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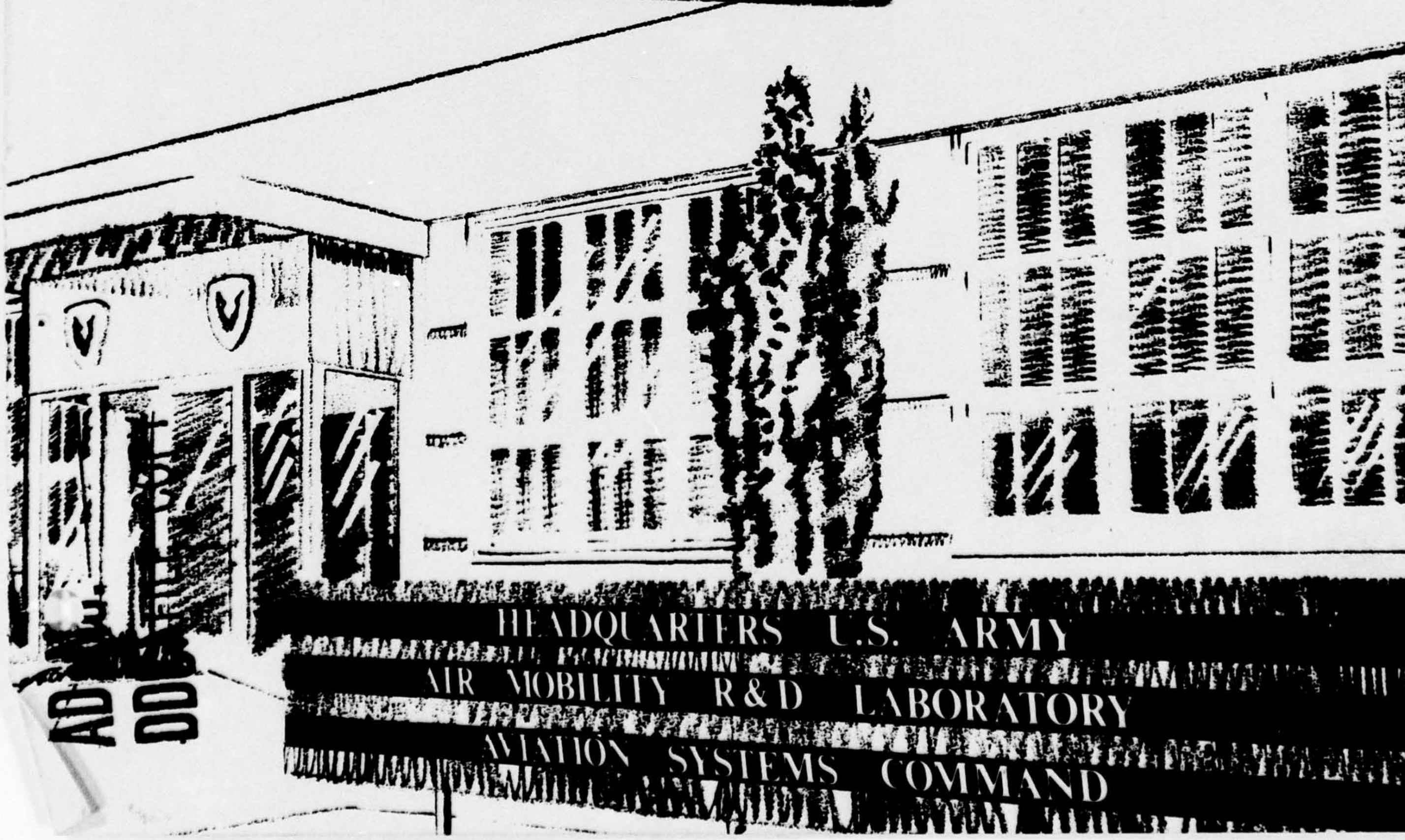

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SINGLE-ROTOR HELICOPTER
DESIGN AND PERFORMANCE ESTIMATION PROGRAMS

VOLUME I - METHODOLOGY

SRIO REPORT NUMBER 77-1

JUNE 1977

PREPARED BY THE

3 SYSTEMS RESEARCH INTEGRATION OFFICE.
410 234 1 ~~US~~ ARMY AIR MOBILITY R&D LAB~~ORATORY~~
US ARMY AVIATION SYSTEMS COMMAND
2 ST. LOUIS, M~~ISSOURI 63102~~

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER 14 SRIO ~~Report Number~~ 77-1 – Vol-1	2. GOVT ACCESSION NO.	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle) 6 SINGLE-ROTOR HELICOPTER DESIGN AND PERFORMANCE ESTIMATION PROGRAMS. VOLUME I. METHODOLOGY.		5. TYPE OF REPORT & PERIOD COVERED 9 FINAL rept.
		6. PERFORMING ORG. REPORT NUMBER SRIO Report Number 77-1
7. AUTHOR(s) 10 Milton A. Schwartzberg, ~~Dr.~~ Roger L. Smith, James L. Means, ~~Dr.~~ Harold Y. H. Law, David P. Chappell		8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS Systems Research Integration Office St. Louis, MO 63103		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS 16 1F262209 AH76 ~~Tech Area VII~~
11. CONTROLLING OFFICE NAME AND ADDRESS US Army Aviation Systems Command St. Louis, MO 63103		12. REPORT DATE 11 June 1977
		13. NUMBER OF PAGES 87
14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office) US Army Air Mobility R&D Laboratory Moffett Field, CA 94035		15. SECURITY CLASS. (of this report) Unclassified
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Distribution of this document is unlimited.

12 90p.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Helicopter design, helicopter performance, rotary wing aircraft design, rotary wing aircraft performance, preliminary design, performance computation.

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Computer programs, designated SSP-1 and SSP-2, have been prepared for: (1) the preliminary design of single-rotor helicopters to meet specified mission requirements, and (2) the estimation of the performance of single-rotor helicopters of known geometry and with known engine characteristics. The present volume is one of a series describing those computer programs. It details the methodology on which all of those programs are based.

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Unclassified.

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

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TABLE OF CONTENTS

INTRODUCTION

This report is one in a series that describes the purpose, the background, and the procedures of two computerized programs, designated SSP-1 and SSP-2. These aircraft systems synthesis programs provide a preliminary design and a performance estimation capability, respectively, applicable to single rotor helicopters.

Although aircraft design and aircraft performance estimation are two distinct functions, creation of the mathematical models and a procedure that enables execution of either one of those functions inherently provides the tools required to perform the other function. Consequently, this initial volume in the series describes the mathematical models that are common to both programs, and it details whatever differences in procedure must be observed to satisfy either the design or performance estimation function, as desired. The other volumes in this series are user's manuals which detail the programming of procedures for each of the two computer programs in both a batch version and an interactive version.

Programs with end objectives identical to those of the present programs can be found in the literature, e.g., references 1 and 2. The present programs were prepared independently of any existing ones with the intent of creating a simple, rapid, and responsive tool for performing the functions of aircraft design and performance estimation. These programs do not include the finer details of rotor blade design, rotor loads, or rotor motions.

The equations used are essentially limited to momentum theory considerations to minimize the required input information, procedural detail, and computational complexity without markedly compromising the utility of the solutions.

The computer programs incorporate mathematical models in several areas of aircraft performance; namely, hover, vertical flight, and forward flight. Mathematical models for engine performance, fuel consumption, and aircraft group weights, the latter based on statistical studies of helicopter data, are also contained in the programs. In their present configurations, the programs apply only to helicopters with a single main rotor, single tail rotor, and conventional gas turbine engines (no wings or auxiliary power sources). The performance estimation program, SSP-2, is not limited in the gross weight range of aircraft to which it can be applied. However, the preliminary design program, SSP-1, should be used with caution for gross weights outside of the range of 2000 to 100,000 pounds, which are the bounds of the weights statistical data used as an integral part of that program.

DISCUSSION

The mathematical models used in both SSP-1 and SSP-2 are very nearly identical. The significant differences between the two programs are in their applications of those models, their required inputs, and their outputs.

COMPUTATION SCHEMES:

In the following, the computation scheme used in each program is described to clarify the interrelationships among the several mathematical models in each case.

SSP-1:

This program has the following computation scheme: For a set of desired aircraft performance characteristics (mission and payload), and certain specified design parameters, it computes the required aircraft configuration, and gross and component weights. A flow chart of that computation procedure is presented in Figure 1. The computation begins with three basic types of inputs:

(1) Aircraft parameters which include geometric and aerodynamic characteristics (main rotor disk loading, main rotor blade loading (C_T/σ), number of blades (b) in main rotor, main rotor tip speed (V_T), and rotor airfoils), engine characteristics, a required payload,

and an initial gross weight assumption. All but the last of these inputs are fixed through the course of a complete program iteration cycle.

(2) A mission profile that specifies the duration and requirements of each segment of the mission.

(3) Operating conditions (altitude and atmospheric temperature at which the aircraft operates) for each segment of the mission.

Initially, the engine is sized to meet the most demanding power requirement among the hover, vertical climb, and high speed segments of the mission, including operation of multi-engined aircraft with one engine inoperative. Then the fuel weight and all of the aircraft component group weights are computed. This enables determination of the available payload capacity. A comparison of the available payload with the specified required payload leads to a new gross weight estimation. The program is reentered with this new gross weight estimate and the iteration is continued until the gross weight compatible with the required payload is determined. In the process, the aircraft's most pertinent dimensions (main and tail rotor diameters and chords, and tail rotor arm) and the required engine horsepower will be specified. A further discussion of the gross weight iteration scheme is presented in Appendix A.

SSP-2:

This program has the following basic computation scheme: For a given aircraft configuration including a specified engine, a specified mission, and a specific VROC requirement, the program can compute the maximum compatible aircraft gross weight, the required fuel weight, and the available payload capacity. An alternative is to use the gross weight iteration subroutine to determine the VROC capability of a configuration for which the gross weight is specified as input.

The mathematical models within SSP-2 for the calculation of power and fuel requirements are the same as those used for those purposes in SSP-1. The differences between the preliminary design and performance estimation program inputs, outputs, and internal relationships can be seen by a comparison of the flow chart for the SSP-2 program shown in Figure 2 with the comparable chart for SSP-1 shown in Figure 1.

ASSUMPTIONS:

Many of the assumptions made in the mathematical models of the present programs are inherent in the development of the rotor characteristics on the basis of momentum theory, rather than blade strip theory.

Level Flight:

(1) A value of lift coefficient is computed for the complete rotor, and that value is used as representative of the average blade section lift coefficient.

(2) An average blade section drag coefficient is established on the basis of the average lift coefficient, determined as above, and this value is used to estimate the total rotor drag.

(3) Deviation of the rotor spanwise induced velocity distribution from the ideal uniform distribution is accounted for by a multiplying factor applied to the computation of rotor induced power. A value of the factor for zero flight velocity must be supplied as input to the program. Some guidance in the selection of that value is provided in Figure 3, as taken from reference 3. The value of that factor is reduced, in the program, with increasing forward flight velocity as suggested in reference 3.

(4) Power transmission losses are a constant percentage of the sum of the main and tail rotor powers.

(5) Engine power extractions and losses (due to inlet particle separator, aircraft environmental control systems, etc) are either a constant amount or a fixed percentage of the power being drawn from the engine.

(6) Parasite drag is a function of equivalent flat plate drag area, either input or computed, and independent of fuselage attitude.

(7) Incremental power required due to compressibility effects on the advancing blade of the main rotor is based on an empirically derived relationship, reference 4. That relationship was derived from flight test data obtained with the UH-1H helicopter only.

(8) Incremental power required due to retreating blade stall effects is assumed proportional to the amount by which the retreating blade tip angle of attack exceeds the airfoil section static stall angle.

Hover and Vertical Climb:

(1) As in items (1) through (5) above for level flight.

(2) An estimate of the ground effect on main rotor induced power is obtained from an empirically derived expression, reference 5. The maximum induced power reduction due to ground effect is 30-percent and this occurs at main rotor height-to-diameter ratios $\leq$ 0.28.

(3) The additional power required to climb vertically from a hover is equal to the increase in potential energy per unit of time less the reduction in induced main rotor power attributable to the increased rotor inflow velocity.

(4) Vertical drag of the aircraft elements located in the rotor downwash is a constant percentage of the aircraft's gross weight.

LIMITATIONS:

The programs are limited in applicability to single rotor helicopters with a single tail rotor, conventional gas turbine engines, and no wings or auxiliary propulsion systems. SSP-1 is limited by the weights statistical data used in that program to aircraft gross weights in the range of 2,000 to 100,000 pounds.

Tapered and twisted rotor blades can be only approximately and indirectly accounted for in the present programs. Blade taper can be simulated by the selection of an appropriate input value for an effective blade chord, and blade twist by the choice of the multiplying factor used to allow for deviation of the rotor inflow distribution from a uniform distribution.

Level Flight:

The computation of incremental rotor power required due to retreating blade stall requires that blade flapping harmonics of the second and higher orders be negligible, and that the blade mass and the rotor inflow velocity distribution be uniform along the blade span. Blade twist, if any, is assumed to have a linear spanwise distribution.

Hover and Vertical Climb:

(1) Since main rotor blade lift is assumed to increase linearly with increasing blade attitude, the effects of blade stall and the approach to blade stall are not accounted for. A warning message is issued when a main rotor blade loading, C_T / σ , greater than 0.2 is reached.

(2) Compressibility effects are not considered for hover and vertical climb flight modes.

Engine:

The relation of engine fuel flow rate to engine horsepower must be linear, or very nearly so, to permit the mathematical representation of this relationship that is used in the programs.

Weights:

The group weight models used in SSP-1 are based on weight data available for 59 different helicopters or their derivatives. Nevertheless, these data do not adequately account for the effects of design criteria developed in recent years, whose full impact will be registered on future aircraft. Nor do these data adequately reflect the effects of recent state-of-the-art advances in structures, materials, and design which will not be statistically significant for some time to come.

For design purposes, it is recommended that the weight effects of the following design criteria be assumed as additive to the present group weight formulations:

Crashworthiness - structure, crew seats, crew restraints, fuel systems

Survivability - ballistic tolerance, reduced infrared cross-section, reduced infrared signature, etc.

Reliability and Maintainability - engine particle separator, redundant control systems, etc.

Similarly, an innovation such as the extensive use of composite materials is not within the capability of this program.

Although the weights data base can, and should be, continually updated, it must necessarily always fall short of providing the full capability of accurately projecting future designs which will incorporate new features. The designer must recognize this and make adjustments appropriate to his particular case. It may be noted, however, that intuition suggests that the weight savings due to improved materials, structural concepts, and design approaches will probably keep pace with the weight increases imposed by the more stringent design criteria noted above.

MATHEMATICAL MODELS:

Power Required For Level Forward Flight:

The mathematical expressions and computational procedure presented here are used to estimate the drive system, or shaft, horsepower required by a single-rotor helicopter in level forward flight. That power requirement can be viewed as due to 3 power absorbing components:

$$SHP_{R} = HP_{MR} + HP_{TR} + HP_{XMSN}$$

The equations used to compute each of these 3 components, in turn, are listed in the following paragraphs.

Main Rotor Horsepower, HP_{MR}:

The power requirement of the main rotor is expressed as the sum of 5 contributing factors:

$$HP_{MR} = HP_{P} + HP_{O} + HP_{i} + HP_{CMP} + HP_{S}$$

where:

HP_{P} = parasite power (as defined in this program, this term represents the power required to overcome both the drag of all of the components of the aircraft other than the main rotor and the component of the main rotor H-Force in the free-stream direction).

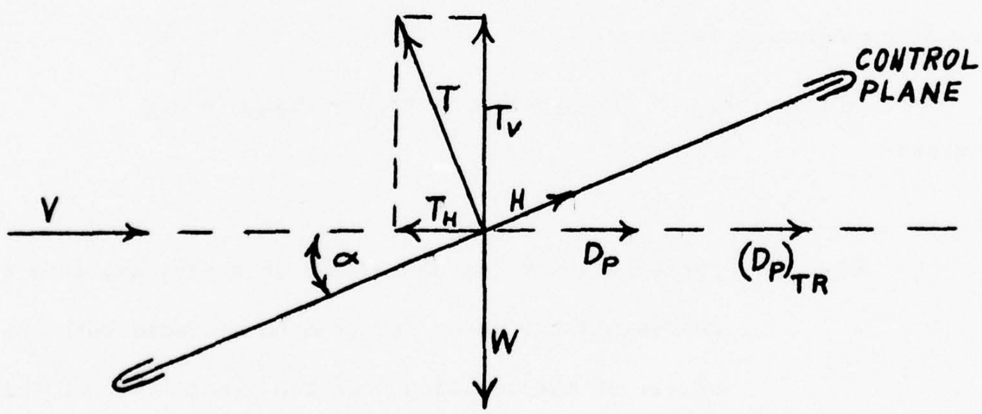
HP_o = main rotor profile power

HP_i = main rotor induced power

HP_{CMP} = incremental power requirement due to compressibility effects on the main rotor

HP_s = incremental power requirement due to stall effects on the main rotor

The thrust and angle of attack of the main rotor, required for level flight at any given aircraft weight and velocity, are determined by an iteration process wherein the balance of forces is assumed to be as shown in the following diagram:



(α as shown is negative)

The equations used in the iteration are:

$$T = \sqrt{T_H^2 + T_V^2} \quad \text{and} \quad \alpha = \tan^{-1}\left(-\frac{T_H}{T_V}\right)$$

where: $T_H = D_P + H\cos\alpha + (D_P)_{TR}$

$T_V = W + H\sin\alpha$

with: $D_P = fq$

where $q = \frac{1}{2}\rho V^2$

and, for SSP-2: f is an input value

whereas, for SSP-1: $f = C(W)^{2/3}$

(See App. B for guidance in the selection of the constant of proportionality, C)

For the main rotor:

$H = \frac{3.65}{B}\pi\rho\sigma\mu C_D v_T^2 R^2$ See App. C

where: $\mu = \frac{V\cos\alpha}{v_T}$

$C_D = C_{D_0} + K\left|\bar{C}_L + 2\pi\alpha_{OL}\right|^{2.7}$ from Ref. 6

$\bar{C}_L = \frac{6\,C_T}{\sigma[B^3 - X^3 + 3/2\,\mu^2(B-X)]}$ See App. D

$C_T = \frac{T}{\rho A v_T^2}$

$B = 1 - \frac{\sqrt{2C_T}}{b}$ from Ref. 7

and for the parasite drag due to the tail rotor:

$(D_P)_{TR} = H_{TR}\cos\alpha_{TR} + T_{TR}\sin\alpha_{TR}$

where: $$H_{TR} = \frac{3.65}{8}\pi\rho\,\sigma_{TR}\,C_{D_{TR}}\,(V_T)^2_{TR}\,R^2_{TR}\,\mu_{TR}$$

(See Appendix E for tail rotor sizing procedure in SSP-1)

with: $$\mu_{TR} = \frac{V\cos\alpha_{TR}}{(V_T)_{TR}}$$

$$(C_D)_{TR} = (C_{D_0})_{TR} + K_{TR}\left|\bar{C}_{L_{TR}} + 2\pi(\alpha_{OL})_{TR}\right|^{2.7}$$

$$(\bar{C}_L)_{TR} = \frac{6(C_T)_{TR}}{\sigma_{TR}\left[B^3_{TR} - X^3_{TR} + 3/2\,\mu^2_{TR}(B_{TR} - X_{TR})\right]}$$

$$(C_T)_{TR} = \frac{T_{TR}}{\rho A_{TR}(V_T)^2_{TR}}$$

and: $$T_{TR} = \frac{\dfrac{550\,HP_{MR}}{\ell_{TR}\Omega_{MR}} - \left(\dfrac{\ell_F}{\ell_{TR}}\right)\left[\dfrac{\rho}{2}V^2(C_{L\alpha})_F\,\alpha_F A_F\right]}{1 - C_{D_{FP}}(S_P/A_{TR})K_F^2}$$

where: K_F is the ratio of induced tail rotor flow velocity at the fin to the final slipstream velocity of the tail rotor flow field. (Plots of K_F as a function of distance between the tail rotor and the fin, and of tail rotor thrust coefficient, for both tractor and pusher configurations, are presented in Figures 4 and 5, respectively, as taken from Reference 8).

and: the second term in the numerator accounts for any anti-torque moment provided by the vertical fin in forward flight, and the denominator increases the tail rotor thrust requirement as needed to overcome the force of the tail rotor induced flow on the vertical fin.

(NOTE: the above equation for tail rotor thrust is one difference between the mathematical models used in SSP-1 and SSP-2. The complete equation, as shown, is used in SSP-2, so that a known vertical fin size and location relative to the tail rotor can be accounted for, if desired. As used in SSP-1, the equation of tail rotor thrust is reduced to

$$T_{TR} = \frac{550\ HP_{MR}}{\ell_{TR}\ \Omega_{MR}}$$

)

The iterative solution for α by means of the preceding equations is obtained by successive substitutions, with convergence to a solution determined by comparison of the difference between successive values of α and an input error parameter. The iteration is initiated by setting $\alpha = H = H_{TR} = 0$. Since H_{TR} is a function of HP_{MR}, the calculation of main rotor shaft horsepower required (see below) must be inside the iteration loop. The tail rotor power required is calculated outside of the iteration for α.

Now: $HP_{MR} = HP_{P} + HP_{o} + HP_{i} + HP_{CMP} + HP_{S}$

$$HP_{P} = \frac{T_H\ V}{550}$$

$$HP_{o} = \frac{\sigma C_D \rho A v_T^3 (1+\mu^2)}{4400}$$ See App. F

$$HP_{i} = \frac{\eta T V K_u}{550 (V/u_H)}$$ from Ref. 3

where: $\eta = \eta_H + K_u (V/u_H)(1.038 - \eta_H)$ from Ref. 3

$$u_H = \frac{1}{B}\sqrt{\frac{T}{2\rho A}}$$

$$K_u = \sqrt{-\frac{1}{2}\left(\frac{V}{u_H}\right)^2 + \frac{1}{2}\sqrt{\left(\frac{V}{u_H}\right)^4 + 4}}$$ from Ref. 3

and: $$HP_{CMP} = \frac{\rho A v_T^3}{550}(\Delta C_P)_{CMP}$$

where: $(\Delta C_P)_{CMP} = 0.009105(\Delta M)^{3.651}$ for $(\Delta M) \leq .19$

$= 0.009338(\Delta M)^{3.686}$ for $.19 < (\ M) \leq .24$

$= 0.1153\ (\Delta M)^{5.445}$ for $(\Delta M) > .24$

See Fig. 6.

$$\Delta M = M_{AT} - M_C$$

M_C = function of $(C_L)_{AT}$

$$(C_L)_{AT} = \frac{2\,C_T}{\sigma}$$

and: $HP_S = (HP_o/4)((\alpha_{tip})_{270^\circ} - \alpha_S)$.

This is analogous to the method presented in Reference 7, except that the latter used $\alpha_S = 12°$ rather than the stall angle of the rotor blade airfoil section. The rotor is considered to have entered unacceptably severe stall when $HP_S = HP_o$. Retreating blade tip angle of attack is

$$(\alpha_{tip})_{270°} = \theta_o + \theta_1 + (\phi_{tip})_{270°}$$

For small ϕ, and assuming that blade flapping harmonics of second and higher order can be neglected, it can be shown that

$$(\phi_{tip})_{270°} = \frac{\lambda + a_1}{1-\mu} \qquad \text{See App. G}$$

With the additional assumptions of uniform blade mass and inflow distributions (the latter is a reasonable assumption at high forward speeds where retreating blade stall will be encountered), rotor longitudinal flapping is

$$a_1 = \frac{\mu\left(\frac{8}{3}\theta_o + 2\theta_1 + 2\lambda\right)}{1-\frac{\mu^2}{2}} \qquad \text{See App. H}$$

Collective pitch is given by

$$\theta_o = \frac{\frac{12C_T}{a\sigma} - 3(B^2-X^2)\left[\lambda + \theta_1/2\,(\mu^2+B^2+X^2)\right]}{2(B^3-X^3)+3\mu^2(B-X)} \qquad \text{See App. I}$$

which incorporates the above assumptions plus linear blade twist and constant chord blades.

Tail Rotor Horsepower, HP_{TR}:

The power requirement of the tail rotor is taken to be the sum of the tail rotor profile and induced power requirements.

$HP_{TR} = (HP_o)_{TR} + (HP_i)_{TR}$. The equations used in each case,in

SSP-1, are of the same form as those presented above for the main rotor profile and induced power requirements; i.e.:

$$(HP_o)_{TR} = \frac{\sigma_{TR} (C_D)_{TR} \rho A_{TR} (V_T)^3_{TR} (1 + \mu^2_{TR})}{4400}$$

$$(HP_i)_{TR} = \frac{\eta_{TR} T_{TR} V (K_u)_{TR}}{550 \left(V/(u_H)_{TR}\right)}$$

where: $\eta_{TR} = (\eta_H)_{TR} + (K_u)_{TR} \left(V/(u_H)_{TR} \right) \left(1.038 - (\eta_H)_{TR}\right)$

$$(u_H)_{TR} = \frac{1}{B_{TR}} \sqrt{\frac{T_{TR}}{2 \rho A_{TR}}}$$

$$(K_u)_{TR} = \sqrt{1/2 \sqrt{\left(V/(u_H)_{TR}\right)^4 + 4} - 1/2 \left(V/(u_H)_{TR}\right)^2}$$

In SSP-2, a factor, K_{DF}, is included in the equation for tail rotor induced power that permits consideration of either an open tail rotor or a ducted fan.

Then: $$(HP_i)_{TR} = \frac{\eta_{TR} T_{TR} V (K_u)_{TR}}{550 K_{DF} \left(V/(u_H)_{TR}\right)}$$

where: $$(u_H)_{TR} = \frac{1}{B_{TR}} \sqrt{\frac{K_{DF} T_{TR}}{2 \rho A_{TR}}}$$

and: η_{TR} and $(K_u)_{TR}$ are computed as above.

The values to be used for K_{DF} are:

1.0 for a conventional tail rotor

2.0 for a ducted fan.

Transmission Power Losses, HP_{XMSN}:

In addition to the power required by the main rotor and the tail rotor, the power available at the engine drive shaft must include that amount which is normally absorbed by the shafting and gearboxes which transmit that power. In the present programs, the power lost in transmission is expressed as a constant percentage of the power required by the rotors,

$$HP_{XMSN} = \mathcal{J}_{XMSN}(HP_{MR} + HP_{TR}).$$

The percentage to be used in any given case must be supplied as a program input. Experience has shown this loss to be of the order of 5-percent.

Power Required to Hover (Out of Ground Effect):

The engine output shaft power required by a single-rotor helicopter in hover can be summed up as that due to the same three power-absorbing components as was the case for level forward flight:

$$SHP_R = HP_{MR} + HP_{TR} + HP_{XMSN}$$

The equations used to compute each of these three components are listed in the following paragraphs.

Main Rotor Horsepower, HP_{MR}:

The main rotor power requirement is expressed as the sum of the profile and induced powers,

$$HP_{MR} = HP_o + HP_i$$

or, in coefficient form,

$$HP_{MR} = \frac{\rho A v_T^3}{550} (C_{P_o} + C_{P_i})$$

where: $$C_{P_o} = \frac{\sigma C_D}{8}$$ See App. F

and: $$C_{P_i} = \frac{\eta_H (C_T)^{3/2}}{B\sqrt{2}}$$ from Ref. 3

with: $$C_D = C_{D_o} + K \left| \bar{C}_L + 2\pi\alpha_{OL} \right|^{2.7}$$

$$\bar{C}_L = \frac{6 C_T}{\sigma (B^3 - X^3)}$$ See App. D

$$C_T = \frac{T}{\rho A v_T^2}$$

$$T = (1 + D_V/W_G) W_G$$

where the vertical drag to gross weight ratio, D_V/W_G, to be used in any given case, must be supplied as a program input.

Tail Rotor Horsepower, HP_{TR}:

The tail rotor power requirement is expressed as the sum of its profile and induced powers,

$$HP_{TR} = (HP_o)_{TR} + (HP_i)_{TR}$$

or, in coefficient form,

$$HP_{TR} = \rho A_{TR} (v_T)^3_{TR} \left((C_{P_o})_{TR} + (C_{P_i})_{TR} \right)$$

where: $$(C_{P_o})_{TR} = \frac{\sigma_{TR} (C_D)_{TR}}{8}$$

with: $$(C_D)_{TR} = (C_{D_o})_{TR} + K_{TR} \left| (\bar{C}_L)_{TR} + 2\pi (\alpha_{OL})_{TR} \right|^{2.7}$$

$$(\bar{C}_L)_{TR} = \frac{6 (C_T)_{TR}}{\sigma_{TR} (B^3_{TR} - X^3_{TR})} \quad .$$

In SSP-1:

$$(C_{P_i}) = \frac{(\eta_H)_{TR} (C_T)^{3/2}_{TR}}{B_{TR} \sqrt{2}}$$

with: $$(C_T)_{TR} = \frac{T_{TR}}{\rho A_{TR} (v_T)^2_{TR}}$$

and: $$T_{TR} = \frac{550 (HP)_{MR}}{\ell_{TR} \Omega_{MR}}$$

whereas, in SSP-2:

$$(C_{P_i})_{TR} = \frac{(\eta_H)_{TR} (C_T)^{3/2}_{TR}}{\sqrt{K_{DF}}\, B_{TR} \sqrt{2}}$$

where, as in the level forward flight case, the factor, K_{DF}, permits consideration of either an open tail rotor or a ducted fan,

and: $$T_{TR} = \frac{550 (HP_{MR}/\ell_{TR})/\Omega_{MR}}{1 - (C_D)_{FP} (S_P/A_{TR}) K_F^2} \quad .$$

Transmission Power Losses, HP_{XMSN} :

As for level forward flight, the power absorbed during hover by the aircraft's shafting and gearboxes is expressed as,

$$HP_{XMSN} = J_{XMSN}\,(HP_{MR} + HP_{TR})$$

Power Required to Hover (In Ground Effect):

The effect of the presence of the ground below the rotor of a helicopter hovering at low altitude is to reduce the induced power requirement of the main rotor, as compared to that needed out of ground effect, to produce a given thrust. The relative induced power requirements can be expressed as,

$$(C_{P_i})_{IGE} = \Lambda\,(C_{P_i})_{OGE}$$

where: $$\Lambda = \left[\frac{1.099\,z/D - 0.104}{z/D + (C_T/\sigma)(.289\,z/D - 0.391)}\right]^{3/2}$$

as determined from a statistical study of flight test data, Ref. 5.

For rotor heights greater than one rotor diameter above the ground ($z/D > 1$), the data indicate ground effect to be negligible. The program establishes an upper bound of $\Lambda = 1.0$ for values of $z/D > 1.0$. In order to maintain stability of the above equation, the program sets $\Lambda = 0.70$ for values of $z/D < 0.28$, the lowest flight test value of that parameter.

The shaft horsepower required to produce a given thrust for hover in ground effect can be computed as for the case of hover out of ground effect with the above indicated adjustment to the computation of C_{P_i} for the main rotor. The power requirements of the tail rotor and transmission are computed as for hover out of ground effect. They will be reduced due to the reduction of main rotor torque in ground effect.

Power Required to Climb Vertically, HP_{VC} :

A helicopter climbing vertically requires power, additional to its hover requirements, which is proportional to its climb rate. Vertical climb also reduces the required induced inflow velocity to the rotor so that the induced power requirement is less than that for hover. The net effect of a vertical climb can be expressed as an incremental power above that required to hover, as follows (from Ref 3):

$$C_{P_{VC}} = \frac{u_H T \left[\left(V_{VC}/u_H \right) + \eta_H \left(K_{VC} - 1 \right) \right]}{\rho A v_T^3}$$

$$u_H = \frac{1}{B} \sqrt{\frac{T}{2\rho A}}$$

$$K_{VC} = \sqrt{\left(1/2 \frac{V_{VC}}{u_H}\right)^2 + 1} - 1/2 \frac{V_{VC}}{u_H}$$

Then,

$$(HP_{MR})_{VC} = \frac{\rho A v_T^3}{550} \left(C_{P_i} + C_{P_o} + C_{P_{VC}} \right)$$

where, C_{P_i} and C_{P_o} are as determined for the main rotor in HOGE.

For the steady vertical climb case, as considered above, it is assumed that the main rotor thrust is equal to the thrust in hover, and that the main rotor profile power is the same in both cases. These assumptions imply similar inflow velocity distributions in the two cases, hence the use of η_H in the above equation.

Uninstalled Engine Power Required:

An engine sizing program must take into account any power demands made of the engine by accessory items, and any losses from the bare engine power that may be incurred by the addition of special installations thereto. Additional power demands of the engine which are normally met before power is applied to the drive shaft can include such items as an oil cooler fan, hydraulic pump, electrical generator, aircraft interior environment control system. There may also be power losses associated with the installation of the engine in the aircraft, an inlet particle separator, or an infra-red countermeasure system.

It may be possible and desirable to assign a constant value to the power requirements of some of the above items, whereas others would be more appropriately expressed in the form of a percentage of the engine power, which varies with the aircraft's flight condition or the environmental conditions. To accommodate each type of power demand, the total engine power required is expressed as,

$$HP_R = \frac{SHP_R + (\Delta HP)_A}{\left(1 - \frac{\varepsilon}{100}\right)}$$

where: $(\Delta HP)_A$ is the power supplied to those accessory items that require a constant amount of power from the engine at all times.

and: ξ is the percentage of engine power supplied to those accessory items which place a variable demand on the engine.

In the current programs, the total engine power required, in terms of a sea level standard day engine rating, is expressed as,

$$HP_{MAX} = \frac{\left[SHP_R + (\Delta HP)_A\right](C_3)}{\left[1 - \frac{\xi}{100}\right](\%_{IPR})}$$

where: C_3 accounts for the engine lapse rate from SLS conditions to the altitude and temperature conditions at which the power requirements were determined.

and: $\%_{IPR}$ allows for the possible requirement that a percentage of the engine power available be held in reserve over and above the power requirements. (e.g., the aircraft specification may require a hover capability with the usable power limited to 95% of that available, then $\%_{IPR} = 0.95$)

Atmospheric Conditions:

A subroutine is included in the current programs for the calculation of the speed of sound, the atmospheric density, and the ratio of the pressure at the altitude of interest to that at sea level on a standard day. The input data required to perform these computations includes the ambient temperature and the pressure altitude of the mission segment.

then: $a = \sqrt{\gamma RT}$

$$\rho = \rho_{SLS}\,\theta^{4.2561}$$

$$\delta = (1 - 6.875 \times 10^{-6}\, h)^{5.2561} \qquad \text{from Ref. 9}$$

where: h is the pressure altitude in feet

T is the ambient temperature in degrees Rankine

θ is the ratio of ambient temperature to standard day sea level temperature (518.688°R)

ρ_{SLS} is the standard day sea level density (2.376919×10^{-3} slug/ft^3)

R is the gas constant for air (1718 ft^2/sec^2 °R)

δ is the ratio of atmospheric pressure at altitude h to standard day sea level pressure

Engine Lapse Rate:

The horsepower rating of an aircraft engine is normally stated in terms of sea level standard day conditions. The hover, vertical climb, or maximum speed condition at which the required maximum engine power is determined by the present programs is likely to be for an altitude and atmospheric temperature other than the SLS condition. In order to size the engine to meet the power requirement at altitude and an elevated temperature, it is necessary to know the engine power variations with atmospheric temperature and pressure. This "lapse rate" has been determined for a number of existing engines by an examination of the data available for them.

For Intermediate Rated Power (IRP),

$$C_3 = \frac{HP_{SLS}}{HP_{Alt}} = \frac{\theta^{C_5}}{\delta^{C_4}}$$

The use of C_3 to size the engine was discussed in the section on "Uninstalled Engine Power Required".

In the programs the lapse rate for the IRP condition of the engine is also used to compare the available and required powers during the hover portions of the mission to make certain that the available power is not exceeded. For the same purpose, the available and required powers are compared during the cruise portions of the mission. In that case, it is the lapse rate of the engine at its maximum continuous power setting that is of interest.

For Maximum Continuous Power (MCP),

$$C_3 = \frac{HP_{SLS}}{HP_{Alt}} = \frac{\theta^{C_7}}{\delta^{C_6}}$$

The engines studied, and the constants obtained thereby, as used in the above equations, are listed in the following table.

MANUFACTURER	ENGINE	C_4	C_5	C_6	C_7
Allison	T-63	1.2	2.6	1.22	3.2
Allison	250-C20B	1.1	1.51	1.1*	2.16*
Allison	250-C20B (Regenerative)	1.1	1.51	1.1*	2.26*
General Electric	T700-GE-700	.983	2.016	1.016	3.126
Lycoming	LTS-101	1.1	2.43	1.1*	3.46*
Lycoming	T53-L13B	1.119	2.854	1.072	3.336
Lycoming	T53-L703	1.077	2.481	.999	3.407
Lycoming	T702-LD-700	1.004	2.362	1.004	3.374
McCulloch	MC-101	1.397	2.635	1.397*	2.635*
Pratt & Whitney	PT6B-34	1.0	2.78	1.0*	4.17*

(NOTE: An Asterisk (*) in the table above indicates an estimated value where no data were available.)

Engine Fuel Flow Rate:

The current programs include a subroutine for the computation of the engine fuel flow rate. The subroutine requires the following input information:

1) Pressure altitude
2) Ambient temperature in degrees Fahrenheit
3) Shaft horsepower for the particular mission segment
4) HP_{MAX}
5) Engine factors, C_1 and C_2

Typical engine fuel flow data are found to vary linearly with engine horsepower at a given atmospheric condition. The effects of altitude changes on this relation can be normalized by the factor, $\delta\sqrt{\theta}$, as shown for a sample case in Figure 7. For current program purposes this relationship has been further generalized by expressing it in the form,

$$\frac{\dot{W}_F / \delta\sqrt{\theta}}{HP_{MAX}} = C_1 + C_2 \frac{HP/\delta\sqrt{\theta}}{HP_{MAX}}$$

so that,

$$\dot{W}_F = C_1 \delta \sqrt{\theta}\, HP_{MAX} + C_2\, HP$$

where: $HP = HP_R = \dfrac{SHP_R + (\Delta HP)_A}{(1 - \xi/100)}$

Examples of the required engine constants as determined from available engine data are listed in the following table.

MANUFACTURER	ENGINE	C_1	C_2
Allison	T-63	0.172	0.509
Allison	250-C20B	.13	.5
Allison	250-C20B (Regenerative)	.122	.376
General Electric	T700-GE-700	.103	.356
Lycoming	LTS-101	.16	.43
Lycoming	T53-L13B	.152	.437
Lycoming	T53-L703	.104	.478
Lycoming	T702-LD-700	.160	.424
McCulloch	MC-101	.091	.87
Pratt & Whitney	PT6B-34	.156	.434

Fuel Consumption:

The current programs include a method for computing the amount of fuel consumed during any desired aircraft mission if that mission is divided into no more than ten separate mission segments. Conceptually, those segments could comprise a typical mission as follows,

Mission Segment

1. Warm-up and taxi (idle power)
2. Takeoff hover
3. Outbound cruise
4. Loiter
5. Forward area hover
6. Loiter
7. Inbound cruise

8. Fuel reserve

9. Landing hover

10. Taxi (idle power)

For each mission segment in the computation, program inputs of time, temperature, and altitude for that segment are required. Other required inputs are the flight velocity for the cruise and loiter segments, and z/D for the hover segments. The order of the mission segments cannot be interchanged, but any unnecessary segment can be eliminated from consideration by assigning a zero time length to that segment.

Although the existing programs confine mission segments 1 and 10 to idle power operation, and segments 2, 5, and 9 to hover power operation, there is some flexibility in the treatment of segments 3 and 4, and segments 6, 7, and 8. In each of these latter cases, the forward flight power required subroutine is used. Therefore, it is possible, for example, to use segment 3 for loiter and segment 4 for cruise by assigning appropriate values to the inputs, particularly the time and flight velocity, for each of those segments.

The fuel burned during any mission segment is determined by an integration of the fuel flow rate from the beginning to the end of the segment's time interval,

$$W_F = \int_{t_i}^{t_f} \dot{W}_F \, dt$$

where the fuel flow rate is computed as described in the preceding section. The aircraft's gross weight decreases as fuel is consumed in the course of a mission segment. Since a reduction of the gross weight, at a given flight velocity, reduces the required engine shaft horsepower, and, therefore, the fuel flow rate, a stepwise solution of the above equation is used to determine the fuel consumed during each segment. A choice of step size is available that permits gross steps, which are sufficiently accurate in most cases, or smaller steps where it is known or suspected that the rotor is operating close to a stalled condition.

The programs permit a discrete change in the aircraft's gross weight to allow for the unloading of stores during segment 5. In the event that these stores were carried externally, a change in the aircraft's drag area, f, can be made for the remainder of the mission.

Aircraft Group Weights:

Aircraft weight estimating relationships have been generated for use in the present programs by correlation of the weight statistics available for 59 differenct helicopters. The pertinent data for those aircraft are tabulated in Appendix J. Some of those aircraft are modifications of an initial design, such as the family of UH-1 helicopters. Others, like the HLH, reached an advanced design stage but were never completely constructed or assembled. Nevertheless, the information available for each aircraft has been used, either in whole or in part, to maximize the sample size for each aircraft weight group.

A new weight estimation study which uses a significantly expanded data base is currently in progress, and may lead to some changes in the weights equations in a revised version of the current programs. However, initial indications suggest that few, if any, substantial differences in estimated aircraft empty weight will result from the expanded data base as compared to the present methods.

The aircraft group weight models, as presented here, are applicable to single-rotor helicopters with various types of main rotors. The group weights equations are listed below. Each equation represents the least squares fit of the data points for a power curve of the form,

$Y = \prod_i^n a_i x_i^{b_i}$.

Articulated Main Rotor:

$$Y = 1.54\ b\ c\ R^{1.5}$$

Teetering or Rigid Main Rotor:

$$Y = 0.94\ b\ c\ R^{1.75}$$

Body:

$$Y = 0.02665\ W_G^{.943}\ R^{.654}$$

Propulsion:

$$Y = 4.44\ (HP_{MAX})^{.86}\ W^{-.098}$$

Transmission:

$$Y = 196\ (DRSYSPWR/RPM)^{.858}$$

Tail:

$$Y = 0.428 \times 10^{-6}\ b^{.728}\ c^{.728}\ R^{3.184}\ (RPM)^{1.456}$$

Skid Landing Gear:

$$Y = 0.44\ W_G^{.63}$$

Wheeled Landing Gear:

$$Y = 0.038\ W_G$$

Flight Controls (Articulated and Teetering Rotors):

$$Y = 0.5045\ W_G^{.689}\ c^{.659}$$

Hydraulic, Pneumatic, and Electrical Systems:

$Y = 152.67\ (HP_{MAX})^{.846}\ W_G^{-.399}$ for Cargo and Utility Aircraft

$Y = 0.1905\ (HP_{MAX})^{.616}\ R$ for other Aircraft

Air Conditioning and Anti-Icing Systems:

$$Y = 0.008\ W_G$$

Instruments:

$Y = 0.000385\ W_G^{1.321}$ for $2000 \leq W < 25{,}000$ pounds

$Y = 3.715\ W_G^{.415}$ for $25{,}000 \leq W \leq 100{,}000$ pounds

Furnishings and Equipment:

$Y = 0.00074\ W_G^{1.298}$ low end of range

$Y = 0.00161\ W_G^{1.298}$ average values

$Y = 0.00296\ W_G^{1.298}$ high end of range

Aircraft empty weight is defined here as the sum of all of the above group weights. It was found that the furnishings and equipment for any given design are not readily categorized by reference to any of the usual aircraft weight correlating parameters. To allow for specific inputs of furnishings and equipment weights an aircraft huli weight has been defined for use in the present programs, where the hull

FIG. 8b. SAMPLE OUTPUT OF PRELIMINARY DESIGN PROGRAM (SSP-1) FOR A RANGE OF PAYLOADS weight is equal to the empty weight less the furnishings and equipment weight. Other definitions are:

Fuel Weight = Consumed Mission Fuel Weight + Reserve Fuel Weight

Payload = Crew Weight + Passenger Weight + Mission Equipment Weight + Avionics Weight + Armor Weight + Armament Weight + Special Provisions Weight + ...

SAMPLE RESULTS:

An arbitrarily selected set of input data has been entered into the preliminary design program, SSP-1, to illustrate the computer output typical of that program. Most of the significant input and output data of this sample case are presented in the following paragraphs and in Figures 8a and 8b. The format used here is not representative of the computer program, but one selected for convenience.

Input:

Single engine, teetering rotor, skid landing gear.

Main Rotor: DL = 3.06, $C_T/\sigma = 0.077$, V_T = 655 ft/sec, b = 2,

$X = 0.21$, $C_{D_o} = 0.0085$, $K = 0.013$, $\eta_H = 1.131$,

$\alpha_{OL} = 0°$, a = 0.10, $\alpha_S = 8.5°$, $\theta_1 = -10.6°$.

Tail Rotor: b = 2, X = 0.21, $C_{D_o} = 0.0090$, K = 0.0135,

V_T = 709 ft/sec, $\alpha_{OL} = 0°$.

Engine: $C_1 = 0.172$, $C_2 = 0.509$, $C_4 = 1.20$, $C_5 = 2.60$,

$C_6 = 1.22$, $C_7 = 3.20$.

Aircraft: C = 0.0479, $D_V/w_G = 0.02$,

$\mathcal{f}_{XMSN} = 0.05$.

Mission: Out of ground effect,

Altitude = 2,000 ft, Temperature = 95°F.

Warm-up and taxi = 0.0833 hrs.

Outbound cruise = 0.23 hrs at 100 kns.

Loiter = 0.3133 hrs at 90 kns.

Hover = 1.25 hrs.

Loiter = 0.3133 hrs at 90 kns.

Inbound cruise = 0.23 hrs at 100 kns.

Reserves = 0.50 hrs at 115 kns.

Taxi = 0.0833 hrs.

Payloads: 500, 600, 700, 800 pounds.

Output:

In addition to the gross weight, fuel weight, engine horsepower, and main and tail rotor dimensions, shown in Figures 8a and 8b, the program provides the following information for each of the required payloads:

1. The group weights.
2. The engine horsepower, the fuel flow rate, and the pounds of fuel consumed during each segment of the mission.
3. The tail rotor moment arm.
4. The main and tail rotor RPM's.
5. The aircraft hull and empty weights.

CONCLUDING REMARKS

Computer programs have been written in both batch and interactive versions, that provide a preliminary design capability and a performance estimation capability for single rotor helicopters. The present report describes the methodology used in those programs. This single description of that methodology is possible since it is nearly identical for both programs.

The program methodology is relatively simple, with the aim of minimizing the required input information and the overall program complexity. Applications of the programs to date have indicated a satisfactory level of accuracy for preliminary design and/or estimation purposes.

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FIG. 1. FLOW CHART OF PRELIMINARY DESIGN PROCEDURE, SSP-1

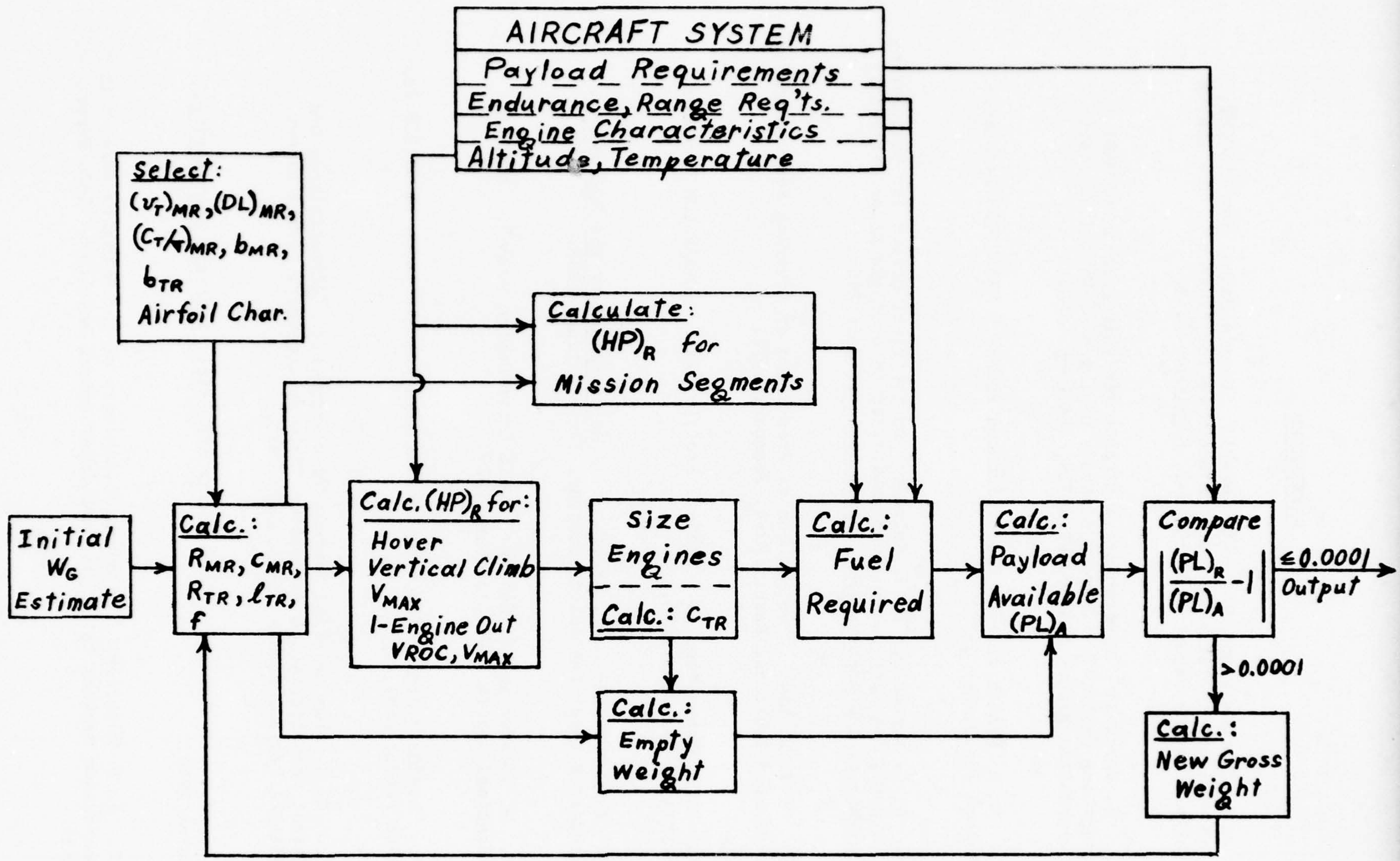


FIG. 2. FLOW CHART OF PERFORMANCE ESTIMATION PROCEDURE, SSP-2

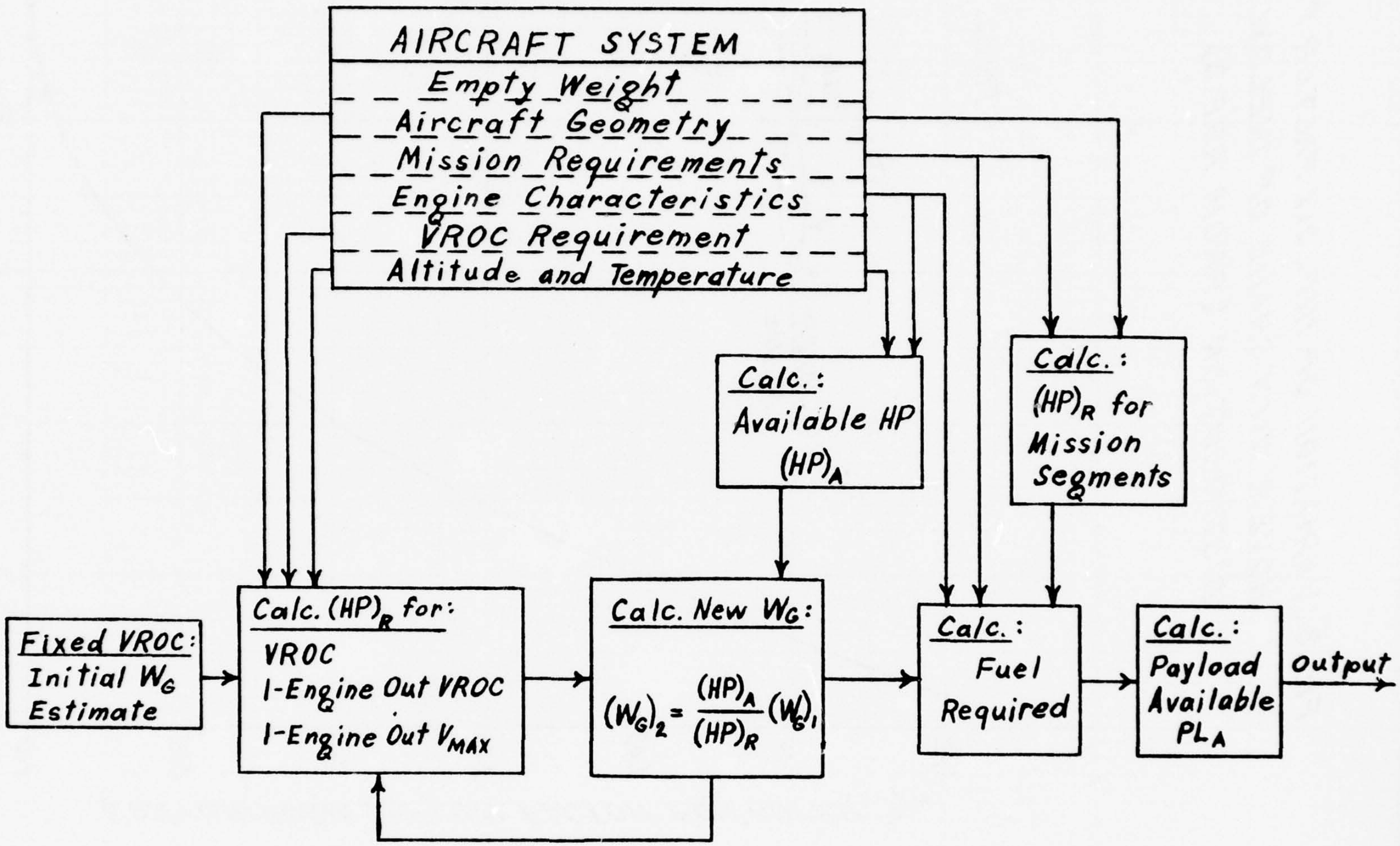


FIG. 3. VARIATION OF INFLOW FACTOR FOR HOVER WITH SHAPE OF INFLOW DISTRIBUTION (FROM REF. 3)

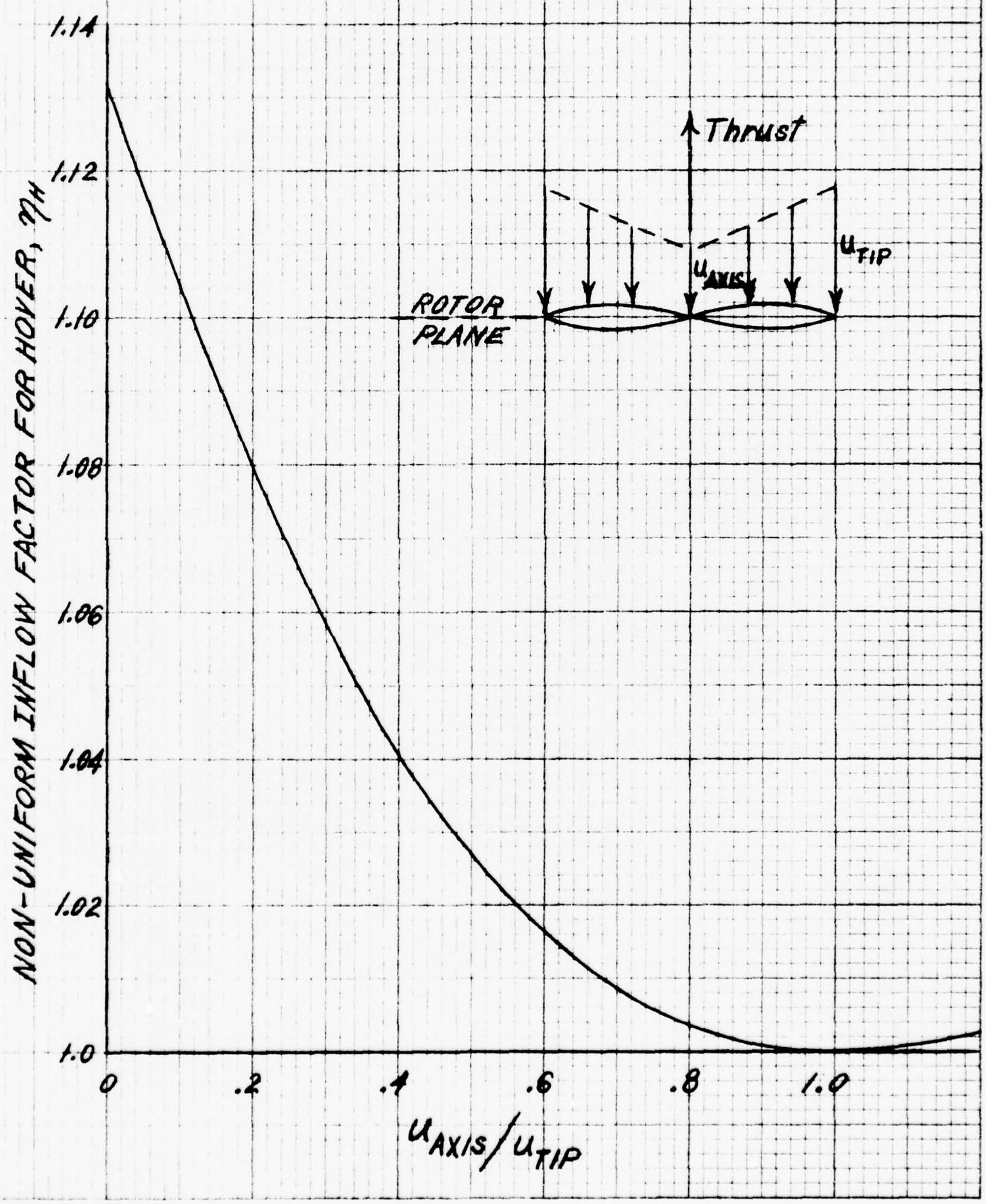


FIG. 4. INDUCED VELOCITY RATIO, K_F, FOR TRACTOR TAIL ROTOR (from Ref. 8)

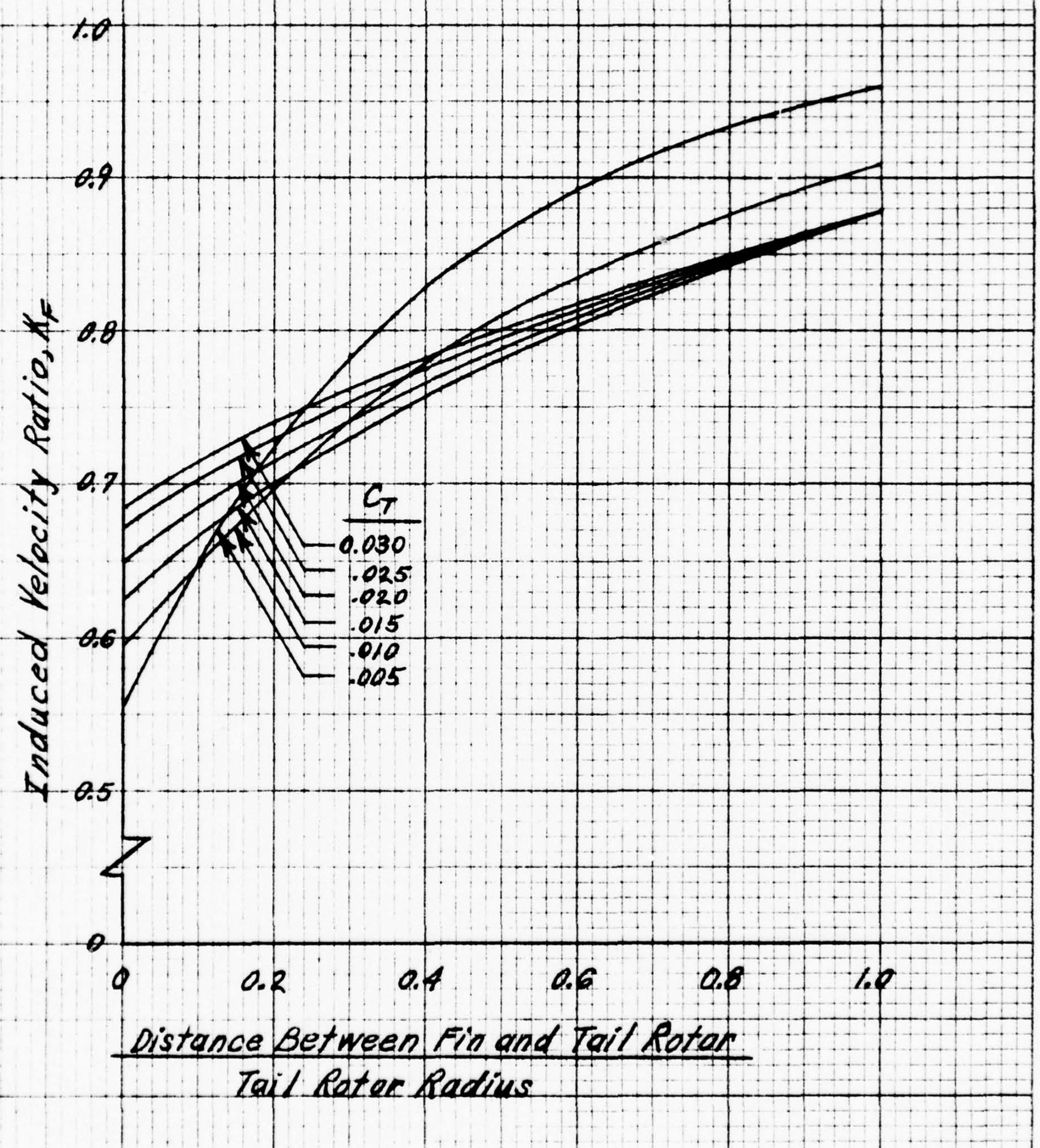
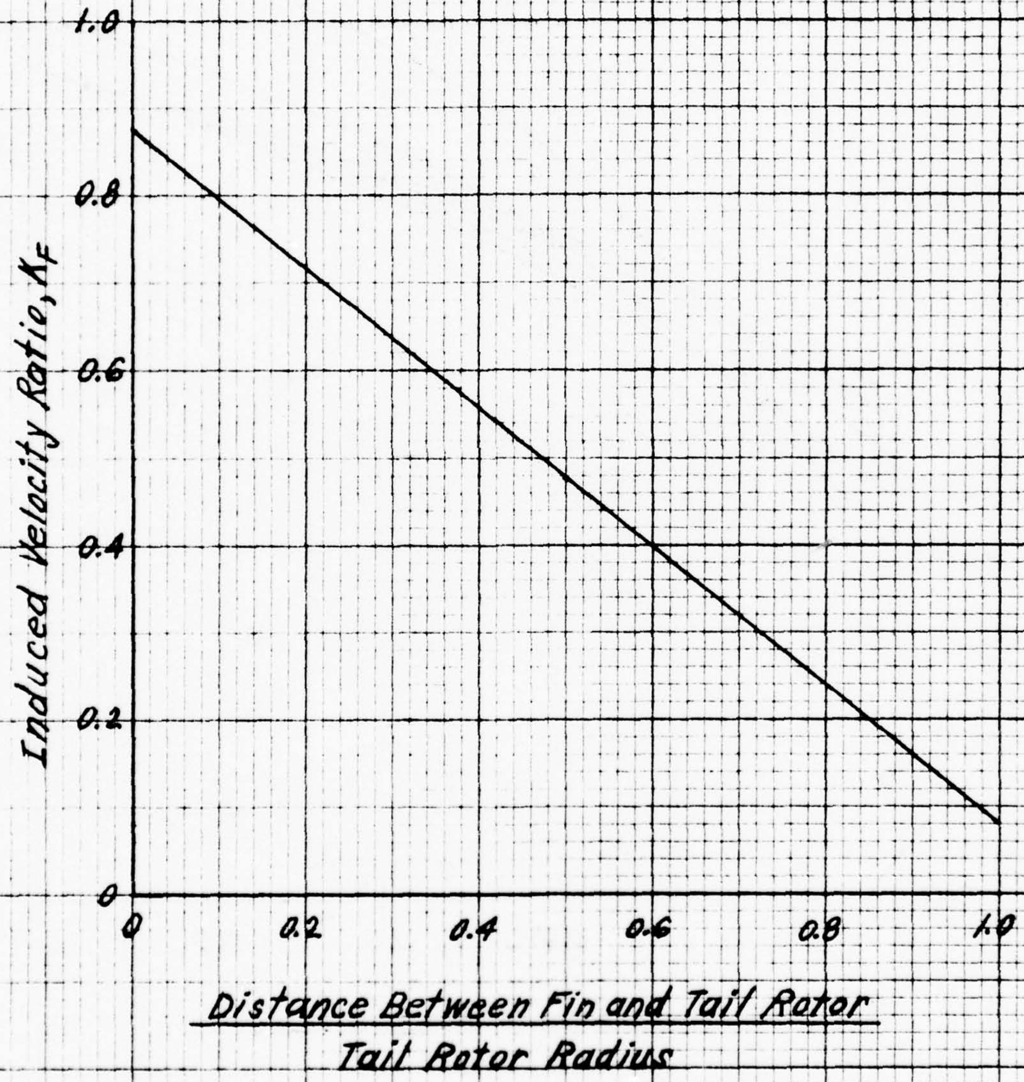


FIG. 5. INDUCED VELOCITY RATIO, K_F, FOR PUSHER TAIL ROTOR (from Ref. 8)



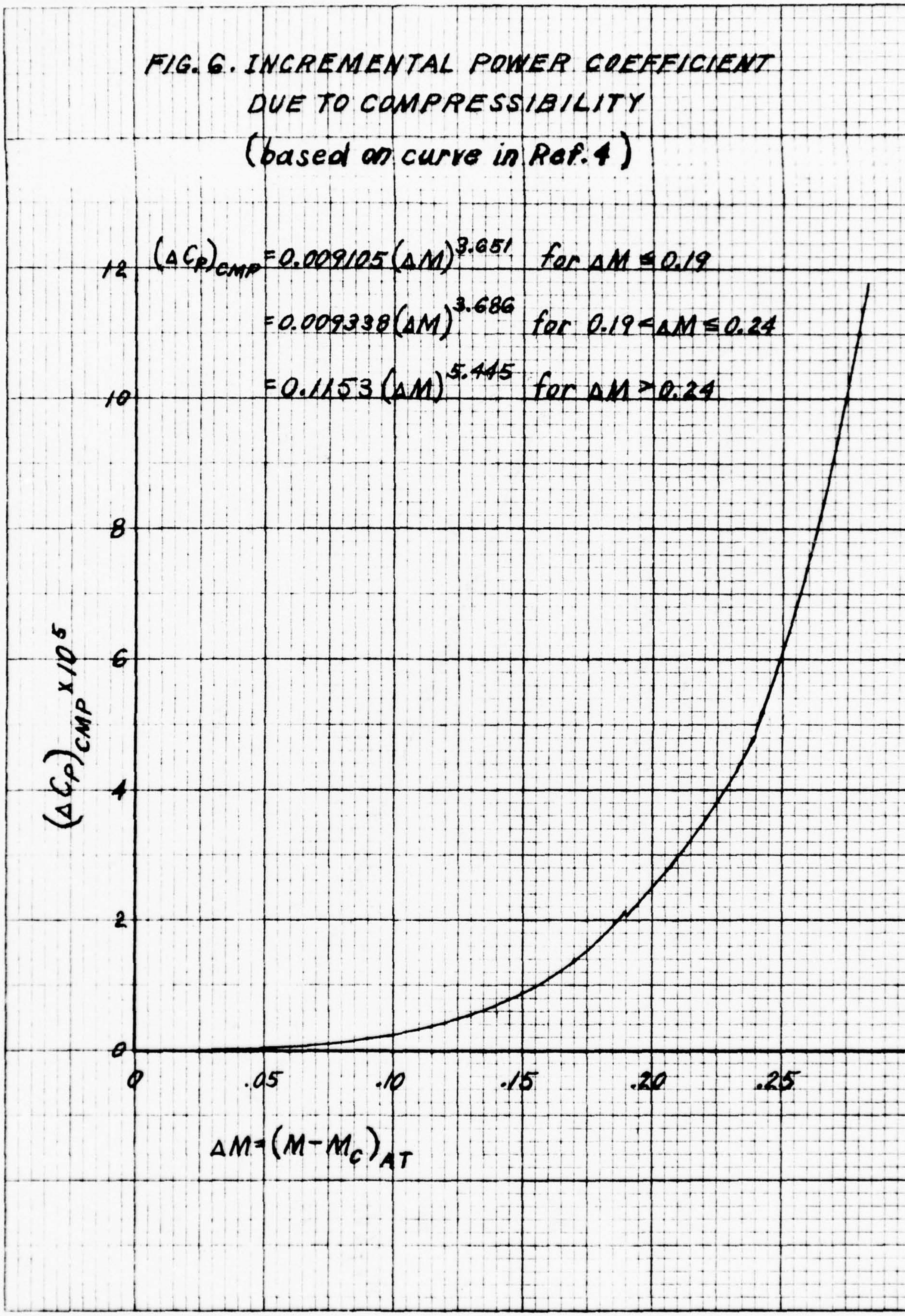


FIG. 6. INCREMENTAL POWER COEFFICIENT DUE TO COMPRESSIBILITY (based on curve in Ref. 4)

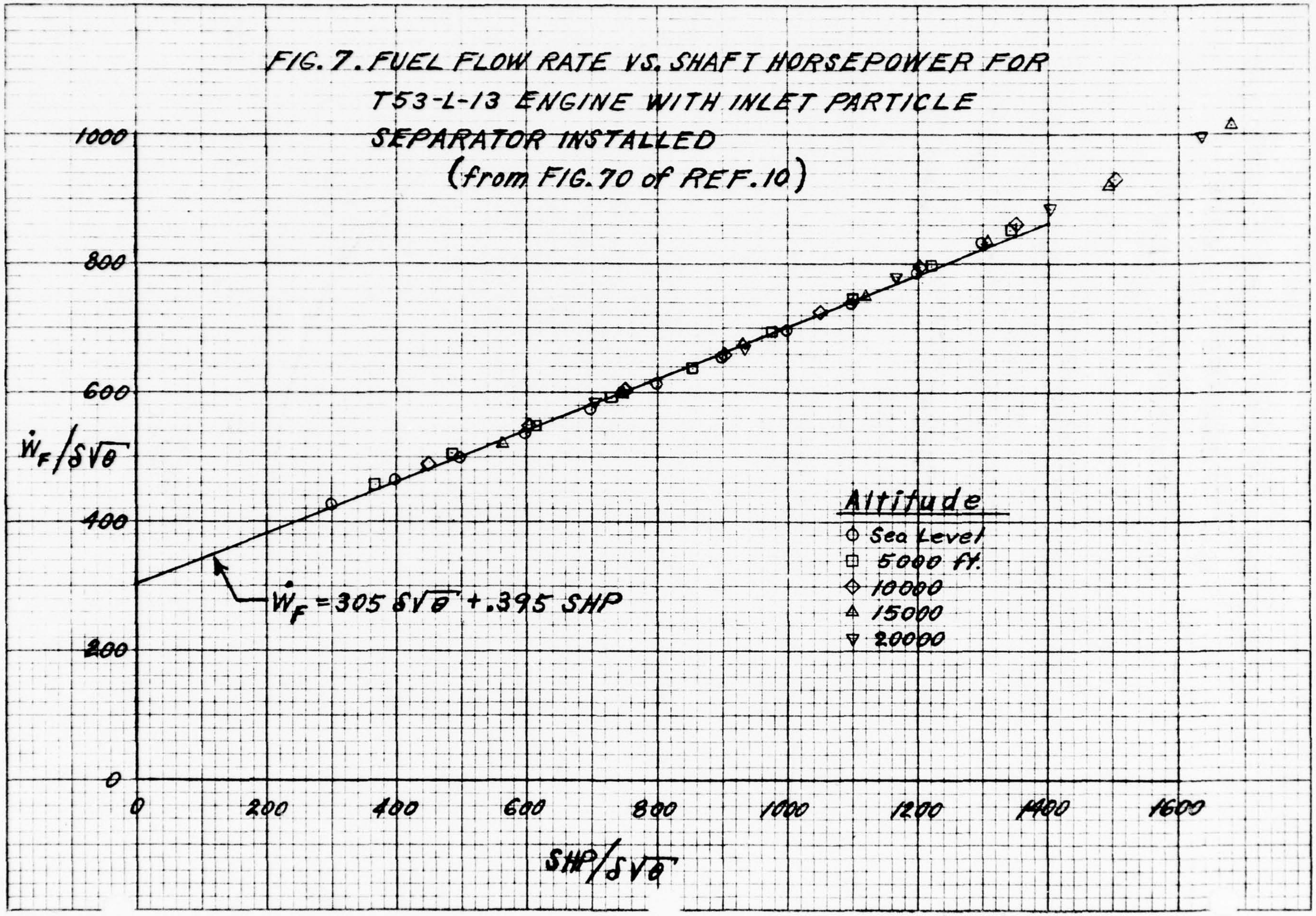


FIG. 7. FUEL FLOW RATE VS. SHAFT HORSEPOWER FOR T53-L-13 ENGINE WITH INLET PARTICLE SEPARATOR INSTALLED (from FIG. 70 of REF. 10)

FIG. 8a. SAMPLE OUTPUT OF PRELIMINARY DESIGN PROGRAM (SSP-1) FOR A RANGE OF PAYLOADS

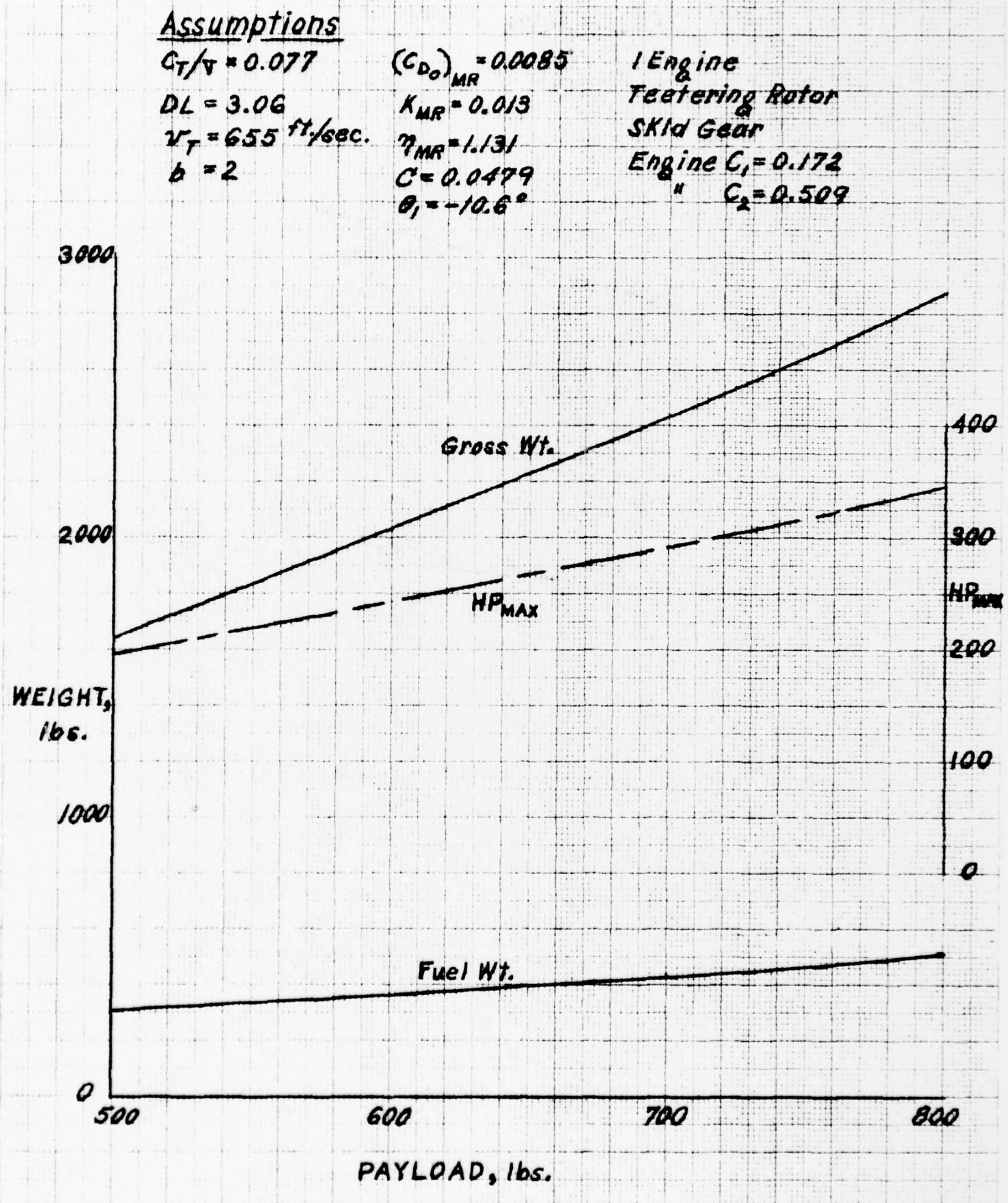
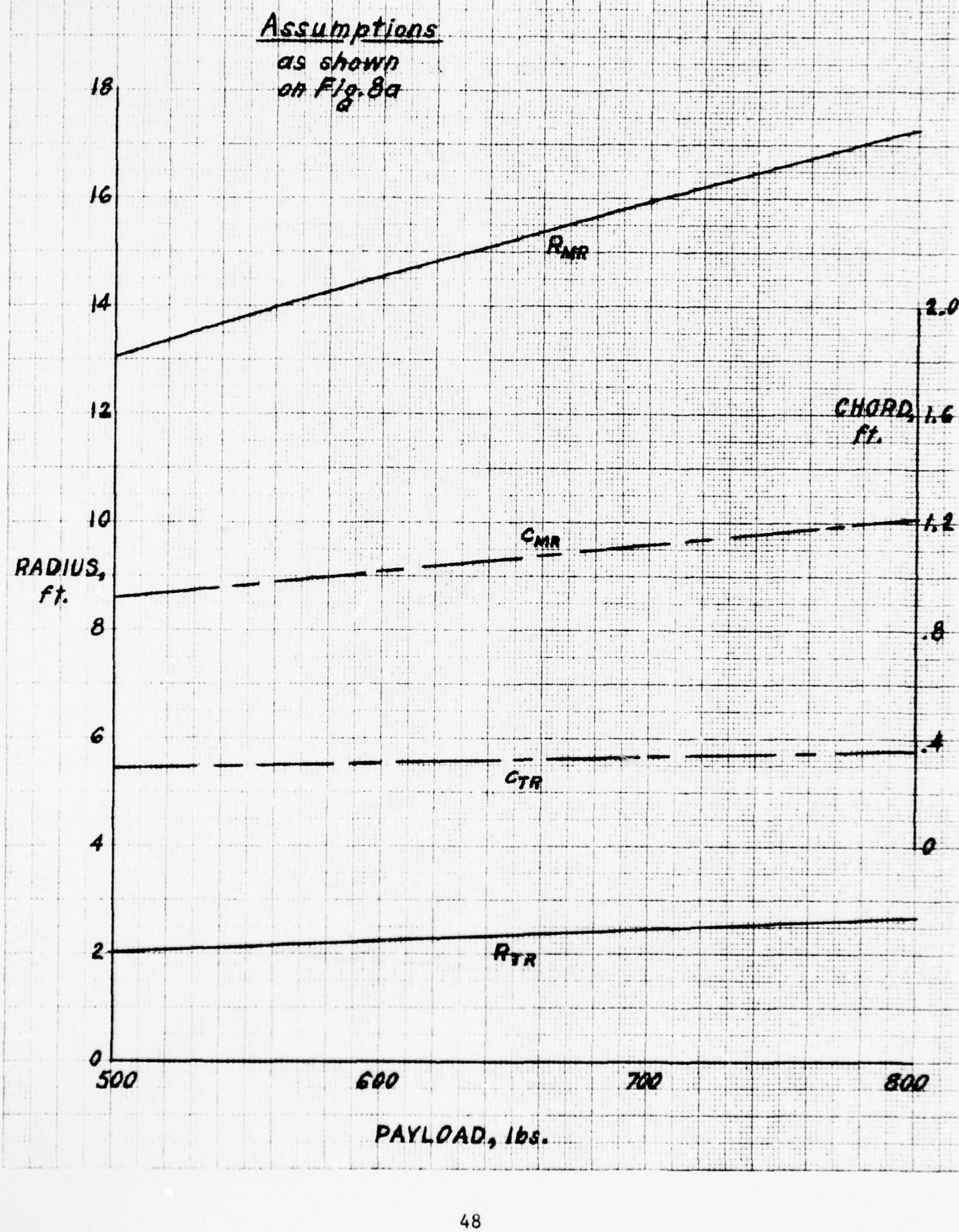


FIG. 8b. SAMPLE OUTPUT OF PRELIMINARY DESIGN PROGRAM (SSP-1) FOR A RANGE OF PAYLOADS



APPENDIX A

GROSS WEIGHT ITERATION SCHEME

The gross weight iteration scheme of SSP-1 is based on the assumptions that the fuel weight to gross weight ratio and the empty weight to gross weight ratio of a given aircraft design are independent of moderate gross weight variations. If we then define the gross weight as

$$W_G = W_E + W_F + W_P$$

it follows from the above assumptions that

$$(W_G)_{i+1} = \left(\frac{W_E}{W_G}\right)_i (W_G)_{i+1} + \left(\frac{W_F}{W_G}\right)_i (W_G)_{i+1} + (W_P)_{i+1}$$

where i represents the ith iteration for $i = 0, 1, 2, \ldots\ldots$

then:

$$(W_G)_{i+1}\left[1 - \left(\frac{W_E}{W_G}\right)_i - \left(\frac{W_F}{W_G}\right)_i\right] = (W_P)_{i+1}$$

or:

$$(W_G)_{i+1}\left[\frac{(W_P)_i}{(W_G)_i}\right] = (W_P)_{i+1}$$

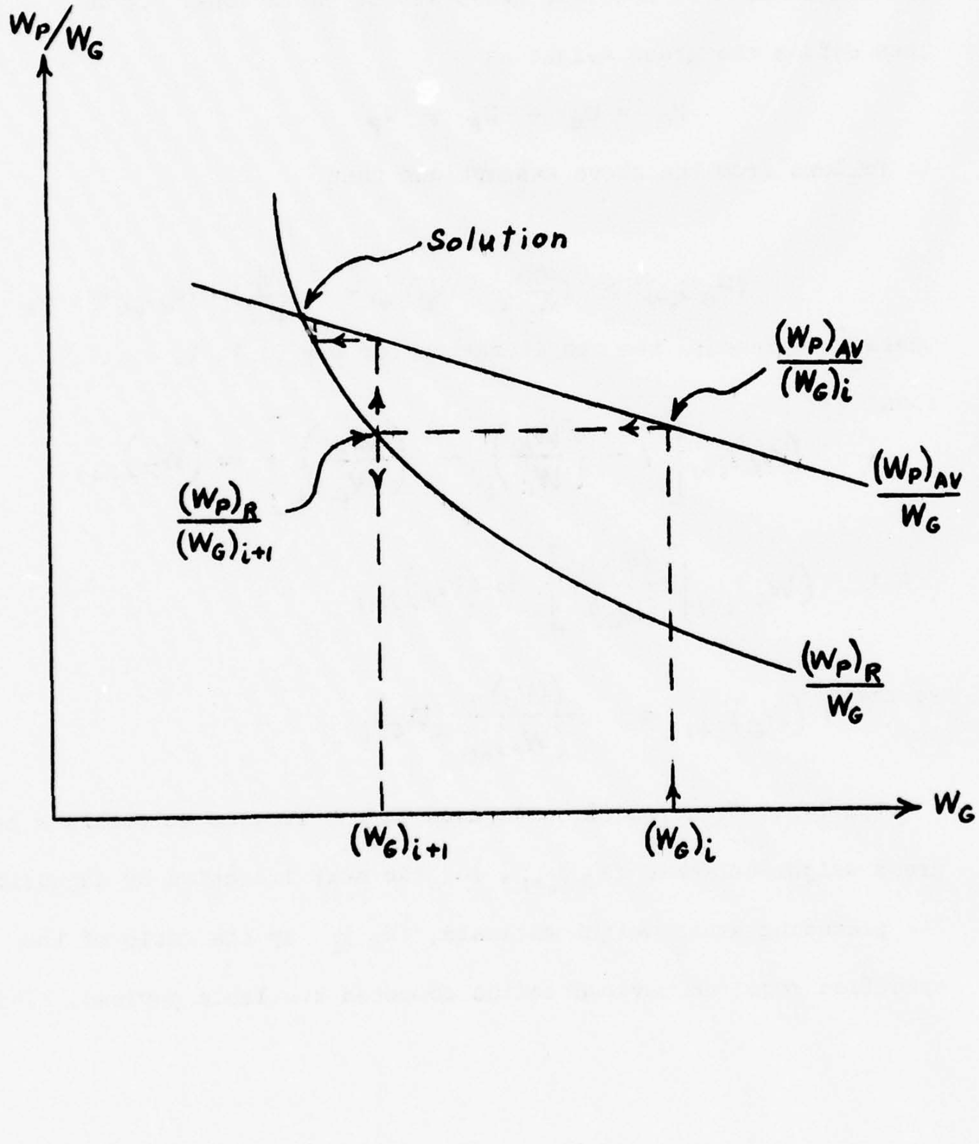
so that

$$(W_G)_{i+1} = \frac{(W_P)_R}{(W_P)_{AV}} (W_G)_i$$

The final equation above is used in the program to obtain a new gross weight estimate $(W_G)_{i+1}$, for the next iteration by adjusting the preceding gross weight estimate, $(W_G)_i$ by the ratio of the specified required payload to the computed available payload. This iteration process continues in the program until the relation

$$\left| \frac{(W_P)_R}{(W_P)_{AV}} - 1 \right| \leq 0.0001$$

is satisfied. The iteration procedure is illustrated in the sketch below.



APPENDIX B

PARASITE DRAG

The parasite drag, Dp, represents the drag force, opposed to the flight direction, of all elements of the aircraft other than the main rotor. It is expressed here as $D_p = fq$, where f is the effective parasite drag area of the aircraft components exclusive of the main rotor.

The current performance estimation program assumes that an appropriate value for f will be provided as input by the user. To assist in the selection of a value for this parameter, where a specific one is not available, flight test results obtained for a wide variety of helicopters are presented in Figure B-1. An average of these data is represented by

$$f = 0.047(W)^{2/3}.$$

It is also apparent that this equation is approximately the middle value of the flight test data available.

In addition to providing absolute values of f where none are available, Figure B-1 suggests a trend for f with W as a particular design is scaled either up or down. That trend is included in the current preliminary design program, where the constant in the equation must be supplied as an input.

The recommendation that $f \sim (W)^{2/3}$ on the basis of the flight test data is compatible with the following rationale:

Aircraft weight is proportional to the aircraft's volume, which in turn can be related to the cube of the aircraft's linear dimensions. Similarly, the aircraft's surface area is proportional to the square of its linear dimensions. Therefore, $W \sim x^3$; $S_w \sim x^2$, so that, $S_w \sim (W)^{2/3}$. Parasite drag can be expressed as $D = C_f \; S_w q$, as well as $D = fq$. Therefore, $f = C_f \; S_w$ or, $f \sim (W)^{2/3}$.

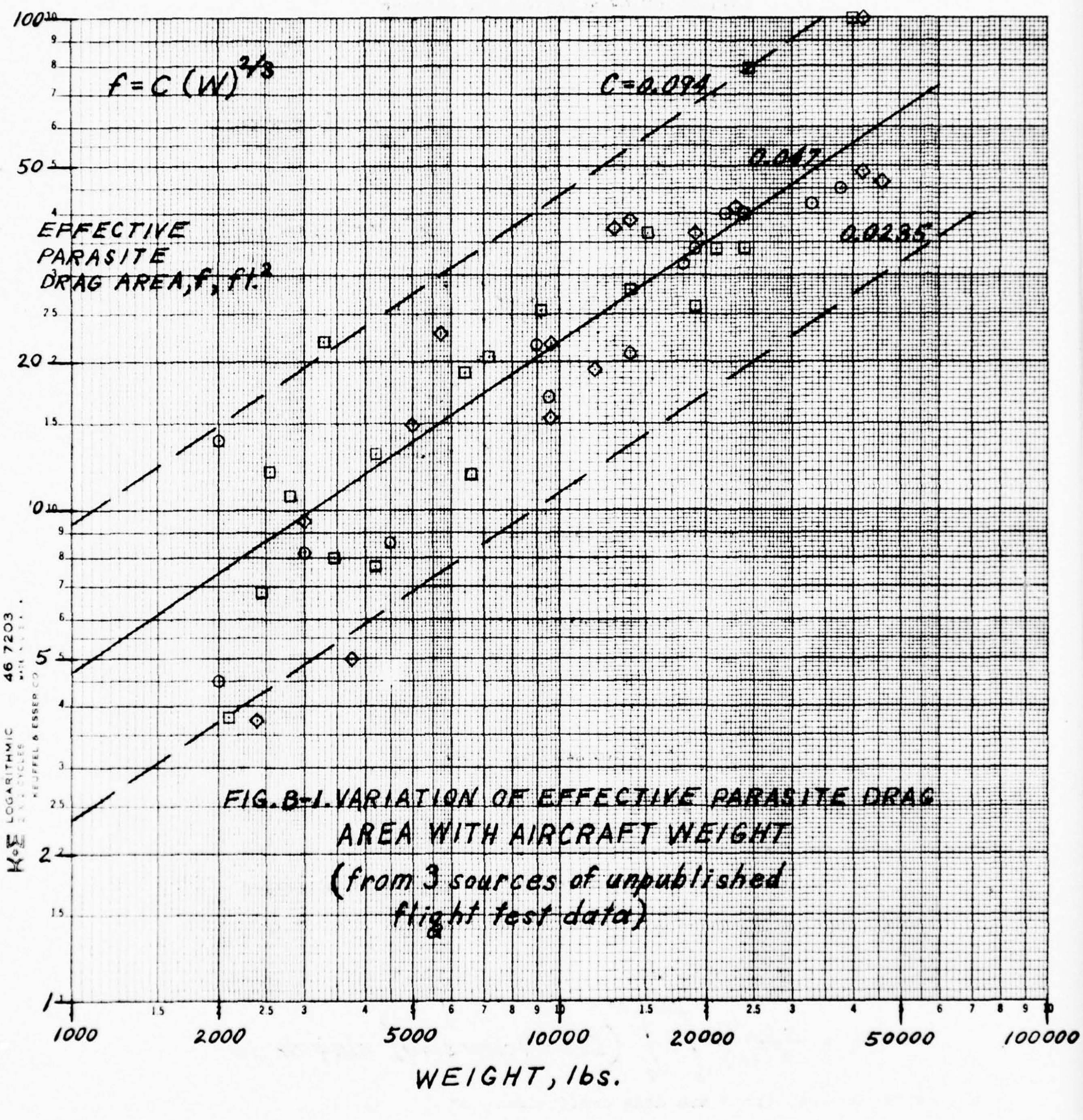
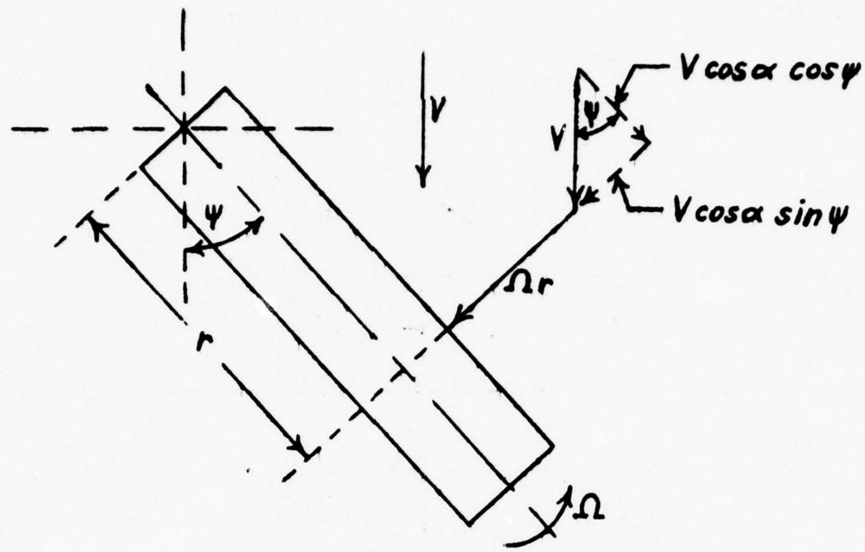


FIG. B-1. VARIATION OF EFFECTIVE PARASITE DRAG AREA WITH AIRCRAFT WEIGHT (from 3 sources of unpublished flight test data)

APPENDIX C

DERIVATION OF EQUATION FOR H-FORCE



The velocity normal to the blade is

$$U = \Omega r + V \cos\alpha \sin\psi \,.$$

The blade profile drag is, then,

$$dD_o = \frac{1}{2}\rho(\Omega r + V\cos\alpha \sin\psi)^2 C_D \;\; c \; dr$$

for an element of the blade.

The component of this drag force in the downstream direction contributes to the H-force,

$$dH_o = dD_o \sin\psi$$

$$\therefore \; H_o = \frac{b c \rho C_D}{2(2\pi)} \int_0^{2\pi}\int_0^{R} (\Omega r + V\cos\alpha \sin\psi)^2 \sin\psi \; dr \, d\psi$$

for a constant chord and drag coefficient, or

$$H_o = \frac{1}{4} \nabla C_D \rho \mu \pi R^2 v_T^2$$

The radial component of the velocity contributes a drag which has a component in the downstream direction. This component is shown in Ref. 11, to be

$$\Delta H_o = \frac{1.65}{8} \rho \nabla \pi R^2 \mu C_D v_T^2$$

$$\therefore \quad H = H_o + \Delta H_o$$

$$\text{or,} \quad H = \frac{3.65}{8} \pi \rho \nabla C_D v_T^2 R^2 \mu$$

APPENDIX D

DERIVATION OF EQUATION FOR ROTOR MEAN BLADE LIFT COEFFICIENT

The total thrust on a rotor can be defined as

$$T_R = \frac{b}{2\pi}\int_0^{2\pi}\int_{XR}^{BR} c\,\bar{C}_L \frac{\rho}{2}(U_T)^2\,dr\,d\psi$$

where it has been assumed that $U_T = U$.

Since, $U_T = \Omega r + \mu\Omega R \sin\psi$

then, $U_T^2 = (\Omega r)^2 + 2\mu\Omega^2 r R\sin\psi + \mu^2(\Omega R)^2\sin^2\psi$.

Then, assuming a constant chord,

$$T_R = \frac{bc\bar{C}_L\rho}{4\pi}\int_0^{2\pi}\int_{XR}^{BR}\left[(\Omega r)^2 + 2\mu\Omega^2 r R\sin\psi + \mu^2(\Omega R)^2\sin^2\psi\right]dr\,d\psi$$

$$\therefore\ T_R = \frac{bc\bar{C}_L\rho\Omega^2R^3}{6}\left[(B^3 - X^3) + 3/2\,\mu^2(B - X)\right]$$

Then, solving for the mean lift coefficient, with the thrust expressed in coefficient form,

$$\bar{C}_L = \frac{6\,C_T}{\sigma\left[B^3 - X^3 + 3/2\,\mu^2(B - X)\right]} .$$

At $\mu = 0$, this expression reduces to the equation used for $\bar{C}_L$ in hover,

$$\bar{C}_L = \frac{6\,C_T}{\sigma(B^3 - X^3)} .$$

It should be noted that the above derivation did not account for the existence of a reversed flow region on the retreating side of the rotor in forward flight. This can be done and will result in a slight modification of the equation for $\overline{C}_L$.

APPENDIX E

TAIL ROTOR SIZING RELATIONSHIPS

As part of the preliminary design program, SSP-1, three geometric parameters associated with the aircraft's tail rotor are computed in the course of the program's operation. These are the tail rotor radius, arm, and chord. The sizing relationships incorporated in the program for this purpose were determined empirically from available geometric information on existing aircraft. The relations used are presented in the following paragraphs:

a. Tail Rotor Radius:

Statistical data gathered for current aircraft have been correlated to provide a basis for estimating the required tail rotor radius. As shown in Figure E-1, these data correlate reasonably well when related to the main rotor radius and disk loading. The relation used in the current program is

$$R_{TR} = \frac{R_{MR}}{7.25 - .27(DL)_{MR}}$$

b. Tail Rotor Arm:

The tail rotor arm is defined as the longitudinal distance between the centerlines of the main rotor and tail rotor hubs. In general, this distance is closely approximated by the sum of the main and tail rotor radii and a 6-inch gap allowed for clearance between the rotors, $\ell_{TR} = R_{MR} + R_{TR} + 0.5$ feet.

c. Tail Rotor Chord:

A correlation of statistical data pertaining to tail rotor chord size has been made as shown in Figure E-2. The correlating parameter used here, $c_{TR} \sim \frac{HP_{MR}}{D_{MR}\, b_{TR}}$, is a simplified expression of the primary influencing variables as deduced from a comprehensive equation for computing the required tail rotor chord.

Since statistical information normally provides the engine horsepower at sea level standard day conditions only, that value has been used in the correlation on Figure E-2. However, for program purposes, an approximation for tail rotor power, transmission power losses, and engine power losses has been made that leads to the following relation,

$$C_{TR} = 0.1 + 0.0592 \, \frac{HP_{MR} \times C_9}{D_{MR} \;\; b_{TR}}$$

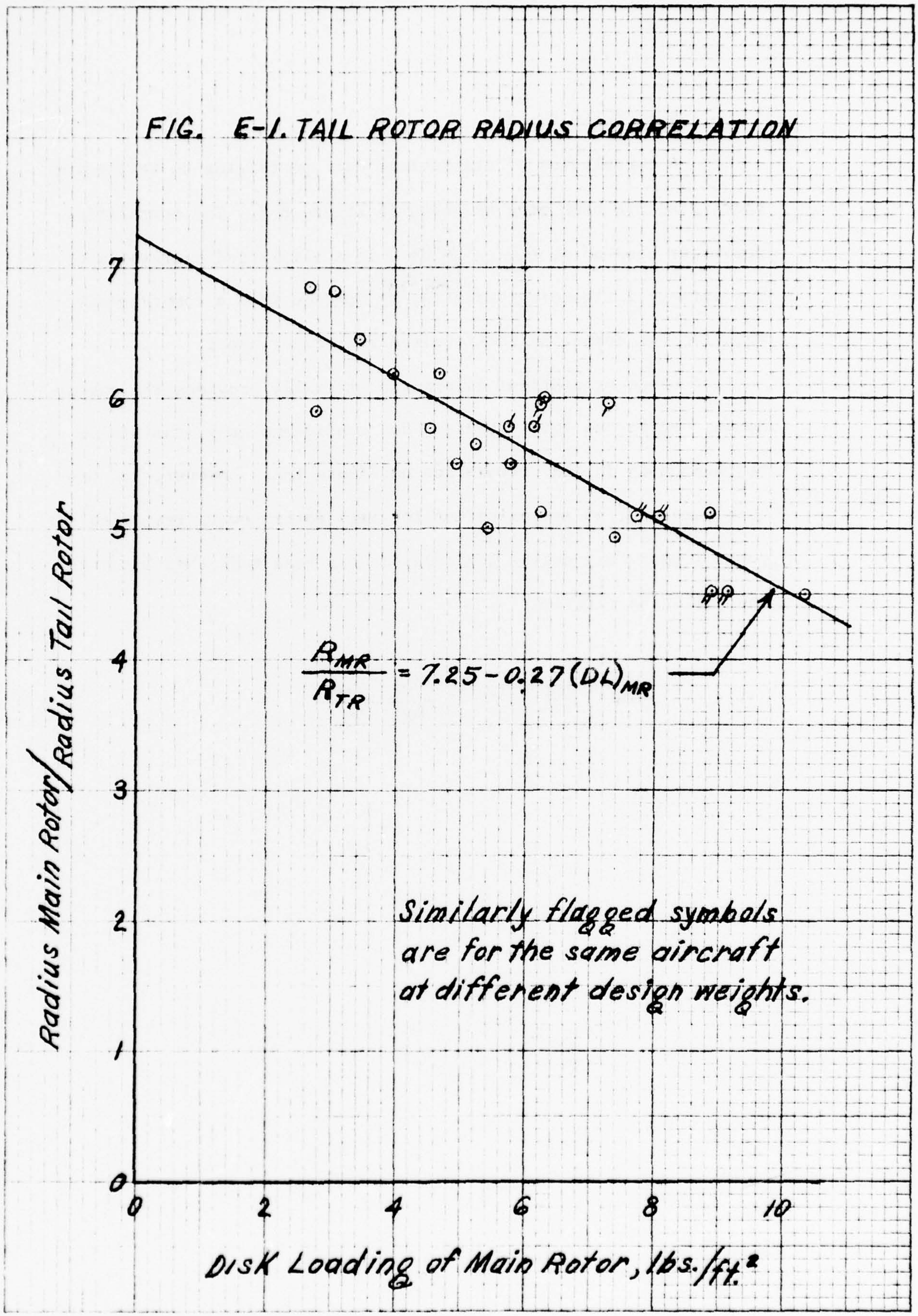


FIG. E-1. TAIL ROTOR RADIUS CORRELATION
Radius Main Rotor/Radius Tail Rotor
0
1
2
3
4
5
6
7
$\frac{R_{MR}}{R_{TR}} = 7.25 - 0.27(DL)_{MR}$
Similarly flagged symbols are for the same aircraft at different design weights.
0
2
4
6
8
10
Disk Loading of Main Rotor, lbs./ft.²

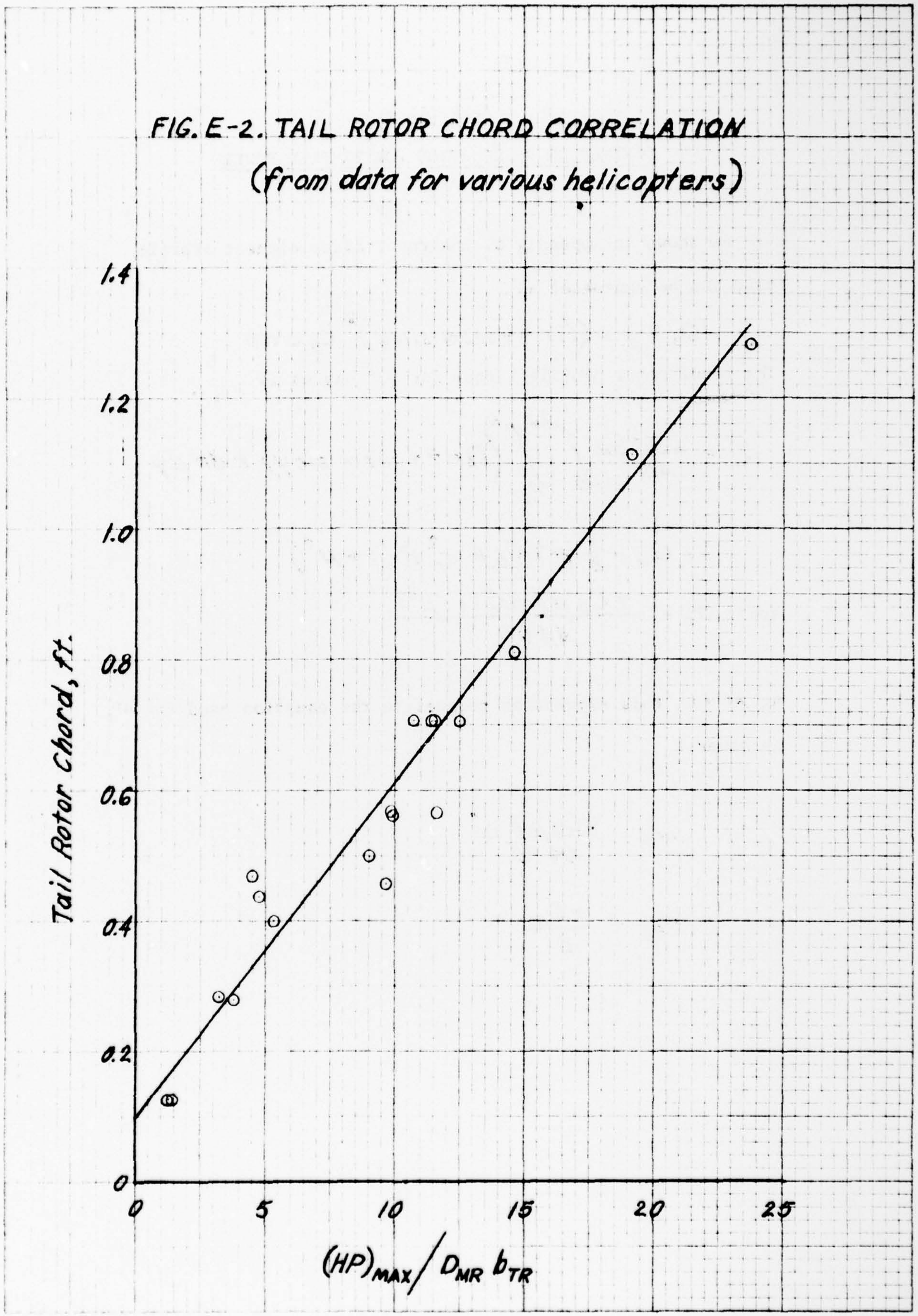


FIG. E-2. TAIL ROTOR CHORD CORRELATION
(from data for various helicopters)
Tail Rotor Chord, ft.
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0
0
5
10
15
20
25
$(HP)_{MAX} / D_{MR}\, b_{TR}$

APPENDIX F

DERIVATION OF EQUATION FOR PROFILE POWER

As shown in Appendix C, the rotor blade element profile drag can be expressed as

$$dD_o = \frac{1}{2}\rho(\Omega r + V\cos\alpha\,\sin\psi)^2\,C_D\,c\,dr$$

The total rotor profile torque for 'b' blades is

$$Q_o = \frac{bc\rho C_D}{4\pi}\int_0^{2\pi}\int_0^R(\Omega r + V\cos\alpha\,\sin\psi)^2\,r\,dr\,d\psi$$

$$\therefore\ Q_o = \frac{1}{8}\rho\,\sigma\,C_D\,A\,v_T^2\,R(1+\mu^2)$$

and: $$HP_o = \frac{\sigma C_D\rho A v_T^3(1+\mu^2)}{4400}$$

At $\mu = 0$, this expression reduces to the equation used for HP_o in hover,

$$HP_o = \frac{\sigma C_D\rho A v_T^3}{4400}$$

or $$C_{P_o} = \frac{\sigma C_D}{8}$$

APPENDIX G

DERIVATION OF EQUATION FOR ROTOR INFLOW ANGLE AT BLADE TIP AND AZIMUTH ANGLE OF 270°

For small values of the rotor inflow angle, $\emptyset$, and neglecting blade flapping harmonics of the second and higher orders,

$$\emptyset \approx \frac{U_P}{U_T} \approx \frac{(V\sin\alpha - u)\cos\beta - r\dot{\beta} - V\cos\alpha\cos\psi\sin\beta}{\Omega r + V\cos\alpha\sin\psi} \quad \text{from Ref. 7}$$

or,
$$\emptyset \approx \frac{\lambda\Omega R\cos\beta - r\dot{\beta} - \mu\Omega R\cos\psi\sin\beta}{\Omega r + \mu\Omega R\sin\psi}$$

where: $\lambda = \dfrac{V\sin\alpha - u}{\Omega R}$ and $\mu = \dfrac{V\cos\alpha}{\Omega R}$

For small values of the flapping angle, β : $\cos\beta \approx 1$, $\sin\beta \approx \beta$.

$$\therefore \phi \approx \frac{\lambda\Omega R - r\dot{\beta} - \mu\Omega R\beta\cos\psi}{\Omega r + \mu\Omega R\sin\psi}$$

Then, for: $\beta = a_0 - a_1\cos\psi - b_1\sin\psi$

and, $\dot{\beta} = \Omega\,(a_1\sin\psi - b_1\cos\psi)$

$$\emptyset \approx \frac{\lambda\Omega R - \Omega r a_1\sin\psi + \Omega r b_1\cos\psi - \mu\Omega R\beta\cos\psi}{\Omega r + \mu\Omega R\sin\psi}$$

At $\psi = 270°$: $\cos\psi = 0$, $\sin\psi = -1$

$$\therefore (\phi)_{270°} \approx \frac{\lambda\Omega R + \Omega r a_1}{\Omega r}$$

Then, at the blade tip, r = R, and:

$$(\emptyset_{tip})_{270°} \approx \frac{\lambda\Omega R + \Omega R a_1}{\Omega R - \mu\Omega R}$$

$$\therefore (\emptyset_{tip})_{270°} \approx \frac{\lambda + a_1}{1 - \mu}$$

FLIGHT HYDRAULIC AIR-CONDIT FURNISHINGS

APPENDIX H

DERIVATION OF EQUATION FOR ROTOR LONGITUDINAL FLAPPING COEFFICIENT

An equation for the longitudinal flapping coefficient is derived in Reference 7 for a rotor of constant chord and no spanwise twist:

$$a_1 = \frac{\mu\left(\frac{8}{3}\theta + 2\lambda\right)}{1 - \frac{1}{2}\mu^2}$$

The same method used to derive this expression can be applied to the case of a blade with linear spanwise twist distribution,

$$\theta = \theta_0 + \theta_1 (r/R)$$

In this case the expression for rotor thrust moment becomes

$$M_T = \frac{1}{2}\rho a c \Omega^2 \int_0^R \Big[\left(\theta_0 r^3 + \frac{1}{2}\theta_0 \mu^2 R^2 r + \lambda R r^2 + \frac{1}{2}\theta_1 \mu^2 R r^2\right)$$

$$+ \sin\psi\left(2\mu R r^2 \theta_0 + \mu\lambda R^2 r - a_1 r^3 + \frac{1}{4}\mu^2 R^2 r a_1 + 2\mu r^3 \theta_1\right)$$

$$+ \cos\psi\left(r^3 b_1 - \mu R r^2 a_0 + \frac{1}{4}\mu^2 R^2 r b_1\right)\Big] dr$$

Integrating, and setting the coefficient of the sine term equal to zero, as in Reference 7:

$$\sin\psi\left[2\mu\frac{R^4}{3}\theta_0+\frac{1}{2}\mu\lambda R^4-\frac{R^4}{4}a_1+\frac{1}{8}\mu^2R^4a_1+\frac{1}{2}\mu R^4\theta_1\right]=0$$

$$\text{then: } a_1\left[\frac{1}{4}-\frac{1}{8}\mu^2\right]=\mu\left[\frac{2}{3}\theta_0+\frac{1}{2}\theta_1+\frac{\lambda}{2}\right]$$

$$\therefore\quad a_1=\frac{\mu\left[\frac{8}{3}\theta_0+2\theta_1+2\lambda\right]}{1-\frac{\mu^2}{2}}$$

APPENDIX I

DERIVATION OF EQUATION FOR ROTOR BLADE COLLECTIVE PITCH SETTING REQUIRED TO DEVELOP A GIVEN LEVEL OF ROTOR THRUST

An equation for rotor thrust that accounts for the negative lift effects on the rotor in the reversed flow region of the retreating blade and blade flapping harmonics of the second order is derived in Reference 12. That equation also allows for a linear spanwise distribution of blade twist and the assumption of a tip loss effect.

A similar derivation has been made for the present purposes; but, with the relatively minor effects of second order flapping harmonics and the reversed flow region neglected. An allowance has been added for possible blade root cutouts.

Then: $$T = \frac{b}{2\pi}\int_0^{2\pi} d\psi \int_{XR}^{BR} \frac{dT}{dr}\, dr$$

For a constant blade chord, and linear blade twist, $\theta = \theta_o + \theta_1 \frac{r}{R}$, the integration yields the expression for rotor thrust:

$$T = \frac{1}{2}\rho a b c \Omega^2 R^3 \left\{ \theta_o \left[\frac{B^3 - X^3}{3} + \frac{\mu^2 (B - X)}{2} \right] + \frac{\theta_1}{4}\left[(B^4 - X^4) + \mu^2 (B^2 - X^2) \right] + \frac{\lambda}{2}(B^2 - X^2) \right\}$$

SYMBOLS

A Planform area or disk area, square feet

Or, in coefficient form,

$$C_T = \frac{1}{2} a \sigma \left\{ \theta_o \left[\frac{B^3 - X^3}{3} + \frac{\mu^2 (B - X)}{2} \right] + \frac{\theta_1}{4} \left[(B^4 - X^4) + \mu^2 (B^2 - X^2) \right] + \frac{\lambda}{2} (B^2 - X^2) \right\}$$

This equation can be rearranged to provide a solution for the collective pitch setting required for a given rotor thrust:

$$\theta_o = \frac{\frac{12 C_T}{a \sigma} - 3(B^2 - X^2)\left[\lambda + \frac{\theta_1}{2}(\mu^2 + B^2 + X^2)\right]}{2(B^3 - X^3) + 3\mu^2 (B - X)}$$

D_p Parasite drag, pounds

APPENDIX J

AIRCRAFT GEOMETRY

AND

GROUP WEIGHTS DATA

NO	MANUF	AIRCRAFT	NO OF ROTORS	ROTOR TYPE	NO OF BLADES PER ROTOR
1	Bell	UH-1B	1	Teetering	2
2	Bell	UH-1C	1	Teetering	2
3	Bell	UH-1D	1	Teetering	2
4	Bell	UH-1E	1	Teetering	2
5	Bell	UH-1F	1	Teetering	2
6	Bell	UH-1G	1	Teetering	2
7	Bell	UH-1H	1	Teetering	2
8	Bell	UH-X3	1	Teetering	2
9	Bell	OH-4A	1	Teetering	2
10	Bell	OH-58A	1	Teetering	2
11	Bell	OH-13S	1	Teetering	2
12	Bell	AH-1G	1	Teetering	2
13	Bell	AH-1J	1	Teetering	2
14	Bell	AH-16C	1	Teetering	2
15	Bell	AH-16T	1	Teetering	2
16	Bell	TH-57A	1	Teetering	2
17	Vertol	CH-46A	2	Articulated	3
18	Vertol	CH-46D	2	Articulated	3
19	Vertol	CH-46F	2	Articulated	3
20	Vertol	CH-47A	2	Articulated	3
21	Vertol	CH-47B	2	Articulated	3
22	Vertol	CH-47C	2	Articulated	3

NO	MANUF	AIRCRAFT	NO OF ROTORS	ROTOR TYPE	NO OF BLADES PER ROTOR
23	Vertol	YH-16A	2	Articulated	3
24	Vertol	CH-21C	2	Articulated	3
25	MBB	BO-105	1	Rigid	4
26	Vertol	347	2	Articulated	4
27	Vertol	HLH	2	Articulated	4
28	Sikorsky	SH-3A	1	Articulated	5
29	Sikorsky	SH-3D	1	Articulated	5
30	Sikorsky	CH-3C	1	Articulated	5
31	Sikorsky	CH-3E	1	Articulated	5
32	Sikorsky	CH34A	1	Articulated	4
33	Sikorsky	CH-37A	1	Articulated	5
34	Sikorsky	UH-19D	1	Articulated	3
35	Sikorsky	H-52A	1	Articulated	3
36	Sikorsky	CH-53A	1	Articulated	6
37	Sikorsky	CH-53A	1	Articulated	6
38	Sikorsky	CH-53C	1	Articulated	6
39	Sikorsky	CH-53D	1	Articulated	6
40	Sikorsky	CH-53E	1	Articulated	7
41	Sikorsky	CH-54A	1	Articulated	6
42	Sikorsky	CH-54B	1	Articulated	6
43	Sikorsky	S-67	1	Articulated	5
44	Sikorsky	AH-X42	1	Articulated	5
45	Sikorsky	HLH	1	Articulated	4
46	Hughes	OH-6A	1	Articulated	4
47	Hughes	OH-6C	1	Articulated	5
48	Hughes	OH-X24	1	Articulated	5
49	Hughes	XV-9A	1	Articulated	3

NO	MANUF	AIRCRAFT	NO OF ROTORS	ROTOR TYPE	NO OF BLADES PER ROTOR
50	Hughes	XH-17	1	Articulated	2
51	Hughes	XH-28	1	Articulated	4
52	Hughes	269A	1	Articulated	3
53	Hughes	HLH	1	Articulated	4
54	Kaman	UH-2B	1	Articulated	4
55	Kaman	UH-2D	1	Articulated	4
56	Kaman	H-43B	2	Teetering	2
57	Kaman	OH-43D	2	Teetering	2
58	Lockheed	XH-51A	1	Rigid	3
59	Lockhead	286	1	Rigid	4

NO	ROTOR CHORD	ROTOR RADIUS	ROTOR RPM	MAX HP	DRIVE SYSTEM HP	GROSS WEIGHT
1	1.75 ft	22 ft	324	1100	1100	6600
2	2.25	22	324	1100	1100	6600
3	1.75	24	324	1100	1100	6600
4	2.25	22	324	1100	1100	6600
5	1.75	24	324	1100	1100	6600
6	1.75	24	324	1300	1100	6600
7	1.75	24	324	1530	1250	6600
8	2.25	25	296	2050	2050	9500
9	1.09	16.7	354	250	250	2900
10	1.09	17.7	354	317	317	3000
11	1.00	16.7	355	260	260	2850
12	2.25	22	324	1300	1100	6600
13	2.25	22	324	1530	1250	6600
14	2.75	24	296	2050	2050	10000
15	2.75	24	305	2050	2050	13000
16	1.09	16.7	354	317	317	2900
17	1.50	25	249	2300	2760	19000
18	1.56	25.5	264.3	2600	3120	20800
19	1.56	25.5	264.3	2600	3120	20800
20	1.93	29.5	223	4400	5280	33000
21	2.10	30	230	4970	5964	33000
22	2.10	30	243	6000	7200	33000
23	2.33	41	147	3600	3300	34000
24	1.50	22	278	1425	1425	13300
25	0.89	16.1	425	600	600	4630
26	2.10	32.5	245	7500	9000	42500
27	3.00	45	159	16200	9720	117000

NO	ROTOR CHORD	ROTOR RADIUS	ROTOR RPM	MAX HP	DRIVE SYSTEM HP	GROSS WEIGHT
28	1.52	31	202.9	2500	2500	18064
29	1.52	31	202.9	2500	2500	20500
30	1.52	31	209	2500	2500	19500
31	1.52	31	209	2800	2575	19500
32	1.37	28	221	1525	1525	11867
33	1.97	36	185	4200	4000	30342
34	1.37	26.5	212	800	800	7100
35	1.37	26.5	212	1050	730	7500
36	2.17	36.1	185	5700	6000	33500
37	2.15	36.1	185	5700	6000	33500
38	2.17	36.1	185	6870	6400	33500
39	2.17	36.1	185	7850	7560	33500
40	2.42	39.5	177	13128	11340	45000
41	1.97	36	186	9600	6600	38000
42	2.17	36.1	186	9600	7900	47000
43	1.52	31	211	3000	2800	17300
44	1.82	31	211	3060	3060	18900
45	4.31	62	115.7			118055
46	0.56	13.2	483	250	250	2400
47	0.56	13.2	497	400	400	3150
48	0.56	14.2	475	400	400	3200
49	2.63	27.6	255	3600		15300
50	5.66	65	90.5			35000
51	5.67	65	90.5			84000
52	0.56	12.5	483.3	180	180	1600
53	5.00	58	115			105000
54	1.80	22	277	1250	1250	7378

NO	ROTOR CHORD	ROTOR RADIUS	ROTOR RPM	MAX HP	DRIVE SYSTEM HP	GROSS WEIGHT
55	1.80	22	288	2500	2500	10187
56	1.31	23.5	235	860	860	6418
57	1.31	23.5	235	600	860	5345
58	1.13	17.5	354	450	550	3500
59	1.13	17.5	354	550	550	4700

NO	ROTOR	BODY	TAIL	LANDING GEAR *	PROPULSION	TRANSMISSION
1	756	858	72.5	106	870	623
2	939	878	94.2	109	874	622
3	742	1038	91.2	121	939	640
4	931	881	103.6	109	860	636
5	740	894	82	106	772	652
6	742	1044	83.9	121	934	660
7	784	1172	99	121	1289	803
8	927	1032	101	119	1111	794
9	250	359	18.5	43	256	159
10	281	332	32	35	240	215
11	284	221	18	54	727	155
12	945	971	103.5	123	926	672
13	951	1043	111.4	123	1167	801
14	1400	1327	151	134	1267	825
15	1515	1291	167.8	148	1234	918
16	277	335	34	45	252	176
17	1138	2910	0	592 w	1132	1933
18	1232	3114	0	592 w	1297	1942
19	1212	3126	0	591 w	1296	2010
20	1498	4487	0	1086 w	1796	3531
21	1674	4687	0	1060 w	1927	3558
22	1700	4730	0	1078 w	2558	3665
23	2269	5424	133	1244 w	4262	3679
24	672	1884	161.7	522 w	2067	1393
25	462	472	55.8	94	434	395
26	2527	6259	0	1114 w	3276	3796
27	5741	9256	0	7197 w	8349	8044

NO	ROTOR	BODY	TAIL	LANDING GEAR *	PROPULSION	TRANSMISSION
28	2328	2009	222.1	748 w	1080	1763
29	2344	1946	215.7	806 w	1283	1853
30	1909	3365	324	690 w	1122	1941
31	2063	3309	326.7	701 w	1600	1971
32	1313	1044	259.9	475 w	2248	1091
33	3251	3247	570.3	983 w	9198	2567
34	786	985	100.8	287 w		1064
35	785	1263	106.2	485 w	557	621
36	4489	5260	672.6	1019 w	2532	3919
37	4788	5145	667	958 w	2516	3865
38	3901	5127	667.9	1017 w	3540	3871
39	4505	5557	678.2	1027 w	2514	4173
40	6164	6491	1292	1177 w	3986	5708
41	4052	2685	519	1794 w	3126	3797
42	3970	2885	632.8	1718 w	3074	3955
43	2348	1695	468	656 w	1499	2123
44	2312	1942	563	617 w	1435	1910
45	12982	8115	986			
46	174	242	22.9	70	236	113
47	217	312	32.6	77	257	133
48	211	307	34.4	70	312	162
49	2805	878	136	476	2428	423
50	10710	4570	6.9	4658 w		
51	25157	4320	324	4554 w		
52	115	125	8.9	53	371	142
53	13700	5315	111			
54	1295	1259	96	343 w	895	733

NO	ROTOR	BODY	TAIL	LANDING GEAR *	PROPULSION	TRANSMISSION
55	1325	1394	216.3	424 w	1375	1361
56	422	903	135.5	187 w	886	730
57	341	774	101.2	144 w	1202	730
58	474	507	56.8	135	394	442
59	726	496	69	142	437	440

* Skid gear unless indicated as wheeled (w).

NO	FLIGHT CONTROLS	HYDRAULIC PNEUM ELEC	AIR-CONDIT & ANTI-ICING	INSTRUMENTS	FURNISHINGS & EQUIP
1	358	386	50	54	180
2	441	415	44	53	397
3	355	409	44	55	404
4	425	420	49	69	240
5	360	509	49	58	180
6	357	406	44	55	417
7	416	481	66	76	338
8	411	481	50	61	383
9					
10	125	85	25	27	42
11	153	130	40	24	30
12	387	279	86	115	82
13	396	436	42	115	210
14	469	510	76	101	130
15	481	374	108	95	277
16	133	110	27	29	64
17	810	770	303	172	833
18	827	787	252	160	1179
19	828	822	257	158	1050
20	1212	767	179	172	1124
21	1470	783	182	162	1851
22	1637	766	190	190	1881
23	1239	932	165	176	562
24	561	404	137	134	303
25	189	168	0	25	47
26	1921	793	202	195	1639
27	3154	1254	935	465	4645

NO	FLIGHT CONTROLS	HYDRAULIC PNEUM ELEC	AIR-CONDIT & ANTI-ICING	INSTRUMENTS	FURNISHINGS & EQUIP
28	654	437	109	368	456
29	688	434	130	430	451
30	613	516	160	233	758
31	618	534	182	407	745
32	378	353	72	108	192
33	965	626	176	191	822
34	164	374	77	70	205
35	353	462	97	124	305
36	1168	733	311	395	1728
37	1168	733	311	395	1728
38	1166	757	315	442	1306
39	1175	759	320	408	1681
40	1474	843	383	484	1870
41	1161	640	114	284	402
42	1165	775	119	264	459
43	780	437	126	187	243
44	886	441	151	192	419
45			0	0	0
46	65	68	10	30	58
47	67	74	10	33	66
48	110	134	15	40	100
49	958	369	0	53	107
50					
51					
52	51	59	11	8	33
53					
54	301	275	81	142	137

NO	FLIGHT CONTROLS	HYDRAULIC PNEUM ELEC	AIR-CONDIT * ANTI-ICING	INSTRUMENTS	FURNISHINGS & EQUIP
55	300	345	68	170	297
56					
57					
58	299	151	3	6	107
59	327	168	14	58	84

SYMBOLS

A	Planform area or disk area, square feet
a	Speed of sound, feet/second; section lift curve slope, per radian
a_1	Rotor longitudinal flapping coefficient
B	Rotor tip loss factor
b	Number of blades in rotor
b_1	Rotor lateral flapping coefficient
C	Proportionality constant in equation for parasite drag equivalent area
c	Chord of rotor blade section, feet
C_D	Drag coefficient
C_f	Effective skin friction coefficient
C_L	Lift coefficient
$\overline{C_L}$	Mean (average) lift coefficient
C_{L_α}	Lift curve slope
C_P	Power coefficient, $P/\rho A(\Omega R)^3$
C_T	Thrust coefficient, $T/\rho A(\Omega R)^2$
C_1	Constant in equation for engine fuel flow rate
C_2	Constant in equation for engine fuel flow rate
C_3	Engine lapse rate, $(HP)_{SLS}/(HP)_{Alt.}$
C_4	Exponent in equation for engine lapse rate at intermediate power rating
C_5	Exponent in equation for engine lapse rate at intermediate power rating
C_6	Exponent in equation for engine lapse rate at maximum continuous power rating
C_7	Exponent in equation for engine lapse rate at maximum continuous power rating
D	Rotor diameter, feet; drag, pounds
DL	Rotor disk loading, pounds/square foot

D_P	Parasite drag, pounds
D_V	Vertical drag, pounds
DRSYSPWR	Drive system power
f	Parasite drag equivalent area, square feet
H	H-force, component of resultant rotor force in plane perpendicular to control axis, pounds
h	Pressure altitude, feet
HP	Horsepower
K	Coefficient in equation for airfoil profile drag due to lift
K_F	Ratio of induced tail rotor flow velocity at vertical fin to final slipstream velocity of the tail rotor flow-field
K_u	Induced velocity factor for forward flight, u/u_H
K_{DF}	Factor used in the tail rotor power computation to allow for either an open or ducted tail rotor
K_{VC}	Induced velocity factor for vertical climb, u_{VC}/u_H
l	Tail arm, feet
M	Mach number
M_C	Critical Mach number
M_T	Rotor thrust moment, foot-pounds
P	Rotor shaft power, foot-pounds/second
Q	Rotor shaft torque, foot-pounds
q	Dynamic pressure, pounds/square foot
R	Rotor radius, feet; gas constant for air = 1718 square feet/(seconds)^2 (degrees Rankine)
r	Radial distance to a blade element, feet
SHP	Shaft Horsepower
S_P	Area of vertical fin projected onto tail rotor, square feet
S_W	Wetted area, square feet
T	Thrust, pounds; temperature, degrees Rankine

t	Time
u	Rotor induced velocity, feet/second
u_H	Rotor induced velocity in hover, feet/second
U	Total velocity at a blade element, feet/second
U_P	Velocity component (at a blade element) in direction of control axis, feet/second
U_T	Velocity component at a blade element (in plane perpendicular to control axis), feet/second
V	Forward flight velocity, feet/second
V_T	Rotor tip speed, feet/second
W	Weight, pounds
W_E	Empty weight, pounds
W_F	Fuel Weight, pounds
$\dot{W}_F$	Fuel flow rate, pounds/hour
W_G	Gross weight, pounds
W_P	Payload weight, pounds
X	Rotor root cutout
x	Representative linear dimension, feet
Y	Aircraft group weight, pounds
z	Rotor height above ground, feet
α	Angle of attack, radians
α_{OL}	Angle of attack for zero lift of airfoil, radians
$(\alpha_{tip})_{270°}$	Angle of attack at tip of rotor blade at 270° of azimuth
β	Rotor blade flapping angle, radians
γ	Ratio of specific heats
Δ	Increment
δ	Pressure ratio
ζ_{XMSN}	Percent of power absorbed in transmission system
η	Non-uniform inflow velocity factor

η_H	Non-uniform inflow velocity factor in hover
θ	Rotor blade pitch angle, radians; temperature ratio
θ_0	Rotor collective pitch angle, radians
θ_1	Rotor blade twist angle, radians
λ	Non-dimensionalized parameter for flow through the plane perpendicular to rotor control axis, $= (V \sin\alpha - u)/\Omega R$
Λ	Induced horsepower ratio $= (HP_i)_{IGE}/(HP_i)_{OGE}$
μ	Non-dimensionalized parameter for flow in the plane perpendicular to rotor control axis, $= V \cos\alpha/\Omega R$
ξ	Percentage of engine power supplied to those accessory items which place a variable demand on the engine
ρ	Mass density of air, slugs/cubic foot
σ	Rotor solidity $= bc/\pi R$
ϕ	Rotor inflow angle, radians
ψ	Rotor blade azimuth angle, radians
Ω	Rotor angular velocity, rad/sec

SUBSCRIPTS:

A	Accessories
Alt	Altitude
AT	Advancing rotor blade tip
AV	Available
CMP	Compressibility
F	Vertical fin
FP	Flat plate
H	Horizontal component; hover
i	Induced; Iteration step
IGE	In Ground Effect
IRP	Intermediate Rated Power

MR	Main Rotor
MAX	Maximum
o	Profile
OGE	Out of Ground Effect
P	Parasite
R	Required
S	Stall
SLS	Sea level standard day conditions
TR	Tail Rotor
V	Vertical Component
VC	Vertical Climb
XMSN	Power transmission system (shafts and gears)