Delta\bar{\theta}$ to a unit control step input $\Delta\delta_{sx} = 1/s$ now becomes

$$\Delta\bar{\theta} = \frac{M_{\delta_{sx}}}{s(s^2 + s/\tau_\theta)} \quad (6\text{-}127)$$

and Eq. 6-124 for the asymptotic response at time t approaching infinity becomes

$$\Delta\bar{\theta}(t)_{t\to\infty} = \infty \quad (6\text{-}128)$$

meaning that the response would grow indefinitely if the control input from trim were held.

Inserting $\Delta\theta = \Delta\dot{\theta}/s$ into Eq. 6-125 gives the transfer function for the pitch rate $\Delta\dot{\theta}$:

$$\Delta\dot{\theta}/\Delta\delta_{sx} = \frac{M_{\delta_{sx}}}{s + 1/\tau_\theta} \quad (6\text{-}129)$$

The pitch rate response $\Delta\dot{\bar{\theta}}$ to a unit control step input $\Delta\delta_{sx} = 1/s$ is

$$\Delta\dot{\bar{\theta}} = \frac{M_{\delta_{sx}}}{s(s + 1/\tau_\theta)} \quad (6\text{-}130)$$

And the asymptotic pitch rate at time t approaching infinity is

$$\Delta\dot{\bar{\theta}}(t)_{t\to\infty} = M_{\delta_{sx}}\tau_\theta \quad (6\text{-}131)$$

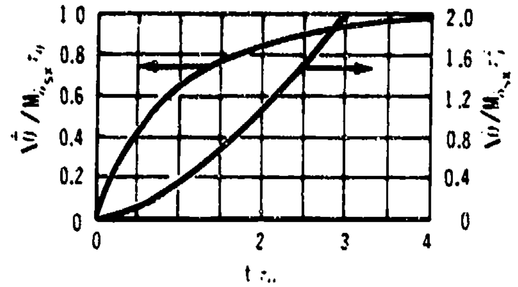


Fig. 6-39. Pitch and Pitch-rate Responses to Unit Step Control Input for Aerostatic Stability

The pitch acceleration $\Delta\ddot{\theta}$ due to unit control step input $\Delta\delta_{sx}$ is given by

$$\Delta\ddot{\theta} = \frac{M_{\delta_{sx}}}{s + 1/\tau_\theta} \qquad (6\text{-}132)$$

Performing the inverse Laplace transformation of Eqs. 6-127, 6-130, and 6-132 into the time domain gives, respectively,

$$\frac{\Delta\bar{\theta}}{M_{\delta_{sx}}\tau_\theta^2} = \exp(-t/\tau_\theta) + (t/\tau_\theta) - 1 \qquad (6\text{-}133)$$

$$\frac{\Delta\dot{\bar{\theta}}}{M_{\delta_{sx}}\tau_\theta} = 1 - \exp(-t/\tau_\theta) \qquad (6\text{-}134)$$

$$\frac{\Delta\ddot{\bar{\theta}}}{M_{\delta_{sx}}} = \exp(-t/\tau_\theta) \qquad (6\text{-}135)$$

Eqs. 6-133 and 6-134 are represented in Fig. 6-39. For $t = \tau_\theta$, a value of 63.2% of the asymptotic pitch rate is reached. The initial pitch acceleration for $t = 0$ is, according to Eq. 6-135

$$\Delta\ddot{\bar{\theta}}(t)_{t\to 0} = M_{\delta_{sx}} \qquad (6\text{-}136)$$

MIL-H-8501 specifies that helicopters should have for visual flight a pitch time constant (Eq. 6-126) of not greater than $I_y^{0.3}/8$ sec, and for instrument flight of not greater than $I_y^{0.3}/15$ sec. For example, a helicopter of $W_g = 10{,}000$ lb and $r_g = 2$ ft should have for visual flight a pitch time constant $\tau_\theta < 1.17$ sec, and for instrument flight $\tau_\theta < 0.62$ sec. The time constant of a helicopter with centrally hinged blades is approximately

$$\tau_\theta = \Omega\gamma r_g^2/(gh) \quad , \text{sec} \qquad (6\text{-}137)$$

where

g = acceleration due to gravity, ft/sec²
h = height of rotor hub above CG, ft
r_g = pitching radius of gyration of helicopter, ft
γ = blade Lock number, dimensionless
Ω = rotor angular velocity, rad/sec

Using typical values of rotor angular velocity $\Omega = 30$ rad/sec, blade Lock number $\gamma = 8$, pitching radius of gyration $r_g = 2$ ft, and rotor hub height h above helicopter CG $= 8$ ft, one obtains $\tau_\theta = 3.7$ sec, which is significantly longer than the specified maximum of 1.17 sec. Hinge offset can reduce the time constant considerably. Blade tip weights, by lowering γ, can reduce τ_θ further; however, an augmentation system providing artificial damping often is called for.

After selecting the time constant τ_θ, the appropriate control power $M_{\delta_{sx}}$ must be provided. MIL-H-8501 specifies that, in hover one second after application of a unit (1-in.) step control input from trim, the pitch angle $\Delta\bar{\theta}$ in degrees must be at least

$$\Delta\bar{\theta} \geq \frac{A}{\sqrt[3]{W + 1000}} \quad , \text{deg} \qquad (6\text{-}138)$$

where for visual flight the feedback constant $A = 45$ and for instrument flight $A = 73$. For $W_g = 10{,}000$ lb and $A = 45$, one obtains from Eq. 6-138 $\Delta\bar{\theta} \geq 2.04$ deg. Eq. 6-133, for $t = 1$ and $\tau_\theta = 1.17$ sec gives $M_{\delta_{sx}} > 5.3$ deg/sec².

For full stick motion from trim in hovering, the initial pitch acceleration and the final pitch rate should be four times the values for 1 in. of stick deflection in order to comply with the MIL-H-8501 requirement that the pitch angle 1 sec after input of all available control be four times the value given by Eq. 6-138.

A feedback loop as used for augmentation systems is shown schematically in Fig. 6-40. The feedback signal is subtracted from the control input signal, yielding a closed loop transfer function

$$\frac{\Delta\dot{\theta}}{\Delta\delta_{sx}} = \frac{1}{\dfrac{1}{G(s)} + KH(s)} \tag{6-139}$$

where

$G(s)$ = system transfer function
$H(s)$ = feedback transfer function
K = feedback gain, in.-sec

Mechanical augmentation systems, such as the Bell stabilizer bar and the Hiller control rotor, provide angular velocity damping in pitch and roll with a first-order dynamic lag. The feedback transfer function $H(s)$ for such systems then is

$$H(s) = \frac{\Delta\delta_F}{\Delta\dot{\theta}} = \frac{K_1}{s + (1/\tau_F)} \tag{6-140}$$

where

K_1 = integral feedback gain, in.
s = Laplace operator, sec^{-1}
δ_F = feedback input subtracting from control input δ_{sx}, in.
τ_F = feedback time constant, sec

For these devices the feedback time constant τ_F is very small, only a fraction of the period of rotor rotation, so that in Eq. 6-140 s can be neglected compared with $1/\tau_F$. For the long-period mode, s has a small stabilizing effect. Taking now the feedback transfer function as

$$\frac{\Delta\delta_F}{\Delta\dot{\theta}} = K_1\tau_F = K \tag{6-141}$$

and the aircraft transfer function from Eq. 6-129, one obtains with Eq. 6-139 the closed loop transfer function

$$\frac{\Delta\dot{\theta}}{\Delta\delta_{sx}} = \frac{M_{\delta_{sx}}}{s + 1/\tau_\theta + KM_{\delta_{sx}}} \tag{6-142}$$

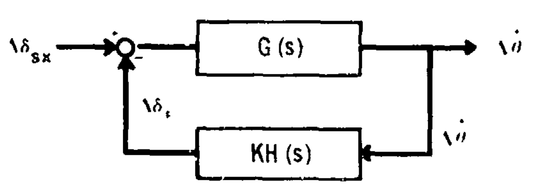


Fig. 6-40. Augmentor Loop

The augmentation system has reduced the time constant from τ_θ to

$$\tau_k = \frac{\tau_\theta}{1 + \tau_\theta KM_{\delta_{sx}}} \tag{6-143}$$

The responses shown in Fig. 6-39 still are applicable if τ_θ is replaced by τ_k. Eq. 6-140 also describes the effect of a rate gyro with lagged signal in the loop. If the time constant of the rate gyro is small compared to the period of rotor rotation, Eq. 6-143 again will be a good approximation.

The combination of a rate and an attitude gyro (ignoring lag) results, instead of Eq. 6-141, in the feedback transfer function

$$\frac{\Delta\delta_F}{\Delta\dot{\theta}} = K(1 + A/s) \tag{6-144}$$

where

A = feedback constant, sec^{-1}

The closed loop transfer function of the pitch attitude $\Delta\theta$ in this case is

$$\frac{\Delta\theta}{\Delta\delta_{sx}} = \frac{M_{\delta_{sx}}}{s^2 + (s/\tau_k) + M_{\delta_{sx}}KA} \tag{6-145}$$

which is of the same form as Eq. 6-123. Responses to unit step control inputs $\Delta\delta_{sx} = 1/s$ now have the form shown in Fig. 6-38 (same as for airplanes), and any desired values of ω and ζ as defined in Eq. 6-120 can be obtained by selection of appropriate values of K and A in Eq. 6-144. The asymptotic response to a step control input from trim is now a pitch attitude angle rather than a pitch rate as before.

6-4.2.2 Normal Acceleration Control

In order to discuss the effects of augmentation systems on normal acceleration control through longitudinal cyclic pitch, the simplest possible representation again is used. Flight speed is constant and all aerodynamic derivatives are zero except for pitch damping and vertical damping. Further, the effect of cyclic pitch upon rotor lift is neglected. The motions of the unaugmented helicopter in response to a pitch control input $\Delta\delta_{sx}$ then are given in body-fixed reference axes by the equations

$$(s - Z_w)\Delta w - U_0 s\Delta\theta = 0 \qquad (6\text{-}146)$$

$$(s^2 - M_{\dot\theta} s)\Delta\theta = M_{\delta_{sx}}\Delta\delta_{sx} \qquad (6\text{-}147)$$

where

U = velocity along helicopter longitudinal body axis, fps
w = vertical velocity, fps
Z_w = vertical damping derivative, $\sec^{-1}$
subscript 0 = initial condition

The second term in Eq. 6-146 is an inertial term required for the description in body-fixed reference axes and takes into account the angular velocity $\dot\theta$ of the linear velocity vector U_0. Eq. 6-147 is identical to Eq. 6-119 except for omission of the static stability term $-M_\alpha$, which is assumed to be negligible. The downward acceleration from trim Δa_z is

$$\Delta a_z = s(\Delta w - U_0\Delta\theta) \quad , \text{ft/sec}^2 \qquad (6\text{-}148)$$

Inserting the values from Eqs. 6-146 and 6-147 into Eq. 6-148 the transfer function for the downward acceleration becomes

$$\frac{\Delta a_z}{\Delta\delta_{sx}} = \frac{U_0 M_{\delta_{sx}}}{\tau_w(s + 1/\tau_\theta)(s + 1/\tau_w)} \qquad (6\text{-}149)$$

where

τ_w = vertical time constant without augmentation, $-1/Z_w$, sec

A typical value of the vertical time constant τ_w for the unaugmented helicopter is 3 sec. The normal acceleration response to a control step input $\Delta\delta_{sx} = 1/s$ will be denoted by $\Delta\bar a_z$:

$$\Delta\bar a_z = -\frac{U_0 M_{\delta_{sx}}}{\tau_w s(s + 1/\tau_\theta)(s + 1/\tau_w)} \qquad (6\text{-}150)$$

The asymptotic normal acceleration response for the time t approaching infinity is obtained by

$$\Delta\bar a_z(t)_{t\to\infty} = [s\bar a(s)]_{s\to 0}$$

$$= -U_0 M_{\delta_{sx}}\tau_\theta \qquad (6\text{-}151)$$

Inverting Eq. 6-150 into the time domain,

$$\frac{\bar a_z}{U_0 M_{\delta_{sx}}\tau_\theta} = 1 + \frac{\exp(-t/\tau_\theta)}{\dfrac{\tau_w}{\tau_\theta} - 1} - \frac{\exp(-t/\tau_w)}{1 - \dfrac{\tau_\theta}{\tau_w}} \qquad (6\text{-}152)$$

Response curves for τ_w/τ_θ = 2, 4, and 8 are shown in Fig. 6-41. It takes considerable time before the final normal acceleration following a step input of cyclic pitch control is reached. For this reason the asymptotic response, Eq. 6-151, is not a very good measure of handling qualities. Instead, MIL-H-8501 specifies that the inflection point in the vertical acceleration response (zero curvature) should be within 2 sec from the control step input. As can be seen from Fig. 6-41, the location of the inflection point is more or less independent of the value of τ_w/τ_θ and can be made to occur earlier if the pitch time constant τ_θ is reduced. A pitch rate augmentation system according to Eq. 6-140 can be used to reduce the time constant, as shown by Eq. 6-143.

6-4.2.3 Long-period Mode

While the essential features of pitch and vertical acceleration control can be obtained by ignoring flight speed changes, the description of the long-period motion must include variations in flight speed. The simplest representation, approximately applicable for low helicopter speeds, is obtained by adding the velocity term $-M_u\Delta U$ to the left-hand side of Eq. 6-147

$$-M_u\Delta U + (s^2 - M_{\dot\theta} s)\Delta\theta = M_{\delta_{sx}}\Delta\delta_{sx} \qquad (6\text{-}153)$$

and by adding the horizontal equilibrium equation

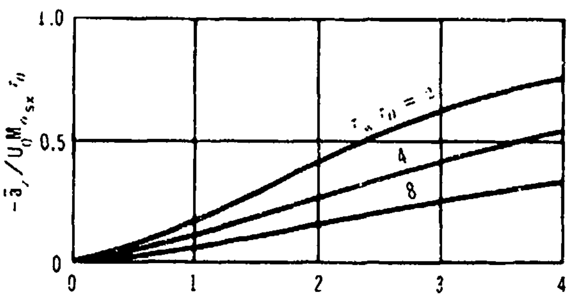


Fig. 6-41. Vertical Acceleration Response to Unit Step Control Input

$$s\Delta U + g\Delta\theta = 0 \quad (6\text{-}154)$$

where

M_u = speed stability derivative, $(\partial M/\partial u)/I_y$, (sec-ft)$^{-1}$

g = acceleration due to gravity, ft/sec^2

The effect of cyclic control upon the horizontal force is ignored in Eq. 6-154. The characteristic equation is

$$s^3 - s^2 M_{\dot\theta} + gM_u = 0 \quad (6\text{-}155)$$

For the long-period mode s is small and the term s^3 can be neglected in a first approximation. One then obtains from Eq. 6-155 the natural frequency of the long-period mode

$$\omega = (gM_u \tau_\theta)^{1/2} \quad \text{, rad/sec} \quad (6\text{-}156)$$

An augmentation system that reduces the pitch time constant τ_θ results, therefore, in a reduced natural frequency ω, or an increased period of the long-period mode. When extracting the roots of Eq. 6-155, one finds a large negative real root that is approximately $M_{\dot\theta}$, and a pair of conjugate complex roots with a positive real part and an imaginary part approximately equal to ω from Eq. 6-156, indicating oscillatory divergence. The effect of a pure rate feedback upon this mode primarily is to increase the period. Significant improvement in the damping ratio of the long-period mode can be made only by adding an attitude gyro into the loop.

Fig. 6-42 shows the root loci of a hovering helicopter as a function of K in Eq. 6-144 for (A) rate feedback and (B) rate plus attitude feedback. The squares indicate the roots of the characteristic equation without augmentation and the arrows on the root curves indicate the direction of increasing gain K.

6-4.2.4 Hardware

The hardware of electronic augmentation systems consists of sensors such as rate or attitude gyros, a signal-modulating network, and electric servos in series with the manual control system (Fig. 6-43). MIL-C-18244 specifies that the control authority of augmentation systems *shall* be limited as far as possible to insure that a "hard-over" signal will not endanger the aircraft. The augmentor servo, therefore, usually has not more than 25% authority over the control travel. In case of augmentor malfunction, the servos often are locked and centered automatically. The power actuator usually is a hydraulic type using two independent hydraulic circuits. The preceding equations are valid only if augmentor servo and power actuator are responding instantaneously. The actual lag of these devices must be considered in a more realistic analysis. Because, in addition to a lagged response, actuators often have considerably nonlinear response characteristics such as deadband or hysteresis, the system sometimes is modeled as a hybrid setup. In such a setup, real time analog or digital elements are mixed with actual hardware components such as electrical servos or hydraulic actuators, which then also must be realistically loaded.

Rate gyros are cheaper and more robust than attitude gyros. In spite of the additional stability improvements obtainable from attitude feedback, helicopter stability augmentation systems (SAS) usually are limited to rate feedback. Another argument in favor of rate feedback only is that the augmented response is of the same type as the unaugmented response.

As long as the SAS must be considered as less reliable than the basic control system, augmentation system failure must be designed for (par. 6-4.3).

When attitude gyros and their associated electrical systems have been perfected to the point where their reliability is sufficient to relax the requirement for a mechanical backup system, the way will be open for control systems with characteristics entirely different from the unaugmented characteristics. For example, a translational control system (Ref. 23) could then be used. With this system, the helicopter is attitude-stabilized and a control force steering system provides rates of change of attitude in proportion to the pilot's stick force input.

6-4.2.5 Gust Alleviation

None of the conventional augmentation systems, which use angular rate or attitude sensors, make full use of the short response time of the blades to reduce or alleviate gust effects before they substantially influence airframe motion.

In order to use the blades effectively for gust alleviation, it is necessary to sense either their motion or the forces that they transmit to the airframe, and to provide fast-acting feedback loops to the cyclic or collective controls. If the rotor is the main source of force and moment inputs into the airframe, linear or angular accelerometers can be used as sensors. If the rotor is only one of several sources of force and moment input, as in compound helicopters, sensors must be provided for blade motions or for rotor-to-airframe transfer forces or moments.

Here again the simplest possible representations are used and the analysis is limited to vertical accelerations

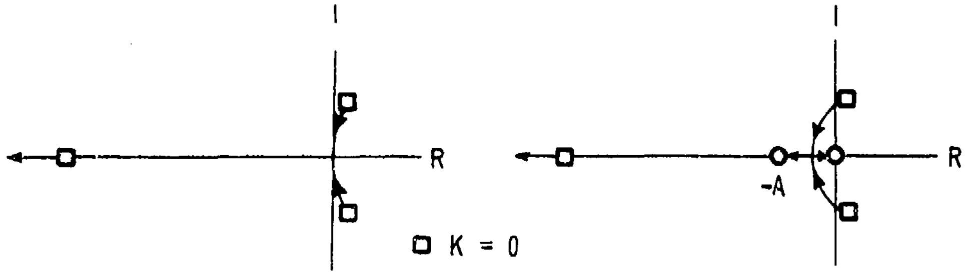


Fig. 6-42. Root Loci for Hovering Helicopter

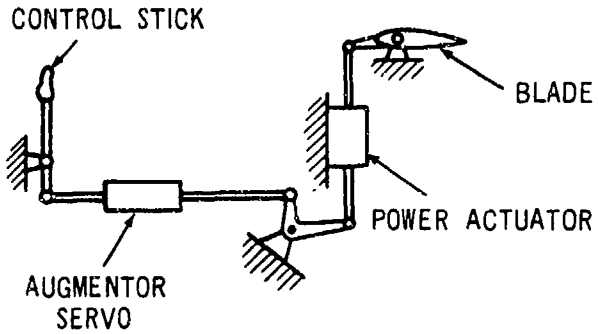


Fig. 6-43. Control Schematic With Augmentor Servo

and aircraft pitching moments induced by gusts. Initially, the airframe is assumed to be fixed and the relative airspeed constant. The description that follows is for a three-bladed rotor with rigid blades, elastically hinged at the rotor center, and subject to uniform inflow velocity $\lambda\Omega R$, ignoring tip losses. All quantities are perturbations from trim. An inertial reference system is used and terms with periodic coefficients in this reference system are omitted. Unpublished studies conducted at Washington University have shown that this omission is justifiable if rotor advance ratio $\mu < 0.5$, and if accurate representation of the frequency response at circular frequencies greater than 0.5Ω is not necessary. In terms of the three variables β_0 (rotor coning angle), β_1 (forward tilt angle), and β_2 (left tilt angle) as a function of θ_0 (collective pitch angle), θ_1 (forward cyclic pitch angle), and θ_2 (left cyclic pitch angle), the equations of rotor motion in Laplace transform notation are

$$\beta_0\left(s^2 + \frac{s\gamma}{8} + P^2\right) + \frac{\beta_2 s\gamma\mu}{12} = \theta_0\left(\frac{\gamma}{8} + \frac{\gamma\mu^2}{8}\right) - \frac{\theta_1\gamma\mu}{6} + \frac{\lambda\gamma}{6} \tag{6-157}$$

$$\frac{\beta_0\gamma\mu}{6} + \beta_1\left(s^2 + \frac{s\gamma}{8} + P^2 - 1\right) + \beta_2\left(2s + \frac{\gamma}{8} + \frac{\gamma\mu^2}{16}\right) = \theta_2\left(\frac{\gamma}{8} + \frac{\gamma\mu^2}{16}\right) \tag{6-158}$$

$$\frac{\beta_0 s\gamma\mu}{6} + \beta_1\left(-2s - \frac{\gamma}{8} + \frac{\gamma\mu^2}{16}\right) + \beta_2\left(s^2 + \frac{s\gamma}{8} + P^2 - 1\right) = \frac{\theta_0\gamma\mu}{3} - \theta_1\left(\frac{\gamma}{8} + \frac{3\gamma\mu^2}{16}\right) + \frac{\lambda\gamma\mu}{4} \tag{6-159}$$

where

P = flapping frequency/Ω, dimensionless

These equations are generalizations to forward flight conditions of the equations used in Ref. 8. A time unit is used for which $\Omega = 1$ rad per unit time. A first insight into gust responses can be obtained by omitting coupling between β_0, β_1, and β_2 and by neglecting μ^2 terms. According to Eq. 6-157 collective flapping from collective pitch θ_0 and from inflow ratio λ are, respectively,

$$\frac{\beta_0}{\theta_0} = \frac{\gamma}{8(s^2 + s\gamma/8 + P^2)} \tag{6-160}$$

$$\frac{\beta_0}{\lambda} = \frac{\gamma}{6(s^2 + s\gamma/8 + P^2)} \quad (6\text{-}161)$$

from which

$$\frac{\beta_0}{\lambda} = 0.75\frac{\beta_0}{\theta_0} \quad (6\text{-}162)$$

Forward tilt from forward cyclic pitch θ_1 and from inflow λ, respectively, are, according to Eq. 6-159,

$$\frac{\beta_1}{\theta_1} = \frac{\gamma}{16(s + \gamma/16)} \quad (6\text{-}163)$$

$$\frac{\beta_1}{\lambda} = -\frac{\gamma\mu}{8(s + \gamma/16)} \quad (6\text{-}164)$$

from which

$$\frac{\beta_1}{\lambda} = -2\mu\frac{\beta_1}{\theta_1} \quad (6\text{-}165)$$

Eqs. 6-160 through 6-165 are valid approximations up to moderate advance ratios for which $\mu^2 \ll 1$. Instead of comparing frequency responses of β_0 and β_1 to gusts, or to vertical velocity λ into the rotor with and without augmentation, it is sufficient, according to Eqs. 6-162 and 6-165, to compare frequency responses to collective pitch θ_0 and to longitudinal cyclic pitch θ_1.

The use of Laplace transforms for the evaluation of the frequency responses of aircraft is described in Ref. 24. Beginning with the collective pitch case having the transfer function Eq. 6-160, and setting $s = i\bar{\omega} = i\omega/\Omega$, one obtains for the absolute value of the frequency response function, assuming $\gamma = 8$, $P = 1$:

$$\left|\frac{\beta_0}{\theta_0}\right| = \frac{1}{[(1-\bar{\omega}^2)^2 + \bar{\omega}^2]^{1/2}} \quad (6\text{-}166)$$

Assuming now a feedback system according to Fig. 6-44 with a coning angle sensor and proportional feedback into the collective pitch control, instead of Eq. 6-166

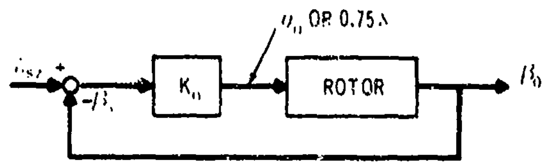


Fig. 6-44. Coning Angle Sensor With Proportional Feedback

$$\left|\frac{\beta_0}{\theta_0}\right| = \frac{1}{[(1 + K_0 - \bar{\omega}^2)^2 + \bar{\omega}^2]^{1/2}} \quad (6\text{-}167)$$

where

K_0 = proportional feedback gain

Figure 6-45 shows a comparison of the coning angle frequency response to periodic inflow, with and without feedback, assuming $K_0 = 3$. It is seen that for values of $\omega/\Omega < 1.0$, which is practically the entire atmospheric turbulence spectrum, the feedback reduces the gust response to about one quarter of the value without augmentation ($K_0 = 0$). For $\bar{\omega}^2 < 1$ the feedback system causes a reduction of Z_ω in Eq. 6-146 in the ratio $1/(1 + K_0)$. The increase in τ_w produced by the feedback does not lead to a delayed inflection point, as can be seen from Fig. 6-41. The reduction in vertical acceleration response from cyclic pitch can be compensated for, if desired, by appropriate selection of $M_{\delta_{LX}}$ and τ_θ.

For the longitudinal cyclic pitch case having the transfer function Eq. 6-163, with $s = i\bar{\omega} = i\omega/\Omega$ the absolute value of the frequency response function, assuming $\gamma = 8$, is

$$\left|\frac{\beta_1}{\theta_1}\right| = \frac{1}{2(0.25 + \bar{\omega}^2)^{1/2}} \quad (6\text{-}168)$$

Assuming now a feedback system according to Fig. 6-46 with a fore-aft flapping angle sensor and integral feedback into the longitudinal cyclic pitch control, instead of Eq. 6-168

$$\left|\frac{\beta_1}{\theta_1}\right| = \frac{\bar{\omega}}{2[(K_1 - \bar{\omega}^2)^2 + (0.5\bar{\omega})^2]^{1/2}} \quad (6\text{-}169)$$

where

K_1 = integral feedback gain

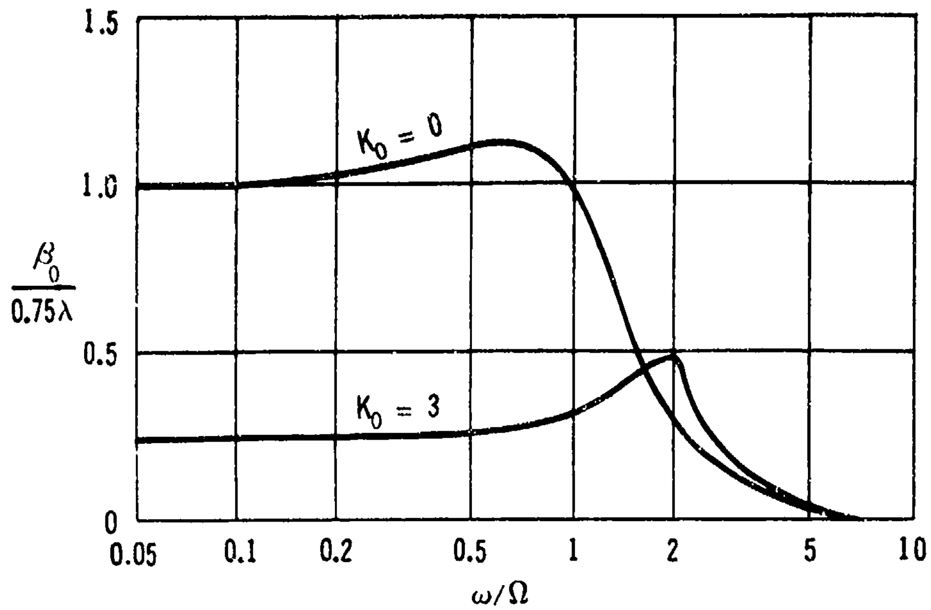


Fig. 6-45. Coning Frequency Response to Periodic Inflow

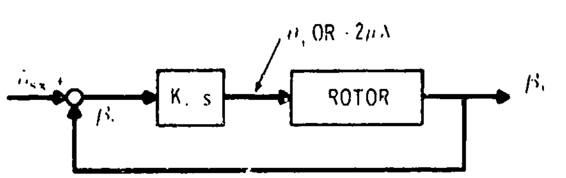


Fig. 6-46. Tilting Moment Sensor With Integral Feedback

Fig. 6-47 shows a comparison of the fore-aft flapping frequency response to periodic inflow with and without integral feedback, assuming $K_1 = 0.5$. It is seen that for values of $\omega/\Omega < 0.3$, which covers a wide range of the atmospheric turbulence spectrum, the feedback reduces the gust response to a small fraction of the unaugmented gust response. For $\omega < 0.1\,\Omega$ the flapping response is virtually zero. This is true not only for periodic inflow but also for all other periodic inputs, including gyroscopic inputs from pitching oscillations. This means that, for low frequencies, the pitching time constant τ_θ approaches infinity. The system of Fig. 6-46, therefore, must be supplemented by a conventional augmentation system with pitch rate feedback, whereby, according to Eq. 6-143 for large τ_θ

$$\tau_k = \frac{1}{KM_{\delta_{sx}}} \tag{6-170}$$

In addition to fixed hub gust alleviation, the body motion contributes to gust alleviation in a manner that also is present without feedback. Considering uncoupled vertical and pitching motions, and assuming the body to behave in vertical motion and in pitch as a first-order system (see Eqs. 6-129 and 6-146 for $\Delta\theta = 0$), the free-body gust alleviation factor κ is

$$\kappa = \frac{\bar{\omega}}{(\bar{\omega}^2 + 1/\bar{\tau}^2)^{1/2}} \tag{6-171}$$

where κ is defined as the ratio of acceleration with damping over acceleration without damping and where the time unit again is such that $\Omega = 1$ ($\bar{\tau} = \Omega\tau$). For high frequencies $\kappa = 1$, and for low frequencies $\kappa = \omega\tau$.

Fig. 6-48 shows the free-body gust alleviation factor κ as functions of frequency ratio ω/Ω for a number of values of reduced time $\bar{\tau} = \Omega\tau$. Typical time constants for vertical motion are $100 < \bar{\tau} < 300$; for pitching motion they are $15 < \bar{\tau} < 50$. Compared with the reduction of responses shown in Fig. 6-47 and Fig.

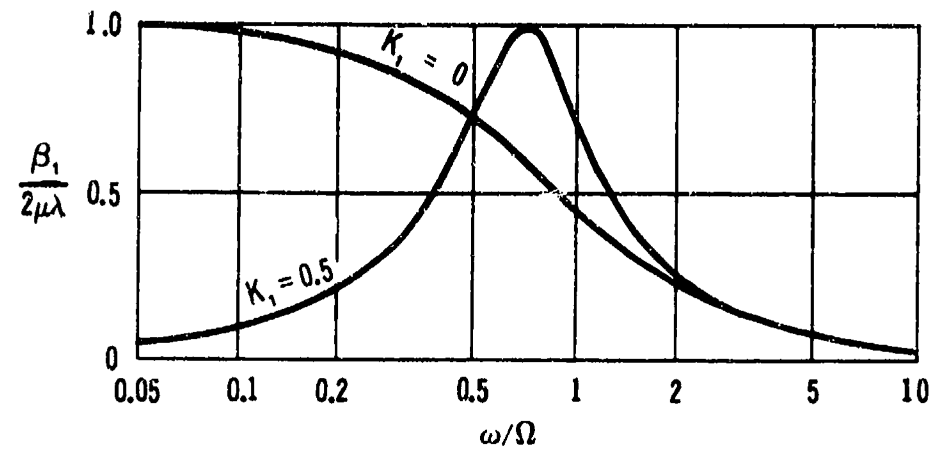


Fig. 6-47. Fore-aft Flapping Frequency Response to Periodic Inflow

6-48, Fig. 6-45 shows, in the frequency range $\omega > 0.05\ \Omega$, very little free-body gust alleviation can be expected.

In order to correlate the frequency ranges in Fig. 6-45 and Fig. 6-47 with atmospheric turbulence frequencies, Fig. 6-49 shows the von Karman vertical gust power spectral density (Ref. 25) with a scale length $L = 400$ ft, which is typical of altitudes above terrain of 300 to 700 ft (Ref. 25). The lower scale in Fig. 6-49 represents ω/Ω for a flight speed of 200 fps (118 kt) and $\Omega = 20$. It is seen that the gust alleviation from Fig. 6-45 and Fig. 6-47 covers a large part of frequency spectrum of Fig. 6-49.

From Fig. 6-45 and Fig. 6-47, it is evident that introduction of feedback results in increased frequency response at high frequencies. While the atmospheric turbulence intensity in the frequency range in which the higher responses occur is low, the response maxima indicate those regions of the frequency spectrum where

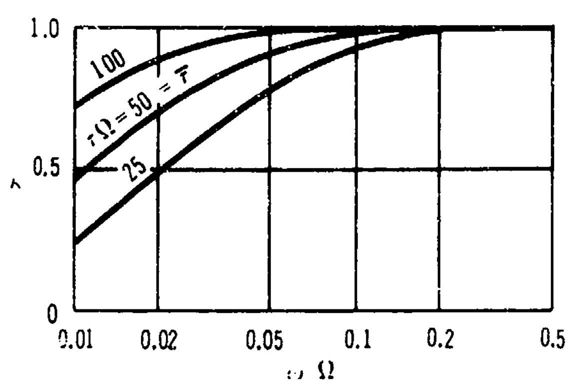


Fig. 6-48. Free-body Gust Alleviation Factor

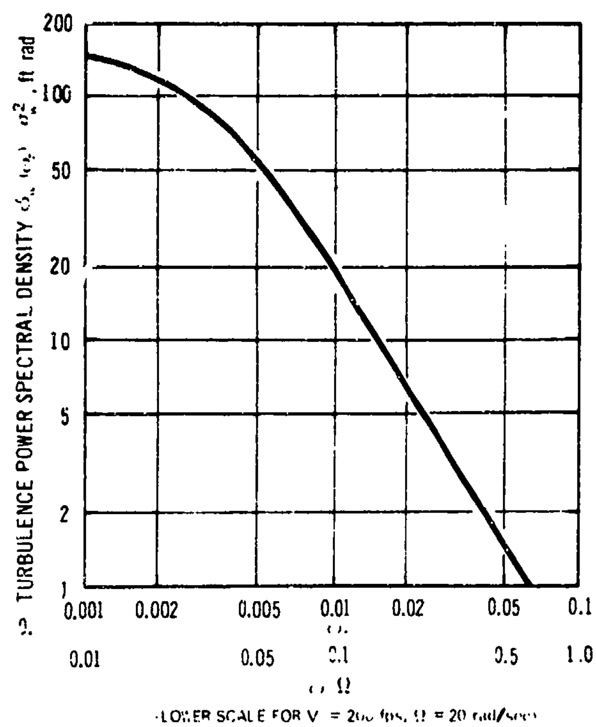


Fig. 6-49. Von Karman Vertical Gust Velocity Spectrum, $L = 400$ ft

coupling with low-damped inplane modes needs attention.

6-4.2.6 Dynamic Instability

Helicopter rotors are subject to four types of dynamic instabilities:

1. Those primarily involving blade torsion modes with frequencies several times Ω

2. Those mainly involving blade fundamental flapping or flap bending modes

3. Those largely involving blade edgewise modes

4. Those mainly involving free-body modes with frequencies substantially below Ω.

Although all of the unstable modes are coupled types and include some blade torsion, some blade flapping or flap-bending, some edgewise motion, and some free-body motion, they usually are denoted according to their principal mode contents. Thus the four types of unstable motions are:

1. Flutter, including stall flutter for unstable conditions with large blade torsion mode contents

2. Flapping instabilities with flapping or flap-bending mode contents

3. Aeromechanical instabilities with edgewise blade mode contents including mechanical instability, or ground resonance

4. Free-body instabilities, which usually are treated in control dynamics.

A review of the first three categories is given in Ref. 27, which also includes 35 other references on these subjects.

All four types of instabilities can occur in both hinged and hingeless rotors. The hingeless type is much more difficult to design properly because of essential nonlinearities; these usually can be neglected in hinged rotors, where the feathering axes participate in the flapping and edgewise blade motions.

When augmentation systems have time constants of the same order of magnitude as those for potentially unstable rotor modes, the augmentation systems influence the rotor mode stability limits and must be included in the dynamic stability analysis. Pitch-flap or pitch-cone coupling, whether obtained by virtual δ_3 hinges or by an augmentation system as assumed in Fig. 6-45 for $K_0 = 3$, has an effect upon the blade flutter limit that can be stabilizing or destabilizing depending upon the shape of the coupled blade flapwise bending mode. High advance ratio flapping instability, occurring with frequencies below Ω, also is affected strongly by pitch-flap or pitch-cone coupling. Large destabilizing effects are produced in a so-called flat-tracking rotor, where tip-path plane tilting is counteracted by a proportional feedback to the cyclic pitch. The integral feedback input into cyclic pitch with which the frequency response at low frequency is reduced, as shown in Fig. 6-47, also produces destabilization of the high advance ratio flapping or flap-bending modes.

Fig. 6-50 shows the integral feedback gain K_I, as defined in Fig. 6-46, at the stability limit as a function of blade Lock number γ and advance ratio μ. The gain K_I is measured in nondimensional time units for which $\Omega = 1$ rad per unit time and must be multiplied by Ω to obtain degrees per second of cyclic pitch per degree of cyclic flapping. Fig. 6-50 (without numerical values of K_I) is from Ref. 7 and has been recomputed by a different method here. The final analysis, in contrast to Eqs. 6-157, 6-158, and 6-159, includes reversed flow effects (Ref. 4), a tip loss factor of $B = 0.97$, and a small stabilizing feedback term not shown in Fig. 6-46. The configuration $K_I = 0.5$, $\gamma = 8$ used for Fig. 6-47 is marked in Fig. 6-50 and shows a maximum advance ratio of $\mu = 1.1$ at the stability limit.

The analysis leading to the stability boundaries of Fig. 6-50 has a minimum of ingredients, i.e., four rigid blades flapping about central, elastically restrained hinges. Incorporation of flapwise and edgewise bending flexibility, torsional and control flexibility, pitch-flap and pitch-lag couplings, and free-body modes leads to numerous coupling terms, many of which have detrimental effects upon the stability limits. Among them are nonlinear terms that are not negligible. In addition to corrections for the effects of torsional, flapwise bending, edgewise bending, and free-body modes, substantial errors also may be introduced by the quasi-steady aerodynamics on which the results shown in Fig. 6-50 are based. A considerable body of analytical, wind tunnel, and flight test work will be required before the problems of dynamic instabilities are understood suffi-

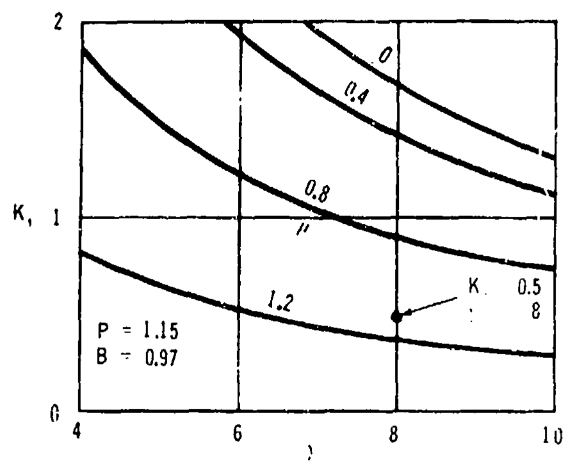


Fig. 6-50. Integral Feedback Gain K_I at Stability Limit

ciently to permit completion of a rational helicopter design with assurance that specified stability margins will be provided with respect to the numerous potential dynamic instabilities.

6-4.3 STABILITY AUGMENTATION SYSTEM FAILURE ANALYSIS

A stability augmentation system normally is a complex mixture of electronic, mechanical, and hydraulic equipment. All of the devices used are subject to failures due to age, stress, or random faults. Thus, for a given mission, certain criteria as to probable failure rates, stability and control characteristics resulting from failure, and operational and maintenance methods to minimize failures must be considered during the preliminary system design. This paragraph covers the consequences of inflight failures of a stability augmentation system (SAS) and design methods that may increase reliability and safety.

The necessity for augmentation of helicopter stability and control is discussed in considerable depth in pars. 6-2.4, 6-2.5, 6-3.1, 6-4.1, and 6-4.2. If the helicopter design has produced a well-stabilized vehicle without a SAS, the addition of a SAS and, subsequently, of automatic stabilization equipment (ASE) only further enhances the handling and control of the aircraft. SAS failures in vehicles with inherent stability usually do not cause divergences that ultimately can destroy the helicopter. However, when artificial means are used to stabilize the motions of a helicopter that definitely are divergent in certain portions of the flight envelope, SAS failures may be catastrophic unless immediate pilot action is taken.

SAS failures can be divided into passive failures, hardover failures, and null-offset failures. Redundancy and monitoring techniques may be implemented to reduce the effects of SAS failures.

6-4.3.1 Passive Failures

All stability augmentation systems have in common the feedback of helicopter angular rate information. Inputs proportional to and opposite in direction from the angular rates of displacement of the aircraft are made to the flight control system. With few exceptions, electromechanical devices with high-speed rotating parts are the basis for angular rate gyros or attitude gyros. Attitude gyros frequently are used in an SAS, with differentiating networks to provide a rate signal for feedback. Alternatively, rate gyros may be used, with an integrating network to provide an attitude signal as well.

These high-speed rotating devices are subject to wear of their rotor bearings, to gimbal failures, and to signal pickoff and motor failures. Motor bearing failures account for the majority of rate and attitude gyro malfunctions, with gimbal bearing failures probably being next most common. Neither of these types of failures produces a hardover output, but both result in loss of the SAS function in nonredundant systems.

For aircraft with inherent stability, a passive failure of the sensors is not critical. If sufficient damping and controllability have been included in the design, the pilot will notice the degraded handling but can complete his mission and return to base. In vehicles with unstable sideslip and/or angle-of-attack characteristics, loss of sensor function may result in upsets that can be disastrous (especially under night VFR conditions) because the proper corrective action is not indicated when external upsets occur. Control of the aircraft after such a failure usually requires that the mission be aborted, or at least be modified by flying at low speeds where the divergence rate is slower. If ASE that relies upon the damping of the SAS is included, the divergence rate following passive failure of the rate sensing system may be increased over the rate for the basic airframe until the ASE is disconnected. With an inherently unstable airframe, a portion of the rate feedback should be included in the ASE to preclude this degradation in performance.

Passive failures of SAS electronics and actuators have the same result as passive failure of a sensor. The electronics can be designed so that passive failures are the predominant type. Electromechanical actuators also fail predominately in the passive mode, but generally have a higher failure rate than do electrohydraulic actuators.

The actuator includes a feedback that matches the actual output with the commanded output. Passive failures of these feedback elements result in conversion of the actuator into an integrating component, and yield a control input equal to the time integral of the input commanded by the SAS. In this case an oscillatory airframe motion will result as long as the SAS remains engaged. However, depending upon the SAS compensation selected, the oscillation may diverge at a lower rate than for passive failures elsewhere in the system.

6-4.3.2 Hardover Failures

Hardover failures of sensors, electronics, and actuators all result in the same airframe response, i.e., a control input followed by the response of the uncontrolled aircraft. Such failures are always worse than passive failures. MIL-H-8501 stipulates that hardover failures must not cause angular rates about any axis greater than 10 deg/sec nor result in changes in normal acceleration greater than ± 0.5 g during the first 3 sec following such a failure in level trimmed flight. This requirement is appropriate for all aircraft intended for use on night VFR missions. For helicopters used for day VFR missions only, the delay time for pilot action may be reduced if the pilot workload is such that he can maintain hands-on flight.

6-4.3.3 Null-offset Failures

Various elements of the SAS can fail in such a manner that a fixed offset is superimposed upon the normal feedback signal. This failure mode results in a transient input with continued normal operation of the system. The pilot must decide whether to disengage the SAS or to retrim the aircraft and continue with degraded performance. An offset actuator may result in nonlinear damping of the airframe, which may be reinforced by the pilot until the system is disconnected.

6-4.3.4 SAS Monitoring

Nonredundant SAS's have been developed that can reduce the transients of a hardover failure. These systems warn the pilot of a passive failure before a large attitude upset can develop. Offset failures also may be detected depending upon the magnitude of the offset.

In an internally monitored system, a test signal is added to the airframe rate signal. The test signal frequency and amplitude are selected so as to be outside of the response characteristics of the augmented airframe. To test the entire system, the test signal is summed with the airframe motion at the rate gyro. The summation is accomplished by torquing the rate gyro via a self-test coil. The resulting high-frequency signal is detected at the output of the electronics as a safe indication

Hydraulic actuator monitoring can be accomplished by comparing the actuator output with an electronic model of the actuator. The model can be designed to include the velocity and position limits of the actuator, as well as the dynamic and static characteristics. With these techniques, many failures of the SAS can be detected; the system may be disengaged and a warning annunciation may be illuminated in the cockpit. The time for detection and cut-off can be less than 0.5 sec, with nuisance disconnects occurring infrequently.

6-4.3.5 Redundancy

When mission or handling characteristics demand full-time augmentation of the airframe dynamics, redundancy of the SAS is considered necessary. The level to which redundancy must be implemented is based upon the allowable SAS failure rate, along with the failure rates of the proposed components that will comprise the system.

The most common form of redundancy is to dualize the complete system from sensors to actuators. Dual systems have been implemented in configurations with both systems engaged at all times (on-on), and with one system operating and the other on standby (active-standby).

The on-on configuration has the advantage that, following a failure of one channel, the other channel acts to attenuate the helicopter response that normally would result. In addition, with two separate systems, authority may be increased while still meeting the failure requirements of MIL-H-8501 for a single fault. The on-on configuration may result in a false sense of security, however. In helicopters with unstable modes without augmentation, a hardover of one channel starts a reaction that is opposed by the remaining system. If means for automatically detecting and disconnecting the failure are not included in the system design, the active channel may saturate in an opposing direction with a resultant divergence similar to a passive failure. This type of passive failure response starts from an untrimmed condition, and although the time to double amplitude may be unchanged, the upset develops to dangerous levels more rapidly.

A monitoring system that compares the outputs of the two channels may improve the failure characteristics by disconnecting both channels before the transient is developed fully. The preferred method, however, determines logically which channel has failed and disconnects only the faulty system. The desired monitoring may be achieved by using two self-monitored channels, or by adding a third set of components to complete the necessary logic. Systems of both types have been developed and tested with excellent results.

Another self-monitoring system that has been demonstrated successfully is a combination of redundancy and monitoring techniques. Triplexed sensors and electronics with monitoring to isolate faults are interfaced to the flight controls via dual actuators, with a single electronic model for comparison. Comparisons at intermediate levels in the electronics, not just at the output, allow all three electronic channels to operate at an intermediate level following a failure by switching the average of the active signals to all three channels downstream of the fault.

All of the self-monitoring systems have the advantages of failure isolation following the first fault and of failure attentuation if a second fault should occur. Of course, as the number of redundant SAS channels increases, so must the number of independent aircraft power sources so that a failure remote to the SAS cannot disable two channels simultaneously.

The active-standby method has no advantages. In the cases where it has been used, it was necessary because the existing actuator could accept only a single input. Several disadvantages preclude acceptance of such a system in future helicopters. First, unless monitoring is included to determine that a fault exists in the active channel, pilot action is necessary to switch to the standby channel. Second, there is no assurance that the standby channel has not failed prior to switching, which may compound an already deteriorating situation.

6-4.3.6 Other Considerations

In addition to the various single-axis failures, the design of an SAS must consider the consequences in one axis of a failure of another axis. For example, if crossfeed of roll rate into yaw is used to enhance turn entry characteristics, a roll failure can affect yaw control as well. In monitored systems, switching off of a failed channel also should remove the crossfeed associated with that channel. In unmonitored systems, provision should be included in the cockpit for switching off both a failed axis and its crossfeed.

The ability to disconnect a single failed channel in flight should be provided in helicopters with critical missions requiring SAS. The switching off of all three SAS axes because of a fault in only one channel should be avoided. The cockpit switching for channel isolation does not need to be in the primary control console, but should be readily accessible to the pilot.

In redundant systems that are operating in active configurations, the system gain is divided equally among the channels. In the event that one channel is switched off, the gain in the remaining channels should be increased as necessary to maintain a suitable level of stability. Some degradation in performance may be acceptable in high-gain systems if actuator saturation during normal maneuvers is likely to occur with the reduced actuator authority. Actuator saturation in high-gain systems may cause excessive angular accelerations as the dynamics change from the free airframe response during saturation to tight response as the actuator becomes unsaturated. The resulting nonlinear response may be oscillatory and may lead to pilot induced oscillation.

6-4.4 AUTOMATIC STABILIZATION EQUIPMENT

As helicopter missions have become more sophisticated, so have the demands upon pilot skills in navigation, tactics, and precision. To enhance the mission capabilities of the helicopter, several special forms of automatic flight control are used to provide pilot assistance and automatic guidance. The stability augmentation system discussed in par. 6-4.2 is intended to improve short-period angular damping; the ASE provides long-period angular damping and flight path control. The modes discussed in this paragraph rely primarily upon a compatible SAS for short-period damping. In helicopters with mission requirements that demand full-time, long-period stabilization, the augmented stability is provided with the ASE.

The ASE function is divided into pilot-assist and aircraft guidance modes for this discussion.

6-4.4.1 Pilot-assist Modes

The pilot-assist modes relieve the flight crew of the task of stabilizing the long-term dynamics, enhance the static control characteristics, and improve the flying qualities by providing responses more suitable to mission accomplishment than are possible with an SAS or with the unaugmented aircraft. The pilot-assist modes discussed are:

1. Attitude hold
2. Heading hold/select
3. Control stick steering
4. Automatic turn coordination
5. Altitude hold/select
6. Velocity vector control.

The pilot-assist modes form the command reference for the guidance modes, which provide the interface computations between various avionic systems and the flight controls.

6-4.4.1.1 Attitude-Hold

Attitude-hold is the primary pilot relief mode and is the basis for many of the guidance and special modes. In general, the purpose of an attitude-hold mode is to stabilize the long-term longitudinal and lateral attitude divergences found in helicopters. The static accuracy of an attitude-hold mode is dependent in large part upon the mission requirements, but in no case should exceed the limits of ± 1 deg over the flight regime of the aircraft given in MIL-F-9490 and MIL-C-18244 . The requirement of MIL-F-9490 that attitude should be maintained up to ± 60 deg of bank and ± 15 deg of pitch is valid for many Army helicopters. In some cases, however, the bank angle limits should be reduced to less than ± 60 deg due to maneuver restrictions.

Upon completion of a pilot-controlled maneuver, the attitude maintained by the ASE *shall* be the attitude at the time the pilot's control forces were removed. In those systems that do not provide synchronization of attitude reference with control stick position, the helicopter *shall* return to the reference attitude commanded by the zero-force stick position or to the reference last established with the flight control equipment. The "wings level" requirements of MIL-F-9490 and MIL-C-18244 generally are not desirable for helicopters due to the need to maintain some roll attitude for unaccelerated flight, especially at hover. If a wings level mode is determined to be undesirable, verni · control of attitude is required via the normal trim controls. A requirement for vernier control precludes trim systems in which control forces are trimmed by release of a spring-centering feel system and re-engagement at a new trim position.

Dynamic accuracy of the attitude-hold mode has not been established by any of the applicable Military Specifications. Until such time as specifications are available, attitude changes in pitch due to power changes for climb or autorotation entry should not exceed ± 5 deg during the first 2 sec. The transient effects of release of stores, weapon firing, and sudden CG shifts due to normal operation should be specified, and will be dependent upon mission requirements. Roll attitude changes due to sideslip, power changes, and release of stores should not exceed ± 2 deg and should be specified in accordance with mission requirements for each vehicle. A smooth return to the reference pitch and roll attitude after a transient change is mandatory, and overshoot *shall* not exceed 20% of the maximum upset as per MIL-F-9490 and MIL-C-18244.

Design of the attitude-hold mode should consider the effects of the system upon ground-handling qualities as well as upon flight characteristics. Landing and takeoff from terrain slopes up to the maximum allowable for the helicopter must not require special pilot techniques when the mode is engaged nor cause motions of the rotor tip path that result in a blade striking another part of the helicopter. In some cases Army helicopters may be required to land on riverine or seagoing vessels. The utilization of an attitude-hold system for the approach reduces the effects of upsetting moments caused by turbulence off the superstructure of large vessels and thus is a desirable feature during approach. Again, special techniques should not be required to engage or disengage the ASE, and deck motion must not cause movements of the tip-path plane that are hazardous to deck personnel.

6-4.4.1.2 Heading-Hold/Select Mode

The heading-hold mode provides pilot assistance throughout the flight regime. In hover, the directional stability and antitorque requirements of most single-rotor helicopters require constant attention to heading control by the pilot, especially in gusty crosswinds. The tandem and coaxial rotor configurations, as well as reaction-powered single-rotor configurations with low directional stability, generally do not place as severe a requirement upon heading control. In cruise flight, heading-hold reduces the navigation task. The ± 1 deg static accuracy requirements of MIL-F-9490 and MIL-C-18244 are reasonable for most helicopter missions. In cases where mission requirements dictate high navigational accuracy, more stringent specifications should be applied.

The most common method of achieving heading-hold in helicopters is via feedback of a synchronizer-referenced heading error to the yaw moment control. As helicopter speeds increase, designs that implement heading-hold via roll control become desirable. Justification of control mode changing is predicated upon the mission requirements, and should be considered carefully.

Dynamic accuracy for the heading-hold mode has not been established by Military Specification. However, in no case should transient excursions greater than ± 10 deg occur during rapid uncommanded decay of rotor toruqe. At airspeeds above the best climb speed, the maximum dynamic upset caused by a sudden uncommanded decrease in rotor torque should not exceed ± 5 deg during the first 2 sec following the power loss. Variations in heading caused by normal power changes, except for jump takeoffs and quick starts and stops, should not exceed ± 2 deg. In the excepted maneuvers, uncommanded heading changes should not exceed ± 5 deg.

Heading-select is a special type of heading-hold and command. In the heading-select mode, the ASE *shall* turn the helicopter automatically through the smallest angle, left or right, to a heading either selected or preselected by the pilot, and *shall* maintain that heading as in the heading-hold mode of MIL-F-9490 and MIL-C-18244. The static and dynamic requirements for the heading-hold mode also are applicable. Heading-select is a desirable mode for many missions, primarily in cruise flight. However, incorporation of this mode into helicopters has not been accepted widely. Because heading-select is achieved through the roll control system, a roll attitude-hold system also is necessary, whereas heading-hold through the yaw control system may be incorporated into an aircraft without roll stabilization.

Turn coordination for the heading-select maneuver is maintained by the ASE turn coordination feature. The heading selector must have 360-deg control. The bank angle while turning to the selected heading should be limited to an amount that provides a satisfactory turn rate, commensurate with the maneuver limits for constant altitude turns and for preclusion of blade stall. This latter requirement may limit the use of this mode near maximum speed if bank angles greater than 25 deg are specified, due to the reduced speed for blade stall onset with increased load factor. In extreme cases, a bank angle limit as a function of airspeed may be necessary.

Turn entry, using the heading-select mode, must be made rapidly with a smooth variation in roll rate, without reversals, up to the bank angle limit. Turn exit should also be made with a smooth variation in roll rate, with reversal permitted only after reaching or overshooting the selected heading. The helicopter *shall* not overshoot the selected heading by more than 1.5 deg as per MIL-F-9490 and MIL-C-18244.

Special procedures for engagement or disengagement of the heading-select mode during normal aircraft operations must be avoided. Engage logic should preclude operation of the heading-hold and heading-select modes during ground operations.

6-4.4.1.3 Control Stick Maneuvering

The pilot should be able to perform maneuvers using the normal helicopter controls while the attitude, altitude, and heading-hold modes are engaged. There is no specific requirement in the Military Specifications for the inclusion of proportional maneuvering. The various forms of control stick maneuvering are:

1. Disconnect or overpower
2. Position proportional
3. Force proportional.

6-4.4.1.3.1 Disconnect Maneuvering

Disconnect control stick maneuvering provides the same maneuvering characteristics as the SAS or basic airframe by disconnecting the ASE error signal from the appropriate axis actuator. Thus, no change i. maneuvering qualities results from the operation of the ASE. To maneuver, the pilot applies a force to the cyclic stick, collective control, or rudder. The force is sensed by switches or a force transducer. The ASE error signal is disconnected while the pilot has control, as indicated by the control force sensors. In some systems, the attitude reference remains constant throughout the maneuver, and the aircraft attitude returns to the previous reference when the force is removed. In these designs, a change in reference may be commanded linearly via a beeper trim, or by releasing the force feel system temporarily and thus commanding resynchronization when the feel system is re-engaged. In either case, changes in reference require positive pilot actions. This mode of attitude operation is very desirable in night VFR missions for the pitch and roll axes because the pilot may return the aircraft to a known airspeed and turn rate merely by releasing the control force. Obviously, this form of maneuvering generally is undesirable for the heading and collective axes because returning to the previous heading and/or altitude seldom is required or desired. An applied lateral or pedal force should result in synchronization of the heading signal. Likewise, collective force should synchronize the altitude reference. A heading-select mode should disengage automatically and the system should revert to a heading-hold mode upon application of lateral and pedal forces. For many day VFR missions, the synchronization of pitch and roll attitudes upon application of cyclic control forces may also be desirable.

Disconnect-type control stick maneuvering systems require detection of force levels within the mechanical breakout range of the flight control system to avoid transients upon switching. Even though reference attitude synchronization with maneuvering force is used, any attitude error at the time of disconnect or reconnection will result in a transient. Transient softening by fade-in/fade-out of system gain is very desirable in a disconnect steering system. However, it may be eliminated if system closed-loop gains are low. Another undesirable feature of some disconnect systems is the change in control system stiffness and, therefore, frequency response, when the error signal is connected. If the forces are released in a dynamic condition, large angular accelerations may be commanded unless transient softening is employed.

6-4.4.1.3.2 Position Proportional Maneuvering

The control position command, or position proportional, system has been used widely in helicopter cyclic ASE maneuver modes and in a few directional systems. This system senses pilot-commanded control motions and forces as a response of the aircraft proportional either to rate or to attitude. Thus, the force-feel system may remain engaged to provide a feel and an attitude or rate reference, or it may be released for reduced pilot effort during transitions and hovering. Generally, this form of control stick maneuvering has been used in aircraft without attitude synchronization and with series actuator systems because of the inherent instability of the control-position sensor, parallel-actuator loop. If means are provided for measuring the difference between the control position commanded by the ASE through a parallel servo actuator and the control position commanded by the pilot, this minor loop can be stabilized. However, the control differential allows operation only with force feel engaged and is much like the force command system discussed below. A series actuator with parallel automatic trim follow-up could be employed with the differential position sensor, providing maneuver command with or without the feel system engaged by disabling the trim system during maneuvers.

As in the case of disconnect systems, the position-proportional maneuvering system must provide for beeper control of lateral and longitudinal attitudes. Beeper control of heading also has been used, but not widely accepted. In a series actuator-controlled system, the usual beeper control commands control position and thus either maneuvering attitude or rate. With a parallel actuator, or series actuator plus trim follow-up, a reference signal proportional to the time integral of beeper switch deflection may be summed with the attitude error signal for trim, thus eliminating actuator offset. Position referenced command of the collective control has been employed using a reference control determined by a self-synchronizing sensor. The reference control position is established at the time the system is engaged and is summed with synchronized attitude information. A similar system could be implemented by synchronizing the attitude signals in the other control axes to reduce series actuator offsets and thus eliminate the tendency for transients upon engagement or disengagement.

6-4.4.1.3.3 Force Proportional Maneuvering

Force proportional control maneuvering is the method specified by MIL-F-9490 and MIL-C-18244. However, the desirability of maneuvering with no force gradient during hovering and translational flight indicates that some exceptions should be considered. As the hovering handling qualities of helicopters are improved through automatic flight control systems, the desirability of maintaining a force feel at all times may make the force proportional system more acceptable to pilots.

Force proportional systems may be implemented with parallel, series, or combination actuation systems. Either step and hold inputs or release of the controls by the pilot requires an actuator response that causes unusual cockpit control motions or forces.

6-4.4.1.4 Automatic Turn Coordination

Both MIL-F-9490 and MIL-C-18244 require automatic turn coordination throughout the airspeed range between 30 kt and the maximum airspeed unless the aircraft mission is such that automatic turn coordination can be deleted. Specifying a single airspeed for all helicopters above which turn coordination is required does not seem justified because large sideslips can be tolerated up to the minimum power-required speed (which may be above 50 kt). Also, sensing a 30-kt speed for switching in the turn coordination function is difficult in view of the poor accuracy of measurement in the very low airspeed regime.

The necessity for automatic turn coordination may come from inherent aircraft characteristics or from the SAS implementation. The yaw SAS generally detracts from turn coordination during the turn entry until the rate signal has washed out. If no washout is employed, the steady-state coordination also is affected. In some helicopters the automatic turn coordination system enhances the apparent static directional stability as well as the maneuvering characteristics; the automatic system is attempting to reduce either lateral acceleration or sideslip angle to zero in opposition to the pilot's control input, and the pilot is required to command a larger pedal displacement per degree of sideslip and thus feels an apparent increase in stability gradient.

Automatic turn coordination has been implemented by lateral acceleration feedback, sideslip sensing, and calculated coordination signal methods. The pilot senses the amount of miscoordination by feeling lateral forces or by reading the cockpit sideslip indicator, normally a pendulous accelerometer. In single-rotor helicopters, to nullify these indications, it generally is necessary to maintain an aerodynamic sideslip and bank angle. A lateral accelerometer is the best sensor choice for a turn coordination system in this type of vehicle.

Tandem-rotor helicopters have been instrumented successfully for turn coordination using either sideslip sensing or an accelerometer as the signal source. Sideslip sensing has been accomplished successfully in operational helicopters by sensing the differential pressure on opposite sides of the aircraft structure at the forward part of the fuselage. In several tandem designs, this method has been used to improve substantially the directional static stability of an otherwise unstable helicopter. The lateral accelerometer feedback can provide both sideslip stability and turn coordination. Because of the relative simplicity of the lateral acceleration sensing system, it should be considered in future designs.

The computational technique for turn coordination uses the relationship between bank angle and forward velocity to determine the required turn rate. However, this method can result in large lateral acceleration errors when airspeed is used in the computation rather than the velocity of the CG referenced to earth axes. In high-speed helicopters, where high bank angles are demanded for turns, the error between the computed turn rate about the vertical axis and the commanded body-axis yaw rate can be significant. This method has been used with moderate success but clearly is not the best system.

Selection of the accelerometer location for turn coordination has a significant effect upon system performance. Effects of angular accelerations about both roll and yaw axes must be considered to obtain best performance. Accelerometers must be of the nonpendulous type, especially in helicopters with large allowable normal load factor ranges. Pendulous accelerometer gain is affected directly by normal acceleration, with gain reaching infinity during a zero load factor maneuver.

The requirements of MIL-F-9490 and MIL-C-18244 for static and dynamic accuracy are consistent with the missions of Army helicopters. To achieve the dynamic requirement limits, a crossfeed of shaped lateral control displacement or roll rate generally is necessary. When the crossfeed is found to be necessary, consideration should be given to the possible effects of roll system failures on the yaw axis characteristics.

Turn coordination systems almost always control the helicopter via series actuation. This technique avoids feedback of rapid control motions during turn entry and allows improvement of static directional stability. A trip follow-up generally is desirable if large actuator offsets are required for control.

6-4.4.1.5 Altitude-hold and Altitude-select

Historically, the altitude-hold mode has been designed to control the thrust of the main rotor directly. In some instances, the penalty in additional fuel consumption has been great enough to preclude operational use of this mode. As helicopter speeds increase, it becomes necessary to consider indirect control of lift via changes in angle of attack as the means of controlling the high-frequency portion of the altitude-hold function. System complexity considerations, of course, may dictate a single mode of control throughout the flight envelope.

The barometric altitude-hold accuracy requirements of MIL-F-9490 and MIL-C-18244 are not stringent enough for flight safety during maneuvering flight in that a 90-ft error is permitted for bank angles greater than 30 deg. For helicopters, altitude errors should not exceed 50 ft up to the maneuvering limits.

The design of a barometric altitude-hold mode must consider the means of synchronization, the actuation system, the mode of commanding changes of reference, and the flight envelope in which the system must perform. Self-synchronizing barometric pressure sensors are available as the system reference. These devices have a bellows arrangement that senses the change in pressure, and a means of clutching the sensitive element to the bellows at the time of engagement. The sensitive element can have a high output gradient because only a small range of altitude variation must be sensed following system engagement. However, the stick steering requirements of MIL-F-9490 and MIL-C-18244, which require an altitude rate proportional to collective force, cannot be met easily with this system configuration. Disconnect stick steering in the altitude-hold mode has been very satisfactory in operational systems and should be considered as a possible alternative.

Electronic or electromechanical synchronization of continuous proportional altitude sensors is becoming more common as a source for barometric altitude input to ASE. As automatic altitude reporting becomes a requirement across the continental U.S., most Army helicopters will be equipped with altimeters that include some form of electrical output. If these sensor signals are encoded externally, the same reference may be used both for reporting and for the ASE. In this case the altitude-rate-proportional control maneuvering is accomplished easily by driving the synchronizer at a rate proportional to control force.

Whichever control maneuvering system is chosen, a positive means must be provided by which the pilot can disengage the vertical mode without removing his hands from the control. This requirement results from the necessity for eliminating ASE vertical axis control during autorotation. In a disconnect control stick maneuvering system, no transient will occur with system disengagement. In a force or displacement proportional system, means must be provided to smoothly but rapidly reduce the commands to zero prior to disconnect. In some helicopters, the stable autorotation rotor speed is indicated by a force detent that might be used to disconnect the altitude-hold mode automatically as the collective is lowered through the detent position.

Barometric altitude-hold via collective control generally may be achieved at low speeds without additional damping. If an integrating servo is used as the controller at high speeds with only collective control, damping may need to be augmented. In most cases, the altitude sensors will provide signals with low enough vibration content that derived altitude rate is usable for this function.

Radar altitude-hold in helicopters, especially under night VFR conditions, is becoming a common requirement. This mode is useful especially in terminal area operations, and in low-speed flight (40 kt) where barometric altitude data are subject to large errors. The radar altitude mode may take the form of a hold or a select and hold system, depending upon mission requirements. If synchronization of the barometric signal is achieved with the ASE, a common synchronizer may be used for both radar and barometric modes. Thus, control maneuvering may have the same effect upon the helicopter regardless of mode.

Radar altitude accuracy requirements of MIL-F-9490 and MIL-C-18244 are given as ±7 ft or 10%, whichever is greater, with respect to the altimeter accuracy. The Navy's general helicopter specification, SD-24, Volume II, allows only a ±2-ft deviation consistent with the accuracy of the radar altimeter. While the latter requirements seem excessively restrictive, the former requirements generally are not adequate. A variation in total system accuracy (including the altimeter) from ±1% at an altitude of 20 ft to ±10% at 300 ft and ±30 ft above 300 ft will produce a more satisfactory system from both operational and flight safety viewpoints. This accuracy should be met in straight unaccelerated flight. These accuracy limits with respect to the radar altimeter indication should be retained in coordinated, turning maneuvers that result in turn rates of 3 deg/sec. The dynamic response requirements of the Military Specifications are acceptable in all cases.

Altitude-select modes using the radar altimeter as a reference have been a part of the ASE for a large number of helicopters. This mode provides means for the pilot to descend or climb to a given altitude while devoting most of his attention to airspeed control and maneuvering during terminal operations. The system compensation for radar altitude-hold and select is the same, but vertical velocity limiting and command smoothing are necessary to prevent excessive load factors and altitude rate commands. The radar altitude-select mode should revert to a radar altitude-hold mode if control maneuvering is used by the pilot. Thus, a change in altitude may be commanded by the pilot without returning to the previously selected reference when the control forces are released, and a pilot action is required before the altitude-select mode can regain control of the helicopter.

Due to the low-pass filter characteristics of radar altimeters, a combination of radar and barometric altitude signals may be desirable for improved system performance. The barometric signals can provide damping and short-period control, with long-term reference established by the radar signals. Thus, the filter characteristics that reduce radar high-frequency performance are compensated. However, the gain of the barometric rate damping signal must be low enough to permit rapid climbs and descents without causing large altitude offsets from the radar altimeter reference.

Control of altitude through the collective axis normally is via a parallel actuator. Limited authority series actuation, with trim followup, also has been employed. However, rapid control inputs by the ASE that are not reflected to the cockpit collective control, but are detected as torque changes, are not readily accepted by pilots, especially if variations of 5% or more are commanded.

In cruise flight, especially in high-speed helicopters, the short-period control of altitude may be accomplished by commanding pitch attitude with long-term trim of thrust. This arrangement is unsatisfactory at hover, and means for programming the method of control with airspeed must be provided. It may be possible to provide a discrete switching point for the changeover, but programming will produce the most desirable system.

A further consideration in all altitude-hold systems is engine governing dynamics. In helicopters with rapid engine response characteristics, a problem seldom is encountered. However, in some helicopter systems, the frequency response of the governor system and altitude-hold loop may cause sufficient phase shift to develop instability, especially at hover. In these cases, additional lead to the engine controls via collective position feed-forward may stabilize the loop.

6-4.4.1.6 Velocity Vector Control

The concept of velocity vector control in the low-speed flight regime has been used in Navy helicopters for several years. Velocity information from Doppler radar systems is summed with pitch and roll attitude to produce a velocity proportional to control position. The resulting system does not really fit in the general category of pilot-assist modes but rather is a navigation or guidance mode.

More recently, several systems have been developed that produce pseudo-inertial velocity feedbacks from autopilot-quality accelerometer and vertical gyroscope systems. The velocity and acceleration data—either in a body axis reference system corrected for gravity effects, or in a vertical referenced system—are summed with attitude and shaped pilot commands to provide short-term velocity responses. Typically, these systems wash out with a time constant of something under 1 min so that inaccuracies in the computation are eliminated.

The advantage of such a system lies in reducing the effects of external disturbances and the amount of anticipation required from the pilot in controlling helicopter velocity. The stability of the airframe with this type of system allows the pilot or possibly a crew member to control the helicopter without reference to the horizon. Thus, instead of hovering with a reference some distance from the aircraft, the pilot can concentrate on a point near the aircraft.

In choosing the best maneuver response for a given aircraft mission near hover, greatly improved apparent speed stability can be provided if necessary, or an attitude response may be maintained like that of the attitude-hold system.

The velocity vector control system requires a series actuator system if apparent speed stability improvement is desirable. The authority requirements of the system are proportional directly to speed stability and the range of speeds over which the system is operated. In most cases, the velocity error signals should be limited to preclude actuator saturation and loss of the SAS function of the ASE.

6-4.4.2 Guidance Modes

Upon the addition of guidance modes, the system becomes a true automatic flight control system. Guidance modes provide complete control of certain parts of the aircraft mission profile, with the pilot acting as a monitor on system function. In most cases the modes have been developed and demonstrated, but with a few exceptions they have not been included in operational helicopter automatic flight control. In all of the modes —except some of the navigational functions—sophisticated sensing and computational requirements will restrict the use of these modes to medium and heavy attack/assault helicopters. The guidance modes are:

1. Automatic navigation
2. Automatic transition
3. Automatic hover
4. Automatic stationkeeping
5. Automatic terrain-following.

6-4.4.2.1 Automatic Navigation

Automatic navigation modes for helicopters have been demonstrated successfully in a wide variety of vehicles. However, the operational necessity for such systems has been established only recently. All of the modes have in common the requirement to maintain a flight path to achieve a given destination, and all require a suitably stabilized lateral-directional inner loop (par. 6-4.4.1.1). The navigational modes discussed are:

1. VOR/TACAN
2. Localizer
3. Ground speed.

The VOR and TACAN navigation modes are similar functionally. The desired course to the station is selected in the cockpit, using the normal procedures for flying either mode manually. Upon engagement of the VOR/TACAN guidance mode, if a valid signal is being received, the helicopter commences a turn to the selected magnetic bearing. As in the heading-select mode, the bearing error commands a bank angle proportional to the bearing angle up to some preset limit. As the error is reduced to zero, the aircraft rolls out to the attitude required for straight flight and flies to or from the beacon as necessary to null the bearing error. Compensation of the beam error signal by adding beam error generally is necessary for suitable system stability.

Because the VOR/TACAN bearing error is an angular measure, the lateral position variation with respect to the beam for a given error is proportional to the distance to or from the transmitting station. Thus, the closed loop response of the helicopter to lateral position changes also is a function of range. When the TACAN receiver is operated in the receive mode, or in conjunction with an omnidirectional beacon without range capability, a compromise loop gain must be selected to provide suitable accuracy at long ranges without lightly damped hunting at short range. If the TACAN receiver is operated in the transmit/receive mode, provision for navigational mode gain changing as a function of distance will provide excellent performance regardless of range. A means should be provided for holding gain at some preselected value as signal strength weakens and the range data consequently become invalid, even though a valid bearing signal may still exist. Also, means should be provided to inhibit commands during the TO-FROM switchover period as the helicopter flies over the ground station. Reversion to heading hold for a set period of time—approximately 10 sec after loss of signal—will allow the aircraft to overfly the station without developing significant errors. This same function will hold the helicopter on heading as loss of signal occurs at the range extremes. If the signal is not regained during the delay time, the VOR/TACAN mode should disengage, with proper annunciation on the cockpit control panel, and the system should revert to a heading-hold mode.

Control stick steering during any navigational mode should cause the system to revert to the basic ASE modes.

The localizer mode is a special radio-beam coupler mode. In this mode, only one course can be flown. A polarized signal proportional to the helicopter position with respect to the beam centerline is detected and processed in the receiver. The processed signal indicates the angular relationship to the beam. As in the VOR/TACAN mode, a turn to the beam is commanded. The initial capture of the beam should be possible from ranges of 4-8 mi and intercept angles of 45 deg. For conventional approaches, i.e., glide slope angles of approximately 3 deg, the intercept should be completed 5-6 mi from the touchdown point. For helicopters, the trend is toward steeper approaches of 6-9 deg with the 4-mi intercept being a maximum requirement under night VFR conditions using a conventional coupler mode. The intercept angle requirement may be increased to suit certain missions. Following the localizer intercept, the coupler gain and compensation should change to assure minimum error as the approach terminates.

MIL-F-9490 requires that the beam intercept overshoot not exceed 50% of full-scale deflection of the localizer indication. The second overshoot and steady-state error must not exceed 10% of full scale.

The velocity vector control method (par. 6-4.4.1.6) has been used successfully during manual approaches at glide path angles of up to 9 deg with good success. Analysis has shown that self-contained inertial damping can enhance the coupled approach mode greatly. Additional system gain modification as a function of radar altitude also may be beneficial during the terminal phase of the approach.

Automatic coupling of the vertical mode through a glide path coupler has not been required in past Army helicopter ASE. The advent of tactical landing beacons with accurate localizer and glide path capabilities may bring forth such requirements in the future. The altitude-hold modes of par. 6-4.4.1.5 can be modified to be compatible with fully automatic approach couplers. Such a system would maintain altitude, either barometric or radar, until the glide path is intercepted. The beam error then would control the aircraft, commanding altitude-rate variations through the same controls as the altitude-hold mode. Some of the beacons also provide range information that can be used to control a programmed deceleration as a function of distance to destination. The basic requirements for the control of speed are covered in par. 6-4.4.2.2.

Automatic control of helicopter drift and heading velocity has been used for a number of years in the Navy antisubmarine warfare (ASW) mission. As Doppler and inertial navigation systems are added to the Army avionic complement, this type of automatic navigation may become a mission requirement.

The Doppler-based systems have been implemented in both speed-hold and speed-select configurations. A ground-speed-select system uses a reference from a cockpit controller to command ground speed and drift angle, or heading and cross-track velocity. The error between the reference and the ground speed commands appropriate pitch or roll attitude changes. Limits upon pitch and roll attitudes that may be commanded during accelerations should be set at not more than ± 10 deg. Control maneuvering by the pilot should disconnect the ground-speed-select mode.

A synchronized ground-speed mode has several advantages over a select type of system. For example, variations in reference speed can be achieved by a velocity-proportional control force, by a displacement-maneuvering system, or by beeper trim. With a synchronized ground-speed system, the automatic transition system (par. 6-4.4.2.2) is a natural outgrowth. Ground speed control of compound helicopters will be achieved by modulation of the horizontal thrust system. Control maneuvering for this type of system is undefined as yet, but a disconnect and resynchronization approach seems logical because speed variations should be controllable easily by thrust variation if altitude-hold is controlling pitch attitude and rotor thrust.

Because of the noise characteristics of Doppler systems, inertial augmentation is a desirable part of the system. The feedback portion of the velocity vector system (par. 6-4.4.1.6) could be used to provide the damping necessary to stabilize the flight path during periods when the Doppler data are noisey or unreliable. Because long-term velocity information is provided by the Doppler system, the pseudo-inertial velocity information may be washed out for a period of 10-20 sec to avoid null and subsequent drift problems. Velocity data from an inertial navigation system may be used as a direct replacement for Doppler information, with a probable improvement in performance of the short-period characteristics.

The ground-speed-hold system requirements of MIL-C-18244 are ± 1 kt or $\pm 2\%$, whichever is greater; those of MIL-F-9490 are ± 5 kt or $\pm 10\%$, whichever is greater. Both specifications are for flight in calm air. The latter specification may suffice for some missions, but performance somewhere between the two levels should be sought. The ± 1-kt requirement indicates a relatively high gain between ground speed and pitch attitude that could produce undesirable angular accelerations in a helicopter in gusty weather and high-speed flight. MIL-C-18244 also specifies that in a ground-speed-select system, the transition from the speed at the time of engagement to the selected speed *shall* be smooth and *shall* not overshoot more than 20% of the incremented change in ground speed. A more realistic requirement might be for an overshoot limit of 20% or 10 kt, whichever is smaller, to preclude the possibility of exceeding helicopter limitations. Because the variation in helicopter velocity is usually a low-frequency function, a parallel actuation system may be used to give the pilot an indication of the trim condition of the aircraft without compromising system performance.

6-4.4.2.2 Automatic Transition

Automatic transitions have been a basic part of the mission requirements of Navy ASW helicopters. The automatic systems, as configured presently, use Doppler velocity and radar altitude select systems. The ASE coupler controls the rate of change of altitude and ground speed to produce a transition to hover once the pilot has established a given set of initial conditions. The resulting response is open loop in that a specific hover position is not obtained automatically; however, an experienced pilot can achieve desired hover position by estimating the distance necessary for the transition and engaging at the proper time.

The Army helicopter mission cannot tolerate the open loop nature of the described type of transition due to the limited area in which the final hover usually must be achieved. The additional requirement for an accurate position following the inbound transition demands that the automatic transition mode rely upon command signals generated in a coupler based upon the navigation system, the obstacle avoidance radar, the ASE, and perhaps an external beacon system, as well as upon air data and power management information. The complexity of the calculations necessary to generate attitude and altitude commands virtually requires inclusion of a digital computer in the aircraft avionic complement.

The navigation system information is the basis for the automatic transition position information. Depending upon the accuracy of the navigation system, and the accuracy to which the final hover point is known, an external reference such as a radio beacon may be required as the absolute reference. The beacon may update the navigation system reference prior to initiation of the approach by being overflown by the helicopter (a technique that is undesirable tactically) or during the inbound approach prior to engagement of the coupler. The engage gate thus becomes a volume in space with vertical and horizontal limits dependent upon the maximum permissible and the desirable maneuver limits for the transition. The length of the engage gate is a function of range and existing airspeed. The pilot must establish a set of initial conditions—altitude rate, ground speed, and heading—within the limits of the engage gate.

The digital computer generates speed, roll, and vertical commands that are referenced to the ASE sensors. The commands are constrained by the maneuvering limits placed upon the system by requirements of flight safety and crew acceptability. These constraints include, but are not limited to:

1. Maximum attitude upsets
2. Minimum and maximum glide path angles
3. Power requirements
4. Power available
5. Limits following engine loss
6. Obstacle avoidance radar limitations.

Acceptable performance of a transition coupler is not covered by a general specification and hence must be developed for each helicopter. As a guideline, the following criteria probably can be met with available technology:

1. Hover position. Horizontal within two rotor diameters of the reference.

2. Vertical position. ± 2 ft with overshoot not more than 20%.

3. Final velocity. Less than ± 2 kt.

4. Attitude excursions. Not greater than ± 10 deg.

5. Power variations. Less than $\pm 50\%$ of the power required for level flight at all speeds.

The outbound transition also can be controlled automatically if desired. Because the end point is not as critical as for the inbound transition, the constraints upon position and final attitude are not as important. However, the power available and power required constraints become critical in achieving a successful outbound transition. Determining these parameters is difficult without flight testing. A power topping check can give an accurate estimate of power available, and power required can be extrapolated from the topping power required. However, the pilot probably would prefer to use handbook data and to apply manual control through the automatic coupler for an initial transition, and to let the computer store the necessary data for subsequent inbound and outbound approaches on a given mission.

6-4.4.2.3 Automatic Hover

The automatic hover mode may be a requirement for future Army helicopters, especially heavy-lift vehicles. As the helicopter reaches the hover position, downwash and recirculation of dust can obscure the horizon and the area adjacent to the helicopter completely; an automatic hover control system thus would be a desirable feature. Several successful systems based upon cable references have been developed and flown operationally; but the desirability of not having to deploy and retrieve any external device is apparent. However, few noncooperative sensors can produce position accuracies of even ±10 ft for periods of up to 10 min over all types of terrain and with the nearest targets varying from perhaps 100 to 1000 ft; and for a hover hold system, the sensor accuracy requirement is in the order of ±2 ft, with drift of 2 fpm or less.

With a suitable reference, the accurate positioning of the aircraft in a closed loop fashion is fairly straightforward. Displacement information from the sensor system and velocity data from either the navigation system or the pseudoinertial system (par. 6-4.4.1.6) are summed to form a coupler error signal. The coupler then commands longitudinal velocity via thrust variation or pitch attitude, and roll attitude and altitude via the ASE.

Pilot control of position must be provided in this mode for maneuvering. Due to the large variation of position maneuvering that might be required, a velocity command system similar to the velocity vector control of par. 6-4.4.1.6 or ground speed control of par. 6-4.4.2.2.1 probably is desirable. The position reference would be synchronized during the pilot overpower of the hover mode. Upon release of the control force, a new position reference would be established. During maneuvers, acceleration greater than 0.2 g seldom will be required in the automatic hover mode. Velocities of up to 30 kt might be demanded occasionally, but most of the maneuvers would be at 10 kt or less. The position overshoot from pilot release of a 30-kt command would be approximately 100 ft for a 0.2 g deceleration. Normal piloting techniques will reduce this magnitude considerably.

6-4.4.2.4 Automatic Stationkeeping

The development of automatic stationkeeping (ASK) has been an area of considerable interest with various Army aviation agencies for several years. Basically, the problem is similar to the automatic hover position problem (par. 6-4.4.2.3) but is compounded by a reference (the leader aircraft) that is moving in space. Sensing the relative motion between the leader of a group of helicopters and all of the followers is quite complex.

Several systems have been proposed for controlling the formation flight of the followers, and limited flight evaluations of equipment have been conducted. All of the systems to date have generated command positions for the follower aircraft based upon the range and angular relationships between follower and leader. Depending upon how the lead aircraft maneuvers, and how the formation is to follow the leader through the maneuvers, a number of system models have been developed. Continuing studies in these areas must be reviewed for each new helicopter mission to determine their applicability.

The mode of control within a follower aircraft probably will be by attitude- and altitude-rate commands. Damping of follower aircraft position will require three-axis velocity information from an onboard navigation system or some other source of inertial velocity information, as in the hover case. Command limits necessarily will be larger for this mode than for other coupler modes if the capabilities of the helicopter are to be realized fully. Bank attitudes of up to ±30 deg and pitch attitudes of ±10 deg are necesssary once the formation has been established. Thrust variations will be required in compound helicopters to allow accelerated changes of up to ±0.3 g along the longitudinal axis.

The rendezvous phase of the mission also may be achieved automatically, using the same limits at lower speeds to perform turning maneuvers in less time and distance. During rendezvous, each aircraft will be directed by the ASK system to a preassigned position placed into the computer by the pilot. Any change in command position following rendezvous will require the pilot to change his position input. The capability for position changes during flight will require a data exchange among all formation elements in the event the desired change interferes with the airspace allotted to another aircraft.

6-4.4.2.5 Automatic Terrain-following

Automatic terrain-following (ATF) for helicopters is another coupler mode that has been developed by the Army in its program of avionic research and development. The function also may be categorized as terrain avoidance. These systems presently are based upon range and angle inputs from forward-looking radar, radar altimeter, Doppler velocity, airspeed, or other inputs unique to the vehicle in which they are installed. Ranging techniques based upon laser technology may interface with the ASE in future designs.

Terrain-following can be accomplished at a varying speed and a fixed altitude with respect to the terrain or at varying altitudes and a fixed speed. The latter method maintains a constant clearance for peaks, but smooths the flight path to avoid the major terrain features.

The bases for terrain-avoidance or terrain-following are the ASE attitude and the radar altitude holds. Forward-looking radar data are processed by various means, and longitudinal maneuvers and/or power variations are commanded. If speed is allowed to vary, many of the required maneuvers can be accomplished by trading speed for climb angle. Thus, a combination of pitch attitude and normal acceleration is commanded by the ATF coupler; as speed is reduced, power may be added via collective or thrust commands to maintain a preselected minimum speed. In some power-limited helicopters, rotor speed sensing may be required to override the ATF system and prevent unsafe conditions during steep climbs.

As the aircraft reaches the top of a hill, the forward-looking radar no longer sees a close-range terrain feature and commands a decrease in climb angle. The radar altimeter prevents a pushover until the hill has been cleared.

The processing of the radar data is the most important aspect of the ATF system. The range-to-terrain features that will cause a climb must change as a function of ground speed and climb or descent angle. The angle from the aircraft to a feature that effects a command also is variable.

The ATF system probably places more demands upon ASE actuators than any other mode of flight. Large pitch changes in the short-period frequency range of the helicopter are necessary to control the flight path, as are power variations from flight idle to intermediate power. The other guidance modes, because of the limited maneuvering requirements, generally can be achieved with limited-authority series actuators and automatic trim follow-up. Automatic stationkeeping requires full-authority actuation for pitch and thrust control; this actuation has both higher rates and better frequency response characteristics than an automatic trim system requires. The safety requirements for such systems are discussed in par. 6-4.4.3.

6-4.4.3 Automatic Stabilization Equipment Failures

Failures of the ASE have the same requirements for allowable transients as SAS failure (par. 6-4.3.2). The detail configuration of the ASE actuators has a significant effect upon the magnitude of the transients.

6-4.4.3.1 Passive Failures

Passive failures of the ASE degrade the mission capabilities of the helicopter, but generally result in nothing worse than a slow divergence from the controlled attitude. The exceptions are failures in the basic pilot-assist modes when a guidance mode is engaged that is dependent upon the attitude or altitude reference of the ASE. In systems where the ASE is controlling the aircraft along a path, especially during approaches or in terrain-following or stationkeeping, redundancy of the critical portions of the ASE is required.

6-4.4.3.2 Hardover Failures

The effect of hardover failures of the electronics of the ASE is largely dependent upon the actuator configuration. Systems that use the SAS series actuators for ASE control should be limited electronically to preclude saturation of the actuators. The limits are a function of maneuvering requirements and SAS authority requirements to control the aircraft following the failure. If automatic trim followup of the series actuator position is used, the trim rate during ASE operation must be a compromise between system operation and failure characteristics.

Parallel position servo actuators are employed in ASE systems used in maneuvering modes requiring large authority. The disadvantage of this actuator method is that it usually is opposed by the SAS series actuator and thus must move through greater amplitudes. The advantage comes in pilot acceptance of the ASE system. Commanded control inputs are reflected to the cockpit controls so that the pilot can monitor the system performance during the maneuver, and may react before the safety of the helicopter is compromised. Because of the large authority (usually greater than 50% of full travel), the rate limits on these actuators must be selected to meet the failure criteria. In modes such as ATF and stationkeeping, the maneuvers quite frequently will exceed the failure limits in less than 3 sec. These modes and the pilot-assist modes must be provided in a redundant or monitored configuration, or deviations from the failure specifications must be allowed.

Overpowering of failed parallel actuator systems must be possible within the limits of the allowable control forces of MIL-H-8501 for hover and low-speed flight. The allowable overpower forces must not exceed the limits of MIL-F-8785 for helicopters at speeds above 150 kt or speeds at which control of the helicopter is achieved through aerodynamic surfaces other than the rotor. Further guidance in this area is included in MIL-F-83300.

A positive means for disconnecting a failed parallel actuator must be included in the system design. If possible, no controls other than the normal ASE controls should be necessary for actuator release following a failure; accordingly, the device must be readily accessible to the flight crew.

6-5 LIST OF SYMBOLS

A = feedback parameter, sec^{-1}
A_b = total blade area, ft^2
A_1 = cosine component of blade feathering with respect to the shaft, positive for down feathering at $\psi = 0$ deg, rad
= sine component of swashplate tilt with respect to a plane normal to the shaft, rad
A_0, A_1, A_2 = generalized coefficients of third order frequency equation
a = blade section lift curve slope, rad^{-1}
a_0 = coning angle, rad
a_z = vertical acceleration, positive down, ft/sec^2
$\bar{a}_z$ = response a_z to unit control step input, ft/sec^2
a_1 = cosine component of blade flapping with respect to a plane normal to the shaft, positive for down displacement at $\psi = 0$ deg, rad
a_{1_p} = pitch-roll coupling derivative, $\partial a_1/\partial p$, sec
a_{1_q} = damping in pitch, $\partial a_1/\partial q$, sec
a_{1_u} = speed stability derivative, $\partial a_1/\partial u$, rad/fps
a_{1_α} = static stability derivative, $\partial a_1/\partial \alpha$, dimensionless
a_{1_μ} = speed stability derivative, $\partial a_1/\partial \mu$, dimensionless
B = tip loss factor, dimensionless
B_1 = sine component of blade feathering with respect to the shaft, positive for down feathering at $\psi = 90$ deg, rad
= cosine component of swashplate tilt with respect to a plane normal to the shaft, rad
b = number of blades
b_1 = sine component of blade flapping with respect to a plane normal to the shaft, positive for down displacement at $\psi = 90$ deg, rad
b_{1_p} = damping in roll, $\partial b_1/\partial p$, sec
b_{1_q} = roll-pitch coupling derivative, $\partial b_1/\partial q$, sec
C_D = drag coefficient of fuselage section, dimensionless
CF = centrifugal force at blade flapping hinge, lb
C_T = rotor thrust coefficient, $T/[\pi R^2 \rho(\Omega R)^2]$, dimensionless
C_T/σ = rotor thrust-coefficient/solidity ratio, $T/[\rho bcR(\Omega R)^2]$, dimensionless
C_1 = pitch-flap coupling parameter, $-\tan \delta_3$, dimensionless
C_2 = linkage ratio, dimensionless
$C(\psi)$ = aerodynamic damping coefficient, dimensionless
c = blade chord, ft
c_x = damping in roll, lb-ft/(rad/sec)
c_y = damping in pitch, lb-ft/(rad/sec)
c_z = damping in yaw, lb-ft/(rad/sec)
c_1, c_2 = coefficients in equation for change of flapping with respect to speed, dimensionless
d = fuselage depth, ft
$E_1 \ldots E_5$ = coefficients in equation for change of trim angle of attack with respect to speed
e = flapping hinge offset, ft
F_i = inertial force due to flapping acceleration, lb
f = speed ratio, dimensionless
$G(s)$ = system transfer function
$\bar{G}$ = gyro specific damping, damping/critical damping, dimensionless
g = acceleration due to gravity, ft/sec^2
$H(s)$ = feedback transfer function
h = height of rotor hub above CG, ft
h_{TR} = height of tail rotor above CG, ft
I_b = blade flapwise mass moment of inertia, slug-ft^2
I_x = moment of inertia in roll, slug-ft^2
I_y = moment of inertia in pitch, slug-ft^2
I_z = moment of inertia in yaw, slug-ft^2
K = feedback gain, in.-sec
= specific damping of blade flapping motion, damping/critical damping
K_0 = proportional feedback gain

K_1 = integral feedback gain, in.
$K(\psi)$ = aerodynamic spring coefficient, dimensionless
k_1 = attitude gain of stabilization system, dimensionless
k_2 = rate gain of stabilization system, dimensionless
k_β = flapping stiffness of roto blade, lb-ft/rad
L = rolling moment about CG, positive down to right, lb-ft
L_H = rolling moment at hub produced by blade flapping, positive down to right, lb-ft
M = pitching moment about CG, positive nose-up, lb-ft
M_H = pitching moment of rotor on hub, positive nose-up, lb-ft
M_A = pitching moment produced by aerodynamic forces on fuselage, wing and empennage, lb-ft
M_g = gyro moment, lb-ft
M_u = speed stability derivative, $(\partial M/\partial u)/I_y$, (sec-ft)$^{-1}$
M_α = static stability derivative, negative for stable behavior, $(\partial M/\partial \alpha)/I_y$, sec^{-2}
$M_{\dot\theta}$ = pitch damping derivative, negative for damped motion, $(\partial M/\partial \dot\theta)/I_y$, sec^{-1}
$M_{\delta_{sx}}$ = control power, $(\partial M/\partial \delta_{sx})/I_y$, sec^{-2}/in.
m = blade mass distribution, slug/ft
N = yawing moment, positive clockwise from top, lb-ft
P = flapping frequency parameter, flapping frequency/Ω, dimensionless
P = nominal flapping frequency parameter, dimensionless
p = rolling velocity about rotor center, rad/sec
q = pitching velocity about rotor center, rad/sec
= dynamic pressure, $\rho V^2/2$, lb/ft^2
R = rotor radius, ft
RF = response factor, dimensionless
r = incremental radius, ft
r_g = aircraft pitching radius of gyration, ft
s = Laplace operator, sec^{-1}
T = rotor thrust, lb
T_D = time required to double amplitude, sec
T_H = time required to half amplitude, sec
T_{TR} = tail rotor thrust, lb
T_0 = period of oscillation, sec
t = time, sec
U = velocity along helicopter longitudinal body axis, fps
u = horizontal velocity of CG of aircraft, positive forward, fps
V = speed, fps
V_s = lateral velocity at main rotor, fps
V_{TR} = lateral velocity at tail rotor, fps
V_v = vertical velocity at rotor, fps
v = lateral velocity of CG of the aircraft, positive to right, fps
W = helicopter weight, lb
w = vertical velocity of CG of the aircraft, positive down, fps
x = horizontal distance from CG, ft
$\bar{x}$ = horizontal location of CG ahead of shaft, ft
x_R = horizontal distance between rotors of a tandem-rotor helicopter, ft
x_{TR} = horizontal distance between tail rotor and CG, ft
Z = vertical force on aircraft, positive down, lb
Z_w = vertical damping derivative, negative for damped motion, sec^{-1}
Z_δ = vertical control sensitivity, (ft/sec^2)/in.
α = angle of attack, rad or deg
α_0 = amplitude of pitching oscillation, rad
β = flapping angle, rad
β_0 = rotor coning angle, rad or deg
β_s = angle of sideslip, rad
β_1 = forward tilt angle of rotor tip path plane, rad or deg
β_2 = left tilt angle of rotor tip path plane, rad or deg
γ = Lock number, $R^4 ca\rho/I_b$, dimensionless
γ_c = climb angle, rad
Δ = phase angle, rad
= incremental
δ = angular displacement of gyro, rad

δ_c = amplitude of cos ψ component of gyro displacement, rad
δ_F = feedback input subtracting from control input, in.
δ_p = pedal displacement, in.
δ_s = amplitude of sin ψ component of gyro displacement, rad
δ_{sx} = longitudinal displacement of control stick, positive aft, in.
δ_{sy} = lateral displacement of control stick, positive to right, in.
δ_T = power control setting, rad
δ_3 = angle of inclination of flapping hinge, rad or deg
ϵ = a constant in generalized coefficients A_1, A_2, and A_3, sec^{-2}
ζ = damping ratio, dimensionless
θ = pitch attitude, positive nose-up, rad
$\bar{\theta}$ = response θ to unit control step input, rad
θ_s = cyclic pitch input of stabilization system, rad
θ_{TR} = tail rotor pitch, rad
θ_0 = main rotor collective pitch, rad
θ_1 = forward cyclic pitch angle, rad
θ_2 = left cyclic pitch angle, rad
$\theta_{1.0}$ = displacement in pitch 1.0 sec following control displacement
λ = inflow ratio, positive for upflow through rotor, dimensionless
κ = free body gust alleviation factor, dimensionless
μ = advance ratio, $U/(\Omega R)$, dimensionless
ν = frequency of pitching oscillation, rad/sec
$\bar{\nu}$ = reduced frequency of pitching oscillation, ν/Ω, dimensionless
ρ = density of air, slug/ft^3
σ = rotor solidity, $bc/(\pi R)$, dimensionless
τ_F = feedback time constant, sec
τ_k = pitch time constant with feedback, sec
τ_w = vertical time constant without augmentation, sec
$\bar{\tau}$ = reduced time, $\Omega\tau$, dimensionless
τ_θ = pitch time constant, sec
ϕ = roll attitude, positive down to right, rad
$\phi_{0.5}$ = displacement in roll 0.5 sec following control displacement
ψ = yaw attitude, positive clockwise from top, rad
= azimuth position of blade, rad or deg
$\psi_{1.0}$ = displacement in yaw 1 sec following control displacement
Ω = rotor angular velocity, rad/sec
ω = angular velocity of gyro, rad/sec
= frequency rad/sec
$\bar{\omega}$ = reduced frequency, ω/Ω, dimensionless
ω_n = blade flapwise natural frequency, rad/sec

Subscripts

A = airframe
eff = effective
i = ith blade
0 = initial condition

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CHAPTER 7

DRIVE SYSTEM DESIGN

7-1 INTRODUCTION

The primary purpose of a helicopter drive system is to transmit power from the engine(s) to the lifting rotor(s) and to the antitorque rotor, if one is provided. Power takeoff from the main drive is used to power the accessories. The basic transmission elements required to accomplish these tasks depend upon the aircraft rotor configuration (single rotor, tandem rotors, coaxial rotors, etc.) and also upon the location and orientation of the engine(s) with respect to the rotor. In general, the largest reduction is taken in a main gearbox, whose output drives the main rotor. Supplementary gearboxes may be used where necessary to change the direction of the drive and speed. Reductions can be accomplished in these as well. Special-purpose gearboxes may be included in the drive system; e.g., the tail gearbox that drives the antitorque rotor in a single-rotor machine, and the intermediate and combining gearboxes necessary to provide a synchronizing link between the main rotors of a multirotor machine.

7-1.1 TRANSMISSION SYSTEMS

Initial considerations that influence the design of a transmission drive system include the missions for which the helicopter is to be used. Such missions might include search and rescue, heavy lift, attack, transport, observation, or any combination of these. Helicopter performance requirements that affect the design of the power train include payload, power requirements of various mission segments, hovering capability, noise level, operational environment, system reliability, and mission altitude. These mission requirements would be converted into specific design requirements such as engine power and speed versus rotor rpm, design life of transmission components, and individual component reliability.

The design of the transmission system similarly is governed by the configuration of the helicopter itself. In the single-rotor machine it is possible to refer to a typical configuration (Fig. 7-1), as the main variation is in the location and orientation of the engine(s).

Multirotor transmission configurations are more varied. The rotors can be placed in tandem or side by side, having a separate engine for each rotor and incorporating interconnecting shafting to synchronize rotor speeds and provide power to both rotors in case of an engine malfunction. The two rotors also can be arranged coaxially and made counterrotating to balance rotor torques.

The loads that the transmission system components must withstand are a function of power and speed ($T \sim$ hp/rpm). Required engine power can be calculated based upon the maximum performance requirements of the mission, e.g., in hover at 4000 ft (out-of-ground-effect) at 95°F (hot-day performance). The input rpm is fixed by the output speed of the engine while the rotor speed usually is determined by the tip speed of the rotor blades. Thus, the overall transmission ratio can be obtained readily if rotor diameters are known. Splitting this ratio among the various transmission elements (bevel gear set, epicyclic, offset spur set, etc.) to obtain the minimum-weight design can be accomplished by preliminary design layout iterations. In general, however, it is better to take the largest reduction in the final stage. Trade-off studies should be made to evaluate different arrangements to determine minimum weight design. These studies should include housing design and should be sufficiently extensive to provide data for plotting a graph of weight versus gear ratio distribution.

The design loads for the drive system components are derived from a load-time spectrum that represents the helicopter mission. This spectrum of operation is similar to, but not necessarily the same as, that used for the structural reliability and fatigue analysis. It has become evident that the simple design information of takeoff, hover, and cruise power and speed requirements no longer can be considered as sufficient description of the design parameters for the drive system. To achieve maximum reliability at minimum weight, the design must be based upon complete aerodynamic analysis of the helicopter mission including complete spectra of maneuver and transient loads and rotor moments and loads as well as the normal power conditions. In addition, the increasing requirements of accessory drives must be considered. These accessories in some helicopter systems must be operated on the ground with the rotors stationary, requiring rotor braking systems and accessory clutching systems.

To judge the reliability of a transmission design, the loads and resulting stresses ultimately must be compared with historical data for similar components. Basic mean load-life relationships are available for most transmission elements. Many of these have become standardized throughout the industry and can be used with confidence. Others may be peculiar to a particular company and have become standard through general usage.

Most of the transmission elements can be broken down into the following main categories: gears, freewheel units (clutches), shafting, bearings, housing, and lubrication systems.

7-1.1.1 Gears

High-performance gears are case hardened and ground with a surface finish of 20 rms or better. Gearing usually is designed for unlimited life with 0.999 reliability or better at the maximum power (other than instantaneous transients) transmitted by that mesh. Primary drive gears should be made from consumable electrode vacuum melt (CEVM) processed steel, which is less susceptible to fatigue failure than is air-processed steel.

One of the primary causes of premature gear failures can be traced to high load concentrations induced by flexible mounting, especially when the housings are made of lightweight, low-modulus materials such as magnesium or aluminum. Experience indicates that, wherever possible, all heavily loaded gears should be straddle-mounted to minimize deflections and prevent end loading.

7-1.1.2 Clutches

Clutches must be provided between engine(s) and rotor system. To meet this requirement, a freewheel unit is located in the drive train between the main rotor and engine so that, if an engine malfunctions, the unit will overrun and allow autorotation or, for multiengine helicopters, will allow operation on the remaining engine(s). Generally the freewheel unit is located at the high speed side of the gearbox because location at the low-speed/high-torque end would result in an unnecessarily heavy design. Several basic types of designs are available and the various types must be compared to determine the best type to use for reliability, wear, weight, overrunning capability, and cost. For high horsepower applications the sprag and the cam-roller types of freewheel units have been used successfully with high reliability. If the freewheel unit is expected to overrun for long periods of time (as would be the case if the accessories were driven by an auxiliary device), the most important design consideration would be wear. To solve wear problems, freewheel unit members have been flameplated with tungsten carbide on the housing bore, and have utilized pressurized lubrication to insure a high oil flow rate during overrunning.

7-1.1.3 Transmission Shafting

Transmission shafting usually is hollow with as high a diameter-to-thickness ratio as is practicable for minimum weight. These shafts are subjected to torsional loads, bending loads, axial tension or compression, or to a combination of all of these. Because the shaft is rotating with respect to the bending loads, this loading

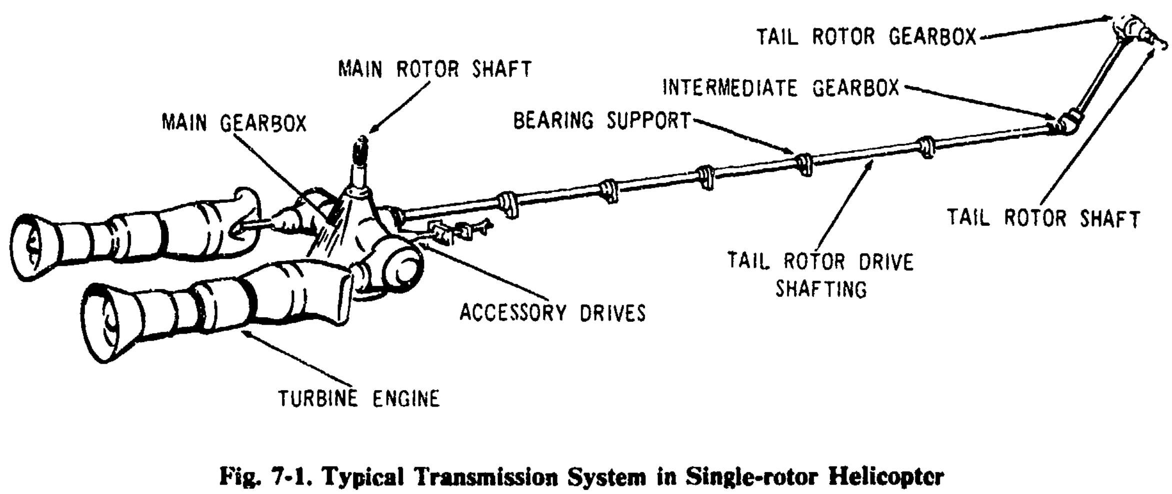


Fig. 7-1. Typical Transmission System in Single-rotor Helicopter

is of a vibratory nature. Due to this combination of steady and vibratory loads, an interaction equation must be used to calculate a margin of safety. Such an equation based upon the maximum shear theory of failure can be used when all three types of stresses are present. The resulting margin of safety *MS* is determined by:

$$MS = \frac{1}{\sqrt{\left(\frac{f_b}{F_{en}} + \frac{f_a}{F_{ty}}\right)^2 + 4\left(\frac{f_s}{F_{sy}}\right)^2}} \qquad (7\text{-}1)$$

where

f_a = axial tension stress, psi
f_b = vibratory bending stress, psi
f_s = torsional shear stress, psi
F_{en} = endurance limit stress, psi
F_{sy} = shear yield stress, psi
F_{ty} = tensile yield stress, psi

In this relation, the vibratory bending stress is compared with the allowable or endurance limit, while the axial and torsional stresses are compared with their respective yield values. This method of computing margins of safety is considered conservative.

Gear shafting usually is designed for unlimited life at a power level and reliability commensurate with the geartooth design. Engine drive and tail rotor drive shafting carry torsional loads primarily, although some bending may be induced by semiflexible couplings spaced along such shafts to accommodate misalignment.

7-1.1.4 Bearings and Housings

The bearings for the power train are selected or designed for overhaul intervals of at least 3000 hr. All critical bearings are made from M-50 type steel made by the consumable electrode vacuum remelt process (AMS 6490), SAE 52100 steel consumable electrode vacuum melted (AMS 6444), or SAE 52100 steel, bearing quality (AMS 6440) to obtain maximum reliability. In high speed applications, bearing life is a function of the centrifugal force imposed upon the bearing rotating elements as well as of the radial and thrust load. Where a stack of bearings is required to support a gear shaft, distribution among the individual bearings must be considered. The selection of high speed bearings often involves a complex computer solution that considers the effects of load and speed as well as of minute changes of internal bearing geometry, i.e., contact angle and radial-axial clearances.

Bearings usually are assembled with separate inner and outer races, although in some cases it may be economical to use the shaft as the inner or outer race. The advisability of using this scheme would depend upon the size, complexity, and resultant cost of the components involved. Integral races are being used extensively in advanced design applications.

With nonferrous housings, pressed-in steel liners are used between the bearing outer race and the housing, and serve many purposes. These liners are installed into the housing with an interference fit and tend to minimize the effect of thermal expansion and contraction upon the bearing. In addition, they act as a buffer between the housing and the bearing to prevent damage to the housing whenever a bearing is installed or removed, and also reduce wear and prevent creep. Liners are designed so that at a maximum expected operating temperature of, for example, 250°F, the liner has a minimum of 0.0005 in. interference fit with the housing. Housing and liner stresses then are checked at the expected minimum operating temperature of −65°F. In accordance with MIL-T-5955, bearing liners (or sleeves) *shall* be restrained from rotation by a positive locking means.

7-1.1.5 Lubrication

Consideration *shall* be given to integration of the lubrication subsystem into the gearbox. Any lubrication subsystem components external to the gearbox *shall* be eliminated to the maximum extent possible. Careful consideration must be given to the distribution of lubricant to and from the gears and bearings. The primary purpose of the lubrication system is to provide cooling oil to remove heat generated due to friction at gear meshes and bearings, and also to provide lubricity between the sliding elements. Pressurized lubrication systems are used in high speed or high load applications to provide lubrication for the gears and bearings, although splash type lubrication systems have been used successfully in gearboxes with low pitch line velocities. Generally, oil jets directed "into mesh" are used to lubricate, while "out-of-mesh" jets are used for cooling.

Consideration must be given during preliminary design to an adequate return path for lubricating oil. A restricted return often results in oil pile-up, which prevents proper entry of cool oil and, consequently, causes overheating.

Another requirement that must be considered is the capability for continuing operation for a minimum of 30 minutes following total loss of the lubrication system. Drive system load for this condition can be assumed to be the minimum power required for level flight at sea level at design, i.e., primary mission, gross weight. Lubrication system redundancy is desirable; however, operation for 30 minutes following total loss of lubrication *shall* be possible.

Lubrication systems incorporating either vane or gerotor type lubrication pumps have proven to be much less susceptible to contamination damage than those using conventional gear pumps. A typical main gearbox lubrication system is shown schematically in Fig. 7-2. This system has low pressure screens at the pump inlet and 40-micron filters between the pump and the cooler. The cooler is provided with a thermostatic and pressure bypass valve.

The heat losses incurred in a gearbox are usually in the order of 0.5% per gear mesh. Overall efficiency is, therefore, from 98 to 99% depending upon the size and complexity of the gearbox. While this efficiency seems high, it means that a sizable amount of power must be dissipated, especially in the larger helicopters where the installed power can be as much as 9000 hp or higher.

7-1.1.6 Accessories

During preliminary design, a review is made to determine the best arrangement for accessories and the engine starting requirements that influence the overall gearbox design. For example, on some helicopters the accessories are driven by the main engine or some auxiliary device (APU or electric motor) while the main rotor remains stationary. Hence, the required startup procedure may dictate the location of additional free-wheel units and the design of the accessory drive. Accessories can be mounted directly on the main gearbox or on a separate gearbox. The separate accessory gearbox is easier to manufacture and results in a "cleaner" main gearbox, but, with this design, at least one generator and one hydraulic pump should be mounted on and driven by the main gearbox for redundancy so that loss of the accessory drive shaft would preclude total loss of hydraulic power and control of the helicopter itself. The separately mounted accessory gearbox generally is superior for larger helicopters while, for smaller machines, the accessory gearbox integral with the main transmission is more practicable.

7-1.2 ROTOR BRAKES

The rotor brake is used to hold the rotor stationary during some ground operations with the engines idling, and also to stop the rotor from full or partial rpm within a specified time with the engines in ground idle

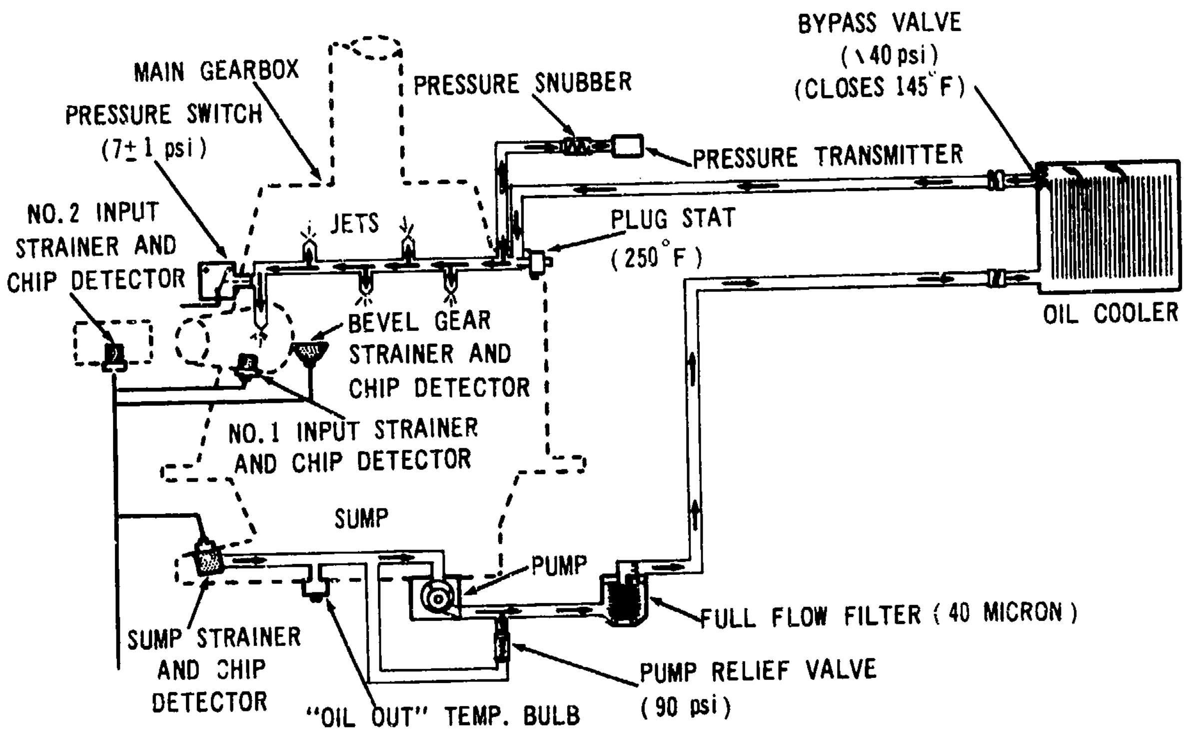


Fig. 7-2. Typical Main Gearbox Lubrication System

or shut down. The total stopping time can be proportioned between a delay in which the rotor rpm decays naturally and the actual braking time. On large helicopters the rotor brake is a hydraulically actuated disk of variable displacement in which lining wear is compensated for by increased volume of operating fluid. In addition to developing sufficient torque to stop and hold the rotor, the brake also must be capable of absorbing and dissipating the kinetic energy of the rotors and other rotating parts. Precautions must be taken to isolate flammable fluids or materials from the vicinity of the rotor brake because temperatures far above ignition temperatures often are reached during braking.

7-2 SYSTEM SELECTION

7-2.1 GENERAL

In addition to gears, other transmission system components include clutches, brakes, and freewheel units. Clutches and freewheel units are used to disconnect power from the engines in the reverse torque direction.

There also are cyclocentric and harmonic drives, which may allow large reductions in a compact space but which also may reduce efficiency over comparable standard gear systems.

Before the components comprising the transmission system can be designed, the transmission designer requires certain basic data. Preliminary design of the transmission commences with the initial design of the vehicle itself. The helicopter designer, by analyzing mission requirements, can establish a basic configuration including type and approximate size and maximum weight of the helicopter; rotor diameters; power plant type, size, and configuration; location of rotors; and rotor speeds and diameters. The maximum available power is known, as are speeds and directions of rotation of input drive shafting from engines. Thus, several transmission system design parameters, limitations, and requirements are established before the preliminary design is begun.

It is during this initial design phase that alternate design solutions are proposed and evaluated on the basis of weight, efficiency, reliability, cost, maintainability, and service life.

7-2.2 BASIC GEARED TRANSMISSION TYPES

Conventional geared transmission systems use several different types of driving gears to achieve the intended function. Often parallel axis gearing is included, requiring the use of spur, helical, or planetary gearing to transmit torque. When intersecting axes are required, bevel gears are used, and when nonintersecting, nonparallel axes exist, crossed helicals are employed. In most helicopter applications, however, such angles usually are small, because the efficiency of crossed helical gears decreases rapidly as the helix angle increases. Crossed helical gearing, if used, normally would be in accessory drive applications. The aforementioned types of gearing are discussed in the paragraphs that follow.

7-2.2.1 Spur Gears

The most commonly used and easily manufactured type of gear in helicopter transmissions is the involute spur gear. The spur gear is used to transmit torque and to change speed between parallel axis shafts. The involute gear tooth profile maintains a constant torque output even though the individual tooth loads are constantly changing in magnitude and location on the tooth face. The involute form frequently is modified a small amount to allow for tooth errors and deflections. For power transmission gearing the pressure angle of spur gears is generally between 20 and 25 deg, although in some applications 27.5 deg or more has been used successfully. The higher pressure angles tend to be stronger in bending, but are a little more critical with respect to scoring because of higher sliding velocities. Higher pressure angles also result in higher bearing reactions. Usually helicopter spur gear teeth are case-carburized and ground to highly precise tolerances resulting in smooth contact, small dynamic loads, and high strength teeth.

In practice, the name spur is given only to gears in which the teeth are straight and parallel to the axis of the gear. Spur gears are the types of gears most frequently used to connect parallel shafts because they do not develop axial loads or thrust as is the case with helical gears. Fig. 7-3 shows the definition of spur gear elements, the geometric relationship of the teeth with respect to the axis, and also the spur gear rack, which is the basis for generation of the system.

External spur gear ratios generally range from 1:1 to 10:1, while helical and double-helical ratios of up to about 15:1 are practicable. Internal spur gear ratios are practicable from 1.5:1 to 10:1 and helical ratios from about 2:1 to 15:1. The higher ratios generally are not used in helicopter power transmission gearing because the size of the driven member becomes too large, or the pinion too small.

The stresses in spur gear teeth are induced by the gear loading. Because the mating teeth are parallel to the gear axis, no axial load is produced. A gear design must be checked for tooth bending and compressive stresses to determine the limiting design condition. When designing a spur gear set, the choice of the number of teeth and pitch usually is left to the gear designer. Smaller teeth will mesh more smoothly and quietly than larger teeth. Provision of hunting teeth is also an important consideration. A hunting tooth is defined as a tooth on the pinion that will not mate with the same spaces on the gear on each revolution. Therefore manufacturing irregularities are not occurring on each revolution in exactly the same spaces, which would result in uneven wear and premature failure of the gear set. A hunting tooth must make many revolutions before mating with the same space again.

There also are degrees of hunting. For example, a 30-tooth pinion mating with a 59-tooth gear will have complete hunting capability and must make 59 revolutions before a given pinion tooth has mated with every available gear space. If the same pinion were mating with a 60-tooth gear, the pinion would have to revolve only twice to mate with the same space again and each pinion tooth would mate with only two gear spaces for the entire life of the gear set. If the 30-tooth pinion mated with a 58-tooth gear, the pinion would have to revolve 29 times before mating again with the same space. Even though the latter case does not provide complete hunting, it is much more desirable than the 2:1 ratio.

The number of pinion revolutions between meshes of a given tooth in the pinion with a given space in the gear is the lowest value of i that satisfies the equation

$$i\,(N_p/N_g) = \text{whole number}$$

where N_p and N_g are the pinion and gear rpm, respectively, and i is an integral number of pinion revolutions, $1 \leqslant i \leqslant N_g$. If i is equal to N_g, perfect hunting exists between the two mating gears.

Another important design consideration is the relationship between bending and compressive stress. In a good design both stresses should be close to the maximum allowable stress. A low bending stress results if too coarse a pitch is selected, and the resulting gears will be unnecessarily noisy. A low compressive stress results if the pitch is too fine, such that bending stress dictates the face width. In this case the gears will be unnecessarily heavy.

The following procedure gives the lightest-weight gear set for a given reduction ratio and pinion and gear diameters:

1. Determine the gear loads for the desired gear diameters and reduction ratio.

2. Solve for the required face width based upon the allowable compressive stress.

3. Select a pitch for the gears such that the bending stress is as close as possible to the allowable bending stress without exceeding it.

The efficiency of precision spur gears usually is 99.5% per mesh.

Precision spur gears, carefully designed, have been successfully used at pitch line velocities over 20,000

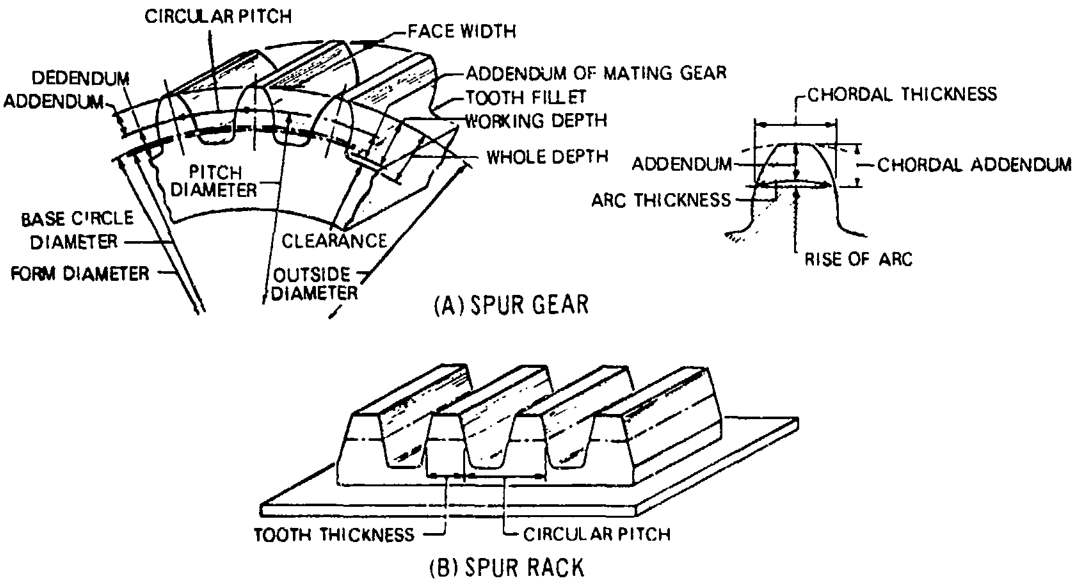


Fig. 7-3. Spur Gear and Rack

fpm. Because many potential spur gear applications are for epicyclic reductions, it is important to remember that the relative pitch line velocity between the mating gears is the important criterion, not the actual velocity. For an epicyclic reduction with a fixed internal gear, the sun or center gear, which is the input for a speed reducer, and the arm or carrier travel in the same direction. Thus, the relative pitch line velocity is a function of the difference in rotational speed.

7-2.2.2 Helical Gears

Helical gears are parallel axis shaft gears with helix angles. In fact, spur gears are a special type of helical gear with a 0-degree helix angle. The helix angle induces axial loads upon shafts. In the axial plane, the teeth of a helical gear are inclined to the axis at some angle greater than 0 deg. For power gearing, the helices of helical gears usually vary anywhere from 5 deg to a maximum of about 35 deg. The terminology of the helical gear and rack elements is shown in Fig. 7-4. For the same face width, helical gears have more load-carrying capacity than spur gears of equal size, are quieter, and have approximately the same efficiency. The overall design is not necessarily lighter, however, because the effect of thrust upon the mounting bearings must be considered. Double helical or herringbone gears eliminate the problem of thrust but are difficult to manufacture and usually "hunt" (chucking back and forth) due to manufacturing errors and tolerances. Sometimes it is advantageous to mount a helical gear on a shaft with another gear such as a bevel gear, such that the thrust loads of the two gears tend to cancel each other.

The form of the working surfaces of helical [illegible] viewed in the transverse plane, or plane of rotation, [illegible] usually an involute curve. Just as [illegible] spur gears, the form generally is modified. The tra[illegible]se pressure angle is usually in the range between 20 and 25 deg, although the angle may be as low as 15 deg and possibly as high as 27.5 or 28 deg. Helix angles can be almost any value, but angles between 7 and 45 deg are most common. Lower helix angles frequently are used for single-helical applications because of the lower thrust reaction, but there is no similar restriction for double-helical gears. The helix angle usually is selected so that two or more teeth always are in contact.

Accepted design practice usually limits face-width-to-diameter ratio to one or less for helicopter applications. Helicopter gearboxes generally utilize magnesium or aluminum housings, which are subject to deflections under load. The resulting shaft slopes and deflections on large face-width-to-diameter ratio gears would cause end loading of teeth and premature tooth failures.

In general, helical gears designed for helicopter use do not offer a tremendous advantage over spur gears of the same size. Shaft and bearing weight should be included when determining whether spur or helical gears offer the best design solution for a particular helicopter application.

In certain cases double helical gears are used to connect parallel shafts. These d[illegible] helical gears are termed herringbone gears and [illegible] on each of two portions of the blank is of o[illegible] hand. The two portions of the blank can be of an integral design or they may be of the type where one-half of the herringbone gear is assembled mechanically onto the opposite hand helix. The helix angle of the two portions usually, but not necessarily, is equal and always is of the opposite hand. By the use of herringbone gears, the advantages of helical gears are achieved, i.e., quieter running plus added strength in comparison to spur gears; but without the disadvantage of the thrust loads that are developed when helical gears are used. All gear types discussed for the parallel axis system of gearing—spur, helical, double helical, or herringbone—may be of either external or internal design.

7-2.2.3 Bevel Gears

Bevel gears are used to accomplish an angular change between two intersecting axes. For all types of cylindrical gears connecting parallel shafts, there is the analogous type used to connect shafts with intersecting axes. The simplest and most commonly used gear until about 20 years ago was the straight bevel gear. The teeth of a straight bevel gear are radial at the point of intersection of the axis and are proportional in size to [illegible] distance from the apex. All the tooth elements con[illegible] at the apex. As in spur gears, there is a line contact between the two mating straight bevel gear teeth and, because of the shaft angle, end thrust is developed and always acts in a direction away from the apex. Although the shaft angle for straight bevel gears theoretically can vary between 0 and 180 deg, shaft angles less than 10 deg seldom are used because of machine limitations. When the shaft angle and ratio between the pair are such that one member results in a pitch angle greater than 90 deg, difficulty may be encountered in machining the straight tooth bevel gear. Gears with approximately 115-deg pitch angle are practicable and can be cut. Fig. 7-5 (left side) shows the terminology that commonly is used to define straight bevel gears and their elements, with the crown gear used as a basis for definition.

Compared with straight bevel gears, which are less expensive to manufacture but are acceptable only for applications in which the pitch line velocity is less than 1000 fpm, spiral bevel gears offer increased load-carrying capability, are quieter and smoother running, and can be operated at pitch line velocities up to or beyond 30,000 fpm. Spiral bevel gears have pitch line contact that spreads under load, forming a contact area. Because contact stresses are proportional to area of contact, the spiral bevel gear can be smaller for the same contact stresses. Also, because bending stress is proportional to tooth thickness, coarser pitch (larger teeth) gears may be used to obtain higher strength in a small space. The efficiency of spiral bevel gears is comparable to that of spur gears and usually is in the order of 99.5% per mesh. Fig. 7-5 (right side) shows the spiral bevel gear and its elements in relationship to a crown gear used as a basis for definition.

Spiral bevel gears must be mounted rigidly with their axes maintained in correct alignment. Although bevel gears have the ability to absorb reasonable displacements without detriment to the tooth action, excessive misalignments reduce the load capacity and complicate the manufacture. Proper mounting is obtained by shimming to theoretical dimensions and checking the resulting load pattern. The load pattern may be changed to any desired form by changing the grinding machine settings slightly. During production all gears are matched against a "master gear". The ground spiral bevel gear thus is suited to a helicopter gearbox because corrections can be made for the deflections and slopes that actually occur under full load on the production housings, whereas straight bevel gears normally are not ground.

Because spiral bevel gears have curvature in three directions, three-dimensional loads are induced. By proper choice of pressure angle, spiral angle, direction of rotation, and hand of spiral the radial and axial loads can be varied to suit the needs of the particular bevel gear application. It always is advantageous (but sometimes impossible) to design the flange of a spiral bevel gear so that the resultant axial and radial gear loads pass through the centroid of the flange. This eliminates flange bending and contributes to the rigidity of the mounting. If possible, the gear and pinion should be straddle-mounted to provide better load distribution. For integral gear and shaft application, it is necessary that the shaft not extend above the root angle cone to permit gear manufacture and grinding. This often creates the need for a separate gear bolted to its supporting

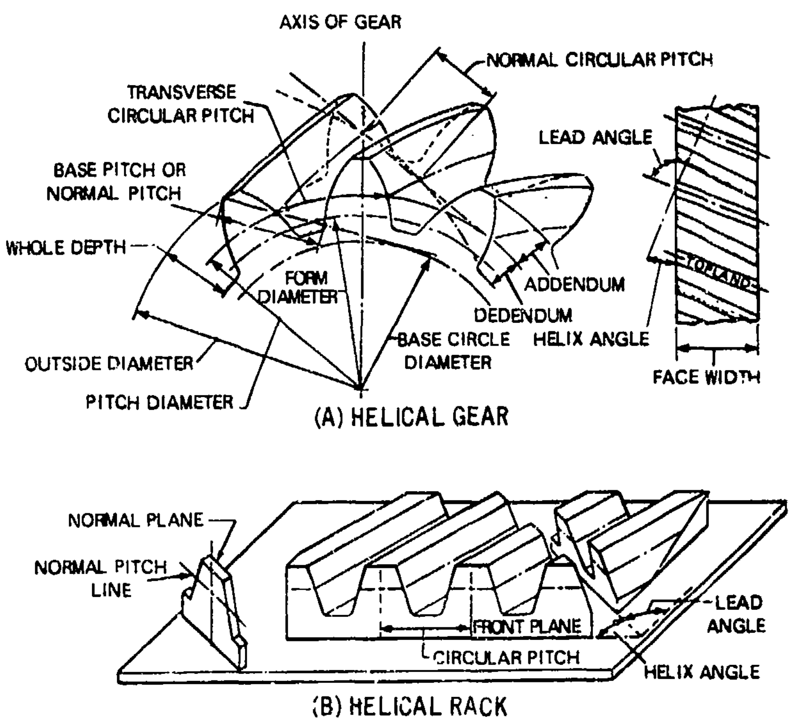


Fig. 7-4. Helical Gear and Rack Terminology

shaft. Such a gear usually is piloted on its shaft and connected by high-strength bolts. It is desirable to design this connection so that the torque of the gear is carried by friction created by the bolts. If this is possible, it eliminates the need for close-tolerance bolts and holes, and reduces the cost of the gear and shaft assembly.

Because the spiral bevel gear must be restrained axially and provisions made for shimming, each member (pinion and gear) usually is designed with the restraining bearings in a separate housing. The thrust bearings may be tapered roller bearings or angular contact ball bearings containing two or more bearings in a stacked arrangement.

As in spur and helical gears, spiral bevel gears are designed to allowable bending and compressive stresses. The disparity between allowable bending and compressive stresses for spiral bevel gears when compared to spur gears is the result of different factors used in the method of calculation and not of any inherent design condition. A factor important to spiral bevel gears is the scoring factor. At high speed, scoring may dictate the design of the gear set.

The bevel gear that is the counterpart of the herringbone gear in the parallel axis of gearing is the so-called Zerol bevel gear. The gear teeth of a Zerol bevel gear have lengthwise curvature, as do spiral teeth. However, the inclination of the teeth with respect to the axis of the part is zero. As a result of the zero tooth inclination, the only thrust loads developed with Zerol bevel gears are due to the conical blank and elements. These loads correspond in magnitude to those of straight bevel gears. These gears were developed primarily for those applications for which there was a preference for straight bevel gears, with the added requirement that the gears be ground. The use of Zerol bevel gears was made possible because the gear teeth could be ground on equipment that is used for the grinding of spiral bevel gear teeth. Fig. 7-6 shows a Zerol bevel gear with its elements and the relationship to the crown gear used

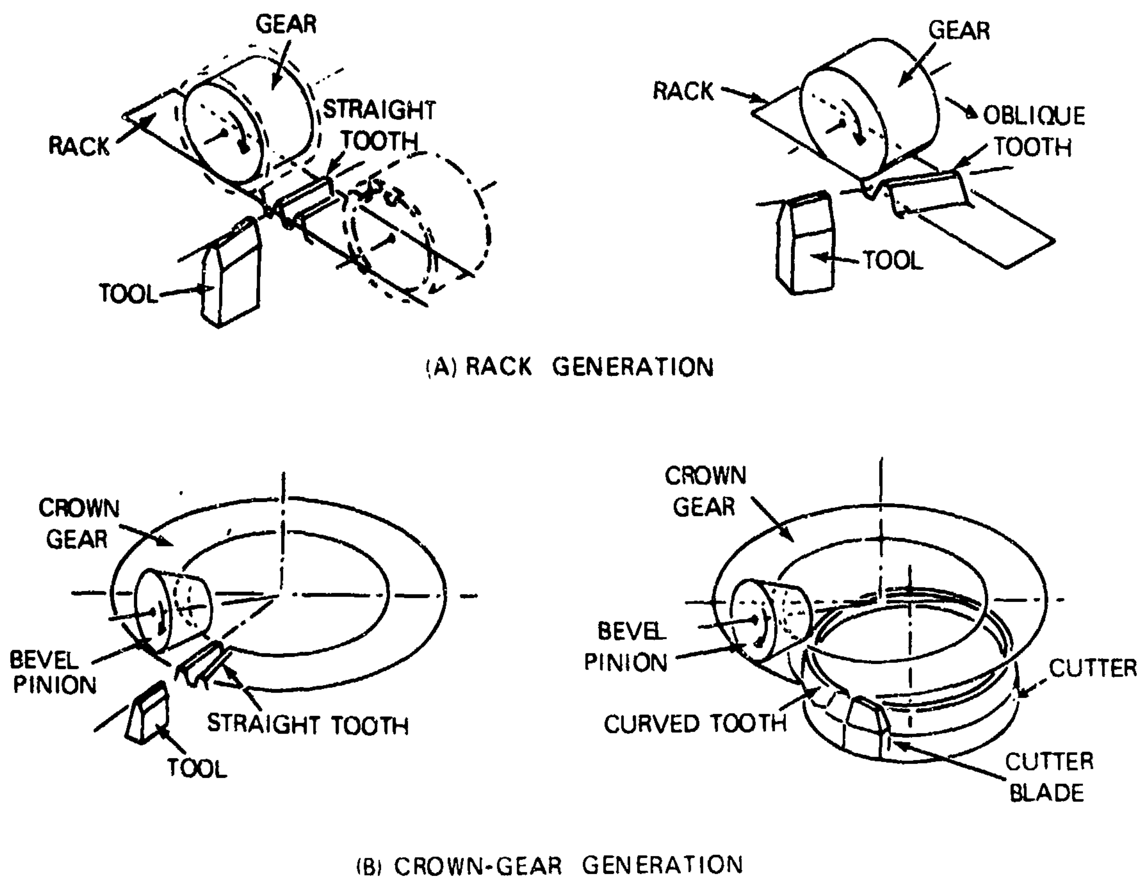


Fig. 7-5. Bevel Gear and Rack

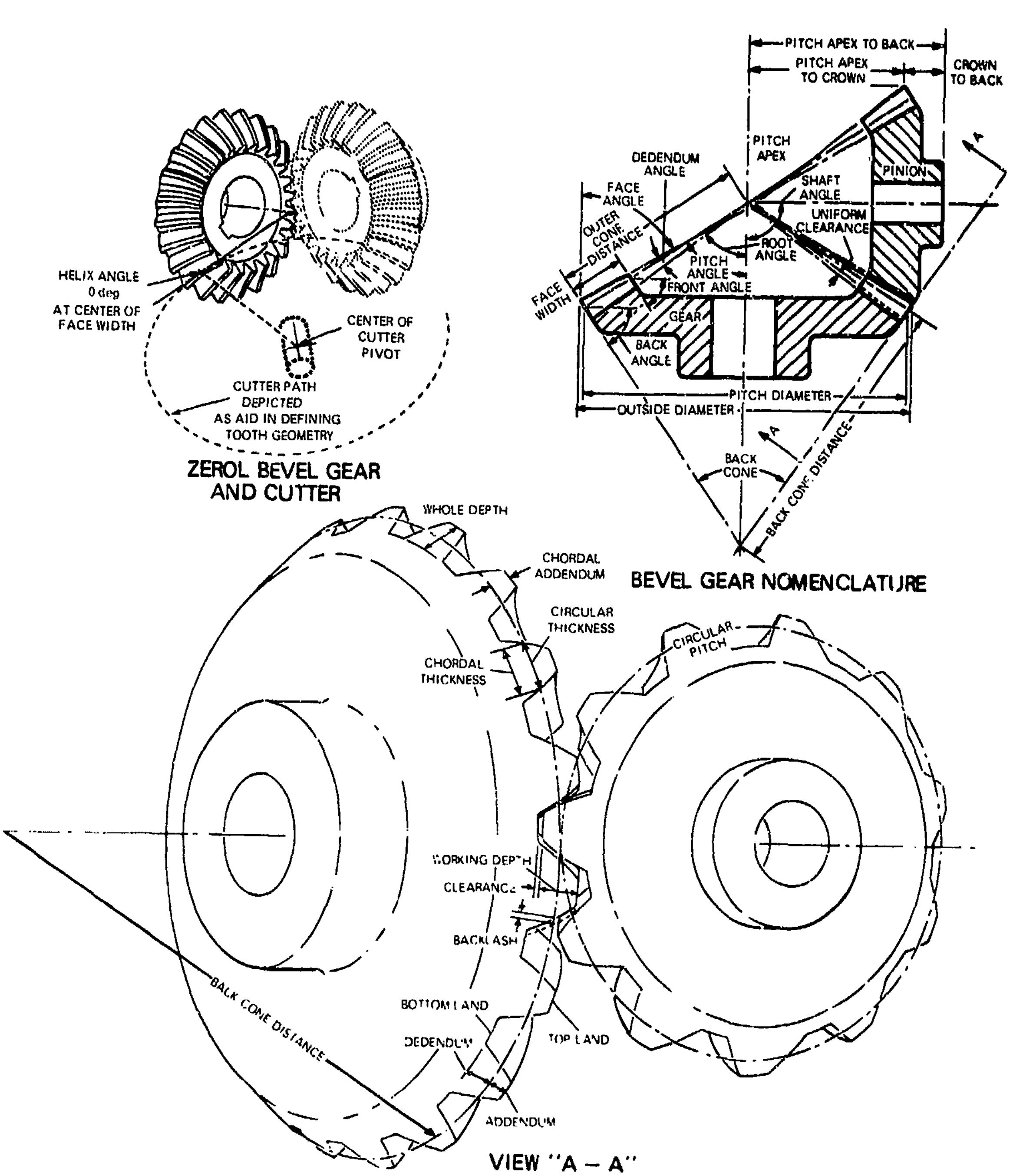


Fig. 7-6. Zerol Bevel Gear

as a basis for definition.

7-2.2.4 Planetary Gearing

Planetary gearing (Fig. 7-7) offers an advantage in helicopter gearbox design in that the load path can be split, thereby offering a compact, lightweight method for carrying large torques through small planet gears. Historically, planetary gear sets have been used at the output stages of a gearbox. They are suited ideally for concentric input and output shafting. By splitting loads to many meshes, the required face widths are reduced. Even though a planetary gear receives more cycles, it usually is designed for infinite fatigue life and the additional cycles have no effect upon life. In production helicopters as many as 18 pinions have been used to divide the load 18 ways, thereby greatly reducing the face width requirement of the gears. In many cases it would be impossible to fit the gears in the helicopter envelope if it were not for the use of planetary gearing.

To function properly, each planetary must have an input member, an output member, and a reaction member. The word "planetary" refers to the analogy between the solar system and the gear set. The center gear about which all others rotate is called the sun gear, while the rotating smaller gears are called planet pinions. The outer internal gear (if there is one) is called a ring gear. The planetary gear usually has a carrier or cage plate. Any member can act as input, output, or reaction and thus many combinations of planetary arrangements are available.

Equal load-sharing among pinions of a planetary gear set is an important factor. In general, load-sharing

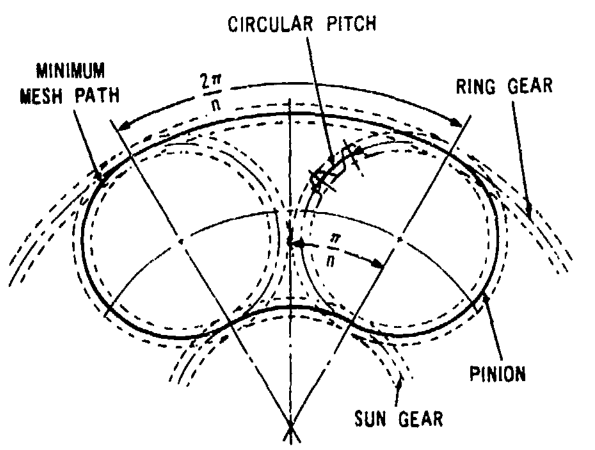


Fig. 7-7. Minimum Mesh Path for a Sun-pinion-ring Mesh

will be better when fewer pinions are used. To aid in load-sharing, the sun or ring gear usually is allowed to float. The face width required on the ring-pinion mesh can be smaller than the face width required on the pinion sun mesh, as internal gear meshes produce inherently lower bending and compressive gear stresses.

Many factors affect the final planetary design. Practicable reduction ratios vary from 2.15:1 to 7:1 for a single stage planetary (sun gear input, ring gear fixed, cage output).

For proper meshing, certain tooth-phasing relationships are required for any planetary gear. Planetary teeth should be chosen so that pinions may be spaced equally and all gears meshed properly. If pinions are spaced equally, all radial and tangential loads on the sun and ring gears cancel each other, leaving only a resultant torque and thereby simplifying bearing design. The planetary pinions can be assembled with equal spacing if the ratio $(n_p + n_s)/n$ = a whole number, where n_p is the number of teeth on each pinion, n_s is the number of teeth on the sun gear, and n is the number of planetary pinions. It is possible to design a planetary gear with unequal spacing of pinions and retain proper meshing but there are no advantages to this type of system. It also is possible to design a planetary gear with equal spacing of pinions that can be assembled but will not rotate if the teeth do not satisfy the requirement that the minimum mesh path divided by the circular pitch must be a whole number. The minimum mesh path for a sun-pinion-ring mesh is shown in Fig. 7-7.

In general the lightest weight for a particular combination of reduction ratio, horsepower, and input rpm will occur when the sun gear diameter is as small as possible. It also is advantageous to "fill up" the available space with planet pinions. To do this the pinion tip-to-tip clearance should be as small as possible. Sometimes stub teeth are used in order to fit the maximum number of pinions into the assembly and to prevent interference.

The tooth loads for planetaries are determined from standard static analysis. The design of the face width for proper stress is then the same as the design of a standard spur gear set and the same equations are used.

Occasionally, tooth modifications are used to correct for misalignment errors caused by tooth deflections under heavy loads. Tip relief usually is used to prevent tip flank interference when the unloaded tooth is just entering or just leaving mesh and the fully loaded tooth has deflected, thereby displacing the unloaded teeth to an incorrect mesh path.

All planetaries require bearings to react the imposed loads. Bearings can be standard ball, roller, spherical, or tapered roller or can use the planet gears themselves as outer races for the bearings. Usually roller bearings are used because their load-carrying capacity is much greater than that of ball bearings for any given size.

Many design variations are possible for the planetary carrier or cage plate. The cage plate carries steady torque, and a bending load is applied by the bearing reaction on the cage post if a single plate is used. Carrier plates on both sides of the planet posts help reduce this load but result in a weight increase. The material of the carrier plate should have a thermal expansion coefficient similar to the materials of the gears to prevent binding of the pinions and ring gear when the gearbox heats up. Titanium carrier plates with steel gears have been used successfully. Aluminum or magnesium carriers with steel gears are not recommended but may be used provided that the gears have sufficient backlash to prevent the binding that can occur under differential temperature expansion or contraction. Plate deflection or distortion must be held to reasonable values to prevent end loading of pinions. Many times plate deflection dictates the final plate design.

Efficiency of a planetary stage is about equal to that for an equivalent spur mesh but in some cases can be higher because of lower relative sliding velocities. Efficiency is not a function of number of pinions but is related to reduction ratio and the individual mesh efficiencies of the equivalent "fixed center" system.

Stresses in each planetary gear rim may be calculated conveniently using ring formulas. On any gear in a planetary system, a fixed point in the gear rim rotates past the load points, thereby creating vibratory stresses in the gear rim. The highest bending moment usually occurs at the point of load application. However, at this instant there is a tooth in this position, thus the highest stress occurs in the root of the adjacent tooth space. Margins of safety calculated by the above methods have proved to be slightly conservative with past designs, but the method readily lends itself to sizing of all planetary backup rings.

Many times two planetaries are used in series to obtain higher reduction ratios. The choice of reduction ratio for first-stage and second-stage planetaries to obtain a specific overall ratio usually is left to the designer. Many times a wide variety of reduction ratios is available. To aid in selection of reduction ratio for each stage for a specific overall ratio, computer programs have been written that design single-stage planetaries. These programs are combined for a two-stage unit and by iteration can choose the proper ratio for each stage for minimum overall weight.

Many types and designs of planetaries are available to the transmission designer. Split path planetaries involving one input member and two concentric, counter-rotating output members have been considered for some applications. Multiple row planetaries or single row planetaries with more than one set of pinions can yield high reduction ratios. Each design has its own special advantages and disadvantages. When dealing with high-reduction-ratio compound planetaries, it is necessary to examine efficiency. For high horsepower applications, as found in modern helicopter main transmissions, the high-ratio compound planetary is not used because of its inefficiency. For example, a reduction ratio of 30:1 might be 88% efficient whereas three stages of simple planetary, while weighing more, would have an efficiency of 97%. Hence, for a 6000-hp helicopter the difference in losses could amount to 500 hp depending upon the type of planetary system used. On the other hand the helicopter hoist offers an ideal environment for the high-ratio compound planetary because the hoist power requirements generally are low and large reduction ratios are required.

7-2.2.5 Miscellaneous Types of Gearing

The harmonic drive is an interesting type of high-ratio planetary but never has been used for high torque applications because the required flexible ring gear cannot withstand the vibratory stresses created with high horsepowers, and the wave generator bearing losses become excessively high.

Another type of planetary arrangement is the roller gear drive. In this planetary, rollers are used whose diameters coincide with the gear pitch diameters. The rollers react induced-radial loading while the gears transmit torque. Roller gear planetary systems can be arranged in multirow patterns to obtain high reduction ratios.

Worm gearing is basically a low-angle helical gear driven by a worm, which is basically a thread form. The worm thread form, in cross section, is usually a basic gear rack. This type gear has only point contact. One modification of this is to wrap the gear teeth partially around the worm; this is called single-enveloping worm gearing. Single-enveloping worm gearing has line contact, resulting in greater load-carrying capacity. If both worm and gear are shaped to conform to their mates, the result is double-enveloping or cone-drive gears.

Pressure angles range from 14.5 to 30 deg for straight worm gearing, generally 20 to 25 deg for single-enveloping and 20 deg for double-enveloping. Lead angles on the worms may range from approximately 3 deg up to 45 deg, depending upon ratio and shaft relationship. Ratios may range from 3.5:1 up to 100:1 for straight or single-enveloping worm sets and from 5:1 to 70:1 for double-enveloping.

Worm gears usually are mounted on nonintersecting shafts that are at a 90-deg shaft angle. Worm gear sets have considerably more load-carrying capacity than crossed helical gear sets. This results from the fact that they have line or area contacts and have high contact ratios. The strongest type of worm gear is the double-enveloping worm gear. Double-enveloping worm gears must be mounted accurately in all directions. Shafts must be at the correct shaft angles and center distance. The axial position of each member of the set is critical. Although at first glance worm gears appear attractive for use in helicopter transmissions, they seldom are used in major drive trains because of their poor efficiency characteristics. Worm gears often have less than 90% efficiency, while 50% is common. Some even tend to lock when overdriven, which of course cannot be tolerated in helicopters.

Hypoid gears also are mounted on nonintersecting shafts, which can be at any angle. In general, hypoid gears are stronger than bevel gears. But, as with worm gears, hypoid gears generate excessive heat and have poor efficiency characteristics, and seldom are used for this reason.

Hypoid gears generally have pressure angles between 12 and 28 deg, and spiral angles between 15 and 55 deg. Hypoid gear sets generally are made in ratios from 1:1 to 10:1. They are quiet-operating, but generally require relatively high viscosity lubricants with EP (extreme pressure) additives.

Hypoid gears, even though they resemble spiral bevel gears, are not bevel gears in the strictest sense of the word. The distinguishing feature of bevel gears is that the pitch elements are conical, whereas in hypoid gears the pitch elements are hyperboloidal. As is the case with spiral bevel gears, the hypoid gear teeth have lengthwise curvature. The teeth usually are asymmetrical, with the drive side pressure angle being lower than the trailing, or coast side, pressure angle. For a given ratio and gear size, the hypoid pinion is larger than the counterpart spiral bevel pinion. As a result, the hypoid pinion will be 15-30% stronger than the counterpart spiral bevel pinion. The characteristic that distinguishes hypoid gears in relationship to spiral bevel gears is that the pinion is offset either above or below the centerline of the gear. Because of the offset hypoid pinion, there is lengthwise sliding in addition to profile sliding between the teeth, which results in greater thrust loads being developed than is the case with spiral bevel gears. Fig. 7-8 shows the terms used to define hypoid gears, their elements, and the basis of the gearing system.

Another main distinction between spiral bevel and hypoid gears is that the spiral angles of a hypoid pair generally are different, the pinion angle having a higher value than that of the mating gear. In spiral bevel gears, the pinion and gear both have the same spiral angle but are of opposite hand, and similarly in the case with helical gears. In hypoid gears, in addition to the two spirals being of opposite hand, there is a difference in the spiral angle value between the pinion and the gear member of the pair.

It is to be noted that the word "spiral" in the definition of spiral bevel gears and also hypoid gears is a misnomer in that it is not a spiral curve in any true mathematical sense. In fact, spiral bevel gears more appropriately might be called circular arc face bevel gears. However, inasmuch as these gears have been referred to by designer, manufacturer, and user as spiral bevel gears for many years, the system of definition by agreement takes preference over a more accurately descriptive or logical name.

All gears considered thus far use involute tooth forms; however, conjugate action can be obtained with forms other than involute. Conformal gears utilize circular arc segments for the basic tooth form. With circular arc gears, area contact can be obtained, thereby increasing load-carrying capacity. If both members of the conformal gear set have the same radius of curvature, the center distance becomes critical. Because of this, the tooth radii deliberately are mismatched, usually to about 10%. This does not affect load-carrying capacity appreciably but greatly reduces the problem of accurate center distance. The center distance problem can arise due to differential temperature expansion and to deflections of gears, shafts, and mounting members. It is estimated conservatively that conformal gears require about half the face width of equivalent spur gears. This gear form has increased resistance to pitting because of the conforming shape of the gears. Efficiency and noise characteristics are similar to spur gears. Conformal gears must have a helix angle and usually are designed for a minimum overlap ratio of 1:3.

At present, there are no well-established design formulas for circular arc gears. Therefore, these and other new designs require extensive development testing to establish a final design.

7-3 TRANSMISSION REQUIREMENTS

7-3.1 GEAR SYSTEMS

The type of engine being used in the helicopter, and its location in relationship to the transmission and rotor, will dictate the type or types of gears to be used. The basic gear train arrangements that are possible or recommended for use in helicopter transmissions include:

1. Simple gear pair with parallel axis
2. Right-angle gear pair
3. Right-angle gear pair with offset axis.

Although there may be other general gear arrangements, they are not appropriate for helicopters.

The category most suitable for helicopter gears, the parallel axis gears, can employ either spur or helical gears. The angle gear pair need not be restricted to a right angle or a 90-deg shaft angle. The pair can include shaft angles either greater than or less than 90 deg, and generally will result in the use of spiral bevel gears and, on occasion—for lower power transmissions and speeds—of straight bevel gears with localized contact.

Bevel gears with intersecting axes, whether the shaft angle is greater or less than 90 deg, can be represented kinematically by pitch cones that roll without slipping at the specified velocity ratio and contact along a common element called the pitch element. The pitch element is, therefore, the instantaneous axis of relative motion of either gear with respect to the other. The two axes and the instantaneous axis all lie in the axial plane

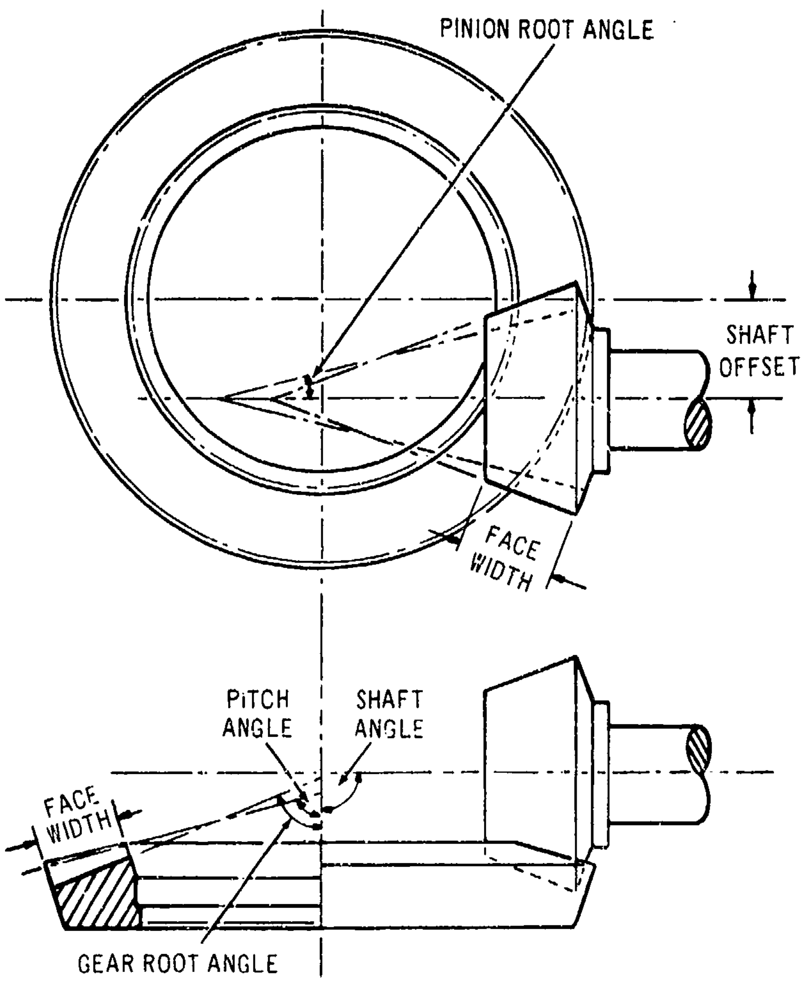


Fig. 7-8. Hypoid Gears

and intersect at the apex. Gears with parallel axes can be either spur gears or helical gears and may be considered an extension of the bevel gears, in which the pitch cones become pitch cylinders and the apex is at infinity. Fig. 7-9 shows the pitch surfaces for the three basic types of gear arrangements.

The pitch cones of bevel gears and pitch cylinders of spur and helical gears have basic properties that make them valuable concepts in analysis and design. These include:

1. Pitch surfaces roll together with no sliding.

2. The pitch element is the instantaneous axis of relative motion.

3. The common tooth surface at any point of contact intersects and is normal to the pitch element.

4. The pitch element is the intersection of the surfaces of action.

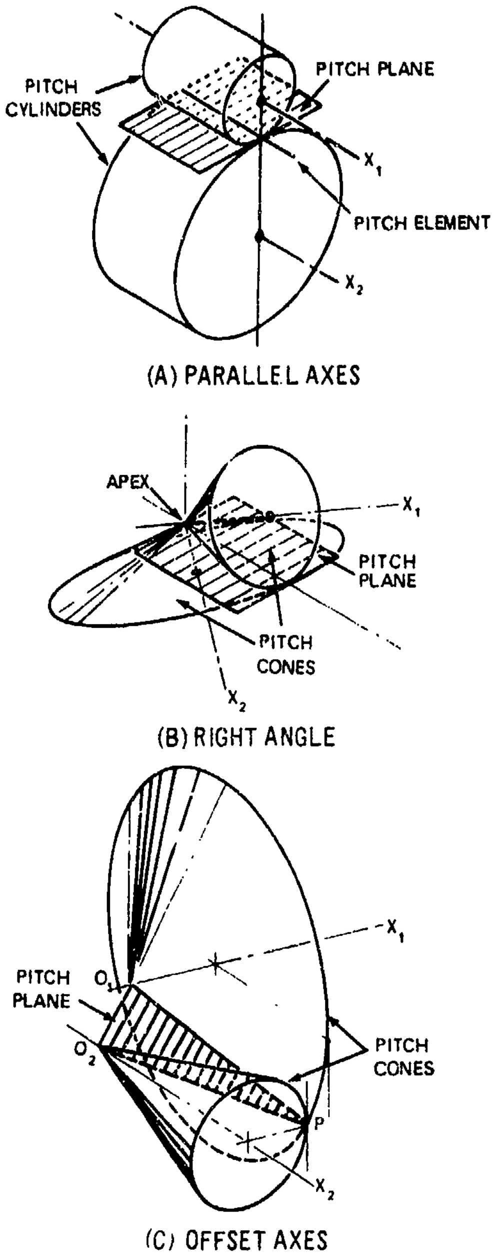


Fig. 7-9. Pitch Surfaces for Basic Gear Arrangements

It should be noted that there are no pitch surfaces for gears without coplanar axes that fulfill any of the first three conditions. The fourth condition can be fulfilled only by choosing specific tooth shapes for the purpose. Pitch cylinders and pitch cones sometimes are referred to in connection with hypoid gears as a matter of convenience, but they must not be considered to be true pitch surfaces.

Because the surfaces of action intersect in the pitch element, it is natural to use the pitch surfaces as a basis for designing gear blanks. Thus, conical gear blanks usually are used for bevel and hypoid gears, and cylindrical blanks are used for spur and helical gears. The gear teeth are positioned around the pitch surface and the intersection of the tooth surfaces in the pitch surface is at the pitch curves. The spacing of the pitch curves on the pitch surface is the gear pitch, which may be measured angularly or linearly and in various directions for various purposes, i.e., pitch may be measured in the plane of rotation or in a direction normal to the gear tooth surfaces.

The pitch plane is the plane that is mutually tangent to the pitch surfaces of a gear pair. It contains the pitch element and is perpendicular to the axial plane. Tooth directions such as pressure angle and spiral angle or helix angle usually are specified with respect to the pitch plane. The system of gearing generally employed can use the basic crown or rack members as a basis for definition. When the pitch angle of a bevel gear is 90 deg, it is a crown gear and its pitch surface is a plane that rotates about the apex while rolling with the pitch cone of its mate.

Similarly, when the pitch radius of a spur or helical gear becomes infinite, it is a rack and its pitch surface is a plane that has translational motion while rolling with the pitch cylinder of its mate. A rack or a crown gear is the basic member for the gear family conjugate to it. Tooth dimensions customarily are specified for the basic member; and they are also the basis of most generating processes even though the actual manufacturing process is that of form cutting or grinding. Be-

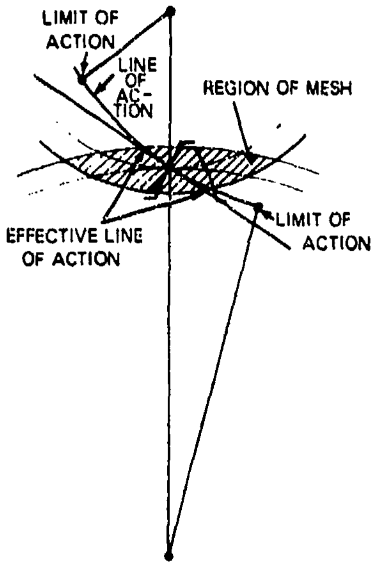


Fig. 7-10. Plane Tooth Action

cause the extension of gear tooth action to a three-dimensional system can be analyzed through the integration of gear tooth action in a given plane, the analysis usually is conducted in a single plane. In order to proceed with the basis of analysis in any one of a number of planes that generally are used, the planes first will be defined. The transverse plane is the plane that is perpendicular to the pitch elements; therefore, it is mutually perpendicular to the axial plane and the pitch plane. It is usually the plane that is used to study spur and helical gear tooth action because any point on the tooth surface remains in that plane during rotation. Such an analysis is, therefore, exact for the particular section or plane of the gear studied. In a bevel gear, a particular point on the tooth does not remain in the same transverse plane during rotation; hence, bevel gear analysis in the transverse plane is approximate. However, the method is used widely and is sufficiently accurate for most purposes. It should be noted that the plane of rotation and the transverse plane are coincident for both spur and helical gears, whereas in the bevel gear system this is not the case.

For a gear that is defined as a machine element whereby uniform motion is transmitted from one member to another, the following fundamental law of gear tooth action applies. The common normal to the profiles at a point of contact contains the pitch point. Fig. 7-10 illustrates tooth action in a plane that supports the definition of a gear whereby uniform motion is transmitted from one member to another.

The line of action is the locus of the points of contact as the profiles go through mesh. The point on the line of action closest to a turning center is a limit of action. The radius of curvature of the profile associated with that turning center becomes zero at this point, and conjugate tooth action cannot proceed beyond it. The limits of action are outside of the region of mesh enclosed by the outside circles of the mating gears. The line of action available for proper tooth action, taking into account the region of mesh and the limits of action, is the length of the line of action.

The angle of rotation of a gear that proceeds from one end of the length of the line of action to the other is the angle of contact. The ratio of the angle of contact to the angular pitch is the transverse (profile) contact ratio and must be greater than one (1.0) for continuous transmission of motion. Fig. 7-11 shows a pair of tooth profiles in contact at a general point M, at a distance S from the pitch point P.

The contact normal at M must pass through P and is inclined at the pressure angle ϕ. K_1 and K_2 on the contact normal are the centers of curvature of the profiles at the contact point M and must be related geometrically as shown in the figure for uniform motion. The algebraic statement of this relationship is

$$\frac{1}{\rho_1 + S} + \frac{1}{\rho_2 + S} = \frac{1}{\sin \phi}\left(\frac{1}{R_1} + \frac{1}{R_2}\right) \tag{7-2}$$

where

ρ_1 and ρ_2 = radii of curvature of the profiles at M

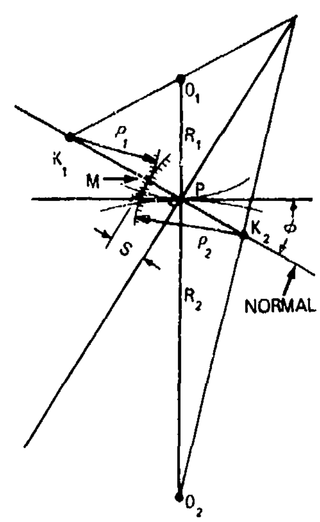


Fig. 7-11. Tooth Profile Curvatures

R_1 and R_2 = pitch circle radii of the pinion and gear, respectively

For a pair of profiles to continue to transmit uniform motion, which is the requirement by definition of a gear, this relationship must hold at all phases of gear tooth mesh. The relative curvature of $1/\rho_o$ between gear tooth profiles is

$$\frac{1}{\rho_0} = \frac{1}{\rho_1} + \frac{1}{\rho_2} \tag{7-3}$$

For contact at the pitch point, the relative curvature is given by the formula

$$\frac{1}{\rho_0} = \frac{i}{\sin\phi}\left(\frac{1}{R_1} + \frac{1}{R_2}\right) \tag{7-4}$$

The simplest and most widely used tooth form has a basic rack with straight tooth profiles. The inclination of the rack profile to a vertical line is the pressure angle. The line of action is straight and normal to the rack profile. The circles about the turning center and tangent to the line of action are base circles. The profile curve on each member is the involute of its base circle (see Fig. 7-12).

The involute of a circle may be defined as the curve traced by a point on a straight line that rolls without slipping on the circle. The circle is called the base circle of the involute. A single involute curve has two branches of opposite hand, meeting at a cusp on the base circle where the radius of curvature is zero. All involutes of the same base circle are congruent and parallel. Involutes of different base circles are similar geometrically. Fig. 7-13 shows the elements of involute geometry. The radius R of the pitch circle is $R_b \sec\phi$, where ϕ is the pressure angle.

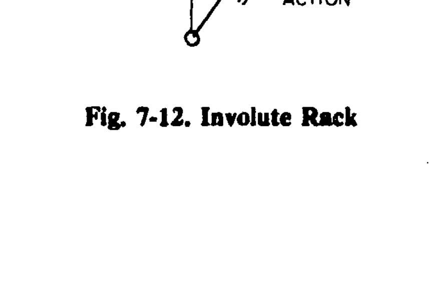


Fig. 7-12. Involute Rack

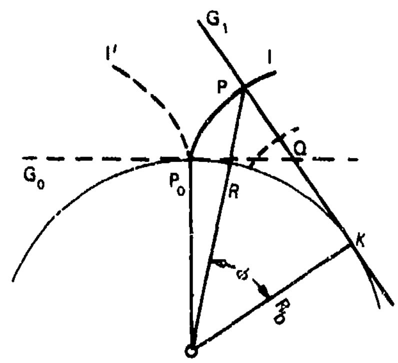


Fig. 7-13. Involute Geometry

7-3.1.1 Design Considerations

Based upon the drive system requirements that establish the initial design criteria, the following factors govern the drive system:

1. Reduction ratio
2. Design torque
3. Operating time schedule
4. Space requirements
5. Source and type of power
6. Type of loading
7. Environmental conditions
8. Material specifications for gears
9. Lubricating media
10. Method of lubrication and cooling
11. Ability to operate following lubrication system failure
12. Projected growth.

Aerospace drive system applications require careful consideration of these factors before preliminary layout for configuration determination can be made. Usually the initial design criteria will require trade-offs. The second requisite involves the selection of a gear train to be used in helicopter arrangements. The following are included as possible gear train configurations:

1. Simple gear train: parallel axis
2. Right-angle gear pair
3. Right-angle gear pair: offset axis
4. Planetary: either simple compound or multistage.

The gears in the train can be arranged in a variety of ways. One of the major design problems in high-performance equipment is the problem of arranging and fitting the gear train to the available space, which usually is dictated by aerodynamic shape, orientation of power plant in relationship to transmission and rotor, etc., while still meeting the design objectives established for the drive system. The gear types that generally are used in helicopter transmissions are:

1. For parallel axes (external and internal):
   a. Spur
   b. Helical
   c. Herringbone or double helical
2. Intersecting axes (straight):
   a. Zerol bevel
   b. Spiral bevel
3. Nonintersecting axes: Hypoid.

The selection of the gear types in the gear train arrangement has a very important effect upon the efficiency, power capacity, and size of the drive system. For example, the following power losses are typical of high-performance gear drives that are used in aerospace applications:

1. Spiral bevel gear mesh: 0.5%
2. One-stage planetary mesh: 0.75%
3. Spur or helical gear mesh: 0.5%
4. Hypoid gear mesh: 0.9 to 2.0%.

The power loss of a gear mesh is dependent upon the amount of sliding motion and the coefficient of friction, which, in turn, depends upon the surface finish of the gear teeth and the ability of the lubricant to reduce the frictional losses. Lower coefficients of friction and increased mesh efficiency result from the use of recess action gears for all three general categories of gears.

Based upon the drive system requirement, the preliminary design activity involves a consideration of the load-rating limitation of the gears, an estimate of the approximate gear capacity, and a consideration of the kind of data that will be required on the gear drawings. The load-rating limitations include:

1. Tooth Strength. The tooth strength is based upon the load the tooth can carry without permanent deformation or fracture. Bending strength is selected so that peak loads are always below the endurance limit. Fatigue failures, when they occur, usually initiate at or near the root fillet.

2. Surface Durability and Scoring Hazard. In present-day helicopter applications, the highly loaded gear teeth are not separated by a fully hydrodynamic oil film during all phases of engagement. It is believed that part of the tooth action takes place under boundary film conditions. The combination of high loads and sliding action of driving pinion teeth and flanks of the driven gear teeth impose the most severe conditions for the lubricating oil. Manufacturing processes also produce tooth profile modifications with variations from tooth to tooth and small surface irregularities that tend to lower oil viscosity and film strength due to frictional heat and dynamic loads in critical contact areas. This results in breakdown of the lubricating film, which permits metal-to-metal contact so that surface deterioration (scoring) occurs very rapidly.

The rating methods that follow are used by most helicopter gear design engineers. In most instances, the basic data are from the various standards and practices of the American Gear Manufacturers Association (AGMA). The tooth strength rating of spur, helical, herringbone, and bevel gear teeth can be determined by using an equation for the tensile stress at the root due to bending f_t applicable to each type

$$f_t = \left(\frac{W_t K_o}{K_v}\right)\left(\frac{P_d}{F}\right)\left(\frac{K_s K_m}{J}\right) \text{, psi} \tag{7-5}$$

where (for tensile stress due to bending)

W_t = transmitted tangential load at operating pitch diameter, lb
K_o = overload factor, dimensionless
K_v = dynamic factor, dimensionless
P_d = diametral pitch, in.$^{-1}$
F = face width, in.
K_s = size factor, dimensionless
K_m = load distribution factor, dimensionless
J = geometry factor, dimensionless

The groups of terms in Eq. 7-5 are concerned with the load, tooth size, and stress distribution, respectively. The relationship of calculated stress f_t to allowable (yield) stress F_{ty} is:

$$f_t = \frac{F_{ty} K_l}{K_t K_r} \text{, psi} \tag{7-6}$$

where

K_l = life factor, dimensionless
K_t = temperature factor, dimensionless
K_r = safety factor, dimensionless

Surface durability rating is a measure of the resistance of the profile to the fatigue phenomenon known as pitting. The following general formula applicable to spur, helical, herringbone, and bevel gears can be used to determine the limiting value of contact stress f_c to prevent destructive pitting:

$$f_c = C_p \sqrt{\frac{W_t C_o}{C_v}\left(\frac{C_s}{dF_e}\right)\frac{C_m C_f}{I}} \text{ , psi} \qquad (7\text{-}7)$$

where (for contact stress)

W_t = transmitted tangential load at operating pitch diameter, lb
C_o = overload factor, dimensionless
C_v = dynamic factor, dimensionless
d = pinion operating pitch diameter, in.
F_e = effective face width, in.
C_s = size factor, dimensionless
C_m = load distribution factor, dimensionless
I = geometry factor, dimensionless
C_f = surface condition factor (includes finish, residual stress, and work hardening), dimensionless
C_p = material elastic properties coefficient, $(\text{psi})^{1/2}$; see Eq. 7-12

The relationship of calculated contact stress f_c to allowable contact stress F_c is given by:

$$f_c \leqslant F_c \frac{C_l C_h}{C_t C_r} \text{ , psi} \qquad (7\text{-}8)$$

where

C_l = life factor, dimensionless
C_h = hardness ratio factor, dimensionless
C_t = temperature factor, dimensionless
C_r = safety factor, dimensionless

Scoring probability rating is a measure of surface capacity and depends upon the oil film strength to prevent metal-to-metal contact. If contact occurs, a welding and tearing apart action results in profile surface deterioration. An empirical formula, applicable to spur and helical gears, used to determine the flash temperature index or scoring hazard, follows:

$$T_f = T_i + \frac{W_{te}^{3/4}}{F_e}\left(\frac{50}{50-s}\right)\frac{Z_t N_p^{1/2}}{P_d^{1/4}} \text{ , °F} \qquad (7\text{-}9)$$

where

T_f = flash temperature index, °F
T_i = initial temperature, °F
W_{te} = effective tangential load, lb
s = rms surface finish, after initial run-in, micro-in.
Z_t = scoring geometry factor, dimensionless
N_p = pinion rpm

Z_t can be evaluated as follows:

$$Z_t = \frac{0.0175\left(\sqrt{\rho_p} - \sqrt{\frac{n_p}{n_g}\rho_g}\right)P_d^{1/4}}{(\cos\phi_t)^{3/4}\left(\frac{\rho_p \rho_g}{\rho_p + \rho_g}\right)^{1/4}} \qquad (7\text{-}10)$$

where

ρ_p = pinion radius of curvature, in.
ρ_g = gear radius of curvature, in.
n_p = number of teeth in pinion
n_g = number of teeth in gear
ϕ_t = operating transverse pressure angle, deg

The effective tangential load W_{te} in Eq. 7-9 must be adjusted to allow for the sharing of load by more than one pair of teeth as follows:

$$W_{te} = K_m W_t \text{ , lb} \qquad (7\text{-}11)$$

where

K_m = load distribution factor for bending load, dimensionless

The design scoring index value is presented in Fig. 7-14. Calculated flash temperatures for gear sets that score and do not score when lubricated with MIL-L-7808 oil or MIL-L-23699 oil (which is basically 7808 oil with a load-carrying additive) have resulted in a recommended safe design scoring index value range up to 276°F, an intermediate or medium risk range from 276° to 338°F, and a high risk range above 338°F. A graph of load capacity versus surface finish is presented in Fig. 7-15.

Allowable stresses are determined best from field experience and qualification bench testing. However, actual tests of gears have shown quite a scatter of test results. A safe stress index is one that is along the lower boundary of the scatter band where test data are plotted as stress versus number of cycles. If field and test data are not available, the designer usually will consult

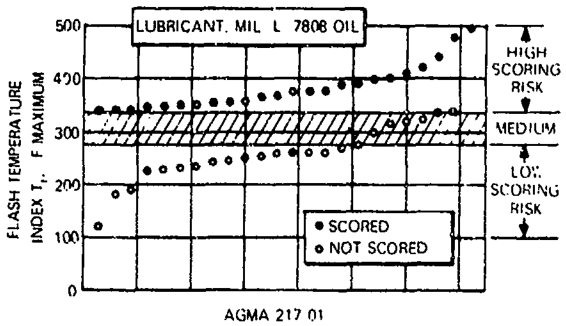


Fig. 7-14. Gear Scoring Design Guide for Aerospace Spur and Helical Power Gears

published data such as AGMA practices and other trade publications for the preliminary gear design and analysis. Caution must be exercised in applying modifying factors in order to select the appropriate value for allowable stress, because consideration must be given to such factors as:

1. Gear material and hardness
2. Tolerance class
3. Mounting accuracy and rigidity of mountings
4. Tooth surface finish
5. Lubricant
6. Surface treatment
7. Dynamic loads
8. Operating conditions
9. Vibrations and resonant conditions
10. Contingency factors.

As suggested previously, gears in a particular transmission should be investigated for all design conditions, namely, bending stress, compressive stress, and the scoring hazard.

After the preliminary design investigation has established that the limits are satisfactory for all conditions,

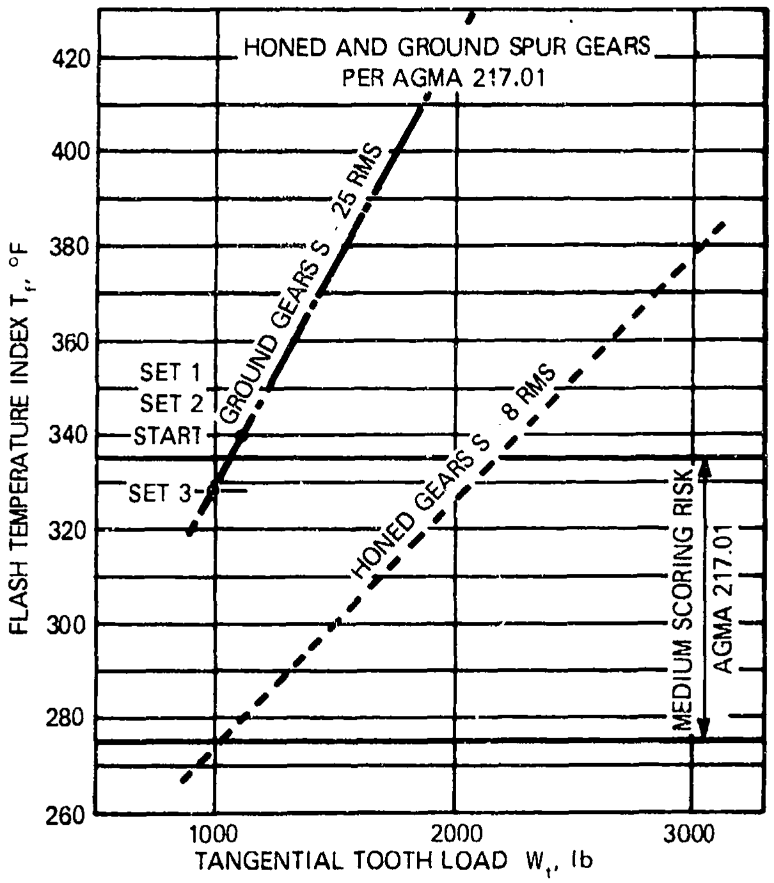


Fig. 7-15. Load Capacity vs Surface Finish

it generally will be found that the capacity of gear teeth to resist pitting actually will establish the gear size and life. The tooth bending strength is determined by the size of the gear tooth; however, long-life gears usually are found to be critical in pitting durability. The general formula (Eq. 7-7) will evaluate whether the gears are critical in pitting durability. In that formula, the constant C_p, which is the coefficient depending upon elastic properties of the material, is given by the expression

$$C_p = \sqrt{\frac{k}{\pi\left(\frac{1-\mu_p^2}{E_p} + \frac{1-\mu_g^2}{E_g}\right)}}\ , \text{(psi)}^{1/2} \quad (7\text{-}12)$$

where

μ_p and μ_g = Poisson's ratio for pinion and gear, respectively, d'less
E_p and E_g = modulus of elasticity for pinion and gear, respectively, psi
k = 1.0 for gears with nonlocalized contact, which is the case for most spur, helical, and herringbone gears
k = 1.5 for gears with localized contact (most bevel gears)

For the typical case of steel gears

$$\mu_p = \mu_g = \frac{E - 2G}{2G} = \frac{29 \times 10^6 - 2 \times 11 \times 10^6}{2 \times 11 \times 10^6}$$

$$= \frac{7}{22} = 0.318 \quad \text{, dimensionless} \quad (7\text{-}13)$$

where

G = shear modulus, psi
k = 1 (nonlocalized contact)

then

$$C_p = \sqrt{\frac{1}{2\pi\left(\frac{1-(0.318)^2}{29 \times 10^6}\right)}}$$

$$= 2270\ \text{(psi)}^{1/2} \quad (7\text{-}14)$$

The formula for contact or compressive stress (Eq. 7-7) may be written as

$$f_c = CC_p \sqrt{\frac{W_tC_o}{C_v}\left(\frac{C_s}{dF_e}\right)\frac{C_mC_f}{I}}\ , \text{psi} \quad (7\text{-}15)$$

which can be written

$$f_c = CC_p \sqrt{\frac{W_t}{dF_eI}} \sqrt{\frac{C_oC_sC_mC_f}{C_v}}\ , \text{psi} \quad (7\text{-}16)$$

The additional coefficient of stress intensity C has been introduced, with the value $C = 1.0$ representing maximum surface stress. Consider the factors under the second radical—namely, C_o, C_s, C_m, C_f, and C_v—as all being equal to 1 for comparative purposes; then a particular transmission system can be evaluated as to whether it will be as good as, or better than, a previous transmission whose history is known. The formula

$$f_c = 2270 \sqrt{\frac{W_t}{dF_eI}}\ , \text{psi} \quad (7\text{-}17)$$

is simplified for purposes of comparison. The geometry factor I is evaluated as shown in Eq. 7-18,

$$I = \frac{C_c}{m_N}\ , \text{dimensionless} \quad (7\text{-}18)$$

where

C_c = contact angle factor for contact load, dimensionless
m_N = 1 for spur gears
m_N = F/L_{min} for helical gears
m_N = $P_N/(0.95Z)$ for helical gears whose contact ratio is > 2 or integer of 1

and

F = face width, in.
L_{min} = minimum length of line contact, in.
P_N = normal base pitch, in.
Z = length of action in transverse plane, in.

The value of the contact angle factor C_c is found from Eq. 7-19 (external spur gears) or Eq. 7-20 (internal gears).

$$C_c = \frac{\sin\phi\cos\phi}{2}\left(\frac{M_g}{1+M_g}\right), \text{ d'less} \tag{7-19}$$

$$C_c = \frac{\sin\phi\cos\phi}{2}\left(\frac{M_g}{M_g - 1}\right), \text{ d'less} \tag{7-20}$$

where ϕ equals the operating pressure angle and M_g is the gear ratio, namely, the number of teeth in the gear divided by the number of teeth in the pinion. For case-hardened gears with a satisfactory depth of hardness and surface hardness, the compressive stresses can exceed 200,000 psi with the use of either MIL-L-7808 or MIL-L-23699 synthetic oils for lubrication and cooling in the transmission. However, in most helicopter transmission designs, where either planet or star system arrangements are used for gear reduction, in order to allow for the unequal load sharing between either the planet or star gears, a satisfactory design limit for pitting is 160,000 psi. Where the design is based upon a surface compressive stress of 160,000 psi, there would be a minimum development time and effort required for the qualification and final acceptance of a newly designed transmission system.

7-3.1.1.1 Life-load Relationship

In evaluating loads versus gear and bearing life, the various load conditions are resolved for the minimum period satisfactory for an operating interval. The structural elements generally are designed to operate below the endurance limit of the material or, at least, to have a life well beyond the useful life of the vehicle. Structural design conditions generally are well covered in MIL-HDBK-5 and other applicable specifications. Changes for structural reasons are made easily by increasing the thickness of material, diameter of a bolt, or by adding a rib, etc. However, gears and bearings should be correct early in the design, since the diameter or face width of a gear is difficult to change at a later date. A bearing change also requires extensive redesign.

Antifriction bearings and case-hardened gear tooth surfaces do not have an endurance limit. These elements become too large when designed for the life of the aircraft. Some individual gears and bearings may last the life of the vehicle while others may last only to a scheduled overhaul because of the wide spread in fatigue life.

The same concept applies to those parts subject to abrasive wear due to microscopic contamination. Thus, the bearing and gear sizes usually will be designed for an overhaul life deemed to be satisfactory. In the past, an overhaul life of 1000 hr was considered a good design objective. Presently, however, an operating interval considerably greater than 1000 hr is required.

Extensive tests of rolling machine elements subject to highly localized compression or "Hertz" stresses follow a statistically determined fatigue pattern. Life L varies inversely with the load P (horsepower) raised to a power,

$$L \propto \frac{1}{P^p} \tag{7-21}$$

Most bearing manufacturers state the exponent p as 3.0; one bearing company uses $p = 4.0$ and two other companies use $p = 10/3$ on roller bearings. The exponent 10/3 also is used for gear-tooth compression fatigue. Some tests have indicated the exponent p to be 3.0 for point contact and 4.0 for line contact. Since most contacts are areas of an elongated elliptical shape, and since there actually is neither point or line contact, an average exponent of 10/3 can be used for design analysis.

All of the various loads, speeds, and times of operation are gathered into one life-load figure. Some bearing companies advise a cubic mean load for the full life; however, a modified life at normal rated power using the exponent 10/3 avoids a variation in bearing and tooth loads between investigators. Only the life is subject to varying opinion. Fatigue damage for the various conditions may be accumulated and averaged to give the rated equivalent time T_r at continuous rated power and rpm:

$$T_r = P_1^p R_1 T_1 + P_2^p R_2 T_2 + \cdots + P_n^p R_n T_n$$

$$= \sum_{i=1}^{n} P_i^p R_i T_i \tag{7-22}$$

where

p = 10/3
P_i = ratio of power used for Condition i to rated power
R_i = ratio of rpm used for Condition i to rated rpm
T_i = ratio of time used for Condition i to total time

In this equation it is essential that

$$\sum_{i=1}^{n} T_i = 1 \quad \text{, dimensionless} \qquad (7\text{-}23)$$

After the gearbox and gears and other components of the transmission have been designed and stress analyzed, the gearing components can be optimized for maximum life in service by certain specifications that will result in an improvement of the life of the gears. The design techniques involved are:

1. Recess action gears usually result in a lower coefficient of friction and reduced noise level and in an increased mesh efficiency with a smoother tooth action (see par. 7-3.1.1.4).

2. Full fillet unground roots result in maximum resistance to bending fatigue. Yet the active tooth profiles, which are ground, can enjoy a dual benefit in that the accuracy of the ground gears will be realized on the flanks of the teeth where gear tooth action takes place, while the maximum benefit of the case-hardened surface will be left intact in the unground fillets.

3. Shot-peening can result in added or increased fatigue life of the gears, both in compressive stress and in bending stress.

4. After the gear teeth have been ground and have completed the manufacturing cycle, they can be subjected to a manganese phosphate surface treatment to break in the gears during the initial stages of operation, minimizing the hazard for scoring. By this treatment the surface of the gear teeth is converted into a nonmetallic oxide that becomes impregnated or saturated with lubricant. The burnishing of the gear tooth profiles during the initial stages of running reduces the risk of scoring.

5. Tip and flank modifications can be incorporated where the deflection test or calculations indicate that a concentration of load at either the tips or the flanks or ends of the gear teeth would result in a shortened gear life.

6. Higher pressure angles generally result in higher bending fatigue strength of gear teeth. Whereas commonly used standards previously employed 14.5- and 20-deg pressure angle gears, present optimized designs generally use 22.5-deg pressure angle gears and, in certain cases, higher pressure angles.

7. The use of deeper teeth for higher profile contact ratio, as well as the use of helical gears for added face overlap, also will result in an increased fatigue capacity of the gears by sharing the load.

7-3.1.1.2 Suggestions for Good Design Practice of Gearboxes

Gear load capacity is dependent directly upon the operating alignment of the gear mesh. Good design dictates that deflections at the gear mesh be kept to a minimum. The total amount of misalignment should not exceed a given allowable value. In most applications the deflections should not exceed 0.001 in./in. of face width for spur gears and 0.003 in./in. for bevel gears at the working surfaces of mating members. Of the two basic types of mounting, straddle and overhung, straddle mounting is preferred. This configuration best reduces both alignment and deflection problems. Wherever possible, the use of two bearings for each gear shaft, mounted as close as possible to the gear faces, tends to minimize large moment loads and reduce vibration problems. In many cases, bearing sizes and shaft dimensions are determined, not by strength and life considerations, but by shaft stiffness to minimize gear tooth displacements.

It is good practice to preload thrust bearings to prevent axial displacements under load, and to allow for thermal expansion of the gear shaft by the use of a cylindrical roller bearing. The preloaded duplex ball bearing reacts with both thrust and radial loads while the cylindrical roller bearing reacts only with radial loads.

Wherever possible, the gear should be integral with the shaft. When gears must be assembled onto the shaft, a double-piloted, locked-up, involute-spline mounting is one method used for minimum weight and good reliability. However, bolted connections also have been used successfully.

The mounting of bevel and hypoid gears can be designed for maximum rigidity by establishing the gear hub directly in line with the resultant of the axial and separating forces. Provisions also should be made for axially adjusting both gear and pinion during assembly. This may be accomplished easily by the use of shims placed between the main housing and the gear shaft subassembly.

The housing, or gear case, acts to form a strong base in which to mount the bearings and support the gears and shafts, and is designed after sizing and locating the gears and bearings.

7-3.1.1.3 Gearbox Lubrication Considerations

A high-power, high-performance, lightweight gearbox dictates a pressure-fed lubrication system and oil jets that direct the oil flow to the gear faces and bearings. Consideration must be given to the proper design of the gearbox to provide adequate internal drainage to the scavenge pump inlet. A tight gear case, in which the housing fits tightly around the gearing, leads to excessive heat being generated due to oil churning, windage losses, and oil entrapment. This difficulty often can be corrected by the development of gear wipers, or shrouds, that act to isolate the gear-toothed members from the sides of the housings. A recommended scavenge ratio, oil-out to oil-in, of from 2.5:1 to 3:1 is considered adequate. Oil distribution within the gearbox by means of drilled or cored passages is recommended from a reliability standpoint in that external lines are susceptible to damage by handling and servicing. The use of externally removable oil jets is extremely desirable because it becomes easy to check the condition of the jet orifice without a complete disassembly of the gearbox. Good lubrication and cooling of the gear mesh are accomplished by spraying oil into both the incoming and outgoing meshes. Oil system design also must include provisions for 30 min of operation after loss of the primary lubrication supply. This may be accomplished by small trapped oil supplies strategically located, consumable bearing cages, or a separate redundant oil system. Grease-lubricated gear systems also are a distinct possibility.

7-3.1.1.4 Recess Action Gears

There are design advantages to be gained from the use of recess action gears. The technique involves the tooth profile shifting to "long and short addendum" design. Whereas in standard design gears the pitch diameter is in the center of the working depth of both the pinion and gear, in recess action gears the pitch diameter is low in the flank of the pinion tooth and high in the flank of the gear tooth. A comparison of standard design and recess action design is presented in Fig. 7-16.

7-3.1.2 Preliminary Gearbox Weight Considerations

A primary consideration in the selection of a gear type for a given application is weight. For a parallel-shaft application, properly designed helical gears will be a little smaller than spur gears of equal capacity. However, the thrust reaction of the helical gears may require larger bearings and a more complex support structure so that the total weight of a spur gear system might be less than that of a helical system.

Double-helical gearing can present real advantages where there is high speed and high power, due to the smoothness of the gear action and elimination of thrust loads. The gears may be heavier than single helical gears of equal capacity because of the extra width of the gear blank required to provide the space between the two helices and/or the material required to assemble the two halves of the gear. The reduced bearing and structural requirements may more than compensate for this.

For applications involving nonparallel shafting there is currently only one choice: spiral bevel gears. In some cases it may be possible to effect real space and weight savings by using two or more spiral bevel gear sets in series. For example, two meshes of spiral bevel gears, each with 45-deg shaft angles, might prove more advantageous than one set of bevel gears with 90-deg shaft angle used in series with a spur or helical gear set or a planetary reduction.

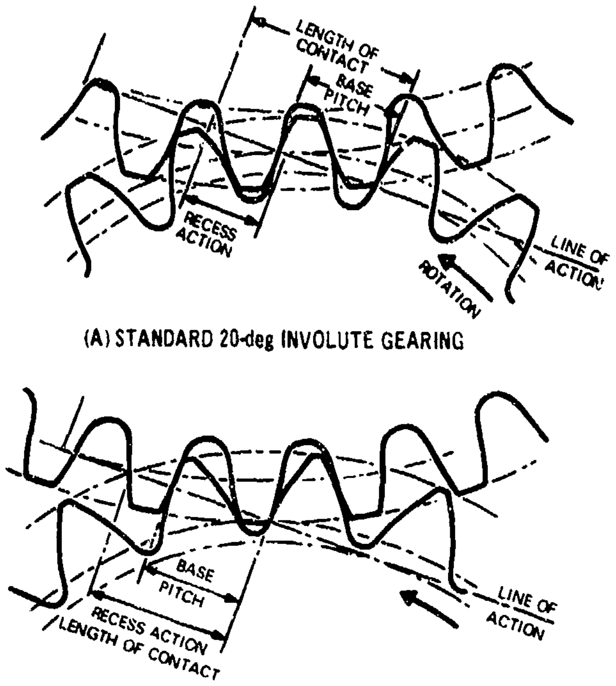
Lightly loaded drives, such as accessory drives or low power main reductions, usually will be served best with spur, or spiral, or Zerol bevel gear drives.

Following the preliminary sizing of the gearing configuration, the empirical expression presented in Ref. 1 provides a means of estimating the weight of the complete gearbox less accessories. The premise is that the weight of a gear system is a function of the total solid rotor volume and that weight estimates are obtained by multiplying the value of ΣFd^2, where F = face width, in. and d = gear pitch diameter, in., by an application factor of 0.25 to 0.30 for helicopter gearboxes with magnesium or aluminum casings of limited-life design, high stress levels, and rigid weight control. This factor considers the gearing, shafts, bearings, and support housing.

7-3.2 BEARINGS

In the selection of bearings for helicopter transmissions, the principal trade-offs are combinations of standardization, initial cost, noise, resistance to shock, frequency and ease of repair or replacement, degree of complication in shaft and housing design, and resistance to contamination. In some instances, it may be necessary to employ special-purpose bearings for reliable high-speed operation and for thrust reversals. Special attention to such trade-offs and requirements is necessary during the preliminary design phase because system weight, cost, reliability, and maintainability are affected significantly by decisions involving transmission bearings and supports.

This paragraph provides general guidance regarding the basic types of bearings and installations in helicopter transmissions. Advantages and disadvantages of the various types of bearings are listed. The discussion includes a brief listing of common mistakes in the selection and handling of bearings and mountings.



(A) STANDARD 20-deg INVOLUTE GEARING

(B) FULL RECESS ACTION

Fig. 7-16. Comparison of Standard Design With Full Recess Action Design

The textual descriptions are supplemented by illustrations of common bearings and supports. The figures illustrate bearing and support concepts, but do not constitute design guidance.

In making the selection of bearing types and sizes, a common solution for bearing-gear combinations is required. These selections should be made early in the design phase on an analytical basis because of the difficulty and expense involved in redesign to accommodate changes in bearing and gear configurations. Par. 7-3.1.1 gives the mathematical relationships to be used in conjunction with manufacturer's ratings and estimated load factors to determine average fatigue damage for bearings and gears. The application of these estimates also is described in par. 7-3.1.1. The effect that bearing support stiffness and location will have upon natural shaft frequencies is described in par. 7-5.2.

7-3.2.1 Typical Installations

7-3.2.1.1 Accessory Bearings

The simple ball bearing, straddle-mounted around gears as in Fig. 7-17, is a system that normally is lightly loaded in radial and thrust directions. The bearings shown are of the "Conrad" type (deep groove) with two-piece machined bronze cages. Their operating speeds run up to 12,000 rpm. These bearings have both radial and thrust capabilities. The ball-to-ball combination is the simplest arrangement for short shafts, with minimum shaft expansion due to differential temperatures.

The simple ball/roller bearing, shown in Fig. 7-18, is used in the hydraulic step-up drive in the spiral bevel drive gear. The ball/roller combination is selected to permit axial freedom, keeping the bearing loads free of the effect of differential expansion between the shaft and the supporting gearbox structure. The unshouldered roller bearing rings allow full float at one end and restraint at the other, ball bearing end.

In Fig. 7-18, the ball bearing is positioned adjacent to the spiral bevel gear where radial loads are light and thrust loads heavy. The bearing holds the axial position of the gear, which is adjusted by shimming. On the opposite end, a roller bearing is used to support the high radial loads from the spur gear. Its outer ring construction is a cylindrical raceway with no shoulder. The roller bearing does not oppose the ball bearing, but it permits changes in length of the shaft due to tempera-

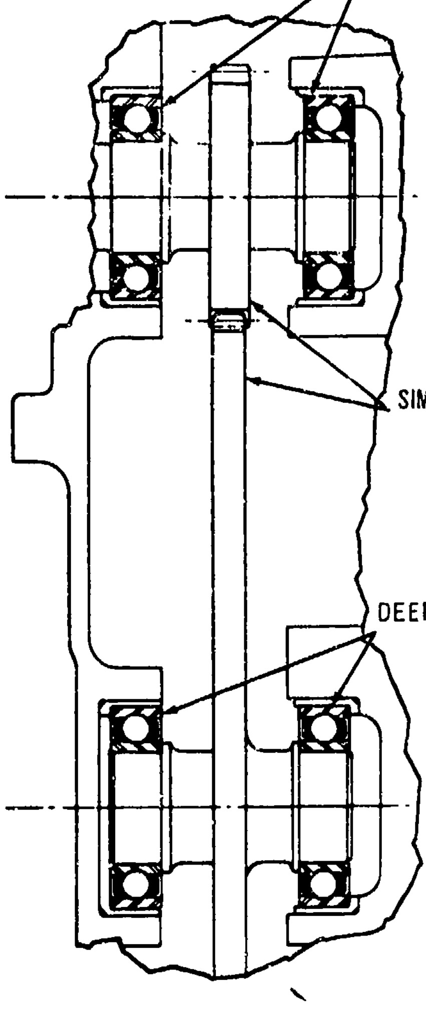


Fig. 7-17. Simple Ball Bearings

ture. The adjacent arrangement, which supports the spiral bevel pinion gear, also uses a ball and roller bearing combination. It differs in this case, however, in the relationship of the ball and roller bearings to the gear. Because the spiral bevel pinion has a greater radial than thrust load, the roller bearing is placed adjacent to this gear. The outer ring is without shoulders, giving freedom of axial movement. The ball bearing positions the gear in the axial direction by shimming at its shoulders.

Another bearing arrangement for bevel gears has a duplex "back-to-back" (DB) ball bearing pair opposite the roller bearing. This duplex pair not only restrains the shaft axially in both directions, but also provides maximum shaft rigidity. This is due to the outward flaring of the contact angles toward the shaft centerline. This form (Fig. 7-19) gives 1.9 times the radial capacity of the single bearing. Duplex ball bearings are used to obtain axial restraint in both directions. Selection of duplex back-to-back ball bearings gives this restraint but also makes a restrained shaft center in the radial direction. In contrast, duplex "face-to-face" (DF) bearings give the same axial restraint but with a "free" or pivoting shaft center by virtue of the contact angle lines intersecting at the shaft center.

The roller-roller combination supporting two straight spur gears is shown in Fig. 7-20. These gears are held by two straight cylindrical rollers. The ball bearing is not needed here because the straight spur gear produces little or no axial thrust, and the shoulders of the roller bearings shown at positions A and B are able to restrain the gears in position and accept the resultant thrust at low speeds. Two different inner ring configurations are used in this assembly. At position A the two-lipped inner ring is pressed on the shaft and held by a locking nut. On the opposite end, position B, the inner ring is ground integrally with the gear shaft. The latter assures raceway-to-gear-pitchline concentricity and eliminates the possibility of fret between inner ring bore and shaft outer diameter (OD), and, of course, no locking devices are required. This roller-to-roller system permits axial freedom and centerline restraint.

7-3.2.1.2 Plate Thrust Bearings

The plate thrust bearing shown in Fig. 7-21 cannot stand high speeds due to its race construction. The principal advantage of the plate thrust bearing is that it affords maximum capacity in one direction. However, angular contact bearings, as in Fig. 7-19, will withstand high speeds in addition to providing good thrust capacity.

7-3.2.2 Special Purpose Bearings

7-3.2.2.1 Transmission Main Drive Train

An example of a double row thrust bearing is shown in Fig. 7-22. It has a single-piece outer ring with both races ground with the counterbore outward from the ring center. The inner rings are two separate pieces with low shoulders facing inboard. Drawing up the inboard faces with the locknut puts a preload into the raceways, making the contact angles assume the DB form. This also yields an inner ring rigidity much

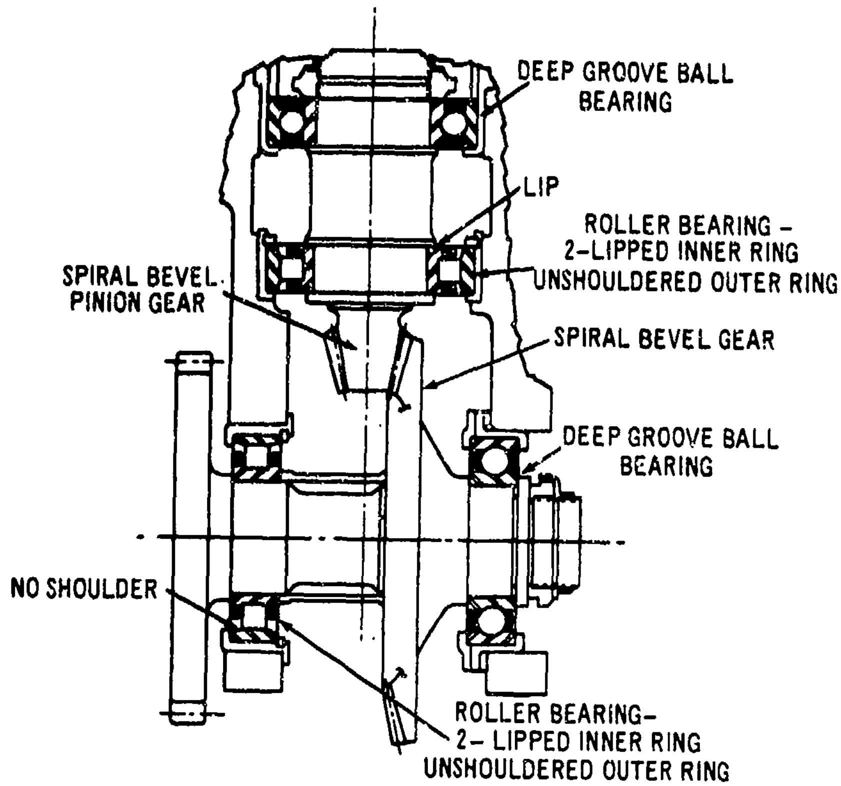


Fig. 7-18. Simple Ball/Roller Bearings

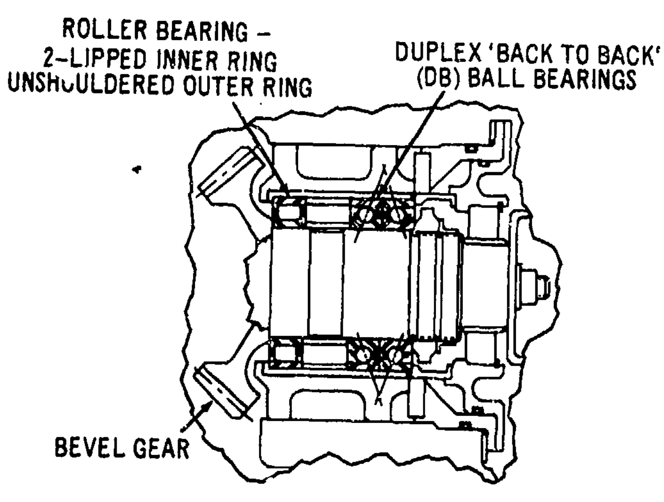


Fig. 7-19. Duplex Back-to-back Bearings

greater than the DF form. The double-row, angular-contact form gives thrust capacity in both directions as well as radial capacity equal to approximately 1.9 times the radial capacity of a single row. By proper grinding of the inner rings, the maximum capacity in both radial and thrust directions can be achieved at high speed. Separate cages are used for each row of balls to permit freedom of rows. The preloaded, double-row ball bear-

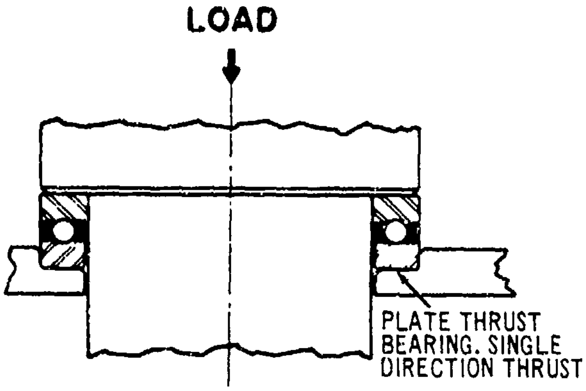


Fig. 7-21. Plate Thrust Bearing

ing is used to give axial restraint in both directions, maximum shaft centerline restraint, and simplicity.

A spur pinion gear support roller bearing with integral inner raceways, shown in Fig. 7-23, consists of two rows of cylindrical rollers on opposite ends of a high-speed spur pinion. In this case, the inner raceway is ground integrally with the pinion to insure roundness and concentricity of the raceway with the gear axis and the gear pitch diameter. The outer ring is double-lipped to give the rollers axial guidance. These two lips also serve as guidance for the broached bronze cage. The outer ring is press-fit into its housing for radial positioning and, in this case, has bolting tabs on one side of the OD to prevent rotation or axial movement. A spherical-ended roller form (see Fig. 7-23, detail A) is used to resist any thrust from gear or clutch combination and to minimize wear at inner and outer ring shoulders. Integral race-to-shaft bearings are used with high-speed gearing to minimize inner ring fitting problems, and to assure concentricity and balance of the centerline.

An arrangement of supporting bearings for a spiral bevel pinion is shown in Fig. 7-24. The forward bearing at position A is similar to that in Fig. 7-23 in that it rides on the gear shaft without a separate inner ring. The gear raceway is without roller guide shoulders, giving it axial freedom and preventing axial loading between it and bearings B and C. This combination is set up so that the radial loads of the spiral bevel go through the two roller bearings, which have no axial restraint. The split inner ring ball bearing at C is set up with radial clearance between the outer ring and the housing so that it cannot be subjected to radial load. Yet the axial retention through the flange of the adja-

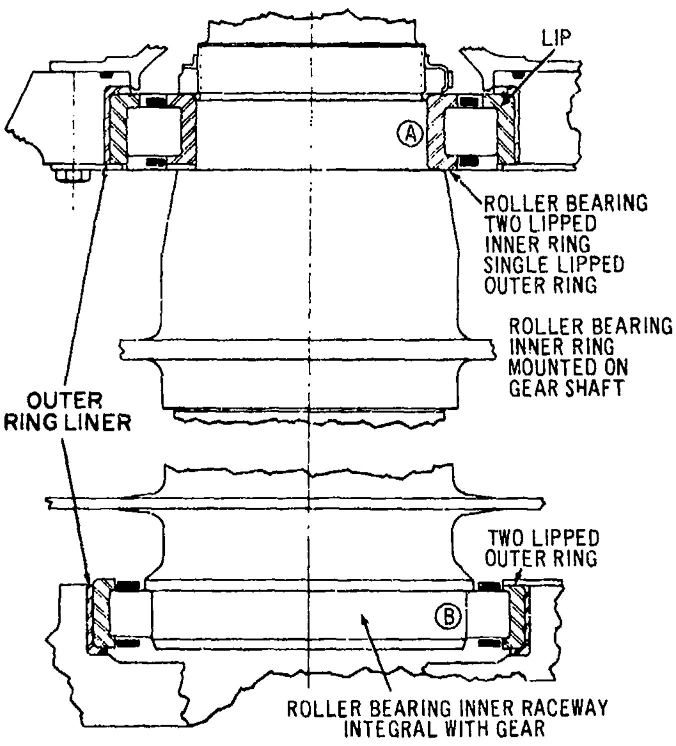


Fig. 7-20. Roller-roller Combination Supporting Straight Spur Gears

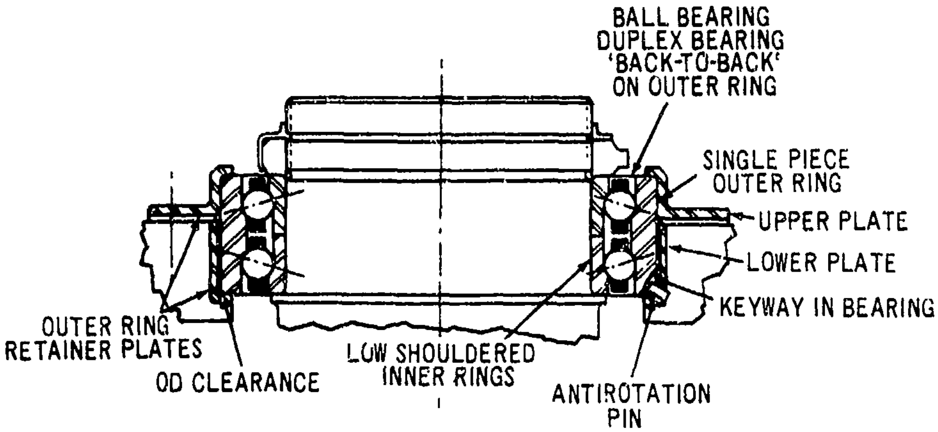


Fig. 7-22. Double Row Thrust Bearing

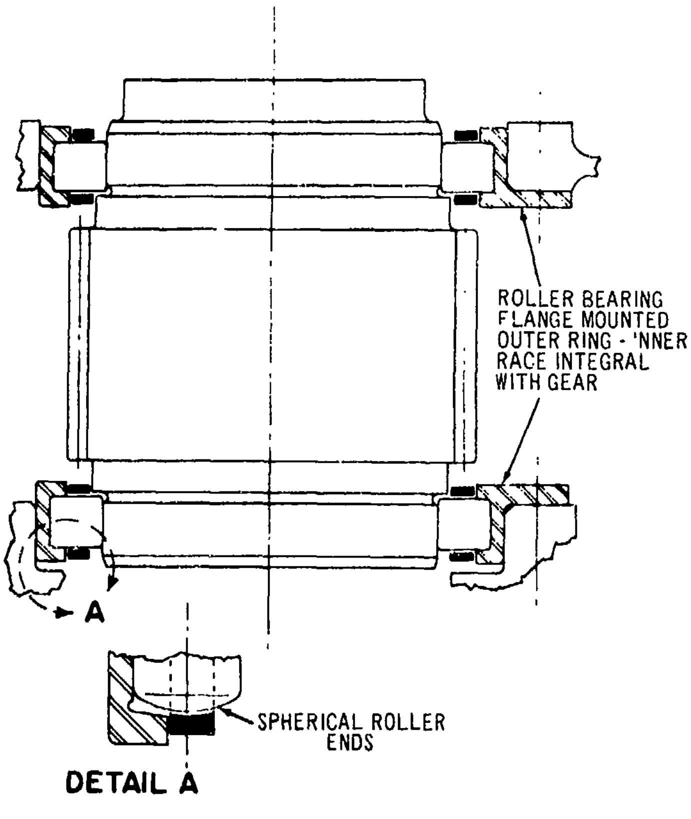


Fig. 7-23. Pinion Gear Support Rollers With Integral Inner Raceways

cent roller bearing and the outer retaining plate hold the outer ring so it can react thrust in either direction, and the inner ring is split radially at the raceway center in order to permit thrust loading in both directions. This construction also permits a high-strength, one-piece outer ring, land-guided cage suitable for high speed work.

Compound planet stages sometimes are used and present special bearing mount problems. The compound planetary gearing, shown in Fig. 7-25, is supported by two cylindrical roller bearings. These are identical in construction but of different size and capacity. The inner rings both are integral with the gear, and are double-lipped for roller and cage guidance. Each outer ring is single-lipped and opposes the other to retain the gear axially in both directions.

The double row spherical bearing example in Fig. 7-26 shows simple planetary gearing with an integral bearing raceway spherically ground in the gear inner diameter (ID). The inner ring raceway surfaces also are ground spherically to accommodate the two rows of barrel-shaped rollers. The cage is one piece and has its pockets broached in a staggered manner for the two rows. Staggering smooths out the action of the rollers to the raceway. The cage is inner-ring-land-guided at the outboard lips of the ring. The barrel rollers, together with the spherically ground inner and outer raceways, permit the misalignment of the gear up to ±2 deg. The integral raceway grinding yields a stiff outer raceway closely concentric with the gear pitch diameter. Double row spherical bearings are used mainly for their self-aligning ability as well as their maximum radial load capacity.

Fig. 7-27 shows a preloaded, duplex pair of angular contact bearings. This arrangement provides axial re-

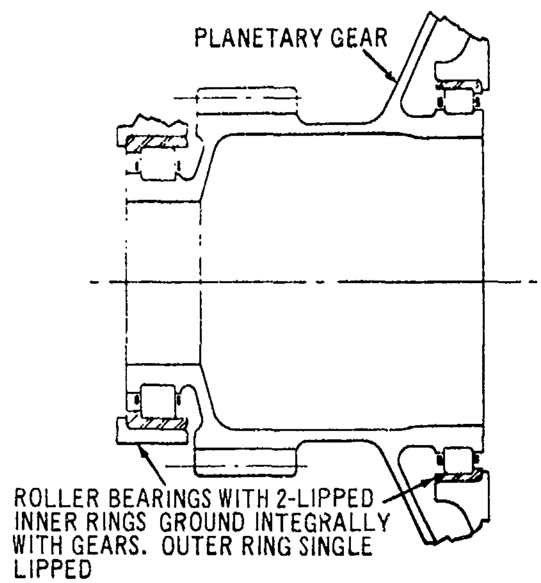


Fig. 7-25. Compound Planetary Gearing

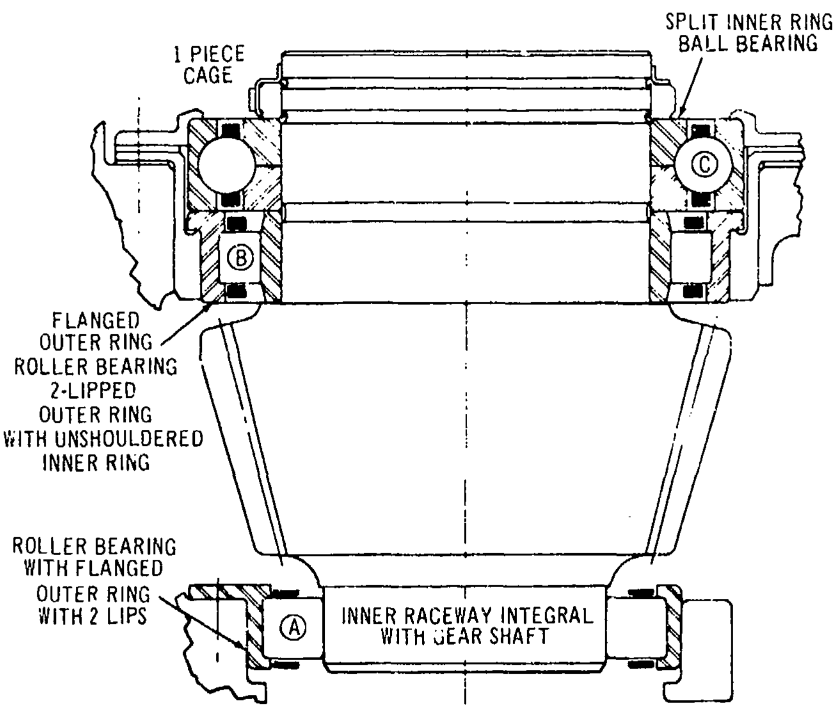


Fig. 7-24. Spiral Bevel Pinion Mount

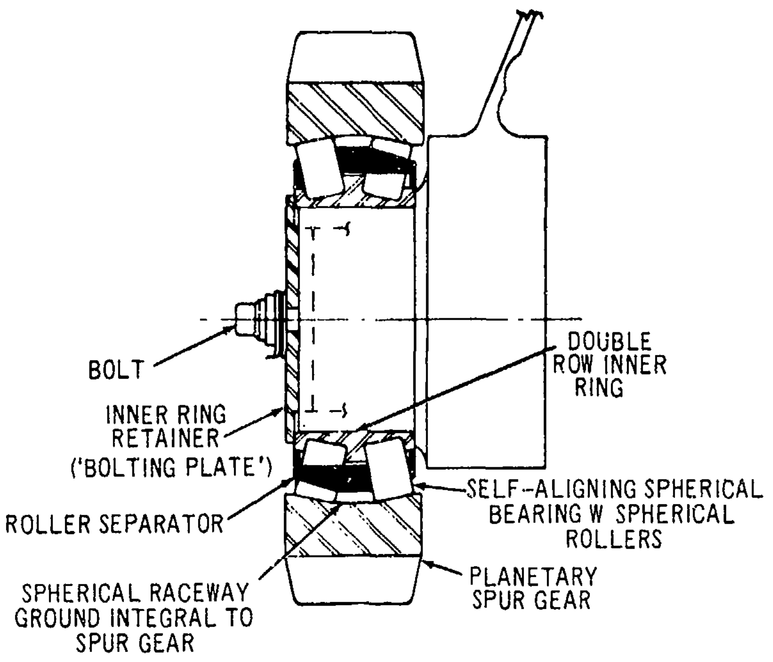


Fig. 7-26. Double Row Spherical Bearing

straint of the shaft, and prevents axial vibrations. This set is matchground DB (with outer ring thrust shoulders duplexed back to back) to give maximum shaft rigidity. The preload is retained by the inner and outer ring spacers at A and B, respectively.

A preloaded duplex tandem set is shown in Fig. 7-28. This set-up is used to resist the thrust from, in this case, a spiral bevel pinion gear. The thrust load acts through the two duplex tandem (DT) ball bearings. The other angular contact bearing at the left is set up DF to preload the DT pair and prevent reverse axial movement. This combination also has high radial capacity by virtue of the three bearings in a row. The DT set is used to obtain maximum thrust capacity in one direction. The DF set gives moderate thrust restraint in the opposite direction with maximum shaft rigidity.

7-3.2.2.2 Oscillating or Limited Motion Bearings

Self-aligning, Teflon-lined bearings (Fig. 7-29) are used in main rotor hinge or pitch rod bearings to permit change in angle of attack of the blades. The Teflon journal at the attaching bolt takes the oscillating pivoting motion. The spherical bearing (also Teflon-lined) permits the initial misalignment of the bearing in both static and dynamic deflections. No external lubrication is required.

7-3.2.3 Mountings

The upper view, Fig. 7-30(A), shows the cross section of a flanged outer ring of a roller bearing. The lower view, Fig. 7-30(B), is the bottom view of this bearing showing two of the three tabs on the outer ring. The flanged face is ground perpendicular to the bearing raceway and serves as a broad shoulder to hold the ring square, and the tabs provide an antirotation device. The holes are drilled oversize to prevent distortion of the ring by the mounting bolts. This construction is a simple means of outer ring retention, both axially and circumferentially. However, care must be taken to avoid introducing distortions in the raceway from the bolts that hold the flange.

The combination of a shaft locknut and locking ring is a common method of retention of rolling contact bearings. Fig. 7-31 shows the locknut with its serrated locking ring. This retains the bearings by clamping them between the washer and a shoulder on the shaft. The result is a bearing system with a DB configuration, consisting of a pair of angular contact bearings mounted with their thrust shoulders opposing each other. Such a system is both radially and axially stiff.

Bearing ring retention using snap rings or spring-type rings is shown in Fig. 7-32 and illustrates the more common type of retention, i.e., the inner ring retained by a ring held by spring tension in a shaft groove.

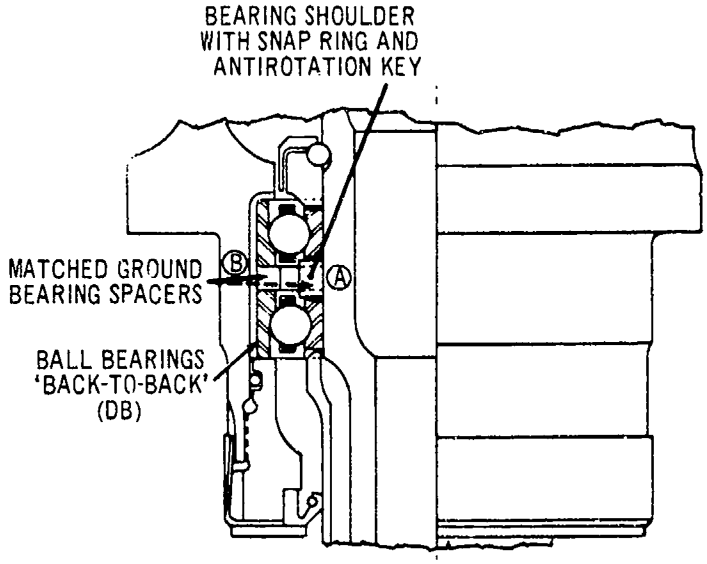


Fig. 7-27. Preloaded Duplex Pair of Angular Contact Bearings for Propeller Drive

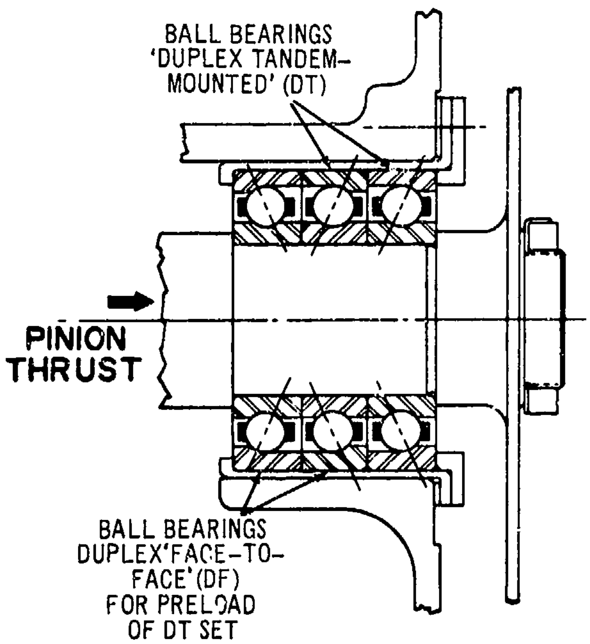


Fig. 7-28. Preloaded Duplex Tandem Bearings

Double row bearings are illustrated in their mounting plates in Fig. 7-33 with resilient laminated bearings or washers at A and B. At point A, the resilience is from an elastomer held between metal laminations. Lips at the extremities of the mounting provide radial restraint from bearing to housing. Point B has resilience in the same directions, but has no radial restraint. Radial loads are not transmitted through this type of mounting. A plain, radially restrained mounting, shown in Fig. 7-34, is a similarly laminated elastomer and metal strip wrapped circumferentially around the

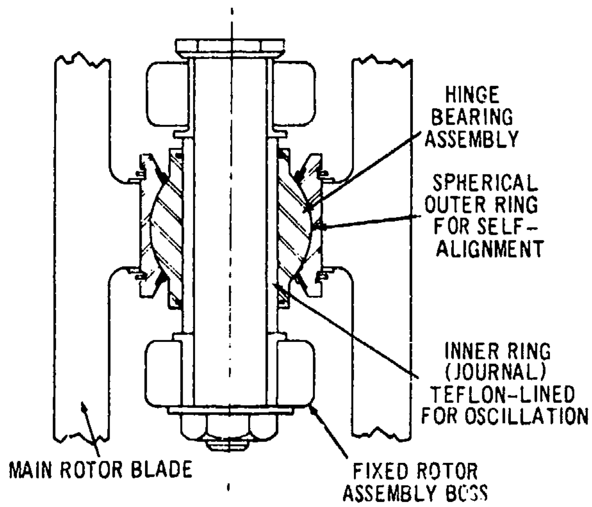


Fig. 7-29. Self-aligning, Teflon-lined Bearing

bearing. The radial displacement allowed by this mounting permits misalignment between the bearing shown and its mate; however, radial loads can be reacted.

7-3.2.4 Seals

All static and dynamic seals *shall* be required to demonstrate a life equal to or greater than the transmission allowable operating interval.

7-3.2.4.1 Integral Type

The integral seals used on rolling contact bearings save in mounting space, retain their own lubricant, and protect against contamination. This minimizes or eliminates maintenance. In many types, seal friction is eliminated by the "noncontact", or labyrinth, construction.

The standard shield (Fig. 7-35) is the simplest form of closure and has a metal disk crimped or held by a snap ring into a groove. Its inner diameter runs at a close clearance to the inner ring shoulder. This type of shield will retain grease lubricants and prevent entrance of external contaminants.

There are two types of labyrinth seals: the "mechani-seal" and the slinger-type seal.

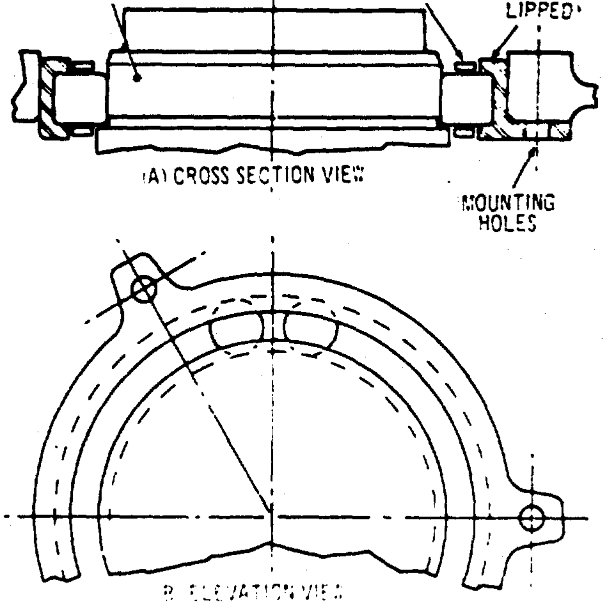


Fig. 7-30. Flanged Outer Ring of Roller Bearing

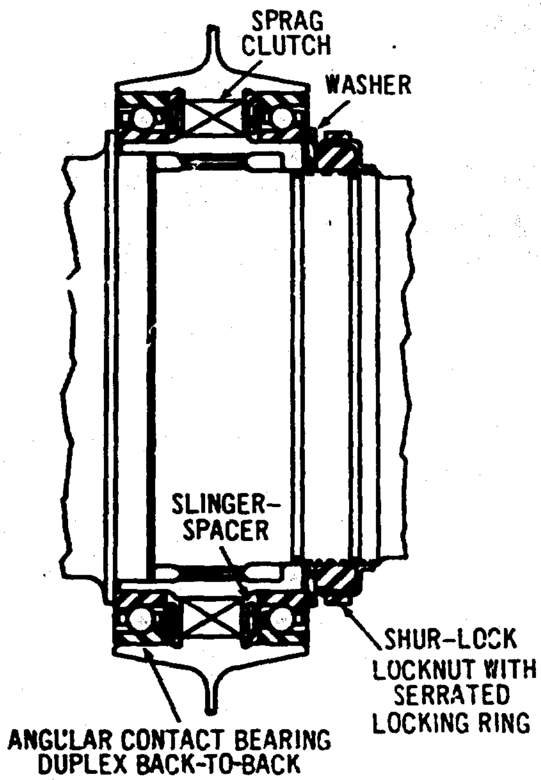


Fig. 7-31. Combination of Shaft Locknut and Locking Ring

The mechani-seal (Fig. 7-36) has the OD of its inner shield pressed into a groove in the outer bearing ring, with a slight clearance to the OD of the inner ring shoulder. The outer shield of this system has its ID pressed to the OD of the bearing inner ring, with a slight clearance between the ID of the bearing outer ring and the OD of the shield. The outer shield employs centrifugal force to keep contaminants away from the bearing, and forms a frictionless, noncontacting labyrinth with the inner seal. This seal is more effective than the standard shield, and also can run at high speeds.

The slinger seal (Fig. 7-37) consists of a metal ring pressed on the land of the inner ring. The ring functions as a slinger between the inner and outer portions of the shield. The centrifugal action of the slinger traps the lubricant in the labyrinth and gives an almost frictionless seal. It prevents external contamination and loss of internal lubrication.

Felt seals employ a felt ring that contacts the ground inner ring land to provide sealing action (Fig. 7-38). The outer metallic or retaining member is crimped into the outer ring groove. Operating temperatures are lim-

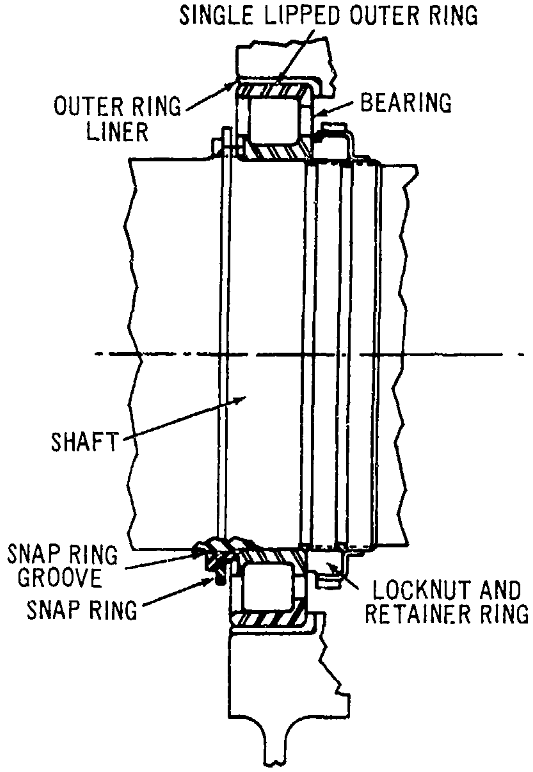


Fig. 7-32. Bearing Ring Retention by Snap Rings

ited to about 275°F, surface speeds are limited to 3000 fpm.

Contacting seals include the land-riding type (Fig. 7-39). Variations of this use external or "armorgard" construction, which prevents external damage in severe exposures to stringy or rough material. In these two types, the external portion is a stamping bonded to a molded synthetic rubber lip seal. The OD of the seal is crimped into the ID of the bearing outer ring. A modification of this, known as the "sentri-seal", is shown in Fig. 7-40. It has synthetic rubber bonded completely around a flat steel ring. Its formed lip contacts the ground chamfered or curved surface of the inner ring OD. These seals are limited to speeds up to 2000 fpm and to temperatures up to 225°F.

An extremely heavy-duty, triple-lip, contact type of seal has been developed for severe external contamination (Fig. 7-41). It has a wide external guard that is crimped into the bearing outer ring and runs at a clearance to the inner ring. Bonded inside this guard is a triple-lip seal of synthetic rubber that consists of three sealing lips contacting the ground land of the inner ring. The effectiveness of the seal is obtained by the multiplicity of the lips and their pressure to the land.

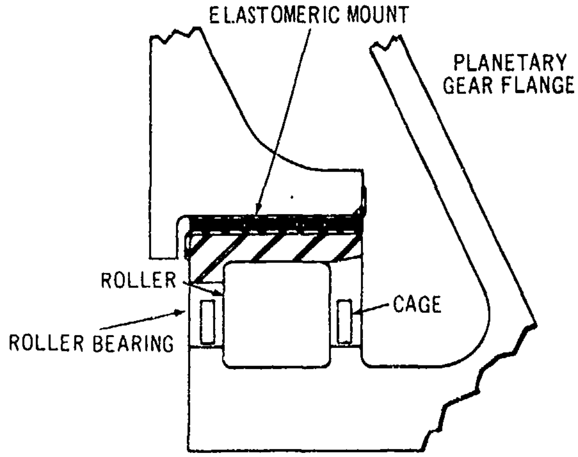


Fig. 7-34. Radially Restrained Mounting

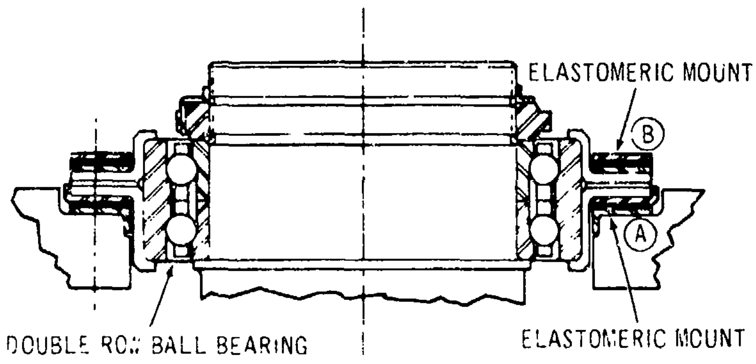


Fig. 7-33. Double Row Bearing Mounted With Resilient Laminate

Consequently, this type of bearing is limited to very low speeds.

7-3.2.4.2 Separate Type

In sealing bearings from contaminants and preventing the loss of lubrication, space limitations are encountered when the seal is integral with the bearing. Many types of separate seals have been developed for these conditions. They consist of (a) wiping or contacting seals, (b) face-rubbing types, and (c) labyrinth types.

The wiping seals commonly are made of synthetic rubber and have different shapes and orientations. The simple wiping lip seal, shown in Fig. 7-42, provides moderate sealing and protection from contamination. Felted or packed woven and graphited elements also

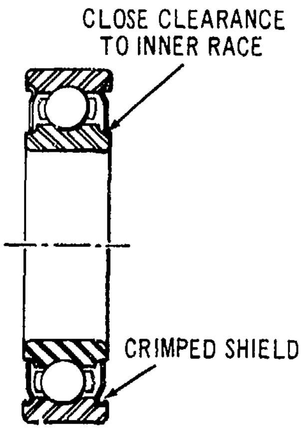


Fig. 7-35. Standard Shielded Ball Bearing

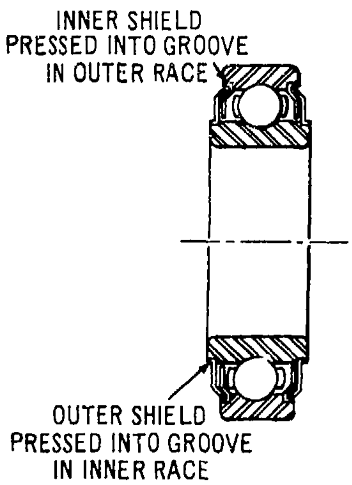


Fig. 7-36. Mechani-seal

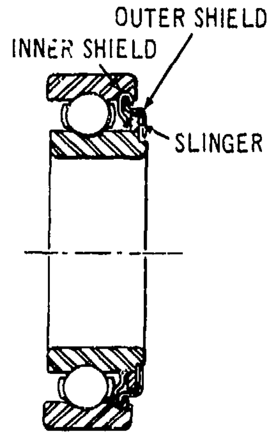


Fig. 7-37. Slinger Seal

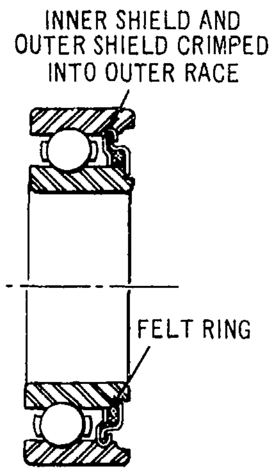


Fig. 7-38. Felt Seal

may be jammed against the shaft to prevent pressure leakages. Both of these types are limited to low speeds.

In order to achieve high-pressure sealing, long life, and high speeds, spring-loaded, face-rubbing seals have been developed. A typical construction is shown in Fig. 7-43. This consists of a rotor with its mounting and rubbing surfaces ground optically flat, and parallel and normal to each other. The active surface of this rotor is hardened and flattened to make continuous, uniform, and minimum-friction contact with the sealing surface. This surface—usually of a carbon compound—also is lapped flat. It is held in contact with the rotor by preload springs. The smooth contacting surfaces constitute the rubbing portion of the seal. In many instances, these surfaces are pressure-balanced to accom-

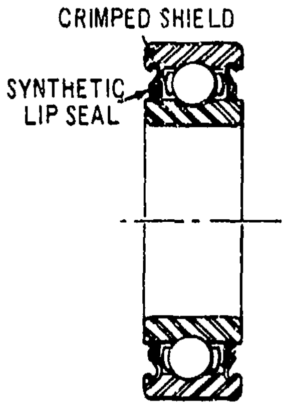


Fig. 7-39. Land-riding Seal

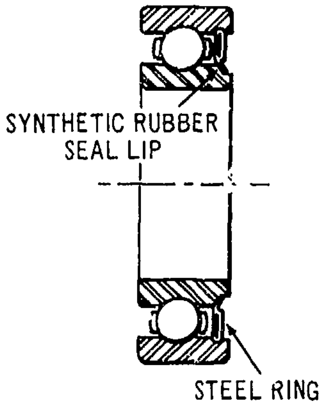


Fig. 7-40. Sentri-seal

modate a range of conditions. The external or static portion of the seal is created by close-fitting the rotor to the shaft. The static portions of the nose of the seal are made fluid-tight and flexible by bellows, either synthetic rubber or metallic. This permits axial motion of the preload spring seals, which can be used in a broad range of temperatures and pressures and at high speeds. Another variation of this form is the pressure-balanced seal whose internal and external areas are designed to balance the local pressures and minimize the thrust loads on the face and rotor contact (Fig. 7-44).

In addition there is the floating seal as shown in Fig. 7-45. This is set up with critical axial and radial clearances to accommodate pressures in opposite directions. This floating rotor rides in a ground steel recess and will contact either side, depending upon the pressure balance.

External labyrinth seals obtain their sealing capabilities primarily from the path through which the fluid must pass. Mating labyrinths, with the internal grooves rotating, are made in two or three sections. The rotating portions not only resist flow because of their shape, but also prevent leakage by the centrifugal forces created due to high-speed rotation. Interlocking labyrinths also are used with mating edges on both internal and external diameters of a rotating surface (Fig. 7-46). Hardened and ground sharp-edged rotating elements with a softer female matrix often are employed as in Fig. 7-47. A radial labyrinth seal is shown in Fig. 7-48.

The effectiveness of most types of seals can be increased by using bleed passages to balance off the leakage flow. This method is used extensively in high-speed labyrinth seals.

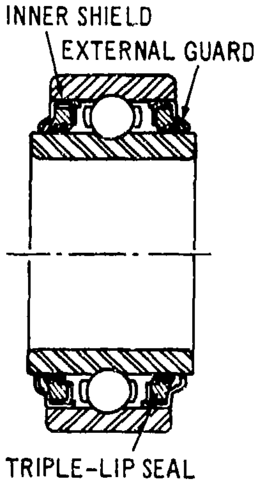


Fig. 7-41. Heavy-duty Seal

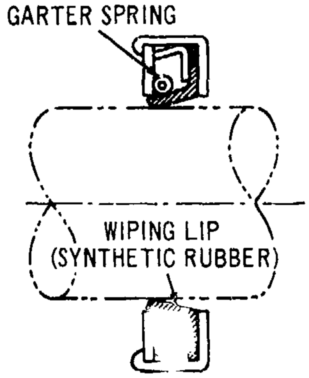


Fig. 7-42. Wiping Seal

7-3.2.5 Installation Considerations

Life expectancy of the bearings and seals described in the foregoing paragraphs is heavily dependent on the care taken in the installation process.

7-3.2.5.1 Lubrication

In addition to lubricating sliding and rolling surfaces, lubricants must dissipate bearing heat. Lubrication requirements are detailed in par. 7-3.3.

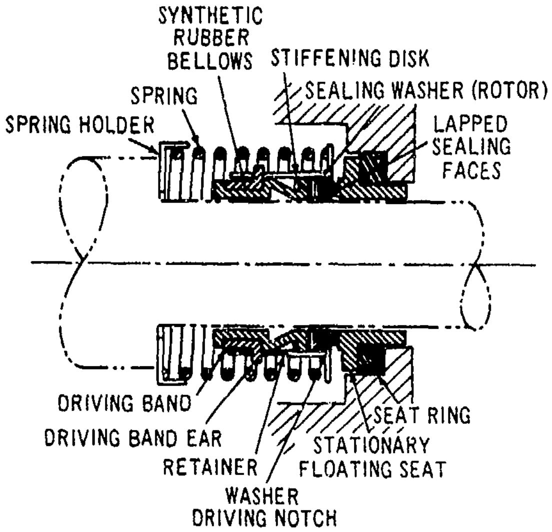


Fig. 7-43. Face Rubbing Seal

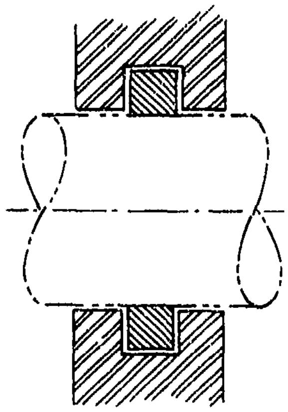


Fig. 7-45. Floating Seal

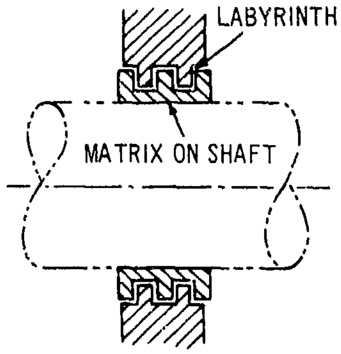


Fig. 7-46. Labyrinth Seal

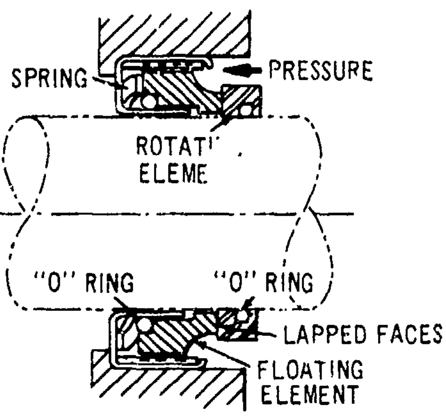


Fig. 7-44. Pressure-balanced Seal

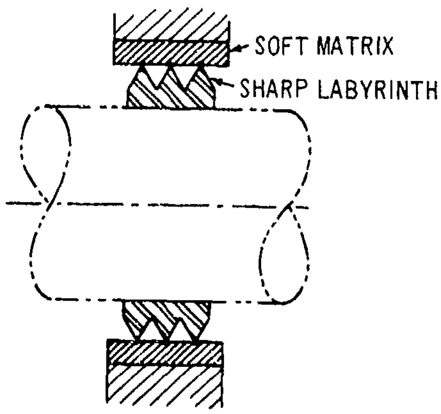


Fig. 7-47. Mating Labyrinth

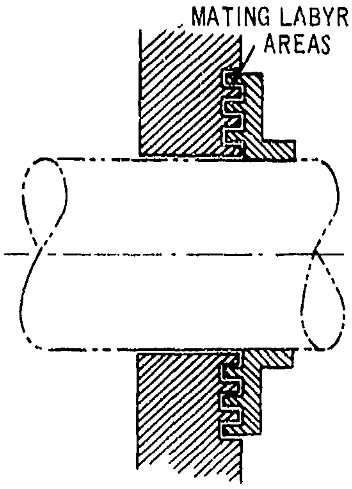


Fig. 7-48. Radial Labyrinth Seal

7-3.2.5.2 Inner and Outer Ring Shaft and Housing Fits

Installations of both inside and outside diameters of bearings must be tight enough to minimize fretting, creeping, or spinning of the bearing rings. Tightness of fit must not eliminate radial play of the bearing in operation. Approximately 80% of the steel-on-steel press-fit on a solid shaft will show up in radial play reduction. External bearing surfaces sometimes are silver-plated to minimize fretting and ring creep.

Inner and outer ring shoulder faces must be mounted to shaft and housing faces that are at least equally as square as the faces of the bearing selected. This insures proper bearing alignment, which is especially important at high speeds or high loads.

Proper selection of materials for inner and outer bearing ring and rolling elements is dependent upon the temperatures and life reliability requirements. Examples include:

1. Vacuum melt or vacuum degassed, SAE 52100 steel: life and reliability approximately 1.5 to 2 times that of air-melt steel

2. AISI 440 C steel: for corrosive resistance and operating temperatures to 400°F

3. Vacuum melt, M-50 tool steel: life factor of 3 to 5 times of air melt SAE 52100; steel also usable up to 500°F.

Cage selections also have a definite influence on life:

1. Stamped and riveted pressed steel: for normal applications

2. Nonmetallics such as synthane: where minimum lubrication exists and maximum quiet is desired

3. Bronze cages: for moderate speeds and temperatures

4. SAE 4340 steel cages, silver-plated: for high speeds, high loads.

7-3.2.6 Common Problems in Bearing and Support Application

Helicopter development and operation have been hampered in some instances by mistakes in the selection and treatment of bearings and mountings. Careful attention to the causes and solutions of these past difficulties should help insure proper design of bearings and mountings in future applications. Several case histories are discussed in the paragraphs that follow.

In the development of a particular helicopter transmission the antirotation pin interfered with the keyway in the outer ring of the bearing (Fig. 7-22). Outer ring misalignment and cage breakage resulted. A proposed solution for this problem was to eliminate the keyway, provide freedom of alignment between bearing rows, and add a resilient mount, as in Fig. 7-33. However, the laminated bearings could not tolerate the environment.

Another example problem was in a compound planetary gear and bearing system, where the lightweight mounting bosses with keyways caused inner ring fatigue by distortion. The end retainer plates were inadequate to hold the rings firm. The solution in this case was to grind inner raceways integral to the gear blank to eliminate keys, support, and bearing fit problems.

Fig. 7-49 shows outer ring support in a compound planetary carrier. Deflection under load caused these bearing outer rings to misalign with respect to the shaft centerline and thus create outer ring fatigue. The solution was to make a one-piece carrier to support the outer rings, and a shorter inner ring shaft assembly, as shown in Fig. 7-50.

Some antirotation and fretting problems have been solved by silver-plating the external surfaces of the bearing rings. The resultant press-fit and separation of the fretting surfaces solved the problem.

Axial thrust in roller bearings has caused extreme roller end wear. Careful attention to methods to prevent thrust loads on these bearings and close control of the alignment of both inner and outer rings solved this problem. Closely related is the problem of outer ring flange mounting; the asymmetrical flange, shown in Fig. 7-30, must be installed with care so that the mounting bolts do not distort the ring. Also, mounting diameters (both shaft and housing) must be round and well-aligned to prevent roller-skewing tendencies. All these problems are amplified by high speed.

The problem of roller skid has been serious in this same roller (Fig. 7-30). The solutions mainly are to reduce the roller size and inertia to prevent skids during the change from unloaded to loaded areas. Minimum radial play and close attention to ring alignments are necessary. Out-of-round shafts or elliptical housings and inner ring cage guidance serve to keep rollers running at constant speed and thus prevent skidding.

7-3.2.7 Fatigue Considerations

Correct bearing retention pla ... an important role in fatigue calculations. In all rolling contact bearings—whether ball, roller, needle, or others—the achievement of normal calculated fatigue life makes certain assumptions:

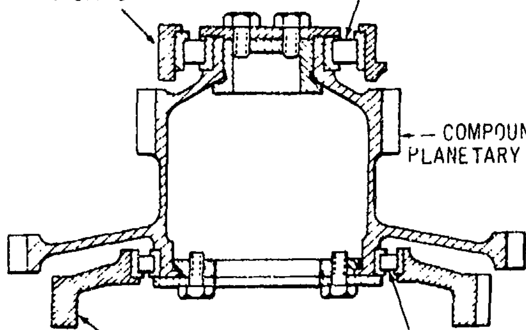


Fig. 7-49. Compound Planetary Gear I

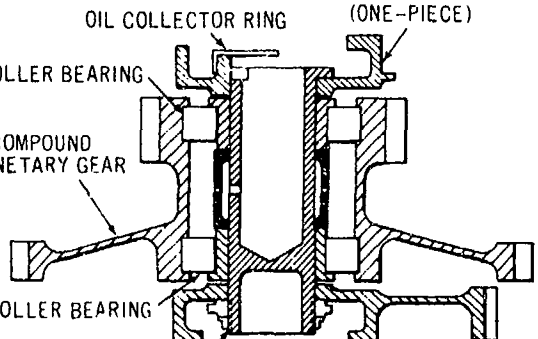


Fig. 7-50. Compound Planetary Gear II

1. That the lubrication is adequate to support the load and carry away the excess heat of friction
2. That bearing geometry of races and clearances is maintained by proper press-fits of inner and outer rings, and by mounting shoulders that are normal to the mounting diameters of the rings
3. That normal loads (radial and thrust) as well as unbalance and critical frequencies are accounted for.

The achievement of all of these is necessary in order for the normal calculated bearing fatigue life to be attained. If these conditions are not maintained, life calculations will have no validity.

7-3.3 LUBRICATION

Lubrication in helicopters serves two purposes. The first and most obvious is to provide a suitable fluid film between metal surfaces with relative motion. These include gears, splines, bearings, and overrunning clutches. The second, and equally important, function is to provide a medium for heat transfer from the working surfaces. The heat usually will be transferred to air either through the surface of the transmission or through an oil cooler.

7-3.3.1 Lubrication Systems

Lubrication systems fall into two basic categories. The first is a simple splash system in which the gears in the unit provide the impetus for oil circulation. The second is a circulating system in which a pump is used to provide oil under pressure to jets that spray onto critical surfaces. The circulating system may be either wet or dry sump.

7-3.3.1.1 Splash Lubrication System

In a splash lubrication system one or more gears dip into the lubricant. Splash or spray from the gears must be directed to any other gears not dipping in oil so that at least one member of each mesh receives lubrication. In addition, a flow of oil must be directed to all bearings. Splash-lubricated systems are cooled only by radiation and convection from the surface of the gearbox, limiting the application to relatively low-power systems. Typical applications include tail rotor gearboxes and simple one- or two-mesh rotor transmissions for small helicopters.

7-3.3.1.2 Circulating Lubrication Systems

In these systems, oil under pressure is passed through a filter and then through jets that direct oil spray onto gears and bearings. It may be possible to use some oil to lubricate two or more parts in series. Oil from bearings and gear meshes in the top of a transmission may drain through other bearings or across the stationary ring gear of a planetary reduction. Designers generally depend upon the mist within the transmission to lubricate accessory gears and bearings unless they are located in isolated areas with respect to the main power gearing.

In a wet sump system, one in which the oil storage area is the bottom of the transmission, only one lubrication pump is required. Some cooling of the oil will occur through the walls of the transmission housing. This can be increased significantly if the sump area is finned and subjected to an air blast. In higher power transmissions, a separate oil cooler probably will be required. This will be in the form of a heat exchanger. Airflow through the heat exchanger may be from rotor downwash or a small fan; such fans are driven by a hydraulic or electric motor or mechanically from the transmission accessory section.

A dry sump system generally is used where the space envelope, control system, or other requirements restrict the area at the bottom of a transmission. Transmissions subjected in service to a wide range of attitudes also often are made dry sump. These systems require an oil reservoir and a minimum of two oil pumps, one for supplying oil under pressure to the system and one to scavenge the sump area.

The scavenge pump generally will be required to have 2.5 to 3 times the capacity of the pressure pump. As a result, there will be air entrained in the oil as it enters the antivortex reservoir. The reservoir design should incorporate an effective entry baffle to de-aerate the oil and should have an antivortex baffle at the outlet.

A common lubrication system for engine and transmission or for more than one transmission *shall* be avoided because there could be a transfer of contaminants from engine to transmission, transmission to transmission, or vice versa.

Oil jets will be used primarily at gear meshes and rolling element bearings in the principal load paths. Oil jets also are used to control and direct oil flow for overrunning clutches and for centrifugal oiling systems for the bearings in planetary gears.

Oil jets on both the entrance and exit sides of a gear mesh have been used satisfactorily for gears operating at more than 20,000 fpm pitch line velocity. However, oil jets at the entering side of the mesh for spur gears with face widths greater than about 1 in. and operating in the range of 10,000 fpm or higher can develop dynamic loading and could cause gear failure (Ref. 2). This may be particularly critical if large amounts of oil are required for cooling. In general, cooling is achieved best by jetting oil at the leaving side of the mesh. Lubrication is achieved best by jetting oil at the entering side. In some critical applications oil is applied at both sides, with perhaps 1/5 of the oil applied at the entering side of the mesh, and 4/5 at the leaving side.

Rolling-element bearings generally are lubricated best by jetting the required amount of oil into the gap between the cage, or separator, and the inner race. The angle of oil impingement is particularly critical for high-speed bearings. Windage effects must be considered for oil entering high-speed bearings and adequate escape passages must be provided so that oil is not trapped.

Overrunning clutches of either the sprag or spring type require submerging the sliding surfaces in oil to achieve satisfactory life. This generally is accomplished by damming the required area and flowing only a small volume of oil into the clutch.

Oil distribution to the various jets of the transmission should be accomplished with internal oil passages. External oil lines are subject to damage in handling or in servicing the unit. Provisions for operation without lubrication *shall* be fail-safe, and *shall* not be subject to deterioration over the life of the transmission, and *shall* not be capable of generating debris that can clog oil jets.

7-3.3.1.3 Lubricating Oil

Current general practice with helicopters is to use the same oil in transmissions as is used for the engines. These are lubricated with synthetic lubricants per MIL-L-7808 or MIL-L-23699. While both of these specifications are satisfactory, MIL-L-23699, having a higher viscosity, may be a slightly better gear lubricant for normal conditions. However, MIL-L-7808 has a lower pour point, making it more desirable for low-temperature operation.

7-3.3.2 Lubrication System Components and Arrangement

Various lubrication accessories, and the requirements for these in different lubrication systems, are shown in Table 7-1.

The relative positions of the items of Table 7-1 for the wet sump and dry sump systems are shown in Figs. 7-51 and 7-52, respectively. The wet sump system in Fig. 7-51 is a system with integral oil cooling at the sump with an air blast. If external cooling were required, it would be connected between the pressure relief valve and the oil distribution gallery.

TABLE 7-1
LUBRICATION SYSTEM COMPONENTS AND FEATURES

COMPONENT	SYSTEM: SPLASH	WET SUMP	DRY SUMP
BREATHER	REQUIRED	REQUIRED	REQUIRED
CHIP DETECTOR	REQUIRED	REQUIRED	REQUIRED
DRAIN	REQUIRED	REQUIRED	REQUIRED TWO PLACES
FILLER CAP	REQUIRED	REQUIRED	REQUIRED AT RESERVOIR
FILTER	NOT APPLICABLE	REQUIRED	REQUIRED
MAGNETIC PLUG	REQUIRED	REQUIRED	REQUIRED
OIL COOLER	NOT APPLICABLE	MAY BE REQUIRED	REQUIRED
OIL LEVEL INDICATOR	REQUIRED	REQUIRED	REQUIRED AT RESERVOIR
OIL STRAINER	NOT APPLICABLE	REQUIRED	REQUIRED
PRESSURE TAP	NOT APPLICABLE	REQUIRED	REQUIRED
PROVISION FOR OIL SAMPLING	REQUIRED	REQUIRED	REQUIRED AT RESERVOIR
PUMP, PRESSURE	NOT APPLICABLE	REQUIRED	REQUIRED
PUMP, SCAVENGE	NOT APPLICABLE	NOT APPLICABLE	REQUIRED
RELIEF VALVE	NOT APPLICABLE	REQUIRED	REQUIRED
RESERVOIR	NOT APPLICABLE	NOT APPLICABLE	REQUIRED
TEMPERATURE INDICATOR	REQUIRED	REQUIRED	REQUIRED TWO PLACES
VENT LINE	NOT APPLICABLE	NOT APPLICABLE	REQUIRED

7-3.3.3 Oil Flow Requirements

Sufficient oil flow must be provided to maintain acceptable temperature levels. The total oil flow V can be calculated:

$$V = \frac{Q}{7.5 c_p \Delta T}, \text{ gpm} \quad (7\text{-}24)$$

where

Q = heat loss, Btu/min
c_p = specific heat of oil, Btu/lb-°F
ΔT = oil temperature rise, °F

Methods of predicting transmission heat rejection are shown in par. 7-3.4. Part of this heat will be dissipated through the transmission housing. Botstiber and Kingston (Ref. 3) show the following conservative method for estimating this heat rejection:

$$Q = \frac{A \Delta T}{C}, \text{ Btu/min} \quad (7\text{-}25)$$

where

Q = dissipated heat, Btu/min
A = exposed area of gear case, ft^2
ΔT = temperature difference between gear case surface and ambient air, °F
C = 30 ft^2-°F-min/Btu for still air
C = 22 for free circulation
C = 10 for forced airflow

If the heat is not all dissipated through the housing, the remainder must be disposed of through an oil cooler. Sufficient oil flow must be provided to maintain acceptable temperature levels.

7-3.4 HEAT REJECTION AND TRANSMISSION EFFICIENCY

Gears are the most efficient mechanical means yet devised for the transmission of power from one shaft to another. With well-made spur and internal gears the power loss is approximately 0.5% of the power transmitted through the mesh. Although it might appear that this small loss could be neglected, it constitutes a heat source that will cause a significant temperature rise. If the heat is not removed properly from gears and bearings, the surface temperatures will rise beyond the metallurgical limitations of the materials and failure will occur.

The efficiency of a transmission is variable and depends upon many factors. These include lubricant, pitch line velocity, gear geometry and surface finish, type of bearings, accuracy of the gears, bearings and mountings, and housing design. The transmission efficiency is dependent upon the magnitude of five types of power losses:

1. Gear mesh losses
2. Bearing losses
3. Losses due to windage and oil churning
4. Seal losses
5. Losses due to lube pumps.

The first two of these are quite predictable once the design details are established. Losses due to windage and oil churning are less predictable.

7-3.4.1 Gear and Bearing Losses

For a given load, the power loss in a gear mesh is dependent upon pitch line velocity, tooth geometry, and tooth surface finish. This means that—with the higher pressure angles—smaller, or finer pitch, gears can be used. These same factors have an effect upon the scoring tendency of gear teeth and are related directly to the current tendency toward finer pitch teeth and higher pressure angles. The tooth surface finish affects the coefficient of friction of the gears and the ability of the oil film to separate the mating surfaces. Surface finish is not usually an appreciable variable in helicopter gear efficiency because the teeth generally are finished by grinding or honing.

The losses in rolling-element bearings are related to load, speed, cage design, and coefficient of friction. The load is a function of the gear design, shaft, and transmitted power. Speed is a function of shaft and bearing diameter. The coefficient of friction is a function of the type of bearing, i.e., ball or roller.

7-3.4.1.1 Preliminary Gear and Bearing Loss Predictions

Losses in gears and bearings cannot be calculated accurately until their design is complete. However, reasonably accurate predictions can be made for transmissions using the parallel and intersecting shaft types of

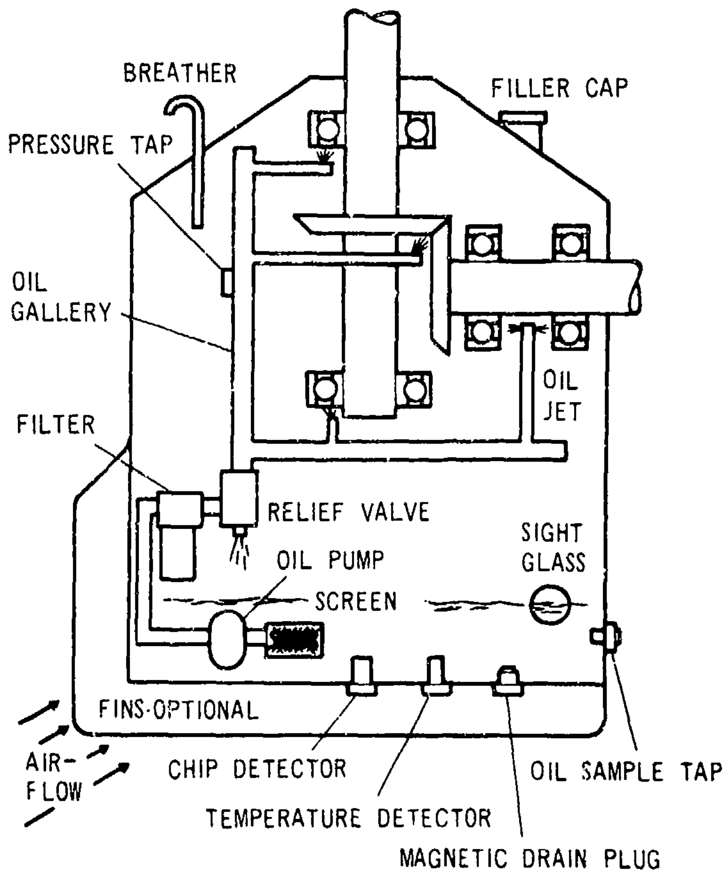


Fig. 7-51. Wet Sump Lubrication System Schematic

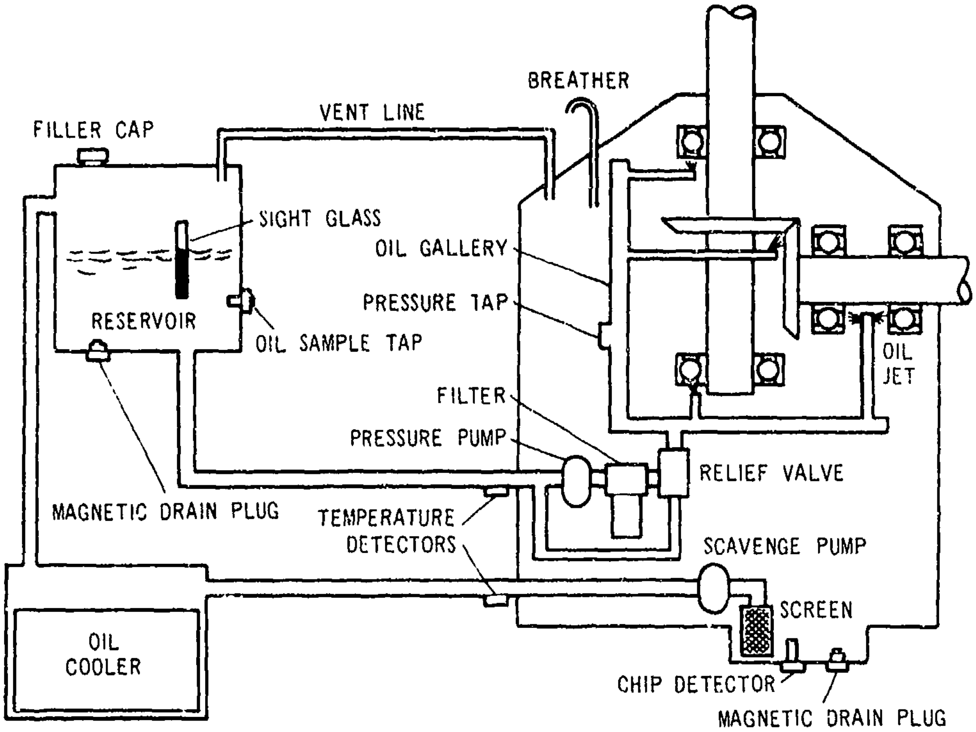


Fig. 7-52. Dry Sump Lubrication System Schematic

gearing most common in helicopter transmissions. For each simple gear mesh the power loss probably will be 0.5% of the transmitted power including gear and bearing losses. For a simple planetary system with the sun gear input and carrier output, the combined gear and bearing losses probably will be about 0.75%. These factors are based upon accurate gears finished by grinding or honing and operating in rolling-element bearings. If the horsepower transmitted by each mesh is multiplied by the probable percentage power loss and the results totaled, a reasonable estimate of total transmission power loss will result.

7-3.4.1.2 Gear and Bearing Loss Calculations

There are many different techniques for computing the power loss in gears. A good system covering all types of gears is presented in Ref. 4. Calculations for losses in rolling-element bearings are reasonably well standardized; a typical set of formulas and supporting data is found in Ref. 5.

7-3.4.2 Windage and Churning Losses

Windage is nothing more than the air movement created by rotating machinery. Losses due to windage are affected by the size, proportions, and speed of the rotating elements; by the housing design; and by case pressure. A further discussion of windage loss, and formulas and supporting data for its approximation, can be found in Ref. 6.

Churning is the effect of mechanical working of the lubricant, either by rotating machinery or by windage. The worst conditions occur when a volume of oil is trapped in some cavity in the transmission housing. The principal concerns of the transmission designer are to eliminate trapped pockets of oil, to avoid situations where windage acts as a screen to prevent oil from draining, and to avoid high-energy impingement of oil on housing surfaces or gears. These design considerations increase in importance with increased gear pitch line velocities. Problems can occur at pitch line velocities of 8000 to 10,000 fpm and the probability of difficulties increases with increased speed.

Although churning losses are not predictable, there are several design approaches that are effective in reducing or eliminating them. Some of these are:

1. Allow a large clearance space around rotating parts such as gears with high surface velocities. This

will help to reduce oil trapping on one side of the windage pattern.

2. Provide oil drain paths from both sides of rolling element bearings, even if oil is applied at only one side.

3. Where oil must drain past a high-speed gear, provide a drain passage sheltered from the windage.

4. Use gear scrapers and deflectors adjacent to high-speed gears to direct the oil flung from the gears toward the transmission sump. This technique makes use of the kinetic energy of the oil to aid in scavenging.

5. Place fine mesh screens around the high-speed gears to dissipate the kinetic energy of the oil flung from the gears and to break up the windage pattern.

Windage and churning losses are not affected by the amount of power transmitted. They are affected by speed and oil flow.

7-3.4.3 Conversion of Losses to Heat Loss and Efficiency

The mechanical efficiency of the system can be obtained readily from the tabulated losses by the formula:

$$\text{Efficiency} = \frac{100\ (\text{Power Input} - \text{Power Loss})}{(\text{Power Input})}$$

The heat loss in Btu/min can be found by multiplying the horsepower loss by 42.44.

7-4 TRANSMISSION DESIGN

7-4.1 MAIN ROTOR TRANSMISSION

The primary function of the main rotor transmission is to transmit the power from the engines or combining transmission to the rotor. The torque supplied to the rotor results in reaction loads of thrust, moment, and side force in addition to torque that must be reacted by the transmission. An alternate solution to the rotating shaft is to provide a stationary mast or standpipe that carries the rotor support bearings and transmits all loads except torque directly to the helicopter structure, independent of the main rotor transmission. A quill from the transmission supplies only torque to drive the rotor.

The transmission normally transmits all loads via its mounts to the helicopter structure except when the stationary mast concept is employed. Where possible, it is desirable to react all rotor loads to the airframe through a forging that is located so as to prevent rotor loads from passing through any gear support housing. Location of mounts depends upon the type of airframe structure.

Usually the transmission is a reducing drive to give low speed and high torque at the rotor. Typically, the axis of rotation of the input is nearly horizontal and the output approximately vertical. Changes in direction of rotation also may be required as in tandem rotor transmissions that must drive rotors in opposite directions to balance torque reaction.

The main rotor transmission may serve as a combining transmission for multiengine inputs or receive only a single input from a single engine or a separate combining transmission.

Positioning of the rotor and rotor controls is another primary function performed by the main rotor transmission or in combination with a stationary mast. The rotor axis usually tilts forward a few degrees in the direction of forward flight. The forward tilt gives a forward force component from the basic rotor thrust (lift) necessary for forward flight. Tilting the rotor reduces the amount of rotor control motion required and minimizes the amount of helicopter nose-down attitude for high-speed forward flight.

For a single-rotor helicopter the rotor shaft also may have a side or lateral tilt of a few degrees to give a side force component from the rotor thrust (lift) to offset the side force component of the tail rotor thrust. Side tilt of the rotor shaft results in level helicopter attitude in lateral or roll direction. Both the forward and side tilt of the rotor shaft result in lower moments and side loads on the shaft and support bearings compared to true vertical mounting of the rotor shaft. The forward tilt usually is built into the transmission in order to hold the input or engines horizontal. However, the side tilt may be accomplished by tilting the entire drive system or at least the main transmission. Offset mounts in the transmission or the structure can be used.

The rotor hub and blade design may dictate a requirement to provide a central hole or tunnel up through the rotor shaft for internal controls, hydraulic power for blade control, and electrical cables for blade folding and blade deicing. The hole also could provide access for instrumentation wires.

A main rotor transmission for a single-rotor helicopter also must drive the tail rotor required to counteract the torque reaction of the main rotor. Typically, the power to drive the tail rotor is 10-12% of the total power.

The transmission system for overlapped tandem rotor configurations must provide mechanical synchronization to index the rotors to avoid collision of blades.

Transmissions now used in helicopters are of the closed type with the housing and seals serving to contain the lubricant and to exclude foreign material.

There are a number of functions that could be considered secondary because they normally are not limited to the main rotor transmission. These include:

1. Drive and support of lube pump, oil cooler blower, power takeoff, rotor brake, and accessories. The lubrication pump should be direct-driven downstream of the input clutch to provide positive lubrication during autorotation or when one engine is shut down.

2. Support for rotor control actuators and related linkage

3. Provision for dephasing rotor drive for manual blade folding

4. Rotor positioning device and lock for power blade folding

5. Axial and radial restraint of external drive shafting

6. Monitoring system (oil temperature, oil pressure, filter pressure drop, screen detector, chip detector, oil level, vibration pickup, and oil jet pressure drop).

With the current trend to multiengine drives for increased safety, a clutch is required for each engine input, thus fixing clutch location on the input side of the main transmission rather than at the output or rotor shaft. Clutches near the engine input serve both main functions of single-engine shutdown and autorotation. A clutch at the rotor shaft adds extra protection in case of seizure of the main rotor transmission, but at the expense of additional weight to achieve the required clutch capacity. A seizure in the main rotor transmission is not likely to stop the rotor, due to the large inertia of the rotor. A quill or shaft shear section will shear and free the rotor. Such a shear section often is included in transmission design to insure drive separation at a known location rather than risking a random separation that could jeopardize the structural integrity of the gear case.

Figs. 7-53 through 7-58 show several currently used main rotor transmissions illustrating typical gear, bearing, and mounting arrangements. Horsepower ratings are approximate and are given for size reference.

1. Fig. 7-53, CH-47 Forward Transmission. This is a forward tandem-rotor drive transmission with single input from synchronizing shaft. First stage gearing is spiral bevel with second and third stages simple spur planetary using a common ring gear. Total reduction ratio is 30.72.1; power rating, 3600 hp. Mounts are in a forged upper cover.

2. Fig. 7-54, CH-47 Aft Transmission. This is an aft tandem-rotor drive transmission with single input from synchronizing shaft. First-stage gearing is spiral bevel with second and third stages simple spur planetary using a common ring gear. Total ratio is 30.72:1; power rating, 3600 hp. Mounts are in a forged upper cover. The high torque output shaft extends the drive to the elevated rear rotor.

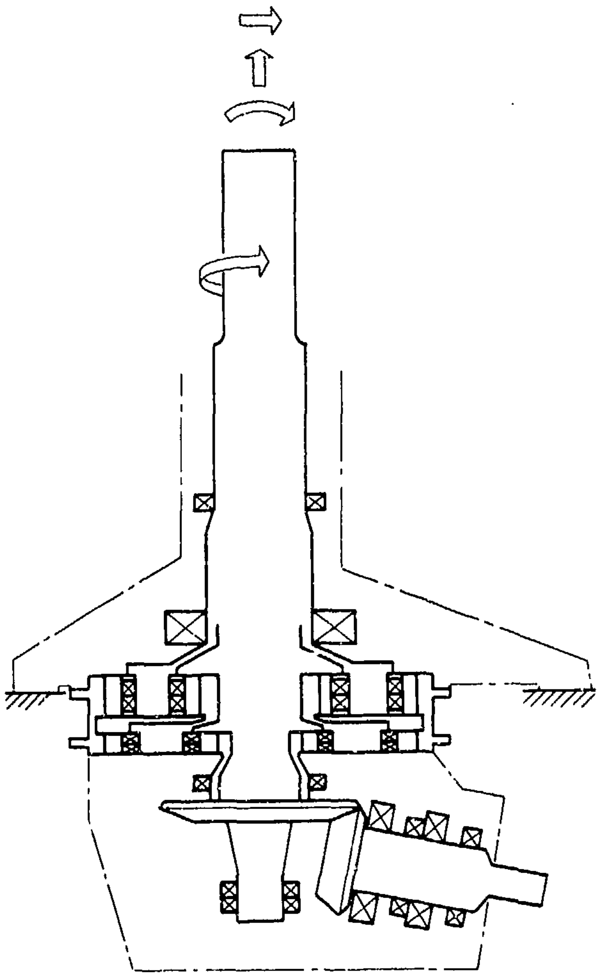


Fig. 7-53. CH-47 Forward Transmission

3. Fig. 7-55, CH-54 Main Transmission. This is a single-rotor drive transmission with two inputs direct from two engines. First- and second-stage gearing is spiral bevel with third and fourth stages of simple spur planetary. Mounts are at the bottom of the cast main housing. Total ratio is 48.6:1; power rating, 8100 hp. Note that planetary stages are below input.

4. Fig. 7-56, OH-6 Main Transmission. This is a single-rotor drive transmission with one input direct from the engine. Two stages of gearing are spiral bevel.

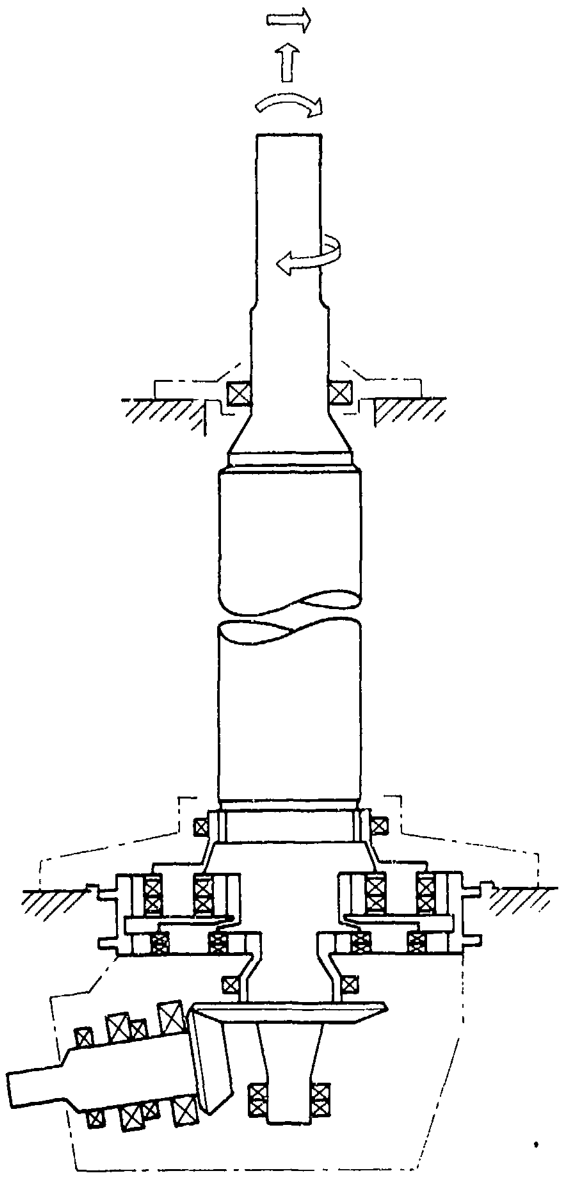


Fig. 7-54. CH-47 Aft Transmission

Rotor is supported by a stationary mast. Transmission mounts are in the upper cover. Total ratio is 12.81:1; power rating is 208 hp continuous and 268 hp takeoff.

5. Fig. 7-57, OH-58 Main Transmission. This is a single-rotor drive transmission with one input direct from the engine. First-stage gearing is spiral bevel with a second-stage simple spur planetary. Total ratio is 15.23:1; power rating is 270 hp continuous and 317 hp takeoff.

6. Fig. 7-58, MIL-10 Main Transmission. This is a single-rotor drive transmission with two inputs direct from two engines. A load-sharing device in each input drives two spiral bevel pinions for the first-stage reduction. The second stage is a spur mesh with four pinions against a common gear. The final stage is a split power spur planetary. Mounts are in the center section cast housing. Total ratio is 69.2:1; power rating, 11,000 hp.

7-4.2 ANTITORQUE ROTOR TRANSMISSION

The primary purposes of the tail gearbox are to achieve the required change in shaft angle, transmit power to the antitorque rotor, and provide adequate mounting for the tail rotor. The lateral thrust produced by the tail rotor counteracts the main rotor torque, thereby providing a positive means of directional control for the helicopter. The tail rotor position (fuselage station) is far enough aft so that the tip paths of the two rotors do not intersect. A typical tail gearbox arrangement is shown in Fig. 7-59.

Input to the tail rotor gearbox is from a drive shaft connected to the input pinion shaft through a flexible coupling. The flexible coupling must be capable of operating at small misalignment angles of up to 1 deg. Misalignment results from mounting tolerances and also from airframe deflections that are caused by inertia loads from maneuvers and by wind loads on the fuselage and tail supporting structure.

The speed reduction in the tail gearbox usually is accomplished by a spiral bevel gear mesh operating at a 90-deg shaft angle. The ratio in this gearbox with single bevel mesh seldom is greater than 3:1. For larger helicopters, it may be advantageous to use super-critical drive shafting, in which case the tail rotor gearbox will require a larger reduction ratio. Several designs have proposed planetary reductions along with the 90-deg spiral bevel mesh to obtain overall reduction ratios of 7 to 1 or more.

Gearbox design follows conventional procedures; the bevel gears are designed for unlimited life at maximum power (excluding transients). The maximum power usually is required during a maneuver such as left yaw at maximum rate and is determined from the design of the tail rotor blades at maximum collective pitch. Under these conditions the gearbox will draw whatever power is required to rotate the tail rotor blades at the pitch and rpm available. The normal maximum steady-state power absorbed by the tail rotor is usually from 10 to 12% of the total helicopter power. For the short-time transient condition, the tail rotor power can exceed 30% of the total usable horsepower. The required strength of spiral bevel gears may be determined by a computer program that analyzes a particular design

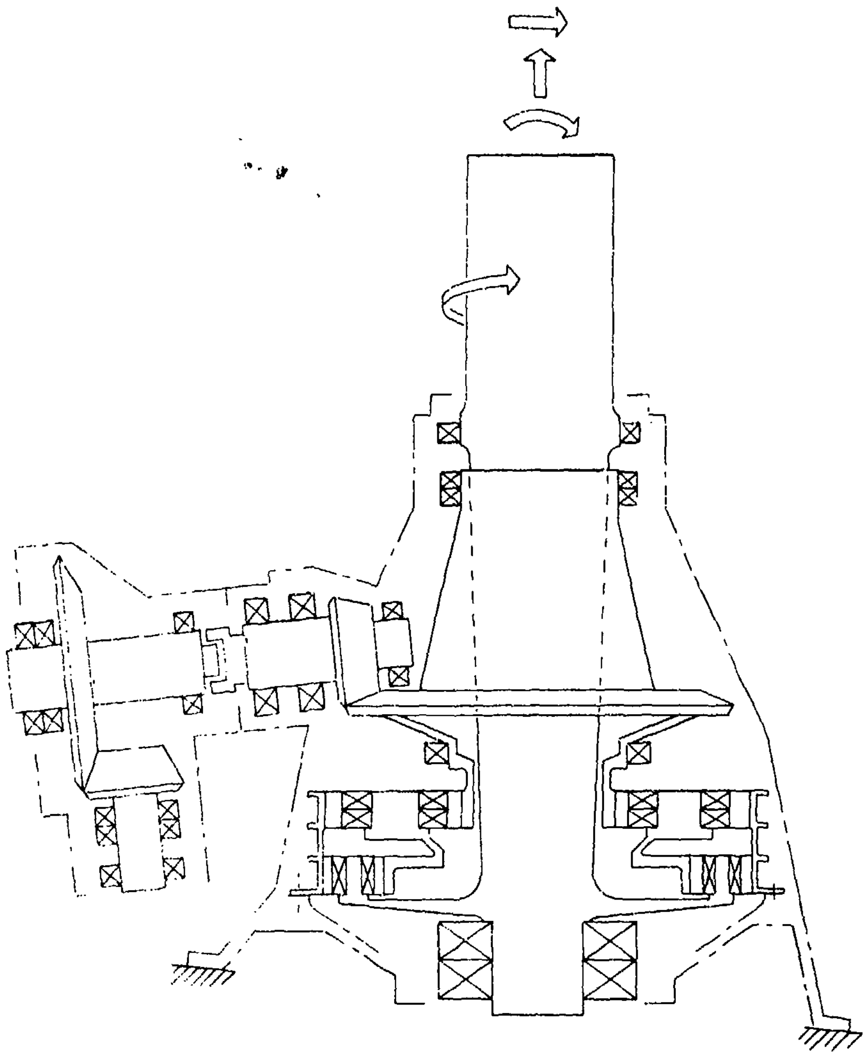


Fig. 7-55. CH-54 Main Transmission

and determines strength factors, contact ratios, and load sharing factors. Design iterations are made until the smallest size is found that is within predetermined limits. Another computer program calculates the basic machine settings required to cut and grind the teeth to size. Small adjustments are made to the basic tooth profile during the bevel gear development test program; the gears are tested and the profile is modified to produce the required load pattern when run with a mate or with the reference master gear.

Bearing loads come from two sources: the bevel gear loads and the aerodynamic rotor forces acting upon the tail rotor, which are transmitted to the structure via the rotor drive shaft bearings and the gearbox casing. The loads acting upon the gears, bearings, and shafting are derived from a complete power and load spectrum that is representative of the aircraft mission.

Due to the remote location of the tail rotor gearbox, the lubrication system usually is of the splash or self-contained type. However, larger designs have included electrical oil pumps or oil pumps driven from a spur gear driven by the input pinion. In the splash-lubricated system the large gear dips into the oil and carries and splashes it to the point of mesh as well as to the supporting bearings in the near vicinity. To lubricate remote bearings, a scupper is provided in the gearbox casing to catch some of the splash and deliver it to the bearings by gravity flow through lines cored in the housing. Optimum oil level must be determined in splash-lubricated systems. Too low a level will result in poor distribution and lack of lubrication whereas too high a level will result in unnecessary churning that wastes power and generates excessive heat. Leakage is prevented by the use of O-ring seals at housing interfaces and by lip-type seals on the input and output shafts.

With force-fed or splash-lubricated gears and bearings, allowable gearbox temperatures nominally are maintained at 250°F or less. Temperatures may exceed 250°F with no adverse effects but must be maintained below 350°F in order to avoid drawing of carburized parts. Refer to par. 7-3.3 for further discussion of lubrication.

Grease-lubricated antitorque rotor gearboxes are a distinct possibility. Gearboxes of this type inherently have a lower vulnerability to ballistic case penetration because of the smaller effect upon gearbox operation from loss of lubricant. However, grease-lubricated gearboxes may require special baffle arrangements and finned areas for proper heat rejection.

Tail rotor thrust is controlled by a pitch control shaft and pitch beam that collectively change the pitch on the tail rotor blades. The pitch control shaft is actuated by a control circuit that extends back to the cockpit. In the gearbox itself the design consists of a push-pull mechanism in which one end is nonrotating and is fixed to the housing, while the other end is rotating with the tail rotor shaft and controls the collective pitch of the tail blades. The nonrotating-to-rotating portion of the

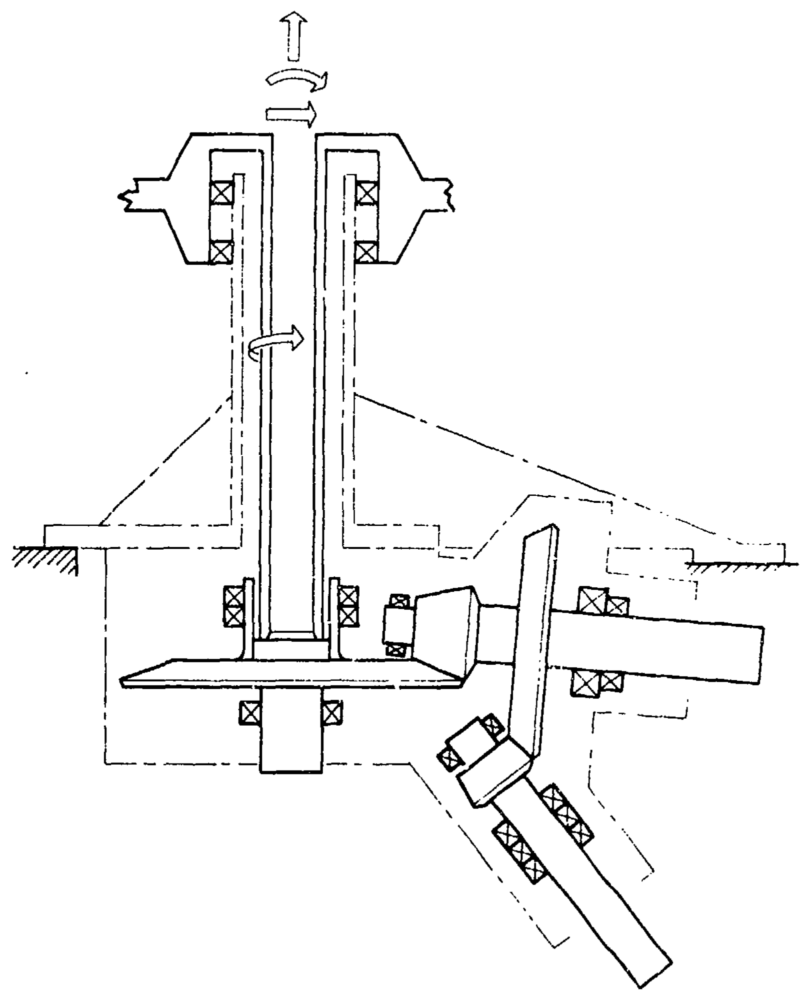


Fig. 7-56. OH-6 Main Transmission

control system usually consists of preloaded ball bearings. The pitch control shaft rotates with the tail rotor shaft by means of a splined connection that prevents relative rotational motion and permits relative axial motion between the tail rotor and the pitch control shaft.

The input bevel pinion shaft usually is straddle-mounted and is subject to steady torsion, vibratory bending, and steady axial loads from the input torque and from the bevel gear loads and reactions.

The output shaft that drives the tail rotor is subject to gear loads at one end and aerodynamic rotor loads at the other; thus, the stresses at any section are a function of hp and rpm (torque) and also of rotor loads that are not associated directly with power. If at any time during a flight the axis of the fuselage is rotated relative to the axis of the tip path plane of the tail rotor blades, a gyroscopic moment is induced upon the tail rotor hub and shaft due to the tendency of the rotor to remain in its present tip path plane location. The induced moment is proportional to the severity of the

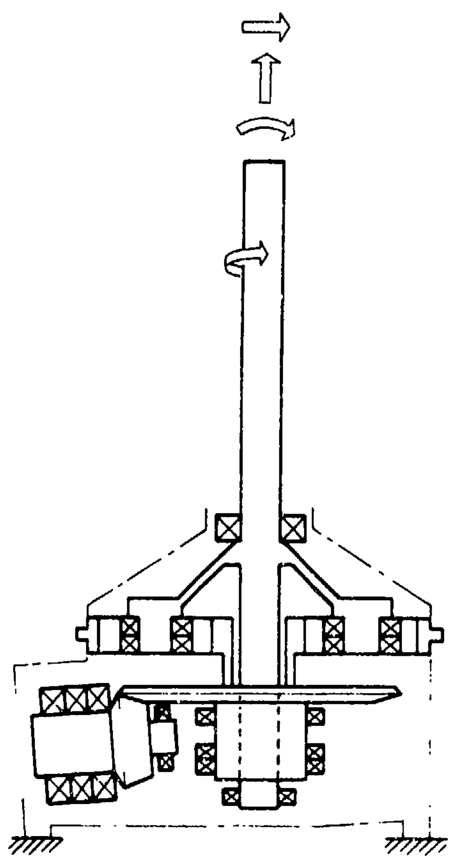


Fig. 7-57. OH-58 Main Transmission

maneuver and can be relatively high. This moment is transitory and only acts for short periods of time. For purposes of calculating loads on mounting attachments and bending moments in the tail rotor shaft, the gyroscopic moments conservatively are assumed to act 100% of the time.

The angle at any instant between the axis of the tail rotor shaft and an imaginary shaft perpendicular to the tip path plane of the tail rotor blades is called the flapping angle. During violent maneuvers the flapping angle can reach 12 deg. An average flapping angle is used to design the tail rotor output drive shaft bearings. The average flapping angle is calculated by the cubic mean method and is determined from an estimated spectrum of flapping angle versus percent time for the mission for which the helicopter is intended.

The tail rotor gearbox is mounted to the fuselage through a bolted flange connection. The loads and moments that this attachment must withstand are those produced by:

1. Input and output torques
2. Tail rotor thrust
3. Tail rotor hub moment (induced by flapping)
4. Weight of gearbox and rotor
5. Control loads.

Because each of these loads varies with maneuver, the worst combination should be used in this analysis. The loads and moments, which will dictate the design of the attachment, then are referred to the CG of the bolt pattern in the mounting plane. The inplane shears and moments produced will create shearing forces in each bolt while the out-of-plane loads and moments result in bolt tensions and compressions. Each bolt then will have an axial load and shear load acting upon it.

Because the bolts are installed with tensile preload, the resulting tension in the bolt P_b is given by:

$$P_b = P_r + KP_a \quad , \text{lb} \tag{7-26}$$

where

P_r = tensile preload, lb
P_a = applied tensile load, lb
K = relative stiffness of the bolt compared to the clamped members, dimensionless

The margin of safety for the bolt is based upon an interaction curve as shown (see Fig. 7-60) and may be expressed by the formula

$$MS = \frac{R_{sa}}{R_s} - 1 \quad \text{, dimensionless} \qquad (7\text{-}27)$$

or

$$MS = \frac{R_{ta}}{R_t} - 1 \quad \text{, dimensionless} \qquad (7\text{-}28)$$

where

R_s = applied shear load/shear strength, dimensionless
R_{sa} = allowable value of R_s (Fig. 7-60), dimensionless
R_t = applied tension load/tension strength, dimensionless
R_{ta} = allowable value of R_t (Fig. 7-60), dimensionless

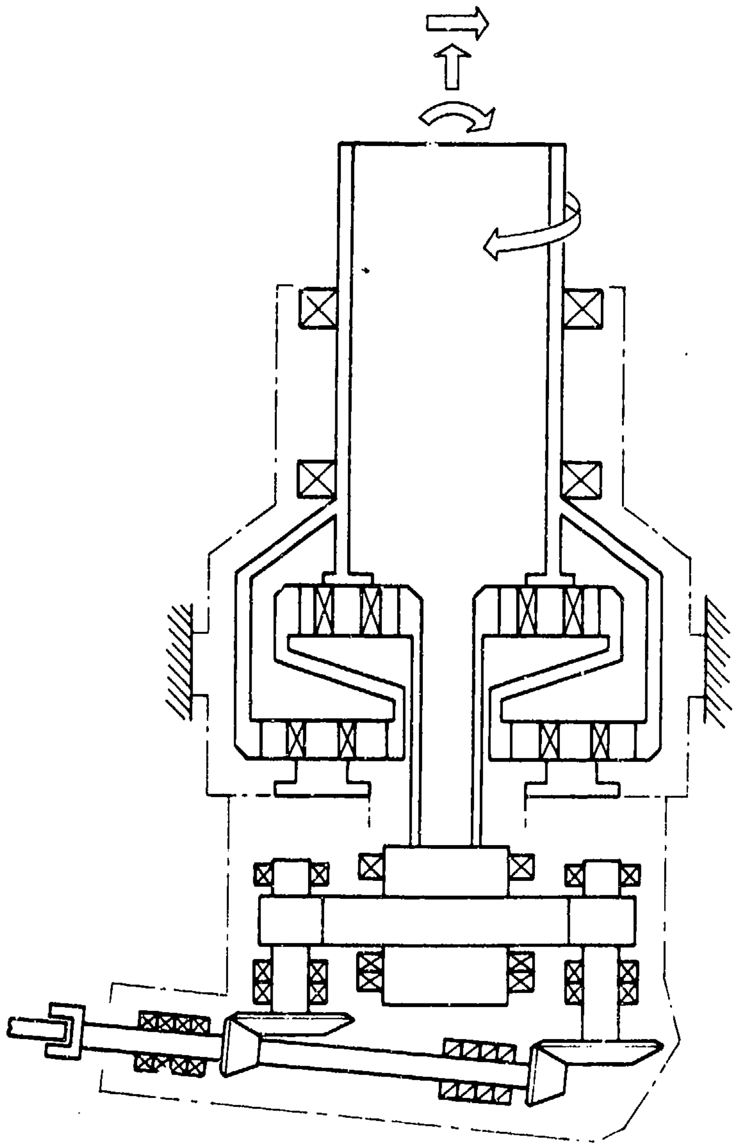


Fig. 7-58. MIL-10 Main Transmission

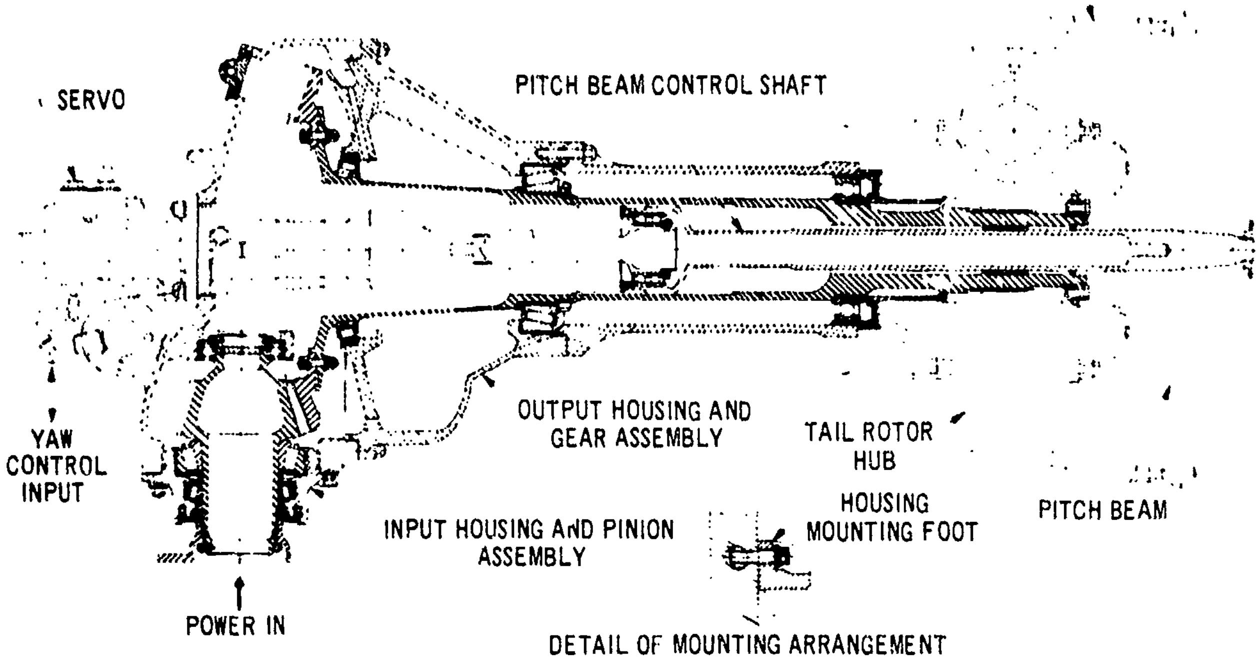


Fig. 7-59. Typical Tail Gearbox

The margin of safety of a bolt or screw may be determined from this curve as follows:

1. Plot the applied values of R_s and R_t (shown as Point 1).

2. Draw a straight line from the origin through this point to intersect the curve (Point 2).

3. The coordinates of Point 2 are R_{su} and R_{tu}, the allowable values of R_s and R_t.

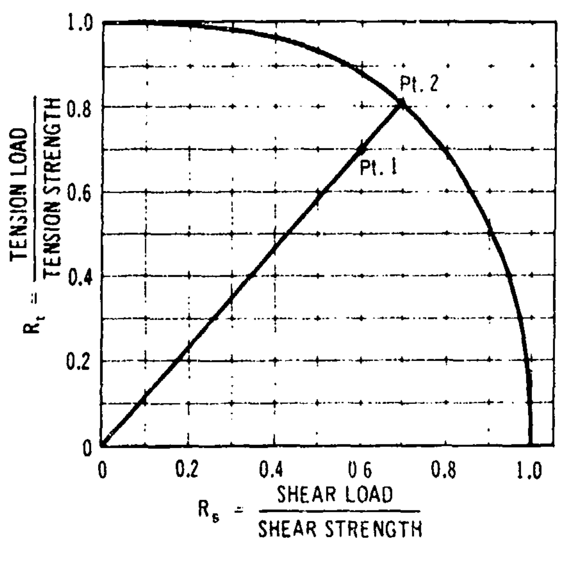


Fig. 7-60. Interaction Curve

The margin of safety is then computed using Eqs. 7-27 or 7-28.

The housing flange thickness at the attachment is based upon either of two criteria: (1) bearing stress in the bolt hole due to the shear load, or (2) bending stress near the casting wall due to the overhang moment due to the axial bolt load.

7-4.3 COMBINING TRANSMISSION

7-4.3.1 Functions

The primary function of the combining transmission is to join multiengine inputs to drive one or more rotor transmissions.

The intermediate transmission is used to transmit power from the main rotor transmission through a change in direction of axis to drive the tail rotor for single-rotor drive systems.

Usually the transmission is a reducing drive that takes advantage of the gearing required to combine inputs and change direction of axis and/or rotation.

Transmissions currently in use are of the closed type with the housing and seals serving to contain the lubricant and exclude foreign material.

The combining or intermediate transmissions normally are mounted to airframe structure and only have to react the torque from the transmitted load. External lift and moment loads from the rotor are not transmitted to the combining or intermediate transmissions.

7-4.3.2 Typical Combining Transmissions

Figs. 7-61 through 7-63 show a few currently used combining and intermediate transmissions with typical gearing, bearing, and mounting arrangements. Horsepower ratings are approximate, and are given for size reference.

1. Fig. 7-61, CH-46 Mix Box. This is a unit that combines the inputs of two 1500-hp engines and drives two main rotor transmissions for a tandem-rotor drive system. The first-stage gearing is herringbone and the second stage is spur. Total ratio is 7.61:1. Mounts are at top of the main casting housing. The clutch is at the second-stage pinion.

2. Fig. 7-62, CH-47 Engine and Combining Transmission. The engine transmission is an intermediate type that reduces speed and changes direction of axis from power plant to combining transmission. Ratio is 1.23:1. Power rating is 3750 hp. Gearing is spiral bevel. The clutch is at the output shaft. The combining transmission combines the input from two engine transmissions and provides two outputs to drive two main rotor transmissions for a tandem-rotor drive system. Gearing is spiral bevel with ratio of 1.70:1. Power rating is 6000 hp.

3. Fig. 7-63, BO-105 Intermediate Transmission. This transmission is a typical single-rotor intermediate type unit with input from a main rotor transmission and output driving a tail rotor. Gearing is spiral bevel with ratio of 1.25:1 step-up. Power rating is 70 hp.

7-4.4 ENGINE/ROTOR CROSS-SHAFTING

Power transmission shafting systems suitable for helicopter applications are high-power, high-speed, and lightweight. In spite of this unique combination of design requirements, however, it is an absolute necessity that the designer recognize that the engine/rotor cross-shafting systems must remain rugged and reliable, and that he place continued emphasis on system safety.

The design goal for each vehicle concept and shaft system *shall* be to use the lightest structure compatible with the concept. To achieve this goal, the designer must consider structural adequacy, component arrangements, and realistic design criteria, and place continued emphasis on weight control. Shaft system design *shall* be based upon general compliance with the guidelines defined by MIL-S-8698, MIL-T-5955, and MIL-T-8679, and with MIL-A-8860 through MIL-A-8870, as appropriate.

The traditional system safety technique used to achieve shaft system integrity is the "safe-life" method. This technique predicts the component service life by means of a rigorous testing program—including functional, structural, and flight tests.

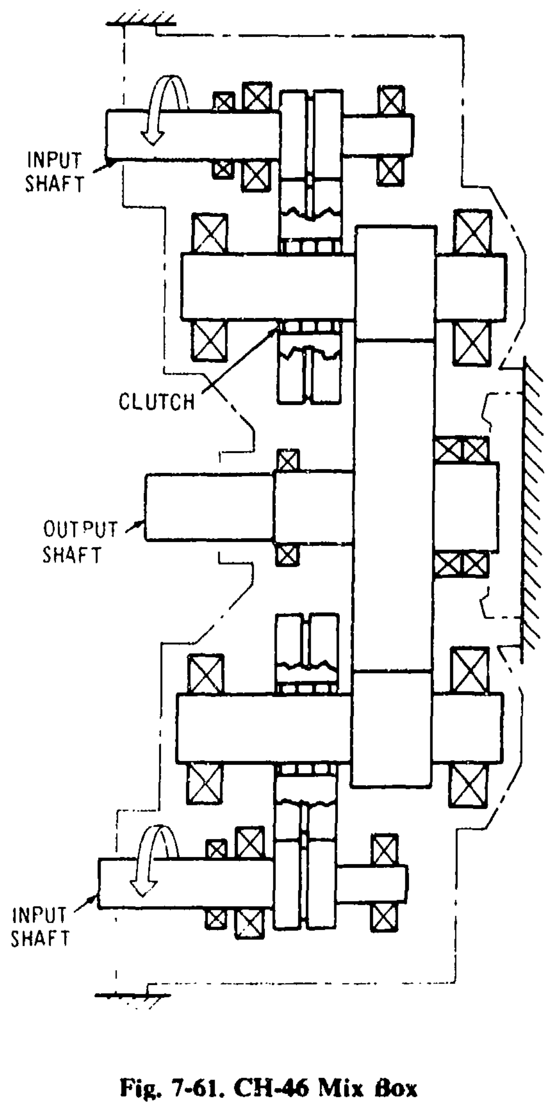


Fig. 7-61. CH-46 Mix Box

Another method employed for shaft systems that has received industry attention is the "fail-safe" design philosophy whereby the demonstrated integrity of the system remains intact even though a portion of the supporting structure has failed. The emphasis is placed upon the ability to detect the incipient failure.

7-4.4.1 Functions and Features

Despite different external appearances, helicopter shaft systems are remarkably similar. The functions and characteristics generally include but are not limited to:

1. Transmitting power from the engine(s), whether wing-mounted or fuselage-mounted, to the helicopter vertical lift and/or forward thrust propulsive systems
2. Torque transmitting capability, stowing, and/or tilting of the vertical lift systems
3. Facilitating the use of multiple engines with provisions for an engine-cut condition
4. Providing autorotational capability in the event of an emergency
5. Using high-speed couplings that prevent loss of shafting even if the coupling has lost its ability to transmit torque
6. Using redundant supports to provide design load capability and stiffness necessary to maintain the operational/structural integrity of the drive system even though failure of a support point has occurred
7. Transmitting power necessary to drive accessories.

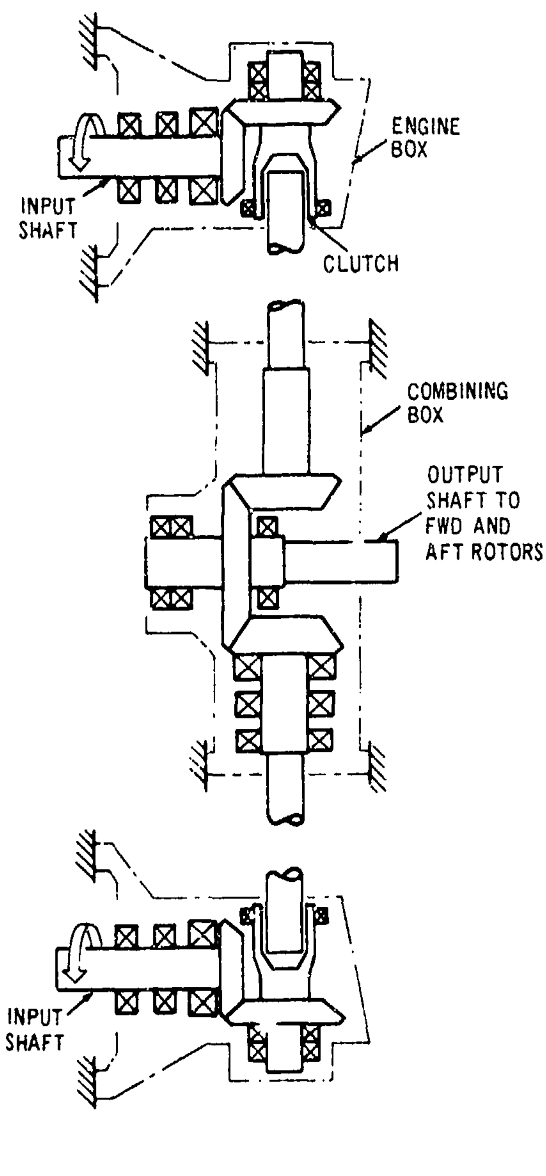


Fig. 7-62. CH-47 Engine and Combining Transmission

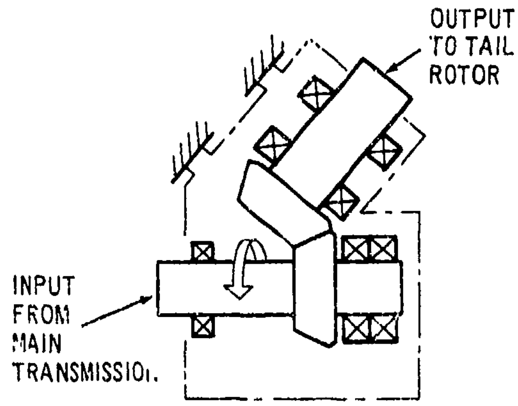


Fig. 7-63. BO-105 Intermediate Transmission

7-4.4.2 Shafting

In general, the lightest weight system is obtained by operating the engine/rotor cross-shafting at the highest practicable speed and considering the elastic properties of the interconnecting shafting. The engine/cross-shaft system constraints impact upon the anticipated integrated propulsion system performance and also are related to components such as gears, bearings, clutches, and couplings that are subjected to such operational conditions as structural deflections, misalignments, load, rpm, temperature, and other environmental conditions.

To date, the mechanical engine/cross-shaft power transmission system remains the optimum method of transmitting power over long distances. The designer of such a system must assess the previously stated operational/environmental conditions; estimate the direction and magnitude of these values; and conduct trade-off studies of the various shaft system candidates to ascertain the optimum system for weight, size, efficiency, and reliability. Three basic types of mechanical engine/cross-shaft systems are:

1. Subcritical (couplings at ends only). This shafting arrangement uses couplings at the end points and, if the shaft is long, uses the bending capability of the interconnecting tubing. Intermediate pillow blocks keep the system operational in the subcritical range. The blocks must be self-aligning and, in some instances, also must be capable of axial movements. This system requires very few complex components and can accommodate reasonable misalignments (1-2 deg).

2. Subcritical (couplings at each support). This shaft system has a coupling at each pillow block as well as at the end points and therefore can accommodate high misalignments (3-5 deg) between pillow blocks and gearboxes. The intermediate bearing pillow blocks must have angular or self-aligning capabilities.

3. Supercritical or hypercritical (couplings at ends only). Studies of subcritical systems have shown that operation at moderately high speeds (of the order of 4000 rpm) with relatively small-diameter interconnecting shafting requires many intermediate pillow blocks. The hypercritical system eliminates this constraint by operating above the second critical speed. The design of this type of system requires special attention to the kinematic properties of the shaft as compared to first and second critical speed systems, which are specially balanced and amplitude-limited. Vibration energy absorption and system tuning are accomplished with damped bearing supports carefully designed for mass, spring rate, viscous damping, and location.

Refer to par. 7-5 for shaft design.

7-4.4.3 Couplings

Basically, engine/rotor cross-shafting couplings can be classified as either rigid or flexible. The designer should investigate carefully the suitability of both types of couplings; conduct an exhaustive review of the design requirements at each shaft location; and select the most suitable and lightest weight coupling available. In general, the coupling selection process should be conducted concurrently with the evaluation of the total engine cross-shaft system design. Both rigid and flexible couplings—for reasons of balance, vibration, and shafting critical speed—normally are located near a support point/major component rather than at mid-span.

The term "rigid couplings" in the context described refers to main drive torque-transmitting couplings that provide a fixed union between shafts, provide axial motion with limited moment (shaft bending) capability, or some combination of both.

Types of rigid couplings, together with their advantages and limitations, follow:

1. Straight splines. These splines can be used if large axial movements during dynamic operation, thermal expansions, or manufacturing tolerances exist. Splines add weight and can reduce the overall system reliability unless proper fits are used together with adequate lubrication to minimize wear and fretting corrosion.

2. Ball splines. Ball or rolling splines have about 1/10 the coefficient of friction of straight splines and therefore reduce the axially induced load. Extreme care must be taken to locate this type of coupling in an area where system bending moments are low.

3. Curvic coupling. This face-to-face type of coupling has the distinct advantage of reducing the overall coupling flange diameter required. Sufficient preload, however, must be applied to negate the effects of the torque separating force.

4. Flanged coupling. This type of rigid-fixed coupling can be used to join shafts. Easy installation, little maintenance, and no wearing parts (except possible fretting at the bolt holes) are its chief advantages.

"Flexible" couplings are designed to transmit torque efficiently when the driving/driven shafts are not always in angular and/or parallel alignment. These couplings also must provide protection against shock and vibration and must compensate for temperature changes, wear of parts, and deflection of support members. The basic criteria that must be considered during the selection of flexible couplings are

1. Misalignment angle. Steady-state and transient misalignment for a particular combination of power and rpm often determines whether a simple crown spline or a more exotic coupling is required.

2. Parallel misalignment. This factor may give rise to shaft bending moments and shear forces with which the coupling must contend.

3. Speed and load. Certain couplings are rated by appropriately reducing the maximum operating torque rating if the coupling is to operate at maximum speed.

4. Lubrication. Certain couplings require some method of lubrication for successful operation.

5. Torsional spring rate. The torsional spring rate of the coupling is an important factor in assessing the dynamic torsional characteristics of the overall system.

6. Other. In high-speed, high-torque applications, other factors of concern are containment, a constant velocity ratio, and a minimum value of torsional/radial backlash.

Types of lightweight flexible couplings for consideration on engine/cross-shaft systems include, but are not limited to:

1. Crown spline. Flexible couplings utilizing "crowned and barrelled" splines generally represent the lightest arrangement, can be used for small misalignment angles (< 1 deg), and will accommodate reasonable axial motion.

2. Laminated disk. These disk couplings consist of two hubs and a laminated disk assembly. The disk assembly is attached alternately to either hub. When misaligned, the laminated disks take the form of a wave-washer. This coupling has moderate angular misalignment capability (1-2 deg), and the torsional spring rate is high.

3. Contoured diaphragm. These couplings use one or more thin disks or diaphragms that provide the torque/bending capability. A stabilizing ball and socket joint is used to maintain a fixed location of the intersection of the centerlines when more than eight diaphragms are used. Because each diaphragm can provide up to 1 deg of angular misalignment, the angular misalignment capability by using multiple diaphragms is high. In addition, for those couplings using the ball and socket arrangement, a shear load capability is available in conjunction with a fail-safe feature that retains the shaft in position even though its torque transmitting ability may have been eliminated.

7-4.5 ACCESSORY DRIVES

The number and type of accessory drives required for the transmission system depend upon the helicopter mission characteristics.

7-4.5.1 Helicopter Accessories

The accessories driven by the transmission vary with the design of the helicopter. For a light, single-engine (observation class), rotary-wing aircraft, the typical accessories driven by the transmission are the transmission lube pump, transmission and/or engine oil cooling fan drive, tachometer generator drive, and possibly a single hydraulic pump drive for the rotor controls.

For the larger, single-engine (utility class) helicopter, in general, a dual hydraulic pump drive is used and an additional generator drive is added.

Where twin engines are employed in helicopters having an all-weather mission requirement, additional drives are required for dual generators or for generators with constant-speed drives for helicopters having wide rotor rpm variations. In addition, single or dual rotor and/or propeller governor drives may be employed in certain configurations.

7-4.5.2 Accessory Drive Arrangement

Although the number and types of accessory drives basically are dependent upon the size and mission requirement of the helicopter, the arrangement of the accessory drives in the transmission system can become paramount in mission effectiveness. Factors such as reliability, maintainability, and vulnerability must be considered carefully in selection of the location and type of drives for the required transmission accessories.

In an observation type helicopter, the drives for the tachometer generator, lubrication pump(s), and hydraulic pump, as required, generally are mounted upon and driven by the main transmission. An individual drive for each of these units, or variations up to a single coaxial drive, can be provided for driving these three accessories. The oil cooling fan for this type of helicopter generally has been designed as a dual-function unit that provides air for the main transmission and engine oil coolers. A direct mechanical attachment to either the main transmission or tail rotor drive shaft has been the preferred method for turbine-engine-powered aircraft, while friction belt drives generally have been used for reciprocating engine installations. Although friction belt drives have been used with some success, and their application appears to be a low-cost and simple approach, they should be avoided where possible.

For a typical single-engine, utility helicopter, accessory drives generally are required for a tachometer generator, a single or dual hydraulic pump, lubrication pump(s), an AC or DC generator, and an oil cooling fan drive. All of these accessories, except for the oil cooling fan, generally would be mounted upon and driven by the main transmission. Where two independent hydraulic pump systems are used, independent drives for each pump should be provided, physically separated from each other with one driven directly by the main rotor gear train. This is to eliminate the potential of losing both systems by a round from a small arms weapon, or by a single mechanical drive failure. If only a single drive can be used for powering dual hydraulic systems, armor plate should be installed around the drive to achieve combat zone protection against a cal .50 projectile. The weight penalty for protective armor should be considered in the proposed empty weight of the helicopter.

The oil cooling fan, if it is mechanically driven, normally would have a direct mechanical connection to the main transmission or tail rotor drive shaft. If proper speed or space cannot be attained for the fan, an offset gearbox or bevel gear drive is used. Belts should be avoided. Where fans are remote, but mechanically driven, the shaft drive must be simple and highly reliable. Where an oil cooling system is designed only for the transmission, the fan can be mounted directly to the input, the tail rotor output drive flanges, or other accessory drives on the main transmission. Because the generators for this class or larger helicopters are too heavy for easy manual handling, the accessory drive for these units should be positioned such that the mechanic need not lift or hold the full weight of the generator while attaching it to the accessory drive pad. In general, the accessory drive pad should not face more than 30 deg down from the horizontal. Adequate clearances must be provided for use of standard tools and for other maintenance requirements.

A twin-engine helicopter, particularly an all-weather vehicle, would require at least one more generator accessory drive than the single-engine utility aircraft previously described. When this type helicopter is large enough to employ an auxiliary power unit (APU) for starting the engines, the hydraulic and electrical systems normally are complex. Therefore, the accessory drives for the hydraulic and electrical systems should be arranged, if possible, so they may be powered by the APU for complete checkout of the accessory power sources and their subsystems or the main transmission for normal operation. This method will eliminate or minimize the requirement for ground checkout equipment and procedures. An arrangement for achieving a complete checkout for two independently driven hydraulic and electrical systems is shown in Fig. 7-64.

This arrangement provides independent drives to each hydraulic and electrical system when the primary engine(s) is operating or both systems are driven simultaneously by shaft power from the APU. Variations of power sources—such as electrical, hydraulic, or pneumatic—may be used in lieu of mechanical shaft horsepower. Variants of this arrangement may be applied to combining gearboxes, accessory boxes, or other boxes, depending upon the drive system configuration.

Advanced engines normally would have an integral lubrication and cooling system that would eliminate the larger fan drives presently used to supply air for both the transmission and engine(s). This feature allows the transmission designer to integrate the total lubrication system with the cooling fan or other items to be included in a single package, thus improving reliability and maintainability and reducing vulnerability.

7-4.5.3 Accessory Drive Pads

The types of drive pads currently used in Army helicopters are the gear-driven Air Force-Navy Aeronautical Design Standard (AND) type pads that employ a multibolt attachment. On small helicopters, the AND pads are more cost-effective when downtime and labor costs to replace an accessory are considered. On larger or more complex helicopters, the use of current AND standard drive pads may produce excessively higher costs when the same cost-effectiveness formula is applied. For these larger or more complex helicopters, the quick attach and detach (QAD) concept of accessory drives may provide large economic gains. The QAD pads feature a V clamp attachment that requires only a single screw and is oriented readily for ease of accessibility. These pads also minimize the volume requirement for accessories by eliminating the required wrench clearances associated with AND pads. The accessory pads, either AND or QAD, or others as required, must have adequate structural integrity to react the accessory overhang moments under the flight conditions specified in MIL-S-8698 and/or crash loads of 10.0 g in all directions or higher as specified by the helicopter system specification.

The accessory pad design and ratings *shall* be in accordance with AND 10230 for the bolt accessories. When QAD pads are used, the details *shall* comply with MS 3325, MS 3326, MS 3327, MS 3328, or MS 3329. The capacity of the drives *shall* be at least 1.25 times the normal rating of the accessory to permit normal growth of the component.

The design life and/or time between overhauls (TBO) for the accessory drives, such as splines, gears, and bearings, should exceed the design life of the related main power components. The interface of the accessory and its drive must be designed to insure that the weakest link, or failure point under overload conditions, occurs on the driven accessory and not the accessory drive. In addition, the maximum attainable misalignment of the drive pad and the accessory should not permit a premature rotating beam fatigue failure of the accessory shaft.

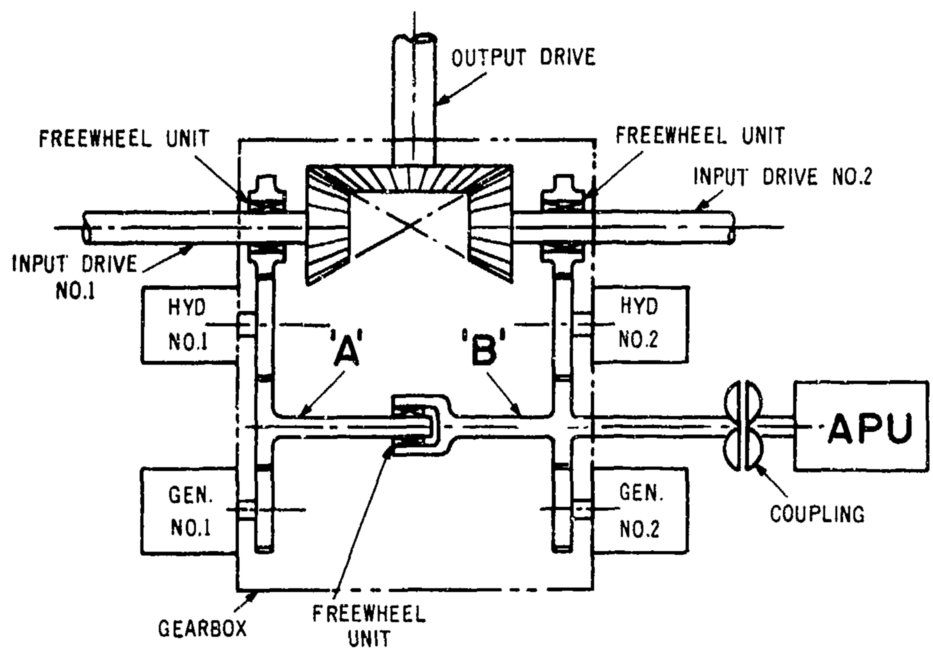


Fig. 7-64. Accessory Drive Arrangement for Hydraulic and Electrical Systems

7-4.5.4 Specifications

MIL-T-5955 specifies the type of drive and covers required when using AND standards. Where the use of QAD pads is beneficial, in general, the requirements of MIL-G-6641 may apply. MIL-S-7471 is not applicable to this phase of accessory drive design because it is used by the manufacturers in the preparation of the aircraft accessory power transmission model specification applicable to a specific model shaft.

7-4.5.5 State-of-the-Art

Although AND standard drives have been in service for many years, designers recently have preferred the QAD drive in order to attain lower helicopter maintenance requirements. Also, the compactness of the QAD design lends itself to lowering the overall weight and space of the transmission gearbox installations. Although QAD adapters—attached to AND standard pads—have and may be used, this results in additional space, weight, and cost.

Recent designs of advanced generator systems also show trends that require a special accessory drive. These advanced generators use an oil-cooling system that is supplied and transferred from the transmission through the accessory drive pad. In addition, some manufacturers propose the use of only one bearing on the generator armature with the accessory drive bearing used to support the other end of the armature.

Though the advanced accessories and their drives promise gains in weight and space, each system must be evaluated from an overall cost-effectiveness standpoint as applicable to the helicopter design, production, and operational requirements.

7-4.6 FAILURE WARNING SYSTEMS

Present failure warning systems for helicopter gearboxes generally are inadequate. The criterion for a successful warning system should be no surprise failures of the transmission system either in flight or in ground runup testing. This stringent criterion, however, does not measure fully the progress of the development of warning systems. To evaluate properly the warning system effectiveness, system failures and the applicable warning system must be defined more fully and categorized. For example, the fact that, practically, no warning system exists to predict a fatigue failure does not detract from the value of the Army Spectrometric Oil Analysis Program (ASOAP) to predict excessive wear failures.

The use of a particular warning system and the alternatives to its use must be considered fully in establishing its degree of success. By use of fatigue failure again as an example, a system that could warn reliably that a fatigue failure is imminent 10 min prior to actual failure would be of extreme value. Conversely, a system designed to indicate excessive wear rates due to a correctible condition such as shaft/transmission misalignment should not be considered successful if it only produced a 10-min-prior-to-failure warning.

An evaluation of the state of the art in failure warning systems must take into account:

1. The failure modes applicable
2. Safety aspects of a failure
3. Maintenance actions available to arrest the trend toward failure
4. The present state of the art in available detection methods when properly applied.

The failure modes that should be detected must be defined well. This must be done not only to facilitate the development of failure warning systems, but also to be able to judge more fairly the success of any system under test. A system that is capable of warning that something is wrong with a gearbox is of definite value. However, a failure warning system coordinated with the maintenance program yields optimum benefits, but this general type of system must be backed up with another that is capable of identifying specific failure modes.

The safety aspect of a failure must influence the requirements for a failure warning system. No false alarm would be acceptable for either a system warning of a catastrophic failure or one indicating that a routine maintenance action is required. This situation also must be taken into account in grading the failure warning results. See Chapter 3, AMCP 706-203, for a detailed discussion of this subject.

Maintenance actions that can be accomplished at the field level will influence the identification of failure modes. If, for example, several failure modes all require the removal of the gearbox and its return to the depot, it may only be of academic interest to know the specific cause of the failure indication, and the added cost and complexity of such a warning system could not be justified economically.

Improvements in detection methods for specific failure modes should reduce the number of short-time warning devices needed for safety reasons and increase the number of long-time warning devices that allow maintenance actions to arrest the trend toward failure.

Present failure warning systems in use on helicopter transmissions basically are limited to two inputs:

1. Chip detection used in the transmission oil system
2. Pilot and/or crew complaints of excessive vibration, noise, oil leaks, nonstandard oil pressure or temperature, etc.

Unfortunately, both of these inputs essentially constitute after-the-fact warning systems. Component failure sometimes progresses to a critical level before chip detector signals are obtained. The same may be true for crew complaints. An added difficulty in relying upon crew complaints is that the source of the problem usually is not localized adequately to permit planned maintenance actions. Further, distinctions between serious and minor problems are difficult to make, and can result in flying helicopters having dangerous failure conditions or sometimes grounding aircraft for minor, easily correctible problems.

Trials of several types of diagnostic systems (Refs. 7-11) have been made in conjunction with the transmission. These include:

1. Vibration monitoring equipment (in flight)
2. Sonic analysis (ground)
3. Spectrometric oil analysis (ground)
4. Temperature warning (in flight).

These systems have not been exploited fully because of the difficulty encountered in effective implementation (as in the case of ASOAP), or because of an unacceptably high rate of false alarms, which can lead to a catastrophic misinterpretation of results.

It is possible that an optimum failure warning system may be constructed using the stated measurements along with oil condition detectors and pressure sensors. The fact is that none of the measurements taken presently or contemplated in the future provides unique indications of particular malfunctions. Further, for any given failure, several of the measurements will respond. This requires that a failure warning system must have the ability to make a logical decision based upon multiple inputs. Additionally, the environment must be taken into account to differentiate normally occurring measurement variations from those related to failure. An example of this is the temperature measurements taken on the transmission. Temperature rises that could be significant in early failure detection are masked completely by normal temperature fluctuations caused by outside air temperature changes, and the variations due to power settings. Temperature inputs to the failure warning system, therefore, must be in terms of temperature rise ΔT over ambient as a function of power settings.

The overall vibration signal received by an accelerometer may be considered as the sum of the signals being generated by all the individual internal components of the transmission, with possibly a different amplification factor applied to each due to differences in their dynamic systems and transmission paths to the accelerometer. As a result, the vibration signal generated by a component that is remote from the accelerometer may increase tenfold due to its transmission path without a noticeable increase in the overall vibration signal. Hence, some type of signal processing —such as band pass filtering—will be required prior to introducing these signals into failure warning systems. Equally important is the accurate interpretation of individual vibration signals and their interrelationships with other vibration signals, other measured parameters, load, and environment. Integration of this knowledge may result in rather complex analysis techniques and equipment in order to arrive at an accurate identification of the failure mode.

The onboard warning system is preferable from the standpoint of operational usefulness; however, the apparent complexities of such a system may preclude its use on a cost-effective basis, particularly for utility and observation helicopters.

The optimum failure warning system may consist of two separate systems—an onboard warning system and either a piece of ground support equipment or one suitable for occasional flight tests. Instruments to provide inputs to each system preferably would be installed permanently in the helicopter, with quick-disconnect capability for the ground failure-warning systems.

The function of the onboard portion of the failure warning system would be to recognize conditions indicating imminent system failure, and also to recognize abnormal conditions that would require an in-depth analysis to identify problem areas. The onboard unit must be capable of performing some signal processing such as ΔT's as a function of torque, and include a limited number of fixed band-pass filters for vibration data processing. Logic analysis is required to set parameter limits close enough to operating values to detect significant variations while still safeguarding against unacceptably high false alarm rates. Closer limits may be set if it is required that failure warning be given as a function of logical combinations of out-of-limit conditions. For example, if the vibration level in a given band exceeds its limit and the temperature rise versus power setting exceeds its limit and the rate of metal buildup in the oil increases, a failure warning is given. The latter parameter would require the addition of oil condition monitoring equipment to the present set of onboard transducers to determine when the metal buildup approaches permissible limits.

The ground unit should be capable of extensive vibration analysis, contain a memory to allow trend analysis, and have limited computational ability. The unit should be used periodically to provide trend data and also on specific occasions when the onboard unit has signaled out-of-limit conditions. A summary of the characteristics of the two systems is included in Table 7-2.

The present state of the art in instrumentation requires very little development work to implement this type of warning system from a hardware standpoint. What is required is additional understanding of the relationship between failures and symptoms, and the logical relationship among symptoms.

7-5 SHAFTS

7-5.1 SHAFT DESIGN

Drive shaft system lengths vary from short to very long, and their function varies from transmitting full engine power to a fraction of installed power at a wide range of rpm. The following applications are typical:

1. Engine to main transmission
2. Interconnection between gearboxes
3. Tail rotor drive
4. Accessory drives
5. Gearbox to main and tail rotors.

Each of these applications requires design specifications peculiar to helicopters. The existing Military Specifications, developed for fixed-wing aircraft, generally are not applicable to the safe design of helicopter drive shaft systems. The purpose of this paragraph is to identify the guidelines so that maximum performance can be achieved in a safe and reliable manner in the absence of adequate Military Specifications.

7-5.1.1 Design Parameters

Because drive shafts are dynamic systems subject to cyclic fatigue stresses affecting safety of flight, design trade-offs must be considered carefully in any helicopter. Minimum-weight systems, consistent with latest proven state-of-the-art concepts, should be designed into Army helicopters.

7-5.1.1.1 Strength/Weight

Materials and geometry must be selected to insure unlimited drive shaft life at maximum power and rpm ratings. Design limit loads should be at least equal to the design value of the driven system. Allowances must be made for expected transient conditions. Ultimate loads should be 1.5 times design limit loads. Fatigue loading is of prime consideration because many shaft end conditions, such as in gear shafts, produce rotating beam shear forces and cyclic bending moments. For shaft assemblies employing flexible couplings, most bending stresses are relieved so that only the torsional cyclic loads from the prime mover and rotors need be considered.

To account for future growth and overload conditions, unlimited life is desired as a design goal for Army helicopters. Also, shafting is subject to handling damage (nicks, dents, and scratches), corrosion, and other stress raisers that may reduce service life. A limited-life part must be serialized and requires an accurate log of flight time to insure timely retirement. When records become lost or misplaced, or when the same shafts are used on several helicopters and several overhaul cycles are involved, a serious paperwork problem develops. The best approach is to design for unlimited life with perhaps some growth potential.

Static strength of shafts must be considered because rotor collisions with birds, loose objects, trees, brush, etc., are common occurrences in the operating environment. The large length-to-diameter (L/d) ratio, thin-walled shafts especially are subject to torsional buckling under peak torque loading that may occur above normal operating spectra. Unusually large axial forces will result from shaft shortening unless provisions are made for adequate coupling travel.

The fatigue strength of shafting must be considered under the highest combination of loading conditions. At the high shaft speeds used on the most recent helicopter designs, damaging fatigue cycles can be accumulated in a short period. Therefore, transient flight conditions must be considered, such as 60%/40% power distributions between rotors of a tandem design and loading to three times the power required at hover for tail rotors.

Because drive shaft system weight affects vehicle performance, the lightest weight system consistent with strength requirements must be employed on the selected design. Variables affecting weight are:

1. Drive shaft geometry and material
2. Damping requirements for operation above shaft critical speeds
3. Number and size of shaft bearing hangers needed for operation below shaft critical speeds
4. Torque requirements including transients (a function of shaft rpm)
5. Number of drive shafts and associated couplings
6. Shaft end configurations and coupling types.

The best performing helicopter also must have the lowest empty weight. It, therefore, will employ the most

TABLE 7-2
FAILURE WARNING SYSTEM CHARACTERISTICS

SYSTEM CHARACTERISTICS	ONBOARD UNIT	GROUND UNIT
INPUTS	1. FLIGHT PARAMETERS 2. TORQUE 3. TRANSMISSION VIBRATION (SEVERAL) 4. OIL PRESSURE 5. TEMPERATURE (SEVERAL) 6. OIL PARTICLE DETECTOR	1. FLIGHT PARAMETERS (IF USED IN FLIGHT) 2. TORQUE 3. VIBRATION (SEVERAL) 4. SONIC (SEVERAL) 5. OIL PRESSURE 6. TEMPERATURE (SEVERAL) 7. OIL PARTICLE DETECTOR
PROCESSING	1. LIMITED VIBRATION BAND PASS FILTERS 2. TEMPERATURE DIFFERENCES BIASED BY TORQUE	1. EXTENSIVE VIBRATION ANALYSIS CAPABILITY 2. TREND COMPUTATION 3. TEMPERATURE DIFFERENCES BIASED BY TORQUE
MEMORY	NONE	SOME FORM OF MEMORY REQUIRED
LOGIC	COMBINATION LOGIC (FIXED)	GENERALIZED LOGIC (PROGRAMMABLE)
OUTPUT	1. SAFETY OF FLIGHT WARNING 2. MAINTENANCE ACTION NOTIFICATION (GENERAL)	1. SPECIFIC PROBLEM IDENTIFICATION 2. SPECIFIC MAINTENANCE ACTION RECOMMENDATION
COST	LOW	MEDIUM

favorable combination of these design factors. The selected arrangement, however, must be sized and stressed for safe operation according to the strength requirements of this chapter.

7-5.1.1.2 Dynamic Considerations

Drive shafting involves high-speed rotation of high-mass systems. Consequently, static and dynamic balance must be controlled so as to avoid high-frequency vibrations. Realistic maximum limits of unbalance can be determined analytically, but should be verified by test during the qualification programs. Balance weights may have to be placed in several longitudinal locations to control shaft displacements at the higher harmonic frequencies.

The latest state-of-the-art in drive shaft design for Army aircraft operating in hostile environments is to employ large-diameter shafts in order to increase survivability. High speeds are used to minimize weight. To assure drive shaft control of dynamic stresses, shaft critical frequencies become important considerations. Shafting *shall* not operate within 10% of actual critical speeds as determined by tests on the vehicle. Analytical determinations of critical speed are of questionable accuracy because of assumed spring rates and damping characteristics of mounting points (bearings, airframe structure, etc.). Therefore, for subcritical shafting, the calculated critical speed *shall* not be within 30% of operating speeds.

For shafting operating above the first critical (supercritical) speed, the design must provide for adequate damping while passing through the resonance rpm. The drive shafts must pass through the critical rpm and not dwell at ground or flight idle ranges to avoid overworking of the dampers.

Hypercritical operation, a possible future development for high-speed, low-weight shafts, must provide damping for the pertinent harmonic resonances. A special technology must be developed to provide a beneficial trade-off. Some considerations are:

1. Proper placement of dampers for optimum efficiency
2. Dual dampers for reliability and fail-safe features at the high energy potentials
3. Weight
4. Dependability
5. Adequate qualification testing.

Torsional natural frequencies and reflected inertia to the engine must be compatible to prevent engine instabilities. Analytical treatment is required at the predesign level, followed by qualification testing on the flight vehicle.

Torsional oscillating loads may become significant when using high-speed universal shafts at appreciable misalignment angles. Self-generated angular velocity loading may combine additively with existing rotor and engine torsional loads to become a significant design problem. Use of this type of joint also requires a soft shaft section, torsionally, to minimize the velocity effects upon driven components of large mass.

7-5.1.1.3 Shaft End Configurations

Couplings normally are required at all shaft joints to allow for misalignments due to tolerances, differential temperature expansions, and helicopter deflections under varying maneuver and power loadings. For simplicity, shaft deflections and misalignment should be kept to a minimum.

Three types of couplings have been successful on recent Army helicopters. The flexible disk, multiple-flex rings, and gear couplings relieve most bending loads and, therefore, eliminate fatigue problems at the drive shaft end joints. Whenever possible, at least one end of a drive shaft should float to relieve axial forces on the couplings.

Practical considerations for quick disconnects and easy shaft removal must not be overlooked. Inspection at periodic intervals and after some flight incidents must be able to be performed with a minimum of maintenance effort. Shafting also must provide for disconnection and displacement so that major subsystems such as transmissions can be replaced easily. Airframe clearances must be adequate to prevent contact during extreme operating conditions or static damage during disassembly or other maintenance.

Drive shaft end configurations are designed on some Army helicopters, e.g., OH-6A, to employ a structural fuse as an additional safety feature should a transmission jam-up occur on the rotor side of the freewheeling clutch. An observable torsional yield deflection also may be used as an overload inspection device whereby drive system damage limits may be determined easily.

7-5.1.2 Vulnerability

Drive shafts represent a large area and, therefore, are vulnerable to small arms fire. Combat employment of turbine-powered helicopters has provided valuable data on drive shaft vulnerability, and many exhibits have been returned intact with up to cal .50 bullet holes. Large-diameter shafts that can accept damage up to cal .50 firepower with a return-to-base or complete-mission capability are desired on all future Army helicopters. All shafting *shall* be capable of surviving 7.62 mm small arms fire unless protection from a greater threat is required by the procuring activity.

7-5.1.3 Large- vs Small-diameter Shafts

Because large-diameter shafts are desirable for vulnerability reasons, a trade-off of advantages and problems must be considered when comparing them with smaller diameter shafts. The basic design factors are:

1. Large section modulus provides more resistance to bending and logistic damage.
2. Damage (scratches and nicks) can be repaired or tolerated more easily on larger surface areas.
3. Balance is more critical at larger radii.
4. Thinner walls, used on the larger diameter shafts, are more prone to torsional buckling and handling dents.
5. Large diameters, requiring no bearing supports, are most adaptable for operation above shaft critical speeds because bearing hanger locations require smaller diameters to permit bearing dN (diameter × rpm) values within grease lubrication ranges. At higher rpms, a weight advantage becomes possible because practical operation can occur between the first and second critical speeds.
6. Space requirements such as bulkhead holes and access doors sometimes restrict shaft diameters; but for vulnerability reasons alone, adequate space should be provided.
7. Long, small-diameter shafts are more susceptible to torsional windup deflections that can become out of phase with rotor or engine torsional oscillations. Drive system torsional instabilities have caused engine surging and excessive fatigue loading.

7-5.2 SHAFT DYNAMICS

The problems of shaft design involve interrelations among the requirements for torsional dynamics (as outlined in par. 5-5), and weight considerations as well as the critical speeds of lateral vibration and other dynamic effects due to the rotation of the shaft. The resultant design is a compromise among opposing demands. For example, it may be desired to operate a tail rotor drive shaft at a high rotational speed relative to the speed of the tail rotor in order to gain one or more of the following advantages:

1. The required torque for a given horsepower varies inversely with the rotational speed; therefore, a smaller and lighter shaft may be used.
2. The effective torsional stiffness of the shaft increases as the square of the rotational speed.
3. Because of the reduced torque load, the thrust load required to slide the splines axially is reduced.

On the other hand, high rotational speeds have certain disadvantages such as:

1. A larger stepdown gear ratio is required to get down to tail rotor speed. This may increase the size and weight of the tail rotor gearbox.
2. If it is decided to operate the shaft "subcritical" (below the first critical speed), the higher operational speed requires that the critical speed be higher. As will be shown, this in turn requires either an increase in shaft diameter or a decrease in the distance between adjacent bearings. Both of these courses of action tend to increase the weight.
3. If "supercritical" operation is chosen, suitable dampers must be developed to enable the shaft to pass safely up through the critical speed ranges to the operating speed. The dampers add weight as well as maintenance problems, while the reduction of the number of bearings required reduces the weight.
4. Gyroscopic effects and the effects of shaft unbalances increase with increased speed of rotation.
5. There may be problems resulting from the operation of bearings at high speeds.

It may be observed from this discussion that to make an optimum decision requires a broad knowledge of all the dynamic problems of the shaft installation. The methods and discussion that follow provide a means of evaluating the major dynamic effects in rotating shafting.

7-5.2.1 Critical Speeds

Among the problems encountered in the design of shafting systems is a divergence phenomenon that variously has been called the "critical speed" or the "whirling speed". The nature of the phenomenon is that, as the speed of shaft rotation increases, residual unbalances in the shaft give rise to centrifugal forces. These forces cause the shaft to rotate in a bent configuration with the centrifugal bending loads being balanced by the elastic forces in the shaft, much in the manner of a skip rope. As the speed of rotation increases, the centrifugal loads increase to the point at which they exceed the elastic forces in the shaft, and divergence occurs. This point in the speed range is called the critical speed. At rotational speeds above the critical speed, the amplitude undergoes a phase change of 180 deg and the shaft center of mass moves to a position between the geometric centerline of the shaft and the axis of rotation. As the rotational speed increases the shaft mass center moves toward the axis of rotation, and, ultimately, the shaft rotates about an axis through its CG with an amplitude equal to the eccentricity of the mass center with respect to the axis of rotation.

Consider now a disk having a mass m mounted on a shaft rotating at a constant angular velocity ω (Fig. 7-65). If the CG of the disk is displaced a distance e from the geometric centerline of the shaft and if the disk is rotating about the geometric centerline of the shaft with the velocity ω, a rotating centrifugal force $me\omega^2$ acts upon the disk. This causes the centerline of the shaft to be deflected a distance r that causes the centrifugal force to increase to $m(r + e)\omega^2$ and this is opposed by an elastic force kr, where k is the spring bending constant, lb/in. Thus :

$$kr = mr\omega^2 + me\omega^2$$

$$\omega_N^2 r = r\omega^2 + e\omega^2$$

and

$$r = \frac{e\omega^2}{\omega_N^2 - \omega^2} \quad \text{, in.} \qquad (7\text{-}29)$$

where

$k/m = \omega_N^2$

ω_N = natural frequency, rad/sec

It may be observed that for small values of ω, r approaches a value of $e\omega^2/\omega_N^2$ which approaches zero as ω approaches zero. When $\omega = \omega_N$, r becomes infinite and divergence is rapid. At large values of ω, r approaches a value of $-e$. The system is stable above the critical speed ω_N because the divergence of the CG sets up a Coriolis acceleration that moves the CG tangentially until ultimately it lies between the geometric centerline of the shaft and the axis of rotation.

The dynamics of rotating shafts are somewhat more complex than indicated because of variables such as

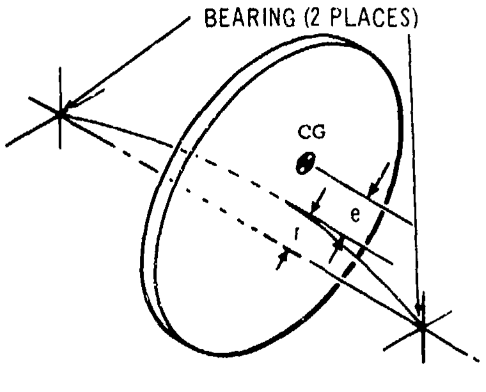


Fig. 7-65. Orientation of CG and Geometric Shaft Centerline of a Rotating Shaft

unequal stiffnesses of the shaft in different planes, unequal bearing stiffnesses, variations in local unbalance, the effects of friction and damping forces, and gyroscopic effects. However, in practice most of the systems encountered may be evaluated by treating the rotating shaft as a beam vibrating in one plane.

7-5.2.1.1 Mathematical Methods

7-5.2.1.1.1 Nonuniform Shafts

There are numerous methods for calculating the critical speeds of shafts having a nonuniform distribution of flexural rigidity and those having concentrated masses or several bearings of arbitrary stiffness. Many of these methods are variations of Rayleigh's method (Ref. 12), which consists essentially of assuming a mode shape, developing expressions for the kinetic and potential energies in terms of the displacements and the displacement velocities, and equating the two expressions to obtain an equation that can be solved for the natural frequency ω_N. Thus, in general

$$\frac{\omega_N^2}{2}\int_0^L u y^2 dx = \frac{1}{2}\int_0^L EI\left(\frac{d^2y}{dx^2}\right)^2 dx \qquad (7\text{-}30)$$

where

I = cross-sectional moment of inertia of area, in.^{4}

u = running mass, slug/in.

x, y = length and displacement coordinates, in.

A common tabular procedure is to replace the shaft of length L by an arbitrary number of concentrated masses m_i representing the distributed mass of the shaft as in Fig. 7-66. The masses are assumed to be connected by shaft elements having flexural stiffness but no mass. A static bending moment M is calculated over the length of the shaft. Then by numerically integrating the function

$$y = \iint \frac{M}{EI}\, dx dx \quad \text{, in.} \qquad (7\text{-}31)$$

with appropriate conditions, the static deflection curve is determined. Using the deflection ordinates y_i of the lumped weights w_i (i = 1 to p), the kinetic energy T may be expressed as

$$T = \frac{\omega^2}{2}\sum_{i=1}^{p} \frac{w_i}{g}\, y_i^2 \quad \text{, in.-lb} \qquad (7\text{-}32)$$

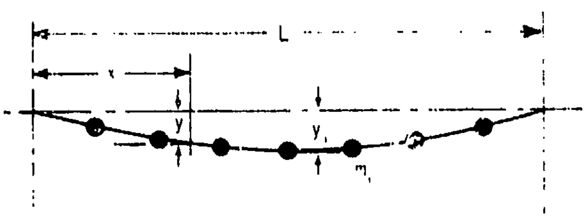


Fig. 7-66. Mathematical Model of a Shaft

The potential energy V is equal to the work done in applying the loads w_i through the distances y_i. Thus, the potential energy is

$$V = \frac{1}{2}\sum_{i=1}^{p} w_i y_i \quad \text{, in.-lb} \qquad (7\text{-}33)$$

Equating the potential energy to the kinetic energy and solving for the natural frequency ω_N results in

$$\omega_\mathrm{N} = \sqrt{g\sum_{i=1}^{p} w_i y_i \Big/ \sum_{i=1}^{p} w_i y_i^2} \quad \text{, rad/sec} \qquad (7\text{-}34)$$

Because it is convenient to express the critical speed N_c in revolutions per minute, we may write

$$N_c = \frac{60}{2\pi}\sqrt{g}\sqrt{\sum_{i=1}^{p} w_i y_i \Big/ \sum_{i=1}^{p} w_i y_i^2}$$

$$= 187.7\sqrt{\sum_{i=1}^{p} w_i y_i \Big/ \sum_{i=1}^{p} w_i y_i^2} \quad \text{, rpm} \qquad (7\text{-}35)$$

Note in this formula that $g = 386.4$ in./sec^2.

This method is restricted in general to calculation of the first critical speeds of shafts operating in two bearings that may be supported simply or may have complete restraint in coordinates other than rotation.

For the case of nonuniform shafts having more than two bearing supports, or for the calculation of the higher order critical speeds, other tabular methods such as Prohl's method (Ref. 13) and Myklestad's method (Ref. 14) are available.

With digital computer facilities becoming available, more sophisticated solutions are possible. One such method involves the solution of equations of the type $M[\ddot{q}] + C[\dot{q}] + K[q] = 0$ where M is a mass matrix, C is a damping matrix, K is a stiffness matrix, and $[q]$ is a column matrix of the coordinates of the system. The methods of constructing the required matrices are shown in Ref. 15.

Another useful method suitable to a digital computer involves the use of transfer matrices and state vectors to describe the relationships between the displacements and internal forces at one point in the system and the corresponding displacements and internal forces at another point in the system (Ref. 16).

7-5.2.1.1.2 Uniform Shafts

In a number of applications, such as an antitorque rotor drive shaft, the shaft has a constant distribution of flexural rigidity and mass, and is supported on several bearings. If the bearings are rigid and equally spaced, and if there is no moment restraint at the bearings, then during the vibration there is no bending moment in the shaft at the bearings. In fact, as there is no reaction at the intermediate bearings, the bearings theoretically could be removed during the vibration and the mode shape would be maintained. For instance, for a shaft running in four equally spaced bearings as in Fig. 7-67(A), the first mode natural frequency and mode shape are identical to the third mode natural frequency and mode shape that would occur if the two intermediate bearings were removed, as in Fig. 7-67(B). It is evident therefore that, for uniform shafts rotating in simply supported bearings, spaced at equal intervals, the shafts may be treated as a beam simply supported on two bearings and having a length equal to the bearing spacing. A beam of length L simply supported at each end having a uniform flexural rigidity EI and a uniformly distributed mass per inch u has a natural frequency ω_N (Ref. 12) of

$$\omega_\mathrm{N} = \left(\frac{\pi}{L}\right)^2 \sqrt{\frac{EI}{u}} \quad \text{, rad/sec} \qquad (7\text{-}36)$$

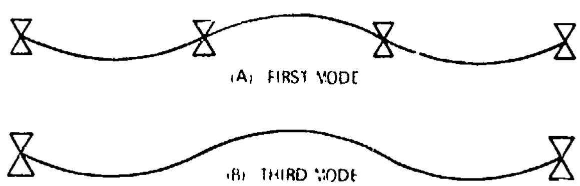


Fig. 7-67. Characteristics of a Shaft Having Four Equally Spaced Bearings

If the shaft is a thin-walled tube, as is common in aircraft design, we may substitute

$$I = \pi r^3 t \quad , \text{in.}^4 \tag{7-37}$$

and

$$u = \frac{2\pi r t \rho}{g} \quad , \text{slug/in.} \tag{7-38}$$

where

r = mean radius of tube, in.
t = thickness of tube, in.
ρ = weight density, lb/in.^{3}
g = acceleration due to gravity, 386.4 in./sec^2

The critical speed N_c expressed in rpm is then

$$N_c = \frac{60\pi^2}{2\pi L^2}\sqrt{\frac{E\pi r^3 tg}{2\pi r t \rho}}$$

$$= \frac{30\pi r}{L^2}\sqrt{\frac{Eg}{2\rho}} \quad , \text{rpm} \tag{7-39}$$

It can be shown that the quantity E/ρ is virtually a constant for commonly used engineering metals. Thus for steel $E/\rho = (30 \times 10^6)/0.283 = 106 \times 10^6$ and for aluminum alloys $E/\rho = (10.5 \times 10^6)/0.1 = 105 \times 10^6$. Making this substitution we find that for practical accuracy $N_c = 13.3 \times 10^6 (r/L^2)$.

It is to be noted that the critical speed is a function of the mean radius of the tube and the length between supports. The material and the wall thickness of the tube have a negligible effect upon the critical speed.

If the shaft is a thick-walled tube,

$$I = \frac{\pi}{64}\left(d_o^4 - d_i^4\right) \quad , \text{in.}^4 \tag{7-40}$$

and

$$u = \frac{\pi\rho}{4g}\left(d_o^2 - d_i^2\right) \quad , \text{slug/in.} \tag{7-41}$$

where

d_o = outer diameter of tube, in.
d_i = inner diameter of tube, in.

Substituting these values in Eq. 7-36 and expressing the critical speed N_c in rpm gives

$$N_c = \frac{60\pi^2}{2\pi L^2}\sqrt{\frac{4gE\pi(d_o^4 - d_i^4)}{64\pi\rho(d_o^2 - d_i^2)}}$$

$$= \frac{7.5\pi}{L^2}\sqrt{\frac{Eg}{\rho}(d_o^2 + d_i^2)} \quad , \text{rpm} \tag{7-42}$$

For a comparison of the two formulas consider a shaft of length $L = 20$ in., an outside diameter $d_o = 1$ in. and an inside diameter $d_i = 0.75$ in. By applying the thin-walled tube formula (Eq. 7-39), the critical speed $N_c = 14{,}550$ rpm, while with the thick-walled tube formula (Eq. 7-42), $N_c = 14{,}890$ rpm.

7-5.2.1.1.3 Effects of Elasticity and Restraints at Bearings

The consequence of reducing the stiffness of the bearing supports is to lower the critical speed of the shaft, because the effect of elasticity at the bearings is that of putting a spring in series with the effective spring of the shaft. Figs. 7-68 and 7-69 show the effect of variation of the bearing stiffness on the natural frequency. When using these curves, the natural frequency ω_N is determined by the formula

$$\omega_N = \frac{\beta^2}{L^2}\sqrt{\frac{EI}{u}} \quad , \text{rad/sec} \tag{7-43}$$

where β is the dimensionless stiffness factor given by Fig. 7-68 or 7-69 for applicable values of the factor $KL^3/(EI)$, with the bearing support stiffness K in lb/in.

If the stiffness of the bearing supports in the horizontal plane is different from the stiffness in the vertical plane, the shaft has two critical speeds corresponding to the two bearing stiffnesses.

Fig. 7-70 shows the effect of moment restraint on the critical speed N of a simply supported shaft. The amount of restraint is represented by the dimensionless parameter $K_\theta L/(EI)$, where K_θ is the equivalent angular spring rate, in.-lb/rad, due to moment restraint of the bearing. The figure shows the ratio of critical speeds N/N_0, where N_0 is the value for $K_\theta = 0$.

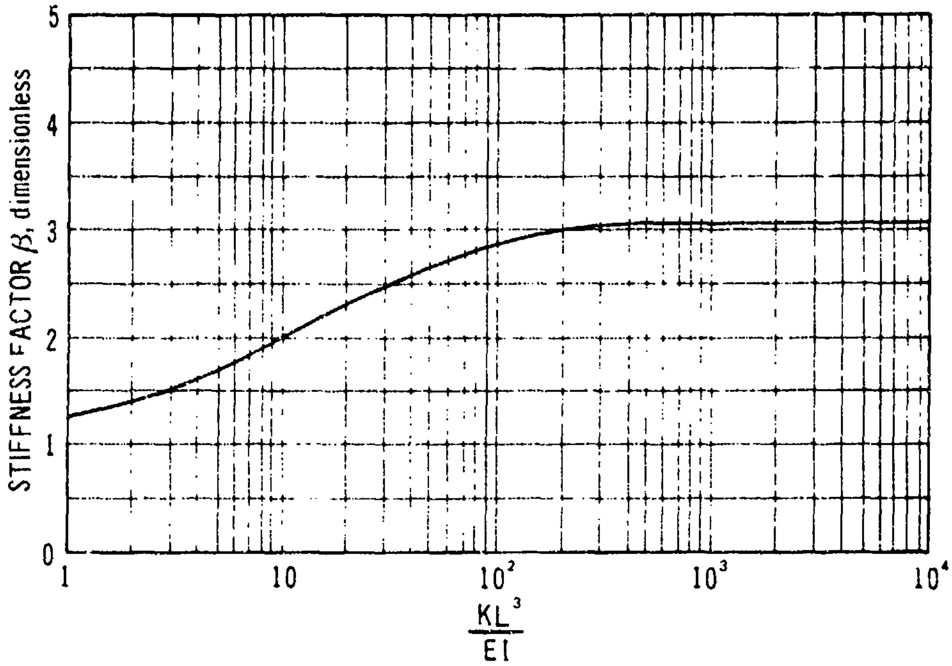


Fig. 7-68. Factor To Account for the Stiffness of Two Bearing Supports of Equal Stiffness (Eq. 7-43)

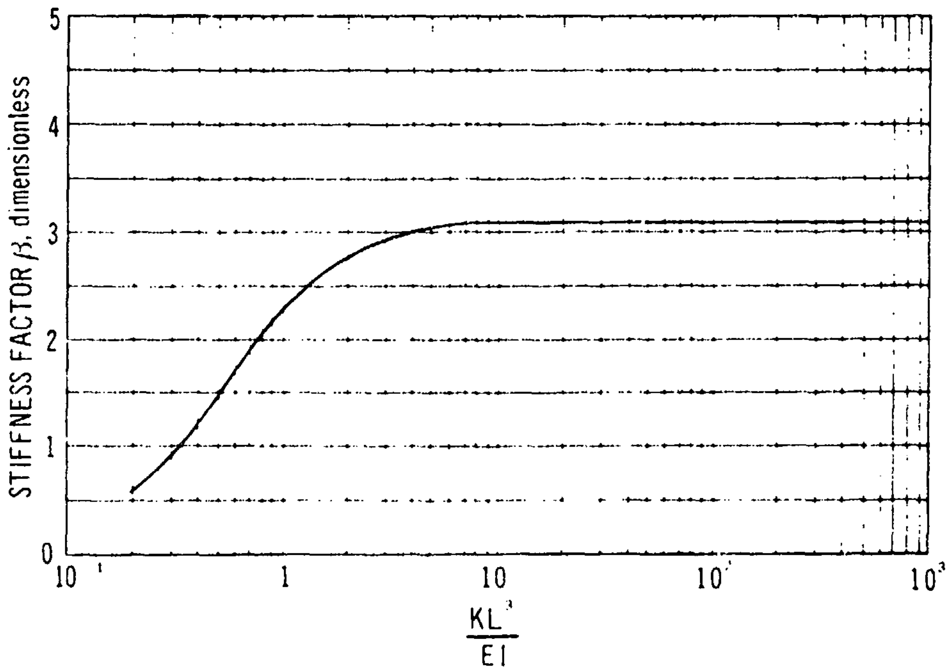


Fig. 7-69. Factor To Account for the Stiffness of One Bearing When the Other Is Considered Rigid (Eq. 7-43)

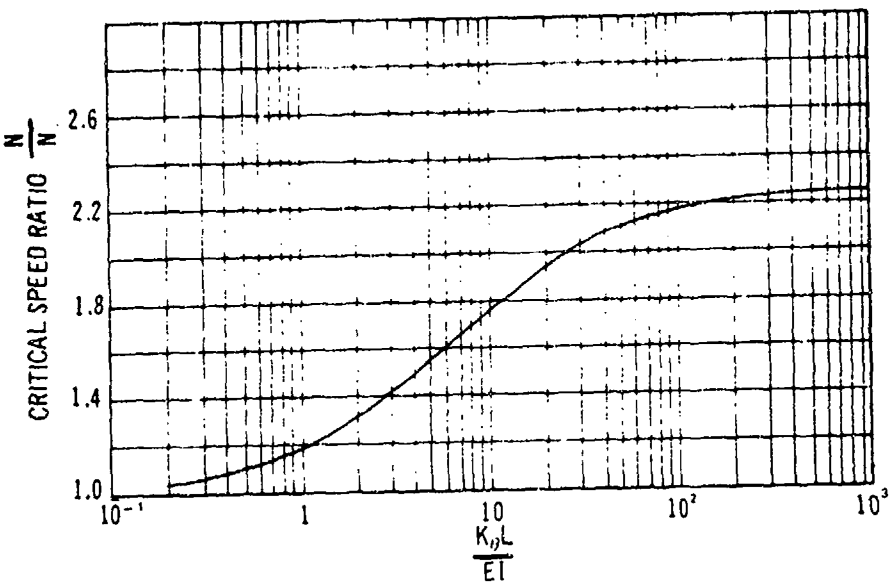


Fig. 7-70. Moment Restraint at the Bearing Upon the Natural Frequency of a Shaft

7-5.2.1.1.4 Effects of Masses Concentrated Between Bearings

If a mass of large magnitude compared to the mass of the shaft is concentrated on the shaft at some point between the bearings, a close approximation (Ref. 17) of its effect may be made by calculating the static deflection δ_{st} of the centroid of the mass and calculating the natural frequency by

$$\omega_N = \sqrt{\frac{g}{\delta_{st}}} \quad \text{, rad/sec} \qquad (7\text{-}44)$$

If the mass of the shaft is not small compared to the concentrated mass, or if there are several masses concentrated on the shaft, the designer may use Dunkerley's formula (Ref. 17), which may be stated as

$$\frac{1}{\omega_c^2} = \frac{1}{\omega_o^2} + \sum_{i=1}^{n} \frac{1}{\omega_i^2} \quad \text{, sec}^2 \qquad (7\text{-}45)$$

where

ω_c = natural frequency of the total assembly, rad/sec

ω_o = natural frequency of the shaft without the concentrated masses, rad/sec

ω_i = natural frequency of the combination of the ith mass with the shaft, which is considered to have flexural rigidity but no mass, rad/sec

7-5.2.1.1.5 Gyroscopic Effects

Masses concentrated on the rotating shaft may show so-called gyroscopic effects dependent upon the speed of shaft rotation. The effect is to cause advancing (in the direction of rotation) and regressing (counter to the direction of rotation) whirl modes in the shaft. The advancing mode has the effect of augmenting the stiffness of the shaft, and thus raising the critical speed, while the regressive mode has the effect of lowering the critical speed. For example, the shaft shown in Fig. 7-71 may be considered to have a nonrotating natural frequency of

$$\omega_{N_0} = \sqrt{\frac{K_\theta}{I}} \quad \text{, rad/sec} \qquad (7\text{-}46)$$

The natural frequency during rotation is (Ref. 12)

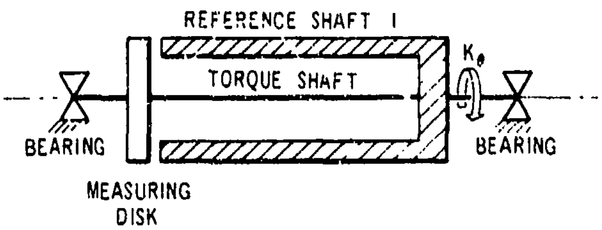


Fig. 7-71. Schematic View of a Torquemeter Shaft

$$\omega_\nu = \mp \frac{J\Omega}{2I} + \sqrt{\left(\frac{J\Omega}{2I}\right)^2 + \frac{K_\theta}{I}}$$

$$= \mp \frac{J\Omega}{2I} + \sqrt{\left(\frac{J\Omega}{2I}\right)^2 + \omega_\nu^2} \quad \text{, rad/sec} \qquad (7\text{-}47)$$

where

I = diametral mass moment of inertia, lb-in.-sec²

J = mass polar moment of inertia, lb-in.-sec²

Ω = rotor speed, rad/sec

Assume that the nonrotating natural frequency of the shaft is 1850 rad/sec (294 Hz). A plot of the rotating natural frequency (Hz) as a function of $J\Omega/I$ is shown in Fig. 7-72.

If we assign values of 2 and 0.2 to the quantity J/I and plot the roots of the equation as a function of Ω, the results are as shown in Fig. 7-73. The line 1P inclined at 45 deg indicates the locus of frequencies occurring at one per revolution of the shaft, and the two branches of the curves show the positive and negative roots of Eq. 7-47. From this it may be seen that, for this particular case, a critical speed occurs at approximately 1100 rad/sec with $J/I = 2$ but this regressive mode critical speed occurs at approximately 1700 rad per sec with $J/I = 0.2$. Unfortunately, means of reducing the ratio J/I also have the effect of reducing the nonrotating natural frequency.

7-5.2.1.2 Other Effects

It is often necessary to consider the effects of such items of hardware as couplings, universal joints, and splines.

Couplings should be located on the shaft as close to a bearing as possible, particularly if the bending stiffness through the coupling is low compared to the stiffness of the shaft. Both the added mass of the coupling and the lack of stiffness in the coupling tend to lower the critical speed of the shaft. With a reasonably stiff bearing support structure, locating the coupling near the bearing places the mass of the coupling at a point on the shaft where the vibratory motion is at a minimum, thus reducing its relative kinetic energy. This also minimizes the effect of the reduction of stiffness through the coupling. It is well to avoid having a misalignment coupling installed on each side of the same bearing because this requires considerable stiffening of the bearing support against rotation, in the plane of the shaft, to react shear forces at the couplings.

A particular type of coupling, known as the universal joint or Hooke's coupling, is used to transmit power from one shaft to another whose axis of rotation is at a relatively large angle to the first shaft. If the driving shaft is rotating at a constant velocity, the driven shaft undergoes a sinusoidal variation of velocity at frequencies that are multiples of the rotating speed. As a consequence, sinusoidal torques and bending moments are produced in both shafts at multiples of the rotational speed. For this reason, the critical speed must not coincide with a multiple of the shaft speed if the shaft contains a universal joint or a misalignment coupling.

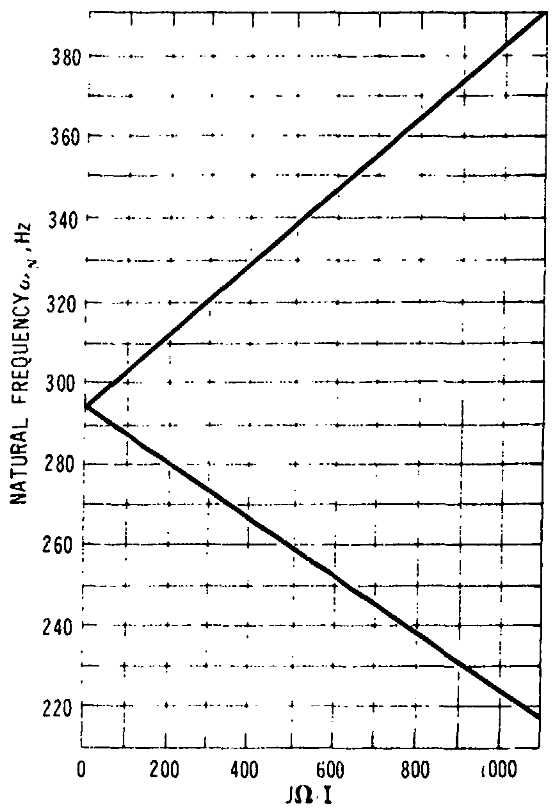


Fig. 7-72. Effect of the Magnitude of $J\Omega/I$ on the Advancing and Regressing Modes of the Shaft Shown in Fig. 7-71

Splines, in addition to the misalignment effects discussed previously, have another effect due to the torque loads and the friction in the splines. Because the coefficient of friction for splines commonly is from 0.1 to 0.15, low-speed shafts carrying high horsepower develop heavy end thrusts because of the high thrust required to overcome the friction and slide the spline axially to accommodate relative motion between adjacent parts of the drive system. The effect of the end thrust is to lower the critical speed. An estimate of the effect of the end thrust may be made by the use of the formula

$$N_c = N_n \sqrt{1 - \frac{P}{P_e}} \text{ , rpm} \qquad (7\text{-}48)$$

where

N_c = actual critical speed, rpm
N_n = critical speed calculated without end thrust, rpm
P = end thrust, lb
P_e = Euler critical column load = $\pi^2 EI/L^2$, lb

If P/P_e is less than 0.1, the reduction in critical speed is less than 5%.

When excessive thrust occurs, it may be reduced by the use of a ball spline. Because this has appreciable weight and suffers a loss of stiffness because of Hertz deformation at the ball races, a bearing usually is required at each end to avoid severe deterioration of the critical speed.

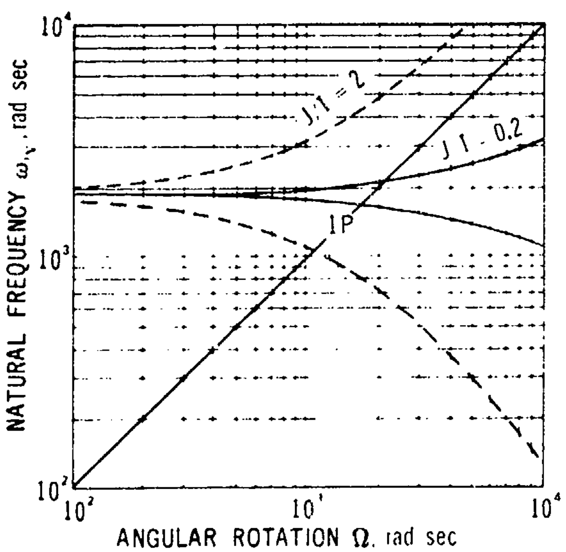


Fig. 7-73. Comparison Between the Effects of High and Low Values of J/I on the Advancing and Regressive Modes of the Shaft Shown in Fig. 7-71

7-5.2.2 Design Criteria

The design criteria for satisfactory operation are:

1. For subcritical shafting, the lowest calculated critical speed *shall* be clear of the highest operating speed by a margin of 30%, and there *shall* be no calculated critical speed within 30% of twice the lowest operating speed or twice the highest operating speed.

2. Floating parts, i.e., parts that cannot be constrained to rotate about the same fixed center in the same orientation at all times, are to be avoided because of the impossibility of achieving a consistent state of balance.

3. For supercritical shafting, means—such as suitable dampers—*shall* be provided for safe passage through all of the critical speeds below the operating speed.

7-6 CLUTCHES, BRAKES, AND COUPLINGS

7-6.1 CLUTCHES

The requirement for clutching devices has been defined earlier in this chapter; however, because current vehicle designs embody freeshaft turbine engines almost exclusively, the use of engagement clutches is unnecessary. Therefore, the only clutching functions required within the engine-rotor drive system are those necessary to provide for autorotation and for single-engine operation in the case of multiengine configurations. This capability is provided through use of overrunning clutches.

The overrunning clutch is a device that transmits torque from an input in one direction only, and that releases automatically when the input direction is reversed or the output overspeeds the input in the drive direction. The latter case is applicable to the helicopter drive system. This function may be accomplished with a sprag clutch or a roller clutch, both of which depend upon a complement of friction wedges disposed between races to transmit torque.

7-6.1.1 Sprag Theory

The sprag overrunning clutch, shown in Figs. 7-74 and 7-75, consists of an outer race, usually the driving member, the inner race, the driven member, a complement of sprags, a retainer, appropriate energizing springs, and bearings to provide the required race concentricity. The term sprag denotes any type of wedge device such as the wedge-shaped blocks used to prevent vehicle motion on sloping surfaces. A disassembled sprag retainer assembly is shown in Fig. 7-76.

The UH-2C combining gearbox with the overrunning clutch installation is illustrated in the partial cross section shown in Fig. 7-77.

Often the overrunning or "freewheeling" clutch is combined with one of the gears in the engine combining gearbox. This approach is shown in Fig. 7-78, a partial section of the "Twin-Pac" gearbox that combines two PT-6 turbine engines. This power package is used in the AH-1J, UH-1N, and S-58T model helicopters.

Power is transmitted from one race to another by the wedging of specially contoured, through-hardened steel sprags. As in the internal arrangement shown in Fig. 7-79, if the outer race is driven in the counterclockwise direction, the expanding force of the energizing spring and frictional force of the sprag against the race combine, causing the sprag to rotate in the counterclockwise direction about its center. Because dimension a is greater than the annular space between races, the sprags wedge, thus transmitting torque. When the inner race rotates in the counterclockwise direction (as in autorotation) at a speed greater than that of the outer race, the sprags rotate so that the portion of the sprag indicated by dimension b is in contact with the races. Because dimension b is less than the annular space, the sprags do not wedge and the outer race may rotate freely.

Torque is transmitted in the drive direction because of the careful choice of the wedging angle, or "gripping angle", of the sprag with respect to the races. The gripping angle is defined by the construction shown in Fig. 7-80. By drawing a line between the points of tangency A and B of the sprag with the inner and outer race, and measuring the angle between that line and a line drawn from the center of the races through the point of tangency A of the sprag at the inner race, the gripping angle GA is obtained.

By employing compound curves for the sprag cams, the gripping angle can be tailored to give optimum value for both the overrunning and drive modes. This initial gripping angle must be such that the radial force produced as engagement commences and the coefficient of friction at the point of contact will combine to resist the tangential force imposed upon the sprag by the applied torque. The proper choice of surface finish and lubricant is required to insure that this condition is guaranteed. Expressed mathematically, the tangent

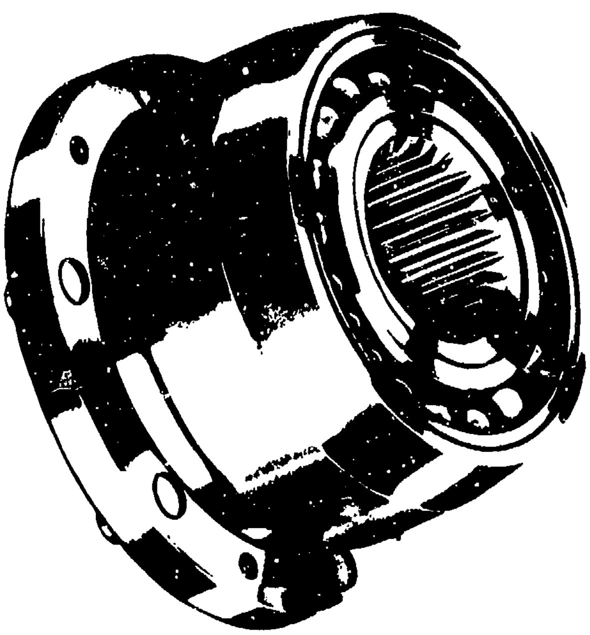


Fig. 7-74. Sprag Overrunning Clutch Assembly Used on UH-2C

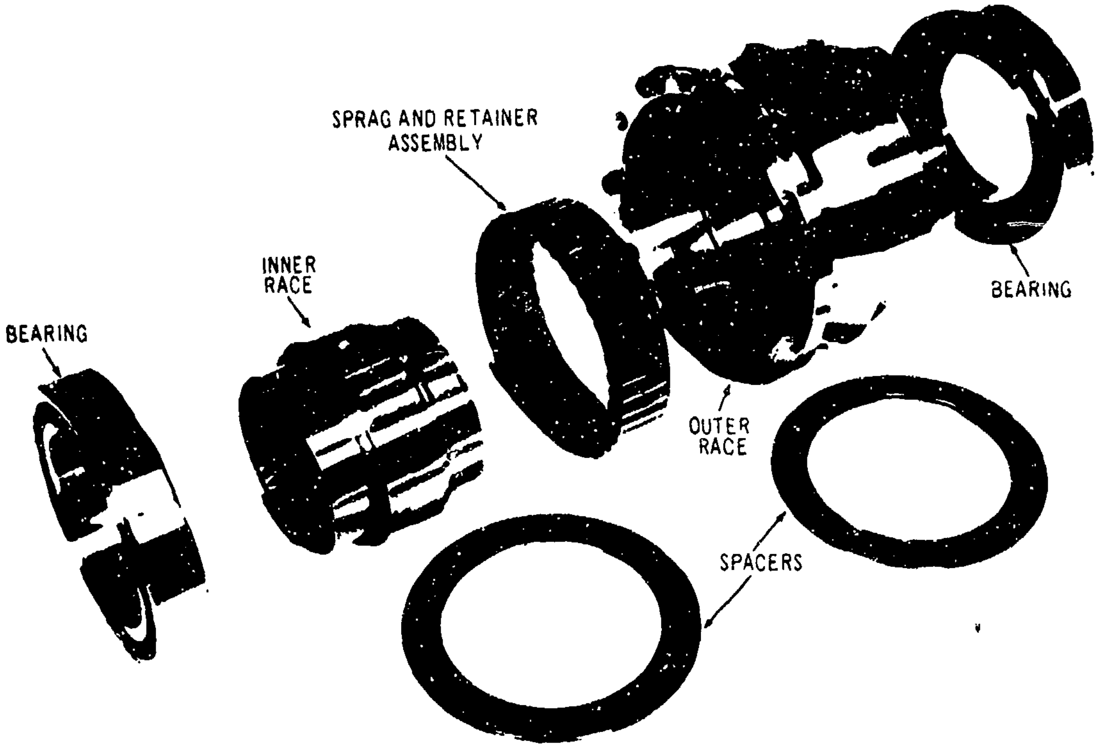


Fig. 7-75. Components of UH-2C Overrunning Clutch Assembly

of the gripping angle must be less than the coefficient of friction.

Sprags are designed to give an increase in gripping angle with an increase in applied torque. This occurs as the sprag rotates under load as a result of the elastic deformation of the races, as shown in Fig. 7-81. A higher gripping angle reduces the radial loads imposed by the sprags, thus permitting higher torques to be transmitted for given values of Hertz (surface compressive) and race hoop stresses.

A typical relationship of gripping angle versus the annular space between races is shown in Figs. 7-82 and 7-83.

7-6.1.2 Sprag Clutch Design

A high degree of refinement is represented in the current state-of-the-art. End-notched sprags are positioned by a precision retainer and energized by expanding garter springs similar to those shown in Fig. 7-76. The retainer keeps the individual sprags in proper axial and angular alignment during overrunning and at the instant load is applied. The optimum configuration provides for individual spring energization of each sprag independent of its neighbor, promoting equal and simultaneous engagement of each sprag in the complement.

7-6.1.2.1 Torque Capacity

Torque capacity is based upon two design considerations: (1) inner race Hertz stress (limitation due to Brinelling) and (2) race design.

7-6.1.2.1.1 Inner Race Hertz Stress

The Hertz stress or surface compressive stress must be maintained within acceptable limits to prevent race distress in the form of Brinelling or pitting due to subsurface fatigue. The highest Hertz stress occurs at the inner race, where the poorest conformity between sprag and race exists. Industry practice is to limit the maximum Hertz stress to 500,000 psi, which will give a life in excess of 500,000 cycles.

Fig. 7-84 shows that the torque T is the summation of tangential components of the load acting through point A of each sprag multiplied by the inner sprag radius $d_i/2$. Thus

$$T = NP\left(\frac{d_i}{2}\right)\tan GA \quad \text{, in.-lb} \qquad (7\text{-}49)$$

where

P = sprag normal force, lb

N = number of sprags

The Hertz stress is based upon the normal component P divided by the effective sprag length. For most retainer designs the effective sprag length L may be taken as the retainer length less 0.20 in. The contact or Hertz stress f_c is expressed by the equation

$$f_c = 0.591\sqrt{\frac{PE}{L}\left(\frac{d_i + d_s}{d_i d_s}\right)} \quad \text{, psi} \qquad (7\text{-}50)$$

where

d_s = outer sprag cam diameter, in.

Solving for P, substituting Eq. 7-50 in Eq. 7-49 and using a value of 500,000 psi for F_c and a value of 30,000,000 psi for E, the Hertz torque capacity is expressed as

$$T = 11.92 \times 10^3 LN\left(\frac{d_i^2 d_s}{d_i + d_s}\right)\tan GA \text{ , in.-lb} \quad (7\text{-}51)$$

This equation expresses the torque capacity if Hertz stress is the limiting criterion and the resultant race deflections at that torque yield the gripping angle assumed.

7-6.1.2.1.2 Overrunning Clutch Races

Because overrunning clutch races usually are integral with shafts and gears, the space available is restricted because of associated components and/or to

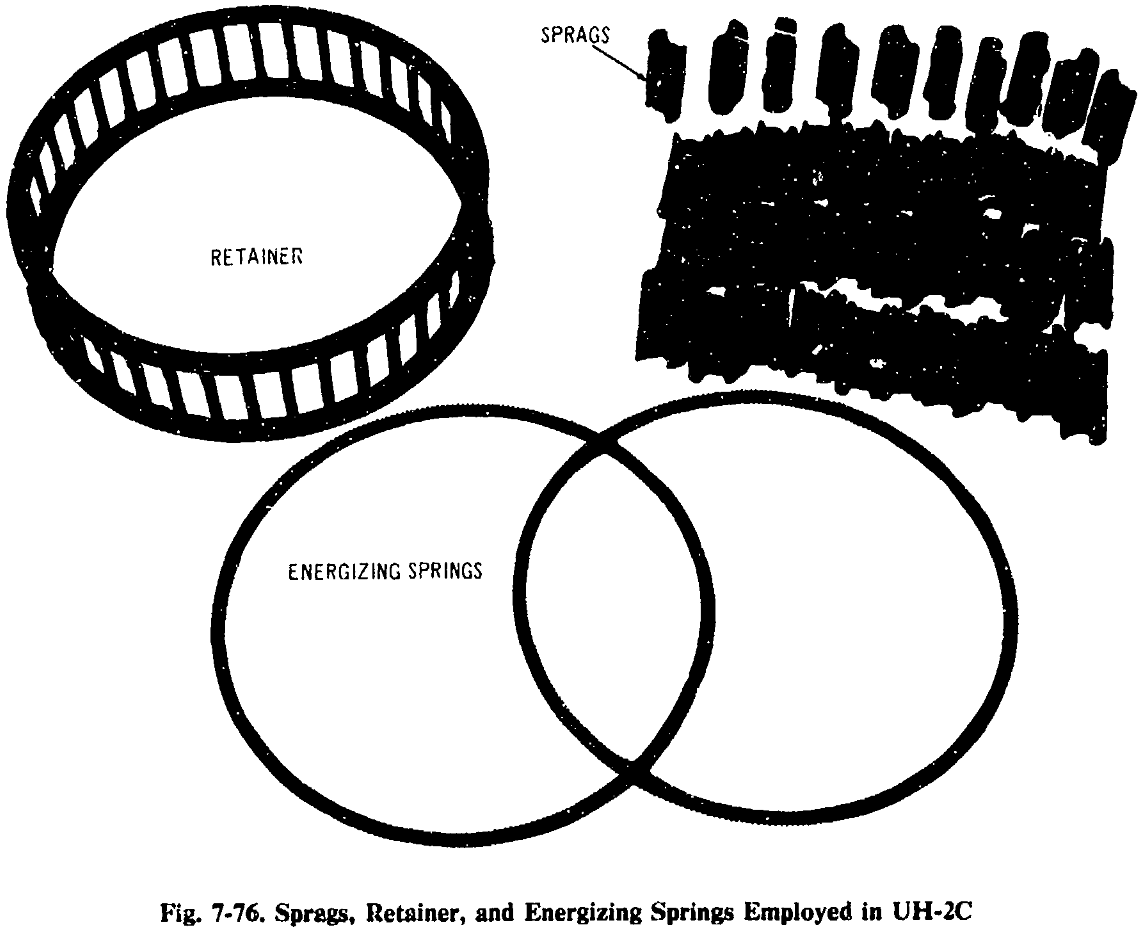


Fig. 7-76. Sprags, Retainer, and Energizing Springs Employed in UH-2C Overrunning Clutch Assembly

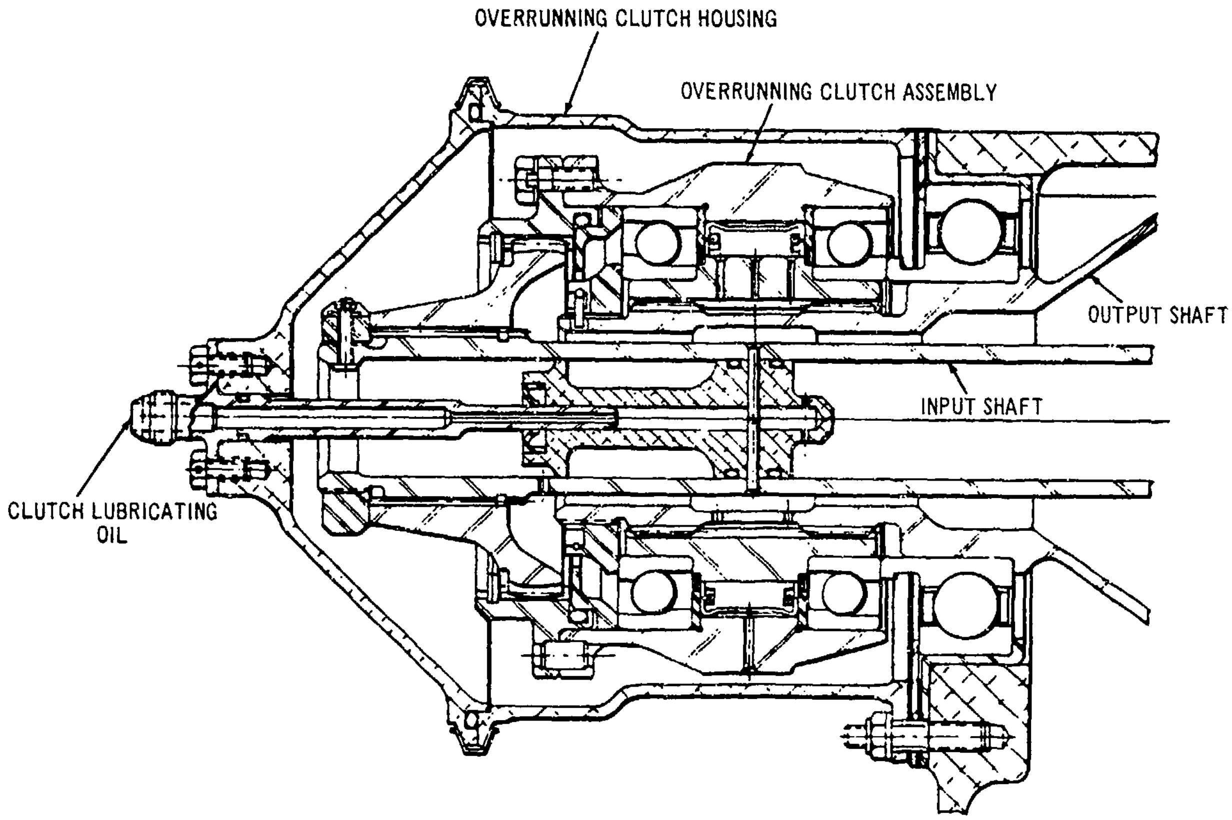


Fig. 7-77. Partial View of UH-2C Gearbox Assembly Showing Overrunning Clutch Installation

minimize weight. Thus the race hoop stresses and/or the resulting race deflection may become the limiting criteria.

The materials generally used are steels such as AMS 6260, AMS 6263, AMS 6265, and AMS 6415. These materials permit adequate case-hardened surfaces required for both inner and outer races while still providing adequate toughness in the core.

The surface hardness should be Rockwell C (R_C) 58-62 while the finished case depth must be 0.050 in. to 0.080 in. when measured to the point of R_C 50. Core hardness should be in excess of R_C 35. A ground surface finish of 10 to 20 micro-inch rms is recommended.

Plain cylindrical races are ideal because their deflection characteristics can be predicted readily. However, it often is necessary to locate the sprag races beneath the web of a gear or over bores of varying diameters. When this is necessary, the races should be designed so that the stiffness is as uniform or symmetrical as possible to minimize bellmouthing of the sprag cavity under load. This condition will result in a nonuniform load distribution along the length of the sprag, producing abnormally high contact stresses and poor load-carrying capability.

Proper clutch function and adequate life also are contingent upon proper control of the annular space between races. The accumulated runout, dependent upon race and bearing eccentricities in addition to bearing clearances, should not exceed 0 003 in. total indicator reading. Axial taper of the sprag race diameters *shall* not exceed 0.0002 in./in.

The magnitude of race deflections must be considered. Current practice is to limit the change in annular space (i.e., the sum of the change in radii of each race) to one-half the total cam rise available. This value is taken as 0.005 in. for the 0.330-in. sprag and 0.006 in. for the 0.375-in. sprag, which are the two sprag sizes common among manufacturers for this type of application.

Allowable design stresses may vary due to in-house practice or experience. The choice of the maximum permissible stress should be tempered by the specific design in question, taking into account torsional excita-

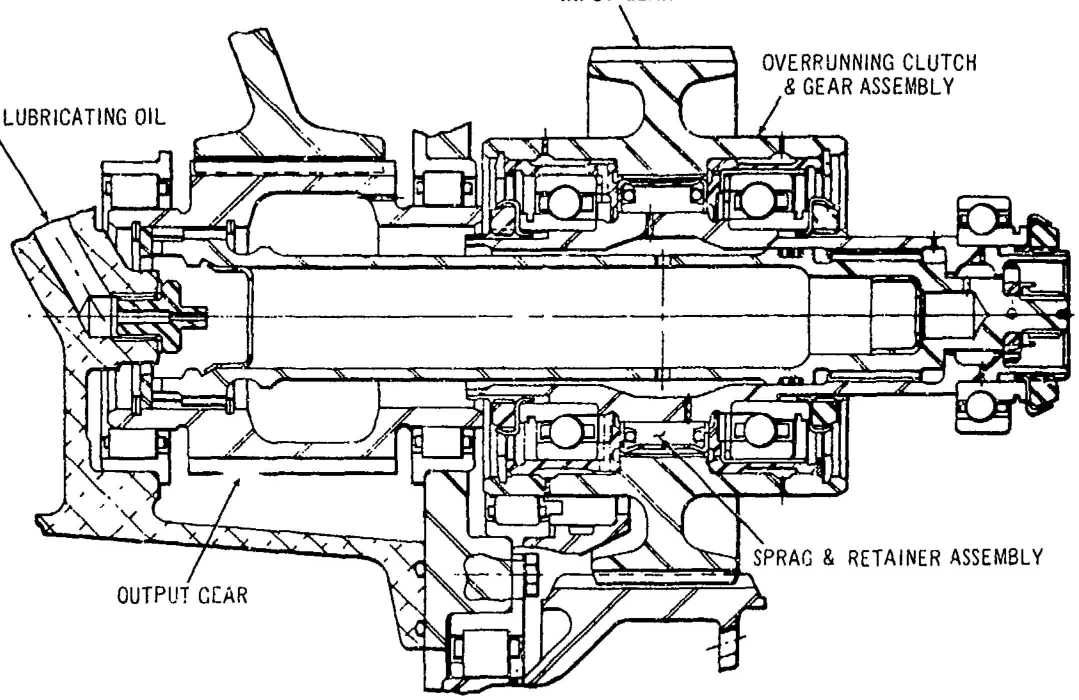


Fig. 7-78. Partial View of Twinning Gearbox Showing Overrunning Clutch Installation

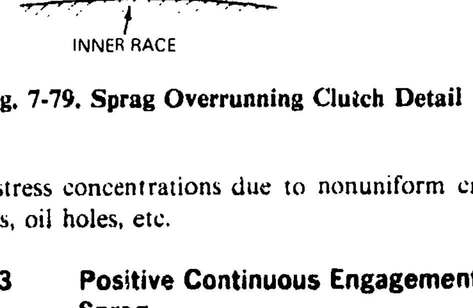


Fig. 7-79. Sprag Overrunning Clutch Detail

tions, stress concentrations due to nonuniform cross sections, oil holes, etc.

7-6.1.3 Positive Continuous Engagement Sprag

The positive continuous engagement (PCE) sprag has been developed recently to meet the increased performance demands of aircraft and helicopter applications. It provides reliable performance under severe torsional and linear vibration as well as high transient overloads.

The unique geometric shape of the PCE sprag limits its movement when it is lightly loaded and is subjected to high torsional and linear vibrations. Control of this movement is made by flank contours that closely limit inter-sprag clearance. Sprag clutch life under these loading conditions is dependent upon how well sprag movement is controlled.

This geometry also offers a sprag that will not "roll over" under excessive torque loads. If extreme overloads are imposed beyond the torque rating of the clutch, the PCE sprag configuration provides a positive sprag-to-sprag abutment. Further sprag motion is prevented by contact of the sprag with the opposite flank of the adjacent sprag.

The three conditions of normal overrunning, normal drive, and extreme overload lockup are shown in Figs. 7-85, 7-86, and 7-87, respectively.

This design requires discrete diameters for a specific number of sprags in the complement to control inter-sprag clearance precisely and to assure lockup under overload. The race diameters given in Table 7-3 are suitable for either conventional "free action" or PCE sprag clutch designs.

7-6.1.4 Overrunning Capability and Lubrication

The overrunning capability of sprag clutches is being challenged constantly as higher shaft speeds and higher power become necessary as vehicle capabilities are extended. The present state-of-the-art is established at an inner race velocity of approximately 10,000 fpm.

Vehicles presently in the design stage will require the development of overrunning clutches that can operate at rubbing velocities as high as 15,000 fpm. This will require increased sophistication in the control of sprag mass distribution, energization, and lubrication techniques.

Lubricants commonly in use are MIL-L-7808 and MIL-L-23699, which are compatible with sprag clutch operation. Lubricants containing antiwear or slippery additives such as molybdenum disulfide, graphites, or phosphates must be avoided. Use of such friction

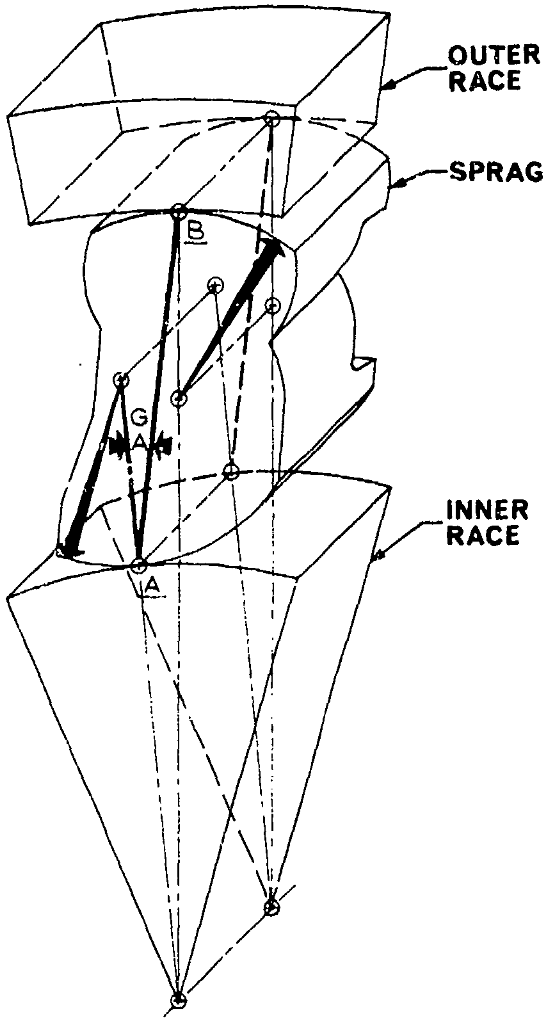


Fig. 7-80. Sprag and Race Detail Showing Gripping Angle

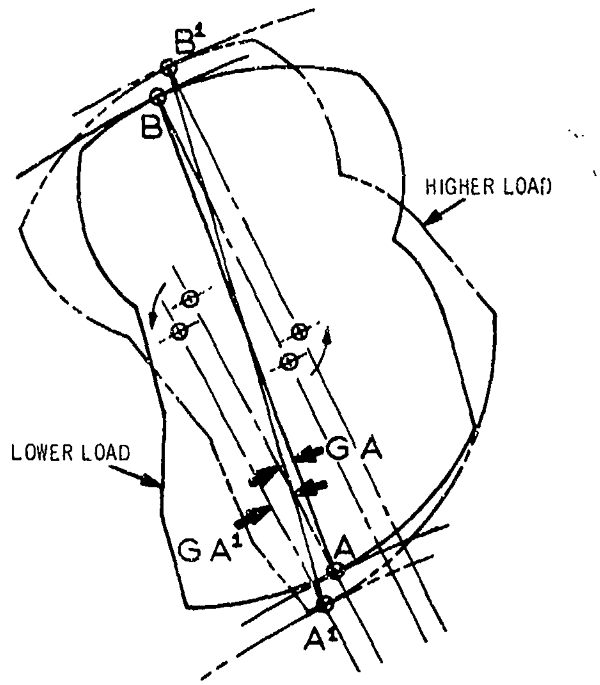


Fig. 7-81. Race Deflection and Increase in Gripping Angle With Increase in Load

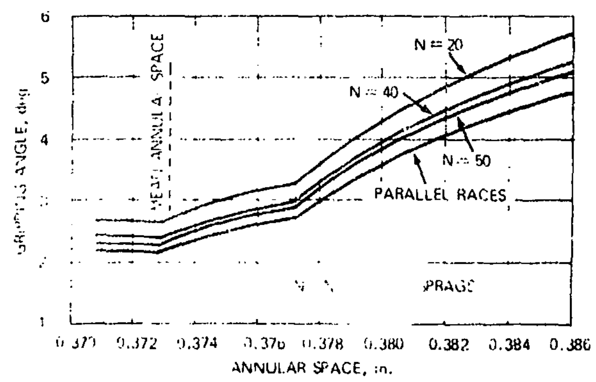


Fig. 7-82. Theoretical Gripping Angle Curve, 0.375-in. Sprag

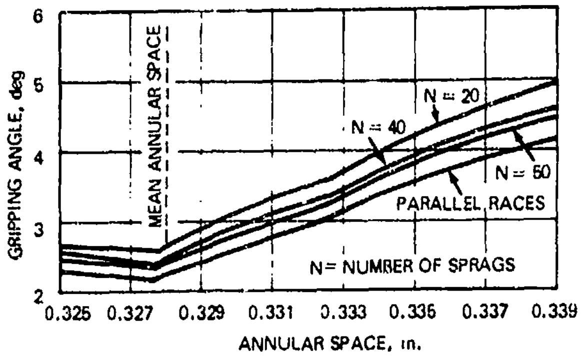


Fig. 7-83. Theoretical Gripping Angle Curve, 0.330-in. Sprag

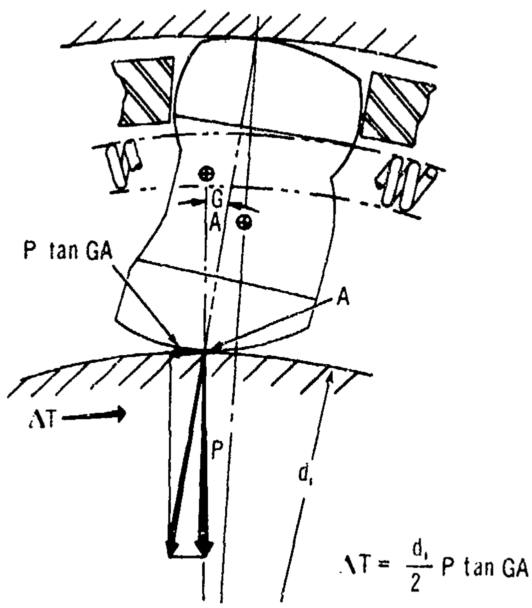


Fig. 7-84. Sprag Force Diagram

modifiers will cause the clutch to function erratically or slip under load.

The quantity of lubricant and the manner in which it is directed are critical. The oil should be directed to the rubbing surface of the inner race through holes directly beneath the sprags. Exit holes also are required in the outer race and should be sized such that a portion of the flow also will flow past the damming washers. The outer race holes are required to purge the sprag cavity continuously, thus minimizing the accumulation of contaminants centrifuged from the lubricant.

An adequate quantity of lubricant must be supplied to keep the inner race wet and to carry away the heat generated during overrunning. The quantity required will depend upon the specific clutch design and the gearbox system with which it is associated, with oil inlet temperature and overrunning speed being most important. Current designs employing a single retainer require oil flows ranging from 0.5 to 1.5 gpm. The actual requirement usually is established during development testing.

Fig. 7-85. PCE Sprags in Normal Overrunning Position

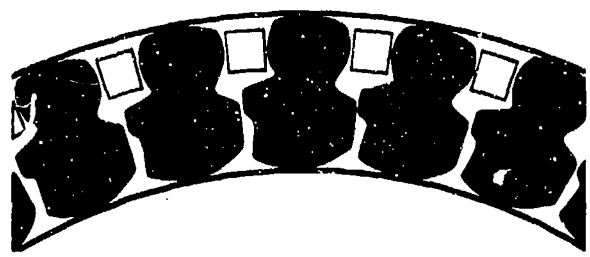


Fig. 7-86. PCE Sprags Driving Under Normal Load

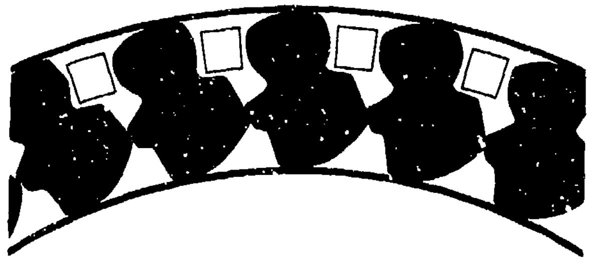
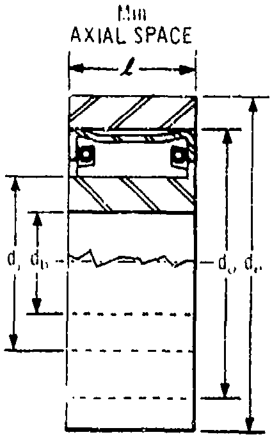


Fig. 7-87. PCE Sprags Driving With Extreme Overload

TABLE 7-3
0.375 in. PCE* SPRAG DATA



*PCE =
POSITIVE CONTINUOUS
ENGAGEMENT

No. OF SPRAGS N	INNER RACE DIA, in. d	INNER RACE BORE, in. $d_{[illegible]}$	OUTER RACE DIA, in. d_o	OUTER RACE OD, in. $d_{[illegible]}$	TORQUE CAPACITY, lb-in. 0.625 in. Min AXIAL SPACE	0.750 in. Min AXIAL SPACE	0.875 in. Min AXIAL SPACE	1.000 in. Min AXIAL SPACE
18	1.3966	0.8100	2.1433	2.6800	2,700	3,260	3,800	4,350
19	1.4931	0.8600	2.2396	2.8100	3,160	3,340	4,480	5,120
20	1.5395	0.9200	2.3361	2.9500	3,660	4,400	5,130	5,860
21	1.6660	0.9700	2.4326	3.0900	4,230	5,080	5,930	6,780
22	1.7624	1.0300	2.5290	3.2100	4,790	5,750	6,700	7,660
23	1.8739	1.0800	2.6255	3.3500	5,430	6,520	7,620	8,700
24	1.9753	1.1400	2.7219	3.4800	6,070	7,300	8,500	9,720
25	2.0718	1.1900	2.8183	3.6200	6,820	8,200	9,550	10,920
26	2.1682	1.2500	2.9147	3.7600	7,600	9,130	10,640	12,180
27	2.2647	1.3100	3.0112	3.9000	8,420	10,100	11,750	13,440
28	2.3611	1.3600	3.1076	4.0300	8,730	10,470	12,240	13,980
29	2.4576	1.4200	3.2041	4.1800	9,250	11,070	12,900	14,760
30	2.5540	1.4700	3.3005	4.3100	9,370	11,250	13,100	15,000
31	2.6505	1.5300	3.3970	4.3500	9,500	11,400	13,300	15,200
32	2.7469	1.5900	3.4934	4.5800	10,420	12,500	14,600	16,680
33	2.8435	1.6400	3.5900	4.7200	10,850	13,050	15,200	17,400
34	2.9355	1.7200	3.6820	4.8700	11,250	13,500	15,750	18,000
35	3.0275	1.7700	3.7740	5.0000	11,700	14,000	16,350	17,720
36	3.1195	1.9000	3.8660	5.2000	12,100	14,500	16,950	18,720
37	3.2115	2.1000	3.9580	5.4500	12,300	14,780	17,250	19,680
38	3.3035	2.2200	4.0500	5.6700	12,680	15,200	17,750	20,280
39	3.3955	2.3200	4.1420	5.8400	12,750	15,300	17,820	20,400
40	3.4875	2.4100	4.2340	6.0000	12,900	15,450	18,030	20,600
41	3.5795	2.5000	4.3260	6.1800	13,000	15,600	18,200	20,760
42	3.6715	2.6000	4.4180	6.3500	13,180	15,800	18,450	21,120
43	3.7635	2.7000	4.5100	6.5400	13,420	16,100	18,800	21,480
44	3.8555	2.8000	4.6020	6.7400	13,650	16,370	19,100	21,840
45	3.9475	2.9000	4.6940	6.9400	13,800	16,550	19,320	22,140
46	4.0395	3.0000	4.7860	7.1400	14,000	16,850	19,650	22,440
47	4.1322	3.1000	4.8787	7.3400	14,200	17,060	19,900	22,740
48	4.2249	3.2000	4.9714	7.5400	14,400	17,250	20,100	23,040
49	4.3176	3.3000	5.0641	7.7500	14,500	17,400	20,300	23,160
50	4.4103	3.4000	5.1566	7.9700	14,600	17,500	20,500	23,400

It is important that the gearbox designer work closely with the clutch manufacturer early in the design stage so that the overrunning function may be optimized without unnecessary compromise to either gearbox or clutch design.

7-6.1.5 Roller Clutch

Because in the past sprag clutches had been known to "roll over", some helicopter engineers selected roller clutches, which do not have the "rollover" characteristics. A roller clutch designed to transmit the same amount of torque, however, usually is heavier than the same sprag type clutch.

7-6.2 SPLINES

A spline is a machine element consisting of integral keys (spline teeth) or keyways (spaces) equally spaced around a circle or portions thereof. Splines are used principally to transmit torque from member to member. They are practically indispensable in helicopter systems, and are used in various forms on almost all rotating parts. The advantages of spline drives over other systems such as keyways and keys, serrations, pins, set-screws, and friction devices may be listed as follows:

1. No loose parts such as pins and keys required
2. High torque capacity for available space; high strength/weight ratio
3. May be self-centering

4. Given spline capacity can be upgraded by easily applied production techniques such as heat treatment, case hardening and grinding, and higher precision tooling.

5. Provides for varying axial positions when required

6. Low cost

7. Easily installed and disassembled in the field.

This paragraph describes the various types of splines and their application to helicopter systems. Applicable standards have been developed that will aid in the detail design. Reference should be made to AMCP 706-202 for detail design data.

7-6.2.1 Types of Splines

Various types of splines have been used in the past on helicopters.

The Society of Automotive Engineers (SAE) standards for involute splines have been used the most generally throughout the aircraft industry. This spline design is a variation of an involute gear tooth but having one-half the effective tooth height and using a 30-deg pressure angle. Many variations of the involute tooth spline have been used in helicopters, from the SAE tooth geometry standards to a low-pressure-angle, full-depth form. This involute spline is the most widely used of the listed types due to its ease of manufacture with standard gear cutting equipment, and the simplicity of inspection techniques. The Aerospace Gearing Committee of the American Gear Manufacturers Association (AGMA) is in the process of forming a new spline standard incorporating lower pressure angle systems that are more advantageous to the latest state-of-the-art helicopter design.

7-6.2.2 Typical Applications

Involute splines, as used in transmission systems, differ in application depending upon the types of members they join.

For involute spline joints, the various applications are:

1. Side fit spline — free floating
2. Major or minor diameter fit spline
3. Side fit spline with piloted members.

The side fit spline may be used as a self-centering type drive where angular freedom is desired and/or where radial looseness is desired to eliminate potential radial forces. Fig. 7-88 shows a free-floating drive shaft, which accommodates small misalignments between the driving and driven component.

The major or minor diameter fit spline is used where accuracy in positioning is required. This type fit may be used for mounting low-power gears or for components having moderate to high speed, where small radial offsets will not produce excessive vibrations. Fig. 7-89 is a typical installation on a moderate-speed drive shaft. A variation of the major or minor diameter fit has been used for flexible ring gear retention. Static radial clearance is provided between the gear and its housing but the major or minor diameter fit limits the radial deflection of the ring gear under load.

A side fit spline, with piloted members, is used typically for mounting a gear to a shaft. The pilots on the mating member are used to react the radial forces from the gear, allowing the spline to carry only the torque. Fig. 7-90 shows a typical gear mounting with pilot diameters. Fig. 7-91 is a curvic spline used for joining shafting with a quick-disconnect feature.

7-6.2.3 Design Considerations

In the preliminary design phase, the prime objective of the designer is to provide a geometric arrangement of a spline drive having adequate power capacity without a potential envelope redesign of the drive or the requirement for exotic manufacturing controls to maintain the required boundary limits. To attain the desired boundary limits initially, and in a simple form, the designer must formulate equations or general rules. For example, the face length-to-diameter (L/d) ratio of splines generally must be limited to insure an effective contact and bearing pattern using various types of manufacturing techniques. As a guideline, a spline length-to-diameter ratio of 2:1 or less is preferred. Where values of 1:1 are used, the basic design objective of load capacity can be achieved more readily without imposing a high degree of machining accuracy.

Most splines are critical from a wear factor (bearing stress) rather than from limit load design values caused from shear or bending. The use of maximum bearing stress allowances of the material at limit load is not a good standard for attaining satisfactory wear life. Also, there is no balance of shear capacity to wear type bearing allowables in the 30-deg pressure angle spline systems. A reduction of pressure angles, as noted by the AGMA Aerospace Spline Task Committee, will provide a better means to balance the spline geometry.

In some cases, standard gear tooth geometry (working depth $= 2.0/P_d$ where P_d is the diametral pitch, or number of teeth per inch of diameter) and moderate pressure angles of approximately 20 deg or less are more suitable spline applications in the light alloys (aluminum). The common $2.0/P_d$ working depth

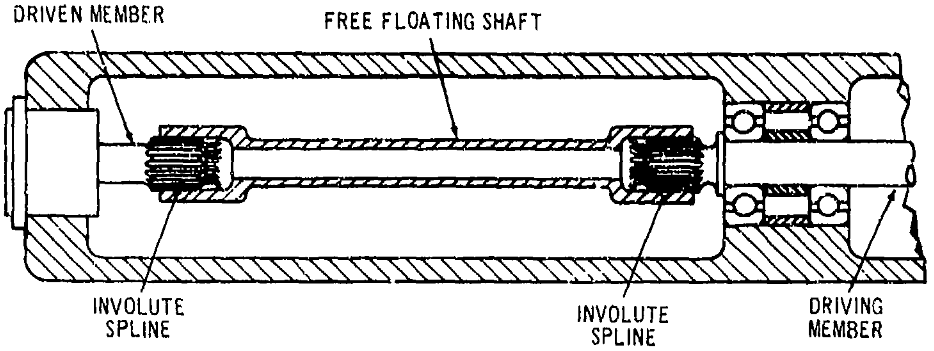


Fig. 7-88. Side Fit Spline

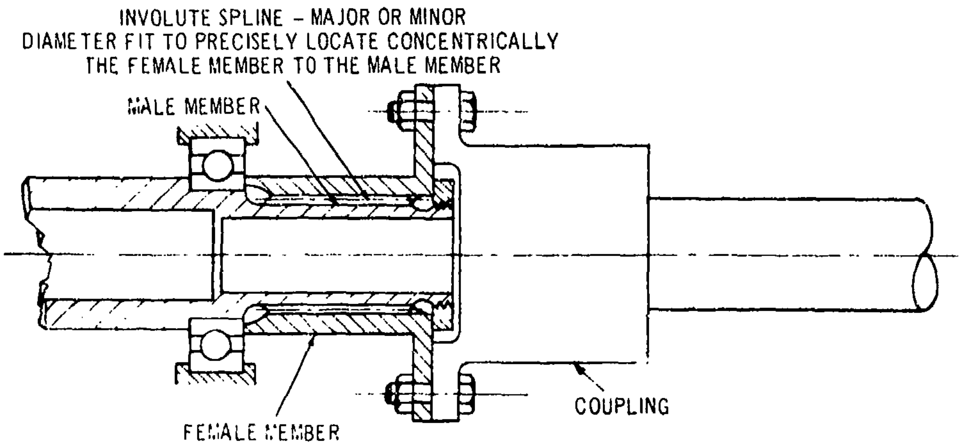


Fig. 7-89. Major or Minor Diameter Fit Spline

reduces the bearing stress by a factor of two, along with the reductions due to the lower pressure angles.

Although many variations can be presented, as a rule the designer can still use the SAE involute standards as the basic geometry in the preliminary design phase for steel-to-steel mating components. Par. 7-6.2.4 presents guidelines in establishing size of SAE splines.

7-6.2.4 Spline Capacity (SAE Type)

There are numerous allowable stresses permitted for splines, depending upon the application and method of design. For example, a clamped-up tight installation will not "work" and, therefore, will not develop readily into a fretting corrosion situation. Safe L/d ratios are dependent upon shaft section properties high enough to prevent excessive tooth end loading from torsional deflections. Accuracy of spline tooth geometry determines the percentage of total teeth capable of sharing the load. Material factors and heat treatments, together with surface finish and lubrication, also affect spline life.

In helicopter applications, most splines are highly loaded because lengths are minimized to reduce weight. These splines generally will be made from high alloy steels, with or without case carburizing and grinding to attain high tooth accuracies.

As in all helicopter applications, fatigue strength must be considered whenever stress concentrations exist. For this reason, a full or large fillet radius is required to minimize stress raisers at the highly loaded tooth root area. This requirement normally eliminates the common automotive practice of using minor diameter fits and flat root areas that require sharp radii.

High-capacity splines usually drive subsystems or large mass components at high speeds and require accurate mounting. The helicopter spline should be designed to accommodate torque only and not be subjected to additional bending stresses. For these reasons, pilot diameters should be used to provide mounting ac-

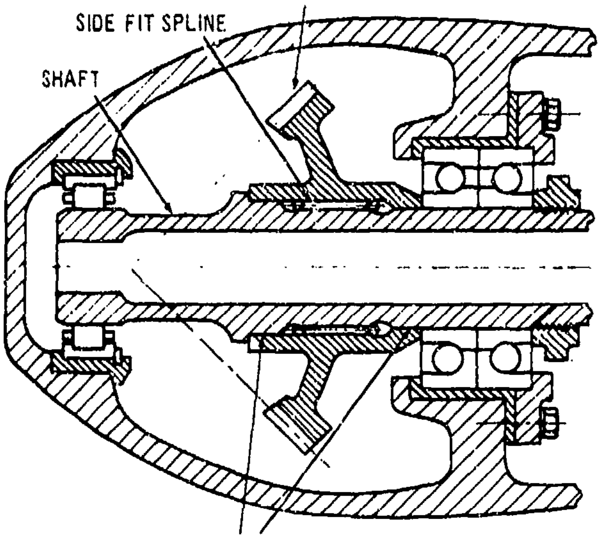


Fig. 7-90. Side Fit Spline With Piloted Members

curacy and load reaction. These diameters, usually line-to-line fits, must be square with the mounting faces to prevent axial clamp-up forces from adding bending loads to the spline teeth.

The sizing of SAE involute splines for preliminary design purposes is limited to the bearing stress values. The bearing stress value has been the limiting factor due to past experience with fretting and wear problems. The bearing stress f_{br} (psi) is a function of torque T (lb-in.) pitch diameter d, and length L (in.). These relationships are shown in Eq. 7-52.

$$f_{br} = 2T/(d^2 L) \quad \text{, psi} \tag{7-52}$$

The allowable values of bearing stress F_{br} for various types of assemblies and manufacturing techniques are presented in Table 7-4. These allowable stresses are based upon maximum operating power conditions.

For steel-to-aluminum spline members, the bearing stress should not exceed values of 500 psi for loose fit and 1,000 psi for clamped-type fit.

7-6.2.5 Materials, Metallurgy, and Lubrication

The important predesign criteria for spline applications involve the required length and its effect upon component size. Shorter spline lengths are possible as allowable bearing stress F_{br} increases. Future helicopter spline loading will increase when better understanding of materials under wear conditions becomes available. Higher precision splines, with improved lubricants or dry film treatments, also will help to reduce component weight.

7-6.2.6 Failure Modes

Splines normally will fail-safe because the load is shared by a large percentage of the teeth. Exceptions have been noted when sharp fillet radii have caused stress raisers, permitting material fatigue failures. This type of failure can be serious as the fracture may open the drive train.

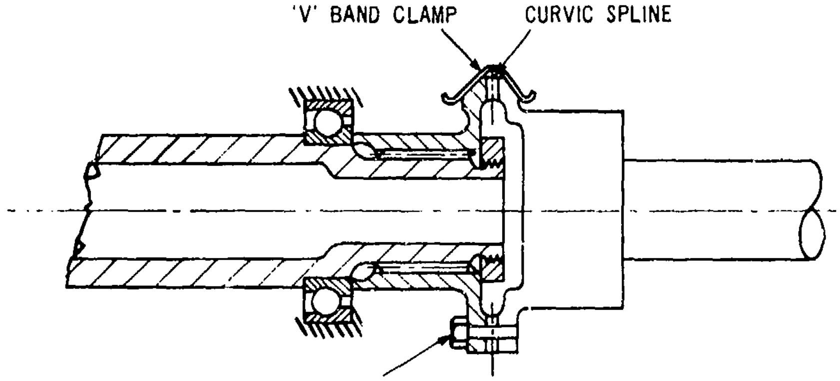


Fig. 7-91. Curvic Spline Shaft Attachment, Quick-disconnect

Failures usually cause premature component replacement, short TBOs, and excessive heat through wear and fretting corrosion. Proper lubrication and surface hardness for some applications are required to reduce wear. A clamped, nonworking spline normally will provide longer life.

7-6.3 MECHANICAL COUPLINGS

Mechanical couplings are used in all helicopters to accommodate the mismatch of drive lines, such as gearbox to shafting or shaft to shaft. The drive line mismatch is angular and axial, with the total amount being a function of installation errors, airframe deflections, and dynamic motions of the related components. Past history has shown that a large degree of attention is required in the coupling design to insure trouble-free performance.

Coupling problems occur from many causes and may result in safety-of-flight problems or excessive downtime and maintenance requirements. For example, a coupling failure on a tandem rotor synchronizing drive shaft is catastrophic. In other applications, if autorotation is successful, only an incident or forced landing results. Some failures may result only in excessive vibration, permitting a get-home capability.

Common coupling problems historically may be listed in order of importance as follows:

1. Fatigue failures, causing a disconnection in the drive train
2. Loss of lubricant or improper lubrication, causing excessive heat and welding of components, which in turn generate high force-low cycle fatigue loading into shaft end fittings
3. Improper maintenance such as shimming errors, inadequate torque on fasteners, scratches or dents in critical areas, operation beyond authorized life, too much or too little or wrong grease, and original quality control human errors.

TABLE 7-4
ALLOWABLE STRESS FOR STEEL SPLINES WITH VARIOUS MOUNTING AND TOOTH PARAMETERS

	ALLOWABLE BEARING STRESS F_{br}, psi		
ASSEMBLY	AS MACHINED NO LUBRICATION	GROUND	HARDENED AND GROUND
LOOSE FIT (FLOATING)	5,000	7,500	15,000
CLAMPED FIT	7,500	10,000	20,000

IN SOME APPLICATIONS PAST EXPERIENCE HAS SHOWN THAT HIGHER ALLOWABLES MAY BE USED DEPENDING ON

1. LOW L/D RATIOS
2. LUBRICATION
3. MATERIALS
4. DEGREE OF PRECISION

ALSO, IN SOME APPLICATIONS, PAST EXPERIENCE HAS SHOWN LOWER ALLOWABLES MUST BE USED DEPENDING ON

1. HIGH L/D RATIOS
2. EXCESSIVE FLEXURAL MOVEMENT OF SPLINE MEMBERS
3. POOR GEOMETRY

In general, the more misalignment a coupling must accommodate, the more complex is the design. Operating and installation deflections, therefore, should be kept to a low value to insure a simple system with a minimum of friction and/or overloading. The design should minimize the requirement for care and skill during maintenance. Long-life components requiring inspection or disassembly only at long periodic intervals will prevent most in-service human errors. Corrosion becomes an important consideration in Army environments where surface protections may be lost or become inadequate to protect against pitting or stress raisers.

The aircraft environment must be considered because couplings often are located near engines and hot exhaust gases. When heat-sensitive couplings (grease-lubricated) are adjacent to engines, heat insulators or cooling provisions may be required for protection from conduction and radiation of engine heat. Seals and other surfaces must be insensitive to exhaust gases.

There are four basic coupling types presently used on helicopter drive systems:

1. Laminated ring flexures
2. Flexible disks
3. Crowned gear coupling
4. Universal joints.

A fifth, the elastomeric type, may become significant if the present R&D programs are successful. The types most commonly used are shown in Figs. 7-92, 7-93, and 7-94. All these types effectively will relieve cyclic bending load at shaft ends caused by misalignments. Each coupling has certain advantages over others depending upon the application requirements. Each type of coupling is discussed in the paragraphs that follow.

7-6.3.1 Laminated Ring Couplings

The laminated ring coupling consists of a dozen or more thin stainless steel rings. The input member is bolted to the ring pack at two or three equal positions and contacts the pack only at these localized areas. Located at an equal distance between the input attach-

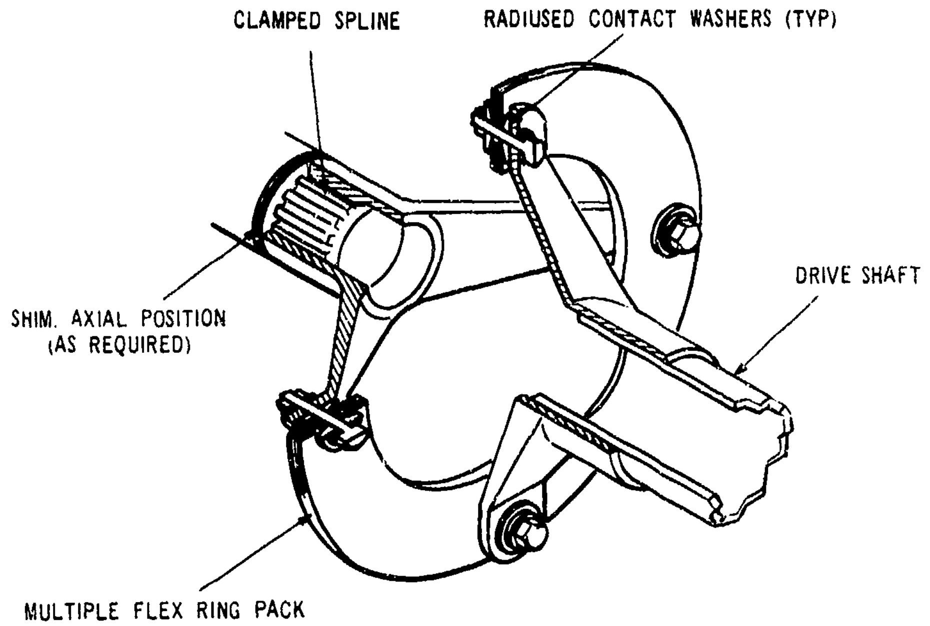


Fig. 7-92. Typical Laminated Ring Coupling

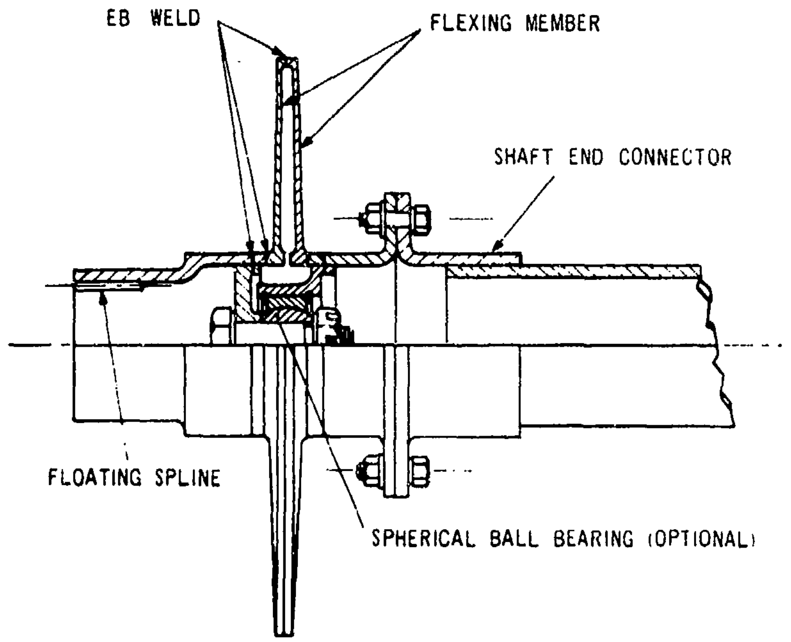


Fig. 7-93. Typical Flexible Disk Coupling

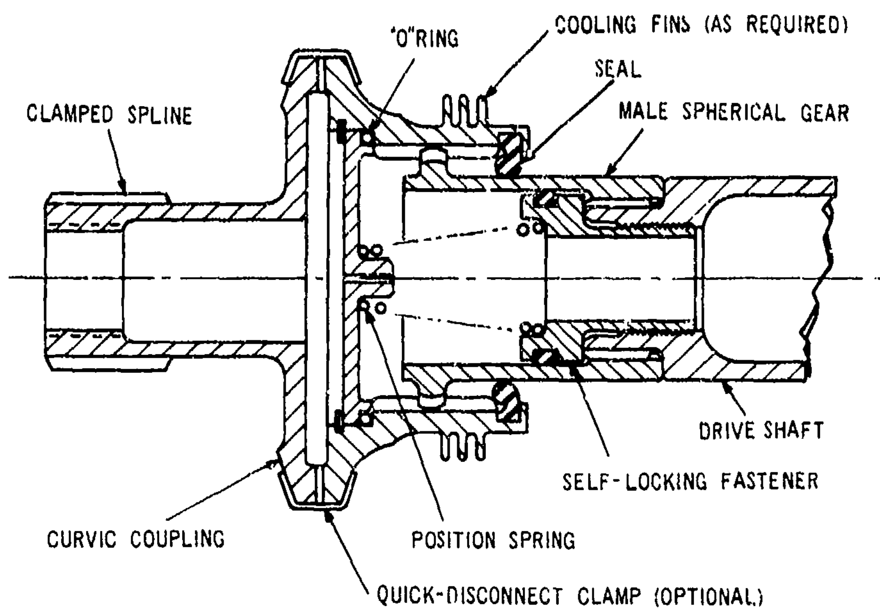


Fig. 7-94. Typical Gear Coupling

ments is the bolted output member. The bolted connections employ specially shaped washers so that flexing of the disk pack is permitted without a stress concentration arising at the corners of the bolted members.

The thin disks are sized and stacked to carry the required steady and cyclic torque, and to bend statically and cyclically to accommodate the misalignment at a stress level below the material endurance limit. Additional laminates are added to improve reliability and permit operation with one or more cracked rings. Each ring is separated by a thin sheet of Teflon to delay or restrict fretting corrosion.

This coupling is applicable especially for systems with small misalignments. It is lightweight, requires no lubrication, and has no moving parts. Because the multiple rings are made from very high-strength stainless steel, environmental problems of heat and corrosion are eliminated. The ring stack is almost immune to handling damage, i.e., scratches and dents, but is subject to fretting corrosion, especially if the connection is not kept tight.

Glass fiber flexure members permit higher operating angles or higher potential life without the fretting problem. They are limited to low to moderate temperature ranges and, therefore, cannot be used in fire zone areas.

The major advantage of this type of coupling over other couplings is in the redundancy of multiple load-carrying members that are inspected easily for cracks on the outer rings. The most highly stressed rings (and the first to fail) are the most easily inspected. A coupling failure occurs gradually and, therefore, will allow a good possibility of safely returning from a mission.

7-6.3.2 Flexible Disk Couplings

The flexible disk coupling also relies upon the flexing of a high-endurance-strength, thin steel member. Torque is transmitted from the input to the ouput member through two or more thin disks, electron beam (EB)-welded together. The reliability of this coupling depends upon an advanced degree of manufacturing and quality control supported by 100% acceptance testing under dynamic load conditions. EB proprietary welding, disk geometry, and material control requirements currently preclude second sources for this component.

Larger misalignments, i.e., greater than 1.5 deg, may be accommodated by stacking two or more EB-welded disk assemblies. A ball joint bearing at the coupling centerline and axially located to permit only angular deflections is required on most applications. By eliminating the steady stresses caused by axial deflections, the ball joint permits higher allowable cyclic stresses from angular misalignment. A floating shaft end is required because no axial deflections can be accommodated except by means of sliding spline action. The ball joint bearing must be capable of withstanding operating torques when subjected to axial forces through spline friction. Wear also must be considered from oscillating motion. Recent Army-funded testing programs have verified that the ball joint floating end concept eliminates the problem of accurately shimming for low axial deflections.

The advantages of the laminated ring coupling apply also to the flexible disk coupling; i.e., no lubrication is required, fatigue stresses can be designed to be below the endurance limit, the unit is lightweight and simple, and reliability is high. Additionally, the flexible disk coupling is not subject to fretting corrosion and can, therefore, be designed for unlimited life. The ball joint supports the shaft in the event of outer disk failure and, except for torque-carrying ability, provides a redundancy to reduce secondary damage.

The Bendix coupling does not exhibit the degree of redundancy possible in the laminated ring coupling but is inspected more easily for stress raisers or crack formation. Battlefield vulnerability is considered about equal for both coupling types.

7-6.3.3 Gear Couplings

The gear type of coupling has been used widely when relatively large axial and angular deflections must be accommodated. Soft-mounted helicopter pylons (UH-1) require couplings that can operate in the region of 3-4 deg of misalignment and with appreciable axial motion. Also, the coupling must permit momentarily higher angles to pylon travel limits without a lockup. The crowned gear teeth on the male member have been developed as a highly specialized spline using gear tooth geometry.

These couplings also are used to replace splines within gearbox and component drives when some misalignment capability is required and oil system lubrication is available. For example, the output drive shaft to the rotor hub may be coupled to the gear shaft with a gear coupling to accommodate angular deflections of the fixed mast hub support. This arrangement is feasible when the drive shaft is loaded in torque only and is not required to react rotor bending loads. Because axial motion can be accommodated easily, no shimming for axial position is required during installation.

Drive shafts employing gear couplings are assembled easily in blind installations with one end floating. Some provisions must be made to insure operation within axial position limits. This has been accomplished through component tolerance control and/or spring arrangements.

The spherical tooth form (barrelling) permits a rocking motion within the tooth contact area, while the outside diameter is crowned spherically to prevent jamming at misalignment limits. At zero misalignment, the full addendum gear teeth act like a spline. As the misalignment angle increases, fewer teeth are in contact until at extremes only a small percentage of the teeth are carrying the load.

Lubrication becomes extremely important because contact pressures are high and increase with misalignment angles when fewer teeth are in full mesh. A sliding action, which must be lubricated and cooled, occurs during misalignment. At angles less than 0.5 deg, pressurized oil jet lubrication may be used. Special high-pressure EP greases have been developed that are suitable for a misalignment range up to 4 deg when operating at moderate temperatures and speeds. At high speeds (10,000-30,000 rpm), a highly developed oil lubrication system may be required.

When compared to the flexing member couplings previously described, the gear usually is more rugged but also is more complex and heavier. It requires frequent servicing and the use of a special grease, which complicates the Army's logistic system.

7-6.3.4 Universal Joints

This automotive joint has been used on Army helicopters when large misalignment angles and relatively slow shaft speeds are the design parameters. Its use is restricted severely because the product of misalignment angle and operating rpm is a detrimental factor in its power ratings. Limiting values of this product normally are listed by manufacturers. Exceeding the recommended values may result in serious developmental problems.

A specialized joint or coupling is required at each shaft end to accommodate the variations in velocity inherent in a universal joint type of system. The resulting accelerations produce high forces and motions that must be accommodated by a bearing arrangement at each end. Angular velocity changes of Cardan and Hooke's type of joints are described in many engineering handbooks such as Ref. 17.

Universal joint components, including the shaft, are rather complex and heavy. Grease lubrication normally is required, which becomes a service problem in the Army environment. U-joints are rigid axially and, therefore, require a floating member to accommodate axial misalignments or deflections.

7-6.3.5 Elastomeric Couplings

Elastic materials have been developed to perform the function of mechanical couplings. They are not developed sufficiently as yet to compete with the previous candidate systems in most helicopter applications. This is because high cyclic torsional stresses occurring from rotor systems cannot be accommodated concurrently with angular misalignment. Also, the softer materials lower the drive system natural frequencies and must be considered dynamically with each interfaced subsystem.

One company has developed a ball and socket arrangement surrounded by layers of rubber, sandwiched between and bonded to thin steel members. Rubber can be made stiff enough to carry the high torque in compression and soft enough in shear to accommodate cyclic motion due to misalignment. It is difficult to carry cyclic stresses superimposed on the steady torque loads. Further development may solve this problem and permit an inherently rugged, fail-safe, and maintenance-free coupling to join the candidate systems for predesign considerations. It is unlikely, however, that an unlimited life will be possible because of material deterioration with time.

For small helicopters or low-power subsystems, belt drives can be considered as elastomeric couplings and they may be used in specialized applications. Whenever components must be driven on a parallel axis from the driver with a limited amount of misalignment (cooling fans), belts may be an efficient arrangement.

Adequate reliability and vulnerability can be attained with proper design and multiple belts can be acceptable. Wear and environmental effects preclude an unlimited life so that accessibility provisions must be made for periodic inspection and replacement of belts. Requirements and provisions for frequent inspection should be considered in the helicopter system specification.

7-6.3.6 Shaft Installation

The large-diameter shaft may be swaged at each end to receive a structurally bonded and/or mechanically fastened end fitting. This fitting may take the form of a bolted flange or splined rigid joint connector to the coupling. A curvic coupling connector provides a well-positioned drive with good alignment. This design provides for a quick-disconnect type of joint.

Couplings should not be attached permanently to a subsystem component or the drive shaft assembly because of a probability of different replacement times, TBOs, service lives, damage, or inspection requirements. Couplings are precision components subject to handling damage and should be protected when shafts or gearboxes are removed.

The connection of the coupling to the gearbox usually is made by means of a spline at the end of the gear shaft. The spline may be a floating type to allow for axial deflections and installation tolerances. In this case, it should be lubricated and sealed to allow axial motion at a low coefficient of friction when transmitting torque.

One end of the shaft usually is connected rigidly through the coupling to the gearbox by means of a clamped-up spline. In this case, no lubrication is required and the spline can accept higher stresses. Care must be exercised that the clamping bolt(s) maintain torque and cannot be backed out since loss of the fastener can cause the fixed end of the shaft to become disconnected and result in a safety-of-flight hazard. The system specification must provide specifically for fastener integrity and attention to design details—including redundancy in design when performance and safety are involved

7-6.4 ROTOR BRAKES

Rotor brakes are mechanical-hydraulic systems used to stop the rotor on shutdown or to prevent the blades from turning until actuated. Such brakes have been used most commonly for shipboard helicopter operations by the U.S. Navy. Army airborne units also will need rotor brakes for the same operational reasons plus those peculiar to the battlefield environment.

Existing specifications such as MIL-T-5955 and FAR 29 are incomplete because most design parameters are left to the discretion of the manufacturer. This paragraph includes the safety features of present specifications and provides general design criteria that establish a baseline for the development of system specifications for future or retrofitted helicopters.

Qualification testing is required for rotor brake systems because loads and torques are introduced into the propulsion system and blades from other than normal sources. System performance must be demonstrated on an instrumented ground test vehicle so that all loads are accounted for when component service lives are submitted in the model specification. The integrity of the hydraulic system must be demonstrated with proof and burst pressure tests.

Reliability testing exceeding the requirements of MIL-T-5955 also should be considered whereby the number of cycles, at the previously determined operating conditions, are run to provide a TBO equal to the drive system. Duty cycles should be established for these test programs.

7-6.4.1 Need for Rotor Brakes

Rotor brakes are needed in order to extend the utility of Army helicopters and to minimize the hazard to ground or deck personnel when securing the vehicles.

Cost-effectiveness of rotor brakes probably can be shown when aircraft ground accidents or damage occur from concentrated helicopter activity in congested areas. Requirements for rotor brakes may be derived from the following partial list of operational demands:

1. Quick dispersal and rapid concealment
2. Shipboard operation in rough seas or high winds
3. Emergency stopping when fast egress is required (water)
4. Prevention of rotors from turning due to winds or downwash from other aircraft
5. Reduction of rotor rotation time on the ground at the lower rpms to minimize hazard to ground personnel
6. Shielding of turning rotors, which attract attention from hostile aircraft overhead.

Because the need for rotor brakes may vary depending upon the aircraft mission and operating environment, provisions for a kit installation should be made on all future model specifications.

7-6.4.2 Full On/Off vs Controllable Systems

A full-on/full-off system is an automatically controlled force system that eliminates the pilot from the control loop except for the on-off function. This system is complex and subject to malfunctions. It also prevents the pilot from using a judgment factor that might be required under emergency conditions.

The controllable magnitude brake allows the pilot to control the rate of rotor deceleration by virtue of the mechanical advantage and input force he exerts on the brake handle. These are the simplest (and safest) systems, and allow faster than normal stopping of the rotor in emergencies such as:

1. Whenever ditching and fast ejection are required as in some water landings
2. The occurrence of a mechanical malfunction in which a dynamic problem or high vibration level is present in the rotor system
3. A tendency to enter a ground resonance condition
4. An operational requirement in a combat situation that makes fast deployment or shutdown necessary
5. Heavy sea states on ship decks where rapid securing is necessary.

A trade-off exists for the pilot whereby he may decide that a fast shutdown is worth a possible short life or excessive heating of the brake system.

7-6.4.3 Description of a Typical Simple System

A typical rotor brake for small helicopters will stop the rotors from 75% power turbine output rpm N_p in less than 30 sec and be capable of actuation at 100% N_p without overheating. Emergency operations will permit faster stopping but may require component inspection or replacement. A minimum duty cycle of 10 actuations per hour or at 5-min intervals will preclude use of excessively heavy systems to prevent disk warpage.

Certain safety features are required to prevent brake operation in flight. This may take the form of a mechanical lever lock or an electrical interlock that must be moved before actuation. A relief valve and/or mechanical stops must be used to insure brake operation at safe load levels.

The brake handle should be placed so that either the pilot or copilot can actuate or release the brake while restrained in his seat harness and equipped with gloves and armor. Human factors engineering considerations of brake operation should be the responsibility of the mock-up review board.

A simple rotor brake installation for a light observation helicopter may be described as follows. The pilot lever operates, through a linkage of appropriate mechanical advantage, the piston of the master cylinder. The resulting hydraulic pressure produced in the master cylinder operates the two slave pistons arranged in caliper fashion relative to the rotor brake main disk. A stator braking disk (puck) is mounted on each of these pistons. Internal retraction springs are provided to maintain adequate clearance between the pucks and the rotating disk, so that the potential damage resulting from puck hangup on the disk is minimized. The rotor brake main disk *shall* be mounted on a separate shaft of the main transmission other than the tail rotor drive shaft or main rotor shaft. Full application of the brake is achieved with a pull of approximately 50 lb on the grip of the pilot's lever. A lock is provided to prevent inadvertent application in flight. An adjustable mechanical stop is incorporated to limit the pilot's applied load on the piston of the master cylinder. A hydraulic system pressure of 1000 psi or less generally is used to minimize development problems and to provide a high degree of reliability. A pressure relief valve dumps hydraulic fluid from the pressurized lines back into the vented oil reservoir to control the applied pressure to the pistons of the caliper-mounted slave cylinders. The installation is designed to permit the use of retrofit kits.

The described system is typical for light- to medium-sized vehicles. Larger vehicles may eliminate the mechanical lever force amplification by an independent hydraulic master-slave piston using a higher pressure onboard hydraulic system. In this case, actuation will be by means of valves and a select lever through perhaps a 3000-psi aircraft hydraulic system. Friction application, by caliper-mounted pistons to the disk, probably would be used.

7-6.4.4 Predesign Sizing Considerations

Standard mechanical engineering design principles for brakes and clutches also apply to rotor brakes. Numerous texts describe proper analysis methods (Refs. 18 and 19). Dynamic friction coefficients may be obtained from suppliers of rotor brakes and will vary depending upon material combinations. A nonscoring or galling material combination should be used between the disk and pucks to prevent chatter or excessive heat generation leading to disk warpage.

The location of the disk in the drive system must be considered carefully to provide adequate speed capability, to attain a desired rotor holding torque capability in a stopped position, and to isolate the brake from flammable fluids or materials. Also, it may be desired for the rotor brake to hold the rotor during engine starting. To attain an optimum rotor brake installation, the mass and geometric shape of the brake disk should be adequate to absorb the rotor energy without exceeding thermal limits of the material or producing excessive warpage. The loads from the rotor brake must be considered in the drive system design criteria.

7-6.4.5 Service Experiences

Disk warpage from overheating, brake grabbing, and fires has been perhaps the most common problem with rotor brakes. Proper material combinations and normal forces can prevent these service difficulties.

Inadequate clearances between pucks and disk have been common faults due to the use of standard aircraft wheel brakes. Positive clearances are required to eliminate drag of the puck on the disk. Clearances far in excess of aircraft brake systems are necessary. Any drag of the puck on the disk, as in aircraft brake systems, may produce a continuous heat source in flight, which could result in a fire.

Rotor brake technology has been developed so that future Army installations should perform safely and reliably with a minimum of weight and cost trade-off penalties.

7-7 ISOLATION SYSTEMS

In general, a helicopter having n rotor blades produces shaking forces and moments at the main rotor hub that occur periodically at multiples of n times the angular speed of the main rotor. These forces are transmitted to the fuselage where they contribute to pilot and passenger discomfort as well as adding to the fatigue load spectrum. Under certain conditions, the vibration level in the fuselage due to rotor inputs may be attenuated greatly by the use of an isolation system. In general, this involves attaching the transmission to the fuselage through elastic elements.

Isolation system needs require studies of the feasibility and effectiveness of such a system for a given aircraft. Small helicopters are characterized by relatively high rotor speeds (300-600 rpm) and have relatively short fuselages, so that the lowest elastic body mode natural frequencies are of the order of 7-12 Hz. On the other hand, large helicopters have low rotor speeds (75-150 rpm) and long fuselages, and have bending and torsional natural frequencies in the range of 2-4 Hz.

Attaching the transmission to the fuselage by means of elastic elements introduces additional degrees of freedom to the airframe mass elastic system. The forces to be isolated occur as sinusoidal functions whose frequencies are multiples of the product of the angular speed of the main rotor Ω and the number of rotor blades n. For maximum effectiveness, the natural frequencies in these degrees of freedom should be one-third to one-half of the lowest exciting frequency (n times the rotor speed) but, in order to avoid ground resonance problems, the natural frequency must be clear of the rotor speed. An additional constraint is the necessity to keep the static and g-load deflections to a minimum. It is apparent readily that the full effectiveness of an isolation system cannot be attained if any mode of the fuselage is in resonance with the primary exciting frequency, as the reduction in force transmitted to the fuselage is offset by the magnification of the resultant vibration because of the resonance conditions.

The probabilities of success in coping with all of these constraints are considerably higher for small helicopters and decrease rapidly as the size of the vehicle increases. In small helicopters, it usually is possible to establish the first bending modes of the fuselage below the $n\Omega$ exciting frequency and the second bending mode considerably above the exciting frequency. However, in the case of large helicopters—which, of necessity, have long, low-frequency airframes—the first, second, and third bending modes, as well as the torsion modes, may all occur quite close together. For instance, there may be six bending frequencies (vertical and lateral) and two torsion frequencies distributed over the range from 0 to 20 Hz. If a rotor speed of 150 rpm is assumed, once-per-revolution excitations occur at 2 5 Hz. A rotor having seven blades would produce exciting forces at 17.5 Hz. Thus, the probabilities of resonance with one of the bending or torsion modes are considerably greater in the case of the large helicopter. Under these conditions, a passive isolation system serves only the purpose of altering the natural frequencies and mode shapes of the several modes.

Manufacturers of large helicopters, in general, have adopted other means than the passive isolation system for overcoming the problem of main rotor-induced vibration. These means include active isolation systems, the use of dynamic absorbers at particular locations on the airframe, and the use of devices such as the bifilar pendulum absorber mounted at the rotor hub to absorb the exciting forces at the source. (see par. 5-2.3).

7-8 LIST OF SYMBOLS

A = exposed area of gear case, ft^2
C = coefficient of stress intensity, dimensionless
= heat rejection coefficient, ft^2-°F-min/Btu
C_c = contact angle factor for contact load, dimensionless
C_f = surface condition factor for contact load, dimensionless
C_h = hardness ratio factor for contact load, dimensionless
C_l = life factor for contact load, dimensionless
C_m = load distribution factor for contact load, dimensionless
C_o = overload factor for contact load, dimensionless
C_p = material elastic properties coefficient for contact load, $(psi)^{1/2}$
C_r = safety factor for contact load, dimensionless
C_s = size factor for contact load, dimensionless
C_t = temperature factor for contact load, dimensionless
C_v = dynamic factor for contact load, dimensionless
c_p = specific heat of working fluid, Btu/lb-°F
d = pitch diameter, in.
d_i = inner race sprag diameter, in.
= inside diameter, in.
d_o = outside diameter, in.
d_s = diameter, sprag outer cam, in.
E = modulus of elasticity, psi
E_g = modulus of elasticity for gear, psi
E_p = modulus of elasticity for pinion, psi
e = distance from mass center to shaft geometric centerline, in.
F = face width, in.
F_{br} = allowable bearing stress, psi
F_c = allowable contact stress, psi
F_e = effective face width, in.
F_{en} = endurance limit stress, psi
F_{sy} = shear yield stress, psi
F_{ty} = tensile yield stress, psi
f_a = axial tension stress, psi
f_b = vibratory bending stress, psi
f_{br} = bearing stress, psi
f_c = contact stress, psi
f_s = torsional shear stress, psi
f_t = tensile stress, psi
G = shear modulus, psi
GA = gripping angle, deg
g = acceleration due to gravity, 32.2 fps^2 = 386.4 in./sec^2
I = cross-sectional moment of inertia of area, in.^{4}
= diametral mass moment of inertia, in.-lb-sec^2
= geometry factor for contact load, dimensionless
J = geometry factor for bending load, dimensionless
= mass polar moment of inertia, lb-sec^2-in.
K = coefficient of relative stiffness, dimensionless
= stiffness, lb/in.
K_l = life factor for bending load, dimensionless
K_m = load distribution factor for bending load, dimensionless
K_o = overload factor for bending load, dimensionless
K_r = safety factor for bending load, dimensionless
K_s = size factor for bending load, dimensionless
K_t = temperature factor for bending load, dimensionless
K_v = dynamic factor for bending

load, dimensionless

$K_{1,2}$ = centers of curvature of gear profiles 1, 2

K_θ = equivalent angular spring rate due to moment restraint at the bearing, lb-in./rad

k = constant describing localization of contact, dimensionless
= spring bending constant, lb/in.

L = life, hr
= shaft length between supports, in.
= spline length, in.
= effective sprag length, in.

L_{min} = minimum length of line contact, in.

M = bending moment, lb-in.

MS = margin of safety, dimensionless

M_g = gear ratio, n_g/n_p

m = point mass, slug

m_N = gear form factor for contact load, dimensionless

N = number of sprags
= critical speed of simply supported shaft, rpm

N_c = critical speed, rpm

N_g = gear rpm

N_n = critical speed without end thrust, rpm

N_p = pinion rpm

N_0 = value of N for $K_\theta = 0$, rpm

n = number of concentrated masses, dimensionless
= number of planetary pinions, dimensionless

n_g = number of teeth in gear, dimensionless

n_p = number of teeth in pinion gear, dimensionless

n_s = number of teeth in sun gear, dimensionless

P = end thrust, lb
= power, hp
= sprag normal force, lb

P_N = normal base pitch, in.

P_a = applied tensile load, lb

P_b = bolt tension, lb

P_d = diametral pitch, in.$^{-1}$

P_e = Euler critical column load, lb

P_i = ratio of power used for Condition i to rated power

P_r = bolt tensile preload, lb

p = exponent in empirical life-load relationship
= number of mass points

Q = heat loss, Btu/min

R_i = ratio of rpm used for Condition i to rated rpm

R_s = applied shear lo[illegible]/shear strength, dimensionless

R_{sa} = allowable value of shear load/shear strength, dimensionless

R_t = applied tension load/tension strength, dimensionless

R_{ta} = allowable value of tension load/tension strength, dimensionless

R_1, R_2 = pitch circle radii of the pinion and gear respectively, in.

r = deflection of geometric centerline of shaft relative to the axis of rotation, in.
= mean radius of a tube, in.

S = distance from gear contact point M to pitch point P

s = rms surface finish, after initial run-in, micro-in.

T = kinetic energy, in.-lb
= torque, lb-in.

T_f = flash temperature index, °F

T_i = initial temperatures, °F
= ratio of time used for Condition i to total time

T_r = rated equivalent time at continuous rated power and rpm

t = wall thickness, in.

u = running mass, slug/in.

V = oil flow, gpm
= potential energy, in.-lb

W_t = transmitted tangential load at operating pitch diameter, lb

W_{te} = effective tangential load, lb

w_i = weight of the ith mass element, lb

x = length variable of a shaft, in.

y = deflection, in.

y_i = deflection of the ith mass element, in.

Z = length of action in transverse plane, in.

Z_t = scoring geometry factor, dimensionless

β = stiffness factor, dimensionless

Δ = change in annular space, in.

ΔT = oil temperature rise, °F
= temperature difference between gear case surface and ambient air, °F
δ_{st} = static deflection, in.
μ = Poisson's ratio, dimensionless
μ_g = Poisson's ratio for gear, d'less
μ_p = Poisson's ratio for pinion, d'less
ρ = weight density, lb/in.^{3}
ρ_g = gear radius of curvature, in.
ρ_i = radii of curvature of gear profiles, in.
$1/\rho_0$ = relative curvature between gear tooth profiles, in.$^{-1}$
ρ_p = pinion radius of curvature, in.
ϕ = gear pressure angle, deg
ϕ_t = transverse pressure angle, deg
Ω = rotor speed, rad/sec
ω = angular velocity, rad/sec
ω_N = natural frequency, rad/sec
ω_{N_0} = nonrotating natural frequency, rad/sec
ω_t = natural frequency of total assembly, rad/sec
ω_i = natural frequency of combination of the *i*th mass with shaft which is considered to have flexural rigidity but no mass, rad/sec
ω_0 = natural frequency of shaft without concentrated masses, rad/sec

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CHAPTER 8

POWER PLANT SELECTION AND INSTALLATION

8-1 INTRODUCTION

Good propulsion system installation design will result in the optimum propulsion system/airframe integration required to achieve all aspects of performance, maintainability, reliability, and durability.

During preliminary design, the configuration of the helicopter will be established. Structure will be determined, engines located, systems defined, and equipment sized. To realize the goals of a good propulsion installation, it is important to establish and adhere to certain basic ground rules and considerations.

The helicopter propulsion system includes the engine, drive system, rotor/propellers, and all related subsystems and equipment. However, the drive system and rotor/propellers will not be included in this discussion (see Chapters 3, 5, and 7 for discussion of these areas). The engine subsystems may include the following: engine installation, air induction system, starting system, lubrication system, power extraction system, power management system, cooling system, exhaust system, fuel system, fire protection system, and auxiliary power units. Ref. 1 provides a good description of the engine and its subsystems.

8-1.1 SELECTION

The selection of the basic engine will be governed by the mission and vehicle performance requirements as defined by the procuring activity. As mentioned in par. 3-3, the selection will be limited to turboshaft engines. The engine selected normally will be one of the candidate engines specified by the procuring activity for a particular aircraft. Refer to par. 3-3.2 for methods of basic engine selection.

A satisfactory engine installation results from careful attention to all design features that affect the operational capability of the aircraft. Particular attention *shall* be given to the specification requirements of the engine manufacturer. These requirements are prescribed by the engine manufacturer to insure that the engine experiences no extreme detrimental effects under the normal usage to which the engine and aircraft will be subjected within the operational profile of the aircraft. Details are given in the applicable engine manufacturer's specifications and installation drawings.

The factors to be considered in the preliminary design stage for establishing the propulsion system configuration are noted in the paragraphs that follow. The special requirements presented in this and subsequent paragraphs represent the criteria by which judgment of the engine helicopter configuration will be made and that should be considered by the engine manufacturer. Recent trends indicate that many of the propulsion subsystems long considered the responsibility of the airframe designer have become integral with engine design and hence the responsibility of the engine manufacturer. This divided responsibility for the design of the various components of installation of the propulsion system makes it imperative that close coordination between the airframe designer and the engine manufacturer be maintained to insure engine/airframe compatibility.

8-1.2 ENGINE POWER OUTPUT SHAFT

The location of the engine output shaft normally will position the engine(s) relative to the rotor drive transmission. Engines with the output shaft at the front end usually will be located aft of the main rotor centerline for the most direct drive. Similarly, with an aft output shaft the engine usually will be located forward of the main rotor centerline. Consideration must be given to helicopter longitudinal and vertical CG requirements. The engines should be located as close as practicable to the rotor shaft to keep the drive shaft short and light.

8-1.3 ENGINE REPLACEMENT

The operational availability of helicopters depends upon factors such as the ease and speed with which engines can be removed and replaced. Quick engine change (QEC) capability is not difficult to achieve if it is an objective in the preliminary design phase of the unit installation. The QFC unit normally includes the complete engine with all equipment—such as mounts, accessories, inlet and exhaust system adapters—attached. Engine-mounted parts of the fuel, lubrication, and power control systems also should be included in the QEC assembly. For multiengine helicopters, interchangeability should be assured by designing installations that are detachable as units to fit any engine position on the helicopter with minimum variations.

Installation and removal procedures, accessibility, and maintenance are prime considerations in the preliminary design phase. The kinematics of engine installation/removal must be defined and evaluated thoroughly. The method of installing and removing the engine will have a profound influence on the surrounding airframe structure.

8-1.4 ENGINE AIR INDUCTION SYSTEM

An early design goal is the development of an air induction system configuration that will provide the maximum possible pressure recovery and uniformity of velocity and pressure distribution at the entrance to the engine inlet. The location of the air induction system must not cause appreciable changes in pressure distribution at the engine inlet with changes in aircraft angle of attack.

For multiengine, side-by-side installations with inlets adjacent to each other, a design objective is to negate any interaction of the inlets relative to engine airflow under any flight condition. The engine air induction system will include an engine air particle separator (EAPS) system compatible with the applicable engine.

8-1.5 ENGINE STARTING SYSTEM

The engine starting system may be electrical, hydraulic, or pneumatic. Selection of the optimum system will be contingent upon the number and size of the engines, the size and mission of the aircraft, and system secondary power trade-offs described in Chapter 9. The procuring activity will establish requirements for self-containment and for emergency starting.

8-1.6 LUBRICATION, COOLING, AND EXHAUST SYSTEMS

The importance of lubrication in achieving satisfactory engine operation is well recognized. The design of the engine oil system should be the responsibility of the engine manufacturer. The system should be complete in itself and should feature, for survivability, an oil cooler with a bypass system to protect against loss of oil because of a hit in the cooler. The oil tank and critical oil lines *shall* be self-sealing and crash-resistant. One consideration for lubrication is its compatibility for system servicing; however, because the responsibilities of the airframe designer and the engine manufacturer are defined by the engine selected for use in the aircraft, it becomes vital for the helicopter system designer to insure that the oil system is compatible with the requirements set by the engine manufacturer.

The engine cooling system must prevent excessive temperature of the engine, accessories, equipment, components, and structure in proximity to the engine installation. Although the proof of the cooling system is provided through testing, special consideration of the cooling requirements of the engine and adjacent structure is necessary during the preliminary design stage.

The engine exhaust system conducts the engine combustion gases from the point of discharge from the engine to the point of discharge from the aircraft. To minimize power losses, the length of the exhaust system should be minimized and the tailpipe exit area sized to comply with the engine specification. If possible, the exhaust system should be an integral part of the QEC unit.

8-1.7 FUEL SYSTEM

Design objectives for a fuel system should include:

1. Meeting the detail design requirement for the aircraft as specified by the procuring activity
2. Conforming to MIL-F-38363.

Fuel system arrangements will be based on supplying fuel to the engine(s) and APU(s). The system *shall* be compatible with fuels conforming to MIL-T-5624, grades JP-4 and JP-5.

The design should insure that all fuel will be available to any engine without interruption of flow. Each detail of the fuel system should insure that the system has the least possible vulnerability with respect to aircraft fires and/or ballistic damage during flight or ground operation. Therefore, self-sealing, inerting, or crash-resistant fuel tanks are required.

8-1.8 SECONDARY POWER SYSTEM

Requirements for secondary power systems, including provisions for onboard pneumatic, electrical, and hydraulic power generation, control, and distribution, should be established in preliminary design. General functional requirements for such a system are shown in Fig. 8-1.

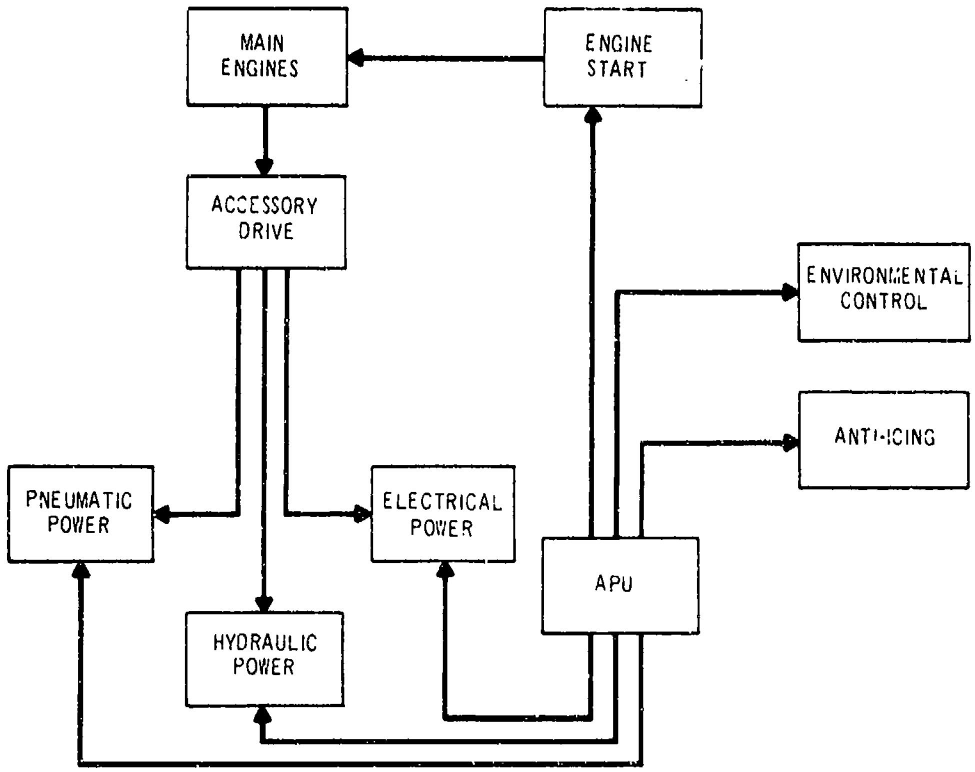


Fig. 8-1. Secondary Power System Functional Requirements

8-1.9 POWER MANAGEMENT SYSTEM

Helicopter rotor speed governing (including overspeed protection), load signal compensation, and power sharing for multiengine installation are firm requirements for a power management system. Development of a fuel control to meter fuel and establish engine power to meet helicopter rotor demands *shall* be accomplished by the engine manufacturer but will demand close coordination with the airframe designer for compatibility with helicopter requirements.

8-1.10 AUXILIARY POWER UNIT (APU)

An auxiliary power unit (APU) may be required for use either on the ground or in flight. Inasmuch as an APU is an engine, its installation involves all the same considerations as does a propulsion engine installation.

8-1.11 SAFETY REQUIREMENTS

The principal safety consideration in a propulsion system installation is fire protection. In addition, the installation must be crashworthy structurally, and the components must be arranged so that danger to personnel performing necessary service on the aircraft is kept to a minimum.

The degree of fire protection inherent in the aircraft will depend upon the extent to which the following are provided for or incorporated in the design:

1. Prevention of fire
2. Resistance to the spread of fire
3. Reliable fire and overtemperature detection systems
4. Positive fire extinguishment.

General rules for fire prevention are aimed at protecting all combustibles from sources of ignition. Fire containment features include shutoff valves, firewalls, and materials that prevent reignition. Sensors capable of detecting fire or dangerous overtemperatures must be provided. The fire extinguishing system should not use toxic or corrosive extinguishing agents.

8-2 ENGINE INSTALLATION CONSIDERATIONS

8-2.1 ENGINE MOUNTING AND VIBRATION ISOLATION

8-2.1.1 Engine Mounting Considerations

The requirements of the helicopter and of the specific turboshaft engine will determine whether the engine is to be mounted rigidly or to employ vibration isolation.

The engine mounting system should be designed to insure the stability of the engine in its mounts and to maintain proper shaft alignment for all flight conditions.

The mounting system for the engine should perform the following primary functions satisfactorily:

1. Safely carry the weight, thrust, torque, and inertial and gyroscopic loads encountered during any condition of operation. In addition, the system should be designed to keep the engine(s) attached to the basic structural member supporting the mounts throughout a survivable crash, even though considerable distortion of the engine and/or support structure may occur (Ref. 2).

2. Maintain engine location and alignment under all linear accelerations and velocities, and torque loadings.

3. Accommodate radial and axial thermal expansion of the engine.

4. Prevent airframe structural deflection from imposing loads upon the engine.

5. Permit ease of installation and removal of the engine from the mount structure, and provide for accessibility to the engine for routine inspection and maintenance.

6. Isolate and/or absorb resonant structural vibrations induced by the engine, or by an engine/propeller combination, so they will not be transferred to the airframe structure; and prevent vibrations induced by the airframe main rotor combination at critical engine frequencies from being transferred to the engine.

Mounting system arrangements vary and are contingent upon the provisions on the engine for its support as well as upon the helicopter requirements. Engine restraining points will be specified by the engine manufacturer, as will the maximum allowable loads and moments at each restraining point for limit maneuver conditions. When the engine is mounted on rigid mounts, it must be restrained in a statically determinate manner. Functional and design requirements for the mounting system usually are satisfied by a three-point mounting system consisting of two main mounting points and a third point of support (the steady rest) to allow for axial expansion of the engine.

Airframe mounting system components such as trunnions, spherical bearings, vibration isolator mountings, and quick attach-detach (QAD) features should provide engine support as indicated by the engine specifications and drawings. These components should facilitate quick engine change, and should withstand safely the anticipated reactions at the engine mounting points. The reactions can be calculated for any specific installation on the basis of engine weight, center of gravity (CG), engine output torque, survivable crash loads, engine moments of inertia (including polar), and anticipated helicopter maneuver conditions.

8-2.1.2 Engine Mounting Forces and Reactions

The reactions on the engine mounts resulting from any flight maneuver will be derived and tabulated for the selected mounting configuration during the development of the basic loads. The critical loading condition(s) will be determined and the structural adequacy of the mounting system demonstrated by preliminary structural analysis.

8-2.1.3 Engine Vibration Isolation

An isolator may be described as a load supporting resilient element having controlled elasticity and damping, and adaptable to integration into a mechanical or a structural assembly. Isolators are used to control engine dynamic forces and motion. Isolators also may be used to reduce induced structural excitations to the engine.

For turboprop applications, a shock and vibration isolation mounting system is a firm requirement. In the case of conventional helicopters, however, a survey should be conducted to ascertain if vibrational frequencies induced by the airframe drive system coincide with engine natural frequencies. If so, consideration should be given to using a vibration isolation mounting arrangement. Critical engine frequencies usually are specified by the engine manufacturer.

The vibration isolation mounting system should be designed to insure the stability of the engine so as to help maintain proper shaft alignment for all maneuvers and flight conditions. The vibration isolating material may be an elastomeric product such as silicone. Silicone materials are preferred because of their ability to withstand temperatures up to 500°F, a temperature level that should not be exceeded for engine front or midframe mounting applications. Metallic types of isolators (such as springs) also can be considered.

To permit compliance with the engine mounting and vibration requirements of MIL-E-8593 and the helicopter vibration requirements of par. 5-2, the engine mounting system *shall* have a natural frequency substantially below any forcing frequency of the helicopter main rotor. The effects of the stiffness of both the engine mount and the airframe support structure *shall* be included in the determination of this natural frequency. The critical forcing frequency will be the main rotor rotational speed (one-per-rev).

Design considerations should include establishing the mount spring rate in such a manner as to minimize induced loads from airframe deflections.

Satisfactory vibration isolation will be insured in most applications if the natural frequency of each mode of the mounting system is maintained substantially below the forcing frequency. The damping characteristics of a single-degree-of-freedom, spring-mass-damper system, shown schematically in Fig. 8-2, are indicated in Fig. 8-3.

8-2.1.4 Engine Mount Installation Considerations

The location of engine mounting points should be selected for ease of engine servicing and maintenance. Engine installation and removal kinematics will be established in preliminary design, and the overall configuration of the mounting system then can be defined. The motion(s) required to remove the engine should be simple and preferably unidirectional. If removal is horizontal, consideration should be given to the use of installation tracks integral with the airframe.

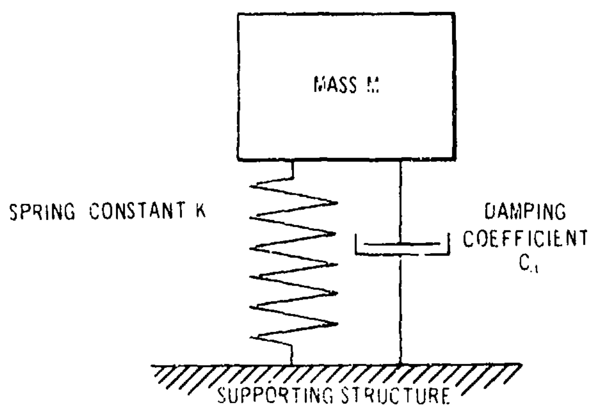


Fig. 8-2. Schematic Diagram of Vibration Isolator

The interface between the engine and airframe for the main connectors should feature an engine-mounted uniball and a structurally mounted mating clamp. This arrangement constitutes a spherical bearing capable of reacting helicopter maneuvering loads as well as allowing for quick engine connection and shaft alignment.

The engine mounting system must react the maximum flight maneuver loads without permanent deformation and must be capable of safely retaining the engine under survivable crash load conditions (defined in par. 4-5.3 based on Ref. 2) or sudden engine seizure. The engine manufacturer should approximate the mount reaction resulting from the torque generated by a sudden engine seizure. In the preliminary design, the spherical bearings should be checked for size relative to their ability to retain the engine under crash or engine seizure conditions.

The steady rest mount should react vertical loads only to permit free engine axial thermal growth.

Vibration mounting system definition must be accomplished during preliminary design because of its effect on the overall envelope of the vehicle. The performance penalties associated with an increase of the fuselage cross-section due to growth of the engine nacelles in order to house a redesigned system may not be acceptable.

Use of the vibration isolator system introduces additional installation problems. Consideration must be given to flexibility between the engine and the structurally supported portions of the air induction and exhaust systems, because the isolating material permits movement of the engine with respect to the aircraft.

Vibration mountings must retain the engines should the isolating material fail. Care must be exercised to insure that the structure to which the isolator is attached has a greater stiffness than that of the isolator. Failure to do so could result in fatigue failure of the brackets. Under transient conditions, deflections may be greater than those due to normal vibration. Under these conditions, the isolating material should be protected for a predetermined maximum deflection by a solid metal-to-metal contact.

8-2.2 ENGINE AIR INDUCTION SYSTEM

8-2.2.1 General Design

The engine air induction system must furnish the required quantity of air to the engine inlet at the highest possible energy level and with minimum flow distortions. A uniform and steady airflow is necessary to avoid compressor stall, which can lead to excessive internal engine temperatures. Energy losses at the engine inlet will result in successively magnified losses through the components (stages) of the engine. Gas turbine engines are more critical in this respect than are reciprocating engines.

The air induction system must recover as much of the total pressure of the free stream as possible. This is known as "total pressure recovery". Ref. 3 is a definitive source for induction system aerodynamic design. The basic equations, as well as pertinent factors for use in these equations, are provided by this reference.

8-2.2.2 Air Induction System Inlet Location

The pressure distribution around the fuselage during flight must be considered during the selection of the engine air induction system inlet location. Inlets placed in locations of high velocities, decelerating flows, or in the wake of aircraft components such as a main rotor shaft must be given special consideration. Changes in aircraft attitude as a result of flight maneuvering should not affect the pressure distribution at the engine inlet noticeably. Also, there should not be any effect upon a given engine inlet as a result of flow interaction with other inlets. Inlet scoops or submerged inlets should avoid areas of boundary layer buildup. The air induction system inlet should be located as high as possible to reduce the intake of leaves, branches, sand and dust, or other foreign objects. A high inlet location also will reduce the potential hazard of ingesting fuel vapors during "hot refueling" operations.

8-2.2.3 Engine Air Induction System Pressure Losses

In addition to airframe induced losses ahead of the air induction system inlet, there are losses caused by duct wall friction, contraction, diffusion, bends, and turbulence arising from duct distortions or obstructions in the duct. A source of an especially large pressure loss is the "air filter" or EAPS system (see par. 8-2.2.6).

Friction and contraction losses usually are very small, provided reasonable care is exercised in providing smooth internal surfaces by avoiding protruding

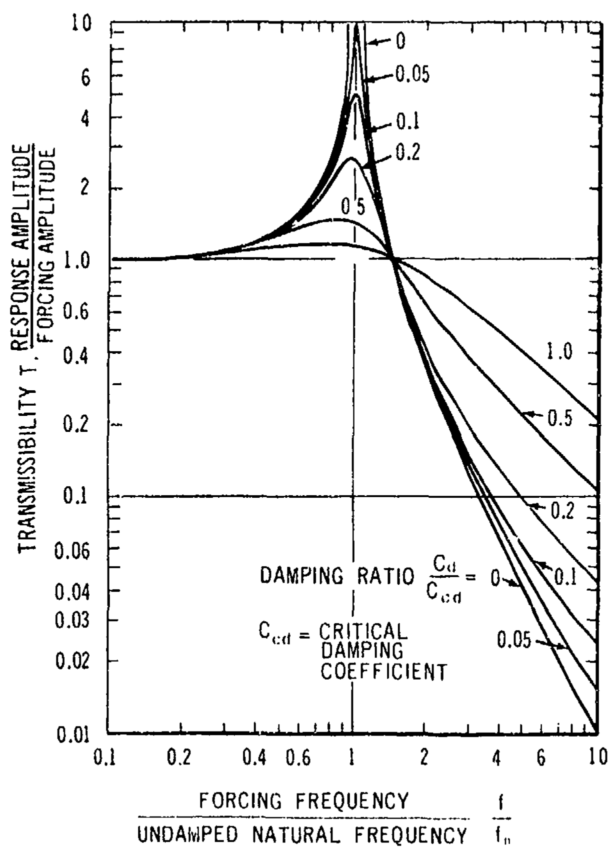


Fig. 8-3. Damping Characteristics of One-degree-of-freedom System

structural reinforcements, bolts, rivets, or generally poor sheet metal work. Internal duct losses increase as the square of the airspeed through the duct; thus, to avoid large losses, it is important that these internal airspeeds be kept low. When sharp bends are required or when the engine inlet face is close to the exit of an air particle separator, a plenum chamber with an airspeed of 60 fps or less is recommended. The engine in this case should be equipped with a bell mouth inlet.

8-2.2.4 Evaluation of Pressure Losses

The total pressure loss ΔP_f in a duct due to wall friction is directly proportional to the friction factor (the so-called Fanning friction factor; the Darcy-Weisbach friction factor is four times the value of the Fanning friction factor), the duct length, and the dynamic pressure; and inversely proportional to the hydraulic diameter (the hydraulic diameter is four times the hydraulic radius).

$$\Delta P_f = \frac{4fLq}{D_h} \quad , \text{lb/ft}^2 \qquad (8\text{-}1)$$

where

D_h = hydraulic diameter, ft
f = friction factor, dimensionless
L = duct length, ft
q = $\rho V^2/2$, dynamic pressure, lb/ft^2
V = air velocity, fps
ρ = air density, slug/ft^3

The friction factor f is a function of the Reynolds number and the roughness of the surface. The value of friction factor ordinarily varies between 0.003 and 0.-007 (Ref. 3). The hydraulic diameter for the noncircular ducts is calculated as

$$D_h = \frac{4\ (\text{duct cross-sectional area})}{\text{wetted perimeter}} \quad , \text{ft} \qquad (8\text{-}2)$$

Similarly, a loss factor k is used to evaluate the pressure losses of bends, turns, diffusers, accelerations, or obstructions as a function of the dynamic pressure (or velocity head) immediately ahead of the turn or obstruction. This pressure loss ΔP_k, based on a constant factor k, is given by

$$\Delta P_k = kq \quad , \text{lb/ft}^2 \qquad (8\text{-}3)$$

where

k = loss factor, dimensionless

The total ΔP_t is as follows:

$$\Delta P_t = \sum kq + \sum \frac{4fLq}{D_h} \quad , \text{lb/ft}^2 \qquad (8\text{-}4)$$

The documentation of the duct losses *shall* include:

1. A plan view and side view of the air induction system with pertinent dimensions

2. The friction factors and loss factors presented in Ref. 3 that are used in the calculations. Factors not covered *shall* be listed and their source and accuracy justified.

3. A table of pressure loss ΔP_t and dynamic pressure q versus the duct station for maximum engine power, at standard sea level static conditions.

In order to simplify the computations, it may be assumed that the duct pressure losses referred to sea level standard density ρ_o remain constant throughout the operating envelope; therefore, calculations for maximum engine power at standard sea level conditions will be satisfactory for all other conditions.

8-2.2.5 Foreign Object Damage (FOD)

The responsibility for providing effective engine protection is shared by the engine and airframe manufacturers. The engine designer is responsible for making the engine as tolerant as possible to sand, dust, rain, foliage, snow, hail, etc. The airframe manufacturer must remove the foreign objects that the engine cannot tolerate, by the installation of appropriate devices or in cooperation with the engine manufacturer if the protective device is an engine-mounted accessory.

Large foreign objects such as bolts, rocks, hail, and rags can put most engines out of operation. Small particles, in the order of 1000 micron diameters, produce nicks and dents in compressor blades, whereas particles of less than 1000 microns can cause damage over extended periods through erosion of the airfoils and casings or by restricting flow passages through accumulation.

Accumulation on the air filter, or screen, of objects such as paper, grass, or plastic sheets can occur and designers should provide for ease of inspection, removal, and maintenance. A substantial blockage of the filter may cause engine stall, and an air bypass may be required.

8-2.2.6 Engine Air Particle Separator (EAPS)

All helicopters *shall* be equipped with an engine air particle separator (EAPS) which may be provided as part of the airframe or integral with the engine. The design goals are high dirt removal capacity, low pressure drop, low weight, and low cost. The designer must trade off these objectives for each installation with the tolerance of the engine and the mission spectrum in mind.

The design criteria outlined in Ref. 4 should be used as a guide. The total pressure at the engine inlet face should be at least 99% of ambient static pressure for all flight operations during which an EAPS is operative.

An EAPS bypass *shall* be provided to allow the inlet air direct access to the engine unless it can be substantiated that the helicopter can be operated safely without the bypass. The engine inlet total pressure during bypass operation should be at least 98.5% of ambient static pressure.

The methods available to the designer to achieve the separation of small particles such as sand and dust from the main engine air stream are:

1. Capture in a porous media
2. Electrostatic separation
3. Inertial separation.

These three methods have been studied and most variations of them tested extensively ever since the helicopter began to receive major combat usage. Of the three, only the inertial separator has proven successful in the severe combat environment. For more complete information concerning the methods of protection, a good source of information is Ref. 5.

The barrier filter (porous media) generally has a poor separation performance compared to that of the inertial separators for the same pressure drop due to the slow ;ration of the fine particles through the filter after ial entrapment. Also, the filters must be replaced uently, especially after they become wet.

he electrostatic filter, barring a major technical kthrough, is not successful because of its excessive ht.

8-2.2.6.1 Inertial Separators

Inertial separators make use of the fact that air can alter its direction rapidly, whereas the high momentum of entrained particles forces them into a trajectory away from the main stream of the air where they can be either captured or directed overboard. An example of this type of separator based upon duct configuration is shown in the schematic of Fig. 8-4.

The separation efficiency of this type of inertial separator is 60-85%, measured with AC coarse dust. There is a relative change in efficiency with a change in particle size. An EAPS with an 85% efficiency with AC coarse dust may have a 75% efficiency with MIL-E-5007 sand.

The separation efficiency increases with increasing scavenge bypass air to a certain maximum beyond which there is no additional benefit. Generally, large scavenge bypass flows (about 20-30%) are required to achieve efficiencies above 70% at acceptable pressure losses (less than 1%). The design may require the use of a scavenger pump.

The advantages and disadvantage of inertial separators are:

1. Advantages:
   a. Good for leaves, grass, water, sand and dust, etc.
   b. Low installation weight
   c. Small frontal area
   d. Usually no need for a bypass
2. Disadvantage: Large power loss for good separation efficiencies (above 85%).

8-2.2.6.2 Vortex Tube Inertial Separators

A vortex tube inertial separator such as shown in Fig. 8-5 consists of a cylindrical tube with a helical inlet swirl vane that gives the air and the entrained particles a high rotational speed. The dirt particles are centrifuged to the wall of the tube. The concentrated dirt passes along the wall in a helical movement to the end of the tube, where it enters an enclosed scavenge space together with a small percentage of the air flow. The clean air exits through a diffuser.

These individual tubes are clustered in groups and panels suitable for aircraft installation. The pressure in the scavenge airspace between the tubes is lower than ambient pressure. The scavenge air and dirt must be pumped overboard by a fan or by an engine bleed air ejector. The overall panel separation efficiency generally is lower than that of the individual tube due to interactions between tubes, unequal scavenge suction, and unequal flow distribution at the EAPS inlet.

The detailed arrangement must be worked out with a filter manufacturer; however, for preliminary design purposes, a trade-off among the following factors is required:

1. Separator efficiency
2. Percent scavenge flow and pressure drop
3. Separator frontal area
4. Scavenge method.

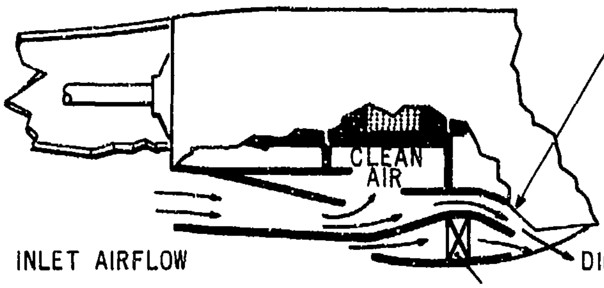


Fig. 8-4. Schematic View of an Engine With Inertial Separator

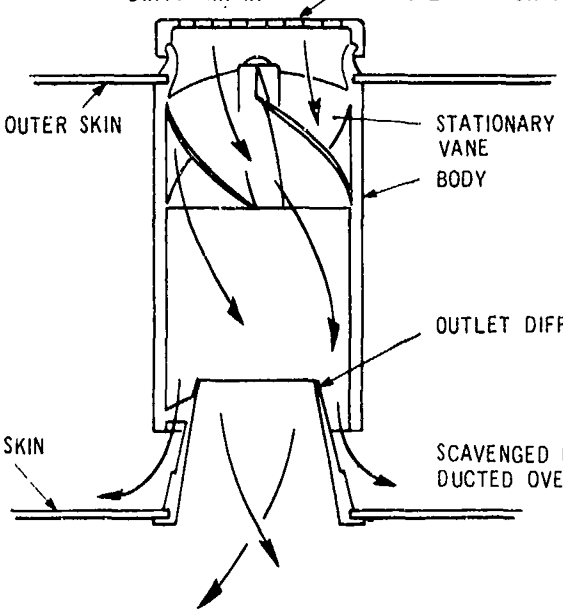


Fig. 8-5. Typical Vortex Tube Inertial Separator

The efficiency of a fixed EAPS design using vortex tubes typically varies between 60% and 85%, depending upon the scavenge flow. The efficiency will rise from the minimum value with no scavenge flow to the maximum value at 5-10% scavenge flow, depending upon the tube diameter. Generally, smaller tubes reach maximum efficiency at a lower value of scavenge flow.

The scavenge chamber pressure drop at maximum separation efficiency will vary from 0.7 to 1.2 times that of the overall panel pressure drop. The overall clean air pressure drop for a fixed airflow is a function of the panel frontal area as shown in Fig. 8-6.

The airflow through each vortex tube is fixed for a given pressure drop. For scavenging reasons a more or less constant number of tubes can be concentrated in a given area: the pressure drop per EAPS frontal area follows the same law as that of the individual tube. The EAPS pressure loss for a fixed flow area, therefore, is proportional to the volume flow rate squared. For a constant airflow, however, the pressure drop is related inversely to the number of tubes squared.

The advantages and disadvantages of the vortex tube inertial separator are:

1. Advantages:
   a. High separation efficiency for small particle sizes
   b. Low pressure losses.
2. Disadvantages:
   a. Large area requirement (only 1-2 lb/sec of airflow per square foot of inlet area)
   b. Poor for leaves, grass, or other foreign objects
   c. Problems with ice and snow.

8-2.2.6.3 Centrifugal (Inertial) Separator

The centrifuge uses the same principle as the individual vortex tube described previously, except that it spins the air as a whole. The types are distinguished by the means with which the rotational speed is imposed upon the air:

1. Nonpowered centrifuge
2. Powered centrifuge.

The nonpowered centrifuge, such as shown in Fig. 8-7, has characteristics similar to the vortex tube inertial separator, and requires a scavenge pump.

The separation efficiency of the centrifuge is equivalent to that of the vortex tube inertial separator for particle sizes above 40 microns; however, the efficiency drops sharply for particle sizes of 30 microns. The poor separation efficiency of small particle sizes is related to the fact that the effect of the centrifugal force on the smaller particles is insufficient to overcome the aerodynamic force and thereby to divert the particle from the primary airstream.

The powered centrifuge, shown in Fig. 8-8, uses the same principle as the nonpowered centrifuge except that a powered axial stage is used to spin the air. This method is advantageous because concentrated dirt and air are scavenged without an additional pump. Generally, the power input is substantial and, therefore, the system can compete with the nonpowered centrifuge only if a high percentage of the rotational energy can be recovered at the engine inlet. Close cooperation with the engine manufacturer is essential for the design of a successful powered centrifuge. The advantages and disadvantages of the centrifugal separator are:

1. Advantages:
   a. High airflow per square foot of inlet area (10 lb/sec or higher)
   b. Suitable for large engines
   c. Low weight
   d. Usually no need for a bypass.
2. Disadvantages:
   a. Poor separation efficiency for small particles
   b. Only fair for separation of leaves, grass, and similar foreign objects

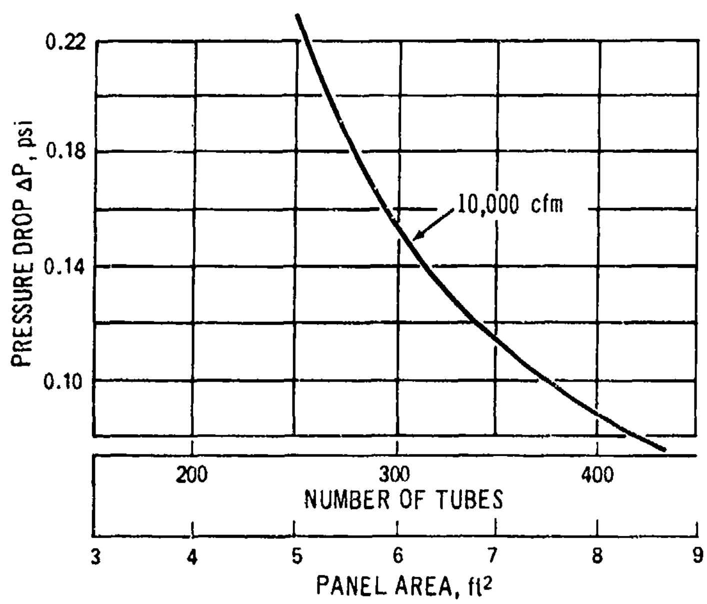


Fig. 8-6. Pressure Drop vs Panel Area

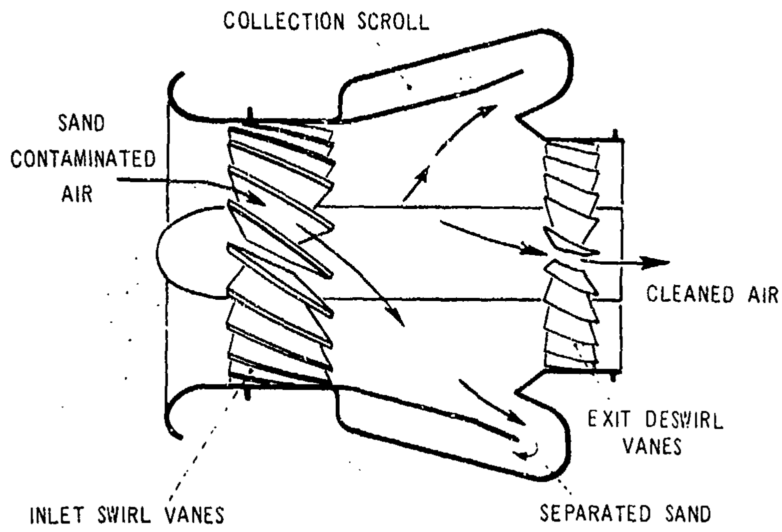


Fig. 8-7. Typical Centrifugal (Inertial) Separator

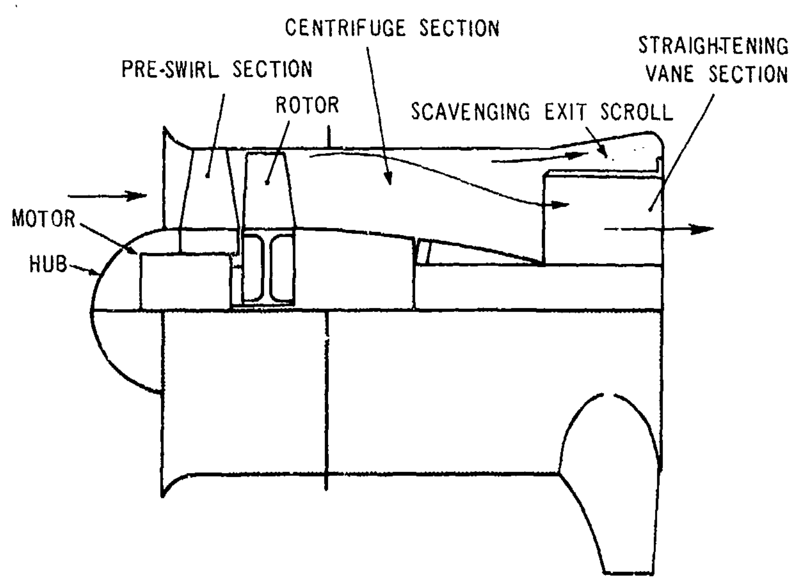


Fig. 8-8. Typical Powered Centrifugal (Inertial) Separator

c. Anti-ice protection required.

8-2.2.6.4 EAPS Scavenge Pump

When required by the EAPS, the scavenge pump is a strong contributor to the weight and performance penalties of a given installation. The following scavenge pumping methods generally are considered:

1. Engine bleed air ejector
2. Mechanically driven fan
3. Electric fan
4. Engine exhaust ejector.

The engine bleed air ejector installation weight is low, and the system operationally is very reliable. However, the engine performance penalty is high. Each percentage point of engine air bleed reduces the available engine output power by 2-4%, depending upon the engine design. Considering that each engine horsepower can lift approximately 10 lb, the payload penalty easily can exceed the weight of other scavenge methods.

The mechanically driven fan requires little engine power, and usually the installation weight can be kept low in a new installation. Experience has shown that sealed bearings and coated fan blades can stand up well in the erosive environment.

Electric fans are heavy and can have adverse effects on the helicopter compass and radio gear.

The engine exhaust ejector uses the residual energy of the gas turbine exhaust. The installation can be lightweight.

8-2.2.7 Power Losses

Typical sources of engine power losses associated with the air induction system are:

1. EAPS pressure loss
2. EAPS scavenge power requirement
3. Ram recovery loss (excluding EAPS)
4. Power for anti-icing requirement
5. Temperature rise.

The effects of the inlet pressure loss on power loss are slightly different from engine to engine, and will be described by the engine manufacturer. For each percentage point of inlet pressure loss, gas turbine engines typically lose 1.5-2.0% of rated power at intermediate power and 1.8-2.2% at 50% intermediate power.

The power losses associated with the scavenge and anti-icing power requirements vary substantially with the installation; however, they should not exceed 1% of the design rated power.

For complex engine air induction system configurations, static model tests may be required to establish the internal pressure losses. Wind tunnel tests of helicopter models usually include simulation of the internal airflow to establish the correct external drag and induction system ram recovery factor. In addition, when the induction system is fully modeled, the pressure distortion at the engine inlet face can be measured (see par. 8-5.4, AMCP 706-203). The engine inlet total pressure variation must not exceed the value specified by the engine manufacturer. Acceptable distortion limits are in the order of 5%. Distortion is calculated as

$$\text{Distortion} = \frac{P_{t1\,max} - \overline{P}_{t1}}{\overline{P}_{t1}} \quad (8\text{-}5)$$

where

$P_{t1\,max}$ = maximum total pressure (absolute) at the engine inlet face, psi

$\overline{P}_{t1}$ = mean total pressure (absolute) at the engine inlet face, psi

Circumferential pressure variation provides harmonic excitation sources for vibration of compressor blades. These blade vibrations can lead to blade fatigue failures.

8-2.2.8 Air Induction System Anti-icing Provisions

Normally, air induction system icing is not the major factor in determining the duration of flight in icing conditions. Rather, the helicopter main rotor blades and tail rotor blades are usually the critical factors. However, this does not preclude the need for an ice protection system for the air induction system. Generally anti-icing is required, as deicing is not acceptable if the engine can be damaged by ingestion of ice chunks.

No particular critical flight condition can be defined, as icing conditions may be worse for forward flight or for hover depending upon the situation. Snow or sleet creates an additional problem. The EAPS system, particularly, is vulnerable to icing conditions, and lack of anti-icing for various types of EAPS systems will limit the cold-weather capability of the helicopter.

Supercooled water droplets may exist in clouds at ambient temperatures far below the freezing point. When those droplets are disturbed by the helicopter, they will impinge upon the air induction system surfaces and freeze. In addition to the normal icing problems, icing may occur at atmospheric temperatures above freezing due to decreases in the static temperature at some stage in the air induction system.

The methods available to prevent ice buildup in critical areas are:

1. Deicing system
2. Alcohol spray
3. Thermal energy.

A deicing system can be used only with engines that can absorb the chunks of ice that will break loose from the surfaces. The alcohol spray method is the most likely to be used as an emergency device.

For all-weather aircraft, thermal energy gives the most effective ice protection. One method uses hot air from the compressor bleed, which is passed through passages integral with the surface being heated. An alternative is the use of electric resistance heating elements imbedded below the surface. As prediction of ice protection requirements for the air induction system is extremely difficult, one must rely mostly on past experience and testing of any new design to verify assumptions.

8-2.3 EXHAUST SUBSYSTEM

The kinetic energy of the exhaust gases leaving a gas turbine engine represents a loss of shaft power. However, the exhaust thrust may be used to overcome some of the aircraft drag during forward flight. The distribution of power between the shaft and the exhaust is governed directly by the exhaust duct area. The best combination of shaft power and exhaust thrust should be determined separately for each individual application.

For most low-speed helicopters it is more efficient to convert as much of the exhaust energy as possible into shaft power and to let the helicopter rotor provide the forward thrust. The exhaust duct should be built for maximum diffusion, which will result in maximum shaft power output.

Exhaust duct design must follow good practice for internal flow aerodynamics. Poor design will result in flow separations at bends and diffusing sections, causing unnecessary pressure losses and reduced effective flow area. The straight conical diffuser is the most effective exhaust duct design for engines with axial outlets. Good exhaust performance also can be achieved for engines with radial outlets by the use of elliptical sections through the bends. The elliptical duct is more expensive to fabricate but the turning losses can be as much as 50% lower than those of a round duct.

To the maximum extent possible, the exhaust system should be included in the QEC assembly. When it is necessary to include part of the exhaust system as a part of the airframe, a QAD coupling should be used that has sufficient flexibility to provide for thermal growth of the engine and the exhaust duct and for angular deflections of the tailpipe at least equal to the maximum anticipated deflection of the adjacent structure. These deflections may result from flight or ground loads, from service and maintenance operations, or from other normal operating conditions.

8-2.3.1 Exhaust Wake

The exhaust duct must be directed away from the fuselage and tail boom to avoid the impingement of hot gases upon the helicopter surfaces and to avoid endangering personnel performing necessary service on the aircraft on the ground. If the gas conditions at the duct exit are known, typical exhaust gas temperatures and velocities may be presented as nondimensional parameters such as those in Fig. 8-9. The engine stations are defined on Fig. 3-53.

8-2.3.2 Engine Exhaust Noise

A small noise reduction can be achieved on the ground when the exhaust duct is directed upward. The exhaust gas velocity is low and can be ignored as a noise source. However, the burner noise that is transmitted downstream by the gas is a strong source of external noise.

Engine exhaust noise can be reduced by lining the exhaust ducts with sound absorptive materials. Absorptive materials are heavy but have good absorptive characteristics over a wide sound frequency spectrum. Resonance mufflers are used only to eliminate specific low frequencies that normally cannot be eliminated by the absorptive muffler.

8-2.3.3 Performance Losses

The performance specifications of an engine are based upon the engine manufacturer-designed exhaust duct. This duct generally is of practical size and can be adapted readily by the helicopter designer. Deviations from this exhaust duct configuration must be documented. A breakdown by section of friction, turning, and diffusion losses should be made based upon the calculation procedure of Ref. 3. It must be substantiated that alternative calculation procedures achieve the same accuracy as that given in Ref. 3.

The engine performance loss calculation due to exhaust system pressure loss follows the same pattern as that of the engine induction system. The effect of exhaust pressure loss upon engine performance varies from engine to engine and will be stipulated by the engine manufacturer. As a guide, it can be assumed that for each percentage point of exhaust pressure loss, 0.5-1.0% of rated power is lost at maximum continuous power, depending upon the engine size, and 0.8-1.2% will be lost at 50% maximum continuous power.

Additional power losses may be associated with an infrared (IR) suppressor. For example, the pumping power for secondary cooling air has to be stated if a cooling fan is driven by the power turbine. In addition, the installation of a cool tailpipe IR suppressor will apply a back pressure to the turbine and create an additional power loss.

8-2.3.4 Infrared Radiation Suppression

One of the threats faced by Army helicopters is attack by IR sensing, or "heat-seeking", missiles. For helicopters whose assigned missions will expose them to this type of threat, an IR suppression system may be required—as a part of the basic helicopter configuration or as a kit installation.

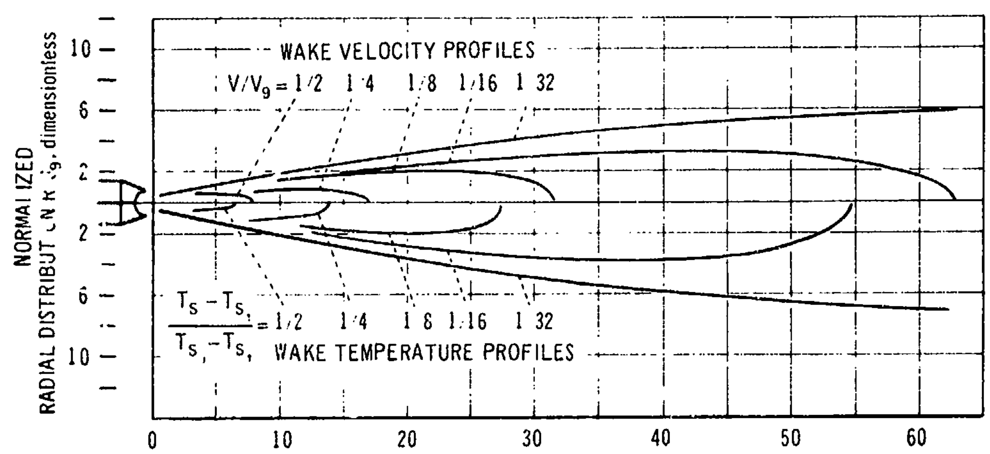
Suppression of IR radiation involves reducing the effectiveness of the source of the radiation. Any heat source is a radiator of IR energy. The exhaust wake, exhaust system, and combustor and turbine stages of the engine are the areas of highest temperature and hence are the primary sources of IR radiation. However, the "cold" engine section, the rotor gearboxes and their lubrication system components, and the hot air being exhausted from heat exchangers and other cooling system outlets cannot be disregarded.

The amount of IR energy radiated from the surface of a given source is a function of the area, the surface temperature, and the surface emissivity. The energy radiated from an exhaust wake is dependent upon the temperature, the chemical composition, and the wake profile. The radiation may be direct or reflected. The amount of suppression required for a given helicopter will depend upon the level of protection required, which will be defined by the procuring activity.

The radiation level, with or without suppression, can be calculated by the projected area method (Ref. 6). The radiation from surfaces follows Lambert's law, which states that the emissive power at any angle to the normal surface is proportional to the cosine of that angle. See AMCP 706-127 and -128(S) for additional information on IR radiation.

Estimation of IR radiation requires definition of appropriate surface emissivity factors. The emissivity factor is a function of surface geometry, material, and surface condition, and cannot necessarily be estimated accurately. Also, reflected radiation from invisible engine surfaces can escape through the exhaust system by means of radiation patterns that can be quite complex. Therefore, final evaluation of IR signatures, with or without suppression, must be made by test.

The most common methods of IR suppression are shielding or cooling of hot surfaces. Shielding involves making the otherwise visible hot surface invisible, and normally results in a requirement to cool the shielding surface. For example, the exposed surface of an exhaust duct may be shielded by surrounding it with another duct—with space between—through which cooling air can flow. Enclosing the hot surface with a layer of insulating material is another method of shielding; the hot surface becomes invisible and is replaced by the cooler exterior surface of the insulating material. Care



L =	HORIZONTAL DISTANCE FROM EXHAUST EXIT, ft
R =	RADIAL DISTANCE FROM EXHAUST CENTERLINE, ft
d_9 =	DIAMETER, ENGINE EXHAUST (STA 9)
T_S =	TEMPERATURE (STATIC), °R
T_{S_1} =	STATIC TEMPERATURE AT ENGINE STATION 1, °R
T_{S_9} =	STATIC TEMPERATURE AT ENGINE STATION 9, °R
V =	VELOCITY, fps
V_9 =	VELOCITY AT ENGINE STATION 9

Fig. 8-9. Exhaust Wake Diagram

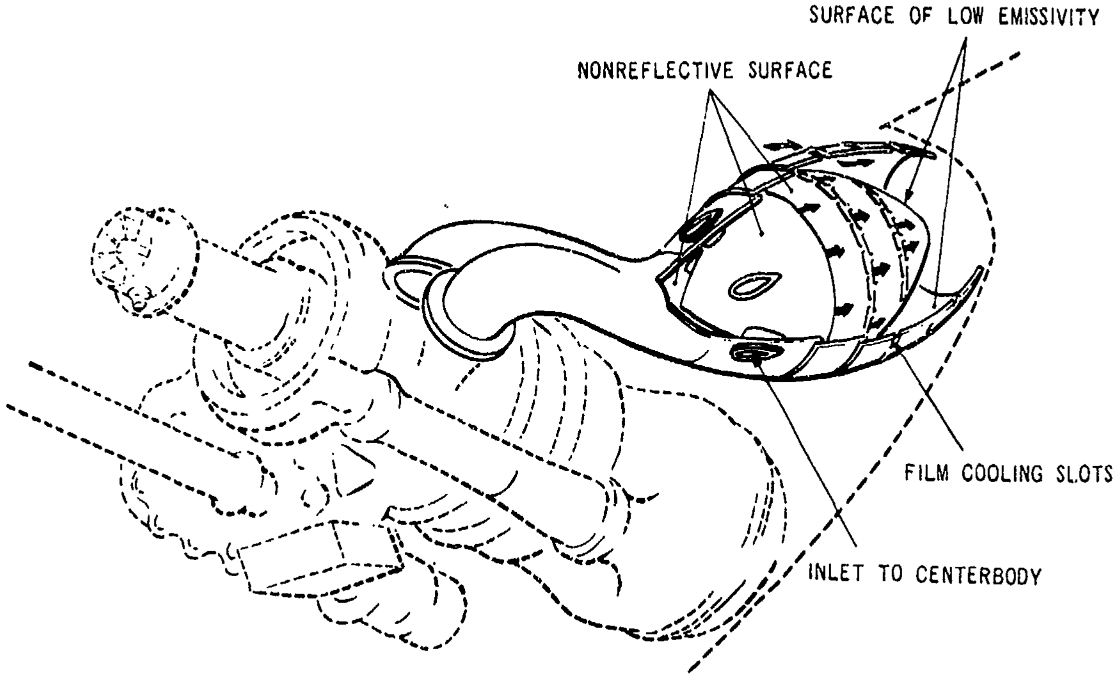


Fig. 8-10. Example IR Suppression Exhaust System

must be used with this method because the equilibrium temperature of the duct may be increased above acceptable limits if the effectiveness of the insulation is high. Alternatively, insulating material may be applied to the internal surface of the duct to cool it. However, in this case, the insulation is in direct contact with the exhaust flow and the efficiency of the duct will be affected adversely by the reduction of area and by the modified inner surface.

Shielding of hot turbine parts can be accomplished by the use of a multiexhaust system. Bends in the ducts serve to shield the engine parts; also, the amount of duct area visible from any given aspect angle is reduced. On the other hand, the design of the bends in the duct to minimize reflection further increases the exhaust system losses.

To minimize reflected radiation from hot metal parts, high emissivity coatings should be applied to the surfaces.

Many design solutions are possible for IR suppression with cooled surfaces used also as shields. One example, shown in Fig. 8-10, uses a conical center plug.

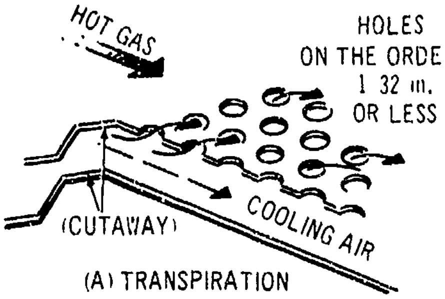
The center plug, which is cooled with air, is used to obstruct the view of the turbine wheels and of the inside surfaces of the hot exhaust duct. The plug is held in place by airfoil-shaped struts that also serve as channels for the flow of cooling air into the hollow plug. The plug and the duct surface may be cooled by transpiration cooling or by film cooling.

Typical examples of transpiration- and film-cooled surfaces are shown in Fig. 8-11.

A second technique is the cooled conical plug in the exit of the tail pipe, as shown in Fig. 8-12. The plug again is supported by cooling-air-carrying struts.

The amount of cooling air required well may be in excess of 10% of the engine airflow. This additional air is used not only to cool the surfaces but also, by mixing, to reduce the exhaust gas temperature. The optimum design for a specified suppression level must be found by a study of trade-offs among the power losses due to duct pressure losses, turbine back pressure, cooling air momentum drag, pumping power, etc.; and the weight of the suppression system components (shields, plugs, insulation, etc.).

Surface plating with low emissivity materials—such as gold—is desirable for hot surfaces that remain visible or may produce reflected radiation, although these coatings may become ineffective after relatively short periods of operation due to deposits of carbon, sand, and dust. Appropriate provision for their cleaning may be necessary.



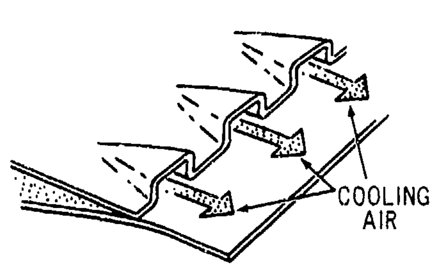


Fig. 8-11. Typical Surface Cooling Methods

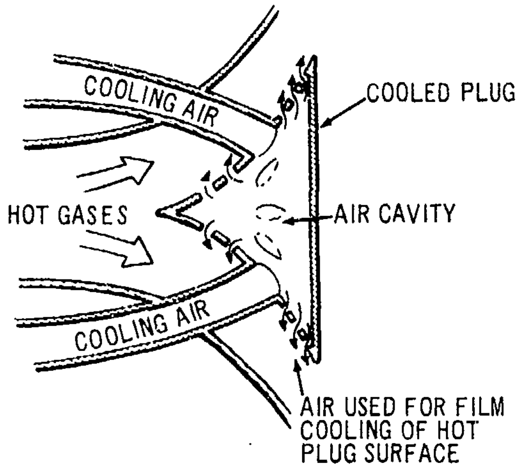


Fig. 8-12. Example of Externally Cooled Plug

Use of a regenerative gas turbine engine is another way of reducing the exhaust temperatures. The fuel saving will help offset some of the weight penalty, especially for long-range missions. The initial cost of the regenerator, however, is high when compared with other IR suppression methods.

The regenerator is a heat exchanger that takes heat from the engine exhaust and transfers it to the air leaving the compressor before it enters the combustor. However, the higher the engine pressure ratio, the higher will be the temperature of the "cold" air, and hence the amount of heat that can be transferred from the exhaust gas to the air at a given value of exhaust gas temperature is reduced. The regenerator, therefore, is effective in reducing this temperature only for relatively low compression ratio engines.

8-2.3.5 Estimation of Exhaust IR Emissions

AMCP 706-127 and AMCP 706-128(S), (*Infrared Military Systems*, Parts One and Two), provide a comprehensive treatment of military infrared technology. Methods for the estimation of IR radiation are provided by this two-part handbook. These methods can be used to evaluate the effectiveness of a proposed suppressor configuration by comparison with the estimated level without the suppressor. The levels of IR radiation also can be compared with the levels defined as mission requirements. However, the final evaluation of suppressor performance will be by test (par. 8-9.2, AMCP 706-203).

8-2.4 PROPULSION SYSTEM COOLING

Cooling must be provided to protect propulsion system components during temperature extremes and to maintain close thermal control when necessary. Thermal control is achieved by regulating the heat transfer to and from the equipment and compartment. The proper management of this heat transfer involves either the singular or combined use of passive and active thermal control techniques.

Thermal control involves the use of inherent methods such as natural convection, heat sinks, emissivities (surface finishes), and insulating materials as well as fans, heat exchangers, or other forced cooling means. Determination of cooling requirements and analysis of engine cooling systems are discussed in detail in Ref. 3.

The engine cooling system normally will include the development and maintenance of a flow of cooling air through at least a part of the engine compartment. This flow should be unidirectional, and careful consideration should be given during preliminary design to the airflow paths to minimize turbulence and its resultant losses while assuring adequate cooling.

8-2.4.1 Insulation

The flow of energy by conduction between a source and a sink is a function of the medium between the source and sink and the temperature difference. Insulation is an example of a medium that regulates the flow of energy by its material properties (such as thermal conductivity and density) and its dimensions. The thermal conductivity of insulation generally is in the range of 0.02 to 1.00 Btu/(hr °F ft). Usually, an insulator is a nonmetal composed of a mixture of gases and solids. While gases are poor heat conductors and solids generally are good heat conductors, the combination results in a usable insulator. Representative conductivities are shown in Table 8-1.

The use of insulation is a common method of protecting temperature-critical components from the heat sources present in the propulsion system. The temperature requirements limit the various types of insulation to those whose properties permit use in the 500-1600°F range.

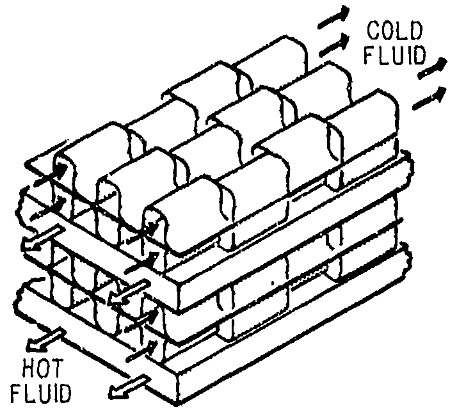
8-2.4.2 Heat Exchangers

A heat exchanger is a device in which thermal energy is transferred (exchanged) from one fluid to another fluid. For example, an oil-to-air heat exchanger (oil cooler) may be used in helicopter applications as a means of maintaining the engine oil and/or the rotor gearbox oil at acceptable temperature levels. This is accomplished by the flow of cooling air through the heat exchanger core while the oil is pumped through other core passages. The rate of heat rejection from the hot fluid to the cooler fluid may be approximated by assuming steady flow, constant fluid properties, constant areas, and one-dimensional flow.

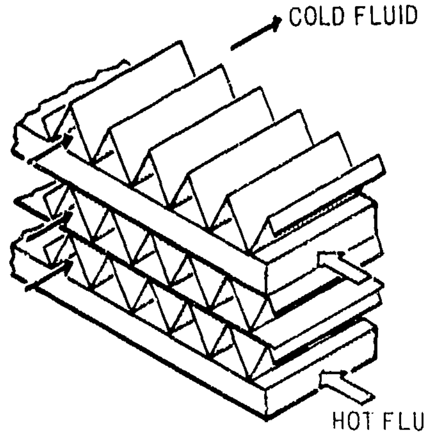
Heat exchanger configurations are numerous; however, most of the current heat exchangers for aviation use are of the compact cross-flow configuration. Although the counterflow is optimum thermodynamically, the overall size and weight of the single-pass cross-flow or multipass cross-flow arrangement, including manifold and transition ducts, will be less than the size and weight of a thermodynamically comparable pure counterflow arrangement (Fig. 8-13). The cross-flow heat exchanger can be fabricated in many configurations such as plate-fin, tubular, finned tube, and shell/tube; each of the configurations may be divided into subgroups, as shown in Table 8-2.

Heat exchangers have a high surface area per unit volume, especially the plate-fin type that is employed for liquid-to-gas applications.

The general design procedure for an optimum heat exchanger is relatively complex. This is due not solely to the mathematical development, but also to the qualitative judgments required of the designer. Qualitative factors include brazing furnace capability, weight, and potential alternative applications. The design must be the result of trade-offs that include the performance effects of the pressure drop, the unit weight, the heat transfer performance, and leakage problems.



(A) COUNTER FLOW ARRANGEMENT



(B) SINGLE-PASS CROSS FLOW ARRANGEMENT

Fig. 8-13. Heat Exchanger Arrangements

TABLE 8-1
SOME REPRESENTATIVE CONDUCTIVITIES

ITEM	CONDUCTIVITY Btu/(hr-ft² (°F/ft))	TEMPERATURE °F
AIR	0.016	100
CORK	0.025	100
ASBESTOS	0.1	100
STEEL (S.S. 321)	9.3	100
ALUMINUM ALLOY	~100	100

Design parameters that relate to heat-transfer performance are:

1. Overall heat conductance coefficient U
2. Area of heat transfer surface A
3. Hot and cold fluid terminal temperatures (both in and out) T
4. Specific heats at constant pressure of hot and cold fluids c_p
5. Weight flow rate of hot and cold fluids W.

Additional considerations include the flow arrangement, i.e., counterflow, cross-flow, parallel-flow, parallel-counterflow, or various combinations of the basic arrangements. Heat exchanger design procedures are outlined in Refs. 3 and 7.

The "ideal" heat transfer q between two fluids may be expressed in the following equations:

$$q = [Wc_p\,(T_{in} - T_{out})]_{\substack{hot\\fluid}}$$

$$= [Wc_p\,(T_{out} - T_{in})]_{\substack{cold\\fluid}} \quad \text{, Btu/hr} \qquad (8\text{-}6)$$

where

W = fluid flow rate, lb/hr
c_p = specific heat, Btu/lb-°F
T_{in}, T_{out} = fluid terminal temperature, °F

$$q = UA\,(\Delta T)_m \quad \text{, Btu/hr} \qquad (8\text{-}7)$$

where

U = conductance coefficient, Btu/hr-ft^2-°F
A = area, ft^2
$(\Delta T)_m$ = mean value of the terminal temperature differences, °F

See Fig. 8-14 for a typical counterflow system.

The efficiency η of the heat exchanger is expressed

$$\eta = q_{actual}/q_{ideal\ max}$$

where

$$q_{ideal\ max} = Wc_{p\ min}\,(T_h - T_c)_{in}.$$

8-2.4.4 Fans

The circulation of fluids, both hot and cold, is an important aspect of cooling systems. Only the fan as the prime mover of cooling air is considered in this paragraph.

The selection of the correct fan is dependent upon the system pressure drop, required volumetric flow rate, weight, size, and power requirements. The pressure drop of the system (excluding fan) is compared with the performance curve of a known fan. If a comparison of the curves indicates an intersection at a point that will provide adequate volumetric flow rate Q, and the fan meets configuration, weight, and power criteria, then the fan design is acceptable. See Fig. 8-15 for typical pressure drop versus volumetric flow rate for a

TABLE 8-2
SOME TYPES OF CONSTRUCTION FOR COMPACT CROSS-FLOW HEAT EXCHANGERS

PLATE-FIN		TUBULAR	FINNED TUBE	SHELL AND TUBE
FIN TYPES:		TUBE TYPES:	TUBE TYPES:	TUBE TYPES:
PLAIN TRIANGULAR		PLAIN	CIRCULAR FINS	PLAIN
PLAIN RECTANGULAR		DIMPLED	SHEET FINS	TURBULATED
RECTANGULAR OFFSET		TURBULATED	OFFSET SHEET FINS	
MATERIALS:		MATERIALS:	MATERIALS:	MATERIALS:
PLATES	FINS	Al and SS	Al and Cu	Al and SS
Al	Al			
SS	SS			
SS	Ni			

NOTE SS = STAINLESS STEEL, Al = ALUMINUM, Cu = COPPER, Ni = NICKEL system and matched fan. Ref. 3 provides additional fan performance data.

The evaluation of pressure losses is the same as outlined in the development of the air induction system (par. 8-2.2.4).

The specific speed, type of fan, and power requirements are estimated by consideration of system pressure drop, blower speed, airflow rate, and the pressure/flow coefficients.

The specific speed N_s commonly is used in relating performance of different types of fans.

$$N_s = \frac{N\sqrt{Q}}{(\Delta P_s/\sigma)^{3/4}} \tag{8-8}$$

where

N = blower speed, rpm
Q = volumetric flow rate, cfm
ΔP_s = system pressure drop, psi
σ = ratio of ambient air density to sea level standard air density, dimensionless

No units are expressed for N_s as the equation is dimensionally impure with N in rpm, Q in cfm, and ΔP_s in psi.

The specific speed of a fan is a reasonable indication of the type of fan blade design that is adequate within the envelope of operating efficiency. The type of blade design is available from charts based upon manufacturers' past experience.

The fan output power hp_f is approximated by the following relationship:

$$hp_f = \frac{\Delta P_s Q}{229} \tag{8-9}$$

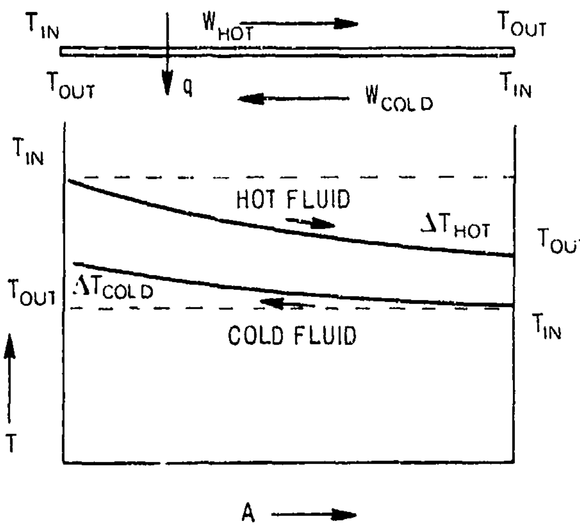


Fig. 8-14. Typical Counterflow Heat Transfer

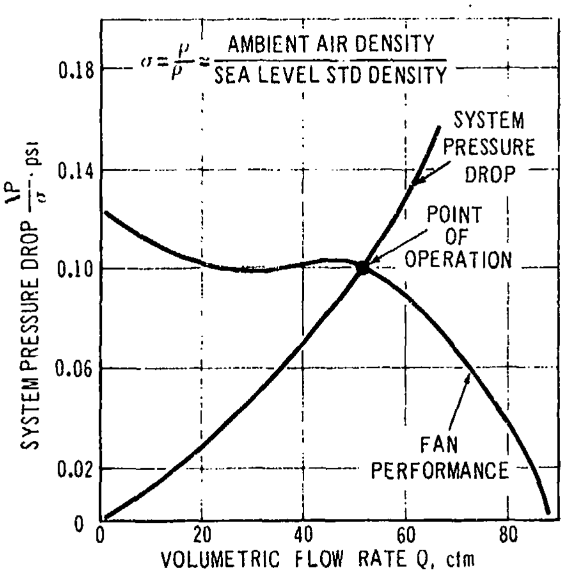


Fig. 8-15. Typical Pressure Drop vs Volumetric Flow Rate

8-2.4.5 Engine Compartment Cooling

The temperatures of the engine compartment air and adjacent structure and components are a result of the engine heat rejection, mainly by radiation and convection.

The temperatures in the engine compartment are calculated by considering:

1. Engine heat rejection for a given temperature
2. Radiation inputs of surface emissivities and geometric shape factors
3. Compartment airflow rate
4. Convective heat transfer film coefficient
5. Environment of the engine and the surrounding compartment structure.

The air, component, and structural temperatures within the engine compartment then may be determined by simulating the engine and the surrounding structure as a cylinder and a concentric shell. The basic solution of the thermal model is accomplished by equating heat in with heat out plus heat storage for transient conditions. If the design is such that air is circulated around the engine, a major part of the engine heat rejection will be to the air; however, some heat still will be transferred to the compartment structure by radiation, convection, and possibly conduction at the attachment points. Methods of analysis are provided by Ref. 8.

8-2.4.6 Cooling System Air Inlet

The basic principles for an air inlet system are discussed in par. 8-2.2. These principles, presented for the engine air induction system, are applicable equally to cooling system inlets.

8-2.4.7 Ejectors

The ejector principle is a method of obtaining the necessary flow of cooling (or scavenging) air. Ram air, bleed, or an auxiliary blower are other possible methods. However, the ejector principle normally minimizes performance loss.

The ejector is a device that creates a secondary flow from a primary flow. The primary flow entrains secondary air at the ejector exit by creating a low-pressure region at the exit. Due to the resulting pressure differential, both primary and secondary air then flow through the ejector. Thus, forced convection that aids in cooling (such as the engine compartment) can be provided by the secondary flow. The primary flow may be engine exhaust, engine bleed air, or any other source capable of producing a region of sufficiently low pressure and of adequate area to induce the required secondary flow.

The functioning of the ejector varies with the primary flow rate and the general design. The design configuration determines the ejector pressure ratio. For engine compartment cooling, ejector flow may be expressed as a percentage of the secondary flow (engine compartment airflow), and typically varies from approximately 2% at sea level conditions to 6% at high altitude. Methods for ejector design and evaluation are provided by Ref. 3. The final ejector design must be subjected to trade-offs, considering thrust loss and weight penalty associated with this method and with alternative means of establishing the desired airflow.

8-2.4.8 Survivability Upon Loss of Cooling System

The most susceptible component of the cooling system, relating to survivability aspects, is the oil cooler. The purpose of the oil cooling system is to maintain oil, either engine or gearbox, at an acceptable temperature level. The loss of a heat exchanger, or a ruptured oil line, negates the stated purpose and thus affects the survivability of the aircraft. Damage to the heat-exchanger system first would manifest itself in higher oil temperatures. Either thermal control or fluid control valves can be used to recirculate the oil through the oil sump by bypassing the heat exchanger, in case of cooler damage. Eventually, the oil temperature will increase to an unacceptable level, as the heat normally rejected through the heat exchanger builds up, but the additional operational time may permit a safe landing.

8-2.4.9 Cooling Power Requirements

The cooling power requirements are dependent upon the means selected to provide the desired airflow. Possible methods specifically discussed are a cooling fan, bleed air ejector, and exhaust ejector.

The cooling fan power requirement is established by the airflow and cooling system pressure drop, and is calculated by applying an appropriate efficiency factor to the fan output power given by Eq. 8-9.

The power loss due to the use of compressor bleed air for a bleed air ejector, is determined from curves contained in the engine specification. Generally, the power loss is about 4% of rated power for each percentage point of bleed air. Thus, this method of cooling is undesirable, and should be avoided.

The power loss calculation for the exhaust ejector is based on the known static pressure at the mixing plane of the primary and secondary gas flows within the ejector. The subsequent power then is calculated from the engine specification for the differential pressure between the mixing plane and the ambient pressure condition. A similar calculation is required for the basic exhaust system whether an ejector is employed or not. Therefore, it is possible that the ejector loss (or cooling power required) be zero, but that the engine suffer an exhaust system power loss.

8-2.5 ACCESSORIES

The requirements for secondary power (electrical, hydraulic, pneumatic, or mechanical) to be provided by the helicopter engine are dependent upon the mission characteristics. An additional consideration is the method of power extraction that would be least detrimental to aircraft performance. There are two principal methods of extracting accessory power from the engine. Power may be supplied pneumatically by compressor bleed air or mechanically either by accessory drives powered by the power turbine (or components of the drive system powered by the power turbine) or by accessory drives powered by the engine gas generator. Accessories for gas turbine engines are divided into these two categories.

8-2.5.1 Bleed-air-driven Accessories

Compressor bleed air may be used for bleed-air-driven accessories. Bleed air is supplied from connections located on the engine case. This air is the source of power for operating helicopter accessory items such as air-turbine-driven alternators, hydraulic pumps, and environmental control units. On multiengine helicopters equipped with pneumatic starters, bleed air from an operating engine also may be used to start the other engines.

The configuration of the engine will establish the individual engine bleed air systems. On engines utilizing either single-shaft axial or centrifugal compressors, one air bleed system consisting of several ports may be available. On multiple-spool (three or more for free turbines) engines, two separate bleed systems—high-pressure and low-pressure—may be available.

Bleed air almost always is used for anti-icing the engine inlet. This air also may be used for the engine air induction anti-icing system. Cockpit and cabin pressurizing and heating units may be operated by a separate air-bleed-driven compressor, although on smaller helicopters the compressor bleed air may be brought directly to the cabin for this purpose. Although more economical, the second method may subject personnel to harmful engine contaminants and extensive testing and monitoring are required before approval for use is obtained.

The first method, although providing uncontaminated air, is expensive in terms of both weight and power. This system is recommended for use on large passenger vehicles.

Pressures and temperatures of the engine bleed air will be contingent upon individual engines and upon the operating conditions of the helicopter. A typical example for present-day dual-bleed engines is low-pressure bleed air of approximately 50 psi at a temperature of more than 250°F and high-pressure bleed air of around 160 psi with temperatures in excess of 650°F. Future technology engines with compressor pressure ratios greater than 20:1 will provide bleed pressures and temperatures of approximately 300 psi and 800°F, respectively.

The quantity of air available for driving accessories and for other purposes in the aircraft is usually 1-6% of the total airflow through the engine gas generator.

The power available for any weight rate of flow can be ascertained from the manufacturer's specification for the particular engine being used. Additional discussion of engine bleed systems is provided by Ref. 3.

8-2.5.2 Mechanically Driven Accessories

The mechanical method of driving accessories involves driving them through the main gearbox, mounting them on the engine on drive pads provided for the purpose, or both. Main-gearbox-driven accessories include units such as the gearbox oil pump, gearbox and/or engine cooling fan, rotor tachometer generator, hydraulic pumps, and generators. Helicopter accessories mounted on and driven directly by the engine include units such as alternators, generators, hydraulic pumps, and other items required to provide power for the aircraft electrical, hydraulic, and pneumatic systems, as well as units—such as the engine tachometer generator—that must be connected directly to the engine.

The drive pads currently used are the gear-driven AND pads, which employ a multibolt attachment, and are listed on AND 10230. The quick-attach-detach (QAD) concept of accessory drives provides a large improvement in maintainability. The QAD pad has a V clamp attachment, requiring a single bolt tiedown and an overcenter (trunk latch) clamping device that may be oriented readily for accessibility. The QAD pad also reduces the volume requirement for accessories by effectively eliminating the wrench clearance associated with the bolted attachments. Current accessories with QAD adapters are available and may be used. This method (conventional configuration plus adapter), however, results in additional space, weight, and cost that must be considered in the preliminary design of the aircraft.

The drives that supply the power may be divided into two categories: variable-speed and constant-speed. Variable-speed drives are satisfactory on helicopters not requiring a great amount of electronic equipment. On aircraft having DC electrical systems, inverters may be used for generation and control of the frequency of the AC power desired. For applications in which frequency control is not a critical parameter, an engine-driven AC alternator will suffice. However, there will be a weight penalty because of these frequency variations (see par. 9-3). For hydraulic systems, this variable-speed drive does not penalize the helicopter because accumulators may be employed to provide a more constant output.

The need for a constant-speed drive has become prevalent in current vehicles because of the extensive use of electronic equipment. Most of this equipment requires a frequency that is constant within $\pm 5\%$ to maintain expected performance. Constant-frequency power may be obtained by a number of methods:

1. Generating AC power at a constant frequency
2. Generating DC power and converting it to AC power through an inverter driven at constant speed
3. Using electronic equipment designed for converting variable frequencies to a constant frequency (solid-state inverter).

8-2.5.3 Mechanical Power Extraction

The electrical and hydraulic subsystems are the principal users of secondary power extracted from the engine. The engine specification will identify the available accessory drives as to type, ratio to gas generator speed, allowable torques, and direction of rotation. The engine specification also will state the maximum power extraction limits for the gas generator as well as the maximum power that may be extracted from each of the drive pads.

The power required for electrical subsystems may be determined by methods outlined in par. 9-3.

The most common type of hydraulic pump is the positive-displacement piston-type that may operate either as a fixed or variable-displacement pump. Pump performance is rated at a speed where the pump is designed to operate continuously. Typical hydraulic pump performance curves are shown on Fig. 8-16. Hydraulic system power requirements are discussed in par. 9-4.

During the preliminary design phase the helicopter electrical and hydraulic load spectra will be established. It should be verified that the individual as well as the combined power demands for these hydraulic and electrical systems do not exceed the engine accessory power output capability. It normally is preferable to install hydraulic pumps and electrical generators or alternators on the main rotor gearbox rather than on the engine.

8-2.5.4 Future Drive Designs

In most current aircraft, accessories required for support of the aircraft systems are mounted directly on the pads of the engine accessory gearbox. Future aircraft will demand more accessory power and, because of this, accessories will be larger. When these larger components are installed on AND pads, undesirable envelope sizes, moments, and configurations will result. Consideration is being given to using higher drive speeds for accessories. Mounting pads providing these higher speeds will be in accordance with MS 3325, MS 3326, MS 3327, MS 3328, or MS 3329, and may provide for drive speeds of up to 30,000 rpm. These drive standards are configured for QAD pads with modification to meet the specific requirement of each accessory relative to lubrication, sealing, and shaft drive size.

To reduce the engine cross-section further, engine manufacturers are considering taking all airframe accessory drives off the engine. This concept provides a remote mounting for all helicopter accessories, including the starting system, and greatly simplifies maintenance, improves reliability, and reduces vulnerability of the accessory package. A remote aircraft gearbox system would have many advantages, such as:

1. The gearbox and accessories can be located where factors such as vibration and temperature are considerably less severe.
2. Easier engine installation, removal, and servicing are afforded.
3. Easier removal, servicing, and installation of the accessories should result.
4. Because of the smaller engine package, the aircraft may be designed for a lower profile and for a smaller frontal area.

8-2.5.5 Effect of Air and Power Extraction

The power extracted for driving accessories is usually a small percentage of the power required to drive the compressor; however, power can be extracted only at the expense of output shaft power and additional fuel. The loss in engine power and the increase in fuel consumption are much greater when an accessory is driven by compressor air bleed than when the same accessory power is extracted mechanically. The use of air bleed also will raise turbine inlet temperature, which in turn will be evidenced by a rise in exhaust gas temperature.

Determination of operating loads for the aircraft electrical, hydraulic, and pneumatic systems is discussed in Chapter 9. The curves, charts, formulas, and methods of analysis for calculating available engine power—as well as the increase in fuel consumption and the gas temperature rise resulting from either mechanical power extraction or air bleed—are contained in the engine specification. Procedures differ for specific engines as well as among different engine manufacturers.

8-2.5.6 Redundancy Requirements

The subject of redundancy as it relates to accessory systems and drives is covered in pars. 7-4 and 9-3. As described therein, special care is necessary to insure that presumably redundant systems are in fact redundant, i.e., so designed that a single foreseeable failure does not result in the loss of two or more supposedly redundant accessory systems. For example, if two gen-

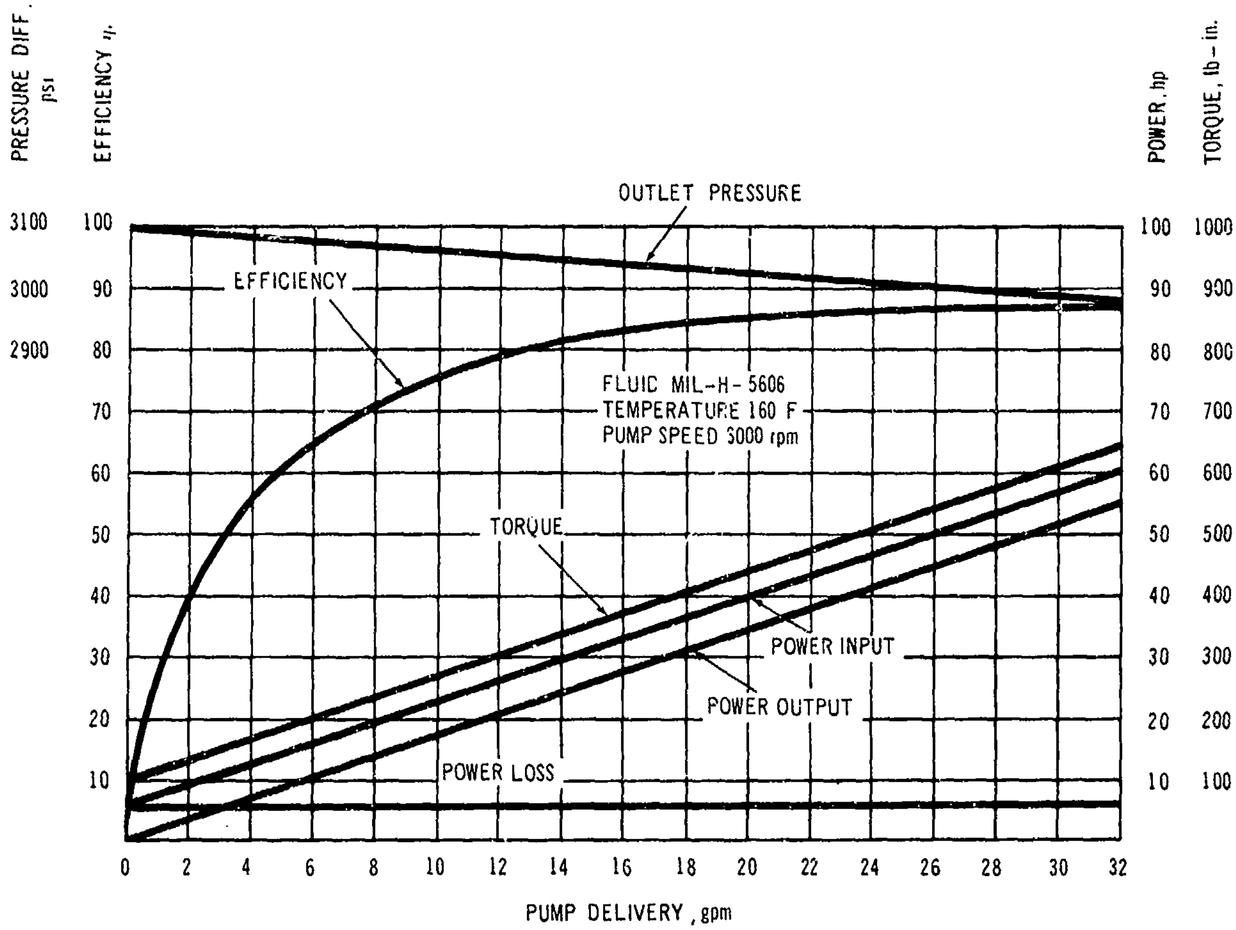


Fig. 8-16. Typical Performance Curves of a Variable-displacement Hydraulic Pump

erators were located on a gearbox, the failure of the gearbox (single failure) could result in the loss of both generators. Therefore, the generators are not redundant. A thorough analysis of accessory systems will lead to a series of functional network representations, from which redundancy can be evaluated and subjected to a variety of optimizing algorithms (Ref. 9).

8-2.6 MAINTENANCE

To facilitate maintenance and insure maximum operational readiness and reliability, the design should provide for quick engine removal, with the disconnect points conveniently grouped and arranged so that they are accessible with minimum interference. Fluids from any disconnected line should not drop on components, field personnel, or decals during maintenance. Other factors that should be considered are:

1. Use of large, quick-opening access doors for engine and engine accessory section servicing

2. Ample space in the engine accessory area for replacement of engine-mounted components; use of approved, quickly detachable connectors for electrical, plumbing, control, and accessory units

3. Maximum accessibility with emphasis placed on preflight and postflight requirements for inspection, cleaning, and adjustment. Items such as drain plugs, filter elements, connector plugs, valves, switches, and visual inspection points should receive special consideration. These items should be located where necessary access is provided.

4. The possibility of a preflight check of the helicopter without need for special stands or ladders. On larger aircraft it is advisable to consider integral working platforms or built-in steps.

In addition, there may be factors required to satisfy varying climatic and environmental factors. For instance, maintenance of the helicopter should be possible by personnel wearing arctic gloves and clothing.

Other servicing features that should be considered, if applicable, include ground level pressure refueling; ground level fuel and oil sight gauges; and ground level connections for electrical, hydraulic, and pneumatic systems checkout.

Use of a mock-up engine to verify the installation is recommended. This mock-up should be made available as early in the design stage as possible. The mock-up engine may be a wooden model or a shell of an actual engine. In any event the CG of the engine and all connections must be identical to the actual article. Use of a mock-up will make it possible to establish a helicopter structural configuration that also provides the accessibility required for engine maintenance and service.

8-3 SUMMATION OF POWER LOSSES

8-3.1 GENERAL

In using the engine specification to determine the installed power available at the engine output shaft, the following parameters that affect power losses should be considered:

1. Air induction system pressure loss due to inlet geometry, filter, and particle separator
2. Exhaust back pressure due to geometry of exhaust duct, and additions to the exhaust system such as infrared radiation or noise suppressors
3. Temperature changes from standard day, including any temperature rise at the engine inlet
4. Power extracted to run accessories
5. Air bleed for the environmental control system, anti-icing/deicing systems, rain removal, fans or blowers, bleed air ejectors, scavenge of particle separator installations, and other accessories
6. Engine anti-ice protection.

Power losses between the engine output shaft and main rotor are:

1. Transmission losses
2. Antitorque requirements (not required in tandem configurations)
3. Accessories that require power from the drive system.

Each of these losses is discussed in detail in the paragraphs that follow.

8-3.2 ENGINE INSTALLATION LOSSES

8-3.2.1 Air Induction System Pressure Loss

The design of the engine air induction system must be analyzed to determine the change in pressure through the induction system. MIL-D-17984 presents a method to be used for this analysis. The necessary equations and factors for use in the equations also are given by Ref. 3. For helicopters in the flight test stage, the pressure loss can be measured.

With the increased use of helicopters in unimproved areas, the installation of particle separators and/or filters has become more important. These installations cause large induction system pressure losses that must be included in the calculation of installation losses.

The model specification for the engine to be evaluated presents charts and methods to be used to calculate power, fuel flow, and engine exhaust gas temperature. The correction factors are functions of the induction system pressure loss divided by the engine inlet (ideal) total pressure. At a given value of gas generator speed, an inlet pressure loss decreases the output shaft power, increases the exhaust gas temperature, and increases the fuel flow. Because the engine power rating is determined by a given value of exhaust gas temperature, the available power is reduced by a greater percentage than the percentage pressure loss. This percentage of power loss typically is greater than twice the percent pressure loss. Also, the fuel flow for a given output shaft power increases due to the air induction system pressure loss.

A second loss associated with the engine air induction system is the momentum, or ram, drag, caused by the change in velocity of the engine airflow. This drag is not an engine installation loss, but should be considered to be a drag increase on the airframe. The propulsive force of a helicopter is provided by the main rotor; therefore, an increase in drag requires a power increase for the same airspeed, which must be determined by the efficiency of the main rotor.

The momentum drag is equal to $(W_a/g)V$, where W_a equals the weight flow of the air into the engine in lb/sec, and V is the helicopter forward speed in fps. Methods and charts for computing engine airflow are provided by the engine manufacturer. The airflow must be corrected for engine installation losses and is a function of engine power condition.

8-3.2.2 Exhaust Pressure Rise

The engine exhaust system design must be analyzed to determine the engine outlet static pressure. Proper design of the exhaust system will result in an engine outlet pressure equal to ambient pressure. In this case, the installation loss due to the exhaust system would be zero. If the exhaust system is designed to decrease the outlet pressure, there would be a gain in power; conversely, an increase in pressure results in a power loss. MIL-D-17984 presents a method of computing pressure changes experienced in the exhaust duct. When the helicopter is in the flight test stage, the exhaust pressure can be measured.

The addition of IR and noise suppressors to the exhaust system causes very large increases in exhaust outlet pressure. These pressure changes must be considered in computing exhaust duct installation losses.

The exhaust installation correction factors are a function of the exhaust outlet pressure minus ambient pressure divided by the ambient pressure (percent exhaust pressure rise). The exhaust correction factors for exhaust gas temperature and fuel flow are the same as the factors for the inlet pressure drop, while the correction factor for shaft power is less than that for the inlet. An exhaust pressure rise reduces the shaft power available at a given value of exhaust gas temperature and increases the fuel flow at a given value of shaft horsepower.

8-3.2.3 Engine Inlet Temperature

Turbine engines are affected greatly by engine inlet temperature. At a given value of gas generator speed, an increase in temperature decreases the shaft power and increases the measured gas temperature MGT. For a typical engine, intermediate power is reduced 18% by an increase in ambient temperature from 59°F to 95°F; whereas, in a reciprocating engine, the power change is equal to one minus the square root of the absolute temperature ratios, or 3%. Because of this, the hot day performance requirement generally governs the design of, or engine selection for, turbine-powered helicopters.

The fuel flow at a given shaft power is little affected by engine inlet temperature.

The engine specification shows methods of correcting for inlet temperatures. A second method that allows quick calculations of temperature effects is "referring". A plot is made of $SHP/(\delta\sqrt{\theta})$ versus MGT/θ, where δ is the ratio of ambient pressure to pressure at sea level and θ is the ratio of ambient temperature (°R) to sea level standard temperature (518.7°R). Thus, for the limit *MGT* and ambient conditions, the referred value of *MGT* can be computed. Knowing the referred *MGT*, the corresponding value of referred *SHP* can be read from the chart and the actual value of *SHP* can be computed. This method of analysis can be used to good advantage in parametric studies performed by computers for preliminary design because it avoids the use of large amounts of data in the form of tables.

In some designs, the inlet is so located that the recirculation of air causes a temperature rise at the engine inlet. This generally occurs in a helicopter hovering in ground effect where the presence of the ground causes changes in the airflow. This temperature rise adds to the ambient temperature and results in an increase in the ambient temperature of the engine inlet. As the temperature rise often is related to proximity of the engine inlet to the ground, it is possible to have different installation losses when hovering in ground effect, hovering out of ground effect, or in forward flight.

8-3.2.4 Power Extraction

Electrical generators and other accessories are run by power extraction. The engine specification presents methods of computing the effect of power extraction. For a given value of power extraction, the reduction in shaft horsepower available at intermediate power is a function of ambient conditions. The power loss increases with ambient temperature. Thus, at 100°F ambient, the power loss can be as much as twice the power extracted.

8-3.2.5 Engine Air Bleed

Air can be bled from the engine for many uses, such as operating environmental control systems and the scavenging of a particle separator. The correction for engine air bleed is a function of percent bleed ($100\,W_b/W_a$), where W_b is the amount of bleed air in lb/sec. As with the other losses, at a constant gas generator speed engine air bleed decreases *SHP*, increases *MGT*, and increases fuel flow.

The loss in power due to air bleed is only slightly dependent upon ambient conditions. Depending upon the engine design, 1% air bleed at intermediate power reduces power by 2-4%.

8-3.2.6 Air Induction Anti-ice

Engine anti-ice air is a special application of bleed air that reduces the power available at a given value of exhaust gas temperature. It also increases the fuel flow at a given shaft horsepower. Generally, the anti-ice air requirement is not a performance-limiting factor because it is needed only on cold days when there is ample power available.

8-3.2.7 Charts for Presentation of Installed Power

MIL-P-22203 includes a format to be used in presenting installation losses and installed power. This specification requires a plot of installation losses as a percent decrease in shaft horsepower, and a percent increase in fuel flow versus forward speed. This is to be presented at standard temperatures for altitudes within the operating envelope, and for intermediate, maximum continuous, and reduced power levels. Plots are required for installed shaft horsepower and installed fuel flow versus forward speed at standard conditions for altitudes covering the operating envelope and for intermediate, maximum continuous, and reduced power levels.

Because MIL-P-22203 is written mainly to cover airplanes, temperature corrections to the available power are not considered important. As noted earlier, engine inlet temperature has a large effect on power available. The ability of a helicopter to hover is an important performance parameter that is especially critical on hot days. To determine the hover capability of a helicopter, the available power must be determined. A plot of intermediate and maximum installed power at zero forward speed should be provided. This plot should show installed intermediate (or maximum) power versus altitude at various values of ambient temperature ranging from 0°F to 130°F. Occasionally, there is an inlet temperature rise due to recirculation of air when hovering in ground effect. In this case, there should be a plot of maximum power available both in and out of ground effect. Range analysis sometimes is required for hot days; therefore, plots of installed fuel flow versus installed *SHP* should be provided for the nonstandard days being considered.

8-3.3 LOSSES BETWEEN ENGINE SHAFT AND MAIN ROTOR

8-3.3.1 Transmission Efficiency

The loss due to power transmission must be considered. This loss is a function of the design of the transmission system and should be estimated by the designer.

8-3.3.2 Power Required for Antitorque

All single-rotor helicopters, except those that are tip-driven, require a method of counteracting the torque used to drive the main rotor. Even those helicopters that are tip-driven must have a method of control and, therefore, require power to drive the control device. A tail rotor has proven to be the most efficient method of providing antitorque and control requirements.

A tail rotor, like any other rotor, requires more power in hover than forward flight. The power requirements of a tail rotor in steady hover flight can be estimated using a typical value of tail rotor Figure of Merit. The Figure of Merit is the ratio of the minimum possible power required to provide a given thrust to the actual power required for a given thrust. The minimum possible power is equal to the induced power, which can be computed easily. The Figure of Merit of a tail rotor typically is equal to 0.6. The following equations show a method of estimating tail rotor thrust T_{TR} and power hp_{TR} for hover:

$$T_{TR} = \frac{Q}{l_t} = \frac{550\, hp_{MR}}{\Omega_{MR}\, l_t}\ , \text{lb} \tag{8-10}$$

$$hp_{TR} = \frac{T_{TR}}{550\, M_{TR}} \sqrt{\frac{T_{TR}}{2\rho\pi R_{TR}^2}} \tag{8-11}$$

where

hp_{TR} = tail rotor power, hp
hp_{MR} = main rotor power, hp
l_t = length from CL of main rotor to CL of tail rotor, ft
M_{TR} = tail rotor Figure of Merit
Q = main rotor torque, lb-ft
R_{TR} = tail rotor radius, ft
T_{TR} = tail rotor thrust, lb
ρ = density of air, slug/ft^3
Ω_{MR} = main rotor rotational speed, rad/sec

The tail rotor hover power requirements for a single-rotor shaft-driven helicopter generally average 8-15% of main rotor power.

In steady forward flight, the tail rotor often is unloaded by the helicopter vertical stabilizer; therefore, the thrust requirements are less. In addition, the tail rotor induced power requirement decreases with forward speed, so the power required at a given thrust is decreased. Tail rotor power averages 4-7% of main rotor power in forward flight.

At the present time, a replacement for conventional tail rotors is being sought. The most likely candidate is a ducted fan. The power requirements for a ducted fan are much greater than for a conventional tail rotor due to the higher disk loading and greater solidity. The power requirements for ducted fans and other unconventional antitorque systems must be estimated for the specific design.

8-3.3.3 Accessories

Accessories that take their power from the drive system must be included in computing power losses. The number of accessories and their associated power requirements are dependent upon the design of the helicopter being analyzed. These power requirements are summarized as secondary power requirements and are discussed in Chapters 7 and 9.

8-4 FUEL AND LUBRICATION SYSTEMS

The selection of fuel and oil system components pertinent to preliminary design, including general safety considerations, is covered in the paragraphs that follow.

8-4.1 FUEL SYSTEM REQUIREMENTS

MIL-F-8615, MIL-T-5578, MIL-T-6396, and MIL-T-27422 are general specifications that should be used as guides for component requirements.

The fuel system consists of tanks for fuel storage, refueling and defueling devices, feed lines between tanks and engines, pumps, and valves. Instrumentation provides sufficient data to permit continuous inflight monitoring of fuel system performance. The system should permit any engine to operate from any tank without affecting the operation of the other engines.

For maximum reliability, the operation of fuel feed systems should be independent functionally of the operation of other helicopter systems. For maximum fire protection and reliability, suction fuel feed systems should be used in place of pressurized systems. Protection of the crew and passengers can be enhanced further by avoiding storage of fuel within, and to the maximum extent practicable, above or below occupied areas.

Main fuel tanks normally consist of one or more bladder cells interconnected to form a tank of the required capacity. The tank configuration and installation must comply with MIL-F-38363. External fuel can be contained in tanks complying with MIL-T-7318 or MIL-T-18847. To minimize turnaround time, it should be possible to remove and replace all external tanks readily and without disassembling the helicopter.

Each engine feed tank should contain a device that warns of low fuel supply. The amount of fuel remaining when the device is actuated must be sufficient for an engine to operate for 30 min at maximum range power setting, unless otherwise specified.

The refueling system is the part of the propulsion system installation used to fill the fuel tanks. If pressure refueling is required, it should refuel all tanks simultaneously from a single connection. Operation of shutoff valves and precheck system must not depend upon the operation of any other helicopter system or an auxiliary power unit. To prevent excessive surge pressure within the refueling system, shutoff valve closure time may be controlled or appropriate pressure relief valves incorporated.

Defueling of tanks should be possible using the pressure refueling system when installed. A low-level shutoff valve can be used to insure complete defueling of all tanks when starting with unequal fuel quantities in each tank.

The fuel system *shall* be designed to be crashworthy. An applicable specification for such systems is not yet available, but the applicable design criteria are provided by Refs. 2 and 10. These references describe the methods of construction and installation required to assure the avoidance of post-crash fire following survivable crashes. These requirements were developed during a comprehensive test program that included crash testing of components as well as of helicopters with prototype systems installed.

Crashworthy fuel tanks *shall* be manufactured and qualified in general accordance with MIL-T-27422, but the "tough-wall" materials described in Refs. 2 and 10 *shall* be used. These references also describe the shapes required to obtain the resistance to tearing, snagging, or other failure of the tanks that could result in fuel spillage. The references also describe in detail acceptable methods of attachment of the tanks to airframe structure as well as crashworthy methods of attachment of other fuel system components both to fuel tanks and to airframe structure.

Fuel lines and fittings suitable for a crashworthy fuel system must be of an acceptable configuration and also must be so installed that the failure modes following an otherwise survivable crash do not compromise the crashworthiness of the fuel system.

The references describe both the component design requirements and the methods of assembly and installation necessary to achieve crashworthiness of the system. Avoidance of post-crash fires can be achieved best by avoiding both the spillage of fuel and the presence of sources of combustion. Attention, therefore, also must be given to the possible failure modes of electrical components and wiring. The installation of electrical system components in the vicinity of any potential fuel spillage area must include protection against sparking in the event of a crash. Sufficient information is provided in the references to design a fuel system that meets all functional and flight safety requirements, while also meeting all criteria for crashworthiness.

8-4.2 REQUIREMENTS FOR OIL SYSTEMS

MIL-O-19838 should be used as a guide for component requirements.

8-4.2.1 Oil Tanks

Oil tanks may be constructed from flexible, crash-resistant material equivalent in strength to that used for the fuel tanks. All tank fittings *shall* have a pullout strength of not less than 80% of the ultimate strength of the tank wall The tank mounting *shall* be able to withstand a 30-g force applied in any direction. Oil tanks should be located as far from anticipated impact areas as possible, and housed in separate compartments built into the airframe.

Tanks *shall* not be located where spilled or sprayed oil could be ingested readily into the engine or ignited by the engine exhaust. Oil tanks *shall* not be located in or above engine compartments, except for engine-furnished oil tanks; in electrical compartments; in occupiable areas; or under heavy masses, such as engines and gearboxes.

8-4.2.2 Oil Lines and Couplings

All oil lines should consist of flexible hose with a steel-braided outer sheath. The number of couplings *shall* be kept to a minimum.

8-4.2.3 Other Oil System Components

Other oil system components (e.g., pumps, valves, filters) *shall* not be located in electrical compartments, occupiable areas, or near the bottom of the fuselage.

Oil filters should not be located in the engine compartment unless they are an integral part of the engine. Other components located in the engine compartment should be restricted to those absolutely necessary for proper operation of the system.

The construction and mounting of oil system components *shall* be able to withstand a 30-g force applied in any direction.

8-4.2.4 Oil Cooler

The oil cooler *shall* not be located in the engine compartment (except engine-furnished oil coolers), under engines or gearboxes, or in any area where oil could be spilled or sprayed onto hot surfaces or ingested into the engine. The oil cooler should be located as far from anticipated impact areas as possible. The oil cooler mounting *shall* be able to withstand a 30-g force applied in any direction.

8-5 PROPULSION SYSTEM FIRE PROTECTION

8-5.1 GENERAL

Many fire protection factors are involved in major system design—e.g., location of equipment, selection of materials, shielding or insulating of possible ignition sources, and compartmentalization of systems. The propulsion system represents the most critical area relative to fire protection due to its flammable fluids and possible ignition sources. The designer must insure that the preliminary design is optimized with respect to the following four basic elements of fire protection:

1. Prevention of fire
2. Resistance to the spread of fire
3. Reliable fire and overtemperature detection systems
4. Positive fire extinguishing.

8-5.2 FIRE PREVENTION

The following general rules and safety precautions apply to propulsion system fire prevention:

1. Isolate all combustibles from sources of ignition.
2. Insure proper drainage of combustibles from the vehicle for all attitudes of ground and flight operation. Incorporate provisions for proper removal or dispersion (by ram or other positive means) of ignitable vapors.
3. Insure that drain lines will drain clear of the helicopter and are sealed to negate impingement and prevent reentry of combustibles.
4. Design fuel vents to insure that spillage will not occur in any attitude. Locate vents in areas isolated from engine exhaust or heated surfaces and away from electrical ignition sources.
5. Insure that all fuel system equipment is isolated from ignition sources by adequate distance or by compartmentalization. All electrical equipment located in fuel tanks must be grounded.
6. Install electrical wiring well above and away from combustible fluid lines.
7. Shield wiring installations to prevent sparking during a crash. Electrical wiring should allow a minimum of 20% slack or permit the wires to elongate

20%, and wires should be attached with frangible clamps to allow for displacement during a crash sequence.

8. Insulate or cool high-temperature surfaces such as gas ducts to reduce the outside surface temperatures to a maximum of 400°F.

9. In areas of possible hydraulic fluid leakage, employ heat shields to provide local protection by reducing temperatures to below the flash point temperature of the fluid.

10. Insure that lines carrying combustible fluids are as direct as possible. Self-sealing, breakaway fittings and valves should be used. Lines should be fire-resistant with ample slack or elongation capability for relative movement.

11. Design and position fluid tanks and lines to withstand all survivable crash impacts. Protect flammable tanks and lines from battle damage. Prevent leaking fuel, oil, and hydraulic fluid from coming into contact with electrical equipment through the effect of gravity, airflow, or battle damage.

8-5.3 FIRE CONTAINMENT

The following fire containment features are applicable:

1. Engine compartments *shall* be isolated from occupiable areas by firewalls. The firewalls *shall* withstand a 2000°F flame for 15 min.

2. All fuel, oil, and hydraulic fluid supply lines passing through firewall bulkheads should incorporate shutoff valves located outside of potential fire areas.

3. Use of absorbent materials or materials considered a source of reignition should be avoided.

4. Consider using inerting systems and/or filler foam for fire suppression.

5. Post-crash fire prevention design should incorporate the features of Refs. 2 and 10.

8-5.4 FIRE DETECTION SYSTEMS

The regions surrounding the engine and APUs are fire zones. Therefore, reliable fire detection systems *shall* be provided. The fire detection and/or overtemperature warning system should:

1. Indicate a hazardous condition (either fire or an abnormally high temperature)

2. Remain on for the duration of the condition

3. Indicate when the condition is resolved

4. Indicate recurrence of the fire or the abnormal temperature condition

5. Operate satisfactorily throughout the flight envelope.

The basic types of fire detection systems include the infrared or surveillance detection system, continuous wire system, the spot or thermocouple system, and combinations of these. MIL-F-7872 contains installation requirements for fire/overtemperature warning systems, and MIL-F-23447 describes installation of radiation sensing fire warning systems.

For any aircraft, various methods of fire detection should be analyzed and the most compatible system selected. The fire detection system *shall* operate throughout the entire flight envelope. The infrared or surveillance detection system provides extensive fire zone coverage but it must be protected from random sunlight or the individual photoelectric cells must be shielded to prevent false alarms. Continuous wire detection systems are routed wherever temperature changes caused by fire are likely to occur. However, the continuous wire element is subject to vibration and maintenance damage. The spot or thermocouple type is more rugged inherently than the continuous wire detector but provides very limited coverage. Spot detectors in reasonable numbers can be employed only in limited-volume fire zones such as combustion heater compartments. MIL-D-27729 is the applicable specification for fire detectors.

8-5.5 FIRE EXTINGUISHING SYSTEMS

A positive, adequate fire extinguishing system *shall* be installed in each fire zone region. The fire extinguishing agent used should not be injurious to the primary structure of the aircraft and should exhibit low toxicity in both its natural condition and when pyrolized. Bromotrifluoromethane (CF_3Br) is one of the more effective agents; is the least toxic of the agents (except for CO_2) in the pyrolized condition; and is noncorrosive to aluminum, steel, and titanium. Design trade-offs should be performed to determine the types and amounts of fire extinguishing capability to be provided.

8-6 ENGINE-STARTING SYSTEMS

8-6.1 GENERAL

This paragraph describes the preliminary design and selection of an engine-starting system for turbine-engine powered helicopters. Because helicopters often are required to operate in remote areas where auxiliary power is not available, their effectiveness would be curtailed severely without a self-contained starting system. System weight is a prime consideration in the selection of engine-starting equipment. Because electrical power is required to operate other helicopter systems, this mode is attractive for use as the starting system in that multiple tasks then can be performed with a single system. However, cold weather starting capability usually is seriously reduced when the system must rely on batteries. The use of pneumatic and hydraulic systems for engine starting usually requires an onboard auxiliary power unit (APU). Ref. 11 gives an index of applicable starting system specifications and standards.

Auxiliary power units are covered in par. 8-6.4. This paragraph concentrates on the selection of battery system for engine starting. Ref. 12 documents a thorough study of starter and battery characteristics for gas turbine engine starting.

The importance of providing an adequate engine-starting system cannot be overemphasized. Whereas a reciprocating engine only needs to be "turned over" for starting, the gas turbine engine must be driven to a finite "lightoff" speed and then assisted in accelerating to a self-sustaining speed. Penalties of an inadequate system include an inability to start and overtemperature, or hot starts, resulting in decreased engine reliability and increased engine overhaul costs.

8-6.2 STARTING SYSTEM SELECTION

The starter selection procedure involves the matching of the battery and starter output torque characteristics to the engine starting torque requirements. Considerations must include the engine acceleration torque required to meet the "time-to-start" requirements over the required ambient temperature range.

Performance characteristics of each component must be determined, followed by analysis of the performance of combinations until the complete system is analyzed. The objective is to construct a performance curve showing starter torque output and engine-starting torque required, as shown in Fig. 8-17. Time-to-start calculations then can be made from the acceleration torque values to insure proper engine starting over the desired ambient temperature operating range.

8-6.2.1 Engine Parameters

Data for the following engine characteristics are presented in the engine model specification for the particular engine considered:

1. Engine drag torque curve (motoring power required) over the required temperature range, including engine lightoff, starter cutoff, and engine idle speed

2. Starter drive pad to gas generator gear ratio

3. Gas generator polar moment of inertia at starter drive pad

4. Maximum allowable starter torques (maximum static and maximum continuous) on drive pad

5. Effect of engine lubricating oil and fuel viscosity on starting torque requirements throughout the ambient temperature range

6. Auxiliary equipment loading not included in engine drag torque curves

7. Required engine acceleration times

8. Engine starter pad weight and overhung moment limits.

The engine torque versus starter drive speed given in the lower curve of Fig. 8-17 is typical of the presentation in an engine model specification. The engine curve can be divided into two segments. The first segment is from zero speed to engine lightoff (minimum firing) speed. The second segment is the lightoff range, which starts at the firing speed and extends to the idle speed. In this segment, the engine torque crosses zero and changes from a resistive torque to an assisting torque. Typically, the engine requirements are shown at two ambient temperatures, −65°F and 59°F; requirements

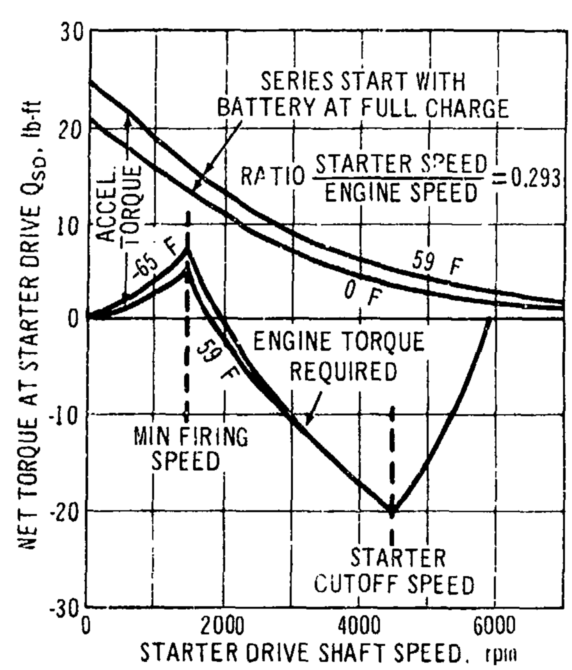


Fig. 8-17. Typical Starter Torque vs Speed Characteristic

for intermediate temperatures fall between the two curves.

It is important to note that the value of torque required at zero speed at −65°F may be much higher than that required at 59°F. This is due mainly to the increased drag caused by the increase in lubricating oil viscosity at −65°F. Restarting a warm engine, with warm oil, in −65°F ambient temperature conditions requires less motoring torque than does a cold start.

The torque requirement curves show steady-state torque values but not torque required to accelerate. Thus, to determine the total starter torque required for acceleration, data from time-to-start calculations must be included. MIL-E-8593 defines required engine ground starting time versus ambient air temperature. The following equation (Ref. 12) approximates the acceleration time for increasing the rotation speed from N_1 to N_2:

$$t = \frac{0.0065J(N_2 - N_1)}{(Q_{s_1} - Q_{e_1}) + (Q_{s_2} - Q_{e_2})}, \text{ sec} \quad (8\text{-}12)$$

where

t = acceleration time, sec

J = polar moment of inertia of the gas generator with respect to the starter drive pad, lb-ft^2

Q_{s_1}, Q_{s_2} = starter torque at points 1 and 2, lb-ft

Q_{e_1}, Q_{e_2} = engine torque at points 1 and 2, lb-ft

$(N_2 - N_1)$ = speed increment between points 1 and 2, rpm

0.0065 = conversion constant, $4\pi/(60g)$

Starter torque curves are provided by starter suppliers and trial calculations can be made to determine the acceleration times with candidate starters.

Eq. 8-12 must be used over several increments of engine speed. The discontinuity in the engine torque curve (Fig. 8-17) at lightoff speed is a limit to an increment, for example. Also, the change in the engine torque from resistive to accelerative must be accounted for by changing the sign of the denominator terms for the engine speed range above the speed at which the engine torque becomes additive.

8-6.2.2 Starter Parameters

The output torque of a DC starter at various values of current and voltage at the starter terminals can be obtained readily from a torque speed grid. The torque speed curve is compiled by selecting a series of voltage and current conditions, and testing to determine the starter speed and torque at these points. Connecting the common voltage points and the common current points results in a grid as shown on Fig. 8-18.

As an example, to determine the voltage and current required at the starter to produce 24 lb-ft of torque at 2000 rpm, locate the intersection of the respective torque and speed lines. At that intersection, read the current and voltage input required, which in this case would be 500 A and 22.5 V. The torque and speed data usually are determined at the starter drive pad. The torque and speed at the engine shaft will vary as a function of the starter-to-engine gear ratio.

With this general procedure, different starter characteristics, such as shunt start and series start, can be studied. The performance characteristics of the starter at various winding temperatures must be considered.

8-6.2.3 Self-contained Power Sources

For a self-contained starting system, when batteries are used to supply the power for a turbine engine starter system, the system batteries must deliver the required voltages and currents during all portions of the start cycle over the required temperature range. Nickel cadmium (NiCad) batteries are unequalled in the durability and performance capabilities required for a self-contained turbine starter system. NiCad batteries are rugged both chemically and mechanically and, in normal service, their life expectancy can be measured in years with thousands of charge and discharge cycles.

A NiCad battery has the ability to deliver a uniform voltage at normal discharge rates and also to deliver high current rates with a relatively small reduction of battery capacity. The NiCad battery also has relatively good performance at low temperatures. Power can be extracted at low rates from a NiCad battery at sub-zero temperatures, and the battery will accept charge at temperatures down to the freezing point of the electrolyte.

The high discharge rate capability of a NiCad battery falls off at sub-zero temperature mainly because of a rapid increase in internal resistance. For this reason engine starting from batteries at a temperature below 0°F is difficult. Sub-zero starting is possible but the batteries must be sized accordingly.

Battery discharge data usually are available from the battery manufacturer in the form of a family of constant current, total capacity discharge curves plotted as functions of terminal volts and ampere-hours at a given ambient temperature. Fig. 8-19 represents these characteristics of a typical NiCad battery.

This data presentation provides the relative relationship of the three needed parameters—voltage, current, and capacity (ampere-hours)—at a given temperature on one data sheet. Initial battery selection can be based upon these graphs.

The battery data most useful to the designer of the engine-starting system is a battery voltage versus current curve (Fig. 8-20).

The values of voltage at any desired current can be plotted directly on the starter-generator torque speed grids (Fig. 8-18) and compared with the engine drag torque curves. Every volt-ampere curve should be accompanied by a corresponding current versus time curve over the applicable start cycle and for a given value of the ambient temperature of the battery.

Temperature requirements are a major factor in sizing a battery. Available battery capacity falls off rapidly at both very high and low temperatures. A typical curve is shown in Fig. 8-21. If engine starts are to be performed at extreme temperatures, a correspondingly higher capacity battery must be used.

For airborne applications, the batteries often represent the heaviest component of a starting system, and there is a tendency to select an inadequate size in order

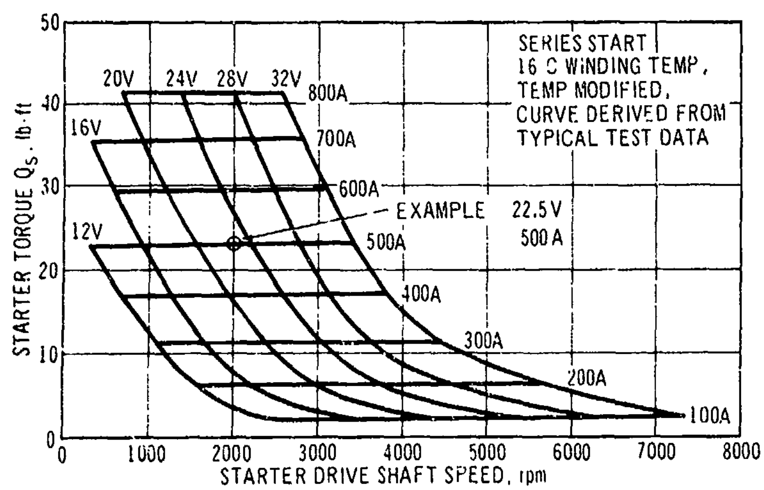


Fig. 8-18. Typical Starter Torque vs Speed

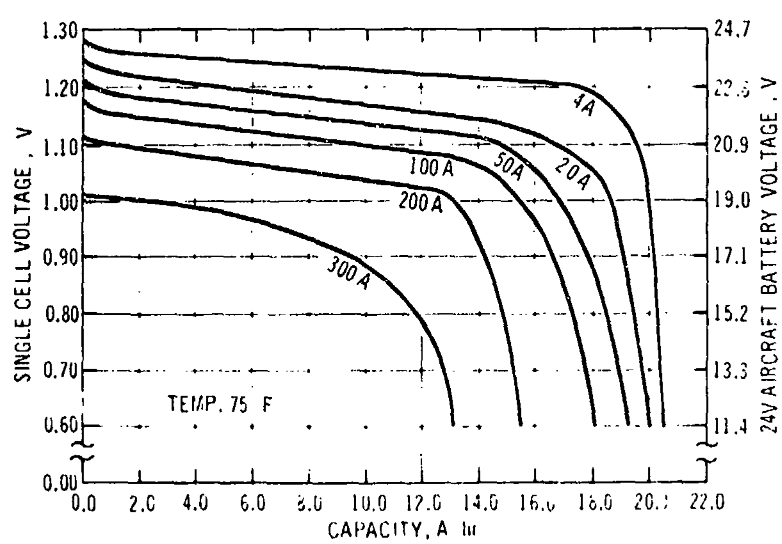


Fig. 8-19. Typical Battery Discharge Characteristics

to save weight. This increases the cost of operation of the starting system through both reduced performance over the required temperature range and reduced reliability of the starting system components, including the engine.

8-6.3 CARTRIDGE-BOOSTED ELECTRICAL STARTING SYSTEMS

At low ambient temperatures, the engine torque requirements increase, but battery energy output decreases. A cartridge boost system for the electrical starter is a possible solution. The objective is to provide an additional supply of energy that can supplement or boost the output of the electrical starting system.

The boost system consists of a gas motor mounted on the existing electrical starter, a cartridge breech, a manifold system that directs the hot gas from the breech to the boost motor, and a cartridge.

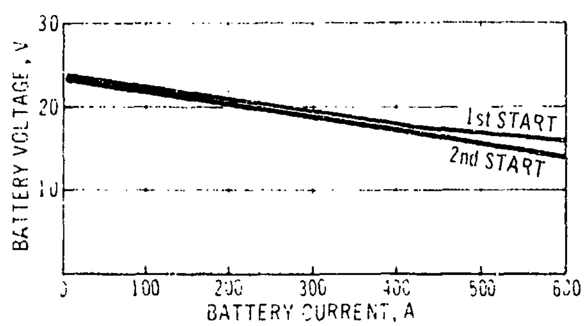


Fig. 8-20. Typical Battery Voltage vs Current

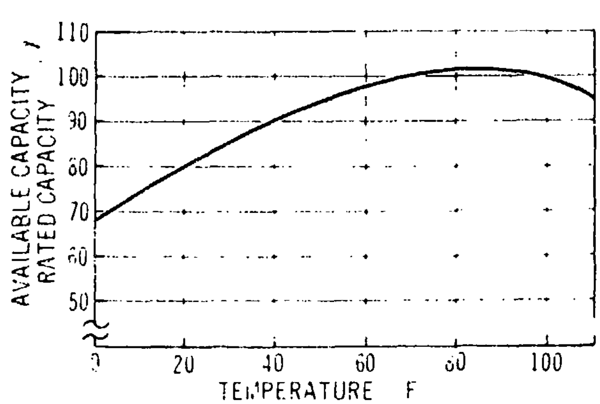


Fig. 8-21. Typical Relationship Between Temperature and Available Capacity for Batteries

The breech is a pressure vessel that houses the cartridge and acts as the gas accumulator. The breech should contain a throttle and a safety burst disk. The gas motor is connected to the aft end of the electrical starter shaft with an overrunning clutch. The gas motor rotates only when the cartridge is fired.

Typically, a cartridge boost start would be the first start of a cold day, and residual heat in the engine and the battery would enable the remaining starts of the day to be made with battery only.

8-6.4 AUXILIARY POWER UNIT (APU) INSTALLATIONS

This paragraph considers the uses and problems of auxiliary power units installed in helicopters. Discussion is limited to helicopter-mounted gas turbine engines conforming to the requirements of MIL-P-8686. Various trade-off considerations pertinent to APU installation and design selection are presented.

8-6.4.1 Major APU Types

The APU is categorized by the method of energy transfer, e.g., direct drive, shaft power, pneumatic. The APU is "auxiliary" in that it provides power in one of these forms, separate from the main engine, for use during some part of the helicopter operational envelope. However, it becomes a primary unit when the power is used to start the main engine, because APU operation then is essential to initiation of the helicopter mission. The APU can be started in a number of ways, including electrically, hydraulically, pneumatically, and manually.

8-6.4.1.1 Shaft Power

APUs of this type include all units used primarily as a source of mechanical power (power takeoff). Single-spool units are used to provide power to the APU reduction drive. In two-spool units (free turbine) the power turbine provides the output shaft power. The free turbine generally is capable of operating more efficiently through a greater range of speeds, and of providing higher power at low output shaft speed, than is the single-spool unit.

The shaft power available from the APU reduction drive output pad(s) is used by the helicopter designer in the forms considered optimum for the particular helicopter and mission requirements. These forms may be:

1 Electrical. Alternating current, direct current, and/or rectified alternating current for direct current uses.

2. Hydraulic. Pumps and pump/motors may be used in systems where power is varied by regulating either hydraulic fluid pressure or flow rate.

3. Pneumatic. Available envelope (size and shape) and pneumatic performance requirements may make it more desirable to use shaft power to drive a blower than to use APU bleed air. This would be likely if the helicopter air supply requirements were for pressure/-flow levels either higher or lower than those available as bleed from the APU compressor (even though air horsepower may be similar).

4. Direct drive. In some cases the helicopter can be designed to use APU power by direct mechanical drive. Main engine starting may be performed efficiently in this way if the APU is the two-spool type (to provide high torque at low main engine cranking speeds).

5. Combination. Weight and size can be optimized by considering combined energy transfer methods. A typical example would be to use direct drive main engine starting (where high power may be needed for short periods), combined with APU accessory drive pads for electrical and/or hydraulic outputs for preflight operational checks and preparations.

8-6.4.1.2 Compressed Air Power

Power may be obtained from some types of gas turbine APUs (usually single-spool) by bleeding air from the compressor discharge. This air will be at a pressure of a few atmospheres, with 400-600°F temperature, and with mass flow rates dependent upon the APU size and aircraft requirements.

8-6.4.1.3 Compressed Air Bleed Combined With Combustion Products

Considerable additional mass flow is available at the APU exhaust but at low pressure levels and high temperatures (typically 1000-1400°F). If this gas is mixed with compressor bleed air, some advantage may be obtained in lower temperatures and higher pressures, with higher mass flow. However, this procedure currently is not common practice.

8-6.4.2 Trade-off Considerations for APU Selection

Ordinarily, the decision to install an APU in a helicopter is based upon the requirements of assigned missions. Some significant trade-off factors are described subsequently, and are related to possible mission profile considerations.

8-6.4.2.1 Self-sufficiency

Because helicopters have become more complicated, the self-contained airborne starters formerly used have been replaced with larger, more complicated ground support equipment. Thus, the mission capabilities of many helicopters have been limited when operation from temporary bases or completely unimproved forward areas was desired.

The availability of comparatively lightweight APUs has resulted in their use to provide self-sufficiency for main engine starting under any expected operating condition or location. In some cases, cartridge starters have been used to provide self-sufficiency. Thus, cold day cartridge reliability will be considered along with potential multiple functions of the APU.

When helicopter availability must be maximized, redundant main engine starting is considered. This may be accomplished with the pneumatic APU system by cross-bleed ducting in multiengine helicopters or by simple valving to use ground support air supply equipment. Redundancy also can be arranged with other APU types. In such cases, it is necessary to design the APU for rapid removal and replacement. No APU trouble-shooting would be performed; helicopter malfunction indicators need only confirm impending or actual APU failure and a replacement APU can be installed in a few minutes.

8-6.4.2.2 Ground Support Functions

The operational helicopter mission profile *shall* include definition of all required functions before takeoff and before main engine starting. Ground checks of items such as flight control systems, avionic equipment, electronic countermeasure (ECM) equipment, and failure warning systems easily can be performed with power supplied by an APU. The hydraulic, electrical, or pneumatic power needed for these checks also can be supplied more economically by an APU than by the main engine(s). This results from the lower fuel consumption of the APU (hence lower cost and logistic support time) and the reduced net operating time on the main engine(s). The increased flight time between main engine overhauls also will increase availability and reduce maintenance costs.

8-6.4.2.3 Flight or Alternate Support Functions

Other functions involving ground or flight support may be needed. Ground maintenance functions often require many hours of APU time, and the APU may compile three to five (or more) times the main engine operating hours. Air-conditioning for personnel or specialized equipment may be needed for all helicopter operating modes. The APU can provide these functions on the ground economically, and use of the APU for these support functions in flight takes some burden off the main engine. Furthermore, with the APU operating in flight, the probability of a successful main engine restart is increased, although inflight APU starting can be provided when needed.

8-6.4.2.4 Weight Optimization

Optimum system weight *shall* be the objective. The system will include the APU, APU starting components, APU-driven equipment, and helicopter system components for energy transfer (such as ducting, wiring, fuel lines, oil lines, and drain lines). Other factors, such as the total weight of the APU fuel used during a typical flight, also *shall* be considered.

Once the helicopter mission profile is established, trade-offs can be examined to choose the optimum weight system. APU size and weight often are determined by the maximum power needed for main engine starting. Other functions performed by an APU probably will require less power, but for longer periods of time. A possible exception is the air-conditioning system, which usually requires total APU power on hot days.

8-6.4.2.4.1 Electrical Energy Transfer

Main engine starting often is accomplished electrically. As the power requirements increase, electrical equipment becomes large. Electrical main engine starting has a weight advantage at power levels below 15-25 hp. Future APU designs may show weight advantages for electrical power levels above this if the generating equipment can be driven at APU turbine speed, thereby reducing both APU reduction drive and alternator weight.

For electrical energy transfer systems, the lightest weight APU starting probably will be electrical, especially if a starter-generator is employed. Batteries can start the APU and also provide start sequencing system power.

8-6.4.2.4.2 Hydraulic Energy Transfer

At APU power levels above 25 hp, energy transfer may be performed more efficiently by using hydraulics. However, this hinges largely on helicopter system needs. Integration with other systems can make APU hydraulics attractive. A suitable APU-driven pump/motor will start the APU, recharge the APU hydraulic starting system, and provide the main engine starting power. The hydraulic energy usually will be used by a motor mounted to drive the main engine reduction drive, or an auxiliary (combining) gearbox. Hydraulic APU starting requires accumulators, valves, lines, recharge system, etc.; but some of these may serve dual functions when other helicopter systems are hydraulic.

Generally, higher pump/motor speeds mean lower weight for both the pump/motor and the APU reduction drive. Higher pressure also means more energy transfer per unit weight. Pressure levels of 3000-4000 psi commonly are used in helicopter control systems and APU energy transfer.

8-6.4.2.4.3 Pneumatic Energy Transfer

Energy transfer by pneumatics (compressor bleed or shaft power-driven compressors) can show a weight advantage when air-conditioning or equipment cooling requirements are large compared with main engine-starting power required. The hydraulic system has higher efficiency of energy transfer but, if large system ducting is needed for air conditioning, overall weight may be saved by using a completely pneumatic APU. APU starting could be electrical or hydraulic, depending on available helicopter components.

8-6.4.2.4.4 Direct Drive Energy Transfer

When power is supplied directly through a reduction drive system, energy transmission efficiency is very high. When a direct drive from the APU is used for main engine starting, the most efficient design would use a two-spool (free turbine) APU. Single-spool engines may be used but a clutch is required to engage the drive shaft at near rated speed (85-95%) into an auxiliary gearbox. A constant-speed drive can be integrated into this helicopter gearbox to start the main engine(s), or a hydraulic pump may be driven to absorb APU power. Direct drive from the two-spool APU to the gearbox eliminates other system components and should be evaluated. With optimum APU location, it can provide the lightest weight package.

8-6.4.2.4.5 Combination Energy Transfer

Almost every helicopter will require some hydraulic and/or electrical power in addition to, and not necessarily concurrent with, the main engine starting requirement. Therefore, the APU should be designed to answer all of these needs. The optimum system will be established by other system requirements. The APU's power per pound may vary with the energy transfer methods, but the overriding effect may be the weight of the complementary helicopter system components.

8-6.4.2.5 Space Optimization

In many cases, the APU can be placed in space otherwise unused. Minor compromises or rearrangement of other system components may be required to provide space for APU inlet, exhaust, and cooling ducting. These penalties must be examined before the APU is chosen.

The shape of available space may dictate the APU configuration. For example, if a pneumatic APU is needed, the compressor air bleed APU has a larger diameter but shorter length for a given output power than a shaft power APU driving a compressor.

APU reduction drives are available to fit driven equipment into the available space. For combination requirements, side pads, axial pads, and "piggyback" pads can be used for fitting the space.

Maximum APU power may be a function of available space. Therefore, it may be desirable to limit APU power requirements so that no space, weight, or other helicopter compromises are needed. This could occur when the APU power level is established by air-conditioning or accessory needs and not by main engine starting requirements.

8-6.4.2.6 APU Performance

Trade-offs often exist between APU specific horsepower and specific fuel consumption. Some APU designs have high power-to-weight ratios, but not very favorable power-to-fuel-consumption ratios. High power-to-weight ratios are satisfactory where operation is primarily on the ground or when the weight or space penalties of APU fuel are not significant. If the APU must operate during extended flight missions, low SFC will show an advantage in lower overall system weight.

8-6.4.3 APU/Helicopter Interface Considerations

Interface relationships between the helicopter and APU must be examined in the preliminary design phase to assess the impact of APU installation. The same considerations are applicable to an APU installation as are pertinent for the installation of propulsion engines. The discussion of these considerations in par. 8-2 also is applicable to APU installations and therefore is not repeated here. Similarly, the discussions of fuel and oil systems in par. 8-4 and fire protection in par. 8-5 also apply to the APU.

8-7 ENGINE/AIRFRAME INTEGRATED CONTROL SYSTEM

8-7.1 GENERAL

The interaction of the helicopter rotor, engine, and propulsion control system requires careful attention if a good or even workable propulsion control system is to be achieved. The flight system requirements reviewed in the subsequent paragraphs are followed by an investigation into the means by which these requirements can be met.

8-7.2 ROTOR SPEED REQUIREMENTS

Lift generated by a helicopter rotor system depends in part upon the speed and collective pitch of the rotor, either or both of which can be varied by the pilot. Because large rotor inertia cannot be accelerated or decelerated rapidly, quick changes in lift needed during helicopter flight are accomplished best by rapid changes in collective pitch. This leads to the primary consideration for the system, i.e., a constant helicopter rotor speed, with the pilot controlling lift by controlling the collective pitch angle.

Optimum helicopter rotor speed depends upon forward speed, altitude, aircraft weight, etc. High rotor speeds usually are required at high forward speeds to avoid retreating blade stall. When landing or maneuvering near obstacles, high rotor speed or high rotor inertia is desired as protection against engine failure or wind gusts, whereas low rotor speed is preferred at cruise altitude to reduce noise, vibration, and sometimes to increase efficiency of the engine and rotor system. However, these operational rotor speeds rarely differ by more than 10%.

Once the pilot selects a particular rotor speed, the control system should maintain this speed accurately unless it is changed by the pilot. Steady-state accuracy reduces monitoring by the pilot and facilitates failure detection.

8-7.3 POWER MANAGEMENT REQUIREMENTS

An additional requirement for multiengine helicopter operation is power management, i.e., the selection of power from one or all engines. An inherent advantage of the multiengine helicopter is its failed-engine capability. Prompt recovery of power following an engine failure is vital. Therefore, the engine load must be shared such that acceleration to full power on each engine is achieved in approximately the same time. This can be achieved by matching torques, speeds, or temperatures. However, the parameter that is used should be matched precisely, thereby leaving the pilot no doubt as to correct operation of the system. Matching of torque is a distinct advantage if the transmission or the engine is torque-limited.

8-7.4 ENGINE LIMIT CONTROL REQUIREMENTS

Engine limit control is the most important function of the engine control system. The engine control not only controls engine acceleration and deceleration but also protects against overspeed, overtemperature, and, therefore, against overtorque. These functions must override the requirements of the main rotor system. However, the responsibility for these protective functions lies with the engine designer and is not discussed further in this handbook.

8-7.5 INTEGRATED CONTROL REQUIREMENTS

A manually operated collective pitch control lever that also contained a twist grip throttle for coordinating the speed requirements of the rotor was installed in early helicopters. This system permits rapid selection of collective pitch angle and enables the pilot to adjust the speed of the rotor separately.

Such a system, however, demands considerable concentration and dexterity on the part of the pilot. Manual control is increasingly complex with free power turbine engine installations, and use of an automatic engine output shaft speed governor is essential.

8-7.5.1 Proportional Control System

The engine output shaft speed governor most commonly used is a proportional hydromechanical type. In this system, engine output shaft speed is sensed by flyweights located in the engine control system. Opening of the flyweights against a spring reduces fuel flow in proportion to the speed error (Fig. 8-22). The force on the spring is then, in essence, the desired or selected engine output shaft speed and is varied manually by the pilot. With an increasing load on the main rotor, a proportional control has a droop in steady-state speed (Fig. 8-23). As the gain of the engine output shaft speed governor becomes higher, the amount of main rotor speed droop decreases. The stability requirements that usually dictate the upper gain limits are complicated by the flexibility of the main rotor shaft system and by the dynamics of a free power turbine engine.

8-7.5.2 Rotor System Response

When a turbine engine is connected to a helicopter rotor system, the resulting combination may be unstable. Careful analysis of the engine, power control and drive system early in the design stage is necessary. Close cooperation between the designer of the drive system and the engine manufacturer is required.

Investigations of this type usually are conducted by combining the transfer functions for the engine and power control system as defined by the engine manufacturer with the transfer functions for the drive systems. A frequency response analysis provides a means of evaluating both the low frequency phase margin and the torsional gain margin.

With respect to torsional oscillation, helicopter rotor systems can be categorized into two major classes. The first of these is the rigid rotor. In most cases the complex spring-mass system can be represented as the lumped parameter system as shown in Fig. 8-24. The transfer function of such a system has a typical frequency response as shown in the Bode plot of Fig. 8-25. The resonant peak occurs at the rotor system natural frequency, which typically occurs in the range of 3-6 Hz. It is this resonant peak that must be attenuated by the engine-control system if torsional oscillation is to be avoided.

The second class of rotor system is the articulated type. This type of system is characterized by a lag hinge in the plane of rotation of the rotor. Lag hinge dampers are placed across the hinge. Although these dampers have a significant effect on the rotor system resonance, their dynamic characteristics usually are defined by other requirements of the rotor system. The articulated rotor system can be represented as shown in Fig. 8-26. The transfer function involves more terms than does that for the rigid rotor. However, the typical Bode plot is of essentially similar form. The articulated rotor with lag hinge dampers does not exhibit as sharp a peak in the region of rotor resonance as does the rigid rotor.

A special case of the articulated rotor system uses a preloaded lag hinge damper. When the amplitude of the torsional oscillation between hub and blade is less than that permitted by the preload setting of check valves in the damper, the damper behaves as a spring. The spring effect is produced by the compressibility of the hydraulic fluid. The effect of this lag hinge damper characteristic is to make the articulated rotor behave more like a rigid system.

The information required to analyze the rotor system dynamics is defined in Ref. 13. This information is to be supplied to the engine manufacturer by the airframe manufacturer in order that the dynamic characteristics of the engine and its control system may be selected so that the torsional oscillations are minimized.

The foregoing description of the analytical techniques has been somewhat simplified. Significant factors have been neglected in the interest of clarity.

One of the neglected factors is the tail rotor. The tail rotor in some cases may be the critical element in the system and also may have a significant effect on that natural frequency which is dependent principally on the stiffness of the main rotor shaft. This effect of the tail rotor, therefore, must be considered. The derivation of the system transfer functions, including the tail rotor, is similar to that for the more simplified system.

Another consideration is the multiple engine installation. For such installations the analysis should consider both the single engine and the multiple engine cases. The multiple engine case can be studied by considering that the engines are identical and thus can be treated as a single engine having power turbine inertia and torque output equal to the sum of these parameters for all the engines.

8-7.5.3 Engine and Rotor Damping

Damping of the helicopter rotor B_R or of the power turbine B_P is defined as the rate of change of torque with respect to speed (Fig. 8-27). Whenever possible, the value of these derivatives should be derived from performance data for each of the components in question. If the required performance data are not available, reasonable estimates can be made.

The torque required to drive a helicopter rotor at a fixed collective pitch may be assumed to be a function of the square of the rotational speed. An approximation of the rotor damping derivative, therefore, can be calculated for any collective pitch from the power absorbed and the helicopter rotor speed.

The torque of a free-shaft power turbine, like that of any impulse machine, is linear with speed. The stalled torque is approximately twice the optimum torque, and runaway speed is twice the optimum speed. The damping can be calculated from the optimum torque and optimum speed (Fig. 8-27). The value of the power turbine damping will be approximately one-half the

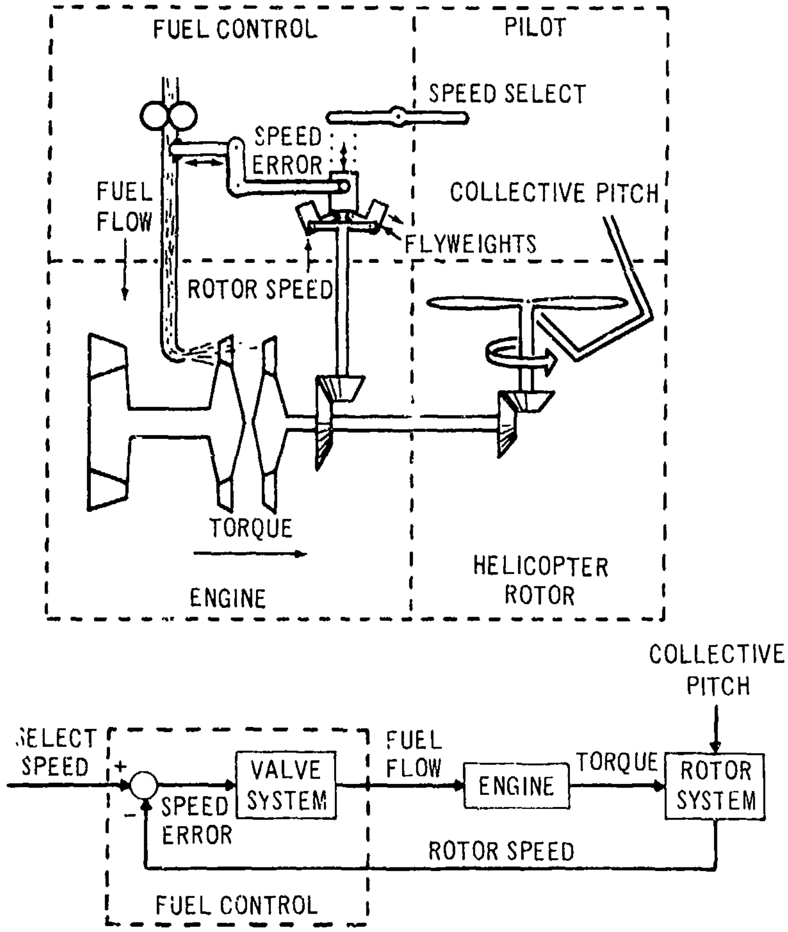


Fig. 8-22. Proportional Control System

value of the rotor damping and both are positive in that they reduce instability.

8-7.6 ENGINE RESPONSE

The gas generator portion of a single-spool gas turbine engine operates at a constant speed and, therefore, can respond almost immediately to a request for power change. In these engines only a very short lag exists as the combustor flame propagates; then full power is supplied.

The gas generator portion of a free-shaft turbine engine does not operate at constant speed and must change speed prior to completing any power change. However, part of the power is available as soon as fuel is burned. This is called the "fast path" and is one-quarter to one-half of the total power change. Linear mathematical models of fixed-shaft and free-shaft turbines are shown in Fig. 8-28.

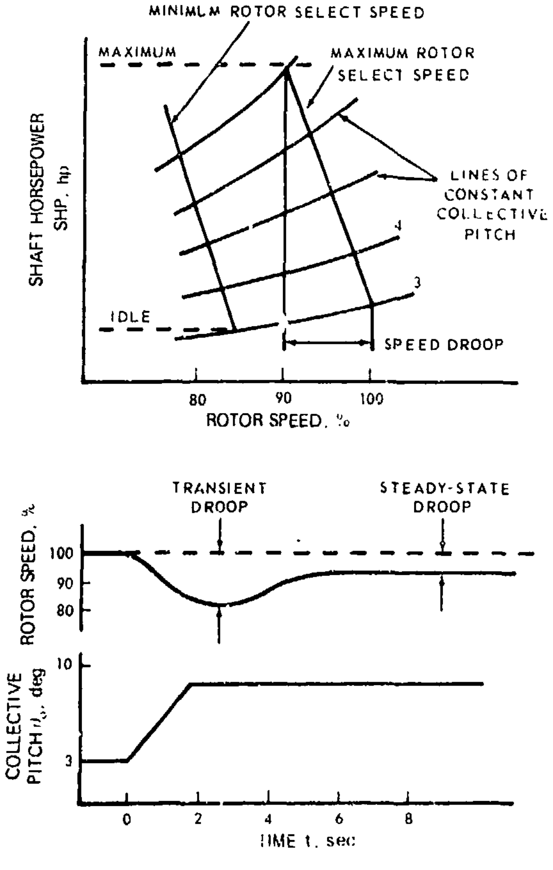


Fig. 8-23. Droop of a Proportional Control System

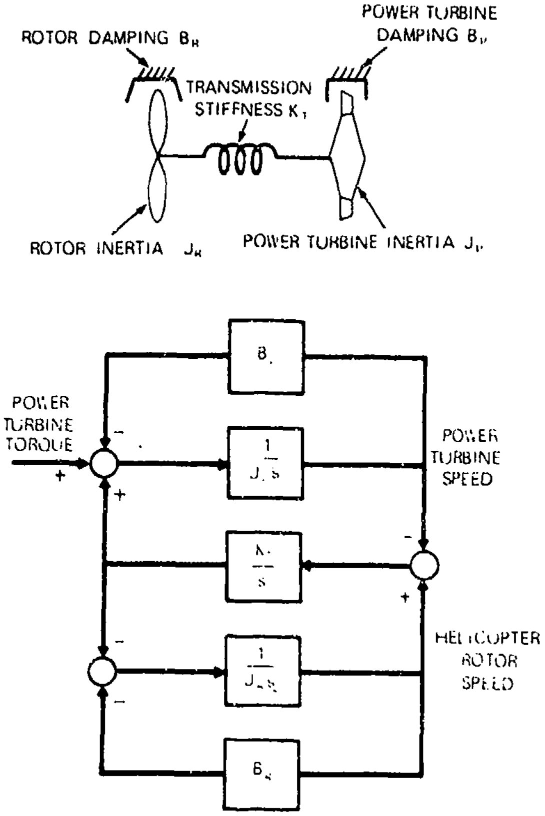


Fig. 8-24. Mathematical Model of a Two-mass Rotor Representation

To best maintain constant rotor speed, a proportional governor should maintain the open-loop gain constant at all conditions.

$$\underbrace{\left[\frac{N_R}{N_e}\right]}_{\text{open loop gain}} = \underbrace{\left[\frac{W_f}{N_e}\right]}_{\text{fuel control gain}} \underbrace{\left[\frac{Q}{W_f}\right]}_{\text{engine gain}} \underbrace{\left[\frac{N_R}{Q}\right]}_{\text{rotor gain}} = \text{constant} \qquad (8\text{-}13)$$

where

N_e = power turbine speed error signal (set speed minus actual speed), rpm
N_R = main rotor speed, rpm
Q = engine output torque, lb-ft
W_f = fuel flow rate, lb/hr

The engine torque-to-fuel flow gain is independent of altitude, but the rotor gain is proportional to air density. The fuel control gain, therefore, should be in-

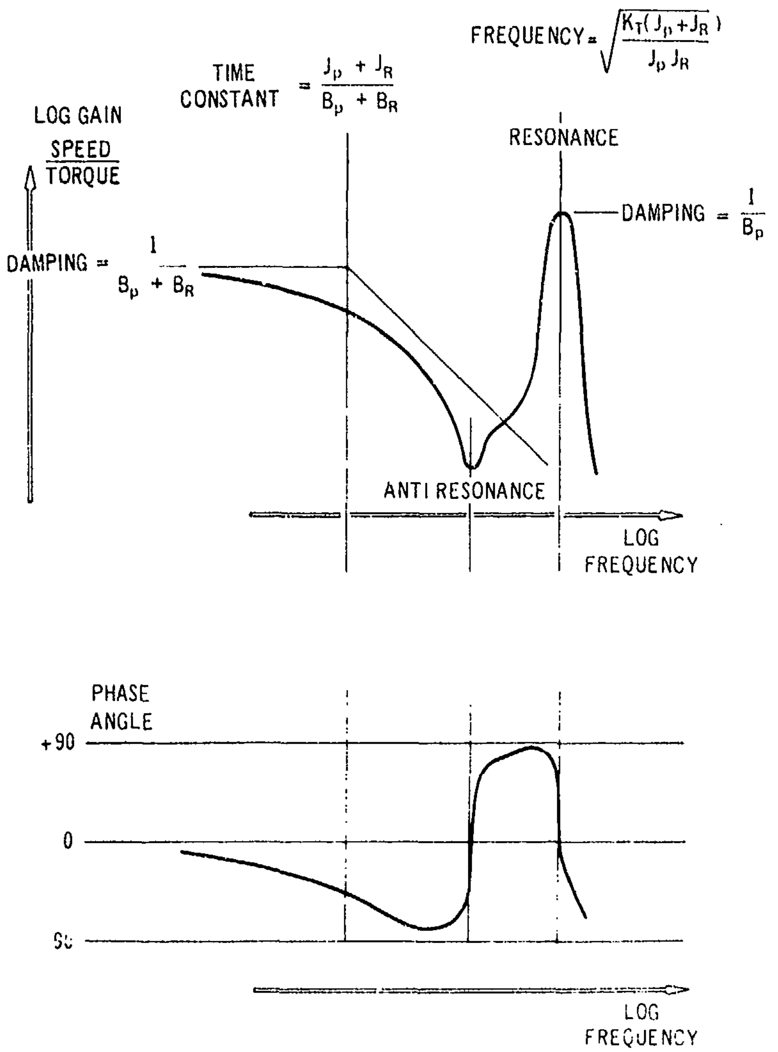
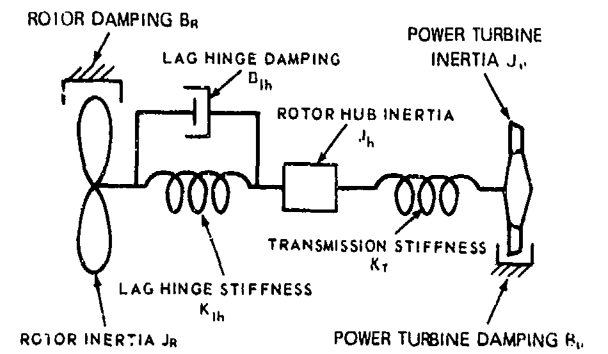


Fig. 8-25. Bode Plot of a Two-mass Rotor System

versely proportional to air density. This is achieved by scheduling W_f/P_{t2} or W_f/P_{t3} with speed error. These parameters also are the most common functions used to protect the gas generator. A W_f/P_t control will approach the stability limit at high power and the W_f/P_{t2} at low power.

A W_f/P_{t3} system has a secondary effect in reducing the transient response of the free-shaft turbine engine—full fuel flow cannot be achieved until P_t has been increased and transmitted to the fuel control. Consequently, the lag due to the free-shaft turbine is increased, and the first break point on a Bode plot of the control and engine combination is reduced as seen in Fig. 8-29. This may be an advantage if a relatively undamped resonance at 3-5 Hz requires a gain reduction. However, in most helicopters, the system dynamics do not require such severe attenuation. The effect of the lag of a free-shaft turbine can be reduced by using reset governing. A reset governor does not change fuel flow directly but resets the gas generator governor to a new desired value. As the gas generator approaches this value, the fuel flow change is reduced. The rate of change of fuel flow with gas generator speed can be varied to suit the application, insuring an adequate stability margin at the first crossover frequency, and attenuating any rotor resonance. Experience has shown that most helicopter systems with W_f/P_{t3} control can use reset governing to achieve faster response.



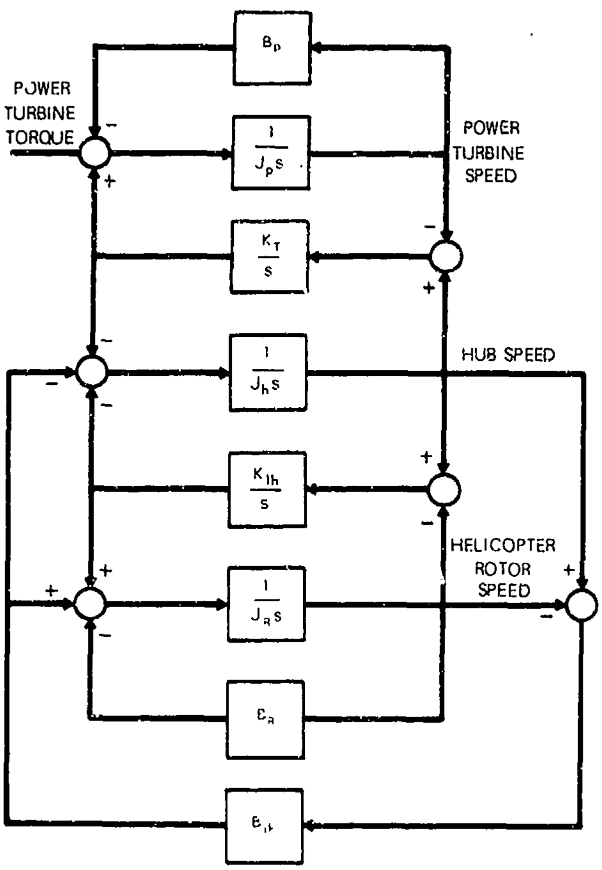


Fig. 8-26. Mathematical Model of a Three-mass System

8-7.7 COLLECTIVE PITCH SIGNAL

When the dynamics of the rotor, engine, and engine control have been studied and a proportional engine control optimized, the response still will not be ideal. There will be excessive transient and steady-state droop that will be unacceptable to the pilot. Additional techniques will be required to obtain a satisfactory control system.

The most successful approach is to monitor the collective pitch. A collective pitch signal, whether of electrical or mechanical linkage, can be used to change the engine output shaft speed governor setting in anticipation of any load (collective pitch) change. Mechanical linkage that connects the collective pitch to the engine output shaft or gas generator speed governor select lever is satisfactory. The pilot's speed trim can be accomplished by an electrical motor-driven link. This system is cumbersome on large helicopters and cannot be relied upon to eliminate completely the rotor speed droop. Altitude and forward speed effects limit the steady-state speed droop reduction to 90% of the uncompensated speed droop.

On large helicopters, the collective pitch usually is monitored electrically, and the governor speed setting is modified according to the derivative of the signal or a straight line approximation of the correct reset. Such a system has to respond quickly if any reduction in transient droop is to be achieved.

8-7.7.1 Variable Gain Devices

The gain of a governor need not be constant and can be designed to vary with engine power or, more commonly, with size of the speed error. Large errors—such as might occur during a jump takeoff that requires high gains—produce large fuel flow changes. Nonlinearities, such as the acceleration limit, prevent any large oscillations. As the system nears the steady-state point, the gain is reduced to prevent any small oscillations that otherwise would occur. An overlapped valve is an example of such a device in a hydromechanical control.

8-7.7.2 Slow Time Constant Governors

If rotor resonance is the predominant stability limitation, the governor can be slowed down to attenuate the speed signal at high frequency. Excessive change in this direction may reduce the stability margin at the first crossover frequency, especially during autorotation. However, experience has shown that the stability margin during autorotation actually is in excess of predictions.

8-7.7.3 Isochronous Controls

An isochronous or integrating control maintains constant steady-state speed with no speed droop. The zero speed droop characteristic is achieved by measuring the speed error and integrating this to give the

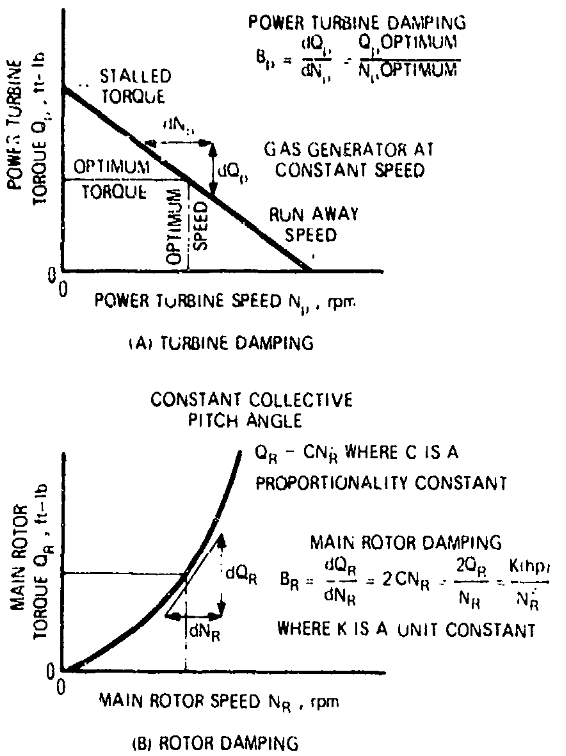


Fig. 8-27. Turbine and Rotor Damping

correcting fuel flow. Thus, the isochronous control only stops changing fuel flow when the speed error is zero. Similar calculations, such as those for a proportional control, will reveal that a simple integrating system responds too slowly to a helicopter rotor pitch change and that a proportional plus integral control is required. The authority of the integral path may have to be limited, or some clamping device incorporated, to prevent overshoots when the control is transferred to the integral governor from the gas generator speed or acceleration limiter.

8-7.7.4 Load-sharing

Load-sharing in a multiengine helicopter equipped with an isochronous governor requires special attention.

Proportional controls inherently will match the loads to a degree of accuracy that depends upon the droop (Fig. 8-30). An isochronous governor will load one engine completely before calling for power on the remaining engine. This condition can be overcome in several ways. A simple means of reducing this one-engine overload is to use only one isochronous governor operating on a proportional control on each engine (Fig. 8-31). The speed signal for the governor ideally should be rotor speed, thus reducing the effect of rotor resonance. However, such a system leads to an additional control unit and does not eliminate the load mismatch.

Failure of such a single isochronous governor can shut down both engines; therefore, authority-limiting is necessary. The proportional governors can be set close to the correct value with knowledge of the collective pitch and pilot's selected rotor speed, leaving the isochronous governor as a limited trim on this signal.

The more common approach to load sharing is to measure engine torque and to readjust the input signal to the separate governors, as shown in Fig. 8-32. An "up only" signal in this torque-matching path prevents the failure of one engine from reducing the power of the remaining engines.

8-7.8 NONLINEAR ANALYSIS

A linear evaluation of the gain and stability of the system is the fastest and most direct means of analysis, but it does not give an accurate picture of the transient response to large changes in collective pitch. A nonlinear analysis, using a computer with the capability of function generation, should be completed as soon as practicable and should include, in order of priority:

1. Lag-hinge damper check valve characteristics and effective spring constant

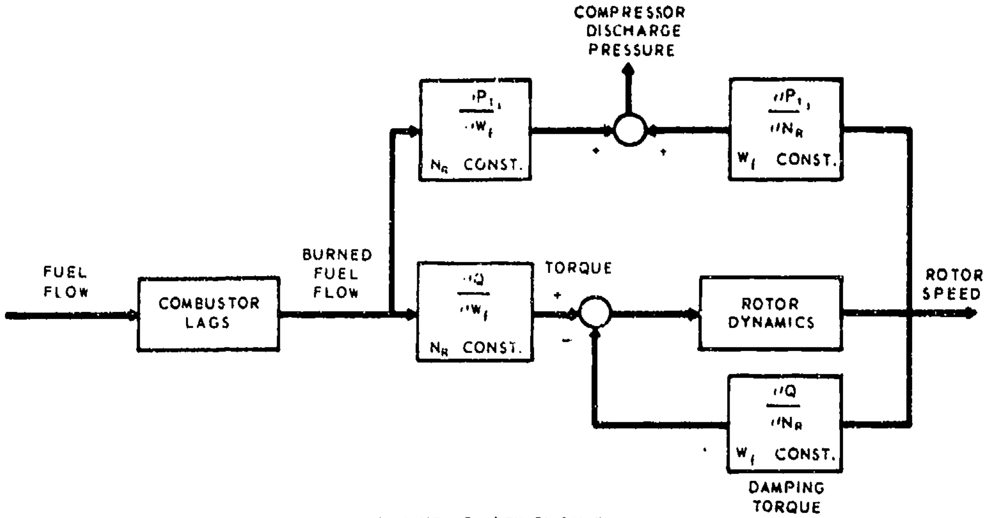
2. Engine acceleration and deceleration limits

3. Selection of maximum or minimum mechanisms in the fuel control, authority limits, control integrator clamps, and servo slew rates.

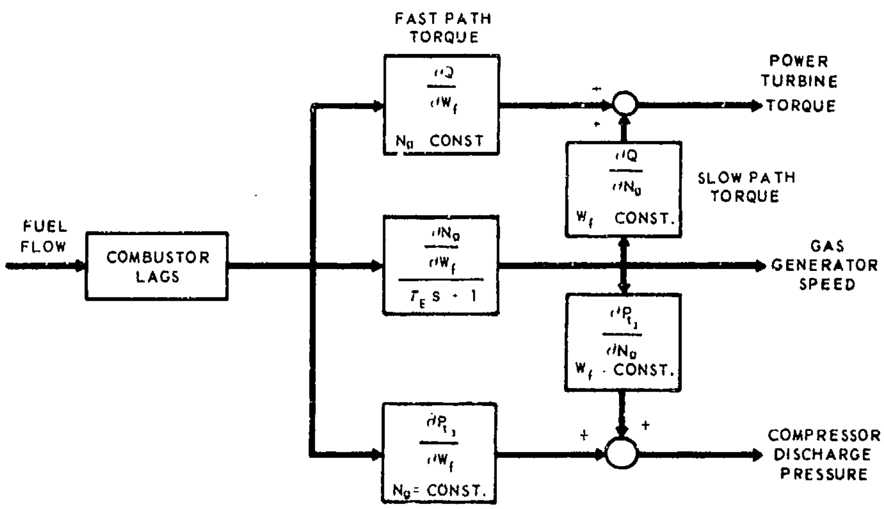
To simulate a helicopter jump takeoff and to evaluate rotor speed droop, a 2-sec ramp change in collective pitch is considered to be maximum. However, to investigate stability, a step change in collective pitch, although unrealistic, gives more definitive results.

8-7.9 EMERGENCY AND AUTHORITY LIMITATIONS

When automatic rotor speed governors first were introduced, it was thought expeditious to limit their authority by making them a limited trim under pilot's control. Experience led to acceptance of the f authority hydromechanical control. Emergency override has been deleted in all but some single-engine helicopters. However, authority must be limited whenever multiple engines are connected for load-sharing.



(A) FIXED SHAFT ENGINE



(B) GAS GENERATOR OF A FREE TURBINE ENGINE

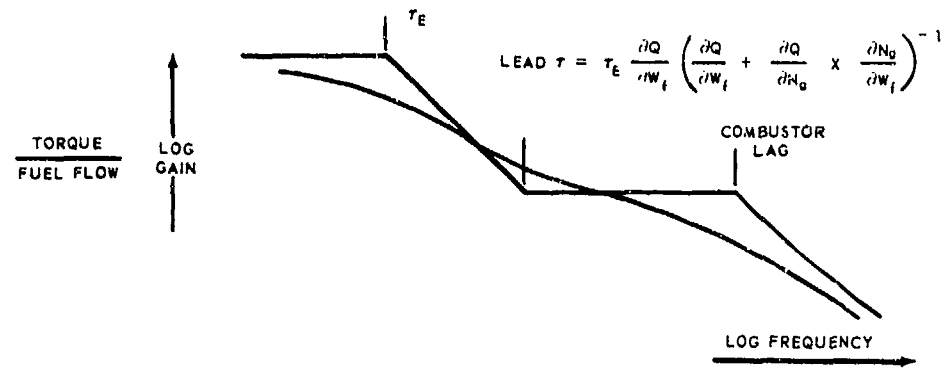
Fig. 8-28. Linear Model of Gas Turbine Engines

8-7.10 ENGINE STARTING AND POWER CONTROL LEVER SYSTEMS

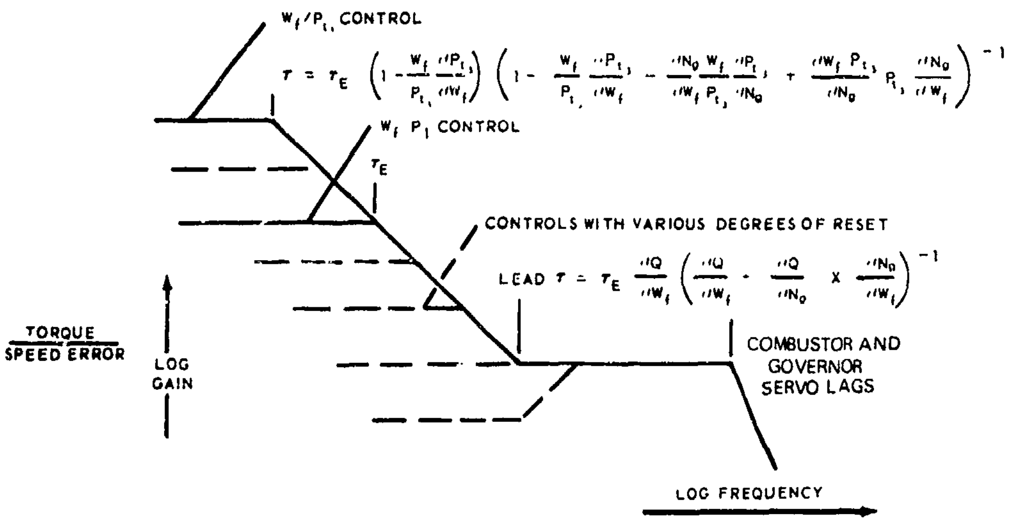
The helicopter engine must be equipped with means for starting, idling, and shutoff without bringing power up to that point required to govern the rotor. Usually, the engine is started with the gas generator governor set to ground idle. When the gas generator governor selector is moved to the maximum or "fly" position, the power is increased until the power turbine speed governor attains control. The transition from ground idle to maximum speed is best left under manual control of the pilot, who can monitor torque and other parameters during this acceleration and can use this form of control in case of a failure of the power turbine governor. A control that leaves the pilot no selection between gas generator ground idle and maximum has been found to be unsatisfactory. The gas generator governor input is usually a mechanical lever on the fuel control that requires mechanical or electrical connection with the cockpit. The lever torque requirements on the control are specified in MIL-E-8593.

The power turbine governor on a hydromechanical control has a similar lever that is controlled from the pilot's trim and the collective pitch motion, although on some systems the collective pitch is introduced separately to the control.

The pilot's speed trim usually is an electrical motor actuator that is controlled from a button on the pilot's collective pitch lever. Future controls may be fully electrical. The pilot's desired speed then would be accomplished by simple rotation of a potentiometer, or similar device, in the cockpit. As such, an electrical system would be isochronous and require no trimming by the pilot once a speed had been selected. Thus, adjustment on the collective pitch stick probably would be unnecessary.



(A) FREE TURBINE TORQUE RESPONSE



(B) FREE TURBINE AND CONTROL TORQUE RESPONSE

Fig. 8-29. Control and Engine Bode Plot

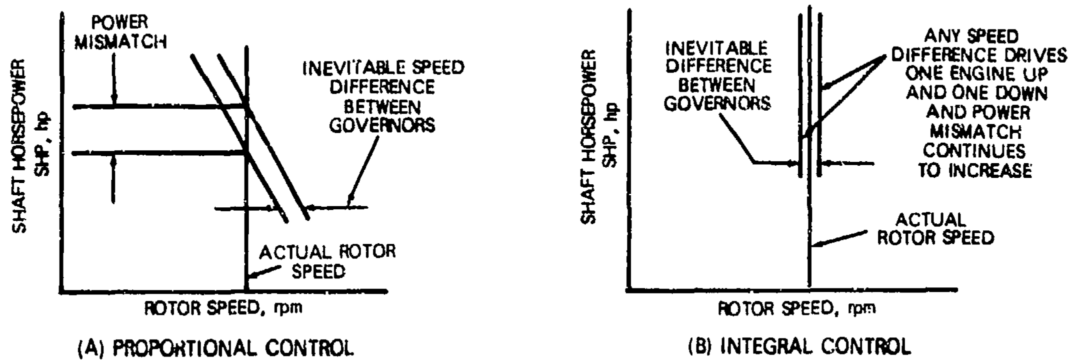


Fig. 8-30. Inherent Load-matching With Proportional Control

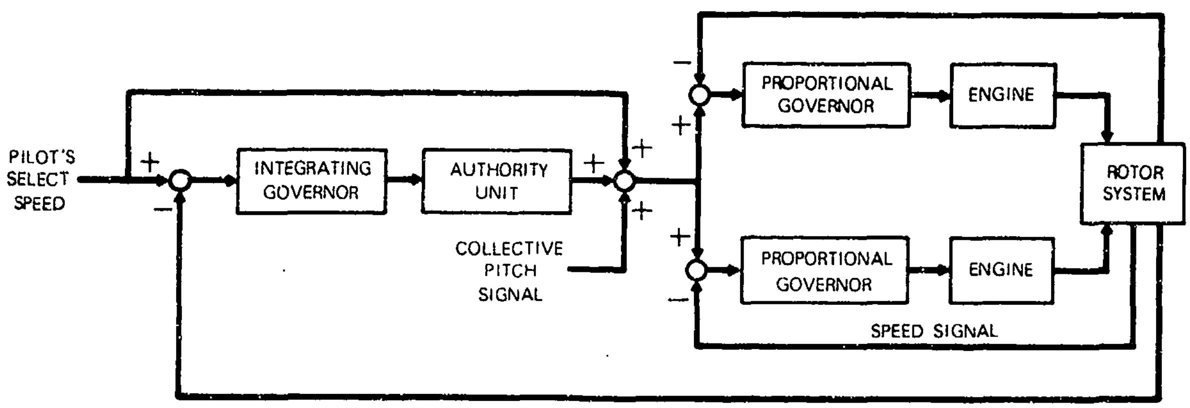


Fig. 8-31. Proportional Plus Integral Control System

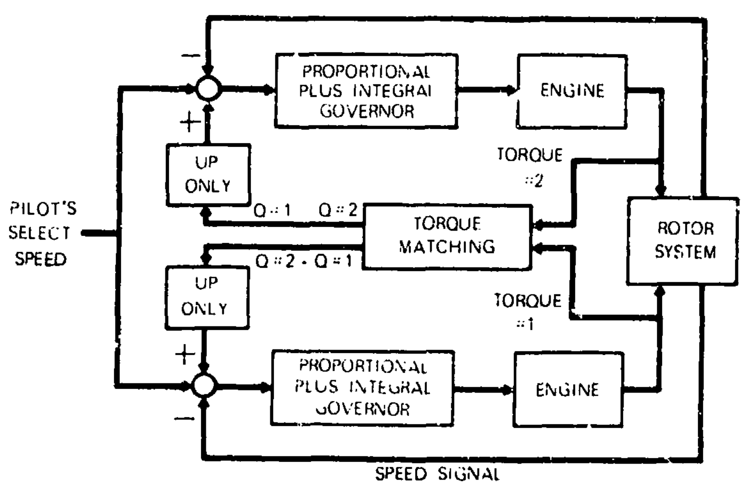


Fig. 8-32. Proportional Plus Integral Control With Load Matching

8-8 LIST OF SYMBOLS

A = area, ft^2
C_{cd} = critical damping coefficient, dimensionless
C_d = damping coefficient, dimensionless
c_p = specific heat, Btu/lb-°F
D_h = hydraulic diameter, ft
e = radial distance to the lag hinge, ft
f = friction factor, dimensionless
= forcing frequency, Hz
f_n = undamped natural frequency, Hz
hp_f = fan output power, hp
hp_{MR} = main rotor power, hp
hp_{TR} = tail rotor power, hp
J = polar moment of inertia of the engine with respect to the starter drive pad, lb-ft^2
k = loss factor, dimensionless
L = duct length, ft
l_t = length from CL of main rotor to CL of tail rotor, ft
M = blade mass, slug
M_{TR} = tail rotor Figure of Merit
N = drive speed, rpm
N_e = power turbine speed error signal (set speed minus actual speed), rpm
N_R = main rotor speed, rpm
N_S = specific speed,
$(N_2 - N_1)$ = speed increment between Points 1 and 2, rpm
P_t = total pressure, lb/ft^2
$P_{t1\,max}$ = maximum total pressure (absolute) at engine inlet face, psi
P_{t1} = mean total pressure (absolute) at engine inlet face, psi
Q = main rotor torque, lb-ft
= volumetric flow rate, cfm
= engine output torque, lb-ft
Q_{s_1}, Q_{s_2} = starter torque at Points 1 and 2, lb-ft
Q_{e_1}, Q_{e_2} = engine torque at Points 1 and 2, lb-ft
q = $\rho V^2/2$ = dynamic pressure, lb/ft^2
= heat transfer rate, Btu/hr
R = radial distance to the blade CG, ft
R_{TR} = tail rotor radius, ft
T_{TR} = tail rotor thrust, lb
T_{in}, T_{out} = fluid terminal temperature, °F
T_c = temperature of cold fluid, °F
T_h = temperature of hot fluid, °F
t = acceleration time, sec
U = overall conductance between fluids in heat exchanger, Btu/hr-ft^2-°F
V = air velocity, fps
W = fluid flow rate, lb/hr
W_a = weight flow of air into the engine, lb/sec
W_b = bleed air flow rate, lb/sec
W_f = fuel flow rate, lb/hr
ΔT_m = mean value of terminal temperature differences, °F
ΔP_f = pressure loss due to friction, lb/ft^2
ΔP_k = pressure loss due to bends etc., lb/ft^2
ΔP_s = system pressure drop, psi
ΔP_t = total pressure loss, lb/ft^2
δ = ratio ambient pressure to standard sea level pressure, dimensionless
η = efficiency, dimensionless
θ = ratio ambient temperature °R to standard sea level temperature °R, dimensionless
ρ = air density, lb/ft^3 or slug/ft^3 as required
ρ_o = air density standard at sea level, lb/ft^3 or slug/ft^3 as required
σ = ρ/ρ_o = ambient air density/air density at sea level, 59°F, dimensionless
τ_e = engine time constant, sec
Ω_{MR} = main rotor rotational speed, rad/sec

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CHAPTER 9

SECONDARY POWER SYSTEMS

9-1 INTRODUCTION

This chapter describes the considerations involved in selecting the secondary power systems of a helicopter. The secondary power systems are classified as electrical, hydraulic, pneumatic, and fluidic. In addition, alternate power sources and environmental control systems also are included in this chapter.

An electrical system is required in any helicopter. Communications and lighting in particular cannot be powered by other means. However, the power requirements of some subsystems may be met in any number of ways. For example, a particular utility function conceivably could be performed by hydraulic, pneumatic, electromechanical, manual, or even pyrotechnic means. Therefore, a system approach to the entire matter of secondary power selection is required. Analysis may show that one or more of the initial possibilities is inappropriate. The choice among the remaining possibilities must be based upon a trade-off study that includes not only the particular utility function under consideration, but also all other secondary power functions. In this way, an overall optimum combination of secondary power systems is defined and incorporated in the helicopter.

For example, hydraulic power clearly may be optimum for some applications. Therefore, the selection of a pneumatic system for some other application, even though it is the optimum choice for that single function, should not be made without thorough analysis, including consideration of the overall additional weight, complexity, and cost of the new power source for a single application.

The system and subsystem trade-off studies are necessary because each type of secondary power system has a unique set of advantages and disadvantages whose relative importance depends upon the specific application and upon the complex set of mission requirements and performance, cost, and weight objectives and constraints. Optimization among these objectives is becoming more demanding as vehicle performance capability advances.

Hydraulic systems find their widest application in helicopter flight control functions. Hydraulic systems also are used for utility applications, such as hoists, landing gear retraction, braking, and gun turret motion. Temperature is the primary limiting factor for hydraulic systems.

Pneumatic systems can operate at high temperatures (above 500°F) where commonly used hydraulic fluids deteriorate. Pnuematic systems are limited primarily by the poor lubricity, low fluid viscosity, and low bulk modulus of the fluid medium. On the other hand, hydraulic fluids have good lubricity, acceptable fluid viscosity, and a high bulk modulus.

Fluidics is a relatively new technology that permits many low-power functions currently performed by electronics to be accomplished by fluids. The same fluid also can supply the power necessary for subsystem operation. Fluidic system attributes include ruggedness with respect to shock, vibration, acceleration, and temperature; radiation hardness; and high reliability. However, fluidic systems have not demonstrated accuracy superior to that of conventional electronic equipment. For applications not requiring high accuracy, rapid response, or high linearity, fluidic systems appear increasingly competitive, with potential savings or improvements in developmental cost, production cost, and simplicity.

The choice of secondary power systems must include consideration of the state of environmental control system (ECS) technology. ECSs may have significant impact upon helicopter power requirements. Economic benefits ordinarily can be derived through a degree of commonality in subsystem power generation and distribution equipment.

Rapid advances in the state-of-the-art and in the diversity of helicopter missions must be considered in compiling a list of trade-offs involved in the selection of secondary power systems. However, in view of the inherent advantages and limitations of alternative secondary power means, it is apparent that the primary trade-offs should include at least the following:

1. Utility application (hoists, gun drives, rotor

braking, wheel braking and steering, fluid dampers, doors and loading ramps, and landing gear retractors). Hydraulic, manual, pneumatic, pyrotechnic, and electromechanical alternatives should be considered.

2. Flight control. Direct mechanical, hydraulic boost, and hydraulic servosystems have been used. When stability augmentation or other automated functions are to be added to the system, consideration also should be given to the use of fluidic or electrical (fly-by-wire) systems to provide the inputs to a final hydraulic actuator.

3. Fuel management. Fluidic and conventional systems compete for this function.

4. Starters. These may be pneumatic, hydraulic, or electrical.

9-2 SUBSYSTEM SELECTION

9-2.1 GENERAL

Many factors must enter into the selection of a secondary power system. Although the mission requirements and the specifications impose constraints and thereby limit the range of alternative solutions, the number of possibilities for a given application remains very large. The helicopter designer must attempt to define an optimum power generation subsystem over a range of conflicting objectives: safety, reliability, weight, cost, maintainability, and performance. These parameters are discussed in more detail in the paragraphs that follow.

9-2.1.1 Maintainability and Reliability Considerations

Maintainability is a characteristic of design and installation that must be considered in system engineering and in determining the effectiveness of the system. In the process of optimizing system design, maintainability is a prime factor and must be integrated with other design criteria in the proper relationships.

Maintainability design is that characteristic of equipment design that contributes to the rapidity, economy, ease, and accuracy of maintenance. It provides features and functions that simplify the task of keeping an equipment in its specified operating condition or restoring it to that condition in the environment in which it will be used. To insure optimum maintainability of equipment, the designer first must be aware of established maintenance methods and practices so that designed maintenance features will be consistent with the procedures in effect. In addition, the designer should know the conditions under which the maintenance technician will work. Only then can he provide equipment maintenance features that will offset adverse effects of the specific environment. Finally, the designer should be aware of the qualifications and limitations of the technicians who will maintain the equipment so that he may keep maintenance methods and test procedures within the capabilities of available personnel.

Maintainability is a design parameter that must be specified and measured, or demonstrated. Maintainability has certain characteristics, both quantitative and qualitative, that make it possible to meet mission requirements with minimum expenditures of maintenance effort and resources under the environmental conditions in which scheduled and unscheduled maintenance will be performed. Quantitative characteristics refer to required turnaround time, reaction time, operational ready rate, utilization rate, etc. Qualitative characteristics are the result of design excellence affecting manpower and skill requirements, test and calibration requirements, and compatibility with aerospace ground equipment and facilities. Maintainability requirements are discussed in detail in Chapter 11.

The reliability of a system is the relative success with which it performs a required task under all conditions of operation. Quantitatively, secondary system reliability is the probability that the system will perform a given task for a given time under given conditions. Reliability requirements and criteria are discussed in Chapter 12.

Maintenance and reliability go hand in hand. The better the reliability, the fewer will be the corrective maintenance tasks encountered for a given component. When preventive maintenance is considered, increased component reliability usually results in fewer inspections. Fewer maintenance tasks and inspections mean fewer maintenance manhours and, hence, lower operating costs

9-2.1.2 Safety

The quality of safety relates to the ability of the equipment to allow the helicopter to perform a given mission without damage to the crew or vehicle. The degree of safety, of course, should be consistent throughout the helicopter. The quantification of safety is discussed in Chapter 3, AMCP 706-203. In addition to hazards associated with, or resulting from, system operation during the conduct of normal missions, careful consideration *shall* be given to crash safety. Ref. 1 provides valuable guidance regarding secondary power system design and installation features essential to minimizing the risk of post-crash fires. It is important, for example, to reduce the opportunities for the electrical system to provide an ignition source following a crash. Detail considerations include possible failure modes of the hydraulic system that might provide a source of flammable fluids, as well as electrical system characteristics that might lead to ignition of such fluid or other flammables.

9-2.2 PERFORMANCE CALCULATIONS

Analysis and selection of a suitable helicopter subsystem require calculation of the penalty imposed by that subsystem upon vehicle performance. Of the several penalty calculation methods available, the most convenient is the takeoff weight penalty method.

The process of analysis and selection of an optimum helicopter subsystem requires a comparison among the various choices available. The comparative evaluation can be more meaningful if an overall helicopter penalty basis is used in order to learn the effects of the subsystem upon the helicopter performance characteristics. Factors affecting helicopter flight performance include weight, drag, and power extraction. These factors can be translated into penalty parameters such as changes in takeoff weight, range, payload, and rate of climb. By so doing, it is possible to assess the relative magnitude of the effect any given system will have upon the helicopter performance and thereby to select the system with the least penalty.

The process described herein allows preliminary calculation of penalties with an accuracy limited only by the assumptions of constant lift/drag (L/D) and specific fuel consumption (SFC). (The effect upon helicopter structural weight—i.e., airframe without equipment—due to the addition of equipment weight is assumed to be negligible.) By taking smaller time segments, some of the nonsteady-state flight conditions can be evaluated. It also should be remembered that the system offering the least takeoff weight or operating penalty is not necessarily the best one for the helicopter. Factors such as reliability, maintainability, and development risk must be considered; sometimes these factors will override selection of a system based upon performances.

The addition of a secondary power system to the helicopter adds weight and requires power. The power extraction can be in the form of shaft horsepower or of bleed air from the turbine engine. Additional power may be required to counteract drag due to protrusions into the airstream, such as inlets or the use of ram air for cooling. Two methods of penalty evaluation will be considered: (1) The takeoff weight penalty method, and (2) the Breguet range equation method, with main emphasis on the former.

9-2.2.1 Takeoff Weight Penalty Method

The total takeoff weight penalty can be estimated by considering the overall flight mission in segments, during each of which the L/D ratio and the engine SFC can be assumed to be relatively constant. For each of these flight segments, the penalties due to fixed system weight, the weight of the expendable medium, ram airflow, bleed airflow, and shaft horsepower can be determined in terms of required takeoff fuel weight. The sum of the penalties for each segment represents the takeoff weight penalty for the whole mission as a result of the subsystem design being evaluated. The accuracy of results can be increased by taking smaller time segments consistent with objectives of the analysis.

The following data are required prior to any penalty evaluation:

1. Specific mission profile including data on altitude, speed, and range or duration
2. Bleed and ram airflow data corresponding to the mission profile
3. Lift-to-drag ratio at the start of each major phase of the mission being considered
4. Specific fuel consumption of the engine.

9-2.2.1.1 Fixed Weight Penalty

The total takeoff weight penalty per pound of fixed weight is given by:

$$\frac{W_{fo} + W_F}{W_F} = \exp\left[\frac{V(SFC)t_m}{325.64L/D}\right], \text{ d'less} \qquad (9\text{-}1)$$

where

W_{fo} = takeoff fuel weight required to carry fixed or variable weight, lb
W_F = fixed weight of system, lb
V = cruise velocity, kt
t_m = mission time, hr

Penalty evaluation charts based upon this equation can be calculated. Because the takeoff weight factor given by Eq. 9-1 is for a unit fixed weight, it includes the fixed weight as well as fuel required to carry the fixed weight. It also includes fuel required to carry the extra fuel.

9-2.2.1.2 Variable Weight Penalty

The variable weight penalty is for cases where an expendable medium is being used up at a constant rate during the mission segment. The relationship between the variable weight W_v and its fuel requirement W_{fo} is:

$$W_{fo} = w_v t_m \left\{ \left[\frac{325.64L/D}{V(SFC)t_m}\right] \times \left[\exp\left(\frac{V(SFC)t_m}{325.64L/D}\right) - 1\right] - 1 \right\}, \text{lb} \tag{9-2}$$

where

w_v = rate of consumption of variable weight W_v/t_m, lb/hr

It should be kept in mind that the expendable medium weight imposes a fixed weight penalty up until the time its consumption starts. It is assumed that the expendable medium is consumed completely at the end of the mission time period being considered. If this is not the case, the remaining unused expendable medium is carried as fixed weight for the next segment of the mission. The weight of fuel required to carry fuel is included in the penalty factor. The total takeoff weight per pound of variable weight is:

$$\frac{W_{fo} + w_v t_m}{w_v t_m} = \left[\frac{325.64L/D}{V(SFC)t_m}\right] \times \left[\exp\left(\frac{V(SFC)t_m}{325.64L/D}\right) - 1\right] \tag{9-3}$$

9-2.2.1.3 Ram Air Drag Penalty

If complete momentum loss is assumed, the ram air drag penalty in terms of takeoff fuel weight is:

$$W_{fo} = \left[\frac{1.689 w_r V(L/D)}{g}\right] \times \left[\exp\left(\frac{V(SFC)t_m}{325.64L/D}\right) - 1\right], \text{lb} \tag{9-4}$$

where

w_r = ram airflow rate, lb/sec
g = acceleration due to gravity, ft/sec^2

The weight of fuel at the beginning of the mission must be carried as fixed weight during the initial mission segments.

9-2.2.1.4 Bleed Air Penalty

If bleed air is extracted from the last stage of the compressor of a turbine engine, the penalty can be evaluated on the basis of increase in fuel flow required to maintain constant power. As a first approximation only, where actual engine data are not available, the increase in fuel flow rate Δw_f due to bleed air extraction can be estimated by

$$\Delta w_f = 0.0335\left(\frac{T_t}{2000}\right) w_b \quad, \text{lb/hr} \tag{9-5}$$

where

w_b = bleed airflow rate, lb/hr
T_t = turbine inlet temperature, °R

For more accurate estimates of bleed air penalty, it is necessary to obtain performance data for the specific engine being considered and the compressor stage that is being bled.

The bleed air penalty in terms of takeoff weight also can be expressed as

$$W_{fo} = \left[\frac{0.005455 w_b (L/D) T_t}{V(SFC)}\right] \times \left[\exp\left(\frac{V(SFC)t_m}{325.64L/D}\right) - 1\right], \text{lb} \tag{9-6}$$

Because Eq. 9-6 is based on the empirical relation given by Eq. 9-5, its use is subject to the same limitation as Eq. 9-5. The penalty for carrying bleed air penalty fuel as an expendable medium is obtainable from Eq. 9-2. However, in most instances, the bleed air fuel penalty factors are available for the engine being considered. Actual engine penalty factors should be used when available, even though they be preliminary, rather than the first approximation method of Eqs. 9-5 and 9-6. The variation of penalty for successively higher stages of the engine compressor being bled cannot be approximated with the Eq. 9-5 method

9-2.2.1.5 Shaft Horsepower Extraction Penalty

To maintain constant net rotor power from the engine, the shaft power extracted must be compensated for by an increase in fuel rate to the engine. If it is assumed that power is consumed at a constant rate during the portion of the mission being evaluated, the takeoff weight penalty is given by:

$$W_{fo} = \left[\frac{325.64(L/D)hp_e}{V}\right] \times \left[\exp\left(\frac{V(SFC)t_m}{325.64L/D}\right) - 1\right] \text{, lb} \quad (9\text{-}7)$$

where

hp_e = power extracted, hp

9-2.2.2 Breguet Range Equation

The range R_A of a turbine-powered helicopter can be expressed by the Breguet equation

$$R_A = \frac{325.64L/D}{SFC} \ln\left(\frac{W_{fo} + W_F}{W_F}\right) \text{, n mi} \quad (9\text{-}8)$$

Eq. 9-8 can be used to determine the effect of a secondary power system upon the range, payload, or gross weight of the helicopter. By taking partial derivatives of the range equation with respect to pertinent helicopter performance parameters as independent variables, the effect of small changes in the parameters permits the calculation of approximate values of the increase in fuel weight.

9-3 ELECTRICAL SYSTEMS

9-3.1 GENERAL

Helicopter electrical power systems may be based upon such electrical power sources as:

1. AC or DC generators driven by:
   a. Main rotor power transmission system
   b. Engine accessory drives
   c. Constant-speed drives (speed controlled by hydraulic or mechanical torque converter)
   d. Constant-speed turbines (speed controlled by air or gas turbines)
   e. Hydraulic motor
2. Inverters
3. Transformer-rectifiers
4. Batteries.

Components of the power system must meet the requirements of their detail specifications, and the overall system must provide the electrical power characteristics specified in MIL-STD-704.

An analysis of helicopter electrical power requirements reveals that 28 V DC power is required for both normal and emergency operation of items such as engine fuel pumps, panel lighting, most avionic equipment, and electrically driven weapons. Constant-frequency, 400-Hz AC power is necessary for some avionic equipment. Although variable-frequency AC power is not mandatory, it can be used for some heating (deicing) and frequency-insensitive loads. To satisfy these requirements, the basic helicopter electrical system can be DC, variable-frequency AC (vf AC), or constant-frequency AC (cf AC).

The trend of new helicopter design is toward increased electrical power as well as increased amounts of constant-frequency power. Many helicopter propulsion system designs will provide the generating system with an input speed to the generator that varies by more than ± 5%.

Engine starting is a major factor to consider when beginning electrical system preliminary design. If the engine maximum starting torque is less than 90 lb-ft, a 400-A, 28 V DC starter-generator powered by two CA-5 or CA-9 nickel-cadmium batteries will provide the simplest self-contained start system. The 90 lb-ft maximum limit defines a small engine, thus a small helicopter.

9-3.1.1 Electrical Load Analysis

Design of the helicopter electrical power system is dependent upon completion of an electrical load analysis. Both the method and format of this analysis *shall* be in accordance with MIL-E-7016. All electrical load components *shall* be identified, and the operating cycles of each tabulated, based upon projected mission profiles. Electrical power requirements then are calculated throughout all anticipated missions. The electrical power sources, both DC and AC, then can be identified. A required portion of the summary of the load analysis is the identification of the minimum excess capacity of each system and the conditions under which it occurs. Each power source or piece of equipment that produces, converts, or transforms electric power, together with its distribution system and applicable loads, constitutes a system for purposes of this analysis. Minimum acceptable excess capacities will be defined by the procuring activity in the pertinent detail specification. Excess capacity is provided in order to assure that electrical power will be available to supply future equipment.

9-3.1.2 DC Systems

The quality of the DC power provided to the load equipment *shall* comply with MIL-STD-704. To insure this compliance, the generating system *shall* comply with MIL-G-6162. Any requirements peculiar to a specific helicopter *shall* be described in the pertinent specification.

Most DC generators in use today (par. 9-3.2) are equipped with brushes and commutator. The brushes and commutator are the primary life-limiting items, followed closely by grease-packed bearings. Research has proven the feasibility of brushless DC starter-generators, which will have additional service life due to elimination of the brushes and commutator; it also will be possible to oil-cool the machine and to oil-lubricate the bearings, further improving reliability. The majority of voltage regulators used are carbon pile devices. Semiconductor regulators have been developed and should be considered. Replacement of rotary inverters with semiconductor converters should also be considered.

9-3.1.3 AC Systems

The AC power provided to the load equipment *shall* comply with MIL-STD-704. To insure this compliance, the AC generating system *shall* comply with MIL-G-21480. Any special requirements peculiar to a specific helicopter *shall* be described in the pertinent specification. Refer to par. 9-3.3 for a further discussion of AC systems.

9-3.1.3.1 Constant-speed Drive System

If the engine or the rotor gearbox speed varies less than ±5%, an AC generator connected directly to the gearbox will satisfy the frequency requirements of MIL-STD-704. If the speed variation exceeds ±5% and the allowable variation of the AC frequency is less than that, either an inverter or a frequency conversion device is necessary.

The most commonly used device is the constant-speed drive (CSD), a hydromechanical, variable-ratio transmission. The CSD will convert the variable engine speed to a constant speed that is optimum for the generator, usually 12,000 rpm. The CSD also will provide cooling and lubricating oil to the generator, and will scavenge it. The oil-cooling capability is valuable for helicopters because it eliminates the necessity of pumping cooling air to the generator. It also prevents foreign material carried by cooling air from eroding generator winding insulation.

9-3.1.3.2 Air Turbine System

A second method of rotating the generator at constant speed when the engine speed variation is greater than ±5% is to bleed air from the engine compressor to spin a turbine wheel at constant speed. The turbine is geared to the generator. However, bleed air is a less efficient power extraction method than is a power takeoff from the engine shaft. For example, one shaft horsepower has the overall effect of adding 0.79 lb to the empty weight of the helicopter while one pneumatic horsepower adds approximately 4.1 lb. Also, making bleed air pressure available at idle engine rpm requires an oversize turbine. For these reasons, an air turbine drive seldom is selected as a generator drive.

9-3.1.3.3 Controlled-speed Hydraulic Motor System

A third method of rotating the generator at constant speed is to use a controlled-speed hydraulic motor (CSM). This method is less efficient than the hydromechanical device because all the power must flow through a hydraulic pump rather than a motor. The CSM is a practical approach if the electrical and hydraulic loads coordinate or if hydraulic starting is used.

9-3.1.3.4 Cycloconverter System

A fourth method of converting variable engine speed to constant frequency is to generate a variable frequency with a high-speed generator whose minimum frequency is 1200 Hz. This variable frequency power then is converted to constant frequency with a cycloconverter. Technical feasibility of this approach has been proven. The desirability of this concept in terms of weight and cost is being established by current research and development efforts.

9-3.1.3.5 High-voltage DC System

A fifth approach is to generate 115 V DC. This method precludes use of brushless AC motors. Brush-type DC motors currently are heavier, more expensive, and much less reliable. Research and development work on the high-voltage DC system is continuing, however.

9-3.2 DC ELECTRICAL SYSTEMS

9-3.2.1 System Description

The most common electrical system in current helicopters is the DC system shown in simplified form in Fig. 9-1.

Primary power is generated by a DC generator driven by the main rotor power transmission. An inverter, operating from the DC bus, provides constant-frequency AC for avionics. Switches or circuit breakers (such as CB_1, CB_2, CB_3, and CB_4) are required to isolate components when failures occur, to apply power to the proper load buses, and to switch in standby components when necessary. In the actual helicopter installation, these switches or circuit breakers can vary considerably depending upon applications; however, this

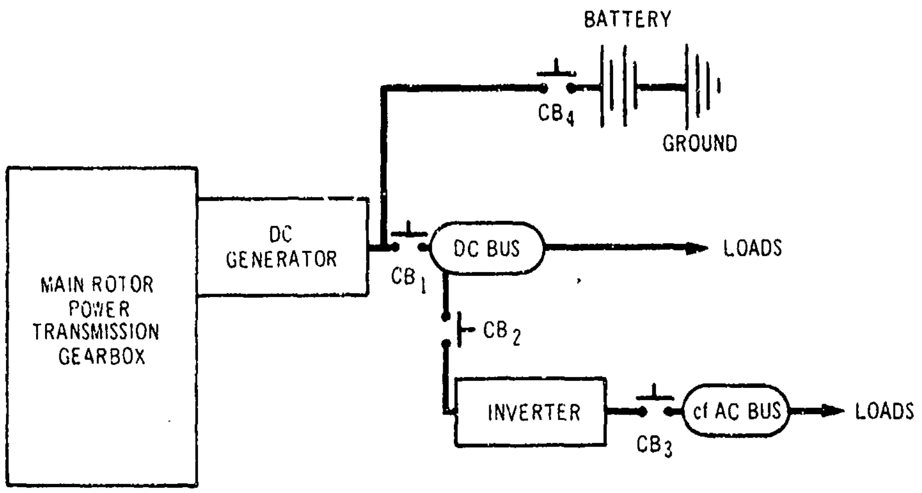

Fig. 9-1. Typical Direct Current System

description will consider a minimum number of switches and will assign a standard weight that will apply to all systems.

The DC electrical system is attractive especially for helicopters employing electrical engine starting because the DC starter-generator—when sized for starting—provides an abundance of DC power during the generating mode of operation. Therefore, low-efficiency inverters normally do not compromise the electrical system. Secondly, the initial cost of inverters is relatively noticeable in the smaller helicopter.

There are certain characteristics of the DC system that make it less desirable as electrical loads are increased:

1. It requires heavy feeders (wires and cables) due to the low voltage being transmitted. As power levels increase, feeder weight can become objectionably high.

2. It requires inverters, which are notably low in efficiency (60%). As cf AC loads increase, it may become necessary to size the DC generator for generating rather than for starting.

3. System weight increases rapidly with an increase in load requirements. The largest helicopter inverter presently available is a 5-kVA unit and, therefore, growth potential is limited. For higher AC load requirements than are practicable with inverters, direct-driven vf AC generators can be added for certain loads. This results in a more complex distribution system, because three types of power must be distributed.

9-3.2.2 Weight Analysis

Weight analyses will be developed considering two load requirements and various distributions of types of power required. First, system weight will be estimated for a total electrical load per generating channel of 5 kVA, with the cf AC portion of this load being variously 0, 1, 2.5, and 5 kVA. In the second case the total load per generating channel is 10 kVA, with the constant-frequency AC portion being 0, 2.5, 5, and 10 kVA.

At 0 kVA cf AC, meaning all transmitted power is DC, the weight of the 5 kVA per channel system is 134.6 lb and the 10 kVA per channel system is 143.6 lb. Table 9-1 tabulates the DC system weight for the two different levels of electrical load.

A weight pattern will develop in favor of AC systems as the system size increases. Curves of component weights for higher-rated system components are included in Figs. 9-2 through 9-7.

The example 5 kVA per channel system includes a 350-A starter-generator, with the rating defined by the starting requirement. In the 10-kVA system the size of the generator increases with increasing amounts of AC power because of the efficiency of conversion. For example, to obtain 5 kVA cf AC with an inverter efficiency of 60%, an input of 8.3 kW DC is required. Therefore, to obtain a system power of 10 kVA, a generator with a rating 13.3 kW is required. The weight for this unit has been estimated in Table 9-1.

Inverter weights in Table 9-1 are catalog weights of existing rotary inverters except for the 10 kVA unit,

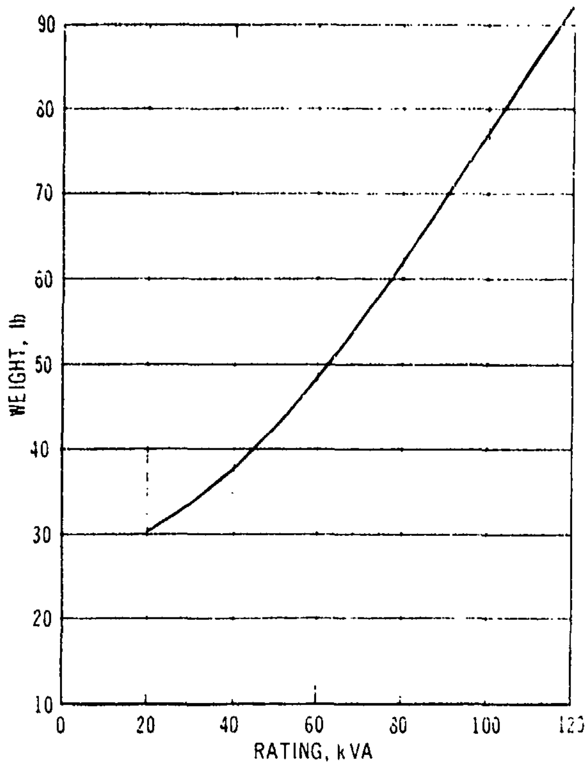

Fig. 9-2. CSD Weight vs Rating

which has been estimated because an aircraft inverter of this size is not available.

The weight of four 1/4-hp motors has been included in the example system. When cf AC power is available, AC motors can be used. These are one-half the weight of DC motors of the same power, less expensive, and significantly more reliable.

Batteries are sized for electrical starting. Feeder weights are estimated by thermally sizing the conductors. As the cf AC power is increased, feeder weights are reduced because the amount of low-voltage power being distributed is decreased.

Switch or circuit breaker weights are established ar-

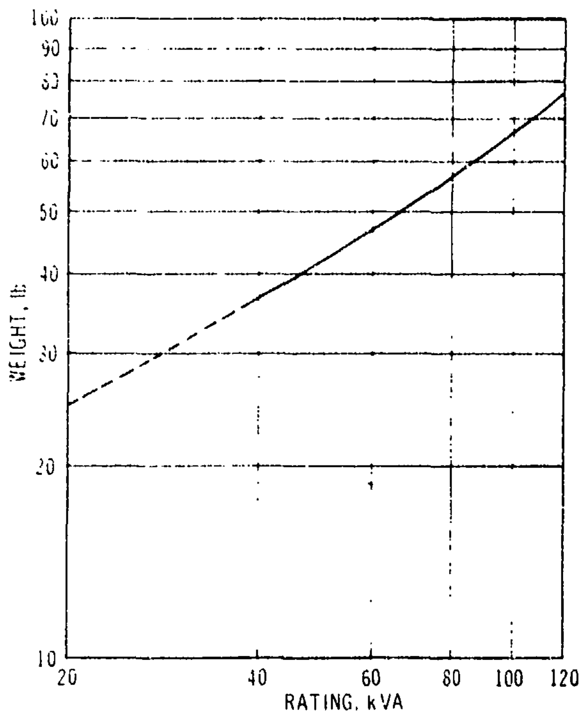

Fig. 9-3. Constant-speed Generator, Weight vs Rating, Spray-cooled, Single Bearing Mag. Housing, 12,000 rpm

TABLE 9-1
TYPICAL DC SYSTEM WEIGHT ANALYSIS

COMPONENT	5 kVA CHANNEL cf AC REQUIRED, kVA			10 kVA/CHANNEL cf AC REQUIRED, kVA		
	1.0	2.5	5.0	2.5	5.0	10.0
DC STARTER-GENERATOR	55.0	55.0	55.0	57.3	61.1	69.2
DC REGULATOR	2.6	2.6	2.6	2.6	2.6	2.6
INVERTER	34.0	44.0	67.0	44.0	67.0	102.3
MOTORS	4.5	4.5	4.5	4.5	4.5	4.5
BATTERY	55.0	55.0	55.0	55.0	55.0	55.0
FEEDERS, 30 ft	8.0	5.3	1.5	16.0	10.6	2.0
CONTACTORS	8.0	8.0	6.0	8.0	8.0	6.0
TOTAL/CHANNEL, lb	167.1	174.4	191.6	187.4	206.8	242.3

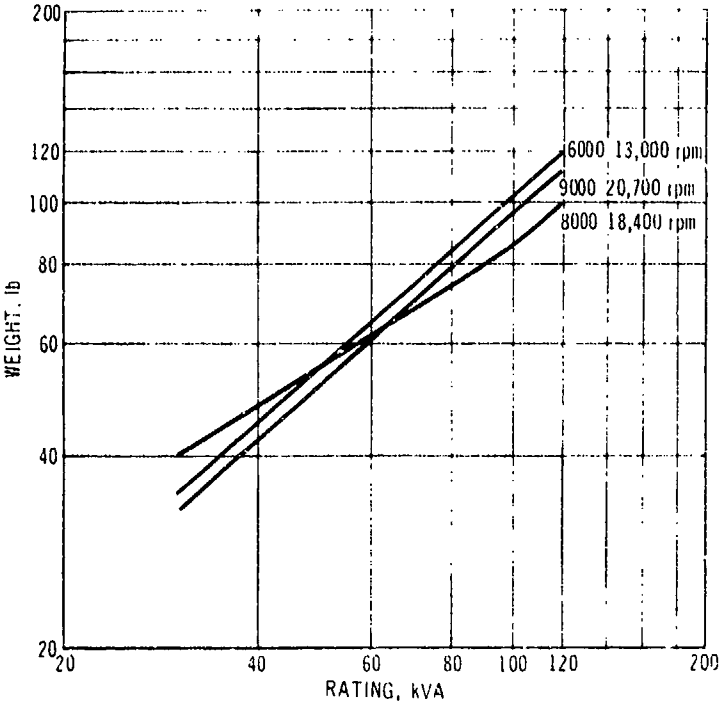

Fig. 9-4. Variable-speed Generator, Weight vs Rating, Spray-cooled, Single Bearing

bitrarily at 2 lb each. The number of units is based upon the simplified diagram in Fig. 9-1.

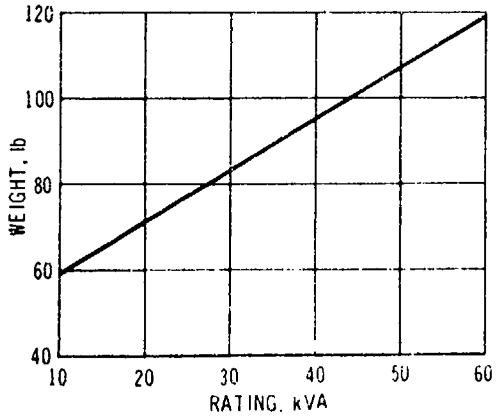

Fig. 9-5. DC Link Converter, Weight vs Rating

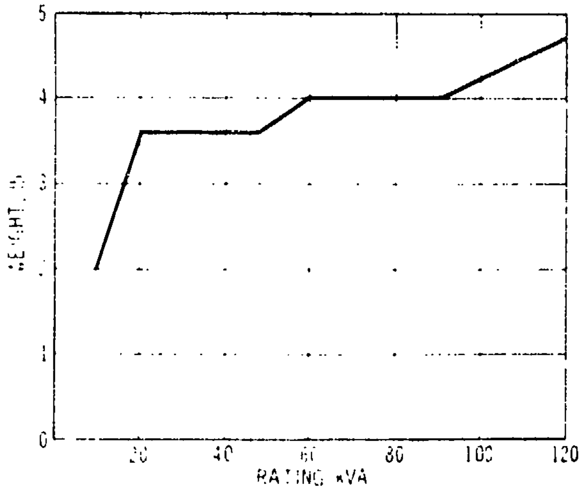

Fig. 9-6. Contactor (Switch or Circuit Breaker), Weight vs Rating

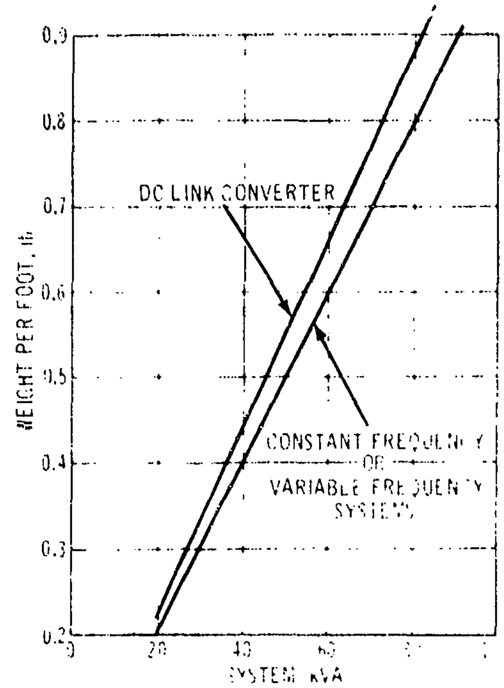

Fig. 9-7. Channel Feeder Weight Per Ft vs System Power

9-3.3 VARIABLE-FREQUENCY AC ELECTRICAL SYSTEM

9-3.3.1 System Description

In the vf AC system, the principal AC power is generated by an AC generator that is driven directly from the helicopter main rotor power transmission. Because the input speed to the generator varies in accordance with main rotor power transmission speed, the output frequency also is variable, being only approximately equal to 400 Hz. However, the vf AC system avoids the use of constant-speed drives. To obtain constant-frequency, 400-Hz power, a converter is used. This converter basically is a rectifier plus an inverter. The system is shown in the simplified diagram of Fig. 9-8.

The vf AC system generally is not competitive with the DC system at low load requirements and is given serious consideration only when the constant-frequency portion of the power requirement is less than approximately 10% of the total requirement. This can occur when a large amount of heating power is required for deicing and anti-icing. Disadvantages of the vf AC system are:

1. Considering hardware available at the present time, this system requires heavy, unreliable conversion equipment for avionics.

2. Because a 2.5-kVA converter is the only size presently available, growth potential is limited.

3. The distribution system is complex when three types of power are being used.

4. A second mounting is required on the engine gearbox when the DC starter-generator is required for electrical engine starting.

An advantage of the vf AC system is that the relatively heavy and unreliable DC generator can be eliminated when other means of engine starting are employed. A small transformer-rectifier unit can be used for the DC power.

9-3.3.2 Weight Analysis

Again, a 350-A DC starter-generator is assumed for engine starting. The vf AC generator weights are estimated. Because the DC generator is required for starting, this source DC power is assumed to supply all electrical loads in the total power per channel requirement except cf AC loads. Therefore, the vf AC generator is sized only to supply power to the converter. This results in the lightest overall system.

The available 2.5-kVA converter weighs 78 lb; however, using state-of-the-art design methods this converter could be made lighter. The weights for converters of other sizes have been estimated on a comparable basis.

Electrical motor, battery, feeder, and contactor criteria are the same as established for the DC system. Table 9-2 tabulates comparable vf AC system weights.

9-3.4 CONSTANT-FREQUENCY AC SYSTEM

In the cf AC system the principal source of power is an AC generator that is driven at constant speed by a hydraulic CSD, as shown in the simplified diagram of Fig. 9-9. The CSD is driven from the engine gearbox, and a second gearbox mounting pad is required when engines are started electrically.

The CSD discussed is a relatively simple hydraulic transmission. The basic elements of the CSD are two hydraulic units that are positioned on opposite sides of a common stationary port plate, as depicted in the mechanical schematic in Fig. 9-10.

The AC generator weights are scaled from existing equipment. The 10-kVA unit operates at 8000 rpm, while the smaller generators are 12,000-rpm machines.

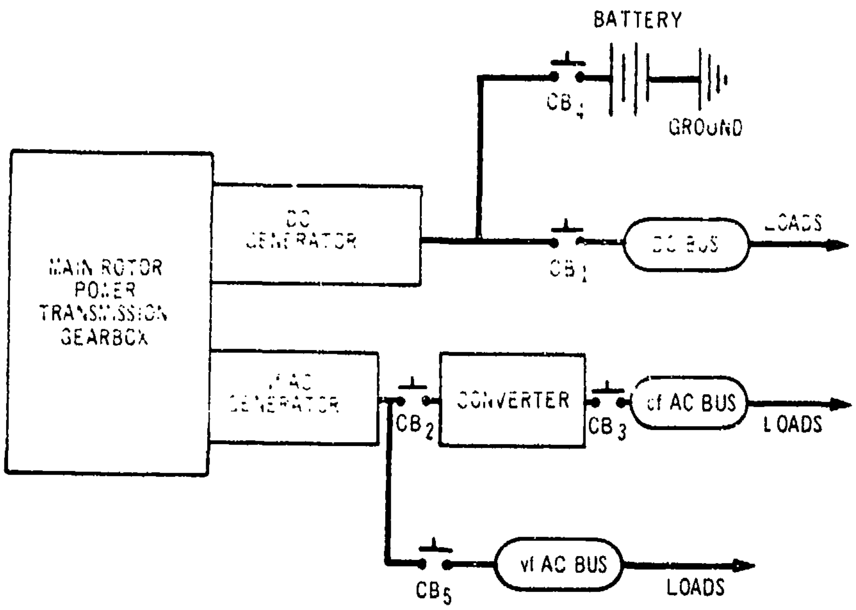

Fig. 9-8. Variable-frequency AC System

TABLE 9-2
VF AC SYSTEM WEIGHT ANALYSIS

COMPONENT	5 kVA CHANNEL cf AC REQUIRED, kVA 1.0	2.5	5.0	10 kVA CHANNEL cf AC REQUIRED, kVA 2.5	5.0	10.0
DC STARTER-GENERATOR	55.0	55.0	55.0	55.0	55.0	55.0
DC REGULATOR	2.6	2.6	2.6	2.6	2.6	2.6
vf AC GENERATOR-REGULATOR	9.0	18.0	30.0	18.0	30.0	50.0
CONVERTER	36.0	50.0	63.0	50.0	63.0	94.0
ELECTRIC MOTORS	4.5	4.5	4.5	4.5	4.5	4.5
BATTERY	55.0	55.0	55.0	55.0	55.0	55.0
FEEDERS, 30 ft	8.0	5.2	1.5	17.0	10.5	3.0
CONTACTORS	8.0	8.0	6.0	8.0	8.0	6.0
TOTAL/CHANNEL, lb	178.1	198.3	217.6	209.1	228.6	270.1

9-3.5 WEIGHT COMPARISON

The results of the weight analysis tabulated in Tables 9-1 and 9-2 are depicted graphically in Figs. 9-11 and 9-12. It is apparent from these figures that, as helicopter load requirements increase, the weight advantage of the cf AC system becomes more pronounced and significant. It is interesting to note, however, that even when cf AC load requirements are as low as 1 kVA, the CSD system is lighter.

Most helicopter applications have more than one generating channel. A two-channel helicopter requiring 10 kVA per generating channel, of which 5 kVA must be cf AC, can realize a weight savings of 65 lb per helicopter by incorporating a CSD electrical system instead of an inverter system, and 95 lb compared with a vf AC system (Fig. 9-12).

These weight comparisons are only for the specific conditions cited and should not be taken as representative of actual conditions or future state-of-the-art.

Weight comparisons of proposed systems must be performed when all electrical loads are known, using actual weights of currently available components.

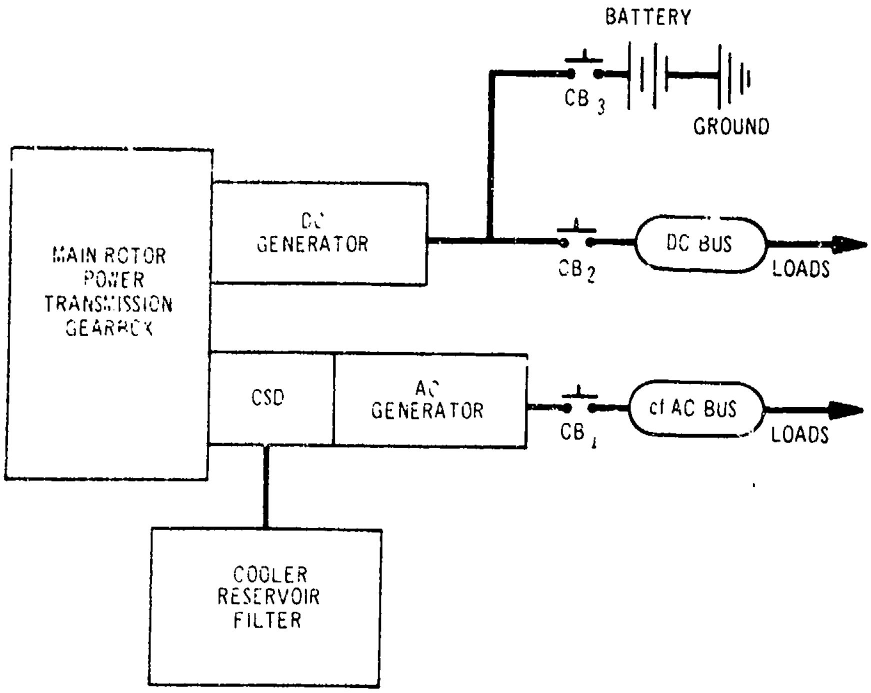

Fig. 9-9. Constant-frequency AC System

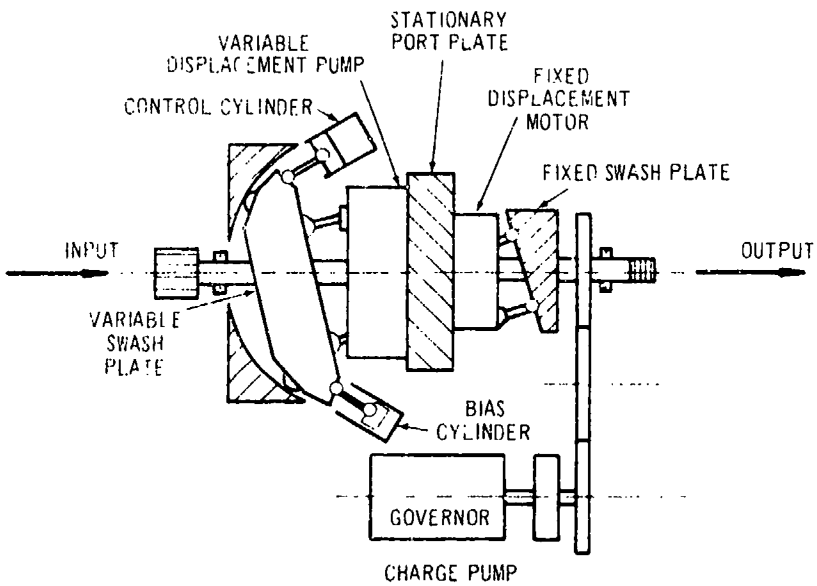

Fig. 9-10. Constant-speed Drive Mechanical Schematic

9-3.6 CF AC SYSTEM WITH ONBOARD APU

9-3.6.1 System Description

Although a significant weight saving can be realized on helicopters with electrical engine start requirements by using a cf AC electrical system, the inclusion of a DC generator and associated batteries substantially penalizes the cf AC system. A DC generator is not required as a source of DC power because a transformer-rectifier can be used instead. However, deletion of the DC starter-generator requires that some other method of starting the main helicopter engines be selected. This is particularly feasible and practicable when an auxiliary power unit (APU) is included in the helicopter as standard equipment. There is a trend developing toward including APUs on helicopters to provide power for self-sufficiency. The APU may be used to provide bleed air to a pneumatic starter or to drive a hydraulic pump for hydraulic starting (see par. 8-6.4). In addition to reducing the weight of the electrical system, the APU may make it possible to start the main helicopter engines at lower ambient temperatures. With electrical starting, battery power is a limiting factor. When using an APU, the key to engine starting at low ambient temperatures is the ability to start the small APU, which has a relatively low start torque characteristic. A cf AC system arrangement using an onboard APU and pneumatic starters is shown in Fig. 9-13.

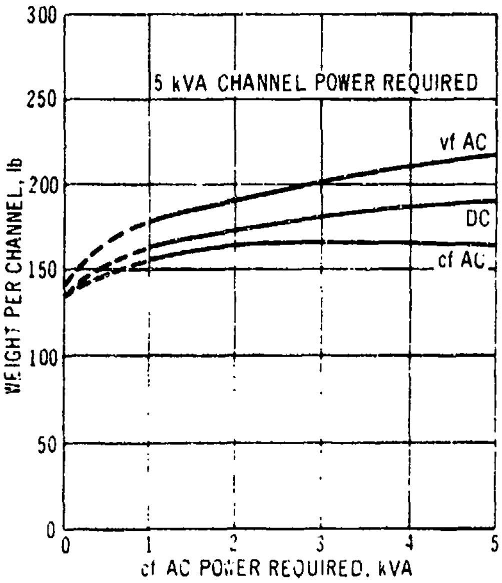

Fig. 9-11. Weight Comparison, 5-kVA System

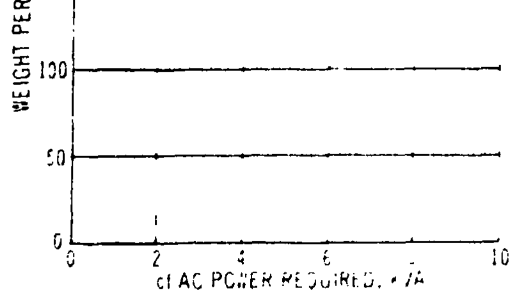

Fig. 9-12. Weight Comparison, 10-kVA System

9-3.6.2 Weight Analysis

In the weight analysis that follows, the cf AC system using a CSD will generate sufficient power to satisfy the total electrical power required per channel. A transformer-rectifier unit (TRU) has been selected to provide 1.5 kW DC, which is assumed to be sufficient to satisfy DC power requirements. Table 9-3 lists the weights.

The pneumatic starter installation weight has been estimated. Battery weight has been eliminated because it is assessed to the APU installation and is substantially less than required for electrical engine starting.

Figs. 9-14 and 9-15 illustrate the weight of this system and show the relationship with the system weights based upon electrical starting (Figs. 9-11 and 9-12).

Because of the significant reduction in electrical system weight that is possible when the APU is used for main engine starting in conjunction with a cf AC electrical system, it appears that a helicopter employing cf AC plus an APU would be lighter than one using a DC system excluding the APU. This, in essence, allows the APU to be installed without increasing aircraft weight. It follows that there would be a similar weight reduction for the vf AC system with APU starting and some reduction for the DC system, although the DC generator still would be required. Battery weight could be eliminated as in the cf AC system, but this saving would be offset partially by the addition of a pneumatic or hydraulic starter.

TABLE 9-3
CSD AND PNEUMATIC STARTING WEIGHT ANALYSIS

COMPONENT	POWER REQUIRED/ CHANNEL 5 kVA	10 kVA
PNEUMATIC STARTER	20.0	20.0
CSD	17.0	21.5
COOLER	3.0	3.0
RESERVOIR AND FILTER	3.0	3.0
OIL	2.0	2.0
AC GENERATOR	16.0	20.0
AC GENERATOR CONTROL UNIT	1.5	1.5
ELECTRIC MOTORS	4.5	4.5
FEEDERS, 30 ft	1.5	3.0
CONTACTORS	6.0	6.0
TRU, 50 A	5.5	5.5
TOTAL CHANNEL, lb	80.0	90.0

9-3.7 PROTECTION

9-3.7.1 Direct Current System Protection

In general, protection of parallel DC generator systems involves two items: (1) removal of a generator when its voltage is abnormal, and (2) short-circuit protection. A third item of interest is overvoltage protection.

The first item is accomplished by a "generator reverse current cutout". This consists of a line contactor and polarized pilot relay mounted in one enclosure. This device senses the difference between generator and bus voltage, and also senses the generator current. When the generated voltage exceeds bus voltage, the pilot relay closes and completes the circuit to the contactor. When the generated voltage drops below normal, reverse current opens the pilot relay, removing the generator from the bus. Service experience has indicated that the contactor of the cutout must be designed with sufficient capacity to interrupt the maximum output voltage and current on a wide-speed-range generator under full field. Such a condition can occur upon either regulator failure or accidental contact between the generator shunt field wire and the positive line wire.

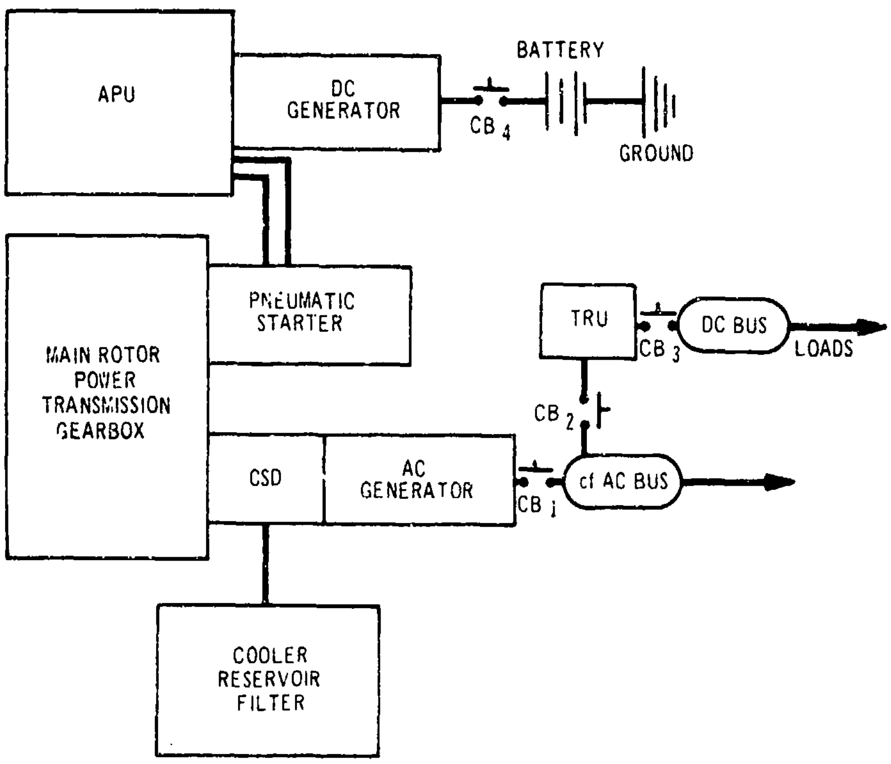

Fig. 9-13. Constant-frequency AC System With Onboard APU

Short-circuit protection is provided in some measure by the reverse current cutout. However, a circuit breaker with a reverse-current trip having sufficient time delay to override normal transients sometimes is used to provide the following functions: interrupting capacity up to 12,000 A; backup protection for a welded contactor or a pilot relay failure; auxiliary contacts to de-energize the generator and prevent it from continuing to feed the fault; and interruption capacity for overvoltage conditions, using a shunt trip coil actuated by an overvoltage relay. In some cases, differential current protection is used to disconnect a faulted generator or generator feeder, or to de-energize the field without time delay, to reduce the hazard of fire. Several successful types of differential current protection for DC generator circuits are available, such as those using series coil current balance relays, a current balance relay operated from special metering shunts in the line and ground leads, or a current balance relay operated from a metering shunt in the line lead and the potential across the generator series windings on the grounded side. Development work also has been performed on methods that compare the transient voltages of mutual reactors in the line and ground leads. This type requires an auxiliary AC source for maximum effectiveness.

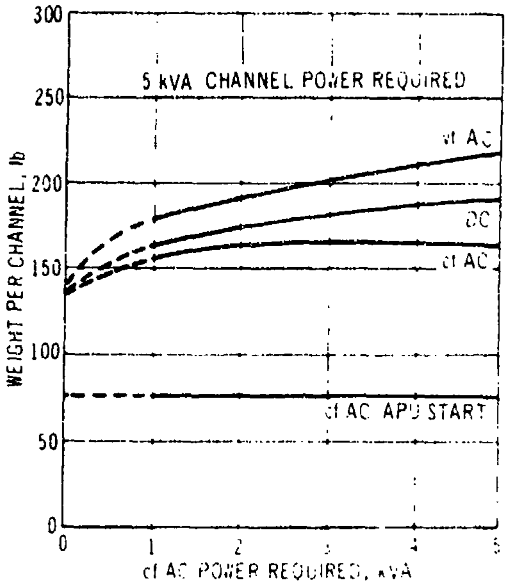

Fig. 9-14. Weight Comparison, Electrical Systems With and Without Pneumatic Starting, 5 kVA per Channel Power Required

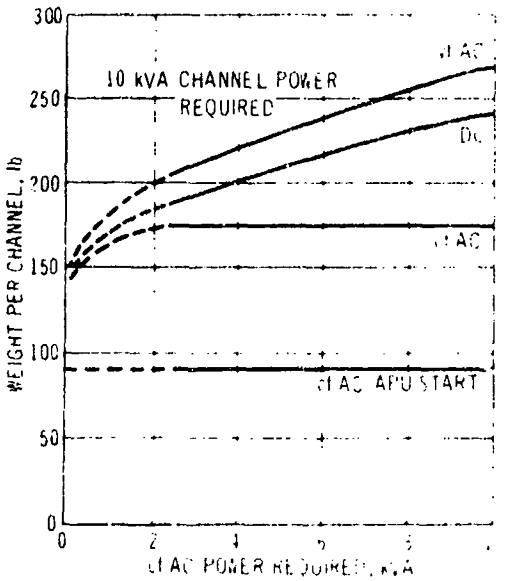

Fig. 9-15. Weight Comparison, Electrical Systems With and Without Pneumatic Starting, 10 kVA per Channel Power Required

The need for overvoltage protection results from the possibility of generator malfunction. Full-field conditions can produce very high voltage on DC systems and, because of the potential damage to load equipment, it is customary to provide fast-acting overvoltage relays to de-energize and remove the faulty generator from the bus. The various methods employed sense either generator field voltage or bus voltage. Sensing generator field voltage is selective inherently because the field voltage of the unfaulted machine always is below rated value. However, it does not protect for a partial failure at high speed, because the field voltage still may be below rated value and yet sufficient to produce an overvoltage at the terminals. Bus voltage sensing protects for all overvoltages above the relay setting but requires a more complex means of selecting the faulty machine. This method requires biasing of the overvoltage relay or the use of a separate selector relay acting on a directional unbalance in the regulator equalizer current.

9-3.7.2 Isolated AC System Protection

The following types of protection are used for isolated AC systems:

1. Overvoltage protection
2. Ground fault protection
3. Undervoltage protection
4. Underspeed or underfrequency protection
5. Phase sequence protection.

The overvoltage circuit may sense the average of the three phase voltages, the highest of the three phase voltages, or some variation of these two, applying the sensed voltage to a relay with an inverse time voltage characteristic. Care is necessary in selection of the sensing method to avoid nuisance tripping of part or all of the generators during a line-to-ground fault.

When an average sensing voltage regulator is used in such a situation, one or both of the unfaulted phases will experience overvoltage. Depending upon the magnitude of the fault current and the sequence impedances of the generators, this overvoltage may result in tripping a high-phase-sensing type of overvoltage relay before the other protection can clear the fault from the system. Use of a high-phase-sensing voltage regulator in combination with a high-phase-sensing overvoltage relay solves this problem.

In reaching a decision on the type of sensing to be used, the following factors should be evaluated for the particular system being studied:

1. The magnitude of single-phase overvoltage attained with various magnitudes and locations of line-to-ground faults
2. The clearing times for these faults
3. The protection coordination problems
4. The effect of overvoltage versus time on sensitive load equipment.

The time voltage tripping characteristic is defined by MIL-STD-704, which gives the maximum voltage that the load equipment may be required to withstand, one of the design criteria being load protection. The generator and regulator must be designed so that normal transient conditions will not cause system voltage to rise beyond the maximum level and cause false tripping.

Short circuits at the generator terminals or at the cables to the bus may cause damage from fire and smoke or cause loss of electrical power. Thus far, some type of differential current fault protection has been considered desirable on AC generators and their feeders because this type of fault protection is inherently selective and thus can be made instantaneous. In an AC system, the current in a phase lead normally is equal to the current in its neutral. If there is a fault somewhere within the loop, there will be a current unbalance between the phase lead and the neutral lead. To detect a fault, the current unbalance in the loop is sensed by using current transformers operating a relay to trip the generator control panel. There may be either three or six current transformers, with the selection depending solely upon the weight comparison of the additional three current transformers in the six-transformer system versus the added length of the neutral leads necessary in the three-transformer system. There is slight difference in the protection level between the systems using three or six transformers.

Use of the six-transformer protection system causes some reduction in reliability because of the possibility of breakage of the small wires used to complete the loop between transformers. An open circuit in the loop causes immediate and nonresettable tripping of that generator. This disadvantage should be weighed against the advantage of quick, selective fault clearing for the generator and its feeder. In a multigenerator parallelled system, the occasional loss of a generator from this cause may not be a serious objection. Protection levels of approximately 20 to 40 A difference in current can be obtained. However, the necessity for a protection level as low as this has not been demonstrated thoroughly. Consequently, other methods of fault protection may be used, as discussed below.

Undervoltage protection can be obtained by several methods, among which are load protection and fault protection. For example, many loads can be damaged by undervoltage. In the case of fault protection, undervoltage sensing gives a measure of protection against solid bus faults. A three-phase solid fault on the main bus to ground will so depress the bus voltage as to cause the undervoltage protection to trip after a slight time delay. The delay is necessary for system coordination; in case of a fault on the distribution system, the overcurrent circuit breakers should trip before undervoltage trips the alternator.

Underspeed and/or underfrequency trips are used with a controlled-frequency system to provide generator switching on and off the bus, and to coordinate with the undervoltage trip when shutting down the system. In addition, some loads must be protected against underfrequency. The coordination of an underspeed trip with the undervoltage relay is necessary so that, in shutting down the system, the underspeed switch takes the generator off the bus and does not permit the undervoltage relay to operate.

Phase sequence relays have been called for in some specifications. In general, phase sequence protection is dependent upon system maintenance requirements. Phase sequence always should be checked when changing a generator.

Anticycling protection is necessary to prevent damage caused by cycling of a system on a fault. Without this protection, when the generator switch is held in the reset position on a fault, the fault will cause the system to trip again and the reset switch will cause it to reclose.

Therefore, a lockout or anticycling relay is used to interlock the reset switch electrically and to permit only one reset on a fault when the reset switch is held closed.

9-3.7.3 Parallel AC System Protection

In order to attain the highest degree of use from a parallel AC system, it is necessary to prevent a fault or a malfunction in one part of the system from destroying the power-producing abilities of the remainder of the system. Moreover, it is advantageous to remove the faulted portion with minimum power interruption to the loads, and quickly enough to prevent further damage to the system.

In designing system protection, the first function of the generator protection panel is to distinguish between failures in the distribution or load system and those in the generator system. If the failure is in the generating system, removal of the faulty machine should take place as quickly as possible. If the distribution and load system are protected and coordinated adequately, the panel should take no action for failures in this area except possibly when protective devices fail to operate properly.

Some of the malfunctions most likely to occur and for which protection is required are:

1. Excitation protection
2. Open-phase protection
3. Bus fault protection
4. Generator short-circuit protection
5. Unbalance real power protection.

One of the foremost problems has been protection against excitation failures. It has been determined that both overexcitation and underexcitation can result in the loss of an entire generating system. In the past, relays operating at the ceiling voltage of the exciter have been used for overexcitation protection. However, because this method gives incomplete protection, other methods have been developed to allow full protection against both overexcitation and underexcitation. Either of these malfunctions results in circulating reactive current flow between machines. However, a system that has an overexcited machine will have a higher-than-average bus voltage or, for a heavily loaded system, at least normal bus voltage. A system with a machine that is underexcited will have a bus voltage lower than normal. These two signals—unbalance in reactive current flow and bus voltage—are used to detect and select an abnormal machine from a system for both types of excitation failures. Furthermore, by the use of a differential circuit and a polarized relay, the magnitude and direction of the unbalanced reactive current flow can be determined.

An open circuit on a phase wire that includes a load division current transformer can cause loss of the entire system. Provided that the system load does not approach the total machine capacity, an open on any other phase is not serious because the effects of the load division circuits are not present. For this reason it is recommended that both the real load and the reactive load division current transformers be located in the same phase.

One method of protection for an open phase when operating isolated is to detect an unbalance in the three phase voltages. This signal may be used with a time delay to remove the generator from the line. In parallel operation, an open phase will result in a high neutral current. It has been suggested that this might be used as a sensing means to isolate the generator if a split bus system is used. Because high fault clearing capacity is desired, this method *shall* not be employed.

Open-phase protection as described also will serve as a method of bus fault protection because the signals for these two malfunctions are similar. Here also, the protection depends upon a split bus configuration, but again would result in isolating all generators when a line-to-ground fault occurs on the system. Another method would be to use total differential current to separate the generators and then to use single-phase undervoltage relay sensing with a time delay to remove the faulted generator.

The probability of having a closely balanced, three-phase fault on a bus system is quite low compared with the probability of a single-phase, or line-to-line, fault. By making use of this fact, most bus faults may be detected by sensing excessive negative sequence voltage and using this to trip the bus tie switches or circuit breakers after a time delay sufficient to provide coordination with distribution system fault protection. This leaves the faulted bus section supplied by one generator. The exciter ceiling voltage relay, mentioned previously, is a relatively simple means of removing this generator from its faulted bus. This relay must have sufficient time delay to coordinate with the other system fault protection devices—such as distribution circuit breakers.

The differential current fault protection described in par. 9-3.7.2 also is applicable to parallel operation. In a parallel system, the protection against failures of the power mover to date has been of three types—overspeed, underspeed, and reverse power. Reverse power is power transmitted into the drive from the system. The total action of the three types of protection is directed toward separation of the faulted drive and generator from the remainder of the system.

The most satisfactory reverse power protection device used to date on AC systems has been a mechanical overrunning clutch installed between the generator and its driving source. Attempts have been made to design reverse power relays to perform an equivalent function by opening the generator contactor. To date, these attempts have not been completely successful, possibly because of the difficulty of designing a relay to measure accurately the net direction of three-phase power under conditions of badly unbalanced voltages and currents found with some fault conditions.

One method of overspeed protection on a hydraulic drive is mechanical speed sensing. If an overspeed occurs, the drive is transferred into its maximum underdrive ratio. On the assumption that the synchronous speed of the alternator is reached at some medium value of drive ratio, action of the overspeed protection would result in a drive speed slower than that required for synchronous operation. However, because of the overrunning clutch in the drive, the alternator rotational speed remains in synchronism with the rest of the system. Because of the possibility of sustained motoring (clutch slippage), it is important that the clutch have proper lubrication under this condition of operation. Mechanical overspeed protection also may shut off the pneumatic or fuel supply to turbine drives.

Underspeed protection has been used to remove the alternator from the bus rather than to produce any reaction on the drive itself. The devices available to perform this function detect actual drive speed, and their actions result in the operation of electrical contacts. These contacts are connected into the alternator control circuits so that the alternator is removed from the bus.

The action of both overspeed and underspeed devices relies upon the overrunning clutch in order to obtain selectivity between good and bad units. As an illustration, given an overspeed condition on a two-generator system, the faulty drive and its alternator can raise the system frequency and will assume all the real load. The other alternator will be rotating faster than its drive, but is able to do so because of the overrunning clutch. After the overspeed device has operated and the faulted drive output speed has been lowered below normal, the system will pull the alternator of the faulted drive away from the drive. The underspeed device on the faulted drive then will operate, resulting in its alternator being removed from the bus.

This type of reverse power protection operates only for failure of the drive and not for mechanical failure of the alternator itself. It is possible, with electrical circuits somewhat similar to reactive sensing in excitation protection, to sense reverse real power. This would protect against alternator mechanical failures. However, the settings of the device must be such that operation is prevented when reverse power appears as the result of an overspeed condition on one of the other alternators and its drive. A mechanical failure of the generator probably will include a ground fault, so that differential protection will remove the generator from the bus. Also, underexcitation will trip the machine when it no longer is generating voltage.

In considering the type and amount of protection to be provided, the designer must consider the type of bus system on the helicopter. It is obvious that bus fault protection provides little useful service on a solid bus system. With the split bus system or a synchronizing bus, protection can be modified such that under some conditions the machines can be separated and each then will supply the system loads successfully on its individual bus.

9-3.8 PARALLEL VERSUS NONPARALLEL OPERATION

Systems with multiple generators of electrical power can be arranged either with the generators in parallel or with the generators isolated (nonparallel). The advantages of each of these arrangements are discussed in the paragraphs that follow. These advantages must be reviewed prior to selection of one of the alternative arrangements for a system to meet given requirements.

9-3.8.1 Parallel Operation

Advantages of parallel operation are:

1. Total electrical load is proportioned among active generators, thus improving reliability.

2. Failure of a unit generator output in a multiple generator system results in no interruption of primary service. Sufficient overload capacity in each generator allows adequate time for load monitoring. Additional safety is provided by continued service prior to monitoring.

3. A larger, interconnected generating capacity allows larger starting and peak load demands for a given time voltage disturbance.

4. Inverse time overcurrent fault protection will operate faster.

5. Installed generator capacity is used more effectively, making possible a reduced number of generators or lower generator ratings.

6. Electrical integrity usually can be maintained in the event of a generator or prime mover failure, only a

portion of the power system is lost in a multigenerator installation.

7. Redistribution of load to remaining generators is automatic upon loss of a generator, thus simplifying operator supervision.

8. Distribution system usually is simplified by providing a common bus for all equipment in lieu of one or more buses from isolated power sources.

9. Complicated switching of loads from a dead bus to an active bus is eliminated.

10. Human element of selecting the power bus for various loads is avoided.

11. Beat-frequency effects in sensitive equipment, such as autopilots and radar, are eliminated by synchronous operation of all AC sources.

9-3.8.2 Nonparallel Operation

Advantages of nonparallel operation are:

1. Automatic load division circuitry and mechanism between generators is unnecessary, thus reducing complexity.

2. Disturbances in a section of the electrical system affect only the section associated with one generator.

3. Fault current of the system is reduced in magnitude, allowing use of smaller and lighter switching and protective equipment.

4. Full individual generator capacity may be utilized because no load unbalance due to meter error or parallelling efficiencies need be considered.

9-3.9 ELECTRICAL SYSTEM RELIABILITY

Each power source on a helicopter should be capable of operating independently of any other power source on the same helicopter. Therefore, it should not be possible for a fault in one electrical power source to affect the ability of another power source to furnish the prescribed type of power to the loads. Furthermore, each power source should be capable of initiating and delivering its power independently of any other power source. If a section of the power distribution system of one power source fails, it must be possible to remove the faulted section before the faulted source can make vital elements unserviceable-- even when demand is transferred to a second power source or cycled between power sources, as in repeated attempts to reset a circuit breaker.

Evaluation of system reliability is discussed in detail in Chapter 12. Selection of the type of electrical system should include consideration of the ability of the system and its components to meet the helicopter reliability guidelines or requirements defined by the procuring activity.

9-3.10 ELECTRICAL SYSTEM SAFETY

An electrical system should be designed so that no fault or probable combination of faults to any of its parts can result in an unsafe condition. Therefore, in the case of equipment necessary for flight, duplication of that equipment or the use of overlapping functions from other pieces of equipment may be necessary. The degree of safety required to perform certain functions should be consistent throughout the electrical power system; thus, if the loads are duplicated or have overlapping functions, it may be necessary to duplicate the supply to these loads.

9-3.11 EMERGENCY SYSTEMS

Measures should be taken to provide power to critical loads in case of failure of the primary electrical system on the helicopter. Automatic or manual monitoring of loads is mandatory when the primary electrical source fails. The following is a list of possible emergency sources:

1. Battery. Small amounts of power may be made available by the use of a battery for a limited period of time. However, if a battery is relied upon, great care must be taken to keep the battery adequately charged. Switching to an emergency bus would bring the battery power to the correct loads. By tri-service agreement, only nickel-cadmium batteries *shall* be used for new design.

2. Separate generator driven by main transmission

3. Hot gas turbine generator units. Auxiliary turbine power to supply mechanical and electrical load may be obtained conveniently from liquid fuels.

4. Monopropellant system. Operates independently of altitude, attitude, and of other generation systems

Helicopters having dual sets of primary flight instruments *shall* have each set powered from separate sources. In some cases this requirement can be satisfied by a separate supply from different sections of a sectionalized multigenerator main bus system. In other cases a small independent power supply, such as a hydraulic motor generator set, can be used for one set of instruments.

9-3.12 DISTRIBUTION SYSTEM

The distribution system is used to transfer available power to the load buses; in a multiple-source system, the distribution system often is designed to make more than one source available to the important load buses.

There are a number of definitions that must be understood prior to a discussion of distribution systems. The term itself, "distribution system", as used herein, is that part of the complete system between the generator line contactors and the load buses. It is comprised of two general areas:

1. The source bus system, which provides the junction points among generator feeders, source bus interconnections, and the transmission system.

2. The transmission system, which is the part of the system between source buses and load buses.

"Emergency loads" in helicopters usually mean those necessary for continued flight and landing.

"Generator feeders" are the lines between the power source terminals and the generator line contactor. They are not, by definition, a part of the distribution system, but enough points of similarity exist between feeder design and transmission system design to warrant further discussion of feeders.

Each feeder must be capable of carrying the full output of the machine. Wire is rated for current and ambient temperatures in such a fashion that an extremely long insulation life is obtained. In most systems, a machine is carrying full load for a comparatively small proportion of the total operating life of the flight vehicle. For these reasons some designers assign a rating to feeder wires in excess of the book value to obtain a lighter system with a probable insulation life still in excess of expected vehicle life. However, as a minimum requirement, the wire size *shall* be sufficient to carry any load or overload up to the limit of machine capability without smoking or overheating. Most AC machines will carry double load considerably longer than the 5 sec rating.

Voltage drop is usually the factor that determines wire size for low-voltage circuits in large-capacity systems such as existing DC applications. In higher voltage systems, such as existing three-phase, 120/208 V AC systems, thermal capacity is usually the factor that determines wire size except where very long lines are used (100 ft or more) or where induction motors with high acceleration rates are required. However, the advent of high-temperature insulations eventually may result in wire sizes small enough to make voltage drop the limiting factor even on 120/208 V systems. With bundle conductors in three-phase systems, the voltage drop primarily is resistive. If the phase conductors are separated from each other, both line reactance and voltage drop are increased.

With single-phase loading, the voltage drop in a given line may be considerably greater than the drop obtained with a balanced load producing the same phase current and, for wire sizes larger than AN-10 or AL-8, predominantly is reactive. Because the zero-sequence reactance for a given wire is dependent primarily upon spacing of the line from the skin, voltage unbalance at the load bus with unbalanced loading is reduced by running distribution lines as close to the skin as possible.

Fault currents (currents in faulted or failed circuits) are used to protect distribution systems. Experience has shown that, for reasons given subsequently, great care must be exercised in the use of fault currents.

With bundled conductors, positive and negative sequence impedances are small. Hence, three-phase fault currents are affected little, and line-to-line fault currents are not affected greatly by distribution system and generator feeder lines of moderate lengths. However, line-to-ground and line-to-line-to-ground fault currents are affected greatly by line lengths in stiff systems, particularly those with large spacing to the skin. For this reason, it is possible to reduce these fault currents by increasing distribution and feeder-line spacing from the structure, or to increase them by decreasing spacing. In practice, installation problems often make it difficult to obtain a large range of control of fault currents by this means. It should be noted that decreasing of fault currents by this method results in increased voltage unbalance at the load bus with unbalanced loading.

9-3.13 ELECTROMAGNETIC COMPATIBILITY (EMC)

Careful consideration must be given to electromagnetic compatibility (EMC) during the design of the helicopter electrical system. MIL-E-6051 defines electromagnetic compatibility as "the capability of systems and all associated subsystems/equipments to perform with required effectiveness, and without degradation, in the total electromagnetic environment encountered during accomplishment of the assigned mission". The program required by MIL-E-6051 must be planned and the pertinent technical criteria defined during the preliminary design phase. These criteria include the definition of required system effectiveness, which will be included in the helicopter model specification. Additional criteria that must be applied during the preliminary design phase pertain both to the electrical system—component, connection, wire, and cable compatibility criteria—and to the helicopter structure and other subsystems. For example, appropriate provision must be made for bonding and grounding, as well as for the selection and installation of components in order to provide protection from lightning and static electricity. Procedures for providing this protection are discussed in par. 7-9, AMCP 706-202. Methods for maintaining electromagnetic interference (EMI) within the maximum acceptable levels are discussed in par. 7-6, AMCP 706-202.

9-4 HYDRAULIC SYSTEMS

9-4.1 GENERAL

Hydraulic applications in helicopters are oriented primarily toward flight control functions. Hydraulics generally provide full power capability plus stability augmentation, allowing low control system force levels while also providing good flight characteristics. Meeting design requirements for handling qualities without stability augmentation can be very difficult if not impossible. If the vehicle has a stability margin without augmentation and also has relatively low control peak load requirements, power boost with manual reversion may be used. The force levels required after reversion to the manual mode should be such that a pilot can continue flight for a reasonable period prior to landing without excessive physical discomfort.

The typical helicopter functions are cyclic pitch, collective pitch, and directional control. The cyclic mode includes control in the lateral and longitudinal planes. Collective pitch provides vertical control.

Utility functions that require or can use hydraulics are varied. Hydraulic systems are used to provide the power and control for hoist or winch systems. Doors and loading ramps can use hydraulics as the optimum approach. Occasionally, the landing gear may be retractable and require hydraulics. Other utility applications include gun turrets and gun drives, rotor braking, wheel braking and steering, fluid dampers, and cargo hooks.

The engine-starting system is an important application for hydraulics. Self-contained, independent engine starting can be provided through the use of a hand pump and accumulator system. The starting motor sometimes is designed to be convertible to a hydraulic pump after starting.

9-4.2 TRADE-OFF CONSIDERATIONS

9-4.2.1 Mission Performance Requirements

The flight control load levels, both estimates and design requirements, must be analyzed to determine if manual pilot effort, boosted pilot effort with reversion to manual, or redundant full-power, irreversible hydraulic systems are required. In addition, the possibility of alternatives to hydraulics in flight control must be evaluated.

Utility functions must be evaluated in a preliminary trade-off relative to the alternative modes of operation. These include hydraulic, pneumatic, electromechanical, manual, and pyrotechnic.

Weight, cost, continuous versus cyclic operation, temperature, and performance limitations are factors that are important in the preliminary trade-off.

9-4.2.2 System Trade-offs

The required trade-offs involve hydraulic system pressure levels; constant pressure, variable-flow systems versus constant-flow, unloader-accumulator systems; and central systems versus packaged, remotely located systems. Also, the type and location of filtration to be used in the hydraulic systems must be considered. The alternative fluids must be evaluated and the optimum one chosen. Survivability and reliability requirements must be analyzed to establish the redundancy required for each concept under consideration. The type of reservoir pressurization for hydraulic systems may be a pertinent trade-off. Weight, cost, maintainability, reliability, and component life are factors to be weighed in all the trade-offs.

Hydraulic systems generally are the only candidates for flight control systems. For utility functions, manual, pneumatic, pyrotechnic, and electromechanical systems can be used in addition to hydraulics and will be involved in the total trade-off. The detail trade-off considerations for these techniques are covered in par. 9-1.

A more detailed discussion of the hydraulic system trade-offs is presented in the paragraphs that follow.

9-4.2.2.1 System Pressure Trade-offs

The system pressure trade-offs are associated primarily with comparing aspects of weight, cost, reliability, heat rejection, and system maintenance. Weight trade-offs for hydraulic systems generally indicate that the lightest system can be attained with pressures in the 3000-4500 psi range, as shown in Fig. 9-16. However, there has been little incentive to go beyond 3000 psi because there is a lack of experience in service with higher pressures and, therefore, components generally

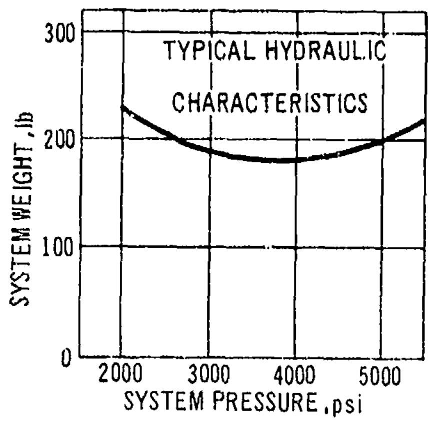

Fig. 9-16. Hydraulic System Weight vs Operating Pressure

are not available. The weight and envelope advantage must be important to justify the extra development, test, and service maintenance equipment expense involved with using higher pressures. In most cases the weight savings in going to 4000 from 3000 psi is 3-5% or less.

The lower pressure systems generally provide reduced maintenance and increased reliability. The system heat rejection levels can be reduced significantly by lower pressures.

The basic motivation is to keep the system pressure as low as is consistent with the emphasis placed upon each of the evaluation factors for a given vehicle preliminary design effort.

9-4.2.2.2 Trade-offs Among Constant-pressure, Variable-flow; Constant-flow; and Load-sensitive Systems

The basic hydraulic system source alternatives are constant-pressure, variable-delivery systems versus constant-flow unloader or relief-type systems. A variation of the constant-pressure design is the use of a load sensor to keep the pressure at low levels except when increased pressures are required for high loads.

The constant-pressure system uses a variable-delivery pump whose pumping stroke is controlled by a hydraulic servo-actuator system. The servo is, in effect, a pressure regulator that tries to maintain a constant pressure within its limits and to meet the flow demands. The difference between maximum full flow pressure and compensated (no flow) pressure can be maintained at 150 psi or less. Fig. 9-17 shows a typical pump performance curve. This system may require an accumulator to keep transient overshoot within 135% of rated system pressure limit. This type of system, with its high flow response characteristics, is required to give adequate flight control dynamic characteristics.

The constant-flow, fixed-stroke pump is used with an accumulator and either an unloader valve or a relief valve. Either valve defines the maximum system operating pressure. This type of pump also can be used with an open center system. In this case the flow is directed back to the reservoir at low pressure until the valve is closed, thus directing the flow to operate a subsystem. The pressure is developed as necessary to meet the load requirements. Upon completion of the subsystem operation, the control valve is returned to open center (bypass) condition. The constant-flow system generally is used for utility functions only.

The load-sensitive, constant-pressure system generally is used in applications that require holding system heat rejection to a minimum. This type of system has a reduced heat rejection capability because its nominal operating pressure can be as low as 500 psi compared with a standard 3000 psi. Therefore, the input energy requirements are reduced significantly.

9-4.2.2.3 Central Hydraulic Systems Versus Remotely Located Packaged Hydraulic Systems

The central system generally is the accepted approach because of the number of using areas or subsystems.

Where there is a single performance requirement, an integrated, electric-motor-driven / hydraulic pump/-reservoir system may prove advantageous. The number of external leak points is reduced and the weight may be less.

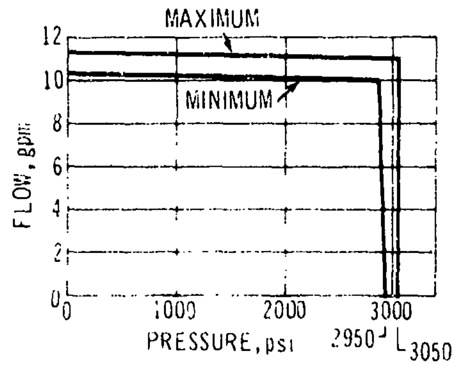

Fig. 9-17. Typical Hydraulic Pump Pressure vs Flow Characteristics

9-4.2.3 System Filtration Trade-offs

Adequate filtration is essential to the satisfactory operation of hydraulic systems. The discussion that follows presents some of the trade-off considerations used when evaluating filters.

The hydraulic pump is the most important consideration because it is the source of practically all system-generated contamination. The filtration level generally accepted in the past has been 10 micron particle size nominal with no absolute maximum, or 10 micron nominal, 25 micron absolute. The trend is currently toward 5 micron nominal, 15 micron absolute filtration.

Filtration can be central or otherwise. A filter just forward of a component, such as the flight control actuator, can be the ultimate in protection, removing contamination introduced into lines and hose during manufacture or maintenance in service. However, where there are several subsystems, this filtration method can impose a penalty in weight and cost.

The current Military Specifications require central filtration without a bypassing relief feature for pressure side filtration. A system then can be configured rather simply with one filter in the return and one filter in the pressure side. The components should incorporate 50 to 75 micron screens to take care of the "nuts and bolts" type of contamination.

Filter elements with a bypass relief valve are used to prevent possible excessive return pressures on the components. However, the pump case drain should not be routed through these filters. Pump pressure transients and/or well-loaded (dirty) filters would allow bypassing of the filter and subsequent contamination of the reservoir along with deterioration and failure of the pump. Therefore, consideration should be given to use of a separate filter for the pump case drain. This would be placed in series with the return filter, thereby minimizing the possibility of a contamination short circuit. The elements in all three areas, as shown in Fig. 9-18, can be the same size or have the same flow rating. The weight-cost penalty of the drain filter is compensated for by two factors: first, only one element need be stocked; second, the filter element life will be increased three to five times over that of an element whose design supposedly is optimum for the job. Consequently, the maintenance and other servicing aspects will be quite acceptable.

Differential pressure indicators normally are incorporated into the filter package. This allows for as-necessary element change. Time-delays are required to avoid destruction of the meter as a result of transients. In addition, the indicator should have a thermal lockout to prevent meter destruction due to increased oil viscosity during cold starts.

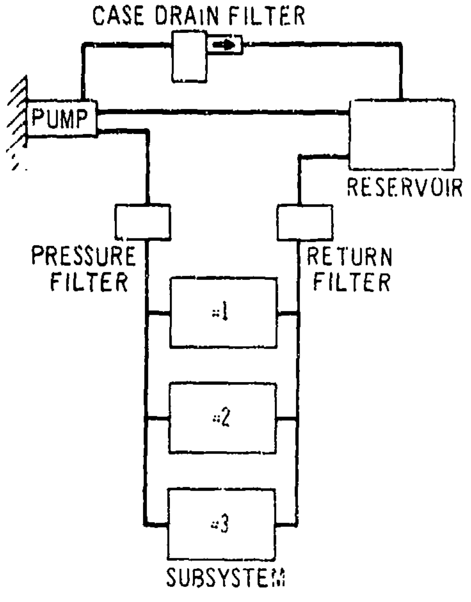

Fig. 9-18. Filter Location, Typical Hydraulic System

Throwaway elements are preferred over cleanable filter elements because cleaning has not proven to be satisfactory.

9-4.2.4 Fluid Selection

The "red oil" (MIL-H-5606) is in general use. The cost of changing ground servicing equipment, including test benches and ground carts, has been a completely inhibiting factor to changing hydraulic fluids. The primary reasons for changing are operating temperature or fire resistance characteristics. For helicopters, the maximum fluid operating temperatures are within the capability of the "red oil". Therefore, the design requirement almost must be unique before a fluid trade-off is required. Safety considerations, including reduction of post-crash fire hazards, are discussed in par. 9-4.5.

9-4.2.5 Survivability/reliability Trade-offs

The survivability/reliability requirements established must be equated with required redundancy in components and systems as necessary. Generally, two, active, independent, parallel usage systems are required. For utility functions, a passive backup to the normal system may be required—such as pneumatic, manual, or pyrotechnic.

9-4.2.6 System Reservoir Trade-off

The basic choices of reservoirs are spring-energized, air-pressurized (separated and unseparated), and bootstrap. The hydraulic pressure system (pump or accumulator) is the pressure source for the latter.

The bootstrap or separated air types are necessary where Type II (275°F max) fluid temperatures are expected. This type will separate oxygen from the oil so that excessive oxidation and breakdown of the oil will not occur. The unseparated gas type can be used for Type II temperatures if an inert gas such as nitrogen is used as the pressurant.

In addition to oxidation problems, air entrainment can reduce the bulk modulus of fluid and may have serious effects upon the dynamics of flight control systems. Therefore, comprehensive and easily followed bleeding procedures are necessary. Adequate air bleed of the closed systems is required and, unfortunately, can be difficult, especially with cyclic utility requirements due to the "trapped oil" condition.

The spring-energized reservoir is not used in high-pressure systems except with volumes of approximately 25 in.^{3} or less because the weight penalty is excessive. Further discussion of reservoirs is found in par. 9-4.4.2.

9-4.2.7 Line Size Optimization

MIL-H-5440 requires "that the average velocity of fluid in pressure and return lines leading to the directional control valves *shall* not be in excess of approximately 15 fps, except where system analysis shows that proper functioning can be achieved even though the rate be higher". MIL-H-5440 also states, "Peak pressure resulting from any phase of the system operation *shall* not exceed 135% of the main system, subsystem, or return system operating pressure . . .". These specification requirements, plus the line length and minimum normal operating temperature, will serve to define the line size to be used in the hydraulic systems.

The general criteria for line sizing are:

1. Minimum fluid temperature for normal operation must be determined. A reasonable temperature level to use in calculations is 50°-70°F.

2. Maximum allowable line velocity V_{max} for flight control and utility systems is:

$$V_{max} = \frac{4050 - p_R}{50}\ \text{, fps} \qquad (9\text{-}9)$$

This meets the specification requirements. However, for utility subsystems where deviation from specification requirements is desirable, Eq. 9-10 may be used.

$$V_{max} = \frac{5000 \text{ to } 5500 - p_R}{50}\ \text{, fps} \qquad (9\text{-}10)$$

where

V_{max} = maximum allowable line fluid velocity, fps. (Fluid velocities are as high as 75-80 fps in vehicles in service.)

4050 = peak allowable transient pressure, psi, using 3000 psi normal system pressure

p_R = pressure immediately upstream of shutoff or control valve during maximum flow rate, psi (see Fig. 9-19)

50 = pressure rise per unit fluid velocity, psi/fps. A typical system may range from 40 to 50 psi/fps with fast valve closing. The larger value was chosen.

5000-5500 = allowable peak transient pressure range, psi. Some utility subsystems may have transient peaks in this range. The key to good service is testing to the level that will occur in normal service. The weight trade-off is the final criterion as to whether deviation from the specification is desirable.

Optimized line sizes are determined as follows for lightest power control and utility systems:

1. Power control functions. Define maximum (no load) rate, then size the pressure and return lines to approximately 2000 psi pressure drop at minimum normal operating temperatures (see Fig. 9-20). The remaining 1000 psi covers losses in the actuator valve and passages.

2. Utility functions. Determine the minimum load rate, then size the pressure and return lines and speed control valve to the pressure drop required to use that portion of the 3000 psi not required for the normal operating load (see Fig. 9-21). The minimum operating temperature *shall* be used.

3. Procedure.

a. Determine the fluid velocity in fps. If more

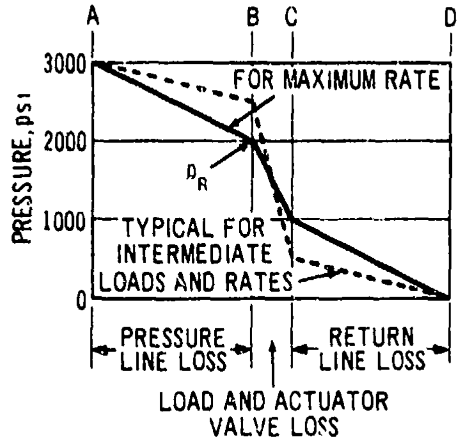

Fig. 9-19. Typical Flight Control System Pressure Loss Characteristics

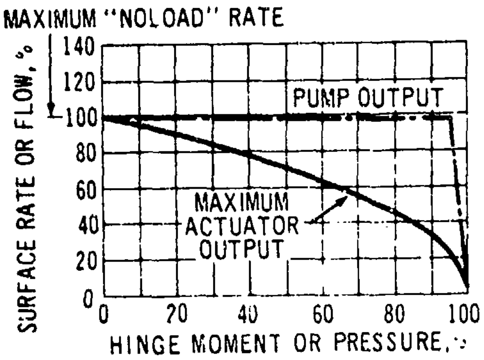

Fig. 9-20. Typical Flight Control System Characteristics

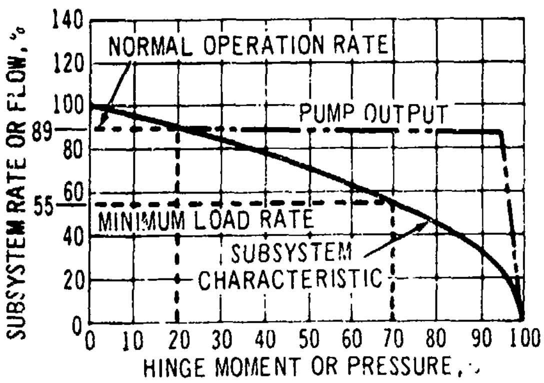

Fig. 9-21. Typical Utility System Characteristics

than one line size is used, the velocity for the line next to the valve or actuator is the important one.

b. Determine the transient peak pressure due to fast valve closing or actuator stopping. Multiply the velocity in fps by 50 and add to the reference pressure p_R, which is the pressure immediately upstream of the valve or actuator. Refer again to Fig. 9-19.

c. If the transient pressure exceeds the criteria specified, it becomes the constraining requirement. The line size must be increased to bring the transient peak pressure within limits.

4. General. Snubbing should be considered for the fast-operating utility actuators to reduce "water hammer" shock. Where slow solenoid valve operation (0.1-0.5 sec) is compatible with requirements, it should be specified in order to reduce "water hammer" effects.

All endurance tests, including those of power cylinders and utility actuators, must be planned to allow simulation of transient peaks expected in normal operation, especially if the peaks are above 4500 psi.

9-4.2.8 System Packaging Trade-offs

In the past, hydraulic systems have been built up by using individual components, with lines and fittings to connect or complete the system. However, the weight and volume were not optimum and the number of possible external leak points was excessive, so the trend has been toward packaging the system components wherever possible. The ultimate, of course, is a complete integration of pump, reservoir, actuator, filters, etc., as necessary to produce a complete system. This is acceptable for a system with a single function. However, from a weight and cost standpoint, the central system is more efficient where several functions or outputs must be supplied. In the central system, packaging of the power supply (pump, reservoir, etc.) and subsystem actuator and valve are used, with lines connecting the packages.

Packaging does have disadvantages in that damage or failure can cause replacement of a relatively complex and expensive manifold. This disadvantage can be minimized successfully by careful detail design and extensive fatigue testing. Cartridge components should be designed to allow rework in those seal areas that may be damaged inadvertently.

9-4.2.9 System Line and Fitting Trade-off

Permanent connections versus threaded, or reusable, connections are a primary trade-off. Permanent connectors can be used where line-to-line connections are required. Reliable connections of this type can reduce system leakage and maintenance as well as reduce system weight.

The alternatives available are brazed, welded, and swaged connections. The brazed and welded connectors require inert gas purge plus extensive ground equipment, and the weld or braze cycle requires close control including certification, X ray, and other means of quality control. The swage fitting does not require a purge and can be checked by a go-no-go gage. The swage fitting has another advantage in that it can be used with titanium, steel, and aluminum. The other two methods have not yet been applied successfully to aluminum.

The main disadvantage of the permanent type of connection is that it demands more design consideration to insure that its location will not interfere with access to other systems and components.

The line material trade-off choice is generally either steel or aluminum. The choice must be steel where there is a heavy vibration environment and/or infinite fatigue life is required.

The possible reusable fittings are MS flareless, AN flared, and several others available commercially; the AN flared fitting generally is used. The flared fitting has a stress raiser at the flare and can be a problem. The commercial fittings shown in Fig. 9-22 should be considered seriously.

9-4.3 SURVEY OF MILITARY SPECIFICATION REQUIREMENTS

A review of applicable Military Specifications is an essential part of the helicopter preliminary design phase. This paragraph summarizes the pertinent basic specification requirements. In general, the text discussion is based upon the requirements of MIL-H-5440. Virtually all of the Military Specifications associated with hydraulic systems and components are referenced in this specification. A thorough study and review of this document are recommended prior to beginning helicopter system and component design. Individual requirements are discussed in further detail in the paragraphs that follow.

9-4.3.1 System Type

System "type" is defined in terms of the design fluid temperature range of the system. There are two such requirements.

1. Type I: −65°F to +160°F temperature range
2. Type II: −65°F to +275°F temperature range.

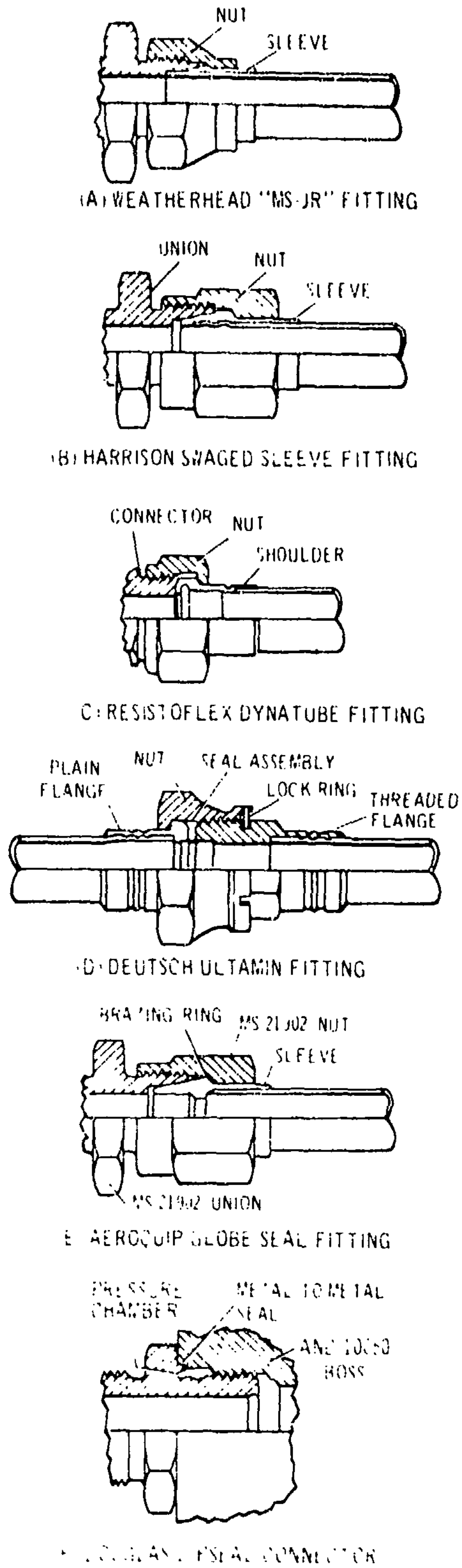

Fig. 9-22. Commercially Available Threaded Fittings

The establishment of one of these types is basic to the design of the system. This decision will affect the Military Specification requirements of many other items in the system such as packings, gaskets, reservoirs, accumulators, pressure regulators, universal fittings, hydraulic fuses, and shuttle valves. Further, if temperatures exceed the range for either type, additional equipment for cooling purposes must be considered.

In general, helicopter systems will be Type I; relatively low system flow rates are necessary and low-speed pumps are used.

9-4.3.2 System Class

System "class" is defined in terms of system operating pressure, as determined by the cutout pressure at the main pressure controlling device. The requirements cover two classes: 1500 psi and 3000 psi. The associated requirements pertaining to peak pressure (ripple or surge), back pressure, brake pressure, pressure regulation, reservoir pressurization, proof pressure, and burst pressure are dependent upon the class of system necessary.

9-4.3.3 System Operation

Pertinent aspects of system operation as covered by specification requirements are:

1. Hydraulic systems *shall* be capable of starting at −65°F, but need not deliver maximum performance.
2. A single system failure *shall* not cause a fire.
3. A malfunction *shall* not occur due to reduced flow or increased flow.
4. System operation *shall* not be affected by helicopter flight loads, including limit load factor.
5. Means *shall* be provided to remove entrapped air from systems.
6. Multiple-rotor helicopters using multiple pumps should have pumps driven by at least two rotors.
7. Flight controls require transmission-driven pumps in order that power will be available during engine-off operation.
8. For single-engine helicopters, backup capability is required by direct mechanical control or an emergency power source independent of the engine.

9-4.3.4 System/Component Design

Flight control systems *shall* comply with MIL-F-9490. System components *shall* comply with the requirements of MIL-H-8775 and applicable detail specifications. The applicable detail specifications for major systems are:

1. Pumps
   a. Variable delivery: MIL-P-19692
   b. Fixed displacement: MIL-P-7858
   c. Electric motor-driven: MIL-P-5954 or MIL-P-5994
2. Reservoirs
   a. Unseparated (gas and fluid in contact): MIL-R-5520
   b. Separated (airless): MIL-R-8931
3. Filters. MIL-F-8815
4. Accumulators
   a. Type I Systems: MIL-A-5498
   b. Type II Systems: MIL-A-8897
5. Actuating cylinders. MIL-C-5503
6. Check valves. MIL-V-25675
7. Relief valves. MIL-V-8813
8. Fluid. MIL-H-5606
9. Brake valves. MIL-B-8584
10. Fittings
    a. Flared: MIL-F-5509
    b. Flareless: MIL-F-18280
11. Tubing
    a. Steel: MIL-T-6845, MIL-T-8504
    b. Aluminum: MIL-T-7081
    c. Titanium: No approved specification available
12. Hose assemblies
    a. Rubber: MIL-H-8790, MIL-H-8795
    b. Tetrafluoroethylene: MIL-H-25579, MIL-H-38360.

9-4.4 DESIGN FEATURES

9-4.4.1 Pumps

The pumps generally used in helicopter hydraulic systems are the variable-delivery constant-pressure type (see Fig. 9-23). The pressure regulator is spring-biased so that when the control pressure is reached, the spring is compressed, allowing the regulator to direct flow to the stroking piston. The stroking piston positions the yoke as necessary to maintain the pressure within the design capability and to supply the flow demanded by a subsystem.

A variation is a pressure-sensitive, variable-delivery pump. In this case the pressure regulator is set at a low pressure, e.g., 500 psi, and the load pressure is fed to the regulator as necessary to increase the output pressure to meet load demands.

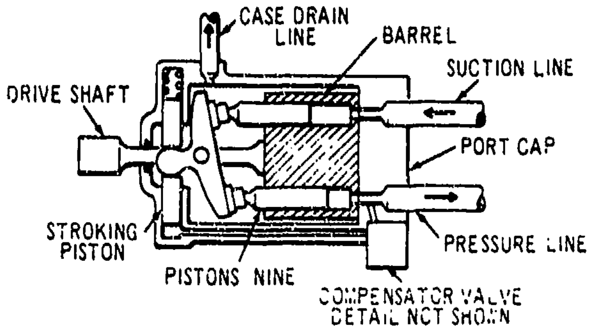

Fig. 9-23. Schematic of Variable-delivery, Constant-pressure Pump

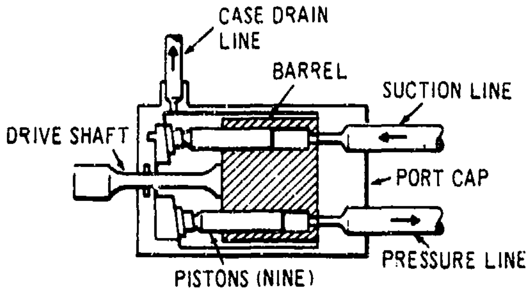

Fig. 9-24. Schematic of Fixed-delivery Pump

The fixed-displacement pump, shown in Fig. 9-24, can be used in conjunction with an unloader valve or relief valve. However, these systems can have poor response and excessive heat rejection characteristics.

Another type of pump, which is used primarily in engine-starting subsystems, is shown in Fig. 9-25. This hand pump, consisting of an oscillating piston and check valves, is employed to charge or recharge the accumulators used for engine starting.

9-4.4.2 Reservoirs

Hydraulic reservoirs *shall* be designed in accordance with MIL-R-5520 or MIL-R-8931. The former specification covers requirements for an unseparated, pressurized type where the pressurizing gas and fluid are in contact. The latter specification covers the requirements for a separated type where no fluid is in contact with gas.

Because of the problems caused by gas entrapped in the circulating loop of aircraft hydraulic systems, the separated type of reservoir is preferred. Reservoir pressurization may be accomplished by gas charge or bootstrap fluid pressure tapped off the pump output side. These two approaches are represented schematically in Figs. 9-26 and 9-27. The detail design aspects of the reservoir are covered in the paragraphs that follow.

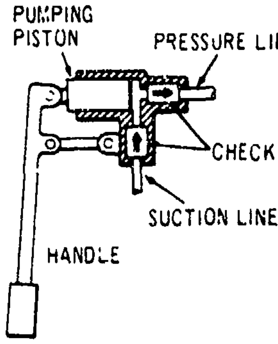

Fig. 9-25. Schematic of Hydraulic Hand Pump

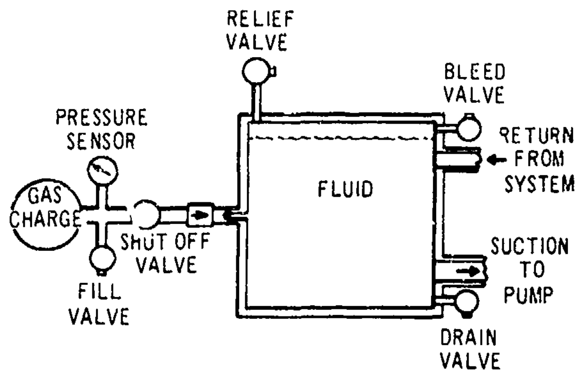

Fig. 9-26. Reservoir, Unseparated Type (Pressurized by Independent Gas Source)

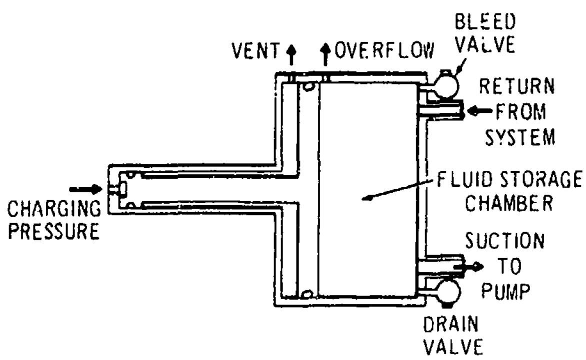

Fig. 9-27. Reservoir, Separated Type (Shown in Overflow Position)

The initial and basic considerations in the design of the hydraulic system reservoir are:

1. Fluid storage capacity. The total capacity *shall* include allowances for

   a. Total system fluid expansion and contraction due to system temperature changes
   b. Fluid compression
   c. Actuator differential fluid displacements
   d. Accumulator charging
   e. Nominal system leakage
   f. Expansion of lines and seals
   g. Quantity measuring fuses.

2. Pressurizing source. The source may be gas or fluid; pressure *shall* be sufficient to maintain pump suction line pressure at a level adequate to prevent cavitation. Another important function of the reservoir can be suction line fluid acceleration. Excessive cavitation at the pump inlet can cause damage and result in short pump life. The pressurization level should be such that adequate pressure is available to accelerate the suction flow to full flow rate in the 20-50 msec required for the pump to move from compensated to full stroke. Line flow losses, if significant, also must be taken into account.

3. Filling provisions. Filling must be accomplished by use of an external force device such as a hand pump or power pump.

4. Draining provisions. Reservoirs must be capable of being drained without breaking tube or hose connections.

5. Bleeding provisions. A bleed capability to remove gas trapped in the reservoir is mandatory.

6. Overflow provisions. Overflow must be provided for in such a manner as to eliminate the possibility of reservoir damage due to overfilling or pressure surges from the system return line. In particular, the overflow capability should be designed to allow sufficient overflow and with response adequate to insure that the fluid chamber pressure will not exceed design proof pressure under maximum return flow conditions, assuming the suction outlet is blocked.

7. Installation and orientation. The reservoir should be designed for installation as close to the system pump(s) inlet as practicable, thereby reducing pump suction line length and associated pressure loss. The reservoir should be oriented to allow air in the system to be entrapped in the reservoir so it can be bled overboard. Fig. 9-28 shows the proper orientation for the system reservoir. The location of the reservoir should be such as to minimize the possibility of combat damage. Further, the orientation should insure adequate capability to monitor fluid.

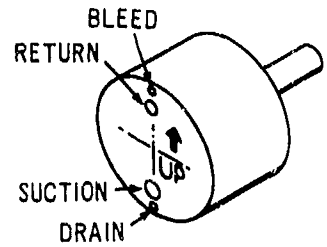

Fig. 9-28. Reservoir Orientation

8. Fluid level indication. Continuous, automatic, visual fluid level indication *shall* be provided. Use of a dip stick, or the manipulation or movement of any device to obtain indication of the level, is not acceptable.

9-4.4.3 Accumulators

System accumulators *shall* be designed in accordance with MIL-A-5498 (for Type I systems) or MIL-A-8897 (for Type II systems). The most common function of the hydraulic system accumulator is as a pulsation damper and/or pressure-volume compensator. Other possible uses include those of auxiliary or emergency pressure sources and leakage compensation devices.

As a pulsation damper the accumulator functions to reduce line shock due to sudden closing of valves, bottoming out of actuators, etc. As a pressure-volume compensator, the accumulator functions to eliminate possible excess system pressures due to thermal expansion or pump surges. Also, the accumulator can prevent sudden losses of pressure due to intermittent pump cavitation or sudden flow transients.

The primary type of accumulator used in helicopter hydraulic systems is the separated, gas-loaded design. Spring-loaded accumulators may be practicable in applications involving small displacement volumes and low pressures because the pressure produced varies with spring compression.

There are three categories of separated gas-loaded accumulators.

1. Bag-type (Fig. 9-29). This unit consists of a cylindrical shell with hemispherical ends, containing a rubber bladder as a separating device. The gas inlet valve is located at one end with the fluid port at the opposite end. The bladder sealing lips may be molded

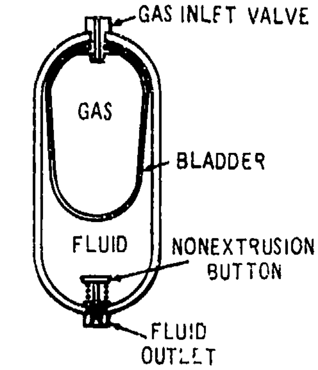

Fig. 9-29. Bag-type Accumulator

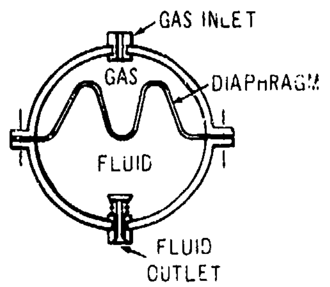

Fig. 9-30. Diaphragm-type Accumulator

or wedged in place by the gas inlet valve to provide a gas-tight seal.

A nonextrusion device is required at the fluid port. This may be accomplished by a spring-loaded button or poppet that will allow fluid flow in and out but will prevent bladder extrusion.

2. Diaphragm-type (Fig. 9-30). This type can best be achieved by a spherical shape that results in a minimum weight-to-volume ratio relative to other shapes of pressure vessels. The diaphragm may be compressed between the two hemisphere bolt flanges to provide a leak-tight seal. This type of accumulator also must contain a nonextrusion device. A primary difference between the diaphragm- and bag-type is that in the former the diaphragm flexes with little stretching, while in the latter case fluid is expelled by the stretching action of the bladder.

3. Piston-type (Fig. 9-31). This type consists of an accurately internal-finished, corrosion-resistant cylinder with end caps. A free-floating piston is incorporated with suitable dynamic seals to provide the gas-fluid separation. Compared to the bag or diaphragm types, this type is less effective as a pulsation damper in low-pressure systems due to the mass and friction associated with the piston. However, the piston type is required for Type II (275°F), 3000-psi working pressure systems.

All accumulator types must incorporate a means of insuring that the unit cannot be disassembled while pressurized. Various means of accomplishing this are possible by using any gas pressure charge remaining in the accumulator to operate a locking device that prevents disassembly until depressurization is accomplished.

9-4.4.4 System Pressure Relief Valves

The system pressure relief valve protects the system by dumping the high-pressure portion of the system into the low-pressure portion in case of a pressure buildup beyond the predetermined relief valve pressure setting, thereby preventing a rupture in the system.

The relief valve *shall* be sized by determining the operational requirements of flow, cracking (initial opening) pressure level, and pressure drop. The relief valve should be designed for maximum rated pump flow and a specific range of adjustable pressure settings. The pressure differential between cracking pressure and reseating pressure should be minimized to insure reseating at a pressure higher than normal system operating pressure. Military Specifications govern the maximum allowable cracking pressure as well as the maximum allowable pressure differential between cracking and reseating pressures. The relief valve pressure drop is essentially the cracking pressure at rated flow.

Relief valves may be direct-operated, pilot-operated or solenoid-operated. They may be loaded in the closed

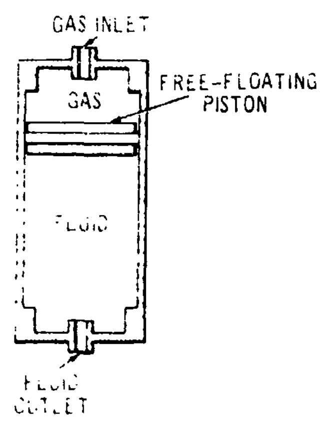

Fig. 9-31. Piston-type Accumulator

position by helical spring(s) or a Belleville spring. Further, they may be designed to insure that reverse flow cannot occur.

An important aspect to consider is the response time associated with opening to the rated flow position. If the response time corresponds to pump cutoff time, i.e., as the pump is adjusting from full flow to null flow conditions, pump output during cutoff can be dumped to the low-pressure portion of the system. Such a capability could permit deletion of the system accumulator. Because the accumulator is a possible source of injecting gas into the system, its deletion can be advantageous from system reliability, weight, and cost standpoints.

9-4.4.5 Filters

The requirements for filters generally used in helicopters are specified in MIL-F-5504 and MIL-F-8815. The former specification covers paper elements with a filtration capability of 10 micron particle size nominal. Although there is no absolute rating or limitation on the maximum particle size that can be passed through these throwaway elements, they incorporate a significant depth, thus providing a tortuous path and performing quite satisfactorily. They are applicable only for Type I systems (−65°F to +160°F).

The filter elements covered by MIL-F-8815 can be used in Type II systems (−65°F to +275°F). This specification provides for either metal screen type, recleanable elements or depth type, composite material throwaway. These units provide filtration capability down to 5 micron particle size nominal with an absolute cutoff on particle size of 15 microns. The filter efficiency is important and must be verified by test; glass beads generally are used. An efficiency of 95% may be required at the nominal filtration level. This means that no more than 5% of the glass beads of the nominal filtration level size may pass through the element.

The filter efficiencies and dirt-holding capacity requirements, including required tests, are described in detail in the noted Military Specifications.

The collapse pressure for the elements is very important and determines whether or not they can be used without a bypass feature incorporated. The Type I element has a 150-psi differential pressure capability and thus requires a bypass relief valve for protection. The Type II system element has a 4500-psi anticollapse capability and can be used in a nonbypass mode.

Differential pressure indicators are in general use. These indicators sense the differential pressure that occurs as the filter is loaded with dirt. The indicator actuates when the design differential pressure is exceeded, indicating that the element is dirty and must be replaced. The indicators may incorporate time delays to assure that transients will not cause them to operate. In addition, a thermal lockout is provided that prevents operation due to excessive differential pressures as a result of cold system startups.

The ground carts used with airborne systems usually have a significantly better filtration capability than the airborne system. The absolute rating of the ground cart filtration can range from 3 to 10 microns.

9-4.4.6 Static and Dynamic Seals

Component seals are covered by MIL-P-25732 and MIL-W-5521 and the glands in which they operate are covered by MIL-G-5514. The basic part number for the Type I system elastomer O ring seals is AN 6227. The Type II elastomer seal basic part number is MS 28775. A dash number is added to each basic number to define a specific size.

These standard seals are used without backup rings for pressures up to 1500 psi. Above this pressure, backup, antiextrusion rings are used. The backup rings also are covered by Military Specifications. The material is usually Teflon(R).

The gland dimensions and the allowable clearances between shaft and housing are specified in MIL-G-5514.

The so-called standard seals generally perform satisfactorily. However, flight control dynamic seal life can be a problem. To improve the life, many types of nonstandard seals have been developed with some success. The cross sections in Fig. 9-32 present some of the more common configurations. They generally are of Teflon(R) with standard O rings used as energizers.

There have been problems of "blow by" of the slipper seals due to pressure transients. This has been solved by grooving the seal on the pressure side (not shaft sealing surface) to allow the pressure to feed behind the slipper to help load the seal positively against the shaft surface.

Loaded Teflon(R) has shown an increased life over the virgin Teflon(R). For flight control actuators, Teflon(R) dual piston rings are used in conjunction with a spring energizer to give long life with minimum leakage (see Fig. 9-33).

Another approach to improving dynamic rod seal life has been the use of a dual redundant seal with a passage connected to the low-pressure portion of the system to allow draining of the cavity between the two seals. The dual seal configuration shows great promise for giving adequate service life.

The various standard and nonstandard seals can be used in any combination as illustrated in Fig. 9-34.

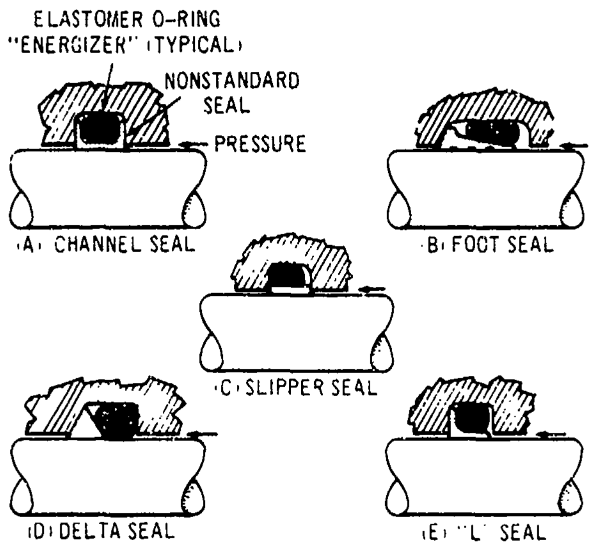

Fig. 9-32. Nonstandard Seals

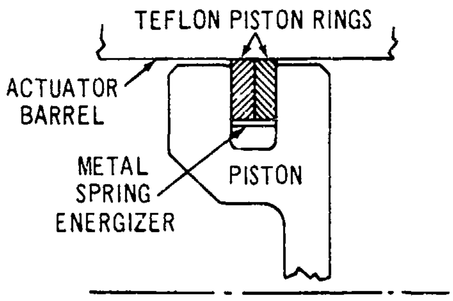

Fig. 9-33. Typical Piston Seal for Flight Control Actuators

Consideration can be given to the use of restrictors and check valves in the return line draining the seal cavity. The restrictor will limit bypassing leakage in the event of first-stage seal failure. The check valve could be used to prevent return transients from affecting the second-stage, low-pressure seal. These alternatives generally are not used, but may be necessary in special cases.

9-4.4.7 Distribution System

The distribution system is made up of the tubing and fittings necessary to supply flow to and return flow from the branch circuit functions. While this portion of the hydraulic system probably will not require detailed consideration during the preliminary design phase, particular aspects should be taken into account.

Because space is a critical factor in any aircraft configuration, it is advantageous to plan in detail the routing of lines and the component porting arrangements early enough in the design phase to allow change without excessive cost or schedule slippage. It may be necessary to plan a tubing mock-up to allow simulation of the actual tube and fitting installation in order to determine optimum location and geometry.

Maintenance capability also must be considered. The distribution system must be so located that maintenance personnel have proper access for installation, repair, replacement, and check-out capability.

Other important considerations applicable to the distribution system are discussed in par. 9-4.2.

9-4.4.7.1 Fittings

Fittings connected to removable components *shall* conform to the requirements of MIL-F-5509 (for flared tubing) or MIL-F-18280 (for flareless tubing) unless otherwise approved by the procuring activity. They must be connected and assembled in accordance with AND 10064 or MS 33566, respectively. Examples of flared and flareless tube fittings are shown in Fig. 9-35. Use of permanent tube fittings requires approval of the procuring activity.

The use of universal fittings on straight threaded bosses conforming to AND 10050 and MS 33649 should be minimized. However, when such is the case for Type I systems, only AN 6289 (nuts), MS 28778 (packings), and MS 28773 (retainers) parts may be used.

9-4.4.7.2 Tubing

Tubing material *shall* be corrosion-resistant steel conforming to MIL-T-6845 or MIL-T-8504, or T6

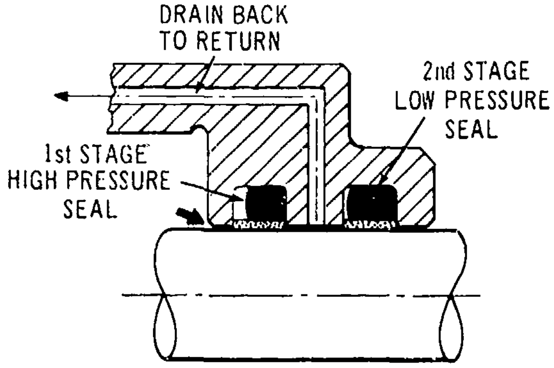

Fig. 9-34. Typical Two-stage Seal

aluminum conforming to MIL-T-7081. Titanium tubing *shall* be used only if approved by the procuring activity. Evaluation of titanium tubing for hydraulic applications is continuing and a specification for the manufacture and qualification of acceptable material is being prepared. Steel tubing may be seamless or electric welded, while aluminum tubing is seamless. Steel tubing is required in 3000-psi portions of the hydraulic system, in fire-hazard areas, or in applications requiring coiled tube or torsion tube installations for motion capability. Aluminum tubing may be used in portions of the system subjected to 1500 psi or less.

9-4.4.7.3 Hose Assemblies

High- and medium-pressure rubber hose assemblies *shall* be designed in accordance with MIL-H-8790 and MIL-H-8795, respectively, while high- and medium-pressure tetrafluoroethylene hose assemblies must conform to MIL-H-38360 and MIL-H-25579, respectively. Hose selections are required to conform to MS 33620.

Hose assemblies can be used advantageously in the following applications:

1. Where relative motion is required between components
2. In portions of the system subject to severe vibration
3. In areas with tight space limitations
4. In areas having troublesome hydraulic pressure pulses
5. In areas having complex contours
6. Where frequent connections and disconnections are necessary.

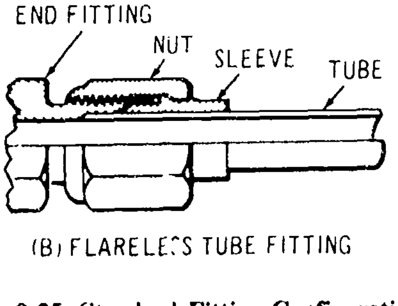

Fig. 9-35. Standard Fitting Configurations

9-4.4.8 Functional Subsystems

The functional subsystems are divided into flight controls and utility functions. These subsystems represent the various branch circuits that receive power from the central system.

9-4.4.8.1 Flight Control Subsystems

The flight control actuators can be reduced to two subassemblies, i.e., the control valve and the output device. The basic control valves are simple, spool and sleeve assemblies, as shown in Fig. 9-36. The input to the control valve can be a simple manual pilot input or may be accomplished by an electrohydraulic input that can be in series or parallel with the pilot input. This electrical control mode generally receives inputs from the stability augmentation system, which is required to keep the helicopter stable throughout the flight envelope. The series mode generally is used because it does not feed back its inputs to the pilot control. See Fig. 9-37 for a schematic of a single-system actuator with manual-series stability augmentation system control modes.

Dual redundancy may be required for adequate survivability or reliability. In this case the control valve would be a dual tandem lapped-fit slide valve. Significant care and effort are required to insure that the two sections of the valve are phased properly to avoid possible pressure intensification. This could occur if the pressure should be applied to two sections of the main ram and if one of the return sides is trapped. The peak out-of-phase pressure permitted for most dual units is 750 psi above the system pressure for a 3000-psi system.

The power boost actuator, which allows manual reversion in the event of hydraulic system loss, also may be used if proven optimum for the particular helicopter. The same types of control valve and output device as are on the full power unit are used. However, provisions are made to lock the control valve in neutral when pressure is lost. The fluid then is passed through a bypass valve from one side of the actuator to the other as it is moved by the pilot. After reversion from boost, the actuator is strictly a mechanical link in the flight control system.

The output device may be a linear ram of the balanced or unbalanced area type. In addition, the output could be either a rotary piston or vane motor (see Fig. 9-38).

9-4.4.8.2 Utility Systems

The utility functions generally are less complicated than the flight control actuators. The control valves

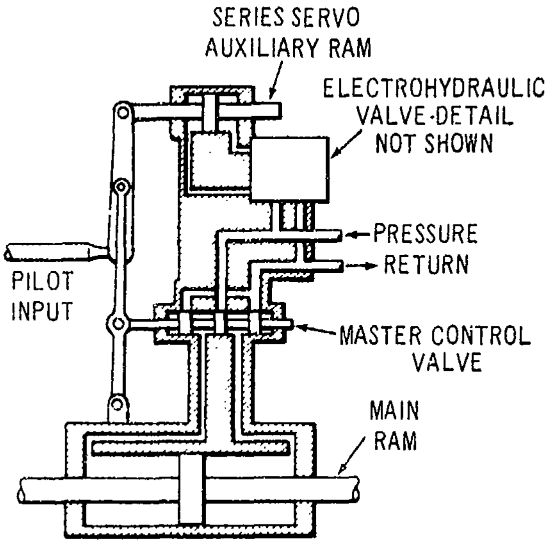

Fig. 9-37. Typical Manual-series, Servo-controlled, Single-system Actuator

can be either manual or electrical. The manual will be quite similar to the flight control type. A schematic of a typical pilot-operated solenoid valve used in utility subsystem applications is shown in Fig. 9-39.

The output devices are the same as for the flight controls: linear output rams, and rotary output piston and vane motors.

9-4.4.9 Engine-starting Systems

Hydraulic systems are used for starting auxiliary power units (APUs) and turbine engines. Such systems provide a self-contained start capability with the added potential of providing multiple starts. Fig. 9-40 presents a schematic of an engine-starting system in its simplest form. The accumulator is pumped up with the hand pump. The start valve then is operated to initiate the engine start sequence.

There are a number of variations on the simple system shown. The accumulator may be used to start an APU through a starter motor-pump. The APU driving the motor-pump then is used to start the main engine through another starter motor-pump. The engine mo-

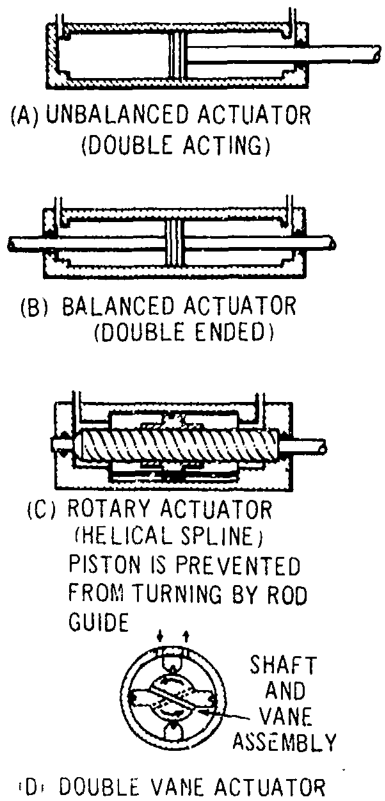

Fig. 9-38. Hydraulic Actuators

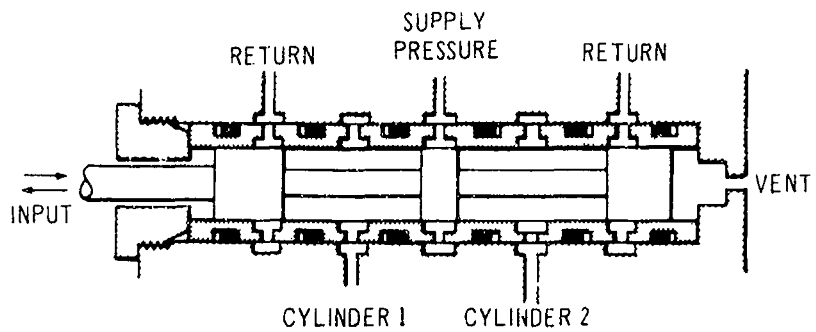

Fig. 9-36. Control Valve Spool and Sleeve Arrangement

tor-pump then functions as a pump for the duration of the flight. Still another alternative would be to start the APU and in turn to start the engine through gearboxes, clutches, and shafts. The starter motor-pump on the APU also can be used on the ground for operational system checkout and maintenance.

9-4.5 SAFETY, OPERATIONAL RELIABILITY, AND SURVIVABILITY CONSIDERATIONS

9-4.5.1 Routing the System

In routing the system and locating components, the benefits of redundant components and subsystems must be realized in actual system usage. The designer

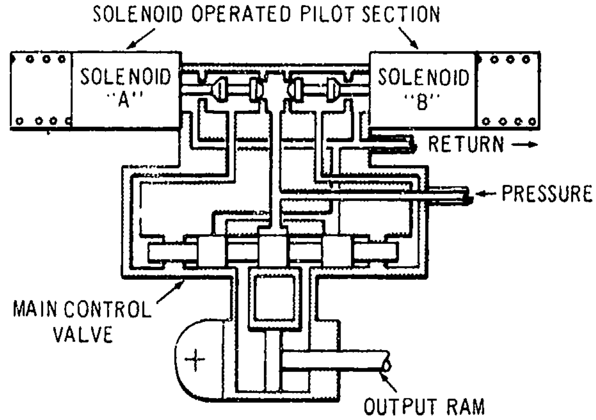

Fig. 9-39. Typical Solenoid-operated Control Valve

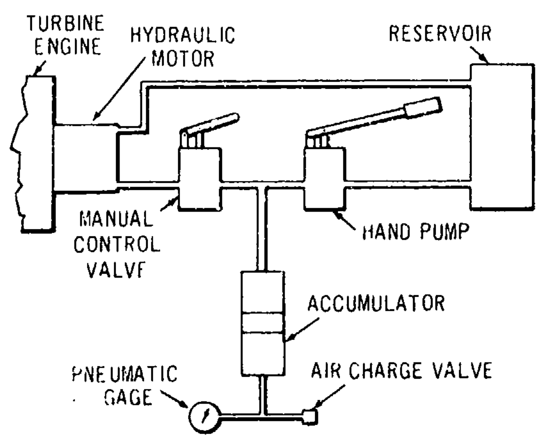

Fig. 9-40. Typical (Simplified) Hydraulic Engine-starting System

shall give adequate consideration to possible installation errors and the effects of such errors on the system. The effects of failures in equipment located adjacent to hydraulic system components should be reviewed. The following items represent some pertinent aspects that *shall* be considered:

1. Steel hydraulic tubing *shall* be used in power plant compartments. The firewall or flame-tight diaphragm *shall* be considered the dividing point for using other than steel tubing.

2. Where two or more lines are attached to a hydraulic component, the fittings *shall* be sufficiently different, where practicable, to prevent incorrect connection to the component.

3. Hydraulic lines should be located as remotely as practicable from exhaust stacks and manifolds and from electrical, radio, oxygen, and equipment lines. In all cases, hydraulic lines *shall* be located below the aforementioned to prevent fire from line leakage.

4. Where small-diameter tubing is used (less than 1/4 in. OD), particular care *shall* be exercised to insure the tubing will be installed, supported, and protected properly.

5. Drain or vent lines from the pump, reservoir, or other hydraulic components *shall* be incapable of being connected to other fluid systems such that mixture of fluid can occur in the components being vented or drained.

6. Emergency air lines *shall* be separated as far as practicable from the normal hydraulic lines connected to a particular component or subsystem.

7. System 1 and system 2 hydraulic lines to dual or tandem actuators *shall* be routed as far apart as possible.

8. The possibility of interconnection supply and return portions of the system *shall* be precluded.

9. Incorporation of components and lines in areas that are subject to being walked on by servicing and maintenance personnel *shall* be avoided.

10. Cushion clamps (Teflon(R) preferred) *shall* be used for all lines. Supports *shall* be placed as close as practicable to bends (but not in bends) to minimize overhang of the tube. In areas of high vibration, the line support spacing *shall* be reduced.

11. Provision *shall* be made for at least a 20% elongation in order to minimize line breakage during a crash.

9-4.5.2 Component and Subsystem Design

Many problems can be averted or alleviated by considering possible component failure modes and causes

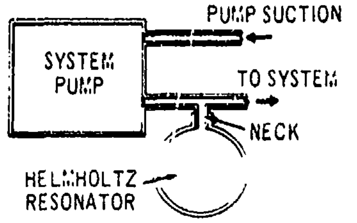

Fig. 9-41. Helmholtz Resonator

in the preliminary design phase and incorporating simple precautionary measures. These items should be considered:

1. Components should be designed so they cannot be installed improperly. A critical actuator *shall* be designed so that it is impossible to reverse the cylinder or cross lines during installation. Care *shall* be exercised in the installation of check valves; these valves, if installed backwards, can cause loss of an entire system.

2. A two-pump system should have a nonbypass-type, sufficiently large, filter in the case drain line of each pump.

3. Flexible hose may be used in the system for:
   a. Insulation against noise and shock
   b. Plumbing in close quarters
   c. Component connections that must move during component operation.

Hoses *shall* be used to connect pumps to the system. The designer should insure that hoses are not flexed in more than one plane of motion and are adequately protected against chafing, also that support clamps are not spaced too far apart (24 in. or less). Slack of 5-8% of the total length between components should be allowed.

4. System pressure pulsations resulting from pump ripple being intensified by system resonance can be attenuated by providing adequate elasticity to the system at the pump outlet port. Use of a Helmholtz resonator design, shown in Fig. 9-41, should be considered. Short, dead-ended lines near the pump *shall* be avoided.

5. Pump cavitation will occur if the reservoir pressurization is not sufficient to accelerate the column of oil in the suction line to a flow rate compatible with the pump displacement and within the response time of the pump compensator. This response demand condition is more likely to be a critical design condition than is steady-state flow.

6. Filters incorporated in valves—e.g., restrictors and T-valves—*shall* be larger in micron rating than system filters to avoid clogging. The normal size is 50 microns. Such component filters *shall* be pencil-type design, rather than flat disk-type, for higher dirt capacity at lower pressure drop. They *shall* be retained adequately, not pressed in, to avoid blow-out due to back pressure surges. In T-valves, central filtration *shall* be used to avoid differential flows as filter pressure drop increases.

7. Lubrication *shall* be provided for all critical joints. Dry lube *shall* be compatible with hydraulic fluid. Graphite-loaded grease in high-temperature applications can result in the grease drying up and leaving a hard graphite collection that can cause interference and contamination problems. Long lubrication paths sometimes result in frozen grease and blocked fittings.

8. Reservoirs *shall* incorporate air bleed vents that are high enough to optimize their capability to bleed the system. The suction outlet should be placed low to reduce suction line loss. The overboard relief flow capacity should be sufficient to prevent reservoir damage when such relief capability is needed.

9. Motors, pumps, or other components operating with different fluids or lubricants *shall* incorporate adequate sealing provisions in separation members. If two seals are used, adequate venting of the common chamber between the seals *shall* be provided for fluid drainage, thereby avoiding pressure buildup, interflow of fluids, and/or noncompatible seal deterioration.

10. Components incorporating face seals *shall* have adequate mounting bolts and lug strength to prevent flexing due to pressure surges. Flexing can cause seal or backup ring extrusion. Face seals can be avoided entirely through the use of transfer tubes.

11. In designing poppet valves, aluminum seats *shall* not be used. Insufficient seat area, combined with high seating forces, can cause Brinelling of the seat. Excessive leakage and/or sticking of valves can result.

12. Snap rings *shall* not be used in components where, under loaded conditions, snap ring failure can cause slippage of internal parts or cause a component to come apart completely.

13. Cold flow materials used for poppet seats *shall* be supported sufficiently to prevent excessive

creep. Malfunctions due to change in poppet travel and/or leakage can result if this is not done.

9-4.5.3 Instrumentation

Military Specifications require, as a minimum, that the hydraulic system incorporate a pressure indicator common to each pump in a system and that each system incorporate a low-pressure indication. These indications must be available for readout in the cockpit. The designer should consider the actions that may be taken by the crew if they are given an indication of an abnormal condition. This is important in that erroneous indications due to an instrumentation failure may cause unnecessary and perhaps hazardous actions to be taken in critical or combat situations. Such an evaluation by the designer may result in consideration of redundant or alternate means of indicating critical system parameters.

9-4.5.3.1 Protection of Pressure Sensors and Transmitters

It is a good design practice to include fuses and snubbers at the upstream ends of the lines leading to pressure sensors or transmitters to protect the devices from surge pressures. Rate- or quantity-measuring fuses may be incorporated to shut off fluid loss overboard in the event of line rupture. Snubbers act to damp forces resulting from pressure surges in dead-ended lines. A possible alternative is to install pressure sensors into fittings in the system line instead of in an appendant line. If this is not possible, the snubber and fuse should be installed at the teeoff point.

9-4.5.3.2 Pressure-sensing Elements

Care should be taken in the design of pressure transmitters with Bourdon tubes and rack and pinion-type gearing. These are susceptible to bearing wear caused by vibration or pressure fluctuations. Helical sensing elements are preferred. Sensing elements may be encased in a compatible fluid to provide vibration and shock protection. A blowout plug should be incorporated into the case to prevent pressure buildup and/or rupture in the event of a leak in the sensing element.

9-4.5.4 Survivability Considerations

Helicopters that may be subjected to combat conditions should be designed to minimize the effects of combat damage. The basic objectives of hydraulic system design in improving chances of survivability in combat conditions are:

1. Minimize the chances of critical equipment being hit by projectiles and fragments

2. Reduce chances of total system loss in the event of system damage

3. Minimize the chances of fire due to large quantities of fluid being dumped onto adjacent equipment or into a fire.

The initial step in minimizing the chances of combat damage is to determine those portions of the system that, if damaged, could cause loss of the system or of a flight-critical function. Critical equipment then can be located and oriented in such a manner as to reduce the area exposed to enemy projectiles. Early in the design phase the use of armor protection should be considered. The design may include armor as an integral part of system/component design or as an add-on, replaceable feature (parasitic armor).

A trade-off study should be conducted to determine whether integral or parasitic armor, or a combination of the two, should be used. In most cases the integral armor design will be lighter than parasitic armor because it will replace a portion of system/component structure. However, integral armor likely will be more costly and more difficult to repair. Parasitic armor has the advantage of allowing addition or removal as determined by the type of mission required. While parasitic armor is easier to repair, its protection envelope may be reduced due to weight considerations. Also, ballast may be required to maintain the proper vehicle center of gravity. The design and evaluation of armor installation are discussed in further detail in Chapter 14, AMCP 706-202 and Chapter 9, AMCP 706-203.

The primary concern in reducing the chances of system loss and fire hazard is to reduce the possibility of hydraulic fluid leakage. The various design approaches and techniques described below should be employed:

1. Check valves should be incorporated in all branch circuit return lines as close to the main return trunk line as possible. A typical arrangement is shown in Fig. 9-42. This will allow isolation of the leak to the particular branch circuit by preventing back flow from the main return line into the leaking branch circuit.

2. Dual actuators that are manifolded in tandem or side-by-side should incorporate a "rip stop" as a separator between the two sections of the actuator that are powered by different systems (see Fig. 9-43). This reduces the chance of structural damage propagating from one side of the actuator to the other; such structural damage otherwise could cause total loss of function. The two system halves may be mated by welding or brazing, or may be a bolted configuration. The latter design allows separate maintenance of the two halves, and provides an easier method of assembly.

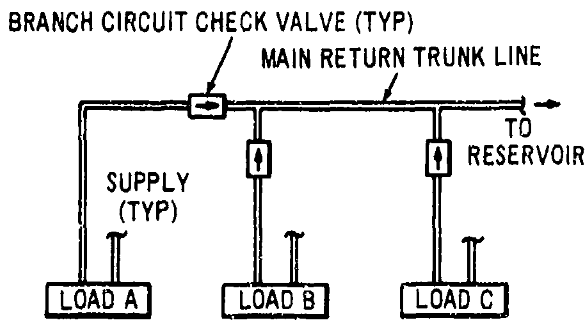

Fig. 9-42. Branch Circuit Check Valves Protect Main System in Case of Branch Circuit Leak Failure

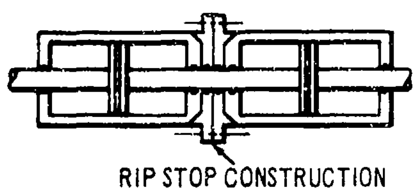

Fig. 9-43. Rip Stop for Dual Actuators

3. Various techniques should be considered for automatically sensing and isolating leaks in hydraulic system branch circuits. Several design approaches may be used depending upon the functional and flow characteristics of the particular branch circuit. These include

   a. Rate-measuring fuse (Fig. 9-44). Designed to sense and shut off flow rates in excess of maximum circuit design flow. Consider for use with cyclic (on-off) constant flow functions such as doors and winches. These fuses also may be used for dead-ended line functions, such as pressure transmitters, which are subjected to system pressure surges. Design should include protection against inadvertent shutoff. Such protection may be in the form of a built-in time delay that is sized according to transients expected in the particular circuit.

   b. Quantity-measuring fuse (Fig. 9-45). Designed to sense a predetermined volume displacement of a particular function. Consider for use in cyclic (on-off) functions of relatively low volume displacement. Insure that the volume measuring element has sufficient safety margin to prevent premature shutoff.

   c. Flow comparator (Fig. 9-46). Designed to measure flow in supply line and return lines. These two parameters are compared by using flow force in each line to balance a comparator device. As long as the supply and return flows occur in accordance with a predetermined ratio, the comparator allows the system to function. If an external leak occurs, an unbalance in flow force is sensed by the comparator, and subsequent shutoff occurs. This concept is applicable to continuous variable-flow functions such as the cyclic, collective, and directional actuators. Because comparator devices are sensitive to system dynamic conditions, some protection must be provided against inadvertent operation. This can be in the form of a time delay, although it should be noted that the greater the delay, the less sensitive the device will be to actual leaks. The device may be designed to be passive during normal system operation and automatically activated to an "armed" mode if a leak occurs in the system. In a system with several such devices, arming of all units could occur upon receiving a low-level signal from the system reservoir. The leaking circuit then would be shut off automatically while the remainder of the system continued operation.

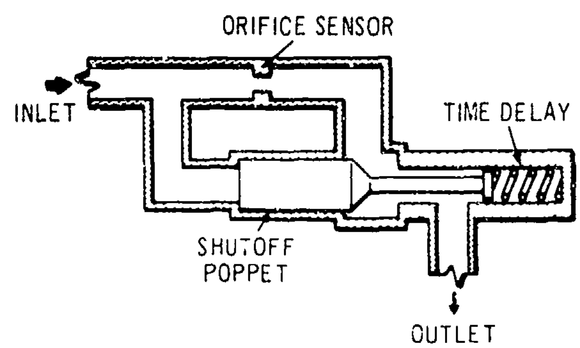

Fig. 9-44. Elements of Rate-measuring Fuse

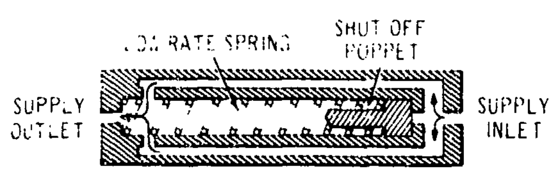

Fig. 9-45. Quantity-measuring Fuse

d. Reservoir level sensing (Fig. 9-47). Another method of system isolation involves use of reservoir level to sense and shut off leaking circuits. An external leak of any magnitude in any subsystem branch circuit causes a decrease of the reservoir supply. When the supply has depleted below the normal minimum level, a signal can be provided, electrically or mechanically, to each circuit shutoff valve in sequence until the leaking circuit is shut off. The spacing of each circuit shutoff sensing point on the reservoir must take into account the normal fluctuations in reservoir level when a particular circuit is shut off. This will prevent unnecessary cycling of adjacent circuit sensors. This approach has the disadvantage of requiring a fixed amount of fluid to be dumped overboard prior to leakage isolation. Also, during the leak search process a nonleaking system may be shut off for a period of time. It is important to note that this approach should be used with redundant systems only. However, for such configurations, where each system has only two or three branch circuits, it provides positive leakage isolation that essentially is insensitive to system flow and pressure transients. Pilot override capability is incorporated easily as is manual operation capability for ground servicing.

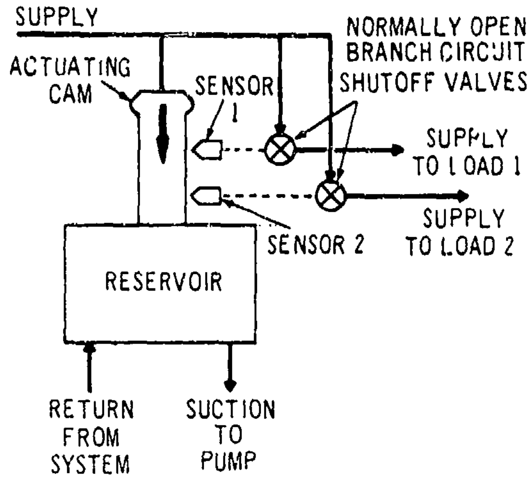

Fig. 9-47. Reservoir Level Sensing and Circuit Shutoff

9-4.6 HYDRAULIC SYSTEM POWER ANALYSIS

A determination of peak and steady-state hydraulic power requirements is required as a part of the preliminary design of any helicopter. A typical example is outlined and means of arriving at a solution presented. This discussion covers the two basic categories of hydraulic power usage—flight controls and utility functions.

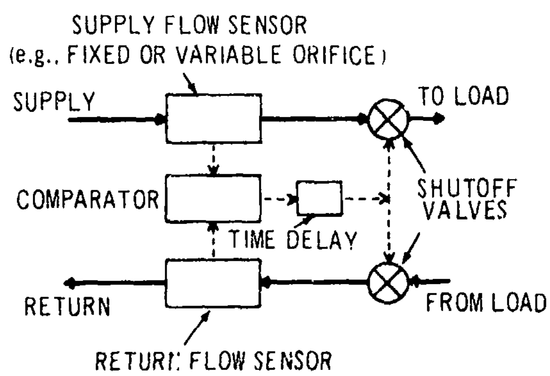

Fig. 9-46. Flow Comparator

9-4.6.1 Flight Control Hydraulic System Analysis

The typical flight control system must provide energy or power to meet demands in terms of force on a moment arm (basic dimension, in.-lb) and rate of motion (basic dimension, deg of rotation per sec). The force levels and the rates of motion associated with these force levels must be defined as closely as a preliminary design effort will permit. Fig. 9-48 shows a graph in which various critical combinations of force and rate are plotted. The rate (deg/sec) of motion is the ordinate and hinge moment (in.-lb) is plotted on the abscissa. The rate is convertible easily to flow in gpm and the hinge moment can be converted to hydraulic pressure in psi. The scales, therefore, include these alternate legends, and are dimensionless.

The hydraulic system energy transmissibility characteristics and the pump output characteristics can be plotted conveniently on the graph showing the system output requirements.

The pump output characteristics shown in Fig. 9-49 depict a pump with an approximate 5% reduction in pressure at maximum flow capability.

The system energy transmissibility curve and the pump output curve start together at the pump compensated-actuator stalled point. As hinge moment is reduced and rate capability increases, energy is lost in the

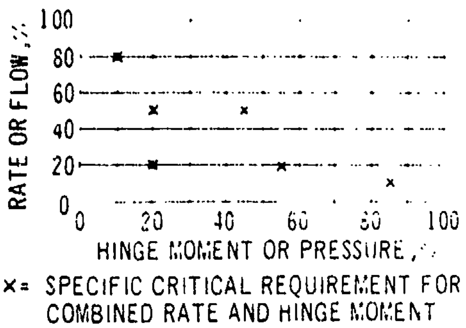

Fig. 9-48. Typical Critical Performance Points for a Flight Control System

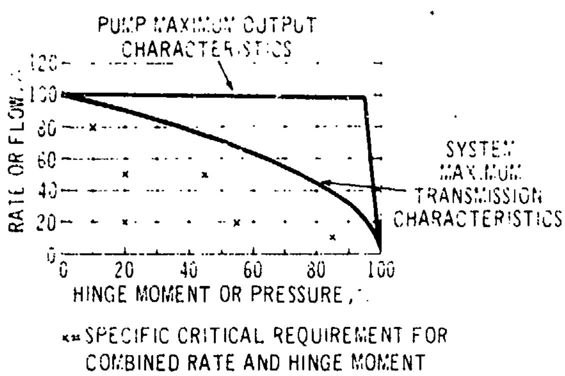

Fig. 9-49. Typical Performance Points and System Characteristics

lines and the energy transmission curve diverges from the pump output curve. If the line sizes are optimum from a weight standpoint, the energy at the maximum no-load rate will be dissipated completely in line and control valve loss. This loss is distributed such that 1/3 will occur in the pressure line, 1/3 in the control valve, and 1/3 in the return line.

If for water hammer reasons, etc., (see a detailed discussion on line sizing in par. 9-4.2.7) the line size cannot be optimized, then the nonoptimum system energy transmissibility curve will represent the system and, normally, the system can operate at this point if a larger pump is available to handle combined subsystem demands.

It should be noted that the output hinge moment discussed is the limit hinge moment. The structural strength requirements should be 1.5 or more times the limit hinge moment for adequate dynamic loading and fatigue life margins.

A further consideration must be assessed. The energy transmissibility curve is a classic, but usually unreal, definition of actual subsystem capability. It states that, for each point, the system has the rate capability corresponding to a particular resisting hinge moment load. However, the load actually will be a steady-state trim load such that as the surface moves, the load increases, causing a rate-of-motion reduction. The actual subsystem rate-load capability is shown in Fig. 9-50. For a given initial trim load, the subsystem has an average rate capability in reaching a new position demanded by the pilot. The rate associated with that hinge moment for the new steady-state null point is the average rate at which the control surface(s) can move from the old position to the new position.

The peak hydraulic power output requirement can be calculated in a straightforward manner. The maximum pump output power $P_{p_{max}}$ is the point so noted in Fig. 9-50, at the knee of the pump pressure flow curve. The equation for pump output power P_p is

$$P_p = \frac{Qp}{1714}, \text{ hp} \tag{9-11}$$

where

Q = fluid flow, gpm
p = pressure, psi
1714 = constant that adjusts dimensions

For a given system, Eq. 9-11 defines the peak pump output horsepower. The input power to the pump may be determined by dividing the result by the pump efficiency. This can range from 0.85 to 0.90 depending upon the fluid temperature and viscosity, pump speed, and fluid density.

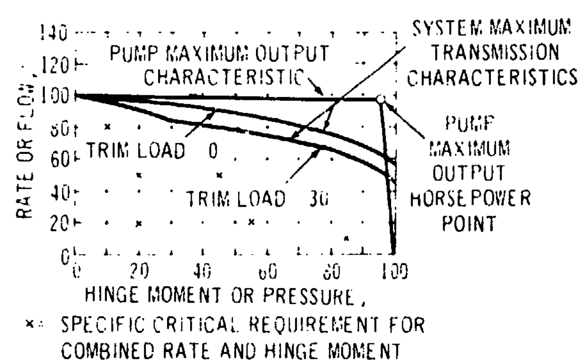

Fig. 9-50. Typical Flight Control Hydraulic System Characteristics

Another more direct means of calculating the pump output is through the use of Eq. 9-12

$$P_p = \frac{TR}{63{,}025} \text{ , hp} \tag{9-12}$$

where

T = torque, in.-lb
R = rate of angular motion, rpm
63,025 = constant that adjusts dimensions.

This takes care of one subsystem. Where more than one subsystem must be supplied, consideration must be given to the combined requirements of the subsystems. This can range from the total combined maximum power peak of the subsystems to one-half to two-thirds of that combined peak. In the case of the reduced peak power, the individual versus combined requirements must be assessed and the factor thus determined then is applied to the combined peak. For example, the combined operating power may be two-thirds of the sum of the individual system peak requirements. This approach provides a close approximation by which to size the pump and assess the impact of its requirements on the power source.

The fluid temperature used in determining system line-loss characteristics is usually the expected normal minimum operating temperature. Temporarily reduced system performance generally is accepted after start-up during a cold soak. The system usually will be operating normally within a few minutes.

It also should be noted that the variable-orifice actuator control valve allows operation at points below the characteristic curve in that it will dissipate the extra energy not required for that specific load-rate condition.

9-4.6.2 Utility Hydraulic System Analysis

The power analysis for utility systems is similar to that for the flight control subsystem. The critical rate-hinge-moment points must be defined, and in this case generally are associated with significant loads, as opposed to the flight controls where a critical point may be near no load. Fig. 9-51 presents a typical rate versus hinge moment plot showing the operating and design points. Rate control is generally of a fixed type such as a restrictor or a flow regulator. The example shown represents a restrictor rate control. The design point, identified as critical, is for a maximum load condition that will not exist normally. The subsystem usually will operate at the noted normal operating point. For this type control, the system will operate along the system

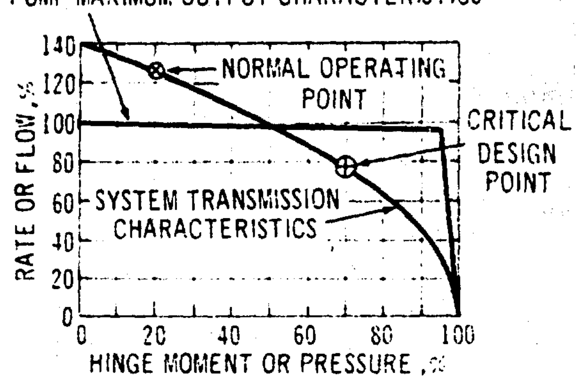

Fig. 9-51. Typical Utility Hydraulic System Characteristics

transmission characteristic curve. As with the flight control subsystem, the ordinate and abscissa will represent flow and pressure, respectively, to complete the description of the pump requirements and system energy transmission characteristics.

The same techniques used in determining peak hydraulic power requirements for flight controls also apply for utility systems. Where there are several utility functions, the pump output will be derived by determination of the sum of the requirements for those subsystems that may be required to function simultaneously.

9-5 PNEUMATIC SYSTEMS

9-5.1 COMPARATIVE CHARACTERISTICS BETWEEN PNEUMATIC AND HYDRAULIC SYSTEMS

Pneumatic systems use a fluid for transfer of power and, in many aspects, are similar to hydraulic systems. A brief comparison of the two types of systems and the significant differences is contained in subsequent paragraphs.

Typically, a pneumatic system uses air or nitrogen as a fluid medium because these are the most readily available gases.

The limiting characteristics for pneumatic systems are attributed primarily to three factors: poor lubricity, low fluid viscosity, and low bulk modulus. The possibility of entrained moisture in the gas also presents a potential icing problem in low-temperature operation. However, this problem can be minimized through proper design.

The most limiting factor for a hydraulic system is temperature. Because most hydraulic systems are recirculating, the result is a temperature buildup, or increase, caused by pressure drops through lines and restrictions. This may result in a requirement to cool the hydraulic fluid. Gases are usable at high temperatures (above 500°F) where commonly used hydraulic fluids deteriorate. On the other hand, hydraulic fluids have good lubricity, acceptable fluid viscosity, and a high bulk modulus.

A comparison of the density of the two fluid media shows that air is considerably lighter than hydraulic fluid. Liquids are considered incompressible (hydraulic fluid is compressed about 2% at 3000 psi), with hydraulic fluid weighing about 52 lb/ft^3, compared with dry air compressed to 3000 psi weighing about 17 lb/ft^3. A direct comparison of a pneumatic circuit and a hydraulic circuit operating at the same pressure shows a substantial weight saving in the fluid for the pneumatic system.

The bulk modulus of the fluid is a measure of the fluid compressibility and is expressed as the reciprocal of compressibility. This factor is important where servomechanisms are considered, with the natural frequency of the system being proportional directly to the square root of the bulk modulus of the fluid. The compressibility of a pneumatic fluid provides a self-damping characteristic, making the pneumatic system less susceptible to shock that results from rapid fluid acceleration or deceleration. With the high bulk modulus, the hydraulic system provides a much stiffer system than the pneumatic; innovations developed to improve pneumatic stiffness have the undesirable effects of higher weight, increased size, and lower reliability.

The lubricity of gases is very poor compared with the lubricity of hydraulic fluid. Thus, the wear of mating surfaces is a problem in pneumatic systems because lubrication is not provided by the working fluid.

9-5.2 TYPES OF PNEUMATIC SYSTEMS

Three types of commonly used pneumatic systems are an airborne compressor-charged system, a ground-charged storage bottle system, and a hot-gas, solid-propellant gas-generator system.

9-5.2.1 Airborne Compressor-charged System

The advantage of a compressor-charged system is that a small compressor-charged storage bottle can supply a system having high peak loads and small average loads. This system is suitable where a normally operating subsystem would be used only one or two times during the flight, with sufficient time in between operations for the compressor to recharge the storage bottle. A typical airborne compressor-charged system is shown in Fig. 9-52. The compressor may be direct-driven by the engine through a gear train and clutch, by a hydraulic motor powered by the hydraulic system, or by an APU. The compression of the air generates heat, and a means of cooling should be provided. A cooling fan is the most common heat dissipator and probably adds the least weight of any method. The inlet air to the compressor should be filtered and may be either regulated engine bleed air or ambient air. If engine bleed air is used, provisions may have to be made for precooling the inlet air. Pressure switches are installed downstream in the system to control the compressor. In a 3000-psi system the compressor would be designed to operate when system pressure drops to 2600 psi and to shut off when the system is recharged to 3000 psi. The removal of moisture from the air is essential to minimize corrosion and reduce maintenance. This may be accomplished by placing a moisture separator at the outlet of the compressor, removing approximately 95% of the moisture, and then placing a dehydrator immediately downstream of the moisture separator to reduce the moisture content further. As the dehydrator is a replaceable item, it should be large enough that frequent replacement is not required. Ground testing and charging provisions are necessary to eliminate the necessity of operating the compressor on the ground. Fittings for such ground support equipment (GSE) are installed downstream of the moisture-removing equipment. A 25-micron filter should be placed in the ground charging line; the storage volume must be sufficient so that the required duty cycles of all subsystems will be obtained. The compressor size *shall* be such that recharging of the storage volume occurs within the required time between subsystem duty cycles. Check valves isolate subsystems from each other and the compressor. Direct-reading pressure gages *shall* be provided where necessary for maintenance personnel to check the system pressure. Pressure relief valves are required in the main supply system and each subsystem to prevent overpressurization due to temperature changes.

9-5.2.2 Ground-charged Storage Bottle System

The ground-charged storage bottle is used most commonly as an emergency or backup unit for a hydraulically powered system. This type of system (see Fig. 9-53) is a simple one, with a minimum number of components. Selection of components is based primarily upon flow rate and pressure requirements. Where a

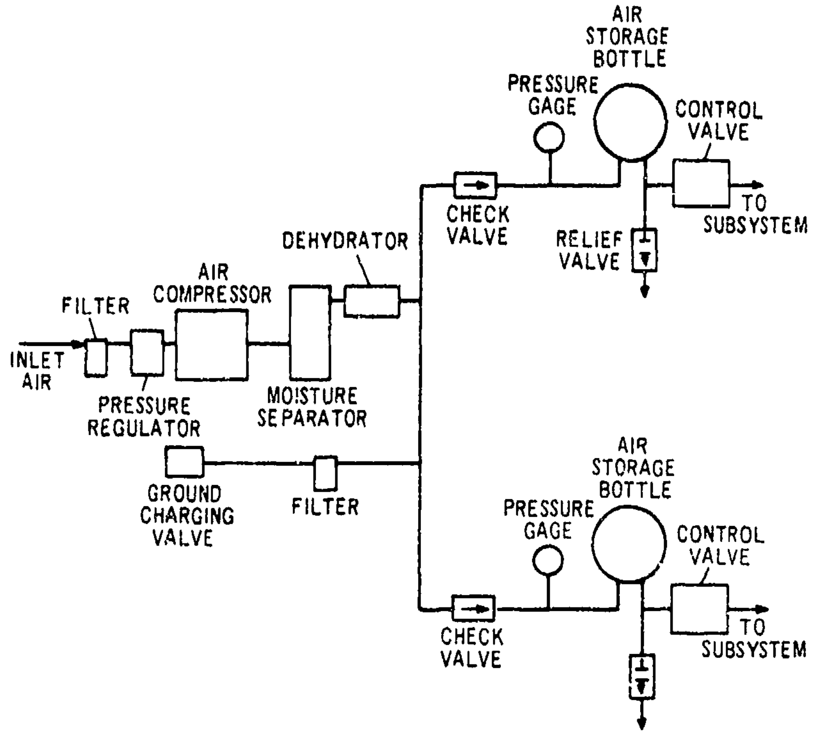

Fig. 9-52. Airborne Compressor-charged System

shuttle valve connects the emergency system to a normally actuated system, the shuttle valve *shall* be designed so that a force balance cannot be obtained on both inlet ports simultaneously, causing the valve to restrict flow from the outlet port. Shuttle valves normally are built into the component; if installation problems exist, it is permissible to have a length of rigid line between the component and the shuttle valve, provided that the rigid line and shuttle valve are attached firmly to the component. A standard pressure gage is installed to allow a check of the storage bottle pressure. The actuation of the control valve may be mechanical or electrical, but electrically operated valves should be provided with a mechanical override. The control valve activation means, such as push-pull rods or cables, should be designed to require little or no adjustment and to prevent overtravel or undertravel by use of internal or external stops. A relief valve is necessary to prevent overpressurization due to temperature changes.

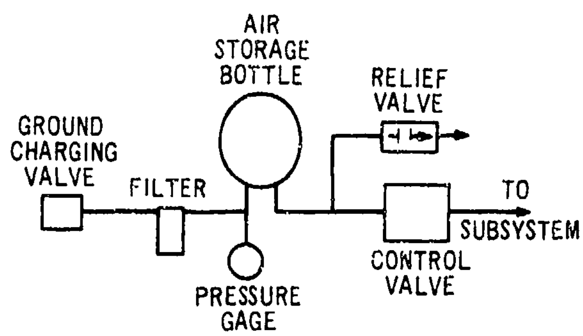

Fig. 9-53. Ground-charged Storage Bottle System

9-5.2.3 Hot-gas, Solid-propellant Gas Generator Systems

Solid-propellant gas generation systems are becoming more widely accepted as emergency systems (see Fig. 9-54). This type of system consists of a solid-propellant gas generator that is ignited through a thermal battery or electrical igniter; the pressure to the actuated subsystem is controlled by the pressure relief valve or back pressure regulator. This type of system must be

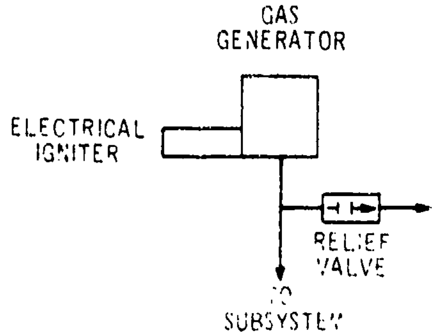

Fig. 9-54. Hot-gas, Solid-propellant Gas Generator

refurbished after usage because very high temperatures occur. Once the generator is ignited, the pressurization rate and duration are fixed. This system does not require periodic checking, and it essentially is unpressurized until used.

9-5.3 SYSTEM INSTALLATION

The system installation considerations for pneumatic compressor-charged systems and ground-charged storage bottle systems are similar. The systems *shall* be as simple and foolproof as possible with respect to design, operation, inspection, and maintenance. Leakage is traditionally one of the main problems for fluid systems. Probably the greatest source of pressure loss in pneumatic systems is the fitting joints. Pneumatic leaks are difficult to locate; however, they may be detected visually by applying special soap solutions and observing bubbles. Leakage can be controlled by careful manufacturing and maintenance procedures, but this means greater cost. Good sliding and rotating seals are difficult to obtain in pneumatic systems because of the low viscosity and poor lubricating qualities of the gases.

The pneumatic component installation *shall* be designed to accommodate the most adverse dimensional and operational conditions. The components should be located at high points in the line to preclude freezing of collected moisture, which will cause the component to malfunction. All components that require periodic servicing should be located in an accessible area. Drain fittings should be installed at low points in the system to permit removal of condensed moisture.

Pressure line tubing *shall* be of corrosion-resistant steel. Alternate materials may be employed in return lines. Consideration should be given to the newer type of tube-joining methods. Permanent joining of tubes by brazing, welding, or swaging joints that do not require disassembly can increase reliability and eliminate a number of potential leak points. Designing for motion of tubing helps to avoid use of swivel joints and hoses, thereby increasing reliability. Relative motion through use of coiled or torsion tubing is allowable in both steel and titanium tubing. Consideration must be given to the combined stress resulting from the Bourdon effect, torsion, tension and compression, and vibration.

9-5.4 SAFETY, MAINTENANCE, AND RELIABILITY

Due to the compressibility of the gas, high-pressure pneumatic systems have the inherent characteristic that structural failure can result in a large and destructive explosion. Therefore, the design should create conditions of pressure and temperature that are not conducive to combustion or explosion. Large cavities in which high-energy combustions or explosions can occur should be avoided, as well as lubricating oils or greases that can induce combustion or explosions. Where the possibility of combustion or explosion cannot be precluded, steps should be taken to minimize them. These steps may include flame arresters or blowout disks.

Where an emergency source is provided for a subsystem, the normal and emergency lines *shall* be separated from each other as far as is practicable so that the possibility of both lines being ruptured by a single projectile is reduced to a minimum. Additionally, the two lines *shall* be so designed that their modes of vibration differ sufficiently to preclude the possibility of both failing due to vibration-induced stresses. Advantage should be taken of the protection against combat damage afforded by heavy structural members and armor plate. Insofar as practicable, pneumatic lines *shall* be remote from personnel stations. Utmost consideration *shall* be given to protecting crew members in case of rupture of storage bottles from gunfire (wire-wound bottles should be used).

Air storage bottles *shall* be accessible and rechargeable without being removed from supporting brackets. If it is essential that flexible hose be used in a pneumatic system, the installation should be so designed that disconnection at the end of the hose downstream of the storage bottle is precluded to the maximum practicable extent. This is to prevent the loose end of the hose whipping about dangerously when pressure is released. Replaceable items such as dehydrators *shall* be easily removed and replaced. Use of components that could be reversed and thereby cause the system to operate improperly *shall* be eliminated to the maximum possible extent. Ground charging of storage bottles should be accomplished slowly because fast charging is a nearly adiabatic process and the temperature within the storage bottle may rise to more than 400°F.

9-5.5 SYSTEM ANALYSIS

To make an accurate pressure drop analysis of a compressible fluid flowing through tubing and components requires a knowledge of the relationship between pressure and specific volume. This is not determined easily. The extremes that are considered are adiabatic flow (Eq. 9-13) and isothermal flow (Eq. 9-14).

$$pV^{\gamma} = \text{constant} \tag{9-13}$$

where

p = pressure, lb/ft^2
V = volume, ft^3
γ = ratio of specific heat at constant pressure to specific heat at constant volume c_p/c_v (This value is 1.4 for air or nitrogen.)

$$p_1 V_1 = p_2 V_2 \tag{9-14}$$

where

Subscript 1 = initial condition
Subscript 2 = final condition

Adiabatic flow usually is assumed in short, perfectly insulated piping where no heat is transferred to or from the pipe except for the minute amount generated by friction. Isothermal flow is more descriptive of a compressor charging a system, while rapid ground charging of storage bottles approaches the adiabatic condition. Storage bottles charged from the ground normally require "topping off"—once the bottle has had time to cool from the initial charging—to raise the pressure to the desired level again.

One characteristic of compressible fluids is that the weight rate of flow with a given upstream pressure will approach a certain maximum rate that it cannot exceed no matter how much the downstream pressure is further reduced. The maximum possible velocity in a pipe is sonic, which is expressed as

$$c = \sqrt{\frac{\gamma g R T}{\text{molecular wt of gas}}}, \text{ fps} \tag{9-15}$$

where

c = sonic velocity, fps
g = acceleration due to gravity, ft/sec^2
R = gas constant, 1544 ft-lb/(lb-mole °R)
γ = ratio of specific heats c_p/c_v, dimensionless
T = absolute temperature, °R

When subsystem operating time is critical, the components should be analyzed to insure that there are no restrictions that will limit flow below the desired rate.

Basically, the components within the pneumatic system restrict or control flow, or produce force or motion. Fixed restrictions such as orifices restrict flow, while a valve such as a solenoid or manual shutoff valve controls the flow of the gas. Manufacturers normally have flow-pressure drop data for their valves, but if data are unavailable, they may be extrapolated from similar valves for preliminary design estimates. A common method for rating a valve is to determine its Equivalent Sharp Edge Orifice Diameter (ESEOD); then calculations for pressure drop can be made easily. The device for producing the force or motion may be a piston, diaphragm, bellows, etc. The fluid volume required to produce the desired force or motion can be determined by analysis of the transmission lines and component volumes. The source is then sized to produce the desired output within the restrictions of Eqs. 9-13 and 9-14.

Generally, the output force requirements first are defined; then pneumatic power requirements, pressure, and volume that will produce the desired output are determined. System components—such as a check valve, shutoff valve, and filter—are added into the design for control of the fluid. Each of these components is analyzed for pressure drop at the desired flow to the output device. The source or storage bottle is sized to produce the desired flow rate considering the pressure drop of the lines and each component along the flow path.

9-6 FLUIDIC SYSTEMS

9-6.1 GENERAL

Fluidic systems use the phenomena of fluids in motion (either gases or liquids) to perform the functions of sensing, data processing, control, and display, without the use of electronics and generally without mechanical moving parts. This technology promises (and in many cases already has demonstrated) cost and reliability advantages over conventional avionic systems. The cost and reliability improvements result from the absence or gross reduction of moving parts. In addition, the following system attributes normally are realized:

1. The ability to operate in extreme environments: high or low temperatures (depending on the fluid used) and high vibration, shock, and acceleration

2. Hardness to radiation exposure and the capability of operation in explosive atmospheres

3. Long shelf life

4. Minimum maintenance.

Because of these attributes, fluidics can be expected to have an important part in future helicopters. To date, fluidic stability augmentation systems already have been flight tested onboard both tandem- (Ref. 2) and single-rotor (Ref. 3) helicopters; commercial production of a hydrofluidic (uses hydraulic oil) yaw damper began in 1971. In addition, an integrated closed-loop engine control and flight control system for VTOL aircraft (Ref. 4) has been demonstrated to be feasible.

Because of the impact of fluidics upon future helicopter avionics, it is essential that helicopter designers be aware of the technology and its capability. To aid the designer, a large number of references is provided.

Fluidic technology has been in existence about ten years, six of which have been applied to system feasibility studies and development programs. These programs have, in general, confirmed the fluidic attribute of reliable performance in environments of severe shock, vibration, acceleration, temperature, and radiation. Except in instances where electronics cannot be used, however, fluidics has not demonstrated superior accuracy. Current indications are that the application of fluidics will be in areas where simple, medium-accuracy controls are suitable and high reliability is desired, or where conventional equipment cannot be used because of some adverse environment such as radiation (ionizing radiation or electromagnetic interference), explosive atmospheres, high or low temperature, or high shock and vibration.

9-6.2 FLUIDICS COMPARED TO ELECTRONICS

One way to indicate the state-of-the-art of fluidic systems is to compare it with conventional electronic control systems. The following discussion compares ten characteristics of the two systems:

1. Applicability. At the present time, fluidic systems are considerably more limited in applicability than electronic systems. For example, the very high accuracy, rapid response, and high linearity attainable with electronic systems cannot be attained with fluidic systems. On the other hand, there are a number of tasks that can be accomplished easily with fluidic systems, but that either are not possible or are uneconomical with electronic systems.

2. System performance. Generally, the maximum performance of a fluidic system is below that of an electronic system. Fluidic system performance, however, is adequate for many applications; in others, cost and other advantages can offset performance disadvantages that may exist.

3. Development cost. The cost of developing a new fluidic system to accomplish a given task is higher today than the cost of developing an electronic system to accomplish the same task. As the fluidic component state-of-the-art advances, however, costs of developing a new fluidic system are expected to compare favorably with costs of conventional electronic system development.

4. Production cost. Because of the basic simplicity of fluidic elements, production costs of fluidic systems may be materially less than those of comparable electronic systems.

5. Complexity. Fluidic components have significantly fewer parts than comparable electronic components. Fluidic devices have no moving parts, and even devices such as fluidic accelerometers have only one moving part.

6. Reliability. Fluidic systems are expected to be significantly more reliable than comparable electronic systems, particularly when operating in adverse environments. In addition, fluidic systems will be able to operate satisfactorily in some environments in which conventional electronic systems cannot operate at all. Finally, because of element simplicity, high reliability will be attainable with much less effort than is required to attain equal reliability in electronic systems.

7. Maintenance. Fluidic systems offer the potential of being easier and less costly to maintain than conventional electronic systems because in mass production low-cost throwaway modules probably will be used.

8. Size and weight. Generally, fluidic and conventional electronic systems are comparable in size and weight.

9. Shape. Fluidic systems may be more difficult to fit into peculiarly shaped volumes than are electronic systems.

10. Power requirements. Fluidic systems require

approximately the same power as conventional electronic systems.

9-6.3 FLUIDIC COMPONENT DEVELOPMENT STATUS

Selected fluidic components are described and performance parameters are presented in par. 9-6.5. Table 9-4 provides a summary of the state-of-the-art of individual components suitable for typical helicopter application (see Table 9-5 for estimated component size, weight, and reliability). The example that follows illustrates interpretation of the data given in Table 9-4. The table contains the following information about fluidic proportional amplifiers.

1. Production models have been built ("D" in Col. 2) and used in operating systems ("X" in Col. 3).

2. Experimental data do not exist regarding shelf life (blank in Col. 15).

3. Size, weight, and reliability predictions for a new proportional amplifier to perform a specified task can be described on the basis of significant experience with developmental models ("3" in Cols. 4, 5, and 6).

4. Cost in production quantities can be described on the basis of a significant amount of experimental data on development models ("3" in Col. 7).

5. Experimental data exist on performance in critical environments ("X" in Cols. 9, 10, 11, 12, 13, and 14) and on response of the amplifiers ("X" in Col. 16).

Overall, the table shows that proportional amplifiers have reached an advanced developmental stage and can be tailored to new applications with considerable confidence.

Fluidic technology, although nearly ten years old, is still in its infancy compared with electronics. The blanks in Table 9-4 and the development status shown in Table 9-5 indicate the general improvement needed. Except for uses in certain simple applications—such as vehicle stabilization, and machine and tool control—most fluidic devices need overall performance improvement.

Table 9-5 presents estimated physical parameters for various fluidic components. Estimates of weight and volume presented in Table 9-5 can be considered as worst-case values for individual components consistent with the current state-of-the-art. Significant reductions, e.g., 10-20%, are possible, especially when several component functions are combined to form a fluidic "circuit" or multifunctional subsystem.

Table 9-6 presents environmental capability of key fluidic components. These data can be used by the helicopter designer in determining the practicality of using fluidics—either components, subsystems, or systems—for any candidate environment by comparing the capabilities indicated with those of existing conventional electronic components.

Table 9-7 shows 16 applications each of which is subdivided into applicable subsystems. Each subsystem has its various functions of sensing, data processing, display, and control judged as to its present or future feasibility. It can be seen that, in some instances, fluidics can accomplish part of the required functions, even though the entire system category has been judged infeasible. An example is the display function of IFF (Identification Friend or Foe) or the proximity warning subsystem of external environmental monitoring. Future development requirements can be seen by noting the blank spaces in the table. Only the components presently deemed feasible are discussed. Those in the other categories are beyond the scope of this handbook.

9-6.4 DESIGN AND TEST SPECIFICATIONS

Fluidic technology is similar to that of electronics in that they both can perform the same general functions of sensing, data processing, control, and display. It would follow, therefore, that for any such application the same functional requirements can be specified for fluidic systems as for electronic devices. The major differences in design and test specification documents will be:

1. The definition of fluidic component power requirements will be in terms of pressures and flows instead of voltage and currents (see Ref. 5 for fluidic-to-electronic analogy).

2. Military Specifications and Standards for hydraulic and pneumatic devices will be used instead of those for electronic equipment.

3. The appropriate pneumatic testing will be used.

In addition, fluidics has its own specification for terminology and symbols.

The following specifications generally are used as the documents applicable to fluidic design and test:

1. A general system requirement document written for the particular system or subsystem involved

2. MIL-STD-1306, which defines terminology, symbols, etc.

3. MIL-H-8501, which delineates helicopter flying and ground-handling quality requirements

4. Documents on hydraulics, including MIL-H-5606, MIL-H-5440, and MIL-H-8775

5. Documents on pneumatics, including MIL-P-5518, MIL-P-8564, and MIL-E-38453

6. MIL-F-18372 or MIL-F-9490, concerning flight controls

TABLE 9-4
FLUIDIC COMPONENT STATE-OF-THE-ART

1	2 State of Development	3 Flight Test or Field Use	4 Reliability	5 Size	6 Weight	7 Cost in Production Quant.	8 General Performance	9 Tests Conducted at Hi-Lo Temperature	10 Vibration	11 Acoustic Noise	12 Acceleration	13 Shock	14 Nuclear Radiation	15 Shelf Life	16 Response
Components															
Amplifiers, Proportional*	D	X	3	3	3	3	3	X	X	X	X	X	X		X
Amplifiers, Bistable	C	X	3	3	3	2	3	X	X	X	X	X	X		X
Amplifiers, Turbulence	C	X	2	3	3	3	3								X
Amplifiers, Supersonic	C	X	2	2	2	2	2	X	O		O	O	O		X
Amplifiers, Vortex*	C	X	2	3	3	2	3	X							
Computer Elements	C	X	2	2	2	2	3	O	O	O	O	O	O		O
Oscillators	C	X	2	2	2	2	3	X	O		O	O	X		O
Resistors, Capacitors, etc.*	C	X	3	3	3	3	3	X	O		O	O	X		X
Vortex Sensor*	C	X	3	3	3	2	3	X	X		X	X	X		X
Accelerometer*	C	X		3	3	2	3	X	X		X	X	X		X
Attitude Sensor	C	X		2	2	2	3		X		X	X			O
Speed Sensor	C	X		3	3	2	3	X							O
Temperature Sensor	C	X	2	3	3	2	3	X							X
Fluid Potentiometers*	B	X	2	1	1	2	2	X	X		X	X			O
α, β Sensors	C	X	2	2	2	2	3	X	X		X	X			O
Pressure Sensor	C	X	2	2	2	2	3	X	X						X
Multielement Devices															
Operational Amplifiers*	C	X	2	3	3	3	3	X	O	O	O	X	X		X
Amplifier Cascades, Bistable	C	X	2	3	3	2	3	O	O	O	O	X	X		X
Counters	B		3	2	2	2	3	X	X	O	X	X	X		X
Comparators	A			2	2	2	1								X
High Pass Network*	C	X	3	3	3	3	3	X	X			X			X
Converters, A/D	B	X		1	1	1	3	X	X				X		X
Converters, D/A	A			1	1	1	3								X
Lead-Lag Network*	C	X		3	3	3	3	X	X						X
Integrators, R.C.*	C	X		3	3	3	3	X	X			X			X
Differentiators*	A	X		1	1	2	3								X
Integrators, Active*	B			1	1	2	3								
Computers, General Purpose	A			1	1	1	1								
Gain Schedulers*	C	X		3	3	2	3	X	X			X			
Function Generators	A			1	1	2	1								
Moment Producers															
Reaction Jets	C	X		3	3	3	3	O	O		O	O			X
Boundary Control	B	X		1	1	1	1	O	O		O	O			
Fluidic/Conventional Control Servos*	D	X		3	3	3	3	X	X		X	X		X	X
Power Sources															
Hot Gas Generators	D	X	3	3	3	3	3	X	X		X	X		X	
Cold Gas	D	X	3	3	3	3	3	X	X		X	X		X	
Pumps	D	X	3	3	3	3	3	X	X		X	X		X	
Compressor Bleed	C	X		3	3	1	3								
Hydraulic Oil	D	X	3	3	3	3	3	X	X		X	X		X	
Transducers															
Fluid/Electrical															
Hot Wire	D	X		3	3	3	3								X
Thermistor, etc.	D	X		3	3	3	3								
Pressure Transducer*	D	X		3	3	3	3	X	X		X	X			X
Electrical/Fluid															
Magnetost. or Piezo Element	B			2	2	1	1		X		X	X			X
Spark Element	B			1	1	1	1		X		X	X			X
Magnetic-Flapper Nozzle*	D	X	3	3	3	3	3	X	X		X	X		X	X
Gas/Liquid															
Switch	B	X		2	2	1	1								
Bellows to Flapper-Nozzle	B	X		2	2	1	1	X	X						X
Fluid/Mechanical															
Jet Recovery on Wheel*	B			1	1	1	1								
Pressure Recovery on Piston*	C	X		3	3	2	3	X	X		X	X			X
Test Equipment	D	X		3	3	3	3								

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KEY

State of Development

- A — Breadboard-experiment models built and tested to prove principles of operation
- B — Developmental - engineer models suitable for system or flight test, built from engineer drawings
- C — Prototype - Design optimized for specific application, models suitable for production fabricated, but without production tooling does not include off-the-shelf components for breadboard use
- D — Production - Models designed for mass production and fabricated with production tooling

Component Description Basis

- X — Tests have been conducted - test data available
- O — Tests considered completed on basis of similarity
- Blank — Experimental data do not exist, factor describable only on basis of general experience and scientific principle involved
- 1 — Describable on basis of analysis or limited experimental data
- 2 — Describable on basis of similarity to other fluid devices for which significant experimental data exist
- 3 — Describable on basis of a significant amount of experimental data on developmental models of the same type device
- * — Includes pneumatic-fluidic and hydrofluidic configurations
- ** — Pneumatic-fluidic only

TABLE 9-5
PHYSICAL PARAMETER VALUE

	Volume, ft^3	Weight, lb	Reliability Failures/10^6 hrs	Dev. Status 1: Now 2: 10 yrs
Sensors				
Mechanical Position	0.004	0.7	0.150	1
Speed	0.022	1.8	0.200	1
Pressure	0.007	1.2	0.150	1
Pressure Ratio	0.003	1.2	0.040	2
Temperature	0.007	0.4	0.1	1
Differential Pressure	0.001	0.06	0.05	2
Pitot Prob.	0.01	2.0	0.50	1
Angle of Attack Sensor	0.03	5.0	6.5	1
Fuel Flow	0.007	1.20	0.15	1
Angular Rate Sensor (Autopilot)	0.01	0.5	0.1	1
Angular Rate Sensor (Inertial)	0.01	1.0	5.0	2
Accelerometer	0.003	0.5	0.[illegible]	1
Attitude Gyro	0.01	1.0	5.[illegible]	2
Attitude Rate	0.05	1.0	0.25	2
Altimeter	0.03	2.0	0.5	2
Vertical Gyro	0.20	12.0	20.0	2
Force Sensor	0.02	1.5	0.25	1
Data Processing				
Amplification - Summing (opl. ampl)	0.005	0.25	0.2	1
- Gain Scheduling	0.005	0.25	0.2	1
Amplifier Basic Single Stage	0.001	0.01	0.001	1
Function Generator (10 ampl. elem.)	0.01	0.5	0.5	2
Division (1 fun. gen. + 1 gain sch.)	0.015	0.75	0.7	2
Synchronizer (2 opl. ampl., 1 act., pos. pot)	0.03	1.0	0.5	2
Schmitt Trigger (5 elem.)	0.001	0.2	0.05	1
Binary Counter (25 elem.)	0.003	0.5	0.25	1
Trim Valve	0.003	0.25	0.20	1
Integrator	0.005	0.40	0.25	1
Analog-to-Digital Converter (30 elem.)	0.001	0.08	0.30	2
Direction Cosine Computer (6000 elem.)	0.173	15.0	60.0	2
Navigation Computer (1500 elem.)	0.043	3.75	15.0	2
Clock - Sequencing (1300 elem.)	0.037	3.25	13.0	2
Digital-to-Analog Converter (15 elem.)	0.001	0.10	0.15	2
Display				
On-off Switch	0.003	0.25	0.25	1
On-off Indicators	0.001	0.05	0.10	1
Analog Indicators	0.01	0.50	1.0	1
Binary - Decimal Conv. (100 elem.)	0.003	0.25	1.0	1
Alpha-Numeric Display (100 indicators)	0.015	1.0	10.0	1
Null Indicator	0.02	0.5	0.25	1
Selection Switches	0.003	0.25	0.25	1
Control				
Servo Actuator	0.08	5.0	5.0	1
Reaction Jet	0.01	1.0	5.0	1
Power Actuators	0.08	5.0	5.0	1
Transducers				
Electrical-to-Gas	0.005	0.25	0.25	1
Gas-to-Electrical	0.002	0.10	0.25	1
Gas-to-Liquid	0.005	0.5	0.25	1
Electrical-to-Liquid	0.005	0.5	0.25	1

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7. Environmental test specifications such as MIL-E-5272, MIL-F-8615, MIL-E-5400, or MIL-STD-810.

9-6.5 EXAMPLE OF A HELICOPTER FLUIDIC SYSTEM

The example system in this paragraph provides a basis for understanding how fluidic components can be incorporated into systems. A three-axis, hydrofluidic stability augmentation system (SAS) is shown in Fig. 9-55. The yaw axis portion of Fig. 9-55 has been selected as the example system and represents third-generation hardware; an experimental model was demonstrated in the laboratory in 1966 (Ref. 6), system reliability (38,000 hr MTBF) was demonstrated in 1967 (Ref. 7), and a successful flight test of a flightworthy model was conducted in 1968 in a UH-1C helicopter (Refs. 3 and 8).

9-6.5.1 System Design Requirements

The function of the yaw SAS is to increase the helicopter yaw axis damping ratio from about 0.30 unaugmented to 0.6 or greater without opposing the pilot's commands during maneuvers. The yaw axis was analyzed without the mechanical stabilizer as part of the basic vehicle.

The yaw rate feedback is high-passed to allow steady-state turns without the SAS fighting the pilot. System authority of 25%, 20% linearity, and a threshold of 0.1 deg/sec were specified. Fig. 9-56 presents the analytical block diagram, and Fig. 9-57 shows the yaw axis response requirements. Detailed analysis and design information can be obtained from Ref. 9. The system was mechanized to meet the stated requirements.

9-6.5.2 System Mechanization

Fig. 9-58 is a schematic diagram of the resultant system mechanization. The system is basically a simple high-passed-rate loop with pedal position input. Fig. 9-59 shows assembled and exploded views of the yaw axis hardware. As shown in Figs. 9-58 and 9-59, the yaw axis SAS consists of (1) a vortex rate sensor, (2) four fluidic amplifiers, (3) a high-pass circuit and lag circuit, (4) the pilot input device, (5) a servoactuator, and (6) power supply elements (not shown). Builtin-

TABLE 9-6
FLUIDIC COMPONENT ENVIRONMENTAL CAPABILITY (PNEUMATIC)

	TEMPERATURE, °F	VIBRATION, Hz	LINEAR ACCEL, g	ACOUSTIC NOISE, dB	SHOCK, g	RADIATION	ALTITUDE, ft	FLIGHT TEST CONDITIONS
AMPLIFIERS, PROPORTIONAL	-65 TO 1500	5 TO 2500 AT 40 g's*	100	150 AT 1200 Hz	---	EXTREME HARDNESS DEMONSTRATED	TO 40,000	①②③
AMPLIFIERS, BI-STABLE	-80 TO 1500	20 TO 2000 RANDOM AT 4.5 g rms	20	-12 dB/OCT TO 2000 Hz	180		TO 70,000	①②③④
VORTEX RATE SENSOR	-65 TO 160	0 to 2000 AT 10 g's 50 to 2000 AT 0.3g²/Hz	30	---	30 AT 10 msec		TO 15,000 VENTED TO 40,000 CLOSED	①②
ACCELEROMETER		0 TO 2000 AT 5g's		---	10,000 AT 400 [illegible]		---	⑤
ATTITUDE SENSOR	---		[illegible]	---	12 AT 10 [illegible]		---	②
[illegible] SWITCH	-65 TO 300	0 TO 2000 AT 10g SCAN	100	---	--	EXTREME HARDNESS DEMONSTRATED	--	---
TEMPERATURE SENSOR	70 TO 2100	5 TO 2000 AT 50 g's	---	---	40	---	---	③
SPEED SENSOR	-65 TO 300	0 TO 5000 AT [illegible]	[illegible]	---	---	---	TO 20,000	[illegible]
PRESSURE RATIO SENSOR	70 TO 400	---	-	-	---	---	TO 40,000	③

*LIMITED TO 0.1 in. D.A. 5 TO 30 Hz

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① AIRCRAFT
② MISSILE
③ ENGINE CONTROLS
④ EXPLOSIVE ATMOSPHERE
⑤ SIMULATION TEST BED

TABLE 9-7
FUNCTION FEASIBILITY RATINGS

Parameter	Sensing Now	Sensing In 10 Years	Data Processing Now	Data Processing In 10 Years	Display Now	Display In 10 Years	Control Now	Control In 10 Years	Overall Rating 1: Now 2: in 10 years 3: No
Engine Control	---	---	---	---	---	---	---	---	
Load Management		X	X	X	NA	NA	X	X	2
Gas Generator Control	X	X	X	X	X	X	X	X	1
Fuel Management	---	---	---	---	---	---	---	---	
Pressure Operation	X	X	X	X	X	X	X	X	1
Quantity Control	X	X				X	X	X	3
Flight Control	---	---	---	---	---	---	---	---	
Manual Control		X	NA	NA		X	X	X	2
Stabilization	X	X	X	X	X	X	X	X	1
Pilot Assist		X	X	X	X	X	X	X	1
Hover Position Control		X		X	X	X	NA	NA	2
Automatic Modes	X	X	X	X	X	X	X	X	1
Guidance	---	---	---	---	---	---	---	---	
Navigation		X		X	X	X	NA	NA	2
Steering		NA	X	X	X	X	NA	NA	2
Ext. Envir. Monitoring	---	---	---	---	---	---	---	---	
Air Data		X	X	X	X	X	NA	NA	2
Terrain Following							NA	NA	3
Formation Flight							NA	NA	3
Wire/Obstacle Det.					X	X	NA	NA	3
Proximity Warning					X	X	NA	NA	3
Target Det. and ACQ				X		X	X	X	3
IFF					X	X			3
Fire Control	---	---	---	---	---	---	---	---	
Aiming							NA	NA	3
Firing	X	X	X	X	X	X	X	X	1
Weapons Readiness	X	X	X	X	X	X	X	X	1
Maneuver Prediction						X	NA	NA	3
Weapon En. Route Delivery	---	---	---	---	---	---	---	---	
Command Techn.			X	X		X	X	X	2
Homing Techn.			X	X	NA	NA	X	X	2
I. P. Following Techn.	X	X	X	X	NA	NA	X	X	1
Target Damage Asses.					NA	NA	NA	NA	3
Surveillance and Penetr.					X	X	NA	NA	3
Protection	---	---	---	---	---	---	---	---	
Armor	NA	NA	NA	NA	NA	NA	NA	NA	3
Radiation Warning		X		X		X	NA	NA	2
Communications									3
Auxiliary Equip.	---	---	---	---	---	---	---	---	
Power - fluidic	X	X	X	X	X	X	X	X	1
Safety	X	X	X	X	X	X	X	X	1
Dependability	---	---	---	---	---	---	---	---	
Monitoring	X	X	X	X	X	X	X	X	1
Inflight Repair	X	X	X	X	X	X	X	X	1
Availability	---	---	---	---	---	---	---	---	
Preflight Checkout	X	X	X	X	X	X	X	X	1
Ground Maintenance	X	X	X	X	X	X	X	X	1

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NOTE: X denotes "feasibility"
Blank denotes "no feasibility"
NA denotes "not applicable"

test (BIT) and rate sensor null-adjustment screws are visible between the two amplifiers in Fig. 9-59.

The top and bottom cover plates are of a sandwich construction. These covers have a center core with channels on each side. Thin sheets of metal are bonded to the center case, resulting in permanently sealed manifold plates. Electroformed (electroplating over conductive wax amplifier castings) fluidic amplifiers are bolted to the manifolds. The center plates contain the rate sensor and shaping network capacitors (bellows) and resistors (orifices) built in as integral parts. External sealing is accomplished with O rings or with the permanent bonding.

9-6.5.3 Component Description

Basic components of the type used in the example system are described in further detail in Chapter 9, AMCP 706-202. Many other component types exist and much descriptive information exists (Refs. 7, 10, and 11). Tables 9-4 and 9-5 present component physical details.

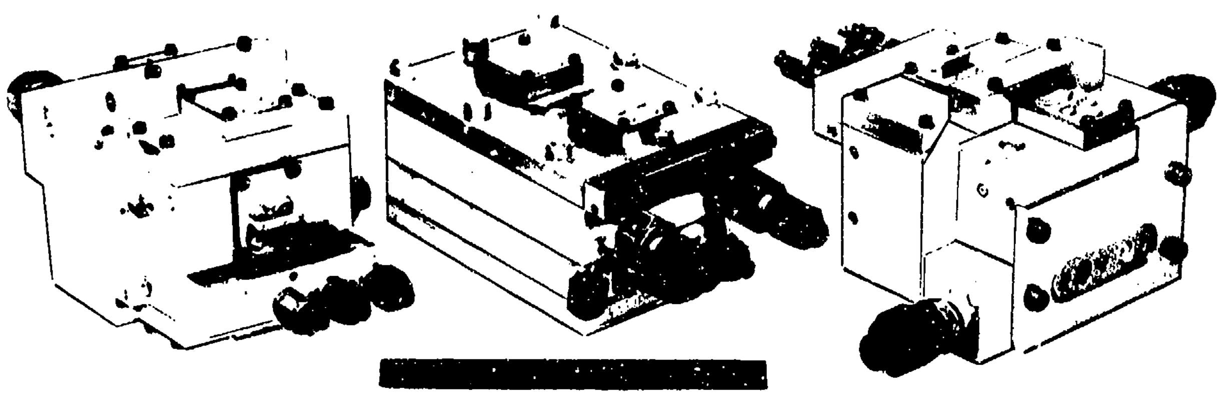

Fig. 9-55. Three-axis Hydrofluidic Stability Augmentation System

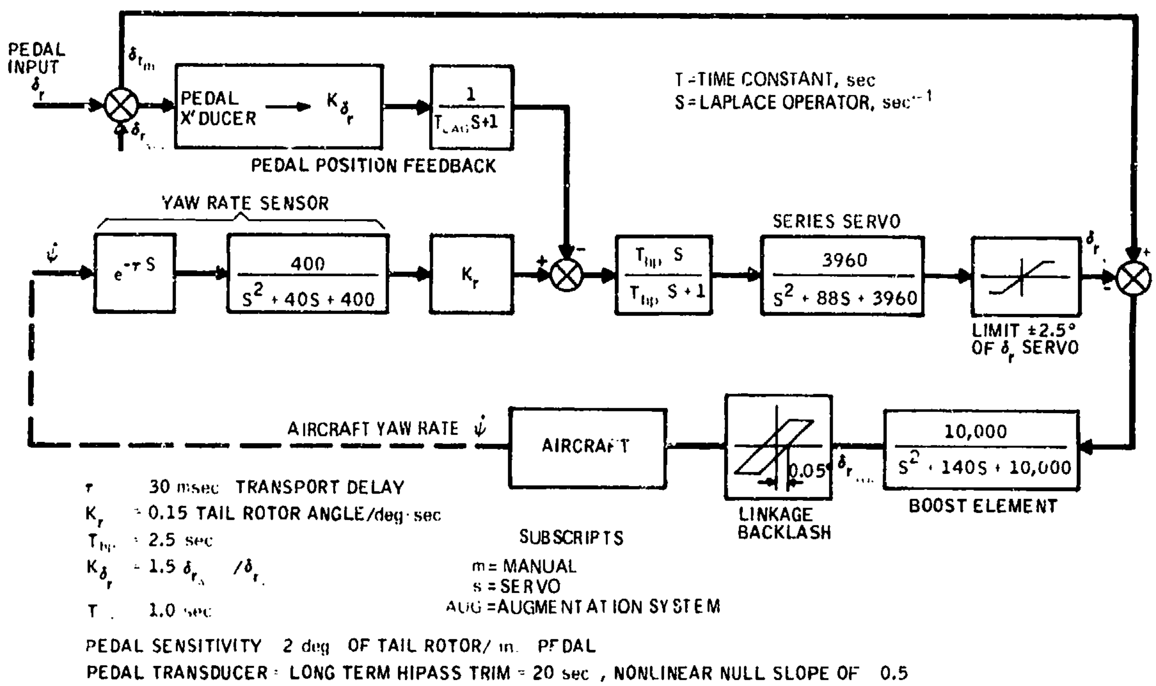

Fig. 9-56. UH-1B Yaw SAS Analytical Block Diagram

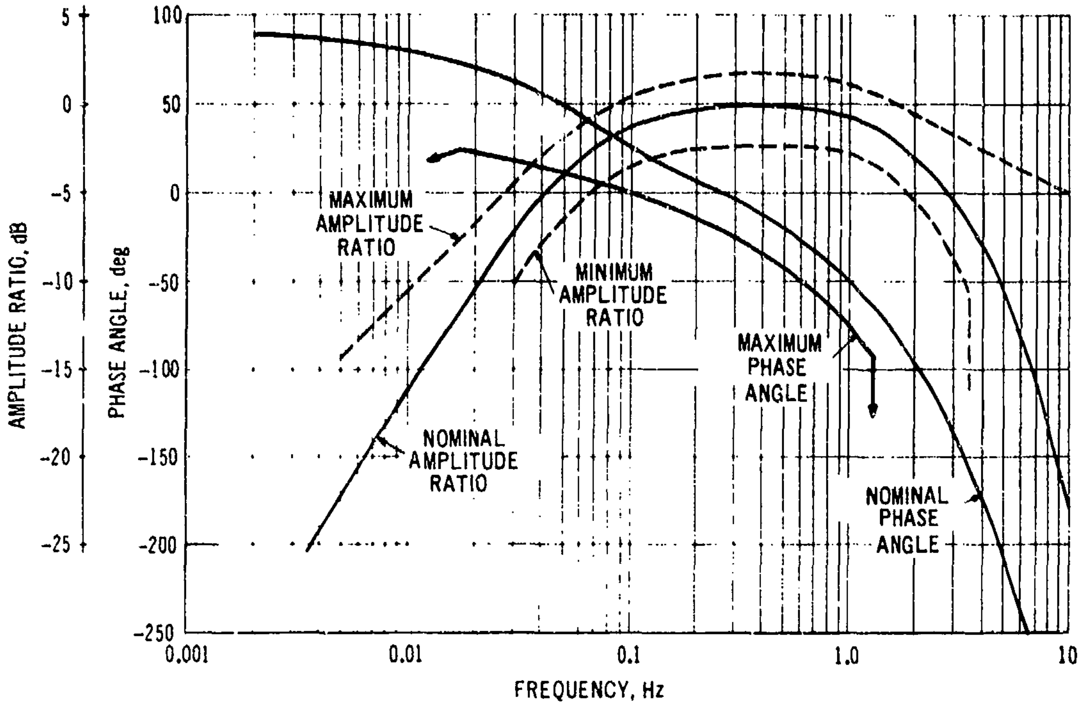

Fig. 9-57. Yaw Axis Response Requirements

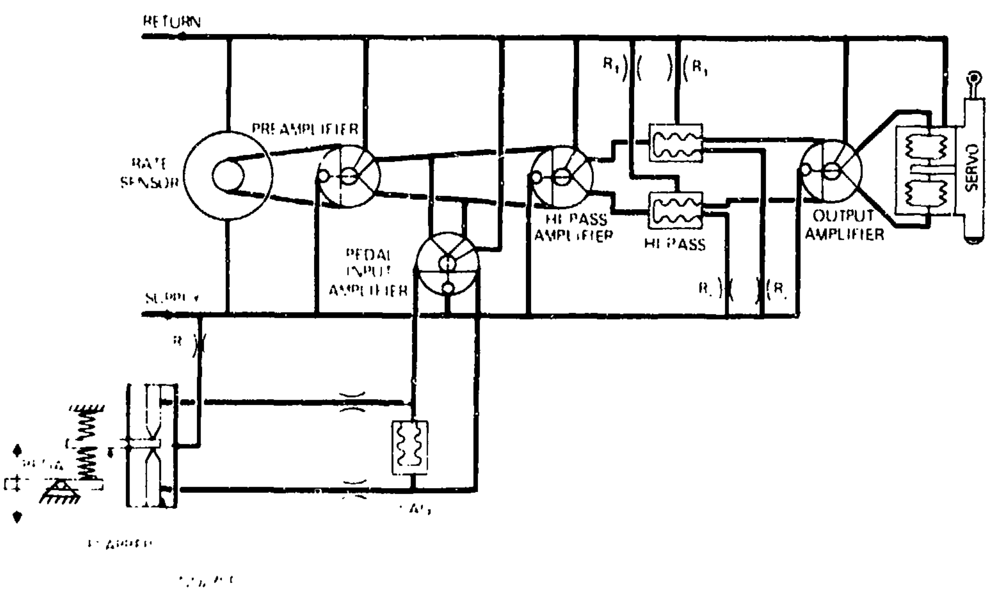

Fig. 9-58. Yaw Axis Hardware Schematic

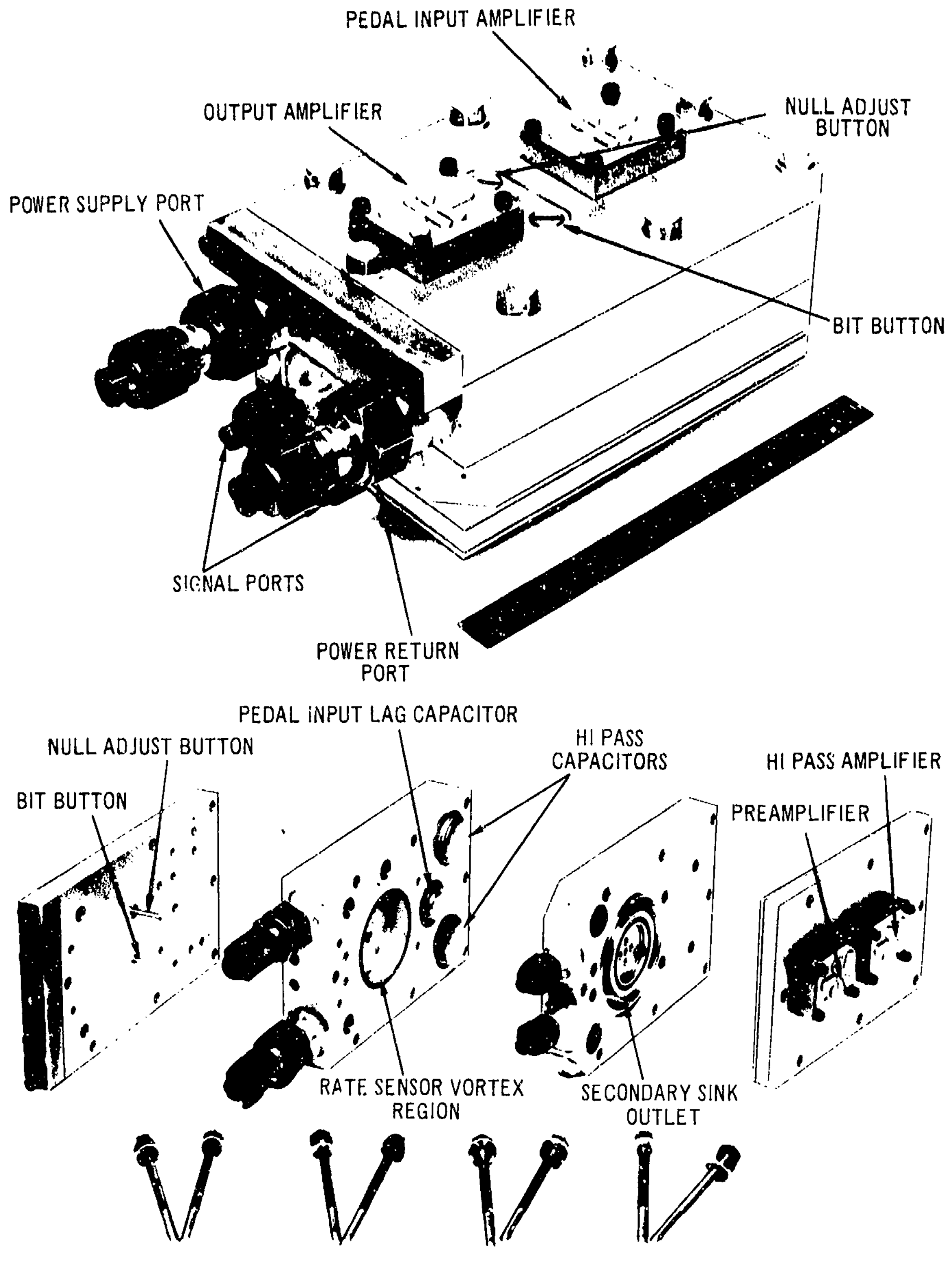

Fig. 9-59. Hydraulic Yaw Stability Augmentation System

TABLE 9-8
COMPARISON OF ELECTRONIC AND HYDROFLUIDIC SAS

PARAMETER	HYDROFLUIDIC	ELECTRONIC
PERFORMANCE	SUITABLE	SUITABLE
RELIABILITY (MTBF)*, hr	89.000	3.500
VOLUME, in^3 *	50	113
WEIGHT, lb	10.0	11.2
COST EST, $	2.000	3.500
POWER, W	500**	250–equiv

* DOES NOT INCLUDE ACTUATOR

** BASED ON 1/3 OF POWER FOR THE 3-AXIS SAS SHOWN IN FIG. 9-55 (2.5 gpm at 1500 psi) A SINGLE-AXIS SYSTEM COULD HAVE TO HAVE A DIFFERENT POWER SUPPLY SYSTEM TO REALIZE THIS POWER LEVEL (SEE PAR. 9-6.6.1)

9-6.5.4 Comparison With Equivalent Electronic System

Table 9-8 presents comparison data for the example system and an equivalent electronic single-axis SAS. Although the comparison is with a hydrofluidic system, similar results would be obtained using pneumatic fluidic systems when compensation techniques have been developed allowing their use.

In general, when comparing simple electronic and fluidic systems for near-term use, a good guideline is that the fluidic system has the advantage in areas of reliability and maintainability, while the electronic system has the advantage in areas of flexibility (use in different vehicles), power, accuracy, and low development risk.

9-6.5.5 Flight Test

Fig. 9-60 presents the results of a flight test on second-generation hardware. Because this system did not have a pilot pedal input transducer, the gain was adjusted to give a compromise between hover and high-speed operation (Ref. 8).

9-6.6 AUXILIARY EQUIPMENT

This paragraph describes techniques of operating fluidic systems with onboard power supplies (jet engine compressor discharge for pneumatic fluidics and hydraulic oil for hydrofluidics) and interfacing with other electronic equipment.

9-6.6.1 Power Supplies

An important feature of fluidic technology is the ability to operate with different fluids. This makes the fluid system versatile and useful in many different applications. The more probable helicopter power sources discussed herein already have been used for a number of flight tests, such as compressor bleed (Refs. 2, 12, and 13), and hydraulic oil (Refs. 8 and 9). Pneumatic pumps also have been used as the power source for several flight-tested systems (Refs. 14 and 15).

9-6.6.1.1 Pneumatic Power

Energized fluid, in the form of pressurized and heated air, can be extracted from the compressor section of the turbine engine. Compressor bleed also may be used to drive auxiliary equipment and for anti-icing of the engine inlet section.

For systems using gas from a central source, contamination should be the first consideration. Gas passages in fluid amplifiers used in the logic circuits necessarily are small; therefore, these gas passages are subject to the collection of foreign matter that can cause a change in operating characteristics and eventually may create a complete blockage.

In terms of both power output and fuel consumption, performance of the turboshaft engine can be affected adversely by compressor bleed. Therefore, this power source should be considered for a fluidic system only if the mass flow required is small. The system characteristics are fundamentally the same whether the pressurized air is supplied by a separate, shaft-powered compressor or pneumatic pump. However, the latter, in addition to being a more efficient use of the helicopter power plant, also permits greater flexibility in selection and control of the air pressure. For example, use of compressor bleed for a fluidic flight control system will be altitude limited, because the compressor discharge pressure drops to quite low levels. Pressure levels suitable for helicopter flight control applications may be available to an altitude of 20,000 ft, but the effect of bleeding an adequate volume of air at this altitude may result in an unacceptably high reduction of shaft power available for helicopter propulsion.

Because an aircraft flight control system must function equally well at all altitudes, discharge conditions of the fluid system must be considered. Discharge from a bleed-powered system will be expelled to the atmosphere. Atmospheric discharge conditions vary with altitude and, therefore, mass flow rate through the fluid system will vary even if the upstream pressure and temperature are controlled. Thus, some form of regulation must be provided on the overboard discharge side of the fluid flight control system. A sonic orifice (such as a venturi) will provide a constant mass-rate discharge. Fig. 9-61 is a schematic of a power conditioner used in a flight test program (Refs. 2 and 13).

If the source of pressurized air is a separate compressor rather than engine compressor bleed, the system may be closed loop. In such a case, the discharge will not be overboard but will be to the inlet side of the compressor. This system, then, will be less altitude-sensitive. Such a system would be similar schematically to one using hydraulic power, which is described subsequently.

9-6.6.1.2 Hydraulic Power

An existing hydraulic power system is an obvious candidate as a power source for a hydrofluidic control system. However, existing hydraulic power systems, for optimum overall efficiency, provide high pressure and low flow rates. Fluidic controllers, on the other hand, require relatively large quantities of low-pressure flow, e.g., 2.0 gpm at a differential pressure of 25 psi. Several types of hydraulic circuits are suitable for providing the required control power.

For some applications, the fluidic control system has been used in series with the powered servoactuator. This arrangement, shown schematically in Fig. 9-62 reduces overall power requirements. Such a system (three controllers, three servoactuators, one flow control, and one priority valve) has been tested (Ref. 9) and no interaction of the separate circuits occurred.

Another approach is shown in Fig. 9-63. Here a jet pump is used as an active throttling device to produce a secondary, or control system, flow at reduced pressure and increased rate. Other throttling devices obviously are possible, but this type is preferable because it provides higher efficiency than do orifices, nozzles, or pressure-regulating valves. In fact, the efficiency level is sufficiently high to make the jet pump arrangement preferable to a system using an additional low-pressure pump in any case in which the actuator power require-

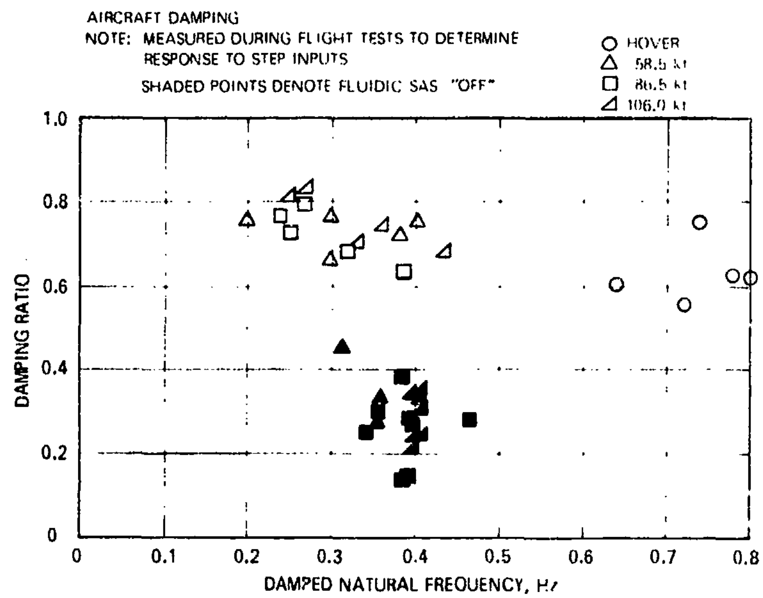

Fig. 9-60. Flight Test Results of Engineering Model Hydraulic Yaw SAS Flown on UH-1C

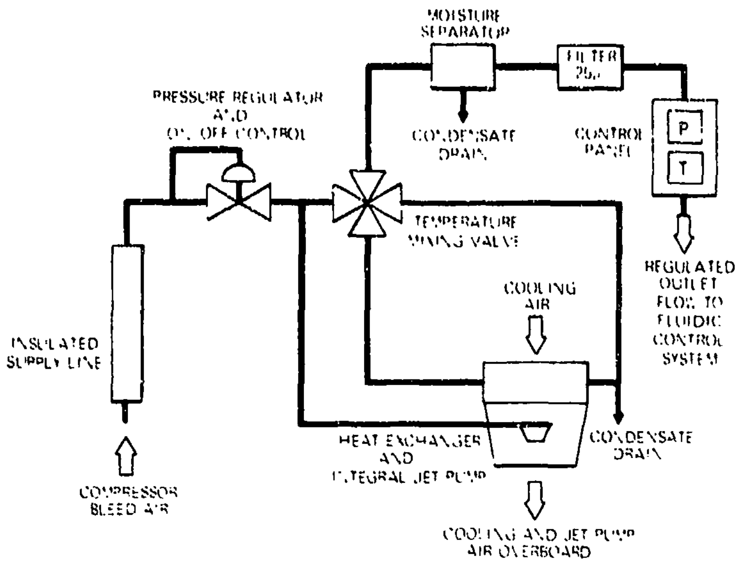

Fig. 9-61. Fluidic Computer Power Supply System

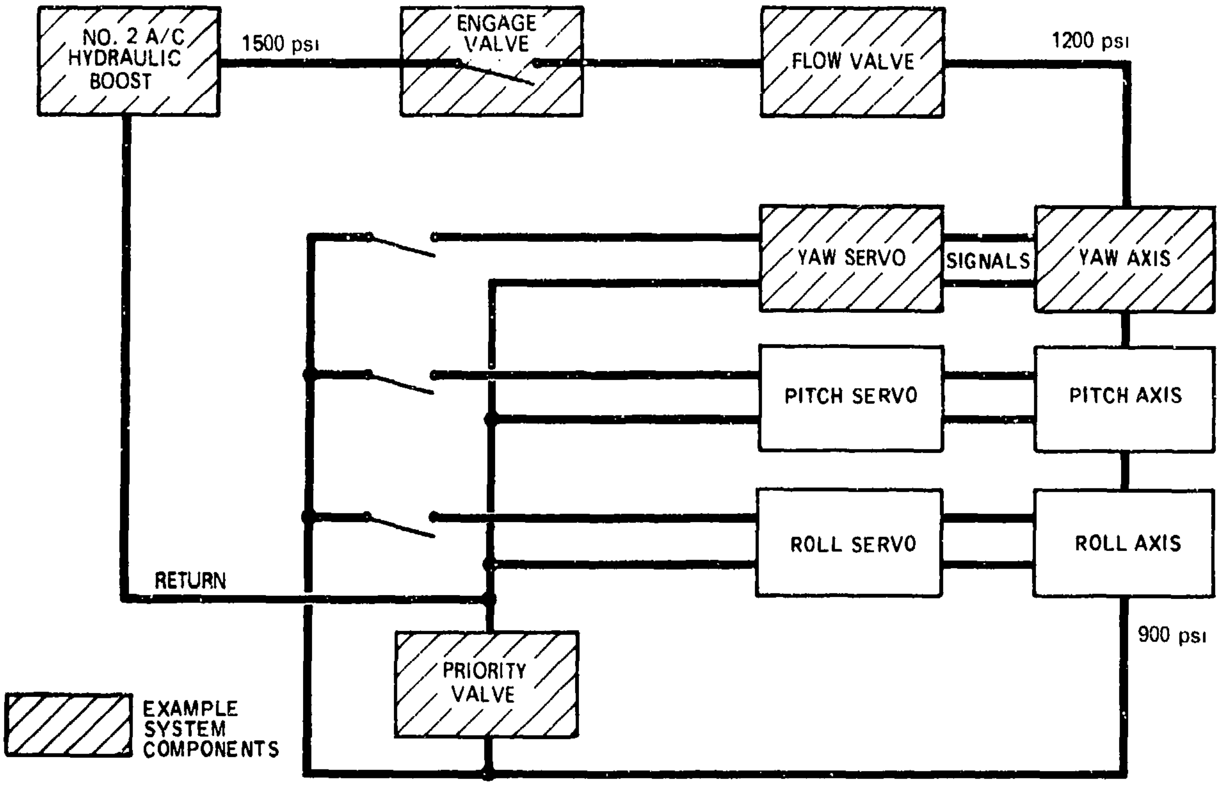

Fig. 9-62. Hydraulic Power Supply for Three-axis SAS

ments are high enough to justify a high-pressure (3000 psi) pump.

Unlike the compressor bleed-powered system, the hydraulic-powered systems operate as closed systems. Changes in altitude, therefore, will not affect hydraulic fluid flow rates.

9-6.6.2 Transducers

It is probable that, except for simple applications, interfacing will be required among fluidic, electrical, and mechanical portions of a total helicopter system. As with actuators, it is expected that conventional transducers, or slight modifications thereof, will suffice for the majority of helicopter applications; no difficulty is seen in providing these interfaces. Table 9-9 presents typical analog and digital fluidic/electrical and electrical/fluidic transducers presently available.

9-7 ENVIRONMENTAL CONTROL SYSTEMS

9-7.1 GENERAL

In providing an environmental control system (ECS) for a helicopter, a primary consideration is its integration with the airframe, the propulsion unit(s), and the secondary power system.

Helicopters in the current inventory use heating and ventilating systems for personnel and avionic environmental control purposes. Avionic components are located and operated in a manner to insure that their rated maximum operating temperatures are not exceeded when using interior ventilation for cooling. Avionic equipment has been affected by high humidity, ambient temperature, and sand and dust contamination. Crews must perform continuously for many hours in these environments, which tends to reduce their efficiency. For these reasons, in helicopters under development and those envisioned for future applications, overall environmental control—including air conditioning for both heating and cooling—is receiving considerable attention.

For worldwide combat operational capability, ECS provisions (heating, cooling, and ventilating) *shall* be specified for personnel and equipment compartments. Complete ECSs for troop and/or cargo compartments can be justified when it becomes more cost-effective than a simple heating and ventilating system or when a special mission requirement must be met.

A preliminary design of an ECS must include the following studies, which are discussed in detail in subsequent paragraphs:

1. Design requirements
2. Conceptual development and synthesis
3. Trade-off studies and selection.

Optimization involves a reiterative cost-effectiveness evaluation of the possible combinations of main and secondary power generation, functional equipment arrangements, and equipment interfaces with the airframe. As a major helicopter subsystem, the ECS influ-

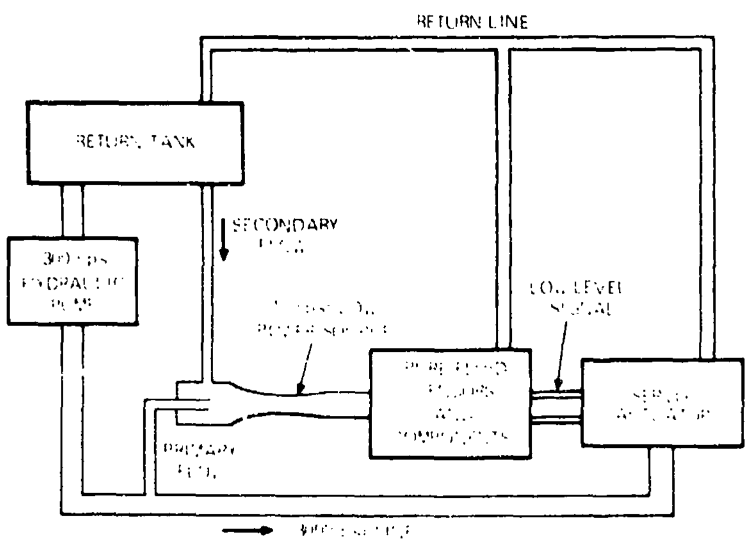

Fig. 9-63. Hydraulic Control System

TABLE 9-9
TYPICAL TRANSDUCER CHARACTERISTICS

	PARAMETER	FLUIDIC/ELECTRICAL		ELECTRICAL/FLUIDIC	
		DIGITAL	ANALOG	DIGITAL	ANALOG
PNEUMATIC	SWITCHING SPEED	10 msec		1.6 msec	
	SWITCH LEVEL	−0.5,+0.15 psi			
	AMPERAGE	20 A		0.25 W @ 12 V DC	
	SCALE FACTOR		40 mV/psi		0.1 psi/V
	LINEARITY		0.5% FULL SCALE		2% FULL SCALE
	RANGE POWER		0.5 TO 100 psi		0.5 TO 1.0 psi
	POWER		10 V @ 35 mA		4 V, 20 mA
	RESPONSE		>100 Hz		15 Hz
HYDRAULIC	SWITCHING SPEED	5 msec		1.6 msec	
	SWITCH LEVEL	0.5 TO 100 psi			
	AMPERAGE	7 A		0.25 W @ 12 V DC	
	SCALE FACTOR		40 mV/psi		0.0015 in.^{3}/sec/mA
	LINEARITY		0.5% FULL SCALE		<7% FULL SCALE
	RANGE		0.5 TO 100 psi		±10 psi (diff)
	POWER		10 V @ 35 mA		±10 mA
	RESPONSE		>100 Hz		125 Hz

ences and is influenced by the balance of the secondary power system (SPS) and the main power subsystems. As such, the most appropriate ECS for a particular helicopter will be the result of cost-effectiveness analyses of the system with regard to its role in the total secondary power system.

9-7.2 DESIGN REQUIREMENTS

Generally, goals relating to reliability, maintainability, survivability, and vulnerability are dictated by the helicopter design criteria and mission objectives. In addition, system loads and operational performance requirements must be established for the design mission profiles.

With the possible exception of pressurization, ECS redundancy for personnel conditioning usually is not considered a requirement. However, redundancy for avionic equipment conditioning might be essential to mission safety. This requirement will depend upon the high-temperature operational characteristics of the equipment, and mission durations also must be considered.

Maintainability, survivability, and vulnerability design requirements affect equipment location, access provisions, protective implementations, and, in some cases, the type of ECS permitted in the design.

ECS operational performance and system loads are dependent upon mission environments and durations; physiological (crew), component (avionics), special cargo requirements, and compartment configurations. Crew station environmental requirements are presented in par. 13-3.2.3. These are defined not on the basis of providing crew comfort but are the maximum acceptable environmental limits before serious crew performance decrements are likely to occur. Avionic cooling requirements *shall* be in accordance with MIL-E-5400. Applicable climatic conditions for worldwide operations are given in MIL-STD-210 and in AR 70-38.

Heating and cooling loads are dependent upon climatic conditions, compartment requirements, heat generation within controlled zones, and heat flux through compartment structures.

In determining the ECS load requirements, it is necessary to evaluate the conditions under which the helicopter will perform its mission. Variations of allowable internal temperatures must be considered with reference to external temperatures, velocities, and altitudes, which affect both thermal load and available ECS capacity. Critical conditions and requirements can be derived from MIL-E-38453 or par. 13-3.2.3.2 for the particular helicopter mission.

A typical helicopter cockpit is shown in Fig. 9-64, which illustrates the various elements that comprise the total steady-state ECS load. Transfer of heat occurs by convection, radiation, and conduction in the following manner:

$$Q_C = Q_{ES} + Q_{IS} + Q_{SL} + Q_P + Q_{IL} + Q_E \quad \text{, Btu/hr} \qquad (9\text{-}16)$$

where

Q_C = total compartment thermal load, Btu/hr
Q_{ES} = heat transfer through external compartment surfaces, Btu/hr
Q_{IS} = heat transfer through internal compartment surfaces, Btu/hr
Q_{SL} = solar heat load, Btu/hr
Q_P = personnel heat load, Btu/hr
Q_{IL} = heat infiltration or leakage, Btu/hr
Q_E = electrical heat load, Btu/hr

Normally, helicopter flight profiles are subsonic and below 50,000 ft altitude. As such, skin temperatures simply are equal to the ram air temperature.

The heat transfer Q_{ES} through external compartment surfaces consists of two terms—heat flow through the walls and structural elements, and internal convection. Thus,

$$Q_{ES} = UA(T_c - T_e) + (UA)_{ef}(T_c - T_e) \quad \text{, Btu/hr} \qquad (9\text{-}17)$$

where

U = overall surface conductance, Btu/hr-°F-ft^2
A = surface area, ft^2
$(UA)_{ef}$ = effective conductance of fins, etc., Btu/hr-°F
T_c = average compartment air temperature, °F
T_e = external surface temperature, °F

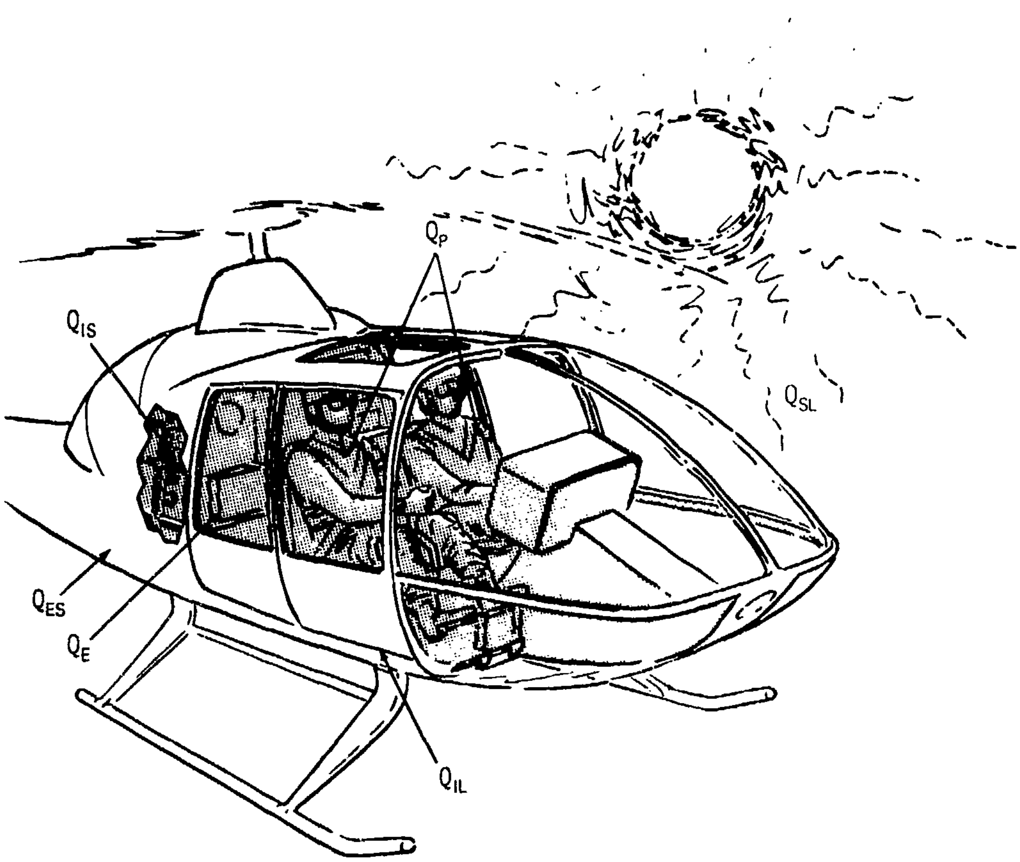

Fig. 9-64. ECS Compartment Loads

The equation accounts for basic conductive and convective heat transfer (wall construction and material characteristics) and fin effects.

Total heat transfer Q_{IS} through internal compartment surfaces—such as floors and bulkheads attached to the helicopter skin—is the sum of the heat flow through these surfaces and is expressed by the equation:

$$Q_{IS} = UA(T_c - T_a) + (UA)_{eb}(T_{eff} - T_c) \quad \text{,Btu/hr} \quad (9\text{-}18)$$

where

$(UA)_{eb}$ = effective conductance of bulkhead, Btu/hr-°F
T_a = average air temperature of adjacent compartment, °F
T_{eff} = effective temperature of bulkhead, °F

The amount of solar radiant energy Q_{SL} transmitted through transparent surfaces is given by:

$$Q_{SL} = -\tau G_t A_p \quad \text{,Btu/hr} \quad (9\text{-}19)$$

where

τ = transmission factor for transparent surfaces, dimensionless
G_t = solar radiation intensity, Btu/hr-ft^2
A_p = projected area of transparent surface normal to sun's rays, ft^2

Fig. 9-65 describes solar heat transmission through a transparent material for a range of surface thicknesses.

Normally, for preliminary estimates of heat loads, heat loss rates H of 400 Btu/hr are used for each crew member and 300 Btu/hr for a passenger, recognizing the different levels of activity. The resulting load Q_P can be expressed as:

$$Q_p = (-HN)_{crew} + (-HN)_{passenger}$$
$$= -400N_{crew} - 300N_{passenger} \quad \text{,Btu/hr} \quad (9\text{-}20)$$

where

N = number of persons
H = heat loss per person, 400 Btu/hr for crew members and 300 Btu/hr for passengers

Infiltration loads Q_I caused by air flowing into the compartments of nonpressurized helicopters may be estimated from the following relationship:

$$Q_I = W_I c_p(T_c - T_e) \quad \text{,Btu/hr} \quad (9\text{-}21)$$

where

W_I = infiltration airflow rate, lb/hr
c_p = specific heat of air at constant pressure, Btu/lb-°F

Moisture also must be considered; i.e., the heat of condensation of the H_2O vapor may constitute as much as 80% of the ECS heat load. Leakage loads Q_L from pressurized compartments may be estimated from the following relationship:

$$Q_L = W_L c_p(T_a - T_c) \quad \text{,Btu/hr} \quad (9\text{-}22)$$

where the leakage airflow rate W_L is

$$W_L = 0.07 V_L^{2/3} + 0.05 \quad \text{,lb/hr} \quad (9\text{-}23)$$

Eq. 9-23, defining leakage flow rate W_L as a function of the volume V_L of the adjacent pressurized compartment, is empirical and its use is described in MIL-E-38453.

The probability of infiltration and leakage occurring in the same compartment is highly unlikely.

The heat load caused by electrical units Q_E is a function of the power consumed by the avionic and electrical equipment. The conversion of electrical power P_e in watts to Btu/hr is given by:

$$Q_E = -3.415P_e \quad \text{,Btu/hr} \quad (9\text{-}24)$$

(The procedure for determining steady-state compart-

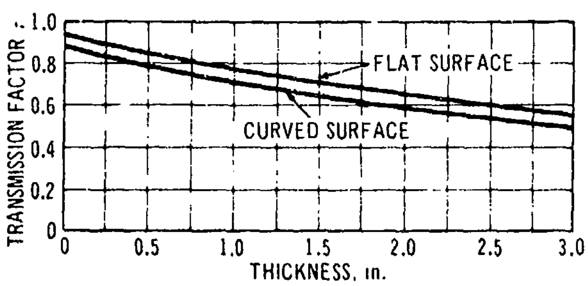

Fig. 9-65. Solar Transmission Factor for Transparent Areas

ment ECS loads is a condensation of that described in Ref. 16.) A sample calculation of heat loads is included in Chapter 13, AMCP 706-202.

9-7.3 CONCEPTUAL DEVELOPMENT AND SYNTHESIS

The development of candidate secondary power system concepts must include consideration of the ECS technology available. As a major part of the secondary power system, each ECS concept must meet certain performance and cost-effectiveness objectives. The impact of system losses on size, weight, and cost of the conversion components, which must be operated from an engine and/or APU power source, is of primary importance in developing comparative factors. Potential economic benefits can be derived by having some degree of commonality in subsystem generation equipment and in power transport media.

It is important to recognize the impact of the ECS upon power requirements to avoid undue penalties in size and installed power, particularly when using an APU as the source of power. Operational methods should be considered, and it may be necessary to limit conditioning capacity for ground operation to selected compartment areas.

Primary sources of air for helicopter ECS are engine bleed, APU bleed, and fan-delivered outside ambient air. Adequate precautions shall be taken to insure that locations of the engines and air inlets will minimize the possibility of ingesting contaminant vapors from fuel expulsion exits, oil drains, gunfire exhausts, engine and/or APU exhausts, etc.

Appropriate ECS subsystems that should be [illegible] gated for possible integration are:

1. Distribution systems
2. Heating and ventilation systems
   a. Blowers and ejectors
   b. Bleed air heating
   c. Electric heaters
   d. Combustion heaters
3. Air-cycle systems
4. Vapor-cycle systems
5. Thermionic cooling (avionics)
6. Combination vapor- and air-cycle systems.

9-7.3.1 Distribution Systems

Distribution systems will affect operation of the ECS as well as personnel comfort, equipment operation, size, cost, envelope, and power requirements of the conditioning equipment. Fig. 9-66 depicts a typical distribution system at a personnel flight station.

The detailed environmental requirements that must be met for personnel compartments are listed in MIL-E-38453 or par. 13-3.2.3.2. The Specification discusses allowable interior wall-to-air temperature differences, air temperature gradients within the compartment, and maximum air velocities impinging upon personnel. Appropriate values of these pa[illegible]eters will govern the number, size, and direction [illegible]utlets and the amount of airflow through ea[illegible]t in accordance with the heat flux being injecte[illegible]o the area. Ducts interconnecting the outlets with the ECS conditioning equipment should be sized for minimum power loss (velocity, bends, roughness, etc.), inasmuch as this loss can increase the size of required ECS equipment. The distribution power loss P_{loss} can be expressed in terms of airflow, specific volume, and pressure loss as follows:

$$P_{loss} = 0.7273 \times 10^{-4} W_a v_1 (p_d - p_c) \quad \text{, hp} \tag{9-25}$$

where

W_a = airflow rate entering air-cycle machine or distribution system, lb/hr

v_1 = specific volume of air at exit of conditioning equipment, ft^3/lb

p_c = compartment pressure, psi

p_d = conditioning equipment exit pressure, psi

0.7273×10^{-4} = constant for conversion of units

[illegible] drop characteristics are determined by the summ[illegible] the pressure losses developed in each partic[illegible]t of the system—such as exit nozzles, ducting, flow balance orifices, bends, turns, expansion joints, and manifolds. Equations, computation methods, and configuration factors of applicable arrangements can be found in various publications such as those listed in Refs. 17, 18, and 19.

9-7.3.2 Heating and Ventilating Systems

Separate heating systems must be incorporated on all helicopters not equipped with total ECS. Applicable heating systems are:

1. Bleed air heating
2. Electrical heaters
3. Combustion heaters.

Because fresh air ventilation is a requirement on all helicopters, a source exists for providing air to the heaters. Mission characteristics, particularly hover opera-

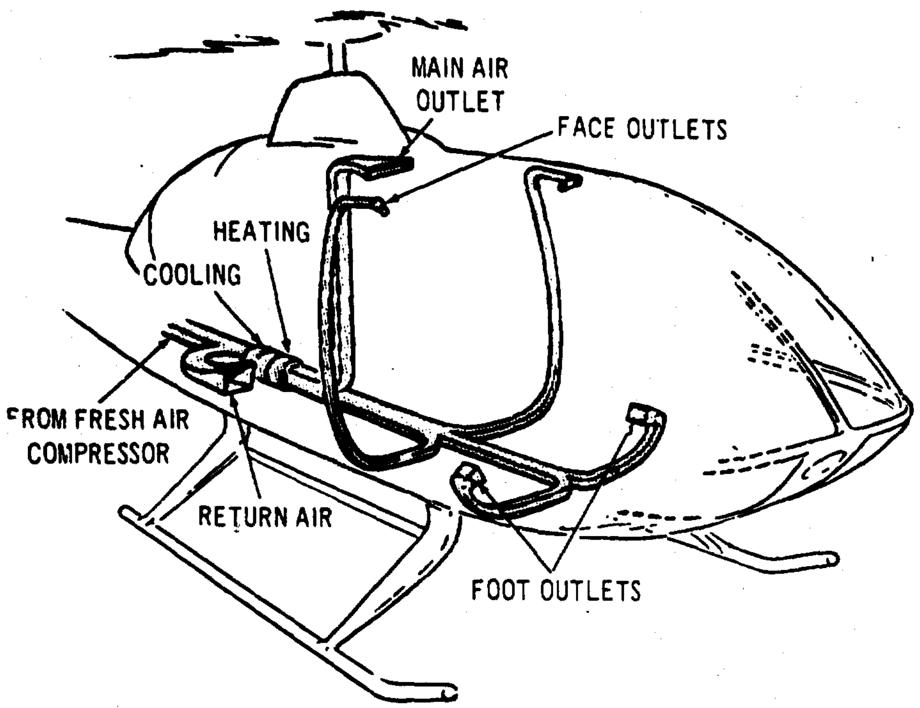

Fig. 9-66. Flight Station Air Distribution Schematic

tion, usually prevent the use of ram air techniques and require the use of blowers or ejectors to draw fresh air through the conditioning equipment and the distribution systems. Both the blower and the ejector can introduce high noise levels, especially when high rotating speeds or high velocities are used for the purpose of minimizing size and weight. Complicated muffler devices that attenuate the noise generated by the blower or ejector might be required to eliminate an intolerable condition in the compartment.

Avionic compartments can be designed to accept cockpit-conditioned air at proper locations and quantities to cool the components; the air subsequently can be ejected from the helicopter by an exhaust blower or fan. Power requirements are based upon the efficiency η_f of the fan or blower, the volumetric flow rate $W_a v_i$ and the pressure head Δp developed, where the fan power P_f is given by:

$$P_f = \frac{0.7273 \times 10^{-4} W_a v_i \Delta p}{\eta_f}, \text{hp} \qquad (9\text{-}26)$$

Methods, formulas, and characteristics of various axial and centrifugal blowers or fans and ejector designs are described in Refs. 18, 19, and 20.

Bleed air can be used simply and inexpensively in heating systems. Heat may be injected directly into the airflow using an ejector to power and mix hot bleed air from an engine or APU. Usually a blower is used to provide the circulating air supply. Simple control systems can be devised to modulate the quantity of hot air necessary to meet the compartment temperature requirements.

Electrical and combustion heaters can be used as heat sources. Electrical heaters, although very efficient in operation, can impose severe weight and power penalties upon the electrical power generating equipment. Control systems can be relatively simple and inexpensive, and can provide complete and rapid heat rate modulation for accurate temperature control.

Combustion heaters, while imposing minimal requirements on secondary power generation equipment, present disadvantages because of the need of a fuel supply and a control system. Location of the equipment is of utmost importance, not only to minimize fuel transport lines, but to minimize infrared signature. Methods must be incorporated to insure that this signature level is consistent with mission requirements.

Often, it is preferable to provide heating in the form of a removable winterization kit. By this means, it is possible to avoid penalizing the payload and/or performance of the helicopter with the weight of heating equipment during operation in those climatic environments where heating is not required. This alternative may be specified by the procuring activity, depending upon anticipated utilization of the helicopter. If not so specified, it still may be shown to be preferable as a result of trade-off studies during the preliminary design phase.

9-7.3.3 Air-cycle Systems

Complete cooling, heating, and ventilating ECS functions can be achieved by air-cycle systems. Numerous system configurations have been developed to accommodate various capacity or temperature requirements. Generally, air-cycle systems applicable to helicopters use an air-to-air heat exchanger with ambient air as a heat sink. In addition, a small, high-speed expansion turbine is used to reduce the cooling air temperature further below ambient conditions. A simple air cycle is shown schematically in Fig. 9-67. Pressurized air is supplied by engine- or APU-powered compressors or by bleed extraction. Condensed moisture in the cooled air is extracted, usually by mechanical means, as it passes through a separator to the entrance of the distribution system. A bypass control system, which also diverts a portion of the hot air to the entrance of the distribution system, is provided to modulate the cooled air for meeting intermediate heat loads or to meet a purely heating requirement.

The generalized approach to initial sizing and air-cycle performance calculations for a total thermal load Q_C is as follows:

$$Q_C = W_a c_p (T_c \quad T_4) \quad \text{, Btu/hr} \qquad (9\text{-}27)$$

The airflow rate W_a, and the conditioned air temperature T_4 leaving the air-cycle machine are determined by conducting a cycle analysis to satisfy the given equation. The dry air rated method of analysis is given in Refs. 17 and 18. Appropriate provision must be made for the energy required to extract moisture.

The analysis requires an iterative solution to find W_a and T_4 for the required Q_C. The equations used also apply for all critically defined conditions for heating and cooling. The analysis requires a knowledge of off-design performance of the heat exchanger and cooling turbine, and the ram and bleed air inlet conditions.

Many variations of the described system are available. One of these, the bootstrap cycle, uses a turbine-compressor arrangement with an additional heat exchanger to absorb the work of the turbine and to provide an additional heat sink for the airflow before expansion in the turbine. Detailed sizing formula analysis methods and component characteristics for the simple air cycle, bootstrap cycle, and various hybrid cycles can be found in Refs. 17 and 21.

Air-cycle systems will impose considerable power requirements upon main engines or the APU, and, therefore, it is essential that air be supplied or extracted from these engines at minimum pressure ratios, consistent with the needs of cooling and/or pressurizing the helicopter.

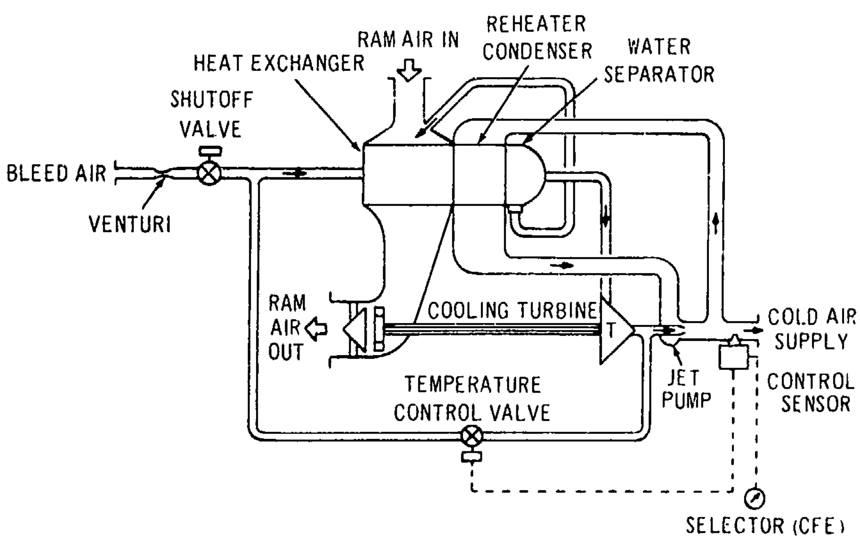

Fig. 9-67. Simple Air-cycle System

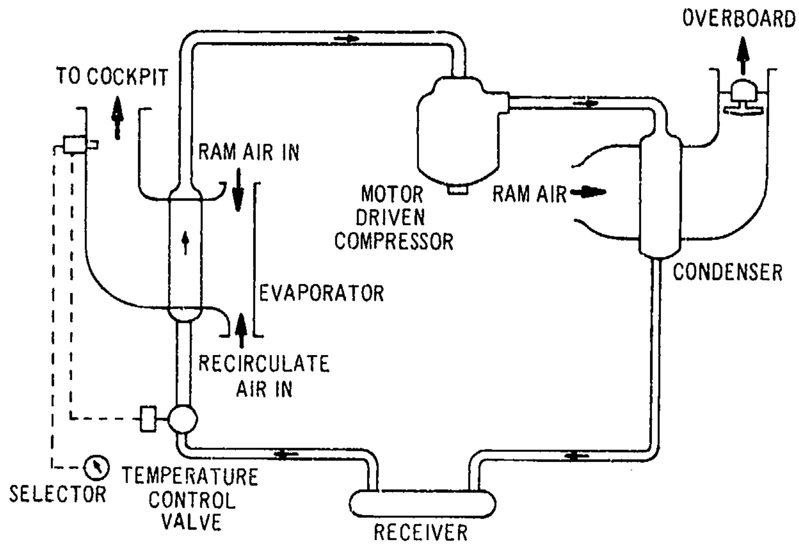

Fig. 9-68. Vapor-cycle System

9-7.3.4 Vapor-cycle Refrigeration Systems

Vapor-cycle systems extract heat in an evaporative heat exchanger, through which the ventilation and recirculation airflow is supplied. In the condenser, ambient air is used as the heat sink, with a motor-driven compressor supplying the energy to circulate the fluid. Vapor-cycle systems have a higher efficiency than air-cycle systems but usually are heavier. Vapor-cycle systems also can provide ground cooling with just an electrical power source (no ram air). A typical system configuration is shown schematically in Fig. 9-68.

The generalized approach for sizing a vapor-cycle cooling system is similar to that for the air cycle. This technique, however, is not dependent upon engine or APU operation, unless the turbine is used to drive the compressor. As in the air-cycle analysis, the cooling demand Q_c that can be met by the cycle is found from

$$Q_C = W_E c_p (T_c \quad T_{ie}) \quad \text{, Btu/hr} \qquad (9\text{-}28)$$

where

T_{ie} = temperature of cooling air leaving the evaporator, °F
W_E = airflow through the evaporative heat exchanger, lb/hr

The matching of W_E and T_{ie} to satisfy Eq. 9-28 should be initiated by assuming T_{ie} to be a minimum of 40°F to avoid icing of the evaporator. Detailed analysis techniques, equipment data, and characteristics for vapor cycles also can be found in Refs. 17 and 18.

The power requirement is the summation of the power to drive the refrigerant compressor, the condenser fan, and the evaporator fan. The compressor power P_{comp} required for the refrigeration system is expressed as

$$P_{comp} = \frac{4.715R}{C.O.P.} \quad \text{, hp} \qquad (9\text{-}29)$$

where

4.715 = conversion constant between tons of refrigeration and horsepower
R = required amount of refrigeration, tons
$C.O.P.$ = coefficient of performance of the refrigeration cycle, dimensionless

The value of $C.O.P.$ is dependent upon many factors, such as condenser, evaporator, and compressor efficiencies; evaporator and condenser temperatures; piping losses; and refrigeration fluid characteristics.

Water content of conditioned air can be maintained accurately by controlling the temperature of the heat exchanger. The same compressor, receiving tank, and condenser can be used in conjunction with several evaporative heat exchangers located at widely separated or remote areas of the helicopter. Such a system can be beneficial for cooling avionic compartments, which require control of contamination and humidity to very low levels.

9-7.3.5 Thermionic Cooling Systems (Avionics)

Cooling systems using thermionic principles are in use today. Thermionics can be described by comparison with a thermocouple circuit. In using a thermocouple, temperatures are measured by the electrical potential generated between the junctions of a pair of dissimilar metals, the two junctions being subjected to different temperatures—one junction at a known and constant (reference) temperature and the other in the environment being measured. The greater the temperature difference between the junctions, the greater the electrical potential. In thermionic cooling, the process is reversed in that an electrical potential is superimposed upon the circuit in opposition to that normally developed in the thermocouple. The result is a cooling effect at one junction (formerly the measured junction). Similarly, the temperature of the other junction (formerly the reference temperature) will increase and release heat to the heat sink. The temperature difference between the cold and hot junctions increases with increasing electrical power in the circuit.

The systems have low coefficients of performance and presently are limited to specific applications where accurate temperature control is required at low refrigeration loads. This method of cooling is most effective when integrated directly into the avionic components. Particular or complete areas of component assemblies can be conditioned to meet a desired requirement. It should be noted that several experimental compartment installations, demonstrating complete air conditioning capability, are in existence today.

9-7.4 TRADE-OFF STUDIES AND SELECTION

The selection of an ECS for the particular mission or application will depend upon comparisons of performance, weight, cost, life, envelope, etc., and upon the ease of integration of the system into the overall secondary power system. Use of advantage-disadvantage summaries, or applying judgment to weighting factors for system attributes, will facilitate initial selections of candidate concepts. A typical weighting factor/trade-off chart is shown in Table 9-10.

TABLE 9-10
TYPICAL FORMAT FOR SYSTEM TRADE-OFF STUDIES

G = GRADE WG = WEIGHT GRADE

SYSTEM ATTRIBUTES	WEIGHTING FACTOR	SYSTEM A		SYSTEM B		SYSTEM C		SYSTEM D	
		G	WG	G	WG	G	WG	G	WG
RELIABILITY									
MAINTAINABILITY									
OPERABILITY									
ACCURACY									
COST									
SCHEDULE FEASIBILITY									
WEIGHT									
SURVIVABILITY									
SAFETY									
COMPATIBILITY									
FLEXIBILITY									
REACTION TIME									
SIZE/VOLUME									
GROWTH POTENTIAL									
POWER CONSUMPTION									
COOLING AIR CONSUMPTION									
[illegible]									
TOTAL SCORE									

Optimization studies must be conducted to establish location and envelope allotments for each candidate system and its relationship to the secondary power generation equipment. Sufficiently detailed design layouts must be generated for each concept chosen to determine transport media losses; power requirements; and overall performance, weight, volume, and installation complexities.

The evaluation and selection process will form a basis for assessing the influence of the ECS upon range and takeoff gross weight of the helicopter, system reliability, vulnerability, maintainability, and flight safety.

This evaluation necessitates the availability of a definitive helicopter model and a detailed mission profile, so that cost differences for the systems and helicopter parameters listed can be established and tabulated. The preliminary design data generated will enhance selection of the best overall system. The interplay of parameters is illustrated by Fig. 9-69, which is an example of relationships to the total helicopter system.

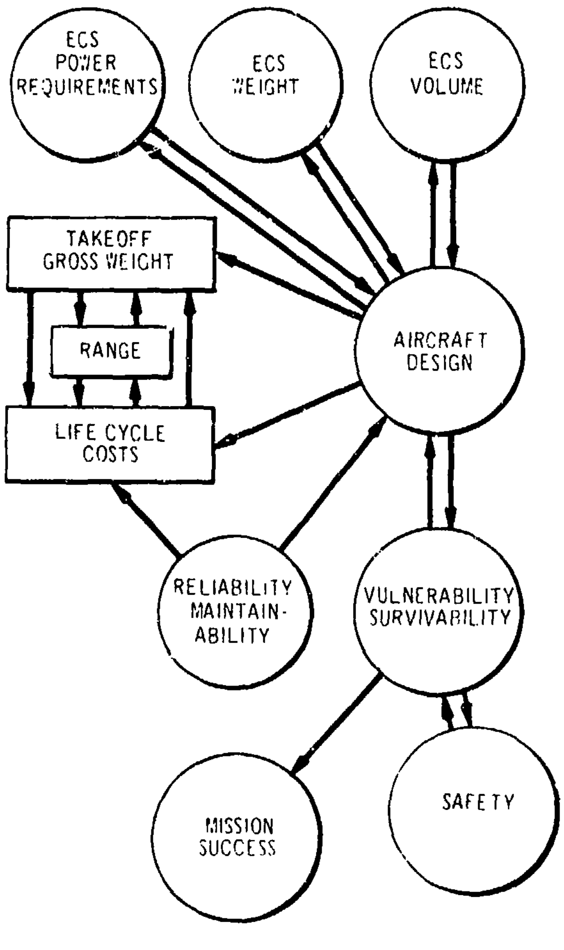

Fig. 9-69. Primary ECS Parameters

9-8 LIST OF SYMBOLS

A = area of surface, ft
A_p = projected area of transparent surface normal to the sun, ft^2
$C.O.P.$ = coefficient of performance, dimensionless
c = sonic velocity, fps
c_p = specific heat of air at constant pressure, Btu/lb-°F
c_v = specific heat of air at constant volume, Btu/lb-°F
D = drag, lb
e = base of natural logarithms
G_t = solar radiation intensity, Btu/hr-ft^2
g = acceleration due to gravity, ft/sec^2
H = heat loss per person, 400 Btu/hr for crew members and 300 Btu/hr for passengers
hp_e = extracted horsepower
L = lift, lb
N = number of persons
P_{comp} = compressor power, hp
P_e = electrical power, W
P_f = fan power, hp
P_{loss} = system power loss, hp
P_p = pump power, hp
p = pressure, psi
p_c = compartment pressure, psi
p_d = conditioning equipment exit pressure, psi
p_R = reference pressure immediately upstream of control or shutoff valve at maximum flow rate, psi
Q = fluid flow, gpm
Q_C = total compartment thermal load, Btu/hr
Q_E = electrical heat load, Btu/hr
Q_{ES} = heat transfer through external compartment surfaces, Btu/hr
Q_I = infiltration load, Btu/hr
Q_{IL} = heat infiltration or leakage, Btu/hr
Q_{IS} = heat transfer through internal compartment surfaces, Btu/hr
Q_L = leakage heat load, Btu/hr
Q_P = personnel heat load, Btu/hr
Q_{SL} = solar heat load, Btu/hr
R = required amount of refrigeration, tons

R = gas constant, 1544 ft-lb/(lb-mole-°R)
= rotational speed, rpm
R_A = range, n mi
SFC = specific fuel consumption, lb/hp-hr
T = temperature, °R
= torque, lb-in.
T_a = average temperature of air in adjacent compartment, °F
T_c = average temperature of air in cabin or compartment, °F
T_e = external surface temperate, °F
T_{eff} = effective temperature of bulkhead, °F
T_{ie} = temperature of cooling air leaving the evaporator, °F
T_t = turbine inlet temperature, °R
T_4 = temperature of air leaving air-cycle machine, °F
t_m = mission time, hr
U = overall surface conductance, Btu/hr-°F-ft^2
$(UA)_{eb}$ = effective conductance of bulkhead, Btu/hr-°F
$(UA)_{ef}$ = effective conductance of fins, etc., Btu/hr-°F
V = cruise velocity, kt
= volume, ft^3
V_L = volume of pressurized compartment, ft^3
V_{max} = maximum allowable line fluid velocity, fps
v_1 = specific volume of air at exit of conditioning equipment, ft^3/lb
W_a = airflow rate entering air-cycle machine or distribution system, lb/hr
W_E = airflow through the evaporative heat exchanger, lb/hr
W_f = fixed weight of system, lb
W_{fo} = takeoff fuel weight required to carry fixed or variable weight, lb
W_i = infiltration airflow rate, lb/hr
W_l = leakage airflow rate, lb/hr
W_v = variable weight, lb
w_b = bleed airflow rate, lb/hr
w_f = fuel flow rate, lb/hr
w_r = ram airflow rate, lb/sec
w_v = rate of consumption of variable weight, lb/hr
γ = ratio of specific heats of air c_p/c_v, dimensionless
Δp = pressure head, lb/in.^{2}
Δw_f = increase in fuel flow rate, lb/hr
η_f = fan efficiency, dimensionless
τ = transmission factor of transparent surfaces, dimensionless

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CHAPTER 10

WEIGHT AND BALANCE

10-1 INTRODUCTION

This chapter discusses the responsibilities and functions related to mass properties and weight engineering in the aircraft design field. Aircraft include helicopter, compound helicopter, fixed-wing, and V/STOL type vehicles. The weight and balance concepts discussed apply to all types of aircraft, but are directed toward helicopters and compound helicopters.

A generalized treatment, rather than a more definitive discussion, has been chosen in order to permit the flexibility necessary in a constantly changing art. Contractors are encouraged to exchange weight data and ideas frequently so as to improve techniques and to keep abreast of the increasing sophistication of aircraft design.

The weight engineer may utilize this chapter as a basis for outlining, planning, developing, and performing the various functions and tasks pertaining to this important discipline of engineering.

The Military Specifications and documents noted herein—MIL-W-25140, MIL-STD-1374—form the basis for most weight engineering requirements in aircraft contracts. Army helicopters will be designed in accordance with Military Specifications and Standards. On occasion, Army aircraft may be procured that meet the civil aviation requirements detailed in applicable Federal Aviation Regulations, i.e., Parts 21, 27, 29 and 133, etc. The complexity of the overall design and weight engineering tasks may be increased in such cases.

10-2 WEIGHT ENGINEERING

10-2.1 IMPORTANCE OF WEIGHT AND BALANCE CONTROL

The requirement to prevent overweight (referred to as one type of growth) prior to first delivery is important. Overweight can cause serious deficiencies in other critical areas such as performance, cost, reliability, scheduling, etc. These degradations usually are compounded after initial delivery due to additional weight required to correct original design deficiencies and to changes necessitated by flight experience and tests.

Such events as product improvement programs (PIP), the addition of new equipment, and new mission requirements also contribute to weight growth. Typical Army experience with weight growth is shown in Fig. 10-1. This experience indicates how important it is to define and consider growth potential in the early stages of design.

Overweight on all types of aircraft has plagued the military services. These overweight problems may be attributed to any number of causes, such as changes in initial design policy, errors in initial weight prediction, engineering management failure, or unwillingness to recognize the need to control weight. Experience proves that removing weight after the fact is both costly and time-consuming. These factors indicate the necessity for vigorous weight surveillance in all phases of a program.

Failure to recognize the overall effect and importance of weight and of the need to develop effective control at the inception of an aircraft development project can place the entire program in jeopardy. Probable effects are increased costs, schedule delay (due to corrective action), and/or reduction in the utility of the vehicle. In some cases the severity of the problem may be cause for program cancellation.

10-2.2 GROWTH FACTOR

Many knowledgeable weight engineers have spent considerable time, energy, and money studying aircraft weight growth and how to predict its effect. Several papers have been published and presented by the Society of Aeronautical Weight Engineers (SAWE) on this subject. The numerical value of the growth factor (GF) varies with the differing characteristics and sizes of aircraft, and each parameter should be evaluated individually. The effects of compromising or reducing margins of safety, independently or collectively with the various performance parameters (speed, range, hover, rate of climb, etc.), will have considerable bearing on the growth factor value and, therefore, must be taken into consideration. Inasmuch as individual contractors have their own methods for developing GF, no further discussion on the subject will be presented in this chapter. Sample techniques for establishing appropriate values of GF are discussed in Refs. 1 and 2.

10-2.3 WEIGHT ENGINEERING GROUP

MIL-W-25140 requires a prospective contractor to establish and maintain a well-organized and competent Weight Engineering Group.

In general, mass properties/weight engineering groups have the responsibility of planning, developing, and performing weight control, assignments during all phases of design, development, and production.

Unlike many other engineering disciplines, weight engineering is required during the entire life of a vehicle, i.e., from the preconceptual through the operational phases. The various weight functions or tasks generally are governed by the type of vehicle being developed and by the overall requirements of the procuring activity.

10-2.4 WEIGHT CONTROL

The Army's increasingly complex and stringent requirements make weight control a factor of prime importance during all design and fabrication phases. The various tasks that must be performed to exercise weight control include, but are not limited to, the following:

1. Weight control program
2. Preliminary weight prediction
3. Weight estimation and calculation
4. Mass property analysis
5. Weight control status report
6. Actual weight and balance determination.

10-2.4.1 Weight Control Program

The weight control program generally functions in conjunction with the efforts employed to estimate weight during all phases of design. This program, repeatedly emphasized in this chapter, is one of the most important functions in the aircraft weight and mass properties field; and for ultimate success it should include, but not be limited to, the following:

1. An accurate preliminary design weight estimate of the proposed helicopter
2. The establishment of an equitable "target" weight system
3. Management of the program by qualified weight engineers
4. Prompt, periodic, accurate reporting of weight and CG status, and related analysis
5. Complete support and backing from the various structural and system design groups

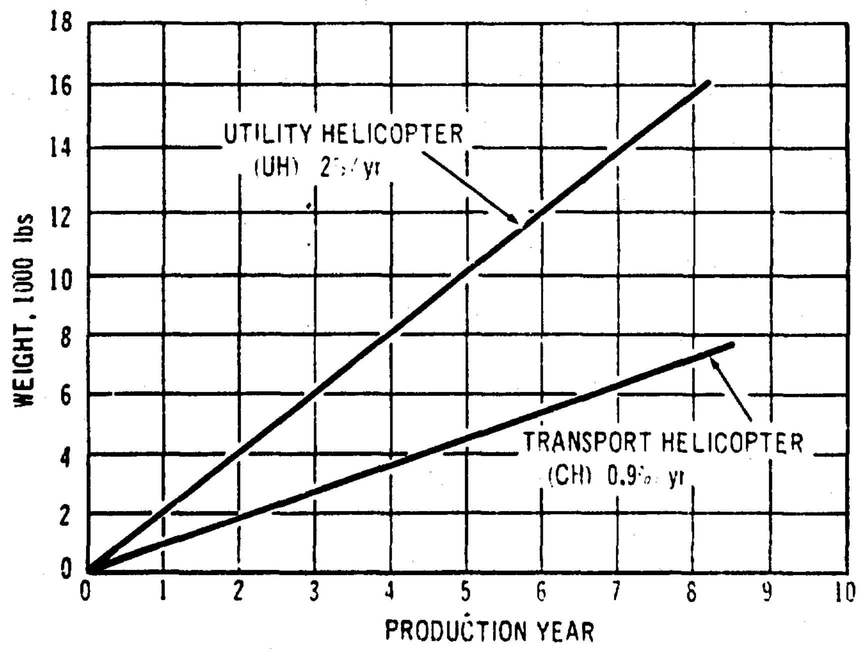

Fig. 10-1. Typical Empty Weight Growth of Production Helicopters (Army)

6. Most important, complete support from management during all phases of the program.

10-2.4.2 Preliminary Weight Prediction

Prediction of weight during preliminary design normally involves the use of purely statistical approaches. The most rudimentary forms of weight expressions are used initially, with more sophisticated weight prediction methods being used as the design advances. In the early phases, little is known of the vehicle except its intended mission, some of its basic equipment, and the applicable design requirements. Therefore, to estimate total weight, very simplified weight expressions that involve a minimum of parameters may be used. These conceivably may involve only gross weight, payload, radius, and hover ceiling or installed power. As the design progresses and additional design details are known, the weight expressions will involve several more parameters, increasing the accuracy of the results. Additional information of value to the weight engineer making weight predictions and estimations is contained in Refs. 3-10.

A commonly used method of weight prediction is the development of a series of equations based upon a logical group of design parameters. The primary equation is obtained by statistical analysis using all the applicable parameters. An example of the validity of such an equation is shown in Fig. 10-2. This typical curve is the result of a regression model using the indicated weights from actual systems and the values of appropriate design parameters.

More rudimentary expressions are obtained by dropping variables in the order of their increasing statistical significance. The result is a series of equations, each having a unique value of standard deviation and coefficient of correlation. An example group of equations may be of the form

$$W_g = K_1 X_1^{N_a} X_2^{N_b} X_3^{N_c} \quad , \text{lb}$$

$$W_g = K_2 X_1^{N_d} X_2^{N_e} \quad , \text{lb} \qquad (10\text{-}1)$$

$$W_g = K_3 X_1^{N_f} \quad , \text{lb}$$

where

W_g	estimated group weight, lb
K	constant of proportionality
X	applicable design parameter
N	exponent

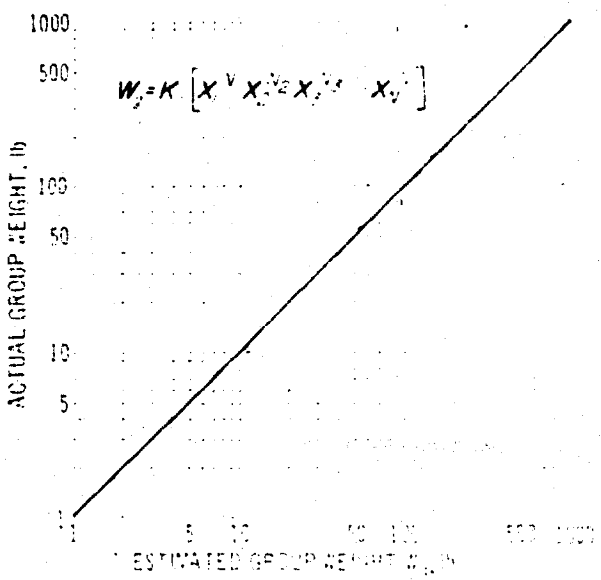

Fig. 10-2. Validation Chart, Typical Group Weight Prediction Equation

In general, the equation with the fewest variables has the largest value of standard deviation and the smallest coefficient of correlation.

This type of analysis can be used for prediction of almost all the group weights that comprise the helicopter empty weight, and is particularly applicable for the structural groups. The weight prediction can be refined as the design progresses and applicable values for increasing numbers of design parameters are determined.

10-2.4.3 Weight Estimation and Calculation

In proceeding to the more definitive stages of weight estimation and calculation usually encountered during response to solicitations, the spectrum of weight determinations widens greatly. Weight prediction equations of numerous types are employed, utilizing a greater number of parameters that by this stage are better defined for the vehicle under study. There usually is a separate equation for each of the structural and system groups.

During this phase, the following three methods of estimations generally are applied:

1. Advanced statistical
2. Analytical
3. Statistical/analytical.

The statistical methods employed at this point, as contrasted with those of the preliminary stage, utilize parameters not previously available or defined for each group. The improved definition provided by these new data narrows the range within which the weight may vary, thus establishing greater confidence in the results.

The analytical method of weight determination can be very useful during contract negotiations, and may be used for both weight prediction and optimization. A typical discussion of weight prediction through structural analysis is provided in Ref. 11, and includes weight optimization by the "structural-index" method.

Statistical/analytical weight estimation employs elements of both methods for its development. This area of weight prediction, therefore, produces a large variety of weight equations based upon the correlation of actual weight, physical and design descriptors, and analytical design parameters. Typical examples of this application are found in Refs. 12-17. The weight prediction methods outlined above often are used as substantiation data for aircraft system proposals.

Finally, layout designs are completed and are the basis for weight estimates and calculations based upon a preliminary structural analysis. Scaling of actual weights from vehicles of similar design may be helpful in furnishing weight predictions in areas not as well defined.

It becomes possible to make more precise weight determinations as design progresses to the release of detail drawings. Aids such as the SAWE Handbook (Ref. 18) generally may be used to facilitate layout calculations, including uniform and established standards of material densities and gage tolerance factors, etc. Most contractors have weight standards based upon many years of experience.

10-2.4.4 Mass Property Analysis

All phases of vehicle development require some form of mass property prediction or analysis to assist other engineering specialities in the analysis of structure, stability and control, dynamics, etc. These mass property analyses may use some of the most rudimentary forms of prediction involving statistics for the preliminary studies. The analyses progress to more exact and precise forms with improved vehicle definition and are discussed in more detail in the paragraphs that follow.

10-2.4.4.1 Moments of Inertia

One of the important tasks performed by weight and mass property control groups is the determination of aircraft mass moments of inertia (MOI) and products of inertia (POI). Formulas for determining moments of inertia can be found in most engineering design handbooks, the most useful being Ref. 18.

For advanced design work, and for applications requiring mass properties of an approximate nature, statistical data may be used. As the design develops, group and component sizes and weights are defined more clearly and a manual calculation of the vehicle mass properties may be performed.

In the layout and detail design stage, the large number of weight elements makes a manual computation of mass properties impracticable. At this point, the manual computations normally are replaced by computer programs and the vehicle mass properties are generated by high-speed computers.

10-2.4.4.2 Center of Gravity Envelope

The center of gravity (CG) envelope is a plot of weight versus permissible CG limits (refer to MIL-W-25140), and is used to determine the potential or actual extreme excursions of weight and CG for a vehicle. The envelope should account for all expected mission configurations, capable of producing these extremes. The plot should show the effects of expendable and variable items of useful load and of alighting gear retraction. The lines connecting these critical weight and CG points form an envelope (sometimes called a potato curve) within which the vehicle *shall* be able to operate without exceeding its design limitations. The weight and balance classification in accordance with MIL-W-25140 can be determined from these various envelopes or curves. Lateral CG limits, which are based upon control capability, are not normally a function of gross weight. Therefore, a chart presentation usually is not required. However, lateral CG limits should be listed in all types of weight reports.

10-2.4.5 Weight Control Status Reporting

The status report provides weight engineering with its primary means of communication among the procuring activity (when required), engineering management, and appropriate design groups. Experience has proven that a timely, accurate, and thorough weight reporting system will contribute significantly to both successful design and weight control programs.

Part III of MIL-STD-1374 presents a typical format for a helicopter weight status reporting and includes all pertinent data. Weight control program documentation requirements are discussed in detail in par. 10-2.5.

10-2.4.6 Actual Weight and Balance Determination

The final weight verification of a vehicle takes place when it is placed on a weight measuring system (scales) to determine its "actual" weight and CG. Prior to this, however, gradual accumulation and compilation of actual data replace the estimated and calculated weights used in establishing vehicle weight and balance predicted for the proposal and specification. Although the methods employed by most aircraft manufacturers are similar, they generally reflect the characteristics of their manufacturing systems, i.e., the actual weighing of manufactured parts. Therefore, it is important to recognize the degree of coordination required between the weight engineering group and the manufacturing division if the program for weighing detail parts is to be effective and efficient. It is the "actual weighing" that finally proves the overall effectiveness and competence of the group.

10-2.5 DOCUMENTATION

Weight data requirements for aircraft are specified in one of the following documents:

1. Request for Quotation (RFQ)
2. Request for Proposal (RFP)
3. Invitation for Bid (IFB)
4. Hardware Contract—Experimental, prototype and production.

Typical weight data and the types of reports required are shown in Tables 10-1 and 10-2.

The preparation, technical content, and format of most of these weight data are established in detail in MIL-W-25140 and include the application of Parts I, II, and III of MIL-STD-1374. MIL-W-25140 outlines the data required to demonstrate contract compliance during design and manufacture as well as the data necessary to establish a system of weight and balance control during service operation of the aircraft.

Additional data requirements, such as substantiation of weight predictions and/or estimates, post-design analysis report, weight control and management plan report, special studies, etc. usually are specified in the

TABLE 10-1
WEIGHT DATA REQUIREMENTS—TYPE OF REPORT

▭ = OPTIONAL

REQUIRED DATA / TYPE OF REPORT — MIL-W-25140 PARAGRAPH	WEIGHT STATUS REPORT 3.7.1	ESTIMATED REPORT 3.7.2	CALCULATED REPORT 3.7.3	ACTUAL - FIRST AIRCRAFT 3.7.4	ACTUAL - INTERMEDIATE AIRCRAFT 3.7.5	APPENDICES TO ACTUAL REPORT 3.7.6	ACTUAL WEIGHT - LAST AIRCRAFT 3.7.8	DESIGN STUDIES AND RESPONSE TO SOLICITATION 3.7.2
SUMMARY TABLE								
GROUP WEIGHT STATEMENT								
DETAILED WEIGHT STATEMENT								
BALANCE CALCULATIONS (DETAILED)								
WEIGHT EMPTY								
GROSS WEIGHT CONDITIONS								
GEAR RETRACTION DERIVATION								
SPECIAL LOAD ITEMS								
REVISIONS TO PREVIOUS CALCULATIONS								
CG ENVELOPE								
LIST OF GFAE WEIGHTS								
GFAE WEIGHTS (SPEC VS CURRENT)								
STRUCTURAL DIAGRAMS								
TABLE AUTHORIZED CHANGES								
DCPR DETERMINATION								
WETTED AREA BREAKDOWN								
TABULATION OF OVER/UNDER WEIGHT								
SUMMARY TABULATION OF CHANGES OVER PROTOTYPE								
SERIAL NUMBERS OF APPLICABLE AIRCRAFT								
ACT. WEIGHT EMPTY & LISTS OF EQUIP. REMOVED & ADDED								
UNUSABLE FUEL AND OIL DATA								
TABULATION OF ALL CHANGES SINCE LAST REPORT								
WEIGHT HANDBOOK DATA (SAMPLE CHTS. A & E)								
WEIGHT SUBSTANTIATION, DERIVATION AND BACKUP DATA								

"How to Respond" section of the solicitation. These types of data have not been standardized in format and usually are outlined by the procuring activity. Detail format usually is established by coordination between the contractor and the procuring activity.

10-2.5.1 Summary Weight Statement

For all levels of weight determination, a summary weight statement comprising the applicable functional groups is to be provided in accordance with MIL-STD-1374, Part I. The useful load condition *shall* be as determined for the primary or basic mission defined by the applicable document; i.e., specification, RFQ, etc. It is important that the dimensional and structural data page of Part I be completed in every detail.

10-2.5.2 Weight and Balance Status Form

Subject to the requirements of the procuring activity, the rotorcraft weight and balance status form (MIL-STD-1374, Part III) may be included in the weight data requirements of an aircraft study or development contract. The importance of monitoring the weight progress of a vehicle development indicates that contractual requirements should include this item.

10-2.6 DEFINITIONS OF WEIGHT TERMS

Fig. 10-3 describes weight terms generally used in weight and balance control work. Some of these terms are technical engineering descriptions that are not employed in the operational phases of weight and balance work.

The definition of guaranteed weight empty is discussed in MIL-W-25140 as applied to both experimental and production contracts. The definition of trapped and unusable fuel and oil also is presented in MIL-W-25140.

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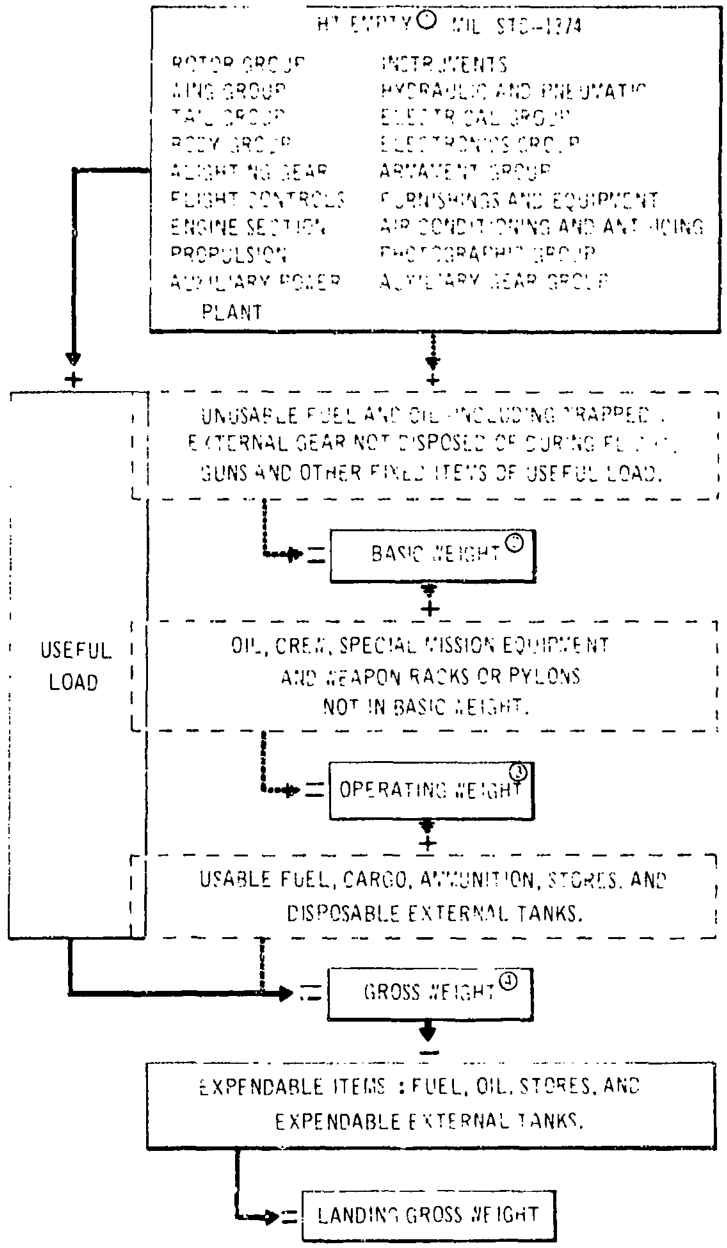
TABLE 10-2
WEIGHT DATA REQUIREMENTS—TYPE OF CONTRACT

☐ = OPTIONAL

REQUIRED REPORTS	TYPE OF CONTRACT: DESIGN STUDY & RESPONSE TO SOLICITATION	PROTOTYPE OR EXPERIMENTAL HARDWARE CONTRACT	PRODUCTION CONTRACT	AMENDED EXPER. & PRODUCTION CONTRACT	PARAGRAPH MIL-W-25140
ESTIMATED WEIGHT REPORT					3.7.2
WEIGHT AND BALANCE STATUS REPORT					3.7.1
CALCULATED WEIGHT REPORT					3.7.3
SAMPLE CHARTS A & E					3.7.9
ACTUAL WEIGHT REPORT, 1ST AIRCRAFT					3.7.4
APPENDICES TO ACTUAL WEIGHT REPORT					3.7.6
FINAL APPENDICES TO ACTUAL WEIGHT REPORT					3.7.7
ACTUAL WEIGHT REPORT, INTERMEDIATE AIRCRAFT					3.7.5
ALTERNATE POWER PLANT DATA					3.6.1(1)
WEIGHT SUBSTANTIATION AND DERIVATION					3.7.[illegible]
ACTUAL WEIGHT REPORT, LAST AIRCRAFT					3.7.8
ENGINEERING CHANGE PROPOSALS (ECP'S)					3.6.3
AMENDED EXPERIMENTAL REPORT					3.6.4
POST DESIGN ANALYSIS REPORT (FORMAT SPECIFIED)					
WEIGHT MANAGEMENT & CONTROL PLAN (FORMAT SPECIFIED)					



① Aircraft complete with all systems in accordance with the system specification.

② Entered on Chart C of Weights Handbook for running log weight.

③ Zero fuel and zero payload, a convenience weight to which operators [illegible] fuel and payload for gross weight.

④ Takeoff weight will vary with mission. It is the sum of the weight empty and the specified useful load.

Fig. 10-3. Weight Definitions

of the Fifth Annual Weight Prediction Workshop, Aeronautic Systems Division, Air Force Systems Command, October 1969.

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17. William H. Ahl, *Rational Weight Estimation Based on Statistical Data*, SAWE Paper No. 791, 1969.
18. *Weight Engineers Handbook*, Society of Aeronautical Weight Engineers, Inc., Los Angeles, Cal.

CHAPTER 11

MAINTAINABILITY

11-1 INTRODUCTION

11-1.1 IMPLEMENTATION

Maintainability is defined in MIL-STD-721 as "a characteristic of design and installation that is expressed as the probability that an item will be retained in or restored to a specified condition within a given period of time when the maintenance is performed in accordance with prescribed procedures and resources". To design for maintainability is to incorporate features that reduce the resources (time, manpower, personnel skills, test equipment, tools, technical data, or facilities) required to perform maintenance. The following specific areas serve in the planning process as a point of departure from which specific measures can be developed, for coordination of these and any other disciplines in a program with respect to maintainability:

1. Analytical techniques
2. Maintainability analysis
3. Maintainability demonstrations.

Implementation of maintainability features in the preliminary design of helicopters and helicopter-installed equipment requires the development of a maintainability program as described in MIL-STD-470. The standard describes in detail those tasks or elements that must be incorporated into a maintainability program to assure effective, timely, and economical accomplishment of maintainability design objectives.

The maintainability program plan should begin with the preliminary design of the helicopter. Included in the program are:

1. Development of a preliminary concept. The preliminary maintenance concept is based upon the proposed operational requirements for the helicopter and the Army maintenance support structure. The maintenance concept is a summary of the total maintenance planned to assure proper functioning of the helicopter and its equipment to meet the operational requirements. Distribution of the maintenance concept to appropriate design personnel is required to insure their understanding of the maintenance environment and resource constraints of the design. The preliminary maintenance concept should be updated whenever factors affecting the original concept are identified.

2. Participation in design trade-off studies. This is required to evaluate the effects that alternative designs would have upon the system maintenance requirements. As a minimum, trade-off studies present the relationships between helicopter alternatives based upon:

a. Maintenance manhours per flight hour (MMH/FH)
b. Mean-time-to-repair (MTTR)
c. Personnel skill requirements
d. Training requirements
e. Maintenance resource requirements.

Ideally, the trade-off study findings would rank helicopter design alternatives in the order of their demand for maintenance resources. The maintainability prediction techniques outlined in MIL-HDBK-472 are useful for establishing the necessary parametric values used in trade-off analysis.

3. Preparation of maintainability design guidelines. Based upon the maintenance concept, maintainability design criteria can be stated and design guidelines established that must be considered in order that the overall design will meet the applicable criteria. The maintainability design criteria of Refs. 1 and 2 are representative of mechanical and electronic equipment, respectively.

4. Participation in design reviews. Participation of maintainability engineering personnel in formal and informal design reviews is required to insure that continuing attention is given to maintenance considerations as the helicopter design evolves. The design review provides a means of exchanging information regarding maintainability problems and solutions; and of coordinating maintainability engineering inputs with those of the structural, reliability, manufacturing, quality assurance, and human engineering disciplines.

5. Identification of potential maintenance prob-

lems. This is a continuing responsibility of the maintainability engineer. Some potential problems are:

a. Use of subsystems or equipment that have known high failure rates
b. Nonoptimum routine maintenance schedules
c. Lack of accessibility to installed equipment
d. Hinderances to equipment fault isolation capability
e. Any apparent excessive maintenance resource demands
f. Undue complexity of maintenance tasks
g. Hazardous conditions and procedures
h. Undue complexity of system design.

6. Demonstration or verification of requirements. As part of any maintainability program, provisions must be made for the continuous monitoring of data that can be compared with predicted values. This is required by MIL-STD-473 and is important especially during the early test phases before the system enters operational service.

Items 1 through 3 are essential to the preliminary design phase and 4 through 6 are essential to the development and airworthiness qualification phases (see Chapter 10, AMCP 706-203).

11-1.2 MAINTENANCE ALLOCATION AND THE MAINTENANCE ENVIRONMENT

The Army's maintenance support structure consists of four levels having specific and distinct responsibilities: organizational, direct support, general support, and depot. The organizational level, that group closest to the daily field use of the helicopter and considered to be the first level, is responsible for the routine servicing and the preventive and corrective maintenance performed on the helicopter. Direct support levels of maintenance sustain the organizational levels, restoring equipment removed from the helicopter to a serviceable condition by replacing failed major assemblies or by adjusting equipment that is out of tolerance or alignment. The general support level restores equipment to service by repairing modules, subassemblies, and piece parts that do not require extensive and complex disassembly or assembly. Depot levels perform all the maintenance requiring special skills, facilities, and the use of costly and complex support equipment. Repairs made at the depot are accomplished by replacing piece parts of nonrepairable subassemblies.

The determination of the type and quantity of spare parts to be stocked at each maintenance level is developed from the preliminary maintenance concept. Several spares stockage and replacement approaches are possible. One approach advocates the replacement of an entire subsystem when service life has expired. This approach could require the stockage of large numbers of costly and bulky spares, and thus may be uneconomical. Also, under this approach the direct support level of maintenance would be responsible for testing entire subsystems to effect a repair, thus duplicating the roles of the higher-level organizations. At the opposite extreme is an approach that provides for the stockage of large quantities of many types of equipment subassemblies and modules at the organizational level and necessitates the introduction of special test equipment and highly skilled personnel at that level. An approach that provides spares for the major equipment functional packages and components—without providing spares for their subassemblies, modules, or piece parts (with the exception of installation hardware)—might prove more efficient. The determination of an optimum maintenance approach to apply at the various organizational levels is a prime objective of the maintainability program.

Direct support spares most often are modular mechanical or electrical subassemblies that can be tested, replaced, or adjusted without requiring special skills, test equipment, or facilities. The replaceable module should be accessible directly when installed on equipment and should perform a complete function or related group of functions when operating within the subsystem.

General support spares normally consist of the assemblies contained within the modules or attached to the modules for which spares are provided at the direct support level. Component spare parts also are provided at the general level when those components can be replaced without requiring special facilities, skills, or tools and equipment.

Depot level spares normally are the piece parts and nonrepairable assemblies required to effect complex repair and overhaul of equipment. Special skills, facilities, and the more complex and expensive test equipment are provided at the depot.

11-1.3 MAINTENANCE-INDUCED FAULTS

Table 11-1 presents the percentage of accidents (total, major, and minor) and of all mishaps (total, major, minor, incident, forced landing, and precautionary landing) attributable to maintenance errors.

Those features of a helicopter design that offer potential for generating maintenance-induced faults are:

1. An installation that can be positioned in more than one way

2. A design that requires unnecessarily complex procedures to maintain

3. Locations of hardware, connectors, and adjustment features in areas that are not readily accessible

4. A design that neither features equipment handling aids nor considers transportability needs.

Specific design guidelines for reducing maintenance error potential are contained in Refs. 1 and 2. Design techniques that eliminate or reduce maintenance-induced faults include:

1. Eliminate the possibility of installing equipment in more than one way. This includes positioning of the equipment in the helicopter; attaching electrical, fuel, hydraulic, or pneumatic lines; and making connections to ground support equipment.

2. Do not permit components or assemblies to be interchangeable unless they are identical functionally and physically.

3. Minimize the dependence upon training of personnel, marking of equipment, or maintenance and operating instructions to eliminate or reduce maintenance error.

4. Do not position those provisions for adjustments that are not to be made by lower levels of maintenance in such a way that they are exposed to lower-level maintenance personnel in the normal conduct of their work.

5. Identify complex maintenance tasks early in design developments in order that design changes that will reduce the complexity and criticality of the task may be effective.

6. Design for ease of handling and include special handling aids, fixtures, cradles, and dollies where they would aid the accomplishment of maintenance.

7. Incorporate builtin test features in equipment whenever possible. Builtin test features should minimize the need for individual judgment in determining either the need for replacement or the status of equipment. Desirable features of highly developed builtin test systems are:

a. Fault isolation capability of 90% or more

b. Builtin self-test capability

c. Low false alarm rates (2% or less)

d. Failure indicators that identify the failed replaceable unit

e. Automatic start-to-finish test capability.

8. Place early emphasis upon the human factors aspects of maintenance work. This includes simplification of maintenance training programs, diagnostic and repair manuals, audio-visual aids, special maintenance tools for the helicopter design and a recognition of the cost-effectiveness aspects of the depth to which builtin test features should be included and described.

The major thrust of efforts to reduce maintenance errors most often is directed at the organizational level of maintenance because errors at this level have a directly visible effect upon helicopter safety and availability. The guidelines applied to the design of equipment maintained at the organizational level, however, also should apply to the higher levels. These guides extend to the special equipment supplied to support the new equipment. For example, consideration should be given to the operational interfaces of special support equipment to insure that the equipment is connected easily to the equipment being tested and that connectors cannot be joined improperly. Special test equipment also could feature utilization of automatic test and fault isolation capability to reduce the dependence upon operator skills and prevent operator error.

TABLE 11-1
RELATIONSHIP OF HELICOPTER ACCIDENTS/INCIDENTS TO MAINTENANCE ERRORS

	PERCENT OF TOTAL ERRORS			
	ACCIDENTS FY		ALL MISHAPS FY	
HELICOPTER	1970	1971	1970	1971
UH - 1	6.24	3.99	12.78	5.97
OH - 58A	0	2.13	4.84	1.56
OH - 6A	7.25	3.77	7.93	4.93
CH - 47	6.45	4.35	13.24	6.22

SOURCE: U.S. ARMY AGENCY FOR AVIATION SAFETY

11-2 INTERCHANGEABILITY AND REPLACEABILITY

11-2.1 INTERCHANGEABILITY

Components are considered interchangeable when they are structurally, physically, and functionally identical and utilize the same attachment provisions. Calibration, alignment, or adjustment of interchangeable equipment components is undesirable. For example, MIL-E-19600 states that electronic modules are designed for interchange without the need for alignment or adjustment of the replacement module or the equipment in which the module is being installed.

The advantages of interchangeable components are obvious. At the two lowest levels of maintenance, the use of interchangeable parts fits well with the objectives of minimizing helicopter down times, minimizing technical skill level demands, and reducing the spare parts requirement. At the general and depot levels of maintenance, the need for such parts can be expected to be less. Any decisions involving the introduction of interchangeable parts into the supply system, however, must be weighed against the impact upon helicopter performance and resource demands.

The principal techniques used to achieve interchangeability in design and manufacture are standard throughout industry. They include:

1. Use of standard engineering drawing practices (MIL-STD-100)

2. Establishment of common tolerances for similar equipment components

3. Use of construction materials that have the same or similar physical properties.

Selected examples of more specialized efforts to design for interchangeability are discussed in the paragraphs that follow.

Alignment of subsystem equipment (such as that required to boresight a weapon subsystem) or calibration of avionic subsystems often are complex tasks. Methods of easing the alignment problem have been found. One involves the incorporation of controlled mechanical hard points and the use of machined surfaces to permit single adjustments in azimuth and elevation. The self-calibrating avionic module, a technique of reducing avionic adjustments in digital subsystems, offers potential for helicopter application. Self-calibrating modules employ digital logic circuitry that allows the regulation and correction of input and output data.

Accuracy requirements associated with air data subsystems sometimes necessitate the use of calibration, or compensatory, circuits mounted on circuit cards, which provides compatibility of a pressure sensor with the subsystem. For interchangeability, the peculiar sensor card and sensor can be packaged together so that the individually noninterchangeable components become a single interchangeable package.

The design of wire bundles and harness assemblies mounted in airframes and within equipment assemblies should receive particular attention in helicopter design. Layout within structures, restraint methods, and size should be considered carefully to ease replacement. One possibility for interchangeability is to use wire harnesses assembled with disposable nylon ties that do not terminate in screwdown or solder terminal boards, and that serve a limited number of functional subsystems. Such assemblies offer the obvious advantage of ease of replacement by lower levels of maintenance.

At the higher levels of maintenance, the use of modular design for interchangeable mechanical and electrical component assemblies can reduce the time required to restore equipment to a serviceable condition. Even turbine engines have been designed as a group of modules that can be repaired by replacement of the failed module. On the flight line, these engines can be replaced as a complete unit rapidly and easily by using an installation technique similar to the rack and panel type. An adaptation of this installation method may be suitable in certain helicopter applications. Rack and panel installations of airborne avionics also provide significant improvements in the ease of interchange of avionic modules over conventional mounting bolt and quick disconnect installations.

The use of Military Standard discrete components and piece part hardware provides for the tolerance, manufacturing control, and process control necessary to insure interchangeability at the lowest component level. Design requirements must be established that assure maximum utilization of such parts and permit deviation only when justified.

Limitations on the use of coatings, potting materials, and other encapsulating materials over components and assemblies must be established. Where cost of the coated assembly permits, a throwaway concept should be recommended. All low-cost, nonmechanical components classed as throwaway items should be encapsulated when practicable to prevent any attempt at repair.

11-2.2 REPLACEABILITY

Components are considered replaceable when they meet all requirements of interchangeability except that their installation may require adjustments or operations in addition to installation of the attaching means. These procedures may include drilling, reaming, filing, trimming, or other operations necessary for installation of the component into the mating assembly. Such operations do not include shearing, bending, forming, or other basic operations that cannot be performed readily with ordinary hand tools.

Structural component installations form the majority of this class, and most are installed with rivets. Plate nuts and other captive fastener devices also are secured with rivets. The structural components of helicopters and of installed equipment are secured in this manner primarily to assure that structural integrity is maintained.

Sometimes, reduction of the shaft length of electrical components such as variable resistors is required to conform with specific installation requirement. Trimming of components generally is required to provide a component that can be used in several applications.

The widespread use of components that must be processed prior to installation is undesirable, but they may be employed to facilitate meeting environmental requirements (vibration, shock, and strength). Their use in other applications is reasonable when the failure rate of the components is low enough to indicate an extremely low probability of repair. The consequences of using components that must be processed prior to their installation are increased MMH/FH and MTTR, additional complexity of maintenance, and increased maintenance [illegible] potential. Any trade-off involving a consideration of their use must be justified on the basis that it is the only acceptable solution to meeting design requirements. Item cost, by itself, cannot be the deciding factor. Any savings in obtaining the component in its noninterchangeable form may be lost in an increase in the cost of performing the work prior to installation and the reduced helicopter availability due to increased maintenance downtime.

Certain items in the replacement part category are not components in the normal sense. Repair stock used to patch the skin or equipment fairings is used commonly and justifiably, preferably by maintenance personnel at the direct support level. Its use permits needed repairs at an austere site and makes it unnecessary to stock large preformed skin sections and fairings. Sheet metal, usually aluminum, of the type used to cover the frame, and Fiberglas kits for fairing repairs are maintained for this purpose. Any major repairs—where extensive forming or rebuilding is required—should be performed at higher levels of maintenance. Avionic and other mechanical equipment requiring repairs of this type should be sent to higher maintenance levels because the suitability of such repairs and the equipment performance cannot be verified with the test equipment normally available at the lower levels.

11-3 STANDARDIZATION

There are levels of standardization applicable to helicopter system design that have separate and distinct but related advantages. The extent to which standardization benefits system development is dependent in one respect upon system and equipment performance and design requirements. Requirements that permit the use of existing developed subsystem equipment or the modification of such equipment without a complete development program are especially advantageous. Subsystems—such as engine, armament, navigation, fire control, communication, and airframe—that can be used in or adapted to the new system requirements provide the following advantages:

1. Decrease in weapon acquisition and ownership costs due to reduction of:
   a. Scope of design development program
   b. Scope of official qualification requirements
   c. Training requirements
   d. Introduction of new spares
   e. New tool requirements
   f. Requirements for special support equipment
2. More rapid development of the total system
3. Product improvement through modification of equipment to correct design deficiencies recognized by using organizations, and/or by taking advantage of state-of-the-art improvements
4. Compatibility with special tools and support equipment
5. Experience of maintenance technicians.

Although all of the advantages described may not result from subsystem standardization for a particular application, only partial benefits are required to realize a significant savings in cost of development and ownership. Attainment of these savings must begin in the proposal design phase. The system specification should be studied in detail and its requirements compared with equipment specifications of procured equipment. Studies of this nature should not be limited to Army inventories alone, but should evaluate other military, federal, or commercial equipment. For example, head-up displays, central computers, instrumentation, and fuel control components used in high-performance aircraft might be adaptable readily to meet helicopter equipment specifications.

A more common form of standardization—the use of standard parts—offers obvious potential for the design of new equipment. Standard parts are identified by Air Force–Navy (AN), Military Standard (MS), Joint Air Force–Navy (JAN), and National Aeronautical Standard (NAS) prefixes followed by a numerical suffix. This designation identifies the mechanical or electrical equipment component as complying with a Governmental specification. The component is designed, fabricated, and finished to the requirements peculiar to that specification and may be available for procurement by order from a number of manufacturers. The production of the standard component is subject to inspections and tests at the manufacturer's facility to assure that the component physical and functional characteristics are interchangeable and as specified.

Requiring incorporation of standard components in new equipment design is essential to reducing the quantities of component spares and simplifying the procurement process. Deviations from the requirement, although sometimes necessary, must be limited in order to avoid the complications associated with nonstandard part use. Nonstandard parts may penalize support and add to development cost because:

1. The parts must be specially manufactured

2. The parts must pass qualification tests prior to use

3. A sole source of supply may exist for the part, in which case the source may go out of business and/or a second source must be found.

One method of attaining maximum part standardization is to establish a master standard part list. There are many standard parts, and any one contractor involved in the design of a helicopter is likely to maintain a list of preferred standard and special parts based upon his part investigations and applications. These preferred part lists can vary radically from one contractor to another. If each contractor designs to his own list, a large number of differing parts is required to support the system. The majority of parts in this case may be standard but the quantities of distinct types are a burden to maintain. Reduction in the number of types required to support the system is obtained through the development of a single master standard preferred part list. Cost reduction also is attained when a single contractor is responsible for the ordering and distribution of parts to associate contractors. Criteria for the selection of preferred parts are established by the contractors as a joint effort that takes advantage of each contractor's experience related to the use of the part. The individual part then is selected on the basis of its reliability, maintainability, and environmental characteristics.

Existing ground support equipment and tools should be usable for the new helicopter system. Design requirements must specify that the helicopter be compatible with electrical power, hydraulic power, pneumatic power, and test and refueling equipment. The performance characteristics of this equipment also must be compatible. Standard mechanics' tools should be sufficient to remove and install all common helicopter equipment and the requirement for special tools should be kept to an absolute minimum.

Compatibility with existing automatic test equipment is another important aspect of standardization. This equipment is used to confirm the presence of malfunctions, locate the malfunction, and perform repair verification testing.

11-4 ACCESSIBILITY

Helicopter accessibility often is limited because of the necessity of meeting physical design constraints imposed upon the system to assure performance and other design requirements. The specific physical design requirements affecting accessibility are weight, volume, CG limitation, and structural integrity. These characteristics in turn affect the density of equipment installations; selection of installation locations; and the size, methods of security, and numbers of access panels/-doors. Flight performance requirements normally do not permit external installation of equipment except for special applications such as releasable fuel and weapon stores. Optimum location of equipment components not always is permissible because of the effects of equipment weight on the CG. Physical configurations often require that equipment be installed in dense arrays wherein equipment access is impeded by the proximity of surrounding equipment. Structural integrity requires the use of braces and ribs to strengthen helicopter structure and also requires that access doors located at certain locations be secured with quantities of structural mounting bolts. The permissible size of access doors at certain critical locations also is limited.

Rapid and easy access to equipment is required to perform servicing and preventive and corrective maintenance within a reasonable amount of time. Therefore, accessibility priorities must be established during the preliminary design phase to insure that special design attention is focused upon access to critical equipment and interfaces. The factors that should be considered when establishing these priorities are, in order of importance:

1. Servicing to satisfy operational requirements. Fueling, munition loading, and other mission-dependent requirements demand that access to helicopter ground support equipment interfaces and preflight-inspected equipment be attained quickly.

2. Frequently performed scheduled maintenance. Daily and weekly inspection tasks account for a significant percentage of all maintenance time expenditures at the organizational level. Providing ready access for performing inspection and servicing tasks facilitates the performance of these tasks and reduces the maintenance load.

3. Frequently performed corrective maintenance. Equipment with a relatively high failure rate should be located in readily accessible positions within the helicopter.

4. Infrequently performed scheduled and unsched-

uled maintenance tasks. Access to support these tasks is given the lowest priority.

The developing installation design should be monitored to assure that access requirements are met. Design features that aid accessibility are:

1. Location of access doors at heights where they may be opened and closed readily at normal working levels without auxiliary stands or other equipment. All high-priority equipment and interfaces should have this characteristic.

2. Incorporation of builtin aids such as walkways, work surfaces, and telescoping ladders that afford access to work areas that are out of normal reach and otherwise would require maintenance stands or other equipment to reach them

3. Provision of spring-loaded, single-fastener, or latch-secured doors at refueling, rearming, and ground power interfaces

4. Location of inspection gages, meters, maintenance panels, high failure rate equipment, fluid fill ports, and support equipment behind readily actuated doors

5. Provision of inspection windows or directly accessible indicators for determining fluid levels, pressures, weapon safing, rounds remaining, and filter status

6. Provision of hinged doors in lieu of panels requiring complete removal to obtain access

7. Placement of access doors within larger structural doors when it is necessary to attain quick access to facilitate high-priority maintenance tasks

8. Use of rack and panel installations. This type of installation is particularly advantageous because the need for access to disconnect and connect the equipment is eliminated or reduced substantially.

9. Location of equipment connectors, metering devices, controls, and indicators on the most accessible equipment surface

10. Provision of mechanical stops or braces to hold access doors open

11. Avoidance of locations (for high-priority access) that feature proximity to bulkheads, rods, wire harness, pneumatic and hydraulic lines, shelves, structural ribs, or other potential sources of access interference

12. Provision of equipment bays wherein arrays of subsystem equipment are installed by rack and panel methods. The equipment installed in such bays should be components of the same subsystem and interfacing subsystems in order to facilitate maintenance.

13. Provision of test points and adjustment controls that are accessible directly without requiring case covers or panels to be removed

14. Use of minimum quantities of captive quick release fasteners to secure case covers and panels

15. Location of subassemblies with high failure rates in a manner that assures immediate access for maintenance after covers or panels are removed

16. Packaging of meters and controls in modules to allow their removal from equipment without requiring cover or panel removal

17. Provision of direct access to mounting screws and bolts, and incorporation of tool guides to them when visual access is limited or a hazardous condition exists

18. Avoidance of the use of cordwood construction and installation techniques

19. Avoidance of box-within-a-box designs except as they are necessary to permit pressurization or provide electromagnetic interference protection.

11-5 SPECIAL TOOL CONSIDERATIONS

The need for special tools such as jigs, fixtures, and templates to support maintenance actions is undesirable. The factors that follow should be considered in the event that a special maintenance tool requirement is being considered in the design:

1. Cost. Use of special tools involves more than the cost of their design and fabrication. MIL-STD-454 specifies that the special tools be mounted accessibly within equipment or in arrays located near the equipment installation. Conformance to this requirement adds to the cost of the design installation and of the securing fixture. A worst-case possibility of satisfying this requirement is the necessity of extensive equipment design to accommodate the tool.

2. Maintenance and human factors engineering. Technicians do not like to use special tools and often find other means of accomplishing tasks that the special tool was designed to support (Ref. 3). Individual refusal to use the special tool also could cause the task to be omitted or performed improperly, with resultant damage to the equipment.

Maintenance time sometimes is increased when a special tool is used. This is because the amount of time to perform the maintenance tasks must include time to secure the tool. At maintenance levels higher than the organizational level, special tools generally are kept in a tool crib with access and inventory control procedures imposed. If a special tool is lost or misplaced, it becomes a critical item if a replacement cannot be obtained readily. The technician's unfamiliarity with the special tool also hampers its use. The technicians available often are involved in on-the-job training programs and have no prior knowledge of the special tool.

11-6 MAXIMUM TIME BETWEEN OVERHAULS (TBO)

11-6.1 FACTORS IN DETERMINING OPTIMUM TBO

Overhauls are required to assure continued performance, arrest degradation, and restore mechanical and electrical subsystems to their specified operating conditions throughout their useful lives. Helicopter engines, transmissions, weapons, flight controls, hydraulic subsystem components, electrical power generating components, and landing gear make up the bulk of helicopter equipment requiring overhaul. Avionic subsystems characteristically do not require overhaul because of their peculiar component life characteristics. The necessity for overhaul of mechanical equipment is inherent within the subsystem components themselves and is dependent upon their operational environment, the physical properties of the materials used to fabricate them, and the processes used to protect them in their environment.

Manufacturers of equipment recommend overhaul intervals. The basis for their recommendations is derived from the failure history of similar components used in like applications, the results of dynamic testing of components, and the analysis of materials. Additionally, the manufacturer must factor in the elements of the new design that affect overhaul.

The new design may use materials and processes having little history by which an accurate overhaul estimate can be made. On the other hand, the materials and processes may have properties that appear to permit a state-of-the-art advancement in the equipment design. Some properties that affect overhaul intervals are wear, fatigue, corrosion, erosion, crystallization, and fracture. Materials that resist failures due to their being less susceptible to these factors are the most desirable for use in mechanical equipment design. The use of these materials should be on such a scale that an even distribution of degradation occurs within the equipment in the prescribed environment and operation. The use of materials with substantial differences in resistance properties will promote a condition wherein the life of the least resistant material establishes the overhaul interval. Less resistant materials have a place in the design and may be used effectively in applications where they are not subject to stress and wear and have finishes that provide resistance to galvanic corrosion and moisture.

The use of modular design permits a reduction in the complexity of overhaul tasks and an increase in intervals between overhauls. This increase will be due primarily to the reduced number of components subject to environmental degradation; consequently, interval selection is less critical.

11-6.2 ADVANCES IN AIRBORNE MONITORING SYSTEMS

Airborne Integrated Data Systems (AIDS) and Engine Control and Monitor Systems (ENCOMS) are recent developments applicable to helicopters that were made possible by component state-of-the-art advancements. Digital and sensor technological improvements are the primary factors permitting the use of these systems in airborne applications.

A typical AIDS consists of sensors, comparators, multiplexer computer, data processor, recorder, and annunciator panel. The sensors and comparators permit the monitoring of temperatures, vibration amplitudes, and acceleration to determine operating conditions and detect deviation of these parameters from prescribed limits. The multiplexer is programmed to sample the sensed data periodically. A computer and processor then convert these data into digital form for recording and display on the annunciator panel. The annunciator panel provides the aircrew with data on equipment condition. The recorder permits retrieval of this information for subsequent analysis if desired.

AIDS equipment provides several features particularly attractive for the performance of maintenance. The typical capabilities of an AIDS are:

1. Monitoring and isolating faults in avionic subsystems

2. Monitoring and controlling engine status and performance. Where sufficient information is provided, the AIDS data can be analyzed to indicate that degradation of engine components has progressed to the point that overhaul is due. The overhaul then can be performed in response to a measured need rather than in accordance with a schedule established on the basis of statistical study of similar engine uses.

3. Gathering data on the stresses applied to structures and the numbers of times that established stress

limits have been exceeded. These data may be used to determine when inspections of structures are required.

4. Gathering other flight history data. These data are used to detect potential design deficiencies and to provide a basis for product improvement decisions.

ENCOMS are similar to AIDS except that the capabilities are limited to the collection and analysis of engine data only.

Incorporation of an airborne monitoring system in smaller helicopters would not appear practical from cost, weight, and subsystem complexity points of view in light of the current state of development. For larger helicopters, the size and weight of the airborne monitoring system might be reduced if ground equipment were provided to process and display selected information.

Where weight, volumes, and cost constraints are less critical, the possibility exists for expanding the monitoring, display, and computer capacity of airborne monitoring systems to store diagnostic and repair instructions. The use of a complete airborne monitoring capability of this scope could provide the potential to:

1. Ease maintenance technical requirements by providing the technician with aids for isolation of faults and instructions to guide the repair
2. Preclude the performance of unnecessary scheduled maintenance
3. Detect potentially hazardous conditions
4. Increase helicopter system effectiveness.

11-7 MINIMUM INSPECTION

Scheduled inspections of helicopters and helicopter installed equipment generally are performed by technicians either by visual observations or with the aid of special equipment such as an X ray or a dye penetrant for airframe structural checks. Maintenance actions normally are initiated to correct deficiencies found during the course of inspection. Inspection thus provides the following benefits:

1. Sustains aircrew confidence in the integrity of the aircraft
2. Uncovers potentially hazardous conditions prior to flight
3. Provides a means of identifying conditions that might prevent successful mission performance
4. Determines the effectiveness of corrosion control programs and other preventive maintenance tasks
5. Detects malfunctions of equipment subsystems, particularly of little-used modes of operation.

The inspection requirements can be justified readily when the benefits of inspection are considered and the consequences of failing to make the inspections are understood. Scheduled inspection, however, represent a significant maintenance load to the maintenance crew at the organization level. Daily inspections in particular tax the capabilities of using units. In actual practice, technicians tend to concentrate on inoperative units; thus, the inspections may be cut short or neglected. The helicopter designer, therefore, should incorporate design features that minimize the necessity of performing inspections and make inspection requirements as simple as possible.

Minimizing inspection requirements is achieved best by using inspection standards such as MIL-M-5096. The inspection criteria become the basis for preliminary inspection requirements. Flight test programs can be used to verify the effectiveness of the inspection requirements. An inspection validation program provides a basis for decisions as to which requirements will be retained and which will be discarded. Additionally, the validation program can be used to identify new inspection requirements. The major task to be performed during the preliminary design period is to translate the inspection requirement criteria into preliminary inspection requirements. Criteria used for this purpose are:

1. Inspections that will guard against hazards that could contribute to equipment damage and affect personnel safety
2. Inspections that insure that systems, assemblies, and components checked or removed during the course of maintenance are reinstalled properly and secure
3. Inspections that will insure early detection and correction of defects
4. Inspections that will assure a thorough and continuing effort to minimize or prevent the development of corrosion and will assure the early recognition of corrosion-producing conditions
5. Requirements that will assure that inspection, test, replacement, etc., needed on a repeating basis are accomplished
6. Assurance that no inspection is required of items that are observed daily during equipment operation or that are monitored during operation (equipment that is automatically monitored by builtin test)
7. Assurance that every inspection of components permits determination of component condition.

Simplification of inspections should be considered a design requirement. One of the most effective ways of simplifying inspection is to provide rapid, simple access to the area to be inspected. The techniques and criteria for attaining the necessary accessibility are discussed in par. 11-4. Another means of simplifying inspection is to provide builtin inspection aids. For example, air vehicle tires have been designed with inspection aids such as grooves or holes in the tread that, when worn to a specified level, indicate that the tire requires change. Hydraulic filters are designed with flag indicators that alert the inspector that they should be replaced or cleaned. Similar aids should be considered and incorporated whenever possible. The procedure of inspection itself provides a means of simplification. Inspection procedures should be written in simple syntax with a limited and consistent vocabulary and using a sequence of steps that yields the most effective use of maintenance time. The instructions for the preparation of inspection procedures and work cards have been standardized in MIL-M-5096.

11-8 GROUND SUPPORT EQUIPMENT (GSE) INTERFACE

The capabilities that make helicopters desirable, such as operation from remote sites and from unimproved fields, also suggest that the needs for ground support equipment be minimized and standardized to permit flexibility of use. Requirements for the GSE should be established simultaneously with the helicopter design development. If self-sufficiency of the helicopter in an austere environment is a primary design goal, it is achieved best if all maintenance support equipment is self-contained. Total self-sufficiency is, of course, beyond attainment within normal economic constraints.

Builtin test (BIT), one of the most promising current technological advances, offers the potential for reducing GSE requirements at the organizational level of maintenance. BIT is discussed briefly in par. 11-1.3. BIT use has been associated predominantly with electronic and electrical subsystem equipment, and it has served to reduce associated GSE requirements substantially. The AIDS and ENCOMS subsystems discussed in par. 11-6 indicate that BIT capabilities adaptable to mechanical subsystems possess further potential for reducing GSE. The self test, fault detection, and fault isolation capabilities of BIT eliminate both the standard and special GSE required to support corrective maintenance and scheduled functional tests of equipment. BIT, coupled with technological improvements in avionic design, has all but eliminated requirements for performing functional tests and has reduced requirements for inspections prior to flight. Automatic BIT, coupled with maintenance instructions (a status panel and failure indicators), also is easier to use than GSE when performing corrective maintenance and has the added advantage that it does not have to be obtained, transported, connected, and disconnected. The penalties realized by its use are increased equipment weight, cost, complexity, failure rate, and volume. However, the development of integrated circuitry and digital techniques is reducing rapidly the disadvantages of BIT and permitting an increasing capability in terms of percentage of fault isolation allowable without serious penalty.

An additional capability of BIT that has not been used to any major extent is its application in detecting and isolating faults in modules of replaceable assemblies. Incorporation of this capability would entail additional BIT complexity but also would serve to reduce GSE requirements at the direct support level and possibly beyond. As the direct support maintenance facility in the field is austere, the advantages realized from this feature would be similar to those described for the organizational level of maintenance. This capability would require that equipment modules be provided with fault indicators similar to those used on the helicopter replaceable assemblies using conventional BIT capabilities.

The elimination of other GSE such as hydraulic power carts and electrical power generating units also can be achieved. Some helicopters are provided with onboard auxiliary electrical power units that permit operating of subsystems on the ground to perform verification. Also, incorporating an electric motor-driven hydraulic pump to permit hydraulic subsystem operation would eliminate two of the largest GSE items normally required, the auxiliary hydraulic and electrical power units. The helicopter still would retain the compatible interface connector for use with existing auxiliary hydraulic and electrical power units to conserve the onboard units for austere site use. However, this auxiliary power feature would be limited to helicopters that could afford the weight and volume represented by the addition. The auxiliary electrical power unit would exact the greatest penalty; electric motor-driven, compact, lightweight hydraulic pumps have been developed that provide pressures and flow rates consistent with helicopter requirements. Currently, gas turbine electrical generators provide the best physical characteristics for use as auxiliary electrical power units.

The smaller GSE, such as levels, gages, thermometers, and meters, also can be built into the helicopter either as modified versions of the GSE it is replacing or as a BIT function, if practicable. The advantages of the builtins are reduced here because supplying these small items at an austere site is accomplished more easily.

As GSE requirements are identified, currently available GSE as defined in MIL-HDBK-300 should be studied to determine which existing items will meet the new helicopter requirements or whether the helicopter design can be changed to enable it to use existing GSE without compromising system objectives. When the GSE capabilities are compatible with the identified design requirements, the design also must provide a compatible GSE-to-helicopter interface. Compatible connectors, fluid ports, etc., should be of a size and type that permit direct connection to the GSE. Failing this degree of compatibility, adapters should be designed to provide the desired interface compatibility.

At the higher levels of maintenance—particularly depot, where automatic support equipment composed of a variety of components is used—compatibility also should influence the design. The types of test connectors used with the equipment are of particular importance. The versatility of the automatic test equipment will not penalize or restrict the equipment design in other ways. As an example, if a requirement to measure a parameter is identified as being beyond the capabilities of the automatic test equipment, this new capability can be added to the GSE. This may be made possible, for example, by the simple reprogramming of a tape-driven test command unit. The number of building blocks and test equipment components, and the range of capabilities of the building blocks, make it unlikely that any additional requirement will be placed on the command unit. When the interface between the GSE and the equipment is compatible, all that is required to permit testing of the equipment is to establish the appropriate levels for the test parameters of time, stimulus, and response to make an equipment test tape.

The direct support and general support levels of maintenance are provided with equipment and module test sets and standard test equipment. Standard test equipment available to these levels includes multimeters, oscilloscopes, voltmeters engine test stands, weapon mounting fixtures, and other electrical and mechanical test sets and fixtures. A new helicopter design introduces other special GSE of the equipment and module tester types with which the technician must become familiar. This new GSE must be compatible with the facility power and standard interfacing power fixtures; otherwise, special facilities would have to be designed and procured to permit operation of the new support equipment.

Handling equipment compatibility also must be considered. GSE such as weapon loaders, engine transport fixtures, and tugs should be compatible with the new equipment or compatible when adapters are supplied. For example, a weapon loader used by all three services (most recently by the Army to support the AH-56A) is equipped with a bomb table that is compatible with a variety of cradles and other special fixtures. This weapon loader is capable of supporting loading and unloading of special weapons, large missiles, smaller air-to-air missiles, a variety of bombs, tow targets, external fuel tanks, and special purpose pods. Providing these stores with hard points compatible with the adapters and tiedowns is all that usually is required to make this weapon loader compatible with new equipment. Engine cradles also must be compatible with new engines if a specific piece of handling equipment is considered when the new engine is designed. Again, appropriate hard points and tiedown areas must be incorporated in the design.

Due to the uniqueness of some equipment designs, special GSE is required to support new helicopter subsystems. The special GSE should be designed to provide maximum ease of utilization, and rapid fault isolation and test capabilities. Many special GSE designs in the past have failed to accomplish their function because, for the sake of design simplicity and economy, they failed to have these desirable characteristics. The GSE was difficult to use or otherwise was undesirable because:

1. Test control required the use of a quantity of interacting, multiposition switches or test jacks and auxiliary test equipment.

2. The GSE was heavy, requiring two or more men to transport it to the helicopter.

3. Modifications of helicopter equipment occurred faster than update of the support equipment capabilities. (This could be avoided if both the helicopter and GSE were procured with the same sense of urgency.)

4. Alternative procedures of maintenance were used by technicians because of the undesirable features previously listed.

The design of special GSE, therefore, should proceed after consideration of past design errors. Accordingly, new designs of GSE might:

1. Be automatic or semiautomatic

2. Employ medium- or large-scale use of integrated circuits and digital technology

3. Be within the one-man-lift capabilities established by human factors engineering standards

4. Employ self-contained maintenance instructions

5. Require minimum judgmental skills from the user.

A design meeting these requirements is integrated readily as a valuable maintenance resource.

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CHAPTER 12

RELIABILITY AND AVAILABILITY

12-1 INTRODUCTION

Technological growth is accelerated by the continuous search for improved military weapon systems. However, realization of this improved capability seldom is achievable without increasing system complexity, a factor normally associated with high cost and reduced availability. The significance of this apparent relationship is clear when one considers that cost and availability are two of the primary variables used to measure ultimate value of a military system, i.e., mission effectiveness (see Chapter 2).

Design methodologies have been developed in recent years that, when properly applied, insure that degradation of availability as a result of increased complexity is held to a minimum. One of the prime techniques employed is referred to as "reliability engineering".

This chapter discusses various aspects of reliability and their impact upon the availability variable used in mission effectiveness measurements. The purpose is to emphasize to the designer the importance of including reliability considerations in the development of Army helicopters.

For the purposes of this handbook the following definitions are applicable:

1. Maintenance reliability. The probability that the helicopter, when operated in the specified environment and maintained in accordance with specified procedures, can launch and complete a specified mission without incurring a malfunction requiring unscheduled maintenance. Maintenance reliability—with its related variable, unscheduled maintenance frequency—is a forcing function for required spares, maintenance personnel and support equipment requirements, aircraft availability, and operation and maintenance (O&M) cost.

2. Mission reliability. The probability that the helicopter, when operated in the specified environment and maintained in accordance with specified procedures, can launch and complete a specified mission without incurring a malfunction that causes delay, cancellation, or abort. Mission reliability is a forcing function for mission planning, aircraft assignment, and mission success rate.

3. Flight safety reliability. The probability that the helicopter, when operated in the specified environment and maintained in accordance with specified procedures, can launch and complete a specified mission without incurring a malfunction resulting in loss or severe damage to the aircraft.

12-1.1 AVAILABILITY, COMPLEXITY, AND RELIABILITY

In Chapter 2 the availability A_s of a system, in this instance a helicopter, was shown to be a function of the combined availabilities of each of its major subsystems, i.e.,

$$A_s = A_1 \cdot A_2 \cdot A_3 \cdots A_N = \prod_{i=1}^{N} A_i \qquad (12\text{-}1)$$

where

A_i = availability of an identifiable (or defined) subsystem (per AMCP 706-134)

A_N = availability of Nth subsystem

A_s = system availability

This same relationship holds for the component parts of each subsystem, e.g.,

$$A_N = A_1 \cdot A_2 \cdot A_3 \cdots A_n = \prod_{j=1}^{n} A_j \qquad (12\text{-}2)$$

where

A_j = availability of an identifiable component

A_n = availability of nth component

If necessary, this concept could be extended to each part of the entire helicopter. The overall availability, then, would become a function of the availabilities of each part.

The complexity of any system is proportional directly to the total number of discrete elements or separate component parts. In highly complex, nonredundant systems there is the clear implication of a probability of reduced availability, if for no other reason than a dependency upon a greater number of parts. If each of the parts was capable of 100% assured performance at all times, no degradation of availability would occur. Unfortunately, perfect components are not obtainable realistically; hence, the probability of lower availability due to larger aggregations of parts is high (Ref. 1).

Reliability R, defined as "the probability that an item will perform satisfactorily for a specified period of time when operated under prescribed conditions", usually is based upon the mean time between failures (MTBF) or its reciprocal, failure rate λ.

As with availability, the reliability R_s of an overall system can be shown (Ref. 2) to be the product of the reliability of each subsystem and, in turn, each component part, e.g.,

$$R_s = R_1 \cdot R_2 \cdot R_3 \cdots R_N = \prod_{i=1}^{N} R_i \qquad (12\text{-}3)$$

and

$$R_N = R_1 \cdot R_2 \cdot R_3 \cdots R_n = \prod_{j=1}^{n} R_j \qquad (12\text{-}4)$$

where

R_i = reliability of an identifiable subsystem

R_j = reliability of an identifiable component

R_N = reliability of Nth subsystem

R_n = reliability of nth component

R_s = system reliability

Because reliability always is less than one, the greater the complexity or total number of parts, the lower the overall reliability and availability of the helicopter. This general relationship is depicted conceptually in Fig. 12-1. Note the exponential relationship of failure to complexity.

12-1.2 IMPROVING RELIABILITY

Given that a certain degree of complexity is necessary if a helicopter is to satisfy its mission requirements, its system reliability may be increased by a variety of design techniques:

1. Redundancy. The provision of parallel or multiple subsystems or components, which assures continuity of performance despite the failure of one; normally these are provided at the risk of increased complexity, size, weight, cost, and additional maintenance requirements.

2. Derating. The operation of a subsystem or component at some level of performance significantly below its maximum design output; normally provided at some sacrifice of size, weight, and operating efficiency.

3. Diagnostic aids. The provision of subsystem or component status monitors that can predict incipient failure so as to allow remedial replacement; normally complex devices in themselves, requiring specialized maintenance provisions. These devices, if installed, must comply with AR 705-50.

4. Conservative design. The deliberate design of generous safety factors so as to insure operational integrity under a wide variety of operating conditions; normally obtained with some sacrifice in size and weight.

5. Component development. The refinement of design by repetitive test procedures so as to optimize the safety factors within the expected operational environment; requires additional testing (costs) and demands accurate specification of expected operating conditions (Ref. 2).

6. Selection. As part of the overall quality assurance program, selection of parts effectively can enhance reliability. Such a technique uses some combination of 100% acceptance testing, burn-in, and overstress "aging". Obviously, there is a concomitant (and sometimes large) increase in cost.

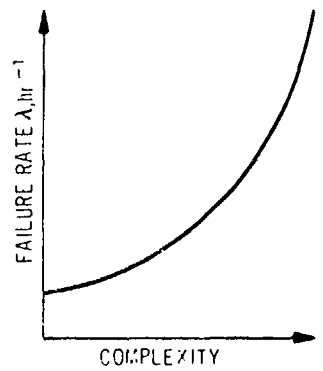

Fig. 12-1. Reliability as a Function of Complexity

The appropriate technique(s) may be selected on the basis of trade-off analyses, the nature of which are dependent strictly upon the characteristics of the particular problem. Factors such as cost, maintainability, size, weight, and producibility—as well as the system mission requirements and operating environment—are variable parameters that must be considered in such evaluations. Obviously, cost exerts maximum influence in most instances. Par. 3-4 describes general techniques for trade-off analyses.

There are a number of trade-offs required among reliability and maintainability, safety, survivability, vulnerability, and human factors engineering disciplines. These generally fall into one of the following categories:

1. Intra-reliability trades. Achieving the proper balance among the various reliability parameters of flight safety, mission success, and maintenance frequency requires assessing the impact of effectiveness and cost on the system life cycle. The three factors, in effect, compete with, rather than complement, each other during the design process. Reliability engineering, working closely with project designers during the performance of design trades, provides quantitative data upon which decisions can be reached that will achieve the established objectives.

2. Reliability, safety, and survivability/vulnerability trades. Survivability/vulnerability disciplines emphasize reduction in the consequences of equipment loss. Techniques that accomplish this include redundancy (backup functions) and equipment capable of failure detection prior to progression to a hazardous state. Unrestricted use of these approaches can compromise reliability because both add to the amount and complexity of equipment in the design.

3. Reliability and human factors engineering trades. The human factors engineering discipline emphasizes the human in relation to ease of operation and maintenance. The human factors discipline aids reliability by considering the reduction in operational and maintenance engineering errors through proper design, which, in turn, reduces equipment malfunctions due to operator or maintenance error. Human factors engineers, in their efforts to reduce crew workloads, can compromise reliability by adding relatively complex automatic features.

4. Reliability and maintainability trades. The maintainability discipline emphasizes the need for ease and quickness of maintenance. By considering such parameters as accessibility, standardization, and diagnostics, to name a few, maintainability directly contributes to equipment reliability by reducing total maintenance actions and the related chance for maintenance errors. As with human factors engineering, care must be taken to insure that maintainability does not compromise reliability by adding complex automatic features.

12-1.3 RELIABILITY PREDICTION

12-1.3.1 Purpose and Scope

Reliability engineering techniques are employed to insure that the degree of system availability is maximized for its military operational role. This process normally is applied throughout the developmental and operational life of the helicopter. Improvements are being sought constantly.

During the preliminary design stage, however, reliability engineering primarily is predictive in nature; i.e., preselected levels of reliability are achieved by application of design techniques. Normally, considerable amounts of reliability data must be accumulated to support this effort.

Overall system reliability need not be predicted on the basis of exhaustive and conclusive tests to failure of each and every component part. The costs and time associated with such tests would in most cases be prohibitive (Ref. 3). This does not mean that testing is of no value. Material, component part, and subsystem tests are fundamental to the science of reliability prediction provided the designer recognizes the inherent limitations of the techniques, namely:

1. The intended operational environment normally cannot be duplicated by testing. This environment includes the actual operating system and the interrelationship of all components in that system.

2. Realistic test limit criteria are difficult to specify.

Regardless of the limitations, reliability data obtained from testing often are the only tangible point of departure in design.

Other data sources can be used. Normally, the helicopter is monitored during all phases of development, test, and service life. Detailed maintenance records provide failure information on many subsystems and component parts. Vendor-supplied component reliability data also may be used.

The designer should be aware of the fact that failure rates are not uniform but may vary significantly with time (par. 2-3). The failure rate data selected for use in reliability prediction must be truly representative of the normal period of operations. With new systems, no such "experience" data are available, and the designer must rely on less valid information.

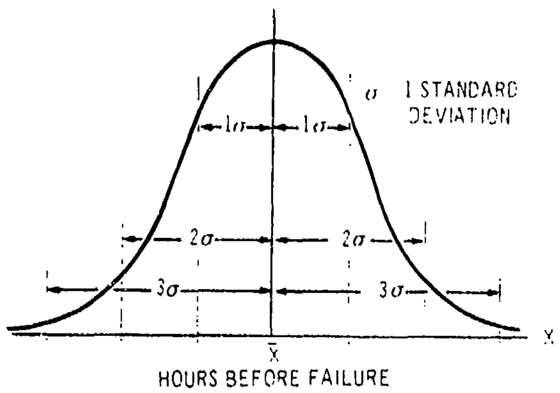

Fig. 12-2. Gaussian Reliability Distribution

Although different systems exhibit markedly different failure rates, as time elapses and a system/subsystem design becomes more refined, a large population of statistical reliability data—i.e., time-to-failure data—tends to result in a bell-shaped, normal, or Gaussian curve as shown in Fig. 12-2. If times to failure are distributed normally, then one can be 68% confident that an item chosen at random will operate within the mean value $\bar{X}$ plus or minus one standard deviation σ, 95% confident that it will operate within the mean value plus or minus two standard deviations, and 99.7% confident that it will operate within the mean value plus or minus three standard deviations.

The achievement of desired reliability is a continuing process that goes beyond the design phase. Regardless of the accuracy of prediction techniques, considerable quality control must be exercised during manufacture and assembly to insure that reliability objectives will be met. Periodic random testing should be employed to insure that the desired levels of reliability are being met (Ref. 4).

12-1.3.2 System and Component Reliability

Theoretically, every component of the helicopter could limit the reliability and availability of the overall system. All equipment is subject to failure, though the failure rates for various components vary greatly, as do the effects of individual failures. Absolute precision in a reliability model for any helicopter would involve an extremely complex set of relationships. For example, simultaneous failure of components A, B, and C, or of A, C, and D, might make a system inoperable, whereas failure of A, B, and D would not. Failure of A and F might make failure of G highly probable, while failure of component H might render the system unavailable for mission Type 1 but not for mission Type 2.

Despite these obstacles, the designer can and must identify and quantify the reliability and availability of the system in terms of the reliability and availability of its subsystems and components. The MTBF of a system is determined primarily by the weakest link or links in the system chain of components, i.e., the component most likely to fail in the shortest time in the environmental conditions to be encountered. The following considerations may be helpful in defining subsystems and components that tend to limit helicopter reliability and availability:

1. Criticality of function. Components operating in a series fashion in performing a critical mission function affect availability and reliability in a direct, readily identifiable manner. The main rotor and its associated nonredundant bearings and linkages are an example.

2. Degree of innovation. Both components and systems show a strong tendency to have high failure rates during the early part of their operating histories. The more radical the departure from conventional design, the greater the likelihood for factors that introduce deficiencies.

3. Environmental conditions. Environmental conditions such as temperature, moisture, shock, vibration, pressure, and contamination have a significant impact upon the mean life of a component or subsystem. Components that must be subjected to adverse environmental conditions will limit helicopter reliability and availability unless great care is exercised and added expense is incurred in strengthening, protecting, or assembling these items. The reliability of most components and materials is highly dependent upon factors such as temperature and rotational speed.

4. Marginal design. In the trade-off process it may occur that the margin of safety for some items is reduced significantly compared with the remainder of the system. For example, critical weight requirements may dictate the selection of lightweight materials for functions for which heavier materials would have given increased reliability.

For helicopter systems that have many types of missions and complicated interrelationships among components and subsystems, the task of system reliability prediction requires much more than the prediction of component reliability. Reliability models are helpful in calculating system reliability from subsystem or component reliability. In general, the process involves the formulation of a block diagram of the system, showing every known relationship necessary for successful operation, and then developing mathematical expressions that relate system reliability to the blocks shown in the system reliability model. Some of the elements will be found to operate in series, insofar as mission success is concerned, and some in parallel (active and standby redundancy). Some subsystems will include both parallel and series functions, and some systems may involve redundant relationships in which the system can operate successfully if particular combinations of elements out of some number n of elements do not fail. All of these combinations, once identified, can be treated by straightforward mathematic ' methods (Ref. 5).

12-2 MISSION REQUIREMENTS AND COMPLETION

Fundamental to achieving adequate reliability and availability of a helicopter is a valid set of mission requirement statements. These requirements become the basis for establishing appropriate reliability design criteria for the subsystems and component parts. The mission profiles and expected operating environments used for reliability analysis, therefore, should represent the actual operating conditions expected if the analyses are to be meaningful.

12-2.1 RELIABILITY REQUIREMENTS

The fundamental questions that must be answered in order to establish reliability requirements are a function of the basic, general definition (par. 12-1.1). These questions are:

1. What is the specified time period of operation?
2. What are the prescribed operating conditions, i.e., the operating environment?
3. What constitutes satisfactory operation, i.e., success or failure criteria? (Ref. 2)

Answers to these questions may appear to be straightforward when the helicopter is considered as a unit; e.g., it may be required that a helicopter have a specified availability rate and a specified maintenance-to-flight hour rate for a series of widely diverse geographic operating environments. In addition, generalized mission profiles, representative of actual operating situations, may be detailed and used as basic guides.

With respect to the subsystems and component parts, however, the problem of stating the necessary criteria for quantifying availability and reliability objectives becomes much more difficult. For example, although the helicopter is operating under arctic conditions, the subsystem in question could be subjected to a localized, extremely high heat and humidity environment. Likewise, a component part could experience an aperiodic vibration or shock that was undetectible otherwise in a relatively smooth-running, well-damped subsystem.

Fortunately, years of operational experience, testing, design, product improvement programs, etc., have enabled designers and users to establish standardized reliability and maintainability specifications for many generally used subsystems and component parts. MIL-HDBK-217 describes sources such as the Bureau of Weapons Failure Rate Data Program (FARADA), the Interservice Data Exchange Program (IDEP), Defense Documentation Center *Technical Abstract Bulletin* (TAB), RADC Reliability Central, and NASA Parts Reliability Information.

The failure rates shown in MIL-HDBK-217 are not complete. In order to use that handbook properly, the directions for techniques such as stressing and derating must be followed in order to approximate the "real world" more closely in the prediction process. When new or special-purpose devices are incorporated, extrapolations of existing requirements may be used as first order approximations until testing programs either confirm or reject the selection.

12-2.2 MISSION COMPLETION

Certainly, no mission can be attempted if there is zero probability ($P = 0$) for availability or reliability. However, there is no assurance that high availability and reliability probabilities will result in a high probability that the helicopter mission will be completed.

Mission completion is dependent upon the same general variables that are associated with mission effectiveness measurements, i.e., availability, survivability, and performance. Chapter 2 shows that availability, survivability, and performance can be estimated or predicted by several methods, and that the probability of mission completion can be described as the product of the individual probabilities assigned to these general variables. Because the value of each of these probabilities is less than one in all real-world situations, the combined product is much less than any of the variables. The emphasis, therefore, should be upon maximizing properly the major contributing elements, and upon evaluating reliability in the context of its impact upon the total utility and overall cost of the helicopter, including the costs of ownership.

12-3 SYSTEM AND COMPONENT RELIABILITY

Reliability theory is based upon four assumptions:

1. Reliability goals and requirements can and must be established prior to helicopter development.

2. System reliability can be forecast while the system is in the paper stage of development.

3. Disparities between the goals of Assumption 1 and the estimates of Assumption 2 can be eliminated systematically via design or at least measured so that design/operational/maintenance compensation can be made.

4. The implied measurement of the disparities of Assumption 3 must be made on a continuous basis throughout system development, and ultimately must culminate in a contractual compliance assessment by the customer so that the contractor knows whether or not he has complied with the goal and/or requirement, or is contractually in default.

Two basic types of effort are implied in these four assumptions. First, technical activity—consisting of design or analysis—is necessary to take the action required in the latter three assumptions. Second, statistical activity is implied in all four assumptions. The reliability function has been considered, mistakenly, by many to consist solely of statistical activity. This misconception undoubtedly arises not only because of the numerical aspects of the four assumptions, but also because reliability most often is defined as a probability that satisfactory operation will occur over a specified time span in a specified environment.

However, statistics is, at best, only a tool for describing the interrelationships of parts and components in a system. A strong effort in design and analytical technology is basic to production of a reliable helicopter.

12-3.1 STATISTICAL SIGNIFICANCE

If a helicopter system had a reliability requirement that failure-free operation in a specific environment would occur 12 hr a day for six consecutive months (permitted a limited amount of preventive maintenance) with probability of at least 0.985 with confidence limits of 0.90, what would this mean in practical terms? It would mean that, if an infinitely large sample of these helicopters were to be built and flown, one could be 90% certain that no more than 15 out of every 1000 would not perform the mission described. At least 985 helicopters would complete this mission, given survivability equal to 1.0.

The statistics of reliability can be kept in proper balance if the probability of success—e.g., 0.985 in the previous example—is treated simply as a measure of the state of knowledge concerning the helicopter. This probability is not a characteristic of each helicopter, in that weight, shape, and color are discernible or measurable for each vehicle, but rather it is a probability figure that can be considered valid for an entire series of helicopters.

Then what does the 90% confidence have to do with the reliability? It is actually a range indicator of the basic reliability number. In other words, it tells us how much faith to put in the basic 0.985 probability of success.

In summary, reliability statistics are reserved for those phenomena associated with adequate amounts of data. These data must be homogeneous, i.e., must be observed under identical conditions on identical systems. If these conditions cannot be met, the usefulness of statistics degenerates rapidly.

12-3.2 ESTIMATING WITHOUT DATA

Frequently, reliability data are not available for a specific design or concept in the paper stage. This is true particularly for innovative design because no previous equipment has been exactly like the new design for which a reliability estimate is desired. Alternative means for estimating reliability must be applied.

Several alternatives can be considered. One alternative is to locate data available from equipment performing similar functions on similar types of helicopters. These data can be extrapolated and utilized with caution, perhaps by a compensatory derating index (sometimes denoted as "k" factors) to allow for the unknown relationship between an operational and a proposed model.

A second possibility is to simulate operation of a helicopter by preparing a mathematical model in which failure data for smaller elements of the system—e.g., parts, components, and subsystems—are interrelated dynamically by computer. In this approach, various types of data can be used or interchanged to measure the overall effect of any given element upon system reliability. Monte Carlo simulation commonly is used for this technique. Monte Carlo simulation uses computer programs with random inputs that are run sufficient times for each problem to obtain the statistical distribution of the parameters of the solution.

Once judgment is converted into some quantitative index (such as a probability of success or failure), there is a temptation to place too much confidence in the number. There is no *a priori* way to distinguish between invalid and valid numbers. The manager or engineer should investigate the basis or source of the numbers used in reliability estimation prior to taking any action based upon those numbers. This point becomes even more critical when reliability estimation has been required without the availability of data.

12-3.3 RELIABILITY MODELING

The ultimate objective of deriving subsystem and component reliability indices is to combine them by some means into an overall system index of reliability. The means most commonly used are mathematical models.

Basically, a mathematical model represents the system characteristics that are unique to the solution of a problem. In this discussion, the problem is the determination of system reliability. Mathematical models, if properly constructed, can be used to evaluate the expected reliability of complex systems such as helicopters. However, trade-offs must be made between model complexity and practicality.

The primary limitations of reliability models are the difficulties associated with mathematically representing the actual environmental conditions that constitute the operating medium of the system, subsystem, components, and elements. Another problem is the validity of the input data, i.e., the reliabilities of the subsystems, components, etc.

MIL-HDBK-217 and AMCP 706-191 describe more completely the use of modeling techniques that can be applied with judgment to helicopter reliability analysis.

The probability density function for occurrence of failures in a system as a function of time (sometimes called the mortality function) is highly dependent upon factors such as the type of system tested, the arrangement of the experiment or test, the environmental conditions of testing, the definition of failure, and even the interpretation of the experimental data. In fact, six basic mortality functions are known to apply to various complex equipment systems (Ref. 6).

The statistical techniques for modeling reliability usually involve a failure probability distribution that is exponential and follows the equation

$$R = e^{-T/M_o} \tag{12-5}$$

where

R = probability of success (reliability)

e = base of Naperian logarithms

T = length of time for which failure-free operation is desired

M_o = mean time between failures (MTBF)

The five assumptions discussed in pars. 12-3.3.1.1 through 12-3.3.1.5, which justify use of the exponential distribution, are not confirmed entirely by reliability data. The modes of possible failure for any item will affect the analytical form of the failure distribution. Early failures resulting from improper design or manufacture and late failures caused by wearout are not predicted by the exponential model.

However, the selection of a failure distribution on the basis of physical considerations is difficult (Ref. 5). The differences among such distributions as the gamma, Weibull, and log normal become most significant only in the tails of the distribution, and knowledge of the shape of the tail comes only through analysis of extremely large samples of homogeneous populations. Early or "birth" failures can be reduced or eliminated by thorough testing and checkout before the equipment is placed in operation. Wearout failures can be excluded by good preventive maintenance provisions and practices (Ref. 7). The remaining failures almost certainly would be chance failures, which would be predictable with reasonable accuracy and convenience by means of the exponential distribution.

The exponential distribution is popular because it lends itself to easy calculation of reliability. By assuming a constant failure rate inherent in the exponential distribution, wearout of equipment is ignored, based upon the assumption of rebuilding, retrofitting, or retiring the equipment prior to entering the period of increasing failure rate. Eq. 12-5 also provides a mathematical means for converting MTBF data into probabilities. This is a convenient conversion because much of the operational data on equipment is collected in terms of MTBF.

When the exponential distribution is used, there are five assumptions that must be true if this distribution is to represent reality (Ref. 8).

1. Constant hazard
2. Random failure
3. Homogeneous equipment
4. Perfect restoration
5. Independence.

Because of the impact of these five assumptions upon the likelihood of MTBF being an accurate and realistic measurement of reliability, each assumption is discussed separately.

12-3.3.1 MTBF Assumptions

12-3.3.1.1 MTBF Assumption No. 1—Constant Hazard

The use of MTBF as a reliability measurement demands the assumption that there exists a constant hazard rate for the equipment under test and evaluation for the entire period during which data are recorded; i.e., there is no wearout occurring in the equipment. Because absence of wearout is assumed, 1000 systems that individually operate without failure for one hour provide the same proof of reliability as one unit that operates successfully for 1000 hr.

As an absurd but valid application of this assumption, it would be possible to prove that a simple 24-hr alarm clock had demonstrated millions of hours MTBF if thousands of clocks were to be wound and measured for only the first 8 hr after winding. If none of the clocks failed in the first 8 hr, eight failure-free hours would be tabulated for each one. Because MTBF is computed by simply adding up the hours of successful operation and dividing by the number of failures that occurred during the same period, the MTBF based on the 8-hr test would be extremely high. Yet the clock was designed to operate only 24 hr without failure (i.e., without being rewound).

12-3.3.1.2 MTBF Assumption No. 2—Random Failures

The use of MTBF as a reliability measurement demands the assumption that all failures are random, i.e., that their frequency of occurrence and the severity of their effect cannot be predicted. In practice, this assumption generally is interpretated as meaning that random failures are "one-time-never-to-occur-again" failures. Further, they often are treated as a class of failures differentiated from a class of wearout failures; therefore, chance is considered to be a cause of failure, as is wearout (Ref. 9).

In real world operation, many of the failures occurring in complex equipment are due to wearout. There is, of course, a large percentage of design or human error, which contributes to wearout failures. Each failure, therefore, should be treated as having a basic, definable cause.

12-3.3.1.3 MTBF Assumption No. 3—Homogeneous Equipment

The use of MTBF as a reliability measurement demands the assumption that the environment in which the helicopter operates during the entire period in which data are recorded is identical in every respect—including concurrent stresses such as vibration, shock, temperature, humidity, altitude, and constant acceleration loading—with the environment to which the helicopter will be subjected in actual field service (Ref. 9).

Environmental simulation of operational stresses in a laboratory (where most MTBF measurements are made) is virtually impossible due to the inability of test equipment to impose a complete envelope of stresses simultaneously upon the helicopter.

12-3.3.1.4 MTBF Assumption No. 4—Perfect Restoration

The use of MTBF as a reliability measurement demands the assumption that, upon repair of any failure, 100% restoration of life for the helicopter occurs; e.g., a helicopter that operates successfully for 630 flight hours before the first failure will be, after the repair of that failure and regardless of its magnitude, in exactly the same condition as it was prior to the start of the 630 flight hours.

It is accepted that all equipment begins to wear out with its initial operation. While some components of a helicopter might have very low rates of wearout, other components would show rates that were very high.

12-3.3.1.5 MTBF Assumption No. 5—Independence

The use of MTBF as a reliability measurement demands the assumption that all failures are independent in both cause and effect; i.e., each failure can be isolated totally and repaired, and no other portion of the helicopter will suffer any degradation as a result of this failure.

Seldom can a failure occur that does not affect, at least superficially, other parts of the system. The effect may be only in loading or unloading some other portion of the equipment, but there is likely to be some change.

12-3.3.2 Exponential Distribution

The exponential distribution, as pointed out previously, is reliable only if the conditions outlined in the five basic assumptions are met. Obviously, these conditions rarely are obtainable in real world operations. For example, care must be exercised in defining the system so that the subsystems or components under examination are operating simultaneously and in the same environment. However, during the accomplishment of a mission, all the subsystems of a helicopter need not operate at the same time and a completely homogeneous environment is impossible to achieve.

Regardless of its limitations, however, the exponential distribution is valuable in quickly establishing initial design objectives if cognizance is taken of the practical environment. It should be noted that reliabilities established by this technique tend to be somewhat pessimistic.

12-4 RELIABILITY PREDICTION, APPORTIONMENT, AND ASSESSMENT

12-4.1 GENERAL

Prediction, apportionment, and assessment are three quantitative techniques or quantitative indices (QIs) that aid in meeting predetermined objectives for reliability and serviceability of helicopters. These techniques are described in the order in which they are employed in the life cycle of a helicopter.

12-4.1.1 Prediction

Prediction is a methodical activity that attempts to estimate future helicopter performance based upon histories of other helicopters and analytical insight into proposed designs. This activity would be described better as estimation, but the term prediction customarily is used. In its technical sense, prediction provides a basis for immediate action to preclude undesirable events that appear probable in the future.

12-4.1.2 Apportionment

Apportionment has two aspects. First, it is the division of the overall helicopter reliability requirement/goal into appropriate requirements/goals for all of the components and subsystems in the helicopter system. Second, these requirements/goals then are assigned to their appropriate elements as targets of achievement that must be met if the overall helicopter reliability objectives are to be reached.

12-4.1.3 Assessment

Assessment is the measurement of the current state of achievement of helicopter reliability objectives. This measurement is based upon data derived from testing actual, rather than generic, elements of the helicopter.

While prediction, apportionment, and assessment appear to be interdependent, they must be kept independent. This is to preclude a circle of logic in which one technique justifies the conclusions reached by the other, and vice versa, and none of the results is trustworthy.

12-4.2 PROGRAM LIFE CYCLE

12-4.2.1 Development Phase

During development, prediction provides a baseline for comparison with apportioned goals. Disparity between the two values gives engineering an opportunity to evaluate constantly the resource allocation for design. This disparity also is useful in planning field maintenance strategy, e.g., the number and types of spare parts, field engineering training requirements, and preventive maintenance policy. Predicted values are also a standard with which assessed values are compared to measure progress.

Apportionment is revised constantly during the development phase as the design continues in a high state of change. Revised goals are communicated to the designers to avoid both underdesign and overdesign. Revisions in apportionment are possible if the overall helicopter reliability objectives are adjusted to reflect changes in mission scenario.

By the development phase, breadboards and other small elements of the helicopter will have been constructed and tested. Data from these tests are the initial data to be used in the assessment process.

12-4.2.2 Qualification Phase

Prediction is the least important of the three activities during the qualification phase (see Chapter 10, AMCP 706-203). Its primary role during this phase is to indicate what maximum performance might be expected during the helicopter test program.

Conversely, apportionment aids the testing activity by furnishing minimum design limits for the tests as well as contributing to the overall design of experiments. In addition, because it is not economical to test all elements of the helicopter system, the apportionment of the helicopter design requirements helps to determine which helicopter elements should be tested.

The test program provides a major portion of the data required for assessment of helicopter reliability. Because the tests are conducted under controlled conditions, the statistical reliability of the data recorded during this phase is higher than that of data from tests conducted either during development or in field operation. However, unless actual operating conditions are represented properly, the results may be overoptimistic.

12-4.2.3 Manufacturing Phase

Although prediction has diminished in importance long before the helicopter reaches the manufacturing phase, prediction can be helpful in establishing reasonable yield levels for manufacturing. Further, it may give some indication as to the types of problems to be anticipated during manufacturing.

Apportionment contributes information vital to the establishment of minimum acceptance levels as well as the types of manufacturing verification tests required for high confidence.

Additional information will be included for assessment during the manufacturing phase. However, assessment can be used effectively in combination with other manufacturing data to determine whether adequate manufacturing integrity of the helicopter has been achieved.

12-4.3 SCOPE OF APPLICATION

The level of effort for prediction, apportionment, and assessment is not a constant, with respect either to time or to depth of application.

The purpose of prediction is to influence designers to avoid pitfalls of the past, as well as to optimize, economically, the allocation of resources for the helicopter. Thus, the predictive effort should start almost concurrently with the development of a helicopter concept. It is a mistake to wait until hardware has been created because many design decisions are irrevocable by that time.

Apportionment can begin prior to the completion of a system prediction. The timing for the initiation of apportionment is nearly as critical as that for prediction, because designers must have some indication of their individual contributions to overall helicopter reliability objectives as early as possible.

Assessment covers a considerably longer time period than either prediction or apportionment. Because assessment is dependent upon having actual helicopter elements available, the effort is not initiated nearly as early as are the other two activities.

Although all three activities involve component parts in the helicopter, the extent to which this is true depends upon the economics of the individual helicopter program. A significant consideration is the direction that each of these activities takes.

Prediction, which is built up from generic data, runs from component parts to system, while apportionment starts with a system value and breaks it into component parts. Assessment goes in both directions but rarely below the component level, and can be used as one means of evaluating both the prediction and the apportionment.

The direction of activity involves the depth of application. There are no easy rules for establishing the level of hardware that will be included in a prediction. In some designs, even eyelets and rivets are considered in a reliability prediction.

If the overall helicopter reliability requirement is severe, there is a practicable limit upon subdividing this already rigorous goal. Therefore, an element of common sense is involved in establishing the depth to which either prediction or apportionment is applied.

The depth of assessment seldom is established, *a priori*, or controlled, by those responsible for it. By its nature, it is dependent upon all types of test data from tests conducted by various organizations for a variety of reasons under all kinds of test conditions. The important point regarding depth of assessment is that no data should be rejected if they are technically sound.

12-4.4 EXECUTION OF ACTIVITY

The preparation of the three quantitative indices (QIs)—i.e., prediction, apportionment, and assessment—has several requirements, some of which are general and others more specific.

12-4.4.1 Antecedent Requirements

Several prior activities can affect the preparation of predictions, apportionments, or assessments.

12-4.4.1.1 Helicopter System Definition

Prior to the preparation of a QI, the helicopter system must be defined in parameters that are consonant with the reliability requirements for that system. System definition generally is considered to consist of five steps:

1. Definition of purpose or intended use of the helicopter
2. Specification of system and subsystem operational parameters and limits
3. Determination of physical and functional boundaries of the helicopter system
4. Definition of what constitutes helicopter system failure
5. Establishment of reliability and availability criteria for each intended use of the helicopter.

Detailed discussion of these five steps is available in MIL-HDBK-217.

12-4.4.1.2 Profiles

Although environmental and operational parameters sometimes are included as part of system definition, they are sufficiently important to be discussed separately. Profiles are two-dimensional, graphic representations of events occurring in chronological time.

Operational profiles particularly are helpful in illustrating the amount of time, for example, that maximum stresses are imposed upon a helicopter. Therefore, they serve as an aid in optimizing duty cycles as well as assisting in the preparation of maintenance strategies. The area under the profile gives information useful in estimating wearout thresholds.

In environmental profiles, simultaneous environmental factors are superimposed to illustrate the type of testing required to simulate operation. An environmental profile is an ideal way to show simultaneity of stresses. These profiles additionally aid in stimulating imagination during analysis.

12-4.4.1.3 Data Base

All three QIs require an extensive source of failure and success data. If the contractor does not possess a data center that can provide these types of data, other sources may be utilized. Using other source data is risky because the conditions under which the data are generated often are unknown or unavailable.

To provide adequate data for QIs, the concept of an integrated data system has become well-known (Ref. 10). Providing an adequate data base involves five separate types of activity:

1. Data gathering and reporting
2. Data control
3. Data processing
4. Data recovery
5. Data utilization.

12-4.4.2 Prediction

Several different techniques are used for reliability predictions. These techniques can be ordered into a spectrum that ranges from close correlation to a previous helicopter design to essentially no correlation with any previous helicopter. All are defined in MIL-HDBK-217.

12-4.4.2.1 Techniques

For a helicopter that is evolving in an orderly fashion and with minor changes from a previous helicopter, the Similar Equipment Technique may be the most meaningful. To use this technique, the new helicopter must be similar to an older design, and the small differences between the existing and proposed helicopters must be capable of being isolated and evaluated easily.

A second method is the Similar Function Technique. This method considers the differences in achievable reliability levels that can be attributed to the functions to be performed by the helicopter. This technique does not have a widespread application due to its limitations in comparing functions that are dissimilar.

The Minimum/Maximum Method also is available. It is the least definitive of the three techniques and is used primarily in the earliest stages of prediction. Instead of using point estimates for reliability values, it stipulates ranges for these values. Thus, engineering is furnished with the most pessimistic and the most optimistic values for reliability.

These three techniques have been discussed in an order running from most meaningful to least meaningful. But as the advancement in the state of the art increases in scope from an existing helicopter to a proposed one, the less meaningful techniques become more appropriate.

12-4.4.2.2 Activity Sequence

On the assumption that the helicopter system has been defined and that operational and environmental profiles have been prepared, the first step in prediction is the construction of a reliability model. This model defines what is required for helicopter system success and translates these success criteria into system success diagrams. These diagrams actually are graphic representations of system functions to which QIs of success have been assigned.

The second step is to develop an equation for each system success diagram. Several methods are used to derive these equations. Four of these methods, as described in MIL-HDBK-217, are:

1. Redundancy equations
2. Boolean truth tables
3. Probability maps
4. Logic diagrams.

All four of these methods can be used with either single-function or multifunction systems. However, for multifunction systems there obviously will be one equation for each function.

The third step in prediction is to calculate probabilities or failure rates. This is done by using one of the three techniques discussed in par. 12-4.4.2.1. Use of failure rate calculation as the basis for probability prediction is easier than calculation of failure probabilities directly.

The fourth and last step in the prediction process is the computation of the reliability prediction. This computation is done by taking the calculated probabilities of the third step and substituting them into the reliability equations developed during the second step.

12-4.4.3 Apportionment

Apportionment has an advantage over prediction early in design because the reliability requirement (the starting point for apportionment) is an established value whereas prediction ultimately must culminate in a number that is unknown when the activity begins.

The weakness of an early apportionment is that the system model may not be sufficiently complete to permit consideration of the true complexity of the helicopter. Thus, apportionment, like prediction, is an iterative process.

12-4.4.3.1 Techniques

Apportionment of reliability requirements can be accomplished by several different techniques, namely:

1. Time/Data method
2. Time Only
3. Minimum Information

The most meaningful technique is the Time/Data method (Ref. 11), which requires accurate data on not only the operating time of the helicopter, but also of all the subsystems. In addition, available failure rate data are assumed to apply to the proposed helicopter. By combining failure rates of subsystems with the assumed operating time and operating modes, reliability requirements are subdivided and allocated among all subsystems on the basis of areas such as relative failure rate, criticality, and proportion of use.

A second apportionment method is the Time Only approach (Ref. 11). This technique assumes that failure rate data are not available for either the helicopter or its subsystems. The allocation of subgoals for reliability is made only on the basis of operating times for the various subsystems. If the subsystems are entirely in series, this simply means that each subsystem will have an equal reliability subgoal.

The Minimum Information technique is a third approach (Ref. 11). In this case, neither failure rate nor time information is available for subsystems. Therefore, the reliability requirement is subdivided on the basis of the number of elements in the helicopter, whether they are in series or in parallel.

In these three techniques, only two factors are considered: operating time and failure rates. A good apportionment, however, also must consider other factors such as complexity, criticality, maintainability, and repairability. These factors often are accounted for by weighting or "k" factors (Ref. 11).

It should be obvious that apportionment can be carried to a point of diminishing return. A relatively high helicopter reliability requirement can result in astronomic numerical goals at levels below the subsystem level. Therefore, the motivation that apportionment is intended to provide can diminish when the goals become totally unrealizable or inconceivable.

12-4.4.3.2 Activity Sequence

If an apportionment preceded the preparation of the prediction, the reliability model discussed in par. 12-4.4.2.2 would have to be constructed as the first task of apportionment. However, this would be an exceptional case. In either event, the same reliability model is the framework for both prediction and apportionment.

The second step is to allocate an appropriate portion of the overall helicopter reliability requirement to all subsystems (and subassemblies, if appropriate) using one of the techniques discussed earlier. By means of Boolean logic, for example, the various subsystem goals can be computed.

The third step is to compare the allocated values (only after the entire allocation has been completed) with the equivalent predicted values. By noting the system elements with the greatest discrepancies between predicted and apportioned values, the allocation of resources in the helicopter design becomes obvious. For example, if the improvement of an existing system. apportioned value is relatively equal to the predicted value for a given subsystem, few or no development resources should be required. On the other hand, if there is wide disparity between the apportioned value and the predicted one, further design effort appears to be mandatory.

12-4.4.4 Assessment

The function of assessment gradually supersedes prediction during helicopter development. Both prediction and assessment are measurements, while apportionment is a standard for measurement. Therefore, both prediction and assessment are compared with apportionment.

Assessment can be improved by increasing test resources to yield more data, while prediction is based entirely upon historical data. Whereas prediction and apportionment primarily influence helicopter design, assessment is most useful in the improvement of an existing system.

12-4.4.4.1 Techniques

The techniques used for assessment have not been categorized as neatly as have the techniques for prediction or apportionment. However, three loose categories of techniques for assessing reliability are described:

1. Integrated Assessment
2. Fractional Assessment

3. Fragmentary Assessment.

The preferred approach could be called Integrated Assessment (Ref. 10). This technique combines two major sources of information, i.e., tests run on the product and analyses of failures in the helicopter. The test data either can be from tests specifically conducted to measure reliability parameters or from other tests in which reliability requirements were included. The failure analysis data used in this method come from analyses in which attempts were made not only to ascertain the cause of failure but also to measure reliability requirements. Obviously, these data are one product of the integrated data system discussed in par. 12-4.4.1.3.

The second technique is Fractional Assessment. Portions of an Integrated Assessment are included in a Fractional Assessment. The difference between the two is primarily in the amount of design or planning beforehand to insure that assessment information will be available. In the absence of such preplanning, the Fractional Assessment method is the next best means of obtaining assessment data.

All remaining assessment efforts can be grouped into a classification called Fragmentary Assessment. This is the only method available if there has not been adequate planning, not only for the generation of data but also for the processing of it by those responsible. In practice, this technique consists of gathering scraps of data from tests run for other purposes and extrapolating from these data. The value of this technique is highly questionable.

The value of the assessment depends upon the amount of applicable data available for the assessment. Mathematical statistics use this same basis in establishing confidence intervals or limits. If the data available for assessment are either too scanty or irrelevant to the helicopter under consideration, the investment of resources for assessment may not be warranted.

12-4.4.4.2 Activity Sequence

Assessment, like the other two QIs, is based upon the reliability model of par. 12-4.4.2.2. Because this model undoubtedly will have been prepared previously for the other two QIs, the first step in assessment involves either design of specific tests that will yield reliability data or active participation in the design of tests being conducted for development, qualification, manufacturing, quality assurance, and customer acceptance. This step includes following the progress of these tests once they are designed to insure that the desired data will be available

The second step in assessment is to:

1. Analyze and study Failure Mode and Effect Analysis (FMEA), and Fault Tree Analysis to gain insight for designing the failure analysis criteria

2. Prepare failure analysis criteria that will yield reliability measurement

3. Follow through the failure analysis process to monitor the data desired for assessment.

The third step is to combine the test data and the failure analysis data into the reliability model for an overall assessment of achieved helicopter reliability. Because many assumptions will be necessary in the early stages of assessment, they should be documented for review.

12-5 RELIABILITY SUBSTANTIATION

12-5.1 GENERAL

The previous discussions have dealt with fundamental reliability engineering techniques that may be applicable to a helicopter development program. However, there are certain basic requirements regarding application of reliability engineering that must be satisfied if proper consideration of reliability is to be achieved. Generally, any development program must include a reliability substantiation program that is divided into two major segments: substantiation during design and substantiation during test. The following paragraphs provide specific and general requirements for the reliability substantiation program.

12-5.2 RELIABILITY PROGRAM PLAN

The integration of all engineering activities relative to reliability normally is addressed in the reliability program plan. MIL-STD-785 provides specific plan requirements and also lists certain minimum engineering functions considered essential to the proper development of reliability. The reliability substantiation program described herein augments the reliability program plan.

12-5.2.1 Reliability Substantiation During Design

For purposes of the reliability substantiation program, design includes those actions associated with establishment of detail design criteria and concepts, finalization of design drawings, fabrication of flight quality hardware, and release of hardware for initiation of reliability testing. The design substantiation program *shall* assure that the initially fabricated articles have the maximum probability of achieving all reliability objectives. Specific areas of concern are:

1. Apportionment/prediction
2. Assessment of design criteria relative to quantitative requirements
3. Utilization of detailed design review
4. Utilization of early design support.

12-5.2.1.1 Apportionment/Prediction

Each system reliability requirement *shall* be apportioned to the subsystem/component level through application of techniques similar to those described in MIL-STD-785.

12-5.2.1.2 Assessment of Design Criteria

All quantified apportionments of reliability requirements *shall* be substantiated through detail assessment of pertinent design criteria, and comparison with qualitative/quantitative reliability history of similar equipment where possible. The suitability of comparison data *shall* be established prior to their use. Special attention should be directed toward establishment of valid environmental "k" factors for converting experience data for use in quantitative predictions. Similarly, other "k" factors may be necessary to adjust experience data properly for changes in operational parameters such as stress, speed, pressure, voltage, and current.

12-5.2.1.3 Design Reviews

Early and continuing in-house design reviews *shall* be used by the contractor to substantiate that reliability requirements are reflected properly in all designs. A listing of all potential failure modes *shall* be available for each subsystem and dynamic component prior to design review and used as a checklist during the review. Potential failure modes *shall* be grouped as they relate to each reliability requirement, e.g., mission reliability and dynamic component MTBR (mean time between removals). All failure modes identified by the contractor's Failure Mode, Effects, and Criticality Analysis *shall* be considered during all design reviews.

12-5.2.1.4 Design Support Tests

Early design support tests at the component, subassembly or part/piece level *shall* be used wherever practicable to substantiate achievement of reliability design objectives prior to release of drawings for fabrication of flight test hardware.

12-5.2.2 Reliability Substantiation During Test

Reliability substantiation during test begins immediately upon availability of flight quality hardware and concludes upon delivery of test or production hardware to the procuring activity for evaluation. Final substantiation of reliability *shall* be achieved through a combination of extensive reliability testing and a design reaction process that requires an action for each failure mode discovered. Appropriate analytical/judgmental techniques *shall* be employed to provide continuous and instantaneous assessment such that all reliability requirements are being or have been achieved.

12-5.2.2.1 Assessment

A technique for continuous and instantaneous assessment of subsystem and dynamic component reliability status with respect to objectives *shall* be established. This technique *shall* be keyed to the concept that substantiation of reliability during development testing is achievable through statistical inference and engineering judgments. Integral to this approach *shall* be a concept of risk analysis for all proposed design changes developed as a result of test findings that cannot be evaluated prior to conclusion of engineering development. A mathematical model *shall* be utilized in support of reliability assessment. Predicted reliability growth curves such as those described in Ref. 12 should be used where practicable. Dynamic component MTBR growth may be predicted/assessed through the application of techniques such as described in Ref. 13. In either case, reliability growth should be predicted as a function of accumulated test hours. Maximum attention must be directed toward assuring that all test hours are suitable for satisfying the growth process.

12-5.2.2.2 Substantiation Testing

The objectives of reliability testing are to identify those failure modes inherent to the final design that prohibit achievement of reliability requirements, and to validate, where possible, the responsiveness of design changes necessitated by discovered failure modes. Additionally, the reliability test program *shall* be arranged to provide reasonable substantiation that those potential failure modes not experienced during testing are not, in fact, inherent to the design. Types of reliability tests (bench, ground test vehicle, subsystem, flight, etc.) for each subsystem and dynamic component *shall* be combined, structured, and scheduled such that maximum achievement of reliability substantiation is achieved during engineering development. These tests *shall* be based upon detailed subsystem and component failure mode analyses and the capability of various test techniques to detect potential failure modes. Appropriate consideration *shall* be given to integrating reliability testing with other test requirements (specifically, environmental).

The following *shall* apply in establishing the reliability test program:

1. Abusive overload and environmental extreme testing may be considered for early detection of certain failure modes. Prior analysis must provide the basis for using these types of tests and must establish firm ground rules for interpretation of results.

2. Subsystem testing may be satisfied through a combination of subsystem test rigs, utilization of the ground test vehicle, and utilization of flight test vehicles.

3. Component reliability bench testing may be utilized to augment system and subsystem tests.

4. Maximum consideration of the stated operational environment conditions must be included in finalizing all test setups.

5. Inherent constraints of any particular test type must be considered in establishing total test hours.

6. All reliability testing must be integrated with other system test requirements where practicable to minimize total development cost.

7. The number of test samples must be based upon a combination of test schedule requirements and consideration of potential failure modes due to manufacturing quality or wearout.

8. Full scale flight quality hardware *shall* be used for these tests.

12-5.2.2.3 Substantiation Testing Process

Prior to the initiation or restart of any reliability test, a listing of all failure modes that could be detected *shall* be prepared and used for comparison with actual test results. Following each test period, the contractor should conduct an analysis of test procedures and results to validate the test adequacy.

12-5.2.2.4 Reliability Model

The contractor *shall* use a mathematical reliability model to assist in providing continuous and instantaneous assessment of reliability characteristics throughout engineering development. This model *shall* include provisions for mathematically assessing the extent of reliability achievement at the subsystem and dynamic component levels and must be compatible with the objectives set forth above for reliability substantiation. Use of techniques such as instantaneous MTBF computation, and best and worse case projections, *shall* be considered in establishing detail model capabilities.

12-6 AVAILABILITY

12-6.1 GENERAL

There are many definitions of availability. Perhaps the simplest could be expressed by the equation

$$\text{Availability} = \frac{\text{Hours Available}}{\text{Total Hours}} \qquad (12\text{-}6)$$

However, availability generally has a more sophisticated meaning. Most often, it is considered to be one of the major elements of a larger measurement or index. Availability in the broader context of mission readiness is discussed in Chapter 2. The "mission capability" aspect of mission readiness relates to the correspondence between the helicopter equipment and configuration and the types of missions the helicopter may be expected to perform. The other aspect of mission readiness—availability—is defined as the probability, under stated conditions, that the operation of the system is satisfactory at any point in calendar time when it might be needed.

12-6.2 DESIGN INFLUENCE ON AVAILABILITY

The most effective and economic point at which to influence availability is during preliminary design of the helicopter. Availability is not an accidental by-product of good design; it must be designed deliberately into the helicopter. Three specific aspects of availability are discussed in detail:

1. Serviceability
2. Modularization
3. Standard versus nonstandard parts.

12-6.2.1 Serviceability

Availability is influenced greatly by the amount of time required to repair the helicopter. Trade-off studies should be made during preliminary design to evaluate field repair (possibly under combat conditions) versus Army depot level maintenance. Three primary factors must be considered during preliminary design—calendar time, maintenance manhours, and level of maintenance skill. Calendar time is important because it is the measure of how long the helicopter will be down (unavailable); maintenance manhours because this factor affects the number of logistic personnel who must be on hand to maintain the helicopter; and maintenance skill level because of the need to train the personnel to maintain the item.

12-6.2.2 Modularization

During preliminary design, basic decisions are reached regarding repairable versus nonrepairable (throwaway) units. Rather than simply evaluating the repairability of the defective module, this trade-off must be extended to include consideration of the ease of removing and replacing the module. Maintenance personnel skills also must be considered during preliminary design when decisions are made concerning modular replacements.

12-6.2.3 Standard Versus Nonstandard Parts

The problem of whether to use standard parts or to design uniquely applicable parts is a major factor in designing availability into the helicopter. The use of standard parts not only reduces the logistic problem, but also enhances the probability that proper tools will be available in field locations for maintenance personnel and that lower maintenance skill levels can be employed to repair the helicopter.

12-6.3 FACTORS IN HELICOPTER DOWNTIME

Minimizing downtime is the key to maximizing availability. Although some of the factors—e.g., logistics, maintainability, and administrative policies—involved in helicopter downtime are beyond the control of the design engineer, their impact upon availability depends to some extent upon design efficiency.

12-6.3.1 Logistics

The downtime may be due as much to lack of parts on site as it is to difficulty of repair. Likewise, lack of consumables—such as propellants, fuels, and lubricants—may cause the helicopter to be inoperative. Items such as batteries and supplies of gases and fluids are likely sources of downtime if they are depleted or contaminated. The lengtn of supply lines, in terms of both distance and time, also influences availability.

12-6.3.2 Maintainability

In addition to repairability and serviceability (discussed previously because they are established primarily during the design phase), other maintainability facets are important in keeping downtime minimal. For example, the scheduled maintenance program must be optimized so that helicopters are repaired prior to failing in service and yet are not being repaired unnecessarily. Unscheduled maintenance must be estimated and its effect upon logistics measured so that allocation of spares is adequate. Other maintainability aspects include access to tools, both standard and special, wherever they may be needed.

12-6.3.3 Policy Factors

Administrative policies can be as responsible as design factors in influencing helicopter downtime. Inspection policies, for example, can result in unnecessary downtime if scheduled maintenance is too frequent. In the same way, diagnostic testing policies can enhance availability through the reduction of repair time. One way for this to occur is to provide advanced fault isolation systems.

12-7 SYSTEM SAFETY

The concept, requirements, design integration, and planning of system safety are discussed in Chapter 3, AMCP 706-203. Because system safety must be an indigenous part of preliminary design, placement of the major discussion of system safety in AMCP 706-203, the Qualification Assurance part of this handbook, could be misleading. However, the importance of initiating system safety activity during conceptual design phases cannot be overemphasized.

Fig. 12-3 shows the relationship of the System Safety Program Plan (SSPP) to the three parts of the *Helicopter Engineering Handbook*. Note that the role of the SSPP changes during preliminary design (AMCP 706-201), detail design (AMCP 706-202), and qualification assurance (AMCP 706-203).

The SSPP will have a significant interface with reliability. The following specific areas serve as a point of departure in the planning process from which specific measures for coordination and integration of these and any other disciplines in a program can be developed:

1. Analytical techniques
2. Failure modes and criticality analyses
3. Failure probabilities and prediction
4. Failed item experience, analysis, and corrective actions
5. Critical item tests and demonstrations.

System safety activity is different in each phase represented by the three handbook volumes. The safety objectives must be established during the preliminary design. These objectives may be expressed as goals, targets, or requirements.

Much of the actual work of designing safety into the helicopter occurs during the detail design phase covered in AMCP 706-202. Typical tasks performed at this time are listed in the SSPP.

AMCP 706-201,
PART ONE
PRELIMINARY DESIGN

ESTABLISHMENT OF SAFETY OBJECTIVES

(GOALS, TARGETS, OR REQUIREMENTS)
SYSTEM SAFETY PROGRAM PLAN
IS PREPARED HERE

AMCP 706-202,
PART TWO
DETAIL DESIGN

ATTAINMENT OF SAFETY

(TASKS PERFORMED TO MEET OBJECTIVES)
SYSTEM SAFETY PROGRAM PLAN
IS IMPLEMENTED HERE

AMCP 706-203,
PART THREE
QUALIFICATION ASSURANCE

VALIDATION OF SAFETY

(MEASUREMENT, ESTIMATION, PREDICTION)
SYSTEM SAFETY PROGRAM PLAN
IS MEASUREMENT TOOL HERE

Fig. 12-3. Role of System Safety in Helicopter Engineering Handbook

By the time the design reaches the qualification assurance stage, whatever safety the helicopter ultimately will possess essentially is fixed, and the emphasis is then on measuring whether or not the achieved level of safety is commensurate with the required level.

12-8 LIST OF SYMBOLS

A_i = availability of identifiable subsystem
A_j = availability of identifiable component
A_N = availability of Nth subsystem
A_n = availability of nth component
A_s = system availability
e = base of Naperian logarithms
M_θ = mean time between failures (MTBF), hr
R = probability of success (reliability)
R_i = reliability of identifiable subsystem
R_j = reliability of identifiable component
R_N = reliability of Nth subsystem
R_n = reliability of nth component
R_s = reliability of complete system
T = length of time for which failure-free operation is desired, hr
λ = 1/MTBF, time^{-1}
σ = standard deviation, dimensions as required

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Reliability Measurement Program, Department of the Navy, Special Projects Office, Washington, D.C., May 1965, pp. 2-39 through 2-58.

11. PUB 549-T, *Statistical Techniques for Reliability*, North American Aviation, Space and Information Systems Division, Downey, Calif., November 1965, p. I-19.
12. *General Electric Reliability Planning and Management Concept*, ASQC/SRE Seminar, September 26, 1970.
13. TR 71-18A, *Helicopter Development Reliability Test Requirements*, USAAMRDL, Eustis Directorate, Ft. Eustis, Va., April 1971, AD-725 595.

CHAPTER 13

CONFIGURATION AND ARRANGEMENT

13-1 INTRODUCTION

A completed preliminary design should include the basic drawings describing the helicopter configuration and the arrangement of components and systems. Drawings that depict the configuration must consist of at least a three-view, a general arrangement, and an inboard profile. Other drawings normally are included to provide additional details in areas of primary interest peculiar to the design.

The data required for the completion of these drawings derive from the background data and techniques discussed in the foregoing chapters of this volume. This information first should be used for configuration studies consisting of such items as program objectives and mission requirements. Overall configuration restraints then should be considered along with detail considerations such as the designated power plant and type of landing gear. Finally, all data must be integrated and summarized in the preliminary design drawings and reports.

13-1.1 CONFIGURATION RESTRAINTS

The final preliminary design should represent an optimization of all design variables but must be held within certain well-defined configuration restraints. The quantity and nature of these configuration restraints vary according to the specific mission and the design requirements and objectives. Pertinent restraints normally will include, but will not be limited necessarily to:

1. Rotor clearance
2. Engine installation
3. Interior arrangement
4. Transportability
5. Personnel emergency egress
6. Fuel storage
7. Aerodynamic requirements
8. Turnover angles
9. Step height
10. Weapon clearance
11. Rear ramp.

These configuration restraints are summarized in more detail in the paragraphs that follow.

13-1.1.1 Rotor Clearance

The design and location of the rotors *shall* provide sufficient clearance for the blades above the ground support equipment or personnel on the ground, from each other, and to other parts of the helicopter. In order to provide head clearance for ground personnel, a minimum distance of 85 in. *shall* be provided between the blade tip and the ground when the blades are on the droop stop under a 1-g load condition. Any auxiliary rotor, such as a tail rotor, not meeting the 85-in. ground clearance should be protected adequately to prevent inadvertent injury to ground personnel. In general, because of blade length, flexibility, and articulation, the clearance during operation will be different from static clearance. During operation under all normal conditions, including ground/air taxi, flight maneuvers, and landings (power-on or autorotation touchdowns), the clearance between the main rotor blades and other parts of the helicopter *shall* not be less than 9 in. For landings up to and including 120% of the reserve energy sink speed, clearance shall exist between the main rotor blades and other parts of the helicopter (a symmetrical load distribution *shall* be assumed). The clearance between auxiliary rotor blades and other parts of the helicopter *shall* not be less than 6 in., and between blades of auxiliary rotors and other rotors not less than 4 in. under all operating conditions. The above clearances are based upon a conventional helicopter configuration. When novel rotor configurations—such as stopped, stowed, or telescoping rotors—are used, special conditions for such clearances will be specified in the solicitation or should be requested of the procuring activity by contractors as required.

13-1.1.2 Engine Installation

The engine installation establishes more configuration restraints than does any other helicopter system or component. The exact engine and/or number of engines to be used usually is specified by the procuring activity. In addition, the following must be considered:

1. Inlet. The engine air inlet *shall* be designed to prevent formation of ice and ingestion of snow, and must be located to minimize ingestion of foreign objects and dirt. The most frequent causes of foreign object ingestion in Army helicopters are tools and materials left in or near the inlet by maintenance personnel and foreign objects picked up by rotor airflow circulation. Other factors to be considered in the location of engine inlets are the proximity of external hoisting systems and the trajectory of materials thrown or ejected from the helicopter in flight.

2. Exhaust. The exhaust system must be designed to minimize
   a. the impingement of hot gases on the structure
   b. the recirculation of engine exhaust gases into the engine inlet or into inlets for ventilation of crew and/or passenger compartments. The exhaust gas exits *shall* be located to prevent impaired crew vision at night.

3. Firewall. All engine compartments, including hot section and compartments carrying combustible fluids subject to ignition, *shall* be separated from the helicopter structure by a firewall.

4. Personnel Protection. All personnel—including crew, passengers, and ground personnel—must be protected from flying objects due to possible failure of engine components having a high rotational speed.

5. Infrared Suppression. The location of the engine hot section and exhaust system within the helicopter structure may have a significant effect upon the ease of providing suppression of the IR signature.

6. Separation. The separation of each engine and its systems from other engines and their systems *shall* be such that, for multiengine installations, a failure of one engine due to battle damage or other causes may not in itself induce the failure of a second engine.

13-1.1.3 Interior Arrangement

The interior arrangement is determined by the mission requirements and constitutes a configuration restraint because accommodation must be provided for the applicable crew, troops, equipment, and disposable loads. The anthropometric data of par. 13-3.1.1 *shall* be used in the sizing and space allowance for the crew and other personnel. Ingress and egress of troops, seat size and orientation, and installation of mission-essential equipment such as suppressive fire weapons are all major considerations in the overall design.

13-1.1.4 Transportability

It is normal for the design requirements to include the capability for the helicopter configuration to be transported and shipped on other vehicles, including a cargo aircraft. This requirement establishes restraints on size and volume so that the helicopter can be prepared easily for shipment. Items such as stowage of rotor blades, removal of aerodynamic surfaces, and removal of tail boom if applicable must be considered in establishing a configuration that meets or exceeds the transportability requirements. The undercarriage clearance and wheelbase of helicopters must be capable of clearing knuckles formed at the intersections of 14-deg ramps and loading surfaces. The sizes of cargo vehicles normally considered for transportation of Army helicopters are discussed in par. 13-2.

13-1.1.5 Personnel Emergency Egress

Emergency exits must be placed in a location convenient for use by the operating personnel; consideration of these exits in the preliminary design is essential because load paths and major structure may be affected. The requirements summarized in par. 13-3.2.1 *shall* apply. Another design restraint consideration involving personnel is the location of the personnel where maximum attenuation is provided under crash conditions, and hence where there is the least likelihood of injury from displaced helicopter systems and components.

13-1.1.6 Fuel Storage

Few helicopters have a surplus of space for fuel storage; accordingly, the initial consideration of locations for the fuel tanks is of major importance. It is extremely desirable to have all the fuel stored at the helicopter normal CG as related to the longitudinal and lateral axes, and it is mandatory that consumption of the fuel does not cause a significant CG location change. Tradeoffs of locating the fuel near the bottom of the fuselage to enhance minimum turnover tendency versus locating the fuel tanks to minimize the probability of rupture due to crash impact must be considered carefully.

13-1.1.7 Aerodynamic Requirements

The aerodynamic configuration necessitated by mission performance requirements introduces manufacturing and cost restraints as related to body contours, surface smoothness, and requirements for fairings.

Three-dimensional contours should, in general, be avoided on the major skin areas, thus permitting the use of flat wrap construction for maximum cost-effectiveness. In those areas where three-dimensional contours cannot be avoided, minimum structural requirements should be investigated and alternate manufacturing methods such as molding should be studied in conjunction with establishing the final preliminary design configuration. Another aerodynamic restraint applicable to some helicopters would consist of the size requirements for aerodynamic surfaces necessary to provide inherent flight stability.

13-1.1.8 Turnover Angle

Army helicopters normally have a requirement for landings on slopes up to 15 deg in any direction. Compliance with this requirement often is demonstrated by landing on a slope while holding partial thrust of approximately 1/3 the weight of the helicopter on the main rotor. Because an operational requirement exists for 15 deg, a minimum turnover angle of 30 deg has been established. This constitutes a configuration restraint in regard to the distance between the landing gears and the relative vertical position of the CG.

13-1.1.9 Step Height

Recent tests have established a maximum step height of 25 in. for fast boarding and unboarding of troops. Step heights for single steps, therefore, *shall* be a maximum of 25 in. and multiple steps *shall* be spaced per MIL-STD-1472.

13-1.1.10 Weapon Clearance

The weapons must be located and restrained in bore excursion so that a projectile or any other fired missile *shall* not pass through the rotor disk area, damage the landing gear, or strike any other part of the helicopter. Provisions also *shall* be made to prevent damage to nearby structure due to propellant gases and muzzle blast. In addition, spent cartridges *shall* not be permitted to be disposed of in a trajectory that might damage portions of the helicopter, such as the tail rotor, or interfere with the operation of flight control mechanisms. Other installation requirements include location of the weapons where personnel can service them conveniently, possibly in flight, and sufficiently remote to preclude personnel injury.

13-1.1.11 Rear Ramp

The rear ramp, if applicable, *shall* have a slope of 8-12 deg. An 8 deg slope is preferred.

13-1.1.12 Other Restraints

All helicopter configurations have design restraints that are peculiar to the type and model. These must be considered on an individual basis as applicable. Some possible examples include rotor system isolation, ground resonance considerations, phasing of rotor blades, and externally mounted equipment provisions.

13-1.2 PRELIMINARY DESIGN CONSIDERATIONS

Once the general configuration has been agreed upon and the configuration restraints have been established, it becomes possible to analyze the major subsystem requirements. There is no single Figure of Merit for any of these major subsystems. Decisions on them are interrelated, and the final design should be an intelligent compromise, based upon fundamental design goals (i.e., safety, performance, cost, and reliability).

The discussion that follows describes examples of criteria commonly used in establishing subsystem characteristics. The design process is similar to that involved in the development of other military systems. Its special characteristics include a unique emphasis upon the dynamics of the total system. Dynamic considerations in helicopters involve rotor system and blade design, control systems, structures, and landing gear.

A primary design task is to maintain cognizance and control of the interrelationships among the individual subsystems in order to insure that the efforts of the various subsystem design groups do not diverge. Of the 11 major subsystems of a single-rotor configuration, the main rotor, tail rotor, flight control, electrical, communication, and instrument subsystems can be exchanged freely among airframe arrangements with few exceptions. The fuel, power plant, transmission, structure (including transparent areas), and furnishing subsystems are interdependent, and never should be selected without first making preliminary layout of all of them to determine the effect of each one upon the other.

As the design effort progresses, each feature of the proposed helicopter should be evaluated continually for conformity with design criteria. In particular, cost, weight, human factors, and dynamic considerations are vital elements about which objective data rapidly accumulate as the design details evolve. Continual compromise and iteration are required in order to insure that system goals are met.

13-2 TRANSPORTABILITY REQUIREMENTS

13-2.1 GENERAL

The policy of the Department of Defense with regard to the transportability of materiel is set forth in AR 705-8. It directs that "transportability will be a major consideration when formulating the priority of characteristics to be considered in the design of any new item of materiel and equipment". Accordingly, helicopters must be designed to permit ready handling and movement by available transportation facilities—air, road, ship, or rail.

Important factors that affect design for transportability are the limitations imposed upon size, shape, and weight by existing facilities, equipment, and practices of normal transportation modes. Special or unique arrangements of rights-of-way, clearances, or other operating conditions will be undertaken only in exceptional cases and after first obtaining approval from the appropriate transportation agency.

For these reasons this discussion of helicopter design for transportability is organized around considerations of the constraints of major transportation modes. Those aspects that are peculiar to helicopters *per se* are presented under the discussion of the transportation modes.

The factors to be considered in the selection of a transportation mode for a helicopter are distance of the move, urgency of the need at the new station, cost of preparation and shipment, and the feasibility and availability of the various transportation modes for the particular move required. The normal transportation modes that can be considered are self-delivery and transport aircraft. Highway transport by truck or trailer, ship or barge transport, or rail are not used frequently, but also should be considered.

13-2.1.1 Sectionalization

Sectionalization refers to that characteristic of a helicopter design that enables subsystems, e.g., rotor blades, tail boom, to be removed or reoriented to reduce weight or shipping dimensions. This may be necessary for large helicopter systems to assure that the vehicle may be accommodated within normal transportation envelope dimensions. For smaller systems, sectionalization should be minimized to facilitate rapid deployment and reaction.

Water movements generally require that the item fit into the hatch opening for stowage below deck in the between-deck area. If a helicopter is to be used by both the Army and Navy, operational compatibility with aircraft carriers, such as the LHA, is an important sectionalization consideration. Compatibility with the aircraft carrier deck elevator is typically the most critical constraint, particularly with heavy lift helicopters (HLHs). Any sectionalization must be accomplished in a manner that will permit reassembly in the field with a minimum of special tools, personnel, and handling equipment.

To date, sectionalization of helicopters has included blade folding and/or removal of rotor blades and tail boom and removal of external antennas and mirrors. Folding of blades and tail booms generally is more desirable than removing them. Removed blades and tail booms require boxes, fixtures, or special cushioning for stowage during shipment. Further sectionalization usually requires removal of mechanical equipment such as the rotorhead, main shaft, and control rod parts. In some cases, pylons may be removed. Transmissions usually are part of the structure of the helicopter and generally are considered inappropriate for sectionalization. Large helicopters, if divided between the landing gear, should have easily applied supports or wheels installed prior to the separation to reduce or eliminate the need for a crane as a sectionalization aid. Each section should incorporate handling provisions. In all designs, quick-disconnections, electrical plugs and receptacles, and other easy-to-separate and reconnect devices should be incorporated at separation or folding points and designed to be connected in only one way.

13-2.1.2 Preparation for Shipment

Except for self-delivery over relatively short distances, every helicopter will require some sort of preparation for shipment (see MIL-P-116). Long ferry flights may require installation of additional fuel tanks either internally or externally. Shipment by surface will require the folding or removal of rotor blades and tail booms; removal of externally mounted antennas and mirrors; and protection of plastic and glass surface from abrasive particles. In some instances pointed appendages also will need protection. Exposed machined metal surfaces will need a protective coating to prevent corrosion. Dissimilar metals will set up a galvanic action, and must be insulated from each other in the original design or as part of the preparation for shipment by ship. Fuel systems may have to be drained, purged, and preserved. These features must be considered during the design of the helicopter to reduce the time required to accomplish these tasks, ease their performance, and reduce the possibility of damage.

13-2.1.3 Tiedown and Lifting Points

The design of a helicopter limits the points at which attachments may be placed for lifting and tiedown. Certain tiedown practices may cause extensive damage to the helicopter structure. Therefore, consideration must be given to the design, location, and marking of attachments for lifting, ground handling, and restraint for surface transportation. MIL-STD-209 provides guidance for the design and location of lifting and tie-down eyes. Restraint criteria must be met for all modes of shipment applicable (e.g., aircraft, ship, and truck). However, the designer also must consider the tiedowns and any other appurtenances or possible attachment points from the viewpoint of the person charged with moving the helicopter to determine how an inexperienced person would go about restraining it. Certain parts must be marked to preclude improper use as restraining points, and other parts marked for use as restraining points.

13-2.1.4 Assistance

AR 705-8 designates the Military Traffic Management and Terminal Service (MTMTS) as the agency to implement the DOD Engineering for Transportability Program for the Army. The U.S. Army Transportation Engineering Agency of MTMTS is a field activity organized to provide assistance in transportability; its services are available without charge and should be requested before, during, and after, engineering design drawings are prepared.

13-2.2 AIR TRANSPORTABILITY

AMCP 706-130 summarizes the requirements imposed to provide air transport and/or air drop capability for materiel. The equipment and systems used are described, including the characteristics and capabilities of the Army and Air Force aircraft that may be used for air transport of Army equipment.

13-2.2.1 Aircraft Size and Weight Limitations

Physical limits for air-transportable equipment currently are governed by the existing military transportation aircraft that probably will be in use for 10-20 yr. These aircraft include the C-130, C-141, and C-5. Unrestricted cargo compartment lengths, widths, and heights for these aircraft are illustrated in Figs. 13-1, 13-2, and 13-3.

Load density and tread or footprint area of the cargo are important factors in air transportability. Flooring of each type of aircraft has different structural support sections that limit load concentrations. When an item does not have sufficient footprint area, shoring and/or load spreaders are required to distribute the bearing pressure upon the flooring surface until it is within acceptable limits. Total cargo load in pounds for different aircraft versus the mission range indicated in AR 70-39 is shown in Table 13-1.

13-2.2.2 Terminal Facilities

Cargo destined for air movement generally must be compatible with highway movement restrictions in order to be transportable to and from the air terminal. Packaged cargo normally is palletized for ease of handling. The Military Airlift Command (MAC) terminals are now capable of using the 463L Materials Handling System, which provides a rapid loading and unloading roller conveyor apparatus. The heart of this system is the 108-in. × 88-in. pallet with a 10,000-lb capacity.

13-2.2.3 Restraint

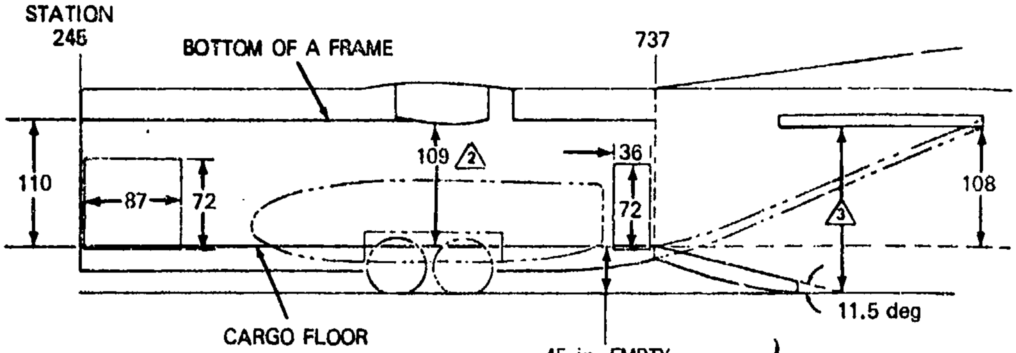
DA TB 55-100 provides appropriate guidance for restraint of air cargo. AR 70-39 and the applicable TO/TM contain load factors for which all cargo must be restrained in each model aircraft.

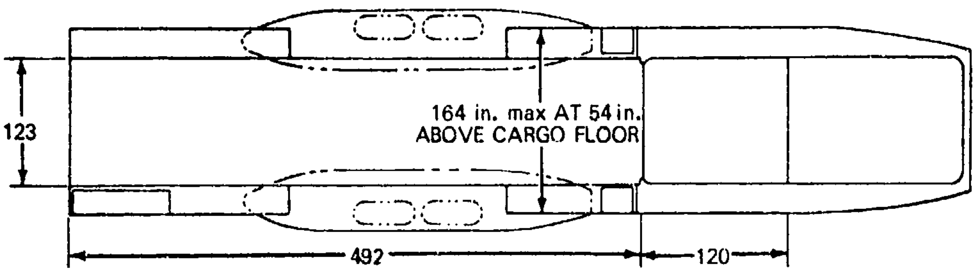
13-2.3 HIGHWAY TRANSPORT

Land movement considerations require careful analysis to allow effective and efficient movement of cargo under all physical, legal, and environmental conditions. The vehicle and cargo must be compatible in size and weight with the highway systems involved. Specific areas of interest for all highway movement are discussed in the ensuing paragraphs.

In many areas of the Continental U.S. (CONUS) and other areas of the world, the only means of mass movement of persons and things is the highway transport system. In some instances this is the only mode of transportation that can be used to move unserviceable aircraft. Most movements that employ other modes of transportation begin and end the journey on the highway (see TM 55-650 for CONUS information and routing restrictions). The largest cargo item that can be transported by highway is determined by the administrative limitations of the individual states in the U.S. and by the host country outside of the U.S. For specific information on permissible size and weight restrictions see par. 13-2.3.2.

For cargo that exceeds the legal size and/or weight limitations, the states and foreign countries review each such load and may grant movement permits. Each state and foreign country must be contacted separately, and





NOTE:

1 DIMENSIONS IN in.

2 108 in. WITH LITTER BRACKET INSTALLED.
WITH DUAL RAIL AIRDROP SYSTEM INSTALLED DEDUCT 2.625 in.

3 AIRCRAFT ON HARD SURFACE, LANDING GEAR STRUTS AT STANDARD INFLATION, NOMINAL DIAMETER TIRES AT 35 PERCENT DEFLECTION, THIS DISTANCE IS:
C–130A AND D: 146 in. AT 108,000 lb
C–130B: 147 in. AT 135,000 lb
382B AND C–130E: 143 in. AT 155,000 lb

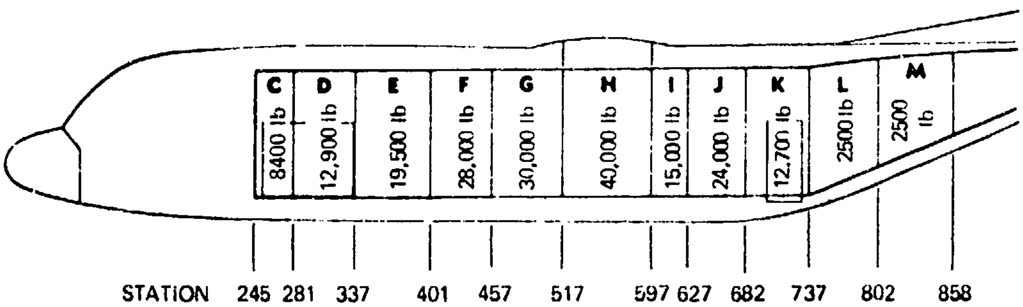


Fig. 13-1. Cargo Compartment Envelope and Weight Limits, C-130/382B Aircraft (from AMCP 706-130)

NOTES:

1. DIMENSIONS IN in.
2. THE HEIGHT AND WIDTH OF THE COMPARTMENT ARE REDUCED 2 in. ACROSS THE UPPER OUTSIDE CORNERS OF THE COMPARTMENT AT FUS STA 1198 AND 1238 BY THE TROOP DOOR TRACKS.
3. WHEN THE ROLLER CONVEYORS ARE TURNED UPRIGHT, USABLE HEIGHT WILL BE REDUCED 1.5 in.

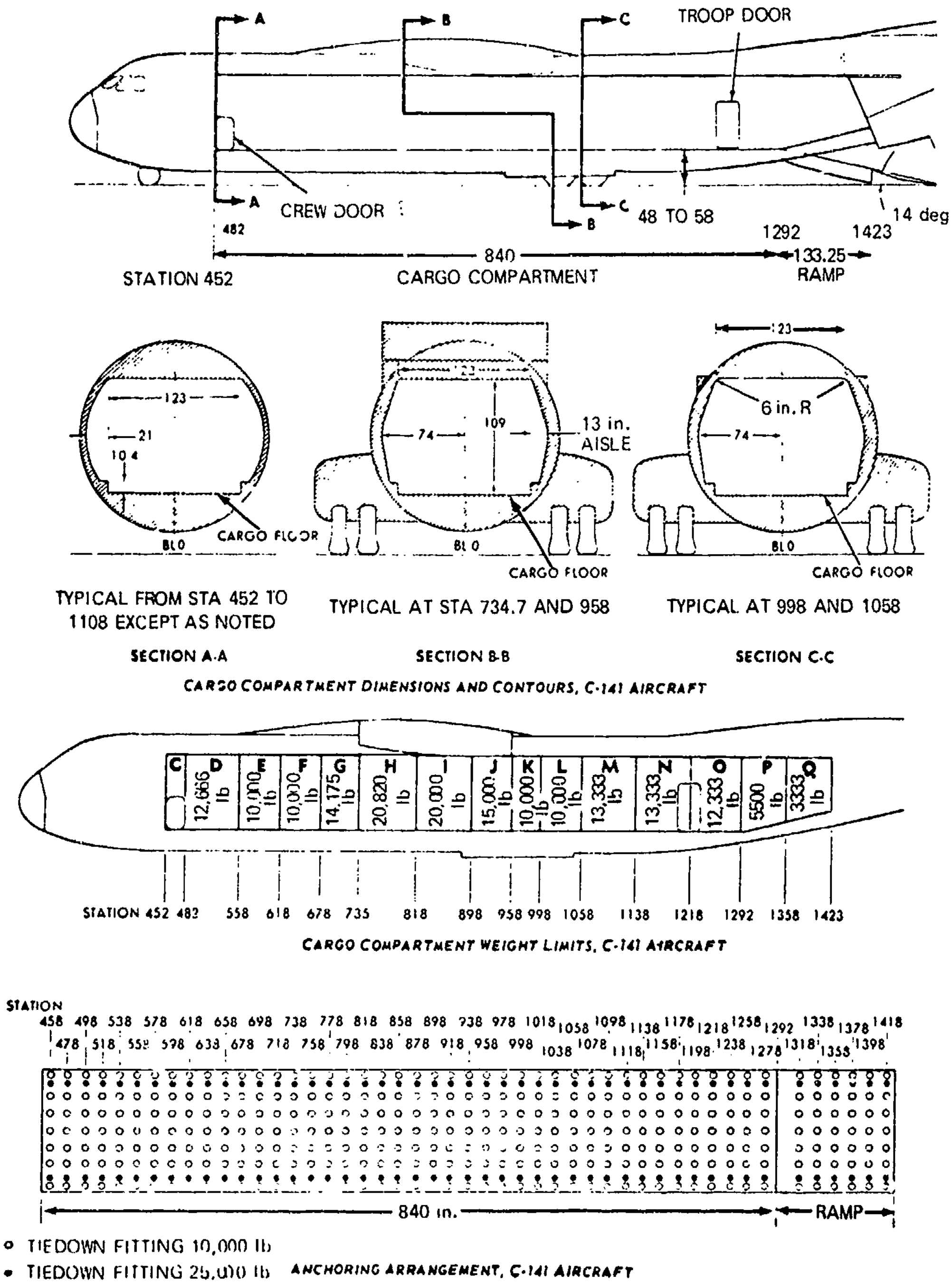
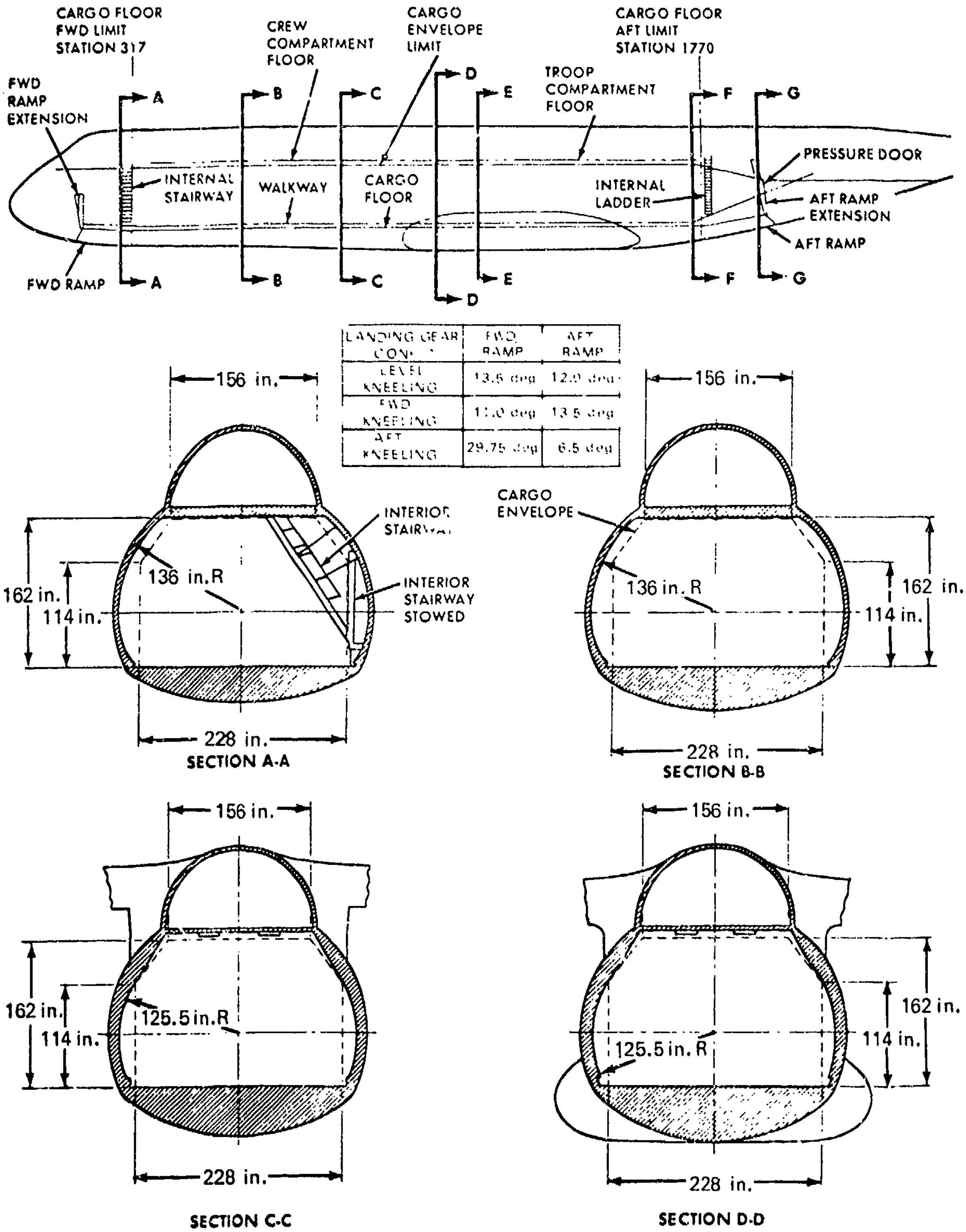


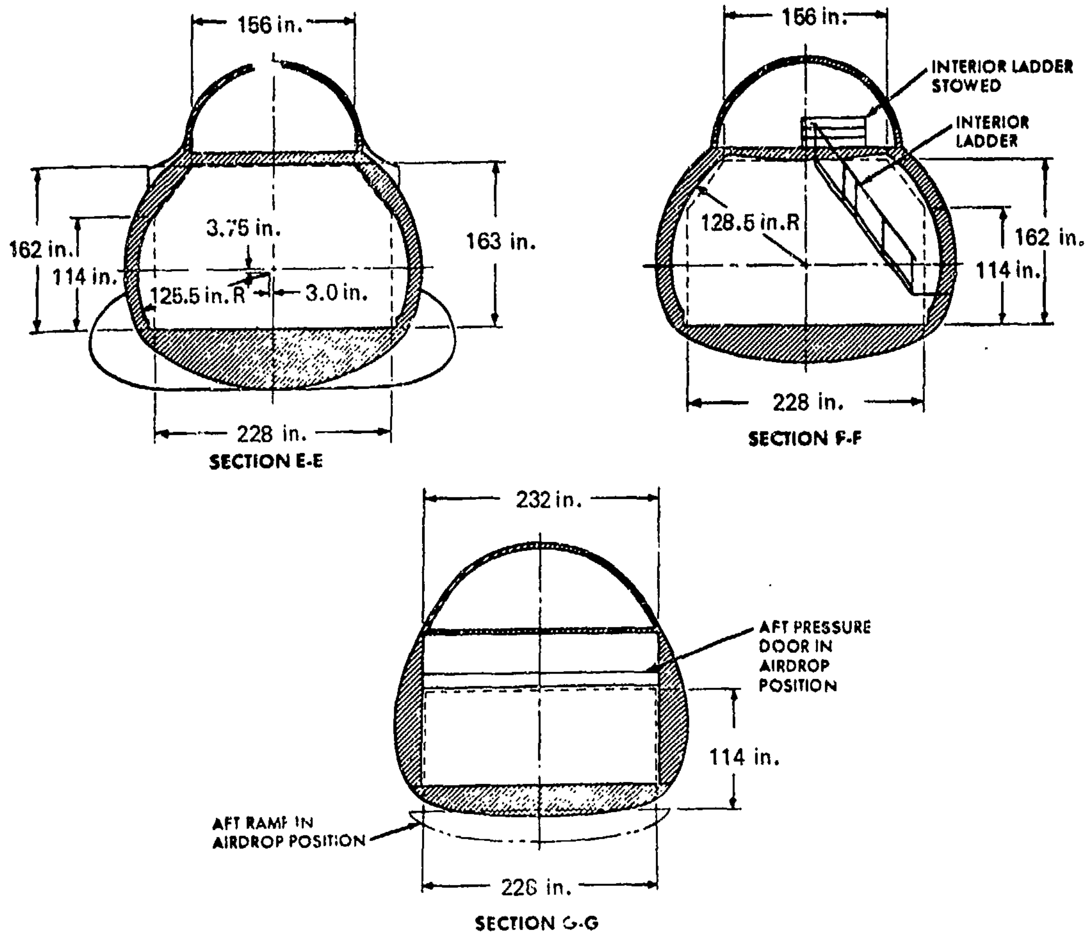
Fig. 13-2. Cargo Compartment Envelope and Weight Limits, C-141 Aircraft (from AMCP 706-130)



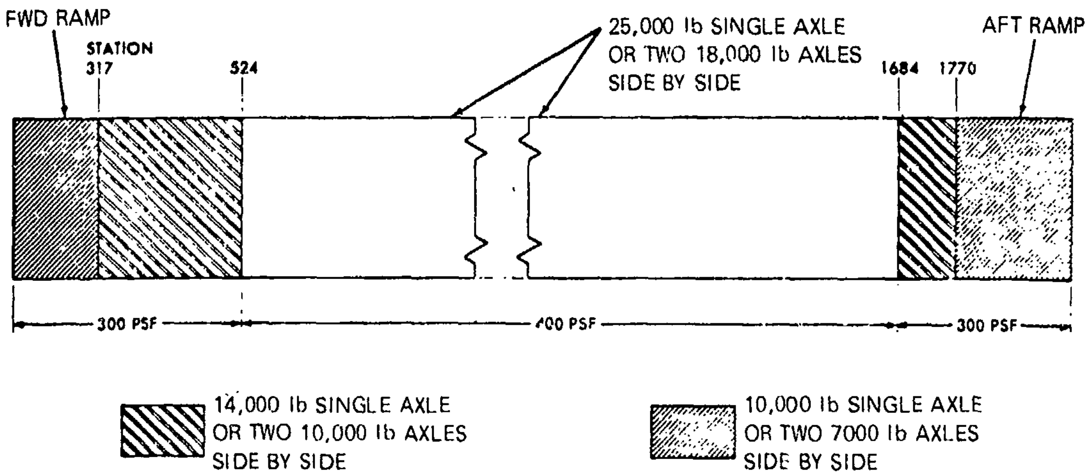
LANDING GEAR CON	FWD RAMP	AFT RAMP
LEVEL KNEELING	13.5 deg	12.0 deg
FWD KNEELING	11.0 deg	13.5 deg
AFT KNEELING	29.75 deg	6.5 deg

CARGO COMPARTMENT, C-5A AIRCRAFT (1 OF 2)

Fig. 13-3. Cargo Compartment Envelope and Weight Limits, C-5 Aircraft (from AMCP 706-130)



CARGO COMPARTMENT, C-5A AIRCRAFT (2 OF 2)



NOTE:
A 25,000 lb SINGLE AXLE MAY BE DRIVEN OVER ENTIRE FLOOR AND RAMP AREA DURING LOADING OPERATIONS.

CARGO COMPARTMENT WEIGHT LIMITS, C-5A AIRCRAFT

Fig. 13-3. Cargo Compartment Envelope and Weight Limits, C-5 Aircraft (from AMCP 706-130) (Cont.)

TABLE 13-1
CARGO LOAD FOR U. S. AIR FORCE TRANSPORTS

	RANGE, n mi	
	1000	2500
AIRCRAFT	CARGO LOAD, lb	
C-130E	38,492	27,000
C-141A	64,120	64,120
C-5A	207,046	207,046

each makes its own decisions. A publication of maximum dimensions and weights for which permits will be issued does not exist. Each movement is considered in relation to the dimension and weight of the item and vehicle, the weather, and the exact route over which the vehicle will move.

13-2.3.1 Cargo Vehicles

Highway vehicles can be classified by type, by axle arrangement, by size, and by weight-carrying capacity. Most straight trucks do not exceed 35-40 ft, although recent construction tends to the longer lengths. In weight-carrying capacity, trucks can carry up to 50 tons. Regardless of the weight rating of the vehicle, the resulting axle load must comply with the individual state or country in order to permit unrestricted movement.

13-2.3.2 Size and Weight Limitations

The data that follow generally fit all CONUS and worldwide conditions. For movement over a given route, the restrictions of that route should be determined.

For practical purposes, the vertical height of a vehicle or vehicle and its load for unrestricted movement in CONUS is 12.5 ft. For world-wide movement on the same basis, the height is 11.0 ft. The legal width of a vehicle and/or its load is 8 ft in CONUS. This same width generally is used for foreign countries. The length dimension introduces more variables, but 35 ft for a single-unit truck generally is accepted for both CONUS and foreign areas. In CONUS the combination truck tractor with semitrailer is limited to 50 ft overall, and the same limit is applied to a truck tractor with semitrailer and full trailer, where permitted. In highly developed foreign areas the truck tractor with semitrailer has a general length limit of 14 meters, or 45.93 ft. There is no single figure that is considered to be representative of other vehicle combinations.

Weight limits generally are stated with respect to single axle weight, tandem axle weight, gross vehicle weight as related to the number of axles and their distances apart, and maximum gross weight. As stated previously, almost all figures are compromises. In CONUS all states permit 18,000 lb single-axle loads and, with certain limits, all states permit 32,000 lb on tandem axles spaced 4 ft apart.

In foreign areas, 16,000 lb is an acceptable single-axle load for first class pavements. For tandem axles the limit is 28,500 lb.

13-2.3.3 Van Loading

Interior van dimensions include a 93-in. width, 91-in. door width, and 7-ft 8-in. height. Average load bed height of a cargo van is 56 in. Specialized equipment in the heavy hauler inventory may be as low as 18 in. from the ground.

13-2.3.4 Restraint

Consideration must be given to transportation environments encountered over public highways. DA TB 55-100 provides appropriate guidance in this area.

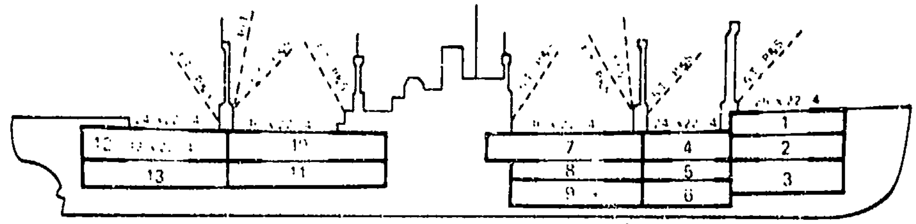
13-2.4 WATERBORNE TRANSPORT

In general, materiel that is transportable by air, highway, or rail modes is accommodated readily by waterborne transport equipment. Special problems that arise in waterborne transportation include the potential for exposure of the helicopter to water and salt spray and the requirement for hoisting of the helicopter during loading/unloading operations.

For general cargo vessels the dimensions of the clear hatch opening at the main deck level limit the size of materiel to be stowed below deck. These dimensions, which vary between hatch locations on a given ship, and the between-deck heights for various ships are shown in Fig. 13-4. The largest cargo item that can be transported by a given ship is determined by the clear-deck loading space available.

13-2.5 RAIL TRANSPORT

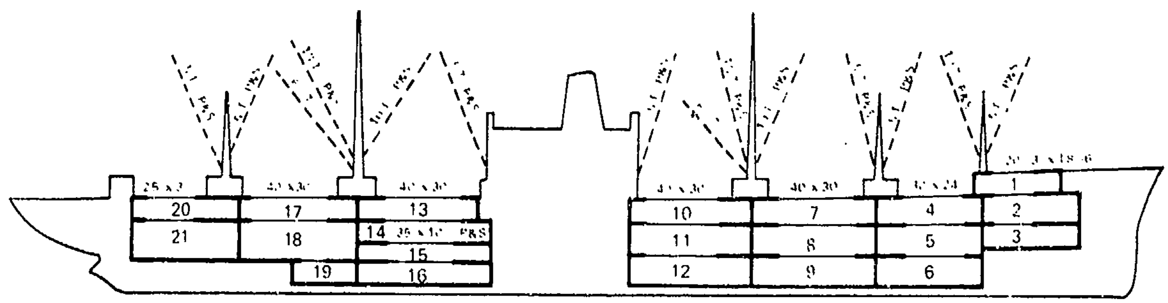
Transportation of Army helicopters by rail generally is not undertaken, unless no other alternatives are available. The legal and environmental conditions are not as critical for movement by rail as with other



COMPARTMENT No	1	2	3	4	5	6	7	8	9	10	11	12	13
CLEARANCE UNDER HATCH GIRDER	6'-7"	9'-8"	15'-4"	8'-5"	7'-7"	11'-4"	8'-1"	7'-7"	11'-4"	8'-0"	10'-4"	10'-7"	10'-4"

P – PORT S – STARBOARD T – LONG TONS

HOLD HEIGHT LIMITATIONS FOR VC2-S-AP3-VICTORY CLASS SHIPS



COMPARTMENT No	1	2	3	4	5	6	7	8	9	10	11	12
CLEARANCE UNDER HATCH GIRDER	7'-7"	10'-8"	11'-7"	9'-7	14'-3"	13'-10"	8'-9'	12'-9"	13'-5'	8'-4"	12'-4"	13'-5"

COMPARTMENT No.	13	14	15	16	17	18	19	20	21
CLEARANCE UNDER HATCH GIRDER	8'-3"	7'-7"	7'-11"	10'-6"	8'-2"	16'-10"	10'-6"	7'-10"	18'-3"

P – PORT S – STARBOARD T – LONG TONS

HOLD HEIGHT LIMITATIONS FOR C4–S–1a MARINER CLASS SHIPS

Fig. 13-4. Hatch Openings and Between-deck Heights

modes; however, certain clearances and/or restrictions due to physical limitations imposed by obstructions (Fig. 13-5) may be encountered along the right-of-way.

Fig 13-6 shows the outline diagram for generally unrestricted railway clearances, referred to as the Berne International Clearance Diagram.

Other areas of the world each have specific clearance diagrams pertaining to the railroads in those areas. Fig. 13-7 shows rail clearances for three sizes of narrow gage track (36-in., 39 3/8-in., and 43-in. gages).

Railway cars generally are classified by type and nominal capacity. Among these the most common types of rail equipment are boxcars and flatcars. Typical boxcars have a capacity of 50 ton and are 40-50 ft long and 9-9.5 ft wide. Typical flatcars have a capacity of 56.5 ton, with a loading surface 52.5 ft long and 10.5 ft wide.

The door openings in the average boxcar vary in size from 6 ft wide by 9 ft 4 in. high to 8 ft wide by 9 ft 10 in. high. Larger boxcars of up to 90 ft in length with 70-100-ton capacities have door openings up to 20 ft wide by 12 ft 7 in. high. These larger and more sophisticated cars are not available as readily as the average cars. Railway cars of foreign countries are generally smaller in size and vary in capacity from 20 to 40 tons.

13-3 MAN/MACHINE CONSIDERATIONS

13-3.1 HUMAN FACTORS ENGINEERING

13-3.1.1 Anthropometrics

Anthropometry is the study of the "measure of man". Because military equipment is used by a wide range of personnel, it must be designed with reference to the ranges in strength, and physiological makeup of those likely to operate or maintain the hardware. Areas to be considered in the optimized anthropometric design include those that follow.

13-3.1.1.1 Body Dimensions

Static measurements to torso, head, and limbs in a standardized position and dynamic measurements taken in working position or task movement must be predicted. These measurements can be obtained directly from the body, or from precision photographs or models. Anthropometric instruments are well standardized and include measuring sticks, calipers, metal tapes, and foot-measuring blocks. Designers also have found scale model articulated mannequins useful in measuring work space in mock-ups.

Figs. 13-8, 13-9, and 13-10 deal with typical values of basic human body measurements (anthropometrics). The document defining anthropometry to be used in the design of a particular helicopter generally will be defined in the solicitation.

The larger number in Fig. 13-9 represents the 95th percentile and the smaller number the 5th percentile. The reach characteristics of a seated crew member (5th percentile) are shown in Fig. 13-10. These are relative to the distance from the center of plane. A common seat reference point is the base point for the dimensions. The charts show the locations at which a control can be grapped between the thumb and forefinger while the shoulders are against the backrest of the seat (inclined 13 deg from the vertical). If the layout is such that the smaller man can reach and grasp within this envelope, any larger crewman generally will find all controls within his reach. Unless otherwise specified by the pro-

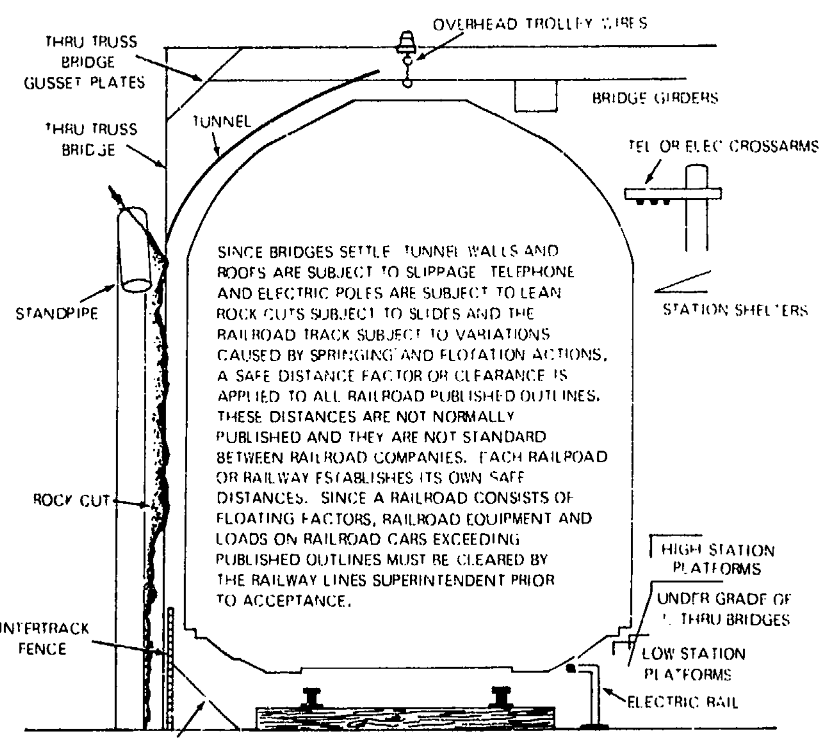


Fig. 13-5. Types of Physical Barriers That Control Railroad Equipment and Loading Sizes

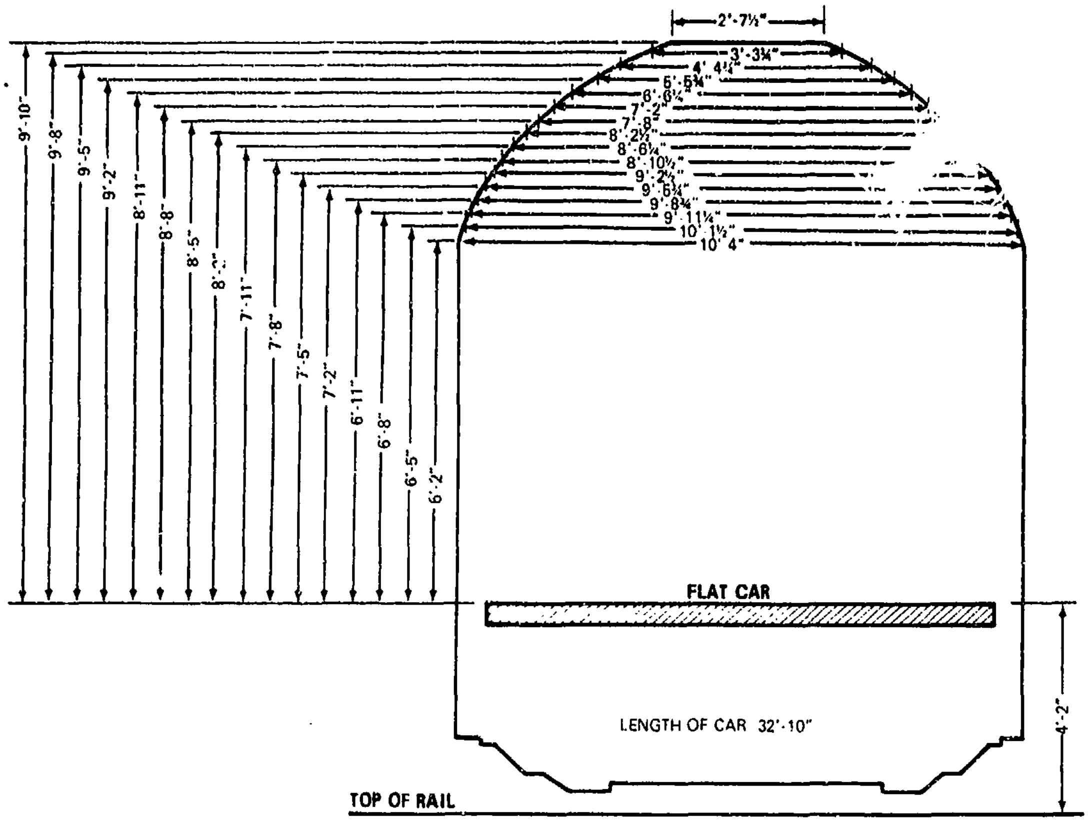


Fig. 13-6. Berne International Clearance Diagram

curing activity, the flight crew stations for Army aircraft *shall* be designed to accommodate the 5th through 95th percentile U.S. Army aviator (Fig. 13-8). Additional data are available in Ref. 1. The seat/control relationships *shall* be in accordance with MIL-STD-1333. The general arrangement of flight crew station controls and displays *shall* be in accordance with MIL-STD-250.

Fig. 13-8 shows selected dimensions for U.S. Army aviators from Ref. 1. The center of mass of the human body varies somewhat due to weight, stature, and build. It varies downward for shorter men and upward for taller ones. The CG or center of mass is slightly more than 50% of the height above the soles (standing). It does not vary with age.

From these illustrations, Ref. 1, and the applicable Military Standards, a first approximation of the crew duty station can be determined.

13-3.1.1.2 Muscular Strength

The maximum force that can be exerted by the weakest 5% of normal American males aged 18 to 30 should be the design criterion for crew-operated systems. Human strength varies with:

1. Age
2. The plane in which the force must be exerted with relation to the rest of the body
3. Hand used (the average individual's preferred hand and arm are about 10% stronger than his other)
4. The direction of force
5. The length of time of exertion
6. The shape and size of the manipulated object.

(See Tables 13-2 and 13-3 and Figs. 13-11 and 13-12.) These data will be of value during systems analysis of flight crew station requirements (par. 13-3.1.3). Ranges of joint motion are shown in Fig. 13-13. Range of head motion is shown in Fig. 13-14.

13-3.1.2 Vision

The human visual system is highly sensitive and adaptive. The lower absolute threshold (laboratory investigations with dark-adapted subjects) is on the order of 10^{-6} footlambert (0.000001 millilambert). The upper absolute threshold determinations are limited because of the danger of injury to the retina. Effectively, the upper threshold can be stated as 16,000 footlamberts. Exposure for even a few minutes to white light above this level is painful and may result in retinal damage.

Acuity is in part a function of how it is measured. The finest width of line that can be detected subtends an angle of approximately 1 sec of arc. Vernier acuity involves the detection of a break in a single line (as lateral displacement), and experiments give values of approximately 5-10 sec of arc as the lower threshold. Landolt ring studies reveal about 1 min of arc as a threshold for existence of space between contours. This is a good absolute minimum figure to remember.

MIL-STD-850 lists the aircrew vision requirements for military helicopter design. Controls, consoles, and instrument panels *shall* be located so as not to restrict vision—particularly over-the-nose visibility. MS 33575 specifies the minimum angles of unimpaired vision that *shall* be available to the pilot. The crew vision requirements on all missions envisioned for a specific helicopter *shall* be investigated to determine if the above criteria are adequate.

Visual acuity varies as a function of brightness (Fig. 13-15). The contrast ratio for light-reflecting crew station displays *shall* be at least 80% for both red and white illumination. Although there is some disagreement among sources as to how to compute contrast ratio, the safest method is shown in Eq. 13-1.

$$\text{Contrast ratio} = \frac{(B_1 - B_2)}{B_1}\,100\ ,\% \qquad (13\text{-}1)$$

where

B_1 = brightness of brighter portion

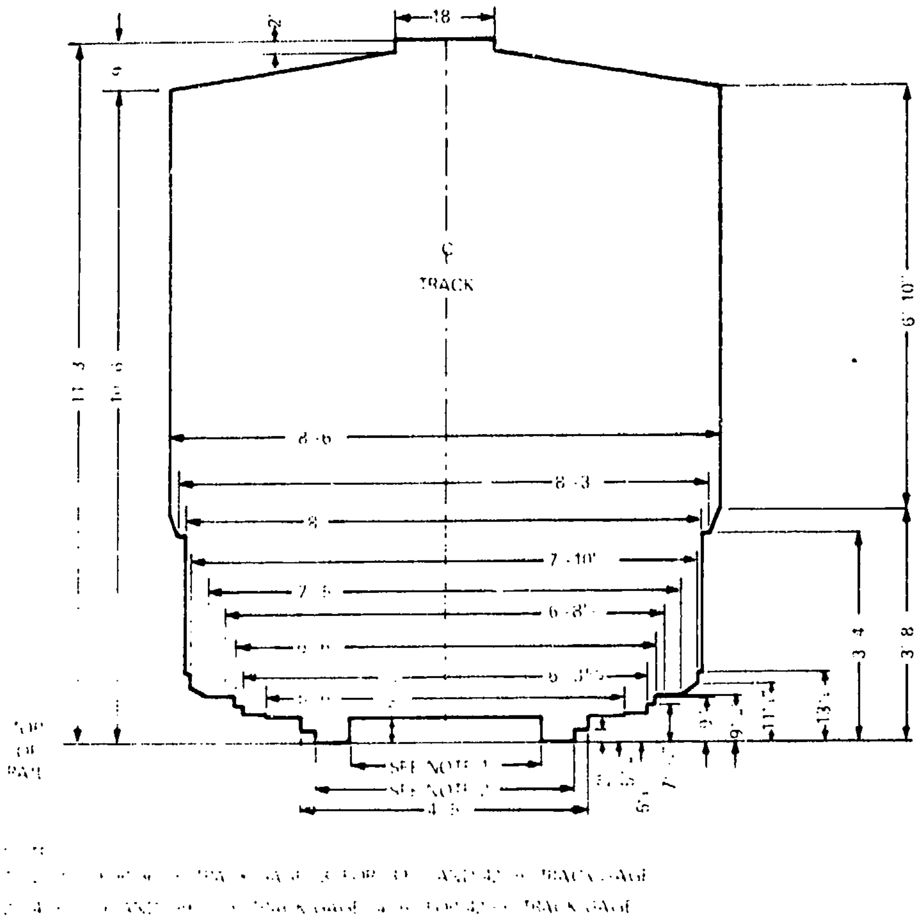


Fig. 13-7. Example Rail Clearances

KEY	MEASUREMENT	PERCENTILE 5th in.	50th in.	95th in.
A	STATURE	64.6	68.7	72.8
B	SHOULDER HEIGHT	52.5	56.3	60.2
C	WAIST HEIGHT	38.4	41.8	45.0
D	CROTCH HEIGHT	29.4	32.2	35.2
E	KNEECAP HEIGHT	18.4	20.2	22.0
F	FUNCTIONAL REACH	28.8	31.1	34.2
G	CHEST DEPTH	8.0	9.5	11.0
H	VERT. ARM REACH	52.8	56.5	60.3
I	SEATED HEIGHT	33.7	35.8	37.9
J	EYE HEIGHT	29.0	31.0	33.1
K	MID-SHOULDER HEIGHT	23.0	24.8	26.6
L	ELBOW REST HEIGHT	7.4	9.1	10.8
M	THIGH-CLEARANCE HEIGHT	4.9	5.8	6.7
N	KNEE HEIGHT	19.3	20.8	22.6
O	POPLITEAL HEIGHT	15.1	16.6	18.3
P	BUTTOCK-KNEE LENGTH	22.0	23.7	25.4
Q	ELBOW-FINGERTIP LENGTH	17.6	19.0	20.3
R	HIP BREADTH	13.2	14.8	16.7
S	FOREARM-FOREARM BRDTH	17.0	19.9	22.8
T	SHOULDER BREADTH	17.0	18.7	20.3

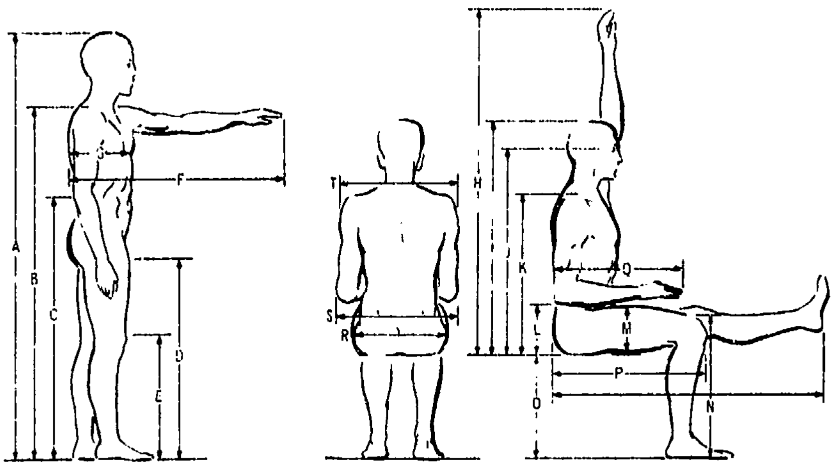


Fig. 13-8. Static Human Body Measurements, U.S. Army Aviators (1970)

(e.g., letter of "white on black" display), lambert

B_2 = brightness of the less bright portion (e.g., background of "white on black" display), lambert

Fig. 13-16 shows the relationship between the minimum visual angles discernible and contrast ratio in percent under three levels of illumination.

13-3.1.2.1 Glare

Glare is the most unwanted effect of illumination. It is controlled or mitigated through use of nonreflective coatings; by dull, minimally reflecting surfaces in panels and other cockpit structures; by proper placement of light sources; and by shielding. Overhead illumination should be shielded to about 45 deg to prevent direct glare.

Eyeglasses cause reflections unless the light source is 30 deg or more above the line of sight, 40 deg or more below, or outside 15 deg laterally from a straight line drawn from outside the corners of one's spectacles.

13-3.1.2.2 Light Transmission, Reflection, and Fogging

Maximum transmission of light through a windscreen bubble is required to assure the best possible external night vision. Polarized glass, nonreflective coatings, and the use of polarized goggles must be considered. Matte, nonreflective paint and surfaces, and glare shields, should be used whenever necessary.

Prevention and removal of windshield fogging and icing that would reduce visibility through transparent surfaces is an important priority. The instrument panel must be positioned and shielded to prevent reflections on the interior surfaces of the windscreen, transparent hatches, and canopy.

13-3.1.2.3 Adverse Visual Effects

While contrast between figures and background should be maximal within a light reflecting display, the general illumination of the crew's field of vision should have "hot spots" reduced to a minimum. At night the panel should be fairly uniform. This is best accomplished through the use of integrally lighted instruments and panels. It is preferable that instruments be illuminated in groups, so that the crew member can adjust the intensity by group. Research has shown individual differences in pilot preference as to how the illumination intensities of his display panel are "contoured". When flying under changing external ambient lighting conditions, the crew member must be able to adjust the illumination levels as his degree of light or dark adaptation progresses. Unless otherwise specified, the primary instrument panel lighting *shall* be integral red lighting in accordance with MIL-L-25467. Although red lighting is more fatiguing than white lighting and does not lend itself to color-coded maps and displays, it is used as the primary panel lighting color in order to achieve maximum dark adaptation. A secondary panel flood lighting system *shall* be provided that is selectable red or white. The white flood lighting *shall* provide a maximum intensity of at least 150 footcandles to provide sufficient illumination after exposure to lightning or nuclear flash.

Patterns of flashing or moving lights can produce hypnotic effects and should be avoided. Particular attention should be given to placement of navigation, formation, and anticollision lights. Single points or patches of light when surrounded by total darkness can appear to float or move (the autokinetic effect). This condition can cause vertigo and must be avoided. Therefore, special consideration should be given to the selection of paint color and finish for rotor blades in order to minimize the reflection of light into the cockpit.

Parallax should be minimized by placing the faces of instruments in planes perpendicular to the normal line of sight of the pilot or other crew members. Similarly, the design for the shape of the windscreen should be established with proper concern for minimizing the distortion of vision caused by differential refraction of light through curved surfaces.

13-3.1.2.4 Optimal Visual Areas

The optimal visual area is the solid angle through which the operator can view displays by eye movement alone. The maximal visual area is limited by the amount of movement of both head and eyes that can occur without muscle strain. Under conditions of acceleration, vibration, or other stresses, this field is narrowed.

Table 13-4 gives optimum and maximum viewing angles as well as the optimum vertical and lateral dimensions. The dimensions correspond to the optimum angles at a 28-in. viewing distance. The main instru-

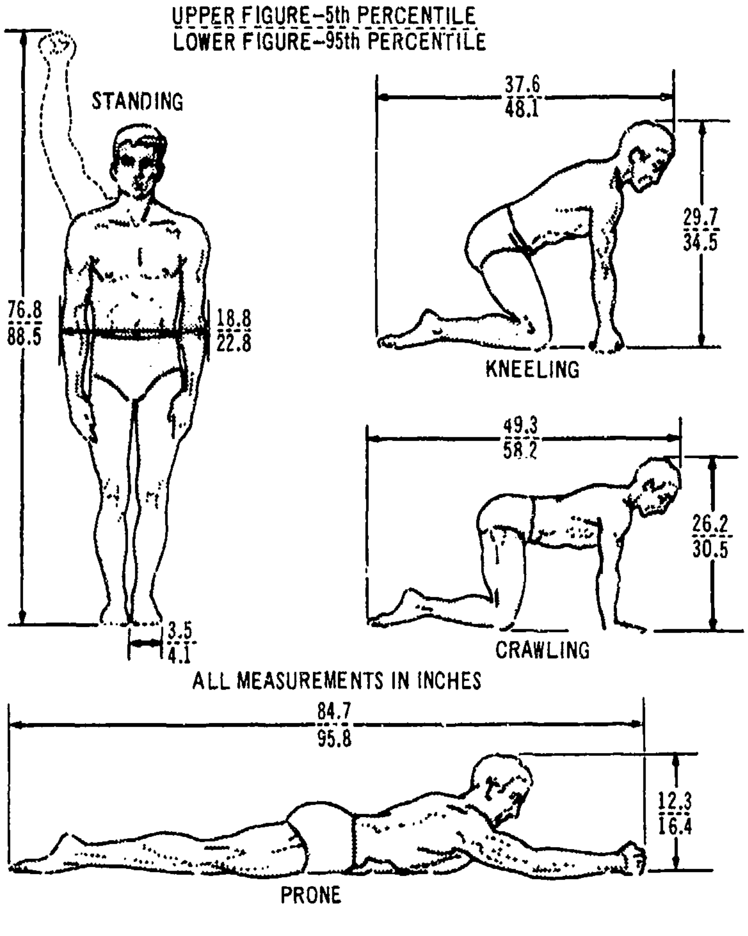


Fig. 13-9. Human Body Measurements of Working Positions

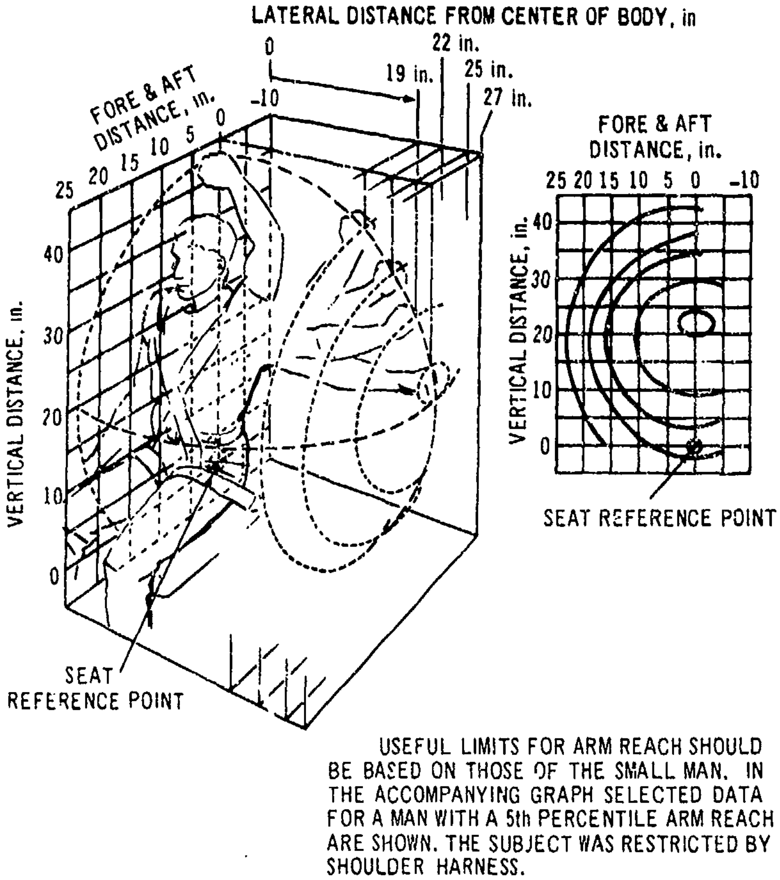


Fig. 13-10. Arm Reach Envelopes

ment panel *shall* be from 28 to 30 in. from the design eye position.

13-3.1.2.5 Cathode Ray Tube (CRT) and Other Light-emitting Displays

The following design recommendations apply to CRT:

1. Scope brightness should be variable between 0.05 and 10 foot lambert.
2. Phosphor excitation caused by ambient illumination should be eliminated.
3. Brightness ratio between CRT and other displays should not exceed 1:10.
4. Specular and diffuse reflections from surfaces of the tube, overlay, and filters should be eliminated.
5. Design should permit an observer to view the scope from as close as he may wish (see MIL-STD-1472).
6. Design should provide the appropriate degree of resolution.

Electroluminescent (EL) displays require a contrast ratio between figure and background of 20-30%. EL displays with appropriate dimming controls are extremely well suited for night flying. In bright sunlight, some have a tendency to "wash out". Appropriate high-contrast construction, combined with antireflective coatings or filters, can meet the contrast and legibility requirements.

13-3.1.3 System Analysis

Frequently, in the design of new helicopters, mission requirements will require trade-off decisions to be made regarding the allocation of certain functions to man or machine and the arrangement of controls, displays, and other mission equipment within a crew station. In the case of a flight crew station where no new or special controls or displays are required, MIL-STD-

250 may prove sufficient. However, when special equipment or unique mission requirements are involved, system analysis should be performed, using the techniques that follow, as required.

13-3.1.3.1 Flow Charting

The broad analysis of information required for the mission should be represented in flow chart form. As these analyses progress, more refined and detailed charts should be developed.

13-3.1.3.2 Function Analysis

From the flow charts, points must be identified where information transformations and/or transfers are required. The load, accuracy, and rate and time delays, as well as cost and other known constraints,

TABLE 13-2
ARM FORCE DESIGN VALUES FOR HELICOPTER CONTROLS

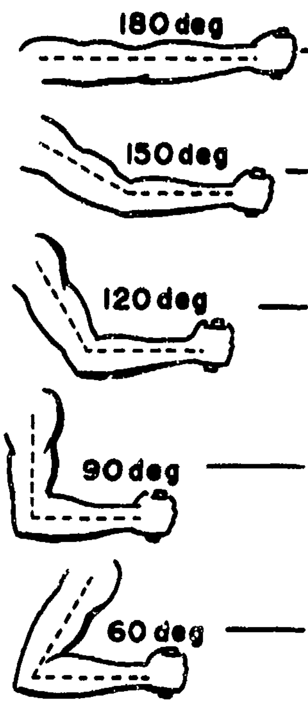
RIGHT ARM ON HELICOPTER CONTROL STICK, lb

DISTANCE, in., FROM SRP*	DISTANCE, in., FROM MIDLANE	PUSH	PULL	IN	OUT
9	8 (LEFT)	12	26	24	34
	$4\frac{1}{2}$ (LEFT)	18	28	31	31
	0	26	34	30	23
	$4\frac{1}{2}$ (RIGHT)	34	39	26	15
	8	37	39	26	12
$12\frac{1}{2}$	8 (LEFT)	18	33	23	31
	8 (RIGHT)	43	49	22	16
$15\frac{1}{2}$	8 (LEFT)	23	39	20	25
	0	43	54	24	20
	8 (RIGHT)	53	55	24	13
$18\frac{3}{4}$	8 (LEFT)	36	45	16	22
	0	64	56	8	15
	8 (RIGHT)	70	58	22	14
$23\frac{3}{4}$	8 (LEFT)	29	51	11	19
	0	54	62	14	13
	8 (RIGHT)	56	58	20	12

*FORWARD CONTROL IS $13\frac{1}{2}$ in. ABOVE SEAT REFERENCE POINT (SRP)

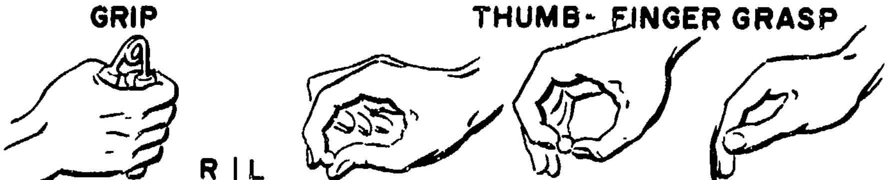
TABLE 13-3
DESIGN VALUES FOR MOTOR PERFORMANCE (MUSCLE STRENGTH)

ARM STRENGTH, lb.



SEATED / PRONE	PUSH R	PUSH L	PULL R	PULL L	UP R	UP L	DOWN R	DOWN L	IN R	IN L	OUT R	OUT L
180 deg (seated)	**50**	**42**	**52**	**50**	**14**	**9**	**17**	**13**	**20**	**13**	**14**	**8**
180 deg (prone)	31	26	31	18	8	5	13	7	12	10	9	4
150 deg (seated)	**42**	**30**	**56**	**42**	**18**	**15**	**20**	**18**	**20**	**15**	**15**	**8**
150 deg (prone)	29	24	29	21	13	7	15	10	15	8	12	5
120 deg (seated)	**36**	**26**	**42**	**34**	**24**	**17**	**26**	**21**	**22**	**20**	**15**	**10**
120 deg (prone)	29	21	31	22	13	11	15	11	15	9	11	6
90 deg (seated)	**36**	**22**	**37**	**32**	**20**	**17**	**26**	**21**	**18**	**16**	**16**	**10**
90 deg (prone)	26	18	24	23	15	15	16	12	16	13	13	6
60 deg (seated)	**34**	**22**	**24**	**26**	**20**	**15**	**20**	**18**	**20**	**17**	**17**	**12**
60 deg (prone)	24	17	21	17	13	13	13	10	16	11	12	8

HAND STRENGTH, lb



GRIP

	R	L
MOMENTARY	59	55
SUSTAINED (1 min)	42	38

THUMB-FINGER GRASP

PALMAR	TIPS	LATERAL
13	13	15

LEG STRENGTH, lb

	R	L
MOMENTARY	387	413
SUSTAINED (2 min)	300	300

ANGLE: KNEE 111 deg ± 5 deg
ANKLE 60 deg ± 5 deg

should be analyzed to determine the optimal allocation of functions to men and machines, as well as the degree of automation introduced. Sufficient justification presenting analyses of crew locations and workload *shall* accompany any proposals to automate crew tasks. A major part of this determination is a function of the sensory and psychomotor capabilities and limitations of the human.

13-3.1.3.3 Operational Sequence Diagramming

Operational sequence diagrams differ from flow charts in that the latter depict decisions/operations without reference to any specific machine implementation or level of human involvement. The various columns of the operational sequence diagram identify the man or machine and the type of information processing that takes place at that point in the system. Sequentially diagramming and symbolically representing the processing of operational information are the preferred approaches and should be followed wherever feasible.

13-3.1.3.4 Definitions of Operator/Maintainer Information Requirements

In order eventually to define control, display, and communication requirements, it is necessary that a comprehensive determination be made of information requirements for all major or critical operator and maintenance positions. The sources of information for this purpose are the operational sequence diagrams previously constructed. The information requirements are inputs, processing, outputs, load, accuracy, rate, and time delay for each major operator and maintainer position.

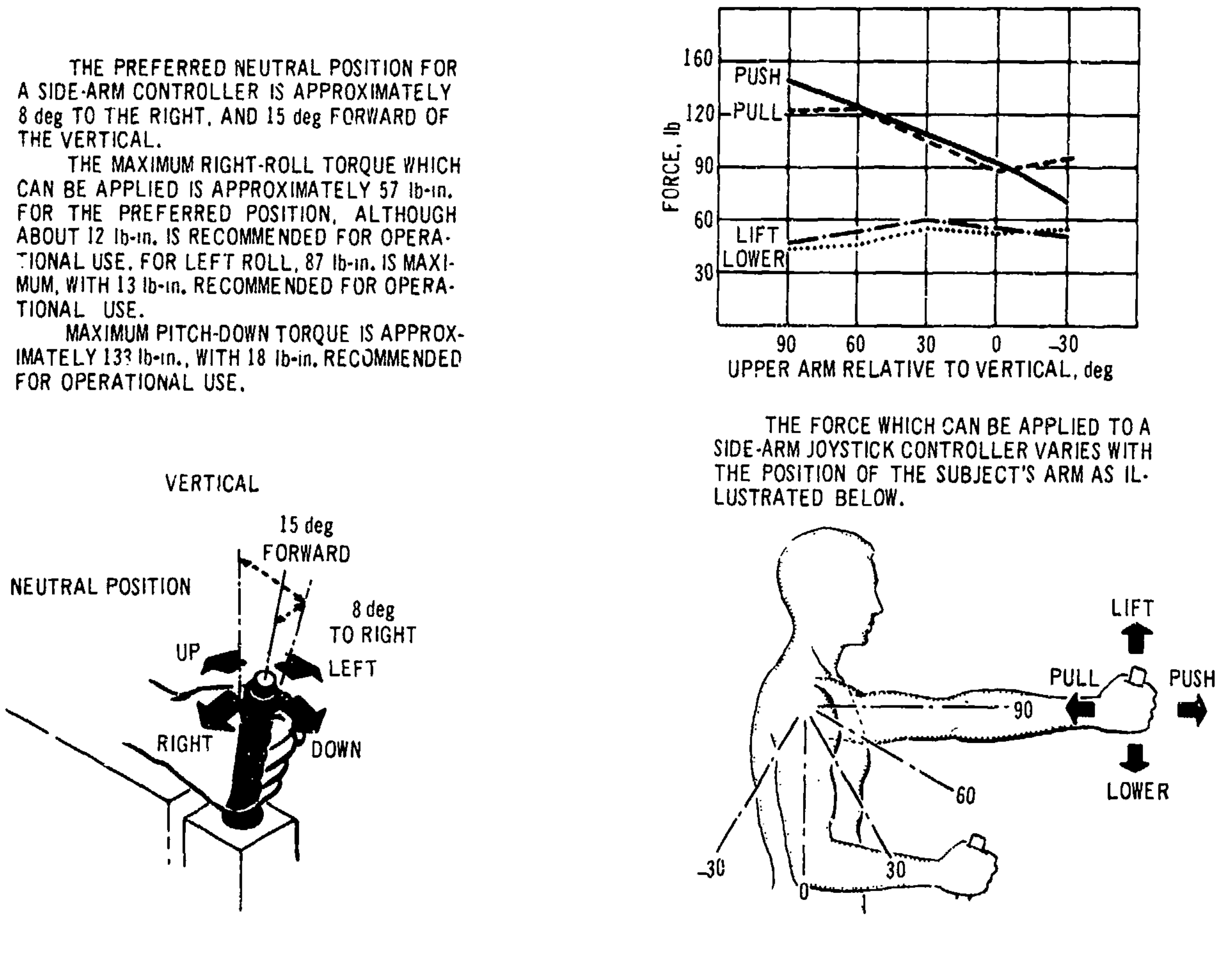


Fig. 13-11. Arm Strength With Elbow Flexion (Right Arm/Forearm-Wrist Strength)

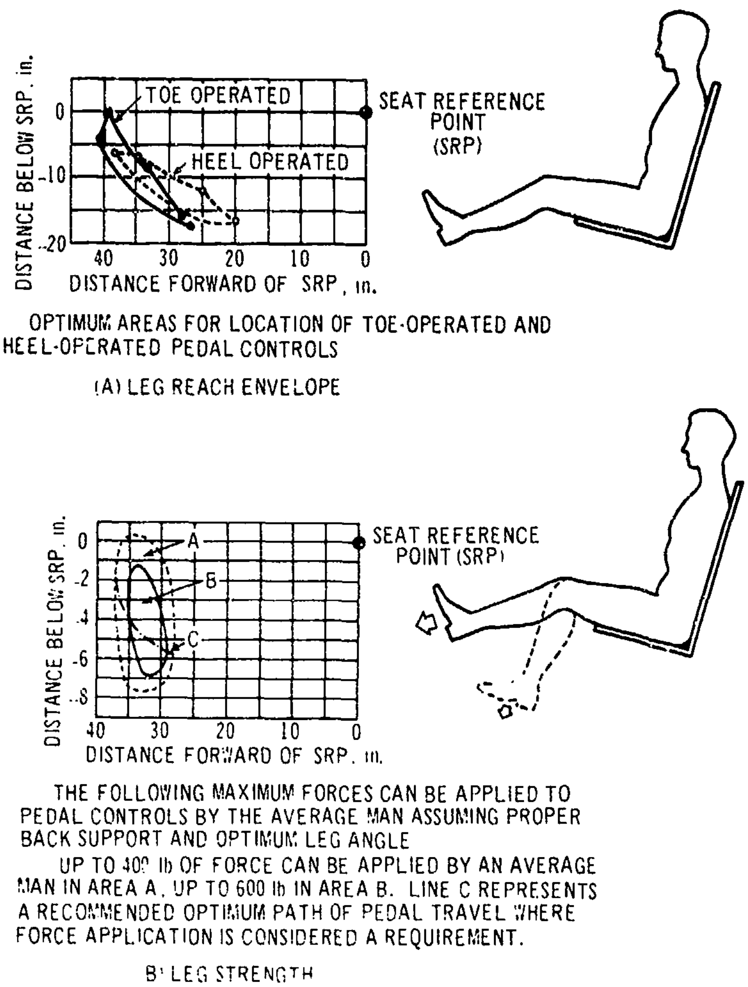


Fig. 13-12. Leg Reach Envelope/Leg Strength

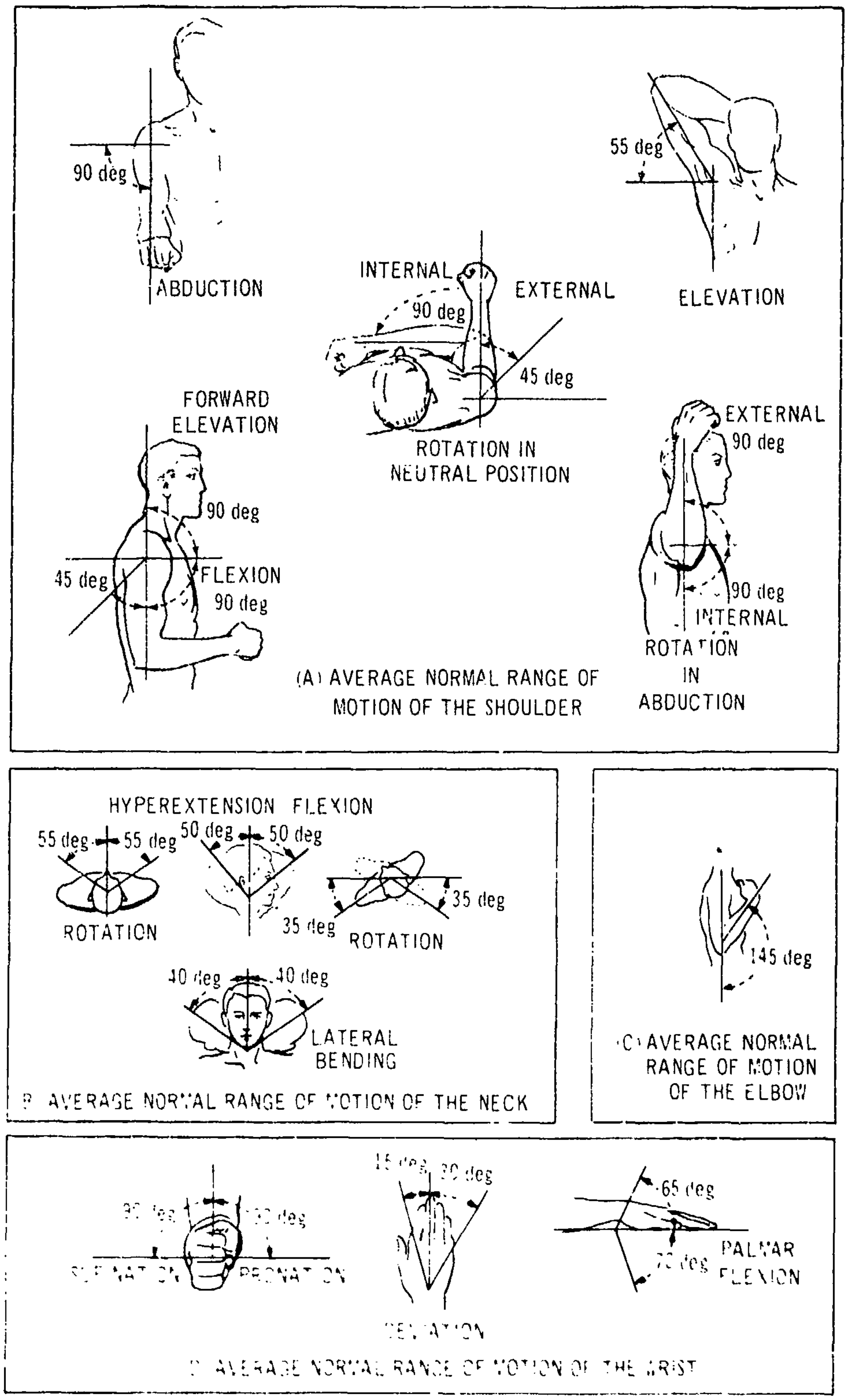
13-3.1.3.5 Operator/Maintainer Task Descriptions and Analyses

Task descriptions should be developed from the decisions that determine the assignment of functions to man and to machines and that, therefore, influence both hardware operations and human tasks within the system. Once developed, task descriptions should serve as a basic reference for all later designs and plans for the personnel subsystem. The summaries of operational sequence diagram requirements may serve as the prime source of information from which task descriptions and task analyses may be compiled. Task descriptions may be prepared as narrative—describing, through specific statements, all the interactions of man with machines and with the system environment—or symbolic format with appropriate comments. The task description specifies, along a time scale, the cues the human should perceive in a task environment and the related responses that the human should make in his task environment.

Where greater detail is needed—as with tasks that are new, difficult, dangerous, or contain elements of critical human involvement—a more detailed task analysis will be required. The analysis should include operator interaction where more than one operator is included.

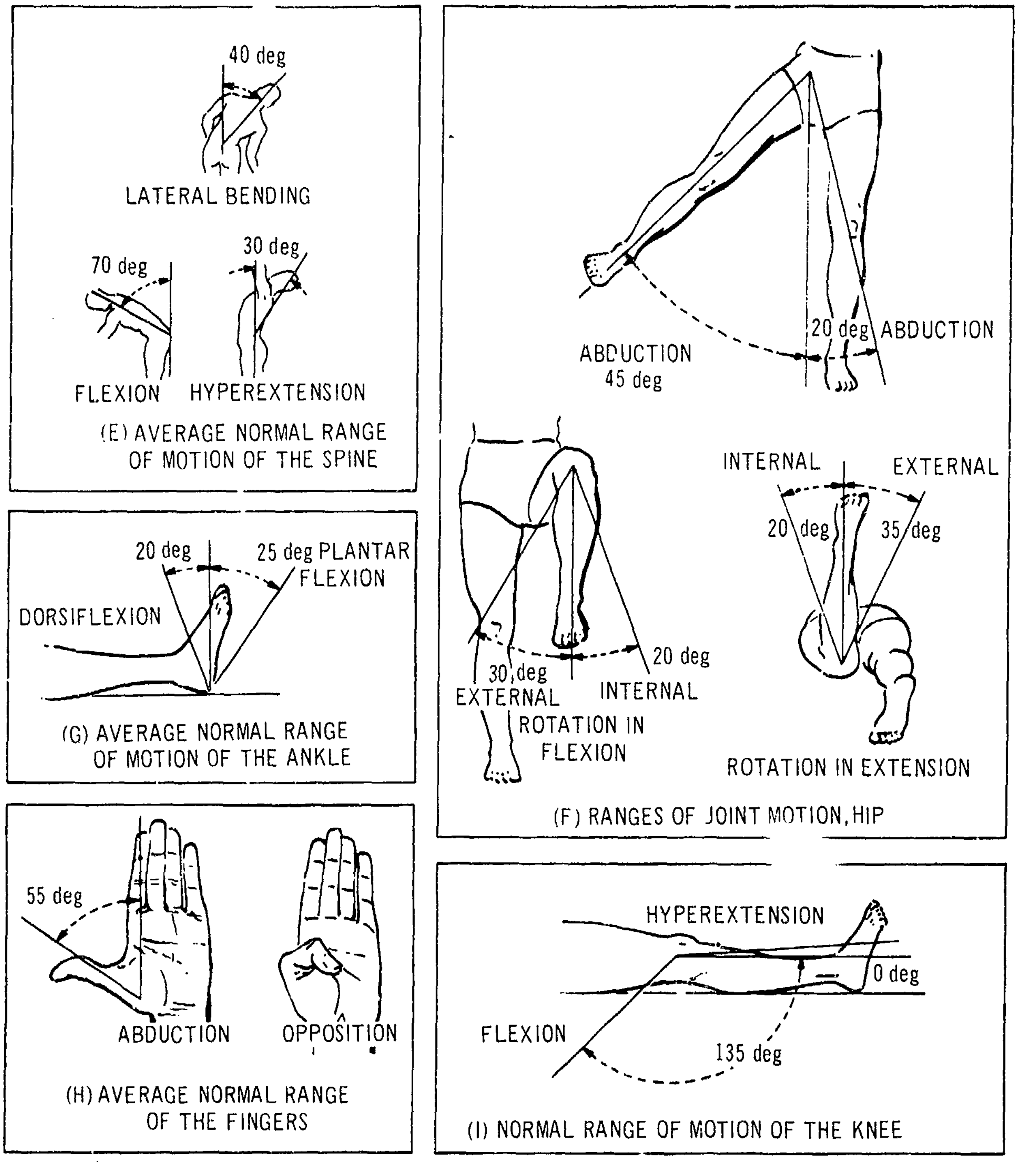
13-3.1.4 General Control Principles

The following guidelines for direction of control movement refer to increasing the magnitude of, or "switching-on", the controlled element:



(continued)

Fig. 13-13. Ranges of Joint Motion



(E) AVERAGE NORMAL RANGE OF MOTION OF THE SPINE

(F) RANGES OF JOINT MOTION, HIP

(G) AVERAGE NORMAL RANGE OF MOTION OF THE ANKLE

(H) AVERAGE NORMAL RANGE OF THE FINGERS

(I) NORMAL RANGE OF MOTION OF THE KNEE

Fig. 13-13. Ranges of Joint Motion (Cont.)

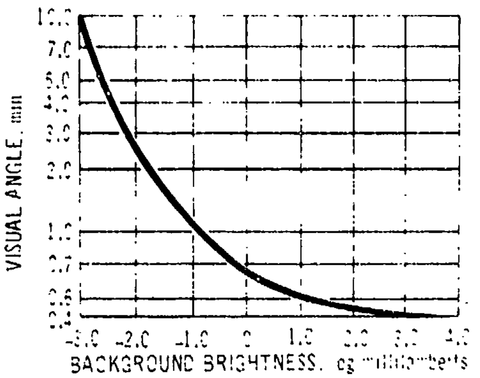


Fig. 13-15. Minimum Visual Angle Resolved at Various Brightness Levels

1. Linear controls should move forward, upward, or to the operator's right.

2. Rotary controls should move clockwise.

3. Movement of controls associated with visual displays should correspond to the display movement.

Primary and emergency controls should be distinguishable readily from secondary controls and from each other both visually and tactilely. Additional information on control location and actuation is provided in MIL-STD-250.

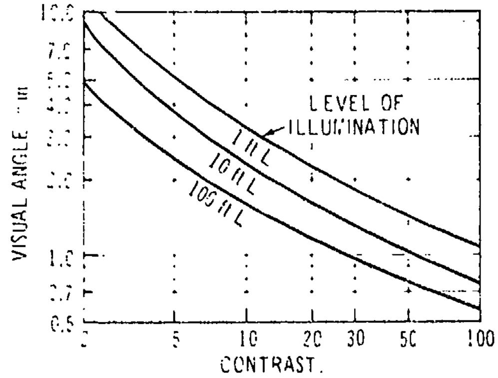


Fig. 13-16. Minimum Visual Angles for Various Contrast Ratios

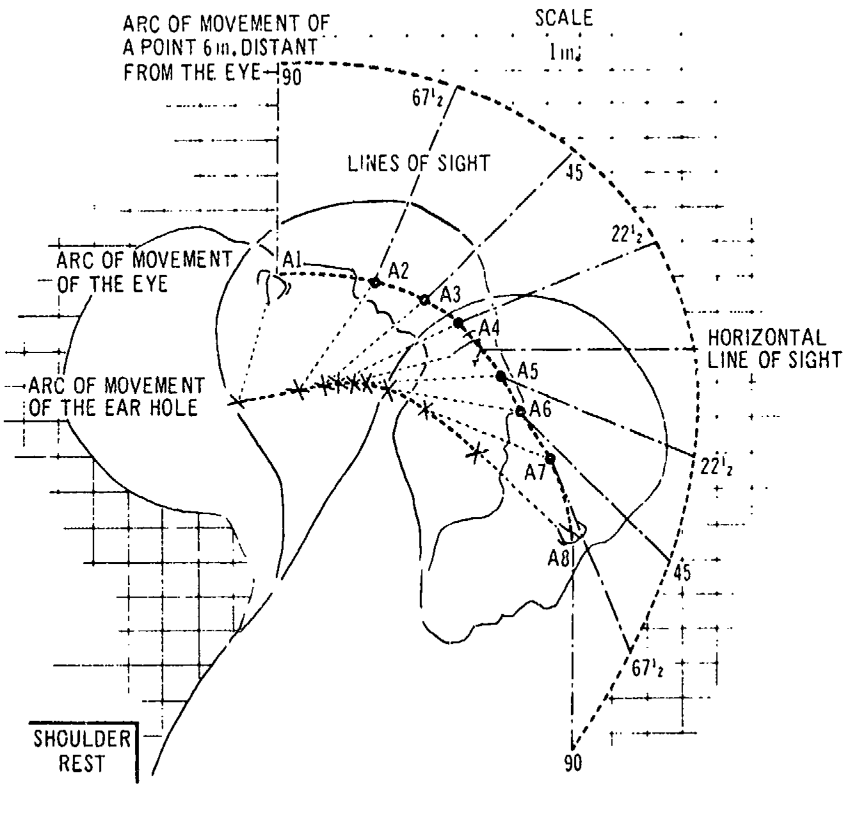


Fig. 13-14. Range of Head and Eye Movements in the Vertical Plane

13-3.1.5 Vibration

The effects of vibration on a body depend upon the physical parameters of the impinging energy; its direction of application with respect to the axes of the body; and the mechanical impedance and absorption coefficient of body tissue, organs, and the rest of the body, plus the consideration of the body as a whole. The resonant frequencies of the body and its parts are of special importance. High amplitude-low frequency (up to 20 Hz) vibrations cause discomfort and can cause physical damage. The main range of human resonance is 2-5 Hz. Cushions used for comfort and to damp vibrations also must consider crashworthiness. See Chapter 5 for additional discussion on the effects of vibration on human performance and the various design techniques used to minimize its effects.

13-3.2 SAFETY AND EFFECTIVENESS OF PERSONNEL

The primary method of identifying hazards and prescribing corrective actions in a timely, cost-effective manner is a system safety program in accordance with MIL-STD-882. Some typical safety considerations that must be addressed during the preliminary design phase are discussed in the following paragraphs.

13-3.2.1 Emergency Egress

13-3-2.1.1 Requirements

Ref. 2 establishes that emergency evacuation of helicopters *shall* be accomplished within a maximum of 30 sec for aircrew and passengers on all multiplace and cargo transport vehicles, using one-half of the available exits. This requires a minimum of one exit for every ten persons expected to occupy any independent section. This requirement for an occupant/exit ratio of 10:1 is based upon the possibility that at least 50% of the exits may be blocked if the aircraft should come to rest on its side, thus leaving an effective occupant/exit ratio of 20:1. Ideally, it is expected that evacuation of all occupants can be accomplished at an exit rate of 1.5 sec per man, on the assumption that there is no occupant debilitation and that all exits are open. In addition, the location, size, and actuation of emergency exits should be in accordance with Ref. 2.

TABLE 13-4
DIMENSIONS OF VISUAL ANGLES

UNITS	DIRECTION OF MOVEMENT		
	LEFT OR RIGHT	UP	DOWN
DEGREES			
OPTIMUM	15	[illegible]	30
MAXIMUM	95	75	85
INCHES			
OPTIMUM	5.5	9	10

13-3.2.1.2 Emergency Environments

Design of the emergency evacuation provisions of a military helicopter should consider the full range of potential environmental conditions through which emergency situations can arise and progress. Examples of abnormal environmental conditions that could exist before or during the evacuation are severe attitude; abnormal maneuvers including spins, sideslips, and rolling or phugoid oscillations; buffeting and vibration; fire, heat, and smoke; darkness; noise; the presence of combat threat; structural damage; or blocking debris. Emergency evacuations may be required when the helicopter is airborne or after it is on the ground or in water.

13-3.2.1.3 Human Factors Engineering Considerations

Three major human factors engineering (HFE) considerations are established by MIL-H-46855 as basic to developing an optimum emergency evacuation system, i.e., anthropometric, physiological, and psychological.

Ref. 2 deals with dimensional clearance requirements for passage through hatches and other avenues of egress, and the reach limitations of crew and passengers as they interface with structural and exit release mechanisms. It is extremely important that the design include the added requirements of clothing, especially protective garments that increase clearance requirements and/or restrict body and limb reach, mobility, and agility.

Psychological considerations include the limits of human strength and reaction time for the operation of specific controls, and abnormal response characteristics that may be induced by various environmental factors, i.e., air sickness, disorientation, loss of postural control, asphyxiation, fatigue, or degraded visual capacity (brought about by severe vehicle motions and attitudes, fumes, fire, heat, smoke, or lack of illumination).

Psychological considerations encompass both normal and abnormal perceptual-motor and cognitive behavior of crew and passengers. Because most emergency evacuations will occur under conditions of extreme stress, emergency egress systems must be designed on the assumption that crew and passenger may not react according to normal behavior patterns. They likely will not think clearly, be prone to error, "freeze" into inactivity, forget recent training, and revert to previous habit patterns.

13-3.2.1.4 Emergency Egress Preliminary Design

The development of an adequate system for emergency egress should be based upon thorough analysis of egress procedures versus hardware configuration. Although the design specification of a given helicopter will not be complete when the analysis is begun, the general configuration concepts will be sufficiently well defined to establish egress design requirements and constraints.

Ref. 3 points up the importance of considering the emergency egress early in the overall design development. If this is not done, commitments to design details in other areas may limit the opportunity for egress system optimization.

It is likely that the design team would wish to enlist the aid of human factors specialists, familiar with human capabilities and limitations, in the conduct of system analyses (see par. 13-3.1.3) and design selection. Consideration should be given not only to operation, but also to maintenance of the emergency evacuation design elements.

The following should be considered during preliminary design:

1. Hatch Openings. Hatch opening size and shape must be based upon the dimensions of crew members, including the special garments and gear that they are expected to wear or use during evacuation (see Refs. 1 and 4). The shape and position of the opening also are affected by the mobility of the crew member and accessibility to the opening. Abnormal, as well as normal, access situations should be considered. Further guidance to hatch opening design is provided by Ref. 2.

2. Exit Doors. Outward opening doors are preferred; however, inward opening doors may be considered. Generally, sliding doors are undesirable for emergency exits because they are subject to jamming in a crash. Door attachment and release must be designed with priority consideration given to crew survivability and rapid egress. Doors should be designed so they will open easily even with ice accumulation, seal vulcanization, or reasonable amounts of distortion. Prime consideration must be given to the location of inside operating handles and to the need for the unlocking mechanism to operate under structural distortion following crash landings.

Emergency exits must be designed similarly and operable under crash landing conditions. Design consideration should include outside handles and latches capable of being operated by personnel equipped with the standard firefighter's asbestos gloves. Emergency exit closures must be capable of being removed within 5 sec; Ref. 5 points out the desirability of removal in even less time. Opening mechanisms should be simple to operate to allow for crew debilitation. Opening by one person using one hand exerting a force of no more than 30 lb is required. The exits should be designed so that only a single pulling or pushing of the exit closure into the clear is necessary once the release handle has been actuated. Up-and-down sliding action should be avoided. Actuation of an internal release handle must not preclude simultaneous actuation of an external release handle. A "locked position" indicator should be provided on both internal and external handles to indicate positive locking of the door in accordance with Ref. 5.

Rapid ingress/egress is mandatory for troop operations under emergency and tactical situations.

3. Explosively Created Exits. The feasibility of applying shaped charges to create emergency exits has been demonstrated. The significant design considerations involved are high reliability in the arming and firing of the explosive and protection against inadvertent actuation.

4. Canopy Enclosures. Sections should be removable by means of latched assemblies of shaped charges, thus eliminating the need for crew members to cut transparent canopies with tools, knives, axes, etc.

5. Location of Exits. Exits should be distributed equally on each side of the vehicle; but, if possible, should not be directly opposite one another. The exits should be located so as to be least exposed to distortion of surrounding fuselage structure, and so that it is not necessary to handle equipment, cargo, or furnishings to gain access to the exit. External access to the exits should not be inhibited unnecessarily by external components, armament, and landing gear, nor should exit location expose rescue crews to fuel spillage or major ignition sources. External access should be viewed in terms of distance from the ground, or other normal support for the rescue crew.

6. Exit Access Routes. Access from aisles to exits must not be obstructed by troop seat components, seat back webbing and webbing support, litter installations, or other protrusions. The aisles should be wide enough for passengers or crew to run to an exit; and the location and direction of opening of doors separating compartments should not impede or block passage to other

exits or interfere with access to or use of special emergency and survival equipment.

Aircrew members should be able to reach and unlock emergency release handles without first removing their seat restraints. (This should not be construed as a recommendation to remove emergency exits prior to an impending crash.)

Strategically located locomotion aids (handholds, rope slides, or escape reels) should be provided to assist crew and passengers in gaining access to the emergency exit or to the ground. This is important particularly where an exit may be located in an inaccessible place (in the event the aircraft is lying on its side, the exit may be above the person trying to reach it).

7. Evacuation Aids. Special evacuation aids (ropes, slides, etc.) should be provided at exits that normally are 90 in. or more from the ground. If inflatable slides are used, the design should insure that the slide is available for immediate use simply by opening the exit and actuating the inflating mechanism. The design must preclude improper installation and should be operable manually in the event the automatic inflating mechanism fails. Impact activated emergency lighting, with provision for activation also from the pilot's compartment, should be included. This system should be independent of the primary electrical system of the aircraft.

The designer should take maximum advantage of the helicopter mock-up in order to verify the adequacy of emergency exits (see Chapter 5, AMCP 706-203). Also, Ref. 2 should be consulted and used throughout the designer's crash survival analysis and design.

13-3.2.2 Acoustical Noise Level and Suppression

Excessive helicopter interior noise can impair mission performance directly by interfering with crew comfort and communication. This excessive noise, continued long enough, can cause permanent damage to hearing. Noise reduction is of direct economic interest to the military due to the large number of individuals receiving military disability retirement payments due to hearing loss. The discussion that follows deals with hearing damage, including speech interference effects.

13-3.2.2.1 Acoustical Concepts

The ear is sensitive to the rapid variations of air pressure above and below ambient pressure; these variations constitute the physical cause of the sensation of sound. The most common description of that sensation is loudness. It then is natural to seek a measure of the physical magnitude of the sound that has a more or less direct relation to loudness. This measure is the sound level L, defined as:

$$L_p = 20 \log_{10} \frac{p}{p_o} \text{ , dB} \quad (13\text{-}2)$$

where

p = sound pressure, dyne/cm²
p_o = reference pressure, dyne/cm²

By international agreement, $p_o = 2 \times 10^{-4}$ dyne/cm². In English units, $p_o = 2.9 \times 10^{-9}$ psi. Ordinarily, conversational speech has a sound pressure level of 60-65 dB at a distance of about 3.28 ft (1 m).

In the calculation of the sound pressure level in enclosed spaces, a more useful quantity is the total sound power radiated into the enclosure. This is obtained by integrating the intensity (power per unit area) over the area concerned. Intensity level L_I is defined by

$$L_I = 10 \log_{10} \frac{I}{I_o} \text{ , dB} \quad (13\text{-}3)$$

where

I = sound intensity, W/cm²
I_o = reference intensity, W/cm²

The reference intensity I_o is standardized at 10^{-16} W/cm² or 6.45×10^{-16} W/in.²

The psychoacoustic attribute of sound that is called pitch has a close physical equivalent in frequency. The sense of pitch arises from the position of maximum vibration on an inner-ear structure called the basilar membrane. Male voices usually have a normal pitch in the 100-to 150-Hz range. A 30-blade compressor turning at 10,000 rpm would have a pitch of 5 kHz. The importance of frequency in damage due to noise is that the basilar membrane apparently is most easily damaged in the 2-to 6-kHz range.

Psychologically, noise is any sound undesired by the recipient. Physically, noise is regarded as a chaotic sound in which single-pitched tones may be present, but do not predominate. Noise usually can be described best by its spectrum. This is the statement of the amount of sound within given frequency intervals.

13-3.2.2.2 Hearing Loss

The end organs of hearing are nerve ends that are stimulated mechanically by sound. Intense and continued sound first will fatigue the nerves, causing a temporary loss of hearing. If acoustic inputs are repeated too often, permanent loss of function of the ear can be caused because nerve tissue does not regenerate when subjected to this environment. Hearing loss also can arise from decreased capability for conducting sound to the inner ear. The first loss is more important because it is irreversible.

Hearing loss usually is measured by the increase in sound pressure level of pure tones of different frequencies that just barely are audible. The values are compared with an internationally accepted set of reference levels that give the threshold of hearing for an average young population. These threshold shifts are measured with an instrument called an audiometer. Hearing loss does not produce noticeable impairment of ability to understand speech until the average of the losses at 500, 1000, and 2000 Hz exceeds 26 dB. A hearing aid will be necessary starting at the 40-dB loss level.

13-3.2.2.3 Noise Exposure

Noise exposure is the physical measure of noise from which hearing damage risk reasonably can be predicted. For noise without marked puretone or impulsive components, the noise exposure essentially is proportional to the total energy transferred to the ear. For industrial noise, the amount of noise permitted is based on an 8 hr/day, 5 day/wk, 50 wk/yr, 20-year work life.

Experience in the replacement of octave band measurements by a single-number statement of noise magnitude has resulted in increasing acceptance and use of the A-scale reading, in dB(A). Recent limits on industrial noise exposure issued by the Department of Labor under the Walsh-Healey Act provide that for each 8-hr day, the maximum exposure times permitted at each level (in dB(A)) *shall* not exceed those given in Table 13-5.

If the noise levels vary, then the total exposure is computed as a time fraction. Let t be the time for which the sound level is L_p dB(A). Then from the given data (or from the appropriate smooth curve drawn with those data) obtain t_o, the maximum time permitted to that level. If for all noise segments the quantities (t/t_o) when summed exceed unity, then the limiting noise exposure has been exceeded. A rough measure of the equivalent exposure level (EEL) is

$$EEL = 90 + \log_{10} \sum_{i=1}^{n} \left(\frac{t}{t_o}\right)_i \quad , \text{dB} \qquad (13\text{-}4)$$

where

t = time, hr

t_o = maximum permissible time, hr

The Walsh-Healey limit will result in moderate occupational hearing loss for a 20-yr work life. Some would like to see it eventually reduced by 5 to 10 dB(A), so that a very safe condition would be attained.

13-3.2.2.4 DOD Noise Limits

The applicable DOD limits on noise in aircraft are given in MIL-A-8806, which was derived from an Air Force study (Ref. 6). This specification gives limits in terms of octave band levels from 31.5 to 8000 Hz, for four operating conditions. If the maximum octave band levels define a spectrum, these can be expressed in terms of dB(A) by procedures given in the Walsh-Healey limits. In terms of the four operating conditions of MIL-A-8806, the levels and maximum permitted (Walsh-Healey) times then would be:

1. Maximum continuous power 100 dB(A): 2 hr
2. Maximum short duration conditions 105 dB(A): 1 hr
3. Maximum with protective devices 105 dB(A): see comment that follows
4. Normal cruise power (Naval aircraft) 94 dB(A): 4.5 hr.

No time duration is stated for the condition with protective devices; if these afford at least 15 dB protection, then the permissible exposure time would be 8 hr. Again, recall that these would be for 250 days/yr, for 20 yr. For less total flying time, the levels can be higher. However, above 115 dB(A) damage is so certain that it constitutes a very real ceiling on noise of a duration of at least 15 min.

The main consideration of MIL-A-8806 was not the effect of noise on hearing damage, but its effect upon intelligibility, annoyance, comfort, and ability to ac-

TABLE 13-5
MAXIMUM NOISE-LEVEL EXPOSURE TIMES (WALSH-HEALEY ACT)

SOUND LEVEL L_p, dB(A)	EXPOSURE per 8-hr day, hr
90	8.00
92	6.00
95	4.00
97	3.00
100	2.00
102	1.50
105	1.00
110	0.50
115	0.25

complish the mission. It generally has been agreed by the military services that the sound pressure levels at the various octave bands quoted in MIL-A-8806 are achievable without serious weight and cost trade-offs.

13-3.2.2.5 Control of Noise at Source

There are a number of means of exercising engineering control over the noise in helicopters. Choice of quiet equipment and components, reduction of driving force, isolation of vibration, reduction of fluid velocity, reduction of air turbulence, use of total or partial enclosures, and attenuation along transmission paths are the usual schemes. The airborne noise from engines, rotor, and propeller may be attenuated by appropriate cabin wall design and increased thickness of windshields and canopies. The attachment of these components to the cabin structure often can be resilient to reduce vibration transmission. Rotor noise is minimized by design techniques such as reduced blade loading, reduced tip speed, and special-geometry blade tips. Aerodynamic noise is reduced by assuring that no protuberances or holes in the aircraft structure are placed in the slipstream. However, the requirements of stability and alignment conflict with this suggestion.

Cabin wall construction for noise attenuation is complex, with much design information being empirical. Generally, division of the available wall material into several decoupled layers produces the most attenuation for a given mass per unit area. There should be at least two imperforate skins, each with some damping material adherent to it, and a layer of absorptive material between. The requirement of decoupled layers is inconsistent with that of strength and stability. Vehicle performance degradation from added weight must be evaluated against crew performance degradation from increased noise. If crew communication performance is critical, then added emphasis must be placed upon noise reduction. This should be introduced at the earliest stages of specification and design.

Every opening in the cabin—windows, doors, access panels, and holes for controls and wiring—introduces the possibility of a considerable leak. Such leaks can compromise seriously the performance of a superior wall construction. For example, if the wall attenuation is 20 dB, then a leak of about 1% of the total cabin wall area will pass as much noise as the entire cabin wall. If the cabin is to be pressurized, such leaks, of course, ordinarily will not exist. Hole closures should have at least as much attenuation as the wall proper.

Some cabin panels may be excited into vibration by aeroelastic effects. Spot damping treatments usually reduce the sound thus radiated. If stiffening of skin sections is required, the stiffeners should be placed to divide the skin into areas of unequal size, shape, and orientation. Thus the new, higher frequency resonances of the divided panel will occur at different frequencies, reducing chances of excitation.

Transmission quieting remains a very difficult problem. Components preselected for quietness can be used in a new vehicle. The noise directly radiated can be reduced by building an isolating, lined enclosure around the transmission. However, it is likely that most of the noise is radiated from the cabin boundary itself, owing to its rather solid connection to the transmission. The best solution is noise-resistant initial design.

Shaft noise similarly is difficult to isolate. The added radiation from torque fluctuations at twice shaft speed from a nonconstant-velocity universal joint can be reduced by using a constant-velocity joint. Journal bearings can be used in place of ball or roller bearings where operating conditions permit.

If the helicopter has a large electronic load, the cooling fans may contribute excessive noise to a vehicle that otherwise may be quiet. This eventually can be handled by using a centralized cooling system ducted to the electronics. Also, a lower velocity air system can be used.

Absorptive material in the cabin interior can be helpful in the frequency range of 500-4000 Hz, to which speech interference and hearing damage effects are sensitive. Materials using a thin plastic septum over a resilient layer of fine fibers can be tuned for maximum absorption at selected frequencies in this range. Spacing such materials away from a wall improves the low-frequency absorption. Absorptive material is used most efficiently in the vicinity of the crew members to be protected.

Baffles are useful in specific locations to shield a localized source of noise. They should be at least three wavelengths in minimum transverse dimension at the lowest frequency to be shielded, and should be covered with absorptive material. Baffles should be close to either the source or the receiver to increase the size of the shadow zone.

Electronic noise reduction by introducing out-of-phase cancelling noise often has been suggested. This may be effective if a stable pure-tone component is to be removed. Feedback information then can be fed to phase and amplitude controls for cancelling at a given location.

13-3.2.3 Environmental Considerations

Much effort has gone into determining conditions under which people feel comfortable, i.e., will be unaware of their environment. The criteria evolving from this effort are set forth in Ref. 7.

When designing a military system, however, the main focus is not on comfort but on performance. Environmental design criteria should define the points to which human performance begins to deteriorate, rather than the range of comfort. Thermal conditions and atmospheric contaminants are primary considerations in the designer's environmental consideration.

13-3.2.3.1 Thermal Stress

Human comfort and ability to perform are functions of the heat balance maintained within the body. Heat balance is defined as that condition existing when the heat generated by the body is equal to the heat passing outward through the skin. The heat balance equation can be stated as follows:

$$S = M - (R + C + E) + Q \quad \text{, Btu/hr} \qquad (13\text{-}5)$$

where

S = rate of storage of heat in the body, Btu/hr
M = rate of metabolic heat production, Btu/hr
R = rate of heat loss from radiation, Btu/hr
C = rate of heat loss from convection, Btu/hr
E = rate of heat loss from evaporation, Btu/hr
Q = rate of heat gain from surroundings, Btu/hr

If heat is lost more slowly than it is being produced, the storage heat S will increase and the body's internal temperature also will increase. If the physiological processes (e.g., perspiring) cannot compensate for the rise in internal temperature, the body temperature will continue to rise, ultimately causing a heat stroke. Conversely, where heat production is less than heat loss, S will decrease and body temperature will decrease. If mechanisms for the conservation of heat prove inadequate, the man eventually will freeze to death.

There is ample evidence that performance is related to body heat content. That is, if S can be maintained at or close to zero, performance will not suffer. Where the value of S is either positive or negative, human performance suffers. Also, the longer man is exposed to the extreme condition, the worse his performance becomes.

The transfer of heat through clothing depends upon the thermal qualities of the material. This usually is expressed in "clo" units. One clo is defined as the clothing insulation required to keep a resting-sitting man whose metabolism is 50 kcal/(m^2-hr) indefinitely comfortable in an environment of 70°F with an air movement of 20 fpm and with a relative humidity of less than 50%. This is equal approximately to normal indoor clothing. The insulation of most clothing materials is about 4 clo/in. of thickness.

Because heat transfer to the surface of the body takes place by conduction, convection, evaporation, and radiation, the thermal conditions of a satisfactory environment are determined not only by the temperature, but also by humidity, airflow velocity, temperature, configuration, and distance of objects radiating or absorbing heat in the vicinity of the crew.

If cockpit environments are designed so that crew members can carry out their tasks while wearing heavy clothing (3.8 clo), cockpit temperatures can drop to about 10°F. At these temperatures (and with 3.8 clo protecting them), crew members can be expected not to lose efficiency for approximately an 8-hr period. Protective clothing heavier than 4 clo will not permit the crew sufficient flexibility of movement to carry out their tasks. In high-temperature environments, onset of performance decrement can be delayed through use of water-cooled protective clothing.

There are only two Military Specifications that cover heating and ventilating requirements for new vehicles: MIL-H-18325 and MIL-E-38453.

13-3.2.3.2 Environmental Control System

Unless otherwise stated in the RFP, the cockpit/cabin environmental control system (ECS) must meet the performance requirements for the ambient environments as follows:

1. Heating: Dry bulb temperature = −25°F (−65°F with heating kit)
Humidity 98-100% relative humidity (RH)
Solar radiation 0

2. Cooling:
   a. Condition 1: Dry bulb temperature 120°F
   Humidity 5% RH (Point B on Fig. 13-17)
   Solar radiation 360 Btu/ft^2-hr
   b. Condition 2: Dry bulb temperature 95°F
   Humidity 74% RH (Point A on Fig. 13-17)
   Solar radiation 360 Btu/ft^2-hr

13-3.2.3.2.1 Heating Requirements

Unless otherwise specified in the RFP, the ECS *shall* be capable of maintaining a cockpit/cabin tem-

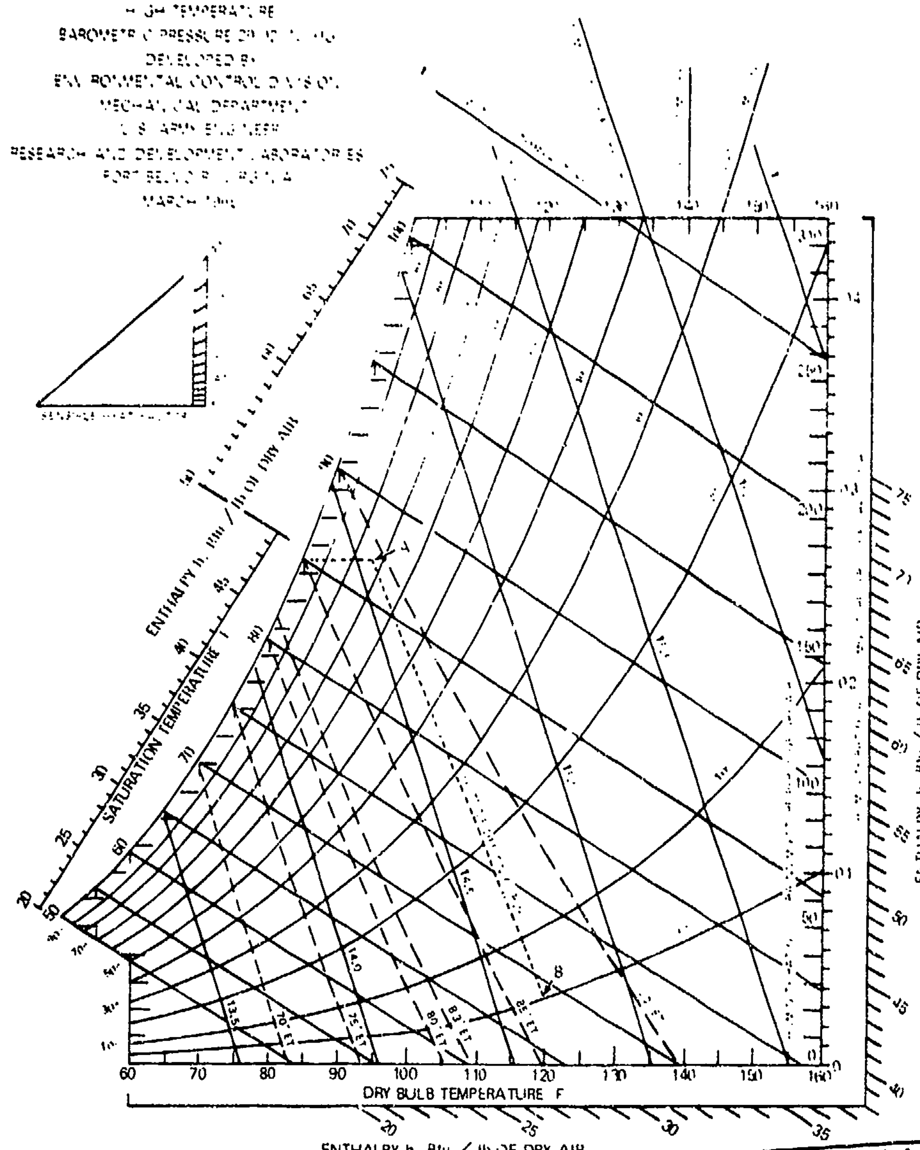


Fig. 13-17. U.S. Army Psychrometric Chart

perature of 40°F. Temperatures measured at head, foot, and waist levels at any occupant position *shall* not vary more than 10°F. For extreme cold-weather operation (−65°F) a heater kit may be used to provide the required cockpit/cabin temperature.

13-3.2.3.2.2 Cooling Requirements

ECS cooling requirements for Army helicopters are defined in terms of Wet-Bulb Globe Temperature (WBGT). WBGT index is defined as follows:

$$WBGT = 0.7 \times \text{wet bulb temperature} + 0.2 \times \text{globe temperature} + 0.1 \times \text{dry bulb temperature}, \ ^\circ F \quad (13\text{-}6)$$

The detailed procedures for measuring WBGT are contained in Appendix I of TB MED 175. The specific ECS cooling requirements are as follows.

For ambient conditions defined in par. 13-3.2.3.2 (cooling), the ECS shall decrease the cockpit/cabin WBGT to 85°F or less within 1 min after takeoff and maintain this condition throughout all representative mission flight profiles. (A two-minute ground run prior to takeoff may be assumed.) For ambient conditions defined in par. 13-3.2.3.2 (cooling), the ECS *shall* decrease the cockpit/cabin WBGT to 88°F within 5 min for ground operation with all doors and windows closed and aircraft engines (except an APU) operating at no higher than the FLIGHT IDLE throttle setting.

13-3.2.3.2.3 Ventilation Requirements

An alternate air ventilation system will be installed to provide outside ambient air to each crew member as a secondary system in the event the primary air conditioning system fails. The ventilation system should meet the following design criteria:

1. It should distribute air uniformly over each crew member from 0 to a maximum airflow within a range of 300-800 linear fpm. The maximum ventilation airflow must be obtainable while the aircraft is flying at maximum endurance speed.

2. The source of the air *shall* be outside the helicopter, free from possible contamination from the helicopter engine exhaust system, hydraulic and fuel lines, and any other helicopter subsystems.

It should be pointed out that these environmental design criteria are not based upon providing crew comfort, but rather are the maximum acceptable environmental limits before serious crew performance decrements are likely to occur.

13-3.2.3.2.4 Thermal Analysis

When performing a heating, ventilation, or air-conditioning analysis, at least the following heat sources or heat sinks must be taken into account:

1. Metabolic heat given off by the occupants
2. Heat given off by electrical equipment
3. Solar radiation effects
4. External ambient temperature
5. Ventilation rate of fresh air (varies with altitude)
6. Rate of cabin air infiltration
7. Humidity of ambient air
8. Mechanical heat sources (power plants, transmission, etc.)
9. Fan work in ventilation ducts.

Ambient temperature *shall* be determined from the "outside design curve" shown in Fig. 13-17. The ventilating system should be combined with the heating system so that the same fresh air intake, blower, and set of ducts and diffusers are used for both. An example thermal analysis is presented in Chapter 13, AMCP 706-202.

13-3.2.3.3 Atmospheric Contaminants

During the design of the heating and ventilating system, consideration also should be given to possible sources of pollutants in the cabin and cockpit of the vehicle.

Possible sources of contamination are engine exhaust systems, fuel vapors, heater exhaust, hydraulic fluid, oil, overheated insulation materials, and weapon exhaust (gun fire and rocket).

In general, avoidance of pollution problems is a matter of providing pathways for spilled liquids that lead them away from cockpit or cabin areas, locating exhaust ports and air intakes so that uncontaminated air is brought into the vehicle, and providing a good ventilation system so that any contaminants that are present cannot build up concentrations to unacceptable levels.

In general, the current Military Specifications have been found adequate for helicopter use. USAF Specification Bulletin 526 contains a listing of various toxic chemicals and allowable concentrations that are permitted in crew compartments. This bulletin also contains references that describe approved test methods for assaying air samples and addresses from which the references can be obtained. Specific requirements are given by the following Standard and Specifications:

1. MIL-STD-800 describes maximum allowable concentrations and specific ground and flight tests required for carbon monoxide gas.

2. MIL-H-18325 contains requirements for the elimination of fuel vapors and other flammable gases.

3. MIL-E-38453 states that contaminants should be prevented from entering crew-occupied areas or, if such penetration cannot be prevented, then adequate ventilation should be provided.

13-3.2.3.4 Oxygen Requirements

In the past, helicopter missions generally have been limited to flights at altitudes that have not necessitated installation of oxygen equipment. It is felt that no unique helicopter requirements exist and, therefore, standard fixed-wing practices with regard to the use of oxygen are applicable and satisfactory.

AR 95-1 is the governing Army Regulation regarding the use of oxygen and should be used by the designer.

13-4 LIST OF SYMBOLS

B_1 = brightness of the brighter portion, lambert
B_2 = brightness of the less bright portion, lambert
C = rate of heat loss from convection, Btu/hr
E = rate of heat loss from evaporation, Btu/hr
I = sound intensity, W/cm^2
I_o = reference intensity, W/cm^2
L_I = sound intensity level, dB
L_p = sound pressure level, dB
M = rate of metabolic heat production, Btu/hr
p = sound pressure, dyne/cm^2
p_o = reference pressure, dyne/cm^2
Q = rate of heat gain from surroundings, Btu/hr
R = rate of heat loss from radiation, Btu/hr
S = rate of storage of heat in the body, Btu/hr
t = time, hr
t_o = maximum permissible time, hr

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5. SAE ARP 577, *Emergency Placarding-Internal and External*, Society of Automotive Engineers, New York, N.Y., 1959.
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CHAPTER 14

DOCUMENTATION

14-1 INTRODUCTION

Two types of data are required during development of the helicopter. First, the contractor is required to submit data in his response to the solicitation. The procuring activity will evaluate the contractor's proposed design; therefore, engineering data will be required to be submitted with the proposal response. The amount of detail will vary among development programs. The solicitation will state the requirements; however, the contractor should be prepared to substantiate his methodology and design.

The second type of data is the data that will be required to substantiate the contractor's design after the contract has been awarded. The requirements for these data will be listed in the Contract Data Requirements List (CDRL) that will be a part of the final contract. Chapter 4, AMCP 706-203, provides a list of this type of data that may be required.

In the paragraphs that follow, the proposal data requirements are discussed in detail.

14-1.1 COMPETITIONS

General guidance and basic principles for solicitation, proposal, evaluation, and source selection are provided in AMCP 715-3. The primary purposes of the competitive formal evaluation and selection process are to insure (1) that there is an impartial, equitable and comprehensive evaluation of the proposals, and (2) that the contractor or source whose proposal offers optimum satisfaction of the Government's performance, schedule, cost, and other objectives is selected.

14-1.2 SOLICITATIONS

There are three basic types of solicitations: Invitation for Bid (IFB), Request for Proposal (RFP), and Request for Quotation (RFQ). The definitions, uses, and requirements of each may be found in ASPR 3501. Issuance of an IFB, RFP, or RFQ is the Government's first official solicitation.

The bidder's response must answer the solicitation clearly, and as concisely as possible, and backup data must be available as required. Each bidder must answer to the best of his ability. A bidder who has started preliminary design considerably before issuance of a solicitation should be able to respond more intelligently and with more pertinent detail than one who performs his initial preliminary design efforts coincidentally with his response to the solicitation.

The solicitation must be understood thoroughly by the bidder so that his response will be complete enough for an accurate appraisal by skilled military and civilian specialists, yet concise enough to provide a balanced appraisal of technical, cost, and management aspects of the proposal. The paragraphs that follow discuss the types of information that should be developed during preliminary design and should be included—if not formally, at least in essence—in the bidder's proposal.

The data and documentation discussed in the remainder of this chapter represent typical engineering requirements during the preliminary design phase of a new development program. The Army's general data and documentation needs during the detail design and qualification assurance phases are defined in Chapter 4, AMCP 706-203. For programs involving minor or major modification of a previously qualified helicopter, these requirements may be tailored to include only significant changes.

For unsolicited proposals and any other presentations for which a solicitation has not been issued, sufficient information to allow the Army to determine possible initial interest should be provided. (The data for full preliminary design or contract definition are deferred until positive interest is shown and mutual agreement on a desired program is reached.) For such proposals, the importance of the various data submitted depends upon the particular design being presented. As a minimum, however, the data described in pars. 14-3.1, 14-3.3, 14-3.4, 14-3.5, 14-3.7, 14-4.1, and 14-4.2 are desirable.

14-2 DATA REQUIREMENTS

Data and documentation *shall* be submitted at the completion of the preliminary design phase in accordance with the requirements listed subsequently (summarized in Table 14-1), except as modified or amplified in a solicitation.

The proposal should describe the helicopter configuration in sufficient detail to evaluate the ability of the proposed system to meet the Army's objectives. If significant items are not covered by the specific data listed, additional reports, drawings, or schematic diagrams may be submitted.

All proposed deviations from solicitation requirements *shall* be indicated in the pertinent paragraphs of the proposal documents. A separate summary listing of, and justification for, such deviations must be included in each of the proposal reports subsequently described. A statement must be made that the listing contains all of the bidder's proposed deviations from the provisions of the solicitation and other applicable documents.

TABLE 14-1
SUMMARY: ENGINEERING INFORMATION REQUIREMENTS

ITEM	
14-3	REPORTS, PLANS, AND MISC DATA
14-3.1	PRIME ITEM DEVELOPMENT SPECIFICATION (PIDS)*
14-3.2	AIRWORTHINESS QUALIFICATION SPECIFICATION (AQS)*
14-3.3	WEIGHT AND BALANCE REPORT*
14-3.4	PERFORMANCE DATA REPORT*
14-3.5	STABILITY AND CONTROL REPORT*
14-3.6	STRUCTURAL DESCRIPTION AND DESIGN DATA REPORT
14-3.7	POWER PLANT ANALYSIS REPORT
14-3.8	POWER PLANT INSTALLATION AND FUEL AND LUBRICATION SYSTEM REPORT
14-3.9	FLIGHT CONTROL SYSTEM REPORT
14-3.10	LANDING GEAR SYSTEM REPORT
14-3.11	HYDRAULIC AND PNEUMATIC SYSTEM REPT
14-3.12	ELECTRICAL SYSTEM REPORT
14-3.13	ELECTROMECHANICAL ACTUATION SYSTEM REPORT
14-3.14	ELECTRONIC SYSTEM REPORT
14-3.15	FLUIDIC SYSTEM REPORT
14-3.16	ENVIRONMENTAL CONTROL SYSTEM (ECS) AND CREW SERVICE REPORT
14-3.17	ARMAMENT SYSTEM REPORT
14-3.18	HYDRODYNAMIC REPORT
14-3.19	ESCAPE SYSTEM ANALYSIS REPORT
14-3.20	HUMAN FACTORS ENGINEERING REPORT AND PLAN
14-3.21	SURVIVABILITY VULNERABILITY (S V) REPORT AND PLAN
14-3.22	ELECTROMAGNETIC COMPATIBILITY (EMC) CONTROL PLAN
14-3.23	SYSTEM SAFETY PROGRAM PLAN
14-3.24	RELIABILITY PROGRAM PLAN
14-3.25	MAINTAINABILITY PROGRAM PLAN
14-3.26	PRODUCIBILITY PLAN
14-4	DRAWINGS SKETCHES, AND SCHEMATIC DIAGRAMS
14-4.1	GENERAL ARRANGEMENT DRAWING AND SKETCH*
14-4.2	INBOARD PROFILE DRAWING*
14-4.3	STRUCTURAL ARRANGEMENT DRAWINGS
14-4.4	POWER PLANT INSTALLATION DRAWING
14-4.5	FLIGHT CONTROL SYSTEM DRAWINGS
14-4.6	LANDING, BEACHING, AND AUXILIARY GEAR DRAWINGS
14-4.7	HYDRAULIC AND PNEUMATIC SYSTEM SCHEMATIC DIAGRAMS
14-4.8	ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC DIAGRAMS
14-4.9	CREW SERVICE DRAWING
14-4.10	ELECTRONIC EQUIPMENT DRAWING
14-4.11	INSTRUMENT PANEL ARRANGEMENT DWG
14-4.12	INSTRUMENT SYSTEM INSTALLATION DWGS
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14-4.14	ANTI-ICING SYSTEM SKETCH
14-4.15	ARMAMENT INSTALLATION DRAWINGS
14-4.16	CARGO HANDLING ARRANGEMENT DRAWING
14-4.17	TOWING, JACKING, MOORING, AND TIEDOWN SKETCH
14-4.18	ESCAPE SYSTEM DRAWING

*DATA NORMALLY REQUIRED FOR INITIAL SUBMITTAL OF UNSOLICITED PROPOSALS, INCLUDING 10 COPIES OF THE PIDS AND TWO COPIES OF OTHER REPORTS AND DRAWINGS INDICATED BY (*)

14-3 REPORTS, PLANS, AND MISCELLANEOUS DATA

14-3.1 PRIME ITEM DEVELOPMENT SPECIFICATION

The bidder's proposed Prime Item Development Specification (PIDS) *shall* delineate, in the format of the type specification furnished in the solicitation, all applicable requirements, design criteria, and characteristics of the bidder's proposed overall helicopter system, subsystems, and associated equipment. The PIDS *shall* identify all Government-furnished Equipment (GFE) upon which the proposal is based, including quantity per helicopter, designations, and unit weight.

14-3.2 AIRWORTHINESS QUALIFICATION SPECIFICATION

This document, submitted by the bidder, *shall* contain sufficient qualification assurance data to define for contractual purposes the proposed qualification assurance program. Chapter 2, AMCP 706-203, describes the requirements for this specification. The contemplated analyses; investigations; airframe and lighting mock-ups; reviews; wind tunnel, component, structural and fatigue articles and testing; and all applicable formal demonstrations of the total system and subsystems, based upon the requirement of AMCP 706-203 as amplified and amended by the solicitation, *shall* be included.

14-3.3 WEIGHT AND BALANCE REPORT

This report *shall* be in accordance with the requirements of MIL-W-25140 for an "Estimated Weight Report". The summary weight statement defined in Part I of MIL-STD-1374 *shall* be complete, including the "Dimensions and Structural Data". The detailed weight statement described in Part II of MIL-STD-1374 *shall* have sufficient data to show that all items of the power plant systems and equipment groups have been included. Justification of estimates *shall* be submitted, including calculations; formulas; basic curves with model designations for plotted points; allowances for special features and items not included in values obtained from basic curves; comparative tabulations and extrapolations from weight data of actual helicopters; or other means used for deriving the estimated weights of the structural, power plant, and equipment groups.

In addition, the bidder *shall* submit a weight control plan that outlines briefly and discusses the major features of his proposed weight control management program to implement the objectives and requirements of MIL-W-25140 and MIL-STD-1374. The plan *shall* reflect the proposed weight control staff and organization; their roles and responsibilities; and their relationships to design groups and monitoring and review levels throughout the layout, optimization, selection of material and processes, drawing release, manufacture, qualification assurance, and production phases. The plan also *shall* propose weight control procedures for GFE associate relationships, where appropriate, for subcontractor efforts and for purchased equipment; general nature of other techniques, such as analysis and testing for weight optimization, stimulation of weight consciousness among design groups, and establishment of value per pound criteria; and procedures for monitoring weight targets and overall status.

14-3.4 PERFORMANCE DATA REPORT

This report *shall* be in accordance with the requirements of MIL-C-5011, and *shall* include a set of Standard Aircraft Characteristics and Performance Charts (proposal category). A Characteristic Summary Chart is not required. The report *shall* include the performance values that the bidder is willing to guarantee. The performance data *shall* be based only upon power plant ratings obtained from approved model specifications or from other engine data that have been approved by the Army, and losses from the power plant analysis report (par. 14-3.7). Analyses *shall* include the derivation or establishment of all aerodynamic data used in calculating helicopter performance. The basis for establishment of these data *shall* be documented and referenced completely. If the performance is based upon wind tunnel or flight test data, the procedures followed in adjusting these test data for differences such as configuration, Reynolds number, surface conditions, ground effects, and vertical drag regimes *shall* be explained fully. The report *shall* provide a complete description of the helicopter aerodynamics and performance, including:

1. Geometry of rotors, rotor blades, wings, and control surfaces—such as areas, radii/spans, chords, hinge lines and offsets, twists, and tapers

2. Detailed description of all airfoil sections, including coordinates, leading edge radius, amount of camber, location of maximum camber, and location of maximum thickness, for other than identifiable standard airfoils

3. Wetted areas of all components and total helicopter

4. For helicopters carrying external ordnance,

pods, fuel tanks, or range extension kits, dimensional scale drawings adequate for use in the preparation of drag estimates for these items

5. All applicable performance data in accordance with USAAVSCOM ADS-10

6. For helicopters with hulls, or for those capable of alternate configuration with floats, analyses showing the derivation of aerodynamic and hydrodynamic data used in the performance calculations, including water resistance versus velocity and takeoff time and distance versus velocity, based upon characteristics of floats or hulls as defined in the Hydrodynamic Report (par. 14-3.18).

14-3.5 STABILITY AND CONTROL REPORT

This report *shall* provide adequate data to substantiate that the proposed helicopter will meet the flying quality requirements, and must provide the background information concerning the evaluation of the proposed configuration from a stability and control viewpoint. The stability and control computer simulations *shall* contain at least the following information and data:

1. Quantitative data showing compliance with the flying quality requirements of Chapter 6 of this document and of AMCP 706-202, as amplified or amended by the solicitation, at normal and maximum alternate gross weights; at most critical forward and aft CG locations; with and without stability augmentation; and over the applicable speed range

2. V-n diagrams for sea level and best cruise altitude for normal and maximum alternate gross weights, within which the bidder warrants that the helicopter will be suitable operationally and free of undesirable blade stall effects such as noticeable loss of control, erratic changes in flight characteristics, excessive control loads, vibration, or other limiting characteristics

3. Graphs for all flight conditions, speeds, altitudes, and configurations defined in the flying quality requirements. The graphs *shall* show variations of the stability and control derivatives over the speed and angle-of-attack ranges for which the helicopter is to be designed, including effects of aeroelasticity, power, and significant external store arrangements, as applicable.

4. Significant flight control system characteristics and control kinematics, such as control rates; feel forces; gearings; control system schematics; longitudinal, lateral, and collective control travels and cyclic/collective pitch ranges at 0.75R; directional control travel and tail rotor pitch range at 0.75R

5. Special features programmed as functions of configuration, angle of attack, speed, dynamic pressure, etc., to improve or alter flying qualities

6. All applicable stability and control data in accordance with USAAVSCOM ADS-10

7. Minimum, normal, and maximum alternate gross weights, CGs and longitudinal and lateral CG limits measured from the centerline of the rotor shaft, and moment-of-inertia data as required for flight characteristic calculations, including the effects of various significant external store loadings. The source or derivation of weight and CG data must be referenced or furnished.

8. A summary table of nondimensional stability and control derivatives, referenced to vehicle body axes, and flight control system characteristics for takeoff, climb/cruise, approach/descent, and dive conditions for several significant altitudes, including the effects of significant external load and/or pod arrangements. Data *shall* be furnished for forward, rearward, and sideward flight as applicable, including the effects of significant external load configurations.

9. Discussion of major problem areas, their impact upon the final configuration, and information showing why certain design approaches were abandoned or pursued

10. Information showing the source of the data presented in Item 3. When these data are obtained directly from wind tunnel tests, the actual data *shall* be provided as appropriate. When the wind tunnel data are obtained on a model similar to the proposed design, the method of converting the data to the proposed configuration *shall* be shown. Supporting data calculated theoretically or on the basis of semi-empirical methods must be discussed clearly and referenced. If bidder-generated references have not been furnished to the procuring activity already, one copy of such referenced material must be submitted.

14-3.6 STRUCTURAL DESCRIPTION AND DESIGN DATA REPORT

This report *shall* contain all required structural design and analysis data, including that for the wing group, tail group, body group, power plant cowling, engine mount, rotor system, transmission installation, landing gear, operation of the control system, external hard points, and cargo-carrying and restraining arrangements on cargo floors. Structural arrangement drawings required by par. 14-4 may be referenced instead of being included in this report. The location, skin-panel size and material gage, and material condition of the skin, stringers, longerons, ribs, formers, frames, and spars *shall* be indicated. Typical structural design data and design criteria to be included *shall* meet the pertinent requirements of this document, as amplified or amended by the solicitation, as follows:

1. A summary of gross weights used for structural design, an envelope of design and operational gross weights versus CG limits, and applicable data concerning mass moment of inertia

2. Flight load design criteria, including a summary of all related speeds, as defined in par. 4-2 of this document, and their derivation; altitude-Mach number diagrams; V-n diagrams including gust curves and buffet boundaries; criteria for carrying external stores in symmetrical and asymmetrical flight, at weights up to and including the maximum alternate gross weight for takeoff with maximum internal and maximum external loads; and flight load conditions proposed for design of the various loading configurations. Any limitations restricting takeoff and flight with all store stations loaded to capacity in combination with maximum internal fuel *shall* be stated explicitly. Data shall show that release of any combination of stores, including salvo release of total helicopter capacity store load, does not produce rigid body or elastic responses that result in exceeding specified limit strength. The flight criteria also *shall* include airspeed-altitude envelopes and a summary of gust, symmetrical flight, asymmetrical flight, and autorotative flight conditions proposed for design, with related speeds and load factors

3. Crash load design criteria, including a summary of crash conditions and the design approach used to comply with these conditions

4. Ground and water load, as applicable: design criteria for landing, ground handling, tiedown, and miscellaneous conditions; the proposed landing design load conditions; and the derivation of the loads and related speeds

5. An evaluation of the helicopter relative to flutter, divergence, mechanical instability, other elastic and aeroelastic instabilities, vibration (including vibratory levels at all personnel stations), and sonic effects

6. Repeated load design criteria including normal flight spectrum, maneuver load spectrum, gust load spectrum, landing load spectrum, and other typical repeated load spectra consistent with the expected environmental history of the helicopter; and substantiating data for selection of loads

7. Stress analysis data consisting of a brief, preliminary stress analysis indicating sizes, loads in members, and stresses in the major structural components

8. Special weapon criteria

9. Description of the nature and extent of proposed compliance with fail-safe design requirements from a structural standpoint

10. Description of components, parts, or elements of the airframe that must be replaced, with stated intervals for these replacements, in meeting the repeated load and fatigue design and construction criteria.

14-3.7 POWER PLANT ANALYSIS REPORT

This report *shall* describe the derivation of available power for use in calculating helicopter performance. All power losses and the effects of speed, altitude, and temperature thereon *shall* be shown. The power losses *shall* include those due to the induction system (including induction system drag), airframe, oil temperature system, exhaust system, IR suppression (if applicable), air bleed and power extractions, transmission, gearboxes, propellers (including slipstream and blockage drag), and the effects of fuel temperature extremes. A thorough analysis showing the derivation of these losses also *shall* be furnished. For turbojet performance or turboprop engines, refer to MIL-D-17984 for installed engine performance and air induction system data presentation requirements.

14-3.8 POWER PLANT INSTALLATION AND FUEL AND LUBRICATION SYSTEM REPORT

This report *shall* describe the power plant installation; the power transmission system, including shafting, gearboxes, couplings, and free-wheeling units; and the cooling, inlet, exhaust, engine control, engine starting, engine lubrication, and fuel systems. The report *shall* include the items that follow:

1. Engine installation features, mountings, and accessibility data, including engine accessories and contractor-furnished airframe accessories that supplement the engine and transmission system installations

2. Details and schematics of engine installation features such as anti-icing, use of engine bleed air, engine and rotor and propeller control systems, intake and exhaust systems, engine and accessory cooling systems, special engine configurations, fire detection and suppression, IR suppression, and all other applicable systems. Bleed air system descriptions *shall* include a brief load analysis of available bleed air and the demands of systems using bleed air for appropriate configurations and flight conditions

2. A listing of all connections (fuel, oil, hydraulic,

drain, electrical, pneumatic, mechanical, etc.) that must be disconnected for engine removal

4. Description of the starting system with explanation of the functions of the various components, identification of the starter, external power requirements, estimated starting time, auxiliary power unit installation, etc.

5. Complete fuel system schematic diagram and lubrication system schematic diagram with detailed descriptions of both systems

6. Calculation of fatigue and ultimate stresses of drive shafting, gear shafting, and main rotor shaft. Transmission gear stresses and bearing life calculations also *shall* be included.

14-3.9 FLIGHT CONTROL SYSTEM REPORT

This report *shall* discuss the flight control system design and philosophy, and explain the functions of the components. Simplified schematics *shall* be provided to show actual function of the system and all related components. Pilot control, rotor control, and control surface travels *shall* be shown on the schematics. When automatic stabilization and/or other automatic flight controls are proposed, block diagrams and automatic control system authority *shall* be included.

14-3.10 LANDING GEAR SYSTEM REPORT

This report *shall* describe the landing system, including controls, provisions for emergency operation, steering system, anti-skid brake system, supplemental or alternate deceleration devices, etc. Calculated maximum loadings of wheels, tires, and brakes produced by the most critical operating conditions must be presented, and the adequacy of the proposed sizes must be substantiated for static load, fatigue life, and soft-ground characteristics. Load stroke and tire deflection data for both the main and auxiliary gear *shall* be shown. For skid, ski, bear paw, and beaching gear, pertinent data *shall* be presented to substantiate the adequacy of soft-ground flotation characteristics and the size of ground-handling wheels.

14-3.11 HYDRAULIC AND PNEUMATIC SYSTEM REPORT

This report *shall* specify the type and class of the systems, as defined by MIL-H-5440, MIL-H-8891 and/or MIL-P-5518, describe the functions and operation of all major components with reference to schematic diagrams, and provide substantiation that the systems are adequate to provide the required services.

14-3.12 ELECTRICAL SYSTEM REPORT

This report *shall* present a preliminary electrical load analysis substantially in accordance with MIL-E-7016. The report *shall* contain a block diagram of the electrical power and distribution systems, showing the main components and listing component ratings, weights, and locations. The report *shall* include an analysis stating the reasons for selecting the proposed power, conversion, distribution, and lighting systems and components. If constant-speed drives are proposed, such units and their supporting components (remotely mounted controls, oil coolers, reservoirs, filters, etc.) *shall* be included in the block diagram and analysis portions of the report. The report also *shall* present information on proposed high-density harness construction, if used.

14-3.13 ELECTROMECHANICAL ACTUATION SYSTEM REPORT

This report *shall* delineate the purpose, principles of operation, and control relationships of each electromechanical actuation system. Environmental parameters and schematic diagrams of mechanical linkages and electrical circuits *shall* be included.

14-3.14 ELECTRONIC SYSTEM REPORT

This report *shall* describe and summarize the electronic system as a whole, including Government-furnished equipment. The following items shall be included: system radio interference problems, heat transfer or cooling problems, moisture problems arising from the cooling system, and the proposed installation configuration, arrangement, and operation. In addition, this report *shall* detail the equipment to be designed and furnished as contractor-furnished equipment (CFE) and *shall* summarize design rationale, performance requirements, reliability, maintainability, microelectronics to be used, type of cooling, antennas, etc. If contractor-furnished or contractor-modified Government-furnished items are proposed, their physical characteristics *shall* be presented.

14-3.15 FLUIDIC SYSTEM REPORT

This report *shall* describe the technology, functions, and operation of proposed fluidic systems, including sketches and block or schematic diagrams as necessary; and provide substantiation that the system is adequate to provide the required services.

14-3.16 ENVIRONMENTAL CONTROL SYSTEM (ECS) AND CREW SERVICE REPORT

This report *shall* describe the environmental control system for maintaining the crew and equipment environments at desired levels, including services such as oxygen, emergency oxygen, suit pressurization, ventilation, anti-blackout air, and communications. Analyses *shall* substantiate the adequacy of the proposed systems to provide the required services. Capacities, flow rates, temperatures, pressures, and ambient and local conditions *shall* be considered in the analyses.

14-3.17 ARMAMENT SYSTEM REPORT

This report, in conjunction with the armament installation drawings described in par. 14-4.15, *shall* provide a unified and coherent presentation of the proposed armament system. The report *shall* discuss the reasons for the particular disposition of guns, bomb racks, rocket and missile launchers, and special armament equipment. The armament control system *shall* be described with respect to the helicopter mission and human factors engineering. A discussion of the rearming capabilities of the helicopter *shall* be included to illustrate the loading time, effort, and personnel required. The anticipated armament maintenance procedure, from the standpoint of accessibility and durability, *shall* be described.

14-3.18 HYDRODYNAMIC REPORT

This report, required only for helicopters to be equipped with floats or hulls, *shall* include the following:

1. Float or hull line drawing (half sections and profile), and hydroski, hydrofoil, and supporting strut drawings, if appropriate
2. Static water line and static stability data
3. Estimated stability and control characteristics (longitudinal, directional, and lateral)
4. Spray characteristics
5. Takeoff time and distance as derived in the performance data report (par. 14-3.4)
6. Water resistance curve as derived in the performance data report (par. 14-3.4)
7. Behavior in design sea state.

14-3.19 ESCAPE SYSTEM ANALYSIS REPORT

This report *shall* present the results of an analysis of the escape system; it *shall* describe the system capabilities as required for the helicopter under flight conditions throughout the performance envelope (underwater escape, ditching, etc.). A system capability analysis *shall* be based upon the use of the specified anthropometric body size ranges and required personnel support equipment. If applicable, the components that comprise the ejection system and the operational sequence, including sequencing of multiple ejections, also *shall* be described.

14-3.20 HUMAN FACTORS ENGINEERING REPORT AND PLAN

The results of the bidder's preliminary analyses, reflected in the proposed configuration, *shall* be described in the human factors engineering report. This study *shall* identify the operation of all aircraft subsystems in sufficient detail to identify the function of all controls and displays, the interfaces between related subsystems, and crew coordination requirements.

In the human factors engineering plan, the bidder *shall* present his proposed program for ongoing human factors engineering efforts to implement the requirements of MIL-H-46855 and this chapter during the development program.

14-3.21 SURVIVABILITY/VULNERABILITY (S/V) REPORT AND PLAN

The S/V report submitted in response to a Government solicitation *shall* include, as a minimum, the following data:

1. Vulnerable area tabulation by type of kills for the specified threats (six to ten threat directions), with and without armor installed
2. Scale drawings showing major components critical to S/V, armor, routing of hydraulic lines and major electrical buses

3. Description of all proposed protective armor (including fixed, flexible, and transparent)

4. Description of emergency lubrication provisions

5. Description of the IR suppression system, including the effectiveness of the system and associated weight and power/performance penalties

6. Preliminary acoustic detection analysis and predicted detectability distances (polar plot, or equivalent)

7. Description of design features that minimize radar cross section

8. Description of design features that enhance crash survivability and minimize the probability of post-crash fires.

A S/V Program Plan *shall* be submitted in accordance with the requirements of USAAVSCOM ADS-11.

14-3.22 ELECTROMAGNETIC COMPATIBILITY (EMC) CONTROL PLAN

In this plan the bidder *shall* discuss and present his proposed program for controlling the electromagnetic environment, and the testing of all components and subsystems contributing to the total electromagnetic environment, in accordance with the requirements of par. 9-11, AMCP 706-203.

14-3.23 SYSTEM SAFETY PROGRAM PLAN

In this plan the bidder *shall* present a system safety program plan that will implement the requirements of MIL-STD-882 and Chapter 3, AMCP 706-203, during the development phase. This plan *shall* be prepared in accordance with Ref. 1.

14-3.24 RELIABILITY PROGRAM PLAN

In this plan the bidder *shall* discuss and present his reliability program proposed to implement the requirements of MIL-STD-785, Chapter 12 of this volume, and par. 10-1 of AMCP 706-203 during the development phase. The areas to be covered include allocations; predictions; design reviews; specification reviews; vendor and subcontractor controls; manufacturing controls; individual tests; flight tests; failure reporting, analysis, and corrective action procedures; training plans; reports; and schedules proposed by the bidder to meet the design goals. The proposed methods of demonstrating reliability at the "black box" level and at the system level *shall* be described. The report also *shall* include an analysis of the effectiveness of the helicopter weapon system and the operational model used in making the analysis. Block diagrams for reliability, maintainability, logistics, and manpower of the operational model *shall* be included. Input information for effectiveness evaluation *shall* be included and must be capable of being used on a per-sortie basis, as well as on a multiple—or continuous operation—basis.

Minimum acceptable reliability levels for the overall system, subsystems, or components, and prediction and allocation calculations required by the RFP and the specifications or documents provided therewith, *shall* be summarized in an appendix to the plan.

14-3.25 MAINTAINABILITY PROGRAM PLAN

The bidder *shall* present his proposed maintainability program to implement the requirements of MIL-STD-470 and MIL-STD-471, Chapter 11 of this document, and par. 10-2, AMCP 706-203, during the development program. Minimum acceptable maintainability levels and maximum times between overhauls for the system, subsystems, or components, as required by the RFP and the specifications or documents provided therewith, *shall* be summarized in an appendix to the plan. A separate section of the plan *shall* describe any special tools and ground support equipment anticipated or special personnel skills envisaged for maintenance of the helicopter.

14-3.26 PRODUCIBILITY PLAN

A producibility engineering plan *shall* be prepared and submitted to assure the compatibility of component and assembly design with selected manufacturing materials, processes, and assembly techniques. It should address the following:

1. The techniques employed in the engineering design phase in consideration of the eventual adaptation to production

2. Program management authority, policies, responsibilities, and controls for implementation of producibility decisions or determination

3. Program trade-offs in terms of weight, equipment selection, manufacturing and assembly techniques, and production costs

4. Producibility engineering controls, which, as a minimum, should include:

a. Design controls

b. Critical engineering terms
c. Quality control and inspection

5. Coordination of producibility engineering requirements
6. Engineering and manufacturing interfaces
7. Other producibility engineering interfaces.

14-4 DRAWINGS, SKETCHES, AND SCHEMATIC DIAGRAMS

The drawings listed *shall* be provided, as applicable. All major items and features *shall* be shown, and means of access to items requiring adjustment or removal presented. The scale must be indicated on all drawings. These drawings *shall* not be reduced in size. The sketches and schematic diagrams should be larger than report or foldout page size, for detail and clarity of information, and may be referenced in the applicable reports of par. 14-3 in which they are discussed. Where necessary for clarity, similar size schematic or block diagrams may be substituted for those already called out in the individual report requirements in par. 14-3.

14-4.1 GENERAL ARRANGEMENT DRAWING AND SKETCH

The general arrangement drawing *shall* be prepared in accordance with the requirements in Appendix A of Chapter 4, AMCP 706-203.

The general arrangement sketch *shall* be a simple, three-view drawing, of report page size, for general reference purposes and for use in the PIDS described in par. 14-3.1. Dimensions (in feet) *shall* include rotor(s) diameter; wing span, including wing folded/-swept, if applicable; tail span; height of rotor, tail and fuselage; tread; and wheel base.

14-4.2 INBOARD PROFILE DRAWING

This drawing *shall* show major structure, power plant, drive system and gearboxes, equipment, armament, useful load items, entrances, emergency exits, escape hatches, location of the crew and their equipment, major instrument sensors, computers, gyro platforms, automatic flight control components, etc. The drawing *shall* include an elevation and a plan view, and *shall* have numerous sectional views to show the locations of the items of equipment. Each item *shall* be labeled plainly. The information shown *shall* agree with the weight and balance data submitted, and the reference data for the longitudinal and vertical CG information used in the Weight and Balance Report *shall* be shown. The scale of this drawing *shall* be one-tenth unless otherwise specified in the solicitation.

14-4.3 STRUCTURAL ARRANGEMENT DRAWINGS

These drawings *shall* show the essential characteristics of the structure of the major components, including the wing, body, hull, floats, empennage, and rotor systems, as applicable. Centerlines of the main members and sections necessary to clarify the arrangement *shall* be included. Solutions to difficult structural problems such as major discontinuities *shall* be indicated. Wing folding, blade folding, and/or any other structural part of the helicopter that is folded for storage (or flight) *shall* be shown in detail, with a description of the operation and locking provisions. Details of the locks and lock controls also *shall* be shown.

14-4.4 POWER PLANT INSTALLATION DRAWING

This drawing *shall* indicate the complete propulsion system that include engine(s); fire protection provisions; shrouds; fuel system (including fueling, defueling, fuel jettisoning, and venting arrangements); oil system; induction system; exhaust system; cooling system; water injection system; starter and starter system components; auxiliary power unit; and power transmission system, including clutches, brakes, and gearboxes for rotor drives and accessory drives. Access, servicing, and removal provisions *shall* be shown.

14-4.5 FLIGHT CONTROL SYSTEM DRAWINGS

These drawings *shall* present the complete flight control system as it will appear in the helicopter. The drawings *shall* depict the pilot's controls, cable runs, pulleys, bellcranks, pushrods, idlers, torque tubes, guards, etc., and their locations in the airframe. All components of the automatic flight control system *shall* be shown.

14-4.6 LANDING, BEACHING, AND AUXILIARY GEAR DRAWINGS

These drawings *shall* show the structural arrangement of the main, auxiliary, tail skid or bumper, and beaching gear; and of the retracting mechanism and locks, if applicable. The length of stroke for shock struts *shall* be noted, with dimensions shown for the landing gear in fully extended, static, and fully compressed positions. Data *shall* be provided, as applicable, for skids, skis, and bear paws.

14-4.7 HYDRAULIC AND PNEUMATIC SYSTEM SCHEMATIC DIAGRAMS

On these diagrams, the pneumatic systems *shall* be presented separately or on separate diagrams. The flight control and utility systems, when completely independent of each other, also *shall* be shown separately. Where practicable, line diagrams of mechanical or electrical sequencing or other functions *shall* be presented.

14-4.8 ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC DIAGRAMS

The pressurization and air-conditioning systems of occupied spaces and of equipment spaces *shall* be shown on a schematic layout in sufficient detail to allow system operation analysis. Portions of the system that are included with individual related systems, such as radar pressurization, may be omitted.

14-4.9 CREW SERVICE DRAWING

This drawing *shall* present crew services and supplies including oxygen, emergency oxygen, suit pressurization, ventilation, anti-blackout provisions, and communications.

Tubing, disconnects, methods of routing, and attachments to the helicopter structure *shall* be included. Access and provisions for servicing and removal of equipment *shall* be indicated. Similar data *shall* be shown for each crew member, as applicable.

14-4.10 ELECTRONIC EQUIPMENT DRAWING

This drawing *shall* show the location, size, and identity of each component of electronic equipment including antennas, radomes, and all other units carried externally. Cooling provisions and accessibility provisions also *shall* be indicated.

14-4.11 INSTRUMENT PANEL ARRANGEMENT DRAWING

This drawing *shall* show the arrangements and locations of all instruments on the pilot's, copilot's, navigator's, and other crew members' panels, as applicable.

14-4.12 INSTRUMENT SYSTEM INSTALLATION DRAWINGS

These drawings *shall* show the location and identity of components of instrument systems such as pitot-static systems and fuel quantity gaging systems. Wiring and tubing connections for such systems also *shall* be shown.

14-4.13 EXTERIOR LIGHTING SYSTEM DRAWING

This drawing *shall* show the location and candlepower for each exterior light assembly. The angles of obstructed visibility of the anticollision light(s) *shall* be included. Special attention *shall* be given to the light locations to prevent situations detrimental to the crew's vision.

14-4.14 ANTI-ICING SYSTEM SKETCH

This sketch *shall* show the area of the wings, fuselage, empennage, enclosure bubbles, canopies, rotor systems, radome, power plant air induction systems, and heating and ventilating systems, as applicable, to be protected against ice accretions. The type of ice protection system selected for each of these areas *shall* be indicated.

14-4.15 ARMAMENT INSTALLATION DRAWINGS

These drawings *shall* present the armament installations including fixed guns, flexible guns and turrets (with effective-field-of-fire diagrams), bombs, torpedoes, rockets, guided missiles, and locations of displays and controls. If contractor-furnished or contractor-modified Government-furnished items of equipment are proposed, the characteristics of those items *shall* be presented. Principal clearances in critical helicopter attitudes and locating dimensions *shall* be shown, as well as the location of fire control equipment with optical and radar field of action diagrams. The methods and equipment, both primary and backup, to be used in arming the helicopter *shall* be shown, particularly new or unique equipment and/or techniques.

14-4.16 CARGO-HANDLING ARRANGEMENT DRAWING

This drawing *shall* indicate the internal and external cargo-handling and restraint features and installation, including loading and unloading equipment, rails, ramps, special or roller floors, tiedown restraint patterns, external slings, cargo hooks, winches, controls, and any hard points for external carriage.

14-4.17 TOWING, JACKING, MOORING, AND TIEDOWN SKETCH

This sketch *shall* present the towing, jacking, mooring, and tiedown provisions for the helicopter. For helicopters equipped with hulls or floats, the anchor and mooring features while waterborne *shall* be shown.

14-4.18 ESCAPE SYSTEM DRAWING

This drawing, if applicable, *shall* show the configuration of the system, means of actuation, location of seats and components; a schematic diagram of the ejection seat clearance also is required.

REFERENCE

1. USAAAVS TR 72-8, *Preparation of a System Safety Program Plan for Aviation Systems Development*, US Army Agency for Aviation Safety, Fort Rucker, Ala, July 1972.

GLOSSARY

Absolute altitude. The measured distance above a ground reference, as would be recorded by a radar altimeter. Also called tapeline altitude.

Acceleration:

Angular. The time rate of change in angular velocity

Engine. The time rate of change in gas generator speed

Rotor. The time rate of change in rotor speed.

Advancing mode. Multiblade mode where the oscillation of the *k*th blade, counted in direction of rotation, lags in phase by $2k\pi/n$ as compared to 0th blade.

Air resonance. Same as ground resonance, but with the helicopter in flight, or without weight on landing gear.

Airworthiness. A demonstrated capability of a helicopter or helicopter subsystem or component to function satisfactorily when used within the prescribed limits.

Anthropometry. The study of the dimensions and ranges of motion of the human body and its parts.

Arizona Road Dust. Uniquely defined particulate material compositions used to test hydraulic and pneumatic filters and fuel system components (see MIL-F-5504 and MIL-F-8615).

Articulated rotor. A rotor consisting of a central part rigidly attached to the drive shaft or rotor mast and a number of blades attached to the central part by flapping and lead-lag hinges.

Attitude flare. The second of the three stages of helicopter landing, preceded by approach and followed by deceleration. During attitude flare the rate of descent is reduced and the pitch attitude is increased above the flight trim value.

Availability. The probability that the system is operating satisfactorily at any point in time when used under stated conditions, where the total time considered includes operating time, active repair time, administrative time, and logistic time.

Blade azimuth angle. Position angle of blade in the rotor plane measured from the aft position in the direction of rotor rotation.

Blade chord. Width of the blade: the dimension in the plane of the blade at right angle to the lengthwise axis.

Blade element theory. Rotor analysis procedure in which the rotor blade is considered to be made up of individual blade elements, each of which contributes to performance. Each spanwise element is then considered as a two-dimensional airfoil section, and the thrust and torque are found by integration.

Blade incidence. The angle between the zero-lift line of the blade section and the plane of the rotor disk. For a blade having built-in twist, the blade incidence is measured at a reference section, either root or 0.75R.

Blade pitch. See: *Blade incidence.*

Blade retention system. The mechanism that keeps the blade attached to the hub against centrifugal loading while permitting blade pitch variations.

Blade section. The cross-sectional shape of the blade in a plane perpendicular to its lengthwise axis.

Bode analysis. A method of analyzing the stability of a closed-loop control system by using plots of the response and the phase angle versus frequency to determine both gain and phase margins as measures of system damping characteristics.

Boundary layer. Region of fluid flow adjacent to a solid surface which is influenced in its velocity profile by surface contact and fluid viscosity.

Breakout force. The minimum force required at the cockpit for a pilot to initiate motion at a prescribed control surface.

Camber. Curve of the centerline of the blade section, i.e., the distance between the mean line of an airfoil and the chord line.

Chord line. Line joining the ends of the mean-camber line.

Climb efficiency factor K_p. Multiplier that adjusts theoretical forward rate of climb to account for transmission efficiency, rotor efficiency, and increases in fuselage download. Usually obtained from flight test data.

Collective mode. Multiblade mode with equal phases for all blade oscillations.

Combat ceiling. The maximum altitude, at a given temperature, at which the aircraft can climb at a rate of at least 500 fpm.

Compound helicopter. Vehicle that has a wing to relieve, partially or totally, the rotor of its lifting requirements, and that also has an auxiliary propulsion device to relieve, partially or totally, the rotor of its propulsion requirements.

Confidence level. A concept of statistics which states, as a percentage, the probability that a true value of a quantifiable parameter is to be found between prescribed limits (termed confidence limits) in relation to a quantitative evaluation of a statistical sample.

Control harmony. The relationship among pitch, roll, and yaw controls—in terms of displacements—related to breakout forces, control force gradients, human capability, and overall helicopter flight response and damping.

Control power. Control moment per unit of control displacement.

Control sensitivity. Maximum aircraft angular velocity per unit control input. Also, ratio of control power to damping moment per unit of angular velocity.

Coriolis acceleration. The tangential acceleration caused by a radial displacement of a mass moving in a rotating plane.

Cost-effectiveness analysis. A technique that defines alternative means of accomplishing a stated objective, and that analyzes both the costs and effects of each alternative as a basis for decision-making.

Cost-estimating relationship (CER). A quantitative statement describing how a specified cost of a helicopter system, subsystem, or component is influenced by one or more changes in the design or operating variables of the system.

Crashworthiness. The quality of a helicopter design which aims at assuring the safety of pilot and passengers in the event that the vehicle is involved in an accident or receives damage as the result of an abnormal condition.

Critical decision point (CDP). A combination of speed and altitude at which, if an engine of a multiengine helicopter became inoperative, the takeoff could be rejected and a safe landing made, or the flight could be continued with a prescribed obstacle clearance. An equivalent definition applies to landing.

Cyclic flare. Precision maneuver associated with autorotation, in which the pilot uses cyclic and collective controls to place his vehicle within narrow limits of height above the ground, rate of descent, forward speed, and angle of attack.

Damping ratio. Ratio of actual damping over critical damping for which the mode becomes aperiodic.

Density altitude. Pressure altitude adjusted for temperature deviation from standard.

Differential collective mode. Multiblade mode where adjacent blades oscillate in counterphase; only possible for even-bladed rotors.

Divergence. Nonoscillatory instability; may occur in a single degree of freedom system due to negative spring-like forces or moments.

Drag. Aerodynamic force parallel to relative wind and acting opposite to the direction of flight.

Droop stop. The mechanism that is loaded by the weight of the rotor blade at rest.

Dynafocal mounting. A mounting arrangement applicable to helicopter engine installations, which is designed to "isolate" the helicopter drive system vibration frequencies from those of the engine by damping transmissible forces at resonance frequencies.

Dynamic instability. An oscillation for which the amplitude increases with time without external excitation.

Dynamic rotor model scaling. An attempt to match, in a model rotor, the natural frequencies and ratios of aerodynamic-to-inertia parameters of the actual rotor. This permits investigation of flutter, mechanical vibration and stability, performance, and flying qualities.

Emissivity. The ratio of the rate at which radiant energy is transmitted from a given body to the rate at which radiant energy would transmit from a blackbody of equal temperature in the same medium.

Endurance. The length of time a helicopter can remain airborne while using a specified quantity of fuel.

Endurance limit. The magnitude of the repeated load (or stress) which the structure can endure indefinitely.

Entry into autorotation. The maneuver that begins at the instant of helicopter power failure and continues until steady autorotation is achieved.

Fail-safe. A principle of design to assure continued safe flight in the event of a structural failure of a component or part.

Failure Mode and Effect Analysis (FMEA). That analysis of a helicopter system or subsystem design which traces the probable types of failure that could occur and the criticality of failure in regard to the accomplishment of system operational objectives.

Fatigue. Characteristic of a material that causes it to fail at a repeated load smaller than its static load capacity.

Fatigue damage. Ratio of the number of repeated load applications sustained to the number at which fatigue failure occurs.

Figure of Merit. Index of rotor efficiency: minimum possible power required to hover (ideal rotor) divided by actual power to hover.

Figure of Merit Ratio Method. Method of predicting rotor performance based on empirical evaluation of isolated rotor whirl stand test data. Particularly useful at high rotor loading conditions.

First harmonic motion. A motion proportional to the sine or cosine of the azimuth angle.

Flapping motion. Blade motions in a plane perpendicular to the plane of rotation.

Flare angle of attack. Highest angle from which a helicopter can be rotated nosedown to a level attitude in the time for which rotor energy can be used to develop hovering thrust.

Flexure; flexplate. Part of the rotor hub having a low bending stiffness in the flapping plane, allowing flapping motions of the blade under load.

Flight idle (power). The power required to maintain governed rotor speed at minimum rotor load conditions.

Flight spectrum. A descriptive array, including estimates of the frequency of occurrence, of each maneuver that a given helicopter design is expected to perform in operational use.

Floating hub. Rotor unit in which the assembly is mounted to the vertical mast by means of a universal joint, through which power to the rotor is passed.

Flueric device or system. A fluidic device or system which accomplishes its function without the use of moving mechanical elements.

Fluidics. The use of fluids to sense, process, display, and control functional changes in physical systems.

Flutter. The self-excited oscillation of an aerodynamic surface and its supporting structure, caused by a combination of aerodynamic, inertial, and elastic effects in such a manner as to extract sustaining energy from the air stream.

Flutter, blade. Aerodynamically induced self-excited blade oscillations with essential torsional mode content.

Flutter, stall. Aerodynamic self-excitation of a blade torsion mode by nonlinear pitching moment characteristics at high angles of attack.

Flutter, wake. Blade flutter induced by the rotor vortex wake at zero forward speed and low thrust. It disappears with forward speed and moderate thrust loading.

Flutter, whirl. Prop-rotor airframe self-excited oscillations in a multiblade advancing or regressing out-of-plane mode. Whirl flutter is not a form of blade flutter, since blade torsion is not essentially involved.

Foot-lambert. Unit of luminance equal to $1/\pi$ candle per square foot.

Free-turbine engine. A turboshaft engine that incorporates an independently mounted turbine to transfer gas generator power to the output shaft.

Frequency response function. Complex valued function obtained by setting $s = i\omega$ in the transfer function.

Fuselage. The structure that supports the useful load, supports and connects the dynamic components, and provides an interface between the aerodynamic and ground environment.

Gas generator. The basic power-generating section of a turboshaft engine consisting of a compressor, combustion chamber, and turbine.

Generalized coordinate. A time function representing the contents of a particular mode (usually a normal mode) in the motion of an elastic system.

Gimbal-mounted rotor. Attachment of the rotor hub to the rotor mast, allowing tilting of the rotor in two planes perpendicular to each other.

Ground idle (power). The rated power output of the engine with a control setting at ground idle.

Ground resonance mechanical instability. Rotor-airframe self-excited oscillations in a multiblade regressing lead-lag mode not involving aerodynamic effects.

Gust model. Mathematical representation of atmospheric disturbances to which the helicopter could reasonably be subjected.

Height-velocity diagram. That combination of heights and speeds from which a safe landing can be made if an engine becomes inoperative.

Helical gears. Gears that run on parallel axes but mate on helical surfaces.

Hertz stresses. Highly localized compression stresses characteristically found in rolling machine elements.

Hooke's coupling. A type of universal joint used to transmit power from one rotating shaft to another whose axis is at a relatively large angle to the first.

Human factors engineering (HFE). The development and application of scientific methods and knowledge about human capabilities and limitations to the selection, design, and control of operations, environment, and material, and to the selection and training of personnel.

Hunting tooth. A tooth on a pinion which will not mate with the same space on the gear on each revolution.

Impulse. For ordnance or rocket devices, the measure of total energy of the device, expressed as the product of the average thrust multiplied by the duration of energy release.

Inplane soft blade. Blade chordwise natural frequency below frequency of rotor revolution.

Inplane stiff blade. Blade chordwise natural frequency above frequency of rotor revolution.

Interchangeable components. Components that are structurally, physically, and functionally identical and use the same attachment provisions.

Isochronous control. A device that maintains a system constant speed by continuously changing the speed control input in response to any detected error between the actual and the desired speed.

Laplace transformation. A linear transformation allowing a time derivative d/dt to be replaced by the Laplace operator.

Lead-lag motion. Blade motions in the plane of rotation.

Life-cycle cost. The total cost of ownership of a helicopter system or subsystem over the system life cycle, including all research, development, test, and evaluation (RDT&E); initial investment; and operating and maintenance costs.

Lift. Aerodynamic force perpendicular to the relative wind.

Load factor. The ratio of a given load to the weight of the structural element which serves as the load reference.

Loadpath. The structural elements effective in carrying loads and moments to the supporting stations, going from the extremities to the center section.

Load station. Coordinate of (resultant) load application point.

Long period mode. Aircraft free body pitching mode with substantial flight velocity changes.

Mach instability. Torsional moments on the rotor system, caused by distortion of the airflow across the blade as the blade tip speed approaches the speed of sound.

Maintainability. A characteristic of design and installation that is expressed as the probability that an item will conform to specified conditions within a given period of time when maintenance action is performed in accordance with prescribed procedures and resources.

Maneuver. Any flight condition different from straight and level flight.

Maneuver stability. A measure of helicopter stability in terms of pilot-required control force per unit of normal acceleration.

Mathematical model. A quantifiable representation of a system operating in a prescribed context. A mathematical model generally can be expressed by a set of equations where the known factors are constants, the independent variables are inputs, and the data sought are the dependent or output variables.

Maximum power. The maximum power the engine will produce for a period of time not in excess of ten (10) min per application.

Mean line (airfoil). The locus of points one-half the distance between the upper and lower surfaces of an airfoil.

Mean-time-to-repair (MTTR). The expected value of time required to restore a system to a condition of satisfactory operation.

Mechanical flight control system. A reversible control system in which the cockpit controls are mechanically linked, by a series of rods and bell cranks, directly to the control horn of the rotor blade.

Mission capability. That characteristic of a design which provides for the accomplishment of the functional requirements of a prescribed mission.

Mission effectiveness. The value of a helicopter design in terms of the combination of its mission availability, survivability, and performance.

Mission profile. A quantitatively defined representation of the sequence of maneuvers which a helicopter must perform to accomplish a designated purpose, e.g., observation, transport, and medical evacuation.

Mission readiness. A measure of the degree to which a helicopter is operable and committable to start a particular mission at a random point in time.

Modal representation of blade displacements. Method of representing blade dynamics, based upon the assumption that blade displacements can be represented by the sum of a series of orthogonal functions.

Multiblade mode. A mode where all blades oscillate with the same frequency and amplitude in a certain phase relationship to each other.

Normal mode. An oscillation where all masses of an elastic system have the same frequency and constant phase relations to each other.

Oil scavenge ratio. The ratio of the rate of oil flowing out of a gear case to the flow rate of incoming oil.

Operations research (OR). That branch of management science which seeks to improve the planning and operations of physical systems through analysis.

Overtorque (condition). The transmission of torque loads greater than the maximum allowable torque loads.

Performance. A measure of how well a helicopter system is able to accomplish a designated mission profile.

Pilot-induced oscillation. Pilot involved self-excited oscillations of any type; for example, air resonance caused by the pilot being in the feedback loop.

Pitch element. The instantaneous line of contact between two mating gears.

Power-operated flight control system. An irreversible system in which the pilot, through a set of mechanical linkages, actuates a power control which in turn moves a linkage attached to the control horn of the rotor blade.

Power spectral density. A frequency domain measure of the probabilistic structure of a stationary random process.

Precone angle. Angle between the spanwise axis of the blade and the plane of rotation, built into the hub, that aligns the blade with the resultant of average lift and centrifugal load.

Prescribed Wake-Momentum Analysis (PWMA). Theoretical three-dimensional rotor performance prediction method. Uses the basic strip momentum theory modified to include the effects of the near wake by adding a wake-induced "interference" velocity to the calculated inflow.

Pressure altitude. The absolute altitude above mean sea level in a standard atmosphere at which a given pressure is to be found. Recorded by a standard altimeter when its zero setting is adjusted to standard sea level pressure of 29.92 in. of mercury.

Proportional control. A device that maintains a system constant speed by changing the speed control input in proportion to detected change in the output speed.

Proportional feedback. A feedback signal directly proportional to the output of a given loop of a dynamic system.

Pylon. Structure, mounted on the fuselage, which supports a main rotor at sufficient height to provide clearance over the tail boom, tail rotor, fuselage, and personnel on the ground.

Quasi-static aerodynamics. Contrary to unsteady aerodynamics, the lift is assumed proportional to angle of attack.

Ram recovery loss. That portion of the inlet dynamic pressure which cannot be recovered because of losses of energy due to turbulence and wall friction.

Range index method. Method of determining range capability of a helicopter, based upon an integral expression for initial and final gross weights. Normally evaluated by numerical integration.

Rate feedback. Feedback signal proportional to the time rate of change of the output of a given loop of a dynamic system.

Refraction. The bending of a light ray as it passes obliquely from one medium to another of a different density.

Regressing mode. Multiblade mode when the oscillation of the kth blade leads in phase by $2k\pi/n$ as compared to 0th blade.

Reliability. A quantitative expression of probability that an item or system of items will perform its intended function for a specified duration under stated conditions.

Reliability apportionment. The process of allocating quantitative reliability goals to components or subsystems of a system. The process is used to control the attainment of the quantitative reliability objective for the total system design.

Replaceable components. Components that are interchangeable, except that their installation requires use of ordinary hand tools to make adjustments.

Rigid rotor. A rotor whose blades are rigidly attached to a rigid hub.

Rotor mast. Drive shaft that imparts rotation to the rotor and transmits flight loads to the airframe.

Rotor solidity. The ratio of the total blade area of a rotor to the disk area swept by the rotor.

Rotor speed. Rate of rotation of the rotor around its axis.

Ryder scoring index. Result of a standard test procedure to measure the load capability of a lubricant.

Scavenge flow. The portion of airflow through an air particle separator which carries dirt particles overboard.

Scoring hazard rating. A measure of gear design quality which expresses the tendency for an operating gear set to break down oil film or otherwise wear excessively or destroy itself.

Sectionalization. Those provisions in a helicopter design to allow ready removal or reorientation of subsystems—rotor blades and tail boom—to reduce weight or protrusion and thus ease transportation and storage and preclude handling damage.

Seesaw hinge. Hinge in the hub of a two-bladed rotor; perpendicular to rotor mast and lengthwise axis of the rotor. (Also called "teetering" hinge).

Service ceiling. The maximum altitude, at a given temperature, at which the aircraft can climb at a rate of 100 fpm.

Short-period mode. Aircraft free body pitching mode with little flight velocity change.

Shuffling. Sometimes used for phenomena described under air resonance.

Single-spool engine. A turboshaft engine with the output drive mechanically connected to the gas generator (also known as fixed shaft engine).

Southwell diagram. Method of plotting blade frequencies against rotor speeds to determine coincidence of exciting and resonant frequencies.

Specific fuel consumption (SFC). The ratio of fuel weight consumed per unit of time to the delivered power of the engine. For helicopter engines, SFC is measured in pounds of fuel per brake horsepower hour; for turbojet engines, pounds of fuel per hour per pound of thrust.

Specific power. Power produced in the turbine cycle by each pound per second of airflow. This index, which tends to establish the overall engine volume, is a function of both cycle pressure ratio and turbine inlet temperature.

Spectrum of flight (of loading). Tabulation of time spent in each flight condition in an average flight.

Speed stability. A measure of helicopter stability as the variation in the rate of change of pitching moment with speed.

Spline. A machine element consisting of integral keys (spline teeth) and keyways (spaces) equally spaced around a circle or circular segment.

Stability. That quality of a helicopter which determines its ability to return to an established flight condition following a disturbance.

Stability derivatives. Those terms in an equation of motion which describe the relationship between the aerodynamic forces or moments acting on a helicopter and the linear or angular displacements or velocities from which they result.

Static stability. A measure of helicopter stability as the variation in the rate of change in pitching moment with angle of attack.

Steradian. The unit of solid angle or the solid angle that subtends a surface on a sphere equivalent to the square of the radius.

Subharmonic mode. A mode with a natural frequency equal to a fraction of the rotational frequency $\Omega/2$ or $\Omega/3$ . . . etc.

Supercooled water. A condition in which, in the absence of a nucleus for crystallization, water droplets can remain in a liquid state even though the temperature is below the freezing point.

Survivability. A measure of the degree to which a helicopter system or subsystem can withstand a prescribed manmade environment or inadvertent operational accident, and still accomplish a desired mission.

Swashplate. Structural element with a rotating and a nonrotating portion which transfers control inputs to the rotating blades.

Synchropter. A helicopter design which uses two intermeshing laterally displaced rotors.

System life cycle. A quantitative estimate, usually expressed in years beyond the delivery of the first production unit, of the expected or intended period of operation of a helicopter system.

Tail boom. Structure that attaches longitudinal control and/or directional control surfaces or devices to the fuselage, e.g., a tail rotor and its mechanical drive.

Task analysis. An analytic process used to determine the human behavior requirements of a man/-machine system. The process involves the development estimates of the sequential tasks required of man and machine. Within each task, behavior steps are isolated in terms of perceptions, decisions, memory storage, and motor inputs as well as errors expected.

Thermionic cooling. A process by which heat can be transferred from one side of a bimetallic junction to the other by imposing an electrical potential across the junction.

Thrust. Resultant of airloads on a rotor in a direction perpendicular to the plane of rotation.

Time between overhauls (TBO). Duration, usually expressed in hours of operation, of a mechanical or other system which defines when the system *shall* be overhauled. Initially arbitrarily set, the duration is adjusted based on experience.

Topping. The maximum gas generator speed at which the engine is allowed to run.

Transfer function. Ratio of output over input in terms of the Laplace operator.

Trim. A helicopter is "trimmed" in steady-state flight when the resultant forces and moments on the helicopter are equal to zero, and the force on the pilot's controls is equal to zero.

Trim controls. A system of pilot set devices used to stabilize a helicopter for a given flight condition or reduce a pilot's workload by nulling forces on primary contro[illegible]

Underslinging. Distance betwe[illegible] [illegible]etering hinge or gimbal center of [illegible] rot[illegible] [illegible]he point of intersection o[illegible] [illegible]ngthwis[illegible] of the blades.

Variable geometry en[illegible] [illegible]ngine that [illegible] [illegible]iable stators in the compressor and/or variable vanes in the turbine section.

Vertical bounce. Rotor-airframe self-excited oscillations in a multiblade collective mode.

V-n diagram. A plot of helicopter forward velocity V against normal load factor n. Generally, this diagram is used to define the regime of structural adequacy of the design.

Vortex ring sta[illegible] [illegible] in which rot[illegible] carries with it a [illegible] air rotating down thr[illegible] [illegible]he middle [illegible] rotor and up on the [illegible]de, becoming very unstable after a short period and making the helicopter difficult to control.

Vortex theory. Basis for including three-dimensional considerations in rotor blade performance calculations. Includes an analysis of the rotor vortex system and the character of the generated wake structure.

Wargaming. A technique which predicts the probable outcome of a prescribed military conflict by simulating the engagement.

Weaving. Sometimes used for phenomena like those described under whirl flutter (see *Flutter, whirl*) when they occur in helicopter flight.

Weight fraction. The ratio of the weight of a designated helicopter subsystem to the helicopter design gross weight.

Weight growth factor. An empirically developed multiplier that is used to estimate the increase in design gross weight that results from an increase in payload weight.

Wind tunnel wall corrections. Corrections to be applied to wind tunnel test results. For rotors, a function of the wake skew angle and the dimensions of the tunnel test section with respect to the rotor diameter.

Windage. The air movement caused by rotating machinery.

LIST OF ABBREVIATIONS AMCP 706-201

AC — alternating current
a.c. — aerodynamic center
ADF — automatic direction-finder
AE — absorbed energy
AFBMA — Anti-Friction Bearing Manufacturers Association
AFCS — automatic flight control system
AGARD — Advisory Group for Aeronautical Research and Development
AGMA — American Gear Manufacturers Association
AH — attack helicopter
AI — autorotative index
AIDS — Airborne Integrated Data Systems
AN — Air Force-Navy Aeronautical Standards
AND — Air Force-Navy Design Standards
APU — auxiliary power unit
AQS — Airworthiness Qualification Specification
ASOAP — Army Spectrometric Oil Analysis Program
ASE — automatic stabilization equipment
ASK — automatic station-keeping
ASPA — Armed Services Procurement Act
ASPR — Armed Services Procurement Regulation
ASW — antisubmarine warfare
ATF — automatic terrain-following
BIT — builtin test
BPR — bypass ratio
CAS — calibrated airspeed
CDP — critical decision point
CDRL — Contract Data Requirements List
CER — cost-estimating relationship
CEVM — consummable electrode vacuum melt
cf — constant frequency
CFE — contractor-furnished equipmen.
CF/S — convertible fan shaft
CG — center of gravity
CH — cargo or transport helicopter
COFC — container-on-flatcar
COP — coefficient of performance
CONUS — Continental U.S.
CP — center of pressure
CRT — cathode ray tube
CSD — constant-speed drive
CSM — controlled-speed hydraulic motor
DAVI — Dynamic Antiresonant Vibration Isolator
DB — duplex "back-to-back"
DC — direct current
DF — duplex "face-to-face"
dN — diameter times rpm
DT — duplex tandem
D/T — diameter-to-torque ratio
EAPS — engine air particle separator
EB — electron beam
ECM — electronic countermeasure
ECS — environmental control system
EEL — equivalent exposure level
EL — electroluminescent
EMC — electromagnetic compatibility
EMI — electromagnetic interference
ENCOMS — Engine Control and Monitor Systems
EPNL — effective perceived noise levels
ESEOD — Equivalent Sharp Edge Orifice Diameter
FARADA — failure rate data
FMEA — Failure Mode and Effect Analysis
FMR — Figure of Merit Ratio
FOD — foreign object damage
FORTRAN — formula translation computer language
GF — growth factor
GFAE — Government-furnished airborne equipment
GFE — Government-furnished equipment
GRP — Generalized Rotor

Performance
GSE – ground support equipment
HFE – human factors engineering
HLH – heavy lift helicopter
HOGE – hover out-of-ground effect
H-V – height-velocity
HV – heating value
IAS – indicated airspeed
ID – inner diameter
IDEP – Interservice Data Exchange Program
IFB – Invitation for Bid
IFF – Identification Friend or Foe
IFR – instrument flight rules
IGE – in-ground effect
IR – infrared
ISO – International Organization for Standardization
IVSI – instantaneous vertical speed indicator
JAN – Joint Air Force-Navy Standards
L/D – lift/drag
L/d – length to diameter
LDP – landing decision point
LOH – light observation helicopter
MAC – Military Airlift Command
MC – mission capability
MGT – measured gas temperature
MMH/FH – maintenance manhours per flight hour
MOI – moment of inertia
MS – Military Standard
– margin of safety
MTBF – mean-time-between-failures
MTBMA – mean-time-between-maintenance actions
MTBR – mean-time-between-removals
MTMTS – Military Traffic Management and Terminal Service
MTTR – mean-time-to-repair
NAS – National Aeronautical Standards
NASA – National Aeronautical and Space Administration
O&M – operation and maintenance
OD – outer diameter
OEI – one engine inoperative
OGE – out-of-ground effect
OH – observation helicopter
OR – operations research
OSD – operational sequence diagram
OWE – operating weight empty
PCE – positive continuous engagement
PIDS – Prime Item Development Specification
PIO – pilot-induced oscillation
PIP – product improvement program
PNL – perceived noise levels
POI – product of inertia
PTO – power takeoff
P-V – pressure-velocity
PWMA – Prescribed Wake-Momentum Analysis
QAD – quick attach-detach
QC – quality control
QEC – quick engine change
QI – quantitative index
R&D – research and development
RFP – Request for Proposal
RFQ – Request for Quotation
ROC Required Operational Capability
R/C – rate of climb
RH – relative humidity
SAC – Standard Aircraft Characteristics
SAE – Society of Automotive Engineers
SAS – stability augmentation system
SAWE – Society of Aeronautical Weight Engineers
SDP – System Development Plan
SFC – specific fuel consumption
S-N – failure load to number of cycles to failure
SPS – secondary power system
SRP – seat reference point
SSPP – System Safety Program Plan
S/V – Survivability/Vulnerability
TAB – Technical Abstract Bulletin
TACAN – tactical air navigation
TAS – true airspeed
TBO – time between overhauls
TEQ – twin-engine climbout torque
TH – training helicopter
TIT – turbine inlet temperature
TOFC – trailer-on-flatcar
TOGW – takeoff gross weight
TRU – transformer rectifier unit
TSFC – thrust specific fuel consumption
UH – utility helicopter
vf – variable frequency
VFR – visual flight rules
VOR – very high-frequency omnidirectional range

V/STOL – vertical/short takeoff and landing

VTOL – vertical takeoff and landing

VWS – voice warning system

WAT – weight-altitude temperature

WBGT – Wet Bulb Globe Temperature

SPECIFICATIONS, STANDARDS, AND OTHER GOVERNMENTAL DOCUMENTS

The listed Governmental documents are referenced in the text.

MILITARY SPECIFICATIONS

MIL-P-116, *Preservation, Methods of*

MIL-C-5011, *Charts, Standard Aircraft Characteristics and Performance, Piloted Aircraft*

MIL-W-5013, *Wheel and Brake Assemblies; Aircraft*

MIL-T-5041, *Tires, Pneumatic, Aircraft*

MIL-W-5044, *Walkway Compound, Nonslip, and Walkway Matting, Nonslip*

MIL-W-5050, *Walkway, Coating and Matting, Non-slip, Aircraft, Application of*

MIL-M-5096, *Manual, Technical, Inspection and Maintenance Requirements, Work Cards, Inspection and Lubrication Requirements, Manual, Technical, Acceptance and Functional Check Flight*

MIL-E-5272, *Environmental Testing, Aeronautical and Associated Equipment, General Specification for*

MIL-E-5400, *Electronic Equipment, Airborne, General Specification for*

MIL-H-5440, *Hydraulic Systems, Aircraft Types I and II Design, Installation and Data Requirements for*

MIL-A-5498, *Accumulators, Aircraft Hydropneumatic Pressure*

MIL-C-5503, *Cylinders: Aeronautical, Hydraulic Actuating General Requirements for*

MIL-F-5504, *Filters and Filter Elements Fluid Pressure, Hydraulic Micronic Type*

MIL-F-5509, *Fittings, Flared Tube, Fluid Connection*

MIL-G-5514, *Gland Design; Packings, Hydraulic, General Requirements for*

MIL-P-5518, *Pneumatic System; Aircraft, Design, Installation and Data Requirements*

MIL-R-5520, *Reservoirs: Aircraft Hydraulic, Nonseparated Type*

MIL-W-5521, *Washer, Aircraft Hydraulic Packing, Back-up*

MIL-T-5578, *Tank, Fuel, Aircraft, Self-sealing*

MIL-H-5606, *Hydraulic Fluid, Petroleum Base, Aircraft, Missile, and Ordnance*

MIL-T-5624, *Turbine Fuel, Aviation, Grades JP-4 and JP-5*

MIL-P-5954, *Pump Unit, Hydraulic, Electric Motor-driven Fixed Displacement*

MIL-T-5955, *Transmission Systems, VTOL-STOL, General Requirements for*

MIL-P-5994, *Pump, Hydraulic, Electric Motor-driven, Variable Delivery*

MIL-E-6051, *Electromagnetic Compatibility Requirements, Systems*

MIL-G-6162, *Generators, 30-volt, Direct Current, Aircraft Engine-driven, General Specification for*

MIL-T-6396, *Tank, Fuel, Oil, Water-alcohol, Coolant Fluid, Aircraft, Non-self-sealing, Removable, Internal*

MIL-G-6641, *Gearbox, Aircraft Accessory Drive, General Specification for*

MIL-T-6845, *Tubing, Steel, Corrosion-resistant (304), Aerospace Vehicle Hydraulic System, 1/8 Hard Condition*

MIL-E-7016, *Electrical Load and Power Source Capacity, Aircraft, Analysis of*

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SUPPLEMENTARY

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AMC PAMPHLET
No. 706-201
CHANGE 1

21 July 1980

Engineering Design Handbook

HELICOPTER ENGINEERING, PART ONE, PRELIMINARY DESIGN

AMCP 706-201, Engineering Design Handbook, Helicopter Engineering, Part One, Preliminary Design, August 1974, is changed as follows:

Remove pages 3-9 and 3-10, and insert the attached pages 3-9 and 3-10.

(DRCMT)

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OFFICIAL:

H. S. JOHNSON
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USA DARCOM Svc Spt Actv

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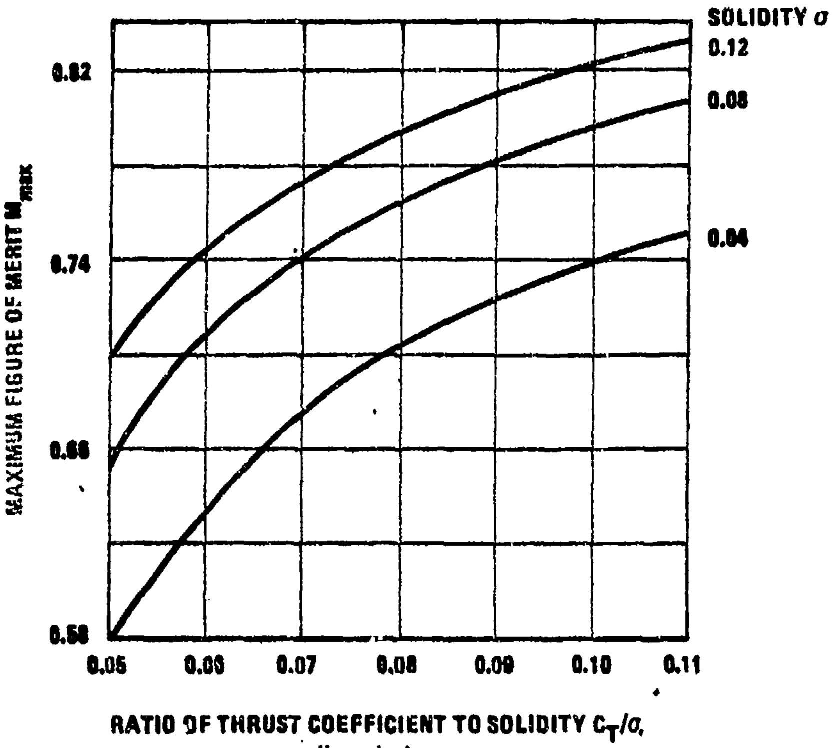

Fig. 3-8. Maximum Figure of Merit

cutout corrections. In reducing the test data, rotor whirl stand performance is adjusted for ground effect using the Cheeseman and Gregory correction (par. 3-2.1.1.8), and the results at discrete tip Mach numbers are cross-plotted to provide performance at the three Mach numbers presented.

There are inherent variants, which should be recognized whenever this method is used, and which contribute to the scatter on the plotted data points. These include:

1. Tip shape
2. Airfoil section
3. Leading edge abrasion strip geometry
4. Reynolds number.

Blade planform and section camber are two other important parameters that need to be considered in detail rotor design. Blade taper has an effect upon rotor performance similar to that resulting from increased blade twist. Both factors tend to provide a more uniform inflow distribution; this relieves the blade loading at the tip, which, in turn, reduces tip losses by decreasing the strength of the tip vortex. In addition, the combined power saving derived from lower induced and profile drag losses can be significant at high rotor loadings.

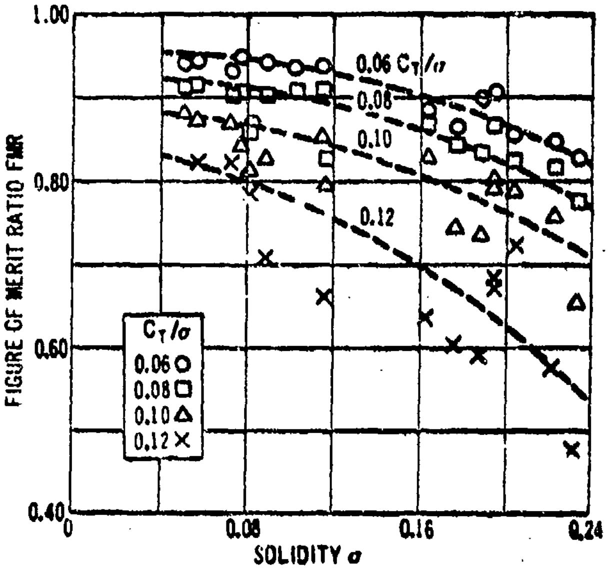

Fig. 3-9. Figure of Merit Ratio for M_t = 0.60

Section camber design considerations usually are

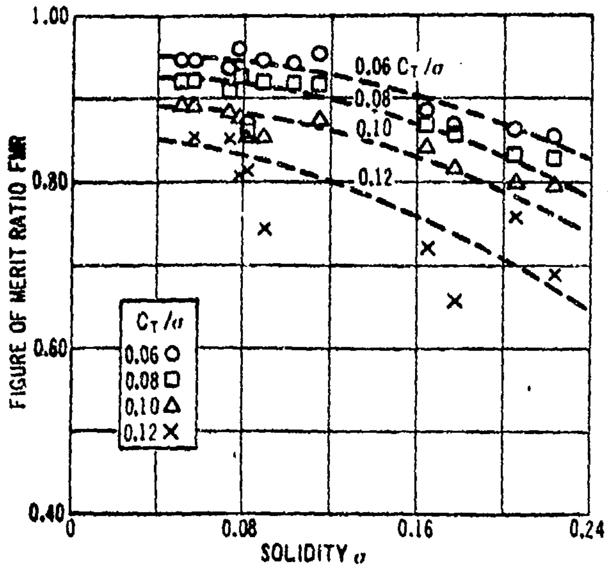

Fig. 3-10. Figure of Merit Ratio for M_t = 0.55

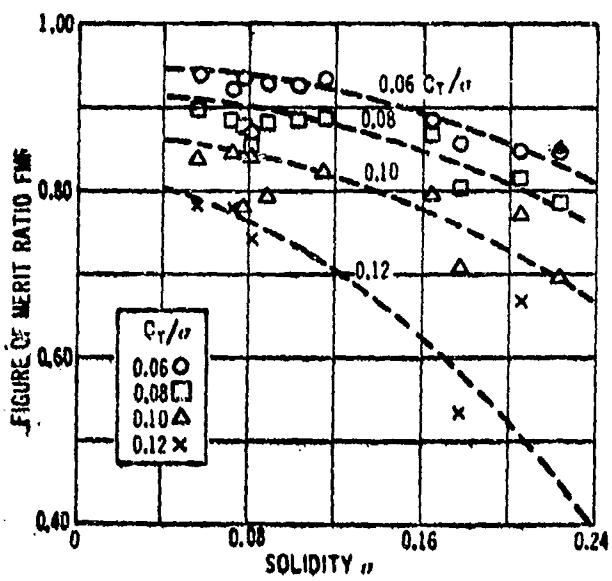

Fig. 3-11. Figure of Merit Ratio for M_t = 0.65

geared to meet specific performance requirements for a particular flight regime, and wind tunnel test programs are conducted to optimize the section characteristics sought. Section pitching moment characteristics normally are the undesirable side effects of camber in a rotating wing, insofar as rotor control load and blade stiffness requirements are concerned. However, means of minimizing inherent pitching moments have been demonstrated. Effects of various airfoils upon hov performance (horsepower required) can be appro imated by use of Eq. 3-7 with $\overline{C}_D$ adjusted for t appropriate section.

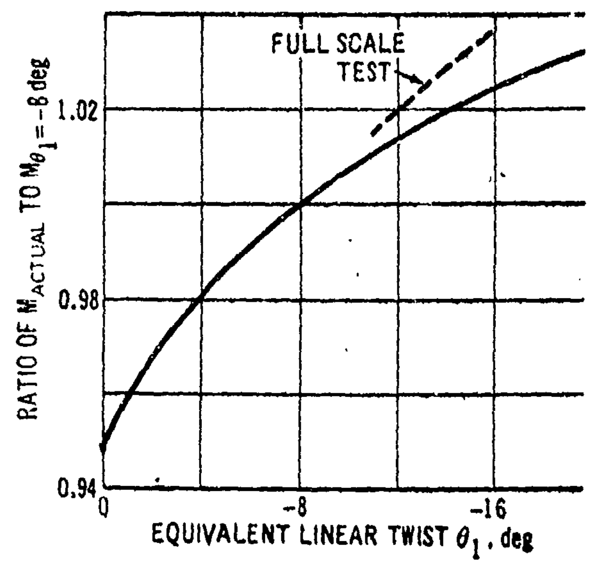

Fig. 3-12. Blade Twist Correction — Baseline: θ_1 = −8 deg

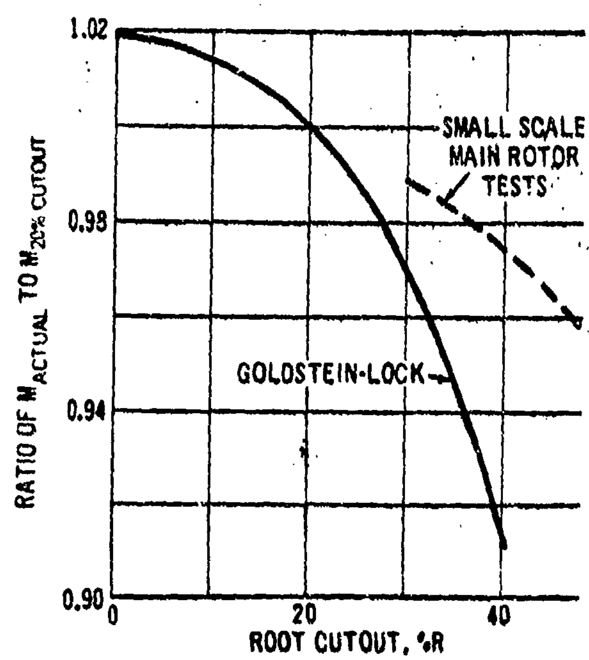

Fig. 3-13. Blade Root Cutout Correction — Baseline: 20% Cutout