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AFFDL-TR-71-78

DESIGN CRITERIA FOR HIGH-AUTHORITY CLOSED-LOOP PRIMARY FLIGHT CONTROL SYSTEMS

R. C. Hendrick, A. J. Bailey, L. D. Edinger, et al.

Honeywell Inc.

TECHNICAL REPORT AFFDL-TR-71-78

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Air Force Flight Dynamics Laboratory
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

FOREWORD

This document is the final technical report on a study titled "Design Criteria for High-Authority Closed-Loop Primary Flight Control Systems". The work was performed from June 1970 to June 1971 by the Government and Aeronautical Products Division of Honeywell Inc., Minneapolis, Minnesota, under Air Force Contract F33615-70-C-1293. The contract was initiated under Project 8226, "Experimental Synthesis and Demonstration of Flight Control System Techniques for Military Aircraft," Task 822601, "Advanced Flight Augmentation System Techniques for Tactical and Strategic Aircraft." The sponsoring agency was the Air Force Flight Dynamics Laboratory (AFFDL/FGL), Wright-Patterson Air Force Base, Ohio 45433, with Duane P. Rubertus as Project Engineer.

Honeywell engineering personnel who performed the study and contributed to the report include R. C. Hendrick, A. J. Bailey, L. D. Edinger, C. L. Kuivanen, R. F. Rasmussen, and C. R. Zimmer. A portion of the study included a survey of operational problems, involving many Air Force personnel, both military and civilian. The majority of these people are listed in Appendix I along with their areas of specialty. They contributed much valuable data and advice and were most receptive and cooperative. This support is gratefully acknowledged. Also to be recognized are technical contributions from the airframe and actuator manufacturers in the area of flight control actuator design data. Detail information on their products adds materially to the value of this report as a reference for advanced flight control design.

The report was submitted by the authors in May 1971. The associated Honeywell report number is 21572-FR.

This technical report has been reviewed and is approved.

George H Purcell

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DESIGN CRITERIA FOR HIGH-AUTHORITY CLOSED-LOOP PRIMARY FLIGHT CONTROL SYSTEM

R. C. Hendrick, A. J. Bailey, L. D. Edinger, et al.

ABSTRACT

A study to develop improved design criteria for primary flight controls which feature feedback techniques is reported. The study consisted of eight parts, including a survey of operational problems, a review of system-gain changing (requirements and techniques), stabilization criteria for high-frequency control modes, an analysis of stall/spin maneuvers, a catalog of dominant performance characteristics which affect flying qualities, an analysis of system/airframe compatibility testing, definition of criteria for built-in test equipment, and a catalog of flight control actuator designs. Operational problems include high angle-of-attack stability and potential control loss. A math model of a spinning F-4 was used to study basic effects and associated control criteria. Nominal control laws in pitch and yaw tended to be beneficial for departure inhibition; roll control degraded controllability. Spin recovery demands full surface deployment without detraction by feedback. The controllability limit for spin recovery was defined. The compatibility test analysis featured closed-loop simulation of structural response. Compensation for surface aerodynamics and special airframe support to avoid bending mode distortion were justified. Criteria for built-in test equipment in redundant flight controls to produce adequate flight safety and mission reliability were expressed in terms of test thoroughness, latent failure probabilities, and false indication rate. Test quality was shown to have a highly significant effect on system reliability that becomes more critical with the number of redundant channels and system life.

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SECTION I

INTRODUCTION

High-authority feedback control techniques offer new dimensions of performance capability for advanced military aircraft. In addition to the historical benefits of airframe dynamic augmentation, contributions in areas of structural efficiency, mission-optimum control modes, combat survivability, control retention, flight safety, and mission reliability are evident. The potential of these benefits has generated wide industry interest and related development efforts, in particular the USAF fly-by-wire and control-configured-vehicle programs.

Also recognized is the need for associated studies in a number of key technologies which influence the design criteria applicable to the high-authority closed-loop Primary Flight Control System (PFCS). The Air Force Flight Dynamics Laboratory responded to this need by defining appropriate areas of investigation, conducting a competitive procurement of the necessary technical services, and administering the program reported by this document.

The program is made up of the following areas of investigation:

(1) Survey of Operational Problems

(2) Review of System Gain-Changing Requirements and Techniques

(3) Stabilization Criteria for Structural Flexure and Other High-Frequency Dynamics

(4) Analysis of Stall/Spin Maneuvers

(5) Control System Characteristics Affecting Flying Qualities

(6) System/Aircraft Compatibility Testing

(7) Criteria for Built-In Test Equipment

(8) Flight Control Actuator Designs

Each of the above topics constitutes a section of this report. Section II, Summary, describes each study and presents significant results.

After concluding the overall investigation, MIL-F-9490C (USAF) was reviewed to assess potential revisions in light of the study. The resulting comments are documented in Appendix III.

SECTION II
SUMMARY

The contents of the eight areas of investigation are briefly summarized in the following paragraphs.

SURVEY OF OPERATIONAL PROBLEMS

The objective of this activity was to obtain data on existing flight control operational performance and problem areas. Such information would provide design guidance for future systems and be of value to the other study efforts. Eleven USAF organizations were visited during the program, including flight test groups, operational squadrons, and air material areas. Personal interviews and the USAF 66-1 Failure Reporting System were major sources of information.

The acquired results may be classified into two broad areas, pilot recommendations and reliability/maintainability. A body of factual data on the "survey" aircraft (the F-4, F-101, F-111, and A-7) and systems was also acquired and documented.

Pilot Opinions

The following opinions and conclusions are considered especially pertinent.

- Pilot assessment of out-of-control conditions and applications of recovery procedures has been unsatisfactory.
- Aircraft should either be spin-resistant or incorporate a spin recovery system.
- Automatic trim functions must not have excessive settling times.
- Turn coordination systems must accommodate high angle-of-attack and weapon delivery requirements.
- Passive angle-of-attack warning devices are generally deficient under stress; an overrideable command limiter should be considered.
- Aircraft status information masked by the control system (e.g., available control authority) must be suitably displayed.
- Pilots will select alternate control modes which offer superior performance and adapt to associated response changes.

Reliability/Maintainability Data

These consist predominately of figures for the survey aircraft flight control systems (primary, secondary, and automatic) expressing:

- Mean time between failure (MTBF);
- Mean time between maintenance (MTBM);
- Maintenance manhours per flight hour (MM/FH);
- Mean time between abort (MTBA).

Also recorded (Appendix II) are the individual device figures for the above systems. Of particular interest are the MTBA figures for the PFCS, these having value as a measure of adequacy of future fly-by-wire systems. The data with greatest statistical value is from the F-4, which demonstrated a MTBA of 1447 flight hours for the PFCS over a period of about 300,000 flight hours. Of comparable interest are safety data giving the aircraft loss rate from various sources, the F-4 PFCS producing 3.8×10^{-6} catastrophic failures per hour based on 3,000,000 hours.

REVIEW OF SYSTEM GAIN-CHANGING REQUIREMENTS AND TECHNIQUES

This review analyzes gain-changing requirements for rigid-body augmentation systems, categorizes and abstracts available gain-control techniques, and relates requirements and techniques in a set of application criteria. The requirements are addressed primarily for the case of the high-bandwidth system which constitutes the expected basis for the closed-loop PFCS. The need for gain changing is related to the surface effectiveness range, the attainable bandwidth, the desired augmentation frequencies, and the required response bandwidth. This criteria is applied to the pitch axis of the F-4 to illustrate its application. The marginal performance of a fixed-gain C* control loop is demonstrated.

The difficulty in achieving closed-loop turn coordination with conventional feedback sensors is also demonstrated by computing required magnitudes and ranges of accelerometer gains.

The available gain-control techniques are categorized as employing either steady-state or dynamic sensing, the former including classical air data scheduling and the latter including the various closed-loop adaptive and modeling concepts. The functions and constraints of each concept are described briefly.

Finally, the application criteria derived from the requirements and techniques studies are related to the dominant influences of the controller properties, the aircraft properties, and the performance specifications.

STABILIZATION CRITERIA FOR STRUCTURAL RESPONSE AND OTHER HIGH-FREQUENCY DYNAMICS

This study addressed the problem of providing adequate closed-loop stability in the presence of aeroelastic modes, actuator and sensor dynamics, and surface inertial effects. A set of design tools was developed for use in defining high-frequency compensation and analyzing stability. These consisted of (1) a general analytical model of the first three symmetric bending modes, (2) sets of gain and phase uncertainties established as a function of frequency, and (3) a procedure for establishing potential phase and gain margins as a function of the available aeroelastic model and the dynamic uncertainties.

The aeroelastic model provided is intended for use when no other bending data is available. It was based on YF-12 flexible data and evaluated using F-4 flexible data. Although satisfactory correlation was demonstrated, lack of suitable data for other comparable aircraft precluded a thorough evaluation. Further refinement of this model is recommended.

The gain and phase uncertainties computed were those associated with current analog hardware in a system of typical complexity. The variations produced by bending mode data tolerances were also determined. It was shown that major phase uncertainties occur due to surface inertia effects around the so-called "tail-wags-dog" frequency, precluding phase stabilization in this region.

The procedures for establishing stability margins were identified in terms of three cases:

- Those using only rigid aircraft data;
- Those using rigid data plus the general aeroelastic model;
- Those having both rigid and flexible data available for the particular application.

Establishment of a quantitative general specification on phase and gain margins is not recommended because of major differences in component tolerances, aircraft data tolerances, controller type (e.g., conventional or adaptive), and point of application to the control loop.

ANALYSIS OF STALL/SPIN MANEUVERS

This study was directed toward obtaining a better understanding of aircraft behavior in the stall/spin flight regime both with and without the influence of feedback controls. Such knowledge will contribute to improved criteria for feedback control in abnormal flight.

The key analysis tool used was a complete six-degree-of-freedom, all-attitude computer simulation of an F-4 aircraft. The simulation included all the nonlinear equations and rotary balance derivatives necessary to simulate stall/spin maneuvers at subsonic conditions. Aircraft inertial properties were held constant throughout the study, and symmetry was assumed.

Four aircraft operating regions were defined as follows:

- Normal flight
- Stalled flight - recovery possible with conventional surfaces
- Stalled flight - recovery possible with auxiliary devices
- Stalled flight - nonrecoverable

Of these regions, the first two are of primary interest in the analysis of feedback controls (since they use the conventional surfaces as force producers). Also of interest is the establishment of the regional boundaries for at least the first two regions. The boundary between the first two regions is characterized by aircraft aerodynamic stall or loss of normal static stability. The boundary between the second and third regions is defined by the controllability limit of the conventional control surfaces (i.e., the limiting condition for recovery with conventional controls).

The analyses were divided into two major categories -- basic aircraft analysis and development of control criteria. The basic aircraft analysis was further divided into studies of departure, spin evolution, controllability limits, and recovery processes. Control criteria development was divided into study of control strategies for normal flight (primarily in terms of stall inhibition), for flight in the recoverable stall region, and for transition among the flight regions.

Departure from normal flight on the F-4 is characterized by a loss of lateral-directional stability at an angle of attack (AOA) around 23 degrees. At approximately 37 degrees AOA, lateral-directional stability is regained due to the stabilizing effects of aircraft dihedral. Continued application of departure-producing controls will result in a developed spin. Conventional feedback controls in the pitch and yaw axes were found to be beneficial for inhibiting departure. Roll rate to aileron tends to aggravate departure; however, this effect appears minor on the F-4. Recovery from an incipient spin can be accomplished by application of full down elevator and neutralized ailerons and rudder. Recovery from more developed spins requires full pro-spin aileron, full anti-spin rudder, and full down elevator. Use of conventional controls for recovery becomes ineffective as the vehicle becomes more spin stabilized. Ailerons were found to be the most effective means for recovery at higher spin rates because of their direct influence (yawing moment) on spin rate. Feedback controls were found to be detrimental for recovery from spins because they consumed surface deflection better used for full deflection moments.

A set of requirements for a stall inhibitor were specified in the study. These requirements featured the assumption that intentional stalling of the aircraft by pilot override would be permitted. A preliminary stall inhibitor was defined and tested using the simulation. Consideration of control recovery from the stalled flight mode led to the conclusion that both a manual and an automatic recovery mode should be provided. The manual modes would most likely retain conventional feedbacks in pitch and yaw, with direct surface control in roll. The automatic mode would be manually engaged at pilot option or automatically engaged on approach of the controllability boundary. It would utilize full surface deflection sets based on rate and acceleration logic statements.

Complete concepts were not established for transition between flight regions. Questions regarding preferred trim states and stalled-to-normal mode change remain.

CONTROL SYSTEM CHARACTERISTICS AFFECTING FLYING QUALITIES

This sect. ntroduces the question of performance requirements for closed-loop pri ry flight controls with two contributions:

- A categorization of the dominant performance characteristics affecting flying qualities relative to particular mission tasks;
- An assessment of the effects of current augmentation systems on each of these characteristics.

The information used in these areas was gained primarily through pilot interviews conducted as part of the Survey of Operational Problems.

The categorization of characteristics produced the following 14 properties:

(1) Stick breakout and deadspot;

(2) Response rate for small-amplitude inputs;

(3) Response rate for large-amplitude inputs;

(4) Overshoot and damping;

(5) Stick force gradients;

(6) Stick force level .

(7) Turn coordination;

(8) Performance limits;

(9) Control effect at maneuver extremes;

(10) Response to turbulence;

(11) Small-amplitude oscillations;

(12) Trim properties;

(13) Speed stability;

(14) Response to rudder pedals.

These properties were related to the following mission tasks:

- Formation flight;
- Air combat maneuvering;
- Flight tracking -- ground-controlled intercept or manual; terrain following
- Target tracking, air-to-air;
- In-flight refueling;
- Target tracking, air-to-ground;
- Flight path tracking - ILS, VOR, TACAN;
- Landing;
- Takeoff.

The above 14 control characteristics were then evaluated for each of the survey aircraft, the F-4, F-111, A-7, and F-101. The recommendations and significant opinions which resulted from this evaluation are included in the operational survey discussed previously.

SYSTEM/AIRCRAFT COMPATIBILITY TESTING

This study is concerned with testing to assure functional adequacy of the closed-loop PFCS prior to actual flight. To this end, it concentrates on closed-loop simulation procedures involving to a maximum extent the actual flight hardware. This form of testing is viewed as the last step prior to actual flight. As such it is considered as supplementary to rather than a replacement for the usual formal testing that is routinely performed on aircraft equipment, such as qualification and reliability demonstration.

The closed-loop testing was divided into rigid-body and flexible categories. The former type is well established and required little investigation. The latter tests for structural stability are rarely performed, however, and pose several questionable influences such as the lack of aerodynamics during ground tests and the effect of the airframe support. These were analyzed by computing and comparing the airframe transfer functions in flight with those on the ground. Parametric variations were made in the airframe support to assess relative test fidelity.

The following conclusions have been drawn from the investigation:

- Closed-loop testing using simulated airframe properties at various flight conditions with the actual actuation system and flight computer should be performed to verify stability margins, response to commands and disturbances, and limit-cycle acceptability. The latter should be evaluated in terms of MIL-F-9490C (USAF) criteria by computing related airframe variables.

- Given proper means of airframe support and suitable correction for lack of surface aerodynamics, closed-loop system operation using actuators, sensors, and electronics in their flight configurations is an effective and accurate procedure for verifying system stability in the presence of structural flexure.

- In general, the support provided by conventional landing gear contributes considerable damping to the lower-frequency bending modes, making structural stability tests under this condition of questionable value.

- Support means with low damping and located near the nodes of significant bending modes are desirable.

- Compensation for lack of surface aerodynamics must be provided in closed-loop structural testing.

- Total closed-loop testing for simultaneous evaluation of flexure and rigid performance is subject to considerable error around intermediate (between short-period and bending) frequencies unless particular support qualities and correction factors are employed.

CRITERIA FOR BUILT-IN TEST EQUIPMENT

This study views the high-authority closed-loop PFCS as equipment which is both flight- and mission-essential. These qualities will demand some degree of redundancy for at least the next decade, hence means of fault detection and removal. Built-in test equipment (BITE) is considered to include the fault detection function for both in-flight and preflight (on-ground) testing. It is evident, therefore, that the primary purpose of BITE is to contribute to the necessary system reliability, both from a flight safety and mission completion standpoint. This study assesses BITE qualities and criteria in terms of these attributes.

It is recognized that BITE criteria are to a considerable degree dependent on system design. Consequently, the study analyzed two prominent system configurations conceivable for the PFCS application, the triple-channel system and the quad-channel system. The factors contributing to total failure and to partial failure (the abort condition) for each system type were identified along with the influence of BITE qualities. The latter include test thoroughness, probability of latent test failure, and false failure indication rate. Comparisons between redundant system types and a conventional PFCS are made.

The study has produced a number of quantitative conclusions applicable to current configurations of redundant systems. Possible variations to the studied configurations are many, and the design criteria must be critically reviewed for each application. There are, however, certain qualitative generalities which have been deduced which are believed to be universally useful. These are listed below:

- All major failure sources must be considered when determining total and partial failure probabilities, including multiple-channel failures, single-point failures, latent failures in both prime equipment and the BITE, and nuisance disengagements.

- Requirements for test quality should reflect the objective of the tests (e.g., flight safety, maintenance, etc.) and should be imposed only on that equipment in the system which affects the objective.

- Test quality (i.e., thoroughness and elimination of latent failures) has a highly significant effect on total failure probability which becomes more critical as the number of channels of redundancy and the system life increase.

- Nuisance trips reduce flight safety and (except for the triple system) mission reliability. Their effect is more pronounced for the quad system than the triple.

- The reliability degradation due to latent failures can be avoided by testing at regular intervals (on ground or in flight). It is advantageous, however, (particularly for the quad system) to test as thoroughly as possible in flight.

- BITE must be designed either with very low probability of having a latent failure (i.e., "fail-safe" qualities) or be tested at regular intervals by auxiliary equipment. This requirement becomes more vital with more channels of redundancy.

FLIGHT CONTROL ACTUATOR DESIGNS

A survey of primary flight control hydraulic actuators was made to produce a catalog of related design data. Airframe and actuator manufacturers were contacted to acquire physical characteristics (e.g., strokes, forces, dynamics, etc.) and determining criteria (e.g., aerodynamic loads, key maneuvers, etc.) for actuators of stabilizers, elevons, rudders, and spoilers. Data is tabulated (Appendix V) in whole or part on actuators for the A-7, B-58, C-5A, C-141, F-4, F-14, F-15, F-106, and F-111. Calculations were made based on these data to determine input and output powers and associated efficiencies.

The following trends were noted from the tabulations:

- Dual-redundant, fully powered actuators are nearly universal choices for the subject aircraft.

- The general practice of achieving stiffness via force capability (usually oversized) is becoming less prevalent with optimization of actuator/structure parameters and use of hydrodynamic stabilization techniques.

- Surface velocities are generally based on maneuver requirements of the aircraft and are increasing in accordance with vehicle performance demands.

- Although superior design compromises are evident in newer aircraft, the use of a constant-pressure central hydraulic source continues to propagate low operating efficiencies due to the low average power requirements of most aerodynamic surfaces.

SECTION III

SURVEY OF OPERATIONAL PROBLEMS

SURVEY CONTACTS

The data contained in this survey was obtained by visiting USAF operational bases, Air Material Area Headquarters, Air Command Headquarters and USAF flight test centers. The individual interviews were with pilots, USAF maintenance and safety officers, squadron commanders, and maintenance and logistic technicians at the various command, operational and Air Material Area (AMA) locations. A brief summary of the bases visited and the particular aircraft on which data was obtained is given below:

Organization	Base	Aircraft Specialty
AFFDL	Wright-Patterson AFB	Flight Dynamics Laboratory - all aircraft
AFLC HQ	Wright-Patterson AFB	USAF 66-1 Failure Reporting System
OOAMA	Hill AFB	F-101, F-4 AMA
OCAMA	Tinker AFB	A-7D AMA
SMAMA	McClellan AFB	F-111 AMA
ADC HQ	Ent AFB	F-101B/F
TAC HQ	Langley AFB	F-4, A-7, F-111
TAC	Nellis AFB	F-111, F-4
TAC	Luke AFB	A-7D
ADC-ANG	Hector Field	F-101B/F
ADC	Tyndall AFB	F-101B/F
MAC	Norton AFB	All Aircraft - Safety

The pilots interviewed were generally veterans of the Vietnam action, thereby having the opportunity to evaluate aircraft handling under combat conditions. Several of the pilots interviewed were assigned to special test squadrons to evaluate new techniques for air-to-air and air-to-ground weapons delivery. The commanding officers of squadrons at several of the locations also volunteered information regarding desirable handling characteristics and their views on improving aircraft maintainability and reliability.

The reliability, maintainability, and safety personnel interviewed were actively performing duties at the depot, field squadron and flight line levels of maintenance. The majority of the statistical data obtained was taken from USAF maintenance reports which are compiled periodically and issued to the various operational and support organizations. This data is collected and distributed under the Air Force Maintenance Data Collection System defined by Air Force Manual 66-1. A brief description of the AFM 66-1 system and the particular reports utilized in this survey are included in a later subsection.

The majority of the squadron and flight line maintenance personnel interviewed had experience with flight control system maintainability on several aircraft and at several locations. This experience was a definite asset in determining whether the particular maintenance actions they were experiencing were peculiar to their environment or whether they were of a general nature. It was also noted that maintenance procedures varied considerably between test squadrons and operational squadrons, with the maintenance workload and type of maintenance action reflecting the difference in environment.

A tabulation of the personnel interviewed at each base and the general information discussed is given in Appendix I.

AIRCRAFT DATA

This subsection summarizes much of the pertinent factual data related to the survey of aircraft mission requirements and their flight control system mechanizations. This information was used as background in preparing the pilot and maintenance personnel interview questionnaires and in developing the proper perspective for the handling and reliability-maintainability data obtained.

The simplified block diagrams and data tables are intended to give an overview of the complexity and functional requirements of the flight control systems included in this survey. The primary and automatic flight control system block diagrams are included because their functions encompass those of the high-authority closed-loop PFCS of the future. The secondary flight control system (flaps, slats, speed brakes, wing fold/sweep system, etc.) has not been included in the block diagrams as its functions are not within the scope of the current study. Details of the surface trim components, mechanical and hydraulic dampers and other feel system components and the hydraulic actuators are not presented here. However, these details are available in the pertinent aircraft technical orders if additional detail is desired.

Aircraft data in tabular form is provided in Tables 3-1 through 3-6, and symbols used in the block diagrams are defined in Table 3-7. The block diagrams are shown in Figures 3-1 through 3-8.

Table 3-1. Aircraft Data

Department of Defense Designation Design vintage Popular name	F-101 B, F-101F 1955 "Voodoo"	F-4C, F-4D, F-4E RF-4C 1958 "Phantom II"	F-111A, F-111E FB-111 1962	A-7D 1966 "Corsair 2"
Manufacturer	McDonnell Douglas St. Louis, Mo.	McDonnell Douglas St. Louis, Mo.	General Dynamics Fort Worth, Texas	LTV Aerospace Corp. Dallas, Texas
Operational organization	USAF - Air Defense Command Air National Guard Canadian Air Force	Tactical Air Command USAFE Navy	Tactical Air Command	Tactical Air Command
Number of aircraft in survey data	210	1753	135	58
Primary mission	Interceptor	Fighter, attack reconnaissance	Attack,fighter	Attack
Typical mission time (hrs)	1.5	2.0	3.0	1.2
Mach range	To 1.6	To 2.2	To 2.5	To 0.95
Altitude range (ft)	0 to 50,000	0 to 60,000	0 to 60,000	0 to 40,000
"G"-loading range-max	-2.0 to +6.0	-3.0 to +8.5	-3.0 to +7.0	-3.0 to +7.0
(typical)	-1.0 to +4.0	-1.9 to +4.2	-1.5 to +4.5	-1.5 to +5.0
Maximum length (ft)	71.1	58.2	75.6	46.1
Maximum height (ft)	18.0	15.9	17.1	16.1
Wingspan (ft)	39.7	38.4	32 (swept); 63(extended)	38.7
Wing area (sq ft)	368	530		375
Weight - basic airframe (lbs)	31,000	28,000	48,250	22,000
Maximum weight (lbs)	53,000	63,000	83,000	42,000
Crew	(2) Pilot, radar observer	(2) Pilot, copilot	(2) Pilot, weapon system officer	(1) Pilot
Engines	(2) P&W J57	(2) GE - J79	(2) P&W, TF30P-3	(1) Allison, TF-41-A-1

Table 3-2. Primary Flight Control System Data

Item	F-101B/F	F-4C/D/E	F-111A/E	A-7D
Surfaces - pitch axis	Stabilator - 19° TEU 11° TED	bilator - 20° TEU 8° TED	Stabilizer - 30° TEU 15° TED	Stabilizer - 26° TEU 6.75° TED
Surfaces - roll axis	Ailerons - ± 21°	Ailerons - 30° TED 1° TEU Spoilers - 45° TEU - flush down	Stabilizer - ±8 each differential surface; spoilers used when wings forward of 45°	Ailerons - ± 25° Spoilers - 0 to 60° TEU (operate with ailerons)
Surfaces - yaw axis	Rudder - ± 25°	Rudder - ± 30°	Rudder - ±30°	Rudder - ± 18.5°, ± 24.5° with flaps extended
Hydraulic power supplies	Dual-redundant-3000 psi	Dual-redundant-3000 psi	Dual redundant-3000 psi	3000 psi - triple-redundant (PC-1, PC-2, PC-3), Emergency ram air turbine (RAT)
Linkage construction	Control cables in pitch and yaw axes; dual controls in F-101F only	Control cables in pitch and yaw axes; dual cockpit controls in all aircraft	Bellcranks, push-pull rods; cable assembly in yaw; spoilers are electrically controlled	Bellcranks, push-pull rods, servo cages; yaw axis uses cables for rudder control
Stick movement - pitch	± 5.23 in.	5.9 in. aft; 1.5 in. fwd	6.5 in. aft; 4.0 in. fwd	± 5.24 in.
Stick movement - roll	± 4.62 in.	± 1.56 in.	± 4.5 in.	± 4.53 in.
Rudder pedal movement	± 3.2 in.	3.24 in. aft; 3.24 in. fwd	± 3.1 in.	± 3.13 in.
Pitch axis feel	Bobweight 5 lbs/g, Q bellows provides force increase with dynamic pressure	Bobweight 3 lbs/g, Q bellows provides force increase with dynamic pressure	Springs and nonlinear linkage, 1.7 to 40 lbs aft, 1.7 to 50 lbs fwd	Bobweight - 3 lb/g at high g and 7 lb/g at low q's; nonlinear linkage stick to surface
Roll axis feel	Spring cartridge 2.0 to 15 0 lbs	Spring cartridge 2.3 to 12.2 lbs	Springs and nonlinear linkage 1.2 to 15 lbs - 0 to 2° δa 23 to 31 lbs - 2 to 8° δa	Spring cartridge 1.0 to 12.0 lbs
Yaw axis feel	Spring cartridge 10 to 80 lbs	Airspeed hydraulic switch 2.6 lbs/degree rudder below 235 knots, 11.5 lbs above 235 knots	Spring cartridge 12 to 80 lbs	Mechanical spring: cruise - 10 to 70 lbs; landing - 10 to 30 lbs

NOTES:

TEU = Trailing edge up
TED = Trailing edge down

Stick Forces - Forces are in pounds from centered position to maximum travel position

Table 3-3. Automatic Flight Control System Functional Characteristics

Item	F-101B/F	F-4C/D/F	F-111A/E	A-7D	Other Factors and Definitions
Surfaces and Authority					
Stability augmentation and control augmentation systems	Yaw SAS	3-axis SAS	2-axis CAS yaw SAS	2-axis CAS yaw SAS	Stability Augmentation System = SAS Control Augmentation System = CAS Autopilot = A/P Automatic Flight Control System = AFCS
Pitch	---	Stabilator - ± 0.5°	Stabilizer - ±13°	Stabilizer - ±4°	
Roll	---	Ailerons ± 7.5°, Spoilers - 11.25°	Stabilizer - ±6° differential	Ailerons - ±10°, Spoilers - 26°	
Yaw	Rudder - ± 5°	Rudder - ± 5.0°	Rudder - ± 15°	Rudder ± 12.5°	
Autopilot authority					
Pitch	Stabilator - 11°TED, 19° TEU	Stabilator - 8° TED, 20° TEU	Same as CAS and SAS	Same as CAS and SAS	
Roll	Ailerons - ± 21°	Same as SAS			
Yaw	Rudder - ± 5°	Same as SAS			
Autopilot Modes					
Heading hold	Yes, Pre-select heading also	Yes	Yes, Auto-Nav mode also	Yes, Navigation select mode	1. Mission Completion All aircraft have certain envelope restrictions imposed when CAS or SAS have failed.
Attitude hold	Yes	Yes	Yes	Yes	
Altitude hold	Yes	Yes	Yes	Yes	
Mach hold	No	No	Yes	No	2. F-101B/F The F-101B/F has an angle of attack, g limiter system which is considered part of the AFCS. The system is commonly referred to as the Redundant Limiter System (RLS). This system also provides angle-of-attack maneuver limits in the manual aircraft control modes.
Other modes	Automatic instrument landing, automatic intercept/attack modes	---	Terrain following automatic ground track	---	
Failure-Safety Provisions					
Automatic "G" Disengagement	AFCS at ± 4.5g, -1.5g	AFCS and SAS @ -2, +4g	No	No	
AFCS Emergency Disengage Switch	Pilot's control stick	Pilot's control stick	Pilot's control stick	Pilot's control stick	
Warning lights	Yes	Yes	Yes	Yes	3. F-4 The F-4 has an Aileron-to Rudder Interconnect (ARI) System which provides aileron inputs to the rudder during low airspeed maneuvers.
Other	---	---	Triple redundant - fail operational/fail safe	Dual redundant - fail safe Roll SAS disengage - angle-of-attack	
Modes required to be operational by USAF for:					
Aircraft takeoff	All except Mach hold	All 3 axes of SAS	Triple-redundant, CAS and SAS and A/P	Dual redundant CAS and SAS	
Mission completion	Yaw stab. aug. and redundant limiter system	Pilot's discretion	Triple-redundant, CAS and SAS	Yaw SAS and pilot's discretion	
Angle-of-attack system	Redundant limiter system provides maneuvering command limits - Note 2	Warning horn, rudder pedal shakers, aural warning system, indexer	Angle-of-attack indexer and stall warning rudder pedal shakers	Indexer and rudder pedal shaker	

Table 3-4. Automatic Flight Control System Hardware Characteristics

Item	F-101B/F	F-4C/D/E	F-111A/E	A-7D
Computer Mechanization				
Shock mounted	Yes	Yes	No	Yes
Cooling	Air-conditioned compartment	Cooling air in compartments	Cooling air in compartments	Separate cooling fan
Vacuum tubes	Yes	No	No	No
Magnetic amplifiers	Yes - 95%	Yes	Magnetic modulators only	No
Relay logic	Yes	Yes - 60%	Feel and trim computer only	No
Solid-state - logic	No	Yes - 40%	Yes	Yes
- amplifiers	Yes - 5%	Yes	Yes	Yes
Moisture resistance	Hermetically sealed metal modules on chassis	Hermetically sealed metal modules on chassis	Potted, encapsulated cubes	Semi-sealed boxes, moisture resistant coating on modules
Sensors Utilized			Triple-redundant packages	Dual-redundant packages
Rate gyros	1 - 3-axis package	3 - single-axis packages	3-packages, P, R, Y	3-packages, P, R, Y
Accelerometers	3 - normal, 1 lateral	1 - normal, 1 - lateral	1 - normal, 1 - lateral	1 - normal, 1 - lateral
Attitude gyros	1 - pitch and roll	1 - pitch and roll	Navigation system	Inertial measurement system
Surface position	Servo position potentiometers	Potentiometers and LVDTs	LVDTs in servos	Potentiometers
Stick force	Pitch force potentiometer	Pitch force LVDT	Pitch and roll LVDTs	Pitch and roll LVDTs
Servo Actuators				
Power source	3000-psi hydraulics	3000-psi hydraulics	3000-psi hydraulics	3000-psi hydraulics
Redundancy	2 - actuators in pitch, single - roll and yaw	Single - all axes	Triple - 2 active valves and 1 model valve	Dual - all axes identical actuators
Aerodynamic Transducers				
Mach Altitude Dynamic pressure	Central air data computer and separate Mach transducer for RLS	Central air data computer	Central air data computer	Central air data computer
Angle-of-attack	Immobile probes and aerodynamic force transducers	Pencil-type probes with aerodynamic force rebalance	Pencil-type probes with aerodynamic rebalance	Vane positioned by air-flow
Signal Scheduling Required				
Mach Altitude Dynamic pressure	All required by A/P and yaw SAS	All required by A/P only	Self-adaptive gain Scheduling-aerodynamic Scheduling not required Gain goes to minimum when system is disengaged	All required by A/P

Table 3-5. Built-in Test Equipment Summary

Item	Aircraft Type[a]		
	F-101 B/F [b]	F-111 A/E	A-7D
BITE control panel	Cockpit	Cockpit Each computer	Cockpit
BITE test sequence	Tests are manually sequenced	Tests are manually sequenced	Tests on manually sequenced
BITE test output display	Pitch boundary indicator; Control stick movement; Engage switches and lamps	Servoactuator position indicators; failure warning lamps; surface deflections; LRU test meter for maintenance testing	Servoactuator position indicators; failure warning lamps; surface deflections
Percentage of device functions tested by BITE			
Computers	90%	90%	90%
Sensors	70% Estimated values	90% Estimated values	60% Estimated values
Servos	80%	80%	80%
BITE time requirement - preflight	10 to 30 sec	20 to 30 sec maintenance - 15 to 20 minutes	20 to 30 sec
Fault-isolation capability	System level only	Line replaceable unit	System level only

[a]The F-4 Automatic Flight Control System does not have any BITE capability

[b]The F-101 B/F BITE Values listed above pertain only to the Redundant Limiter System (RLS) portion of the Automatic Flight Control System. The RLS comprises approximately 20% of the total F-101 Automatic Flight Control System.

Table 3-6. AFCS Test Equipment Summary

Item	Aircraft Type			
	F-101 B/F	F-4 C/D/E	F-111 A/E	A-7D
Type of test equipment				
Flightline	2-wheel trailer type tester (UG637)	Suitcase tester (ASM-73B)	None required	Suitcase tester (ASM-244)
Field shop	4-wheel trailer automatic tester (UG 897) Flyaway test bench (UG677)	Suitcase tester (ASM-73B)	Computer-automated test consoles	Suitcase tester (ASM-245)
Depot level	Manual test consoles	Manual test consoles	Automated test consoles	Automated test consoles
Testing concept	Intercept testing -- system mockup tests at field shop	Parallel testing - component testing only at field and depot levels	No system tests except at the flightline	No system tests except at the flightline
Sensor tests in aircraft				
Rate gyros	Null only	No aircraft tests	Self-test-torque	Self-test-torque
Accelerometers	Null only	Specified	Self-test-torque	Self-test-torque
Air data computer	Self-test	Self-test	Self-test	Self-test

Table 3-7. Definition of Symbols

Symbol	Definition
- - - -▶	Indicates a mechanical linkage (push-pull rods, cable assemblies, etc.)
——▶	Indicates an electrical signal path (servo actuator control/feedback signals etc.)
⇑ H_1	The H indicates a hydraulic system power supply, while the number designates which hydraulic system is providing the power
↑ q	The arrow indicates gain or feel system scheduling is used; the letter indicates whether pitot pressure (q), altitude (alt) or Mach number (Mn) is used to provide the scheduling.
	The box in solid lines indicates one major device or component is used to provide the particular function; the dashed lines indicate separate functions provided within a major system device.

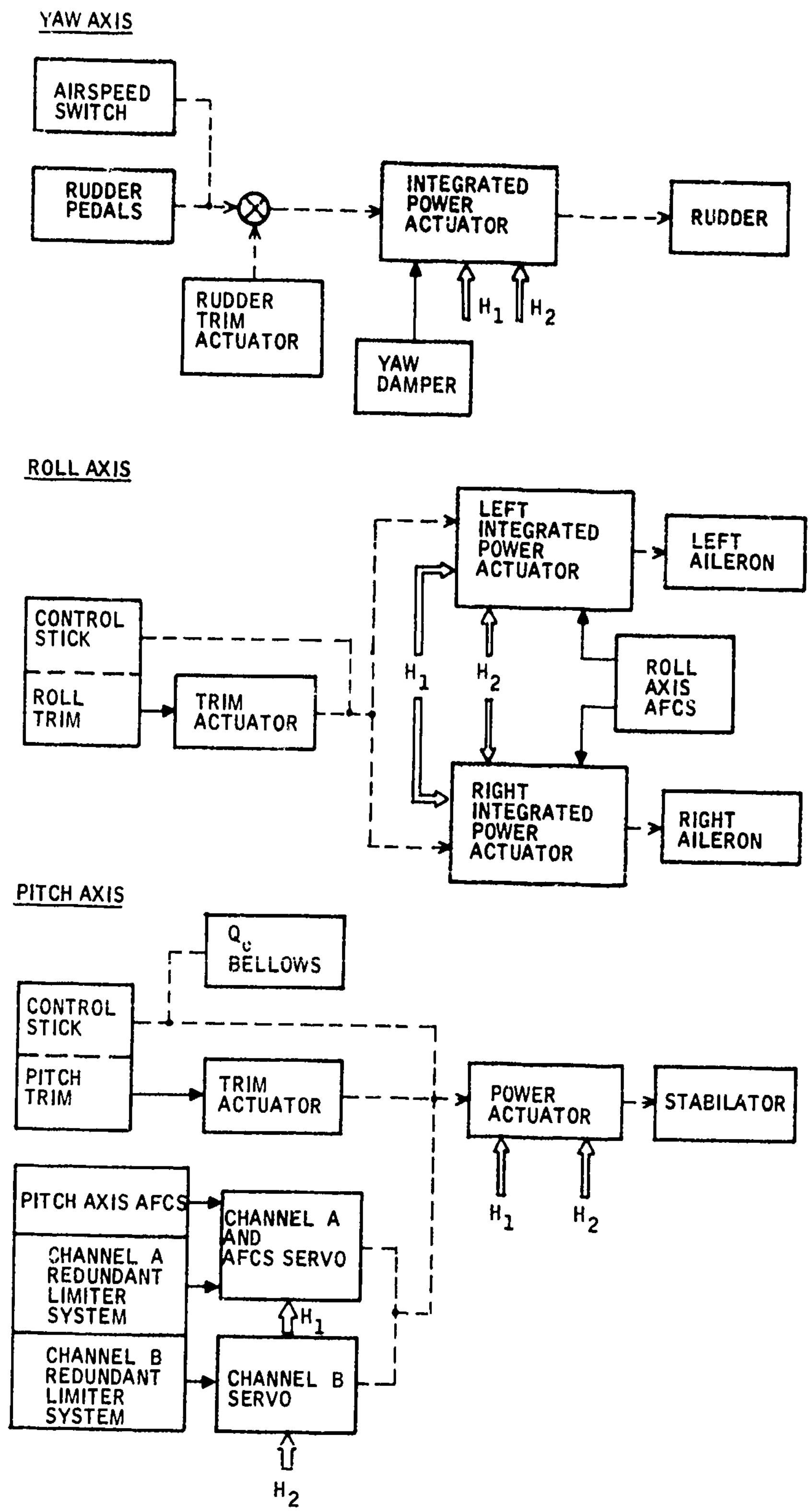

Figure 3-1. F-101B/F Flight Control System Block Diagram

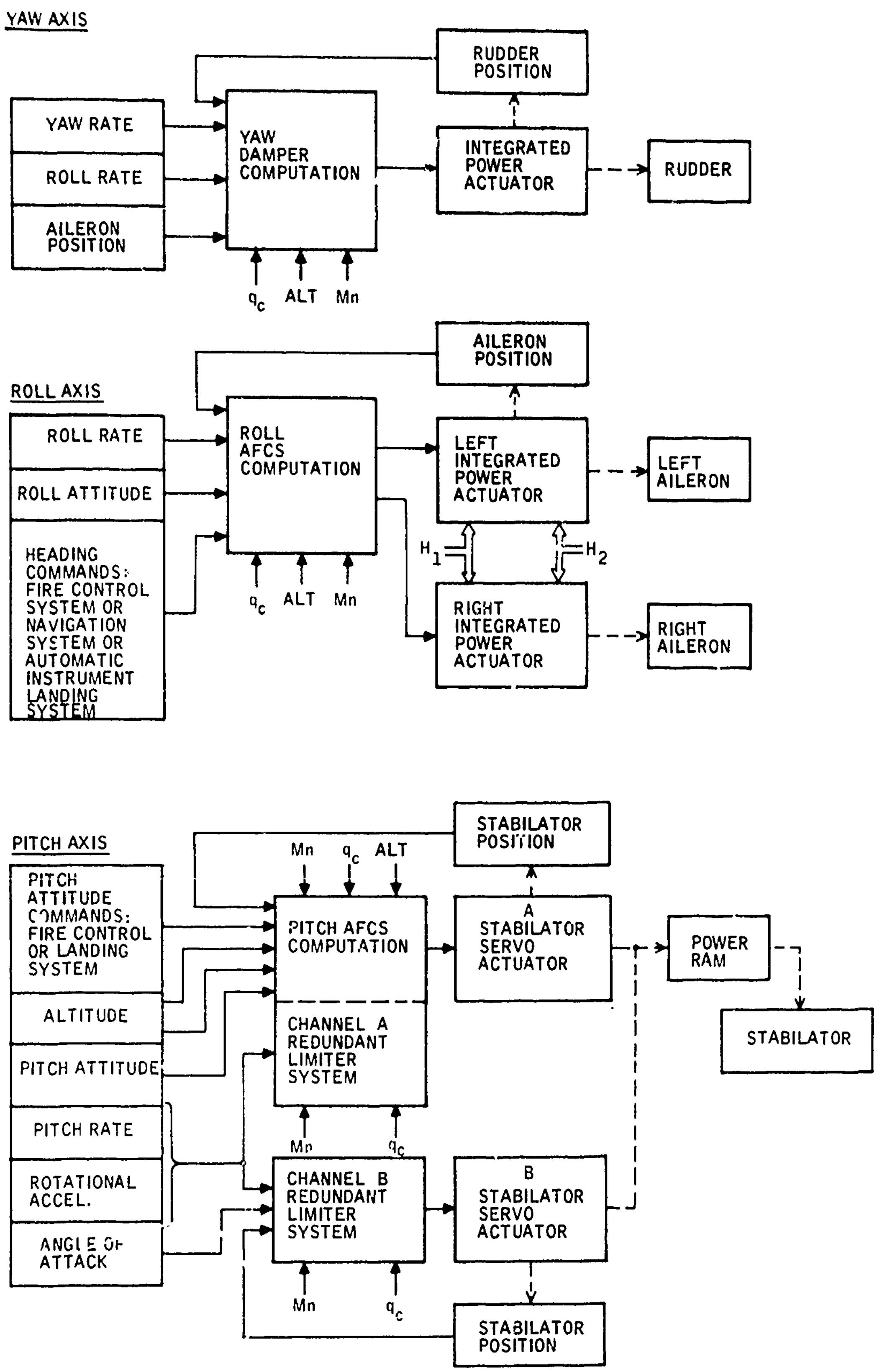

Figure 3-2. F-101B/F Automatic Flight Control System Block Diagram

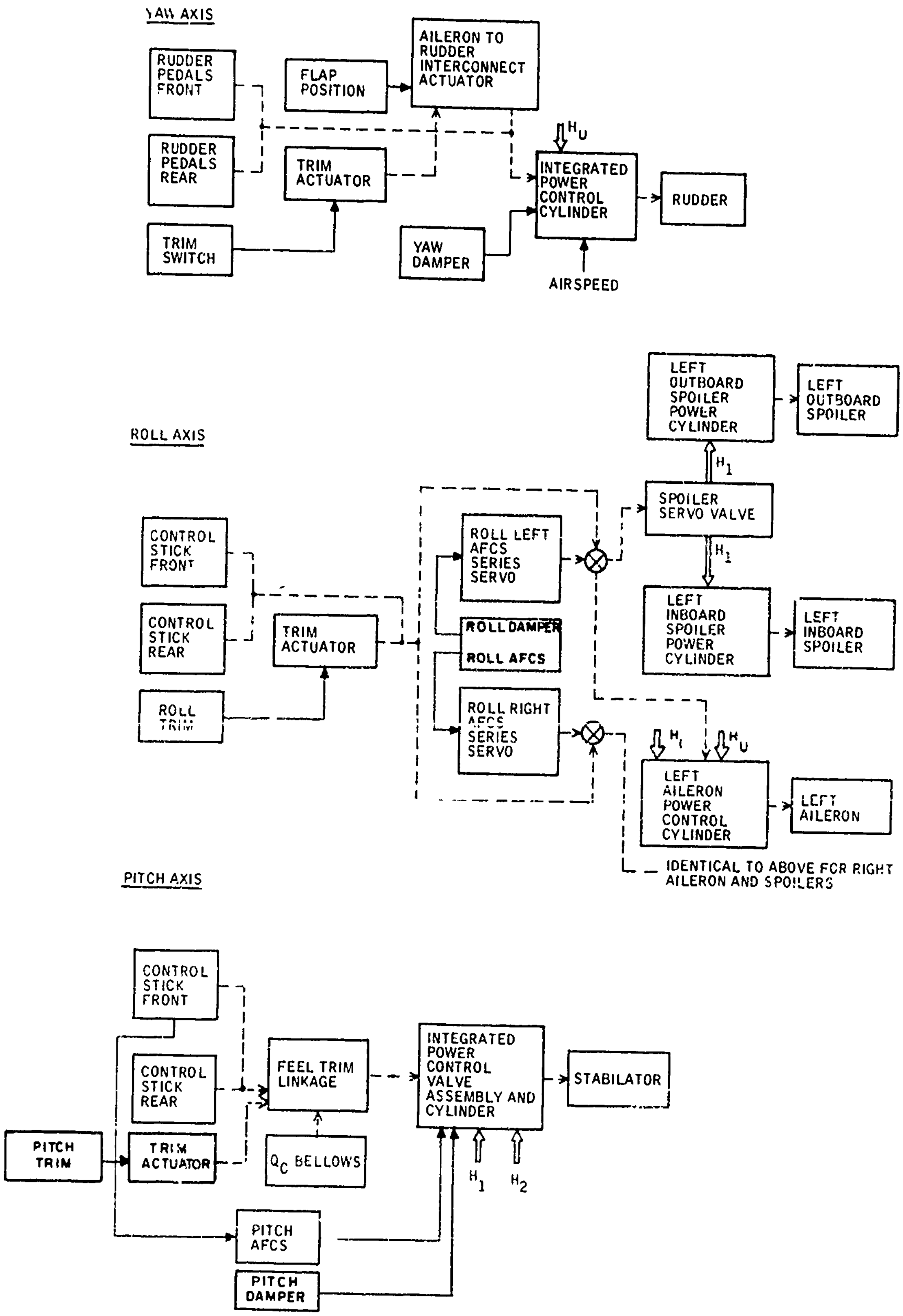

Figure 3-3. F-4 Flight Control System Block Diagram

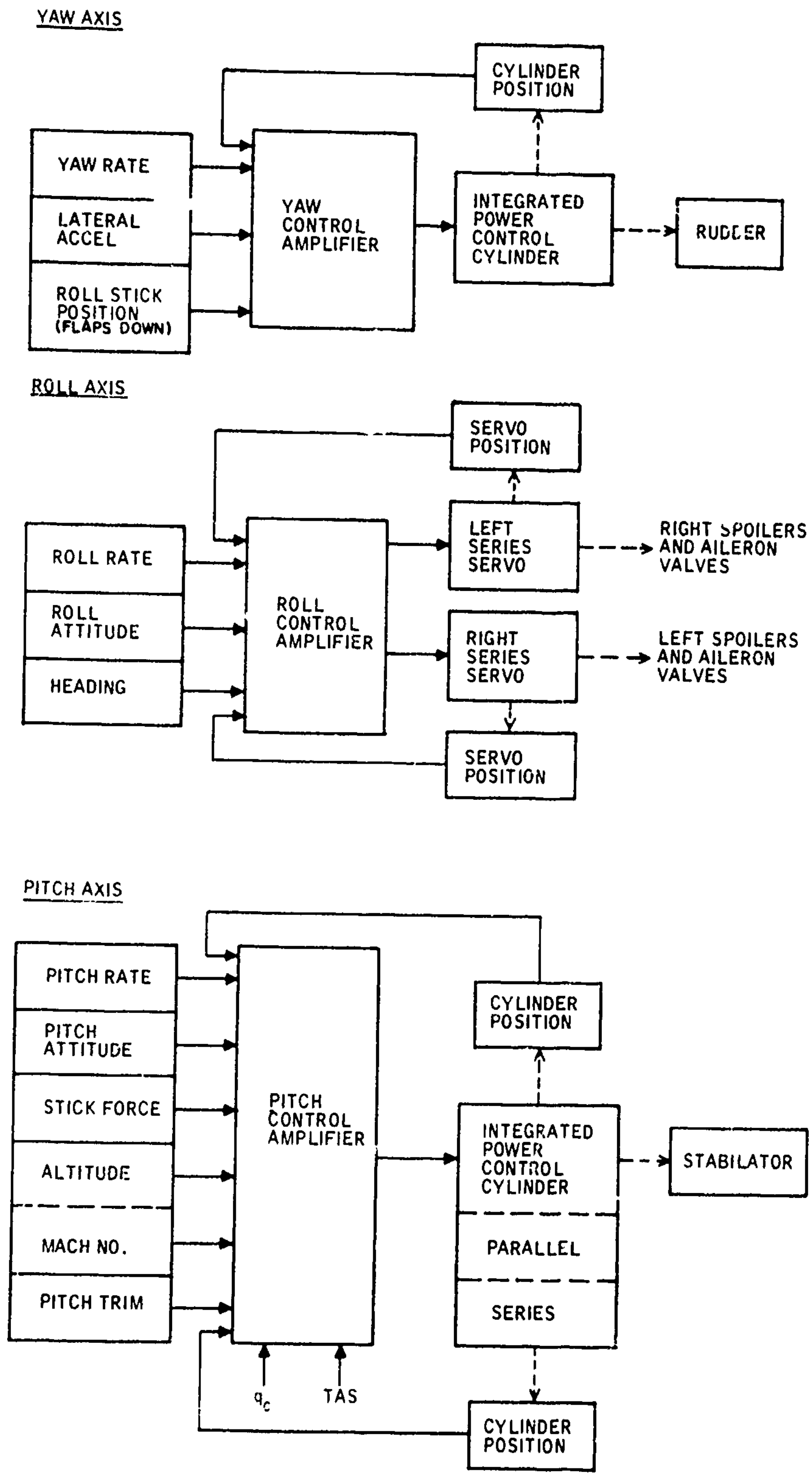

Figure 3-4. F-4 Automatic Flight Control System Block Diagram

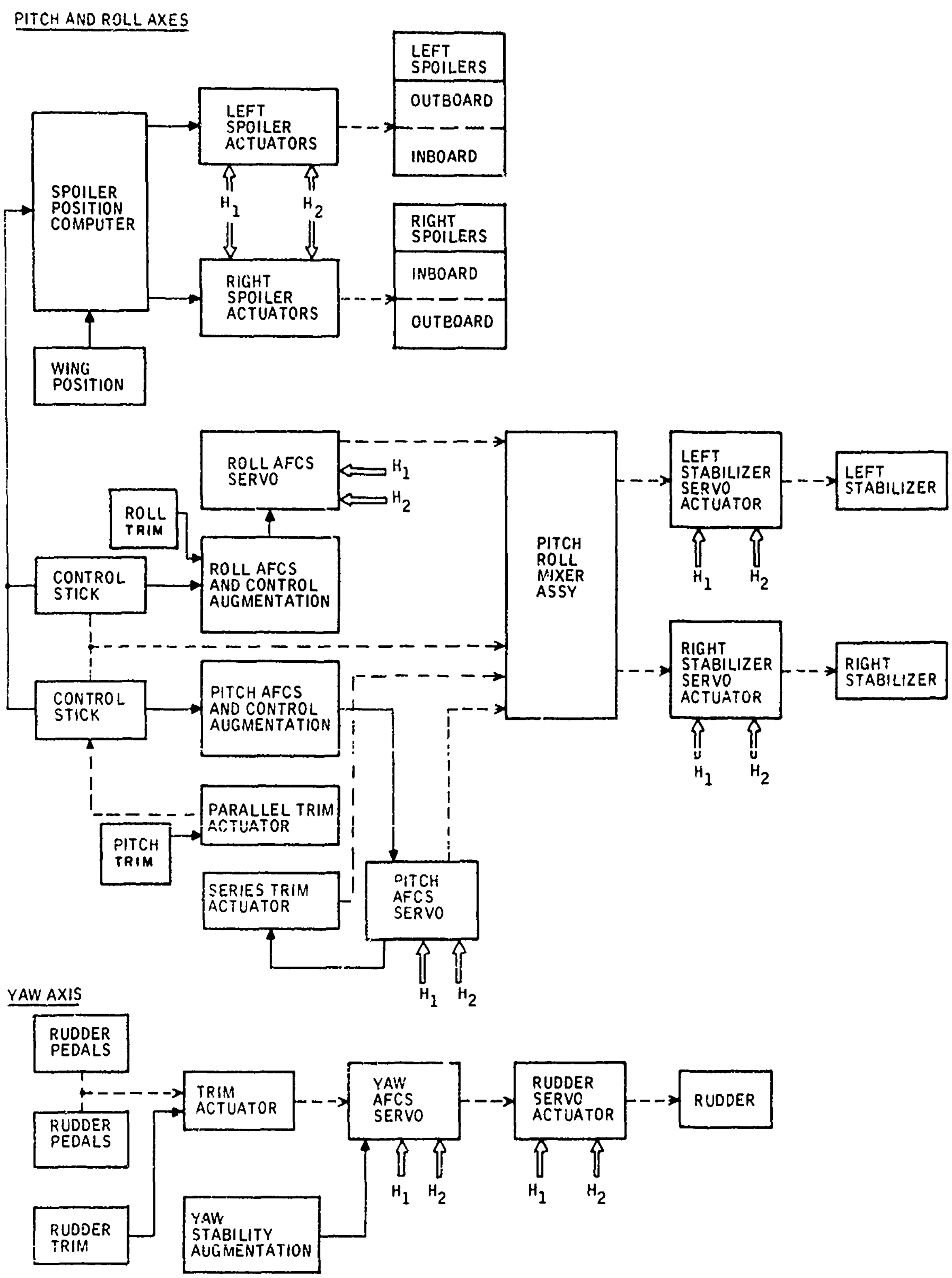

Figure 3-5. F-111 Flight Control System Block Diagram

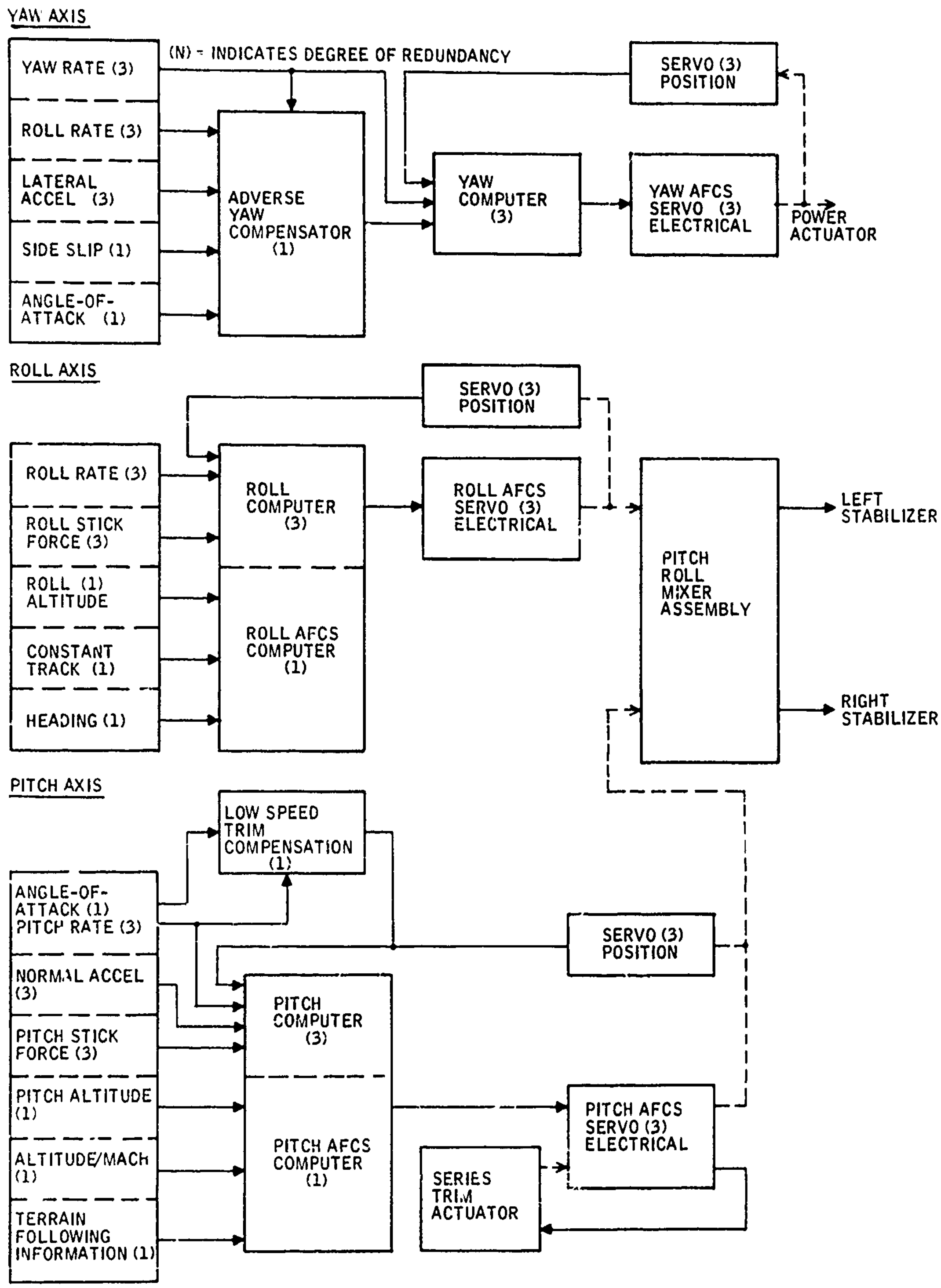

Figure 3-6. F-111 Automatic Flight Control System Block Diagram (Triple Redundant - Adaptive System)

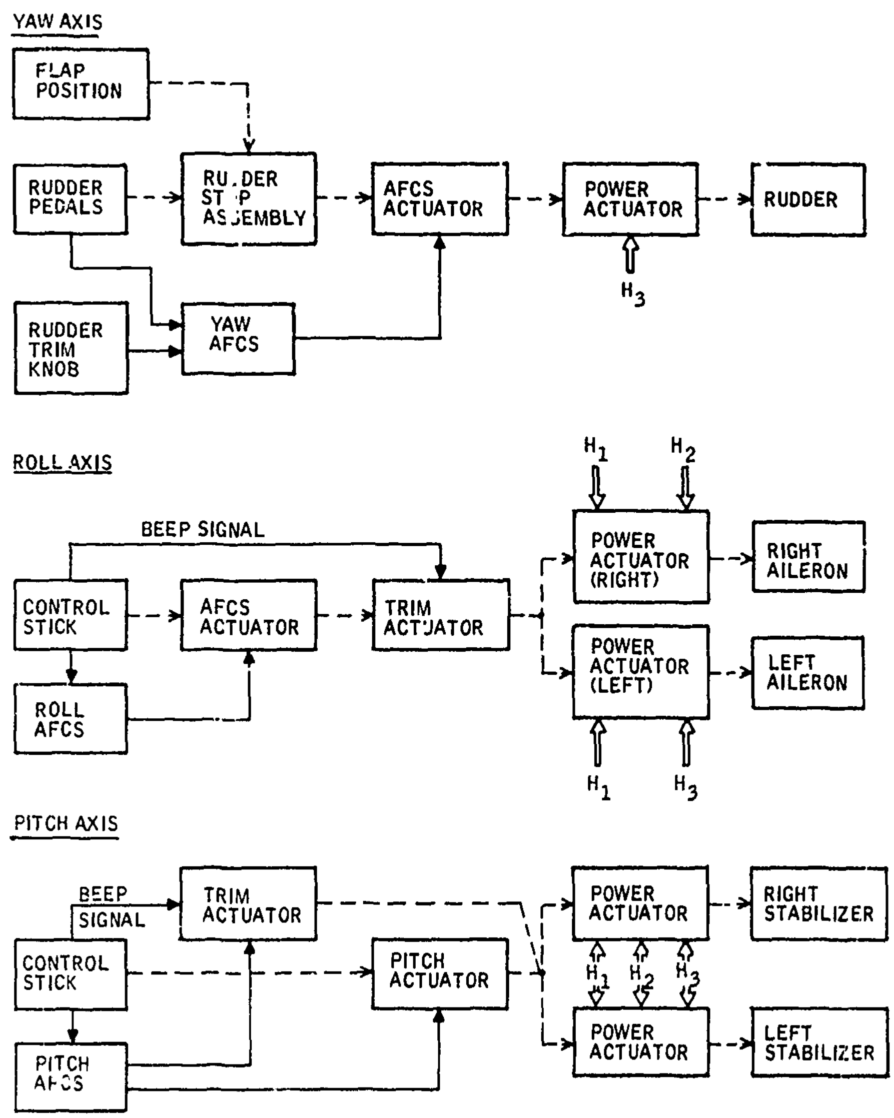

Figure 3-7. A-7D Flight Control System Block Diagram

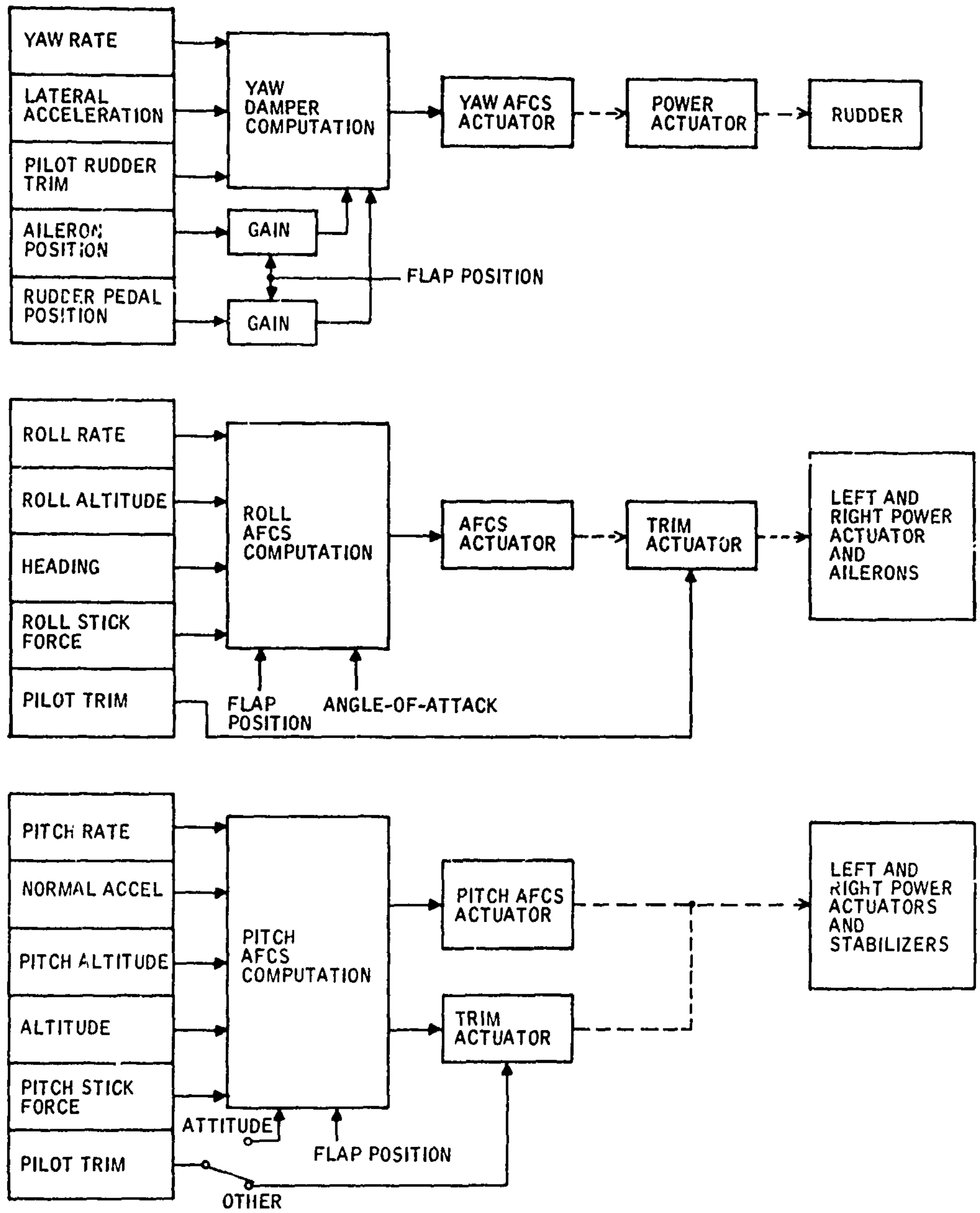

Figure 3-8. A-7D Automatic Flight Control System Block Diagram (Dual Redundant System - Single Channel Only Shown)

PILOT INTERVIEW SURVEY

This portion of the study includes first-hand information related to the performance of flight control systems which has largely been obtained from a select group of USAF pilots. A list of these pilots, their experience, and affiliation is given in Appendix I.

The average experience level of the pilots participating in the survey was about 2000 hours of jet fighter time, one combat tour in Southeast Asia and five to ten years of military flying. This breadth and depth of experience along with the fact that most of these pilots are currently holding squadron or test billets makes them an excellent source of information.

The pilots interviewed know their aircraft and how to make it perform under the most demanding mission requirements.

The following list of facilities visited illustrates the extent of this investigation:

- 4950th Test Wing, WPAFB (F-4)
- Project Test Group, OOAMA, Hill AFB (F-4)
- 422nd Fighter Weapons Squadron, Nellis AFB (F-111)
- 414th Fighter Interceptor Squadron, Nellis AFB (F-4)
- 58th TAC Fighter Training Wing, Test Detachment, Luke AFB (A-7)
- TAC Headquarters, Langley AFB (F-111)
- Defense Weapons Center, Tyndall AFB (F-101B)
- USAF Safety Center, Norton AFB (F-4, F-111)
- ADC Headquarters, Ent AFB (F-101B)
- 119th Fighter Group ANG, Fargo North Dakota (F-101B)

The pilot interviews were centered about a questionnaire which provided a discipline to direct discussions into meaningful, pertinent areas. These areas concerned characteristics of current primary flight controls and augmentation systems, their contributions and detractions to mission accomplishment, and needed improvements for future aircraft. Each interview usually lasted for an hour and was conducted on an individual basis.

Pilot Recommendation Summary

The following conclusions were extracted from the interviews and discussions conducted during the investigation. Expansion of these points is presented in the following subsections.

- If superior performance is available by selection of alternate control characteristics, the pilots will exercise the option and adapt to associated response changes.
- Aircraft response to pilot commands must be smooth, fast, and well-damped in all axes.
- Pilot assessment of out-of-control conditions and application of prescribed recovery procedures has been unsatisfactory.
- Aircraft should be designed to be spin resistant, or an automatic spin recovery system should be considered.
- Automatic trim function must not produce excessive settling times to avoid significant increase in pilot workload.
- Turn coordination systems must accommodate high AOA (angle-of-attack) maneuvers and weapon delivery requirements.
- Passive AOA warning devices (aural, tactile, and visual) are generally deficient under stress conditions. Command limiters, overridable by the pilot, should be considered.
- Fly-by-wire is acceptable to the majority of pilots.
- Intensive air combat maneuver training is favored by pilots.
- Aircraft status information (e.g., available control authority) masked by the flight control system must be restored via a suitable display.

Primary Flight Control Comments

Pilot interview comments on the primary flight control systems (PFCS) and the unaugmented aircraft handling behavior are presented in the following discussion.

F-4 PFCS -- In this report it is not practical to develop detailed discussion on the flight handling or primary control systems of each model of the F-4. Therefore, this discussion will be confined to major properties generally applicable to most production versions of the aircraft.

The F-4 is a two-place (tandem), supersonic, long-range, all-weather fighter bomber and interceptor. It is now the primary air superiority fighter for both the U.S. Navy and the USAF. The aircraft is distinctive in appearance having a low swept wing with pronounced anhedral for the approximate outer one third of the wing span and a one-piece stabilizer with pronounced cathedral. Dual, irreversible hydraulic power control cylinders position the stabilizer, ailerons and spoilers. A single, irreversible hydraulic power control cylinder positions the rudder. The control feel systems have trim actuators which, through the power cylinders, move the entire control surface. Secondary controls are leading edge flaps, trailing edge flaps and wing-mounted speed brakes.

The large flight envelope and variety of possible external loads results in a wide range of aircraft flight handling behavior for the pilot to cope with.

The general feeling among pilots is that the F-4 has acceptable control and handling qualities at all flight conditions provided the pilot is informed of his augmentation status. A tendency toward PIO (pilot-induced oscillation) at low altitude in pitch does exist. A former McDonnell pilot who has flown a number of the F-4 tests at high Mach number says, "It is a good handling aircraft from 130 knots indicated airspeed up to 2.5 Mach number". These comments are all fine, but trouble with control systems and handling qualities begin to show up when the aircraft is placed in an operational environment. Tactical situations often require maneuvering at maximum performance, and often heavy external loads produce cg and stability margins that tax the skill of the pilot. The F-4, like many of its contemporaries, is being used in ways that were not expected when the aircraft was designed.

The interviews with pilots at the Air Combat Training Center, Nellis AFB, and at the Flight Safety Center at Norton AFB tend to point up the following problem areas which are relevant to this program of relating pilot comments to control system design criteria.

The problems are: pilot loss of control; pilot failure to recognize departures and incipient spin characteristics; buffet recognition and pilot technique at maximum performance, high angle of attack; and longitudinal stability.

The loss of control or out-of-control situation surprisingly extends across all pilot experience levels. The history for out-of-control situations is discussed in the following paragraph.

There have been 51 aircraft lost because of pilot loss of control. Nine of the 51 aircraft were lost in air combat training maneuvers, and the rest were scattered through all types of maneuvers, mostly at high aircraft gross weights with near-aft cg limits and high load factors. Contributing to pilot loss of control is the difficulty in recognizing departures due to the buffet being severe enough to mask the normal stall. Rapid rolling moments often accompanying loss of control causes the pilot to assume he is in a spin -- the use of the anti-spin technique at this point usually results in pro-spin action on the aircraft.

A very important fact brought out is that in 51 cases of lost control, not in one single case did the crew execute the correct recovery technique in total sequence. In seven cases the pilots evidenced retrograde amnesia and could not remember anything following the takeoff. The rest of the pilots just said the aircraft went wild and could not be controlled.

The USAF, recognizing these problems, conducted during 1970 a test program at Air Force Fighter Test Center, Edwards AFB, utilizing an instrumented F-4E. The results of these tests are pertinent to this program, and it is recommended that the report be obtained if the reader wishes to thoroughly cover the F-4 flight characteristics. To make this program discussion more meaningful, portions of the Air Force test program are discussed and repeated herein.

The information presented in these reports (Ref. 3-1, 3-2) is good information on the stall and near stall characteristics of the F-4 aircraft. The information in the reports shows good correlation with the F-4 six-degree-of-freedom computer study being done at Honeywell as part of this study program.

Pertinent comments from the AFFTC report are repeated here for the reader: An investigation of the stall and out-of-control characteristics of an F-4E test aircraft indicated that major revisions of the F-4 flight manual are required. The results of 233 departures from controlled flight demonstrated that forward stick (full forward when necessary) was the primary recovery control for all out-of-control events (rolling departures and spins). The drag chute was an effective recovery aid for rolling departures and the steeper spins. Full aileron applied in the spin direction was an effective recovery aid for the steep spins. Natural stall/loss-of-control warning was unsatisfactory. Departures were the result of lateral-directional stability breakdown, and susceptibility to departure was significantly affected by aileron or rudder inputs. The wide variety of stall entry conditions, center-of-gravity locations, and store loadings resulted in five distinct, upright spin modes. Only the flat mode was not aerodynamically recoverable. Maneuvers and control techniques were developed for possible incorporation in the F-4 combat crew training program.

Unfortunately the test aircraft in this program was lost in a flat spin, leaving the following tests incomplete: evaluation of incorrectly applied controls after departure; additional high supersonic stall entries at aft cg with asymmetric loadings; evaluation of aft stick spin recovery from steep spin encountered with clean loading.

A training film (Air Force TF6553) is being prepared for distribution to F-4 aircrews.

It is interesting that a stick pusher was discussed in the report as being a useful aid to reduce out-of-control situations. It was acknowledged that a pusher is not popular with pilots because they just do not like any device that is capable of taking the controls away from them.

Pilot interviews at the Air Combat Maneuvering Training Center, Nellis AFB, emphasized techniques for getting maximum turning performance. Buffet recognition and a good understanding of adverse yaw and dihedral effect are necessary for effective control of the aircraft.

Mild buffet appears at about 12 units AOA (angle of attack)[1], the air over the wing tips starts becoming turbulent. From 12 to 18 units AOA, the shudder level continues to increase, but it is difficult to feel a specific angle of attack within this range.

Moderate buffet and mild wing rock occur at 18 to 20 units AOA. The turbulent air over the wing tips starts to spread inboard along the trailing edge of the wing, causing a definite shake throughout the aircraft. Also, the air over the wing tips becomes extremely turbulent which causes a mild wing rock which is accentuated by the operation of the lateral dampers. The lateral dampers can command 25 percent of available roll control. Any aircraft rolling motion will cause the dampers to counteract the roll. The roll SAS is generally disengaged for maximum performance maneuvering.

Buffet at this flight condition can be felt through the whole airframe. The wing rock is random and usually returns to neutral. Any attempt to fight the wing rock with ailerons aggravates the situation by inducing adverse yaw. The pedal shaker starts at 22.3 units AOA but may not be detected because of the aircraft buffet. Difficulty in detecting pedal shaker action has been a comment made by all pilots in all aircraft that use a pedal shaker. Heavy buffet and moderate wing rock occurs at 23 to 26 units AOA.

The aircraft nose rise will occur at 26 to 28 units AOA. The wing tip and trailing edge are in stall, and the aerodynamic center moves forward causing the nose to rise. After the nose rises, it will slice in either direction resulting in an out-of-control condition unless AOA is reduced. Nose slice always precedes post-stall gyration. If the nose starts to rise and then slice. the AOA must be reduced and the slice corrected with rudder. This should prevent a departure. This is considered to be beyond the safe-flight region and requires good pilot technique and judgment.

The above mentioned characteristics are altered to varying degrees depending on the rate the stick is moved aft or if the ailerons are used abruptly. Even at 15 units AOA, a large aileron deflection can induce sufficient adverse yaw to cause a snap roll and from there on into an out-of-control situation.

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[1] One "unit" AOA change corresponds to 1.05 degrees of noseboom AOA change in the F-4E; 15 units corresponds to 11.4 degrees. Data based on Air Force Flight Test Center measurements.

The optimum unit AOA for maximum performance is between 19 and 20. If this value is exceeded, lift peaks out and drag increases.

In maneuvering the F-4 through maximum performance turns, the techniques employed are basically the same as those employed with other swept wing fighters. Low angle-of-attack maneuvering requires aileron/spoiler as primary control surfaces for initiation of the change of direction and rudder as required to keep the maneuver coordinated. As the intensity of the maneuver increases more rudder is required and less aileron/spoiler. In the high-angle-of-attack regime, the rudder becomes primary for initiation of a change of direction, and the aileron/spoiler combination must be used very judiciously. This variation in control techniques is necessary due to the adverse yaw induced at high angles of attack. Adverse yaw comes from the downward-deflected aileron and the aircraft rolling. By pulling the rudder trim circuit breaker, rudder pedal forces decrease from 77 pounds per inch to 33 pounds per inch of pedal travel. The reduced force makes it easier for the pilot to control the aircraft. This practice of pulling the circuit breaker has recently been forbidden, since at 13 units AOA, a large rudder deflection can cause structural failure if the q is high.

If the above discussed techniques are not used and aileron/spoilers are deflected at high AOA, adverse yaw will produce a rolling moment opposite the turn direction due to dihedral effect. Countering this with more aileron/spoilers will increase the tendency to roll out and may result in a snap which can, if the aircraft is being maneuvered tactically, result in a disaster.

During maximum-performance maneuvering, proper trim technique is extremely important. Although some positive pressure on the stick is desirable, excessive pressures should be minimized by use of trim. This procedure will decrease the possibility of inducing ailerons while at high angles of attack, thus avoiding moderate to heavy wing rock and adverse yaw. Tracking during a gun attack becomes exceedingly difficult if stick forces are not reduced to a comfortable level.

The use of an aural warning to inform the pilot of his AOA, primarily for high-AOA maneuvering, is being installed in one block of USAF F-4 aircraft. Not much experience has been obtained at this time, but initial pilot acceptance seems to be good. Starting at 15 units AOA, the tone, frequency of interruption, and volume are changed as the AOA is increased. This is still a passive system, and it remains to be seen how effective it will be under a stress situation. It is interesting that one of the major U.S. airlines is using a tone for altitude information from the radar altimeter during flareout and landing.

The Category II stability and control evaluation of the F-4E was completed in May 1969. The evaluation in brief concluded that the longitudinal handling qualities were poor with an aft cg, high AOA, or high "SIN" (T.O. IF-4C-1-1 Stability Index Number relating to the total destabilizing effect of wing-mounted stores). The usual external load for air-to-ground attack consists of two wing tanks outboard and a bomb dispenser and rocket pad mounted inboard. This configuration was found to be one of the least stable configurations for maneuvering.

The Category II tests further concluded that stick force cues are inadequate for control of AOA with a high SIN and/or with an aft cg. The 3-pound bobweight system has been installed in the aircraft to correct the tendency toward instability at the low-altitude and high-speed regime. This modification has improved the handling qualities; however, with an aft cg the force gradients were not improved, and the undesirable characteristic of decreasing stick force per g exists.

Interviewed pilots indicated that pilot-induced oscillations at low-altitude high speed on the primary pitch control system (augmentation off) are a concern. Also, when decelerating through 0.96 Mach, a transonic dig is experienced, causing a steady applied stick force to suddenly command an increase in normal acceleration. It is estimated the stabilator effectiveness increases by a factor of 1.4. This is particularly disconcerting during a tactical maneuver in which a high turn rate is being attempted, as it will usually result in a normal acceleration overshoot which will cause a speed reduction.

F-111 PFCS -- The F-111 is a two-place (side-by-side) long-range fighter bomber. The aircraft is designed for all-weather supersonic operation at both low and high altitude. Mission capabilities include: long-range high-altitude intercepts, long-range attack missions and close-support missions. An automatic low-altitude terrain-following system enhances penetration capability. The wings, equipped with leading edge slats and trailing edge flaps, may be varied in sweep, area, and aspect ratio by the selection of any wing-sweep angle between 16 and 72.5 degrees. This feature provides the aircraft with a highly versatile operating envelope. The empennage consists of a fixed vertical stabilizer with rudder for directional control and a horizontal stabilizer that is moved symmetrically for pitch control and asymmetrically for roll control. Stability augmentation incorporates triple-redundant features.

There is actually very little information available on the aircraft behavior while flying on the basic mechanical aircraft flight control system. The primary mode to fly the aircraft is on the control augmentation system (CAS). The probability of flight being required without the CAS in either pitch, roll or yaw is extremely remote. Basic redundancy, failure monitoring, and self-test of the system enhance the full-time operation of the system. Pilot instructions are that in the event of a flight control system malfunction necessitating turning any of the three-axis CAS off in flight, the aircraft speed should be reduced to 40[illegible] KIAS or Mach 0.75 whichever is less. Continued flight should be accomplished with a wing sweep of 26 degrees and a landing made as soon as practical. With an instruction such as this in the Pilots Handbook, T.O. 1F-111-1, there are not many pilots who will explore the flight envelope in the CAS-off control mode.

The flight envelope of this aircraft has not been fully investigated, and General Dynamics is just getting started with the stability and control flight test program. The F-111 aircraft that are now operating in the USAF are restricted to 15 degrees AOA, and the rudder shaker starts at 18 degrees AOA. This is a very limited and restricted envelope.

In August 1970 at ASD, representatives from NASA, AFFTC, AFFDL, USN, TAC and AFIG met and formed an ad hoc team to objectively evaluate the total F-111 spin test program. This report as presented represents the opinion of individuals whose credentials are outstanding in the area of stall/post-stall/spin prevention. The information is pertinent to this program study, and there is correlation and mutual confirmation in conclusions reached. The F-4 and A-7 are both discussed in the ad hoc report.

The summarized recommendations from the ad hoc report that are pertinent to the F-111 are presented as follows:

The F-111 spin program as presently defined complies with MIL-S-25015 which requires tests to show the fully developed spin characteristics of an airplane. The program is not recommended. Testing to define stalls and post-stall gyrations, entered out of tactical maneuvers, is recommended. Entry into these out-of-control maneuvers including incipient spins should be made from flignt conditions which simulate actual use of the aircraft. The test results will then be directly applicable to the needs of the operational pilots.

A-7 PFCS -- The A-7 is a single-engine, single-place, transonic light attack aircraft with an all-weather capability. Principal recognition features include a shoulder-mounted wing with a marked degree of negative dihedral, an all-moving horizontal stabilizer, a low-profile fuselage and a rounded nose radome mounted above the engine duct. It is a short coupled aircraft with the ailerons on the outer section of the wings.

The pilot interviews were conducted at the 58th Tactical Training Wing Test, Detachment I, Luke AFB. This squadron is charged with the responsibility of evaluating the aircraft and developing techniques and procedures to make the aircraft an effective weapons system. Several of the pilots including the commanding officer (Lt. Col. Chuck McLarren) have served a tour in the Navy in A-7's. Practically all the A-7 experience in the USAF is within this pilot group.

The unaugmented aircraft can be flown at all points of the flight envelope, but precise control and mission accomplishment would not be possible. Pitch and yaw are so underdamped that the pilot must use smooth deliberate inputs to control the vehicle.

Regarding the pitch axis, the pilots commented frequently that the artificial feel system tended to produce a varying pilot stick feel almost from flight to flight. Records show that 5 to 6 percent of the writeups were about the changing feel in the pitch axis. Also, the pitch feel spring has been a frequent replacement item.

The yaw and roll damping on the basic aircraft are unacceptable for anything except emergency flying. With aft cg and external stores, the aircraft is barely controllable -- it wallows and feels like it wants to swap ends. It is interesting to note that directional stability (C_{N_β}) goes to about 0 and becomes negative at around 23 units AOA. However, the aircraft does not stall until 24 or 25 units AOA. Therefore flight occurs often in a region of negative directional stability at high angles of attack. Also C_{L_β} or dihedral effect reaches its lowest value at 23 units AOA. This adds up to a potential departure situation which will be manifested by an abrupt nose slice with little roll until a high value of beta is reached.

Optimum maneuvering is around 20 units AOA, and, in a panic, 22 units AOA can be used, for example in a dive recovery where airspeed loss is not a consideration.

Since the aircraft does not stall until 24 to 25 units AOA and directional stability is as previously mentioned, it is to be expected that the aircraft is going to be maneuvered on occasion at 23 or 24 units AOA with sometimes startling results. The cue to the pilot on what his airframe is doing other than the AOA indicator comes to him in the buffet characteristics. Buffet characteristics do not, however, provide a very clear indication of the specific AOA. Buffet starts early at 17 units AOA and increases to a fairly high level. Just before departure from controlled flight, the buffet is severe. The rudder shaker starts at 20 units AOA, but aircraft buffet almost completely masks the artificial stall warning.

The U.S. Navy is at this time conducting a comprehensive spin investigation of the A-7. The project pilot is Lt. Commander D. D. Smith, USN. The Navy program is particularly detailed in the high AOA flight region with a wide variety of external ordnance. The attack mission requires vigorous maneuvering at low-altitude high speed with heavy external loads. The Navy will use the information from the spin investigation to develop a pilot training program (including a film) to familiarize squadron pilots with the A-7's high-angle-of-attack and departure flight characteristics. Results of the tests so far indicate that the A-7 can be departed from controlled flight safely without significant danger of going into a spin.

F-101B PFCS -- The F-101B is the oldest aircraft in the survey, becoming operational in the 1957 period at the time the F-4 was making its first flight.

The unaugmented aircraft can be flown safely throughout the established flight envelope. The roll axis is very sensitive, and at high-q low altitude there is a tendency toward pilot-induced oscillations. Several studies were performed during early development of the aircraft, and it appears the ailerons are large for high-q subsonic flight but just right for supersonic flight. The problem is easily accommodated in pilot training. The basic yaw axis is underdamped at all flight conditions, with a 10 to 12 cycle to damp being required in smooth air. If the air is rough, the unaugmented aircraft has a continuous wallow which although it is not serious, precise control for tracking is impossible and flight in weather may be difficult.

The aircraft has a thin swept, low-aspect-ratio wing and high-mounted stabilizer. The high stabilizer is, by position, free of airflow disturbances throughout the normal operating flight envelope. This high stabilizer position provides good pitch stability in the normal envelope. However, at high AOA an undesirable characteristic occurs called pitch-up. The aircraft destabilizes at low speed and/or high "g" loading and cannot be full-stalled short of longitudinal pitch-up. This defines the extreme boundary of the aircraft capability.

Pitch-up, briefly described, is caused by wing-tip stall and down-wash airflow over the horizontal stabilizer. As the tip stall progresses inward and forward along the wing, that area still producing lift is centered forward. Tip stall in effect also decreases wing span and concentrates lift distribution in the inboard wing and fuselage area. This results in a sharp upward airflow deflection at the wing which then washes downward across the tail. Negative stabilizer angle of attack, in relation to downwash flow direction, will produce negative (downward) lift at the tail and, combined with the forward-shifted center of lift, will cause a nose-up tendency. If nose-up movement is allowed to continue, forward stick will become progressively less effective, and pitch-up will occur. In the subsonic flight region, approach to pitch-up is well advertised by buffet, but, in supersonic flight, there is little or no buffet prior to pitch-up; and if the q is high, structural failure is a very likely possibility.

Pitch-up is a serious flight problem, and considerable training and engineering effort has been expended over the years to minimize the problem. At the time this report was written 26 aircraft had been lost due to pitch-up.

It is appropriate to discuss these various pitch protective devices in some detail, since at least two of them utilize techniques that could be considered to be applicable to new aircraft. These two, the manual command signal limiter and the pitch boundary indicator have received good pilot acceptance. The horn and pusher systems preceded these devices.

The horn and pusher systems were the first protective systems developed and came into being shortly after the serious nature of the pitch-up characteristic was fully appreciated. The recognition of pitch-up came after the airframe was finalized and very close to the time the first aircraft were to be delivered to the USAF.

The horn and pusher systems are separate channels taking angle-of-attack information each from vanes mounted on opposite sides of the aircraft nose. Each channel takes stick rate and angle of attack and matches it to carefully flight-investigated maximum angle-of-attack boundaries. These boundaries parallel each other with the horn at a slightly lower angle of attack. The boundary is scheduled with Mach number. As the aircraft reaches the horn boundary, the horn blows providing an aural warning to the pilot. If the angle of attack increases beyond the horn, the pusher drives the stick forward at a fixed rate. This system has never been well accepted by pilots, mainly because pilots dislike the stick being taken away from them. Inadvertent pusher actuations can occur frequently, mainly because of the stick rate signal which becomes large when the stick moves quickly, even though the aircraft is not near the critical angle of attack.

The horn and pusher systems were installed in all F-101 aircraft, but the command signal limiter and pitch boundary indicator were installed only in the F-101B interceptor version. The reason was that the interceptor mission places severe maneuver requirements on the aircraft at high altitudes where the aircraft may be supersonic and up against the maximum angle-of-attack capability.

The pitch boundary indicator (PBI) is a cockpit instrument which shows on one needle the existing angle of attack and on another needle the maximum angle of attack scheduled with Mach number that is attainable at that particular flight condition. Therefore, at a glance the instrument tells the present angle of attack and what is available for maneuvering. This helps the pilot judge quickly at any flight condition how much aircraft maneuver capability he has available.

The command signal limiter (CSL) evolved from a pitch limiter mode of the AFCS to its present configuration as a full-time dual-channel limiter on the basic aircraft. This evolution followed the path of first being noted as an effective AFCS mode for maneuvering the aircraft on control stick steering (CSS). The USAF was pleased with this pitch protection concept, and a program was instituted to provide this same concept of protection for the basic aircraft. Later it was decided to remove the pusher and horn and replace it with a dual-channel command signal limiter which is now the primary operational pitch protection system in all F-101B aircraft.

The following is a brief simplified description of the dual-channel command signal limiter concept as it is now mechanized in the F-101B aircraft.

The redundant pitch limiter system in F-101B aircraft is composed of two channels of sensors, power supplies, electronic computation, separate hydraulics and servos. Self test provides the pilot with a simple end-to-end test to assure himself, with no ground assistance, that his system is working correctly. The information being sensed and used in the computation very completely describes the aircraft behavior -- forward and aft accelerometers, stabilizer position, pitch rate and angle of attack with appropriate q_c and Mach scheduling.

The limiter system is on at all times during flight and is considered a basic part of the primary control system. It is always ready; and, as the critical angle of attack is reached or rapidly approached, one channel will engage and hold the aircraft smoothly on the boundary which defines the maximum angle of attack for the existing flight condition. A 60-pound stick force is required to penetrate the boundary, thereby providing a smooth, firm resistance to hold against. The second channel will only engage if a failure occurs in the first channel.

Experienced F-101B pilots consider that the limiter gives them not only protection from pitch-up but it gives them the capability of holding this aircraft, even during rapidly changing flight conditions against a firm boundary limit. This permits the aircraft to be smoothly maneuvered head-out-of-the-cockpit at the maximum safe angle of attack for any flight condition. It is part of the Air Defense Command tactical maneuver procedure following a snap-up attack to ride the boundary during the recovery.

Pilot acceptance of this concept of limiting has been excellent and extends over an operational period beginning in 1958, which certainly provides a time base sufficient to draw firm conclusions as to pilot acceptability.

This limiter concept was discussed extensively with all pilots interviewed, and, after a good explanation, even the pilots not experienced in the F-101 commented that the concept seemed good, and that in a new aircraft they would welcome some kind of a limiter which would permit complete head-out-of-the-cockpit maximum stick commands to the safe limit of the aircraft performance capability.

Pilot Opinion on Contribution of Augmentation Systems

In reviewing the four aircraft in this survey, pilot consensus is that all four aircraft can be controlled safely with one SAS axis off at all points of their respective flight envelopes by military pilots of average skill, provided they exercise smooth deliberate control inputs and they know which axis of augmentation is not operating. This in no way means that the assigned mission can be accomplished, since precision control would probably not be possible, and pilot workload would be very high since the pilot will have to carefully supervise each control input. He may even have to resort to a sampling or trial-and-error technique of small commands to assess the vehicle response.

In the case of the F-111, the flight handbook presents a very limited portion of the flight envelope that may be flown without control system augmentation. However, after a discussion with the General Dynamics F-111 project pilot, it was found that actually the aircraft could be flown at any of the established flight conditions if the precautions set forth in the preceding paragraph are observed.

It is significant to this study that, in every one of the survey aircraft, one or more axes of augmentation was considered by pilot opinion to be essential to mission accomplishment. A summary of this opinion is discussed in the following paragraphs by aircraft.

F-101B SAS -- The F-101B has augmentation only in the yaw axis. Pilot comment is that the yaw damper is essential for mission accomplishment. At most flight conditions in smooth air, 10 to 12 cycles are required for the free aircraft to damp the yaw oscillations coupled with roll. If the air is turbulent, this causes a continuous roll and yawing which is of sufficient amplitude and frequency to be disconcerting to the pilot, particularly when flying on instruments. For the F-101B there is no aiming task, so the requirement for the damper is primarily to improve aircraft handling and was not designed for tracking tasks.

The redundant limiter system (RLS) in the pitch axis is considered to be part of the basic aircraft and is mandatory for flight. The RLS is used during recovery from "snap-up" attacks permitting the pilot to pull on the stick against a firm resistance at the maximum safe angle of attack.

F-4 SAS -- The F-4 pilots consider the pitch damper to be essential to mission accomplishment, primarily at high airspeed and low altitude. The basic aircraft exhibits a tendency toward pilot-induced oscillations, and the damper reduces this threat.

When in a tactical situation, either air combat maneuvering or precise air-to-ground ordnance delivery, the recommended procedure is to turn off the roll damper. The damper seems to offer some resistance to small commands as used in precise tracking, thereby raising the pilot workload and reducing his ability for precision control.

At high angles of attack, the roll damper may also aggravate departure tendencies.

The yaw damper apparently works in an acceptable manner, and the only pilot comment concerning rudder action is that, at high AOA when a lot of rudder is used by the pilot, the pedal forces are considered to be heavy.

A-7 CAS -- The A-7 pilot comments were that augmentation is essential in all three axes for mission accomplishment. The yaw axis is particularly underdamped, and several unsatisfactory performance reports have been submitted by the 68th Test Detachment at Luke AFB regarding this problem. In addition to being underdamped, there is a hold-off of several seconds in yaw which seems to be caused by the lateral accelerometer taking 5 to 6 seconds to trim. This is particularly difficult to live with in a tracking air-to-ground situation when there are often only a few seconds available for gun firing. It is essential to have an aircraft that can be maneuvered precisely and quickly, with all axes being well damped for a good gun platform. The optimum coordination could be realized if the aircraft could be made to roll about the gun barrel axis instead of about the velocity vector with minimum sideslip.

In turbulence in the lateral axes with augmentation at certain airspeeds and external loadings, the oscillations are of an amplitude of ±10 degrees in heading which is very loose. The damper definitely needs to be tightened up.

The roll damper is automatically disconnected at 22 degrees AOA. This is to eliminate the pro-spin characteristics of the damper which, according to pilot comments, can be all that is needed to lock into a spin.

F-111 CAS - The F-111 was designed to use control augmentation in all three axes as the primary mode of flight.

The pitch and roll channels are adjusted by a self-adaptive gain changer. This gain-changing technique attempts to maintain the system gain as high as possible within the constraints of system stability. This tends to hold the short-period characteristics near optimum throughout the flight envelope. During periods of extended flight in smooth air, the gain can drift to a critical level which can, on occasion, manifest itself with a momentary instability of several cycles of the control surface and even into the control stick at a frequency of 2 Hz at a small amplitude. On the other hand, if the aircraft is being flown in turbulence or in formation where many control inputs are required, the gain will be lower than optimum, and the aircraft will be less responsive to stick commands and the damping will be degraded.

The system gain is also affected by acceleration and deceleration of the aircraft which is rapid enough so that the gain changing characteristics cannot keep up. Pilots have commented that the only place this gain-changing characteristic is objectionable is the situation where, as they pull up to start an air-to-ground ordnance delivery, the gain goes up as the aircraft slows down. Then, as the aircraft noses down, it accelerates very rapidly, and the gain goes critical and the oscillations are objectionable. To get around this problem, the pilots have developed a practice of momentarily turning the pitch CAS off before starting with acceleration. This drives the gain down to a lower level and prevents the oscillations from occurring.

The mechanical flight control system in pitch has a fixed-feel spring and provisions for trimming the aircraft. Primary flight control is on the automatic mode, and automatic series trim is provided. At one time a low-speed trim-compensation system was used whenever the slats were extended to provide speed stability during takeoff and approach. An interesting fact regarding this low-speed compensation device is that it has been disconnected for some time, and very few of the pilots have noticed that it was inoperative. During normal flight operation, the stick is maintained at a near central location by the auto trim.

Pitch stick command per inch is held uniform by the CAS which varies the elevator deflection per inch of stick throughout the flight envelope. The combination of auto trim and uniform aircraft response to stick commands has met with some mixed pilot comment. One aircraft has been lost due to a failure in the fuel transfer system which caused an aft cg (52 percent) condition to develop. The series autotrim and lack of stick position change completely masked the fact that the stabilizer was almost fully trimmed in one direction, and the degraded stability of the basic airframe (masked by the CAS) was not apparent to the pilot. As he slowed down, the aircraft went out-of-control and was lost. The system had taken away the basic pitch-feel cue which most pilots depend on. The autotrim does take away the speed-change cue, and pilot comments seem to be divided on whether speed-stability characteristics are really necessary. If the pilot is occupied with a particular problem which is absorbing most of his attention, then the loss of the speed-stability cue can be detrimental to vehicle control. In the roll axis, the command augmentation system tends to minimize the variation in roll rate per pound of stick force with flight condition.

Pilot Recommendations and Discussion

Pertinent pilot recommendations are summarized in this section. Briefly, pilots want a vehicle that is reliable, has the performance sufficient for a mission success, and has a high degree of survivability included in the design. To achieve these objectives, the interviews revealed that most pilots at this time expect and look forward to large changes in the configuration of new aircraft. The operational survey attempts to answer some of the questions pertinent to the development of advanced control technology.

Abnormal Maneuvers -- The pilot wants to be able to command the aircraft rapidly and safely to the maximum performance limits. Control limiters, shakers, force detents, horns, pushers. varying tones in the earphones and panel indicators are all part of the array of devices attempting to keep the pilot within the prescribed flight limits.

The ultimate control system would permit the pilot to command the aircraft to maximum performance limits and then smoothly and safely hold him within the prescribed flight limits.

Pilot comments are strong for future aircraft to be designed to have spin-resistant characteristics.

The out-of-control situation is the main concern of the pilot, and it is difficult to determine how much of the aircraft performance is actually compromised by the pilot as he attempts to play it safe. This is a dangerous practice, because his success on some missions will be determined by how effectively he can use the aircraft performance right up to the limit. If he is not familiar with the aircraft behavior, he is a candidate for an out-of-control situation.

"Out-of-Control" Situation -- Spirals, stalls, post-stall gyrations, departures, incipient spins, upright and inverted spins, and flat spins are all "out-of-control" maneuvers, and each must be recognized and correct control recovery action taken. Fifty-one F-4 aircraft have been lost due to out-of-control condition. This becomes very significant when it is believed that, in all 51 situations, the pilot did not exercise all aspects of the "correct" recovery technique (which in itself was subject to considerable confusion). The difficulty in identifying an out-of-control state indicates a need for an automatic recovery system or an aircraft with inherent recovery properties. Provision of the latter may entail unacceptable performance compromises.

Reports and pilot comments indicate that the pilot has inadequate timely information that a critical flight condition is approaching. Also, the situation can deteriorate rapidly at a moment when the pilot may be applying his full concentration on another task.

Information obtained in pilot interviews indicates that major factors causing aircraft losses through out-of-control conditions are:

- Design deficiencies
- Insufficient flight testing of aircraft performance
- Incomplete pilot flight manual information
- Inadequate aircrew training

A major contributing factor to the problem in the case of the F-4 and F-111 are the six or seven years between aircraft service acceptance and the completion of full stability and control flight investigations.

Flight Limiters and Monitors -- All the aircraft in the survey are equipped with AOA warning or limiting devices. The F-4 is equipped with a rudder shaker and an aural tone. The A-7 and the F-111 are equipped with rudder shakers. The F-101 has had or has a horn, pusher, pitch boundary indicator and command signal limiter.

The rudder shakers in the F-111, A-7 and F-4 are considered to be of little use due to the fact that they are masked by heavy aircraft buffet.

The stick pusher system was used in the F-101 until a few years ago when it was replaced by a dual-channel command signal limiter. Stick pushers are not very popular with pilots since they resent having the stick taken away from them. Then there is always the concern that the pusher will inadvertently actuate at an inconvenient moment.

The pitch boundary indicator which tells the pilot his present angle of attack as well as the maximum AOA attainable has received good pilot acceptance. However, the effectiveness of this device is limited since it must be looked at for information.

In the F-4 an aural tone which sounds at 15 units AOA and increases in frequency and volume as the AOA increases appears to be well accepted by pilots. This system is new and has not had sufficient operational use to prove the concept.

A stall-inhibiting device is being considered for the F-111 in addition to the rudder shaker. In the F-111 there is little degradation in handling qualities as it approaches stall, and during the post-stall gyrations the engines may not provide enough power to the flight control system to effect a recovery.

Pilot comments were favorable on the closed-loop command signal limiter as installed in the F-101B. This system has been in operational use for 12 years. It is now dual-channel on the F-101B and is the primary pitch protection system since the pusher has been removed. It is used tactically during recovery from snap-up attack maneuvers in order to get the maximum safe change in the aircraft flight path. It can be used in any tactical situation to aid the pilot in attaining and holding a maximum safe AOA. An advantage of a physical limiter is that it will act to reduce or hold a safe angle of attack with no input from the pilot. During the stress of a departure or post-stall gyration, the pilot may have difficulty identifying the situation. In the confusion, passive monitors such as shakers or horns can be overlooked.

The pilot wants to be able to command the aircraft in any axis to the maximum aircraft capability without looking in the cockpit, during severe buffet, and with his entire senses concentrated outside the cockpit on perhaps another aircraft that could be closing with him at 3000 mph.

Augmentation Systems --

Aircraft Response Characteristics -- Pilot opinion on desired aircraft response characteristics is greatly influenced by the aircraft he is flying or has flown. This is unfortunate since, with older augmentation systems, the pilot has learned that increased damping is usually accompanied by a slow response which he finds impossible to accept. Therefore, in order to have a rapid responding aircraft, the pilot will develop his techniques and judgment so he can live with an underdamped aircraft. It can be noted that, in all of the four survey aircraft, response rate and damping were less than the pilot really would like and that, at some flight condition, performance was degraded enough to affect mission accomplishment.

The interview discussion emphasized that, with new control technology, vehicle response could be very rapid and that the pilot's physical makeup would probably define the maximum response rates. The desired response is rapid, smooth and well damped -- specifically, response times of less than one second and damping ratios of over 0.6.

As background for the previous paragraph, consider how a pilot flies his aircraft. The pilot works a good part of the time on a digital-sample-type technique. Particularly during instrument flight, he operates on an instrument-scan technique where he observes a certain number of bits of information and makes control inputs to achieve a desired result, repeating this cycle until the desired vehicle situation is accomplished. Then he goes into a monitoring mode to keep the vehicle at the desired condition. The best aircraft response behavior for this type of flying is smooth, well-damped, fast, and without drift. Thereby fewer corrections are required and the workload is lower.

If the airplane/system combination has a tendency to drift during this "sampled-data" control process, many scan cycles are necessary before the errors are reduced to within some personal "tolerance" band. Such a property may be manifested by "automatic trim" systems which have long (e.g., 4 to 6 seconds) settling times. These prove to be very objectionable unless the pilot can devote his entire attention to the related control axis, obviously not possible in most situations. The available "time on target" in most weapons delivery situations dictates that response times be fast -- less than a second for superior aiming.

If the pilot is maneuvering with respect to another aircraft or ground object, he will lock his eyes on the object and attempt to command or adjust the aircraft flight path to reach the desired point. The desired response from the aircraft is the same as before -- rapid, smooth and well-damped.

Turn Coordination -- The most stringent performance requirement for turn coordination occurs during air-to-ground or air-to-air tracking.

Maximizing the time the weapons are on the target is the objective. This time can be as low as 2 to 4 seconds per mission.

Performance that is particularly objectionable to the pilot is adverse yaw, low yaw damping, and slow-drool-type sideslip trim. Anticipating and using a control technique to neutralize these problems presents a sizable pilot workload in a compressed time period. Somewhat less objectionable is the requirement for pilot skill and judgment in the use of the rudder at high AOA.

An important aspect of the question of turn coordination is the objective of the maneuver. In the case of gunnery, good control of the projectile stream is desired. The concern is coordination of the aircraft about the gun barrels to increase the time on target. For example, if the gun barrels are pointed below or above the aircraft velocity vector, the gun aiming point will be either adverse or proverse respectively to a roll input. Therefore, to have the gun response optimum, the turn coordination would need to be designed with the boresight as the rotation axis rather than about the velocity vector (even at the expense of sideslip). A study and flight test is now in progress for such an aiming mode by SAAB and Honeywell in a Viggen 37 in Sweden.

Because provision of a good lateral aiming response will in general compromise coordination in the conventional sense (i.e., zero sideslip), it is evident that the flight control could provide selectable response modes which better satisfy variable mission tasks. Pertinent to the subject survey is the question of whether the variable response qualities resulting from such an implementation would meet with pilot acceptance. There is considerable evidence that it would, judging from the practice of pilot disablement of roll SAS and yaw trim prior to air combat maneuvering in the F-4. Comparable activities have been reported by A-6 pilots in their use of a spin-recovery switch and by "Mohawk" pilots in their use of slight flap extensions, both of which offer extra surface authority for improved maneuvering capability. The lesson here is that if a pilot can obtain improved performance by any means, approved or otherwise, he will exercise this ability at appropriate times and adapt to the associated response changes. The desire for "better performance" in a military airplane is never satisfied.

Control Margins -- Reflection on the properties of conventional flight controls shows that they provide more than just a means for adjusting airplane attitude and flight path. They also provide indications to the pilot of available control margins in terms of how much surface deflection yet remains. Unfortunately, information does not come without penalty in terms of inconvenient stick positions in some cases. As flight controls become more advanced, the relationship between stick position and surface deflection becomes less firm (much as the relationship between stick force and hinge moment was destroyed by the fully powered surface). This loss in implicit surface information must be replaced by a display function, for available control authority is as an important a resource as fuel and oxygen.

The consequences of not knowing control margin can be severe, as exemplified by a lost F-111. An aft cg developed (52 percent) as a result of incorrect fuel sequencing -- the autotrim maintained the stick position as the aft cg developed, and the high-gain adaptive pitch CAS masked the deteriorating pitch stability. When the trim ran out, the aircraft pitched up and was lost.

Reliability -- Pilots agree that equipment should always work. If it must fail, they want to be advised beforehand; and once advised, they want alternate or backup equipment.

Fly-by-Wire -- Most pilots interviewed agreed that FBW sounds like a logical and reasonable development. This favorable attitude could be a significant factor in acceptance of FBW in forthcoming development programs.

Pilot Training -- The discussion of air combat training or maneuvering (ACM) has come up in almost all interviews. There appears to have been a reluctance to put emphasis on ACM for the reason that it is somewhat hazardous and must be tightly controlled by supervision and rules. However, there is no question that both the Navy and Air Force must develop ACM programs if we are to maintain air superiority. Recent experience of a one-for-one kill ratio in SEA has brought the problem into focus. It is the old story of how a good, tough, realistic training program produces pilots who can survive an enemy engagement, destroy an opponent, destroy a target, and return safely.

RELIABILITY-MAINTAINABILITY-SAFETY

Data Sources and Definitions

USAF Data Reporting System -- The primary source of statistical reliability-maintainability data for this program was the USAF Maintenance Data Reporting system commonly referred to as the AFM 66-1 system.

The AFM 66-1 maintenance reporting system generally operates in the following manner:

- The individual who performs a maintenance action records the action taken by a Work Unit Code number, the amount of time expended, the type of action taken, equipment type, etc.

- The above information is transferred to data processing cards along with additional information such as weapon system designation, base location, etc.

- The data cards are transmitted to Wright-Patterson AFB and are read onto a magnetic tape. An estimated 5,000,000 maintenance actions are recorded on this magnetic tape each month.

- The data processing center at Wright-Patterson uses a program to "edit" the raw data to eliminate duplicate reports, reports from the incorrect time periods, etc. The deleted data is documented under a set of reports designated RSC-XX-Log K 260 to allow retrieval of this data if desired.

- The edited magnetic tape is now used to generate two basic types of reports:

  (1) RCS-XX-K261 "ON" Equipment Maintenance Action Reports - These reports contain the failures, maintenance actions, aborts, etc. for equipment which required maintenance at the aircraft level. Approximately 25 standard reports giving various analyses of the raw data are distributed. Specific reports used for survey data were the following:

RCS-06-K261	Failure, Maintenance, Abort Analysis
RCS-07-K261	Abort Analysis
RCS-15-K261	Accident/Incident Analysis
RCS-25-K261	Maintenance Manhour/Flight Hour Analysis

(2) RCS-XX-K262 "OFF" Equipment Maintenance Action Reports - These reports contain the failures, maintenance actions, repair actions taken, time expenditure and other data associated with devices which are not being utilized on the aircraft. The depot-level repair actions and failures are the prime data basis for the "OFF" equipment reports. This data was not utilized in the operational survey as its goals are more oriented toward repair time, etc.

The AFM 66-1 data system provides a large amount of statistical maintenance data which is updated monthly. The majority of reports cover a 6- to 12-month period; hence, the impact of Technical Change Order modifications and other maintenance practice changes can be assessed. The environmental conditions, reporting methods, base workload, training schedules, T.O. adequacy and various other items can affect the statistics reported to the 66-1 system; consequently, the environmental situation surrounding any given analysis report should be examined to properly evaluate the maintenance data. Since all AFM-66-1 maintenance data is recorded against a Work Unit Code, a brief description of Work Unit Codes is given below.

The Work Unit Code (WUC) is a five-character alphanumeric designator used to define a given piece of equipment within a major weapon system (i.e., aircraft, missile, etc.). The first two numbers in the WUC represent a major subsystem. WUCs are assigned to each weapons system which uses the AFM 66-1 maintenance reporting system. For the aircraft involved in this survey, the WUCs are defined in the 06 Aircraft T.O. (i.e., the WUC Technical Order for the F-4C is T.O. 1F-4C-06).

The majority of the statistical data presented in the following paragraphs was obtained from the AFM 66-1 data pool described in Table 3-8.

Table 3-8. AFM 66-1 Data Pool

Aircraft	Report Source	Failure Reporting Time Frame	No. of Flight Hours	Average No. of Aircraft
F-101B/F	OOAMA	3-1-70 to 8-31-70	15,006	155
F-4C	OOAMA	3-1-70 to 3-31-70	42,748	344
RF-4C	OOAMA	3-1-70 to 8-31-70	57,962	348
F-4D	OOAMA	3-1-70 to 8-31-70	97,230	576
F-4E	OOAMA	3-1-70 to 8-31-70	92,990	485
A-7D	OCAMA	6-1-70 to 12-31-70	7,358	30
F-111A	SMAMA	6-1-69 to 12-31-70	14,149	60

The actual data obtained from the 66-1 reports are the following items:

- Maintenance Action - Each time effort is expended on a component or system, a maintenance action is recorded. Items such as lubrication, visual inspections, removal for access to other components, etc. are all recorded as maintenance actions.

- Mean Time Between Maintenance Action (MTBMA) - This number is computed by dividing the total number of flight hours by the number of maintenance actions. This gives a feel for the number of times a component is handled, tested, removed, repaired, etc.

- Maintenance Manhours - The number of maintenance manhours expended on each task is recorded and reported. The manhours are divided into scheduled, unscheduled or troubleshooting, and shop maintenance time.

- Maintenance Manhours per Flight Hour (MM/FH) - This number is computed by dividing maintenance manhours by the number of flight hours; hence system to system and aircraft to aircraft maintenance effort comparisons are available.

- Failure - A failure is recorded whenever a system or component does not perform to its Technical Order specifications or when physical damage has been incurred by a given component. These failures can be divided into various hazard classifications which relate the failure to its vehicle safety implications. These classes and typical types of failures in each class are defined in a subsequent subsection.

- Mean Time Between Failure (MTBF) - This number is computed by dividing the number of flight hours by the number of failures.

- Abort - Whenever a scheduled flight is canceled or terminated because of equipment failures during preflight or in flight, an abort is recorded in the AFM 66-1 data. The aborts are categorized by time of occurrence, i.e., preflight or in flight.

- Mean Time Between Abort (MTBA) - This number is computed by dividing the flight hours by the number of aborts.

- Accidents/Incidents -- Accidents and incidents are recorded whenever aircraft damage or personnel injury is incurred or threatened. The accident/incident reports are also used in preparation of safety reports by the various USAF Command Headquarters.

Accident/incident data was also obtained from Norton AFB for a 5-year period for flight control system failures. The 5-year period provides an acceptable confidence level in the statistical flight control accident data. The safety source data consisted of the final analysis report on each of the accidents/incidents reported to Norton AFB during the 5-year period.

Flight Control System Classification -- The data collated was oriented toward definition of criteria for high-authority, closed-loop primary flight controls. This type of system includes "fly-by-wire" system mechanizations. Figure 3-9 indicates which existing aircraft systems were of primary interest for this study. Data on the complete aircraft flight control systems, the hydraulic and electrical power supplies and on the aircraft instrument system were obtained to provide a broader perspective in evaluating the existing operational flight control system problems.

The aircraft flight control systems consist of three major subsystems referred to as the primary flight control system, the secondary flight control system and the automatic flight control system.

The following definitions will be used for the remainder of this section:

- Manual Flight Control System - This system consists of all items which comprise the primary and secondary flight control systems.

- Primary Flight Control System (PFCS) - The PFCS is defined as the total set of hardware used to provide vehicle maneuverability and trim. The PFCS therefore includes the control stick, rudder pedals, linkages, actuators, feel and trim components and the actual flight control surfaces. The surfaces and associated control and trim systems defined as primary for each of the survey aircraft are:

    - F-101B/F - rudder, ailerons, stabilator
    - F-4 - rudder, ailerons and spoilers, stabilator
    - F-111A, E - rudder, spoilers, stabilizers
    - A-7D - rudder, ailerons, stabilizers

- Secondary Flight Control System (SFCS) - The SFCS is defined as the hardware which is used to alter the long-term aircraft aerodynamic configuration. The SFCS typically consists of wing leading and trailing edge lift devices (flaps, slats, speed brakes, wing sweep/fold mechanisms and landing/takeoff flaps).

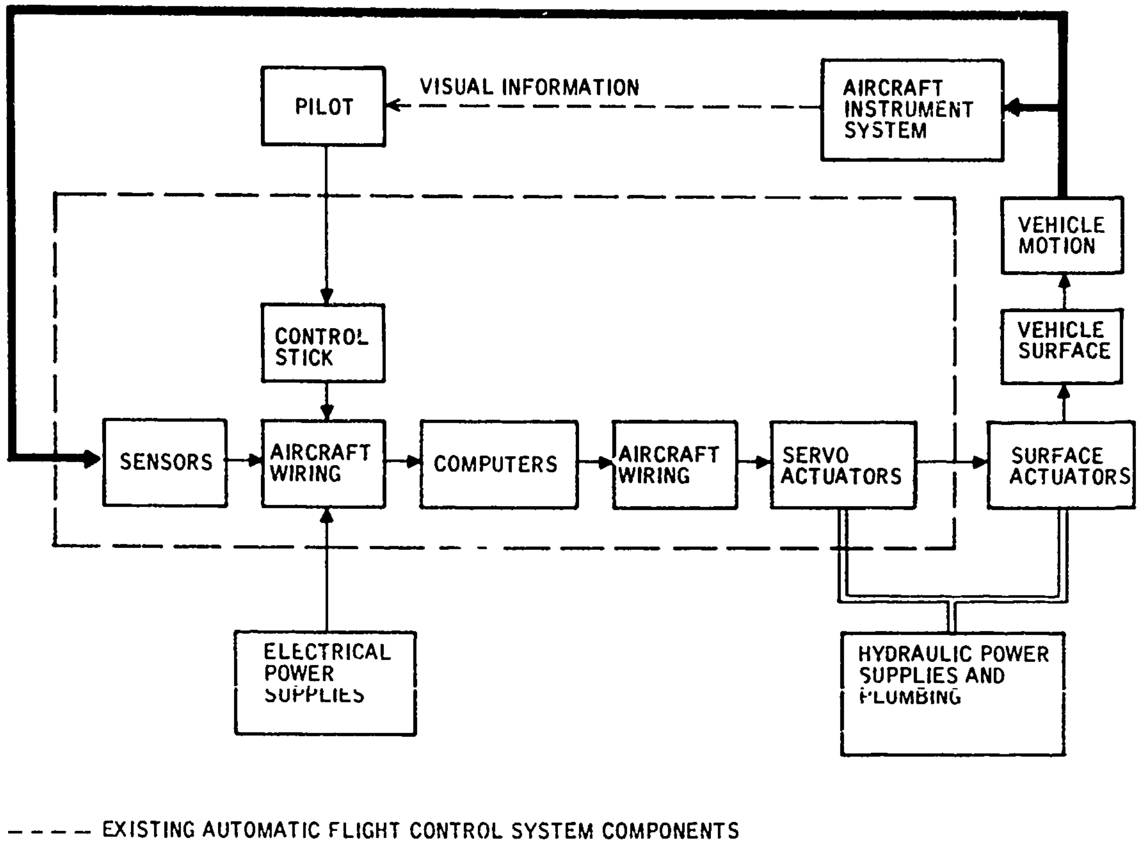

Figure 3-9. Future High-Authority Closed-Loop Primary Flight Control System -- Simplified Block Diagram

- Automatic Flight Control System (AFCS) - The AFCS is defined as the set of hardware which provides aircraft control through the use of closed-loop feedback information. The AFCS normally consists of a stability augmentation system and an autopilot. The stability augmentation system (SAS) is also commonly referred to as a damper. If the SAS is mechanized such that manual pilot commands are electrically summed with the stability augmentation signals, the system is referred to as a command augmentation system (CAS). The autopilot (A/P) provides the long-term vehicle control (attitude, altitude, Mach hold, etc.) and the mission control modes such as automatic attack, terrain following, ground track following, etc. The hardware included in the AFCS consists of control panels, pilot force/position transducers, computers, actuators, and sensors which provide vehicle motion or aerodynamic information. The sensors included as part of the AFCS in this study are all those which are unique to the AFCS or those which are primarily in the vehicle to provide AFCS information. Sensors and computers associated with the radar altimeters, heading information systems, air data computers, and bomb delivery systems are not included as part of the AFCS.

- Hydraulic Power Supply - This system includes the hardware associated with the various 3000-psi hydraulic supplies on the aircraft. The hydraulic pumps, accumulators, valves, plumbing lines, filters, etc., are all included as part of the supply system.

- Electrical Power Supply - This system includes the hardware associated with supply of the 115-vac 400-Hz and the +28-vdc electrical power. The hardware includes the generators, switching gear, rectifiers, etc.

- Aircraft Instrument System - The instrument system is composed of all instruments and displays used by the flight crew. The instrument system varies considerably as a function of the aircraft mission; however, representative reliability-maintenance values were obtained.

The above systems were subdivided further in many cases, and detailed data for further divisions is included in Appendix II.

Failure Classifications -- The relationship of system failures to aircraft mission completion and flight safety is a prime concern in PFCS reliability specification. The AFM 66-1 data system reports a failure whenever a device does not perform to its Technical Order specifications; consequently, the reported failures can have a wide range of effects on system performance and on vehicle safety. The System Safety Engineering Requirement Specification (MIL-STD-882) provides a method of relating failures to the safety hazard which they create. The classifications defined do not precisely fit the requirements of this report, so slightly modified MIL-STD-882 definitions as given below will be utilized:

- Class I - Negligible -
  - Failures which do not result in personnel injury or system damage.
  - Failures such as slight hydraulic seal leaks, high rate gyro nulls, etc.
  - These failures do, however, impact the logistics and maintenance workload; hence, they are a major concern of the USAF maintenance officer.
- Class II - Marginal -
  - Failures which can be counteracted or controlled without injury to personnel or major system damage.
  - Failures such as loss of one channel of a triple-redundant system, low hydraulic pressure level, etc.
  - These failures may or may not be reported as aborts depending on when they are detected. For training missions, these failures may result in a mission abort.
  - These failures directly affect the operational personnel because additional missions, vehicles or both must be programmed to provide the desired training levels.
- Class III - Critical -
  - Failures which will cause personnel injury or major system damage or will require immediate corrective action for personnel or system survival.
  - Failures such as loss of one channel of a dual-redundant system, loss of navigation gear, loss of flight control system feel and/or trim functions, etc.
  - These failures affect operational performance to the extent that mission success is improbable. This class of failure will result in a mission abort unless discovered on the return leg of the mission.

- Class IV - Catastrophic

  - Failures which cause death or severe injury to personnel or major aircraft damage.

  - Failures such as loss of all hydraulics or pitch axis control, etc.

  - These failures are recorded as major accidents by the various command headquarters.

Operational Base Interviews -- The interviews at the operational bases were oriented toward obtaining data that:

- Would supplement the statistical data obtained from AFM 66-1 system;

- Would indicate which particular environmental conditions have a major effect on system operation and maintainability,

- Would give a subjective feel for the various types of BITE/ self-test hardware;

- Would give a subjective view of the ASE and Technical Order (T. O.) adequacy for a given system;

- Would allow suggestions for system maintenance improvements to be submitted;

- Would relate system deficiencies to aircraft incidents and accidents.

Many of the comments and the general concepts expressed during these interviews are included in the following subsections.

Reliability/Maintainability Data

This subsection summarizes the reliability/maintainability data as it pertains to maintenance and logistics. The effects of failures on mission success and flight safety are described in the next subsection.

System Data -- Comparative data for each of the major flight control systems are given in Figures 3-10 through 3-13. Details relating to individual component failures, high-failure-rate items and maintenance problems associated with the various components are discussed later.

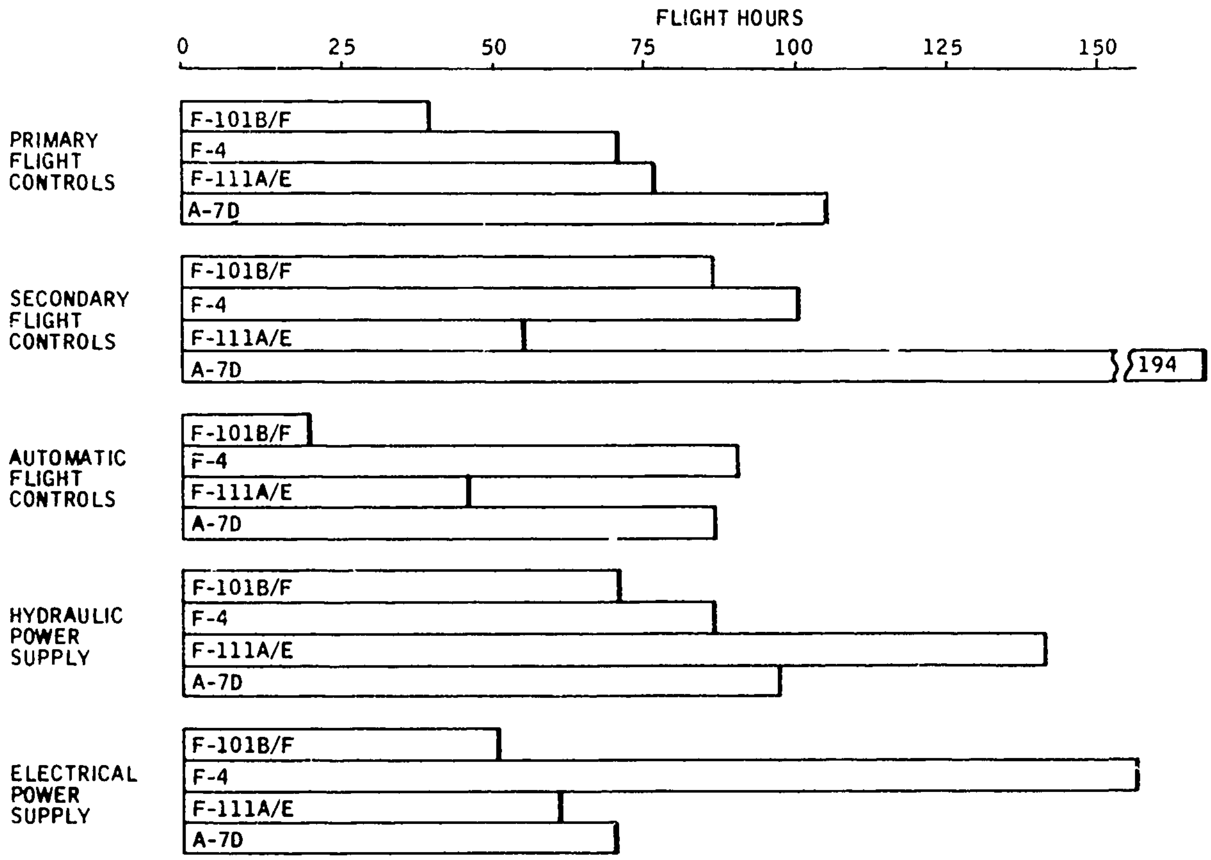

Figure 3-10. Mean Time Between Failure Summary

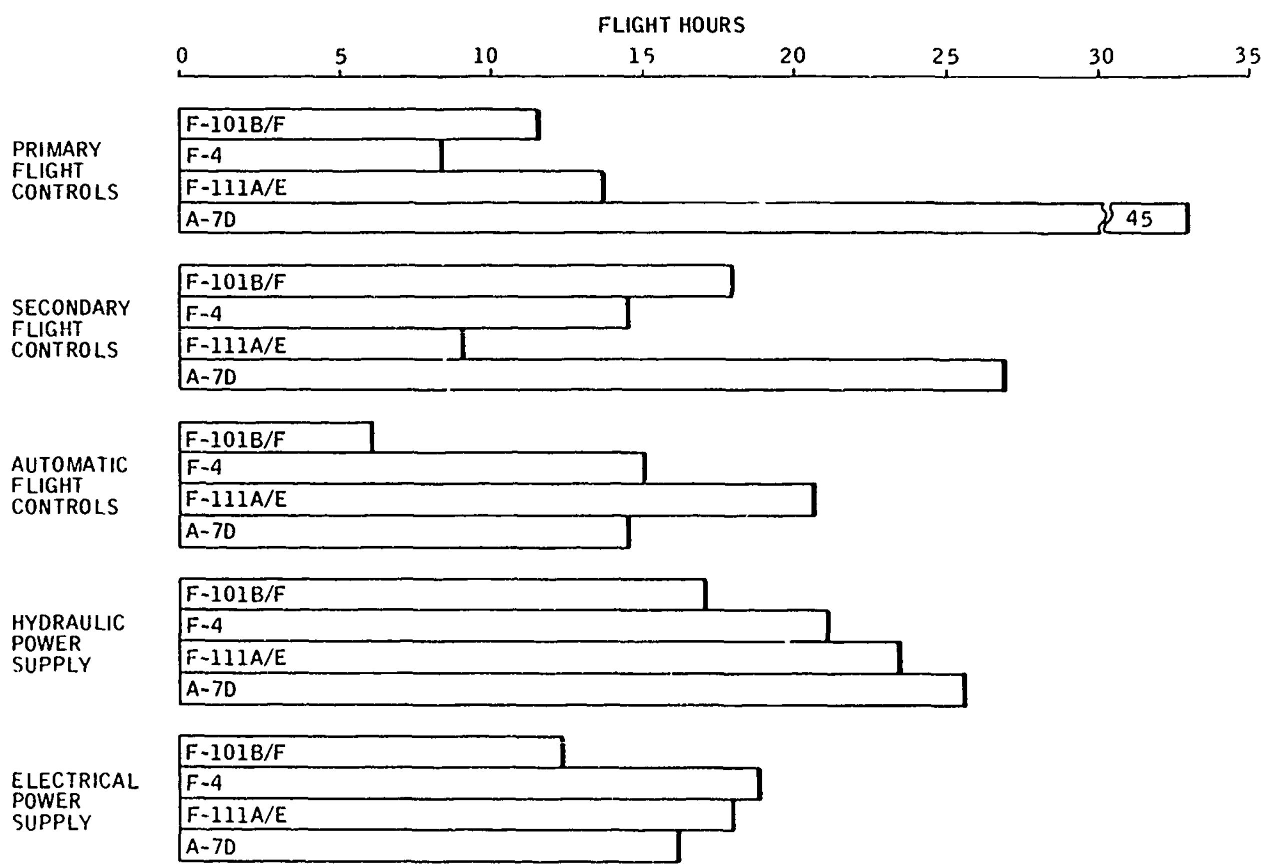

Figure 3-11. Mean Time Between Maintenance Action Summary

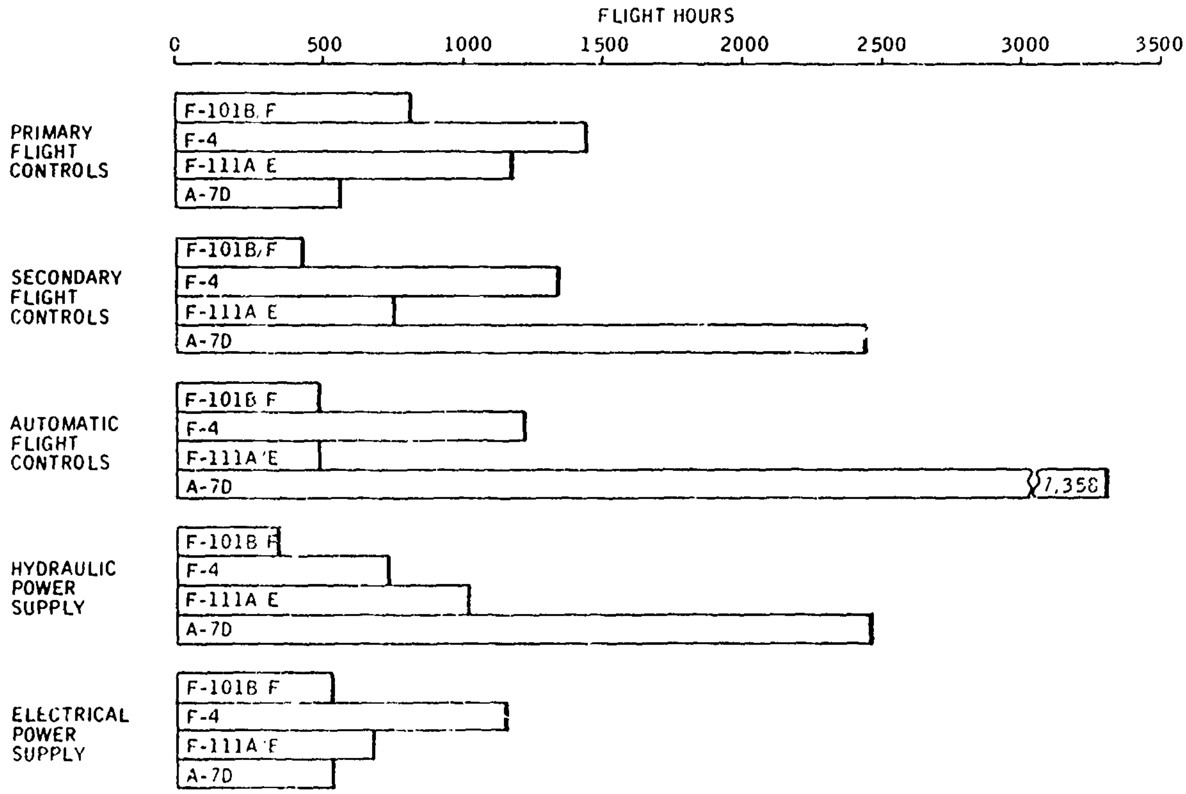

Figure 3-12. Mean Time Between Abort Summary

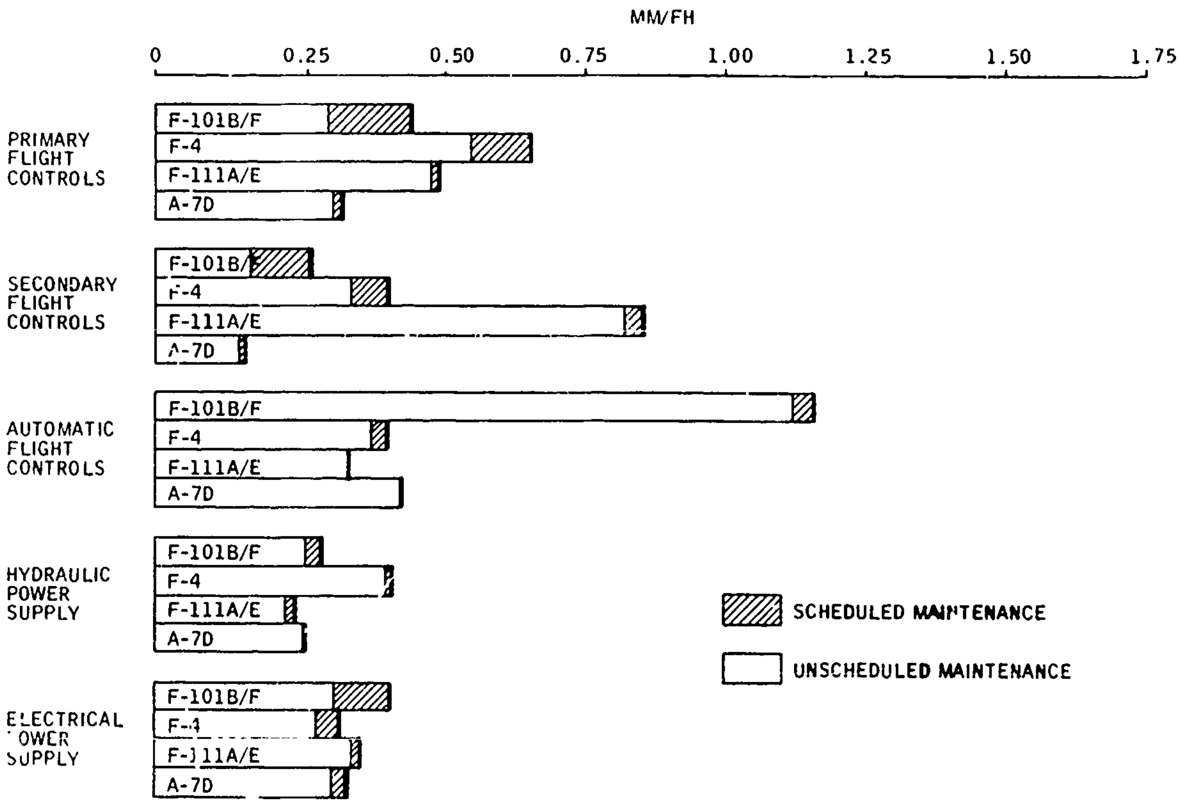

Figure 3-13. Maintenance Manhour per Flight Hour Summary

Some general observations on the data shown in Figures 3-10 through 3-13 are:

- The data bases are large enough to provide a h.gn confidence level in the values stated for the systems.
- Device wearout and fatigue are not as much of a factor in the F-111A and A-7D data as they are for the F-101B/F and F-4 data.
- The MTBF values in Figure 3-10 are computed by including all AFM 66-1 reported failures. The failure and safety hazard classifications are discussed in later subsections.

The effects of wearout and technological improvement on virtually identical systems in the F-4 series of aircraft are shown in Table 3-9.

System Complexity and Design Vintage -- Each of the control systems examined was designed for a specific aircraft; consequently the aircraft flight envelope and mission requirements are reflected in the flight control system mechanization. The design vintage also covers a period of approximately 10 to 12 years (1954 to 1966), and the technological advances which have occurred lead to considerable differences in actual hardware. The effects of the above conditions must be kept in mind when comparing reliability, safety and maintainability of the various flight control systems. The following paragraphs detail the major factors affecting control system complexity and then provide a complexity ranking for the aircraft associated with the survey.

The major factors which affect the various control systems complexity are:

- Primary Flight Control System

  (1) The number of control surfaces and associated linkage and actuators;

  (2) The mechanical control input assembly construction (i.e., control cables versus push-pull rods);

  (3) The number of trim surfaces, trim modes and the hardware associated with the trim system;

  (4) The amount of aerodynamic, mechanical and hydraulic components used to provide proper aircraft handling (pilot feel) characteristics. This factor is somewhat proportional to the size of the aircraft flight envelope.

Table 3-9. Effects of Service Life, Wearout and Technological Improvement on System Reliability/Maintainability

System	Aircraft Type and Production Vintage			
	F-4C 1959-1963	F-4C 1961-1969	F-4D 1963-1965	F-4E 1965-1969
Mean Time Between Failure (flight hours)				
Manual flight controls	28.6	37.6	40.0	63.2
Automatic flight controls	73.8	77.0	88.8	114.0
Electrical power system	55.0	76.0	79.0	157.0
Hydraulic power supply	108.0	147.0	150.0	218.0
Aircraft instruments	26.0	34.0	31.0	53.0
Total of above systems	8.73	11.26	11.39	18.39
Mean Time Between Maintenance Action (flight hours)				
Manual flight controls	3.5	6.8	4.0	8.0
Automatic flight controls	11.0	13.0	15.0	21.0
Electrical power system	13.3	19.9	19.3	34.9
Hydraulic power supply	10.8	22.1	17.0	30.1
Aircraft instruments	9.0	15.0	12.0	19.0
Total of above systems	1.52	2.59	1.95	3.48

- Secondary Flight Control Systems

  (1) The number of surfaces and associated actuators used (related to size of the flight envelope);

  (2) Changes in aircraft design environment; i. e., carrier or short-field landing capabilities, etc.;

  (3) The amount of system usage time; i. e., F-4 spoilers exhibit wearout earlier than the wing-fold system.

- Automatic Flight Control Systems

  (1) The stability of the basic airframe; i. e., whether stability augmentation is essential for aircraft control, mission success or aircrew comfort only; the degree of redundancy is directly related to this criteria;

  (2) The number and type of aircraft "hold" modes, i. e., attitude, altitude, Mach, heading, etc.;

  (3) The number and type of mission-essential modes, i. e., automatic intercept, terrain following, etc.;

  (4) The amount of gain scheduling required which is related to the system design approach. The F-111 uses an adaptive gain system to eliminate complex aerodynamic scheduling and its associated hardware.

Table 3-10 provides a feel for relative system complexity based on the amount of functions to be performed and the amount of hardware required. The design vintage of the aircraft is also included to aid in seeing the effects of technological progress.

Safety Data

Data Sources -- The primary source of aircraft safety data was the Inspector General Aerospace Safety Group (IGDSFR) at Norton AFB. Norton AFB serves as the major safety center for all USAF operations and consequently maintains a large data bank of accident/incident information.

Other data sources utilized were the safety reports issued semi-annually by the Tactical Air Command (TAC). These reports presented overall accident/incident rates for all TAC aircraft for individual years, 5-year periods and various aircraft types. The specific TAC reports used are listed in Reference 3-3.

Table 3-10. Relative System Complexity

Aircraft Type	Design Vintage	Complexity Rating[a]		
		Primary Flight Controls	Secondary Flight Controls	Automatic Flight Controls
F-101B/F	1954	4	3	2
F-4	1958	2	2	4
F-111A	1965	1	1	1
A-7D	1966	3	4	3

[a] 1 = Most complex

4 = Least complex

The A-7D aircraft has not had sufficient flight time to provide meaningful safety data; consequently the A-7D statistics have been omitted from all safety data and calculations.

Definitions -- The safety data is normally submitted to the data center via an accident or incident report. The distinction between major and minor accidents and incidents requires a subjective decision in many cases; however, the following general guidelines apply:

- Major Accident - An accident in which substantial property damage, aircraft loss, or major personnel injury is incurred.
- Minor Accident - An accident in which the damage or personnel injury is not as severe as in the major accident. Minor accidents are usually reported in an attempt to prevent these accidents in the future.
- Incident - Incidents are reported whenever an inflight safety hazard occurs even though no property or personnel injury may have occurred.

The failure safety hazard classifications related to the above reports are the Class III and Class IV failures. Class III (critical) failures are related to incident reports caused by material failure/malfunctions while Class IV (catastrophic) failures would normally be associated with material failures/malfunctions which cause major accidents.

Tactical Air Command Data -- The following data was taken from the TAC Safety Reports (Ref. 3-3). The TAC reports provided a 5-year base covering approximately 4,000,000 flight hours, a 1968 data base of 935,000 flight hours and a 1969 data base of 865,000 flight hours. The major accident rates[2] for all causes for the 5-year period for various types of aircraft are:

Fighter-type aircraft - 12% per 1000 hours

Cargo-type aircraft - 2% per 1000 hours

The above rates indicate that a major accident occurs every 8,300 flight hours for fighter aircraft and every 50,000 flight hours for cargo aircraft. Analysis of 1967, 1968 and 1969 data indicates that the primary accident cause factors were:

Pilot	- 40%
Material failure	- 30%
Maintenance	- 5%
Supervisory and miscellaneous	- 10%
Undetermined	- 15%

The 1969 TAC data gives further detail enabling some identification of flight control accident causes. Of the 45 major accidents of fighter-type aircraft which occurred in 1969, 10 were caused by loss of aircraft control and 2 of the 10 were malfunctions of the flight controls (422,000-flight-hour-base).

Utilizing the above data, the following values can be calculated for fighter type aircraft:

Accident Cause	Accident Rate (%/1000 hours)	Mean Time Between Accident (flight hours)
All causes	10.7	9,400
Loss of control	2.4	42,200
Malfunction of flight controls	0.48	211,000

USAF Safety Center Data -- The accident/incident data from Norton AFB includes the 1966 to 1970 time period with the following flight-control-related accident, incident and flight hour data:

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2 Rate $= \left(\frac{\text{Accidents} \times 10^5}{\text{Flight Hours}}\right)$

This provides percent per 1,000 hours or number of accidents per 100,000 flight hours.

Aircraft Type	Major Accidents	Reported Incidents	Flight Hours
F-101B/F	3	58	216,643
F-4C/D/E	25	476	2,621,089
F-111A/E	6	125	70,706

The individual reports of the above accidents/incidents were reviewed and the causes and/or most probable causes were divided into three categories; primary, secondary or automatic flight control system malfunctions. The accident/incident rates were computed for the individual systems using the standard failure rate formula:

$$\text{Rate (percent per 1,000 hours)} = \frac{\text{Accidents or incidents} \times 10^5}{\text{Flight Hours}}$$

Table 3-11 provides detailed accident/incident rates by aircraft and system.

Table 3-11. 1966-1970 Accident/Incident Rates

Aircraft	System	Accident Rate	Reported Incident Rate
F-111	Primary flight controls	2.8	12.7
	Secondary flight controls	2.84	149
	Automatic flight controls	1.42	7.1
F-101B/F	Primary flight controls	0.92	10.1
	Secondary flight controls	0	1.84
	Automatic flight controls	0.46	10.1
F-4C/D/E	Primary flight controls	0.38	8.1
	Secondary flight controls	0.08	3.12
	Automatic flight controls	0.12	6.4

Accident/Incident Data Trends -- Review and analysis of the safety data points out the following items:

- In selecting an allowable catastrophic failure rate for an advanced PFCS (which could be a FBW system), some improvement over the experienced rates for the current PFCS would be anticipated. The experienced catastrophic failure rate of 0.38 percent per 1,000 hours for the F-4 PFCS appears to be a reasonable rate for a mature primary flight control system.

- The ratio of flight control system catastrophic failures caused by component failures versus maintenance malpractices appears to be approximately 4 to 1.

- The F-111 accident rate substantiates intuition that higher than normal failure rates will be experienced during the initial years of aircraft operation. This also agrees with the initial portion of the bathtub-shaped failure rate curve used for failure predictions.

- The number of loss-of-control accidents (primarily due to pilot error) is high compared to the number of accidents due to control system failures. Methods of increasing flight control system capability to reduce the pilot workload and the number of loss-of-control accidents should continue to be investigated.

Failure/Maintenance Data Analysis and Survey Comments

The failure/maintenance data presented in the preceding paragraphs was discussed with the personnel at each of the various maintenance levels to obtain additional details and suggestions. Their comments and pertinent information are noted for each of the major aircraft subsystems categorized in the preceding data. The detailed statistical data on which many of the observations are based is given in Appendix II.

Manual Flight Control System -- The highest failure rate items on the manual flight control system are the various surface actuators. Consequently, an individual section has been devoted to actuator data. The remainder of the system has a relatively low failure rate; however, the MTBMA is still low because the various mechanical portions of the linkage, pivot points, etc., require periodic inspection and lubrication. The inspection and lubrication requirements associated with the control surface pivot points cannot be completely eliminated; however, the use of sealed bearings and technological advances in the field of lubrication and friction reduction should be explored and utilized in future designs.

Primary Flight Controls -- The major maintenance items in the PFCS, next to the actuators, are the feel and trim system components. The feel system components fall into four broad areas.

(1) Mechanical springs and lever arrangements - The major problem associated with these items are high breakout friction and bearing and seal wearout.

(2) Aerodynamic bellows - The major problems are bellows leakage, heater burn-out or plugging of the aerodynamic inlets. The loss of the aerodynamic feel has very severe handling effects and can easily result in the loss of aircraft and aircrew.

(3) Aerodynamic and hydraulic switching - This has the same problems associated with (1) and (2) above. In addition, a discrete difference in feel and/or surface authority is made at a given aerodynamic condition. Consequently, the pilot must condition his reflexes to the new handling qualities.

(4) Bobweight to provide stick force/g feel - This mechanization is very simple and requires virtually no maintenance; however, several incidents were caused by maintenance malpractices which resulted from personnel working in that area.

The surface trim mechanizations varied considerably between the survey aircraft; however, a relatively large amount of relay logic was used in each aircraft to support the trim function. The relay logic consequently contributed many of the failures associated with the trim system. It would appear that the use of solid-state electronic logic in place of the relay logic would eliminate many of those failures. The trim system actuators normally position the PFCS linkage rather than positioning the surfaces directly. The actuators do not exhibit wearout failures as rapidly as the primary surface actuators, probably because they are normally electromechanical actuators and are not exercised as continuously as the primary surface actuators.

The concept of providing trim through the AFCS actuator (A-7D) yaw axis) eliminated many of the trim system components, hence improving reliability and maintainability; however, malfunction of the yaw AFCS can prevent all yaw trim capabilities.

Several of the survey aircraft used push-pull control rod linkage construction rather than cables for the pitch and roll axes. The maintenance personnel feel that this change has simplified the rigging and maintenance of the primary flight controls. They also suggested that push-pull rod linkage construction be investigated for yaw-axis control on future aircraft.

Another point mentioned frequently was that much of the PFCS linkage is nonredundant; consequently one system failure can be catastrophic. The nonredundancy makes the aircraft vulnerable to battle damage. Reference 3-1 discusses the failure probabilities of the F-4 pitch axis PFCS and indicates that, while a dual-redundant, dispersed linkage system would be desirable from a safety standpoint, the additional weight, space, cost, and maintenance effort penalties exceed the value of the safety improvement.

Secondary Flight Controls -- The secondary flight controls are not directly related to the study goals; however they do cause a substantial portion of the flight control maintenance effort. The following maintenance comments were obtained.

- The SFCS is normally easy to troubleshoot and/or repair. The systems are relatively simple and are not integrated with other systems.

- The SFCS failures do not normally cause loss of aircraft control, but different aircraft handling techniques must be utilized by the pilot to maintain aircraft control.

- The SFCS actuators show a larger flight hour lifetime than the PFCS actuators which can be attributed to the much lower usage rate.

- The failures and maintenance effort expended on the SFCS are directly related to the SFCS complexity. The F-111 SFCS maintenance difficulties required the assembly of a maintenance group which was specifically trained to provide proper SFCS maintenance.

Automatic Flight Control Systems -- Initial discussions with the field personnel indicate that the automatic flight control system (AFCS) should be subdivided into three major categories for failure analysis. The AFCS was divided into the following areas which were common to each system:

- Rage Gyros and Accelerometers - Rate gyros and accelerometers were further examined individually as they are used to supply stability augmentation information for each of the survey aircraft.

- Electronics - Control panels, computers, surface or servo potition,and stickforce sensors and schedulers were included in this group. A considerable variation in the amount of hardware and functional complexity was apparent between the AFCS's of the survey aircraft.

- Actuators and/or Servo Actuator Valves - These devices were grouped with the PFCS actuators for analysis because many similar maintenance and failure analysis concepts apply.

The effects of the technological advances can be shown by a brief comparison of the F-101B/F and the F-111A AFCS reliability/maintainability data. The F-101B/F has a single-channel nonredundant AFCS with a large number of pilot-selectable outer-loop modes and complex gain scheduling. This system is mechanized using 1954 to 1956 vintage electronic and electromechanical hardware, thus utilizing vacuum-tube amplifiers, magnetic-core amplifiers, relay logic and motor velocity generators to perform the various computations. The F-111A has a triple-redundant command augmentation system (CAS), a larger number of pilot-selectable outer-loop modes and uses a self-adaptive gain-scheduling concept. This system is mechanized using 1964 to 1966 vintage solid-state electronics to perform the logic and functional computation and provide the various self-test capabilities.

The F-101B/F and F-111A AFCS reliability/maintainability values are compared in Table 3-12.

Table 3-12. Comparison of F-101B/F and F-111A AFCS Electronics Reliability

Aircraft	Flight Hours			
	MTBF	MTBMA	MM/FH	MTBA
F-101B/F	23.2	6.8	1.07	565
F-111A	46.1	20.7	0.33	488

The F-111A AFCS MTBF is approximately double that of the F-101B/F AFCS. This is a definite indication of the improved reliability of the newer hardware. The F-111A AFCS MTBMA improvement is related to several factors, one being fewer device "remove and re-install" maintenance actions because the mean time to repair requirements imposed on the entire F-111 improve device accessibility. In addition, there are very few calibrations and adjustments or electronic amplifier "drift" problems with the newer vintage electronics, eliminating additional maintenance actions.

The addition of built-in-test equipment in addition to some of the factors listed above accounts for the F-111 AFCS's lower maintenance manhour/flight hour value.

The reasons for the F-111 AFCS's lower mean time between abort are related to two particular items:

- The use of BITE during preflight by the flight crew detects failures which would not otherwise be detected until inflight or possibly not even then, depending which AFCS modes were engaged. The detected failures result in a higher ground abort ratio.

- Even though the F-111 AFCS is a triple-redundant fail-safe system, present USAF regulations require an abort upon loss of any one channel. If this requirement were relaxed, the number of aborts would certainly decrease.

The field maintenance personnel made the majority of their comments and suggestions on the following items:

- Ladders and Work Stands - The elimination of ladders and work stands to reach system components is one of the largest labor savers that can be designed into the aircraft. This allows more individuals to work on an aircraft at the same time and also eliminates the need for storage space for all of the work stands. The F-111 aircraft received many favorable comments in this area.

- Device Connectors and Mounting Hardware - The many variations in device connectors and mounting hardware requires different types of tools and also creates logistic problems. The MS3112, 3116 type connectors which rotate 30 degrees to 45 degrees and then snap into place are preferred over threaded connectors. The snaplock feature emits a sound and feel which gives assurance of full connector engagement. Threaded-type connectors are subject to over and under torquing, cross-threading and, in addition, must be safety wired in many cases. The safety wire occasionally turns out to be an electrical hazard.

- Ground Electrical, Hydraulic, Air Conditioning Power - The very need for the above units causes fault isolation and repair delays. In addition, it was pointed out that a variety of hydraulic, electrical and air conditioning connections are standard; consequently, individual maintenance units must manufacture interface adapters to allow usage on different types of aircraft. The possibility of using a single ground-power card with a single-point hookup to supply all the required ground power would eliminate many of the time delays which currently exist. It should also reduce the total number of ground-power units required and their associated maintenance.

- Aircraft Wiring - There is presently no USAF tester which can be used to check aircraft wiring at the flight line level. The PSM-6 multimeter would be perfectly adequate if proper-size test probes were supplied to interface with the aircraft connectors. The possibility of providing very simple electrical breakout boxes or cables to aid fault isolation might also be considered.

A second common comment was that the aircraft wiring T.O.'s seldom adequately describe the aircraft wiring. Splice locations and wire bundle routing appear to be the items most commonly neglected in T.O. preparation.

- External Cooling Air - The very need for external cooling air causes some system/component failures or overstresses because of the times when it is advertently not connected. In addition, in-flight AFCS system failures can occur should the air conditioning system fail.

- Environmentally Related Failures - High humidity and moisture related failures cause a large percentage of the maintenance workload. Connector corrosion, short-circuit and low impedance paths and component degradation due to moisture accumulation were evident in some areas of each AFCS in the survey. Humidity created enough problems on the F-111A AFCS that the computers are stored in the AFCS shop during some high-humidity environments. Temperature changes and the use of potted modules cause solder breaks due to a difference in the coefficient of expansion of the modules and the printed circuit boards on which they are mounted.

  The humidity/temperature problems are often difficult to isolate and repair unless the environmental conditions can be duplicated. The application of gaskets and hermetic seals on devices to eliminate moisture problems were strongly advocated by the field maintenance personnel interviewed.

Rate Gyros -- The major portion of gyro failures are due to the bearings within the gyro which simply wear out from usage or suffer damage from shock, vibration and temperature effects. It should be noted that wearout is not yet a factor in the F-111 and A-7D gyro failure rates, hence those rates reflect more of a random gyro failure rate. A summary of the rate gyro data is given in Table 3-13.

The environmental effects on gyro reliability are very pronounced as evidenced by the difference in MTBFs of the F-4 roll, pitch and yaw gyros. The roll gyro is mounted in the cockpit; consequently is operates in a relatively mild environment. The pitch and yaw gyros are mounted in the right and left main gear wheel wells and consequently are subject to greater shock and vibration, wider temperature ranges and moisture. The effects of these environmental changes on MTBF are:

Table 3-13. Rate Gyro Summary

Aircraft	Number in System	Total Gyro Group MTBF (hrs)	Average Individual Gyro MTBF (hrs)	Percent of AFCS Failures	Percent of AFCS Manhours
F-101B/F	4	431	1726	4.9	3.1
F-4	3	369	1110	24.3	9.3
A-7D	6	1226	7456	7.1	1.5
F-111A	9	1768	15912	2.6	2.4

F-4 roll rate gyro - MTBF = 1724 flight hours

F-4 pitch rate gyro - MTBF = 1238 flight hours

F-4 yaw rate gyro - MTBF = 755 flight hours

The reason for the difference between the pitch and yaw MTBFs cannot be totally explained without a further analysis; however the environmental effects on the gyro MTBF do become obvious.

Use of the recently developed vibrating rate sensor with an estimated MTBF of 40,000 hours and no wearout factor will virtually eliminate the gyro wearout failures experienced today. This type of gyro could therefore increase the system MTBF by 5 to 25 percent.

The use of self-torquing gyros has substantially aided in reducing the amount of time required to troubleshoot the rate gyro failures; however the major percentage of the maintenance manhours expended on the rate gyro maintenance are on removal and remounting of the gyro package. The possibility of providing a simplified mounting system, rather than the three or four screws or bolts presently used, could reduce the overall maintenance manhour effort.

<u>Accelerometers</u> -- Accelerometers are a second sensor common to each of the survey aircraft stabilization systems. It was interesting to note that there were many adverse field comments concerning rate gyro reliability, while very few concerning accelerometer reliability were made. The data obtained, however, indicates that the accelerometer MTBF is nearly equal to that of the rate gyros -- hence, the complaints must be based on the other factors. The fact that there are more gyros and that they may be difficult to install probably accounts for the additional comments. Accelerometer data is summarized in Table 3-14.

Table 3-14. Accelerometer Summary

Aircraft	Number in System	Total Accel. Group MTBF (hrs)	Average Individual Accel. MTBF (hrs)	Percent of AFCS Failures	Percent of AFCS Manhours
F-101B/F	3	1047	3131	1.9	2.04
F-4[a]	2	803	1608	11.1	2.90
A-7D[a]	4	668	2672	13	3.0
F-111A	6	3537	21,222	1.3	1.2

[a]The F-4 and A-7D accelerometers were experiencing abnormally heavy failure rates because of design and production problems which are being corrected.

It should be noted that the percent of effort expended on accelerometer maintenance is very small. Recent accelerometer developments using peizo-resistive and other solid-state elements which are acceleration sensitive to replace the mechanical sensing portion of present accelerometers should lead to an even greater MTBF as wearout will also be eliminated. The piezo-resistive transducer and accelerometer assemblies presently in production have a predicted MTBF of 40,000 hours.

Pilot Control Panels -- The pilot control panels were separated because they provide the human/mechanical/electrical interface between the pilot and the automatic flight control system. The control panels exist in a relatively mild environment; however they are subject to damage from personnel using the control panel console as a cockpit exit/entry step. The complexity of the control panel or panels and their associated reliability/maintainability values are shown in Table 3-15.

The major field comments obtained were:

- Do not include lamps on the control panels unless the panel can be located in a favorable viewing position.
- Most of the control panels in the survey were well designed from a human factors standpoint; however when the self-test functions are added to the control panel, caution must be exercised such that the pilot can perform the required switch movements with one hand in a relatively short time period. If the above requirements are not met, the pilot has a tendency to skip or delete portions of the self-test procedure.

Table 3-15. Control Panel Summary

Aircraft	Panel Complexity	Flight Hours			
		MTBF	MTBMA	MTBA	MM/FH
F-101B/F	Most complex[a]	253	114	4890	0.034
F-111A	↓	14,149	1572	14,149	0.0028
A-7D	↓	7358	2453	7358	0.0003
F-4	Least complex	8087	1146	54,989	0.0047

[a] Includes the redundant limiter system panel which contains three lamps as well as the switching functions

Computers, Force and Position Sensors, Signal Schedulers -- The computers, sensors and schedulers account for the majority of the AFCS failures, aborts and maintenance actions. The computers consist of the electronic "black boxes" which are used to perform the stability augmentation and autopilot computational functions and also the various logic computations which are required. The sensors included in this section consist of surface position transducers, stick force transducers, and aerodynamic transducers whose output is used to provide signal scheduling. The schedulers themselves consist of motor-generator-potentiometer or synchro follower devices which are driven by the aerodynamic transducers or by the aircraft air data computer (ADC) outputs. It should be noted that the ADC is normally considered part of the aircraft instrument system for failure reporting purposes; however failures can affect aircraft handling and impose a restricted flight envelope. The exact effects of ADC failures on the flight control system performance on the survey aircraft is beyond the scope of this study; however, it should be noted that the F-111 avoids this problem completely through use of the self-adaptive gain scheduling.

The following observations and field comments were obtained:

- Computers -

  (1) Solid-state logic and amplifiers are a great improvement over magnetic amplifiers and relay logic.

  (2) Elimination of electromechanical devices simplifies system maintenance and improves reliability (above observations are borne out by reliability data).

(3) Moisture in electronic assemblies causes a large percentage of the system malfunctions and some component failures. The field suggestions were:

   (a) Ensure that all black boxes have a moisture-proof seal when installed in the aircraft;

   (b) Ensure that all connectors are mated in a horizontal rather than vertical plane to prevent moisture from running into the connectors and causing corrosion;

   (c) The use of potted modules to eliminate moisture causes temperature problems and does not completely eliminate the moisture problems. The potted module and the printed circuit board and component leads all may have different coefficients of expansion; hense solder breaks and component failures can occur with temperature variations. These failures also are often difficult to troubleshoot because they are often intermittent depending upon temperature, etc.

(4) Isolate aircraft functions to particular black boxes or portions of the black box. This procedure was used by computer designers on each of the survey aircraft.

(5) Provide a greater number of easily accessible test points on the front panel of each computer. This aids positive fault isolation in contrast to the "guess, remove and replace" type of maintenance action. The test point requirements were not expressed by the F-111A maintenance personnel, indicating that the addition of a good self-test system reduces the need for a large number of test points. The F-111A personnel did indicate, however, that test points to verify proper operation of the various power supplies are desirable.

(6) The concept of throw-away modules was not accepted very well at the field shop level. The basic reasons given by the shop personnel were that:

   (a) The cost of throwing away a module "appeared" to be excessive compared to replacing an individual component. It should be noted that none of the individuals contacted actually had repair cost data available.

   (b) The effort in replacing a module consists of unsoldering six to ten leads while replacing a component only requires soldering and unsoldering two to eight leads.

   (c) Failure isolation to a given component requires only slightly more effort and time than isolating the failed module.

(d) A large number of different modules had to be stocked at the base level to prevent long repair delays. It was suggested that the number of different modules required be compared to the total number of different components required if a throw-away module concept is to be used. The normal electronic piece parts which fail can usually be ordered through the Federal Stock Inventory with delivery in several days while module delivery may take weeks.

(e) The throwaway module concept was not integrated among all the avionics systems; hence individual electronic components had to be procured for repair of many of the instruments, communications, and radar/bombing navigation computers which were being repaired in the same field shop.

- Sensors -

(1) The location of many of the surface position transducers is such that they are susceptible to hydraulic fluid contamination. The comments indicate that better sealing procedures to eliminate the above problem were needed.

(2) The angle-of-attack and/or sideslip sensors used to provide switching, gain scheduling and various instrument readouts are a major source of maintenance effort. The sensors are very susceptible to ground handling damage. They are also difficult to ground test because aerodynamic conditions cannot be simulated easily. The sensor friction affects its dynamic response and static accuracy, and constant exposure to the elements aggravates that effect over a period of time; however the friction failure may be difficult to detect.

(3) The control stick force transducers operated satisfactorily in all of the survey aircraft and no improvement comments were obtained at any of the survey locations.

- Signal Schedulers -

(1) The schedulers used on the F-101, F-4 and A-7D aircraft were all of the electromechanical variety. The major problem is their dependency on the air data computer for positional data and the wearout which the electromechanical devices inherently face. Signal scheduling in the F-101 is considerably more complex than that in either the F-4 or the A-7D. The schedulers consequently account for a high percentage of the total F-101 AFCS failures and also make aircraft handling qualities dependent on operation of the central air data computer.

(2) The F-111A uses self-adaptive gain scheduling. This mechanization was ideal from the AFCS maintenance standpoint as troubleshooting and fault isolation are considerably easier. The self-adaptive system also eliminates the AFCS dependence on the air data computer.

PFCS/AFCS Actuators -- Field interviews and comments indicated that the actuators consumed a considerable portion of the PFCS maintenance workload, hence separate data was obtained on the actuators. The PFCS/AFCS actuators included in this summary are the control surface or linkage actuators used by the PFCS and the AFCS only. Trim activators and secondary flight control system (SFCS) actuators have not been included in the above category. The individual actuators are all 3,000-psi hydraulic actuators controlled by valves which are mechanically moved by the control stick linkage (PFCS actuators), by electrical control signals from the AFCS computers (AFCS actuators) or by mechanical and electrical inputs (integrated actuators).

Detailed actuator failure modes are beyond the scope of this study; however, the majority of actuator failures fall into two categories:

- Seal and gasket wear allowing slow hydraulic leakage -- This failure has no effect on aircraft control or performance, but if allowed to continue, could lead to a catastrophic actuator failure.

- Failure of the control valves due to wear about their center position -- The control valves which are used to provide aircraft stability exhibit this tendency since they are continually cycling about their center position. This condition is aggravated by the fact that extremely close mechanical tolerances must be held on these valves to provide an adequate servo frequency response.

- The data gathered in this survey did not show any definite correlation between actuator complexity and the failure rate encountered. Reference 3-4 provides considerable additional information on the characteristics of various F-4 longitudinal actuators and also on the overall reliability-maintainability-safety of the F-4 longitudinal control system.

A brief summary of the actuator data is given in Table 3-16.

Table 3-16. Actuator Summary

Aircraft	Average Actuator Values (flight hours)					
	PFCS Actuators[b]		AFCS Actuators[c]		Integrated Actuators[d]	
	MTBF	MTBMA	MTBF	MTBMA	MTBF	MTBMA
F-101B/F	473	158	1630	863	343	118
F-4	4422	930	930	736	796	222
F-111A[a]	1074	251	2122	786	---	---
A-7D[a]	2452	350	2760	582	---	---

[a]Wearout will be a very small percentage of actuator failures

[b]PFCS actuators - Mechanically controlled, hydraulically powered actuators

[c]AFCS actuators - Electrically controlled, hydraulically powered actuators

[d]Integrated actuators - Mechanically or electrically controlled, hydraulically powered actuators

Note: The MTBFs listed above include failures such as leaking seals, etc. Only a small percentage of the failures actually cause loss of function.

The field interviews pointed out that:

- The aircraft hydraulic system design can have a significant effect on actuator life. The A-7D hydraulic system initially became so hot that the paint on the actuators actually darkened. The heat caused seals to begin leaking within 50 to 100 flight hours. Modifications have been made to the A-7D to add coolers to the hydraulic system and thus keep the hydraulic fluid temperature to an acceptable level.

- While many hydraulic technicians and many electrical technicians exist in the USAF, very few persons are trained technicians in the electro/hydraulic area for work on the integrated actuators. This could lead to actuator removals and failure where some other device may actually have failed.

Electrical Power System -- The electrical power system essentially consists of a-c generators, a power control panel and automatic switching gear, an a-c distribution system, an ac-to-dc power rectifier and a d-c distribution system. Only the generators, a-c control and distribution, and the d-c rectifier and distribution systems have been subdivided to give a relative feel for the reliability/maintainability of the electrical system. An example of electrical system failure modes and effects on F-4 flight controls is given in Reference 3-4.

Hydraulic Power Supply -- The hydraulic power supply data was included since nearly all surface actuators are hydraulically powered. It should be noted that the reliability/maintainability values given are of interest only from a maintenance standpoint since system loss of function and system failure, as given in the data, are not directly related. Reference 3-4 provides a brief description of functional failure rates and a failure mode and effects analysis for the F-4 longitudinal-axis hydraulic systems.

Aircraft Instrument System -- The aircraft instrument system data was included because sensors which provide some of the instrument display data also provide AFCS information for outer-loop modes. Examples of the above are Mach number, altitude, attitude and heading hold modes, terrain following modes, various bombing and automatic attack modes and automatic instrument landing modes.

The corresponding sensors are the air data computers, attitude and heading gyros or inertial measurement systems, radar altimeters, or airborne radar systems.

The data gathered indicates that the reliability/maintainability values of the aircraft instrument system are approximately of the same magnitude as those of the automatic flight control system.

Built-In Test Equipment/Self-Test Experience

Three of the four survey aircraft (F-101, F-111, A-7D) automatic flight control systems had some built-in test equipment. There is a considerable difference in BITE capability, yet maintenance personnel regarded BITE as one of their best maintenance tools on each system. The BITE in each of the aircraft is a ground maintenance tool which is manually initiated by the pilot or maintenance technician with stick movement, surface position indicators and warning lamps providing the test results. Table 3-5 provided a summary of the BITE operation in each of the survey aircraft.

The following field comments and suggestions were obtained:

- F-101B/F Aircraft - A recent modification program has installed a redundant limiter system (RLS) in the F-101B/F aircraft fleet. This system limits pilot command inputs to predetermined angle of attack or normal acceleration limits. Since the RLS is a safety-of-flight system, BITE was incorporated to test the system during preflight. The BITE tests approximately 85 percent of the RLS; and because portions of the AFCS are used by the RLS, approximately 20 percent of the AFCS is tested during the RLS test procedure.

  The BITE is used to aid in device-level fault isolation at the flight line and also by the pilots in their preflight system tests. Both the maintenance personnel and the pilots' primary suggestion was that the rate gyros and accelerometers should have a self-test torquer capability, thus increasing the system test capability. It should be noted that the A-7D and F-111 systems have this capability, but a sensor retrofit program would be required to add that capability on the F-101B at this time.

- F-111A Aircraft - The self-test system on the F-111A AFCS is virtually ideal from the maintenance technician's standpoint. This comment was based on several factors:

  (a) The self test procedure is manually controlled, hence the technician knows which test is in progress and its relation to the system squawk.

  (b) The manual procedure allows a test to be re-run if the results are marginal.

  (c) The meter readout provided is simple, and with use of a fault isolation chart, the failed LRU can be located within 15 to 20 minutes with a 90 to 95 percent confidence level.

  (d) The self-test of the self-test circuitry gives assurance that the self-test is operating properly.

(e) The self-test hardware consists of several manual switches, solid-state logic circuitry and a meter readout. None of these items are difficult to troubleshoot or repair in case of a self-test hardware failure.

(f) The rate gyros, accelerometers and air data computer have self test features, thereby testing the devices which formerly were most difficult to troubleshoot.

- A-7D Aircraft -- The A-7D self-test hardware provides a quick, simple test of functional operation of the various AFCS components. The self-test does not isolate failures to a line-replaceable unit; consequently some additional test gear must be used to troubleshoot and fault-isolate failures.

  The self-test system does simplify operational testing and fault isolation, the largest benefits being the capability to torque the rate gyros and accelerometers and indicate failures in these components if the proper outputs are not obtained.

  The maintenance personnel indicated that the suitcase tester designed for A-7D flightline use is seldom utilized because a PSM-6 multimeter in conjunction with the self-test capability can normally be used to isolate the failure and replace the failed device.

Support Equipment Experience

The comments made by maintenance personnel can be divided into two categories:

- Equipment problems common to all aircraft;
- Problems peculiar to test equipment for a given aircraft.

The common equipment problems are briefly summarized below:

- The need for individual ground hydraulic, electrical and air conditioning power units and the associated time delays required to obtain that hardware.

- The need for work stands and ladders to achieve access to various devices (F-111 was designed to eliminate this problem, and replacement of only one or two components requires work stands)

- Manually selectable or manually initiated test sequences are preferred to completely automatic testing sequences. The manually selectable test procedures allow fault isolation and repair actions to be completed while changing environmental conditions in case where the failures occur only under a peculiar environment.

- Flight-line and field shop personnel should be located in the same facility. This improves communication for field device repair, aids in assuring correlation of flight line/field shop failure isolation and eliminates or reduces device transportation time lags and the associated possibilities of damage in transit.

  With the above in mind, a complete aircraft mockup tester should be available in the shop to simulate actual device usage in the system. System testing also aids in obtaining assurance of proper fault isolation and repair action procedures.

- The use of semi-automatic/automated testing in the field or intermediate shop has greatly reduced the maintenance man-hour effort and has also removed much of the drudgery associated with manually testing hardware. Better T.O. descriptions of the automated test objectives, stimuli applied and results desired would aid fault isolation procedures.

- No testers of any type are provided to test or fault isolate the aircraft wiring. The PSM-6 multimeter is adequate for continuity checks; however, no foolproof method of preventing connector damage exists. Female connectors are very susceptible to damage if improper size probes are used while checking continuity. The possibility of providing a set of connector pins or breakout boxes to the maintenance organizations should be considered.

- Virtually all of the maintenance personnel prefer several limited-capability, small, one-man-portable testers to a single, large, trailer-type, multiple-capability tester for flight-line usage.

- The majority of the flight-line personnel prefer checklist type pocket size T.O.'s rather than the detailed full size 8-1/2 x 11-inch T.O.s.

- One of the major fault detection and isolation aids is a detailed problem description (flight squawk) by the pilot. The maintenance personnel indicated that the validity and accuracy of pilot flight squawks were almost directly related to pilot experience.

- The electrical testpoints provided on individual devices aid the overall system fault isolation procedures. The maintenance personnel expressed a desire to have more testpoints made available on the systems on each of the survey aircraft.

The following comments were obtained on the individual aircraft testers:

- F-101B/F AFCS Testers - The four-wheel trailer, tape programmed, UG897 Automatic Test Set is too large and hookup time is too great to use the tester for flight-line maintenance. The tester is used extensively in the field shop and provides a considerable reduction in overall system test time.

  The UG677 Flyaway Test Bench which simulates the complete AFCS is regarded as a valuable fault isolation-repair assurance tool.

  The major complaint on the F-101B/F tester hookups are that the connectors are the threaded type rather than snap-lock; consequently much maintenance time is expended connecting and disconnecting testers to the aircraft.

- F-4 AFCS Testers - The suitcase tester used for flight-line maintenance performs its functions adequately. The tester however does not have an internal voltmeter; consequently a separate voltmeter must be used in conjunction with the tester. The tester also has an internal high-temperature problem which exhibits itself very rapidly in a high-temperature environment, causing tester component failures. The above problem has been brought to the attention of the pertinent AMA.

- A-7D AFCS Testers - The suitcase tester provided was designed to be used for both flight line and field shop maintenance. The maintenance personnel felt that the tester had several unnecessary functions for flight line testing and did not have enough functions for proper fault isolation and repair assurance in the field shop. The tester obviously was compromised to perform both duties; consequently neither of the maintenance personnel are satisfied.

- F-111A AFCS Testers - The F-111A personnel did not use any testers on the flight line; however, they had built a testpoint breakout cable to aid in troubleshooting the feel and trim assembly computer. The only other equipment normally used on the flight line was the PSM-6 multimeter. The maintenance personnel were well satisfied with the above flight-line maintenance concept.

The F-111A field shop provides a substantial avionics repair capability. The shop uses a USAF version of automated test equipment which was originally designed for the SKYBOLT missile system. The various electronic computers are tested on individual test station consoles which are tied into a Burroughs time-share computer system. The device T.O. s normally consist of the computer programs and a brief description of the individual tests. Considerable training and experience are required to obtain a competent technician.

A large number of test stations and auxiliary test gear is required, and it is felt that major support and logistics problems could result if the maintenance shop had to be moved. The shop maintenance personnel comments obtained were:

(a) The automated testing greatly decreases the total testing time expenditures and eliminates much of the dog-work.

(b) Troubleshooting of environmental or intermittent problems is difficult. More manual test capability and a greater selection of test points would aid in troubleshooting and fault isolation of problems.

(c) Devices are never tested in a system environment. Consequently there are occasions where the devices pass the device-level tests but fail system tests and vice versa. These discrepancies are being corrected by T.O. and tester revisions, but this has been a very time-consuming procedure.

Conclusions and Recommendations

Failure-Safety Relationships -- The design and analysis of any high-authority closed-loop flight control system will at some point require a failure mode and safety effects analysis. The document which defines failure hazard classifications and a method of accomplishing a system safety analysis is MIL-STD-882, "System Safety Program for Systems and Associated Subsystems and Equipment: Requirements For". MIL-STD-882 provides the following four failure safety classifications (previously defined in more detail). The failure classes are defined by "conditions such that personnel error, environment, design characteristics, procedural deficiencies, or subsystem or component failure or malfunction:

- Category I - Negligible ... will not result in personnel injury or system damage;
- Category II - Marginal ... can be counteracted or controlled without injury to personnel or major system damage;

- Category III - Critical ... will cause personnel injury or major system damage or will require immediate corrective action for personnel or system survival;
- Category IV - Catastrophic ... will cause death or severe injury to personnel or system loss."

The failure data gathered on the existing flight control systems has been placed into the above categories to provide a field-experienced failure rate baseline which can be used to aid in defining a realistic design and failure rate for a high-authority closed-loop flight control which might replace the existing aircraft primary flight control system. The secondary and automatic flight control system data are included for comparative purposes.

A summary of the failure rates and the corresponding mean time between failure for the survey aircraft flight control systems is given in Table 3-17.

The ground rules, data bases and assumptions used in developing the failure rates for each failure classification are given in the following paragraphs. These figures represent averages weighted by flight hours.

Catastrophic Failures (Class IV) -- The aircraft major accident data provides the data for computation of the catastrophic failure rate. The F-4 aircraft data base of approximately 3,000,000 flight hours gives a catastrophic failure rate (percent per 1000 hours) of 0.38 for the PFCS, 0.08 for the SFCS and 0.12 for the AFCS. The above rates appear to be reasonable catastrophic failure rates for a mature flight control system. The F-111 data indicates rates of five to ten times the mature rate can be anticipated during the first several years of system usage. The F-111 SFCS is also much more complex than that on any of the other survey aircraft; consequently a higher catastrophic failure rate might be anticipated, and has occurred to date. Analysis of the accident reports indicated that approximately 90 percent of the accidents were caused by materiel failures while the remaining 10 percent were caused by maintenance errors. The above factor and a factor relating flight hours to operating hours must be taken into account in defining a high-authority closed-loop PFCS catastrophic materiel failure rate.

Critical Failures (Class III) -- The flight control incident report data provides an estimate of the number of Class III failures which have been experienced. The data obtained from Norton AFB provides the following incident rates:

Aircraft Type	Incident Rate (%/1000 hrs) PFCS	AFCS	SFCS
F-111	12.7	7.1	149
F-101	10.1	10.1	1.8
F-4	8.1	6.4	3.1

Table 3-17. Safety Classification Failure Rates and MTBFs

Failure Hazard Classification	Flight Control Systems - Survey Aircraft Failure Data Summary					
	Primary Flight Control Systems		Secondary Flight Control Systems		Automatic Flight Control Systems	
	MTBF[a]	λ[b]	MTBF	λ	MTBF	λ
Class I - Negligible (requires maintenance)	60	1675	74	1350	50	2000
Class II - Marginal (in-flight aborts)	2860	35	1675	60	1330	75
Class III - Critical (flight incidents)	10,000	10	20,000	5	12,500	8
Class IV - Catastrophic (major accidents)	254,000	0.38	1,000,000	0.10	835,000	0.12

[a] MTBF = Mean Time Between Failure = $\frac{\text{Flight Hours}}{\text{No. of Failures}}$

[b] λ(%/1000 hrs) = Failure Rate (% per 1000 hours) = $\frac{\text{Failures} \times 10^5}{\text{Flight Hours}}$

Based on the above figures, a PFCS rate of 10 percent, an AFCS rate of 8 percent and an SFCS rate of 5 percent are estimated values which can be expected from a mature flight control system. The F-111 SFCS rate of 149%/1000 hrs is due to a very high number of problems during the initial stages of aircraft deployment. Maintenance and hardware changes have reduced the rate considerably in the last year.

Marginal Failures (Class II) -- The data which best defines the Class II failure rate is the system abort data. The abort data obtained in the survey included both ground and inflight aborts and both are included in the mean time between abort values presented in Appendix II. The Class II failure rate is obtained by modifying the abort rate for two factors:

- Failures which occur on the return leg of a mission which normally would require an abort are not recorded as aborts because the aircraft is already homeward bound. Since the return leg of the mission is normally about one-third of the total mission time, the inflight abort rate must be multiplied by a factor of 1.5 to obtain the Class II inflight failure rate.

- The aborts which occur during ground operation were not included in the Class II flight hour failure rate, as the failures causing ground aborts occurred during system ground operating time, and those operating hours are not included in the flight hour failure rate calculation.

Average Class II experienced failure rates were calculated using the above ground rules. The weighted average Class II failure rates and MTBFs for the survey aircraft are:

PFCS - 35%/1000 flight hours (MTBF = 2860 flight hours)

SFCS - 60%/1000 flight hours (MTBF = 1675 flight hours)

AFCS - 75%/1000 flight hours (MTBF = 1330 flight hours)

The abort data which is related to weapon system effectiveness is the sum of the ground and in-flight abort rate. The ground abort rate is an important factor in determining operational readiness, and the in-flight abort rate can be related to the probability of having the vehicle complete the mission assignment. A summary of the F-111, F-101, and F-4 flight control system abort rates is given in Table 3-18. The average survey aircraft flight control system abort rates and mean time between aborts in terms of flight hours are:

PFCS - 72%/1000 flight hours (MTBA = 1390 flight hours)

SFCS - 82%/1000 flight hours (MTBA = 1220 flight hours)

AFCS - 95%/1000 flight hours (MTBA = 1050 flight hours)

Total - 249%/1000 flight hours (MTBA = 400 flight hours)

Table 3-18. Flight Control System Abort Rates

Aircraft and System		λ_A - Abort Rates (%/1000 Flight Hours)			
		Ground λ_A	Inflight λ_A	Total λ_A	Total MTBA
F-111	PFCS	64	21	85	1179
	SFCS	84	49	133	749
	AFCS	142	64	206	488
F-101	PFCS	82	41	123	815
	SFCS	136	95	231	431
	AFCS	136	75	211	473
F-4	PFCS	41	28	69	1447
	SFCS	37	37	74	1341
	AFCS	33	49	82	1206

$$\lambda_A \text{ (\%/1000 flight hours)} = \text{Abort Rate} = \frac{\text{Aborts} \times 10^5}{\text{Flight Hours}}$$

$$\text{MTBA (flight hours)} = \text{Mean Time Between Abort} = \frac{\text{Flight Hours}}{\text{Aborts}}$$

The F-4 flight control abort data indicates that approximately one of every 200 missions is aborted because of a flight control system failure, while one of every 19 missions is aborted because of all vehicle system failures.

Negligible Failures (Class I) -- The Class I failures are those which require some maintenance effort but do not affect flight safety or mission accomplishment. The class I failures are then all failures reported through the AFM 66-1 system except Class II, III and IV failures. The Class I flight control system failure rates and MTBFs were summarized in Table 3-16.

Two areas which should be taken into account in the reliability/failure analysis are:

- Incidents reported as failures through the AFM 66-1 system but which cannot be confirmed as true failures;
- Failures due to maintenance malpractices, transit damage, improper power application, etc.

Analysis of the 66-1 data indicates that approximately 5 percent of all PFCS and SFCS failures and 7 percent of the AFCS failures fall into the two areas described above. The reliability analysis and predictions must in some manner account for the above to provide correlation between predicted and AFM 66-1 reported failure data. The Class I failure rate variations and the best estimate of the average field-experienced values are:

System	λ - Range	Average	
		λ^a	$MTBF^b$
PFCS	1300 to 2520	1675	60
SFCS	990 to 1850	1350	74
AFCS	1110 to 4980	2000	50

[a] %/1000 flight hours

[b] Flight hours

Aircraft Wiring Failures -- The aircraft wiring failures affect electronic system performance and safety; consequently, an effort was made to obtain statistical wiring failure data. The AFM 66-1 system did not provide wiring failure data; hence no overall failure rate was determined.

The safety data from Norton AFB contained 19 incident reports which listed wiring failures as the most probable incident cause. No major accidents were attributed to aircraft wiring failures. Utilizing the ground rules established earlier, a Class III (marginal) aircraft wiring failure rate of 0.65 percent per 1000 hours is obtained. This is approximately a factor of 10 smaller than the Class III failure rates for the flight control systems and could be used as a baseline for a fly-by-wire system safety analysis.

It should be noted that other AFFDL studies are being conducted to provide wiring techniques and reliability values specifically for fly-by-wire system mechanizations. The results of those studies should be utilized in any final fly-by-wire system reliability-safety analysis.

Failure Detection, Cause, Prediction Data -- The failure data collected was analyzed to determine when failures occurred in the aircraft duty cycle. The "when discovered" failure code was used for the following data:

Failure When Discovered	F-4 PFCS	F-4 AFCS	F-111 PFCS	F-111 AFCS
In flight	8%	58%	15%	49%
Between flights	33%	26%	34%	48%
Scheduled inspections	59%	16%	51%	3%

Some of the conclusions which can be drawn from the data are:

- The ground maintenance procedures (visual inspections between flights and the visual inspections and data conducted during periodic scheduled inspections) detect 92 percent of the F-4 and 85 percent of the F-111 PFCS failures. The high percentage of failures detected during the ground checks is related to the mechanical nature of the existing PFCS. The point which this makes is that visual inspection of the surfaces, surface actuators and linkage of any new high-authority closed-loop flight control system will be a necessary part of the overall system maintenance plan.

- The use of built-in test equipment and its failure detection capability is illustrated by the fact that only 26 percent of the F-4 AFCS (no BITE) failures are detected between flights while 48 percent of the F-111 AFCS failures are detected between flights, with USAF technicians estimating that 95 percent of the ground-F-111 detected failures are detected by BITE. The difference in the number of failures detected during scheduled inspections is largely due to the fact that, because the F-111 AFCS has BITE, the only scheduled inspections are visual inspections of the AFCS actuators, whereas the F-4 AFCS is subjected to a complete functional test during the scheduled inspections. The use of BITE drastically reduces the periodic inspection effort.

- The failure causes are not reported in sufficient detail to allow a thorough categorization of failures by <u>cause</u>. A study to determine the effects of the following items on the system/component failure rates may assist in providing additional accuracy in reliability prediction techniques and also aid in developing more realistic life and reliability demonstration test programs. Item causes which deserve further analysis are the effects on reliability of:

  (a) Application and removal of hydraulic and electrical power and the transient effects

  (b) Environmental effects -- ground, airborne, geographic location

  (c) Maintenance malpractices

  (d) Design deficiencies

  The investigation of design deficiency reports against the system and components may lead to specifications which could prevent those deficiencies in future systems.

<u>Maintainability</u> -- The maintainability information obtained consisted of statistical data such as the mean time between maintenance action and maintenance manhours per flight hour and of field comments and suggestions regarding maintainability. The trends and conclusions from the factual data are:

- <u>Technological Improvements</u> - The F-111 and the F-101 flight control system maintainability values are:

System	MTBMA (flt hrs)	MM/FH
F-111 - PFCS	13.8	0.48
F-101 - PFCS	11.6	0.44
F-111 - AFCS	20.7	0.33
F-101 - AFCS	6.1	1.16

The PFCS data indicates that very little maintenance improvement has occurred between the F-111 and the F-101 at the system level because no major mechanical-hydraulic technological advances which affect maintainability have occurred. The AFCS maintainability values show a distinct improvement which is largely related to the advances in the electronic technology, particularly use of solid-state electronic components. The use of built-in test equipment in the F-111 AFCS is another factor which aids in reducing the maintenance manhour expenditures.

- System Integrity - The MTBMA values indicate that some type of maintenance action is performed on the PFCS and on the AFCS between every fourth or fifth mission. The number of maintenance actions influences system ground operating time and the associated failures. The number of maintenance errors and malpractices is somewhat related to the total number of maintenance actions. Statistical data on the relationship was not obtained.

A summary of the USAF maintenance personnel comments and suggestions pertaining to aircraft and flight control system maintainability follows:

- A single ground-power unit to supply electrical, hydraulic and air conditioning power to the aircraft should be developed. The development of a standard single-point connection to interface all ground power with the aircraft should also be investigates.

- Equipment accessibility on the F-111 is much better than on preceding fighter-type aircraft. The minimization of the necessity for work stands and the ease of achieving access to electrical connectors were regarded as major improvements.

- The manually controlled F-111 AFCS BITE is preferred to a fully automated testing scheme for flightline maintenance. The capability to select individual tests and the fear of not being able to detect BITE hardware failures seem to be the major deterrent to a completely automated test system. This viewpoint may well change as new and more comprehensive BITE systems are designed and built.

- Eliminate all scheduled maintenance on electronic systems except periodic visual inspections for cleanliness, moisture, and corrosion. This maintenance concept along with a postflight and preflight self test has been adopted on the F-111.

- Specify completely sealed bearings for utilization in the flight control linkages, thereby eliminating lubrication tasks. The use of sealed bearings will also reduce moisture accumulations in bearings which might freeze at high altitude and result in the loss of vehicle control.

- Design the control surface actuator linkage such that re-rigging of control linkages in the aircraft is not required. This concept was used in several of the survey aircraft. Consider failure effects in linkages, e.g., potention of hardover surfaces.

- Perform periodic inspections on an engine-hour or flight-hour basis. This eliminates the use of a variety of counters and elapsed time indicators to schedule maintenance and thus eliminates a considerable amount of paperwork and record keeping.

General Information --

Angle-of-Attack Systems -- The AFCS maintenance shops normally provided maintenance of these systems apparently consumes a disproportionate amount of maintenance effort. The field comments often included the problems associated with the various angle-of-attack warning systems used. This aspect of the angle-of-attack systems was discussed in a previous subsection.

The particular maintenance problems experienced are:

- The angle-of-attack sensors are normally rather delicate devices having movable parts on the outer surface of the aircraft. A considerable number of sensors are damaged when work stands, ladders, etc., are used around the aircraft.

- The sensor ground tests are normally only gross checks of proper operation; consequently in-flight tests and system adjustments must be performed periodically. The angle-of-attack warning system on the F-111 is the only electronic item related to flight controls which requires periodic ground maintenance tests.

- The F-101B has sturdy immobile angle-of-attack probes which are not as subject to ground handling damage. These probes however require an aerodynamic pressure test station hookup to verify proper angle-of-attack transducer operation.

AFM 66-1 Data Base Utilization -- The AFM 66-1 maintenance data reporting system provides a large data base which should certainly be consulted for failure/maintenance experience data when revising or developing hardware design specifications. The various maintenance engineers and technicians who monitor aircraft system performance at the various Air Material Areas often have an extensive background in their maintenance specialties; hence their inputs should be solicited and utilized during the hardware specification and procurement phases.

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SECTION IV

REVIEW OF SYSTEM GAIN-CHANGING REQUIREMENTS AND TECHNIQUES

This portion of the study analyzes the origins of gain-changing requirements for flight control systems and the techniques developed to satisfy these requirements. The result is a set of design criteria which provide guidelines for the application of available concepts.

GAIN-CHANGING REQUIREMENTS

Three major influences determine the gain adjustment needs of flight control systems:

- The nature of the controller
- The aircraft properties and flight envelope
- The performance specifications

Each of these is discussed individually to illustrate the basic requirements to be considered in augmentation system design.

Controller Properties

A general block diagram which encompasses all FCS is shown in Figure 4-1.

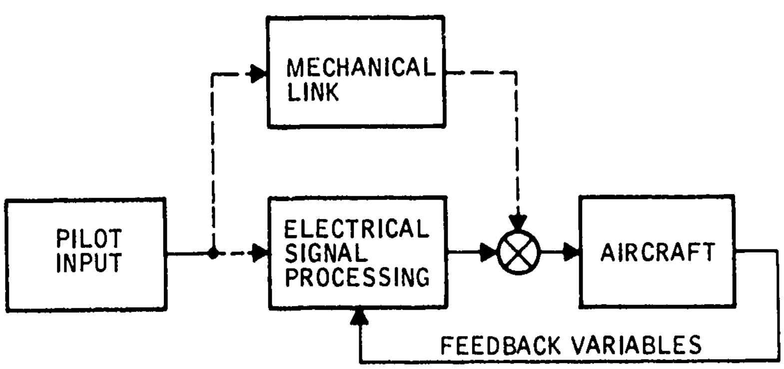


Figure 4-1. General Flight Control Diagram

Signal paths not present in all systems are shown by dashed lines; e.g., a fly-by-wire system has no mechanical link, and the electrical portion of the system might not have an input from the pilot. Use of the latter has evolved along with the trend for higher feedback control gains and higher feedback authorities; its use is most obviously required to avoid excessive counteraction of the mechanical input by the feedback signals. Such systems have become known as "command augmentation systems" (CAS), and their capacity for control gains which tend to minimize the effect of the mechanical link have also given rise to adjectives such as "high gain" and "high bandwidth". The term "high" generally alludes to loop bandwidths that are larger than, and thus insensitive to, the dominant aircraft frequencies. Of perhaps more significance to a study of gain changing are the design objectives applicable to various controller types, and here significant differences occur which directly impact the gain changing requirements.

Two alternate design objectives may be defined for an FCS:

- Provision of minimal levels of corrective feedback signals which, in conjunction with known airplane characteristics, achieve a specified standard of performance.

- Provision of dominating levels of feedback signals which, in conjunction with preset dynamic properties of the controller, achieve a specified standard of performance in spite of variable or unknown airplane characteristics.

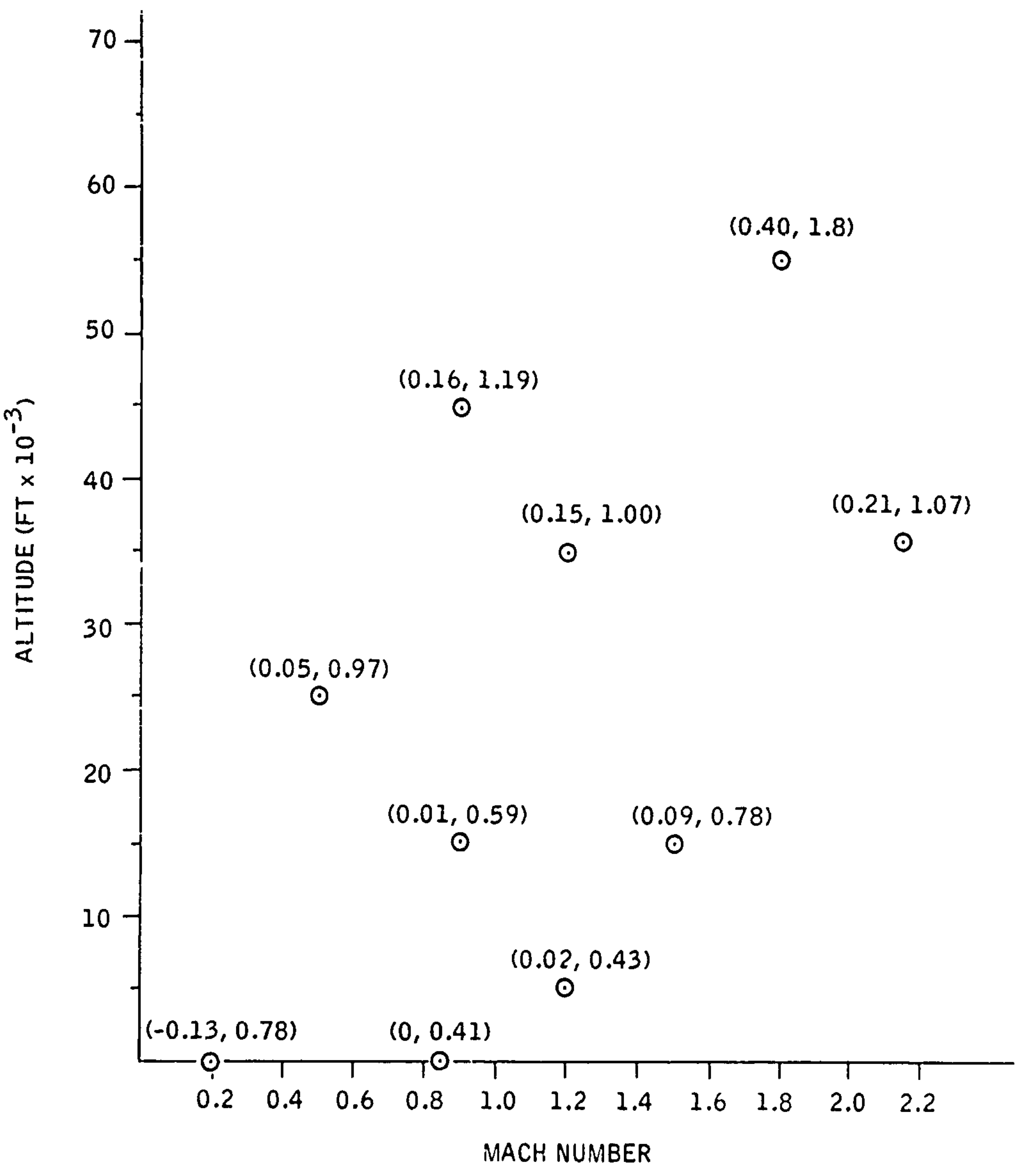
These two objectives are exemplified by the original "stability augmentation system (SAS)" and the current "high-bandwidth model reference systems", respectively, the latter being also a CAS.

Considering a simple SAS to illustrate the potential gain-changing needs, a pitch damper with a pitch rate feedback gain K_q has the characteristic equation

$$s^2 - \left[\frac{Z_\alpha}{U_o} + M_q + M_{\dot\alpha}\right] s - \left[M_\alpha - M_q \frac{Z_\alpha}{U_o}\right] - K_q \left[M_{\delta_e}\right]\left[1 + \frac{Z_{\delta_e} M_{\dot\alpha}}{M_{\delta_e} U_o}\right] s$$

$$+ K_q M_{\delta_e} \left[\frac{Z_\alpha}{U_o} - \frac{Z_{\delta_e} M_\alpha}{M_{\delta_e} U_o}\right] = 0$$

If values of K_q are computed to provide a short-period damping within the MIL-F-8785B (ASG) range of 0.35 to 1.30 (level 1, categories A and C), the situation for the F-4 flight envelope (data per Ref. 4-1) is as illustrated in Figure 4-2. This figure shows that a single gain value exists that



NOTE:
NUMBERS AT EACH CONDITION ARE MINIMUM AND MAXIMUM VALUES OF RATE GAINS, DEG/DEG/SEC

Figure 4-2. Required Pitch-Rate Gains to Provide 0.35 to 1.30 Short-Period Damping (F-4 Aircraft) marginally satisfies all conditions (e.g., 0.40 deg/deg/sec); hence a scheduling function would not be required to meet current specifications. Should a tighter damping requirement be applied, however, a scheduling function with significant tolerance considerations would be required if this simple controller configuration is to be used. One solution is to utilize some air data function to vary the rate gain, an approach dependent on a reasonable correlation between these data and the stability derivative function which determines rate gain. Such correlations between required gain and the air data quantities of Mach, altitude, and pitot differential pressure are illustrated in Figure 4-3. It is evident from this figure that only altitude (static pressure) bears a significant parallel to the gain required for maintenance of constant damping, and even here a two to one variation in damping over the speed range at sea level is likely. It is evident that the tolerance of the system (short-period damping uniformity) to variations in either the controller or the airplane is severely limited, since the design objective is to simply augment the existing airplane properties. By the same token, a discrete gain change will result in a discrete performance variation.

The alternate design objective, typified by the "model reference" system, is dependent on "sufficient" control gain to meet performance; i.e., any gain is satisfactory (within stability constraints) above some lower limit. The block diagram of Figure 4-4 demonstrates this application of elementary servomechanism theory. Here the attainment of adequate loop gain levels to achieve "model" response is limited by stability constraints on the higher-frequency system modes, such as those associated with actuators, sensors, and structural flexure. These influences have limited FCS bandwidths (frequencies below which the open-loop gain is unity or greater) to 3 or 4 Hz. This range is about an octave above basic aerodynamic frequencies and desired response frequencies, a separation which is somewhat marginal if a controller is to be designed with no parameter variation over the flight envelope.

Detailed design studies have produced fixed-gain[1] model reference controllers which satisfy MIL-F-8785B requirements (and C* requirements) over the operational flight envelope of high-performance airplanes, but the compliance tends to be marginal at conditions of low dynamic pressure. A loop gain variation directed towards a constant bandwidth at the 3- or 4-Hz level provides a significant performance improvement. Since the airplane transfer function around these frequencies for most FCS designs approaches a pure rotational inertia (e.g., $M_{\delta e}/S$), the theoretical gain variation is inverse with surface effectiveness. Nearly all of the adaptive systems advanced over the past decade have been directed toward this end. Because the situation is borderline relative to gain-changing need, considerable variation is evident in the applied techniques, ranging from no gain variation (e.g., the A-7 CAS) to a continuous adaptive variation (e.g., the F-111 CAS).

[1]Fixed gain except for a discrete increase often applied in the gear-down condition

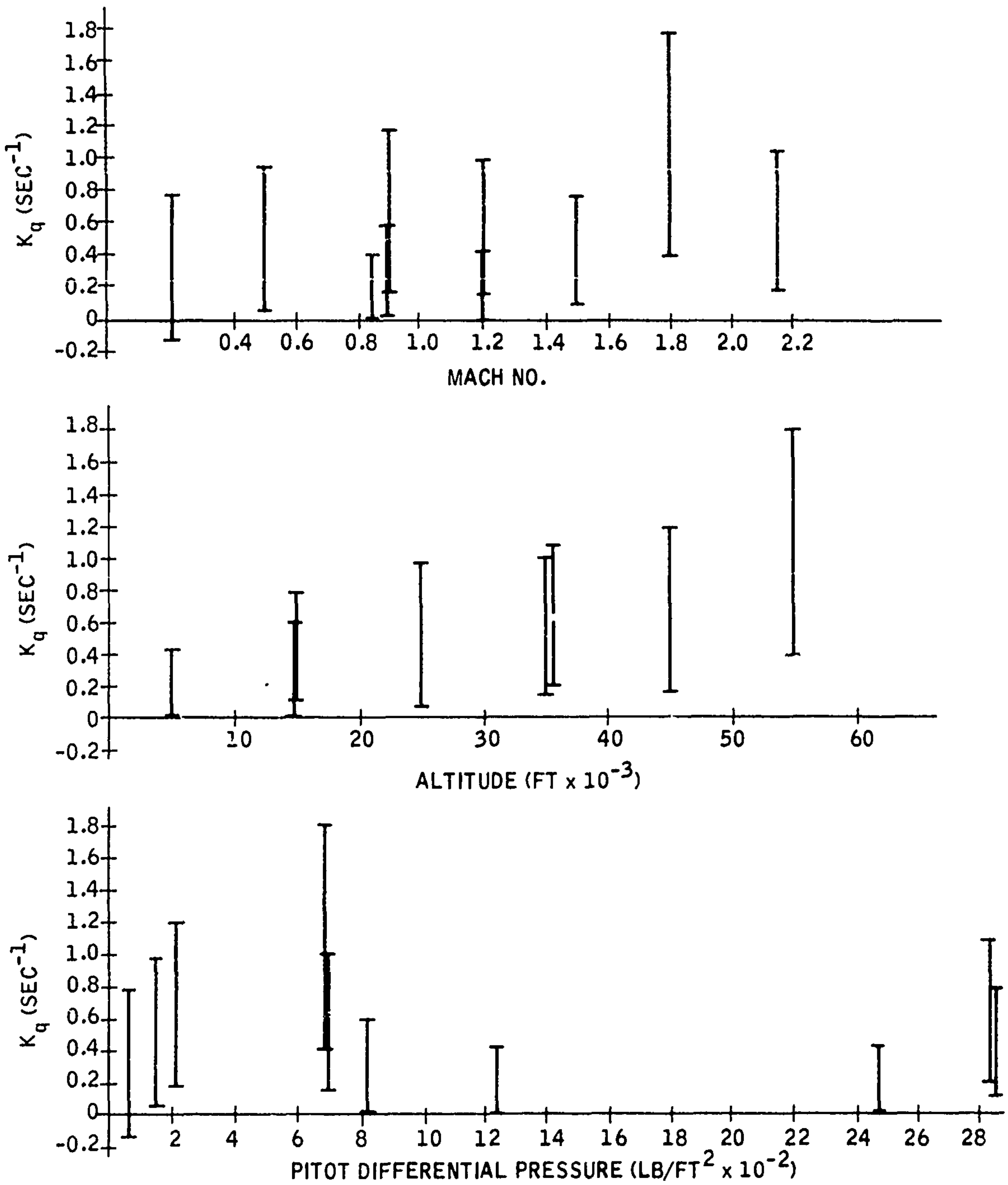
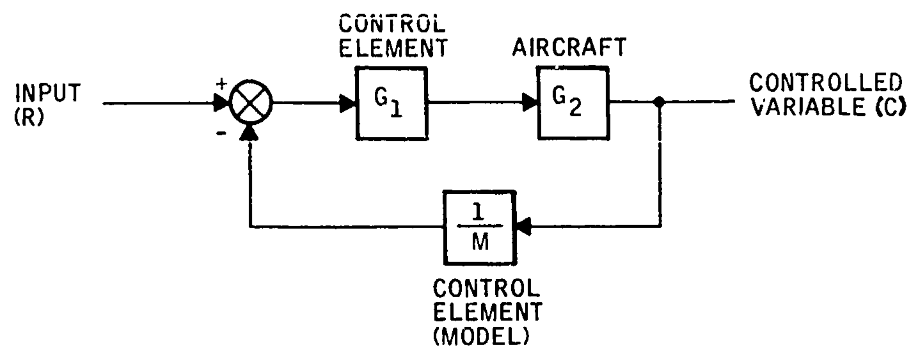
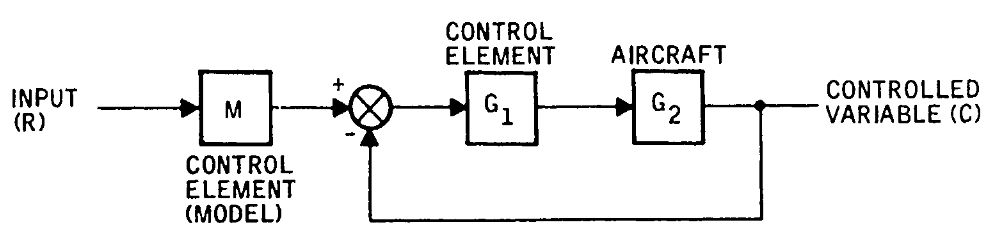


Figure 4-3. Pitch-Rate Gain Correlations with Air Data



$$\frac{C}{R} = \frac{G_1 G_2}{1 + \frac{G_1 G_2}{M}} \approx M \text{ for } \frac{G_1 G_2}{M} > 1$$

(a) FEEDBACK MODEL



$$\frac{C}{R} = \frac{M G_1 G_2}{1 + G_1 G_2} \approx M \text{ for } G_1 G_2 > 1$$

(b) INPUT MODEL

Figure 4-4. Model Reference Feedback Control

The relationship between bandwidth/gain constraint and attainable tolerance to airframe variations can be illustrated by hypothesizing a model reference controller and computing the response properties at various loop gain values. Such a hypothetical control system is shown in Figure 4-5. A first-order feedback model has been selected to fall within the C* Category I allowance of Reference 4-2. Considering the feasible maximum loop bandwidths to fall between 2 and 8 Hz (the range between the highest short-period frequency and the lowest bending frequency for fighter aircraft), the following cases are of interest:

Case	Gain	Loop Crossover (unity gain) Frequency
1	Fixed, $K_q = \dfrac{2(2\pi)}{60\ M_{\delta_{e\ max}}}$	Variable, 2 Hz max.
2	Fixed, $K_q = \dfrac{4(2\pi)}{60\ M_{\delta_{e\ max}}}$	Variable, 4 Hz max.
3	Fixed, $K_q = \dfrac{8\ (2\pi)}{60\ M_{\delta_{e\ max}}}$	Variable, 8 Hz max.
4	Variable, $K_q = \dfrac{4(2\pi)}{60\ M_{\delta_e}}$	Fixed at 4 Hz

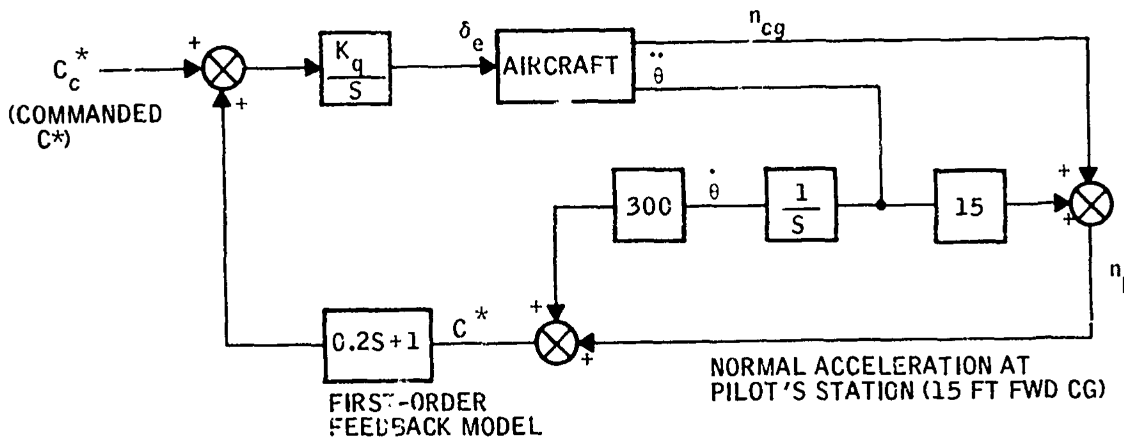


Figure 4-5. Model Reference C* Control

The gain K_q is computed based on the aircraft transfer function C^*/δ_e approximating 300 M_{δ_e}/S at the specified crossover frequencies. The closed-loop rigid-body frequency response at each of the flight conditions shown in Figure 4-2 was calculated for each of the cases specified above and compared to the Category I frequency response envelope specified in Reference 4-2. The results are plotted in Figures 4-6 through 4-9. The Case 1 situation is well beyond the specification region for the lower q conditions (Figure 4-6). The Case 2 situation is marginally acceptable except for the landing flight condition, reflecting the current design practice towards use of gain varied with landing gear position. The Case 3 situation is marginally acceptable with a fixed gain; unfortunately, the 8-Hz bandwidth at the 5000 foot, Mach 1.2 condition would encroach excessively on most structural mode regions. The constant bandwidth of Case 4 shows excellent conformance, as might be expected.

The preceding example illustrates the relationships to be maintained between the control system bandwidth (ω_c) and the desired response bandwidth (ω_d), namely that $\omega_c \geq \omega_d$. Selecting the "slower" side of an acceptable response as a means of determining ω_d establishes a lower bound on ω_c. Further recognizing that the system bandwidth (with fixed control gain) varies directly as surface effectiveness (M_{δ_e}, L_{δ_a}, or N_{δ_r}), suggests the criteria for selection of gain format shown in Figure 4-10. It is assumed in Figure 4-10 that a group of flight conditions is to be evaluated for controllability provided by a fixed-gain control system. The system is to be constrained to a particular bandwidth limit ω_c (perhaps by a known bending frequency range). Knowledge of the surface effectiveness ratio for each condition and the minimum acceptable response frequency at that condition (ω_d), permits placement of a point for each flight condition. Location of these points indicates potential gain changing needs of the system.

For illustration, these criteria are applied to the situation corresponding to Figure 4-7 and plotted in Figure 4-11. Here the minimum desired response frequency is selected (with some artistic license) from the slower bound of the C* envelope as 0.5 Hz and specified as applicable to all flight conditions. The resulting points shown in Figure 4-11 indicate that a gain change is definitely required for condition 1 (see Figure 4-6 for identification), that some of the lower q conditions are marginal, and that the remainder are satisfactory. This agrees with the results of the prior example based on the hypothetical system of Figure 4-5 and also corresponds to other synthesis studies based on the F-4 which involve complete system representations.

Of additional concern to the FCS design is the degree of augmentation. if any, that must be supplied to the inherent dynamic properties of the airplane (e.g., short-period or dutch-roll damping, roll subsidence, etc.). Current specifications relative to existing airframe frequencies are such that conformance to prescribed command response envelopes ensures a

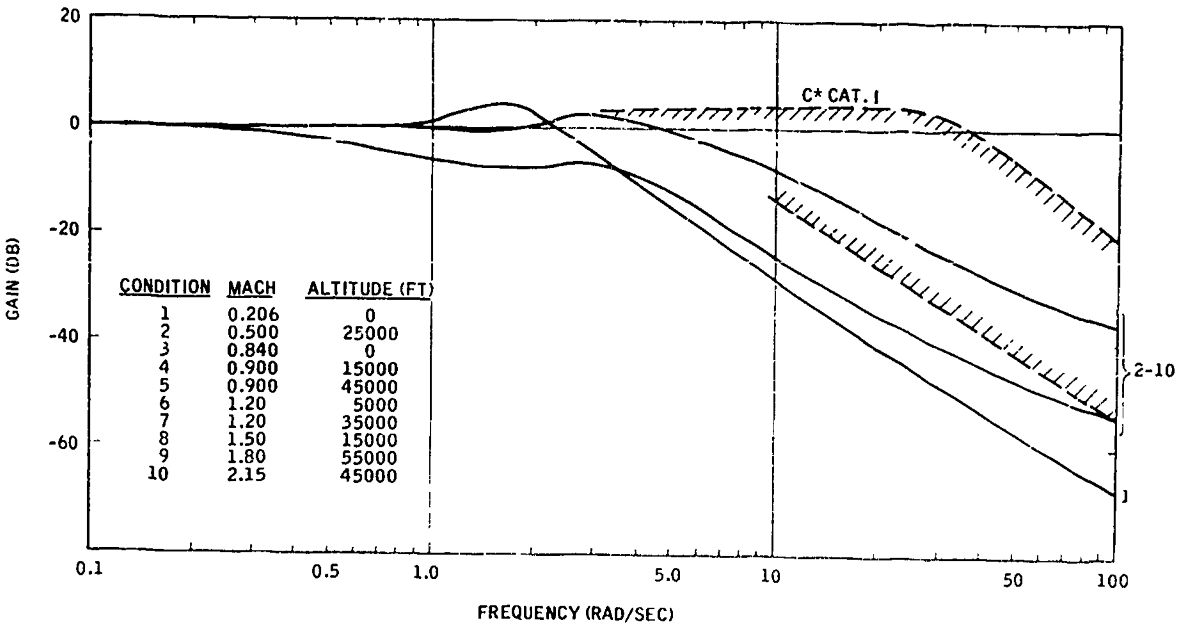


Figure 6. Closed-Loop C* Response to Pilot Inputs - Fixed Control Gain, 2-Hz Maximum Bandwidth

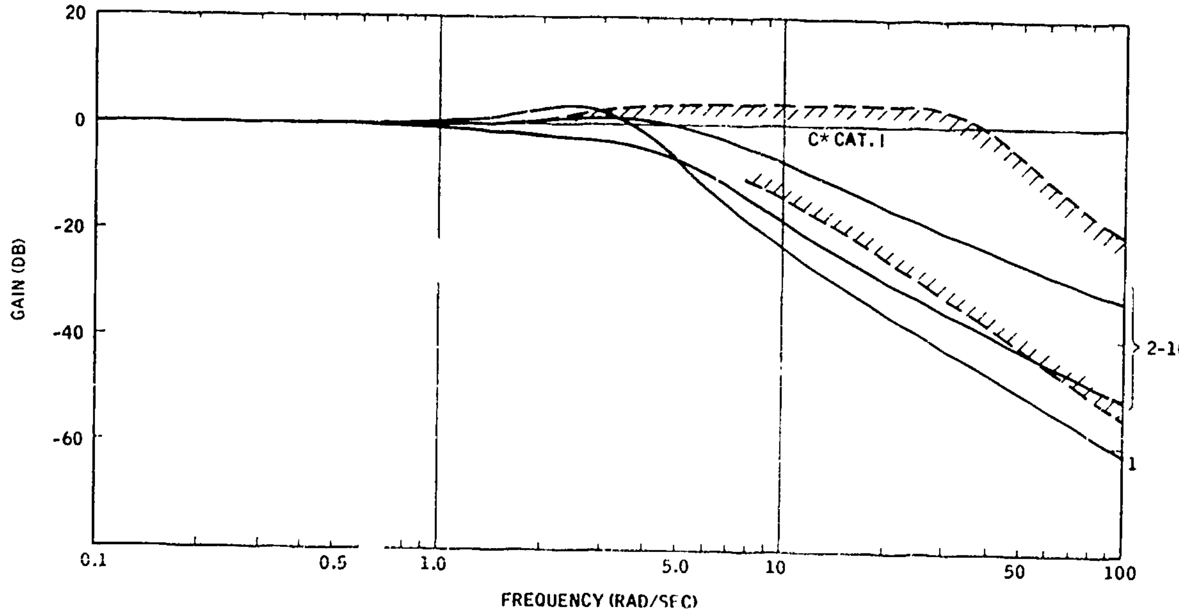


Figure 4-7. Closed-Loop C* Response to Pilot Inputs - Fixed Control Gain, 4-Hz Maximum Bandwidth

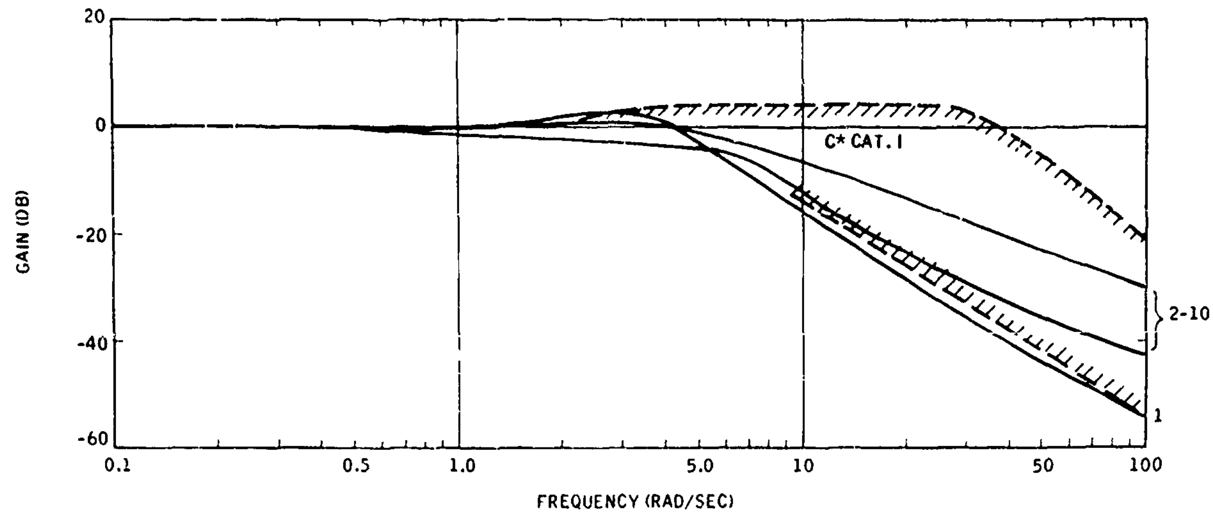


Figure 4-8. Closed-Loop C* Response to Pilot Inputs - Fixed Control Gain, 8-Hz Maximum Bandwidth

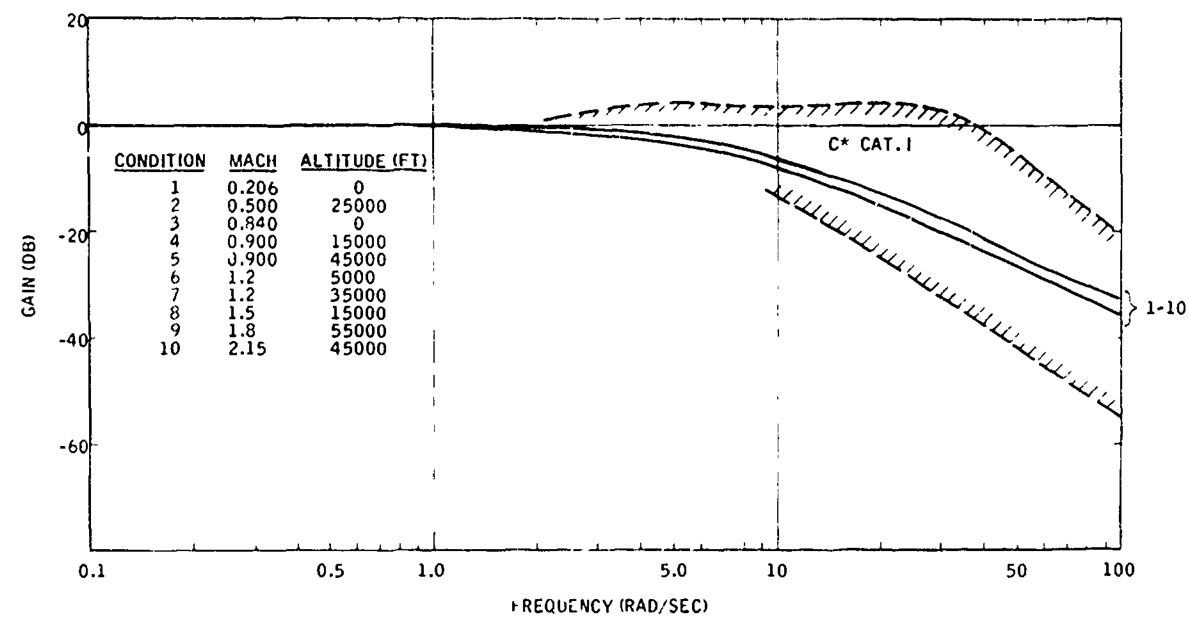
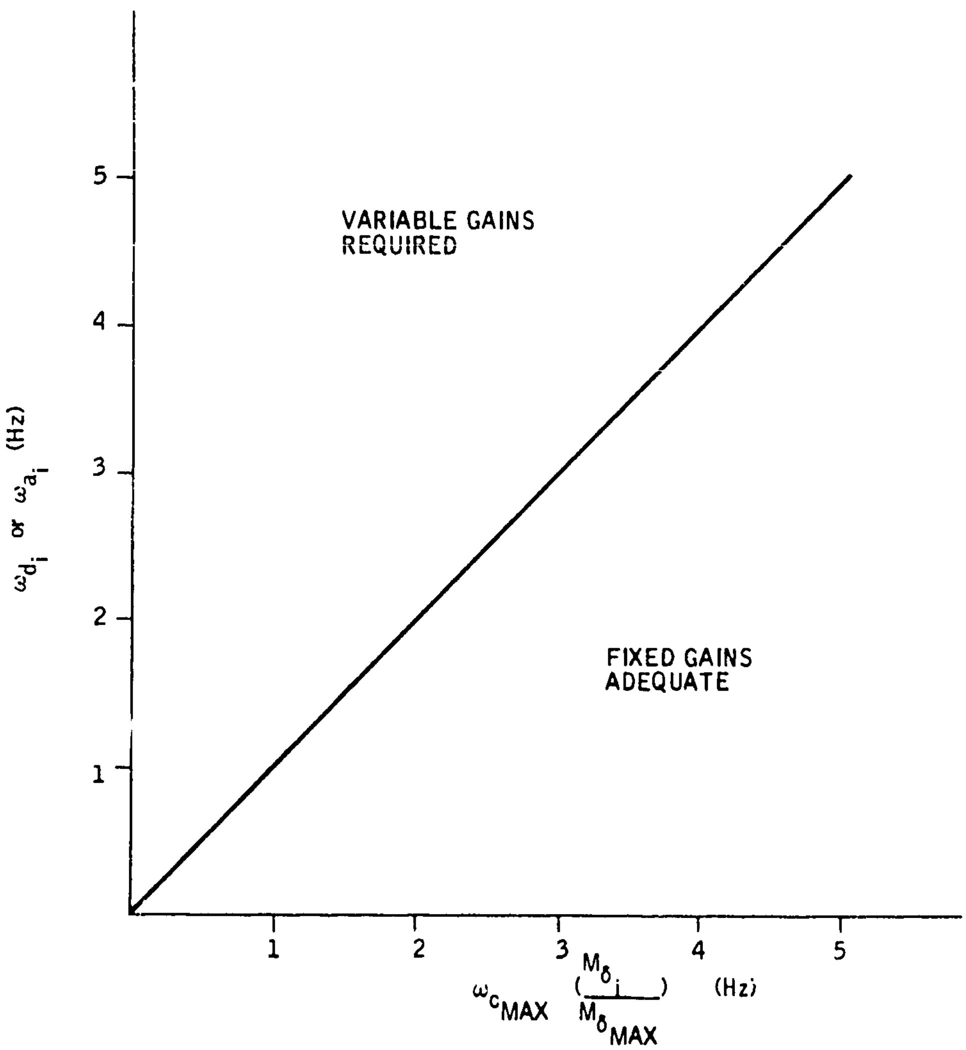


Figure 4-9. Closed-Loop C* Response to Pilot Inputs - Variable Control Gain, 4-Hz Constant Bandwidth



$\omega_{c_{MAX}}$ = MAXIMUM ATTAINABLE SYSTEM BANDWIDTH (UNITS OPEN-LOOP GAIN)

$M_{\delta_{MAX}}$ = MAXIMUM CONTROL SURFACE EFFECTIVENESS

M_{δ_i} = CONTROL SURFACE EFFECTIVENESS AT i^{th} FLIGHT CONDITION

ω_{d_i} = DESIRED RESPONSE FREQUENCY AT i^{th} CONDITION

ω_{a_i} = MAXIMUM FREQUENCY AT i^{th} CONDITION AT WHICH THE INHERENT DYNAMIC PROPERTIES OF THE AIRCRAFT REQUIRE AUGMENTATIONS

Figure 4-10. High-Bandwidth System Gain-Changing Criteria

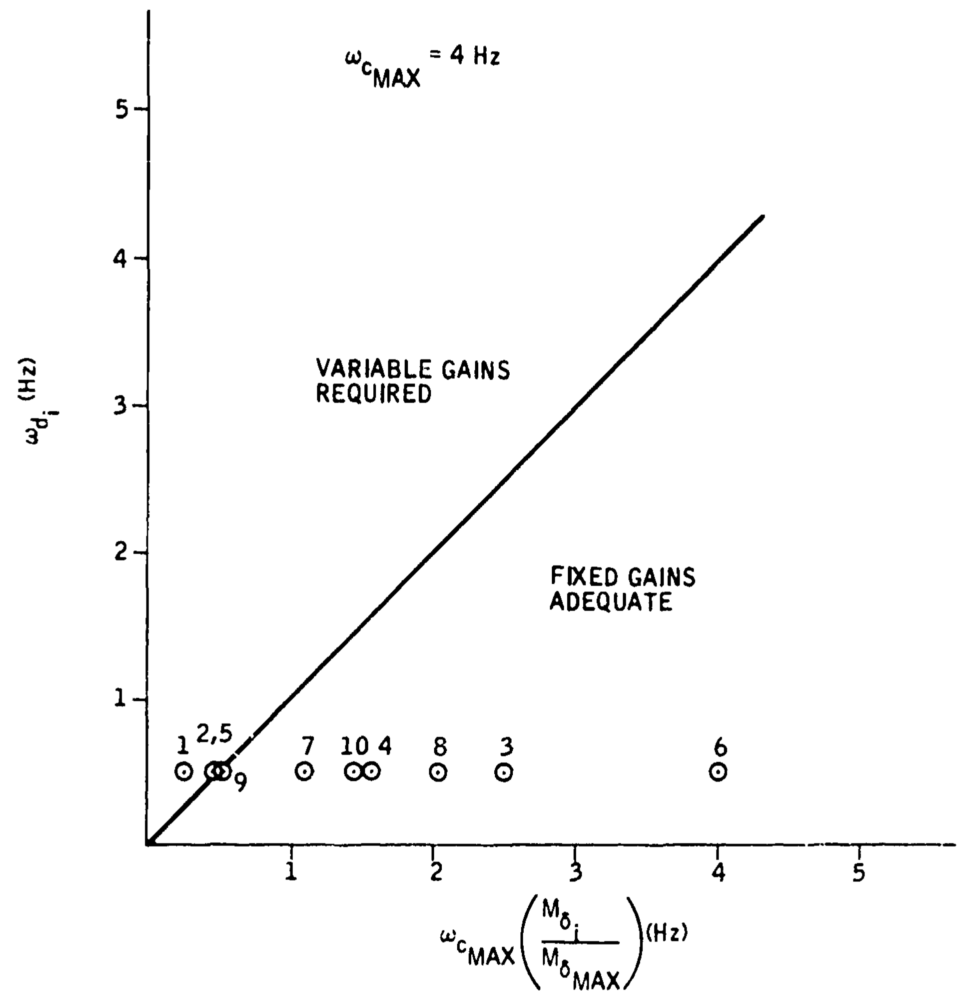


Figure 4-11. Pitch-Axis Control Adequacy to Satisfy F-4 Response Bandwidth

degree of conventional stability augmentation. However, this is not true in general in that adequate control system bandwidth to provide suitable command response may not provide sufficient augmentation of higher-frequency modes (such as bending, for example). Examination of this property by the criteria of Figure 4-10 is performed in terms of ω_a, defined as the maximum frequency at which some dynamic property of the airplane requires augmentation. The criteria are based on the simple premise that a significant degree of augmentation requires significant loop gain, where "significant" is taken to be a feedback contribution equal to the inherent airframe property, or essentially unity open-loop gain. Consistent with prior assumptions, therefore, the system bandwidth must equal or exceed the highest airframe frequency to be augmented, and the latter becomes an additional factor to be evaluated via the criteria of Figure 4-10.

An example of such an evaluation is shown in Figure 4-12. Here the 10 flight conditions of Figure 4-6 are plotted using the short-period frequencies as ω_{a_i}. All fall within the fixed-gain region except the high-speed high-altitude condition, number 9. Consequently, this condition is identified as a potential problem area if a fixed-gain control system is being contemplated. If the basic airframe damping is not greatly deficient (unfortunately it is), the suspect condition may not be a problem. Conversely, a condition may fall within the "fixed-gain" region and, because of greatly deficient inherent dynamics, be a problem to a fixed-gain control. The criteria tends, therefore, to be a necessary but not sufficient condition for effective augmentation.

In addition to the more dramatic differences in control systems as described above, it should be recognized that there are also options available within a given class of controller which can significantly affect the gain changing requirements. For example, a specified response characteristic can be achieved with a variety of feedback sensor arrangements, given sufficient freedom in the controller design, such as replacement of the conventional normal acceleration-rate gyro combination in the pitch axis with only a rate gyro. Here, however, the equivalent in command response could only be maintained in an imperfect fashion and by very involved scheduling. It is evident that the use of more feedback quantities generally reduces the requirement for variations in control law parameters; indeed, such a reduction is a prime reason for increasing the feedback complement.

Aircraft Properties

Variable airframe characteristics relative to the specification are the primary origins of gain changing in flight control systems. These variations are due to flight conditions (speed and altitude) and aircraft loading (magnitude and distribution). The former are dominant, although the latter can be quite significant if the aircraft is intentionally designed with low aerodynamic stability.

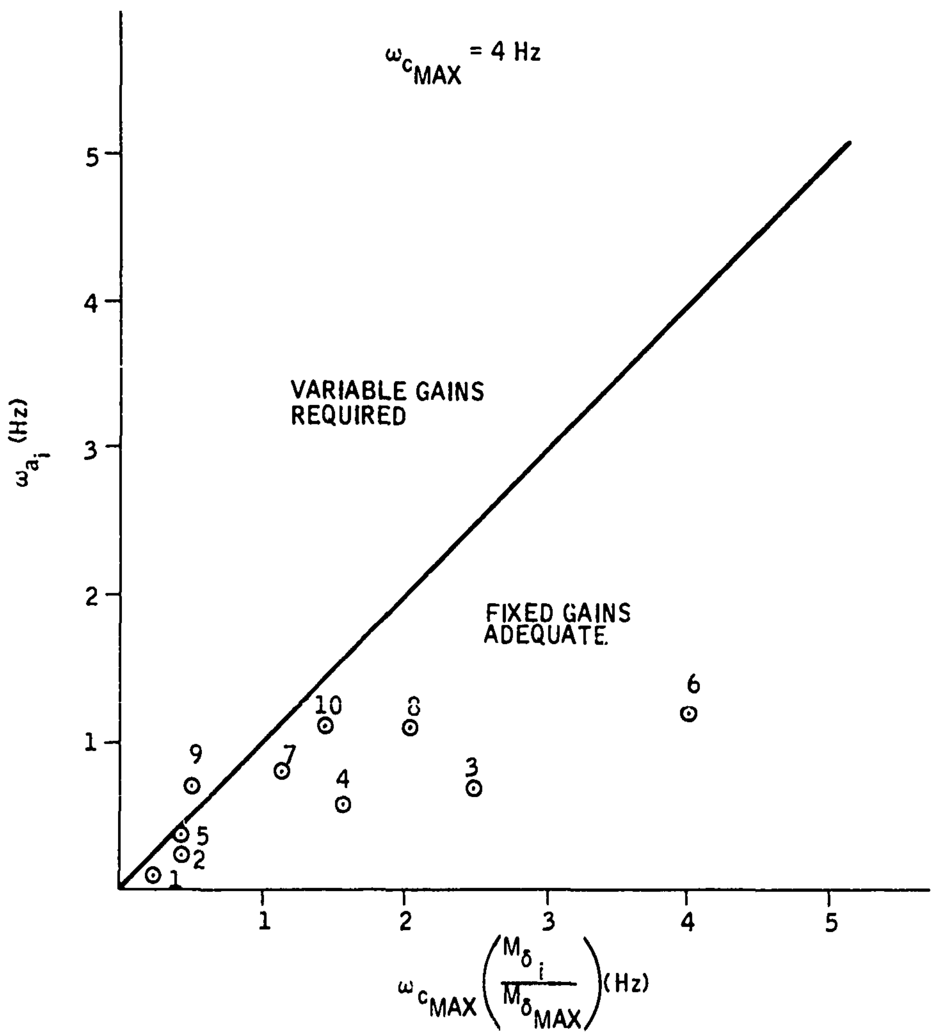


Figure 4-12. Pitch-Axis Control Adequacy to Augment F-4 Short-Period Characteristics

The relevance of surface effectiveness to the gain-changing needs of high-bandwidth system was discussed in the preceding subsection, and examples were given of the performance variation associated with the F-4 pitch axis over the flight range. The variation in M_{δ_e} of about 16 to 1 is considered a normal range for a high-performance airplane, although even greater ranges have been considered in other studies. The effects of aeroelasticity generally act to reduce the range to at least that used here and sometimes much more, even to causing control reversal. Furthermore, the limitations of threshold or resolution in the control surface itself are such that provision of proper resolution at conditions of high effectiveness and significant authority at conditions of low effectiveness limit the useful flight range of the surface regardless of the control gain provided to drive it. A simple example will illustrate.

Position resolution (threshold) on typical electrical feedback servoactuators is in the order of 0.10 percent of full range. Somewhat tighter values (e.g., 0.04 percent) have been specified for one fly-by-wire system.

A control system resolution at the high-effectiveness condition of 0.01 g [a reasonable value in light of the MIL-F-9490C(USAF) limitation of 0.02 g residual oscillation] requires a corresponding elevator resolution of about 0.01 degree. With the 0.01 percent actuator threshold, a ± 10-degree authority results, near the range of most surface controls. Now, if a 1-g authority is to be retained at the low-effectiveness condition, a maximum practical surface effectiveness range of about 10 to 1 is indicated.[2] The point is that control gain increases in regions where the control surface has inadequate authority have little value, a fact easily overlooked in small-perturbation synthesis studies.

A significant deviation to the normal gain-changing relationship to surface effectiveness sometimes occurs in the lateral-directional axes with the problem of achieving turn coordination. This problem is complex, not only because of the involved inter-axis coupling which usually prevails, but also because of the basic sensing limitations which exist.

The preferred sensor complement for the yaw axis consists of lateral acceleration plus yaw rate, the former providing turn coordination and the latter yaw damping. This set is preferred not only for its relationship to the controlled quantities but also because of the compatibility of the sensors with reliability and redundancy objectives, not the least of which is the avoidance of external (airstream) vulnerability. Unfortunately, the use of acceleration plus rate feedback entails limitations for the turn coordination task. These limitations can be exposed by considering some of the basic influences involved in the coordinated turn.

[2] This analysis considers surface effectiveness in terms of short-period g's per degree rather than M_{δ_e}; however, the two quantities correspond sufficiently to validate the argument.

If the basic airplane were assumed to have roll control surfaces which contributed no yawing moment ($N_{\delta a}$) or side forces ($Y_{\delta r}$) and if the yawing moment and side force due to roll rate (N_p and Y_p) were also zero, the relationship between sideslip and bank angle (for $\delta_r = 0$) would reduce to:

$$\frac{\beta}{\phi} = \frac{\frac{g}{U_o}(S - N_r)}{S^2 - S(N_r + Y_v) + (N_\beta + N_r Y_v)}$$

Here the inevitable occurrence of adverse sideslip in the absence of yaw control forces is evident, initiating as the airplane is banked and developing to a sufficient magnitude to counteract the yawing moment due to yaw rate in the turn. The benefit of directional stability (N_β) in minimizing the sideslip is also evident. Development of yaw rate in proportion to bank angle

$$\left(r = \frac{g\phi}{U_o} \right)$$

is, of course, essential to turn coordination, and herein lies the first problem in feedback sensor design for the yaw axis.

In using the rate gyro and accelerometer combination, the former generally dominates for the higher frequencies (around the dutch-roll frequency) and the latter is dominant for the lower frequencies. This arrangement offers dutch-roll damping and is preferable from the standpoint of local vibration pickup.

Unfortunately, it usually results in contrary reactions to the initiation of a banked turn, the net feedback opposing the yaw rate essential for the coordinated turn. Placement of the lateral accelerometer can further aggravate the tendency to initially miscoordinate if the location is forward of the center of rotation for sideslip inputs (point where the initial acceleration of the cg is equal and opposite to the local linear acceleration due to yaw angular acceleration). This location equals $- Y_\beta / N_\beta$, which, for the F-4 airplane, varies from about 6 feet forward of the cg at the landing condition to about 35 feet forward at the high-speed, high-altitude condition. Since lateral accelerometer positions often exceed the former, its initial output during a bank at the low-speed conditions often aggravates an adverse yaw situation.

Of perhaps dominant influence in attainment of turn coordination via a lateral accelerometer is the wide variation in its effectiveness as a contributor of lateral "stiffness" over the flight range. A measure of "artificial" directional stability is given by the product $K_{n_y} N_{\delta_r} Y_\beta$, where K_{n_y} equals the acceleration gain (rudder deflection per unit acceleration).

The dimensional derivative product $N_{\delta_r} Y_\beta$ varies widely over the flight range, by a factor of 100 for the F-4. Since the minimum value occurs at the landing condition, very high acceleromater gains are required to make significant improvements in coordination. For example, doubling the yaw static stability of the F-4 at the landing condition (about a 40 percent increase in dutch-roll frequency) requires an accelerometer gain of 437 degrees of rudder per g of lateral acceleration. This magnitude is generally well beyond practical loop gain values for frequencies around the dutch roll.

The inadequacies of a linear accelerometer as a coordination sensor for the low-speed conditions are recognized in the industry. The unavailability of an alternate sensor and the aversion to use of a sideslip sensor have resulted in employment of various crossfeed or feedforward signals, the most popular of which is the aileron-to-rudder signal. Although these techniques (in conjunction with acceleration) have been used with reasonable success, they do not enjoy the tolerance advantages of the high-bandwidth feedback system which must, of course, be capable of sensing somewhat directly the controlled variable. Lack of this tolerance has contributed to gain-scheduling complexities which are highly dependent on individual airframe peculiarities. Aileron yawing moments can be dominant influences on the coordination problem, varying with flight condition, angle of attack, and placement of the roll surfaces relative to the remainder of the airframe.

Performance Specifications

Current performance specifications for flight controls are centered in MIL-F-8785B(ASG). They are based primarily on pilot assessments of desirable flying qualities for a particular class of airplane and for a particular flight category. There is some reference to particular mission tasks, but knowledge is scarce in this area, and little quantitative data is available to relate performance precisely to a particular task. Current specifications are also largely dependent on the use of conventional control surfaces and conventional approaches to basic airframe stability. That these limitations prevail is indicative of the complex and evolutionary nature of aircraft stability and control. Of interest to this study are the potential influences of current and future specifications on the gain-changing question.

Perhaps the initial reaction to a proposal for more complete specifications on more complex airframes (the anticipated growth trend) is that the gain changing problems will grow accordingly. Such is not believed to be the case, however.

In the first place, examination of the trends in augmentation system design over the past two decades reveals no correlation between gain-changing complexities and system capabilities. The stability augmentation systems for the F-100 and F-101 involved air-data scheduling of reasonable complication. The development of adaptive systems over the last decade has produced substantial electronic complexity in some cases. The need for this complexity may be questioned, however, in light of more experience in applying feedback control techniques and the expanding base of available feedback sensors. Certainly the fixed-gain augmentation systems of the F-4 and A-7 do not support any trend toward aggravation of the gain-changing problem.

A second influence is the basic limitation in control surface operating range discussed previously. Vehicles with expanded flight envelopes (e.g., VTOLs) counteract this limitation by employing alternate forcing functions (e.g., reaction jets). Selection of alternate control elements can be accompanied by adjustment of gain format. By a similar token, selection of alternate control properties (perhaps for use in a particular mission task) can likewise accomplish gain adjustment. It is argued, therefore, that neither vehicles with expanded flight envelopes nor flight controls offering selectable performance properties will afford significant gain-changing complications.

Of most significance to the gain-changing question appears to be the need for higher reliability in flight controls and the associated use of redundancy. Of vital importance to a valid redundancy approach is channel isolation and avoidance of failure modes common to two or more redundant channels. Commonly used for failure detection is comparison of the outputs of redundant elements. Both the isolation and comparison monitoring facets offer particular problems to the use of air-data sensors, since attainment of one makes achievement of the other difficult. For example, output tracking of sufficient accuracy makes location of probes close together desirable, an obvious invitation to common destruction by foreign objects. This problem might be circumvented by a number of methods:

- Using basically fixed gains, with gain adjustment a non-essential performance improvement function;
- Using monitoring not dependent on comparison of outputs;
- Not using air data but rather some adaptive technique based on "internal" measurements.

The latter approach illustrates an added benefit for the "pure" adaptive-type system which utilizes only the feedback signals organic to the FCS. Special attention must be given redundant mechanizations of adaptive gain changers, however, to effect channel tracking with attendant isolation.

GAIN-CHANGING TECHNIQUES

This subsection presents an overview of the principal gain-changing methods developed within the flight control industry and an appraisal of their major features. The treatment here is not intended to be a chronicle of adaptive works, there being several rather complete survey papers published on this subject (e.g., Ref. 4-3, 4-4). The intent is, however, to cover the basics of the major techniques available and give prominent examples of their application. The available techniques are categorized in Figure 4-13.

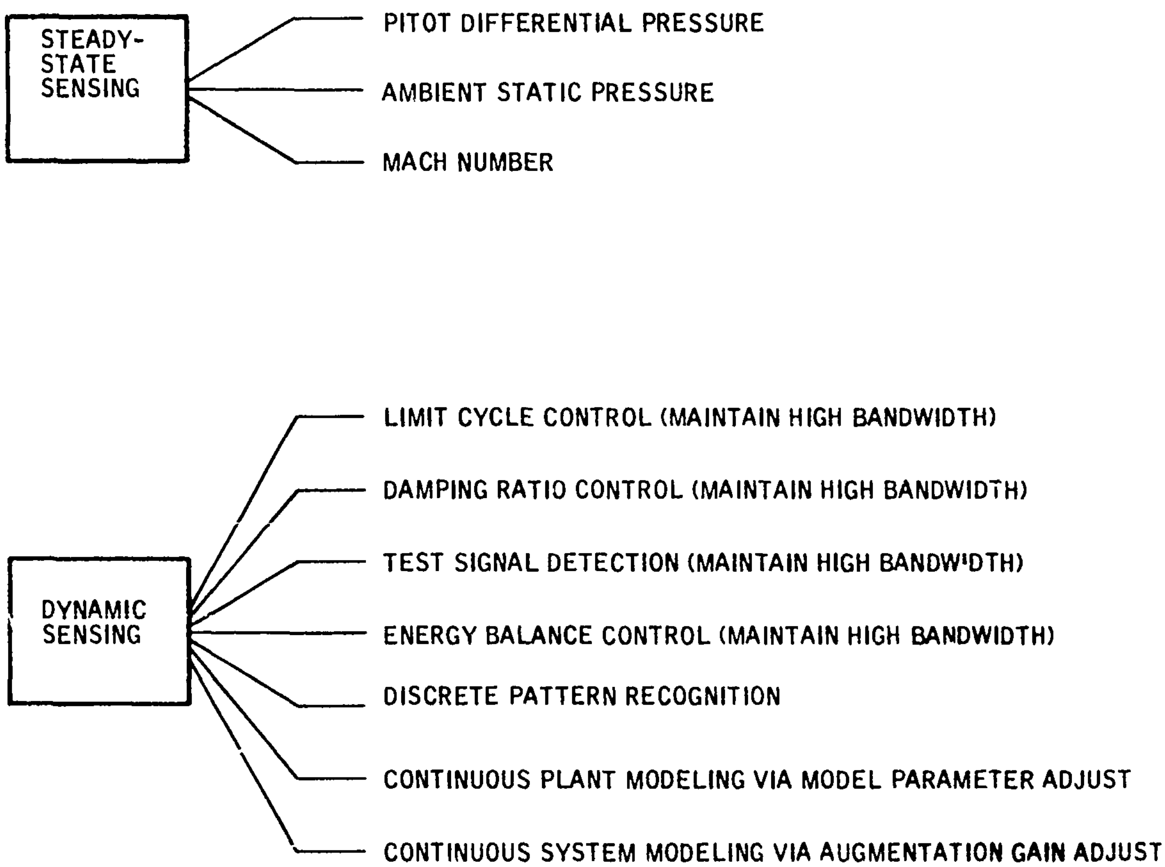


Figure 4-13. Gain-Changing Technique Categorization

The first distinction made relates to the information upon which the gain adjustment is based, categorized as being "steady-state" or "dynamic". The former measures the long-term environment in which the airframe is operating -- speed, pressure, altitude, or functions thereof. Predicted relationships between this environment and key stability derivatives are relied upon to determine gain schedules. Obvious inaccuracies arise because of this prediction, which generally assumes a particular airframe weight and configuration and is insensitive to degradations in numerous elements of the system. It is important to observe, however, that proper feedback system design can often accommodate these inaccuracies, particularly if they are reflected as errors in forward-loop gain elements. Because of the relatively low-frequency nature of the gain-changing data, the potential for adverse coupling with dominant system dynamics is minimized, a major advantage of this type of system.

The techniques which use dynamic sensing base their gain adjustments on measurements which include the dominant system frequencies, at least those around the highest crossover (unity open-loop gain) frequency and sometimes including frequencies below the short period. All of the known "adaptive"-type systems fall into this category, as well as other types which are simply identifiers of key stability derivatives. Techniques in this category are sometimes classified further as being "open-loop" or "closed-loop" gain adjusters, the distinction being the relative ability to assess the results of a particular gain setting and make readjustment if the performance of the control system deviated from some standard. The "closed-loop" adjusters are considered to be advantageous in this respect, although they are generally quite limited in their ability to compensate for many of the parameters that could potentially change. A key characteristic of a closed-loop adaptive technique is that it always is associated with, and a functional part of, a feedback control system, whereas an open-loop technique could function with no actual feedback control system in operation. With the closed-loop adaptation, the object of the process is clearly identifiable in terms of system performance (e.g., maintain high bandwidth), whereas an open-loop technique could adjust gain to accomplish alternate ends.

Another potential classification for the dynamic gain control technique is relative to the nature of their identification signals, i.e., whether they are unique to the identification process (like a limit cycle or applied test signal) or whether they are the "normal" controlling signals. Preference for the latter is common and understandable; however, use of the unique identification signal offers continual (hence up-to-date) and sometimes more accurate evaluations.

The individual techniques identified in Figure 4-13 are discussed further with prominent examples in the following paragraphs.

Air Data Sensing

Steady-state air data variables are conveniently available to the FCS and historically have been the most common means of obtaining gain changing information. A procedure for selecting appropriate schedules was illustrated in Figure 4-3. The desirability of knowing surface effectiveness for gain adjusting in a high-bandwidth system was also discussed. The correlation between pitot differential pressure and surface effectiveness is promising for this function provided that deviations due to supersonic Mach effect can be tolerated. This relationship, using pitch surface effectiveness, is illustrated for four aircraft in Figure 4-14. In the absence of Mach corrections, the procedure would be to assume a subsonic linear relationship and suffer the loss in attainable gain for the supersonic conditions. At most this loss appears to be only 8 db for a Mach 2.0 condition.

Limit-Cycle Control

Limit-cycle control is an adaptive technique designed to hold loop gain at its maximum stable value by detecting and maintaining a small-amplitude limit-cycle oscillation at the crossover frequency (3 to 4 Hz). It indirectly establishes, therefore, a constant-bandwidth system at the maximum attainable frequency for the prevailing system dynamics. The basic block diagram is shown in Figure 4-15. The limit-cycle detection consists primarily of a bandpass-filter-plus-rectifier combination. Limit-cycle amplitudes differing from the set value cause gain variations. Control signals other than the limit cycle itself which pass the detector can cause transient gain reductions -- the magnitudes of which are tolerable for vehicles which have no inherent instabilities capable of divergence below a minimum loop gain setting. Using the intentional limit cycle produces accurate and timely adaption, at amplitudes undetectable by the pilot. Addition of a limit cycle solely for identification purposes is generally viewed with disfavor, however.

This system was developed by Honeywell in 1958. Under USAF sponsorship, it was applied to a three-axis variable-stability CAS used in an F-101 at the Aerospace Research Pilots School. A more advanced model incorporating redundancy was later used in an X-15.

Energy-Balance Control

Energy-balance control is another adaptive control process designed to maintain a constant, high-bandwidth system. It does so by maintaining, via loop gain adjustment, an equality between the output signals of two filter sets. One set measures the amplitude of signals around the maximum crossover frequency, the other about one decade below this. The higher-frequency set is phased to drive the loop gain down; the lower-frequency set drives the gain up. When the system bandwidth is a nominal value (somewhat below

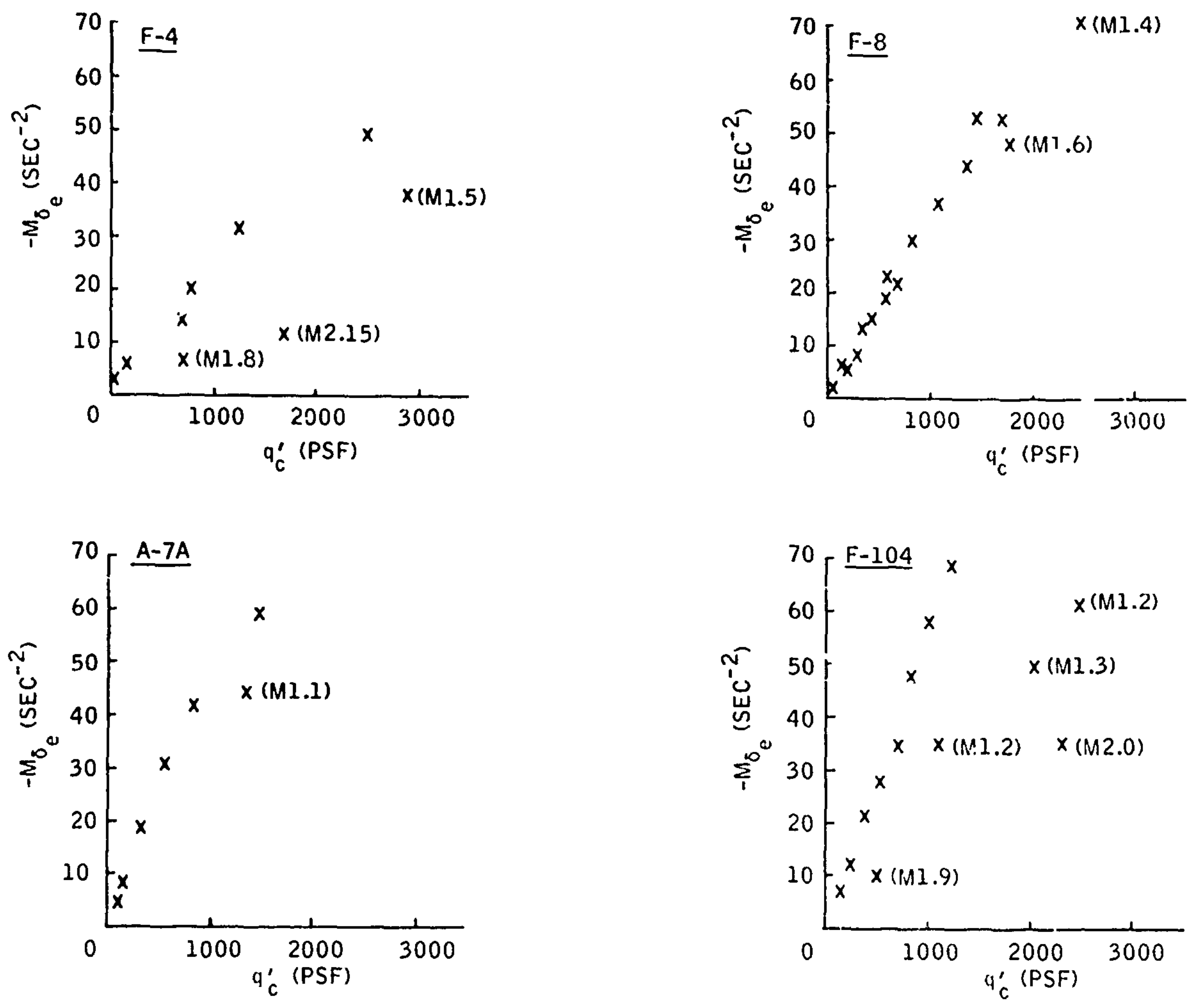


Figure 4-14. Correlation Between Pitot Differential Pressure and Surface Effectiveness

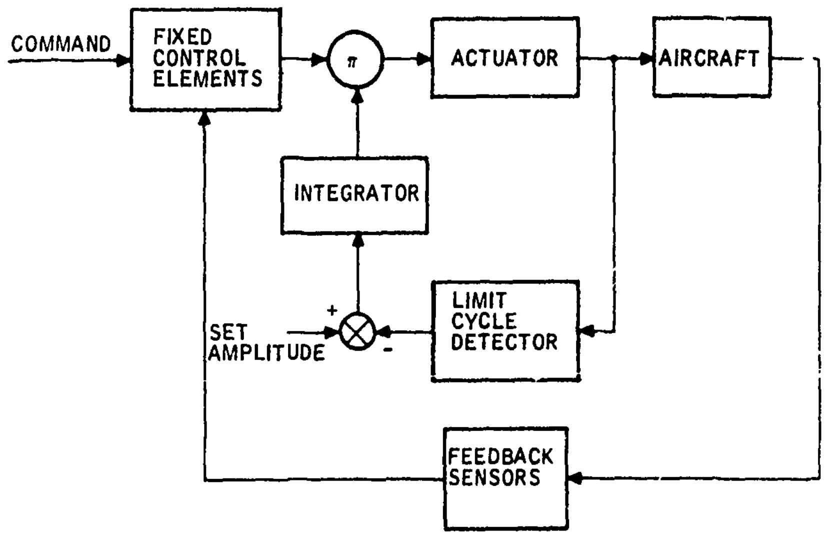


Figure 4-15. Gain Changing via Limit-Cycle Control

maximum) and the applied disturbances are normal (distributed across the entire frequency range), the filter outputs balance, and the gain variation ceases.

A block diagram of this system is shown in Figure 4-16. A common input to both filter sets is illustrated, although alternate versions of this system have used separated inputs (with appropriate changes in filter properties).

This system has the favorable attribute of using normal control perturbations for evaluation. It is sensitive, however, to the spectral distribution of commands and disturbances, and must in general be designed by assuming a particular model for the anticipated inputs. Successful use requires capacity for accommodation of variation in input spectrums, an ability which is often dependent on the nature of the controlled system. For example, an inherently unstable vehicle that can cause control loop divergence at excessively low control gains is uniquely suited for this type of system, since either inadequate gain or excessive gain will result in strong control loop resonances to drive it back to the nominal value.

Both Sperry and Honeywell have conducted developments on this type of system, much of it under USAF sponsorship. Honeywell combined it with a limit cycle control for application to booster vehicles and to the X-20. The latter was mechanized in a triple-redundant configuration.

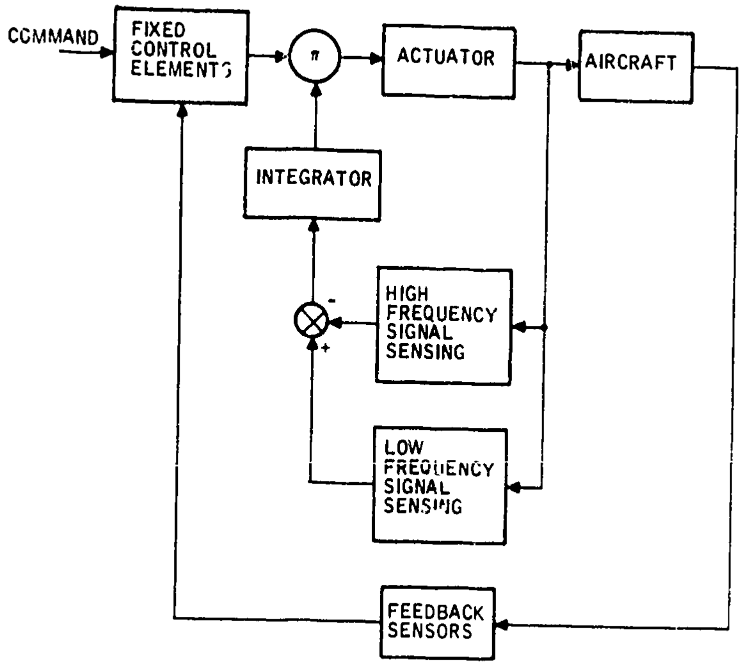


Figure 4-16. Energy-Balance Gain Control

Damping Ratio Control

Another adaptive technique for maintenance of a constant high-bandwidth, damping ratio control does so by holding the resonance associated with the bandwidth limit (the potentially divergent root) at a constant damping ratio (e.g., 0.3). It can be considered, therefore, as being somewhat analogous to the previously described limit-cycle control except that the latter holds the root at zero damping ratio. The basic block diagram is shown in Figure 4-17. Adequate evaluation of the key damping ratio requires sufficient stimulation. Normal control disturbances are inadequate, so a periodic pulse (undetectable by the crew) is applied for stimulus. Detection threshold is varied as a function of turbulence-generated signal level to minimize associated adjustment errors.

This system was developed by General Electric and has been applied to the F-111.

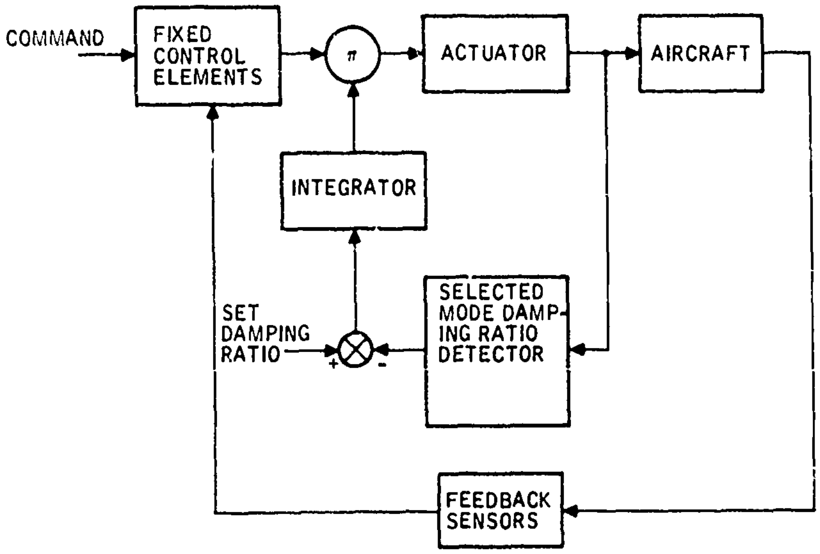


Figure 4-17. Damping-Ratio Gain Control

Test Signal Detection

Test signal detection involves application of a signal of known properties to the control loop and measuring the resulting system output. The signal applied is generally a discrete sine wave at a frequency above the highest short period and near the desired crossover, the object being to assess surface effectiveness and the associated control gain requirement for maintenance of a constant high bandwidth. In addition to gain, phase measurement and corresponding shaping adjustments can also be made. This system alleviates the usual problem of signal discrimination in a noisy environment, with the advantage that better filtering can be used with a known test frequency. A sample block diagram is shown in Figure 4-18.

Primary development of this system has been conducted by Autonetics under USAF sponsorship.

Discrete Pattern Recognition

The discrete pattern recognition technique includes those procedures which discretely categorize vehicle control properties (e.g., stability derivatives) by comparison of sensors output functions with preset performance models. Identification is generally performed using normal control perturbations in a frequency range which encompasses the short-period frequencies

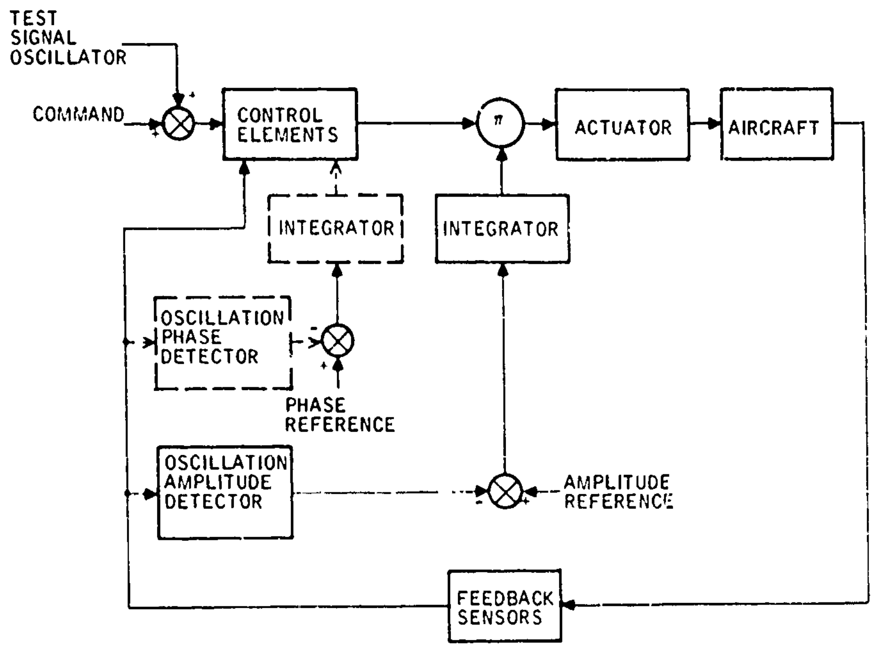


Figure 4-18. Gain Control by Test-Signal Detection

of the aircraft. Alternate combinations of sensed variables are selected to form the model, depending on the properties to be identified and the accuracy desired. The identification can be performed in an open-loop manner without existence of an associated control system.

A simple example of this technique is shown in Figure 4-19. The desired object is to identify the stability derivative Z_α relative to n discrete levels. Here the selected model is based on the short-period normal force equation

$$N_a - \ell_a \ddot{\theta} = - Z_\alpha \alpha - Z_{\delta_e} \delta_e$$

where

N_a = output of a normal accelerometer

ℓ_n = accelerometer location forward of cg

$\ddot{\theta}$ = pitch angular acceleration

α = angle of attack

δ_e = elevator position

Z_α = normal acceleration due to α

Z_{δ_e} = normal acceleration due to δ_e

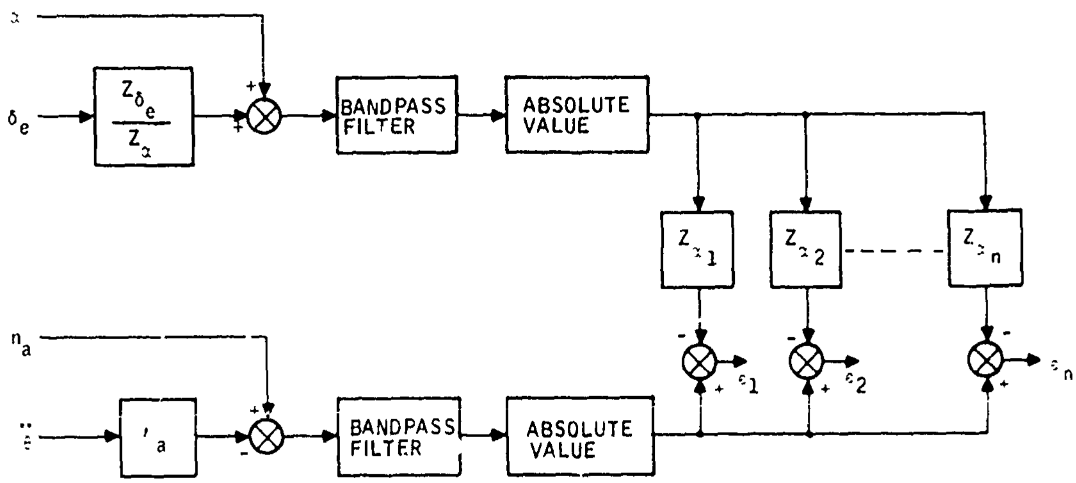


Figure 4-19. Discrete-Pattern-Recognition Technique

With the reasonable assumption that Z_{δ_e}/Z_α is constant, the two sides of the equation are related in magnitude by Z_α. Measurement of the indicated variables, restriction of the measurement to frequencies around the short period (via bandpass filters), and comparison of outputs weighted with discrete values of Z_α constitutes the desired categorization of Z_α.

It should be noted that the selected model in the above example is applicable to either elevator or turbulence-type disturbances; hence valid identification is performed for either input. It is also notable that the identification is valid over a relatively broad frequency range, which increases the amount of stimuli available. These features are indicative of the qualities of a good model, selection of which is a basic design decision. In making this decision, a tradeoff is often available between using more sensed data or using more filtering. Better performance usually favors the former in that filtering is used to correct for model approximations, always an imperfect process. Consider, for example, a desire to identify M_{δ_e} based on the short-period pitch model

$$\ddot{\theta} = M_{\delta_e}\delta_e + M_\alpha \alpha + M_{\dot{\alpha}}\dot{\alpha} + M_q \dot{\theta}$$

It is recognized that, for elevator inputs above the short-period frequency

$$\ddot{\theta} = M_{\delta_e}\delta_e$$

Use of this approximate model instead of the complete short-period equation could be considered, given additional filtering to reject the lower frequencies. Errors would result, however, from turbulence and from imperfect filtering, and the majority of the available identification signals would be lost.

An identifier based on the technique illustrated in Figure 4-19 was developed by Honeywell under Navy sponsorship and flight verified in an F-4 airplane.

Continuous Plant Modeling via Model Parameter Adjust

A close relative of the discrete recognition technique, continuous plant modeling adjusts the model itself to conform to the existing aircraft properties, the resulting model parameters being functions of current stability derivatives. Knowledge of the latter would serve as a basis for system gain adjustment, an added function not germane to the identification process itself.

An example of this technique is shown in Figure 4-20 applied to the pitch short-period equation. The selected model includes all terms significant to the short-period frequency range (added filtering would be necessary for eliminating the phugoid and bending frequencies in an actual system). When the output error (ϵ_O) is finite, the model parameters are driven towards their correct values ($M_{\delta e}$, M_α, $M_{\dot{\alpha}}$, and M_q), at which state the output error is zero. Convergence to the correct solution has been demonstrated (Ref. 4-5) given proper initial conditions and integrator gains. The latter can be varied as functions of signal level to achieve superior identifier dynamics. Various model forms are design options as previously discussed for the discrete identifier.

The majority of the development work on the more complete model forms of this continuous identifier has been done by North American under USAF sponsorship. A simplified version which uses a combination of filtering and single parameter adjustment to compute surface effectiveness has been developed by Bendix, also for the USAF, and is currently included in a redundant CAS in flight test on the F-4 as part of the Tactical Weapon Delivery Program.

Continuous System Modeling Via Augmentation Gain Adjust

An extension of the plant modeling process to include the entire system is the feature of the continuous system modeling adaptive control technique. Here, however, feedback gains are the adjusted quantities; and the object is to achieve conformance to a preset characteristic. This "model" performance does not necessarily entail high loop bandwidths, since feedback correction is only applied to augment existing airplane properties. Correct modeling is attained for both pilot and turbulence inputs.

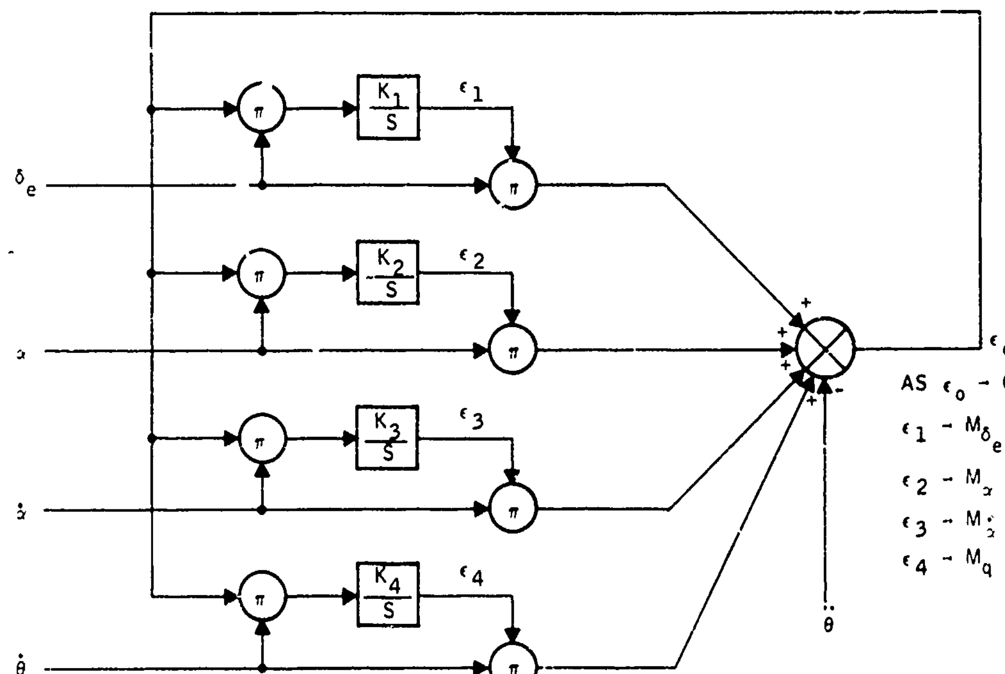


Figure 4-20. Continuous Plant Modeling of the Pitch Axis

An example of this technique applied to achieve an invariant C* response of the pitch axis is shown in Figure 4-21. The system model is formed in accordance with the C* expression:

$$C^* = N_z + U_c\dot{\theta} + \ell_p\ddot{\theta} = K_o C$$

where

U_c = "cross-over" velocity constant

ℓ_p = pilot location forward of cg

C = command signal

K_o = desired proportionality constant between command and C*

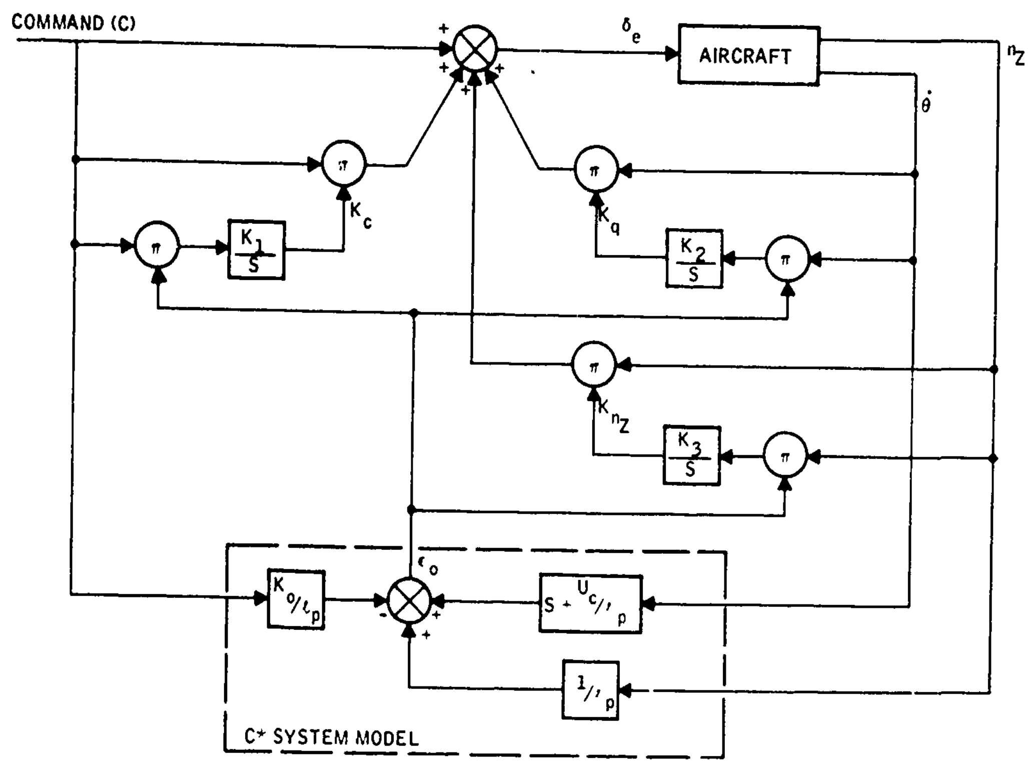


Figure 4-21. Continuous System Modeling via Augmentation Gain Adjust

The parallel between the model and the airplane equation written in corresponding terms is evident:

$$N_z \left(\frac{M_\alpha}{Z_\alpha} + \frac{M_{\dot{\alpha}}}{U_o} \right) - \dot{\theta} \, (M_{\dot{\alpha}} + M_q) + \ddot{\theta} = \left(M_{\delta_e} - \frac{Z_{\delta_e} \; M_\alpha}{Z_\alpha} \right) \delta_e$$

With the addition of two feedback signals and one feed-forward signal, the response can be made to equal that of the model.

An error signal is formed from the C* model as indicated in Figure 4-21 and is applied to vary the gain elements. Convergence to proper gain values to achieve the model response is analogous to that performed by the continuous plant modeling device discussed previously. Proper performance requires attention to initial conditions and integrator gain setting.

Development on this system to data has been limited to computer studies, performed at Honeywell and at the Naval Postgraduate School.

APPLICATION CRITERIA

The foregoing subsections discussed the origins of gain-changing requirements and summarized the techniques developed to satisfy these requirements. The justification and rationale for gain-changing criteria was also developed. There are no comprehensive "handbook" methods for flight control design practice, even for the restricted subject of system gain changing. There are, however, certain major influences and key considerations which collectively determine gain-changing practice. These factors, their associated relationships, and the resulting design criteria are summarized by Figure 4-22.

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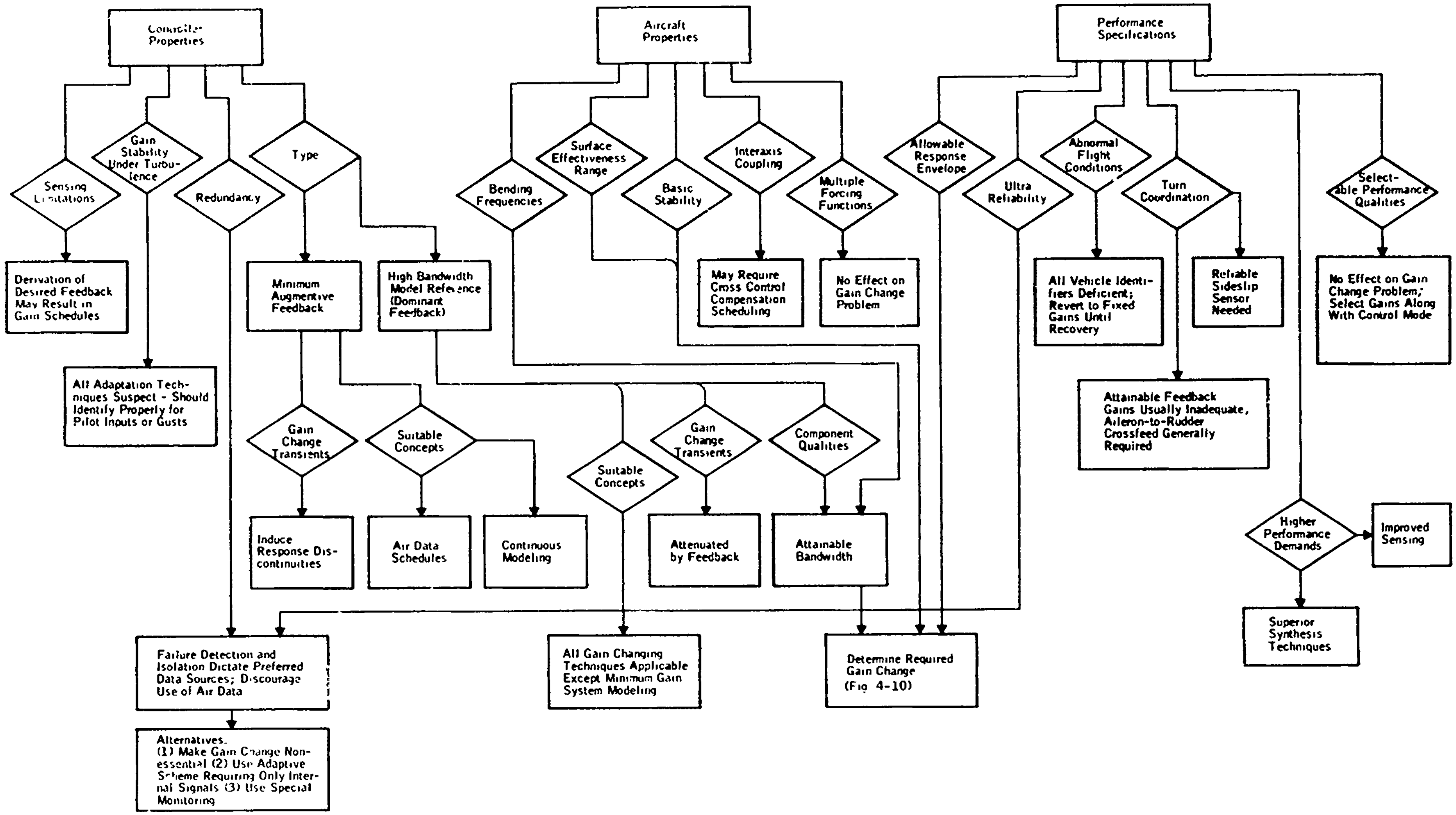

Figure 4-22. Gain-Changing Application Criteria Summary

SECTION V

FLIGHT CONTROL STABILIZATION CRITERIA FOR STRUCTURAL FLEXURE AND OTHER HIGH-FREQUENCY DYNAMICS

One of the elements lacking in present design criteria for airplane flight control systems is a criteria for establishing the high-frequency compensation. Military specifications have recently been updated to reflect significant improvements in the low-frequency response and handling qualities requirements. But in these specifications the only requirement placed on the high-frequency properties of a control system is that there shall be no unfavorable interaction with the aeroelastic modes. In short, no specific requirement is available to assure a safe control system in the high-frequency range (i. e., frequencies above the short-period or dutch-roll frequency range). As a result, each design is generally based on the individual design team's self-imposed criteria. And, of course, the designs meet with varying degrees of success, depending entirely on the experience of the design team and the data available.

The objective of this study is to investigate the uncertainties associated with high-frequency dynamics of closed-loop primary flight controls and to relate these uncertainties to feasible stabilization techniques. To accomplish this end and to contribute an analytical tool for future control designs, an aeroelastic model is defined for the first three symmetric aeroelastic modes. Although this model and the control laws used for tolerance investigations were based on the pitch axis of the high-performance class of airplane, the resulting stabilization criteria are believed to be applicable to other control axes as well in that the tolerance accumulations appear comparable.

PROBLEM DEFINITION

Design of "high-gain" flight control systems for high-performance tactical aircraft is greatly influenced by the design criteria imposed for high-frequency compensation. The criteria imposed establish the attainable gain levels for the system at any given flight condition. Hence, these criteria in essence establish the variation in gain required over the flight envelope to meet the aircraft performance requirements without exceeding stability limits. It has been demonstrated many times that, as the control gain is increased, the system/aircraft performance becomes more tolerant to variations in low-frequency aircraft dynamics. Ideally, if the gain could be made sufficiently high, the system-aircraft performance would be independent of the basic unagumented aircraft characteristics. In this situation, the control gain could be made a constant and hence, the system design made simpler by elimination of gain scheduling. It is evident, therefore, that stabilization of high-frequency modes impacts both low-frequency performance and system complexity.

In the past, the general approach to design of high-gain control systems has been to assume the aircraft to be a rigid body. In this case, the high-frequency compensation was dictated primarily by the dynamics of the actuation system. Gain and phase margins were assumed and applied to rigid aircraft-control system open-loop frequency responses. By applying lead compensation in the control electronics, the lag effects of the actuator could be compensated, resulting in a higher attainable open-loop gain. When the system/aircraft combination went into flight test, it was often discovered that the control gain was too high. The excessive gain was usually attributable to one or both of the following problems:

- The math model used to represent the rigid-body dynamics was in error;
- The control system, by sensing aeroelastic motion as well as rigid-body motion, interacted unfavorably with the aeroelastic modes resulting in limit cycles and unacceptable aircraft vibration.

The problem of excessive control gain was solved by either compromising the system/aircraft performance or by costly modifications to the control system electronics. It is highly desirable to avoid this trial-and-error process, especially when one considers the eventual use of fly-by-wire control. With fly-by-wire, flight test will become more akin to a missile flight test wherein the system must work on the first flight. (See Section VIII for a discussion of system/aircraft compatibility testing.)

The objective, then, is to provide better tools and criteria in the system design phase to minimize high-frequency stability problems. The other aspect to this problem is that the stability criteria should not be overly conservative. An overly conservative criteria can result in unnecessary high-frequency compensation and/or overly complex gain scheduling.

To examine the problem in more detail, consider the block diagram of Figure 5-1 which shows the general aircraft-control loop. The problem is to assure the stability of this loop which includes both control components and aircraft dynamics.

To establish the frequency range considered "high frequency", consider the typical open-loop frequency (assuming a rigid aircraft) response as shown in Figure 5-2. For the high-performance tactical aircraft, the short-period or dutch-roll frequencies usually occur in the range between 1 and 10 rad/sec. Hence, for this study it is assumed that "high frequency" means frequencies above 10 rad/sec. All of the elements shown in Figure 5-1 contribute to the frequency-response characteristic in the high-frequency range. Stability requirements for the control loop have usually been specified (at least informally) in terms of a minimum allowable gain and phase margin. The magnitude of these required margins has varied from application to application.

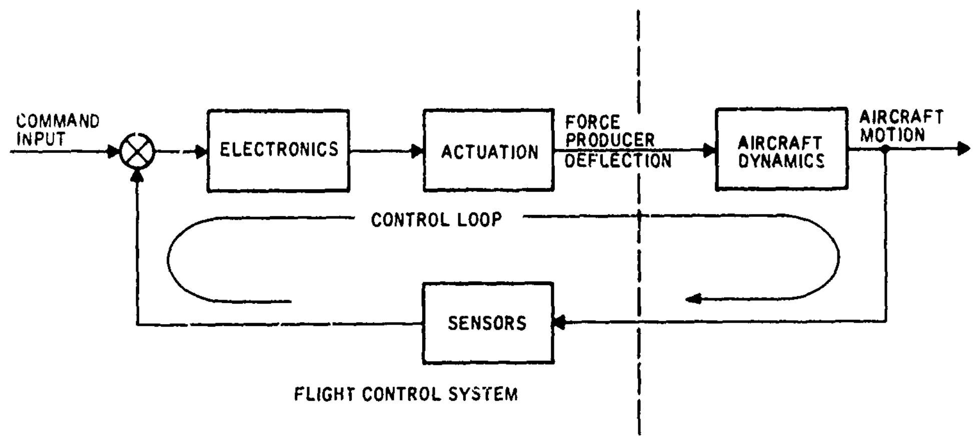

Figure 5-1. Flight Control/Aircraft Elements

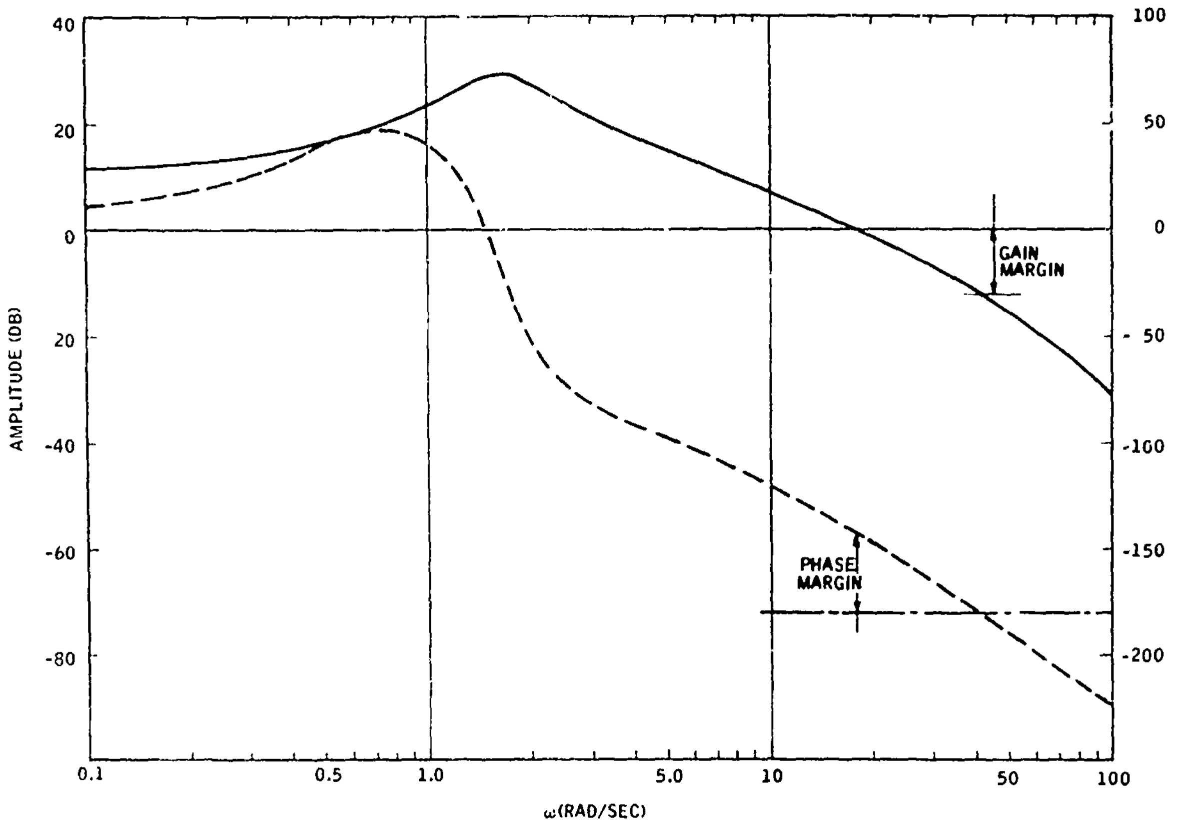

Figure 5-2. Aircraft-System Open-Loop Frequency Response (Rigid Aircraft)

These stability margins have been (and should continue to be) primarily a tool for the system designer rather than a general requirement to be imposed on all systems. The quantities selected reflect a number of factors unique to the particular application, including:

- Tolerances of the control hardware (e.g., analog or digital);
- Nature of the system (conventional or adaptive[1]);
- Accuracy of the vehicle model;
- Point of loop opening in a multiple-loop system.

In spite of these limitations, however, stability margins present a useful measure of system tolerance to variations which enables synthesis in terms of nominal values rather than extremes.

It should also be noted that the ultimate user of the system expects a certain level of performance (which includes not only stability but a certain degree of dynamic acceptability) for all systems, the off-nominal ones included. Hence, a margin of stability from nominal is not of direct interest to the user, particularly since he is probably not aware of the potential tolerance accumulations possible. The general specification should define the dynamic qualities expected from each production model of the controlled airplane over certain flight conditions. These dynamic qualities should in turn imply a certain level of system stability. Unfortunately, current specifications deal primarily with equivalent qualities of lower-order systems (e.g., damping ratio) which are not adequate criteria for the higher-frequency properties indicative of marginal stability conditions.

The stability of all dynamic modes are considered in terms of "damping", a broad interpretation must be used wherein it is a measure of the tendency of the system to "settle down" subsequent to a disturbance. Quantitatively, this might be expressed in terms of decaying oscillations such as percentage reductions in adjacent peak (local maximums or minimums) rates of change of the controlled variable. The nature of this specification is unknown at this time, but certain properties seem desirable:

- The damping quality should be expressed in directly measurable terms relative to specified system variables (probably the dominant controlled variable) and for specified disturbances.
- The damping quality should be independent of system order or linearity.

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[1]Systems such as Honeywell's limit-cycle adaptive system used in the X-15 aircraft make meaningless the gain margin requirement. This adaptive system always sets the gain at its maximum stable value so the gain margin is essentially zero.

- The damping quality should be a function of amplitude. Residual oscillation below specified maximums should be allowed. For discrete disturbances, perturbations of the controlled variable below a specified percentage of the total maneuver increment should be unconstrained.

A comprehensive input/output specification on response of specified variables to specified inputs may ultimately be capable of incorporating the qualities of damping. It would appear, however, that such a specification would need more than a time-response or frequency-response envelope to do so, since the qualities of damping must be measured relative to the existing dominant response. For example, local abnormalties within permissible envelopes can be unacceptable, suggesting perhaps a higher derivative constraint.

The emphasis in the study at hand is on exploration of potential uncertainties in the system elements which contribute to higher-frequency design requirements, in particular those associated with the airframe flexure modes.

These uncertainties are expressed in terms of gain and phase margins due to specified sources, this format being selected to emphasize that each flight control situation must reconsider the tolerance makeup of the system. The magnitude of these required margins has varied from application to application.

Two factors must be considered during design of the high-frequency compensation for a specific system:

- How well can the nominal frequency-response characteristics be defined in the high-frequency range?
- How large should the stability margins be to accommodate variations in the frequency response away from the nominal?

The second factor, the size of the stability margins, of course is dependent on the first factor, how closely the nominal frequency response represents the actual dynamics of the control loop. Knowing the variations in the nominal response due to uncertainties in the modeling, the stability margin required to accommodate the uncertainties can be estimated.

Consider first the accuracy of the nominal response characteristic. Examination of the individual frequency responses of the elements in the control loop reveals that dynamics are added as the frequency is increased. In the vicinity of 10 rad/sec for an aircraft such as the F-4, the aircraft rigid-body dynamics, the control electronics, and power actuator dynamics establish the response characteristic. Increasing frequencies must include aeroelastic modes, higher-order actuator dynamics and sensor dynamics. As dynamics are added, the certainty with which the nominal response is known diminishes. This indicates that perhaps the stability margins should be increased with frequency. In most airplane applications, however, stability margins have usually been specified independent of frequency. On flexible boosters, however, the stability margins do tend to be a function of frequency, primarily because of the presence of significant aeroelastic modes.

In the frequency range between 10 and 100 rad/sec, the greatest unknown is the response characteristic of the aircraft's aeroelastic modes. Depending upon the aircraft's inertial properties, the frequency of the lowest aeroelastic mode can be anywhere in this range. In a flight test evaluation, it is usually the interaction of the control system with the aircraft's aeroelastic modes which necessitates changes in the control system design. An accurate math model of the aeroelastic mode dynamics is usually not available during the control system design phase. This lack of a model results in inadequate high-frequency compensation being incorporated in the design phase. Eventually, it is expected that an accurate model of the flexible aircraft will be available during the design phase. But until this is the case, the control designer must allow for this lack of structural data by increasing stability margins over the frequency range of the aeroelastic modes. The amount of increased margin can be determined from analysis of aeroelastic-mode responses on a similar aircraft for which structural data is available. This approach has been used in some recent programs. In the Navy Adaptable Flight Control Systems Program (1961-1969), a system was designed for an F-4 aircraft using this approach. A gain margin of 6 db was applied at frequencies below the structural-mode frequencies. Over the frequency range of the structural modes, a gain constraint was applied which required the open-loop "rigid body" gain to be below -20 db. A similar type requirement is currently being used on the SAAB J-37 aircraft. This type of high-frequency requirement is designed to provide gain stabilization of the aeroelastic modes. By this it is meant that the sensor pickup of aeroelastic mode signals will be attenuated so as to minimize any feedback of aeroelastic mode signals to the control surface. Stabilization is accomplished regardless of the phasing of the aeroelastic-mode signal through the control system. This is to be contrasted with phase stabilization wherein the response of the aeroelastic mode is not attenuated sufficiently by the control system to avoid potential instability. In this case, the phasing of the sensed aeroelastic-mode signals must be known to ensure stability, thereby requiring increased knowledge of the aeroelastic modes on the aircraft.

Phase stabilization has been used successfully on large flexible boosters in order to extend the permissible bandwidth (i.e., increase the control gain level) of the control system. It has also been applied on large flexible aircraft in recent programs such as the Air Force's LAMS (Load Alleviation and Mode Stabilization) program. In this application, phase stabilization was applied to augment the stability of the lower-frequency aeroelastic modes for the purpose of increasing the aircraft fatigue life. Phase stabilization has not been applied to any great extent on high-performance tactical aircraft, however, for two basic reasons. Adequate "rigid-body" augmentation has been achieved without the need to phase stabilize aerolastic modes.

Also, sufficiently accurate aeroelastic models have not been available for this type of aircraft. However, rigid-body augmentation and handling quality requirements are becoming more and more stringent, which in turn has required higher control system gain levels. Furthermore, developments in structural design of aircraft has resulted in a trend toward increased vehicle flexibility. For example, the F-4 aircraft has a symmetric first-mode frequency on the order of 10 Hz, the F-14 exhibits a 5-Hz first-mode, and the YF-12 has a 2-Hz first-mode frequency. Because of these design trends, it is becoming apparent that phase stabilization may be necessary on future high-performance tactical aircraft to meet "rigid-body" performance requirements.

It appears, then, that a major complication to the task of establishing an effective design criteria is the uncertainty in the basic open-loop response due to a lack of aeroelastic-mode data. Unless an aeroelastic-mode model is available, the designer has no alternative but to allow for this uncertainty in his stability margins. As we have indicated, this currently is being done by increasing the minimum allowable gain margin over the frequency range of the aeroelastic modes.

Another alternative, however, is to develop a generalized aeroelastic model which could be adapted to each aircraft application. If such a model were feasible, a design criteria could be formulated wherein stability margins would be based on using the aeroelastic model. Certainly, an approximate model of the aeroelastic modes should give the designer a better estimate of the high-frequency stabilization problem than having no model at all. If this were the case, it seems reasonable to conclude that the designer could relax his stability margins by using the model. This would then permit him to achieve a more realistic high-frequency compensation. If in fact such a model could be defined, it would at least indicate to the designer where best to place his sensors on the aircraft to either minimize aeroelastic-mode pickup or to possibly achieve some phase stabilization. Use of such a model does not obviate the need for accurate aeroelastic data, but it could serve as a good substitute until such data is made available.

To summarize at this point, it is concluded that the following items should be considered in the definition of a high-frequency design criteria:

- Stability margins should be established which reflect the uncertainty in the nominal aircraft system dynamics at each frequency point.
- Development of a generalized aeroelastic mode model should be explored for use as a means of reducing the uncertainty in the nominal aircraft system response.
- Consideration should be given to both gain and phase stabilization of the aeroelastic modes.

DEVELOPMENT OF DESIGN CRITERIA

The development of design criteria consists of two major tasks:

(1) Definition of gain and phase uncertainties as functions of frequency for expected variations in aircraft system parameters. These uncertainties were computed for the following cases:

(a) No flexible aircraft data or generalized aeroelastic model available.

(b) No flexible aircraft data available, but a generalized aeroelastic mode model available. In this case, consideration was given to bothgain and phase stabilization.

(2) Definition of a generalized aeroelastic model for use in case (b) above.

To derive a quantitative design criteria, it was necessary to pick a representative vehicle/system configuration. The fly-by-wire control system shown in Figure 5-3 was assumed to be representative. The F-4 and YF-12 aircraft were picked as representative aircraft for which at least some aeroelastic-mode data was available. A comparatively good set of flexible aircraft data was available for the YF-12. For this reason, this airplane was selected for computing variations in aeroelastic mode responses.

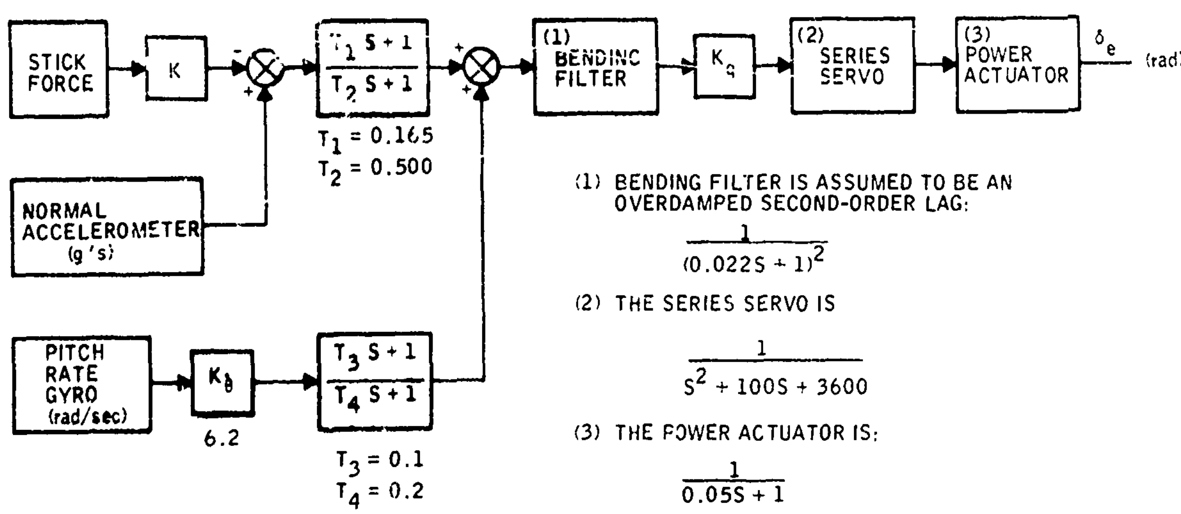

Figure 5-3. Pitch-Axis Fly-by-Wire Configuration

Definition of a Generalized Aeroelastic Model

The procedure followed in defining a generalized aeroelastic model was to take the data provided for the YF-12 aircraft and establish means for adapting it to other aircraft. To accomplish this the following assumptions were made:

- Aerodynamic coupling among the aeroelastic modes and between the aeroelastic modes and rigid-body dynamics can be neglected;
- The control system designer will have a good estimate of at least the lowest one or two aeroelastic-mode frequencies;
- The control system designer will also have knowledge of aircraft rigid-body stability derivatives and the basic inertial properties.

Evidence of the validity of the first assumption is observed in flight test results which often show that the interaction problem with aeroelastic modes is worst at low-dynamic-pressure conditions. It is at these conditions where aeroelastic-mode damping is the lowest due to lack of aerodynamic contributions to damping. Further, it is usually at these conditions where the control gain is highest.

It was further assumed that, for the subject class of airplanes, a model describing the first two or three aeroelastic modes would suffice. Definition of a model to describe higher-frequency modes would be difficult to achieve. Hence, this study was directed towards definition of a model to represent the lowest three symmetric aeroelastic modes. For this task it was assumed the flexible-aircraft dynamics could be represented as shown in Figure 5-4. From this diagram it is evident that to specify the aeroelastic-mode dynamics one must somehow determine mode shapes, slopes, aeroelastic-mode damping, and control surface force coefficients. It was assumed that the aeroelastic-mode frequencies would be provided.

Structural damping ratios are difficult to estimate, but they usually are in the range between 0.01 and 0.05. For the model, a value of 0.025 was picked as being most representative.

Also included as part of the aeroelastic model is the effect of surface inertia as an additional forcing function. This effect is discussed in a later subsection.

To obtain mode-shape data, the YF-12 mode shapes (for three modes) were normalized as a function of aircraft length. This data is shown in Figure 5-5. Thus, given aircraft length, the mode shapes and slopes can be computed from Figure 5-5 for any location. Use of these mode shapes for aircraft other than the YF-12 represents the greatest source of error in the aeroelastic model. Mode shapes are a function primarily of aircraft-mass

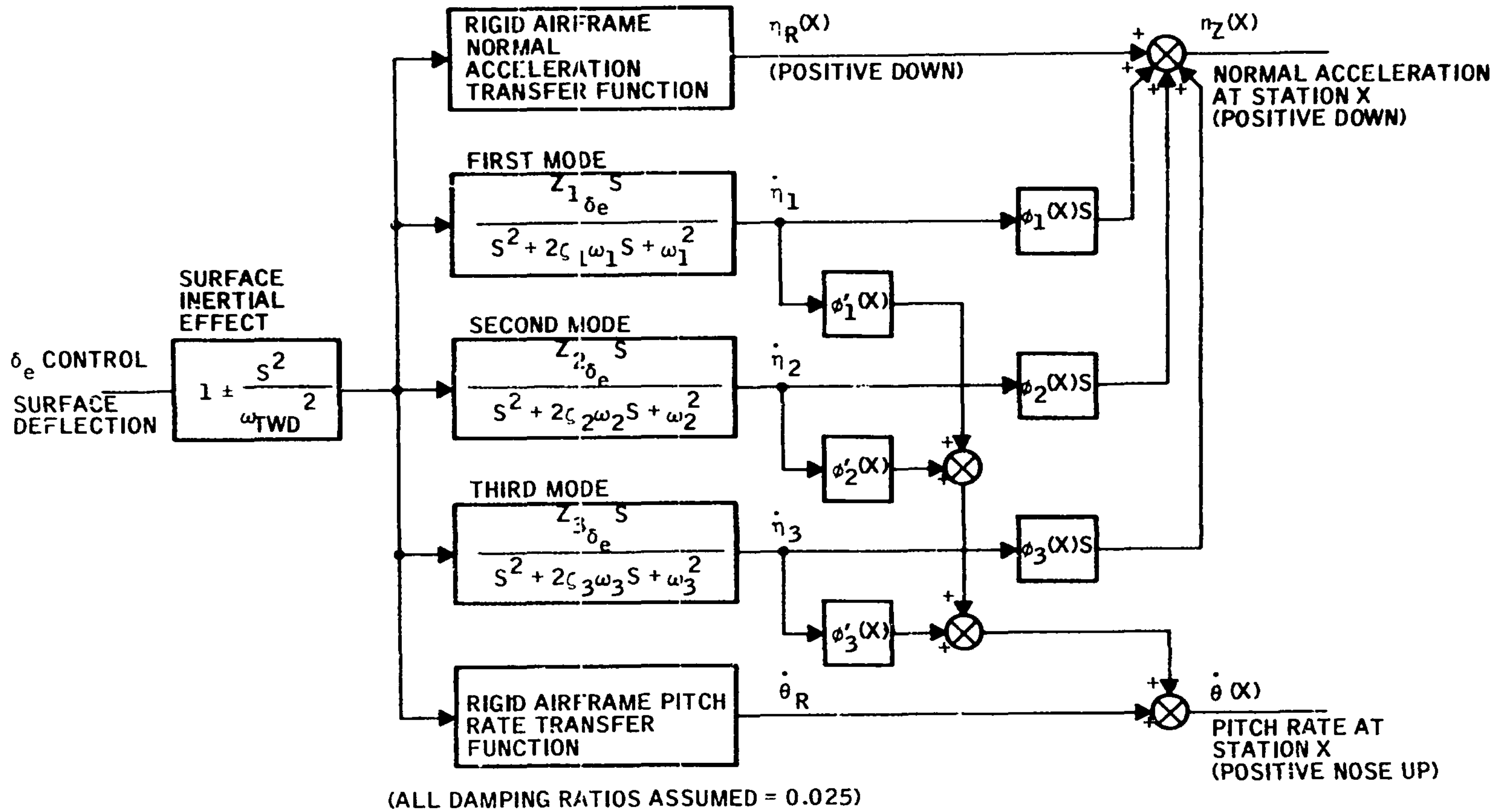

Figure 5-4. Simplified Flexible-Vehicle Pitch-Axis Model

distribution and structural moments of inertia. Some effort was made in the study to devise a way of altering these mode shapes to more accurately represent other aircraft configurations, but no satisfactory procedure was obtained. Hence, by using the mode shapes given in Figure 5-5, the aeroelastic model accuracy will vary with sensor location. Whether or not this is a seriour problem can only be estimated by comparison with another aircraft. Such a comparison will be made in subsequent paragraphs.

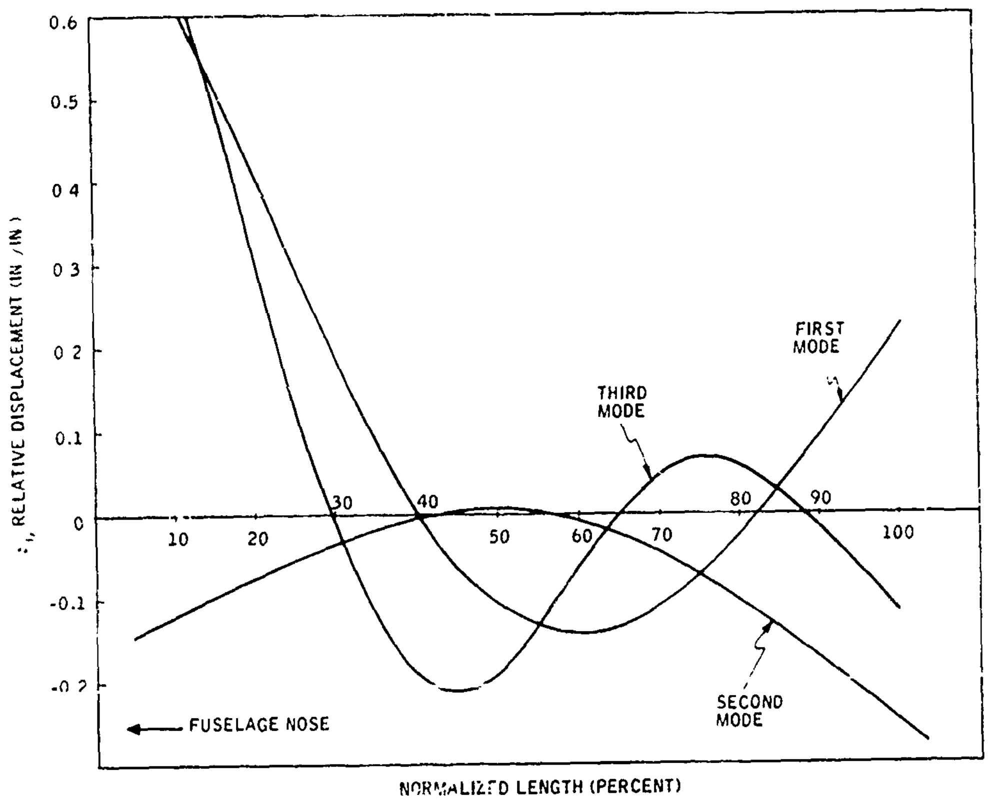

Figure 5-5. Mode Shape as a Function of Fuselage Length

The remaining task is to devise a method for computing the force coefficients $Z1_{\delta_e}$, $Z2_{\delta_e}$, and $Z3_{\delta_e}$ (see Figure 5-4). These terms can be expressed by the following formula:

$$Z_{i_{\delta_e}} = \frac{\phi_i (X_{\delta_e})}{m_i} Z_{\delta_e} m$$

where $\varphi_i (X_{\delta_e})$ is the effective mode shape deflection at the location of the control surface; m_i is the model mass, m is the aircraft mass, and Z_{δ_e}

is the rigid body stability derivative which specifies the normal force due to a unit control surface deflection. The term

$$\frac{\phi_i (X_{\delta_e})}{m_i}$$

is the only unknown in the formula. Fortunately, it is a function only of aircraft inertial properties. Comparison of this coefficient with corresponding coefficients (and mode shapes) for the YF-12 and F-4 aircraft indicated that this coefficient tended to vary inversely with aircraft mass and length. As a result, it was assumed these terms could be computed by ratioing aircraft mass and length to the mass and length of the YF-12. This procedure resulted in the following expressions for the first three structural modes:

$$\frac{\phi_1 (X_{\delta_e})}{m_1} = \frac{2.56 \times 10^5}{mL} \qquad (5.1a)$$

$$\frac{\phi_2 (X_{\delta_e})}{m_2} = \frac{-13.6 \times 10^5}{mL} \qquad (5.1b)$$

$$\frac{\phi_3 (X_{\delta_e})}{m_3} = \frac{1.64 \times 10^5}{mL} \qquad (5.1c)$$

where m is expressed in lb-sec^2/ft and L in inches.

Frequency responses of the system of Figure 5-3 with the loop opened at the actuator output were computed using the aeroelastic model and compared against similar responses using the F-4 flexible-aircraft data[2]. With the relatively high frequencies of the F-4, the third mode is well beyond frequencies of interest. Hence, no comparison was attempted for the third mode. To obtain a comprehensive comparison, the responses were compared at three different rate sensor locations on the aircraft. The stations selected for the comparison were at 30, 50, and 70 percent of the length measured from the aircraft nose. These comparisons are shown in Figures 5-6, 5-7, and 5-8. Surface inertia effects were excluded for the purposes of

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[2] The control system was included in this comparison to provide a realistic combination of rate gyro and accelerometer bending contributions. Because the control system shaping amplifies the rate signal (or attenuates the acceleration signal) at the higher frequencies, the bending effects are primarily due to rate gyro pickup for most sensor locations used in actual aircraft.

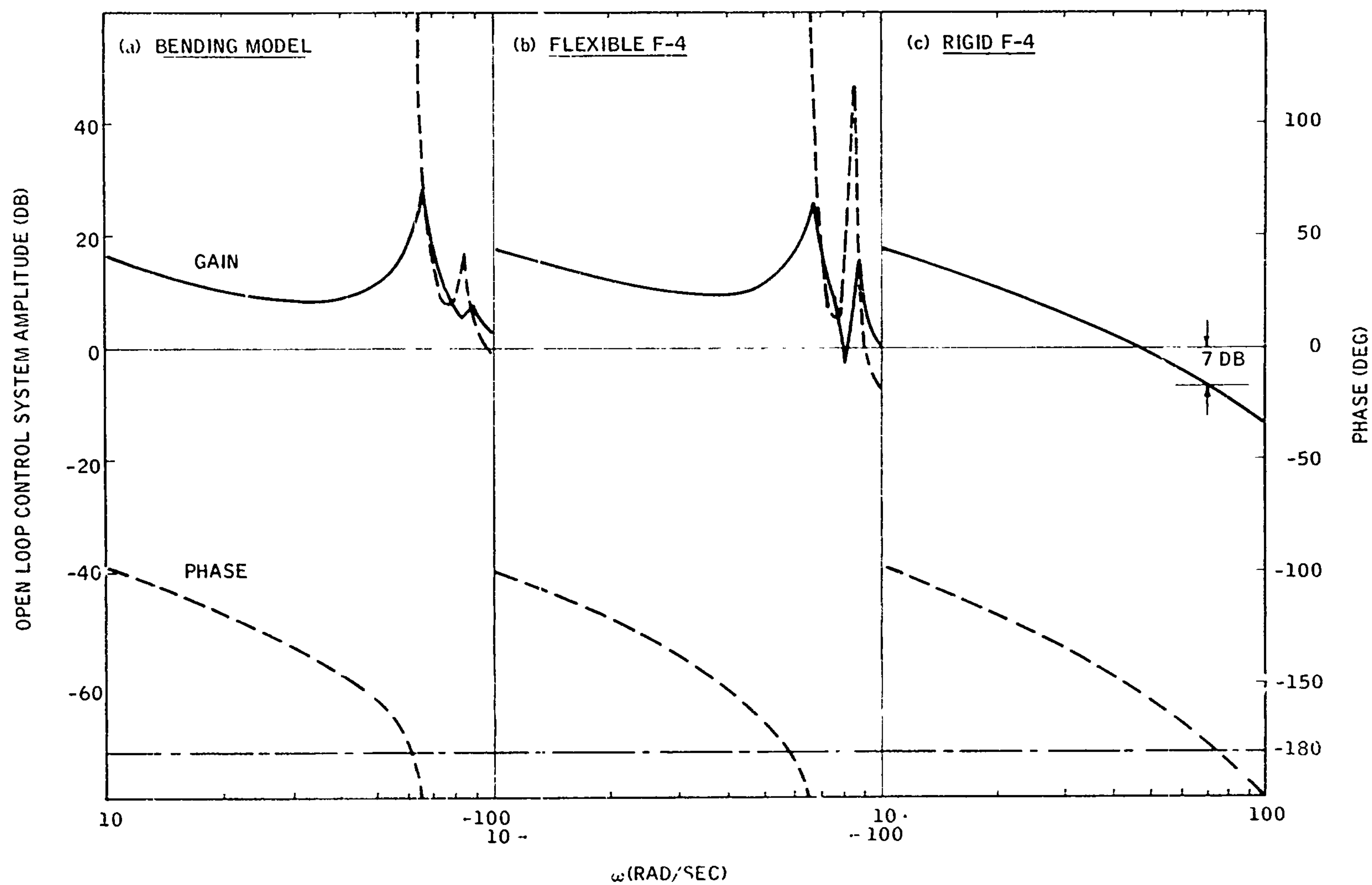


Figure 5-6. Comparison of Bending Model and F-4 Aircraft at 0.30 L

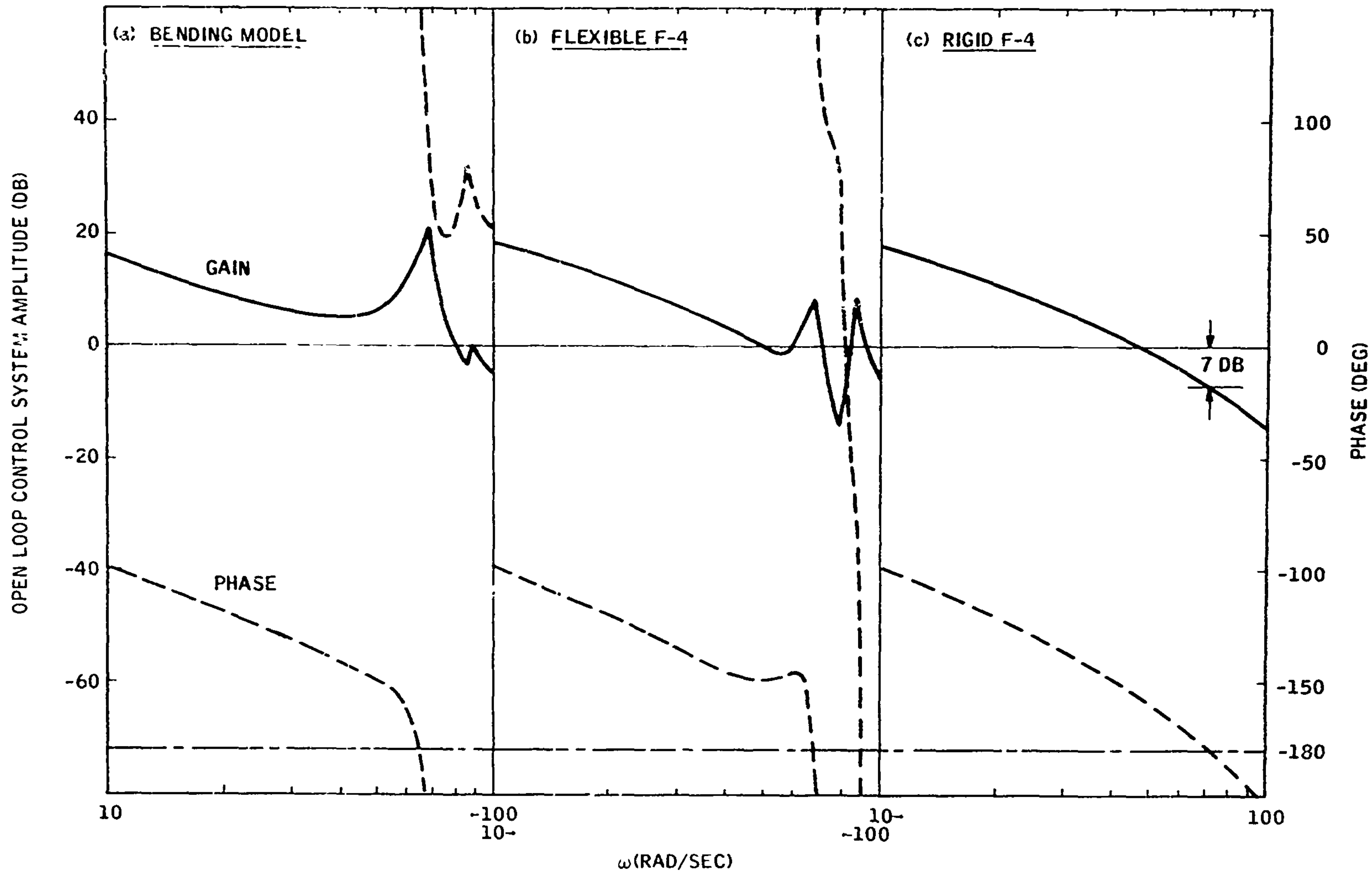


Figure 5-7. Comparison of Bending Model and F-4 Aircraft at 0.50 L

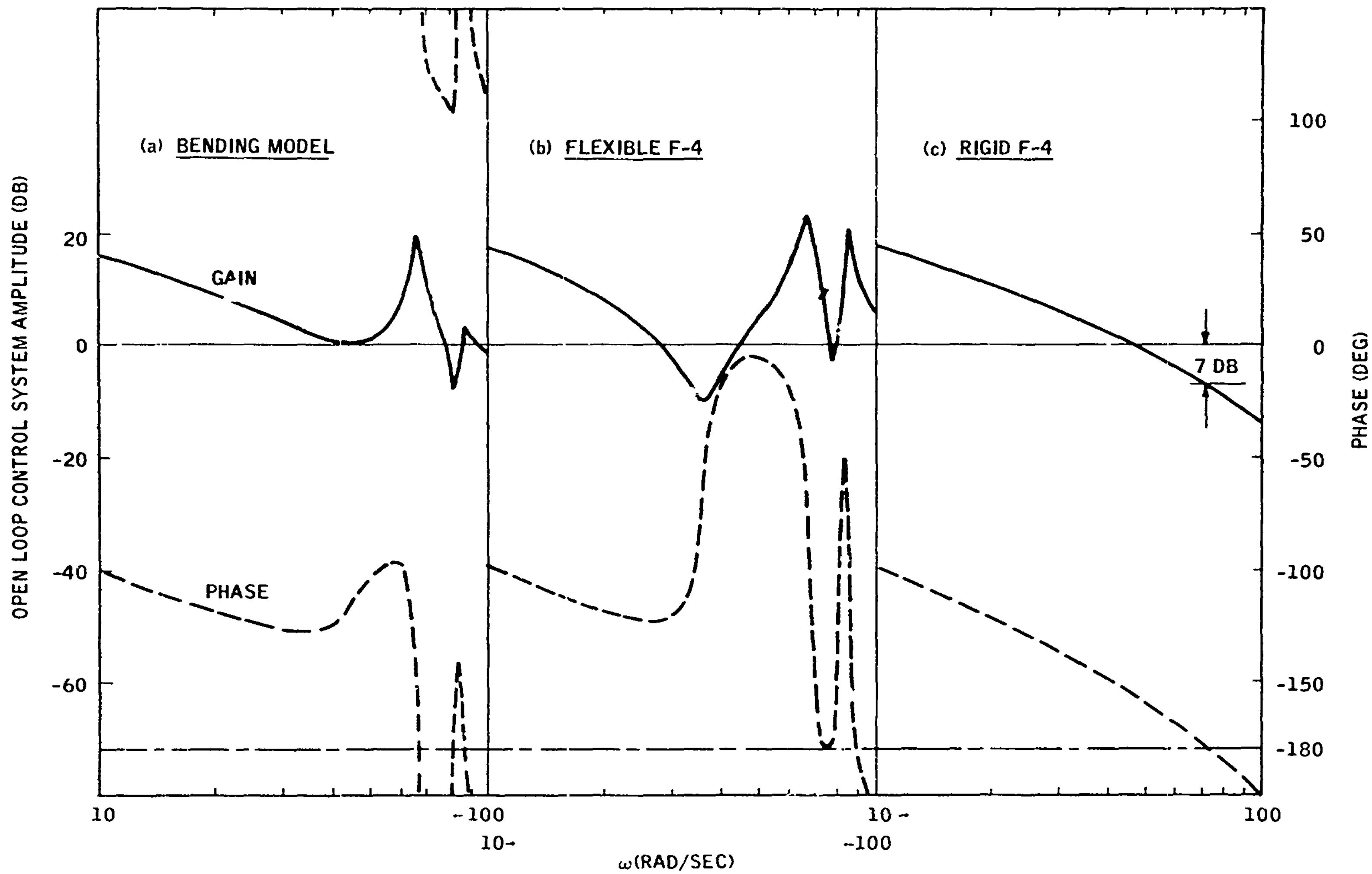


Figure 5-8. Comparison of Bending Model and F-4 Aircraft at 0.70 L

this comparison. Comparison of the model and actual F-4 aircraft responses shows reasonable agreement for the first aeroelastic mode except at 0. 5 L. At this station the model shows a higher first-mode peak amplitude. The reason for this difference is attributed to the fact that the first-mode antinode occurs at 0. 5 L on the F-4 aircraft. Hence, the sensed aeroelastic-mode signal is actually smaller than the model indicates it should be.

The rigid-body F-4 response is also shown in these three figures for comparison. The three responses obtained in each of the figures were obtained using identical control system gains and shaping. Hence, the three responses can be compared in each figure to determine the maximum acceptable gain levels each would require. The rigid-body response shows a gain margin of 7 db. If the same gain margin requirement is applied to the two flexible aircraft responses, the control gains would have to be reduced as shown in Table 5-1.

Table 5-1. Required Control-Gain Reductions

Rate Sensor Station	Rigid Aircraft	Flexible F-4	Aeroelastic Model
0. 30 L	No reduction	-25 db	-28 db
0. 50 L	No reduction	-15 db	-25 db
0. 70 L	No reduction	-20 db	-23 db

Based on the results in Table 5-1, it appears that the aeroelastic model would give a fairly good indication of stability. The largest discrepancy occurs at 0. 50 L (first-mode antinode) where use of the model would establish a gain 10 db lower than necessary. However, use of the rigid response without consideration of bending would result in a too high a system gain at all the locations.

It should be pointed out that the aeroelastic model described above was based on a rather limited set of data. Hence, its general validity has not been well established. It is recognized that the model will need further refinement. Nevertheless, it can be concluded from the study that such a model could offer the designer a valuable tool.

Rigid-Body Properties

The so called "rigid-body" stability margins were established by performing a parameter variations analysis on the elements shown in Figure 5-3. The resulting deviations in gain and phase were root-sum-squared at each frequency point to determine the required margins. The following parameter variations were computed:

Rate gyro natural frequency	±10%
Rate gyro damping ratio	±40%
Rate gyro d-c gain	±10%
Actuator time constant	±20%
Actuator d-c gain	±10%
Series servo natural frequency	±10%
Series servo damping ratio	±20%
Series servo d-c gain	±10%
Pitch rate feedback gain	±10%
T_1, T_2, T_3, T_4	±20%
Bending filter natural frequency	±20%
Bending filter d-c gain	±10%
Accelerometer feedback gain	±10%
Aircraft surface effectiveness	±50%

The above values were estimated based on current analog hardware and design practices applied to military aircraft environments. The tabulated parameters do not in general include effects that become significant above 100 rad/sec (such as compliance of the actuator due to compressibility of fluid and structure, an effect which contributes a pair of lightly damped poles at 182 rad/sec for the F-4 stabilator per Reference 5-2). Consequently, tolerances were not computed above 100 rad/sec, a frequency range beyond interest for flight control system stability. Control surface inertial effects are also excluded (which, strictly speaking, are also "rigid-body" effects). These will vary with frequency and will be discussed later.

Figure 5-9 shows the resulting rss'd gain and phase deviations as a function of frequency. As expected, the required margins increase somewhat with frequency.

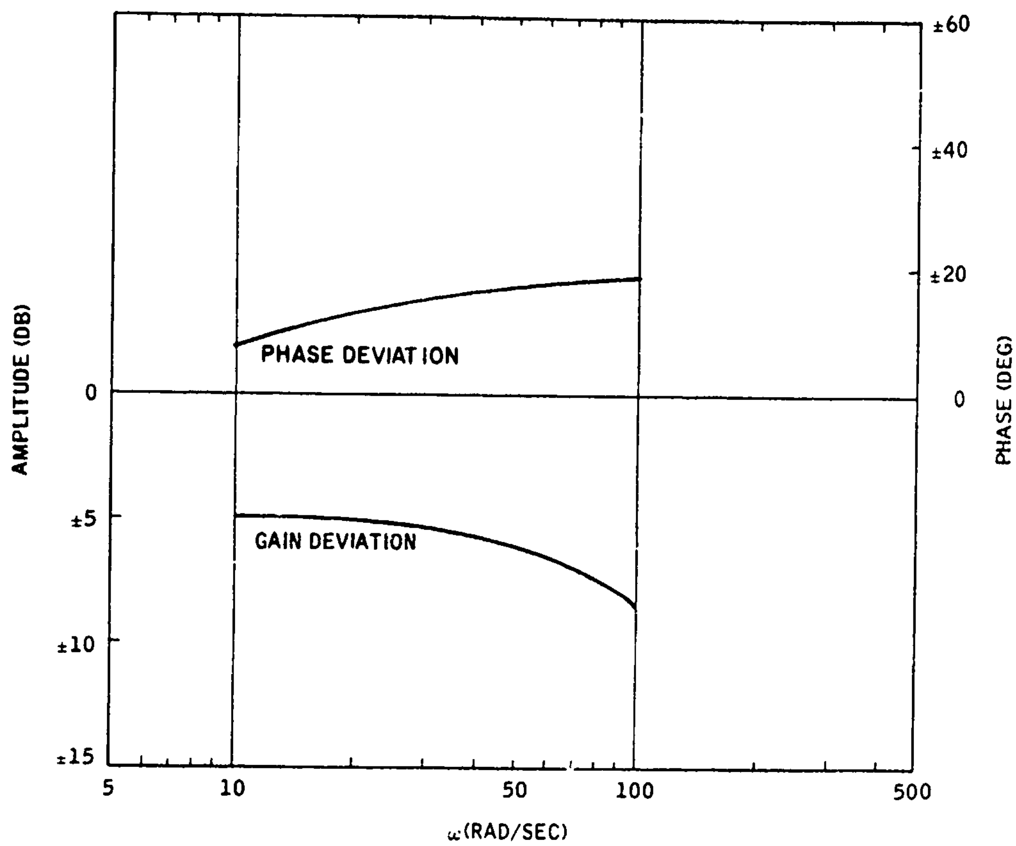


Figure 5-9. Pitch-Axis Rigid-Body Uncertainties

The data of Figure 5-9 indicate that commonly used stability margins of 6 db and 45 degrees are reasonably valid criteria for control loops subject primarily to rigid body tolerances of the sort included in the analysis, at least up to the frequencies of 60 to 70 rad/sec. Other error sources will further expand these uncertainties, as will become evident in the following paragraphs.

Inertial Forces Due to Surface Deflection

The general aeroelastic model shown in Figure 5-4 includes an added transfer function which modifies the control surface deflection input with the operation:

$$1 \pm \frac{S^2}{\omega^2_{TWD}}$$

This factor is intended to account for the mass imbalance of the surface which results in inertial forces in proportion to the surface acceleration. Inertial moments will be applied even with no imbalance due to rotational inertial torques, but this effect appears negligible for aerodynamic control surfaces. The sign option shown for the inertial term is dependent on the surface center of gravity location relative to the hinge line. If the inertial forces act in the same direction as the aerodynamic, a phase lead effect results, calling for the positive sign. If the inertial force acts in opposition to the aerodynamic, the negative sign is used. For conventional surfaces such as ailerons, spoilers, rudders, and elevators, therefore, the sign is positive if the cg is aft of the hinge line. The frequency term, ω_{TWD}, is commonly referred to as the "tail-wags-dog" frequency for obvious reasons. A function of the surface inertial and aerodynamic properties, it may be derived as follows.

Consider the sketch of the F-4 stabilator control shown in Figure 5-10. When the stabilator is deflected, vertical forces are generated due to aerodynamic and inertial forces. The former, taken from Reference 5-1, are tabulated in Figure 5-10. The latter are primarily due to acceleration of the stabilator cg, which includes actuator and linkage masses. The total forces generated by the stabilator are given by

$$F_s = \left(F_a - \frac{W_s \ell_s S^2}{g} \right) \delta_e \tag{5.2}$$

where the inertial component is taken negative (as shown) when the stabilator cg is aft of the hinge. Values for ℓ_s and W_s were determined from verbal discussions with MCAIR corresponding to alternate stabilator designs, also defined by Figure 5-10. Note that ℓ_s is the cg offset parallel to the aircraft longitudinal axis.

Equation (5.2) may be alternately expressed in terms of a frequency

$$F_s = F_a \left(1 + \frac{S^2}{\omega^2_{TWD}} \right)$$

where

$$\omega_{TWD} = \sqrt{\frac{g\,F_a}{W_s \ell_s}} \text{ rad/sec}$$

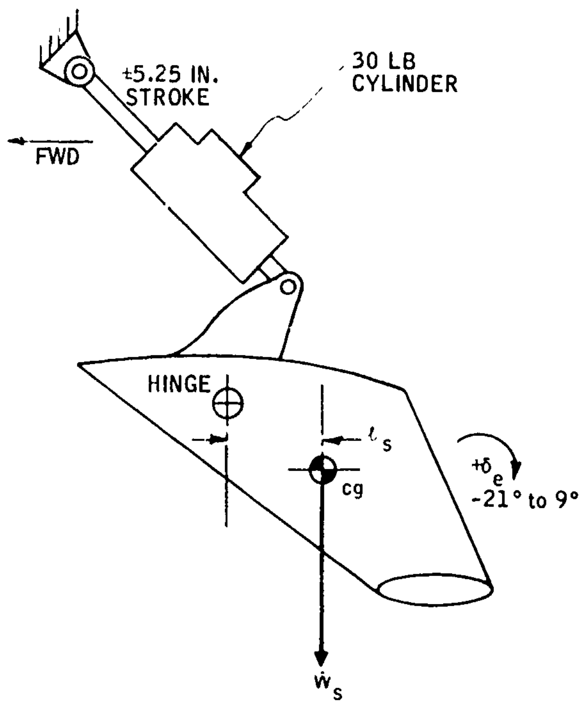
or

$$f_{TWD} = \frac{1}{2\pi} \sqrt{\frac{g\,F_a}{W_s \ell_s}}$$

This frequency has been computed for each of the conditions shown in Figure 5-10 and plotted in Figure 5-11. The fact that the TWD frequencies vary through the bending-mode range over the flight range of the airplane indicates that the phasing of the feedback signals associated with bending will vary by 180 degrees. This makes control of phase difficult, over the frequency range of the TWD dynamics, and indicates that stabilization based on phase rather than attenuation is also difficult and likely to be impractical.

It is interesting to note that since the TWD frequencies are zero at zero airspeed (no aerodynamic forces), the situation wherein transition of bending frequencies occur is common to all aircraft. With better balancing or less mass of the control surface, this crossing of the lower frequencies could, however, occur during the takeoff roll rather than inflight as for the F-4.

The "tail-wags-dog" effect can have significant tolerance implications on the control loop dynamics. The uncertainty is with the tail-wags dog frequency (ω_{TWD}) which is subject to variation with aerodynamic effectiveness of the control surface, the major source of error. Given a ±6-db tolerance on the latter, a variation in frequency (and hence phase around the frequency) as shown in Figure 5-12 will result. No contribution to gain error is attributed to surface inertia, since definition of the inertial properties of the surface should be at least as accurate as aerodynamic definition, and the latter (e. g., M_{δ_e}) has already been included in the "rigid-body" tolerance accumulation of Figure 5-9. The phase uncertainty around ω_{TWD} precludes phase stabilization in this range. Recalling the variation in ω_{TWD} over the flight envelope (Figure 5-11), phase stabilization is likely to be impractical for any frequency above the minimum value of ω_{TWD} unless special effort is made to identify ω_{TWD} as a function of flight condition. For the F-4 a minimum ω_{TWD} of about 9 Hz has been computed, resulting in preclusion of phase stabilization above about 6. 4 Hz after tolerances.



STAB. DESIGN	ℓ_s (FT)	W_s (LB)	I_{HINGE} (SLUG FT^2)
NONSLOTTED	0.299	676	112
SLOTTED	0.178	710	118

MACH	ALT (FT)	AERO NORMAL FORCE, (LB/RAD x 10^{-5})
0.206	SL	-0.610
0.318	SL	-0.403
0.500	5K	-0.607
0.500	25K	-0.267
0.700	40K	-0.240
0.900	45K	-0.294
1.200	45K	-0.491
1.500	35K	-0.920
1.500	40K	-0.569
1.600	40K	-0.777
2.150	45K	-0.656
1.800	55K	-0.371
0.840	SL	-1.830
0.850	5K	-1.550
0.850	25K	-0.690
0.900	15K	-1.157
0.900	35K	-0.486
1.200	35K	-0.795
1.200	5K	-2.830
1.200	15K	-1.920
1.500	15K	-2.210
1.500	25K	-1.455
2.150	36K	-1.012

Figure 5-10. F-4 Stabilizer Data

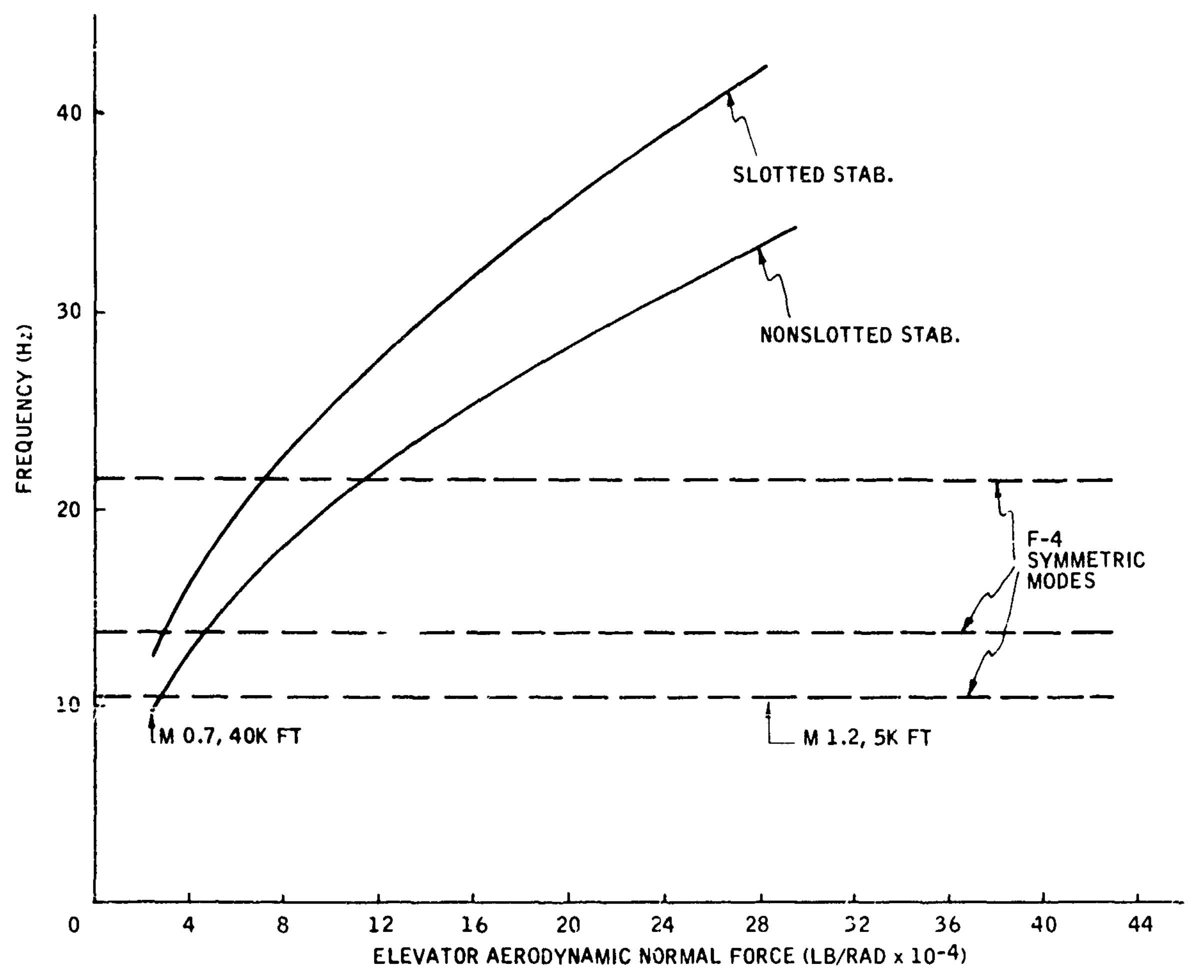


Figure 5-11. F-4 Tail-Wags-Dog Frequencies (Pitch Axis)

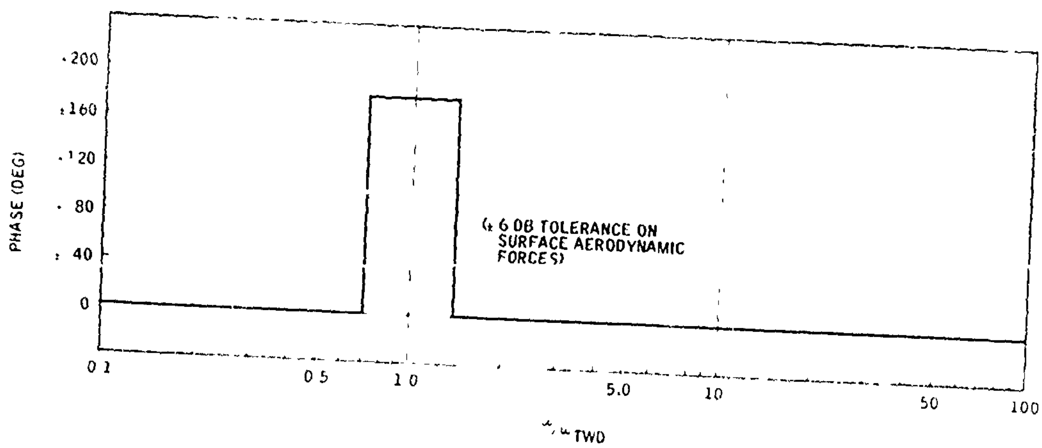


Figure 5-12. Phase Uncertainty Due to Surface Inertia Forces

Aeroelastic Properties

The effect of structural flexibility on potential stability margins was analyzed using the YF-12 flexible-aircraft data together with the fly-by-wire system shown in Figure 5-2. The magnitude of the coupling between the control system and the aeroelastic modes is a function of system sensor location and, for a given set of sensor locations, is also a function of the aeroelastic-mode parameters. Hence, the analysis of the flexible-aircraft data was performed in two steps:

(1) The amplitudes of the aeroelastic modes were computed as a function of sensor location;

(2) For a given set of sensor locations, an aeroelastic-mode parameter variation analysis was performed to determine the variations in the aeroelastic-mode gain and phase as a function of frequency.

The results of these two steps may be combined to determine how the "rigid-body" gain uncertainties (shown in Figure 5-9) should be increased to assure gain stabilization of the aeroelastic modes. Gain margin requirements based on these uncertainties would be used in the case where no aeroelastic-mode data or aeroelastic-mode model is available. No modification of the "rigid-body" phase allowance is necessary in this case.

The results of step (2) can also be used to establish required gain and phase margins for the case where an aeroelastic mode model or actual aeroelastic data is available. In this case, it may be possible to provide a combination of gain and phase stabilization of the aeroelastic modes since the modes themselves are defined. Hence, it is apparent that two alternate design procedures evolve:

- A procedure involving gain stabilization of the aeroelastic modes based upon knowledge of only a "rigid-body" frequency response (i.e., no aeroelastic data or aeroelastic-mode model available);

- A procedure applicable to gain and/or phase stabilization of aeroelastic modes based on using an aeroelastic-mode model or actual aeroelastic-mode data.

Consider first the details of the first situation (i.e., for the case where only a rigid-body model is used). The aeroelastic-mode amplitudes were computed as a function of sensor location by varying the positions about their nominal locations. A nominal rate gyro location was selected to correspond to the existing F-4 aircraft rate gyro location. This location is near the wing root at approximately 0.58 L. The nominal accelerometer location was selected to be in the forward portion of the data-link equipment bay at approximately 0.365 L. The data-link bay is immediately behind the aft cockpit. The sensor locations were varied ± 0.1 L about their nominal locations. These ranges were considered to represent the range of most likely locations for each of the sensors. The range on the accelerometer location assumes the accelerometer could be placed anywhere in the aft cockpit or data-link equipment bay Space limitations preclude any other locations. The range of possible rate gyro locations correspond to the most likely locations where a gyro would be placed in an attempt to minimize lower-frequency aeroelastic-mode pickup. These ranges of sensor locations are also reasonable for the YF-12 aircraft. In fact, the nominal rate gyro location on the YF-12 is essentially at the same corresponding location in terms of percent of fuselage length. Neither aircraft use a normal accelerometer in their standard control systems.

Variations in the aeroelastic-mode amplitudes were computed as a function of sensor location using the YF-12 flexible-aircraft data. The results obtained showed that variations in the accelerometer location over its range had a negligible effect upon the response amplitudes when compared to changes in rate gyro location. Figures 5-13, 5-14 and 5-15 show the effect of variations in rate gyro location. These responses of pitch rate to elevator deflection show that the critical response amplification (i.e., where the phase equals 180 degrees) remains a relatively constant 25 db with respect to the "rigid-body" response. These results indicate (neglecting variations in the aeroelastic-mode parameters for the moment) that gain stabilization of the aeroelastic modes could be assured by simply increasing the rigid-body gain uncertainty by 25 db over the frequency range of the aeroelastic modes. To be complete, however, some additional uncertainty must be allowed to account for variations in the aeroelastic-mode parameters for a given set of sensor locations. These data were obtained from the second step of the analysis as described in the following paragraphs.

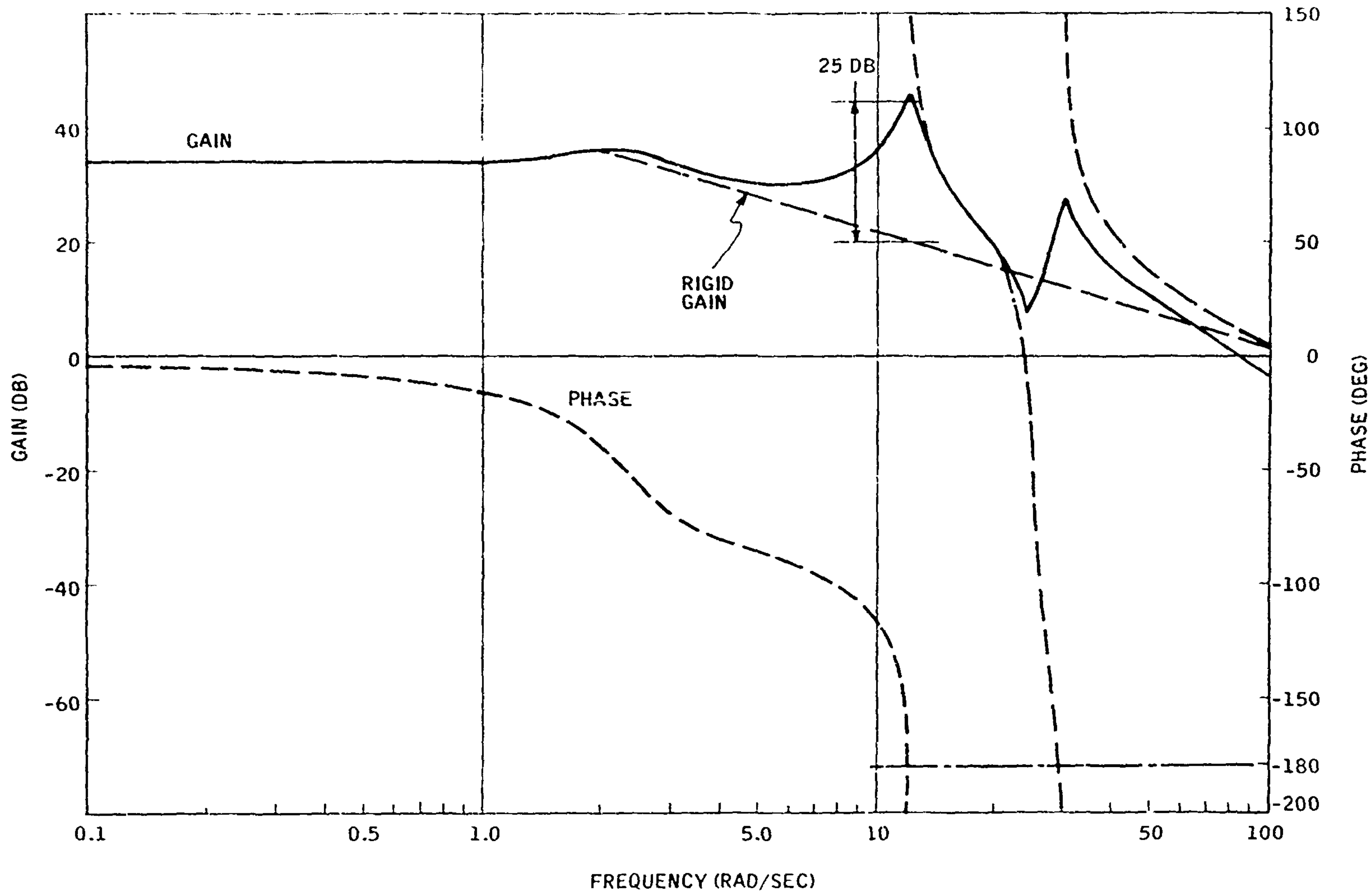


Figure 5-13. Open-Loop Frequency Response, Sensed Pitch Rate to Elevator, YF-12 Aircraft - Rate Sensor Location at 0.48 L

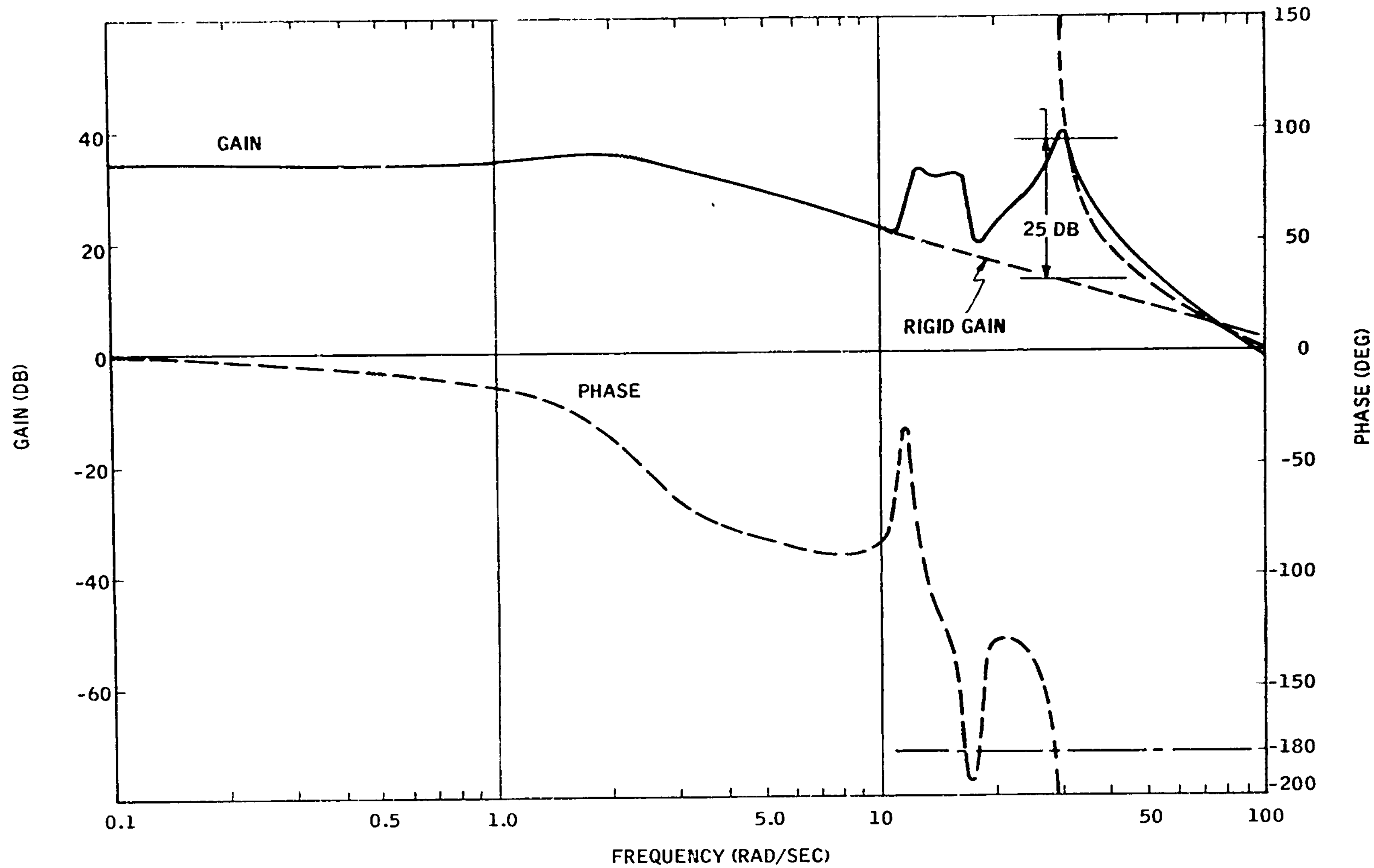


Figure 5-14. Open-Lo' 'requency Response, Sensed Pitch Rate to Elevator, .F-12 Aircraft - Rate Sensor Location at 0.58 L

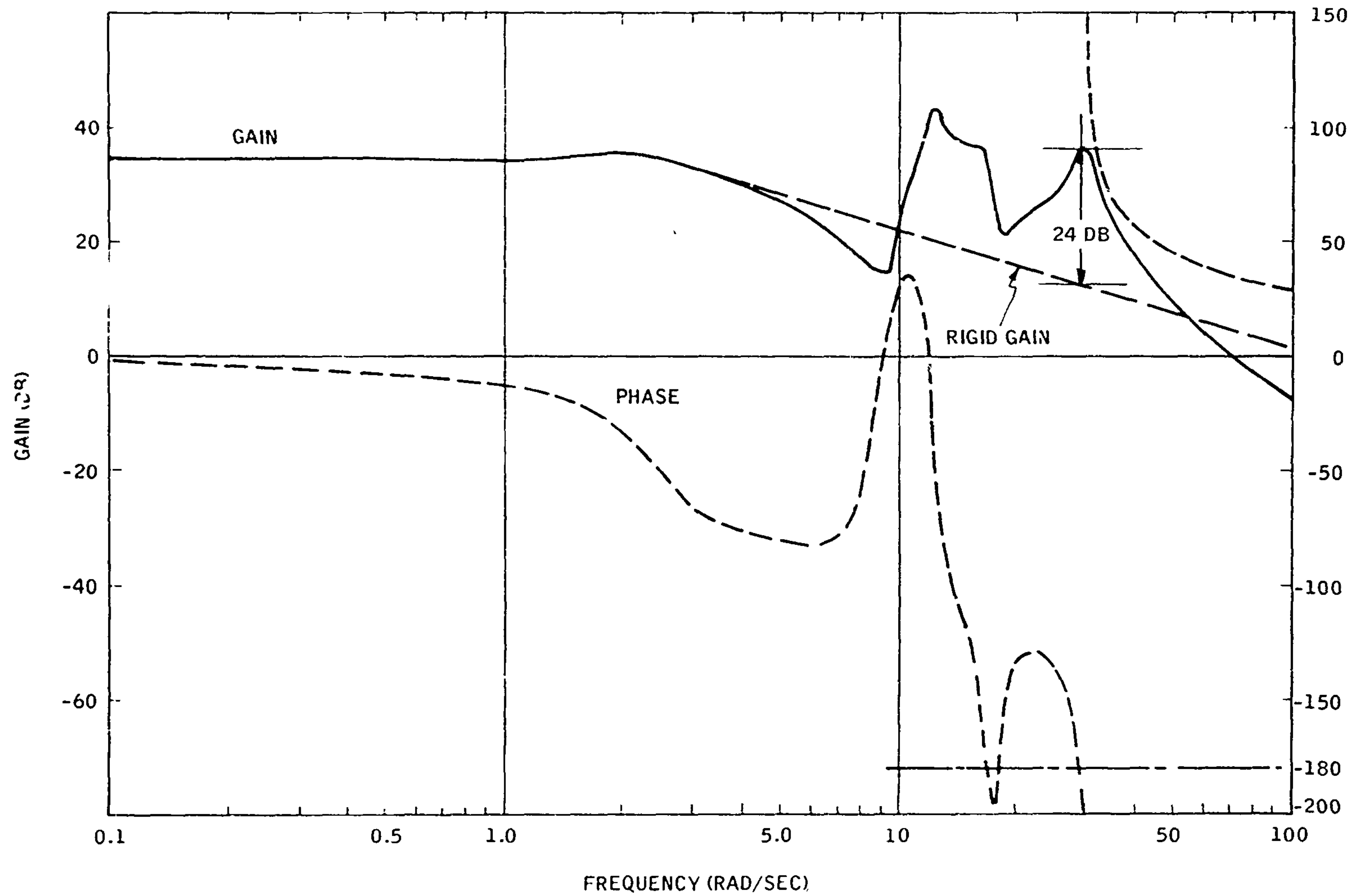


Figure 5-15. Open-Loop Frequency Response, Sensed Pitch Rate to Elevator, YF-12 Aircraft - Rate Sensor Location at 0.68 L

Amplitude and phase uncertainties were computed for variations in aeroelastic-mode parameters for the nominal sensor locations. These uncertainties were computed as a function of the following aeroelastic-mode parameter variations using the YF-12 aircraft data:

Mode shapes (all three modes):	+ 100%, - 50%
Mode slopes (all three modes):	+ 100%, - 50%
Aeroelastic-mode control force coefficients (all three modes):	+ 50%, - 50%
Aeroelastic-mode frequencies (all three modes):	± 10%
Aeroelastic-mode damping ratio (all three modes):	± 50%

The magnitudes of the variations were assumed as being representative of the uncertainty which exists in these data. Figure 5-16 shows the additional phase and gain allowances as a function of frequency required to accommodate these variations. The frequency has been normalized with respect to the lowest aeroelastic frequency. The plot of phase uncertainty shown in Figure 5-16 actually represents an envelope of phase uncertainty. Actual computed phase uncertainties are shown at a number of frequency points. As can be seen from Figure 5-16, there is considerable scatter in the phase-uncertainty data with frequency, especially near the aeroelastic-mode natural frequency points. This scatter is expected because of the steep slope characteristic of the phase-angle frequency response in the range of the aeroelastic-mode frequencies.

The data shown in Figure 5-16 must be used in conjunction with the other sources of phase and gain uncertainty developed previously. The combination of these data provide the elements necessary to evaluate high-frequency control-loop stability. Figure 5-17 is a block diagram of how these data are to be combined as a function of the design objective (e.g., gain or phase stabilization) and the available aircraft data. Sample applications are described in the next subsection.

It is evident that the motivation for phase stabilization of bending modes varies widely among specific aircraft applications. For the F-4 with a minimum bending frequency of about 10 Hz, the bending problem is one of achieving good "rigid-body" control around 2 Hz. This spread in frequencies facilitates the application of filtering which can achieve a suitable compromise between low-frequency phase lag and high-frequency attenuation. The YF-12, conversely, with a minimum aeroelastic frequency of 2 Hz, does not enjoy a comparable separation in control modes, making phase stabilization mandatory. Fortunately, a reduced bending-frequency range is accompanied by better knowledge of phase around the bending modes, tending to make phase stabilization feasible.

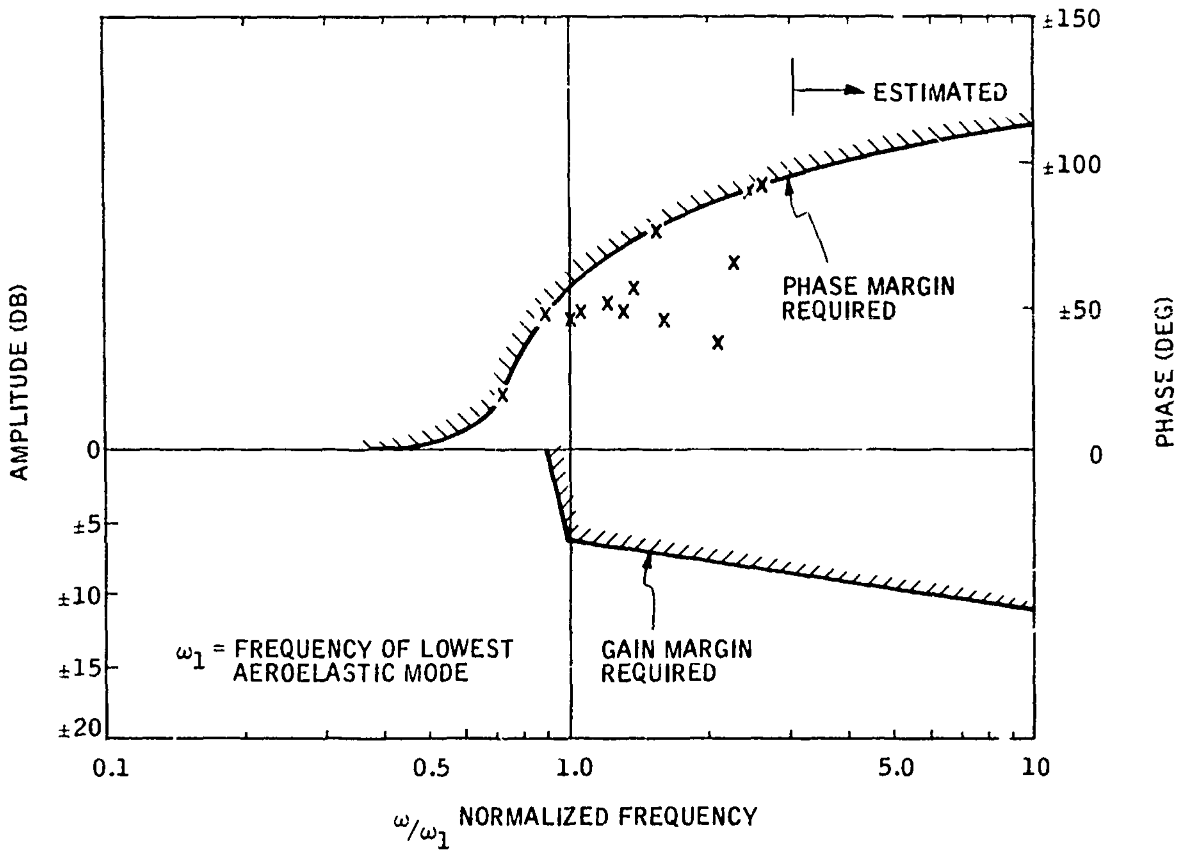


Figure 5-16. Stability Margins Required to Accommodate Flexure Tolerances

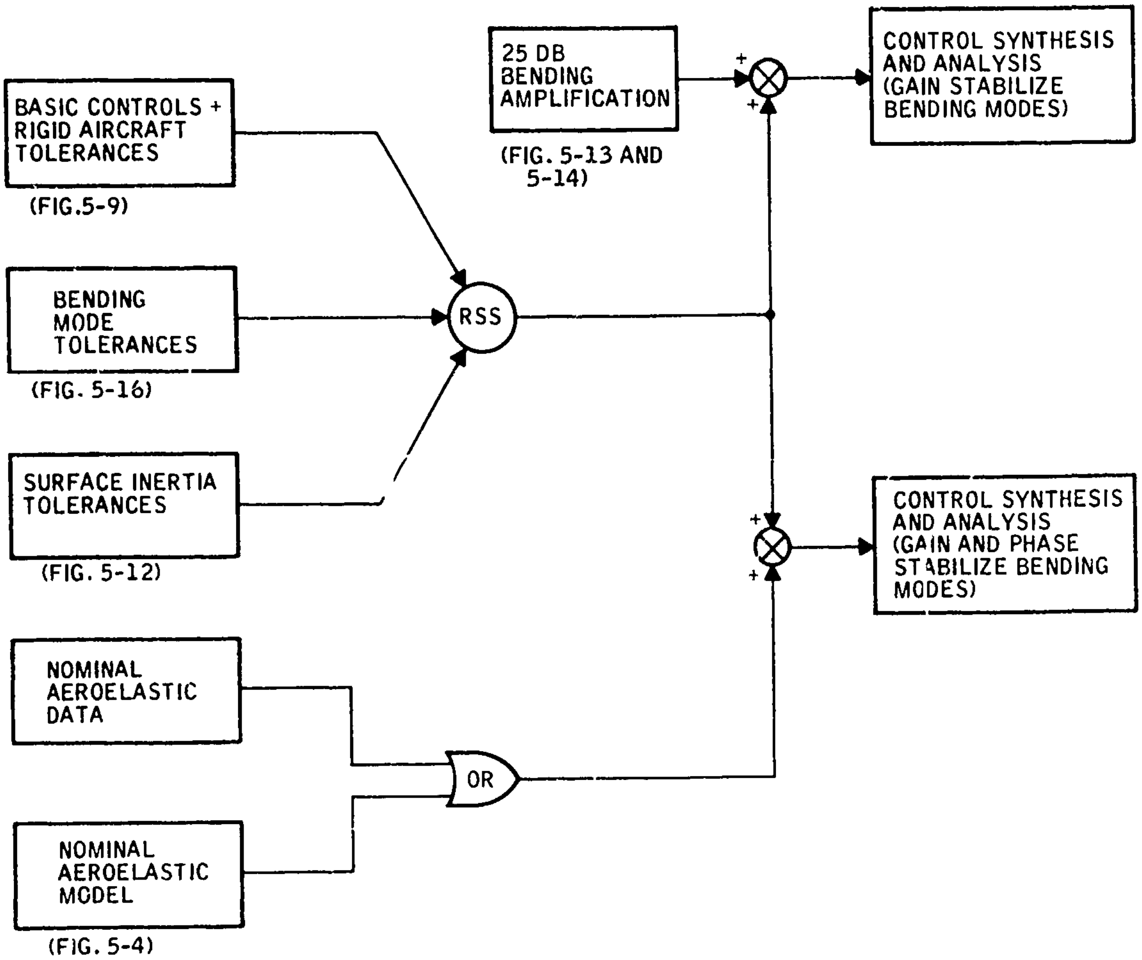


Figure 5-17. High-Frequency Stabilization Criteria

Sample Applications

First consider the case where no aeroelastic-mode model is available. The stability margins required to assure gain stabilization of the aeroelastic modes are obtained by a root sum square (over the frequency range of the aeroelastic modes) of the gain uncertainties of Figures 5-9 and 5-16 and addition of the result to the -25-db margin required for nominal bending amplification. The net result is shown in Figure 5-18 for the F-4 aircraft.

This constitutes the procedure to be followed if no flexible-aircraft model is available. The reader should be reminded that the phase margin requirement of Figure 5-18 reflects the margin required to account for uncertainties. A larger margin may be required in an actual application to assure "adequate" damping.

Now consider the alternate case where an aeroelastic mode model is available. In this case, the -25-db attenuation requirement is replaced by by the aeroelastic-model characteristics. The gain and phase margins required to account for aeroelastic-mode parameter variations are assumed to also be applicable to a aeroelastic-mode model or to actual aeroelastic-mode data. Hence, the stability margins to be used are obtained by combining the "rigid-body" margins of Figure 5-9 with the "aeroelastic-mode" margins of Figure 5-16. This task is done in the same manner as for the case without an aeroelastic model except that now the phase margins must also be combined in an rss manner. As an example, Figure 5-19 shows the margins which would be required for the F-4 aircraft.

To summarize at this point, we have established two sets of stability margin requirements, one for use without an aeroelastic-mode model and one for use with an aeroelastic-mode model. An analysis was performed using these two sets of requirements to determine if an aeroelastic-mode model would provide any significant benefit over not using a model. The two approaches were compared for the F-4 cases shown in Figures 5-6, 5-7, and 5-8. Table 5-2 summarizes the results. This table shows the reduction in open-loop gain required to meet the gain-margin requirement for the three cases: (1) actual F-4 flexible-aircraft data is available; (2) an aeroelastic-mode model is available; and (3) the rigid-body response is used.

The data in this table indicates that the aeroelastic-mode model specifies a gain level between the level actually required (i.e., based on actual F-4 flexible-aircraft data) and the gain level required to accommodate unknown flexure effects. For the forward and aft sensor locations, the use of the model resulted in a gain level within 3 db of the maximum allowable. In the middle case, where the sensor is located at the first-mode antinode, it requires a gain level 10 db lower than necessary. In this case, however, the use of only rigid-body data would result in a gain level 17 db lower than the maximum allowable. These results indicated that the use of an aeroelastic-mode model will produce more realistic gain levels than will the use of only the rigid-body

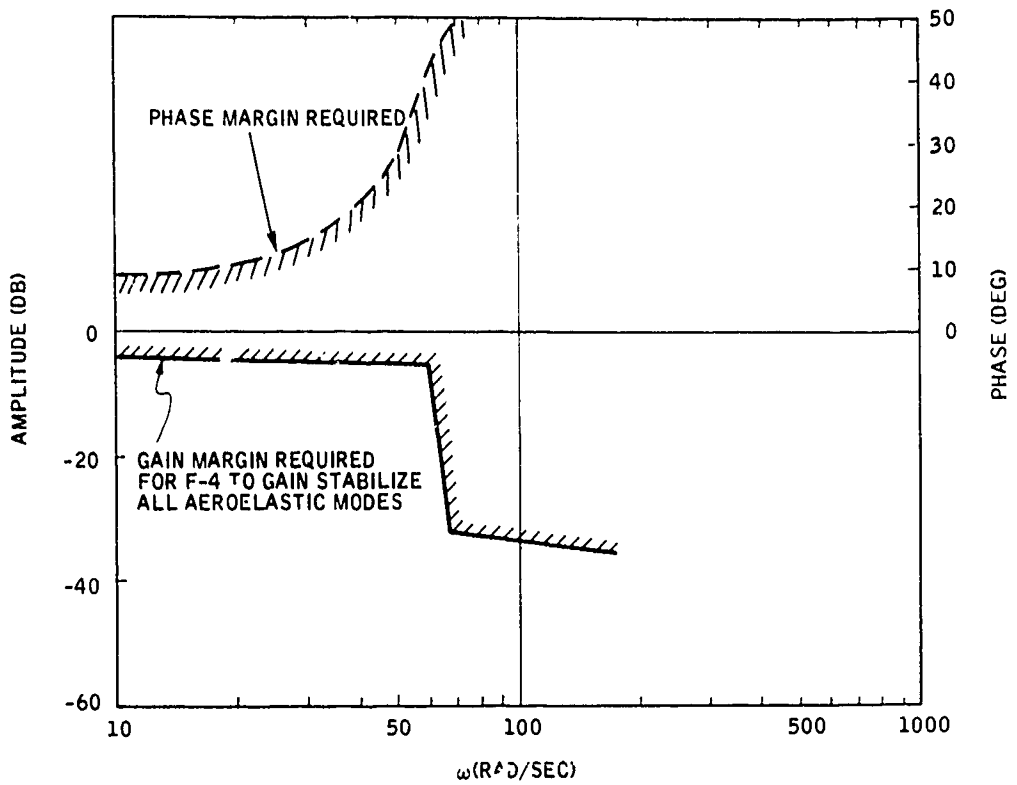


Figure 5-18. Required Stability Margins for F-4 Aircraft System Rigid-Body Open-Loop Response

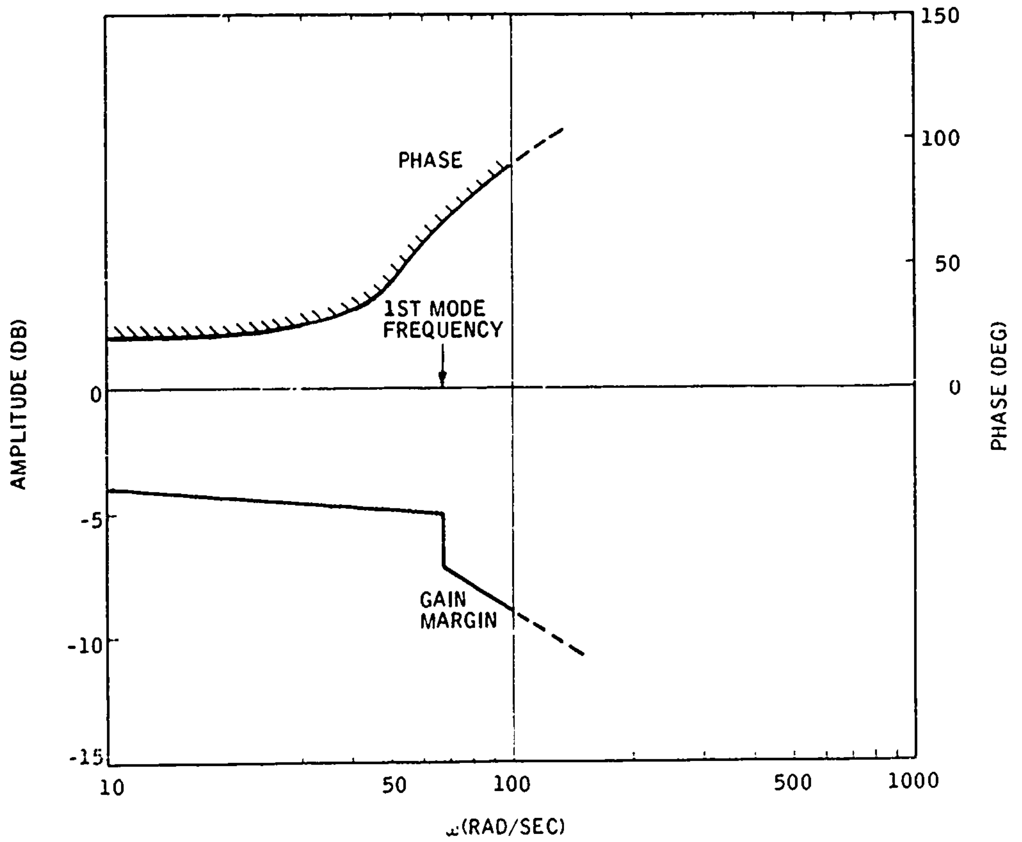


Figure 5-19. Stability Margins Required for Use with Aeroelastic Mode Model on the F-4 Aircraft

Table 5-2. Gain Reduction Required to Accommodate Aeroelastic Modes as a Function of Rate Sensor Location

Rate Sensor Location	F-4 Flexible-Aircraft Data	Structural Mode Model	F-4 Rigid-Body Data Only
0.30 L	- 25 db	- 28 db	- 32 db
0.50 L	- 25	- 25	- 32
0.70 L	- 20	- 23	- 32

response. The results also indicate that if the aeroelastic model mode shapes could be better adapted to the subject aircraft, the gain levels could be more accurately predicted.

SUMMARY AND CONCLUSIONS

This study addressed the problem of specifying design criteria for high-frequency compensation in closed-loop primary control systems. To date, adequate criteria has been precluded primarily by the lack of definition of aircraft dynamics in the high-frequency range (i.e., frequencies above the short-period or dutch-roll range). These higher-frequency dynamics include aeroelastic modes, tail-wag-dog properties, and actuator compliance. Of these, the lack of definition of the aeroelastic modes has been the greatest problem for the flight control designer. This lack of definition has inhibited both the determination of a nominal response characteristic and the establishment of adequate stability margins.

The study served to establish a set of design tools to provide a more accurate definition of the high-frequency compensation. These design tools consist of the following items:

(1) An analytical model of the first three symmetric aeroelastic modes;

(2) Sets of gain and phase uncertainties established as a function of frequency;

(3) A procedure for establishing stability margins as a function of the aircraft data available, using the uncertainties provided in item (2).

These items are referred to as design tools rather than as a general requirement. This approach was taken because the above items were based on assumptions which will vary from application to application. The option should be left to the designer to alter any of these tools to more accurately fit his application. For example, it was pointed out in the study that the gain and phase uncertainties are dependent on such factors as the type of control hardware used (e.g., analog or digital) and the nature of the system (conventional or adaptive). The set of gain and phase uncertainties provided in item (2) are based on an assumed typical analog fly-by-wire control system and YF-12 aeroelastic mode data.

The aeroelastic model provided is to be used with rigid-body data when no better aeroelastic data is available. The model developed in the study was based on YF-12 aeroelastic data and evaluated using F-4 flexible-aircraft data. Lack of comparable quality data for a number of other aircraft, however, precluded a thorough testing of the accuracy of the model during the study. It is recommended that this model be further refined and verified in subsequent studies because it promises to be a useful design tool.

As indicated in item (3), the stability margins are to be established as a function of the aircraft data available for the analysis. A set of stability margins can be determined for each of the following cases:

- Only rigid-aircraft data is available, and the aeroelastic model provided herein is not used;
- Only rigid-body data is available, but the aeroelastic model is used;
- Both rigid and aeroelastic data are available for the design phase.

The study showed the major effect of tail-wag-dog (TWD) (i.e., control surface inertia) dynamics was to introduce a 180-degree phase uncertainty around the TWD frequency. This uncertainty precludes phase stabilization of the aeroelastic modes in the vicinity of the TWD frequency unless the TWD dynamics can be identified as a function of flight condition.

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SECTION VI

ANALYSIS OF STALL/SPIN MANEUVERS

This section describes the results of a study to examine the stall/spin control problem as it relates to high-authority closed-loop primary flight controls.

The value of development work on stall/spin control is unquestionable. The high aircraft losses due to upset, the need to utilize all available maneuvering capability of the aircraft, and the deficiencies of current warning devices are quite adequate justifications. Additional motivation is provided by the fact that little work has been done to date to determine the behavior or desired properties of a closed-loop PFCS near or in a stall/spin condition. Most analysis of these systems has been confined to the more normal flight regime.

The objective of the study described here was to develop control criteria for primary flight controls in the stall/spin flight regime. The study included preparation of a complete six-degree-of-freedom computer simulation of the F-4 spinning aircraft. The analysis was broken into two major areas -- study of the basic aircraft behavior and definition of control criteria. The study established the dominant factors affecting the stall and spin conditions and evaluated the effects of feedback control on the stall/spin behavior.

In the following subsections, the study approach is described in more detail; a description of the computer simulation is presented; results obtained from the basic aircraft studies are discussed; and the control criteria studies are described. A summary and the conclusions, including suggestions for future study, are also presented. A detailed description of the airplane mathematical model and associated data are presented in Appendix IV.

STUDY APPROACH

This subsection describes a general approach to the study of stalls and spins. Basic terminology and aircraft operating regions are established, and the stall/spin problem is categorized as it relates to flight controls. Specific study objectives and scope are then derived.

Definition of Terms

Describing aircraft motion for abnormal (e.g., stalls/spins) conditions requires a set of terminology not generally required by the flight controls designer who is usually concerned with normal flight conditions. This fact plus the fact that the definitions found in the literature tend to vary from airplane to airplane makes it desirable to at least reiterate the terminology, particularly as it appears in this report. The terminology presented here is taken directly from Reference 6-1. It is especially suited to the F-4 aircraft.

- Stall - Stall is defined as the angle of attack (AOA) for maximum usable lift at a given flight condition. This definition recognizes that AOA, not airspeed, is the key to stall. The phrase "maximum usable lift" indicates that there may be a greater lift potential beyond stall but that this lift is not usable for one or more reasons. Most often, severe longitudinal instability or a rapid breakdown in lateral-directional stability serves to limit the airplane to a selected AOA for stall. Attempts to use lift beyond this stall AOA invite loss of control. Many airplanes have artificial warning or control devices based on AOA that are designed to prevent the airplane from being stalled. If an airplane does not have a satisfactory natural stall warning or an acceptable artificial stall-warning/prevention device, the pilot must supply his own definition of stall through appropriate cues. Only in this fashion can the airplane be safely flown to its maximum capability.

- Wing Rock - Wing rock is defined as uncommanded lateral-directional motions, viewed by the pilot primarily as roll oscillations. These oscillations may vary in intensity from small perturbations that degrade precision tracking to an objectionably large dutch roll. The F-4 generally exhibits these characteristics at high AOA's.

- Nose Slice - Nose slice is defined as uncommanded lateral-directional motions viewed by the pilot primarily as an excursion in yaw. In the F-4 at high AOA, the yaw motions may oscillate but can diverge, resulting in a departure from controlled flight.

- Departure - Departure from controlled flight is defined as the first aircraft motions immediately following complete loss of control by the pilot. Departure is often called "pitch-up" in aircraft which have severe static longitudinal instability. For the F-4, departure is characterized by a nose slice that results in loss of control of the aircraft.

- Post-Stall Gyration - Post-stall gyration (PSG) is defined as uncontrollable motions about one or more aircraft axes following departure. For the F-4, PSG is usually a rapid roll after an initial yaw divergence and is referred to as a "rolling departure". However, during high-pitch-attitude entires with very low speeds, PSG may also be characterized by random motions about all axes.

- Spin - Spin is defined as a sustained yaw rotation at AOAs above stall (positive AOA for an erect spin, negative AOA for an inverted spin).

- Incipient Spin - Incipient spin is the initial stage of a spin in which an insufficient balance between aerodynamic and inertial moments exists to allow identification of the spin mode. Spin modes are identified by average values of angle of attack and yaw rate and by the magnitude of the three-axis oscillations.

- Developed Spin - In a developed spin there is a sufficient balance between aerodynamic and inertial moments to allow identification of the spin mode. For the F-4, a spin can become developed within the first turn. The spin is fully developed when the trajectory has become vertical such that the effect of gravity on the spin is constant and no significant change is noted in the spin characteristics from turn to turn.

Aircraft Operating Regions

Figure 6-1 shows the potential flight regions of a fighter aircraft. These regions all fall into either a normal or abnormal category, the latter being considered as encompassing various forms of stalled flight. The center region corresponds to normal flight, and it extends to the stall boundary. The latter represents the maximum usable maneuvering capability of the aircraft and is rarely penetrated except by accident. Combat aircraft often require extreme maneuvers, but the tactical value of an intentional stall is at best rare. The requirement for maximum maneuverability without upset suggests the need for some sort of stall inhibition.

The properties of "abnormal flight" may differ substantially between aircraft. Aerodynamic stall of the lifting surfaces is certainly one boundary to normal flight. Others may be due to static instabilities in pitch ("pitch-up") or yaw, or to dynamic instabilities of buffet or flutter. Angle of attack is at least an important factor in all of these instances, if not the dominant one.

The potential of a departure from normal flight into the stalled region must be recognized. This may occur intentionally under some peculiar tactical situation or accidentally due to an external force such as turbulence. To permit an intentional stall, an active stall inhibitor (i.e., one which physically constrains pilot action) must be capable of being overridden by conscious pilot effort.

The first area of the stalled region to be encountered on departure is that which can be recovered from with the conventional control surfaces. This region could include various modes of operation such as departure,

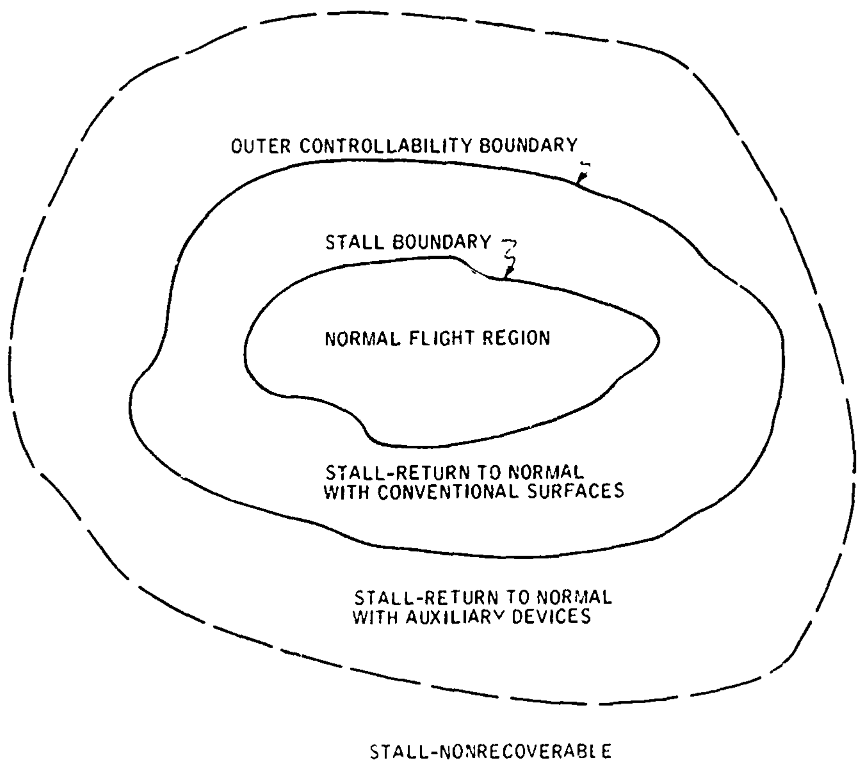


Figure 6-1. Aircraft Operating Regions

post-stall gyration, incipient spin or even a developed spin for some airplanes. The difference between these is somewhat academic when one recognizes that the point of concern is being able to recover normal flight. Consequently, this region is shown by Figure 6-1 as being restricted by an outer "controllability boundary". The controllability boundary is of course dependent on the control forces available.

The above discussion suggests that concepts of "prevention" and "recovery" are most suitably applied to the stall region itself rather than, for example, to spins. Certainly stall prevention also prevents spins, and provision of active stall inhibition would offer both tactical and safety advantages. Accepting the remote likelihood of desiring an intentional stall, the inhibitor would have override capability. Allowing this, the subsequent concern is for recovery from the stall region, be it from a controlled stall, an incipient spin, or a developed spin. The recovery means could be effected in two ways.

- By pilot option if within the recoverable zone
- By automatic engagement of a recovery system if penetration of the outer controllability boundary is eminent

Study Categorization

The study of the stall/spin problem can be categorized as shown in Figure 6-2. Basic to the study is a complete all-attitude six-degree-of-freedom simulation of the aircraft behavior in the stall/spin conditions. This simulation is essential for testing various notions or concepts related to basic aircraft performance and/or control techniques. The more conventional linearized small-perturbation techniques normally used in controls analysis, while still partially useful, are in themselves totally inadequate.

The study was divided into two main categories -- basic (aircraft) studies and control criteria. The basic aircraft studies are directed toward understanding the nature of the stall/spin phenomena. By studying the major influences, more effective use of flight controls and other aircraft elements can result. The control criteria studies are concerned with establishing flight control requirements using results obtained from the basic studies.

Study Objectives

The ultimate objective of the study of stall/spin maneuvers is to establish control criteria for future closed-loop flight controls. A complete achievement of this objective is beyond the scope of this study, which was directed primarily toward obtaining better understanding of aircraft behavior in stalled flight, both with and without the influence of feedback control. It is intended that the results obtained be used as a basis for further analysis in each of the areas described in the study categorization.

The specific objectives of the study were as follows:

- Identify dominant aerodynamics affecting the aircraft stall/spin behavior;
- Determine necessary aerodynamic control surface states for departure and recovery and associated feedback control effects;
- Develop analytical expressions which describe airplane dynamics in stalled flight;
- Develop analytical expressions which describe the controllability limits;
- Establish preferred feedback control laws for normal flight including stall inhibition;
- Establish preferred pitch-axis control laws for the stalled flight mode;
- Examine strategy for transition between normal and stalled flight and vice versa.

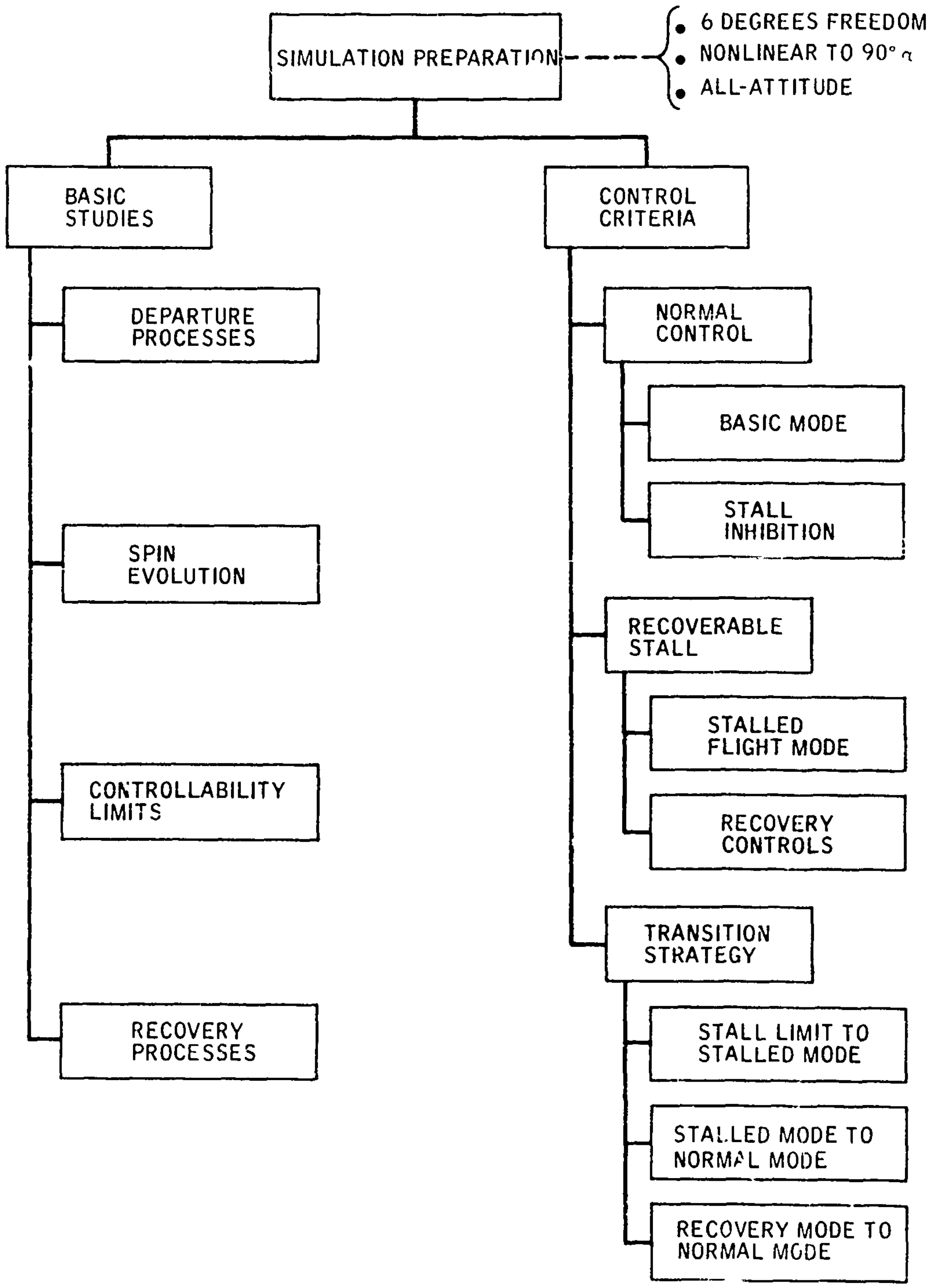


Figure 6-2. Stall/Spin Study Approach

Study Scope

The scope of the study was limited to a six-degree-of-freedom simulation analysis of an aerodynamically clean F-4 aircraft in low-speed flight. Variations in aircraft aerodynamic and inertial proportions were not considered, and controls were limited to existing aerodynamic control surfaces.

SIMULATION DESCRIPTION

This subsection describes the data sources, approximations, and assumptions used to establish the stall/spin computer simulation. Evaluation of the simulation adequacy is also discussed.

General Description

The simulation provided a six-degree-of-freedom, all-attitude representation of the dynamic performance of the F-4 aircraft in low-speed flight. The permitted range on angle of attack was from 0 to +90 degrees.

The simulation was performed on an H-1800 digital computer and was based on Honeywell's existing aircraft simulation software program, THRUST. The simulation was designed for use as either an all-digital simulation or as part of a hybrid analog-digital computer simulation employing the Sigma-5 digital computer facility.

Data Sources

The data used for the simulation was obtained from References 6-2 and 6-3. These reports describe a spin evaluation program performed by the McDonnell-Douglas Corporation for several F-4 aircraft configurations. The program involved the use of a full six-degree-of-freedom computer simulation which was substantiated by comparison with F-4 flight test time histories.

Aircraft Configuration

The airplane used in this study was the F-4D in a clean aerodynamic configuration. The center of gravity was fixed at 33 percent, and the gross weight was 43,000 pounds. This configuration has the following moments and products of inertia:

$$I_x = 28,800 \text{ slug-ft}^2$$

$$I_y = 133,100 \text{ slug-ft}^2$$

$I_z = 153,200 \text{ slug-ft}^2$

$I_{xz} = 4,400 \text{ slug-ft}^2$

Equations of Motion and Aerodynamic Data

The equations of motion, nomenclature, and aerodynamic data are described in detail in Appendix IV. The equations of motion include the effects of engine gyroscopic torques but neglect thrust misalignment, aerodynamic and structural assymetries, and mass variation effects.

The aerodynamic data used represents low-speed flight and includes rotary balance data for spinning flight. The nondimensional derivatives are defined in aircraft body axes. Essentially all these derivatives were approximated by analytic expressions for use in the simulation to enhance the simulation efficiency. These analytic expressions are also described in Appendix IV.

Simulation Verification

The simulation accuracy was checked by comparison with similar spin time histories obtained in the McDonnell-Douglas spin evaluation program (Ref. 6-2). An initial comparison of time histories is shown in Figure 6-3. This comparison was obtained by using control surface displacement time histories identical to those used by McDonnell-Douglas, but with engine angular momentum neglected and with substantial deviations in rolling moment coefficients. The engine momentum effect was subsequently shown to be negligible, but the moment coefficient was significant.

Figure 6-3 shows a significant difference in the roll-rate time history in the initial seconds of the developing spin. Examination of the simulated stability derivative, C_ℓ (the rolling moment coefficient) showed some differences with the McDonnell-Douglas data, particularly in the 20- to 30-degree angle-of-attack range. After making corrections in this approximation and adding engine angular momentum effects, the superior comparison shown in Figure 6-4 was obtained. The match shown in Figure 6-4 is considered acceptable for meeting the objectives of the stall/spin study.

Control System Configurations

"Normal control" is considered as those control laws which apply in unstalled flight and which may also include means for inhibiting exit from normal flight. The normal control mode of a feedback PFCS was investigated in the study primarily to analyze its operation in near-stall conditions and to establish transition strategy. The basic modes of this system are fairly well

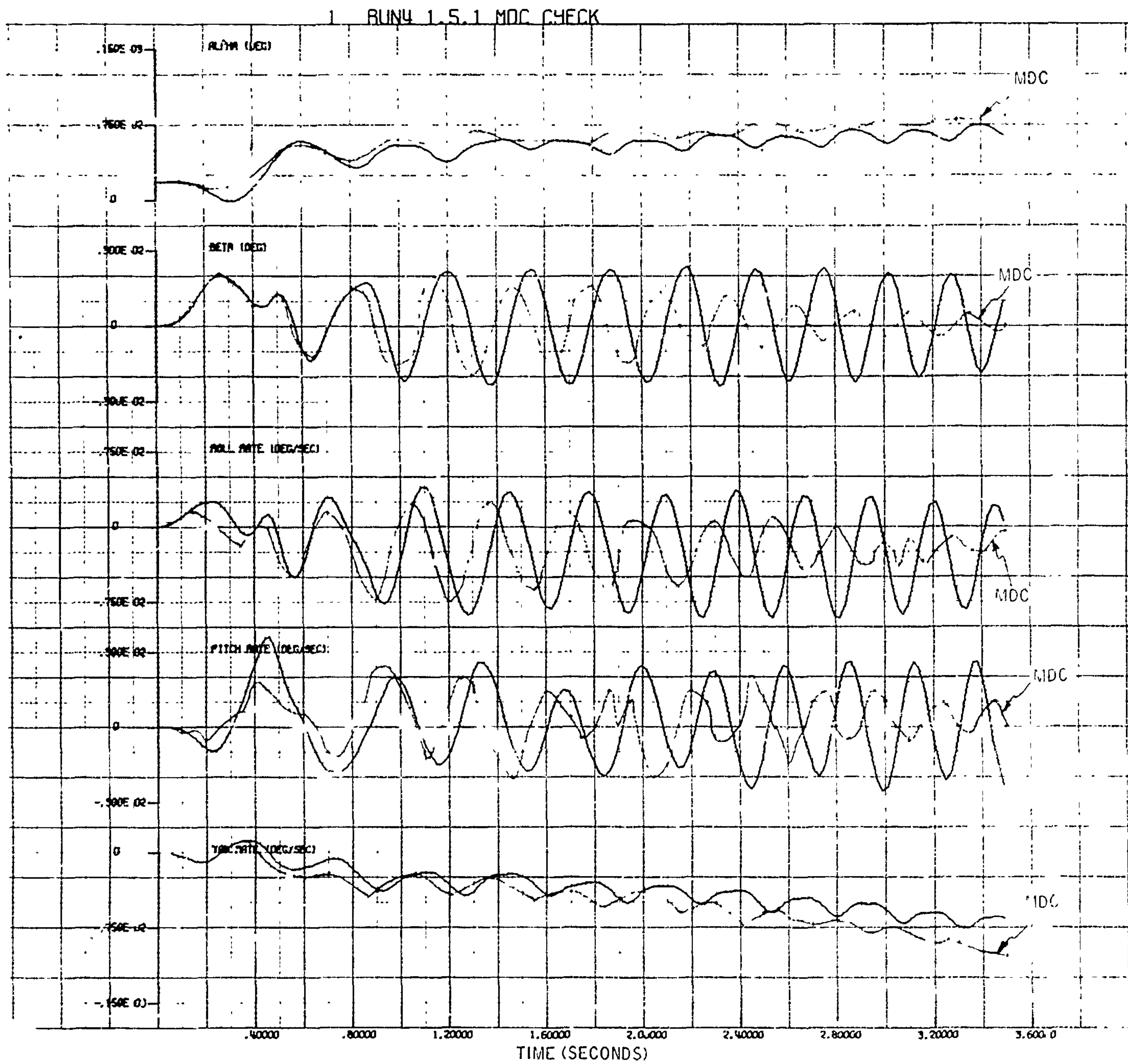


Figure 6-3. Preliminary Comparison with MDC Results

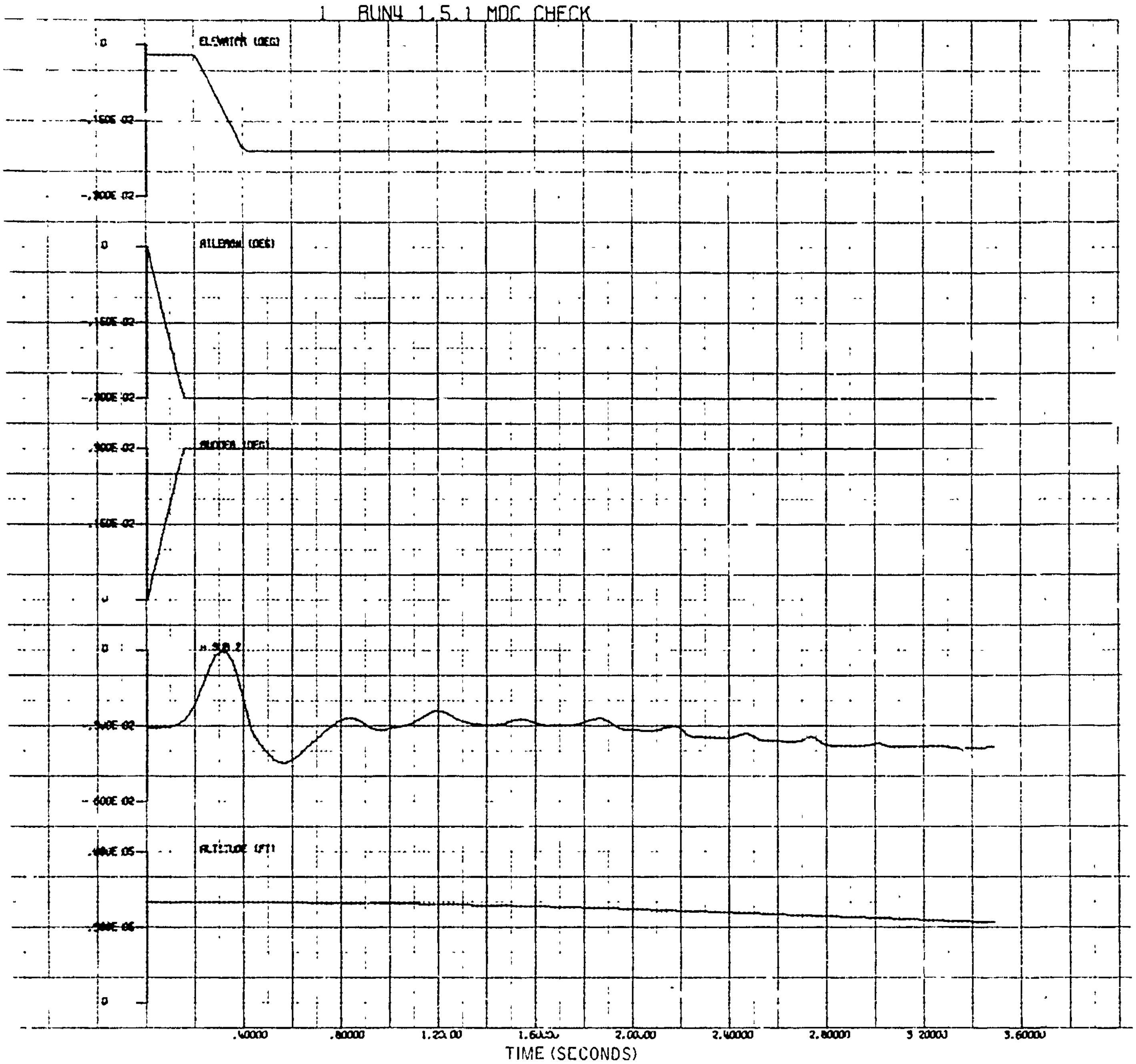


Figure 6-3 Preliminary Comparison with MDC Results (Concluded)

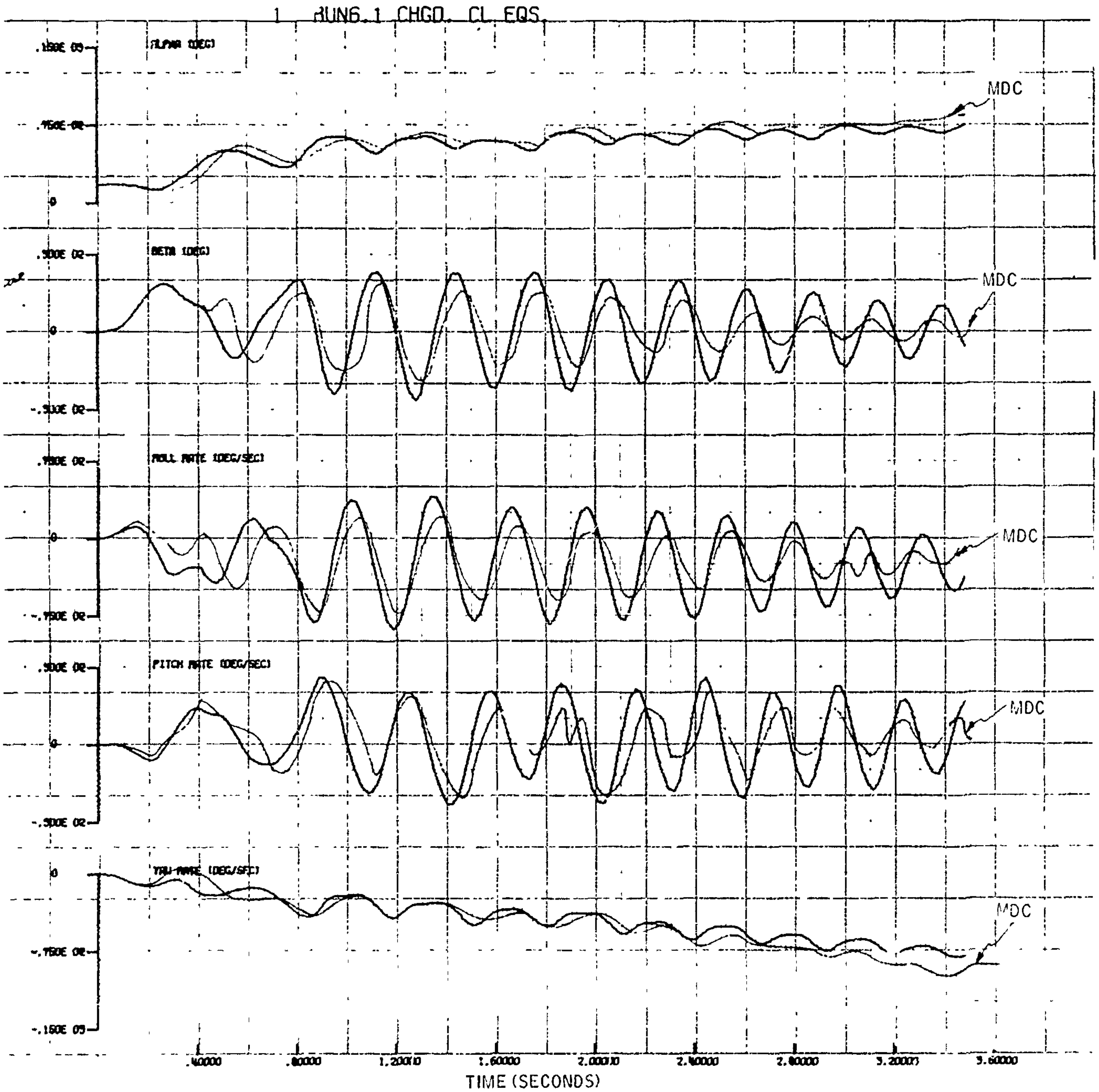


Figure 6-4. Final Comparison with MDC Results

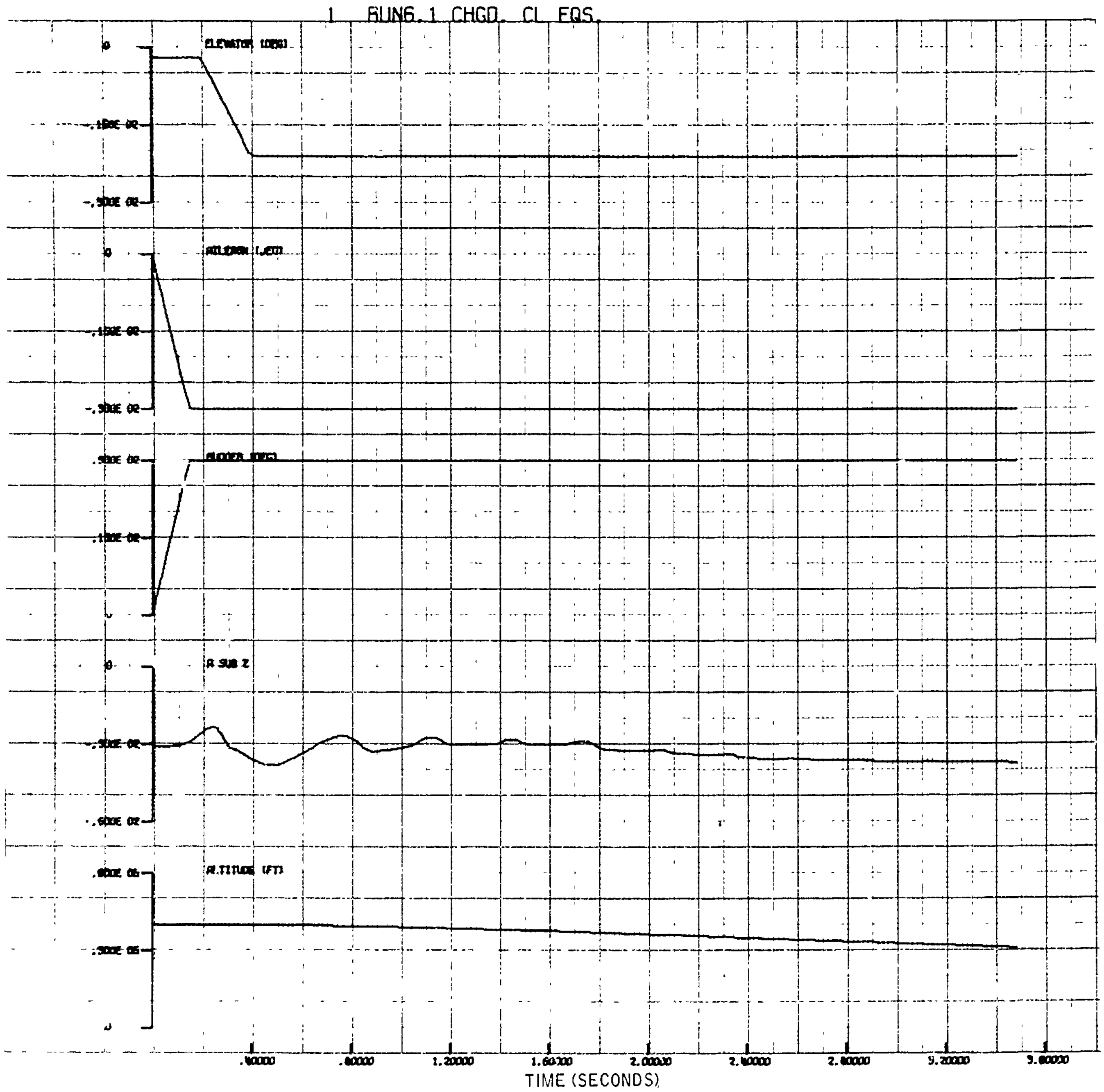


Figure 6-4. Final Comparison with MDC Results (Concluded)

established for normal control. The initial pitch and lateral-directional axis configurations defined for the study are shown in Figure 6-5. Variations in this system were later made to study their effects.

The pitch-axis controller provides a command and augmentation function by comparison of pilot stick force with feedbacks of normal acceleration and pitch rate. The gains and compensation shown are adequate to meet the handling quality criteria of MIL-F-8785B(ASG) and the C* criteria over the entire flight envelope. In this initial configuration the variable gain element, K_α, was designed to operate as a function of angle of attack to eliminate feedback in the stalled flight regime. When K_α is equal to one, the system is in the normal mode of operation. When K_α is set to zero, the system is in a direct surface command mode. Variation of K_α with angle of attack according to the schedule shown in Figure 6-5 provides a sharp increase in the stick gradient as the stall angle of attack is approached. This initial concept for stall limiting was later rejected during the study because many of the basic mode feedbacks contributed desirable stall inhibiting action. Furthermore, these feedbacks provided desirable stability for manual control in the stalled flight mode, thus obviating the need to blend them out at stall. This notion is described further under "Control Criteria" in which the stall prevention concept is presented. Stall limiting by only reducing K_α to zero was also undesirable because it was sensitive to stick trim condition and could not assure angle-of-attack limiting under all conditions. For example, the pilot could continually relieve his stick force by trimming as he reduced airspeed. As a result, he could trim into a stall condition and not have any large force on his stick.

A so-called "neutral speed stability" mode is provided by use of a proportional-plus-integral function. This function should be switched out at some angle of attack prior to stall to obtain a proportional control mode for reasons discussed later (under "Departure Processes"). The sudden (and undesirable) increase in stick force gradient (pitch rate per pound stick force) as the lift coefficient decays near stall was demonstrated by applying progressively larger stick commands and noting the response increase associated with integral control. The optimum angle of attack at which to switch out the integral control was not determined in the study, but it will most likely have to occur at something less than 20 degrees AOA for the F-4. Final design of a stall prevention system will influence the maximum AOA at which integral control can be tolerated.

In the roll axis, a roll command augmentation function is obtained by comparison of roll stick force with a roll rate feedback. As with the pitch axis, a variable gain element, K_α, is used to blend between a normal mode of operation and a direct surface command mode ($K_\alpha = 0$). As the stall angle of attack is approached, the gain K_α is blended to zero to increase the roll stick gradient and to eliminate the roll rate feedback.

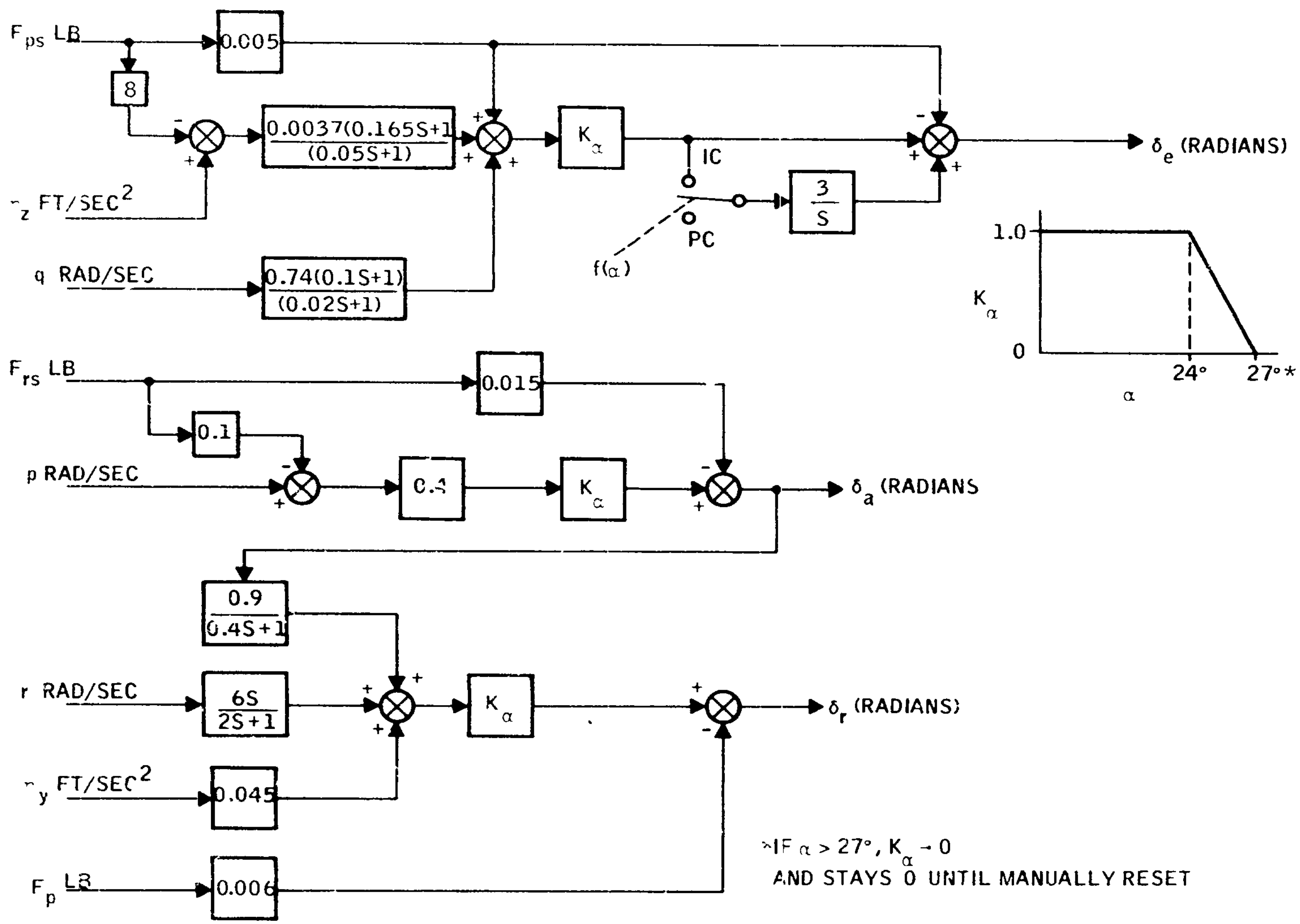


Figure 6-5. Initial Feedback Control System

The yaw axis control uses inputs of yaw rate and lateral acceleration to augment dutch-roll frequency and damping. The yaw rate signal is high-passed to avoid opposition to pilot commands in a turn. The lateral acceleration feedback functions to also provide turn coordination, primarily at the high-dynamic-pressure conditions. Additional turn coordination is provided by a lagged aileron-to-rudder crossfeed. This provides the primary turn coordination at low-dynamic-pressure conditions but tends to degrade turn coordination at high-dynamic pressure. The lag on this crossfeed reduces the degrading effect at high dynamic pressure. The lateral-directional controls shown in Figure 6-5 satisfy MIL-F-8785B (ASG) with fixed gain in both roll and yaw.

The yaw-axis control is also regulated by a variable-gain element, K_{α}, which provides a transfer of control from the normal mode to a direct command mode.

An alternate lateral-directional controller, shown in Figure 6-6, was also considered for the stall/spin study in order to examine performance of the system with a different turn coordination characteristic. This system differs from the one shown in Figure 6-5 in that it has a larger roll-stick-to-aileron gain, and it uses high-passed roll stick to rudder for turn coordination. The high-passed crossfeed is intended to preclude miscoordination of the airplane after a large maneuver such as a 360-degree roll.

Although there is substantial difference between the lateral-directional systems of Figures 6-5 and 6-6 for normal flight, the rapid loss in rudder effectiveness with angle of attack and the relative ineffectiveness of the closed-loop turn coordination properties of either system make this difference trivial for the purposes of this study. The majority of the feedback control data presented here was computed for the system of Figure 6-6, but the general results and conclusions apply equally well for either system. A more thorough analysis of turn coordination effects should be made in a future study wherein increased rudder effectiveness could be hypothesized.

BASIC AIRCRAFT STUDIES

As discussed earlier, the stall/spin study was categorized into two major areas of analysis -- the basic aircraft studies and the control criteria studies. This subsection describes the results obtained from the basic aircraft studies. Analysis of departure processes, spin evolution, controllability limits, and recovery processes are presented.

Departure Processes

The simulation is capable of analyzing aircraft stall/spin conditions initiating at any airspeed and altitude condition in the subsonic speed range

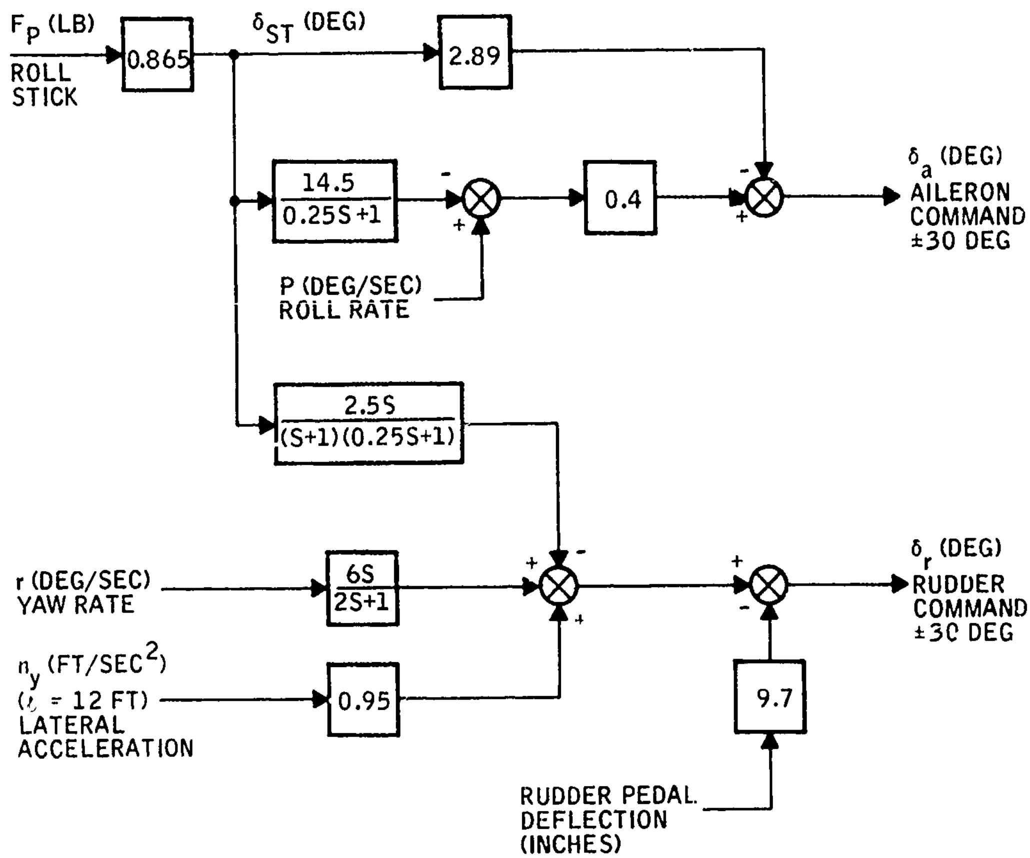


Figure 6-6. Alternate Lateral-Directional Axis Controller

However, to meet the objectives of this study, it was sufficient to restrict the initial conditions to one flight condition. This condition was a one-g trimmed flight at a 40,000-foot altitude with a velocity of 501 ft/sec. The trim angle of attack was 17.9 degrees.

Departure and the subsequent aircraft behavior is very sensitive to the manner in which the control surfaces are deflected. Whether the airplane goes into a rolling departure, a mildly oscillatory spin, a flat spin, etc., depends on the initially applied controls. This phenomena has been fairly well documented in past studies (e.g., References 6-1 and 6-2). For the study of spin evolution, controllability limits, and recovery, most of the spins were established by using full-up (TEU) elevator, full-anti-spin aileron, and full-pro-spin rudder[1]. Figure 6-7 shows a spin time history[2] wherein all surfaces were deflected at the same rate (15 deg/sec) and all three surface deflections were initiated at the same time (t = 1 second). As can be seen in Figure 6-7, these controls produced a mildly oscillatory, high-angle-of-attack spin which progressed into a flat spin. It has been established in past studies that a flat spin is nonrecoverable with the aerodynamic controls. Hence, this spin behavior was of particular interest for determining the controllability limits of the aerodynamic control surfaces and the impact of these limits on the flight controls design.

The departure process for the spin shown in Figure 6-7 is studied by first examining the pitching moment equation for the first 4 seconds after application of controls. The major aerodynamic and inertial pitching moments are plotted in Figure 6-8 for the first 4 seconds of flight. Since angle of attack increases monotonically with time in this time period the moments were plotted as a function of angle of attack for the purpose f the discussion. The corresponding time points are indicated on each curve, however. When trailing-edge-up elevator is applied, the angle of attack begins to increase because of the imbalance in the pitching moment equation. The net aerodynamic pitching moment increases until the angle of attack reaches approximately 22 degrees. Initially the positive pitching moment due to elevator deflection increases more rapidly than the restoring moment due to angle of attack. At approximately 22 degrees AOA, the elevator effectiveness begins to decrease. As a result, the pitch-restoring moment due to angle of attack increases more rapidly than the moment due to surface deflection. Finally, at an angle of attack of approximately 35.5 degrees, the restoring aerodynamic moment due to angle of attack just balances the positive moment due to elevator deflection. Hence, if the lateral-directional axis had remained unperturbed, the vehicle would eventually settle out at a steady angle of attack around 36 degrees; and no spin would result.

[1] The terms "anti" and "pro" refer to the normal surface moment with respect to the existing direction of the spin rate component along the subject axis.

[2] In Figure 6-7 and in the rest of the computer-generated spin time histories reproduced in this section, the horizontal axis is understood to be "time" in seconds.

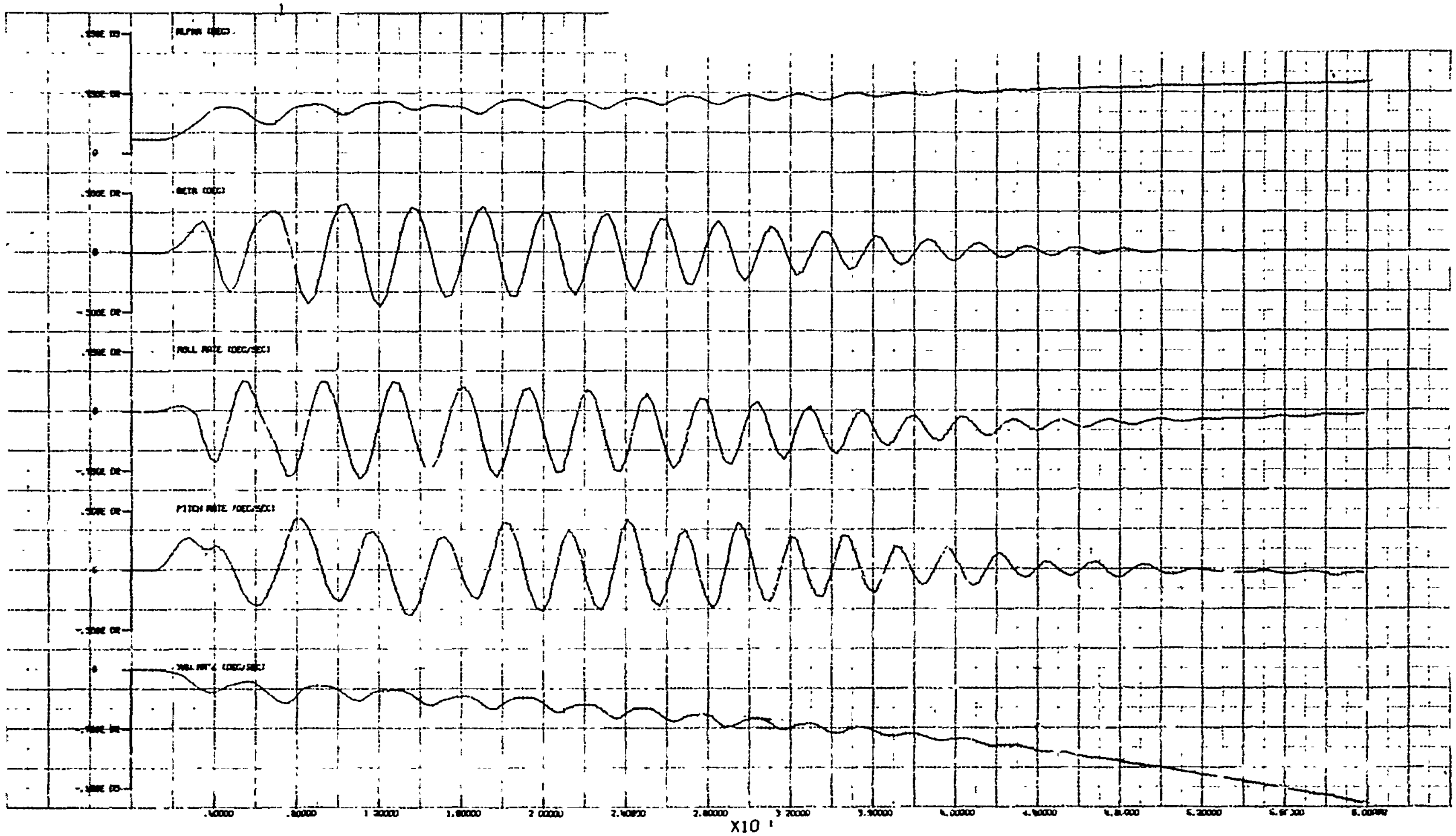


Figure 6-7. Nominal Stall/Spin Time History

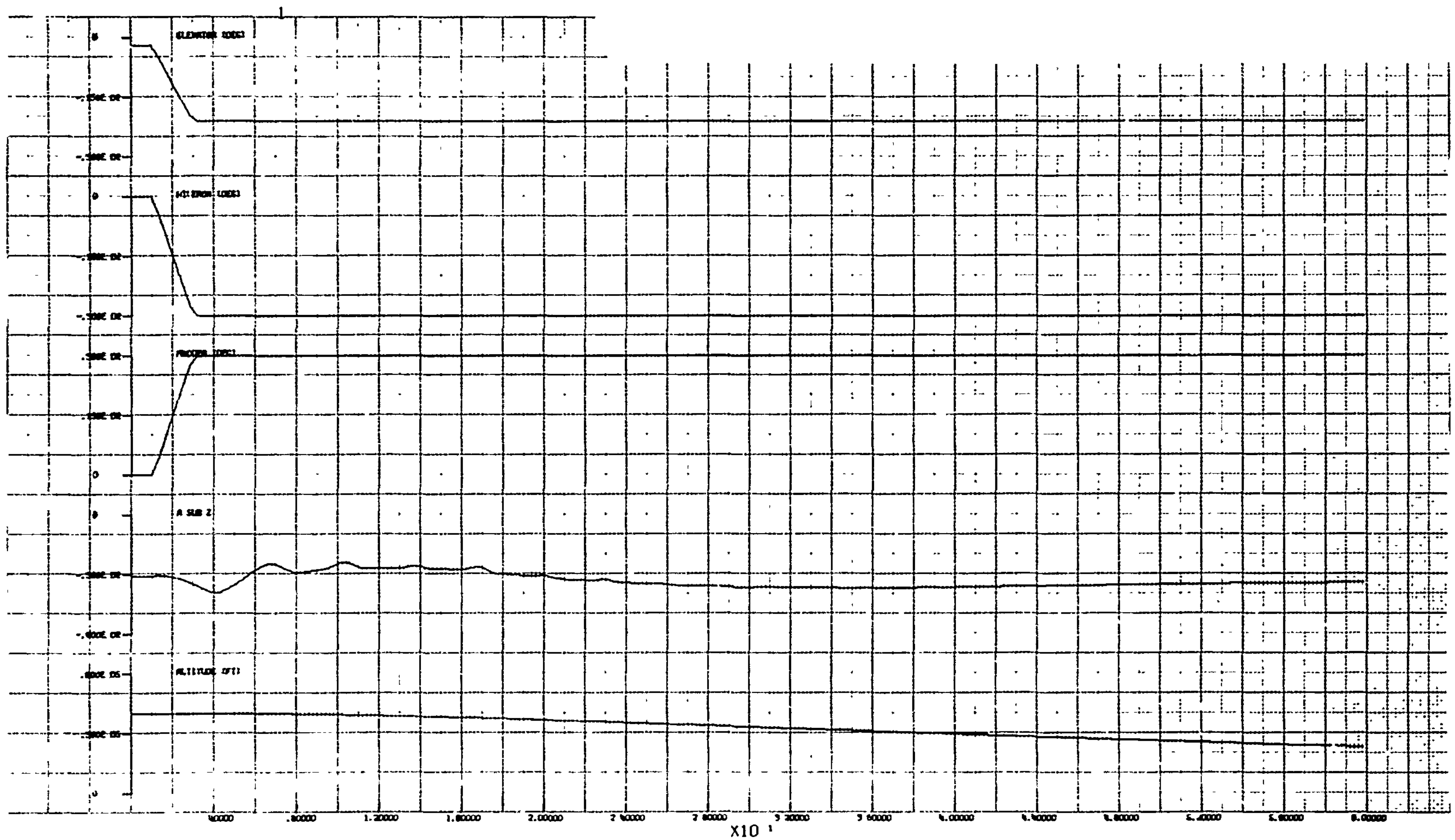


Figure 6-7. Nominal Stall/Spin Time History (Continued)

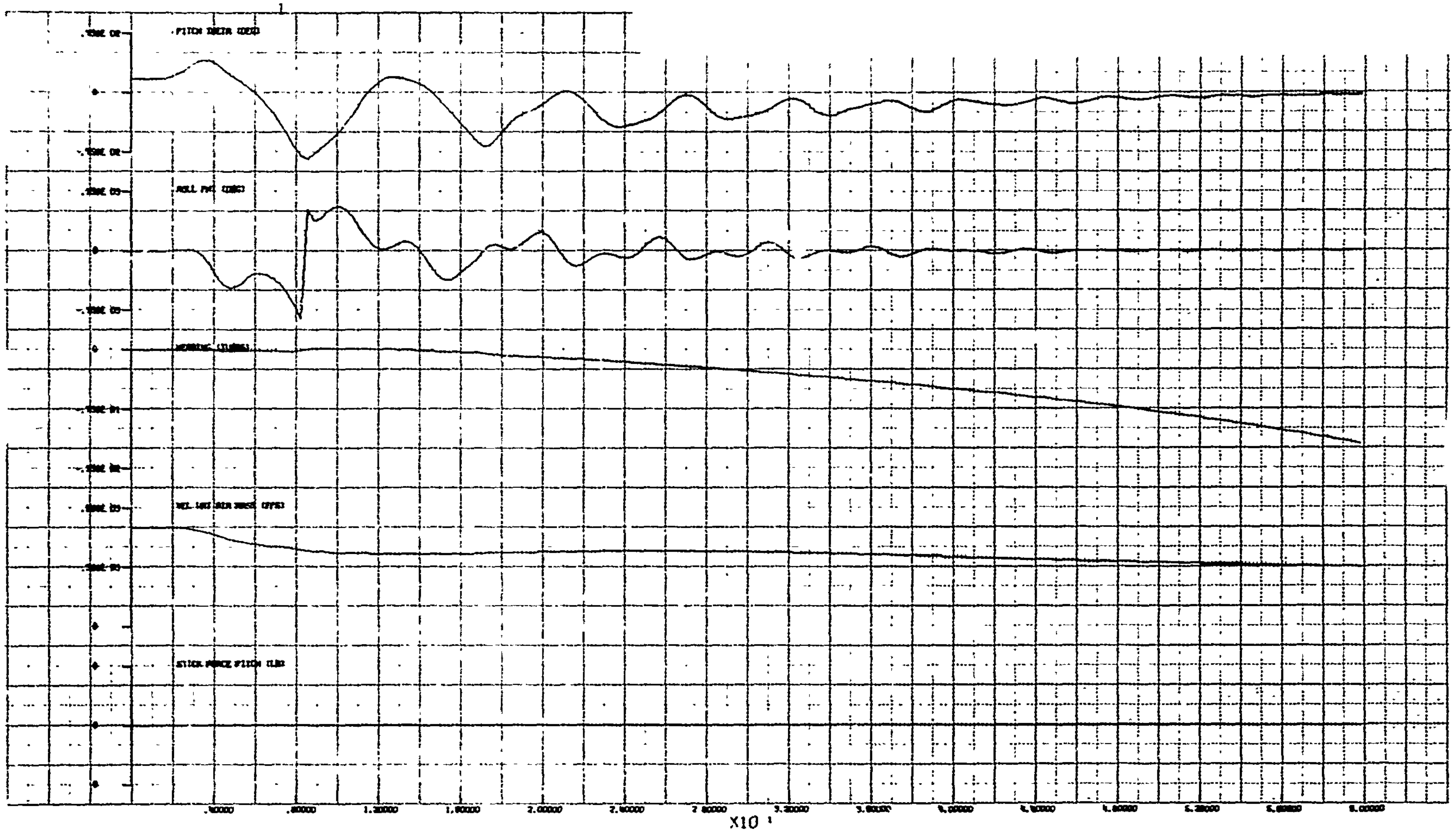


Figure 6-7. Nominal Stall/Spin Time History (Concluded)

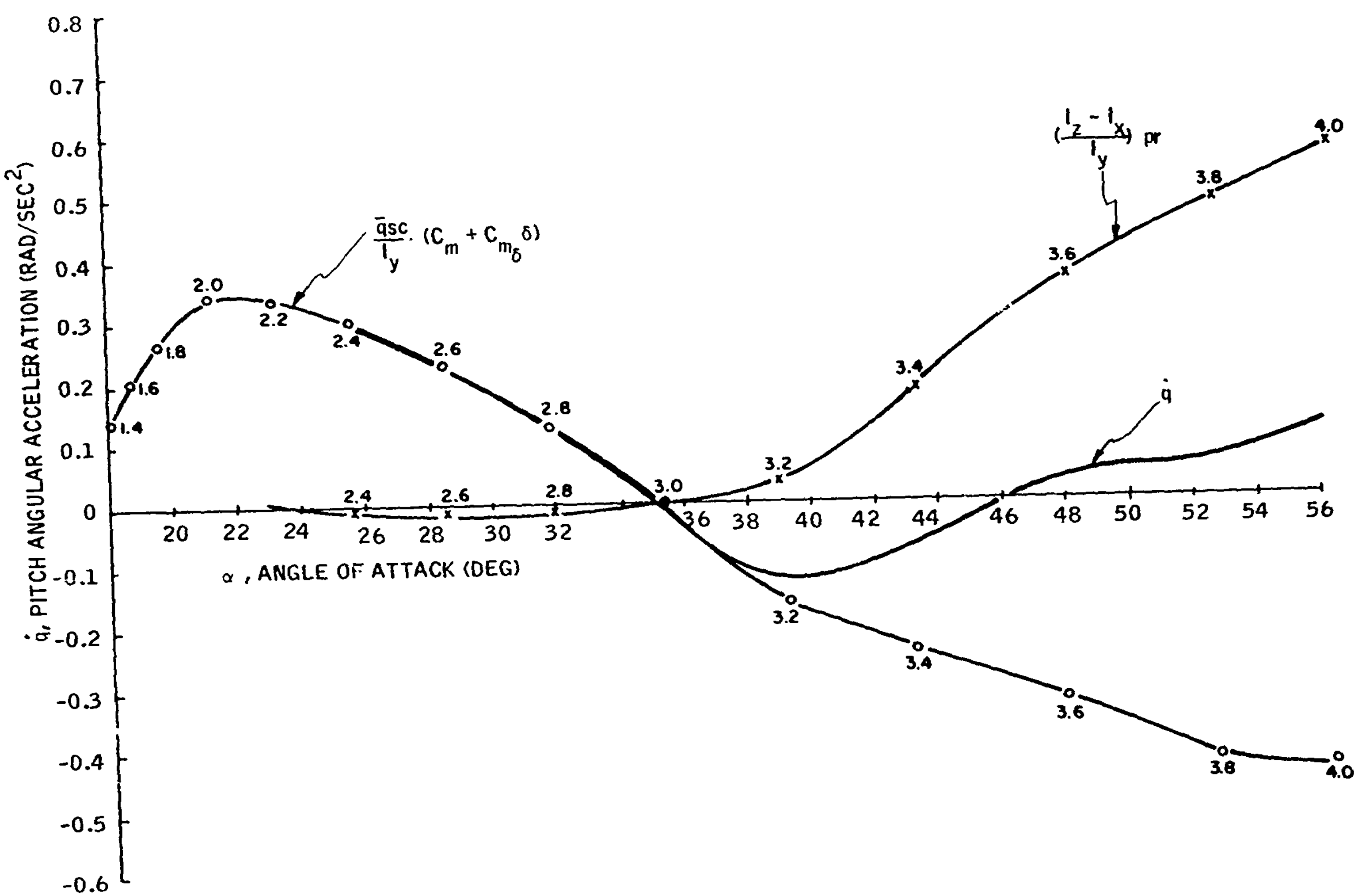


Figure 6-8. Contributions to Pitching Moment as a Function of Angle of Attack

At this angle of attack the engine thrust is insufficient to maintain level flight, and the aircraft would lose altitude until forward stick was applied.

As figure 6-7 indicates, however, the angle of attack does not stabilize near 36 degrees. Figure 6-8 shows that because the lateral-directional axis had also been perturbed, a positive inertial pitching moment resulted from the lateral-directional motions. This positive pitching moment was sufficiently large as to cause the pitch rate to again increase. Angle of attack continued to build up until the inertial pitching moment was countered by the increased pitch-axis aerodynamic restoring moment.

The lateral-directional axis motions were caused by the commanded negative (roll right) aileron and positive (yaw left) rudder deflections. The negative aileron causes an initial positive rolling moment and an initial negative yawing moment. Deflection of the aileron and elevator alone (i. e., without any rudder deflection) is sufficient to cause the airplane to enter a spin. This is shown in Figure 6-9. A positive deflection of the rudder also causes a negative yawing moment which further aggravates the out-of-control situation. The positive rudder deflection also causes some positive rolling moment, but this effect is masked by a larger rolling moment due to aileron. Like the aileron, a full-rudder deflection is sufficient to set up the conditions for a spin. This is exhibited in Figure 6-10.

Roll-axis behavior can be explained by examination of the most significant contributors to rolling moment as shown in Figure 6-11, namely the aircraft dihedral effect, $L_\beta \beta$, and aileron surface effectiveness $L_{\delta_a} \delta_a$. There is also some significant effect due to the roll damping term L_p, but it was neglected in Figure 6-11. The negative aileron deflection causes an initial positive rolling moment. This moment increases as the surface deflection increases; but as angle of attack increases, the surface effectiveness decreases and eventually reverses at an angle of attack of approximately 55 degrees.

The negative yaw rate caused by deflection of the ailerons and rudder in turn caused a positive sideslip to result. This positive sideslip through the dihedral effect, $L_\beta \beta$, causes a negative rolling moment. As both sideslip and angle of attack increase, the rolling moment due to sideslip overpowers the positive moment due to aileron deflection, causing a roll rate reversal (see Figure 6-7). Because $L_\beta \beta$ builds up rapidly, the roll rate reversal is also rapid, causing a 360-degree snap roll to the left. Negative roll rate continues to increase until sideslip changes sign to effect a sign change on the dihedral effect, $L_\beta \beta$.

The aircraft behavior during departure is compounded by the fact that the aircraft is statically unstable in the lateral-directional axis for angles of attack between 23 and 37 degrees. This can be observed by examining the approximate expression for the dutch-roll frequency at high angles of attack. It is given by

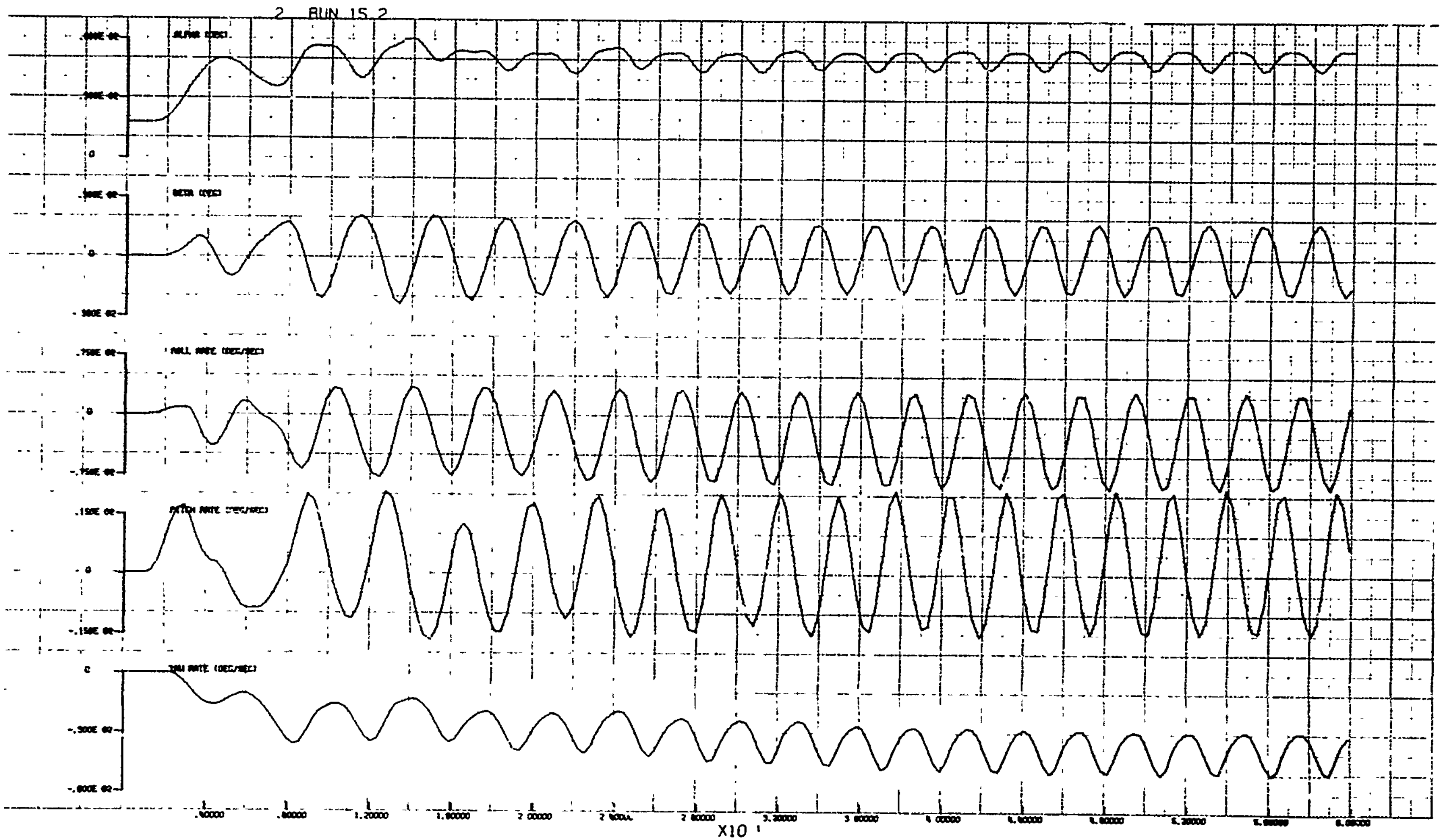


Figure 6-9. Spin Produced by Elevator and Aileron Deflection

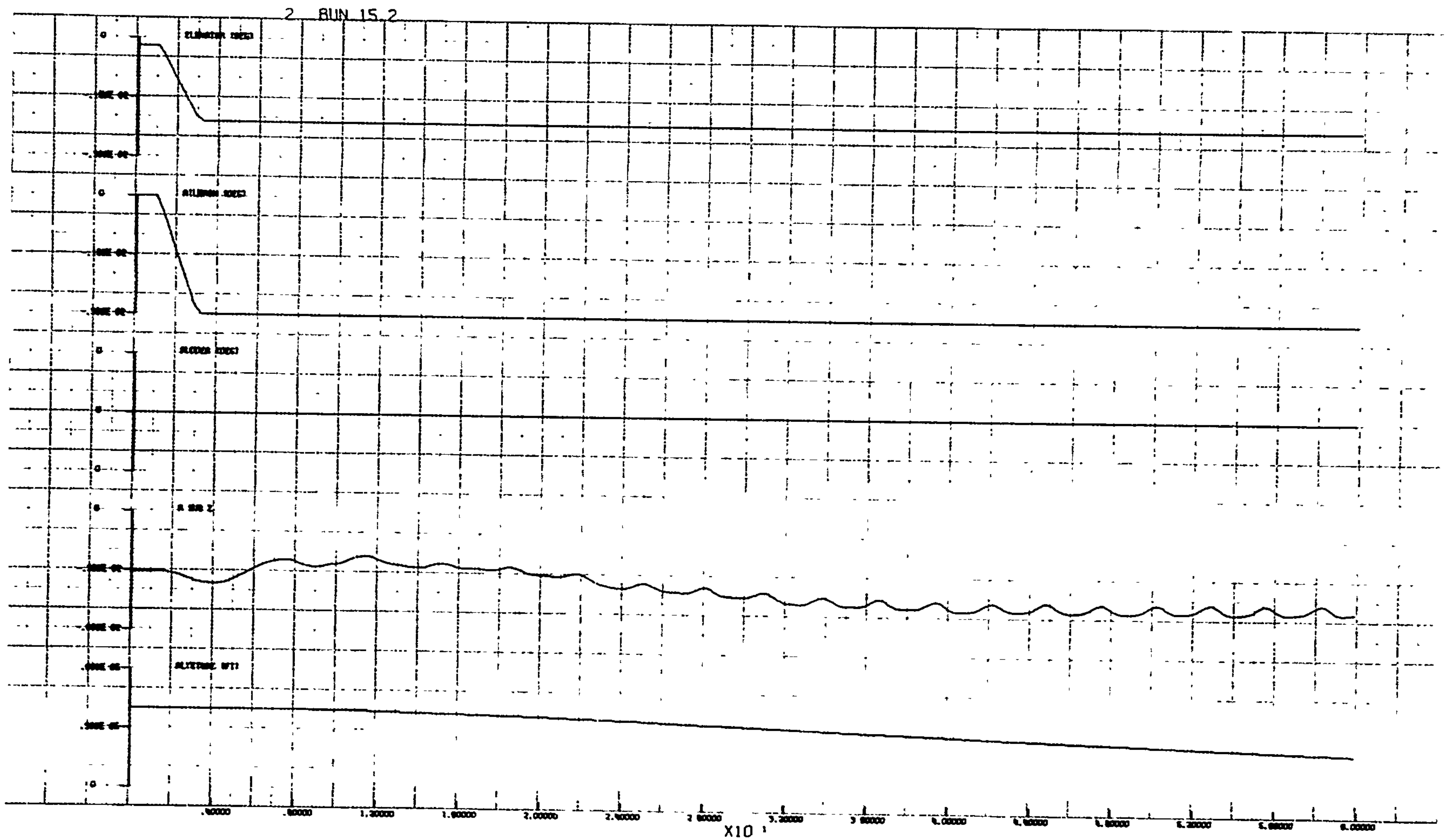


Figure 6-9. Spin Produced by Elevator and Aileron Deflection (Continued)

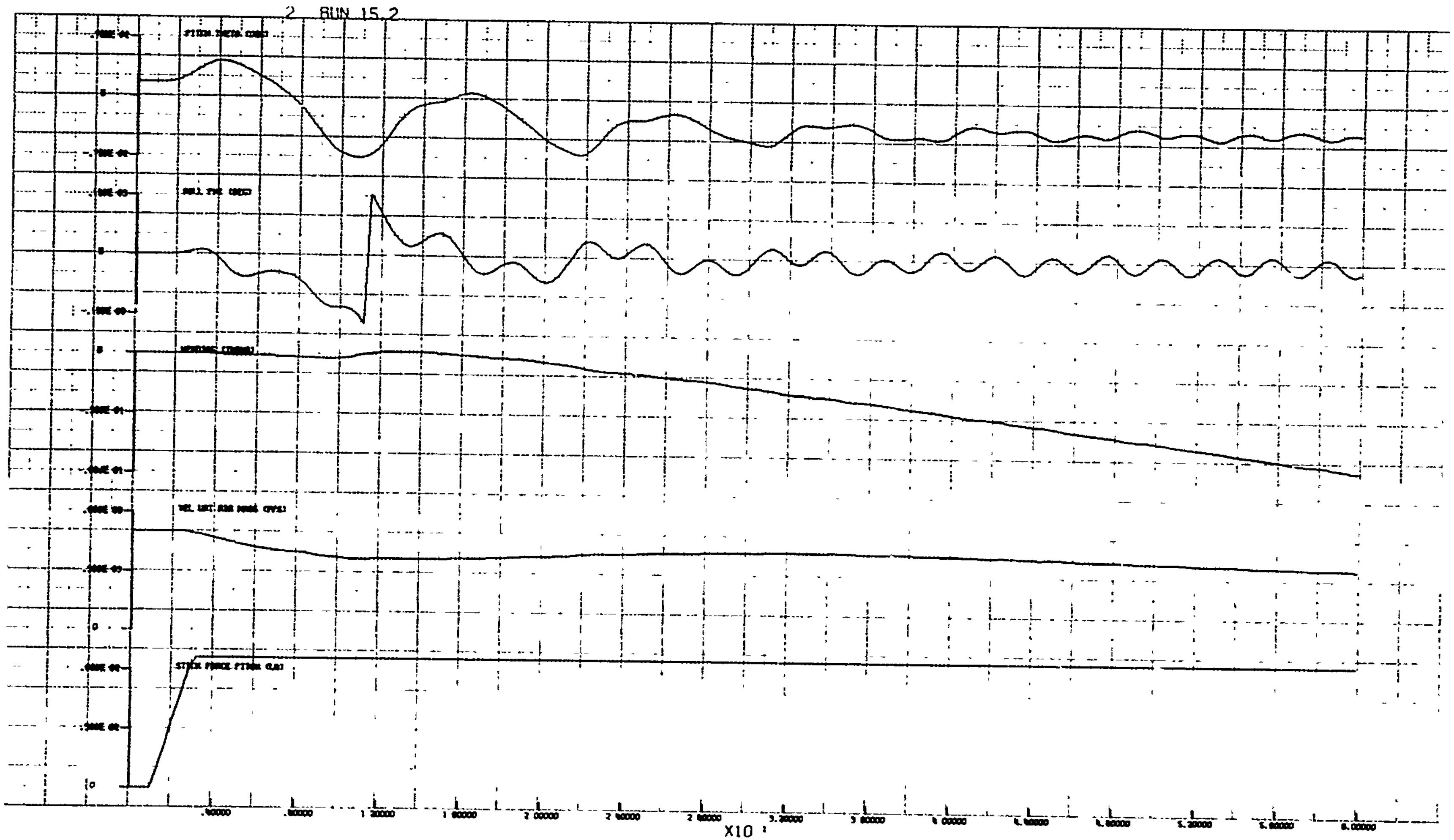


Figure 6-9. Spin Produced by Elevator and Aileron Deflection (Concluded)

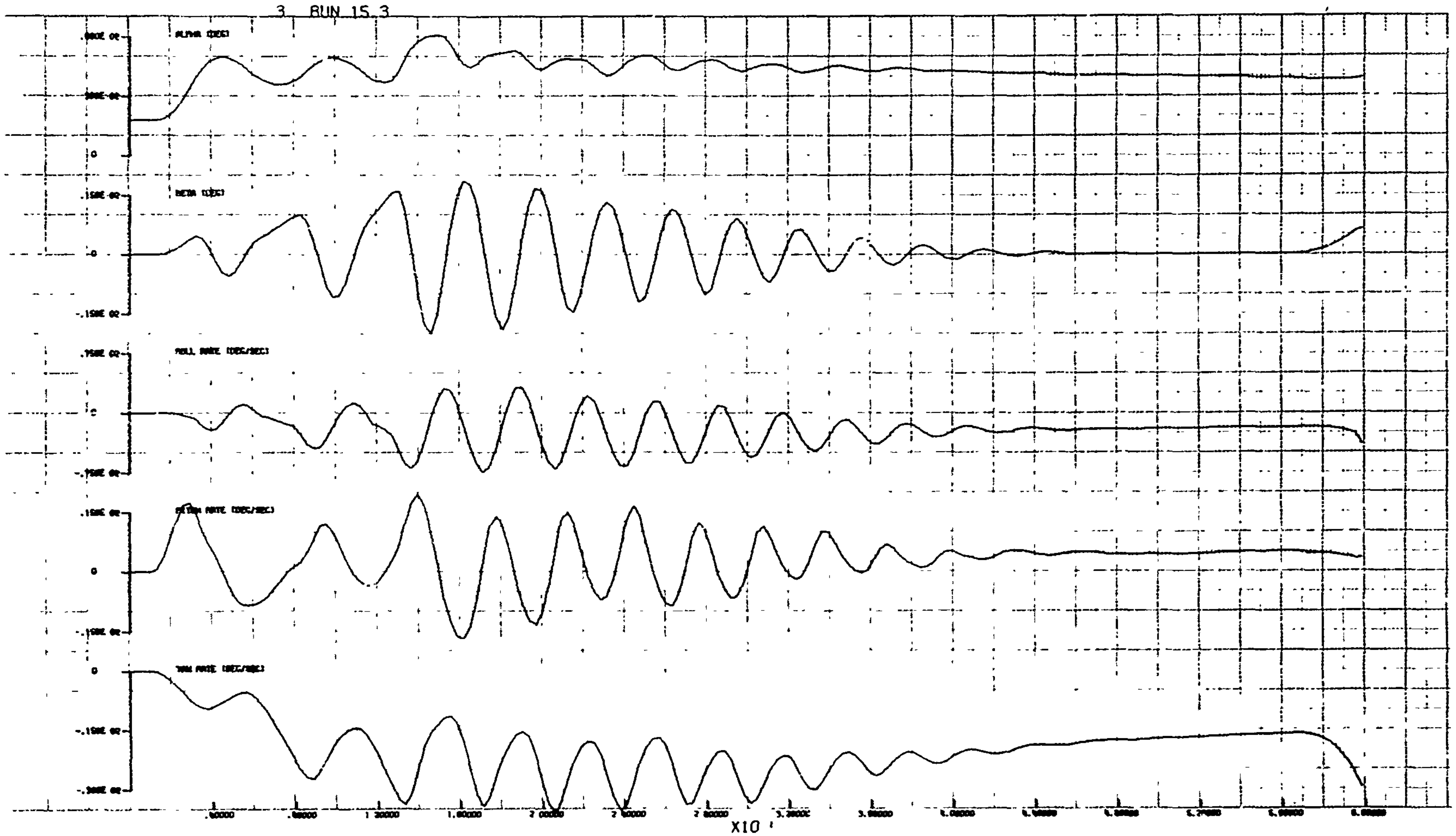


Figure 6-10. Spin Produced by Elevator and Rudder Deflection

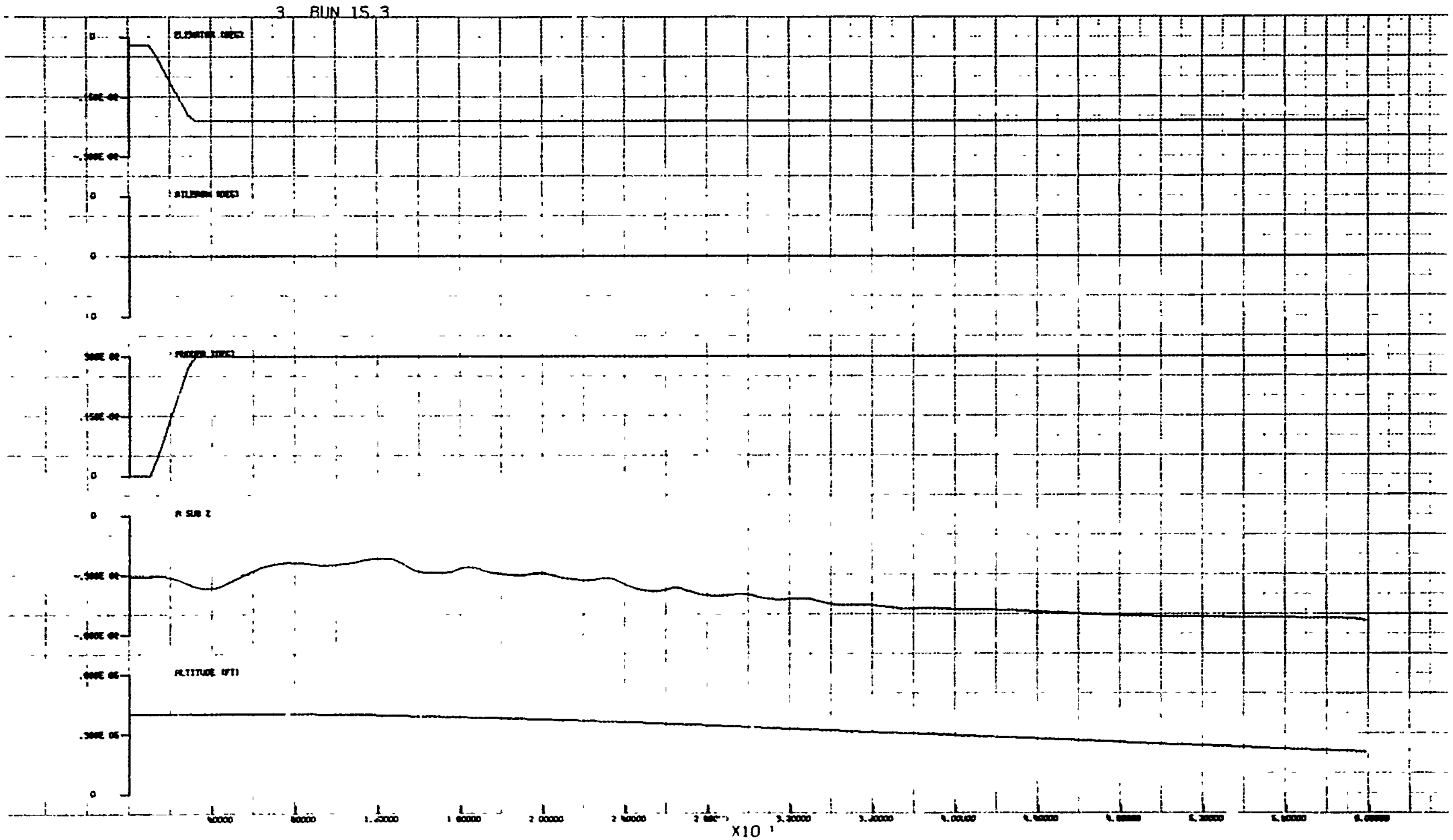


Figure 6-10. Spin Produced by Elevator and Rudder Deflection (Continued)

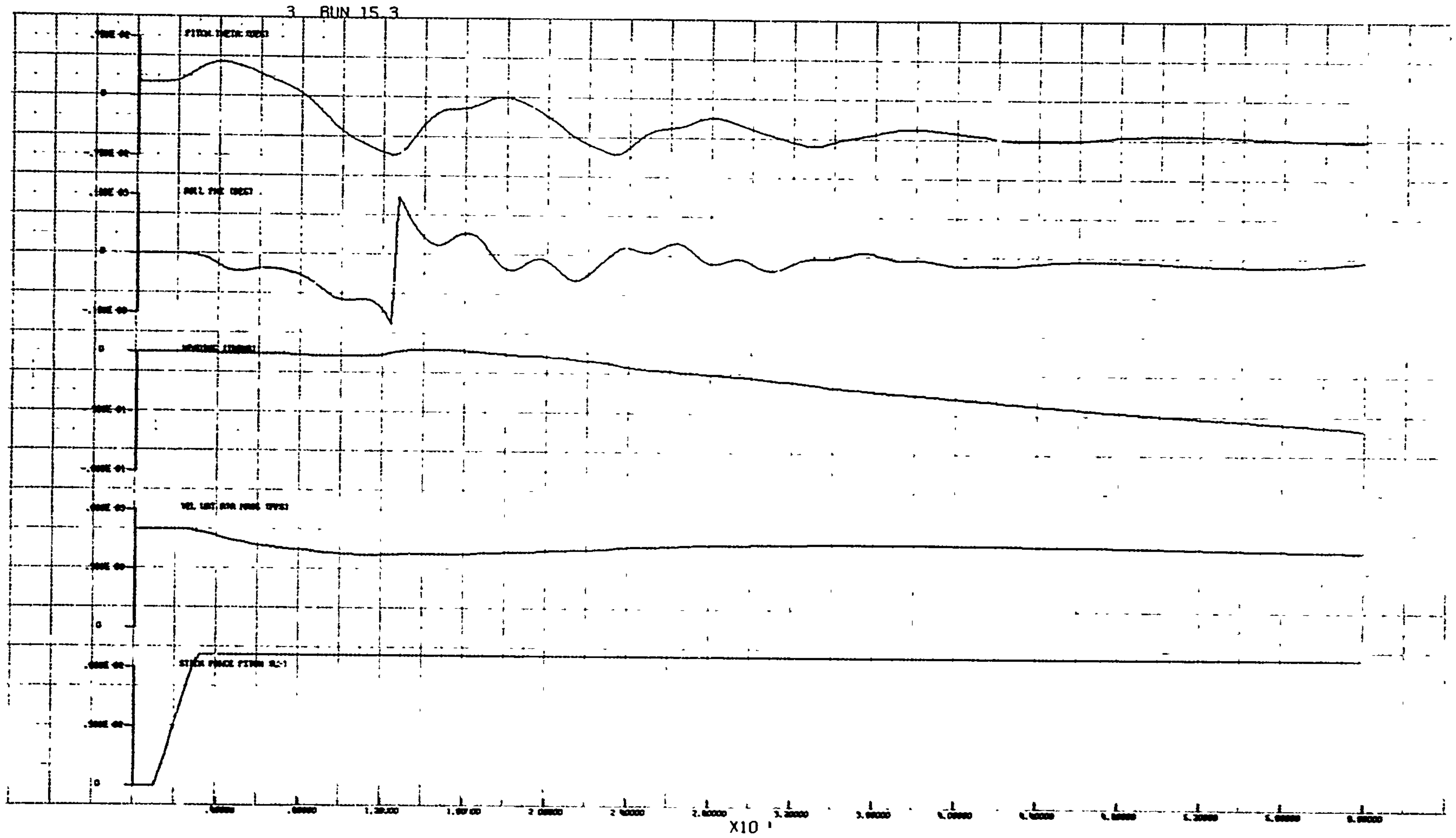


Figure 6-10. Spin Produced by Elevator and Rudder Deflection (Concluded)

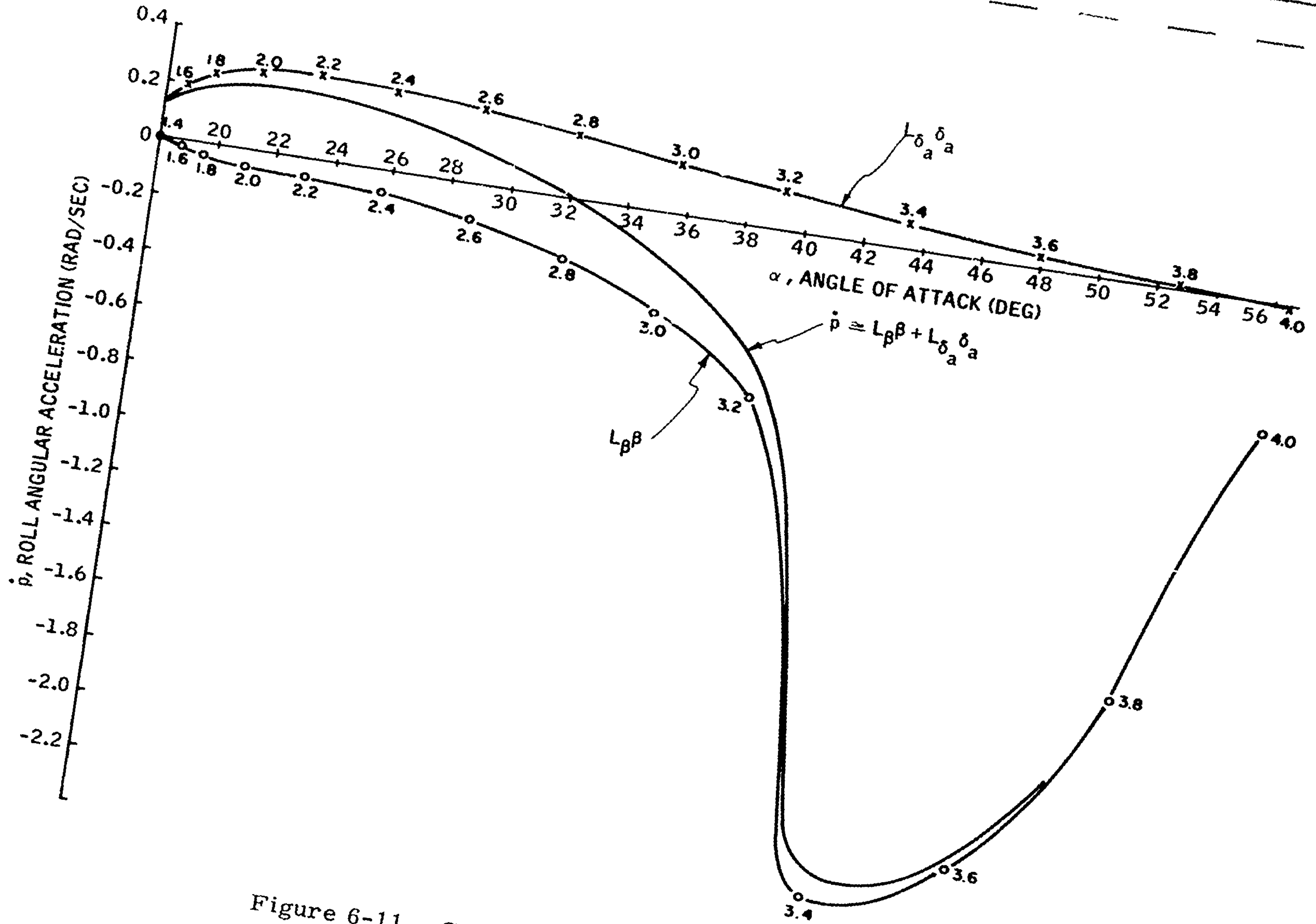


Figure 6-11. Contributions to Rolling Moment as a Function of Angle of Attack

$$\omega_d^2 = N_\beta \cos\alpha - L_\beta \sin\alpha$$

The static instability arises from the fact that the yawing moment due to sideslip, N_β, which is normally positive, becomes negative for angles of attack above 20 degrees. This is exhibited in the plots of C_n as a function of sideslip and angle of attack shown in Figure IV-3 of Appendix IV. Figure 6-12 shows a comparison of the two terms which comprise the above expression for the dutch-roll frequency. As can be seen, the effective dihedral term, L_β, actually delays the onset of static instability from 20 degrees to nearly 23 degrees. Figure 6-12 also shows that static stability is regained at 37 degrees angle of attack when $L_\beta \sin\alpha$ again becomes the dominant term. The effect of the static instability can be observed in Figure 6-7 by close examination of the yaw rate response. A definite increase in the negative yaw rate buildup can be seen to occur around 2.4 seconds. This increased rate of change on yaw rate continues until the angle of attack exceeds 37 degrees (around 3.1 seconds) at which time the aircraft is again statically stable. Static stability is maintained so long as the angle of attack remains above 37 degrees. Yaw rate continues to build up in an oscillatory fashion, however, due to the continued presence of negative aileron and positive rudder deflections. The net positive product of yaw rate and roll rate also provides a positive pitching moment which inhibits the angle of attack from decreasing to below the stall angle of attack. As a result, the vehicle enters into a spin condition.

As mentioned previously the severity of the established spin depends greatly on the manner in which the control surfaces are deflected relative to each other. It is of particular interest to note the spin that results when only ailerons and elevator are used to effect departure. In this case the buildup in yaw rate and angle of attack stabilizes, and an apparent balance between inertial moments and aerodynamic moments is achieved in all axes. Figure 6-9 is an example of this case. When rudder and elevator are used to establish the spin, the vehicle begins to recover from the spin by itself. Recovery continues until the angle of attack becomes less than 37 degrees, at which point the lateral-directional axis again becomes statically unstable. The static instability eventually would cause the vehicle to once again enter an oscillatory spin. A better example of this situation is shown in Figure 6-13 where aileron and rudder deflections were applied for only eight seconds. This figure shows how the lateral-directional static instability reinitiates an oscillatory spin whenever the angle of attack decreases to 37 degrees.

It has been established that a combination of full trailing-edge-up elevator in combination with either an aileron or rudder deflection is sufficient to cause departure and finally a spin condition. This suggests the question as to whether or not an effective turn coordination system could inhibit departure. Normally a turn coordination system would command a rudder deflection to minimize sideslip for a roll rate command. The departure-inhibiting effect of using coordinated aileron and rudder controls is shown in Figure 6-14,

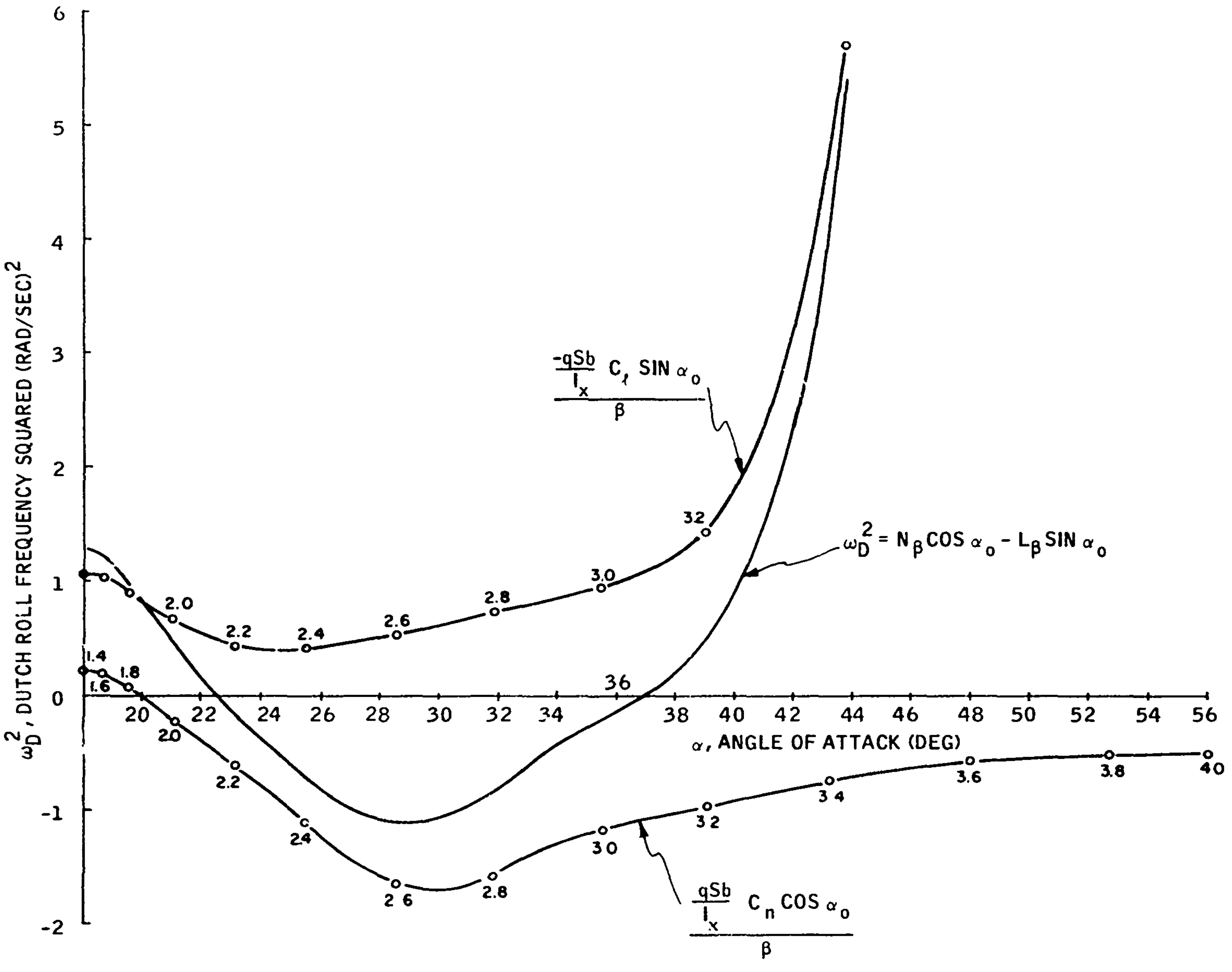


Figure 6-12. Contributions to the Dutch-Roll Frequency as a Function of Angle of Attack

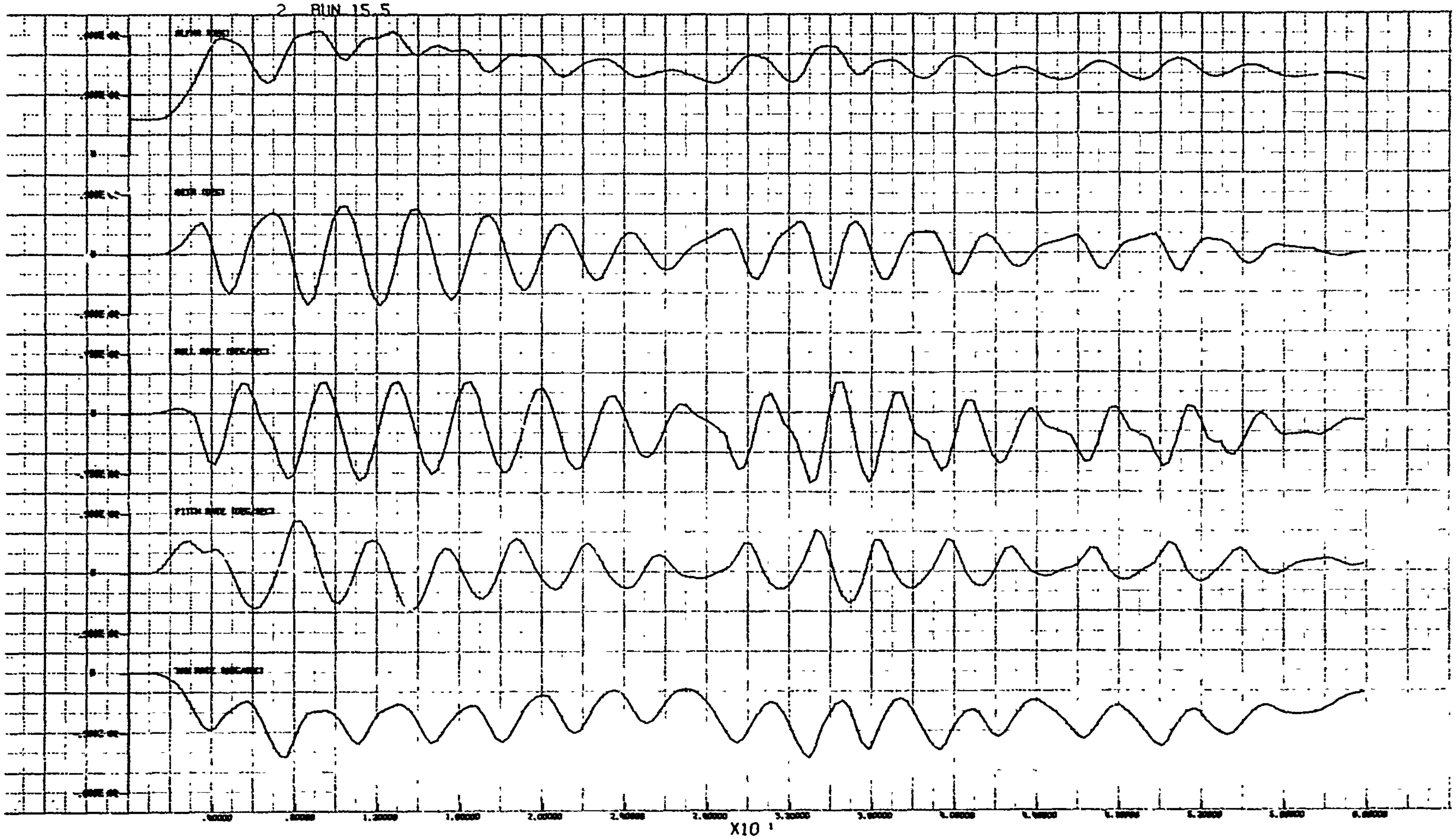


Figure 6-13. Spin with Ailerons and Rudder Neutralized after 8 Seconds

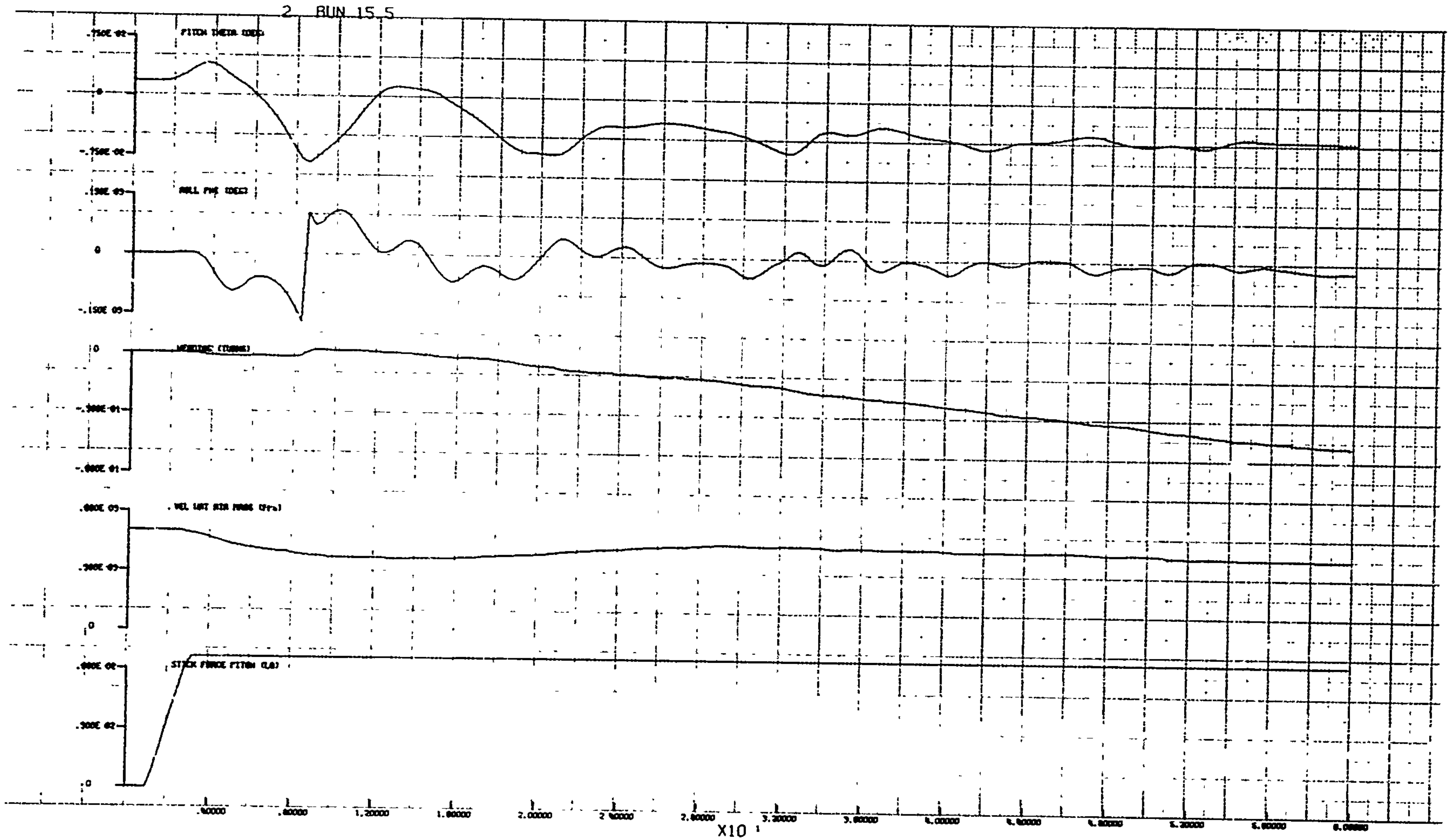


Figure 6-13. Spin with Ailerons and Rudder Neutralized after 8 Seconds (Continued)

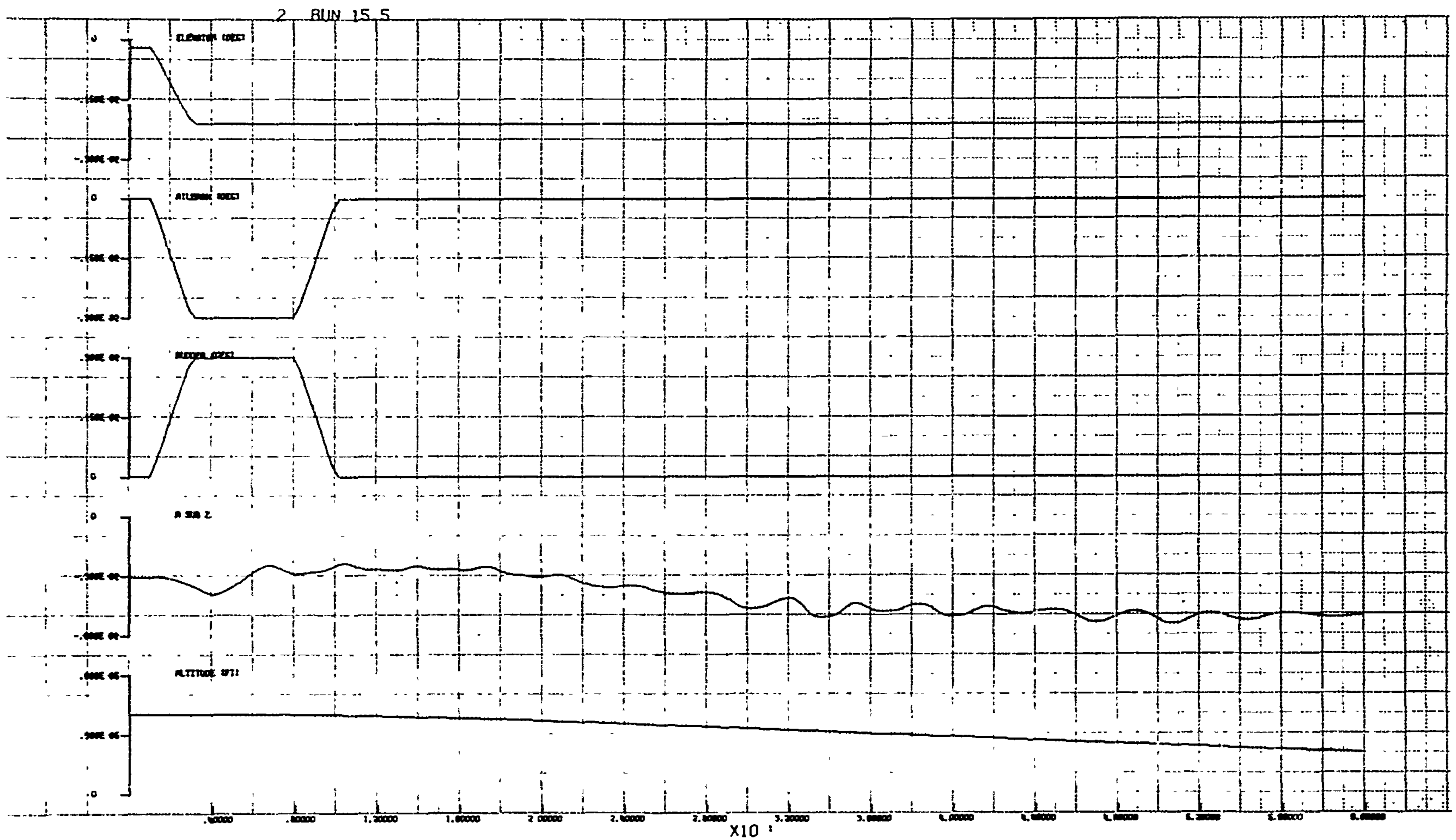


Figure 6-13. Spin with Ailerons and Rudder Neutralized after 8 Seconds (Concluded)

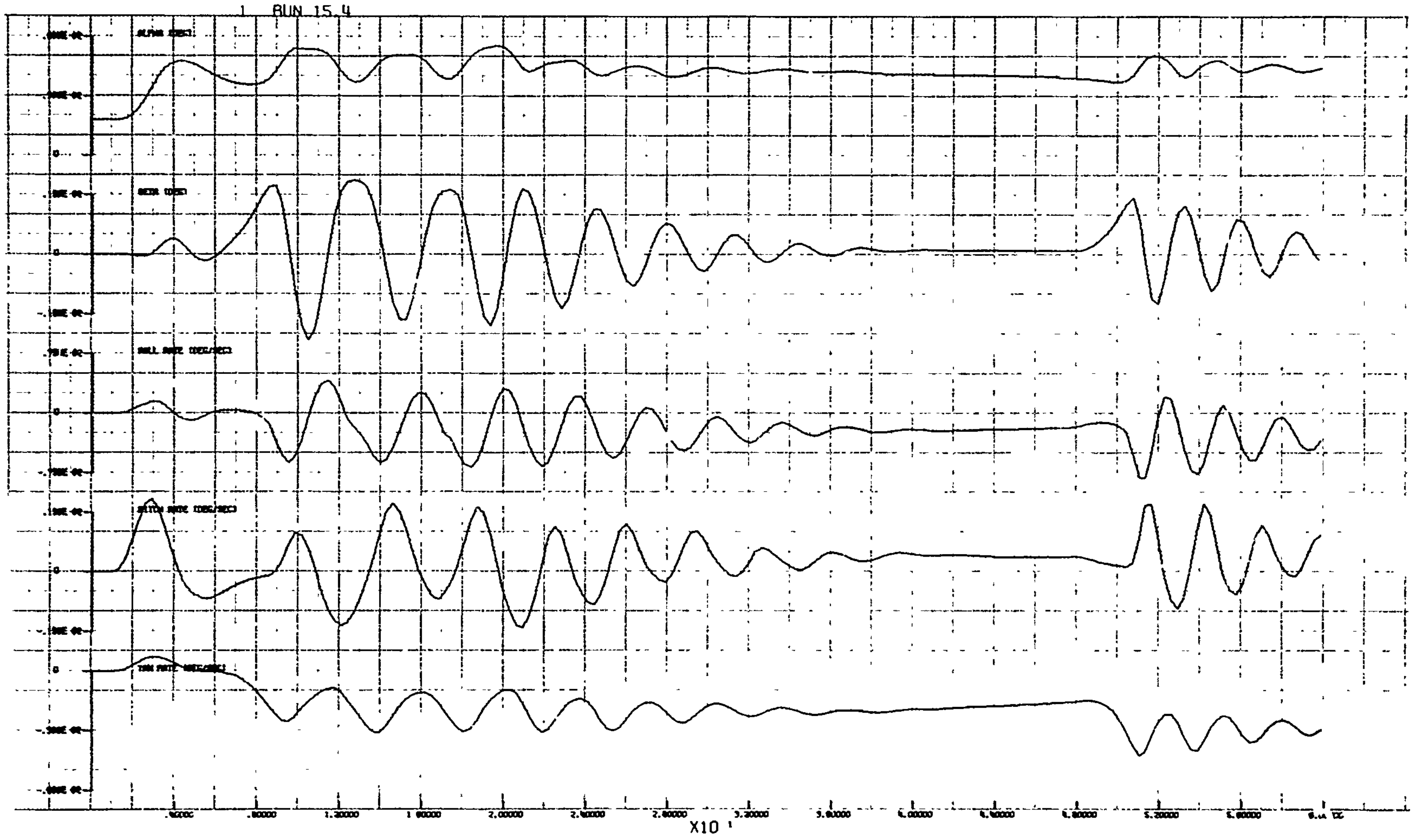


Figure 6-14. Spin Initiated with Coordinated Aileron and Rudder Deflection

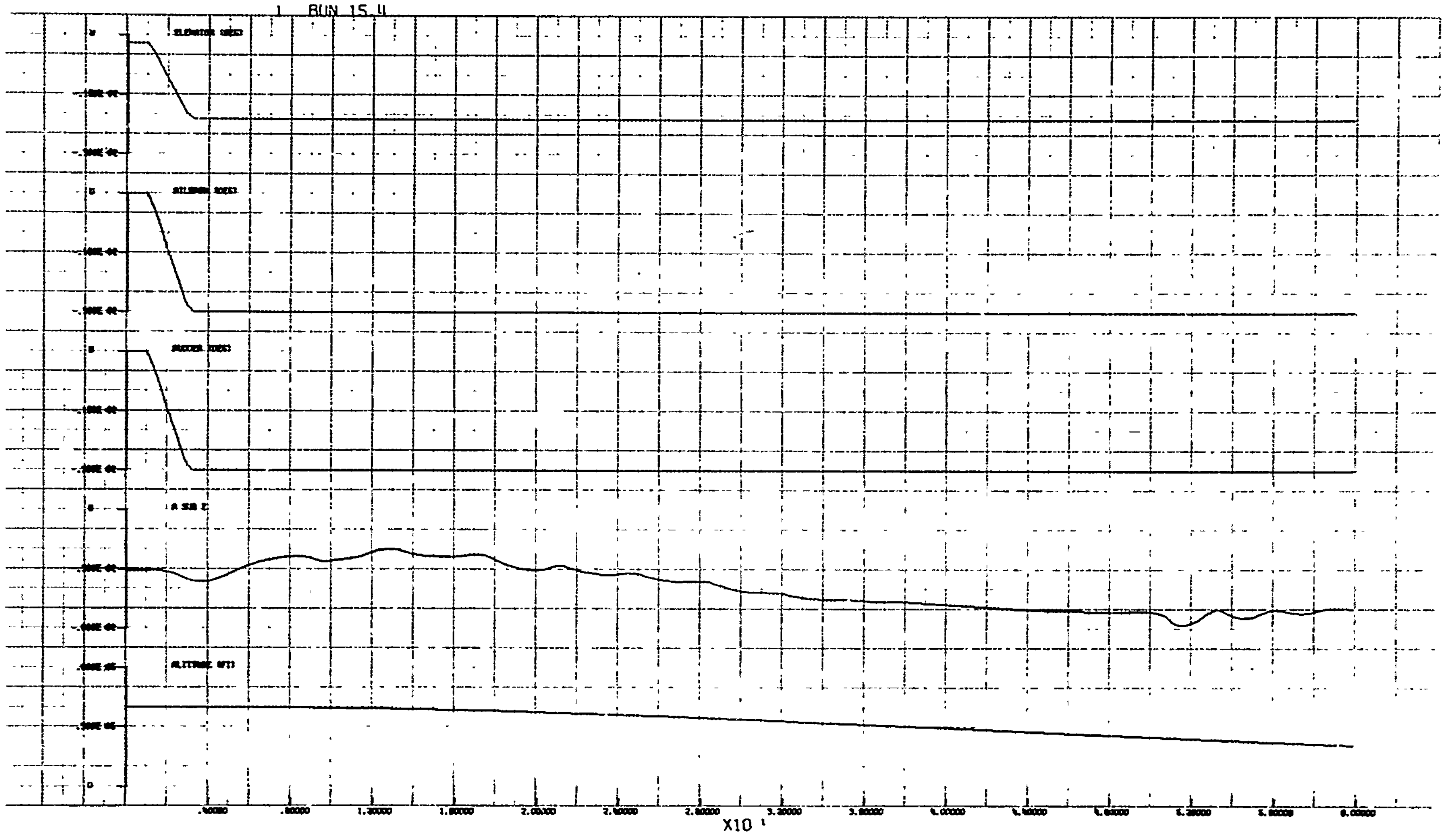


Figure 6-14. Spin Initiated with Coordinated Aileron and Rudder Deflection (Continued)

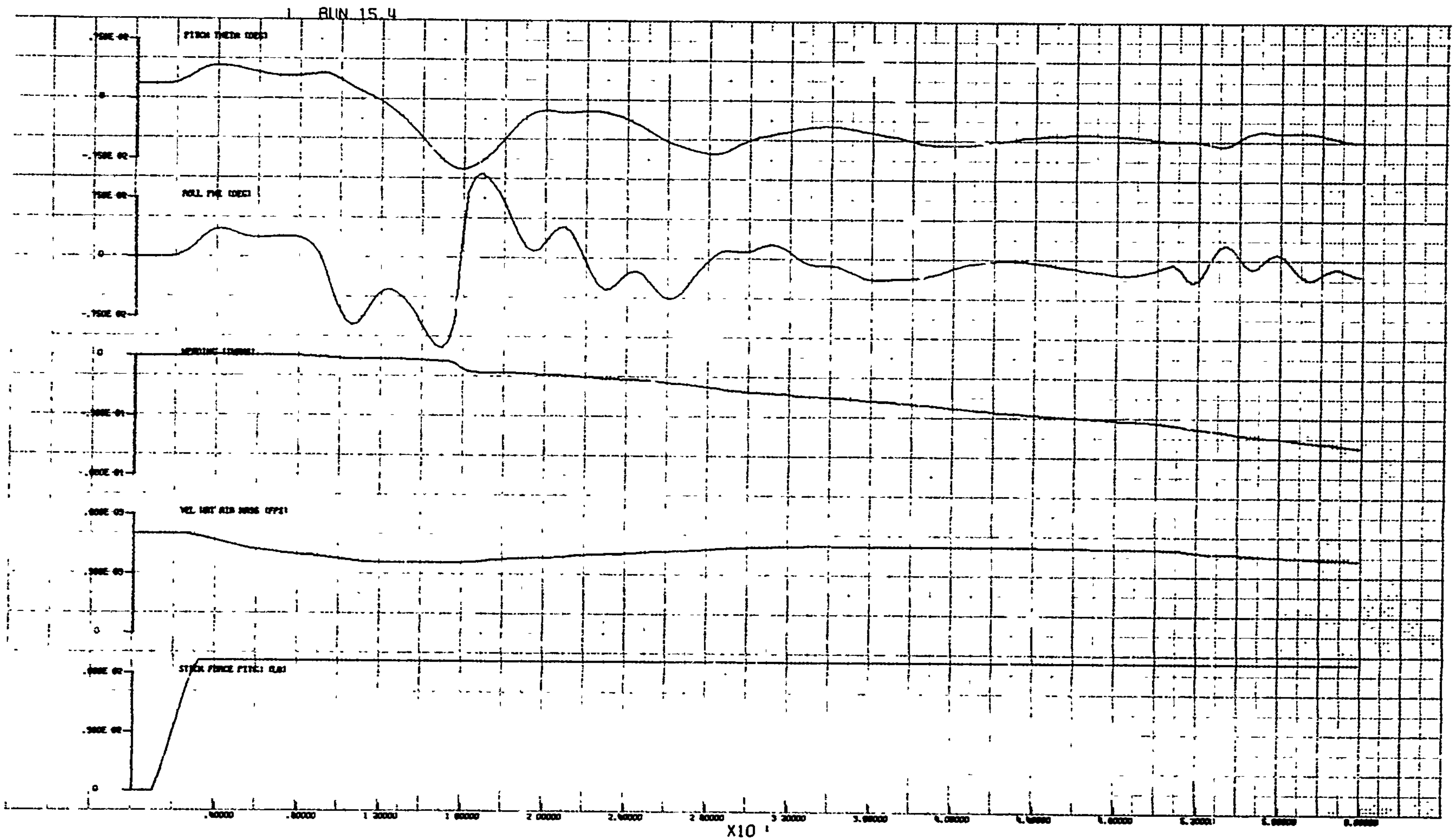


Figure 6-14. Spin Initiated with Coordinated Aileron and Rudder Deflection (Concluded)

where both the aileron and rudder were deflected at the same rate and in a direction so as to cause a positive roll. Figure 6-14 also shows that sideslip was kept reasonably small for the first 6 seconds after controls were applied. After the ailerons and rudder attained their deflection limits, the yawing moment due to aileron could no longer be balanced by the yawing moment due to rudder. This resulted in a negative yaw rate buildup which was aggravated by the lateral-directional instability occurring because the angle of attack was less than 37 degrees. The buildup of yaw rate resulted in a corresponding positive buildup in sideslip, which in turn caused a roll reversal through the aircraft dihedral effect. The increase in a positive product of yaw and roll rates caused a positive pitching moment and, hence, drove the vehicle into a spin condition.

Figure 6-14 does illustrate that, so long as adequate control authority remained, departure was inhibited through the use of coordinated controls. The prevention of departure could have been more effective if the rudder had been deflected in a more opportune fashion (generally leading ailerons), such as would be provided by a closed-loop coordination system. The task of inhibiting departure would have been more difficult than shown in Figure 6-14 if less than full trailing-edge-up elevator had been used. In this case more time would have been spent in the region of lateral-directional static instability. Hence, a greater differential in relative deflections between rudder and aileron would be required to prevent departure.

Now consider the effects of feedback controls upon the departure process. In the pitch axis the augmentation system (acceleration plus rate feedback) will tend to inhibit undesired angle-of-attack developments because of the superior controllability afforded. For departures initiated at low dynamic pressures (the usual case), the effects of a normal acceleration feedback will be essentially negligible when compared with the pitch-rate feedback. Hence, the pitch-axis augmentation will be essentially a pitch-rate command system at low-dynamic-pressure conditions near stall. This type of augmentation is beneficial at these conditions in that it provides superior control of angle of attack.

This conclusion may appear contradictory to some adverse experiences with augmentation systems near stall. Difficulties have arisen with pitch augmentation having forward-loop integration, the latter attempting to maintain a set ratio of feedback to stick force. Because the attainable normal acceleration decreases near stall, the associated pitch rate (per unit stick force) increases, producing an apparent reduction in stick gradient and overshoot into stall. This problem is alleviated by eliminating the integration above some AOA and permitting the normal system "droop" to prevent overcontrolling. Contributing also to apparent feedback control deficiencies near stall may be peculiarities of the particular test system. For example, use of stick-force command signals in the F-4 was inhibited near stall by the loss in feel system forces at high AOA.

The yaw-axis feedbacks are generally beneficial in that they will also attempt to inhibit departure. However, yaw-axis augmentation (usually consisting of high-passed yaw rate and lateral acceleration) is less effective than the pitch augmentation near stall. As angle of attack increases, the rudder effectiveness decreases rapidly as evidenced in Figure IV-4 of Appendix IV. Consequently, attempts to utilize yaw feedbacks on the F-4 to provide static stability augmentation in the intermediate angle-of-attack range will probably be futile. Also pertinent are the attainable coordination gains. As discussed in Section IV, lateral acceleration at low dynamic pressures is practically ineffective. A sideslip sensor in conjunction with an effective rudder would most likely correct the lateral stability problem, however.

Roll-rate feedback to the ailerons tends to aggravate departure. When roll-rate reversal occurs due to the increasing dominance of L_β over $L_{\delta a}$ with angle of attack, the roll-rate feedback signal will increase the aileron deflection to aggravate departure. For example, if as shown in Figure 6-7, the roll rate is negative after reversal, the roll-rate feedback will command a negative aileron deflection which in turn will increase the negative yawing moment. The negative yawing moment will in turn cause more positive sideslip which will result in increasing the rolling moment in the negative direction (through the dihedral effect, L_β). This type of action has motivated roll damper disablement above certain AOA ranges in some operational aircraft (e.g., the A-7). In this study of the F-4 aircraft, however, a roll-rate feedback with a gain of 0.4 rad/rad/sec was found to have a minor effect on departure[3]. This result was determined by comparing the departure caused by rudder and elevator with and without aileron feedback and noting little difference. The same conclusion can be drawn from a comparison of the lateral-directional roots with and without the roll-rate-to-aileron feedback for various angles of attack, as illustrated in Figure 6-15. Here the yaw feedbacks are set to zero. Note that dutch-roll instability appears to occur slightly earlier (at lower AOA) and that the instability is slightly more pronounced with the rate feedback, but the effects are not major. Up to about 20 degrees AOA, the rate feedback contributes desirable damping.

The fact that the roll-rate feedback has little effect is attributed in part to rapid loss of aileron surface effectiveness with angle of attack. Had the roll-rate gain been increased to accommodate the loss in surface effectiveness, then it could be expected that the roll-rate feedback would significantly aggravate the departure. The degrading effects of a roll-rate feedback can be minimized through the use of proper turn coordination through the higher angle-of-attack ranges (assuming adequate rudder authority exists). The turn coordination would act to minimize the sideslip which causes the roll reversal. It is only when roll reversal occurs during the departure that a roll-rate feedback is degrading. Of course, if departure is initiated by a full rudder deflection, turn coordination will be useless.

[3]This gain level is considered to be a nominal gain level for normal flight.

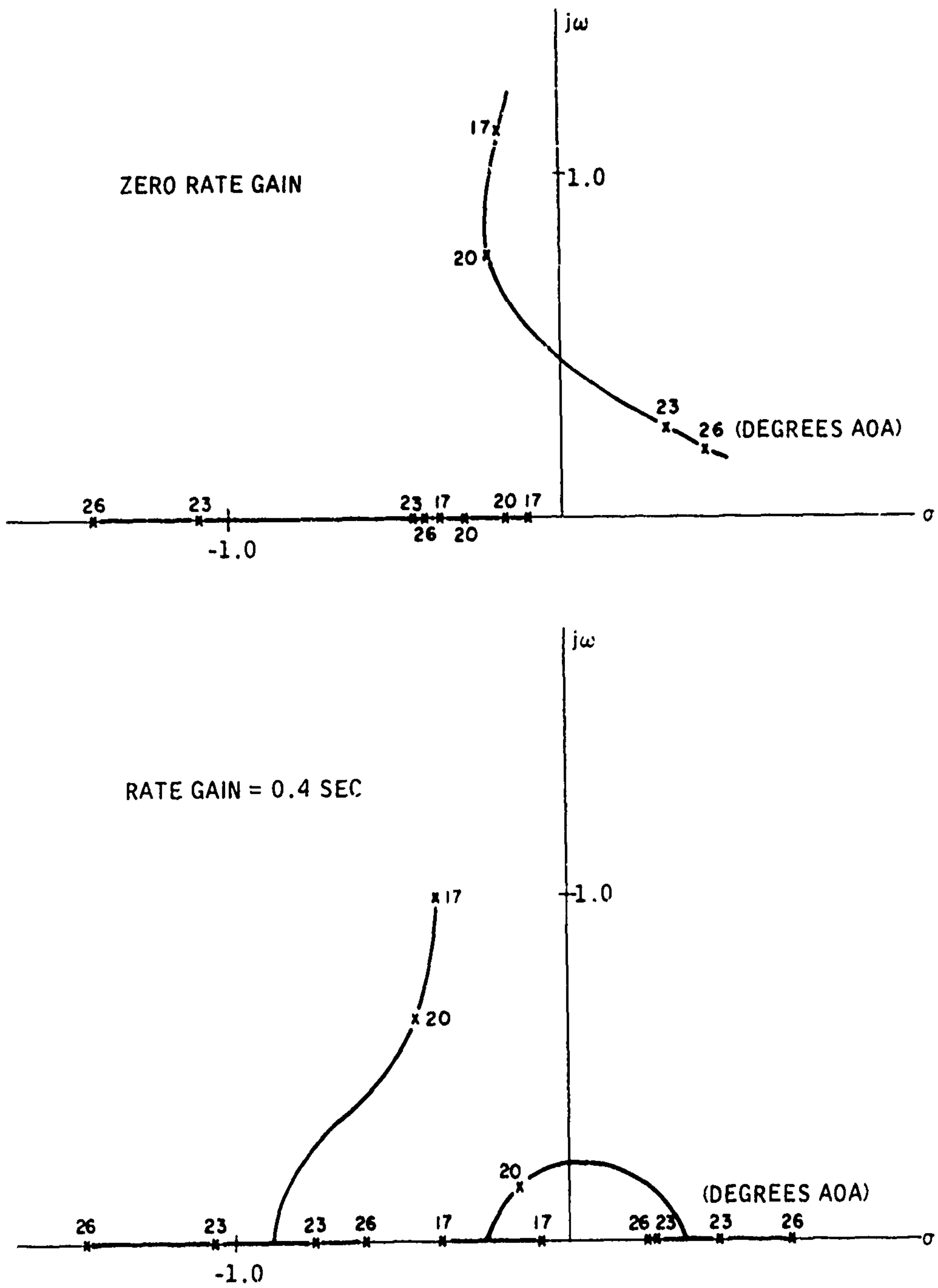


Figure 6-15. Effect of Roll-Rate-to-Aileron Feedback on Lateral-Directional Root Loci at Various Angles of Attack

It is concluded that with the existing aircraft properties and the tendency for pilot input via the rudder at high AOA, disablement of roll rate feedback is a prudent but relatively ineffective means for inhibiting departure.

Spin Evolution

Spin evolution was studied to determine the dominant aircraft parameters and the effects of feedback control on the spin growth. As pointed out in previous paragraphs, departure and subsequent spin evolution are very sensitive to the initial conditions (i.e., how departure from normal flight is executed). Examples of varying spin evolution as a function of applied entry controls were shown in Figures 6-7 through 6-14. Figure 6-7 shows a spin development which resulted in a flat spin. Figure 6-8 shows a fully developed oscillatory spin in which an apparent equilibrium was reached between the aerodynamic and inertial moments. Figure 6-9 shows a spin in which the inertial moments were insufficient to maintain the spin and resulted in partial recovery of angle of attack. However, when the angle of attack was reduced to the point where the lateral-directional axis became statically unstable, the conditions for spin entry were again established. Similar behavior is shown in Figures 6-10, 6-13 and 6-14. Many other types of spin behavior are also possible by changing the aircraft's inertial properties as has been reported in previous studies (see References 6-1 and 6-2).

A single spin time history, namely the one shown in Figure 6-7, was selected for analysis of spin evolution. Selection of a single spin was made in order to achieve controlled conditions for the analysis. From examination of the spin evolution in Figure 6-7, it is evident the spinning motion is comprised of an oscillatory component and a low-frequency drift. The dominant factors affecting the oscillatory component were studied by developing a linearized set of perturbation equations. These equations described the oscillatory motion about a "fixed" condition characterized by an average angle of attack, roll rate, and yaw rate. Table 6-1 presents the linearized equations. They were derived by assuming the aircraft variables could be represented by a linear combination of steady-state term and a perturbation term. Linearization is achieved by neglecting products of perturbation terms and, where applicable, by replacing the sine of an angle by the angle and the cosine by unity. Terms which appeared as bias terms in the equations were also neglected in this derivation. The equation for angle-of-attack rate shown in Table 6-1 was obtained by combining the force equations along the X and Z axes, under the assumption of constant total velocity.

The characteristic eigenvalues obtained by solving these equations were computed for the spin shown in Figure 6-7 at 33 seconds. At this condition the following "steady state" parameters were used to compute the coefficient in the linearized equations:

α_3 = 70 deg θ = -25 deg

p_s = -25 deg/sec V = 372 ft/sec

r_s = -75 deg/sec ρ = 8.24 x 10^{-4} slugs/ft^3

Table 6-1. Perturbation Equations for Low-Dynamic-Pressure, High Angle-of-Attack Conditions

- $$\dot{\alpha} \cong q - (p_s \cos\alpha_s + V_s \sin\alpha_s)\,\beta + Z_w\alpha + \frac{Z_s}{V}\,\delta_e$$

- $$\dot{q} \cong \left(\frac{I_z - I_x}{I_y}\,p_s\right) r + \left(\frac{I_z - I_x}{I_y}\,r_s\right) p + M_\alpha\alpha + M_q q + M_{\delta e}\delta e$$

- $$\dot{\beta} \cong (p_s \cos\alpha_s + r_s \sin\alpha_s)\alpha + (\sin\alpha_s)\,p - (\cos\alpha_s)\,r + y_v\,\beta + \left(\frac{g}{V}\cos\theta_s\right)\phi$$

- $$\dot{r} \cong \left[\left(\frac{I_x - I_y}{I_z} + \frac{I^2_{xz}}{I_x I_z}\right) p_s + \frac{I_{xz}}{I_z}\left(\frac{I_y - I_z}{I_x} - 1\right) r_s\right] q + N'_\beta\,\beta + N'_{\delta a}\,\delta_a + N'_{\delta r}\,\delta_r + N'_r r + N'_p p$$

- $$\dot{p} \cong \left[\left(\frac{I_y - I_z}{I_x} - \frac{I^2_{xz}}{I_x I_z}\right) r_s + \frac{I_{xz}}{I_x}\left(1 + \frac{I_{xz}}{I_x}\left\{\frac{I_x - I_y}{I_z}\right\}\right) p_s\right] q + L'_\beta\beta + L'_p p + L'_r r + L'_{\delta a}\delta a$$

where subscript "s" stands for steady-state value and the primes denote derivatives of the form

$$N'_\beta = \frac{N_\beta + \frac{I_{xz}}{I_x} L_\beta}{1 + \frac{I^2_{xz}}{I_x I_z}}$$

The resulting equations are shown in Table 6-2. The eigenvalues obtained are shown in Table 6-3. The first pair of roots represent the roll subsidence and roll spiral mode. The second part corresponds to the pitch short period and the third pair are the dutch-roll mode. It is the dutch-roll mode which is evident in the time history shown in Figure 6-7. The frequency and damping ratio obtained from Figure 6-7 at 33 seconds are approximately 2.41 rad/sec and 0.045 which are in good agreement with those shown in Table 6-3. The dutch-roll frequency can also be obtained from the following approximation derived from the equations in Table 6-1:

$$\omega_D^2 = N_\beta \cos \alpha_s - L_\beta \sin \alpha_s$$

Table 6-2. Perturbation Equations for t = 33 Seconds

- $\dot{\alpha} = q + 1.38\beta + 0.113\alpha + 0.00593\delta$
- $\dot{q} = -0.409r - 1.22p - 1.54\alpha - 0.825\delta - 0.565q - 0.590\beta$
- $\dot{\beta} = -1.38\alpha + 0.940p - 0.342r - 0.0122\beta + 0.0786\phi$
- $\dot{r} = 0.363q - 0.598\beta + 0.0685\delta_a - 0.0468r - 0.120p$
- $\dot{p} = 0.85q - 5.56\beta + 0.109\delta_a - 0.503p - 0.0072r$

Table 6-3. Eigenvalues Computed for t = 33 Seconds

Roots	Value	Damping Ratio	Frequency (rad/sec)
1	-0.036708 ± 0.003309j	0.996	0.0368
2	-0.326870 ± 1.942594j	0.166	1.9699
3	-0.143422 ± 2.390041j	0.0599	2.3943

This yields a dutch-roll frequency of 2.34 rad/sec which is in good agreement with the value shown in Table 6-3. It is evident from this expression that the dominant term characterizing the oscillation frequency is the dihedral effect, L_β. Significant inter-axis coupling precluded obtaining an accurate analytical expression for the damping ratio of the characteristic oscillation.

Eigenvalues were also computed using the equations for the 33-second time but with simplified roll and/or pitch axis rate feedbacks added. The pitch rate feedback used had a gain of 0.74 rad/rad/sec with a lead-lag compensation of

$$\frac{0.074S + 1}{0.02S + 1}$$

The roll-rate feedback used had a gain of 0.4 rad/rad/sec. These feedbacks were taken from the system shown in Figure 6-5. Tables 6-4 through 6-6 show the resultant eigenvalues. These results indicate very little effect on the short-period or dutch-roll frequencies and damping by the roll-rate feedback. This was anticipated because of the small surface effectiveness of the ailerons at 70 degrees angle of attack. These eigenvalues show both a marked increase in the short-period damping and a corresponding decrease in the dutch-roll damping due to the addition of a pitch-rate feedback term, however.

A linearized analysis of the effects of yaw-axis feedbacks was precluded by the fact that rudder surface effectiveness is directly related to the absolute value of sideslip angle (see Figure IV-4, Appendix IV). Hence, a linearized approximation of rudder effectiveness was not practical. But, because of the relatively small effectiveness of the rudder at high angle of attack, it can be concluded the yaw-axis feedbacks would at best have only a small effect.

Figures 6-16 and 6-17 show the effects of feedback in all three axes. In each of these runs the surfaces were held in saturation by large applied stick and pedal forces until 33 seconds. This was done to assure approximately the same "steady-state" conditions at 33 seconds as existed in the spin of Figure 6-7. At 33 seconds the applied forces were reduced in amplitude and reversed in direction thereby permitting the feedbacks to command surface deflections. The full complement of feedbacks used in an augmentation system were used to obtain the results in Figure 6-16. In Figure 6-17 the same feedbacks were used except the roll-rate feedback was excluded. Comparison of Figures 6-16 and 6-17 show the only noticeable effect of the roll-rate feedback was a small change in the yaw-rate buildup after feedback controls were applied. Figures 6-16 and 6-17 show a reduction in the damping of the oscillatory mode after the feedbacks were added. A reduction in damping was predicted by the linearized analysis and is due in part to the addition of a pitch-rate feedback. The linearized analysis did not predict the complete loss of damping shown in Figure 6-17 due to a pitch-rate feedback. The additional damping loss is apparently due to a change in the steady-state surface positions. The conclusion drawn from this analysis of feedback control is that the lateral-directional feedbacks have at best a small effect on spin evolution. The pitch-axis feedbacks, primarily pitch rate, can have a significant effect on the oscillatory properties of the spin. Addition of pitch-rate feedback tends to reduce the damping of the oscillation. The potential benefits or detriments of this action is discussed in the subsection on spin recovery.

The nonoscillatory behavior in spin evolution can be analyzed by examining the spin time history in Figure 6-7. Once the aircraft has gone through departure, the steady-state rate of growth of angle of attack is constant at 0.77 deg/sec. This buildup in angle of attack is attributed to both a small steady-state pitch rate and an increasing negative flight path angle. The change in flight path is due to the loss in lift. Evidence of the residual positive pitch rate is reflected in the pitch attitude response of Figure 6-7.

Table 6-4. Eigenvalues Obtained with Pitch-Rate Feedback, t = 33 Seconds

Roots	Value	Damping Ratio	Frequency (rad/sec)
1	$-0.035458 \pm 0.003549j$	0.995	0.0356
2	$-0.663638 \pm 1.767666j$	0.351	1.888
3	$-0.086724 \pm 2.467195j$	0.035	2.469

Table 6-5. Eigenvalues Obtained with Roll-Rate Feedback, t = 33 Seconds

Roots	Value	Damping Ratio	Frequency (rad/sec)
1	$s_1 = -0.035$, $s_2 = -0.0388$	--	--
2	$-0.319 \pm 1.938j$	0.162	1.96
3	$-0.1296 \pm 2.396j$	0.054	2.399

Table 6-6. Eigenvalues Obtained with Pitch- and Roll-Rate Feedbacks, t = 33 Seconds

Roots	Value	Damping Ratio	Frequency (rad/sec)
1	$s_1 = 0.033$, $s_2 = -0.0384$	--	--
2	$-0.653 \pm 1.763j$	0.347	1.88
3	$-0.0746 \pm 2.471j$	0.030	2.47

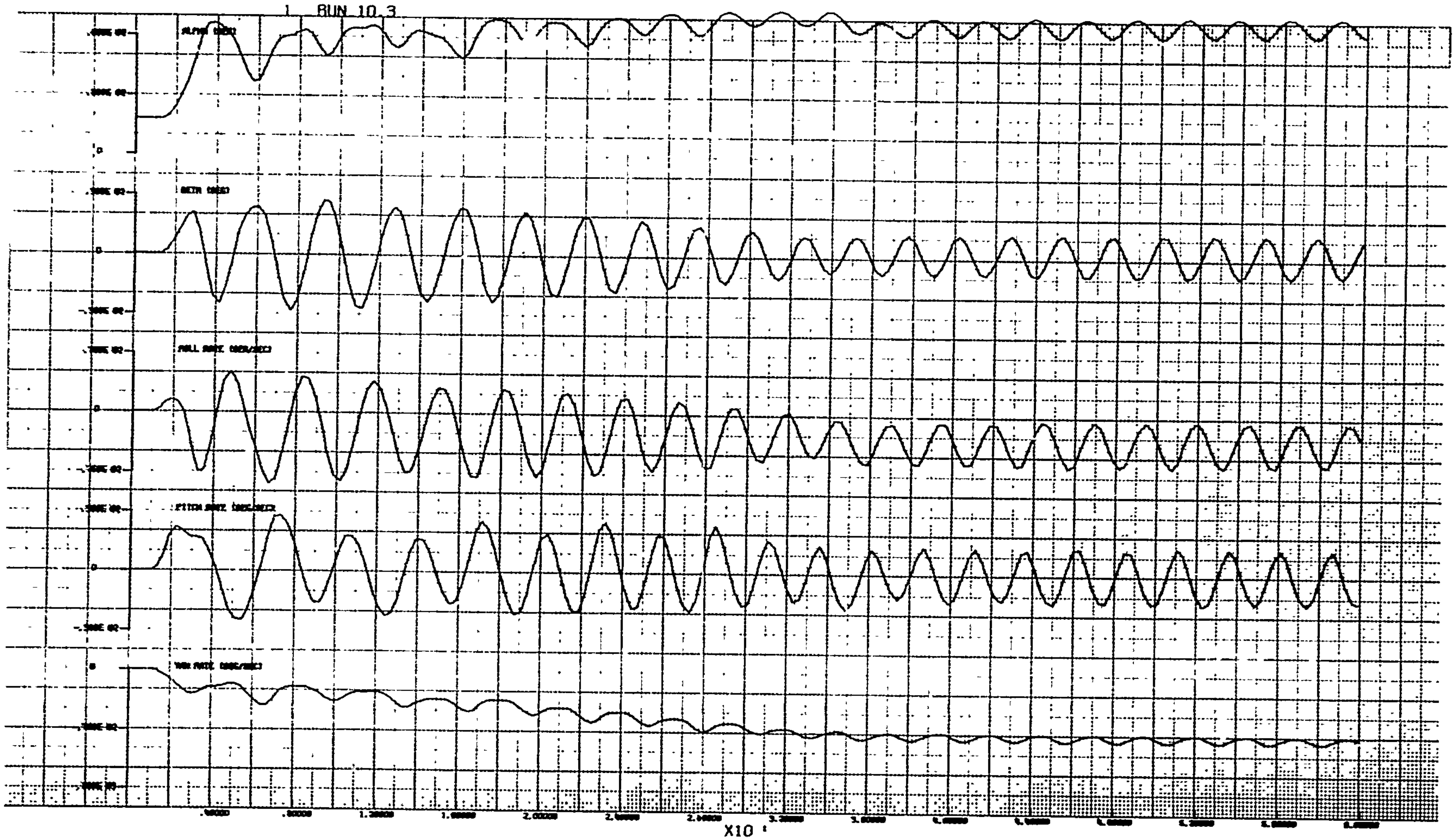


Figure 6-16. Effects of Three-Axis Feedback Control on Spin Evolution

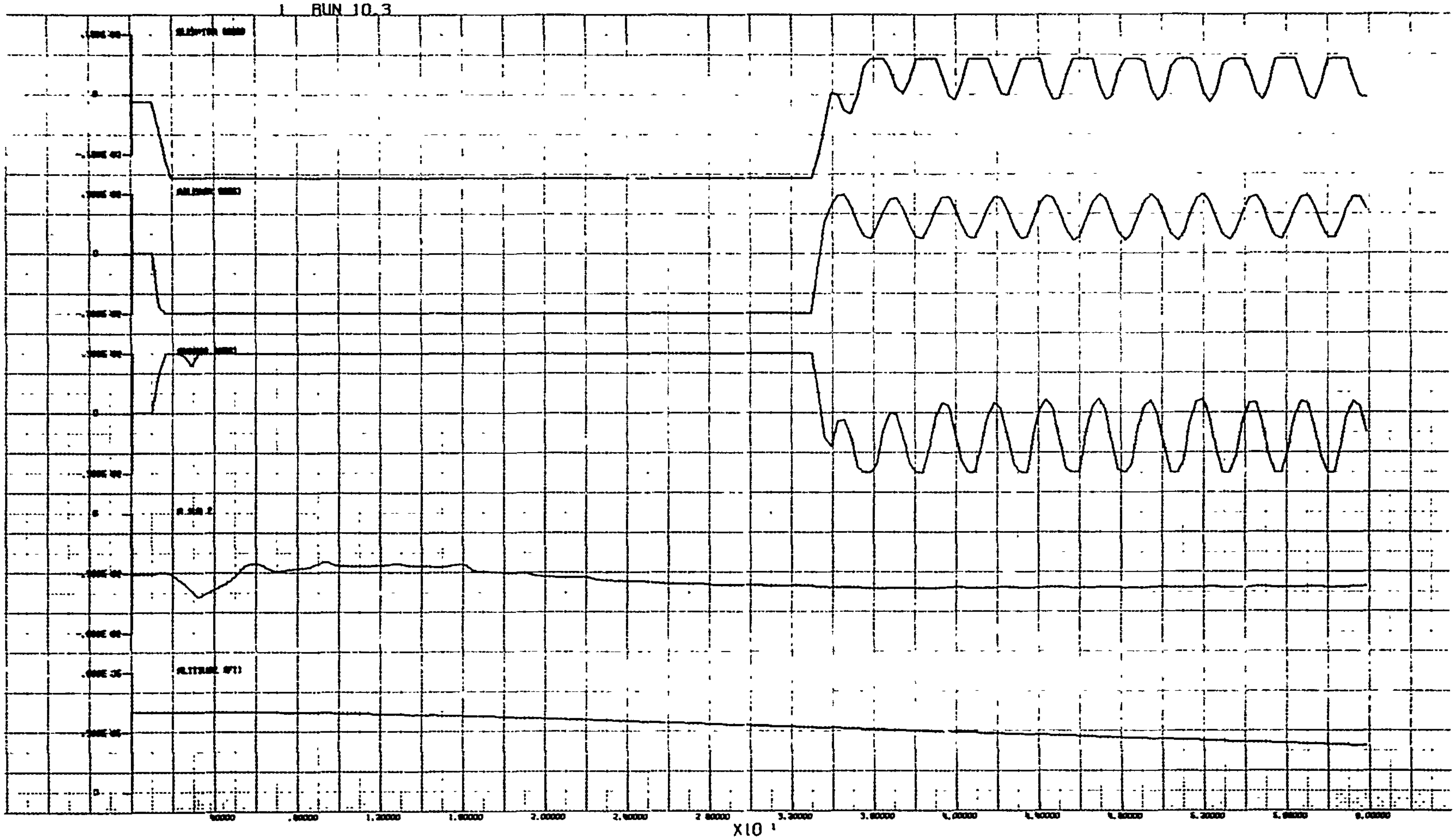


Figure 6-16. Effects of Three-Axis Feedback Control on Spin Evolution (Continued)

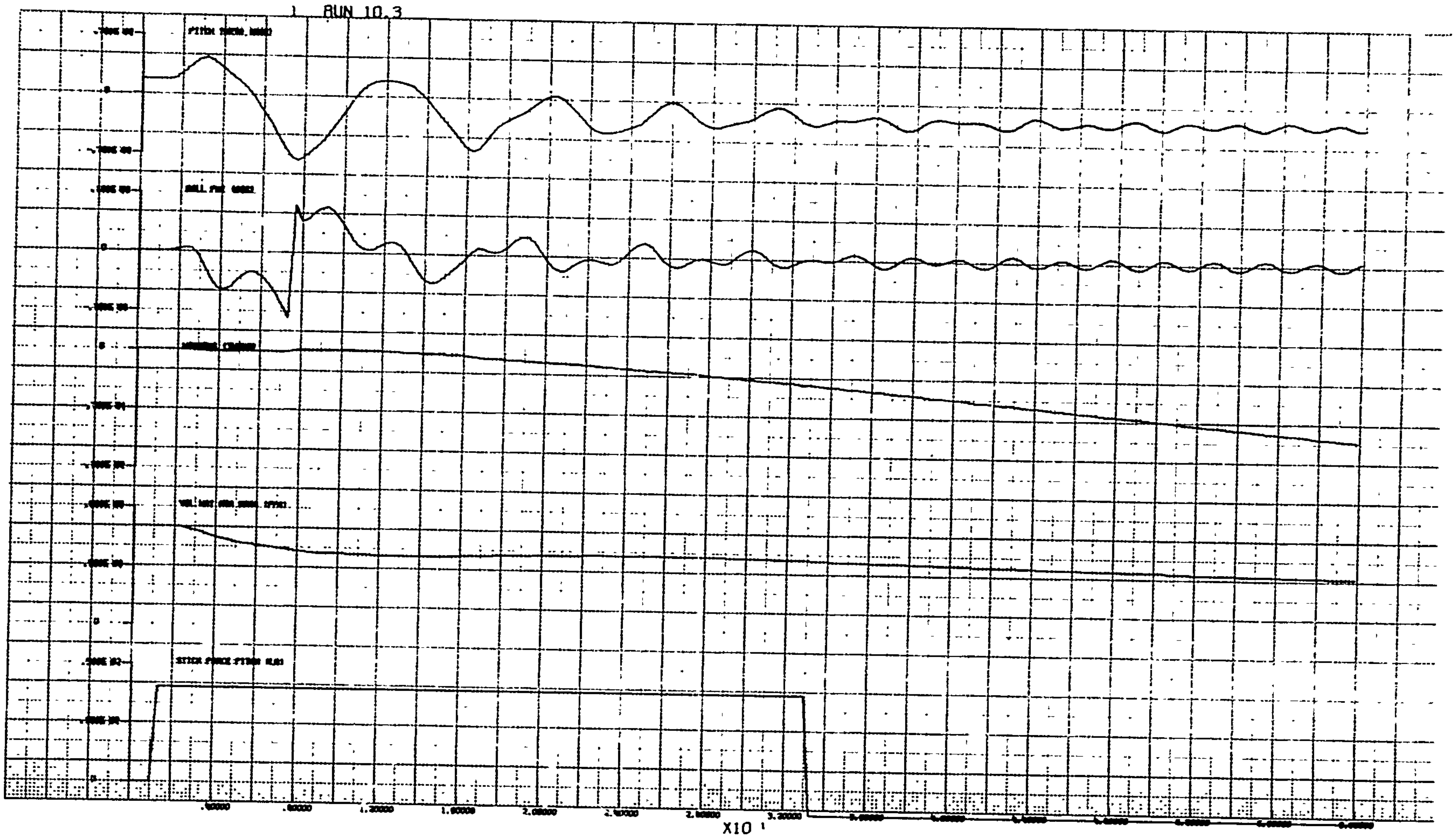


Figure 6-16. Effects of Three-Axis Feedback Control on Spin Evolution (Concluded)

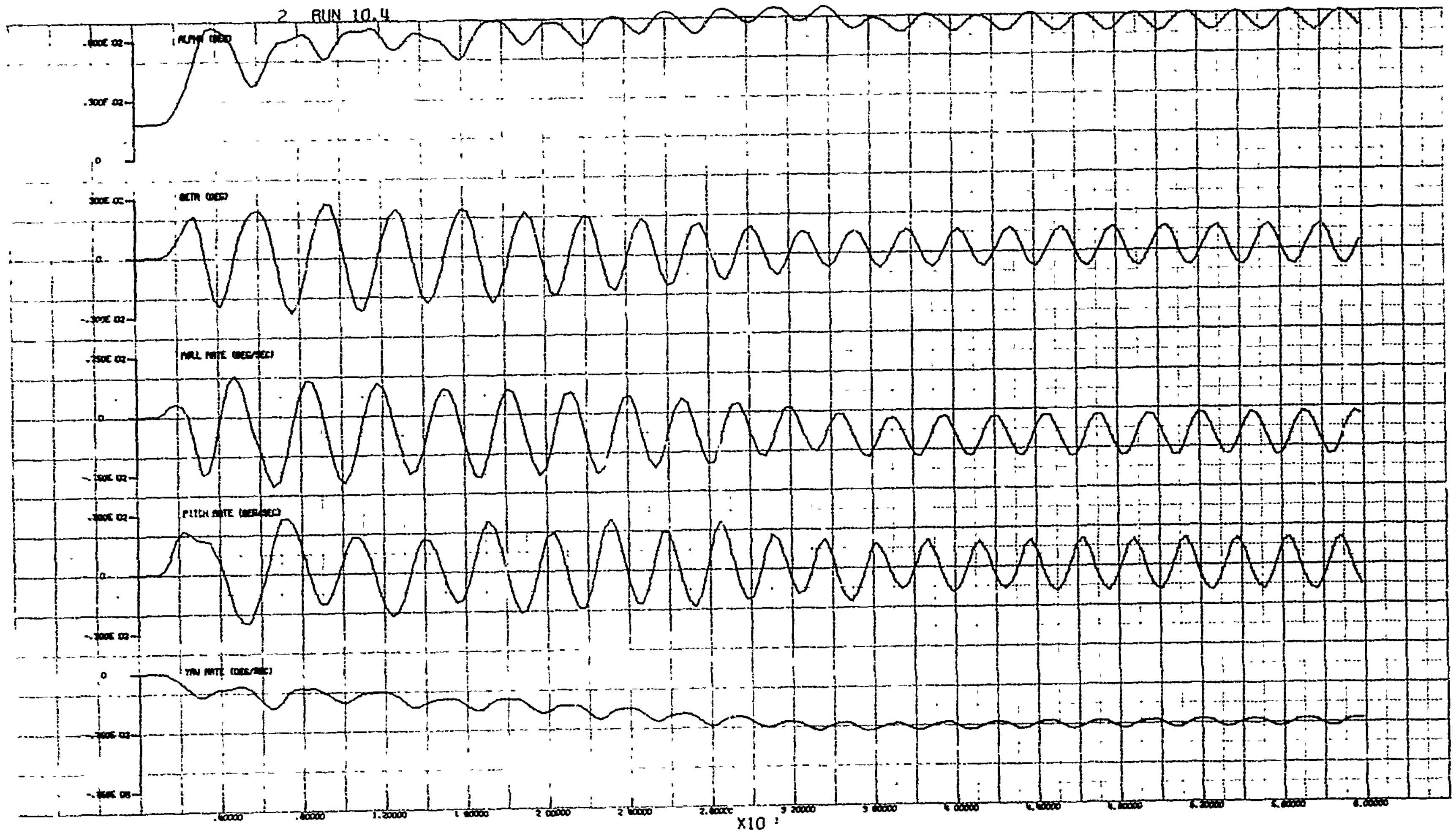


Figure 6-17. Effects of Pitch and Yaw Axis Feedback Control on Spin Evolution

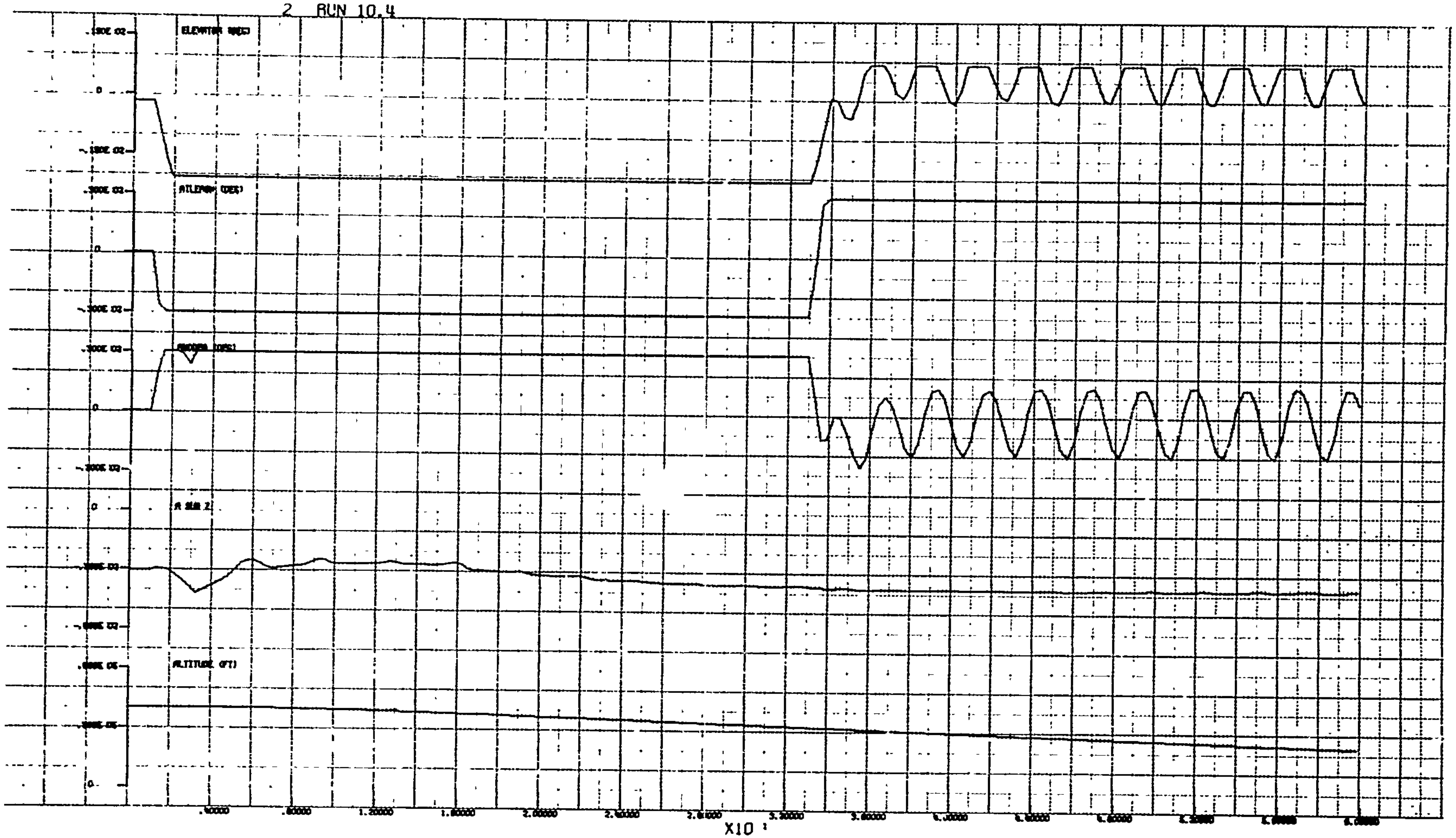


Figure 6-17. Effects of Pitch and Yaw Axis Feedback Control on Spin Evolution (Continued)

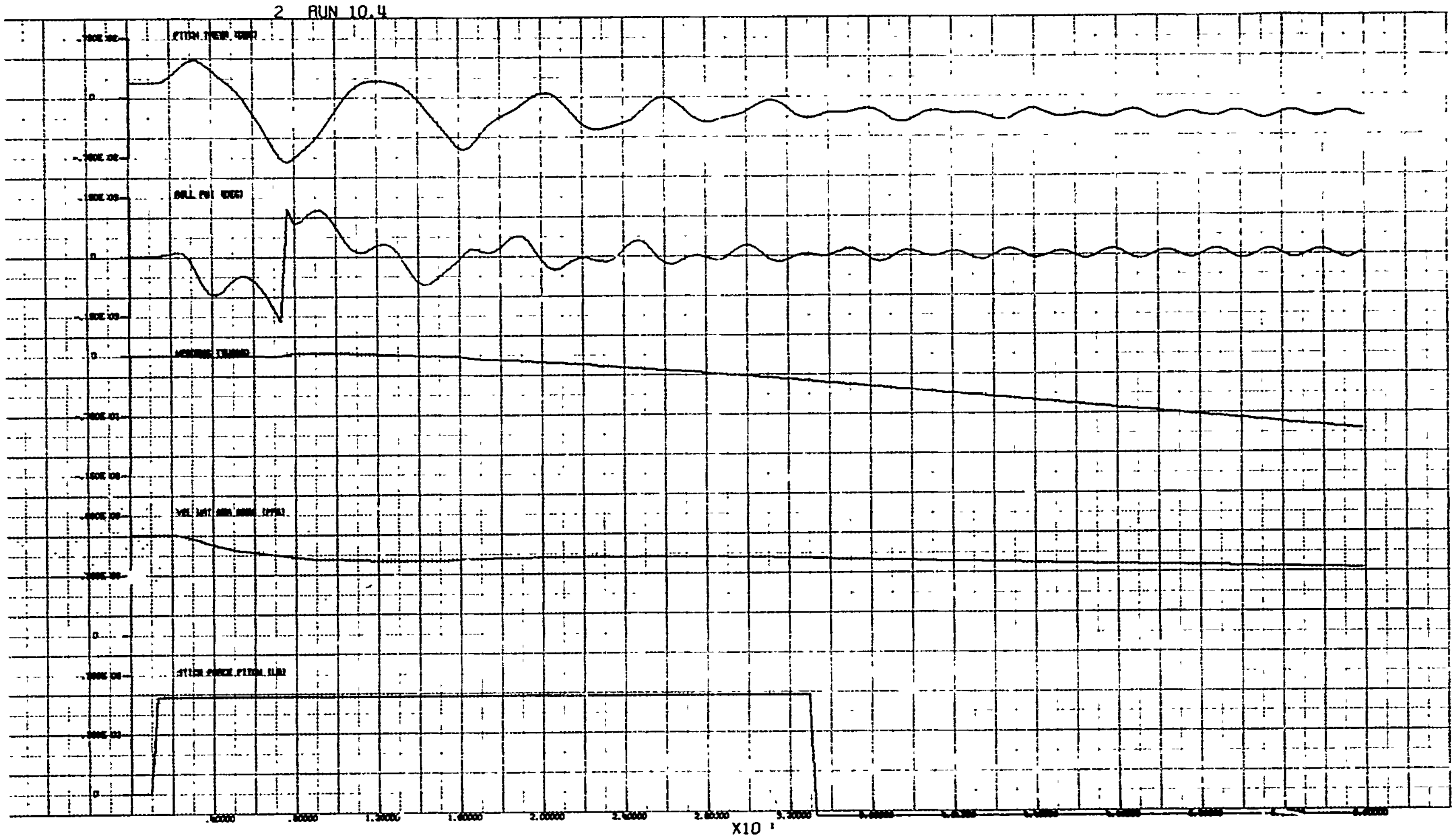


Figure 6-17. Effects of Pitch and Yaw Axis Feedback Control on Spin Evolution (Concluded)

The continuous increase in yaw rate shown in Figure 6-7 indicates that no balance was achieved among the yawing moments over the studied time intervals. Yawing moment due to aileron is the dominant influence on the yaw-rate buildup for at least the first 40 seconds of the spin. Beyond a time of 40 seconds, the yaw-rate buildup increases as a result of yawing moment due to spin rate.

In cases where ailerons were not used to initiate and maintain the spin (e. g., Figure 6-10) the yaw rate development could not be sustained. The combination of decreasing rudder effectiveness and a restoring inertial moment caused the yaw rate to decrease after a nearly equilbrium condition had been reached.

These results on spin evolution clearly point out the dominance of the elevator and the ailerons in a developing spin. Although the rudder plays a significant role in a departure, its reduced effectiveness at high angle of attack minimizes its impact on spin evolution. The contribution to spin evolution by the ailerons is not through production of significant rolling moments but rather through its contribution to total yawing moment via $N_{\delta a}$. Any control system which acts to minimize the adverse deflections of the aileron and elevator should be beneficial in inhibiting spin growth.

Controllability Limits

This aspect of the study was concerned with determining the boundary between recoverable and nonrecoverable stalled flight. If stalled flight is to be permitted, then it is essential to know the controllability limits of stalled flight to prevent a transition into a nonrecoverable condition. These results have direct application in implementation of an ultimate recovery system because they provide the basis for determining the automatic engagement of recovery controls.

The desired objective was to derive an analytical expression for the controllability limit. Ideally, this expression would relate the available recovery control power for all three axes to current values of the aircraft state (e. g., to body rates, angle of attack, etc.). Throughout this analysis it was assumed only the existing aerodynamic control surfaces would be used for the recovery; auxiliary control devices such as a spin chute were not considered. Nevertheless, it was an objective to obtain an analytical expression for the controllability which would be readily amenable to inclusion of auxiliary controls. A number of alternate approaches to establish a criteria were explored during the study. No single comprehensive analytical expression was obtained which would determine controllability of the total airplane. Instead, the resultant criteria established separate controllability limits for each axis of control. These limits were then compared to determine which was the most restrictive, and that limit was selected as the overall airplane controllability limit.

Limits for controllability in each axis were derived by examining the respective moment equations. It was hypothesized that the airplane would be out of control in a given axis whenever it was no longer possible to reverse the existing angular rate. Also implied in this analysis is the assumption that loss of control in any one axis implies loss of control of the entire airplane. A major difficulty in developing this controllability limit is the fact that the angular rates can exhibit both an oscillatory motion and a long-term low-frequency drift. Hence, it is not adequate to simply compare the available control moment against the existing moments at any given time point. A satisfactory comparison of the available control moment and the existing moment (aerodynamic and inertial) was obtained by integrating their sum and examining the slope of the resultant function.

Figure 6-18 shows the results obtained in the yaw axis for the spin shown in Figure 6-7. The existing yawing moment is obtained by measurement of yaw angular acceleration, $\dot{r}$. The available control moment is computed by summing the differential yaw moments that would be produced by deflecting the ailerons in a full pro-spin direction and the rudder in a full anti-spin direction. This combination produces the greatest possible yaw-axis restoring moment.

Figure 6-18 shows that the slope of the function goes to zero at approximately 45 seconds into the spin. This, then, represents the controllability limit in the yaw axis. Beyond this point it would be impossible to reduce to yaw rate with the ailerons and rudder because the resulting control moments will be less than the established moment (inertial and aerodynamic).

Figures 6-19 and 6-20 show plots at corresponding moment functions for the roll axis and pitch axis, respectively. Figure 6-19 indicates that the roll axis is initially uncontrollable and then becomes controllable after approximately 10 seconds into the spin. This behavior is explained by the fact that aileron reversal occurs for angles of attack greater than 55 degrees. The recovery controls used to compute the function shown in Figure 6-19 (pro-spin aileron) are in a direction to augment the spin for angles of attack less than 55 degrees. Both Figures 6-19 and 6-20 indicate there exists sufficient control power to overpower the existing pitching and rolling moments for the time period shown. Hence, the comparison of results in Figures 6-18, 6-19, and 6-20 indicates the yaw axis to be the critical axis in terms of the airplane controllability limit. The validity of this limit was tested by setting up the spin shown in Figure 6-7 and applying the specified recovery controls at differing recovery times. Figures 6-21, 6-22 and 6-23 show attempted spin recoveries initiated at 33, 40, and 50 seconds, respectively. No feedback controls were used. These results show recovery was attained when controls were initiated at 33 seconds but not at 40 or 50 seconds. As Figures 6 through 6-23 indicate, recovery controls were applied at a rate of 15 a sec thereby requiring 4 seconds to achieve full recovery control. Hence, even though recovery was initiated at 40 seconds, full restoring control moments were not achieved until 44 seconds. This is the approximate time at which the controllability limit should be reached. It is concluded that the

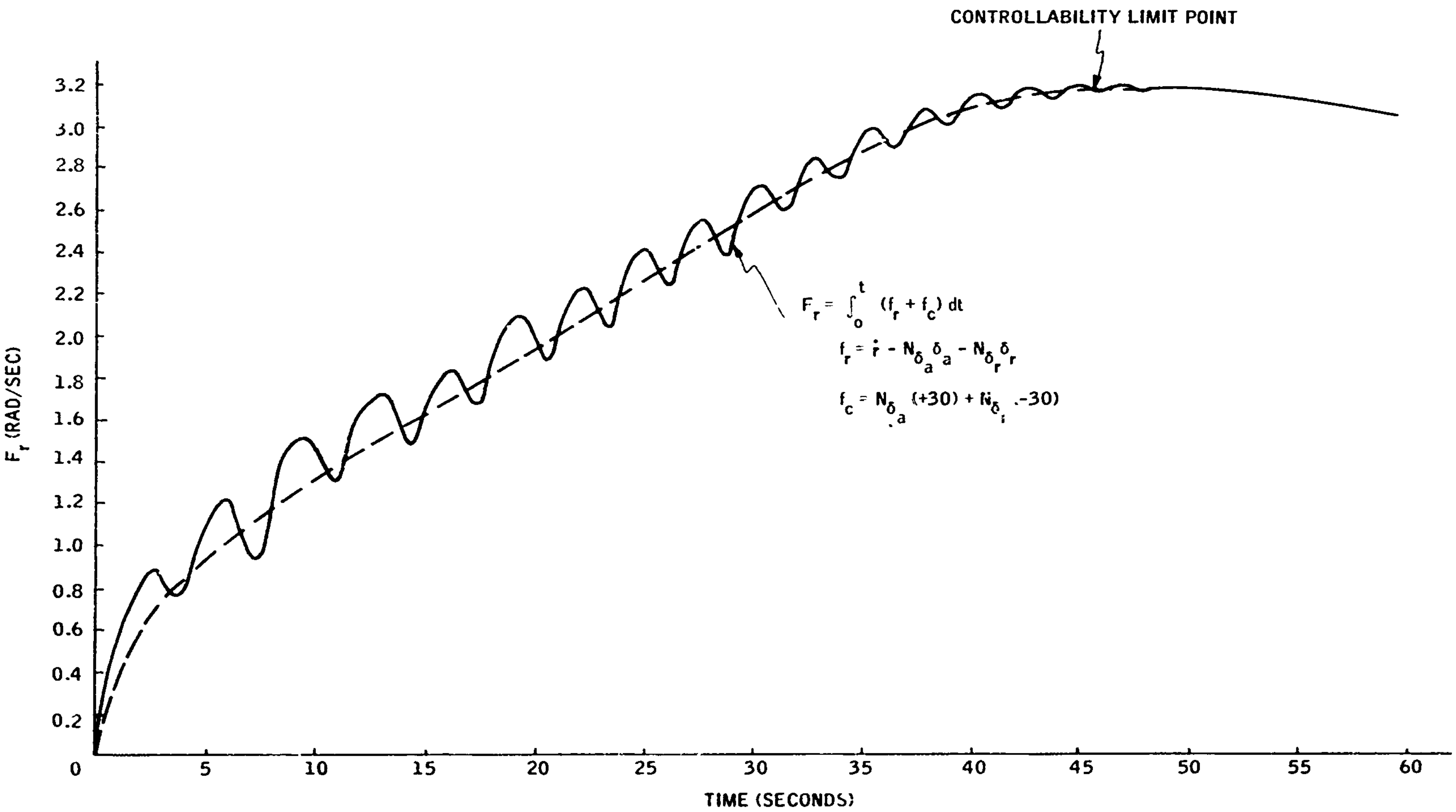


Figure 6-18. Yaw Axis Controllability Limit

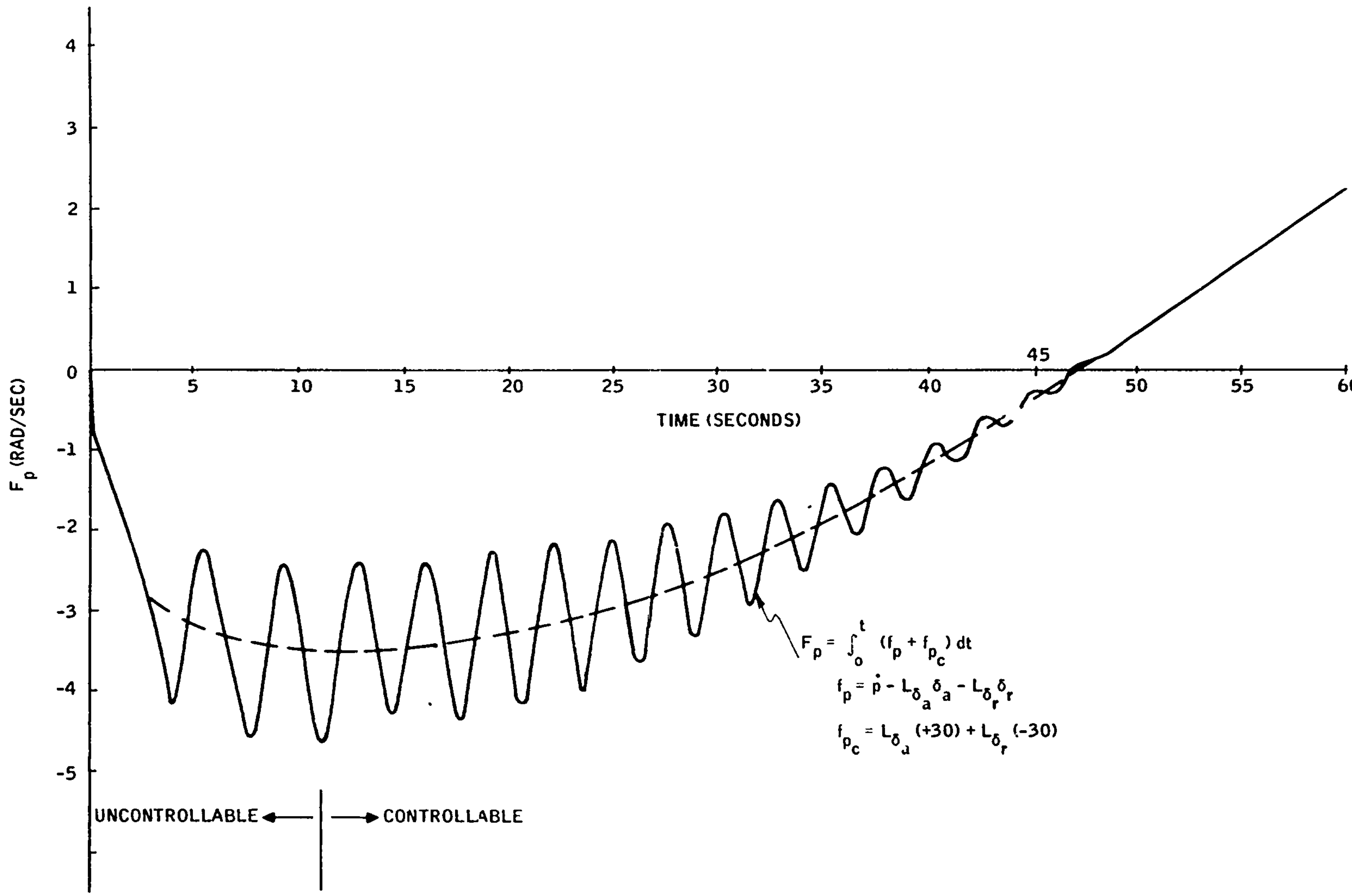


Figure 6-19. Roll Axis Controllability Limit

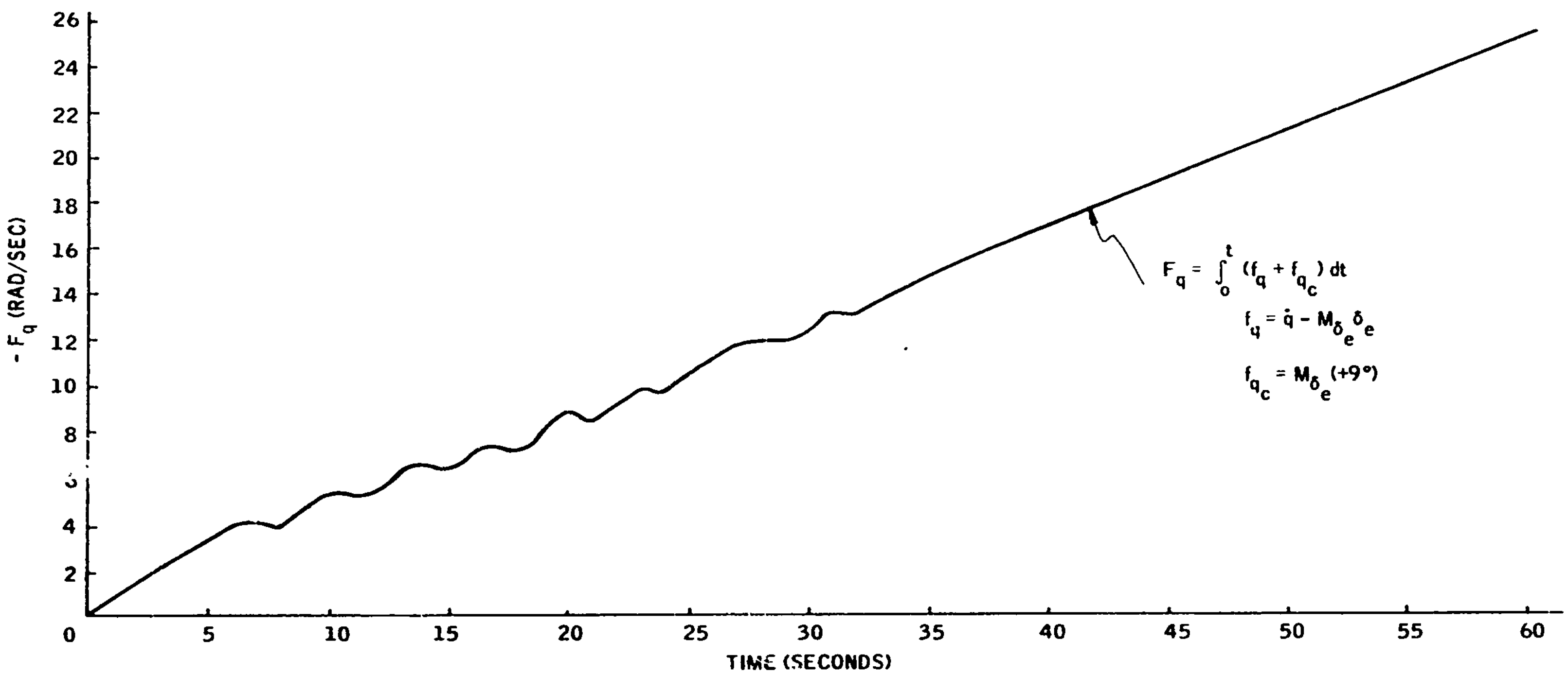


Figure 6-20. Pitch Axis Controllability Limit

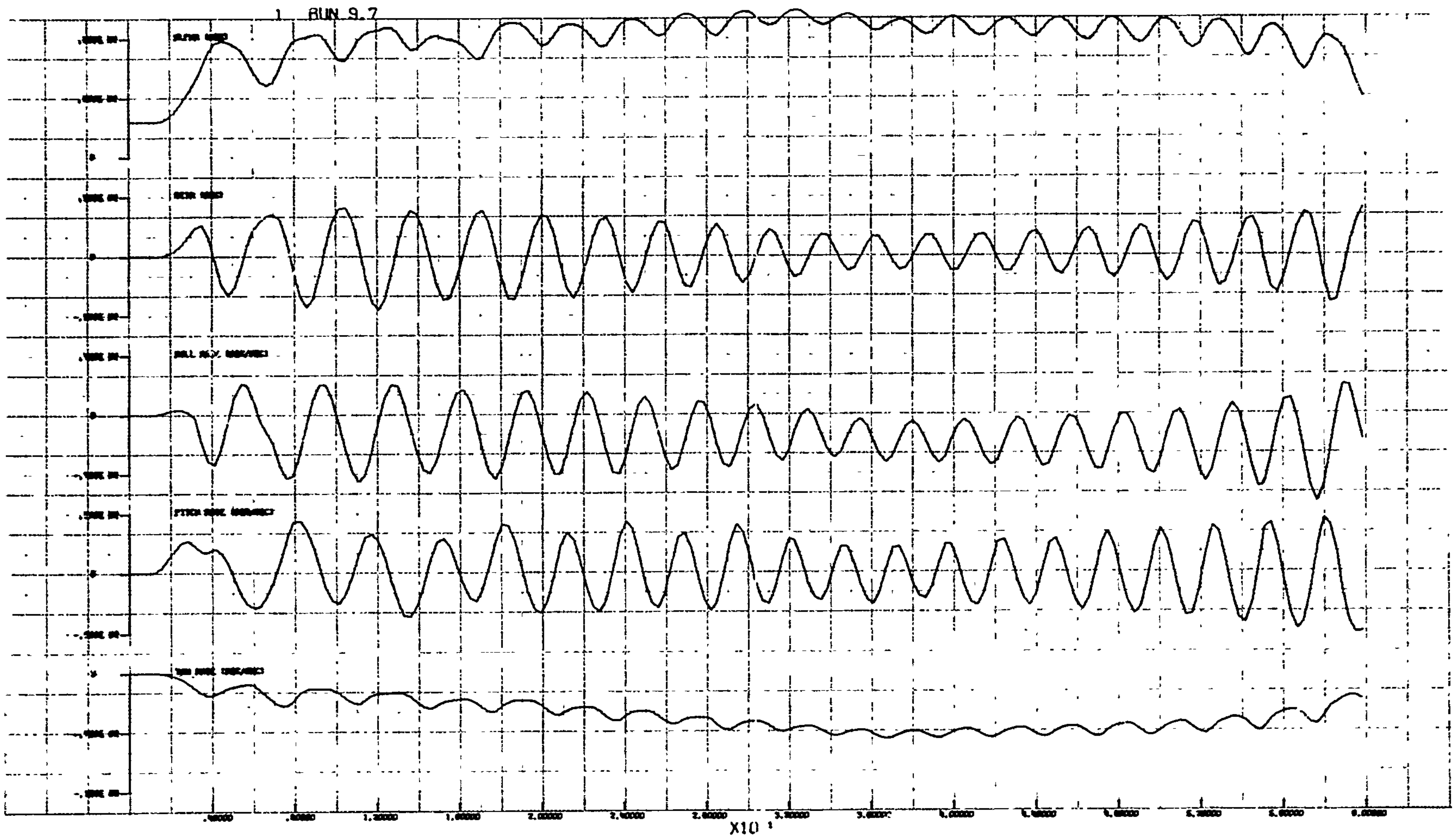


Figure 6-21. Spin Recovery Controls Applied at 33 Seconds

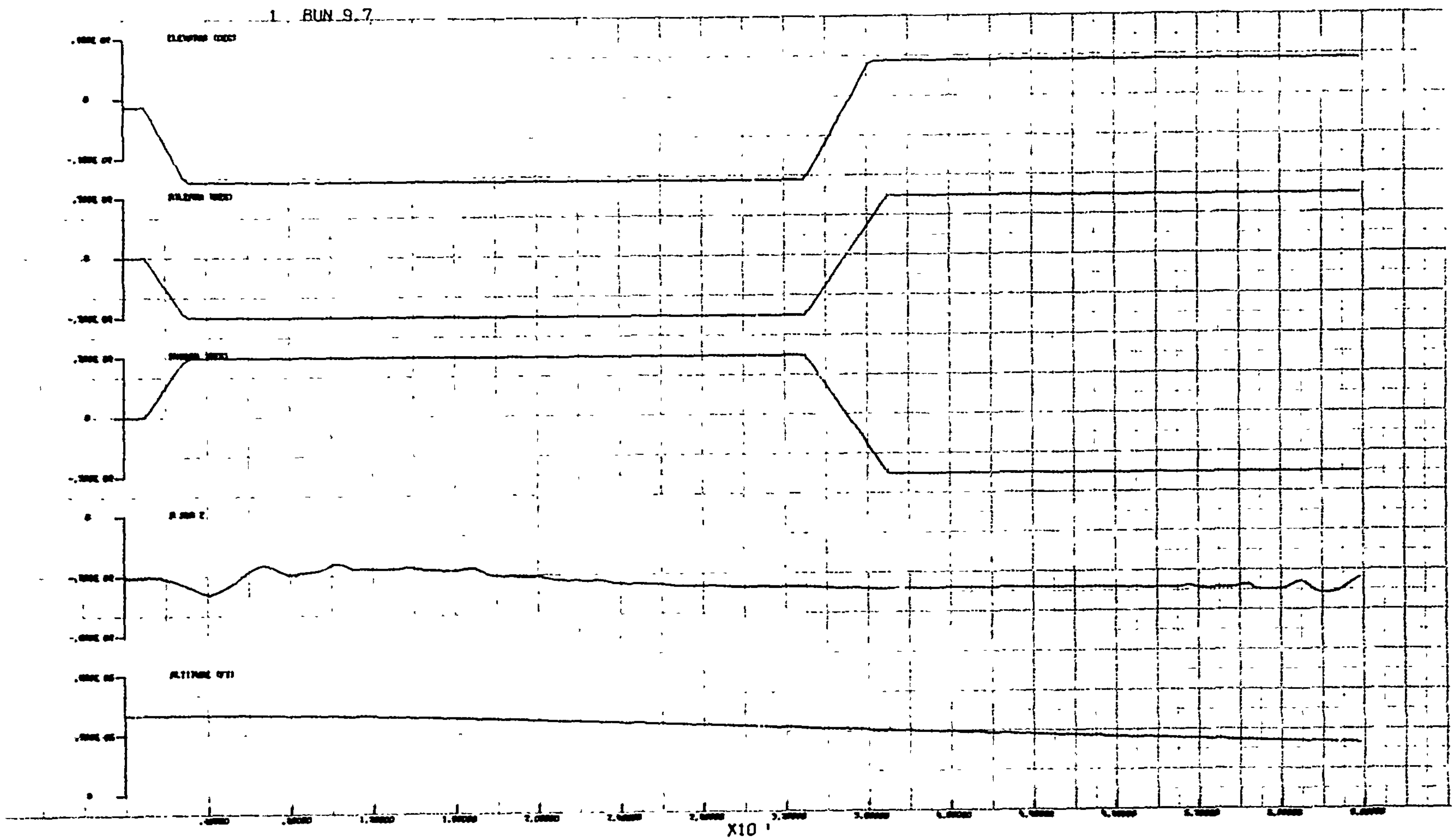


Figure 6-21. Spin Recovery Controls Applied at 33 Seconds (Continued)

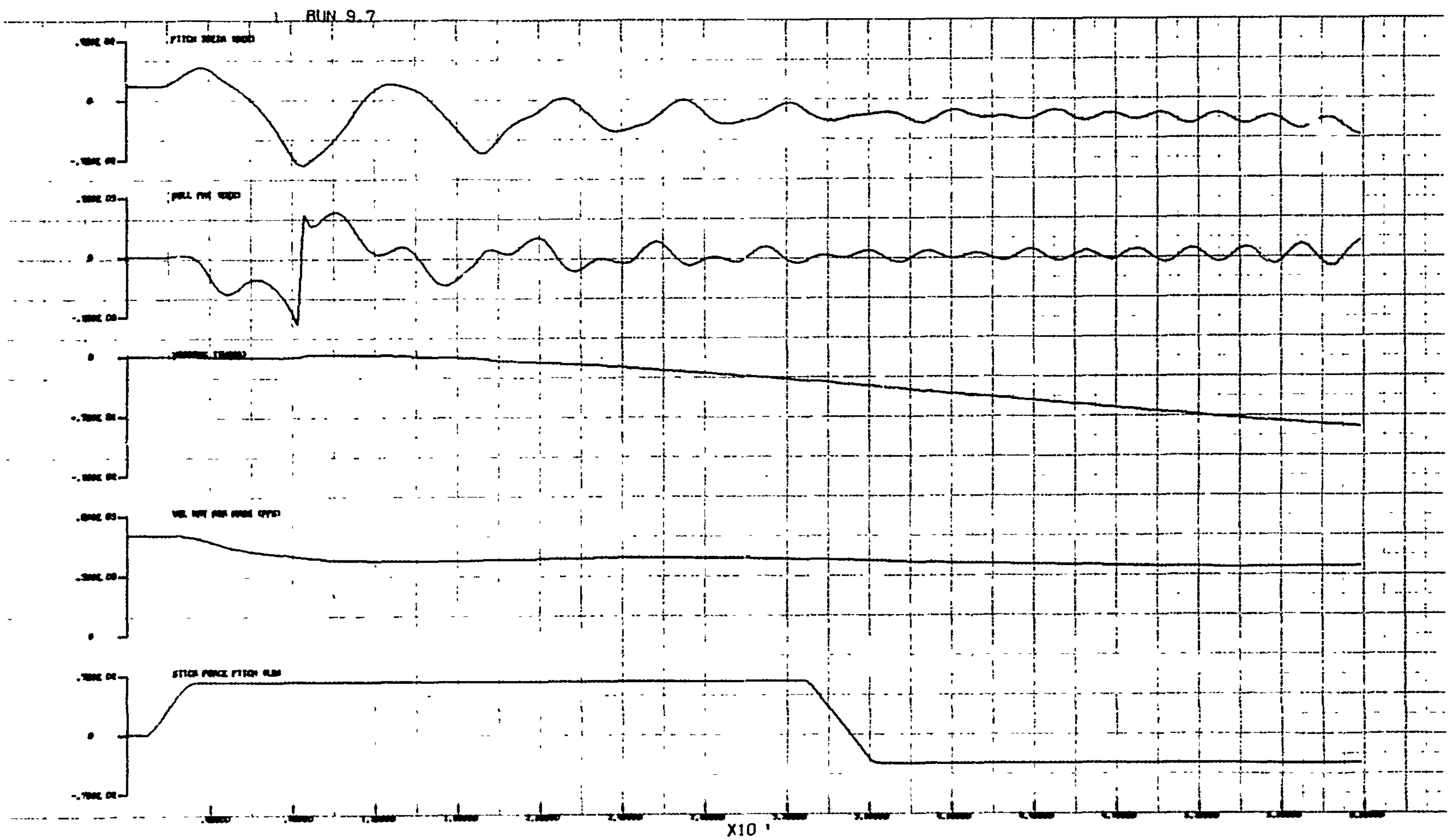


Figure 6-21. Spin Recovery Controls Applied at 33 Seconds (Concluded)

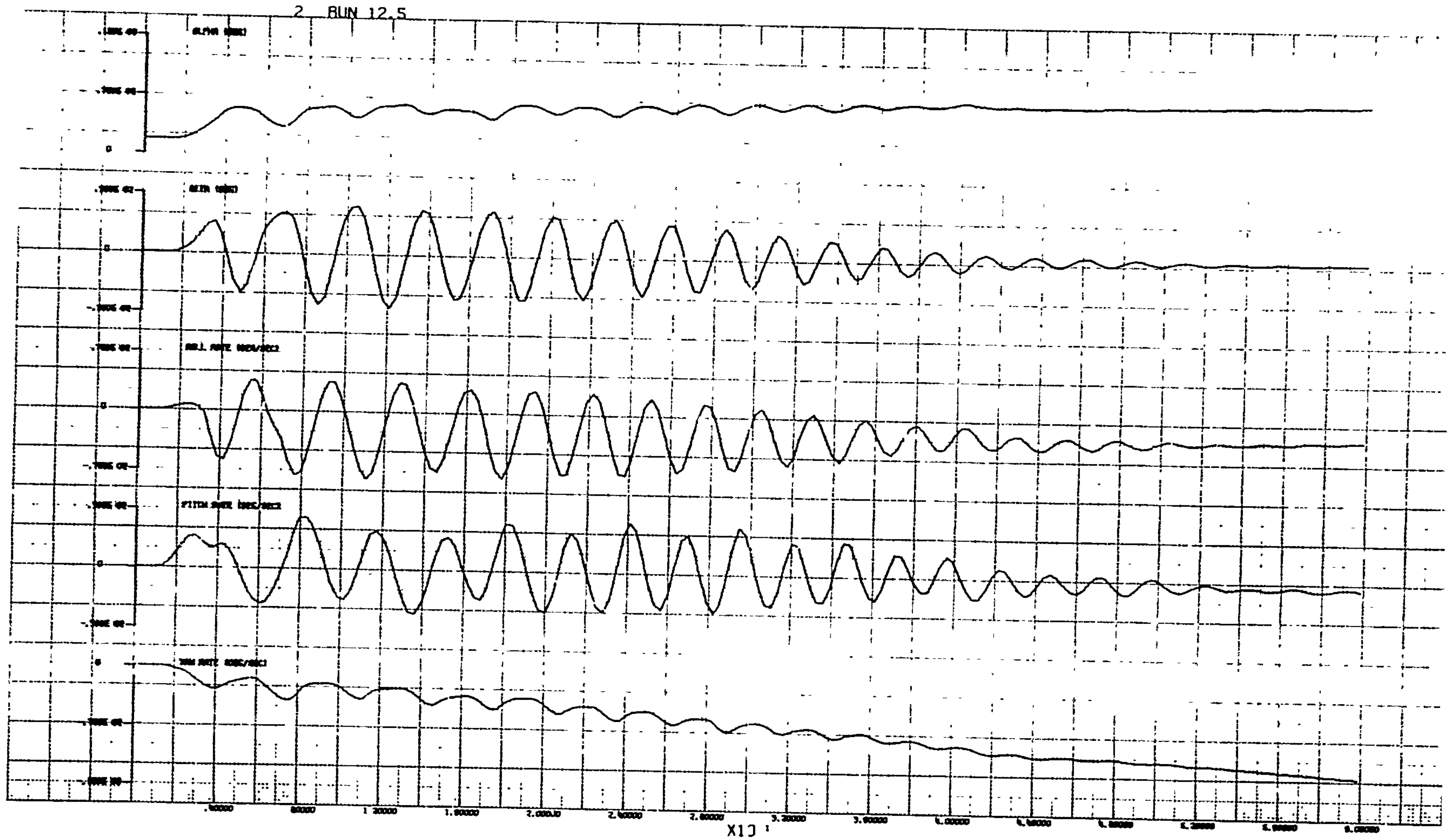


Figure 6-22. Spin Recovery Controls Applied at 40 Seconds

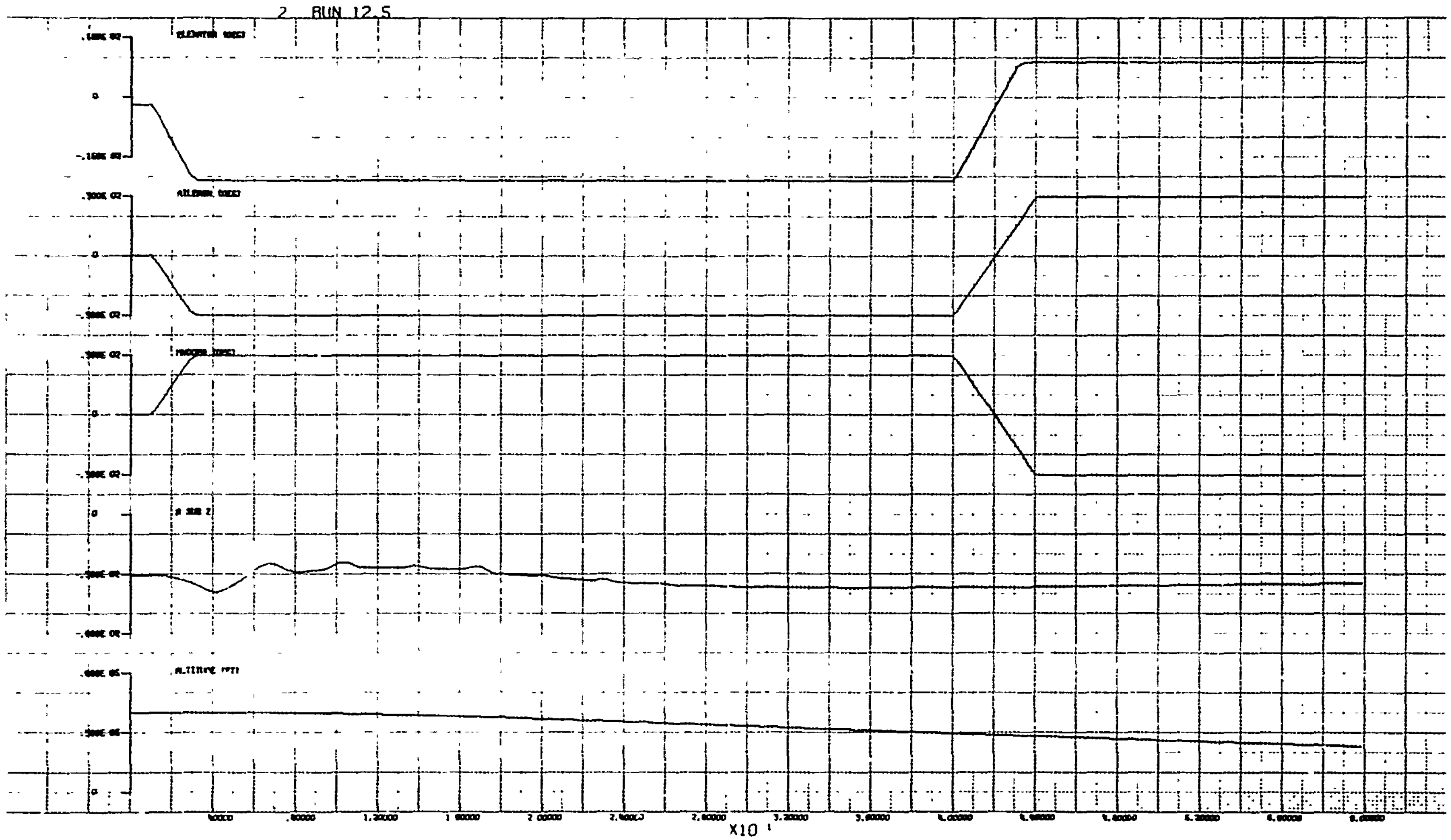


Figure 6-22. Spin Recovery Controls Applied at 40 Seconds (Continued)

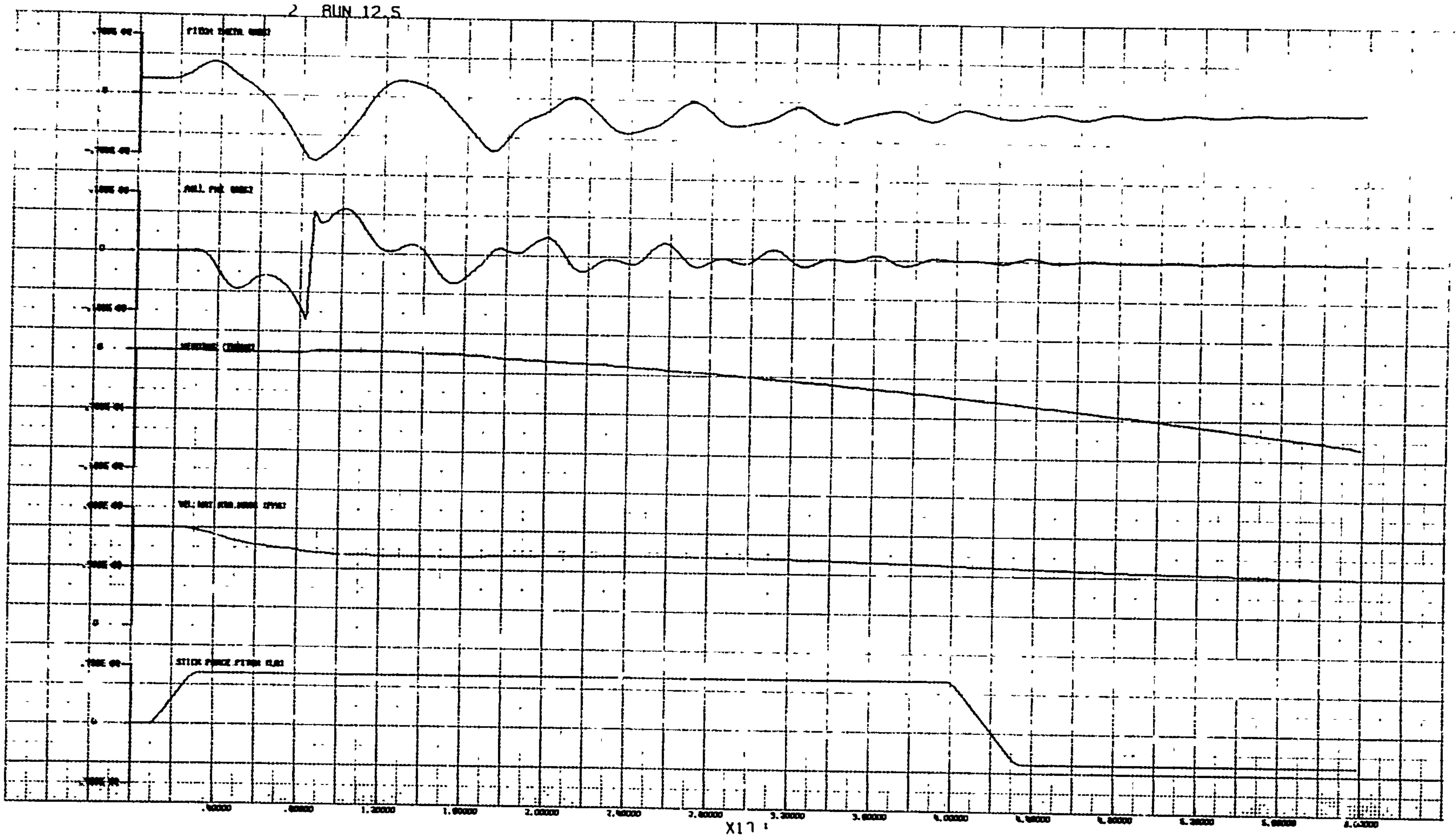


Figure 6-22. Spin Recovery Controls Applied at 40 Seconds (Concluded)

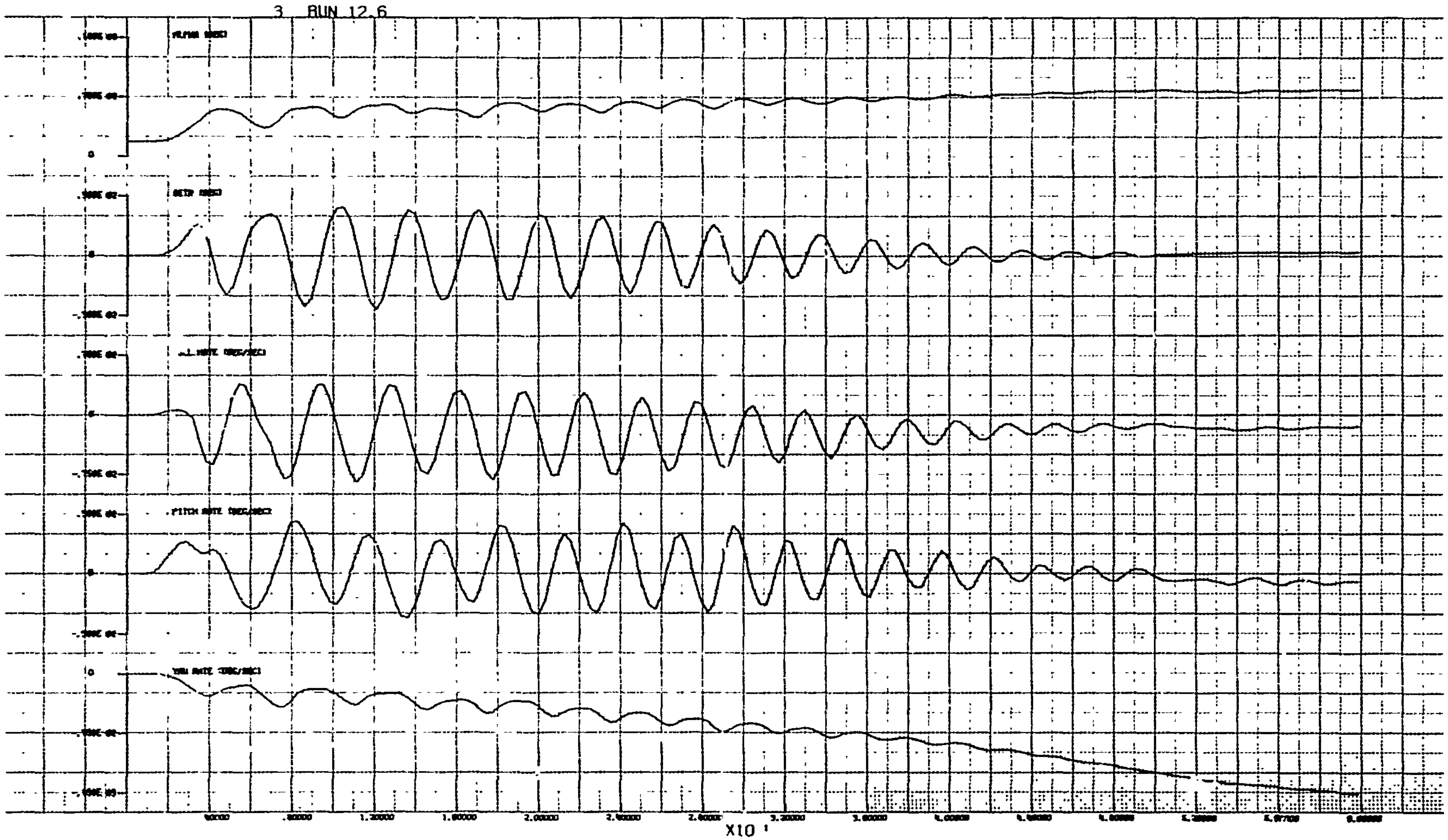


Figure 6-23. Spin Recovery Controls Applied at 50 Seconds

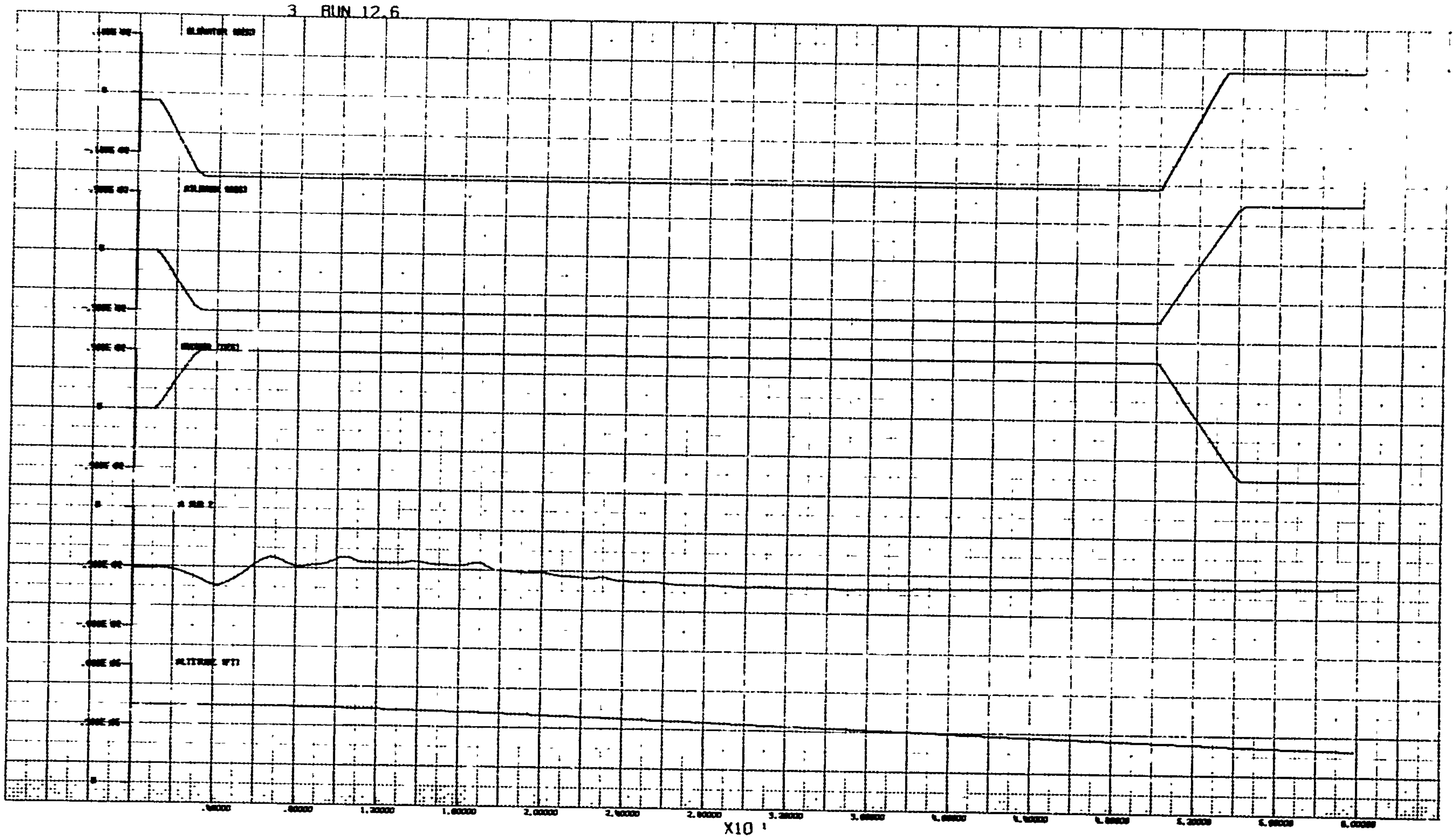


Figure 6-23. Spin Recovery Controls Applied at 50 Seconds (Continued)

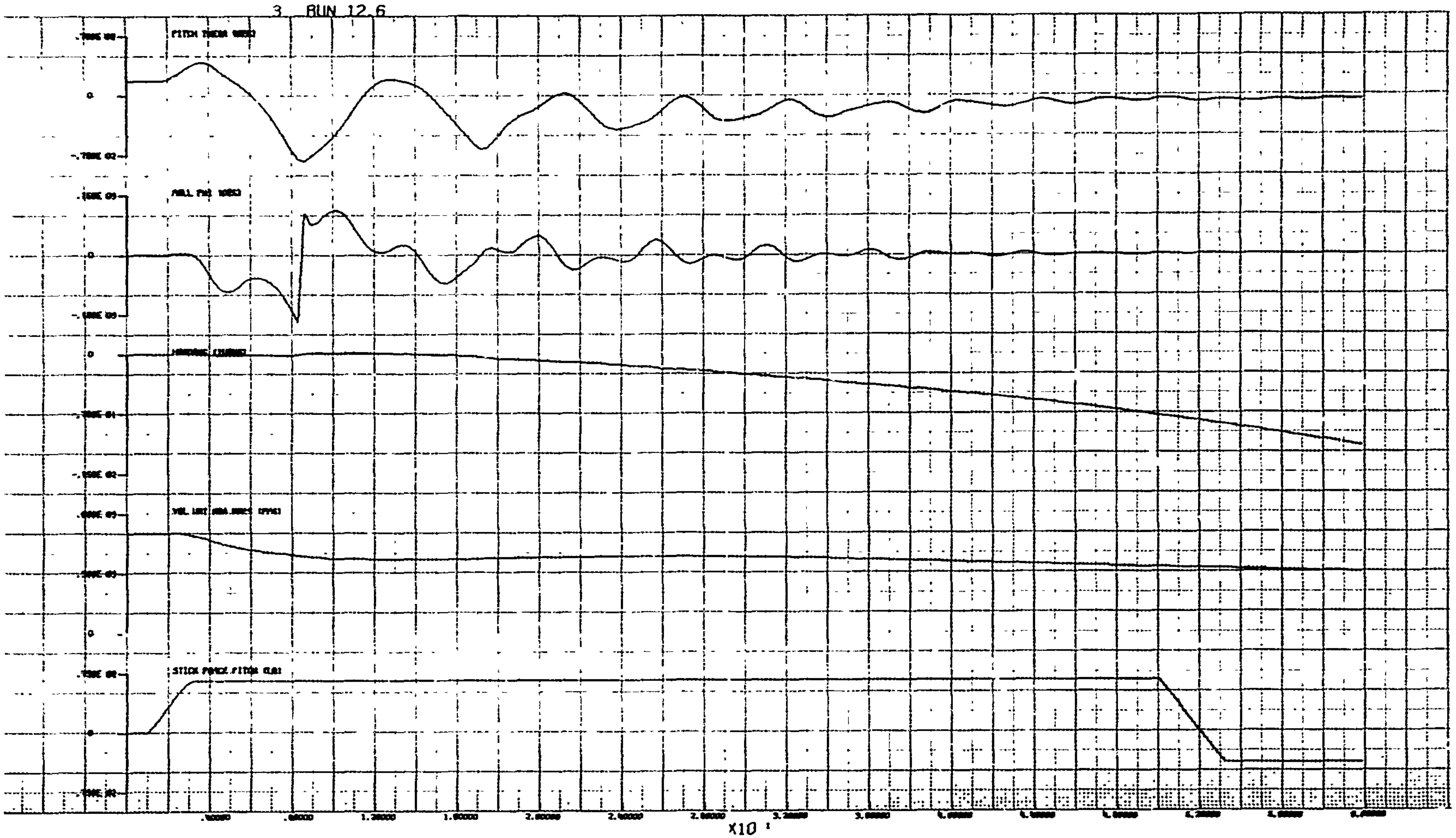


Figure 6-23. Spin Recovery Controls Applied at 50 Seconds (Concluded)

the actual limit must occur between 37 and 44 seconds, agreeing satisfactorily with the predicted limit obtained from Figure 6-18.

It is evident that recovery controls must be applied at some time prior to reaching the controllability limit. When the limit occurs there is no longer adequate control power remaining. This implies that additional criteria is required to determine the proximity of the limit and, hence, the rate at which recovery can be achieved. Also significant is the altitude loss incurred during spin recovery. The rate of altitude loss in the nominal spin of Figure 6-7 is about 300 ft/sec at 40 seconds after spin initiation. The altitude at that time is approximately 30,000 feet. It is assumed that, if recovery is not achieved by 10,000 feet, then the pilot must eject. This then indicates the pilot has about 1 minute to recover. Unless sufficient control power exists at 40 seconds to stop the spin rate within 1 minute, cessation of the spin rate would be futile.

There are two major drawbacks to the criteria described in the previous paragraphs. First, practical implementation of the above criteria would not be straightforward. The involved aerodynamics which ultimately determine surface effectiveness are expected to contribute considerable complexity in a mechanization. In addition, influences of aircraft asymmetry and cg location are expected to be significant factors, indicating either more complication or else some conservatism in the established boundary.

A second drawback with the criteria is the lack of visibility given inter-axis coupling effects. An example of this is evident in the pitch axis. Figure 6-20 indicates that there is, within the studied period, adequate control power in pitch to cause a reduction in AOA. This presumes, however, that the associated inertial moment in pitch does not also increase, which it will (via gyroscopic torques) unless control in the other axes is also successfully exercised. Essential to the proper application of the criteria is a simultaneous satisfaction of controllability in all three axis. While this is sufficient to determine recoverability, the possibility of a preferable strategy is not revealed. A superior criterion would present a unified statement for spin vector control enabling assessment of alternate surface deployment concepts.

Recovery Processes

A critical factor in configuring flight controls for operation in stalled flight is an understanding of the recovery process. Analysis of recovery can encompass a broad range of investigation if auxiliary recovery devices are included. This study considered the use of only the existing aerodynamic control surfaces. The objective of the analysis was to determine a preferred combination of surface controls to effect the recovery and to analyze the effects of feedback control on the recovery process. A further area requiring analysis is the determination of the preferred control procedure immediately following recovery. This was considered in the study but only in a superficial manner.

Most of the analysis of recovery processes was based on recovery from an oscillatory spin such as shown in Figure 6-7. It was assumed that a spin of this type which slowly transitions to a flat spin would be one of the more severe from which to recover. It was also assumed that the preferred recovery controls for this spin would be equally if not more effective in less severe spin conditions. These assumptions were made to keep the scope of the study within practical limits and to establish controlled conditions for the analysis.

Only a cursory analysis was made of departure recovery procedures. Nevertheless, the results obtained substantiate the procedures recommended by the Air Force's Flight Test program (see Reference 6-1). Their recommended procedure during departure is to apply full trailing-edge-down elevator and to neutralize the ailerons and rudder. Full trailing-edge-down elevator is the most effective means available for recovery during departure. The objective of this procedure is to reduce the angle of attack as fast as possible to regain directional stability and normal flight. To recall, departure in the F-4 is primarily a result of lateral-directional instability in the angle-of-attack range from 23 to 37 degrees. This instability causes a buildup of a positive produce of roll and yaw rates which in turn causes a positive pitching moment. This positive pitching moment can be effectively countered by a nose-down elevator. But down elevator must be applied before the lateral motions are allowed to become too large. Hence, less than full-down elevator only lessens the chances for immediate recovery.

Neutralized ailerons and rudder during departure is also recommended, by Reference 6-1. It has been reported that pilots can become disoriented during a departure and cannot be relied upon to apply a more favorable lateral-directional control. In fact, they may aggravate departure with improper use of the controls because of the existing lateral-directional instability. Normal reaction to the roll-reversal characteristic of departure would be to apply ailerons in a direction to counter the rolling motion. This reaction, of course, would only aggravate the situation because of the induced yawing motion and subsequent additional buildup of sideslip. Use of rudder for recovery during the period of directional instability could be effective to stop the yaw-rate buildup, but a prolonged deflection could result in initiating a yaw rate in the opposite direction because of the directional instability which exists.

If full-down elevator is not effective in reducing the angle of attack during departure, the aircraft will likely go into a mild spin at angles of attack above 40 degrees. In this situation the pilot should be able to assess the situation and apply the proper recovery controls. If he cannot assess the situation, an automatic system might take over, either at his option or as the controllability limit is approached.

Recovery from a spinning condition which has developed for some period may not be successful with application of only full-down elevator and neutralized ailerons and rudder. Figure 6-24 shows an attempted spin recovery with

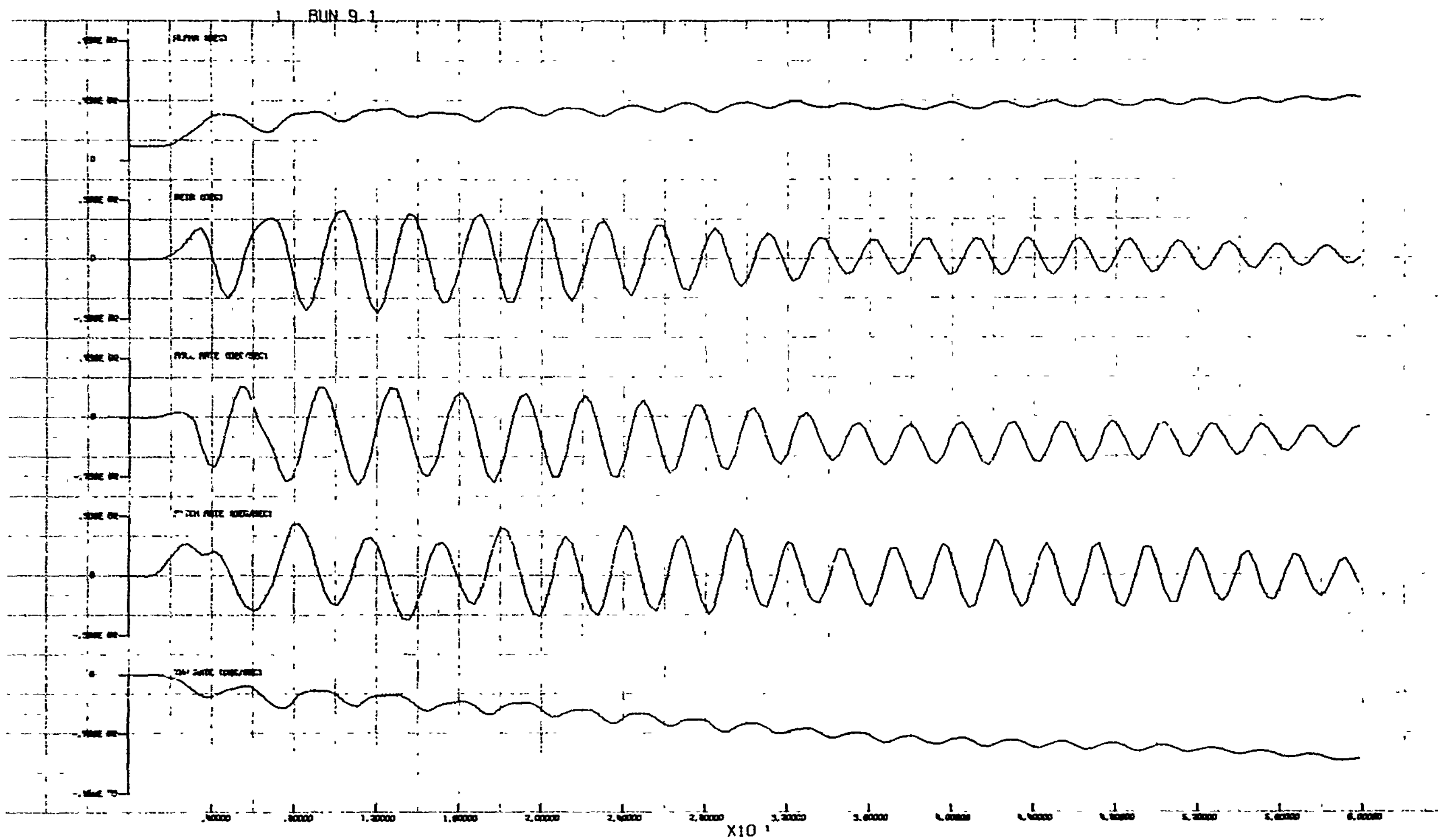


Figure 6-24. Attempted Spin Recovery with Full-Down Elevator and Neutralized Ailerons and Rudder

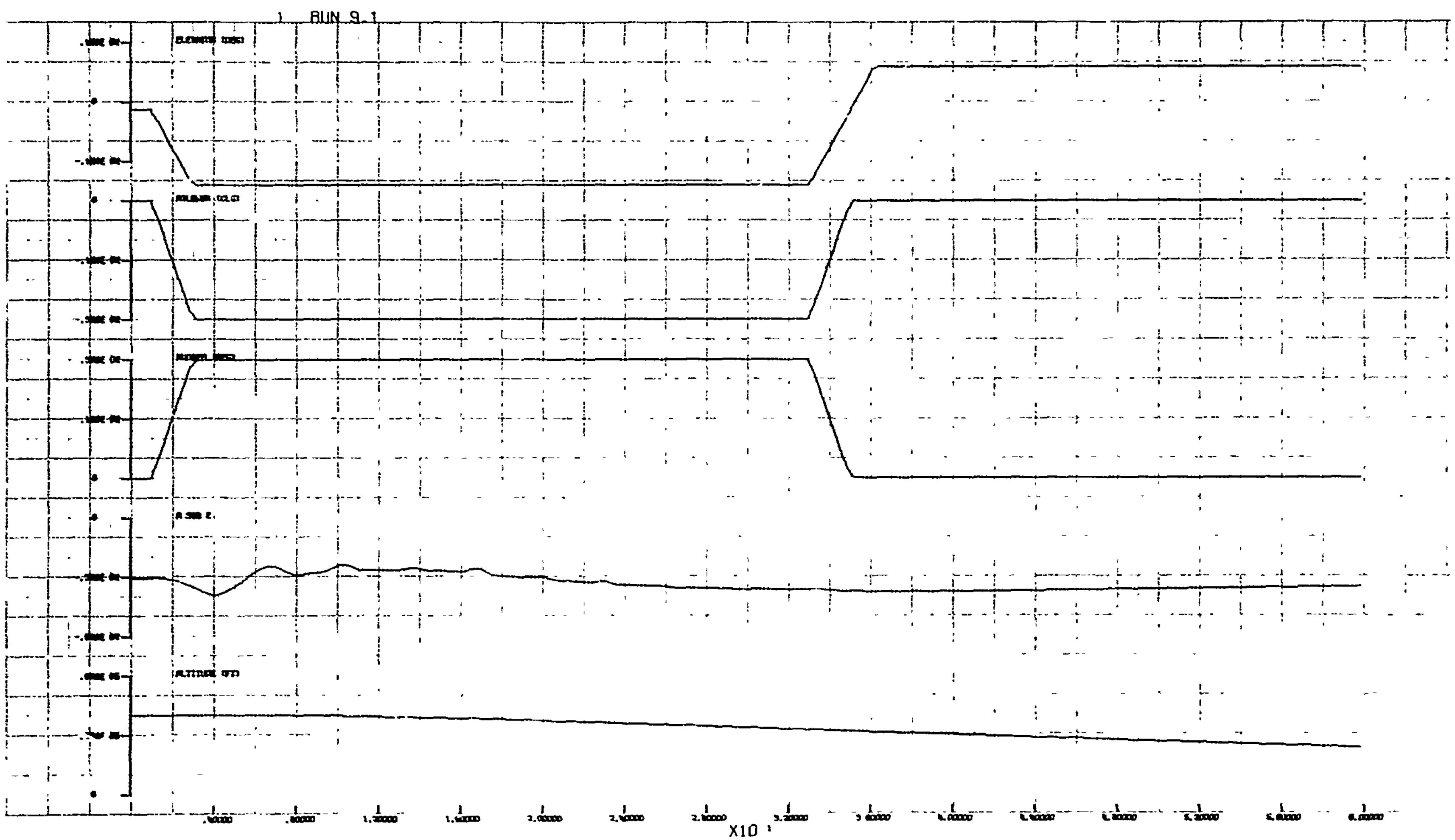


Figure 6-24. Attempted Spin Recovery with Full-Down Elevator and Neutralized Ailerons and Rudder (Continued)

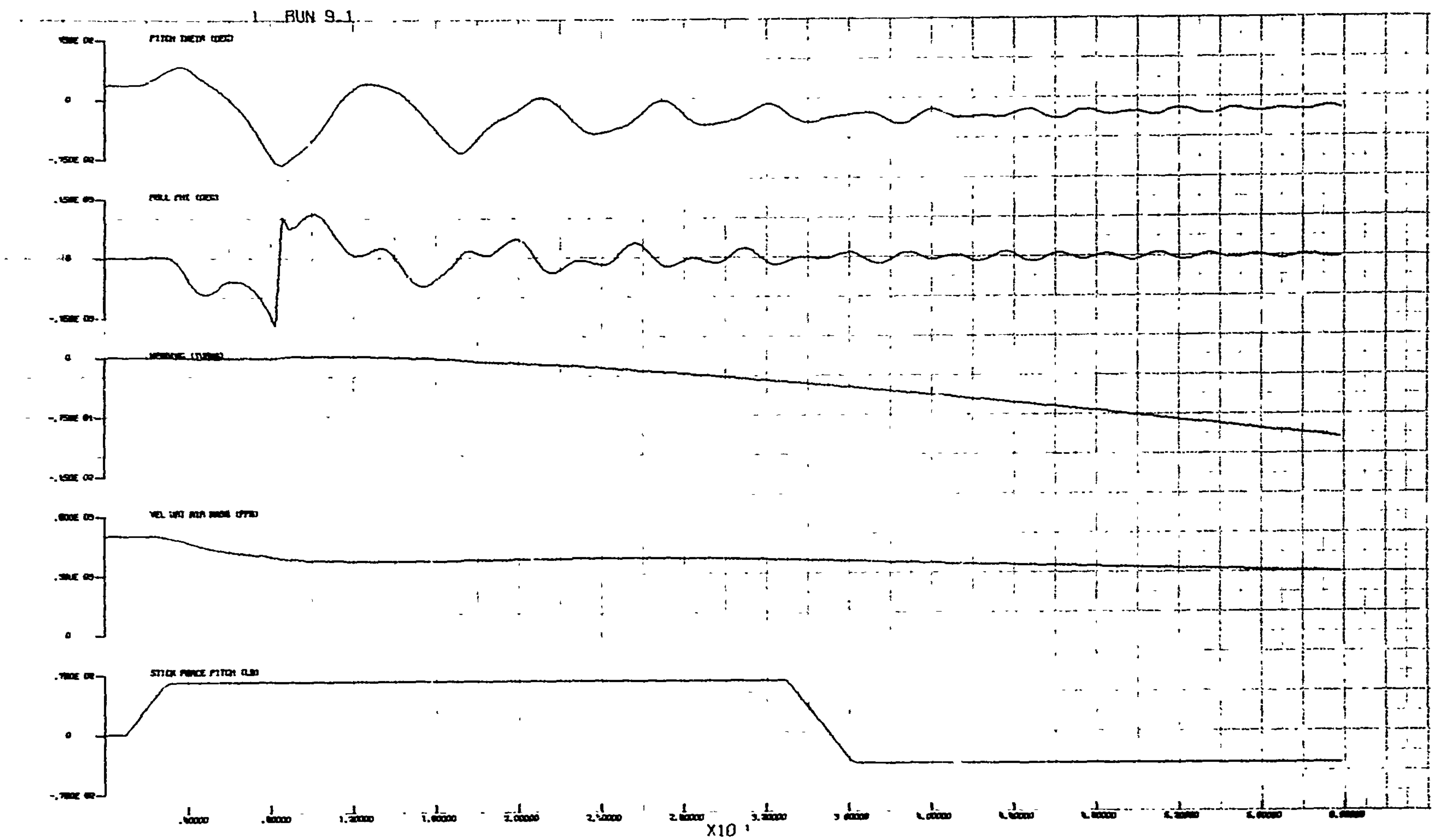


Figure 6-24. Attempted Spin Recovery with Full-Down Elevator and Neutralized Ailerons and Rudder (Concluded)

these controls. The analysis of the dominant factors affecting spin evolution revealed that the ailerons (via their yawing moments) should be the most effective means for stopping the spin. Rudder should exhibit some additional benefits but primarily only at angles of attack less than 50 degrees. Use of rudder at higher angles of attack can be of some benefit but only if the oscillatory motion of sideslip is significant. The effectiveness of the elevator to reduce the angle of attack diminishes as the vehicle becomes more spin stabilized with increasing spin rate. After the vehicle has achieved a sufficient degree of spin stability, the full-down elevator can only effect a precessional motion normal to the applied torque. In this case the only possible recourse (exclusive of auxiliary control devices) is to attempt to slow the spin rate (i. e., predominately yaw rate). This can be achieved by applying opposing control moments in the yaw axis through use of the ailerons and rudder. Application of ailerons in the pro-spin direction will produce an opposing yaw moment as will application of rudder in the anti-spin direction.

Different combinations of recovery controls were evaluated (using the spin of Figure 6-7) to determine the relative effectiveness of each surface. Comparison of Figures 6-25 and 6-26 show pro-spin ailerons to be more effective than anti-spin rudder in stopping the yaw-rate buildup. Figure 6-27 shows that,when the combination of pro-spin aileron, neutral elevator and anti-spin rudder is used, the yaw rate can actually be reversed. This combination is more effective than use of full-down elevator with either (but not both) the anti-spin rudder or the pro-spin aileron. These combinations are shown in Figures 6-28 and 6-29. Comparison of Figures 6-27 and 6-21 show, however, that the use of full-down elevator with pro-spin aileron and anti-spin rudder will produce more rapid and effective recovery.

A final remark should be made about retention of full-up elevator. It is possible to initiate a recovery in less severe spin conditions even with full-up elevator. Figure 6-30 shows that a successful recovery was achieved even though the elevator was left in the full-up position. In this case the recovery was started earlier at a time of 30 seconds rather than at 33 seconds. A potential danger with maintaining full-up elevator, however, is that as angle of attack re-enters the region of lateral-directional instability the spin could be easily reinitiated in the other direction. Hence, retention of up-elevator is not recommended.

The effects of feedback control on the recovery process are primarily detrimental, simply because they consume surface authority better used in a full corrective deflection. A nominal feedback configuration was used to evaluate the feedback effects upon recovery. The system used was described earlier and shown in Figure 6-5. To achieve controlled conditions (i. e., a repeatable spin) with feedback control, the spin was initiated with the surfaces displaced at their maximum rates. This was accomplished by applying large pilot stick and pedal forces. The resulting spin was similar in character to the spin shown in Figure 6-7. Figure 6-31 shows a spin obtained in this manner. Figure 6-31 also shows a successful recovery when recovery

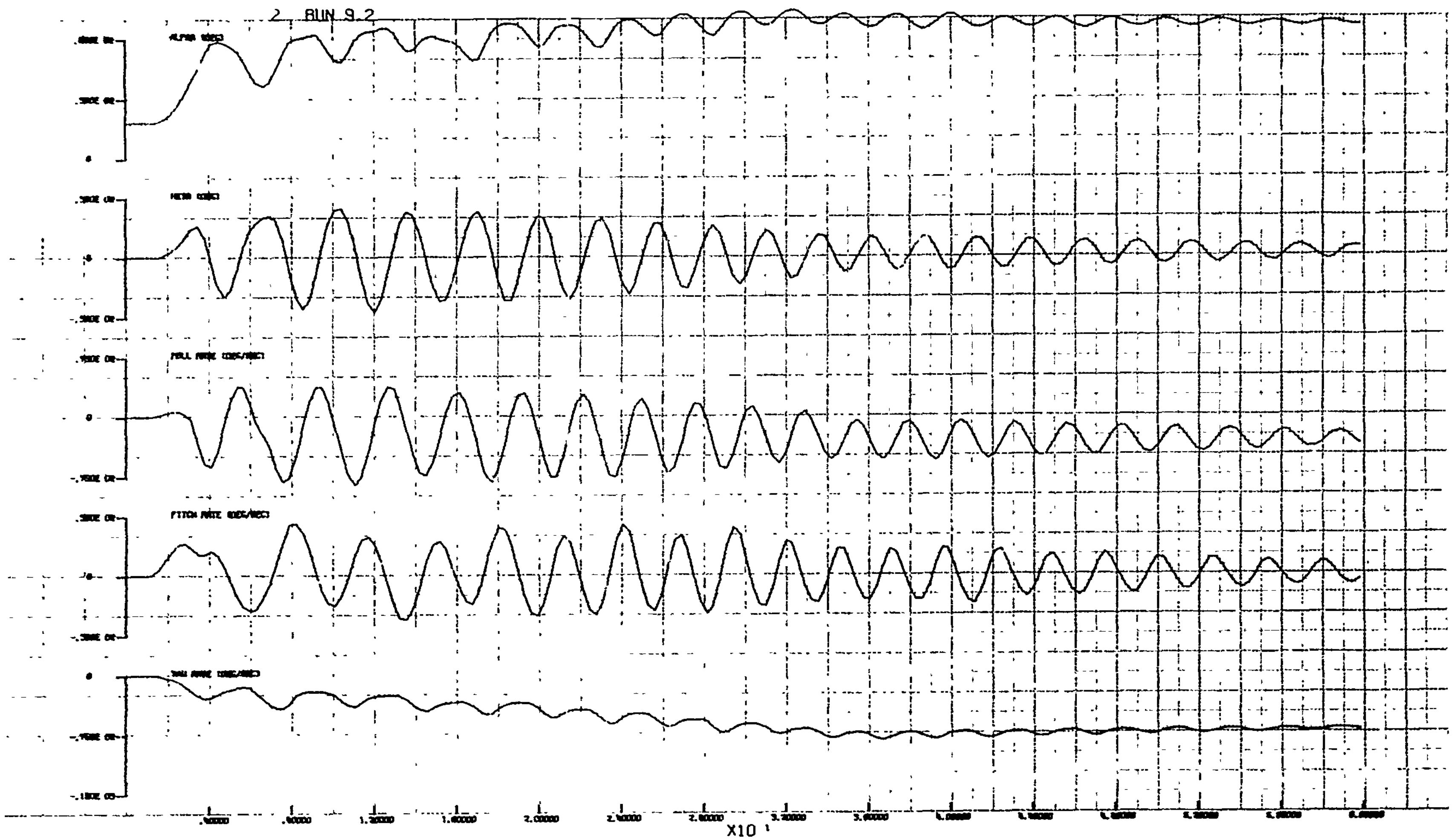


Figure 6-25. Attempted Spin Recovery with Pro-Spin Ailerons and Neutralized Elevator and Rudder

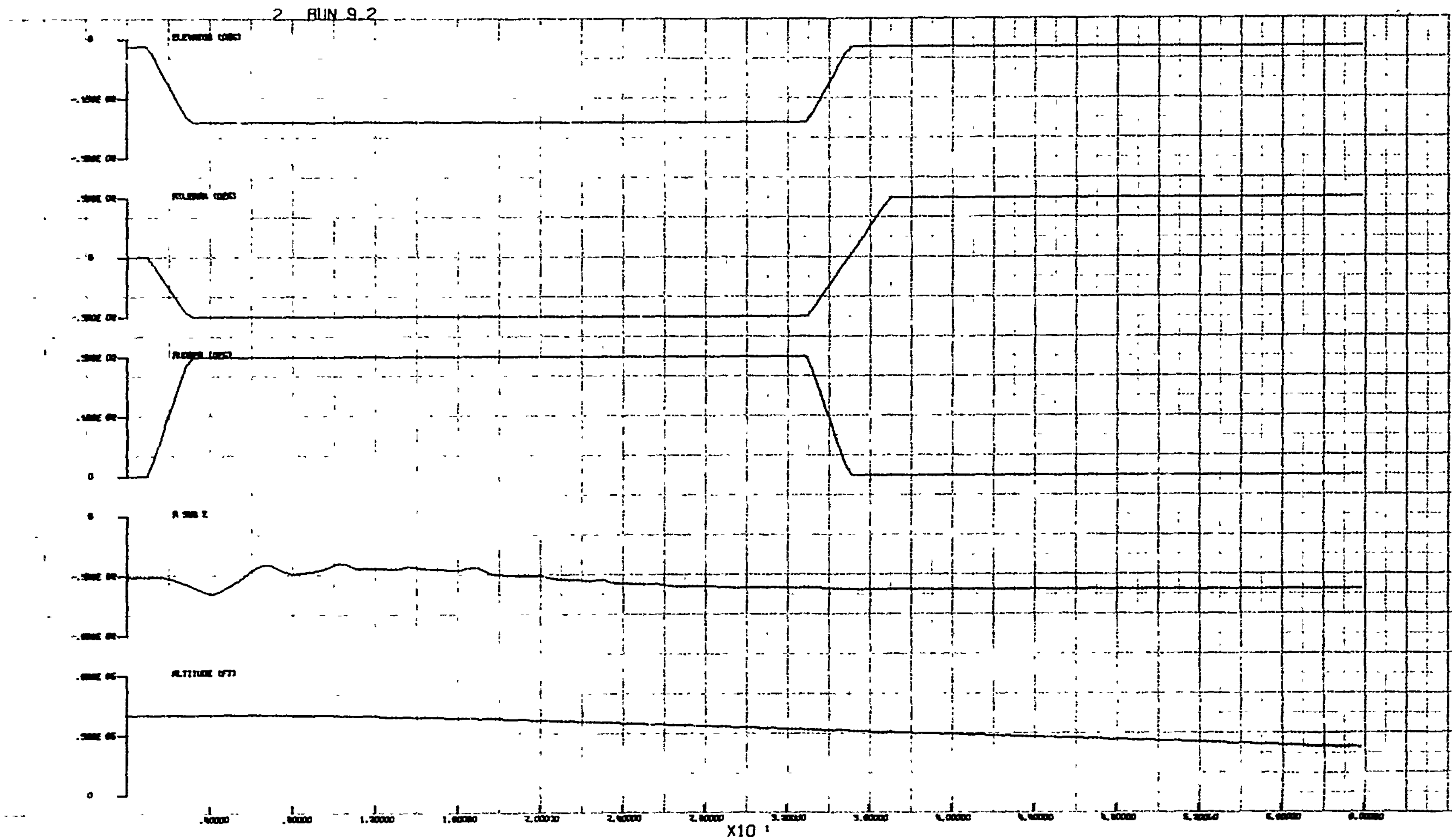


Figure 6-25. Attempted Spin Recovery with Pro-Spin Ailerons and Neutralized Elevator and Rudder (Continued)

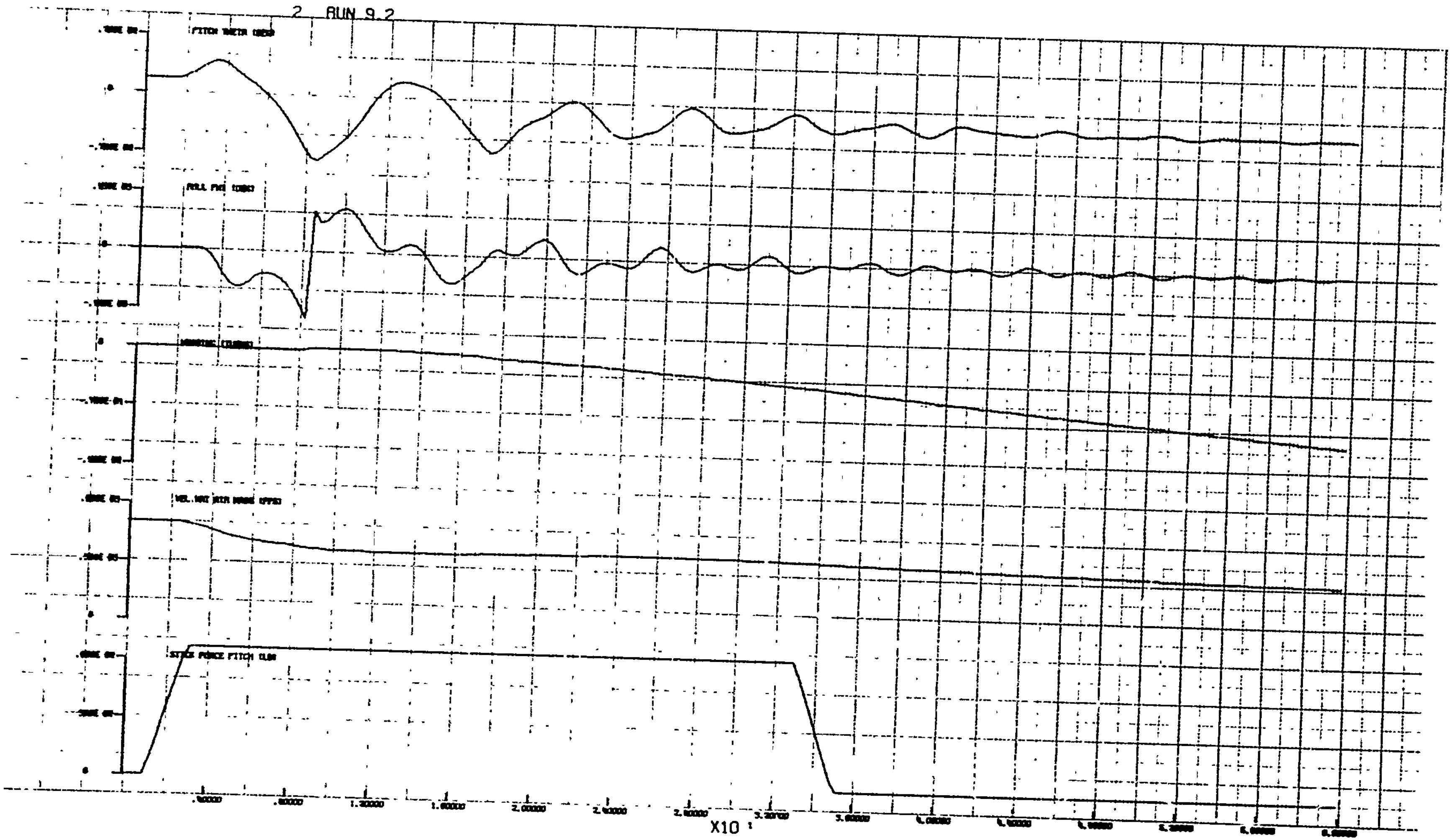


Figure 6-25. Attempted Spin Recovery with Pro-Spin Ailerons and Neutralized Elevator and Rudder (Concluded)

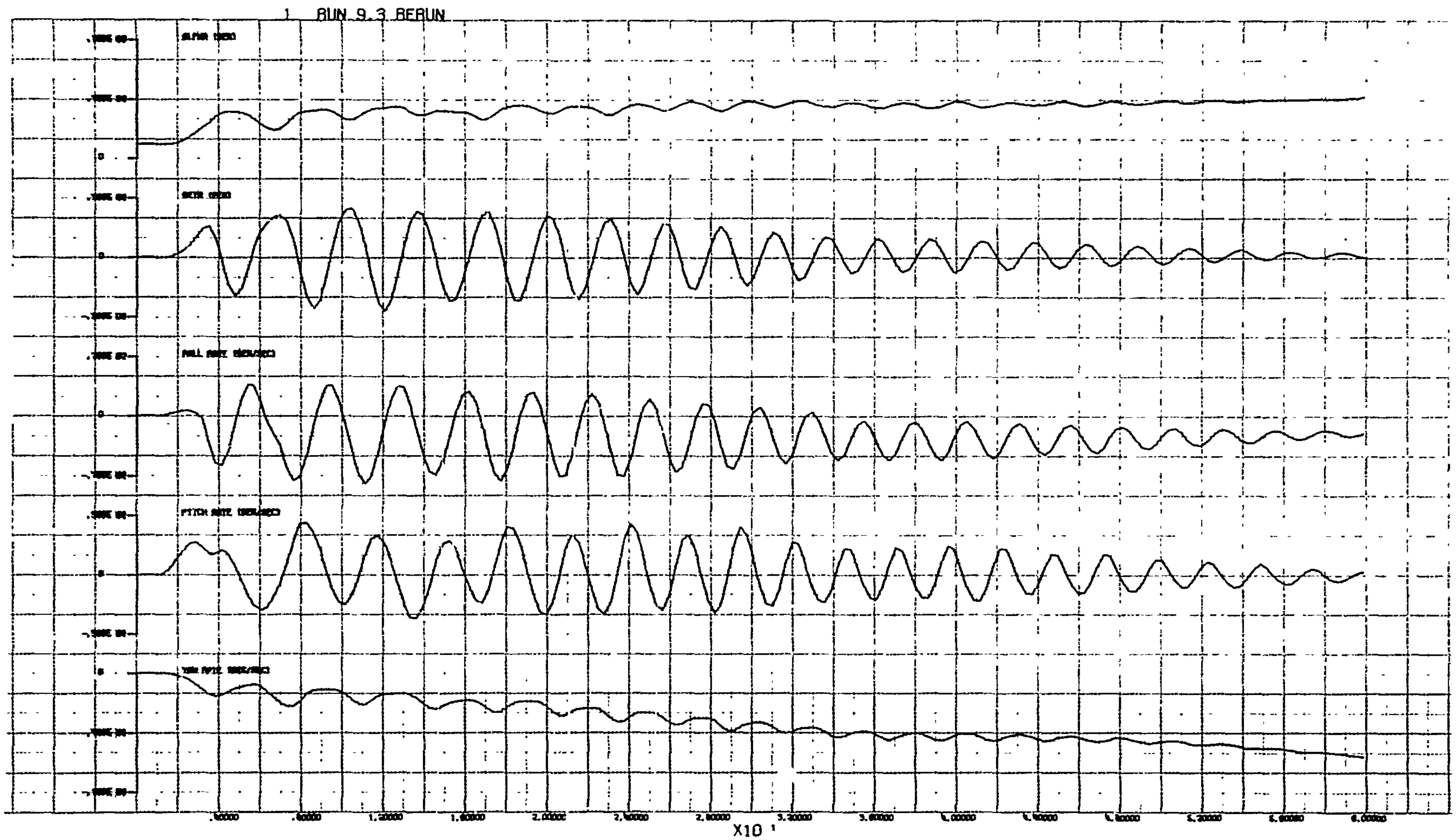


Figure 6-26. Attempted Spin Recovery with Anti-Spin Rudder and Neutralized Elevator and Ailerons

Figure 6-26. Attempted Spin Recovery with Anti-Spin Rudder and Neutralized Elevator and Ailerons (Continued)

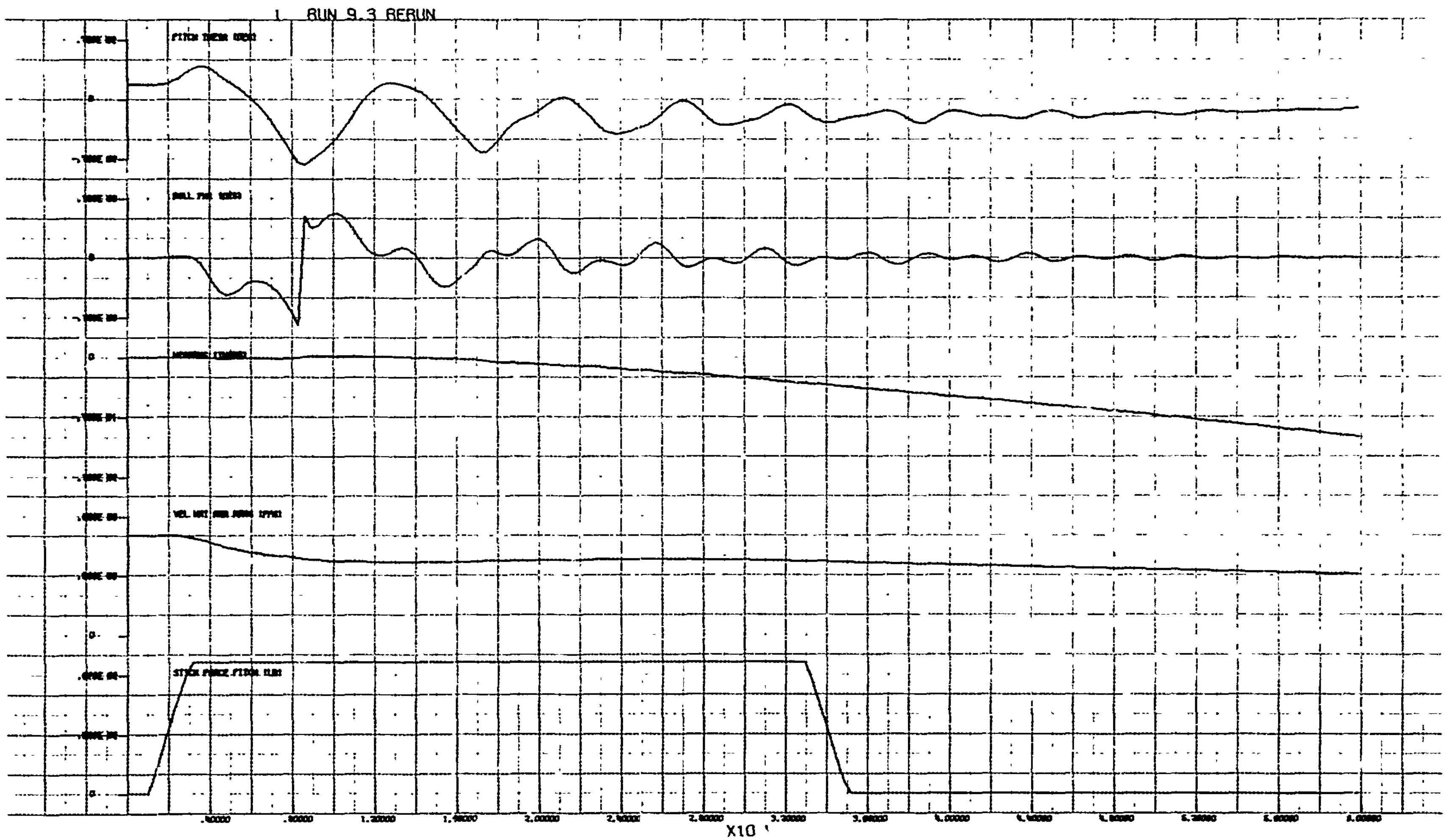


Figure 6-26. Attempted Spin Recovery with Anti-Spin Rudder and Neutralized Elevator and Ailerons (Concluded)

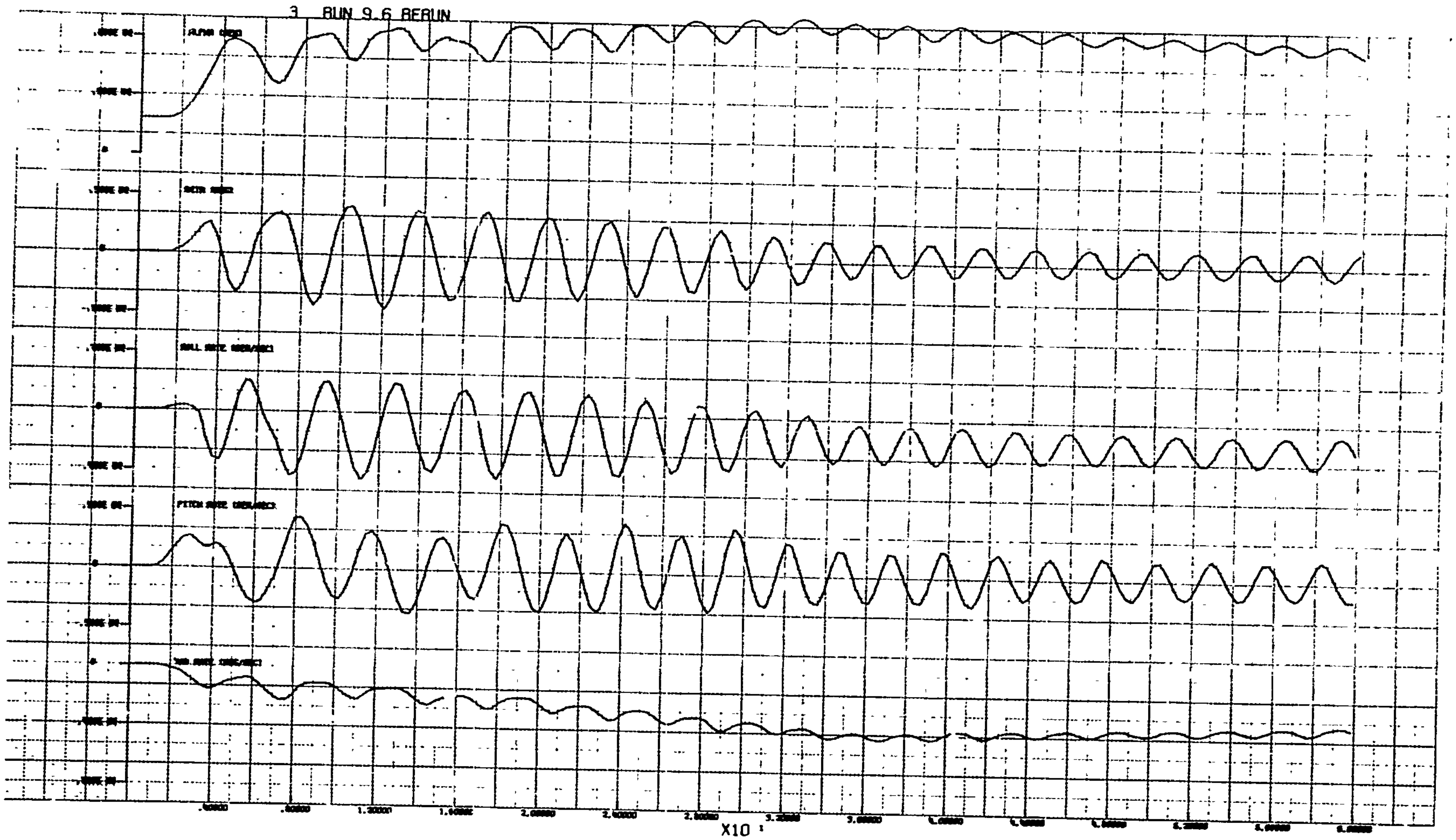


Figure 6-27. Attempted Sp:n Recovery with Pro-Spin Ailerons, Anti-Spin Rudder, and Neutralized Elevator

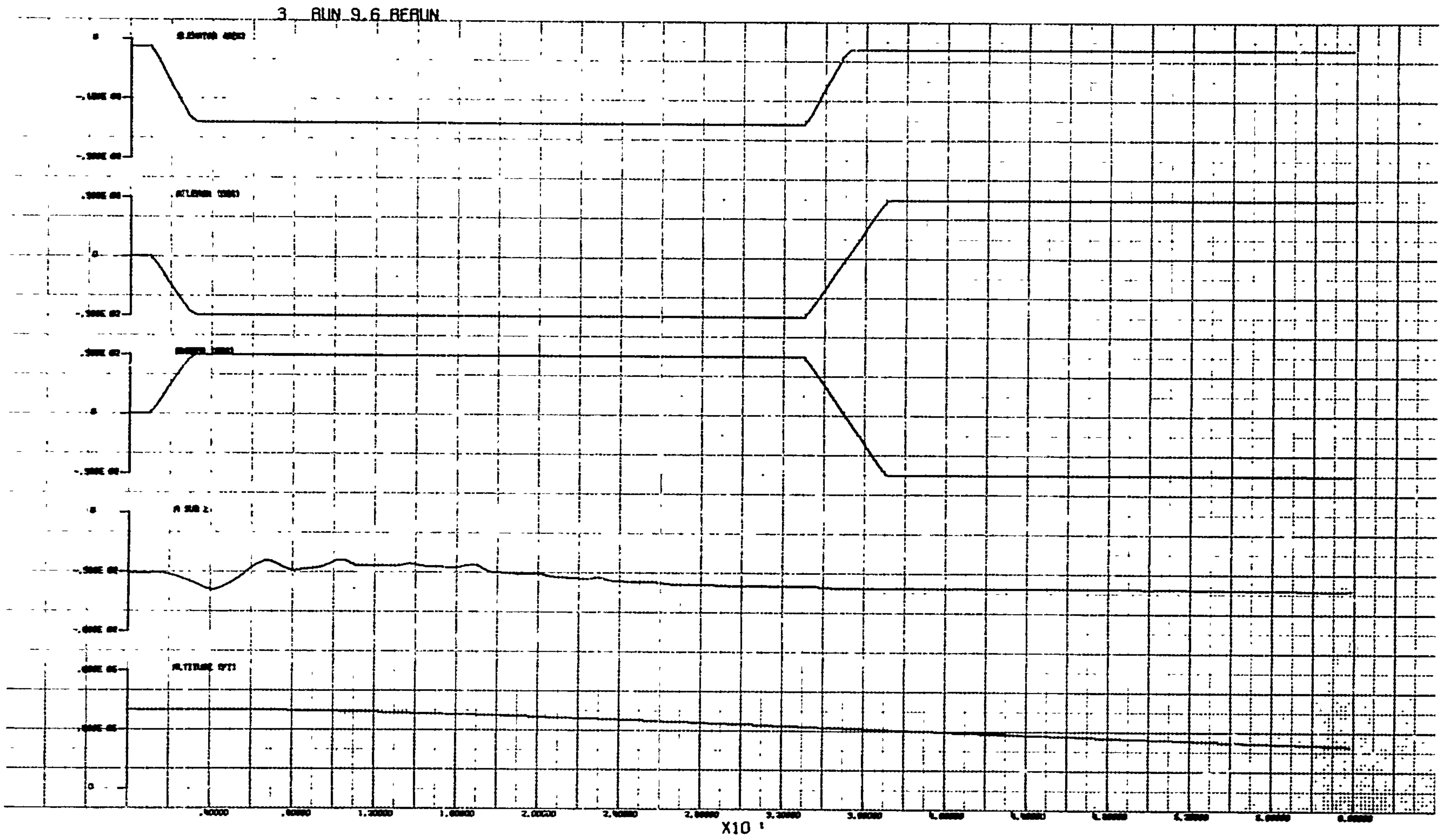


Figure 6-27. Attempted Spin Recovery with Pro-Spin Ailerons, Anti-Spin Rudder, and Neutralized Elevator (Continued)

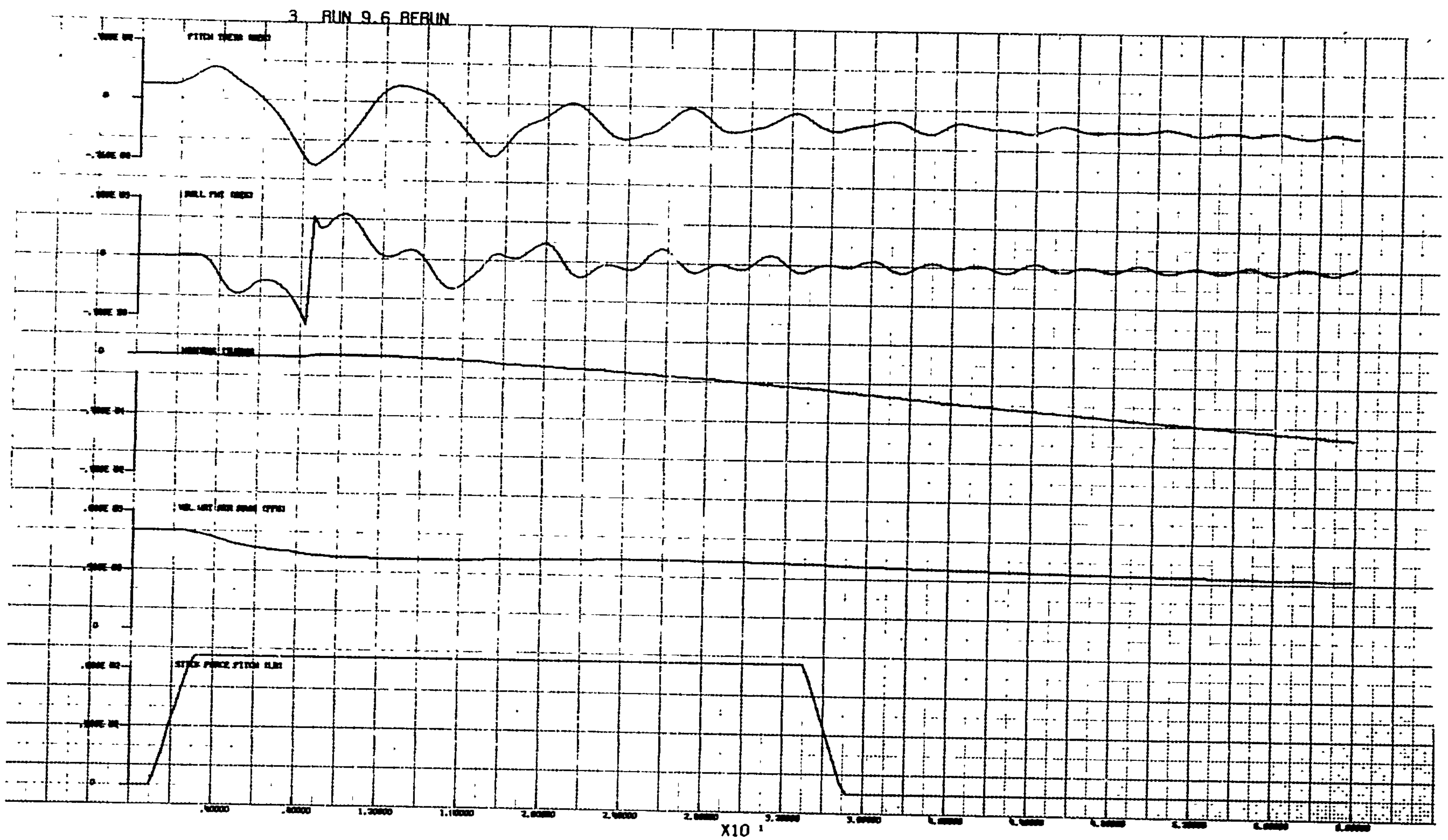


Figure 6-27. Attempted Spin Recovery with Pro-Spin Ailerons, Anti-Spin Rudder, and Neutralized Elevator (Concluded)

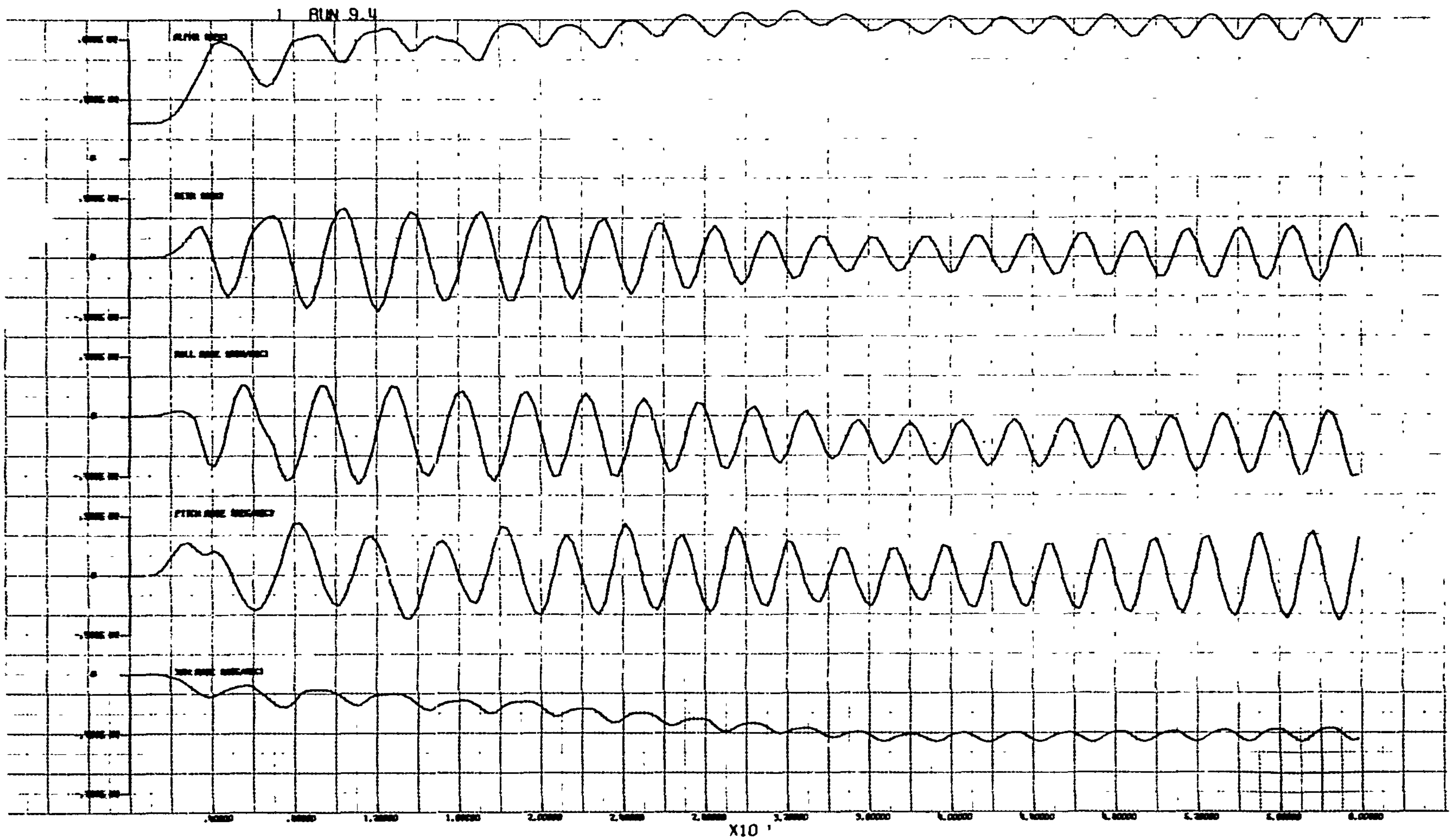


Figure 6-28. Attempted Spin Recovery with Full-Down Elevator, Pro-Spin Ailerons, and Neutralized Rudder

Figure 6-28. Attempted Spin Recovery with Full-Down Elevator, Pro-Spin Ailerons, and Neutralized Rudder (Continued)

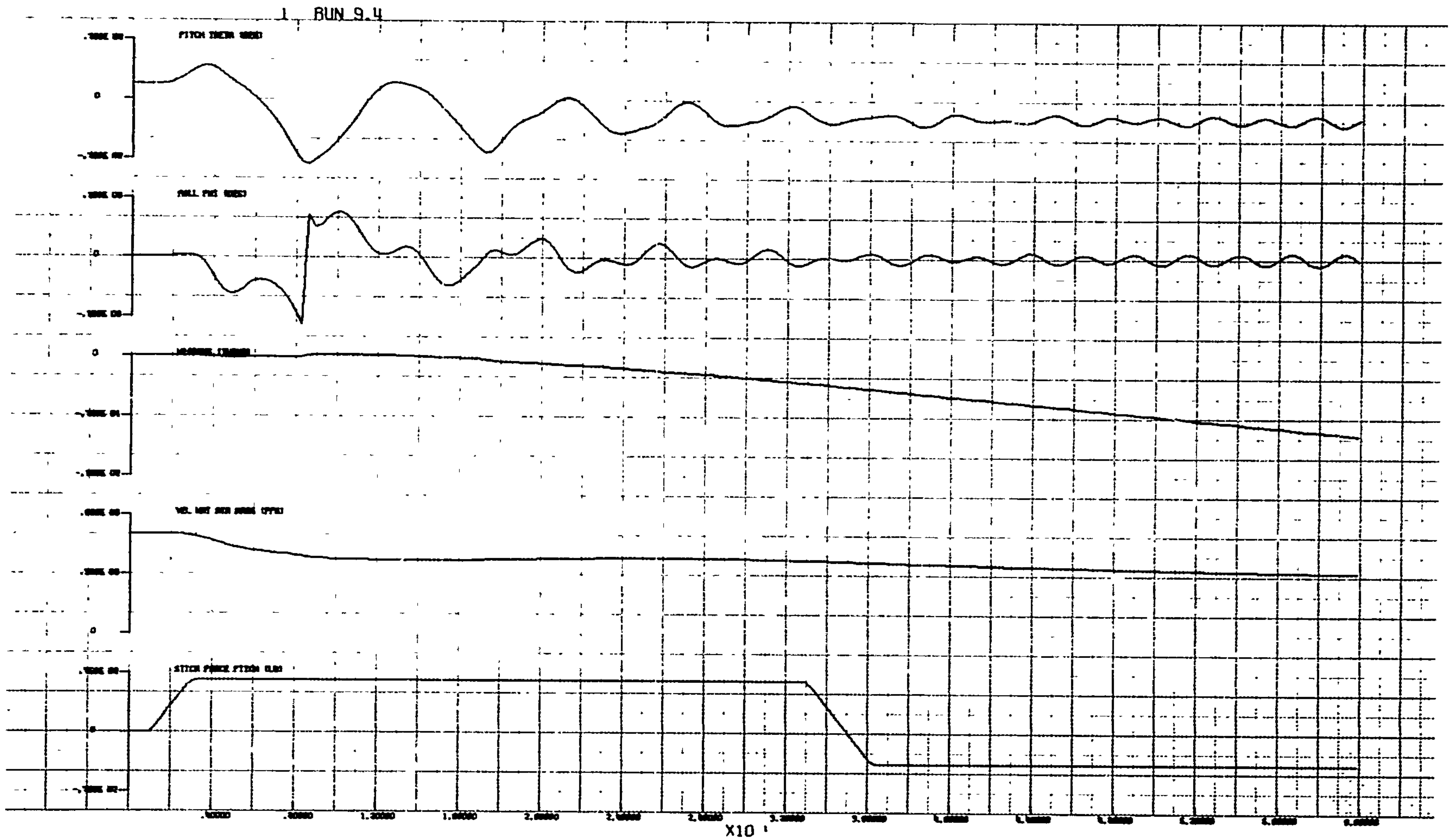


Figure 6-28. Attempted Spin Recovery with Full-Down Elevator, Pro-Spin Ailerons, and Neutralized Rudder (Concluded)

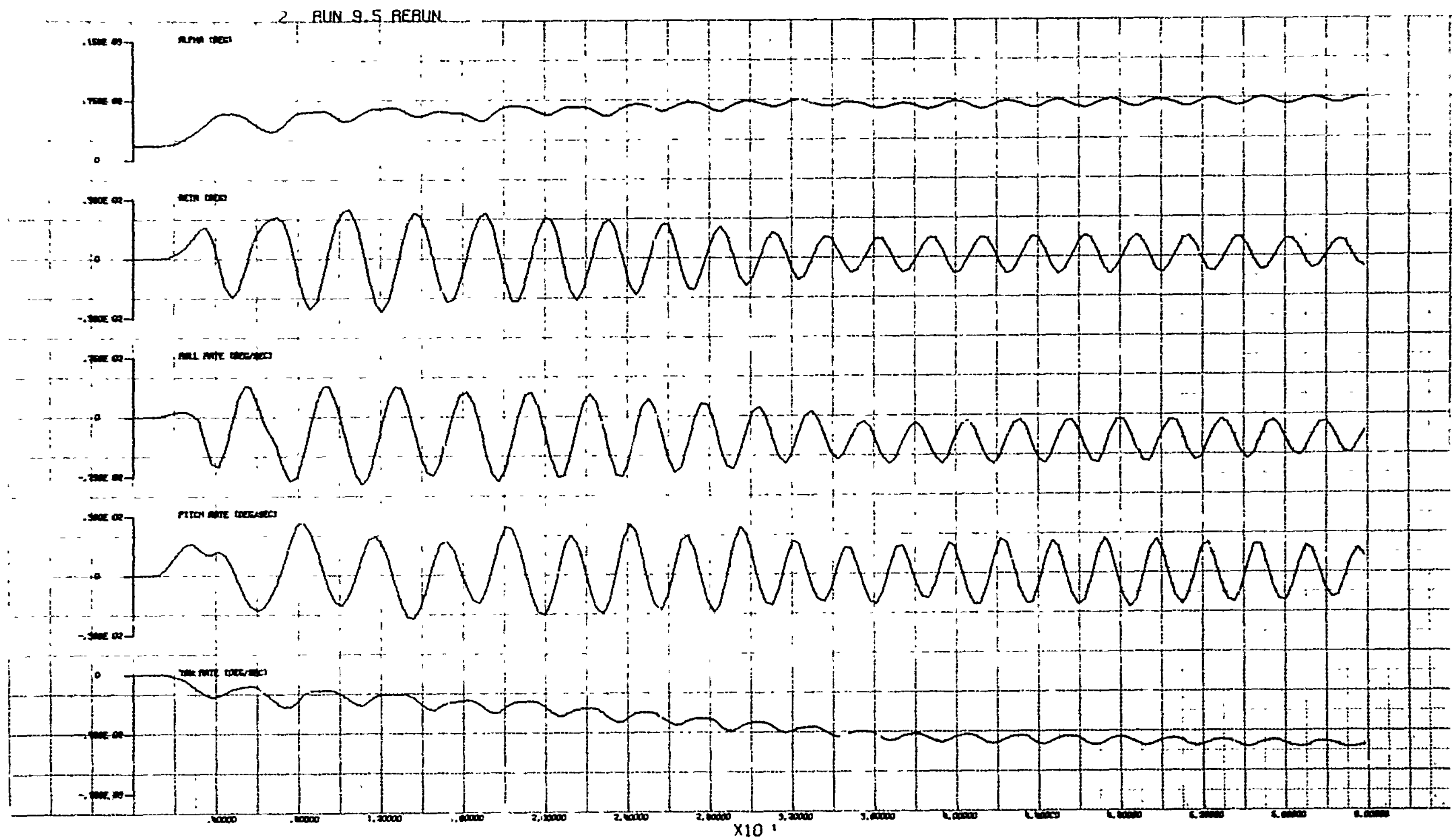


Figure 6-29. Attempted Spin Recovery with Full-Down Elevator, Anti-Spin Rudder, and Neutralized Ailerons

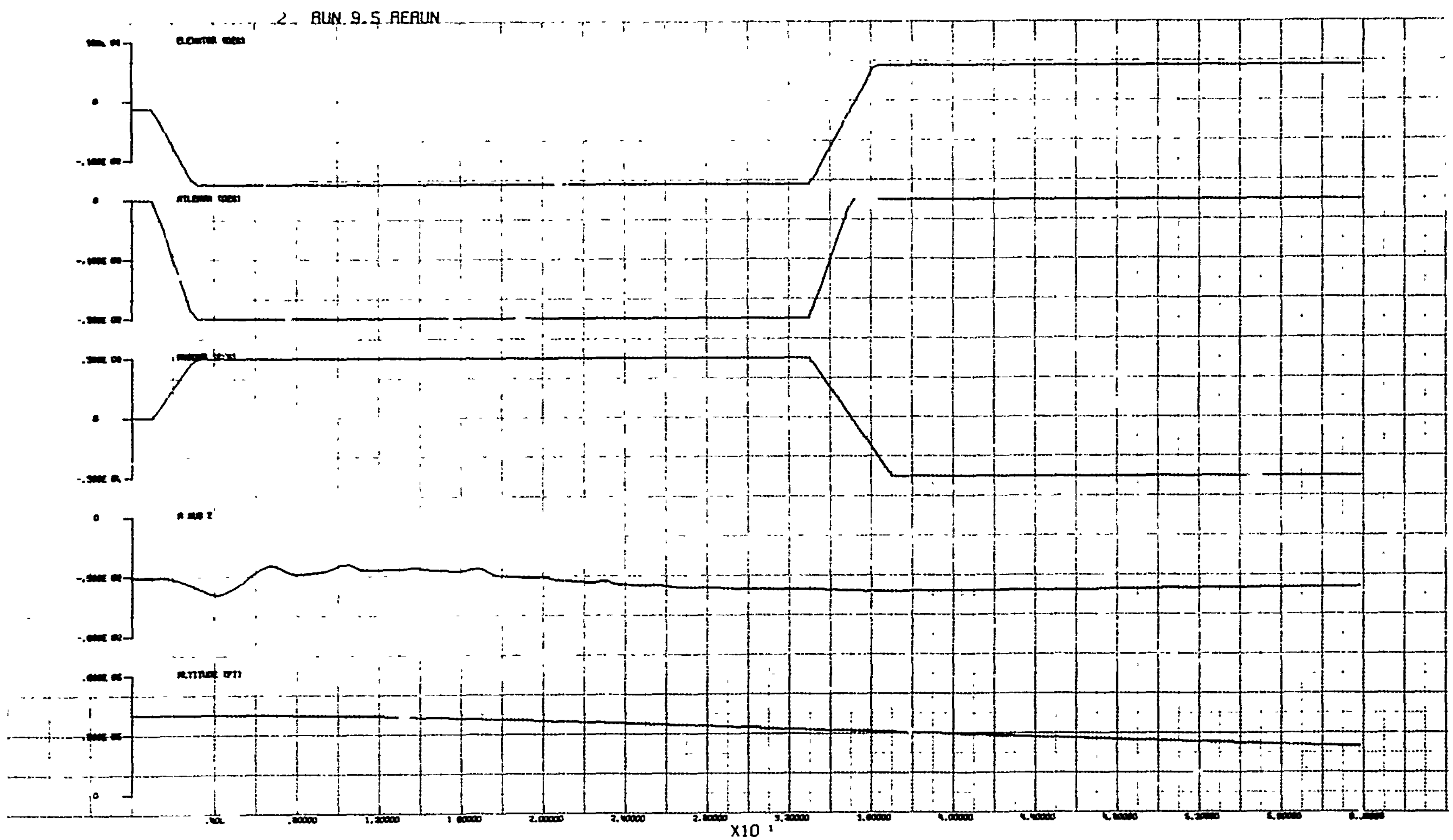


Figure 6-29. Attempted Spin Recovery with Full-Down Elevator, Anti-Spin Rudder, and Neutralized Ailerons (Continued)

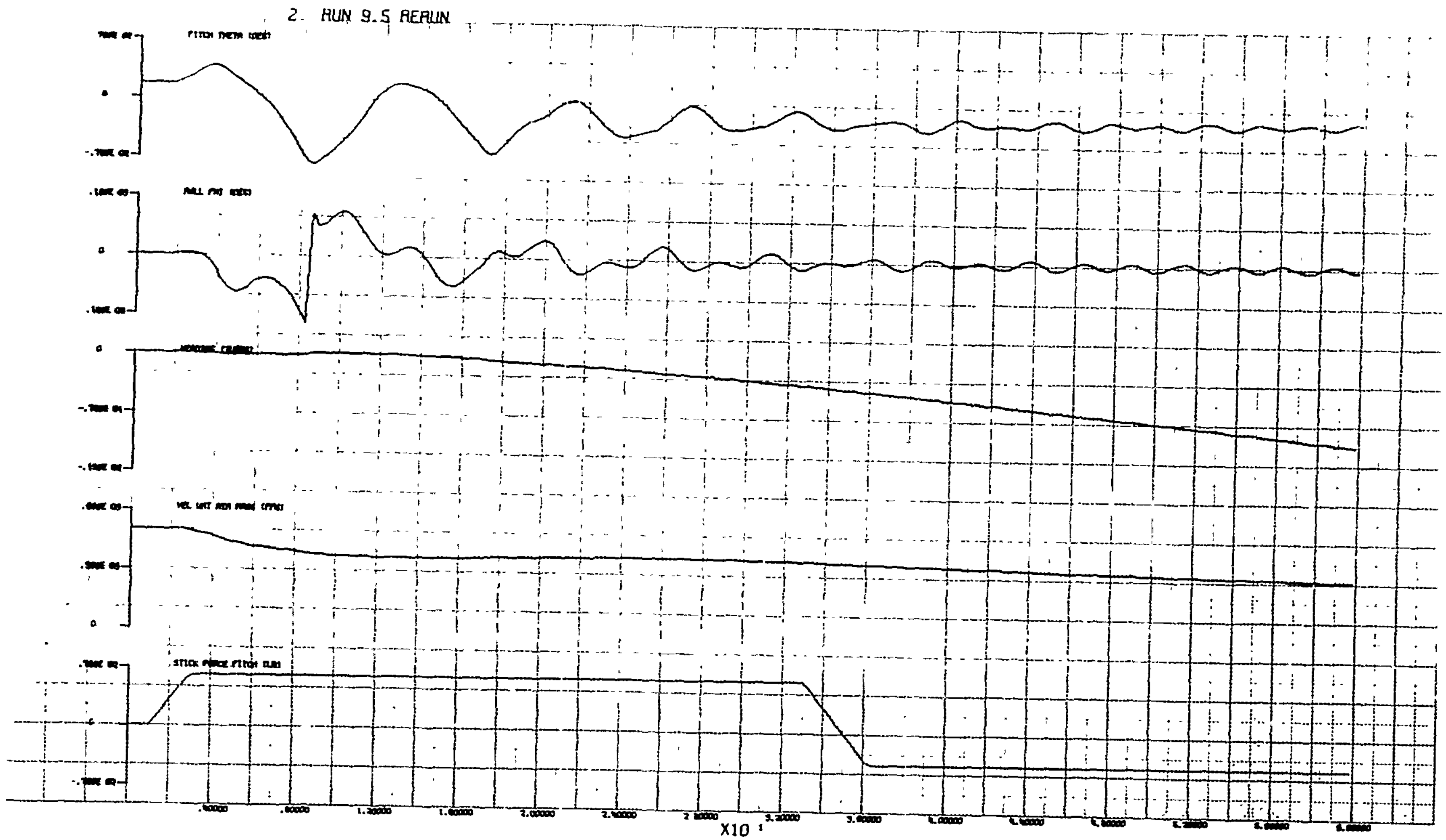


Figure 6-29. Attempted Spin Recovery with Full-Down Elevator, Anti-Spin Rudder, and Neutralized Ailerons (Concluded)

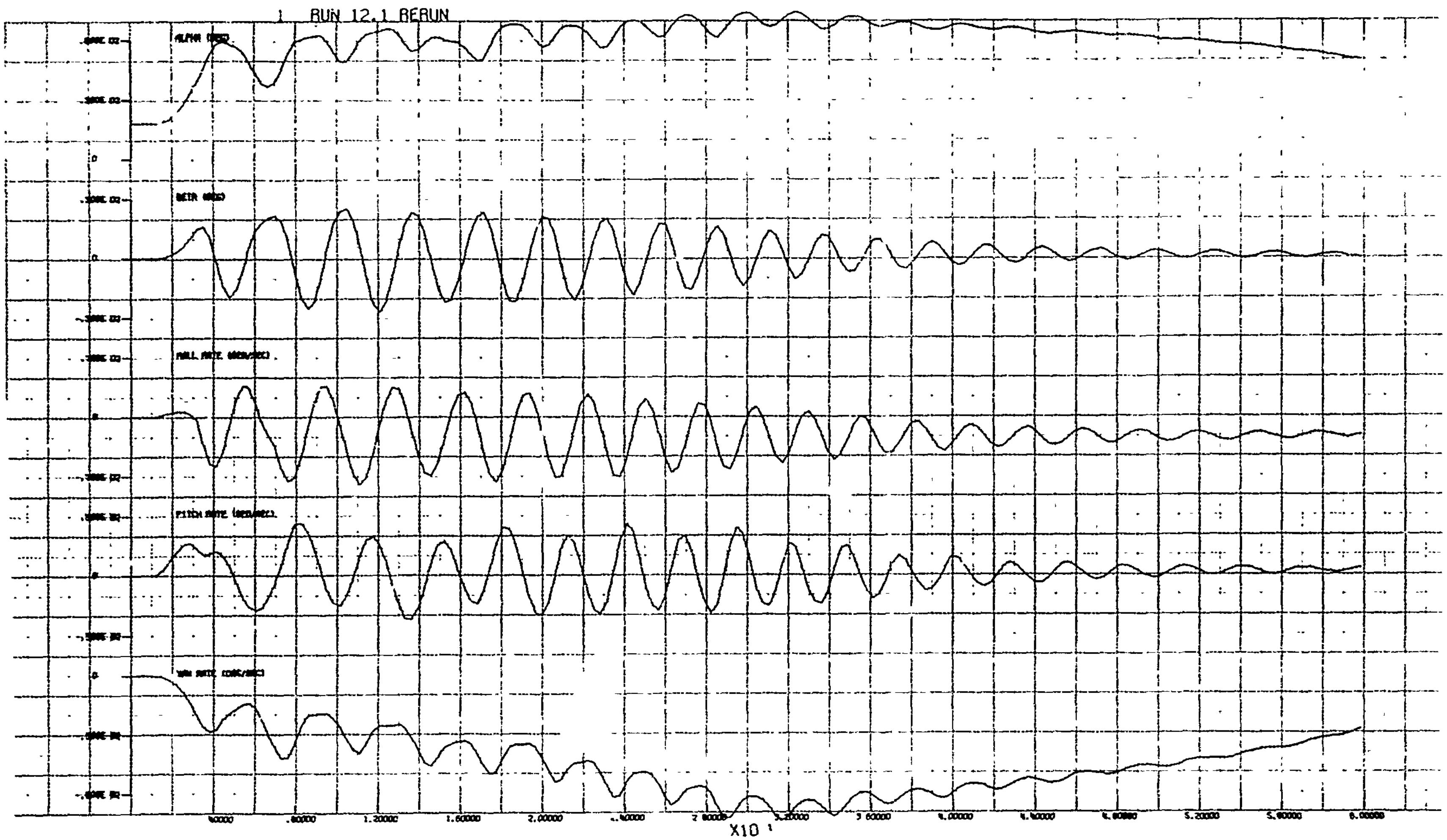


Figure 6-30. Attempted Spin Recovery at 30 Seconds with Full-Up Elevator, Pro-Spin Ailerons, and Anti-Spin Rudder

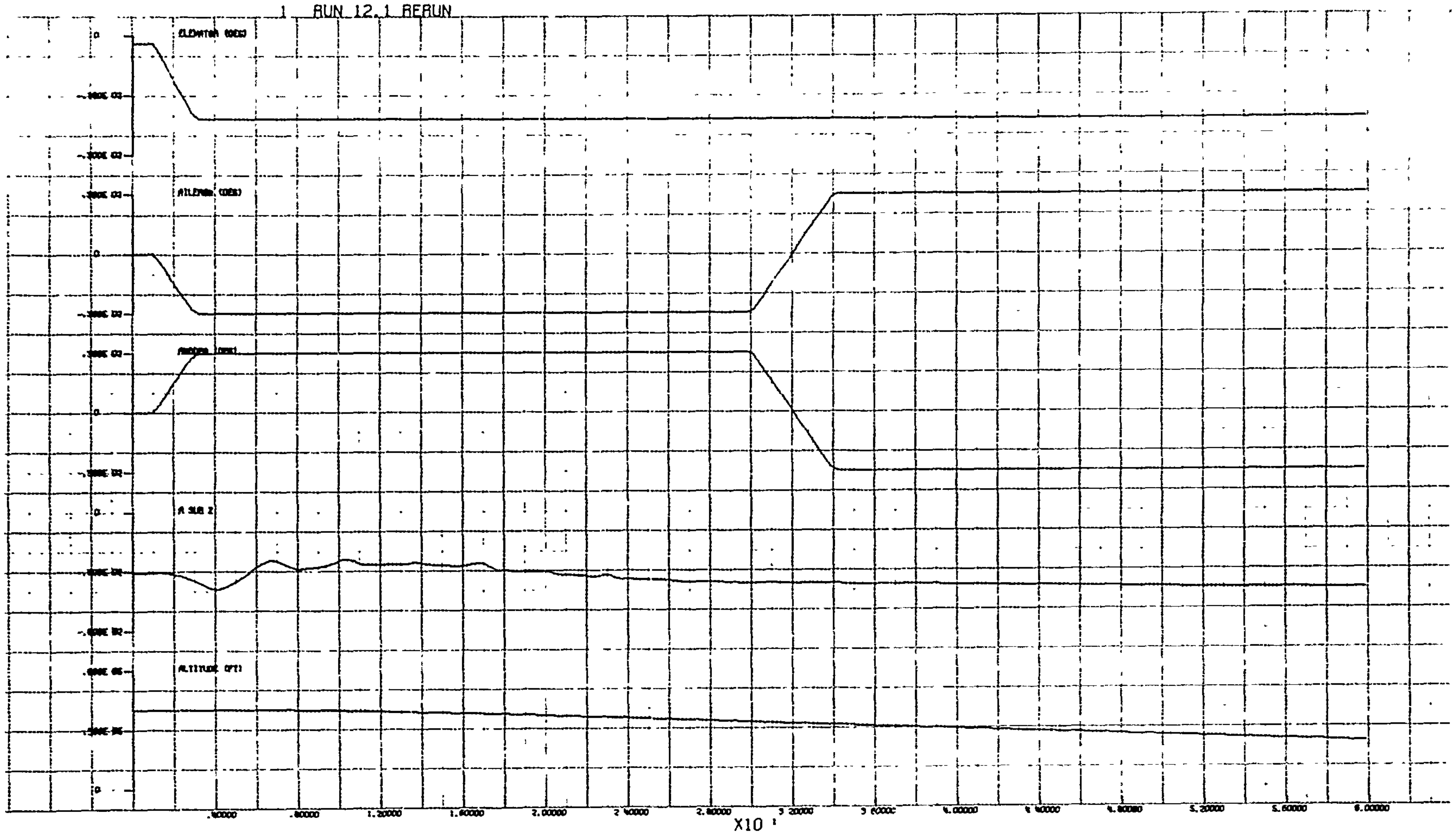


Figure 6-30. Attempted Spin Recovery at 30 Seconds with Full-Up Elevator, Pro-Spin Ailerons, and Anti-Spin Rudder (Continued)

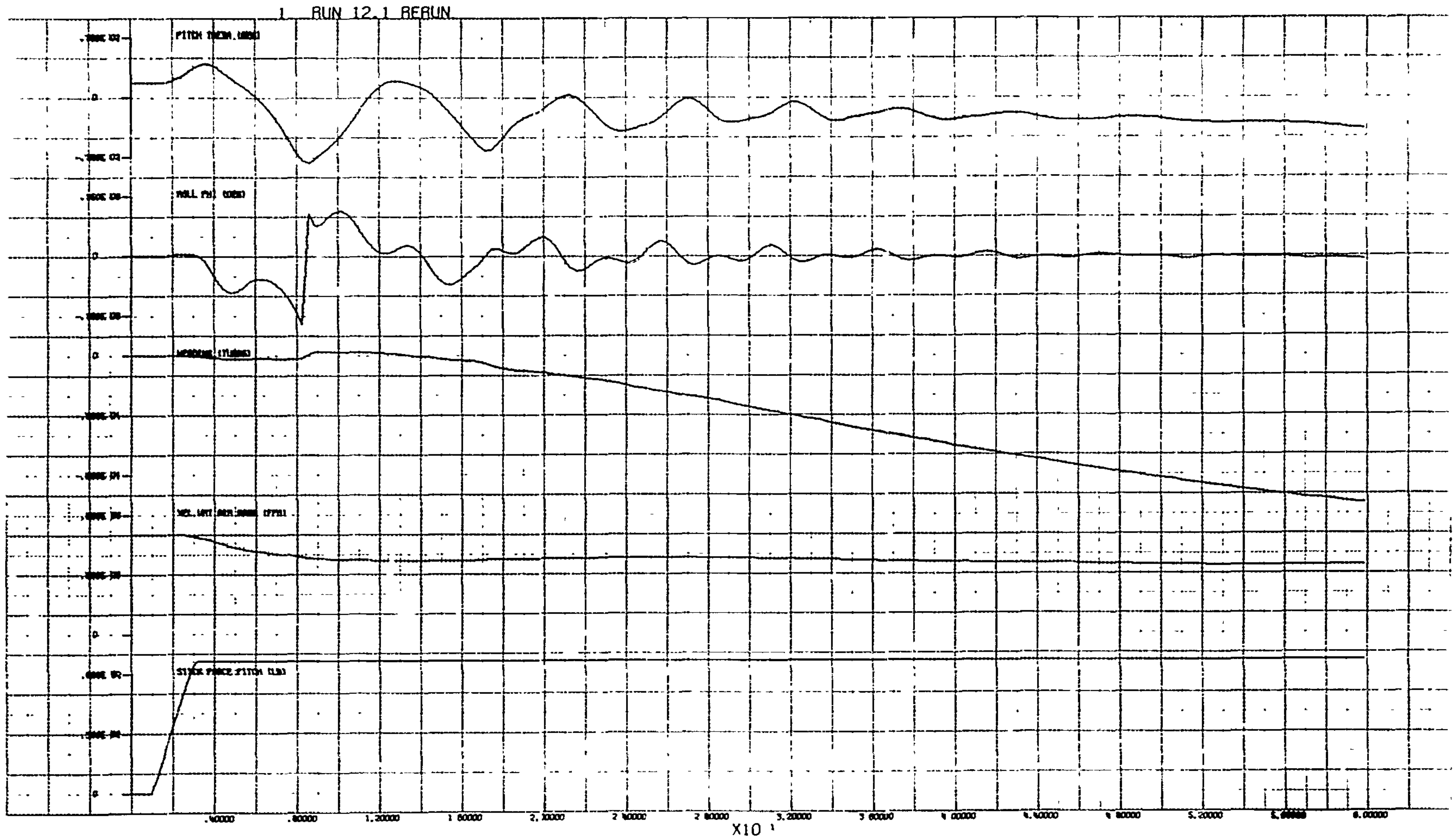


Figure 6-30. Attempted Spin Recovery at 30 Seconds with Full-Up Elevator, Pro-Spin Ailerons, and Anti-Spin Rudder (Concluded)

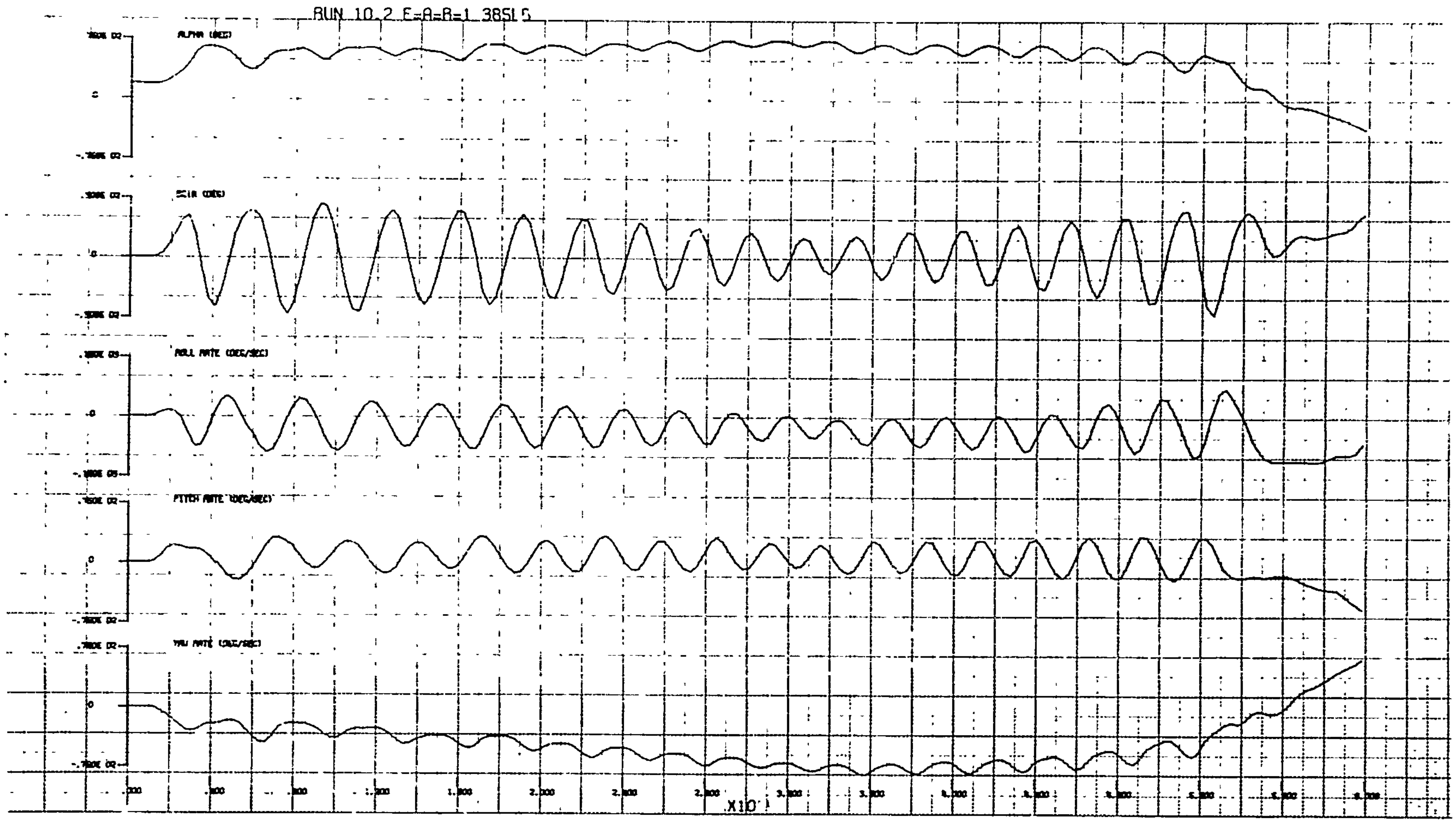


Figure 6-3i. Attempted Spin Recovery with Feedback Controls and Large Stick and Pedal Forces

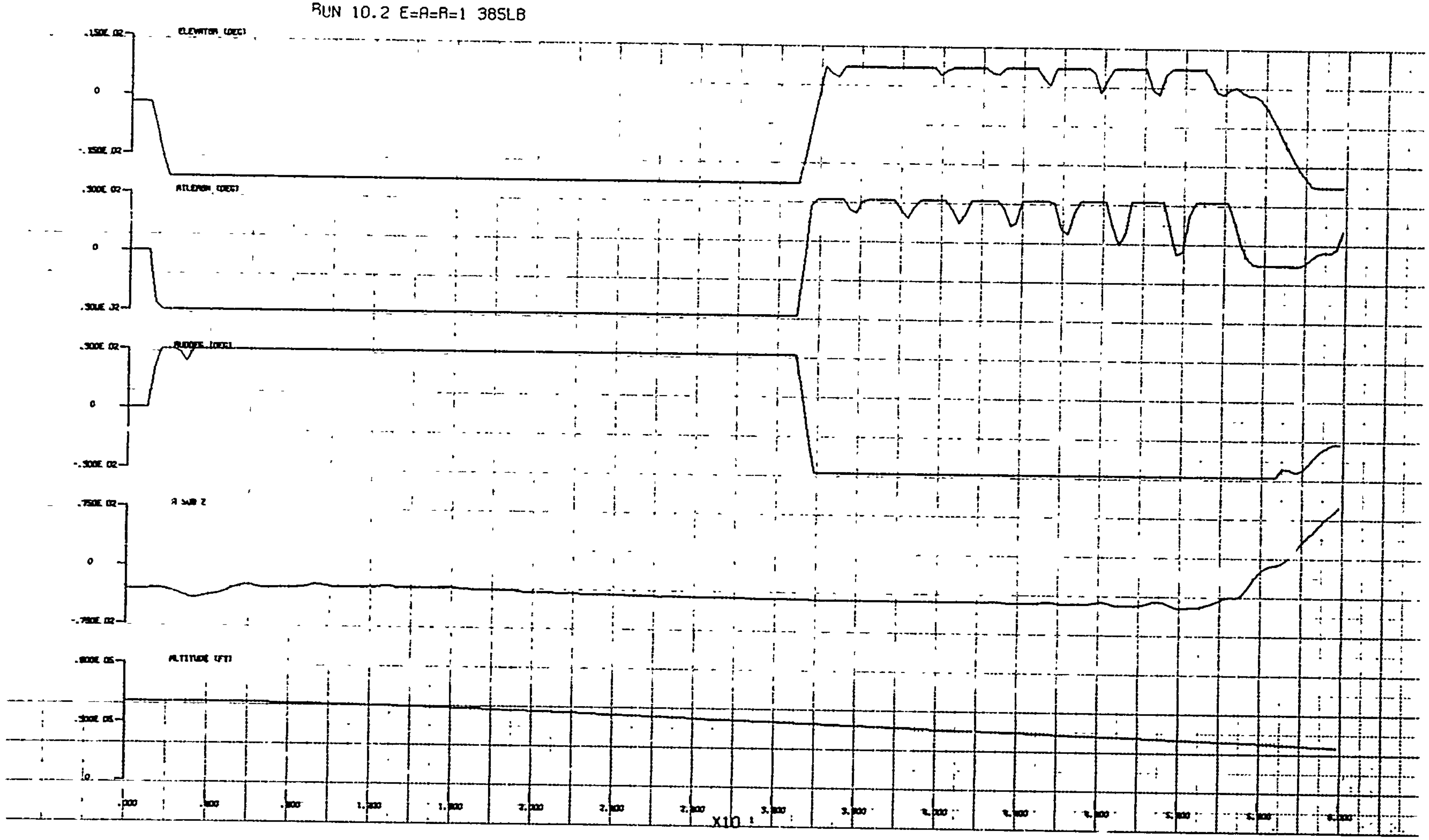


Figure 6-31. Attempted Spin Recovery with Feedback Controls and Large Stick and Pedal Forces (Continued)

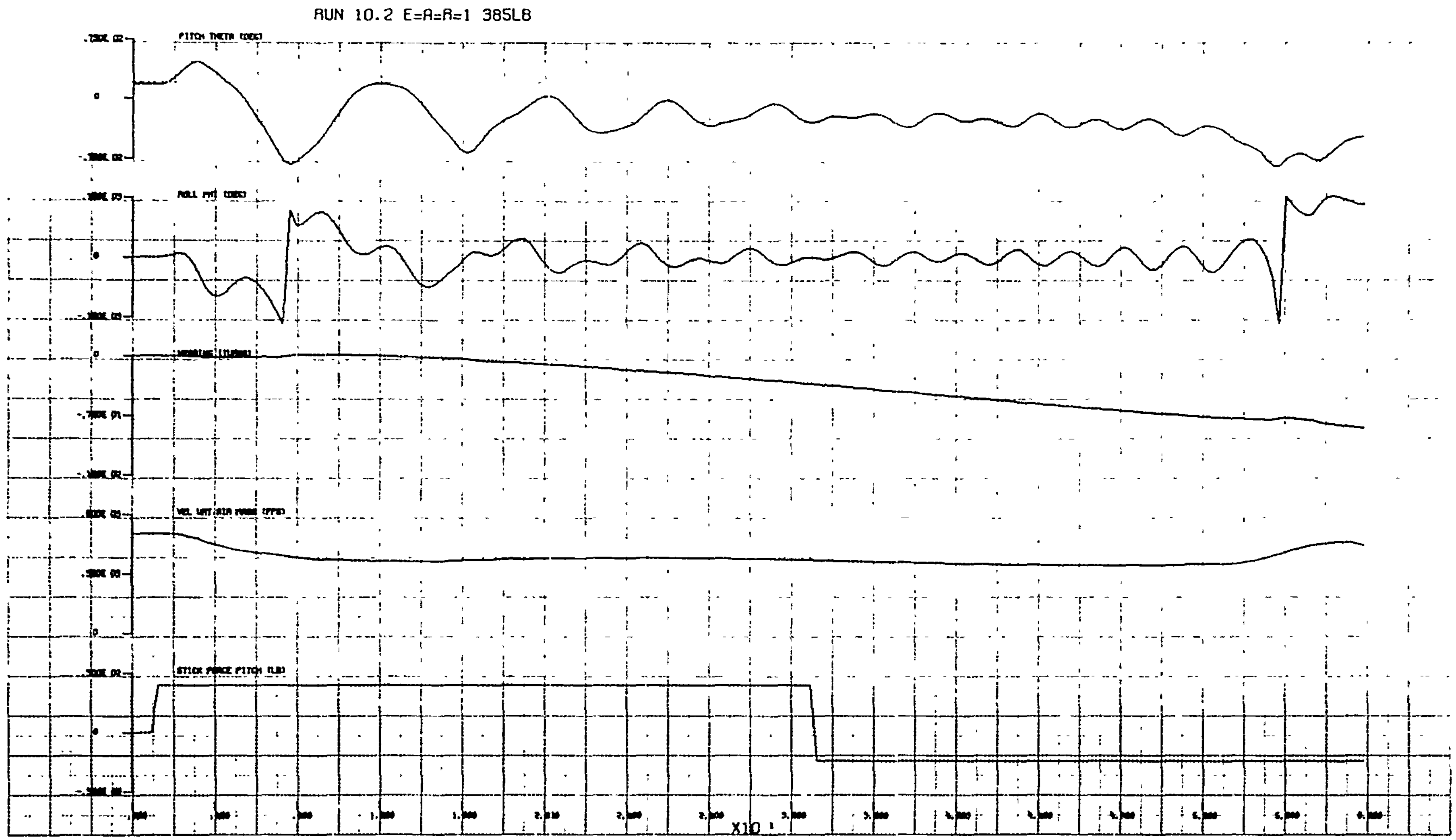
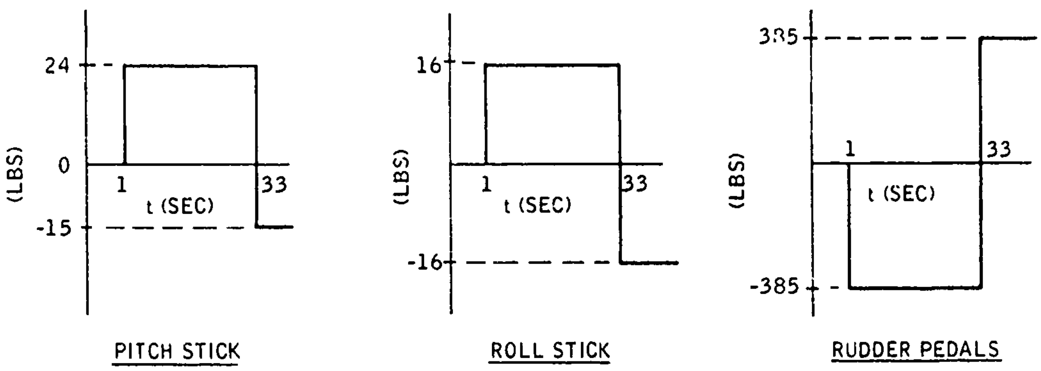


Figure 6-31. Attempted Spin Recovery with Feedback Controls and Large Stick and Pedal Forces (Concluded)

controls were applied at 33 seconds. These recovery controls consisted simply of applying sufficiently large pilot stick and pedal forces to maintain saturated surfaces in the desired recovery position. Hence, the feedback controls, though active, had no impact on either the spin entry and evolution nor on the recovery. This spin then serves as a "norm" for comparison with spins in which the feedbacks are permitted to command surface deflections. The following pilot force levels, applied to the system of Figure 6-5, were used to achieve the spin shown in Figure 6-31.



These were the force levels required to overcome any surface deflections commanded by the feedbacks. An unusually large pedal force, required to achieve saturation of the rudder, is noted. This large force is needed to achieve cross controls (i.e., a positive rudder for a negative aileron command) with yaw-rate and lateral acceleration feedbacks plus an aileron to rudder crossfeed for turn coordination. This large pedal force is probably at least a factor of two greater than that which a pilot can produce. Hence, it is apparent that crossfeed signals can make it very difficult for a pilot to achieve full cross controls. If this is the case, the yaw-axis inputs, unless disengaged, will always have some impact upon the spin behavior simply because they cannot be overpowered by the pilot.

Figure 6-32 shows a spin recovery in which the rudder pedal force was relaxed to +88 pounds at 33 seconds. Figure 6-32 shows that although recovery was achieved, the reduction in yaw rate was noticably slower than in Figure 6-31. Similar to the inhibiting action produced in spin entry, the yaw-rate feedback and the aileron-to-rudder crossfeed both act to also oppose the recovery.

The effect of pitch-axis feedbacks upon recovery is shown in Figure 6-33. In this case the pitch stick force was reduced to -6.5 pounds at 33 seconds. Comparison of Figures 6-33 and 6-32 show the pitch-axis feedbacks to have about the same effect on recovery as the yaw-axis feedbacks. Recovery was accomplished with the pitch-axis feedbacks but at a slower rate than in the case shown in Figure 6-31.

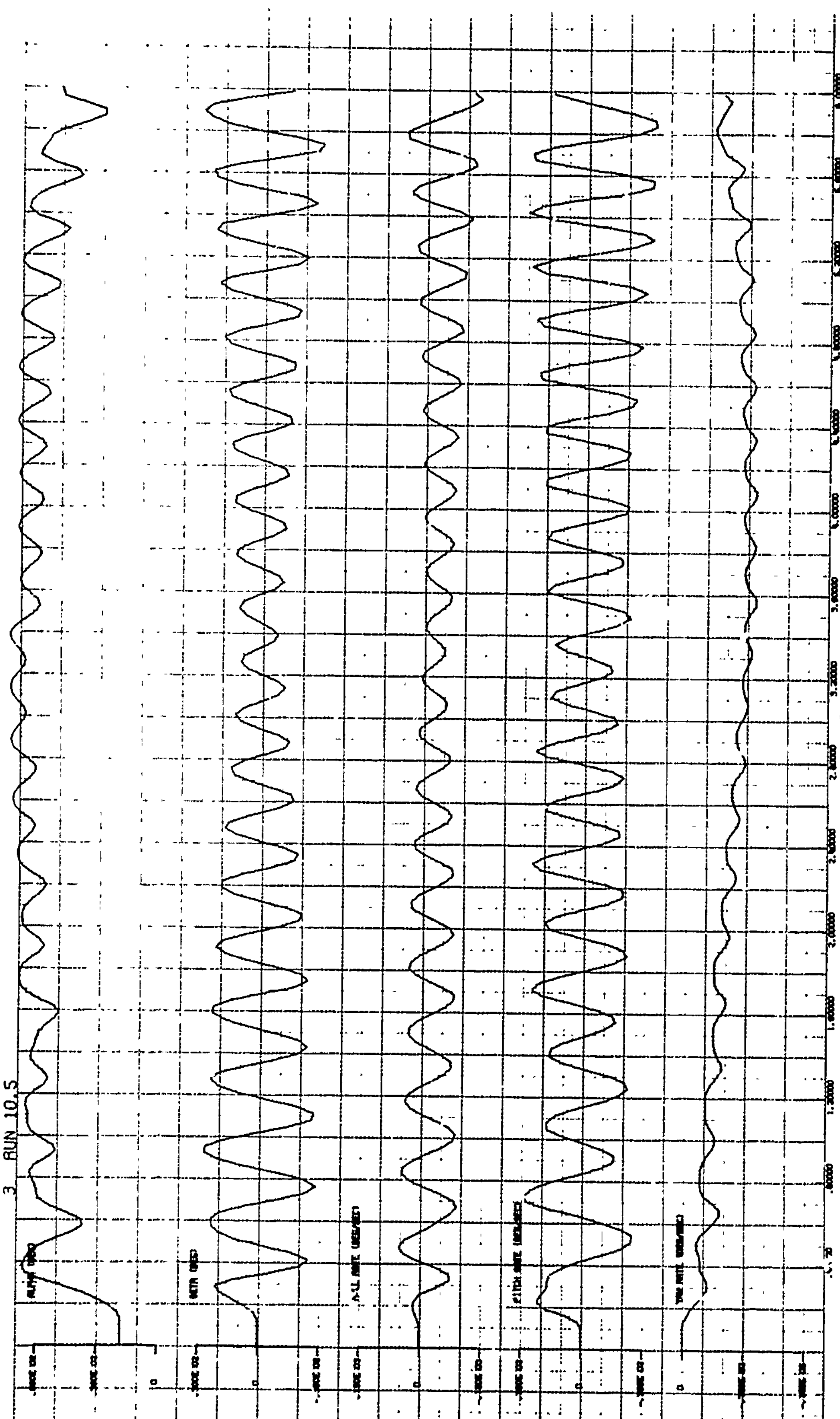


Figure 6-32. Effect of Yaw-Axis Feedbacks on Spin Recovery

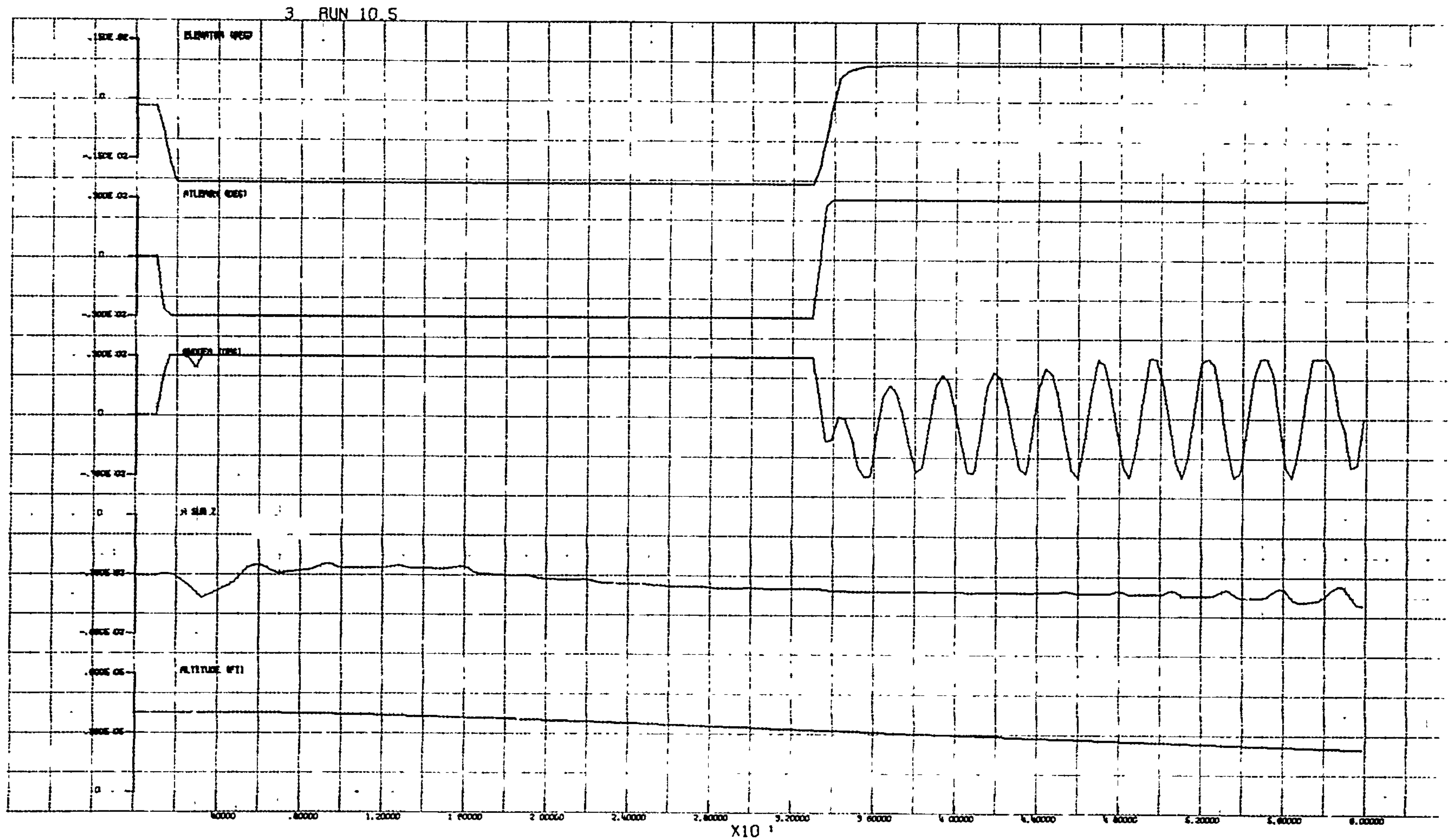


Figure 6-32. Effect of Yaw-Axis Feedbacks on Spin Recovery (Continued)

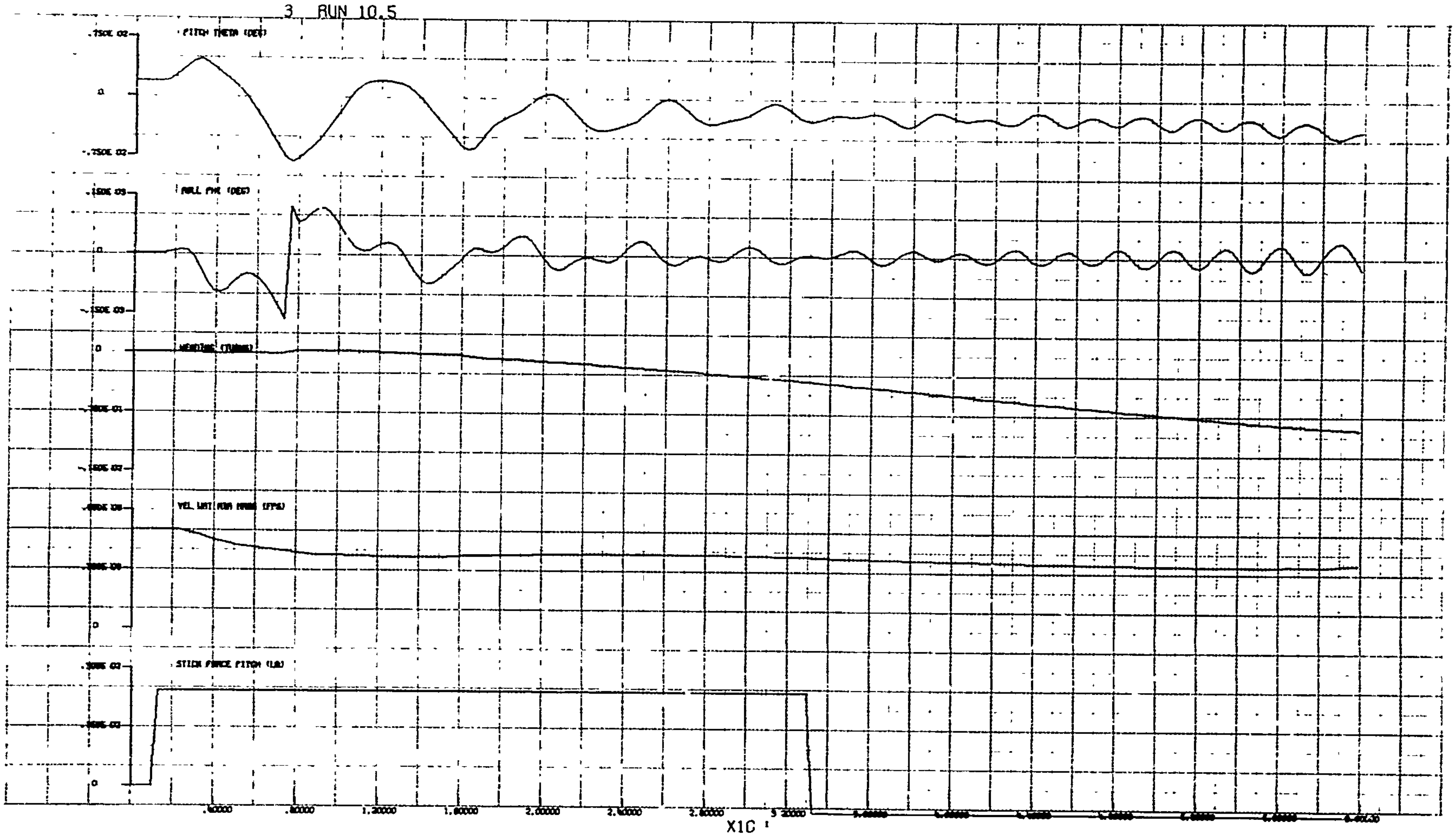


Figure 6-32. Effect of Yaw-Axis Feedbacks on Spin Recovery (Concluded)

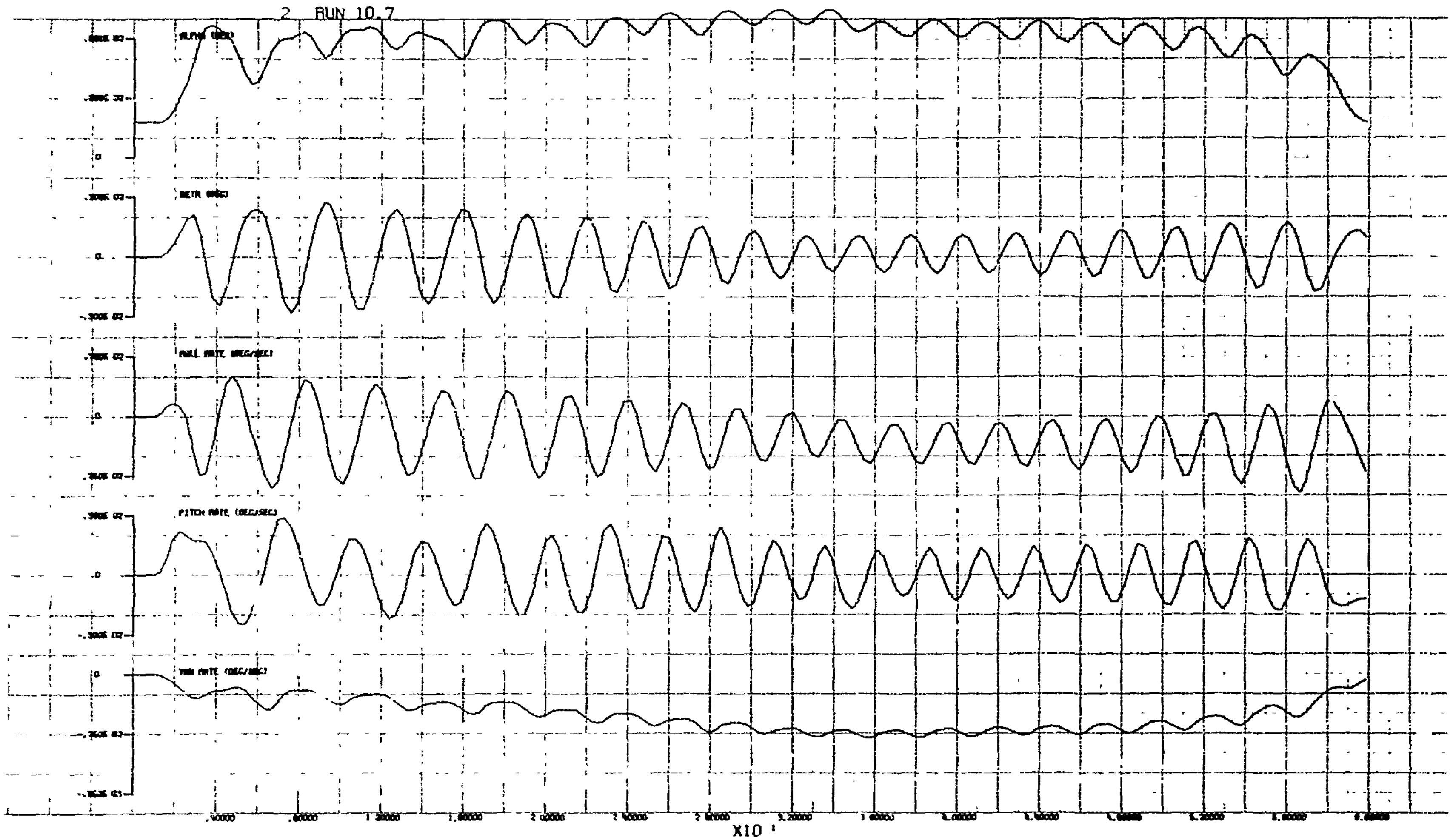


Figure 6-33. Effect of Pitch-Axis Feedbacks on Spin Recovery

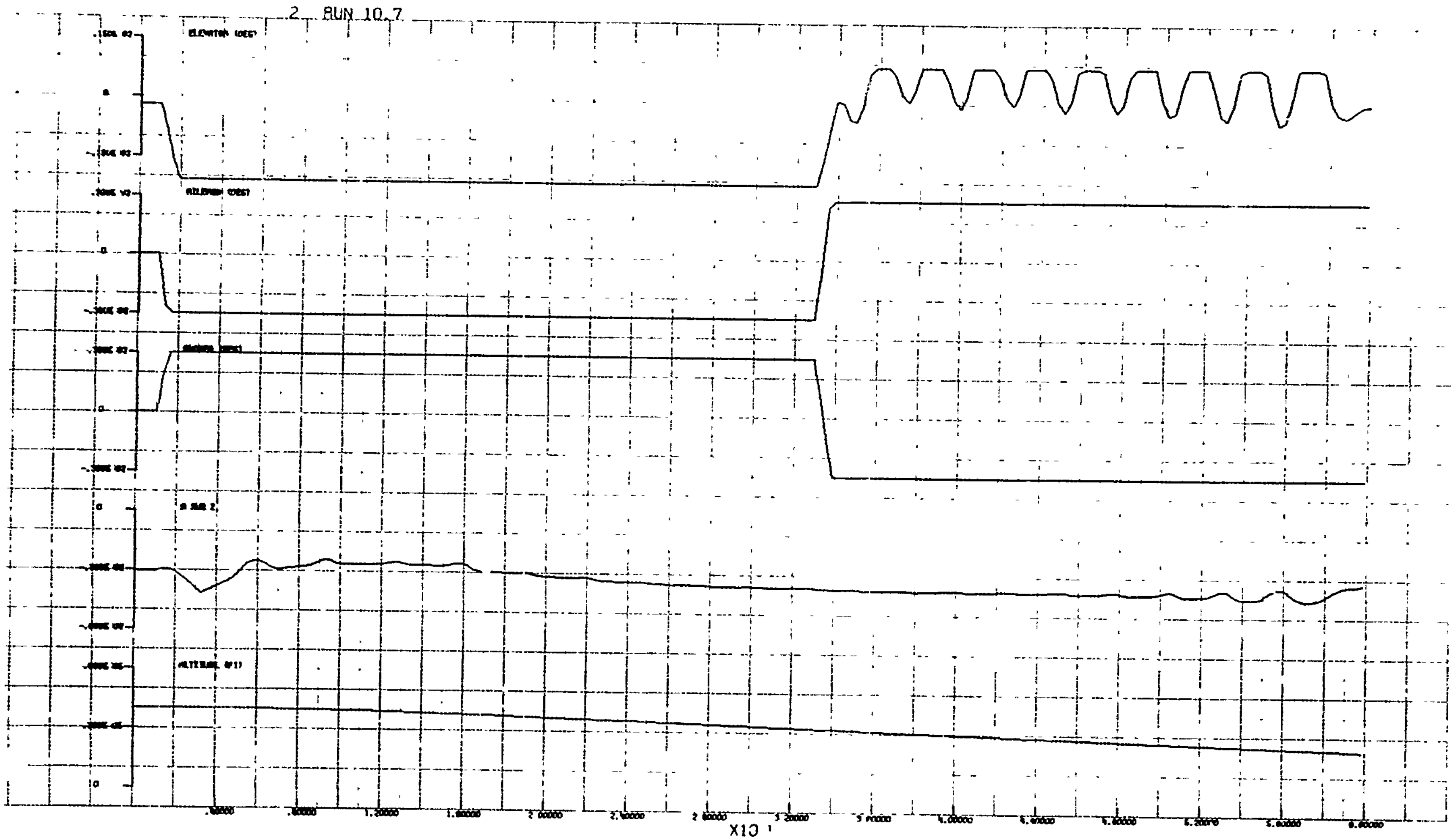


Figure 6-33. Effect of Pitch-Axis Feedbacks on Spin Recovery (Continued)

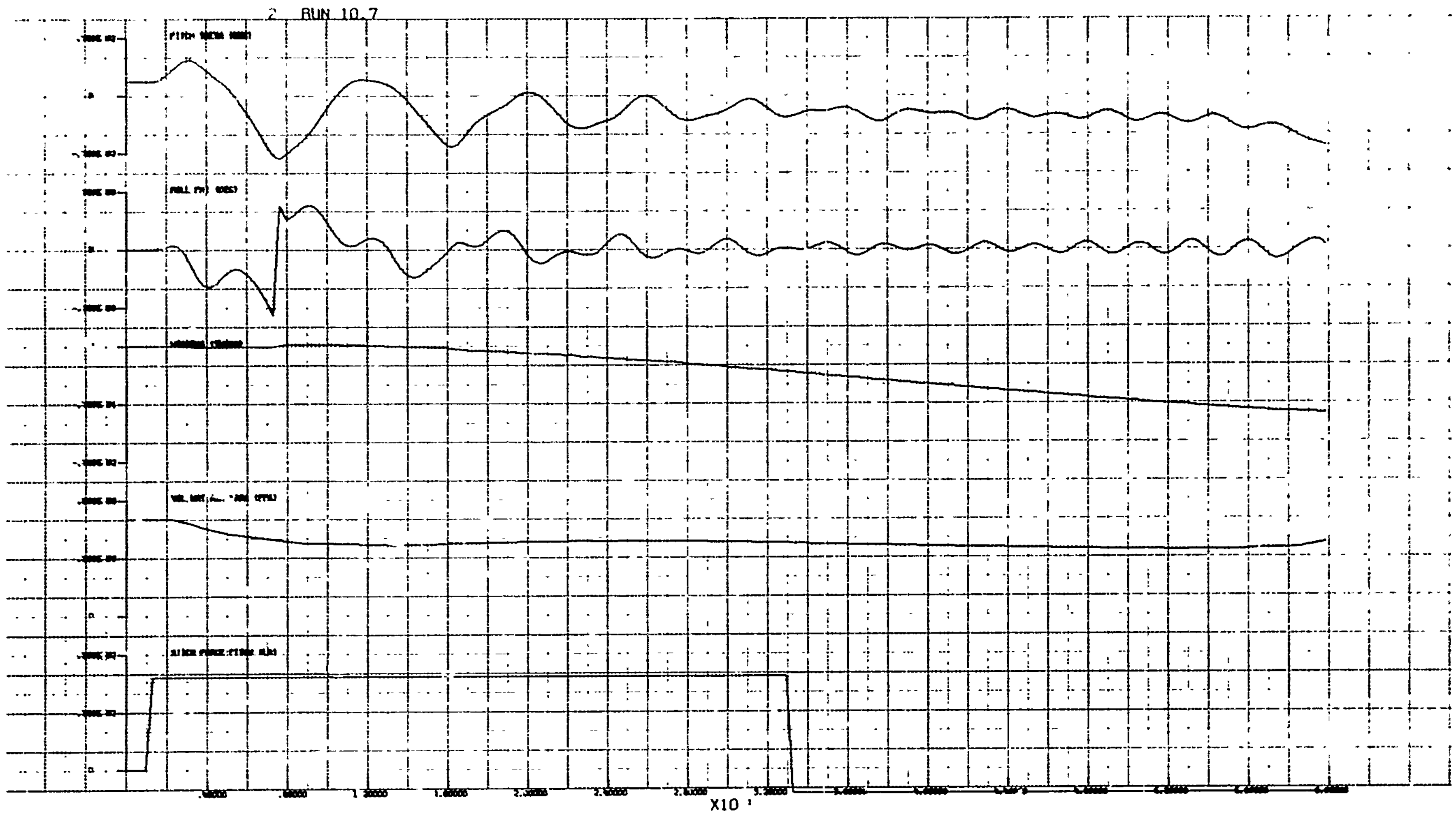


Figure 6-33. Effect of Pitch-Axis Feedbacks on Spin Recovery (Concluded)

Figure 6-17 shows that when both pitch-axis and yaw-axis feedbacks are permitted, the combination of the two prevented recovery at 33 seconds. Thus, their inhibiting action appears to be additive.

Figure 6-16 shows that when the roll stick force is also relaxed (to -10 pounds) at 33 seconds, little change occurs from the spin shown in Figure 6-17. All these results substantiate the notion that the feedbacks have an inhibiting action on the recovery, primarily because they consume surface deflections better used for sustained spin reductions. The action of the feedbacks can be overpowered by the pilot in the pitch or roll axes, but not in the yaw axis. Consequently, if yaw-axis feedbacks are permitted in a manually-controlled stalled flight, it may be necessary to inhibit their action for a manual recovery.

CONTROL CRITERIA

Study of control criteria constitutes the second of the two main categories in the stall/spin maneuver study. This portion of the study was concerned with deriving control requirements from the results of the basic aircraft analysis, the first of the two categories of study.

This subsection describes a nominal feedback control system for normal flight and the impact of stall/spin maneuvers on this system. Preliminary requirements for stall prevention are presented together with a potential stall prevention system. Finally, a control philosophy is discussed for stall recovery and the transition between normal and stalled flight modes.

Normal Mode Control

The basic system configurations for normal flight were shown in Figures 6-5 and 6-6 and described in the accompanying text.

Stall Inhibition

The normal controls shown in Figures 6-5 and 6-6 must be modified to provide the desired control functions at the high angle-of-attack conditions. One aspect of this modification is to provide satisfactory stall inhibition. Automatic stall inhibition is desirable to achieve the maximum maneuvering capability of the airplane in situations where the pilot is under stress. There may also arise situations where the pilot may wish to intentionally stall the aircraft (remotely possible in a combat maneuver). As a result, any eventual design of the stall inhibitor should provide the capability of being overridden by conscious pilot effort. These features and others lead to specification of the following preliminary set of requirements for stall inhibition:

- The stall inhibitor should limit airplane angle of attack to that corresponding to maximum usable lift coefficient, commensurate with limits on longitudinal or lateral-directional stability;

- The pilot should have the capability of overriding the stall inhibitor with intentional application of a large stick force (e.g., 45 pounds for a center stick);

- Stall inhibition should be provided without dependence on the existence of a particular range of pilot stick force;

- A stall inhibitor should function properly in a gust environment;

- A stall inhibitor should be compatible with any control modes used in stalled flight.

For the F-4 aircraft the first requirement restricts the useful angle of attack to approximately 23 degrees even though maximum lift occurs around 27 degrees. In this sense, aircraft maneuverability could be expanded by lateral-directional static stability augmentation to permit increased lift.

The stall-inhibiting action provided by the system shown in Figure 6-5 did not entirely satisfy the above requirements. Stall inhibition in the system of Figure 6-5 was provided by operating the gain, K_α, as a function of angle of attack. As indicated previously, it was determined that the pitch-axis feedbacks should not be removed because they provide desirable stall-inhibiting action and they aid the pilot in controlling angle of attack in the stalled flight mode. An initial modification of the system shown in Figure 6-5 was to sum in the feedbacks downstream of the K_α gain element. The K_α gain logic was also modified to provide additional damping to minimize angle-of-attack overshoot when the stall limit is approached. This gain logic is shown in Figure 6-34. High-passed angle of attack was used for damping augmentation, although pitch rate could probably suffice in an eventual mechanization. This configuration was attractive because of its relative simplicity and its compatibility with desired stalled flight control functions. The proposed control philosophy for stalled flight in the pitch axis is to provide a lower stick force gradient and to retain the normal mode feedbacks. Although the configuration of Figure 6-34 provided effective stall limiting under controlled conditions, there are inherent weaknesses in its design. A major deficiency is that it is dependent on the presence of a large stick force to provide a limiting action. Consequently, if the pilot continued to trim his force as he reduced airspeed, he could trim into a stall condition.

The stall inhibitor was, therefore, modified to include an angle-of-attack feedback to overcome the dependence on the presence of a large stick force. Figure 6-35 shows the stall inhibitor with this modification. The angle-of-attack feedback becomes active only when K_α becomes less than unity and the

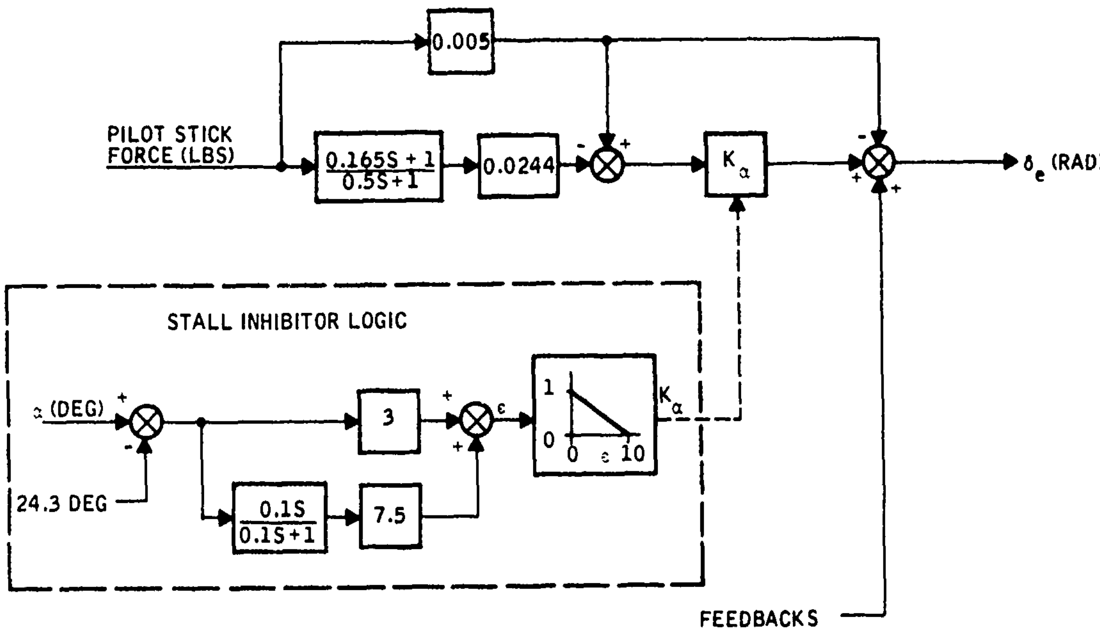


Figure 6-34. Initial Pitch Stall Inhibitor

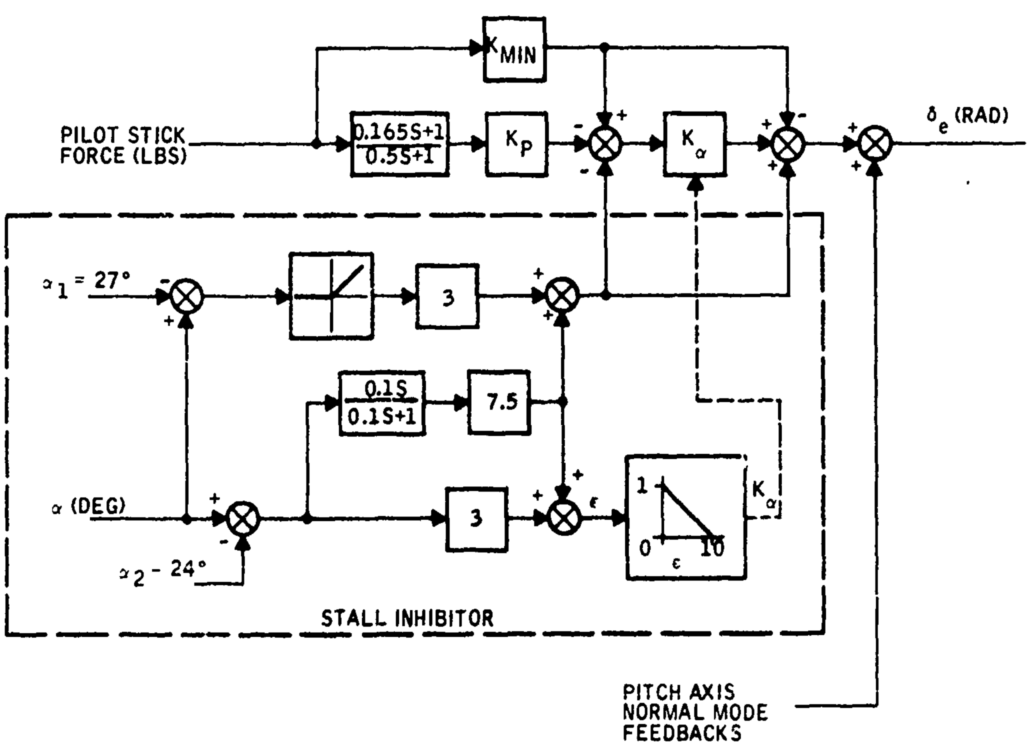


Figure 6-35. Modified Stall Inhibitor

angle of attack becomes greater than the desired stall angle of attack. As can be seen, this configuration was designed to limit at 27 degrees AOA. Unless the lateral-directional stability is augmented, this limit would have to be reduced to around 23 degrees. Figure 6-36 shows the angle-of-attack limiting provided by this system for a 40-pound step pilot stick force. Figure 6-36 shows the angle of attack actually limited around 29 degrees. Subsequent divergence of the lateral-directional axis at this angle of attack resulted in aircraft stall. This could be corrected by reduction of the angle-of-attack limit or possibly by augmentation of the lateral-directional stability. A fault with the design of Figure 6-35 can be seen in Figure 6-36. It required elevator deflection rates greater than the normal limit of 25 deg/sec. Adherence to these limits would degrade the angle-of-attack limiting. The excessive surface rate problem could be solved, however, by changing the set points and gains used in the stall inhibitor logic and by substitution of pitch rate for high-passed angle of attack.

The questionability of an angle-of-attack feedback in the deeply stalled flight region suggests an additional modification of the scheme shown in Figure 6-35. This would entail resetting K_α back to one for intentional penetration of the stalled flight region. Restoring K_α to one for stalled flight could be accomplished by using logic based on existing stick force. If stalled flight is intended, then the pilot must achieve it by applying a large stick force, say for example, a force greater than 45 pounds.

The predominant development problems which remain with the system shown in Figure 6-35 have to do with establishing the transition strategy for an intentional stall and return to normal flight. Unfortunately, the scope of the study precluded finding a final satisfactory solution. This solution must also include a compatible control strategy for the lateral-directional axes, particularly in the stalled region.

Recoverable Stall

To recall, stalled flight was broken into two regions -- a recoverable region and a non-recoverable region. The boundary between them was determined by an outer controllability limit beyond which the aerodynamic control surfaces were insufficient to achieve recovery.

The pitch axis control philosophy for the recoverable region is to provide primarily an attitude rate command mode. Angle-of-attack feedback in the stalled flight region is considered unreliable because of the unknown behavior of the local flow field at the angle-of-attack sensor in this region. It is desirable to maintain the same feedbacks in the pitch axis as are used in normal flight for the sake of simplicity and to accommodate potential static stabilization requirements in future aircraft. The normal acceleration feedback can probably be retained because it has little effect at low dynamic pressures. The control philosophy for the lateral-directional axis in stalled flight was not

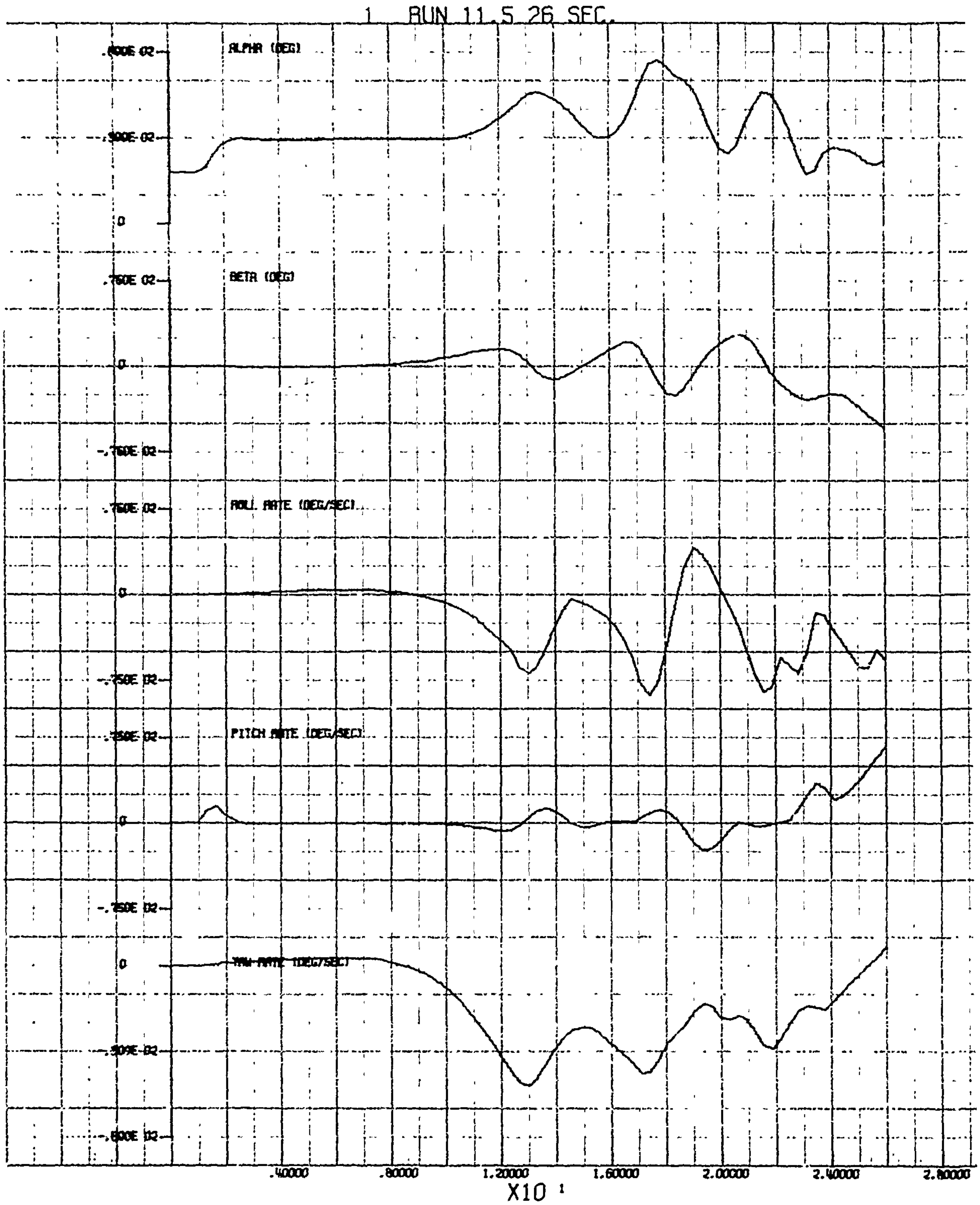


Figure 6-36. Stall Inhibitor Performance

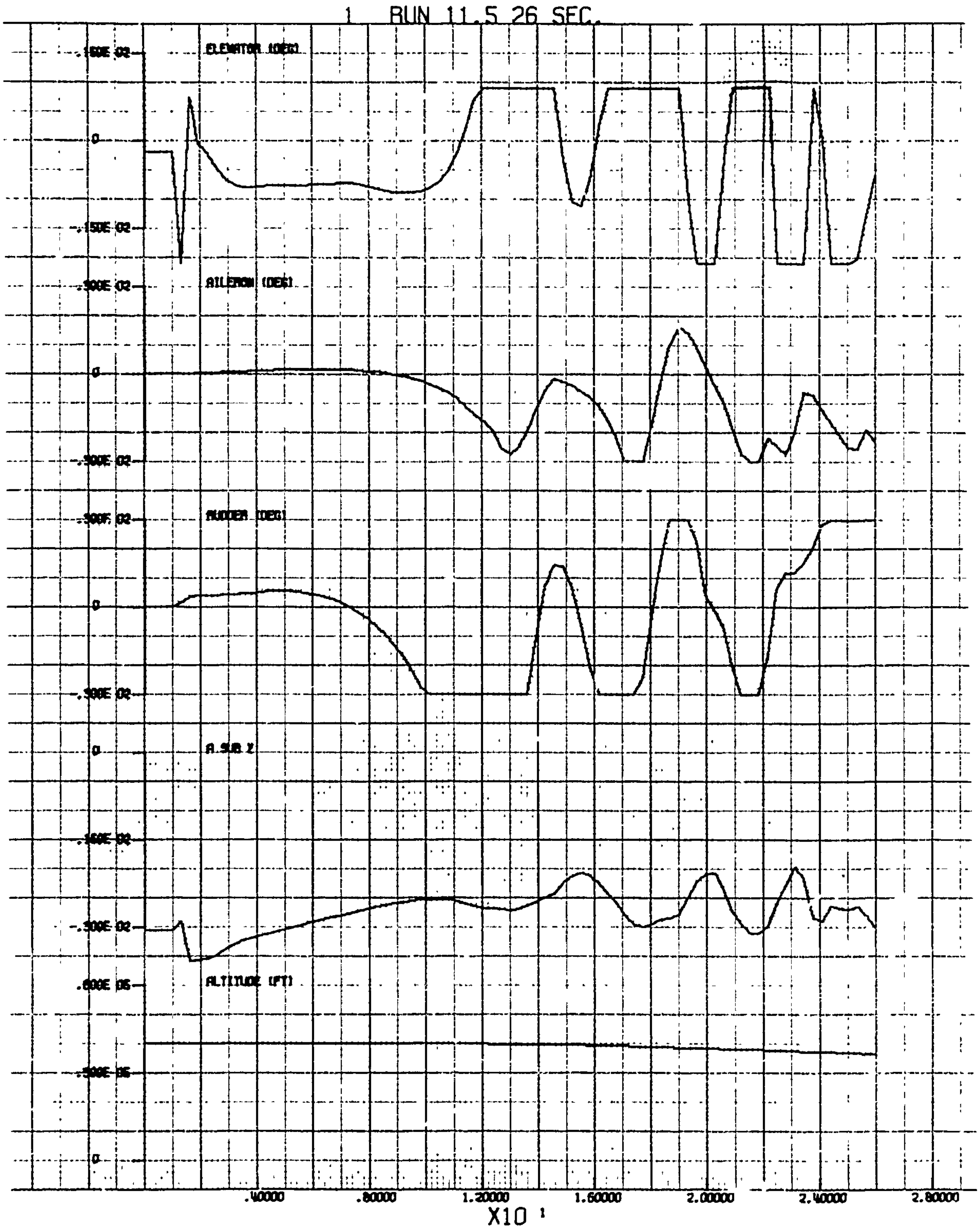


Figure 6-36. Stall Inhibitor Performance (Continued)

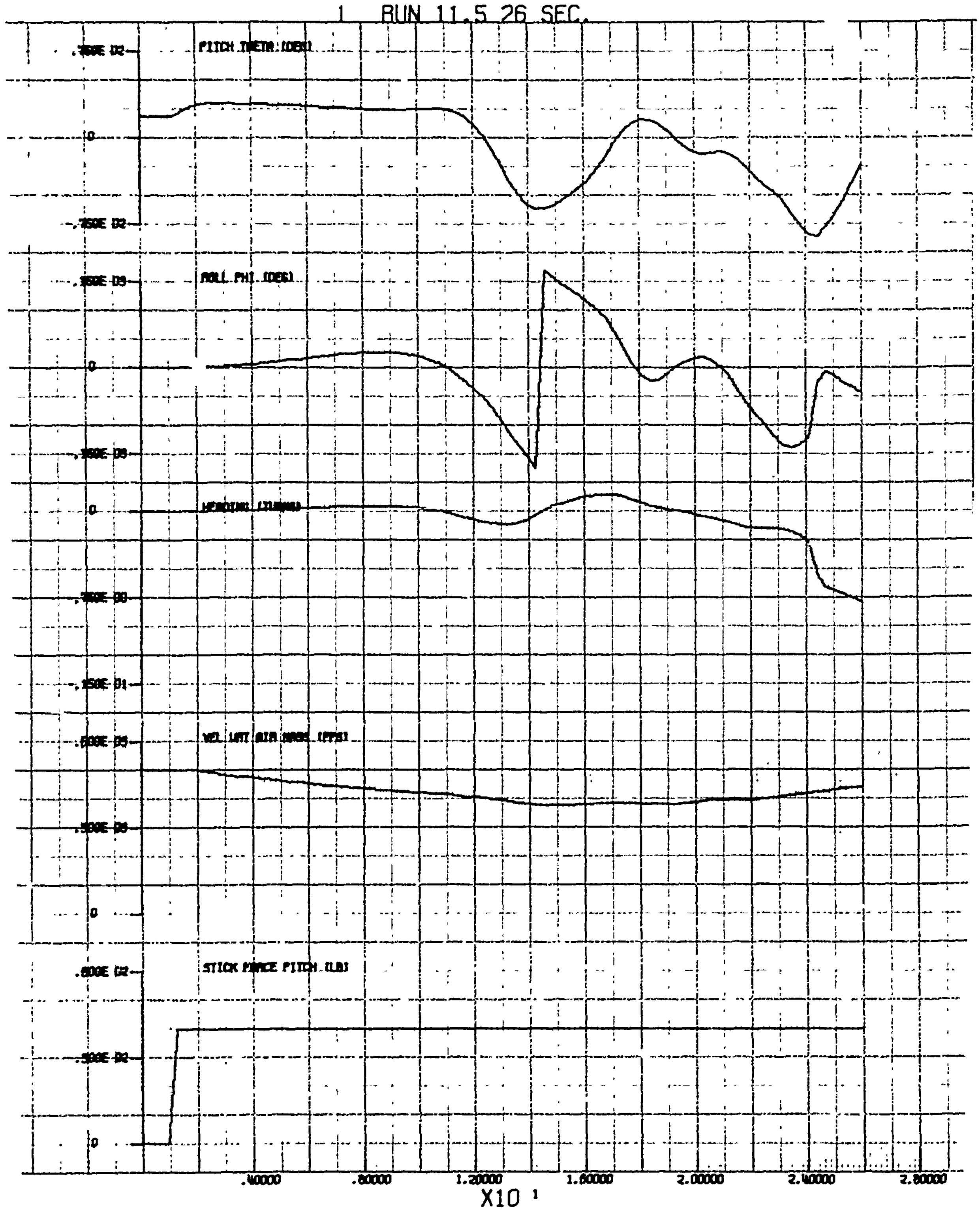


Figure 6-36. Stall Inhibitor Performance (Concluded)

analyzed in the study. It is evident, however, that the roll-axis control will differ from that in the normal mode. Ailerons become the primary agent for achieving directional control at high angle of attack with aileron roll reversal occurring above 55 degrees AOA.

Exit from the stalled flight region is conceivable by two means:

- The pilot manually controls the transition to normal flight.
- Automatic recovery controls are used either by pilot option or automatically when the aircraft nears the outer controllability boundary.

The basic aircraft studies demonstrated a preferred combination of surface positions for stall/spin recovery, i.e., full deflections of down elevator, anti-spin rudder, and pro-spin aileron. Automatic application of these surface positions would assure recovery in all cases.

Transition Strategy

The strategy for transitioning from normal flight to flight on the stall limit, to flight in the stalled flight mode, to a recovery mode, and finally back to normal flight has only been partially answered in the study. The problem of regaining normal control from the recovery mode or from the stalled flight mode is unresolved. Policies regarding the treatment of trim, stick feel, and the degree of automation in the mode transition have yet to be established. It has been conjectured, however, that recovery from the stalled flight region should entail automatic reversion to the normal feedback control mode to provide tractable aircraft stability under a situation which is very demanding of the pilot. This procedure is definitely necessary if the aircraft exhibits any degree of short-period instability. The trim situation indicates a preference for zero g's in pitch and yaw, with zero rate in roll, all of which can be initialized with streamlined average surface positions. This will offer the maximum ability to restore lost airspeed and engine operation.

SUMMARY AND CONCLUSIONS

Figure 6-37 summarizes the procedures and results of the stall-spin maneuver study and recommends future work in the area. The study was broken into two major areas -- basic aircraft analysis and development of control criteria. These two major areas were further divided as indicated in Figure 6-37. Most of the study effort was expended under the first major area - basic studies.

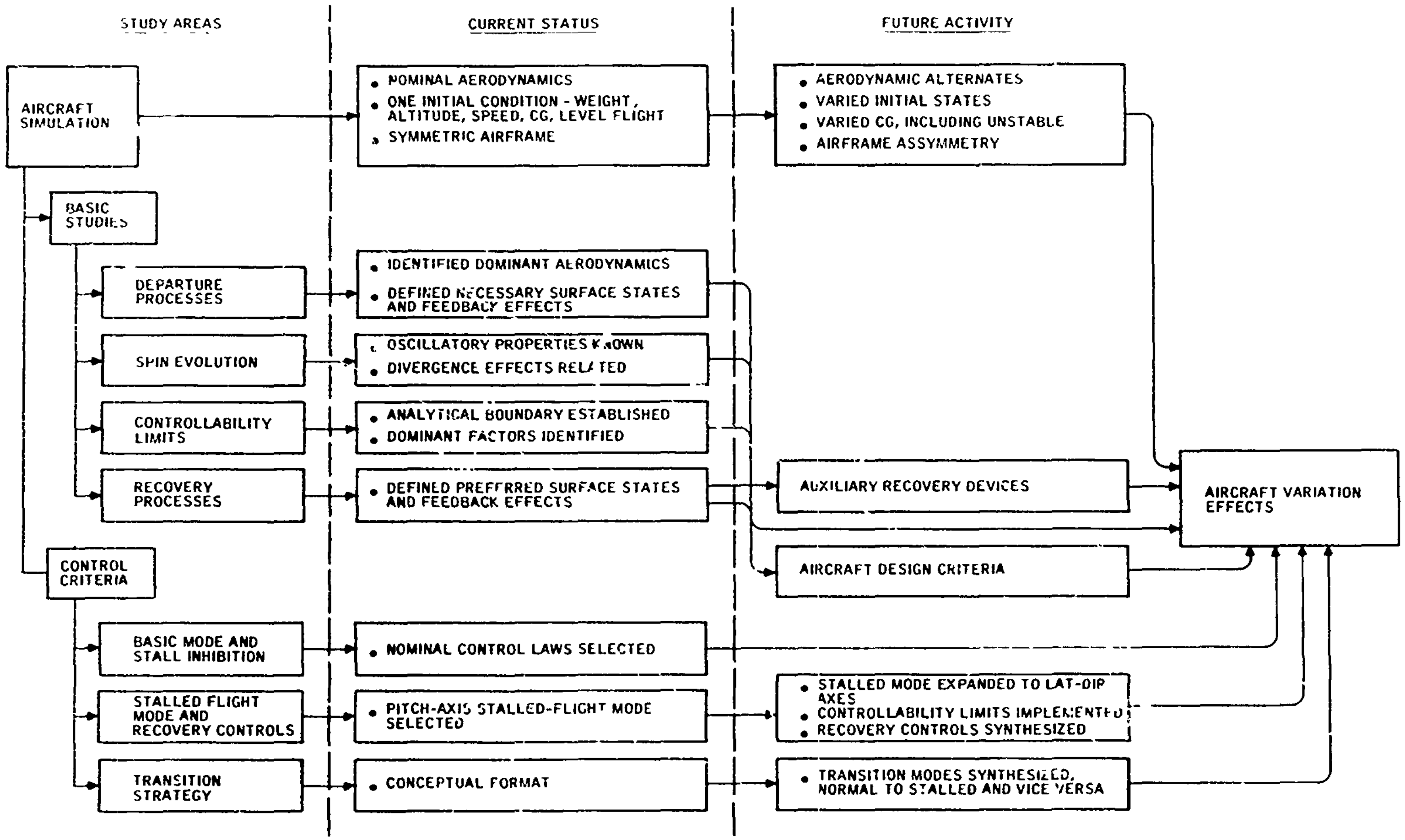


Figure 6-37. Stall/Spin Study Growth

The key analysis tool used in the study was a complete six-degree-of-freedom all-altitude computer simulation of an F-4 aircraft. The simulation included nonlinear equations and rotary balance derivatives (for spinning flight), valid for subsonic conditions. The aircraft inertial properties were fixed for one cg configuration with no external stores.

The majority of the effort was applied to obtain basic understanding of stall/spin phenomena for the F-4. For this airplane, departure from normal flight is exemplified by a loss of lateral-directional stability for the angle-of-attack range from 23 to 37 degrees. Thus, although maximum lift occurs around 27 degrees angle of attack, this instability essentially limits the available angle of attack for normal flight to 23 degrees. Having achieved the AOA adequate for stall conditions, the primary agent for effecting departure is aileron deflection. A dominance of aileron yawing moment in conjunction with dihedral effect produces a spin in a direction opposite to normal aileron action, exemplified by an initial snap roll. The dihedral effect also accounts for restored lateral-directional stability at high angles of attack.

Pitch- and yaw-axis feedbacks of typical augmentation systems provide an inhibiting action on departure. Roll rate to aileron tends to aggravate departure. For the F-4, however, the degradation is relatively minor, although disablement appears advisable to avoid marginal cases and obtain a preferred recovery configuration.

Initial spin development requires continued application of departure-producing controls. Spin recovery can be accomplished in incipient spins by immediate application of full-down elevator and neutralized ailerons and rudders. Recovery from more developed spins requires full-down elevator, full pro-spin aileron, and full anti-spin rudder. After the spin has achieved a sufficient energy state, the aerodynamic controls are completely ineffective for recovery. The criteria for the outer controllability limit developed in the study was based on the moment equations. Analysis of these equations showed that controllability of yawing moments was the critical factor. The most effective means for controlling yawing moments was through the ailerons. The rudder was ineffective at high AOA except when sideslip oscillations were large. Elevator effectiveness for recovery diminished rapidly as the aircraft approached a higher degree of spin stability. Reduction of the spin rate (predominately yaw rate) by pro-spin deflection of the ailerons was the most effective means for recovery at high spin rates.

Feedbacks were found to be primarily detrimental for spin recovery. Surface deflections resulting from feedback rate damping detracted from that available for recovery.

Much of the effort expended in the study of control criteria was in the analysis of stall inhibition. A set of requirements for a stall inhibitor were specified based upon the assumption that intentional stalling of the aircraft by pilot override would be permitted. A preliminary stall inhibitor was defined and tested in the simulation.

Analysis of control criter'a for the stalled flight mode led to the conclusion that both a manual and an automatic recovery mode be provided. The automatic system would be engaged at pilot option or automatically if the aircraft approached the outer controllability limit for stalled flight.

Future efforts proposed for stall/spin studies are considered within the framework of the study approach shown in Figure 6-37. Activities are necessary to complete nominal control criteria and to investigate their applicability to alternate airframe characteristics. This process is illustrated in Figure 6-37 which identifies current status and relates it to logical growth developments.

Expansion of the current airplane simulation is proposed to include the variational effects listed in Figure 6-37. Alternate aerodynamic data is available in the literature (Ref. 6-4) and may easily be applied to the current digital simulation. Of particular importance is the potential impact of the statically unstable vehicle, an expected result of current control-configured-vehicle developments. It is clear that such a vehicle would greatly influence the nature of the control law in the stalled flight region (where no normal acceleration feedback is available for artificial static stability) and for the spin recovery where automatic reversion to feedback control would be mandatory. The very nature of the spin would of course be different, with an AOA greater than 90 degrees likely for developed spins.

Study tasks remaining before nominal control laws can be specified include completion of the stalled flight control modes, synthesis of the recovery controls, and synthesis of the transition means for accomplishing mode switching. The lateral directional control of the F-4 at high AOA (above stall) is essentially limited to only ailerons because of an ineffective rudder. The primary control moment is then $N\delta_a$ and $L\beta$, causing an apparent roll reversal. This will require corresponding changes in the lateral control law if reasonable attitude regulation is to be accomplished during stalled flight.

Synthesis of recovery controls will require implementation of the controllability limit function and selection of the surface control laws. The former will involve tradeoffs between complexity and accuracy. The surface controls for F-4 recovery will be full deflection sets selected by rate and acceleration logic statements.

The required mode transition means currently exist only in conceptual form, and their synthesis will demand much further study and verification by simulation. The transition phases include the following:

- Passage from the stall limit to the stalled flight mode;
- Passage from the stalled flight back through the stall limit to normal control by the pilot;

- Engagement of recovery controls by the pilot in stalled flight and automatic reversion to normal flight
- Automatic engagement of recovery controls at the controllability limit and reversion to normal flight

Having the above controls in at least a nominal form for the F-4, study of aircraft variation effects can proceed as outlined in Figure 6-37.

Also noted on Figure 6-37 are other potential aspects to this investigation which would profit from the basic studies. Auxiliary recovery devices, such as spin chutes and reaction jets, can be applied to extend the available recovery envelope beyond that possible with the existing surfaces (see Figure 6-1). General forcing functions can be applied to the simulation at key points on the airframe, and the recovery potential related to various thrust levels and time histories. This parametric data would then enable evaluation of available devices.

A second related study subject is the development of aircraft design criteria for inherent departure resistance and improved recovery capability. This is obviously a very broad subject area which in itself can require extensive study.

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SECTION VII

CONTROL SYSTEM CHARACTERISTICS AFFECTING FLYING QUALITIES

This section introduces the involved question of performance requirements for closed-loop primary flight controls with two contributions:

- A categorization of the dominant performance characteristics affecting flying qualities relative to particular mission tasks;
- An assessment of the effects of current augmentation systems on each of the dominant characteristics.

An attempt was made to express qualities and requirements in terms most meaningful to the pilot to facilitate communication and minimize translation errors.

The dominant characteristics and their relationship to mission tasks were defined by reviewing current specifications [(e.g., MIL-F-8785B (ASG)] and through discussions with pilot personnel from the organizations identified in Appendix I. Table 7-1 presents these results. It is notable that the average operational pilot, while very sensitive to handling characteristics of his airplane, has difficulty identifying the precise properties which influence a particular mission task. Furthermore, when questioned about his concept of ideal characteristics, he can give only very general impressions, ones heavily biased by the constraints of conventional airframes and systems and, naturally, relative to his own experience in the particular aircraft. In spite of these difficulties, however, pilots are generally keen judges of their relative performance in accomplishing a particular task and in the effectiveness of the automatic control equipment which is supplied to assist them. The principal worth in the data presented here, therefore, is twofold:

- Identification of qualities significant to particular tasks offers a basis for producing mission/dependent characteristics when such capabilities become available from advanced control systems;
- Knowledge of the real contributions made by current automatic flight controls offers a measure of the validity of current design criteria and the compromises elected in their application.

Data for the latter area is summarized in Tables 7-2 through 7-5 for the F-4, A-7, F101, and F-111 aircraft respectively.

Conclusions and recommendations resulting from these data are presented in Section III.

However, to meet the objectives of this study, it was sufficient to restrict the initial conditions to one flight condition. This condition was a one-g trimmed flight at a 40,000-foot altitude with a velocity of 501 ft/sec. The trim angle of attack was 17.9 degrees.

Departure and the subsequent aircraft behavior is very sensitive to the manner in which the control surfaces are deflected. Whether the airplane goes into a rolling departure, a mildly oscillatory spin, a flat spin, etc., depends on the initially applied controls. This phenomena has been fairly well documented in past studies (e.g., References 6-1 and 6-2). For the study of spin evolution, controllability limits, and recovery, most of the spins were established by using full-up (TEU) elevator, full-anti-spin aileron, and full-pro-spin rudder[1]. Figure 6-7 shows a spin time history[2] wherein all surfaces were deflected at the same rate (15 deg/sec) and all three surface deflections were initiated at the same time (t = 1 second). As can be seen in Figure 6-7, these controls produced a mildly oscillatory, high-angle-of-attack spin which progressed into a flat spin. It has been established in past studies that a flat spin is nonrecoverable with the aerodynamic controls. Hence, this spin behavior was of particular interest for determining the controllability limits of the aerodynamic control surfaces and the impact of these limits on the flight controls design.

The departure process for the spin shown in Figure 6-7 is studied by first examining the pitching moment equation for the first 4 seconds after application of controls. The major aerodynamic and inertial pitching moments are plotted in Figure 6-8 for the first 4 seconds of flight. Since angle of attack increases monotonically with time in this time period the moments were plotted as a function of angle of attack for the purpose f the discussion. The corresponding time points are indicated on each curve, however. When trailing-edge-up elevator is applied, the angle of attack begins to increase because of the imbalance in the pitching moment equation. The net aerodynamic pitching moment increases until the angle of attack reaches approximately 22 degrees. Initially the positive pitching moment due to elevator deflection increases more rapidly than the restoring moment due to angle of attack. At approximately 22 degrees AOA, the elevator effectiveness begins to decrease. As a result, the pitch-restoring moment due to angle of attack increases more rapidly than the moment due to surface deflection. Finally, at an angle of attack of approximately 35.5 degrees, the restoring aerodynamic moment due to angle of attack just balances the positive moment due to elevator deflection. Hence, if the lateral-directional axis had remained unperturbed, the vehicle would eventually settle out at a steady angle of attack around 36 degrees; and no spin would result.

[1] The terms "anti" and "pro" refer to the normal surface moment with respect to the existing direction of the spin rate component along the subject axis.

[2] In Figure 6-7 and in the rest of the computer-generated spin time histories reproduced in this section, the horizontal axis is understood to be "time" in seconds.

Table 7-1. Significant Characteristics Affecting Flying Qualities

Mission Task	Significant Characteristics
Formation flight	Stick breakout and deadspot Response rates for small amplitude inputs Overshoot and damping Stick gradients
Air combat maneuvering	Response rates for large amplitude inputs Stick force levels Turn coordination Performance limits (g's, stall, roll rate, and combinations thereof) Control effectiveness at maneuver extremes
Flight path tracking - Ground-controlled intercept or manual terrain following	Turbulence response Overshoot and damping Response rate for small and large inputs Stick breakout and deadspot Stick gradients Turn coordination
Target tracking, air-to-air	Overshoot and damping Response rate for small inputs Stick gradients Stick breakout and deadspot Small amplitude oscillations Turn coordination
In-flight refueling	Stick breakout and deadspot Response rate for small amplitude inputs Overshoot and damping Stick gradients

Table 7-1. Significant Characteristics Affecting Flying Qualities (Concluded)

Mission Task	Significant Characteristics
Target tracking, air-to-ground	Overshoot and damping
	Response rate for small inputs
	Stick gradients
	Stick breakout and deadspot
	Small amplitude oscillations
	Turbulence response
	Trim properties
	Turn coordination
Flight path tracking - ILS, VOR, TACAN	Overshoot and damping
	Response rate for small inputs
	Stick breakout and deadspot
	Stick gradients
	Speed stability
Landing	Response rate for large inputs
	Roll and yaw response to pedals
	Overshoot and damping
	Performance limits (stall)
Takeoff	Response rate for large inputs
	Stick force levels
	Performance limits (stall)

Table 7-2. Effects of the Augmentation System on F-4 Flying Qualities

Characteristic	Augmentation System Effects
Stick breakout and dead spot	None
Response rate for small-amplitude inputs	Adverse effect by roll damper -- it noticeably opposes small tracking commands; roll damper disengages as force increases above 2 pounds and re-engages as force decreases through 2.5 pounds; the re-engagement usually accompanied by slight transient; pitch improved by a slight damper opposition as the axis is basically over sensitive
Response rate for large-amplitude inputs	Pitch reduces sensitivity of basic aircraft, thereby avoiding pilot induced oscillations; negligible effect in roll
Overshoot and damping	Basic aircraft improved in all three axes; damping level satisfactory
Stick force gradients (lbs/g)	Insignificant except at high q in pitch where SAS increases gradient noticeably
Stick force levels	None
Turn coordination	Aileron-to-rudder interconnect improves coordination by feeding in rudder; up to 12° AOA coordination good, and sideslip controlled acceptably; at the high AOA from 15 on up, some pilot rudder input needed
Performance limits -- g's, stalls, roll rate	Roll damper contributes slight pro-spin tendencies at high AOA
Control effect at maneuver extremes	Pitch stability improved; roll stability improvement negligible
Response to turbulence	Improves stability to an acceptable level for mission accomplishment
Small-amplitude oscillations	None
Trim properties	None
Speed stability	None
Response to rudder pedals	No significant effect on manual inputs to the rudder

Table 7-3. Effects of the Augmentation System on A-7 Flying Qualities

Characteristic	Augmentation System Effects
Stick breakout and dead spot	Breakout force and dead spot are reduced, improving precision control Primary flight control system breakout and initial stick feel variable due to viscous damper and spring capsule inconsistencies
Response rate for small-amplitude inputs	Aircraft response to pilot commands accelerated, resulting in more effective tracking
Response rate for large-amplitude inputs	Aircraft response accelerated, thus contributing significantly to the rapid manuever requirement for an attack aircraft
Overshoot and damping	Roll and pitch acceptable on CAS; yaw axis improved by augmentation but still unacceptably underdamped
Stick force gradients (lbs/g)	Pitch and roll gradient variations reduced, making control easier and more consistent
Stick force levels	Stick input command is augmented by CAS, reducing maximum required force levels and improving pilot capability to maneuver aircraft rapidly
Turn coordination	CAS is essential to aircraft maneuvering at all flight conditions; additional improvement still desired at high angles of attack; excessive time required for yaw trim to settle
Performance limits -- g's, stalls, roll rate	CAS in roll is automatically disengaged at 22 units AOA because of the characteristic of introducing pro-spin surface positions
Control effect at maneuver extremes	Unknown
Response to turbulence	Improved from unacceptable to acceptable aircraft behavior, except additional improvement in yaw damping desired

Table 7-3. Effects of the Augmentation System on A-7 Flying Qualities (Concluded)

Characteristic	Augmentation System Effects
Small-amplitude oscillations	No problem
Trim properties	Yaw damper must be engaged for the pilot to manually trim yaw axis -- this requirement is awkward and objectionable to the pilot
Speed stability	Negligible effect
Response to rudder pedals	No deleterious effects at any flight condition

Table 7-4. Effects of the Augmentation System on F-101 Flying Qualities

Characteristic	Augmentation System Effects
Stick breakout and dead spot	No effect (no roll or pitch augmentation)
Response rate for small-amplitude inputs	No effect
Response rate for large-amplitude inputs	No effect
Overshoot and damping	Lateral directional damping significantly improved by yaw damper; pilot opinion evenly divided between damper being essential or important for mission accomplishment
Stick force gradients (lbs/g)	No effect
Stick force levels	No effect
Turn coordination	Lateral directional stability improved throughout flight envelope; sideslip minimized. Performance could stand improvement at high roll rates
Performance limits -- g's, stalls, roll rate	No effect from SAS; command signal limiter system applies stick constraint in pitch to limit both AOA (for pitch up) and load factor
Control effect at maneuver extremes	No effect within pitch limiter boundary
Response to turbulence	Basic aircraft takes 12-13 cycles to damp in smooth air; yaw damper essential for precise control as required in instrument flight
Small-amplitude oscillations	Negligible
Trim properties	No effect in pitch and roll; yaw SAS with lateral accelerometer has caused apparent trim wander due to long setting times

Table 7-4. Effects of the Augmentation System on F-101 Flying Qualities (Concluded)

Characteristic	Augmentation System Effects
Speed stability	No effect
Response to rudder pedals	No significant effect on manual inputs; slight increase in pedal pressure required for cross control as used upon landing

Table 7-5. Effects of the Augmentation System on F-111 Flying Qualities

Characteristic	Augmentation System Effects
Stick breakout and dead spot	CAS is primary pilot control mode; little information available on basic aircraft behavior; CAS breakout and dead-spot characteristics are considered good by pilots
Response rate for small-amplitude inputs	Aircraft response accelerated, particularly at low q; CAS gain does not follow rapidly changing flight conditions -- pitch can be very sensitive and even limit cycle when accelerating speed very rapidly
Response rate for large-amplitude inputs	Aircraft response accelerated, particularily in roll at low q; aircraft is restricted from full-deflection commands
Overshoot and damping	Basic aircraft unacceptable; CAS increases damping to acceptable levels in all axes; yaw could be increased at low q
Stick force gradients (lbs/g)	CAS makes gradients acceptable at all flight conditions
Stick force levels	Force levels are brought by CAS within a pilot-acceptable level at all flight conditions
Turn coordination	CAS makes the aircraft acceptable; some slight adverse yaw occurs during rapid roll commands
Performance limits -- g's, stalls, roll rate	No information; flight envelope is presently restricted, and flight test investigation of stability and control at flight limits is just getting started
Control effect at maneuver extremes	No information
Response to turbulence	Turbulence causes CAS gains to go to a lower level which is desirable; aircraft with wings swept on CAS has excellent characteristics in turbulence

Table 7-5. Effects of the Augmentation System on F-111 Flying Qualities (Concluded)

Characteristic	Augmentation System Effects
Small-amplitude oscillations	Pitch CAS exhibits several cycles of limit cycle if aircraft is rapidly accelerated; this is not considered pilot-acceptable
Trim properties	Pitch CAS provides series autotrim; need display to determine surface position
Speed stability	Neutral speed stability on CAS acceptable to majority of pilots; slight concern about the loss of speed change cues
Response to rudder pedals	Good at all flight conditions; no difficulty with crosswind landings; could use a little more authority at high angle of attack

SECTION VIII

SYSTEM/AIRCRAFT COMPATIBILITY TESTING

The evolution of the feedback control system into the status of flight-essential equipment demands testing to ensure functional adequacy prior to actual flight. Aircraft equipment has always been subjected to reasonably thorough tests, and certain test procedures have become common in the industry, including:

- Qualification;
- Flightworthiness;
- Reliability-demonstration;
- Mockup;
- Iron-bird.

The limitation to all of the above is a substantial degree of isolation from actual operation. This isolation is particularly evident in terms of the actuation system (e. g., actuators, hydraulics, linkages and surfaces) and the flexible structure of the airframe. The objective of the subject study is, therefore, to develop test procedures which can be applied to the total system/airframe combination to provide greater assurance of functional adequacy, particularly in areas of the combined performance of the actuation system with the electronics, and for system stability in the presence of structural flexure. These two areas can be considered as rigid and flexible testing, respectively (or, alternatively, aerodynamic and structural). They will be treated as such in the following subsections, followed in turn by a concluding subsection on total closed-loop testing which combines both aspects.

RIGID-AIRFRAME TESTS

Rigid-airframe tests are designed to demonstrate that the system performance will be achieved in the absence of structural flexure effects. Of particular interest is the action of the actuation system, from initial input to surface deflection output. The actual nonlinearities associated with control linkages will be exercised, and any limit cycles or other undesirable phenomena can be measured. These tests will be both open- and closed-loop in nature.

Actuation System Frequency Response

Frequency response measurements between electrical surface position commands and actual surface deflections should be made over the actuation bandwidth, typically zero to 10 Hz. Lack of aerodynamics aggravates surface backlash, and this can cause pessimistic results if it is not recognized and steps taken to avoid its inclusion in the measurements. Generally, placement of the position transducers directly on the power actuator ram is effective, if one is not already provided as part of the actuator. It is well to obtain some idea of the magnitude of the backlash influenced by aerodynamic loading; for if it is significant (e.g., > 0.05 degree) it can influence the results of the flexibility tests to be run subsequently. Application of temporary antibacklash springs might be considered if a problem exists.

The frequency response testing will verify the math models used in prior system analysis. It will also provide supporting data in interpreting results of subsequent closed-loop testing. To examine limit-cycle potential (due to mechanical nonlinearities), responses should include low-amplitude measurements. Maintenance of uniform effect for the small-amplitude type of mechanical nonlinearity is provided by measuring the responses at fixed output amplitude, i.e., varying the input level as required at each frequency point. Runs at plus and minus 1 percent and 10 percent of full stroke are recommended, out to frequencies where rate or acceleration limits are encountered.

Closed-Loop Rigid-Body Response

A commonly used procedure for obtaining a very comprehensive check of system performance (except for flexibility effects) is to conduct ship-side closed-loop simulation using an analog of the rigid airframe (Ref. 8-1, 8-2, 8-3). Such an arrangement is illustrated in Figure 8-1.

Here the nominal responses to both commands and disturbances as well as stability margins can be evaluated using actual flight hardware. Most important, the actual actuation system is employed, and the associated nonlinearities may be evaluated. The tendency for limit cycles is of prime interest, and their effects in terms of airframe disturbances can be computed directly by the simulated variables. These are limited by paragraph 3.1.3.7.2.1.2 of MIL-F-9490C (USAF) as follows:

- Normal acceleration in cockpit $< \pm 0.02$ g
- Lateral acceleration in cockpit $< \pm 0.01$ g
- Pitch attitude $< \pm 0.1$ deg
- Roll attitude $< \pm 0.15$ deg
- Heading $< \pm 0.1$ deg

These values are reasonable quantities for general flight acceptance [and within the Category A allowances of MIL-F-8785B(ASG)], and their observance by the closed-loop simulation at representative conditions over

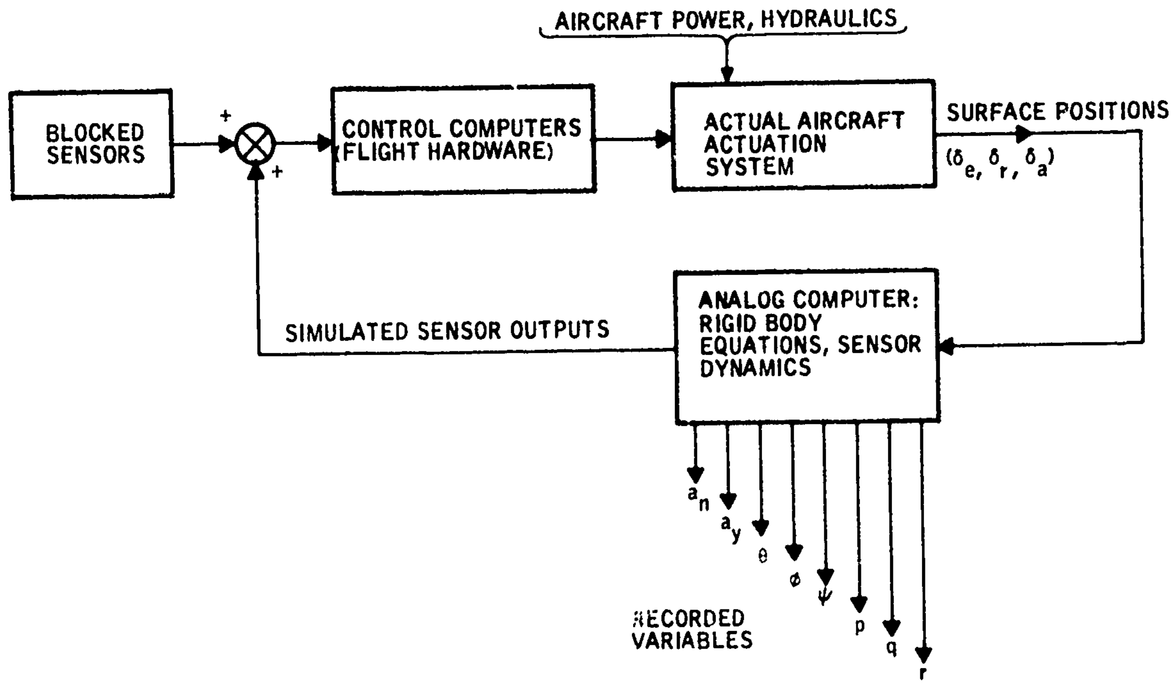


Figure 8-1. Closed-Loop Rigid-Body Testing

the flight range should preserve flight safety. Loop gain perturbations of 6 db at each condition can be applied to evaluate rigid gain margins. The rigid phase margin can be determined by inserting

$$\frac{1 - TS}{1 + TS}$$

in series with the surface position signal, and increasing the time constant until divergence occurs. For each system configuration, step commands and simulated gusts should be applied to ensure excitation of all dynamic modes.

The propensity for limit cycles should be known by prior analysis (and should be eliminated, if possible, by proper system design). This is particularly true for saturation effects from large-amplitude disturbances. If such are a potential problem, appropriate amplitudes should be employed in the simulation.

STRUCTURAL STABILITY TESTS

Determining potential adverse coupling between the feedback control system and the structural bending modes presents the most difficult aspect of compatibility testing. Major differences are evident between in-flight and on-ground structural coupling, for three primary reasons:

- Elevator lack of aerodynamic effectiveness on the ground at zero speeds;
- Airframe support;
- Aerodynamic effects such as intermodal coupling and aerodynamic damping.

The latter effects are relatively easily resolved by the argument that at low-dynamic-pressure flight conditions, they would be negligible. Furthermore, aerodynamic damping would have a stabilizing influence, which would tend to make ground tests conservative. Frequency responses computed for the F-4 at low dynamic measures show little change with or without aerodynamic coupling.

If stability can be verified for the low-q conditions (including, of course, takeoff and landing), the major test hurdle for the high-authority closed-loop PFCS is cleared. The remainder of the flight range can be progressively expanded, with the stability investigated at each increment. This study will, therefore, concentrate on the critical low-q region to evaluate compatibility tests of structural stability.

The questions of elevator aerodynamic forces and airframe support are analyzed in following subsections. Mathematical models are developed for both in-flight and on-ground cases, and the responses produced by each are compared for various influences. Only the pitch axis is studied. Recognizing that the major high-frequency feedback contribution is via pitch rate, the transfer function between pitch rate and elevator deflection at alternate sensor locations is considered.

In-Flight Pitch-Axis Model

The equations relating pitch rate measured at two different locations on the F-4 (stations 383 and 100) are presented by the matrix of Figure 8-2. Station 383 is at the aft section of the wing root (the production gyro location), while station 100 is in the forward fuselage. The former is a position selected for its low first-mode slope, while the latter is selected as having high slopes on the first three symmetric modes to better exhibit response properties. The bending shapes are illustrated in Figure 8-3. Nomenclature for the matrix of Figure 8-2 is given in Table 8-1 (which also includes nomenclature for the following subsection).

$\dot{\theta}_{383}$	$\dot{\theta}_{100}$	α	θ	η_1	η_2	η_3		δ_e
0	0	$-M_{\dot{\alpha}}S - M_\alpha$	$S^2 - M_q S$	0	0	0		$M_{\delta_e} + M_{\ddot{\delta}_e} S^2$
0	0	$+U_o S - Z_\alpha$	$-U_o S$	0	0	0		$+Z_{\delta_e} + Z_{\ddot{\delta}_e} S^2$
0	0	$-Z_{1_\alpha}$	$-Z_{1_{\dot{\theta}}} S$	$S^2 + 2\zeta_1\omega_1 S + \omega_1^2$	$-Z_{1_{\eta_2}} - Z_{1_{\dot{\eta}_2}} S$	$-Z_{1_{\eta_3}} - Z_{1_{\dot{\eta}_3}} S$		$Z_{1_{\delta_e}} + Z_{1_{\ddot{\delta}_e}} S^2$
0	0	$-Z_{2_\alpha}$	$-Z_{2_{\dot{\theta}}} S$	$-Z_{2_{\eta_1}} - Z_{2_{\dot{\eta}_1}} S$	$S^2 + 2\zeta_2\omega_2 S + \omega_2^2$	$-Z_{2_{\eta_3}} - Z_{2_{\dot{\eta}_3}} S$	=	$Z_{2_{\delta_e}} + Z_{2_{\ddot{\delta}_e}} S^2$
0	0	$-Z_{3_\alpha}$	$-Z_{3_{\dot{\theta}}} S$	$-Z_{3_{\eta_1}} - Z_{3_{\dot{\eta}_1}} S$	$-Z_{3_{\eta_2}} - Z_{3_{\dot{\eta}_2}} S$	$S^2 + 2\zeta_3\omega_3 S + \omega_3^2$		$Z_{3_{\delta_e}} + Z_{3_{\ddot{\delta}_e}} S^2$
-1	0	0	+S	$+\phi'_{1-383} S$	$+\phi'_{2-383} S$	$+\phi'_{3-383} S$		0
0	-1	0	+S	$\phi'_{1-100} S$	$\phi'_{2-100} S$	$\phi'_{3-100} S$		0

Figure 8-2. In-Flight Pitch-Axis Model

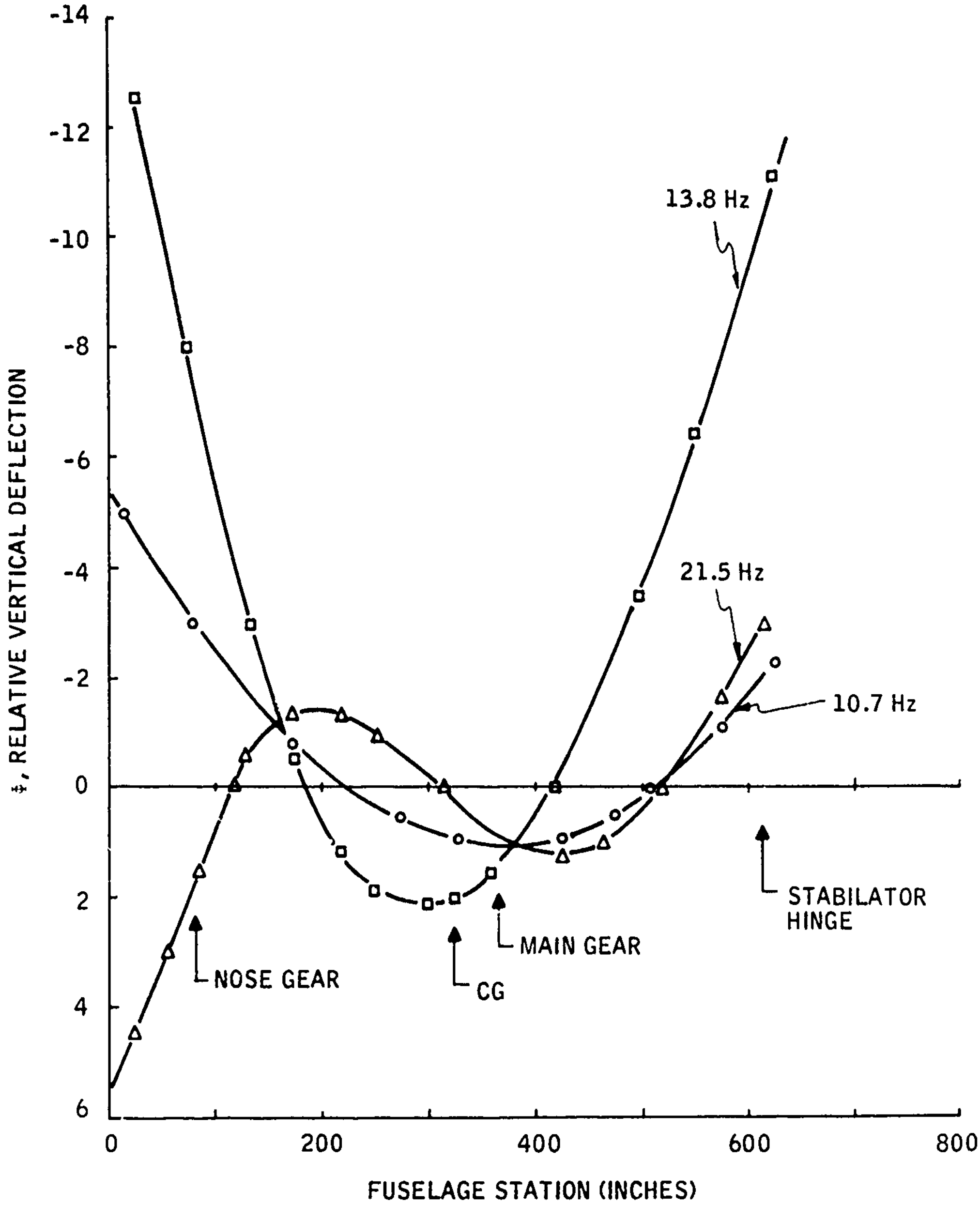


Figure 8-3. Pitch Bending Mode Shapes

Table 8-1. Pitch Model Nomenclature

Quantity	Units	Definition
F_M	lb	Main gear force
F_N	lb	Nose gear force
D_M	lb sec ft^{-1}	Main gear damping coefficient
D_N	lb sec ft^{-1}	Nose gear damping coefficient
K_M	lb ft^{-1}	Main gear spring constant
K_N	lb ft^{-1}	Nose gear spring constant
M_{FM}	lb^{-1} sec^{-2}	Angular acceleration due to main gear deflection
M_{FN}	lb^{-1} sec^{-2}	Angular acceleration due to nose gear deflection
ℓ_M	ft	Distance of main gear aft of cg
ℓ_N	ft	Distance of nose gear forward of cg
m	lb sec^2 ft^{-1}	Aircraft mass
M_α	sec^{-2}	Angular acceleration due to angle of attack
$M_{\dot{\alpha}}$	sec^{-1}	Angular acceleration due to angle-of-attack rate
M_q	sec^{-1}	Angular acceleration due to pitch rate
M_{δ_e}	sec^{-2}	Angular acceleration due to elevator deflection
$M_{\ddot{\delta}_e}$	---	Angular acceleration due to elevator acceleration
U_o	ft sec^{-1}	Nominal aircraft velocity
Z_α	ft sec^{-2}	Acceleration along aircraft z axis due to AOA
Z_{δ_e}	ft sec^{-2}	Acceleration along z axis due to elevator deflection

Table 8-1. Pitch Model Nomenclature (Concluded)

Quantity	Units	Definition
$Z_{\ddot{\delta}_e}$	ft	Acceleration along z axis due to elevator acceleration
$Z_{i_{variable}}$	$ft\text{-}sec^{-2}$/variable units	Acceleration of i^{th} bending mode due to subscript variable
S	sec^{-1}	Operator d/dt
z	ft	Displacement of cg along z axis
z_M	ft	Displacement of main gear
z_N	ft	Displacement of nose gear
α	radians	Angle of attack
δ_e	radians	Elevator deflection
θ	radians	Pitch attitude (rigid body)
$\dot{\theta}_{100}$	$rad\text{-}sec^{-1}$	Pitch attitude rate at station 100
$\dot{\theta}_{383}$	$rad\text{-}sec^{-1}$	Pitch attitude rate at station 383
η_i	ft	Displacement of i^{th} bending mode at station 383
ϕ_{i_M}	---	Relative displacement of i^{th} mode at main gear
ϕ_{i_N}	---	Relative displacement of i^{th} mode at nose gear
ω_i	sec^{-1}	Natural frequency of i^{th} bending mode
ω_{TWD}	sec^{-1}	Tail-wags-dog frequency
ζ_i	---	Damping ratio of i^{th} bending mode
ϕ'_{i-100}	ft^{-1}	Slope of i^{th} mode at station 100
ϕ'_{i-383}	ft^{-1}	Slope of i^{th} mode at station 383

The model of Figure 8-2 is simply a conventional short-period aerodynamic representation with the first three symmetric bending modes added. The inertial forces due to elevator mass are included as described in Section V. The matrix of Figure 8-2 shows aerodynamic-model coupling terms $Z_{i\alpha}$, $Z_{i\theta}$, Z_{in_j}, and $Z_{i\dot{n}_j}$. These were evaluated for the low-q conditions using data from Reference 8-4 and found to be completely negligible, as might be expected. The remainder of the data used for the study are tabulated in Table 8-2. These correspond to F-4 characteristics at the sea-level Mach 0.206 condition.

Pitch-rate frequency responses to elevator deflections for both gyro locations are shown in Figures 8-4 through 8-7. It is these responses which must be simulated or otherwise duplicated during ground tests.

On-Ground Pitch-Axis Model

The airframe properties which are significant during ground testing include the inertial qualities of the actuation system, the inertial and flexure properties, and the means of support. The first two are already included in the in-flight pitch model of Figure 8-2, which is directly applicable after eliminating the aerodynamic effects. The support means considered in this study consist of the basic landing gear and associated modifications. The latter include variation in the damping and spring constants of the shock struts and a presumed shift in gear locations to a hypothetical union at the cg. Although the latter is a physical impossibility, suspension of the airframe by a spring cable (as is done in some cases for ground shake testing) is not, and the math models used herein are applicable to this case.

The resulting system model is given by the matrix of Figure 8-8. Associated terms are defined by Table 8-1. It may be noted in this model that each landing gear effect is represented by a simple spring-and-damper combination which applies forces according to displacement and velocity at the gear attachment point. A rigid member is assumed between the two main gears such that each experiences equal deflections which also equal that at the associated fuselage centerline station.

The landing gear models ignore tire flexibility, which in terms of shock-strut spring constants derived from Reference 8-5 data appears to be at least twice as stiff as the basic strut for perturbations around the nominal load position. The presence of tire flexibility, plus wheel and lower-strut mass, contribute added dynamics which are estimated to be significant for frequencies above 27 Hz, well beyond interest.

The data unique to the on-ground airframe model is presented in Table 8-3. The nominal shock-strut spring constants represent linearized values taken from maintenance calibration data of Reference 8-5. They result in reasonable frequencies of 6.2 rad/sec and 12 rad/sec for the rigid pitch

Table 8-2. In-Flight Model Data

Quantity	Value	Quantity	Value
m	1212 slugs	$Z_{2_{\ddot{\delta}_e}}$	0.00215 ft
M_α	$-0.45\ \text{sec}^{-2}$	$Z_{3_{\ddot{\delta}_e}}$	0.00376 ft
$M_{\dot{\alpha}}$	$-0.137\ \text{sec}^{-1}$	ω_1	67 rad/sec
M_q	$-0.45\ \text{sec}^{-1}$	ω_2	86.7 rad/sec
M_{δ_e}	$-2.93\ \text{sec}^{-2}$	ω_3	135.2 rad/sec
$M_{\ddot{\delta}_e}$	-0.0012	ζ_1	0.025
U_o	230 ft/sec	ζ_2	0.025
Z_α	$-89\ \text{ft/sec}^2$	ζ_3	0.025
Z_{δ_e}	$-15.2\ \text{ft/sec}^2$	$\phi'_{1\text{-}100}$	0.312 rad/ft
$Z_{\ddot{\delta}_e}$	-0.00518 ft	$\phi'_{2\text{-}100}$	0.913 rad/ft
$Z_{1_{\delta_e}}$	$43.3\ \text{ft/sec}^2$	$\phi'_{3\text{-}100}$	-0.576 rad/ft
$Z_{2_{\delta_e}}$	$6.3\ \text{ft/sec}^2$	$\phi'_{1\text{-}383}$	0
$Z_{3_{\delta_e}}$	$11.0\ \text{ft/sec}^2$	$\phi'_{2\text{-}383}$	-0.279 rad/ft
$Z_{1_{\ddot{\delta}_e}}$	0.0147 ft	$\phi'_{3\text{-}383}$	0.095 rad/ft

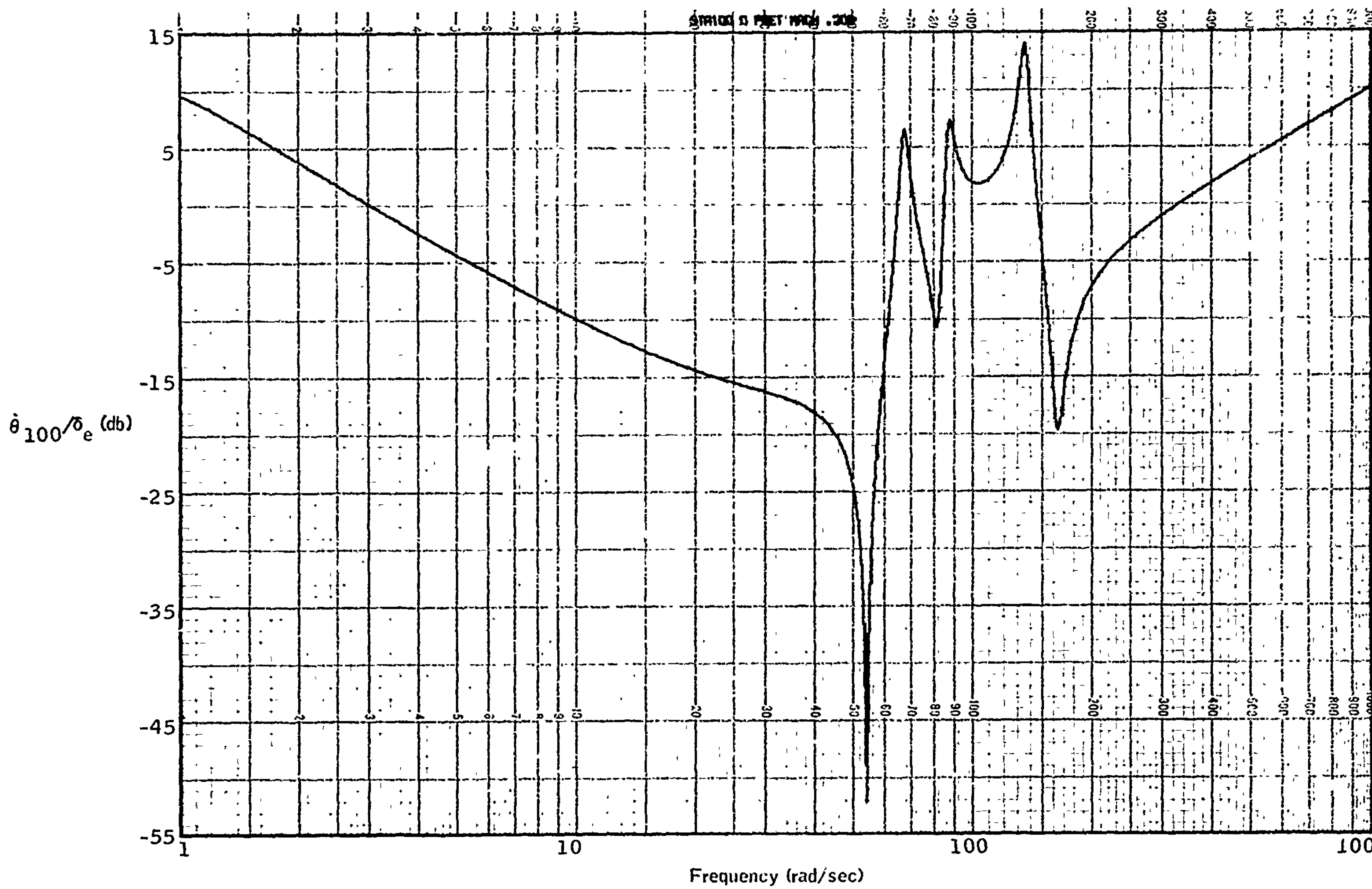


Figure 8-4. Nominal In-Flight Amplitude Response at Station 100

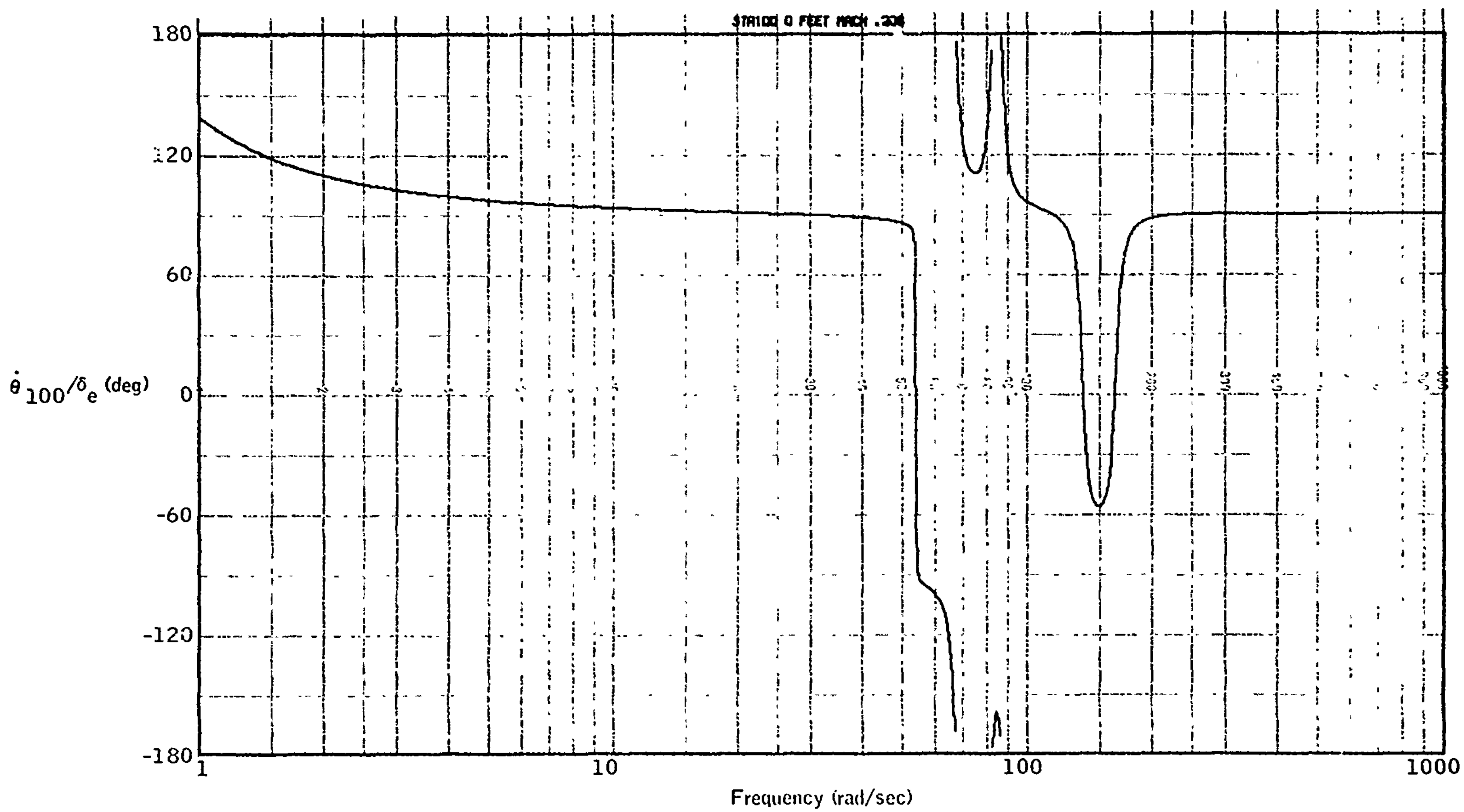


Figure 8-5. Nominal In-Flight Phase Response at Station 100

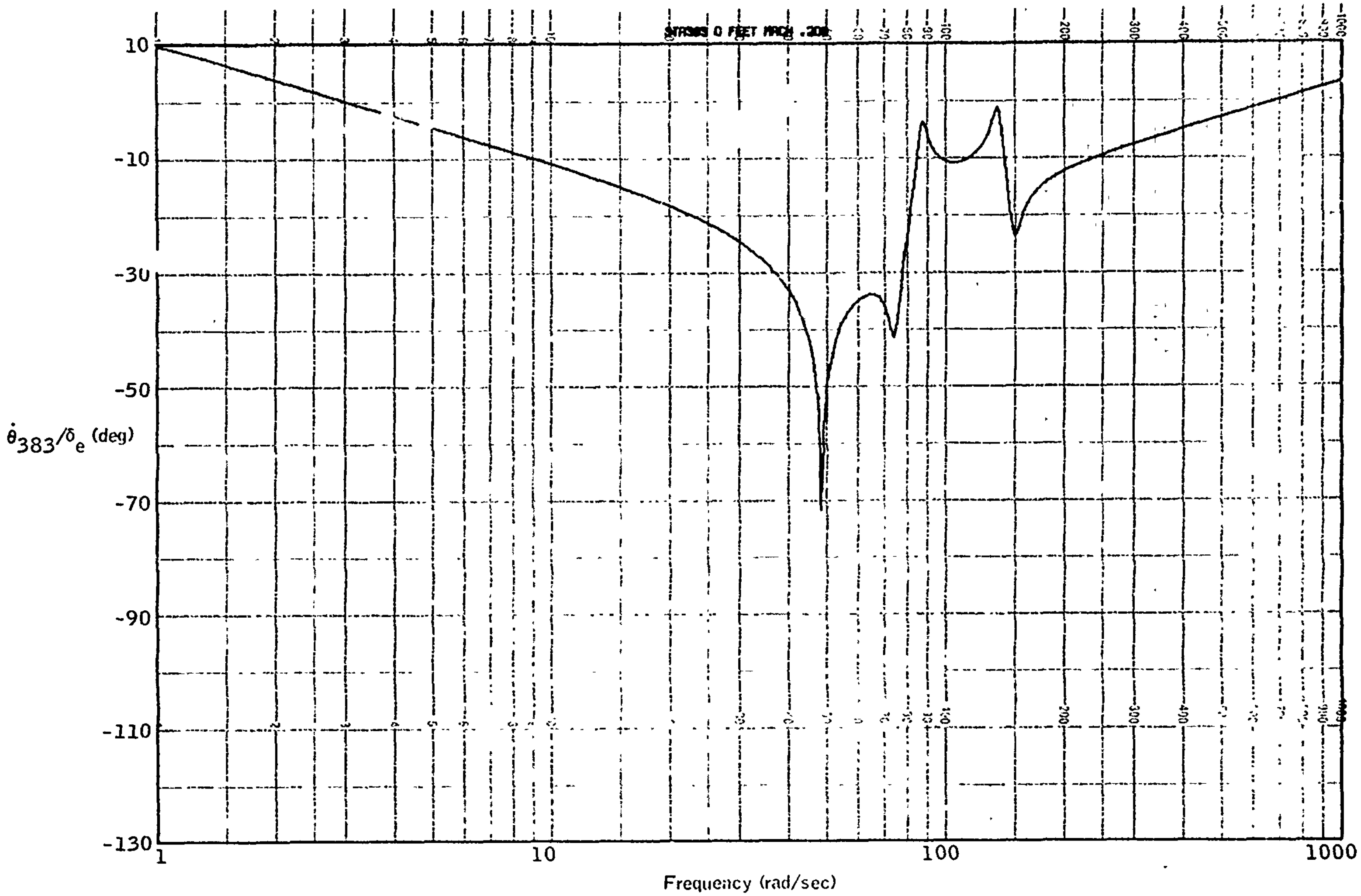


Figure 8-6. Nominal In-Flight Amplitude Response at Station 383

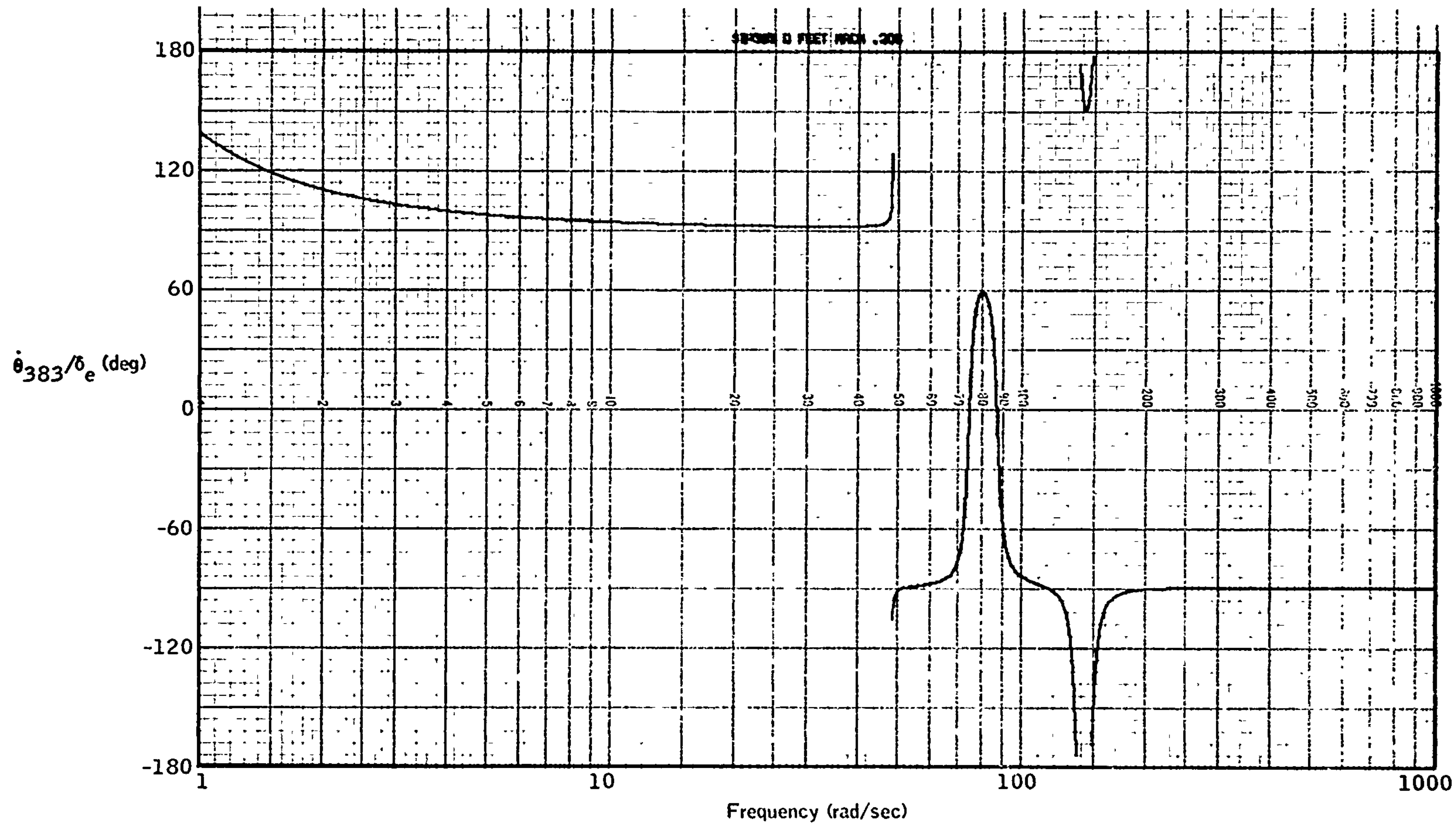


Figure 8-7. Nominal In-Flight Phase Response at Station 383

$\dot{}_{100}$	$\dot{}_{383}$	θ	z	z_M	z_N	F_M	F_N	$'_1$	$'_2$	$'_3$		δ_c
0	0	$+S^2$	0	0	0	$-M_{FM}$	$-M_{FN}$	0	0	0		$M_{\ddot{\delta}_e} S^2$
0	0	0	S^2	0	0	$-1/m$	$-1/m$	0	0	0		$Z_{\ddot{\delta}_e} S^2$
0	0	0	0	$K_M + D_M S$	0	+1	0	0	0	0		0
0	0	0	0	0	$K_N + D_N S$	0	1	0	0	0		0
0	0	$'_M$	1	-1	0	0	0	ϕ_{1M}	ϕ_{2M}	ϕ_{3M}		0
0	0	$-'_N$	1	0	-1	0	0	ϕ_{1N}	ϕ_{2N}	ϕ_{3N}		0
0	0	0	0	0	0	$-Z_{1_{FM}}$	$-Z_{1_{FN}}$	$S^2 + 2\zeta_1\omega_1 S + \omega_1^2$	0	0		$Z_{1_{\ddot{\delta}_e}} S^2$
0	0	0	0	0	0	$-Z_{2_{FM}}$	$-Z_{2_{FN}}$	0	$S^2 + 2\zeta_2\omega_2 S + \omega_2^2$	0		$Z_{2_{\ddot{\delta}_e}} S^2$
0	0	0	0	0	0	$-Z_{3_{FM}}$	$-Z_{3_{FN}}$	0	0	$S^2 + 2\zeta_3\omega_3 S + \omega_3^2$		$Z_{3_{\ddot{\delta}_e}} S^2$
0	-1	S	0	0	0	0	0	$\phi'_{1\text{-}383} S$	$\phi'_{2\text{-}383} S$	$\phi'_{3\text{-}383} S$		0
-1	0	S	0	0	0	0	0	$\phi'_{1\text{-}100} S$	$\phi'_{2\text{-}100} S$	$\phi'_{3\text{-}100} S$		0

Figure 8-8. On-Ground Elevations

Table 8-3. On-Ground Airframe Properties

Quantity	Nominal Support	Soft Support	Nominal Support at cg	Soft Support at cg	Low-Damped Support	Low-Damped Support at cg
K_M ($lb\text{-}ft^{-1}$)	168, 000	42, 000	168, 000	42, 000	168, 000	168, 000
K_N ($lb\text{-}ft^{-1}$)	7200	1800	7200	1800	7200	7200
D_M ($lb\text{-}sec\text{-}ft^{-1}$)	13, 000	6500	13, 000	6500	1300	1300
D_N ($lb\text{-}sec\text{-}ft^{-1}$)	1600	800	1600	800	160	160
ℓ_M (ft)	3. 5	3. 5	0	0	3. 5	0
ℓ_N (ft)	20	20	0	0	20	0
M_{FM} ($lb^{-1}\text{-}sec^{-2}$)	2.76×10^{-5}	2.76×10^{-5}	0	0	2.76×10^{-5}	0
M_{FN} ($lb^{-1}\text{-}sec^{-2}$)	-1.58×10^{-4}	-1.58×10^{-4}	0	0	-1.58×10^{-4}	0
Z_{1FM} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	1.17×10^{-3}	1.17×10^{-3}	1.03×10^{-3}	1.03×10^{-3}	1.17×10^{-3}	1.03×10^{-3}
Z_{2FM} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	4.64×10^{-5}	4.64×10^{-5}	6.64×10^{-5}	6.64×10^{-5}	4.64×10^{-5}	6.64×10^{-5}
Z_{3FM} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	1.72×10^{-4}	1.72×10^{-4}	3.03×10^{-5}	3.03×10^{-5}	1.72×10^{-4}	3.03×10^{-5}
Z_{1FN} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	-3.52×10^{-3}	-3.52×10^{-3}	1.03×10^{-3}	1.03×10^{-3}	-3.52×10^{-3}	1.03×10^{-3}
Z_{2FN} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	-2.42×10^{-4}	-2.42×10^{-4}	6.64×10^{-5}	6.64×10^{-5}	-2.42×10^{-4}	6.64×10^{-5}
Z_{3FN} ($ft\text{-}sec^{-2}\text{-}lb^{-1}$)	3.65×10^{-4}	3.65×10^{-4}	3.03×10^{-5}	3.03×10^{-5}	3.65×10^{-4}	3.03×10^{-5}
ϕ_{1M}	1. 0	1. 0	0. 88	0. 88	1. 0	0. 88
ϕ_{2M}	1. 4	1. 4	2. 0	2. 0	1. 4	2. 0
ϕ_{3M}	0. 85	0. 85	0. 15	0. 15	0. 85	0. 15
ϕ_{1N}	-3. 0	-3. 0	0. 88	0. 88	-3. 0	0. 88
ϕ_{2N}	-7. 3	-7. 3	2. 0	2. 0	-7. 3	2. 0
ϕ_{3N}	1. 8	1. 8	0. 15	0. 15	1. 8	0. 15

and plunge modes, respectively. Damping properties of the struts were unknown, so nominal constants as required to produce 0.5 damping ratios for pitch and plunge were selected. Table 8-3 also defines values associated with the support variations studied.

Comparison of Structural Response on Ground with in Flight

Effect of Elevator Aerodynamics -- The elevator mass imbalance relative to the aerodynamic forces generated by surface deflection is such that the inertial forces dominate over the bending frequency range for the low-q conditions. This situation creates a "tail-wag-dog" zero at about 50 rad/sec for the studied condition, as illustrated in the amplitude responses of Figures 8-4 through 8-7. Although the inertial forces dominate, the aerodynamic effect is still significant, at least for the first bending mode. This situation is illustrated by the amplitude responses of Figures 8-9 and 8-10 which compare the pitch-rate responses in flight with and without elevator aerodynamics. The cases without elevator aerodynamics are comparable (over the bending frequencies) to a ground test where no support constraints exist. It is evident from the 10-db gain error at the first bending mode that ground tests for structural coupling must include correction for lack of aerodynamic elevator forces. The nature of this correction term is:

$$G_{TWD} = \frac{M_{\delta_e} + M_{\ddot{\delta}_e} S^2}{M_{\ddot{\delta}_e} S^2} \qquad (8.1)$$

where G_{TWD} is a transfer function to be applied in series with the elevator command signal.[1] Simulation of G_{TWD} in its current form poses obvious drift problems due to the double integration involved, so a more tractable form would be

$$\overline{G_{TWD}} = \left[\frac{M_{\delta_e} + M_{\ddot{\delta}_e} S^2}{M_{\ddot{\delta}_e} S^2}\right]\left[\frac{S}{S + \frac{\omega_1}{10}}\right]^2 = \frac{\left(M_{\delta_e} + M_{\ddot{\delta}_e} S^2\right)}{M_{\ddot{\delta}_e}\left(S + \frac{\omega_1}{10}\right)^2} \qquad (8.2)$$

Note that the added "double high-pass" filter has a break point selected at one decade below the first-mode frequency (ω_1) to avoid significant error over the bending frequency range. Selection of this filter poses some compromise between accurate bending-mode reproduction and avoidance of

[1] Recall from Section V that the quantity $\sqrt{M_{\delta_e}/M_{\ddot{\delta}_e}}$ has been defined as ω_{TWD}, the "tail-wags-dog" frequency.

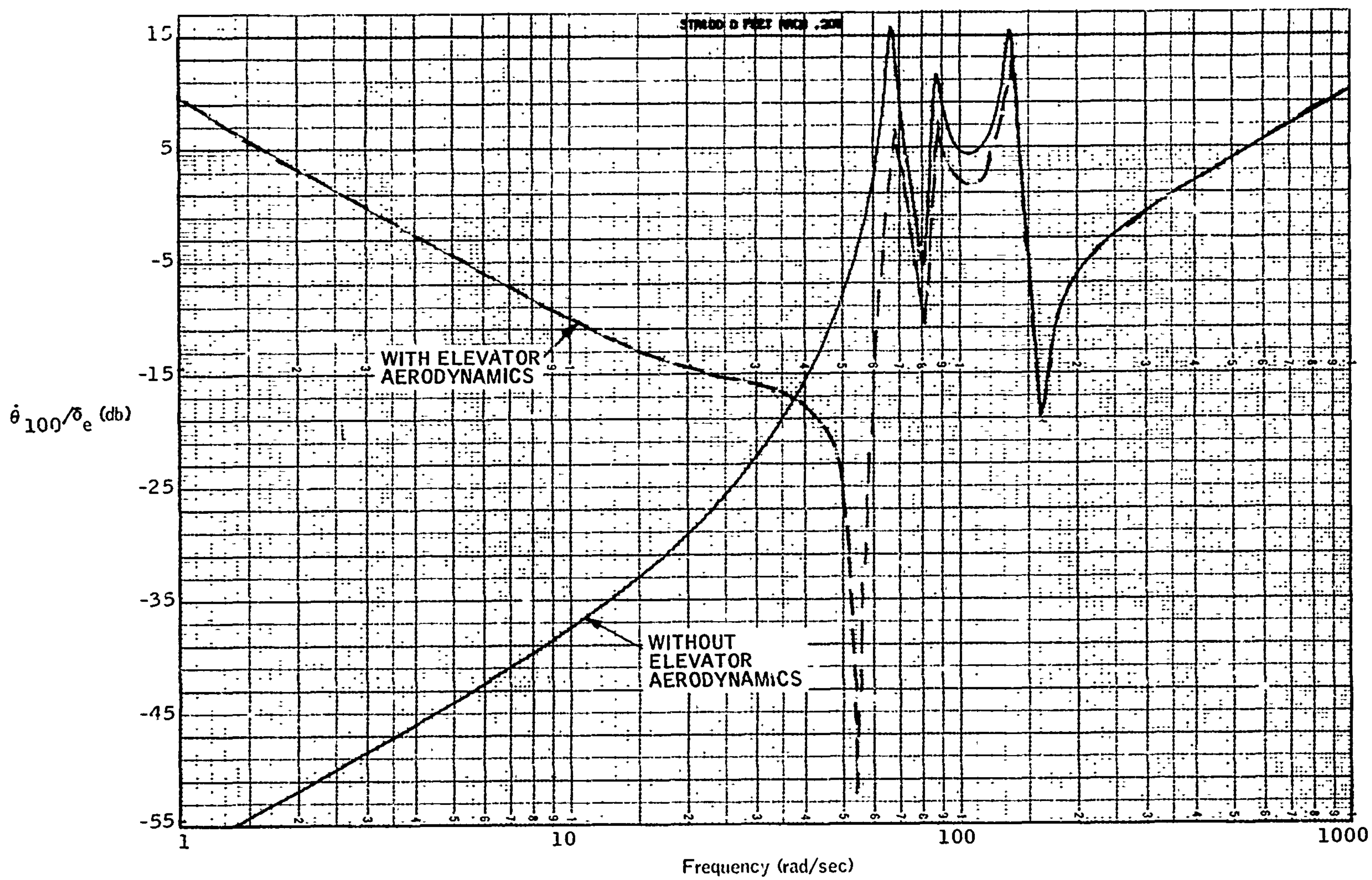


Figure 8-9. Elevator Aerodynamic Effect in Flight at Station 100

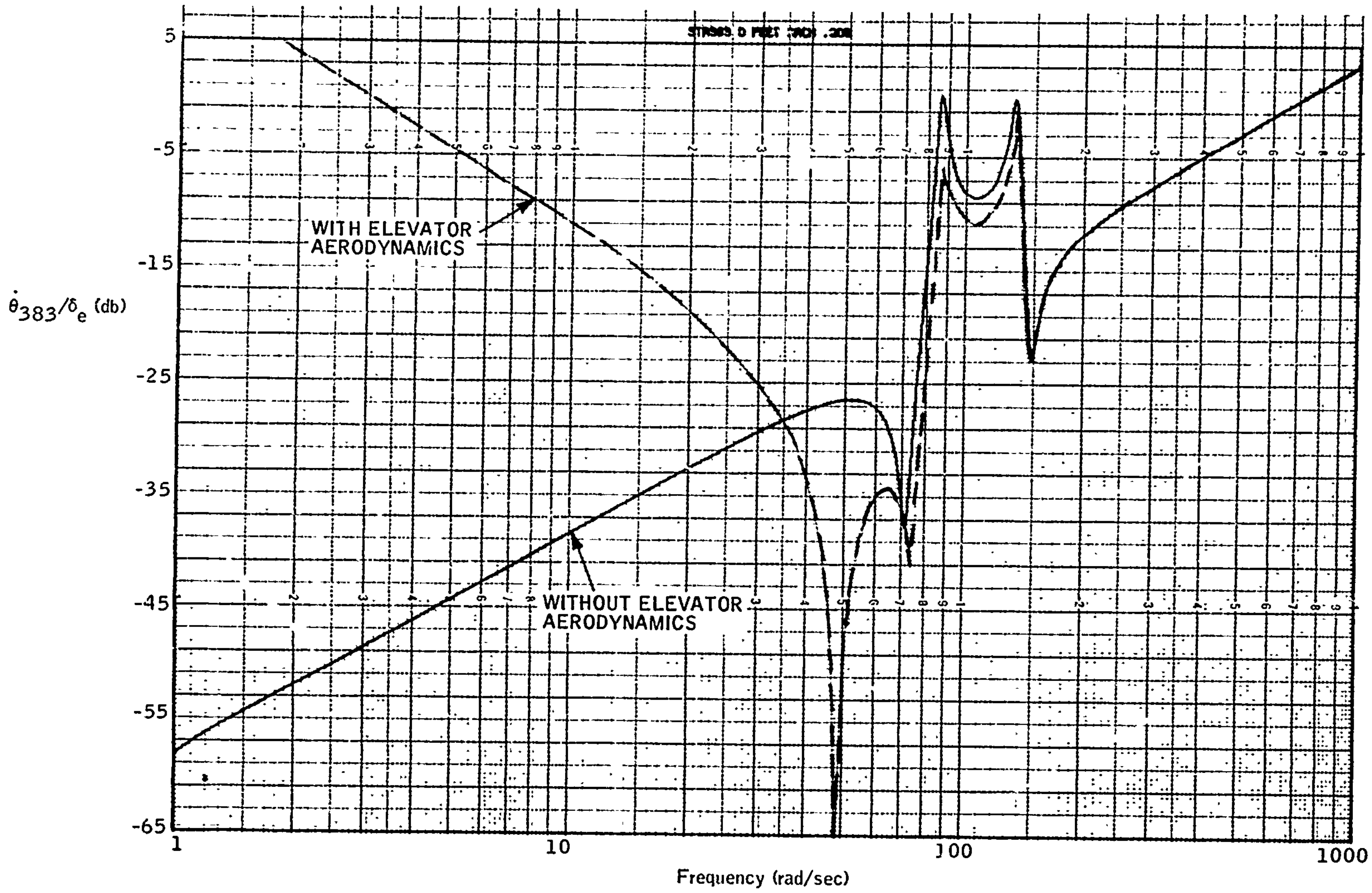


Figure 8-10. Elevator Aerodynamic Effect in Flight at Station 383

excessive low-frequency amplification, the latter potentially creating feedback from the rigid pitch and plunge modes of the grounded airframe.

Another interesting aspect is the question of how to cope with the case where the surface inertial forces are comparatively negligible (i.e., the tail-wags-dog frequency is higher than the bending frequencies). Conceivably, mass could be added to the surface to create an artifically high $M_{\ddot{\delta}_e}$. If this artificial quantity were denoted as $M'_{\ddot{\delta}_e}$, the correction function of (8.2) would be:

$$G'_{TWD} = \frac{M_{\delta_e} + M_{\ddot{\delta}_e} S^2}{M'_{\ddot{\delta}_e} \left(S + \frac{\omega_1}{10}\right)^2} \tag{8.3}$$

The practicality of this artifice is questionable, however, in that mass added to the surface may create unacceptable secondary effects, such as stability problems within the actuator loop itself due to compliance.

<u>Effect of Airframe Support</u> -- The manner in which the airframe is supported during ground tests has a manifold influence on the validity of the structural coupling tests. Avoidance of support influence has led to special support means for structural shake tests, and comparable efforts appear necessary for conduct of meaningful closed-loop compatibility tests.

Of interest to the study is the degree by which conventional landing-gear support can influence structural response. Having determined this, modifications were made in the gear properties and locations to determine the potential for alternate support with superior test qualities. The set of support cases considered is defined by Table 8-3. These may be considered in two categories:

- Support involving the basic landing gear but with alternate spring rates and damping coefficients;

- Support concentrated at the cg (as with a cable) with alternate spring rates and damping.

One means of determining support effects is by the associated roots of the characteristic equation of the structure. This method has the advantage that the influence of elevator aerodynamics (as discussed previously) is excluded. It has the disadvantage that modes with little effect on the control system (like the first bending mode for the gyro at station 383) are not discounted.

Table 8-4 lists the undamped natural frequencies and damping ratios for the studied cases. The following conclusions may be drawn from the table:

Table 8-4. Support Effects on Bending Poles

Case	ω_1 (rad/sec)	ζ_1	ω_2 (rad/sec)	ζ_2	ω_3 (rad/sec)	ζ_3
In flight low-q	67	0.025	87	0.025	135	0.025
Nominal support	68	0.28	86	0.037	135	0.036
Soft support	67	0.15	86	0.034	135	0.030
Nominal support at cg	67	0.13	86	0.035	135	0.025
Soft support at cg	67	0.075	87	0.031	135	0.025
Low-damped support	69	0.048	87	0.028	135	0.026
Low-damped support at cg	68	0.035	87	0.026	135	0.025

- Modal frequencies are not changed significantly by any support case;
- Modal damping is increased by all cases, particularly the first mode;
- If the first mode were not of concern (as for the station 383 gyro location), the nominal landing gear support would probably be satisfactory;
- A concentrated support at the cg is advantageous, probably to avoid nose gear damping;
- Shock-strut damping, rather than spring constant, is the primary influence, suggesting use of undamped spring supports.

Comparisons among the various support means are also made by the frequency-response plots of Figures 8-11 through 8-14. Also shown on each curve is the low-q in-flight response without elevator aerodynamics, which, in effect, represents no support constraints and is, therefore, the ideal case. Of concern are the responses over the bending range, the low-frequency properties being of relatively low magnitude compared to aerodynamic effects, as illustrated by Figures 8-9 and 8-10. It is evident from Figures 8-11 and 8-12 that, with the conventional landing gear, significant error prevails at the first-bending-mode frequency (6 db), even for the best gear properties studied (low damping). This agrees with the data of Table 8-3. From Figures 8-13 and 8-14 the superior response offered by cg support is evident, the maximum gain error being 3 db for the low-damping case.

The results suggest that an ultimate support arrangement might consist of the main landing gear with reduced shock strut damping (say 10 percent of nominal as used for the "low-damped" cases) plus a temporary spring support somewhat aft of the nose gear around station 200 (see Figure 8-3) near the first- and second-mode nodes. Alternatively, the support used for the ground-shake testing might be considered.

Having noted that proper support with low damping offers reasonably good structural response characteristics, the effectiveness of the compensation for lack of elevator aerodynamics proposed earlier (Eq. 8.2) in conjunction with proper support is of interest. This is determined by comparing the ground response using the compensation with the inflight response with normal aerodynamics, as shown in Figures 8-15 and 8-16. An excellent correspondence over the bending frequency range is evident. Also evident is an undesirable effect -- the amplification of rigid-body ground modes (plunge in the case of Figures 8-15 and 8-16) by the correction term G_{TWD}. So long as the amplified response is below inflight values (as with Figures 8-15 and 8-16), no stability problems should occur. It is evident, however, that the associated testing is applicable only to the bending modes, and that no results can be drawn from lower-frequency phenomena.

It may be conjectured that some alleviation of the effects of rigid-body motions on the ground is possible, either by reducing the amplification by altering the compensation filter G_{TWD} or by lowering the plunge frequency via a softer spring support of the grounded airframe. More will be said about this in the following paragraphs.

TOTAL CLOSED-LOOP TESTING

Preceding subsections described individual means of performing closed-loop testing to determine compatibility of both rigid- and flexible-airframe properties with the feedback control system. Each test involves closing the loop through the actual control electronics and the actuation system. For rigid-body testing, an analog computer adds the in-flight rigid aerodynamic

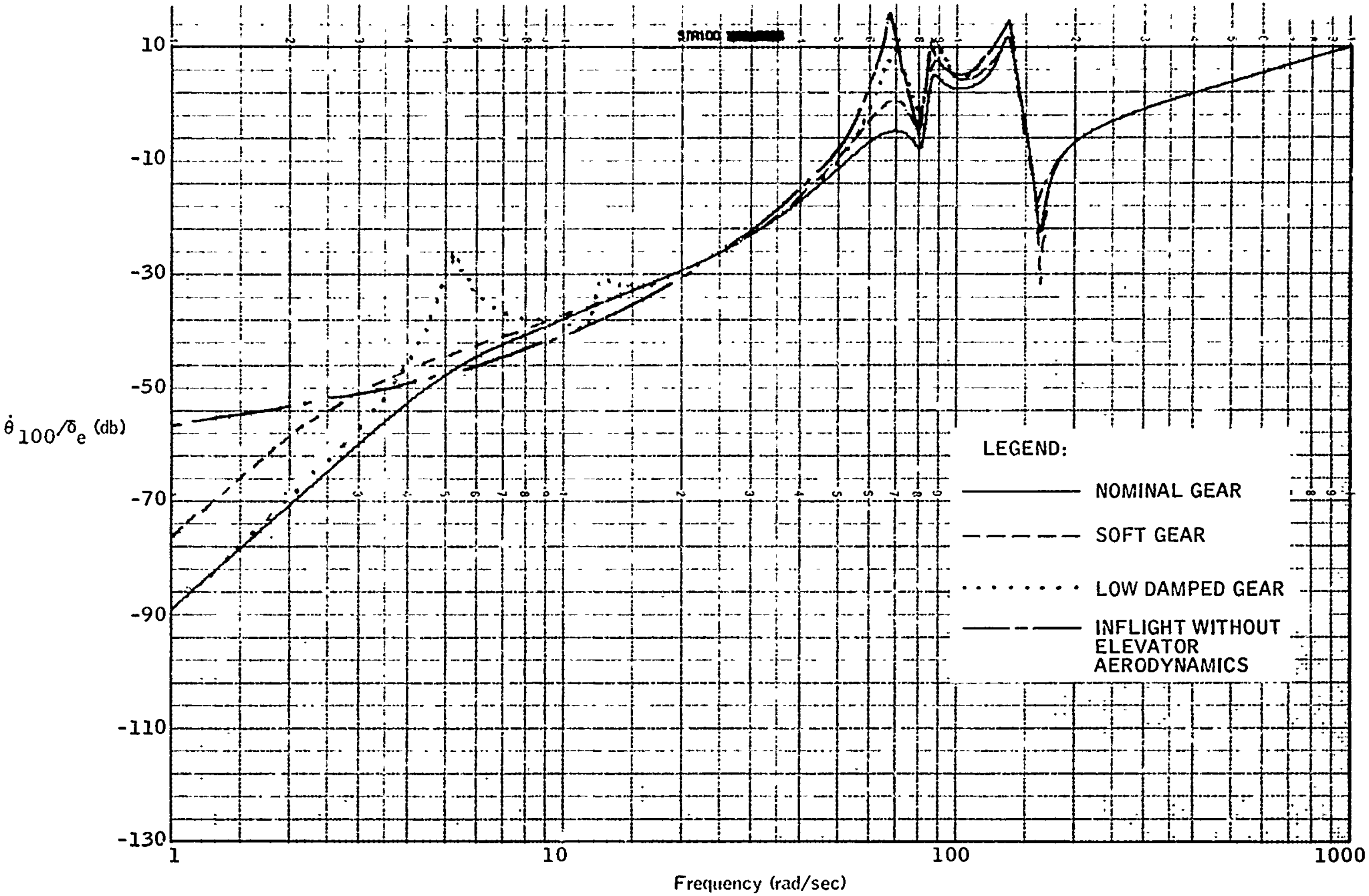


Figure 8-11. Landing Gear Effects on Amplitude Response at Station 100

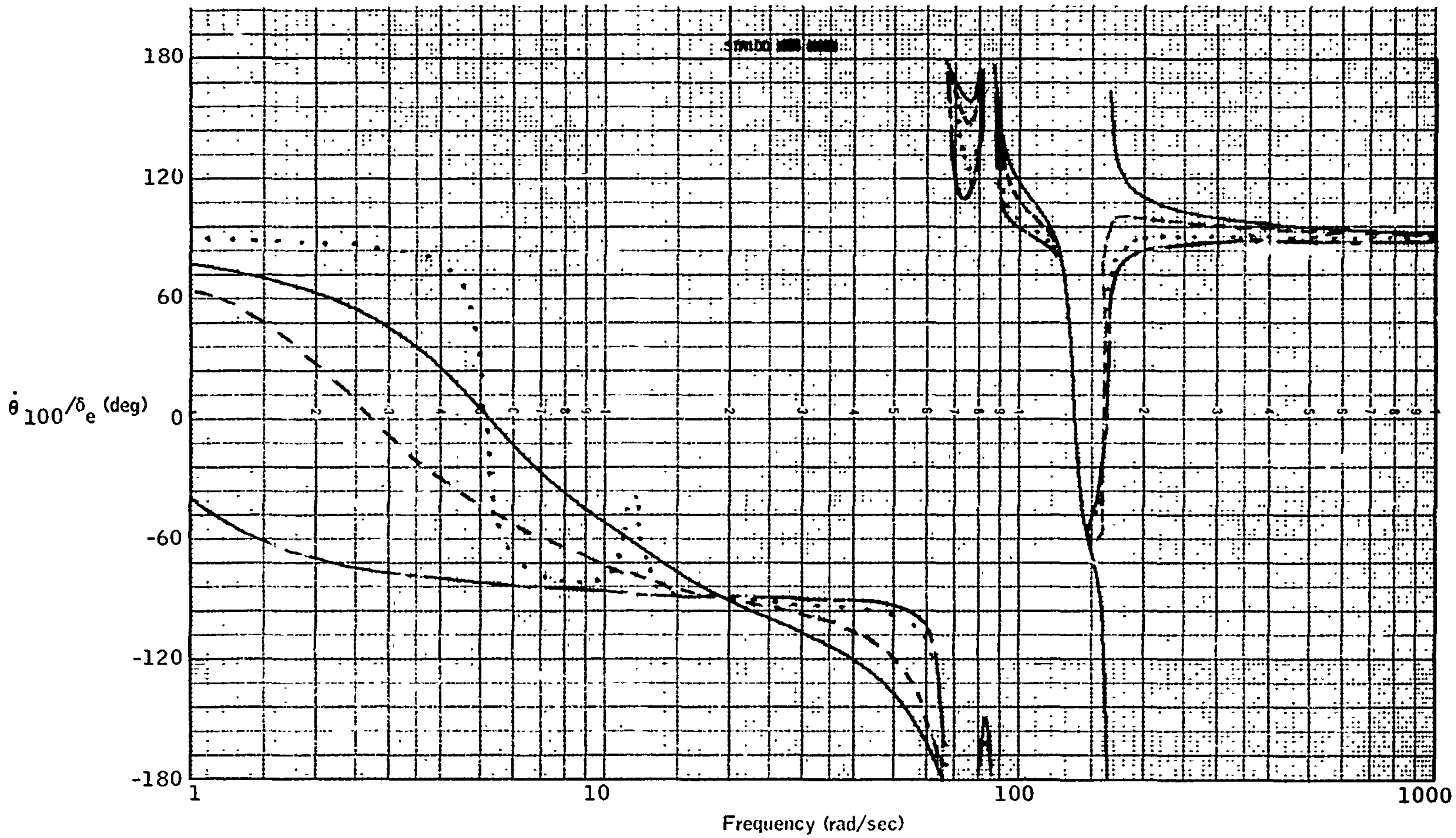


Figure 8-12. Landing Gear Effects on Phase Response at Station 100

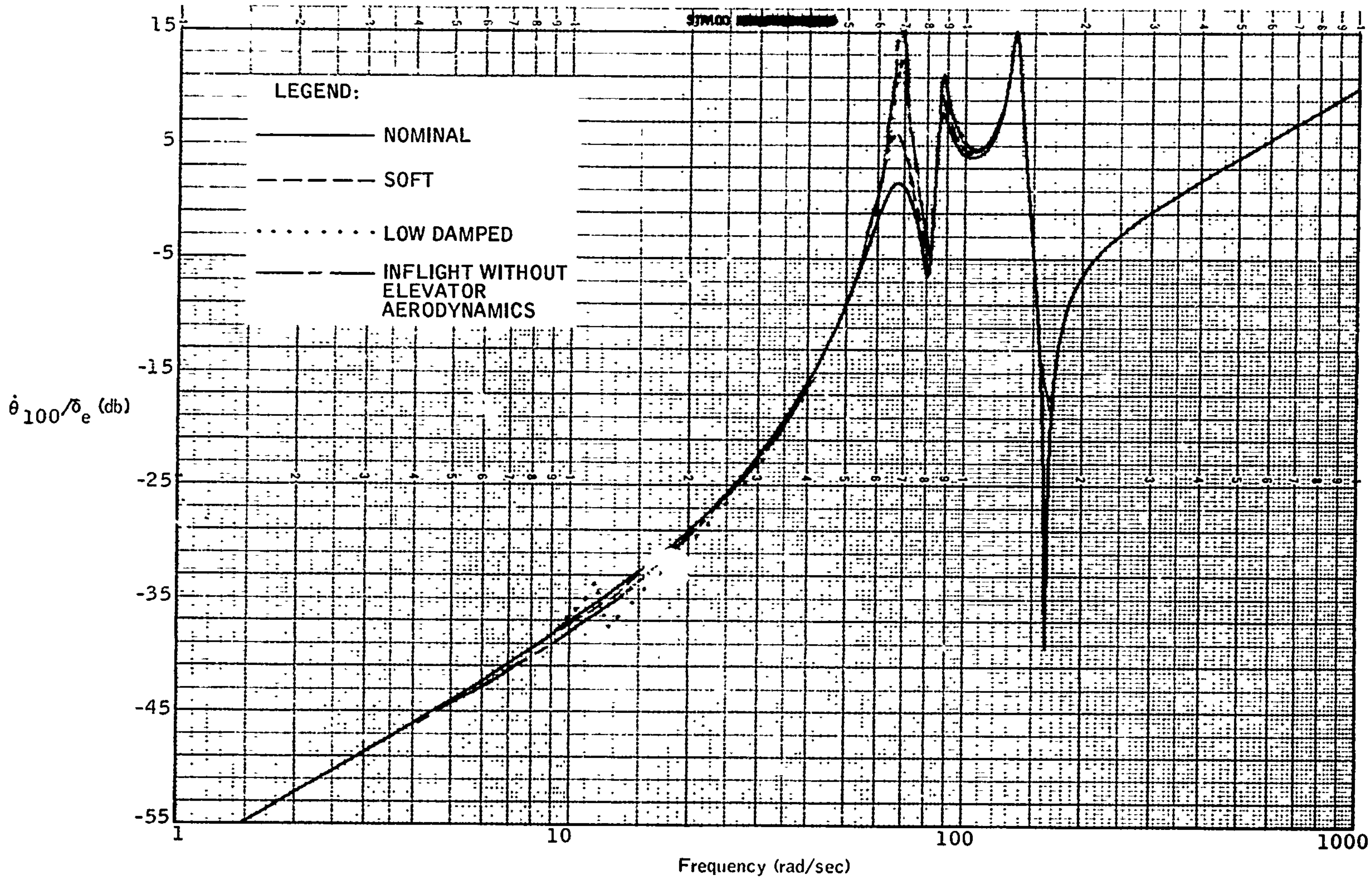


Figure 8-13. CG Support Effects on Amplitude Response at Station 100

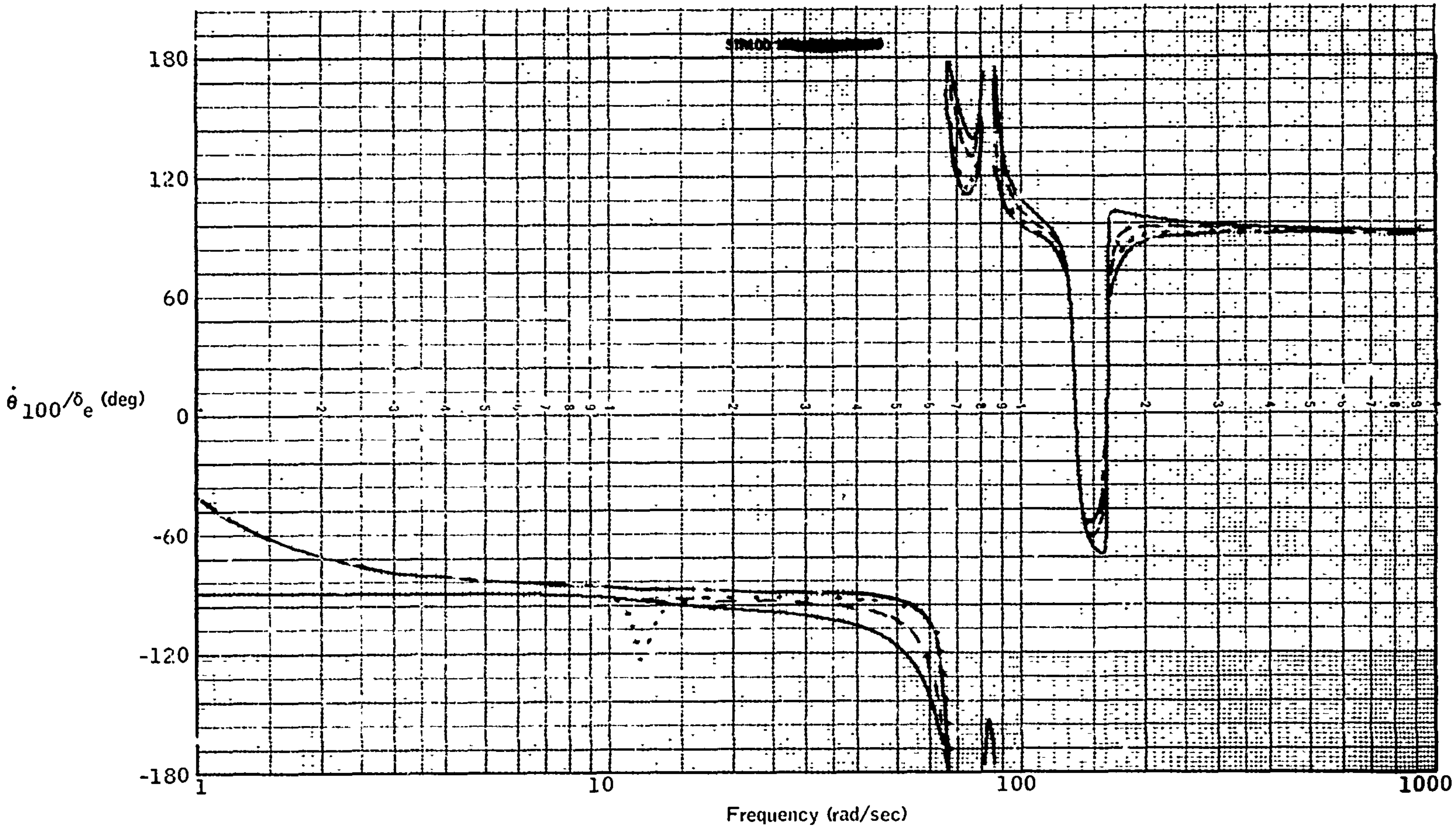


Figure 8-14. CG Support Effects on Phase Response at Station 100

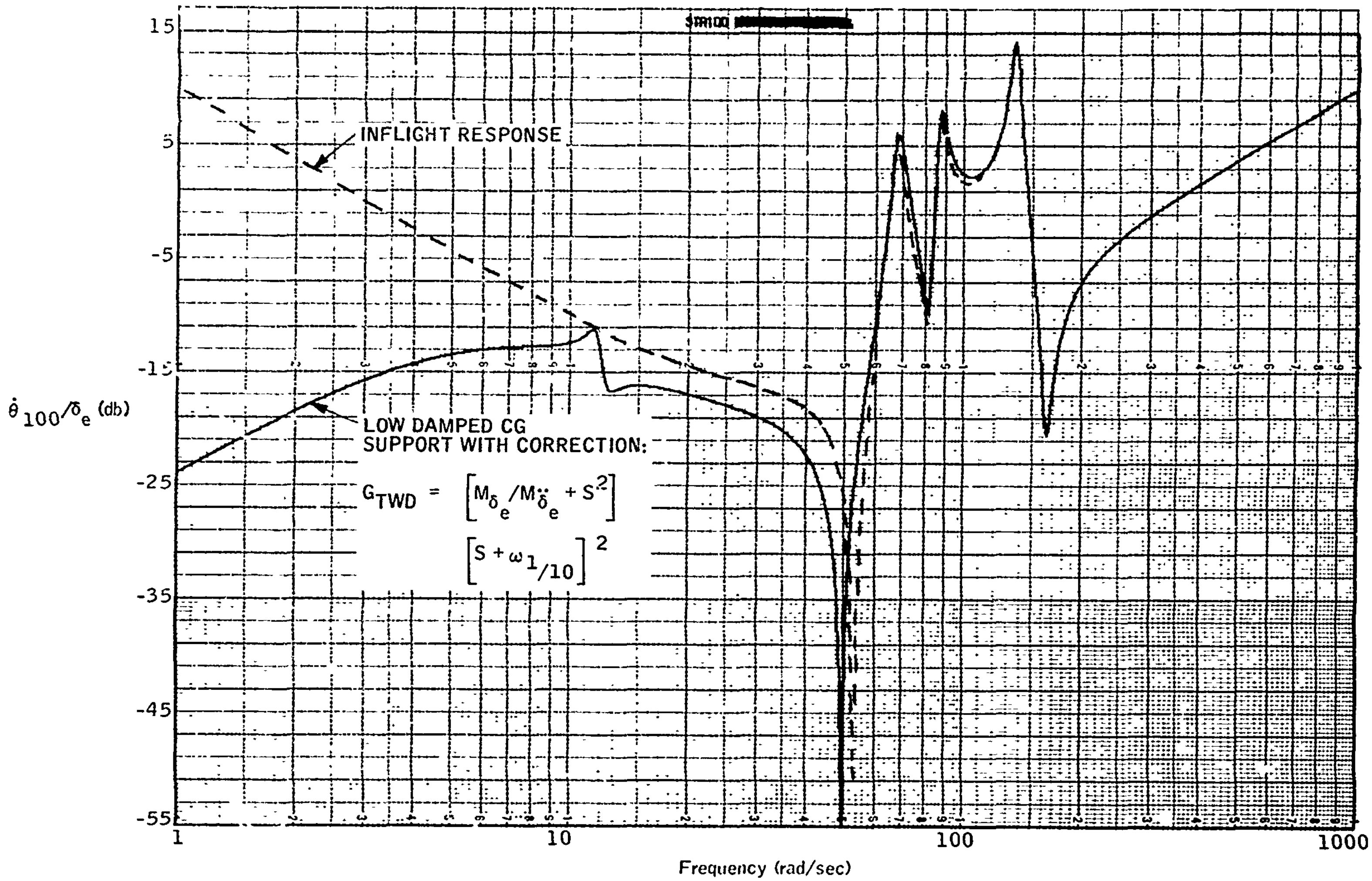


Figure 8-15. *Amplitude Response Comparison, Compensated CG Support versus in Flight*

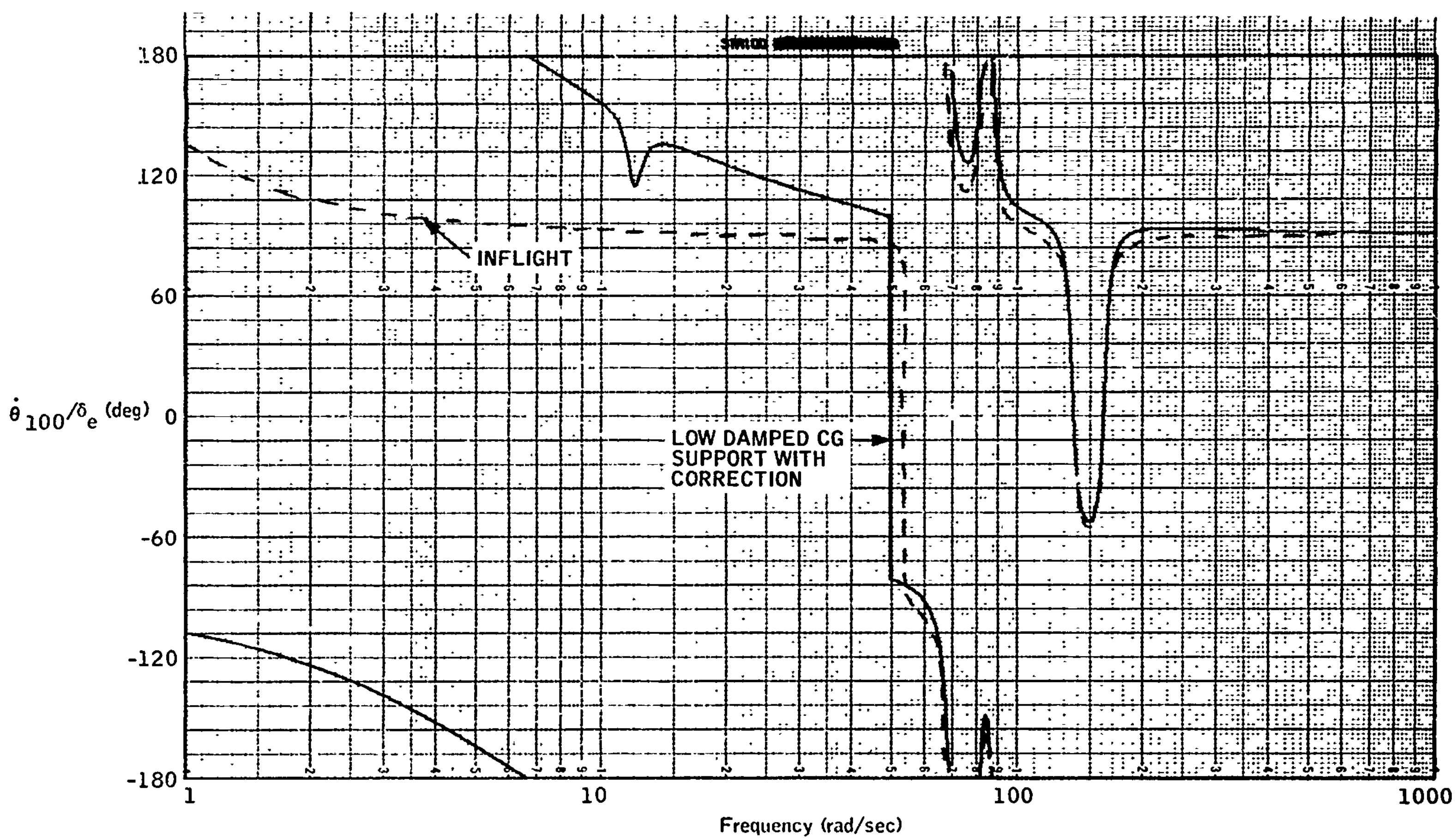


Figure 8-16. Phase Response Comparison, Compensated CG Support versus in Flight

and inertial properties to compute equivalent sensor signals, the actual sensors being bypassed. For structural stability tests, the airframe (with proper support) supplies its own flexure properties which are detected by the actual sensors. These feed the control electronics which drive the actuator system in the conventional manner. The only artificial element in this loop is an added compensation term to correct for missing surface aerodynamics.

These control loops are illustrated in Figure 8-17, which also suggests a potential unified test scheme to accomplish both rigid and flexure testing simultaneously. It might be expected that such a total test scheme would be capable of exercising the lower- and higher-frequency modes with fidelity, but that the middle frequencies would be compromised where the upper and lower ends overlap. Such proved to be the case for the conditions considered in this study, and an entirely satisfactory total simulation was not achieved. The problem involved the previously mentioned rigid-body modes of the grounded vehicle which were amplified by the compensation for surface aerodynamics, as illustrated in Figures 8-15 and 8-16. An adjustment was made in the compensation break frequency ($n = 5$ in Figure 8-17) to reduce the low-frequency gain, and the total response (simulated aerodynamics plus actual structure) for station 100 were computed as shown in Figures 8-18 and 8-19. Also plotted is the in-flight response for comparison. Whereas the amplitude match is satisfactory, the phase error around 20 rad/sec is 27 degrees, an unacceptable amount considering that the nominal loop crossover is often around this frequency.

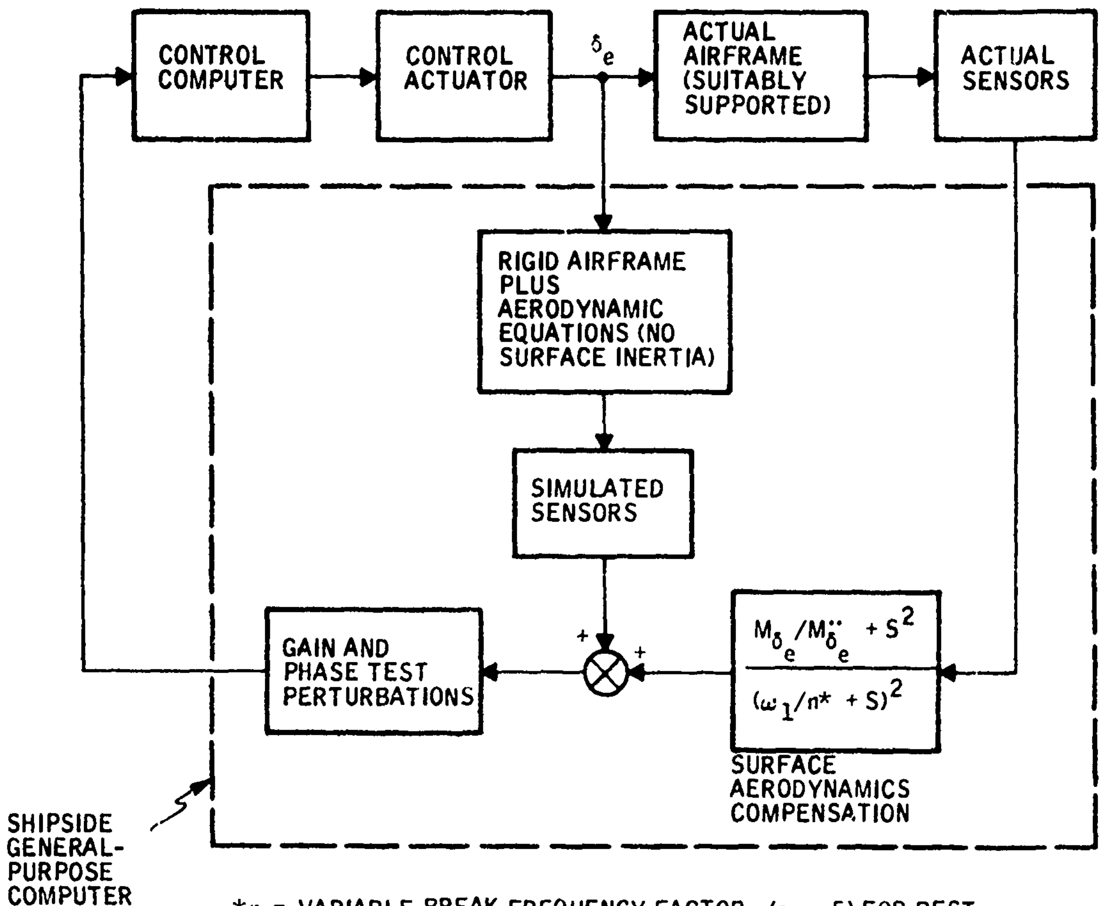
Further adjustments in compensation parameters or in airframe support might improve the simulation validity around the center frequencies. Until this is achieved, however, separate rigid and flexible testing appears advisable.

SUMMARY AND CONCLUSIONS

This study of system/airframe compatibility testing has concentrated on closed-loop simulation procedures involving to a maximum extent the actual flight hardware. This form of testing is viewed as the last step prior to actual flight. As such it is considered supplementary to rather than a replacement for the usual formal testing routinely performed on aircraft equipment, such as qualification and reliability demonstration.

The following conclusions were drawn from the investigation:

- Closed-loop testing using simulated airframe properties at various flight conditions with the actual actuation system and flight computer should be performed to verify stability margins, response to commands and disturbances, and limit-cycle acceptability. The latter should be evaluated in terms of MIL-F-9490C (USAF) criteria by computing related airframe variables.



*n = VARIABLE BREAK FREQUENCY FACTOR (e.g., 5) FOR BEST RIGID AND FLEXIBLE COMPROMISE

Figure 8-17. Total Closed-Loop Compatibility Testing

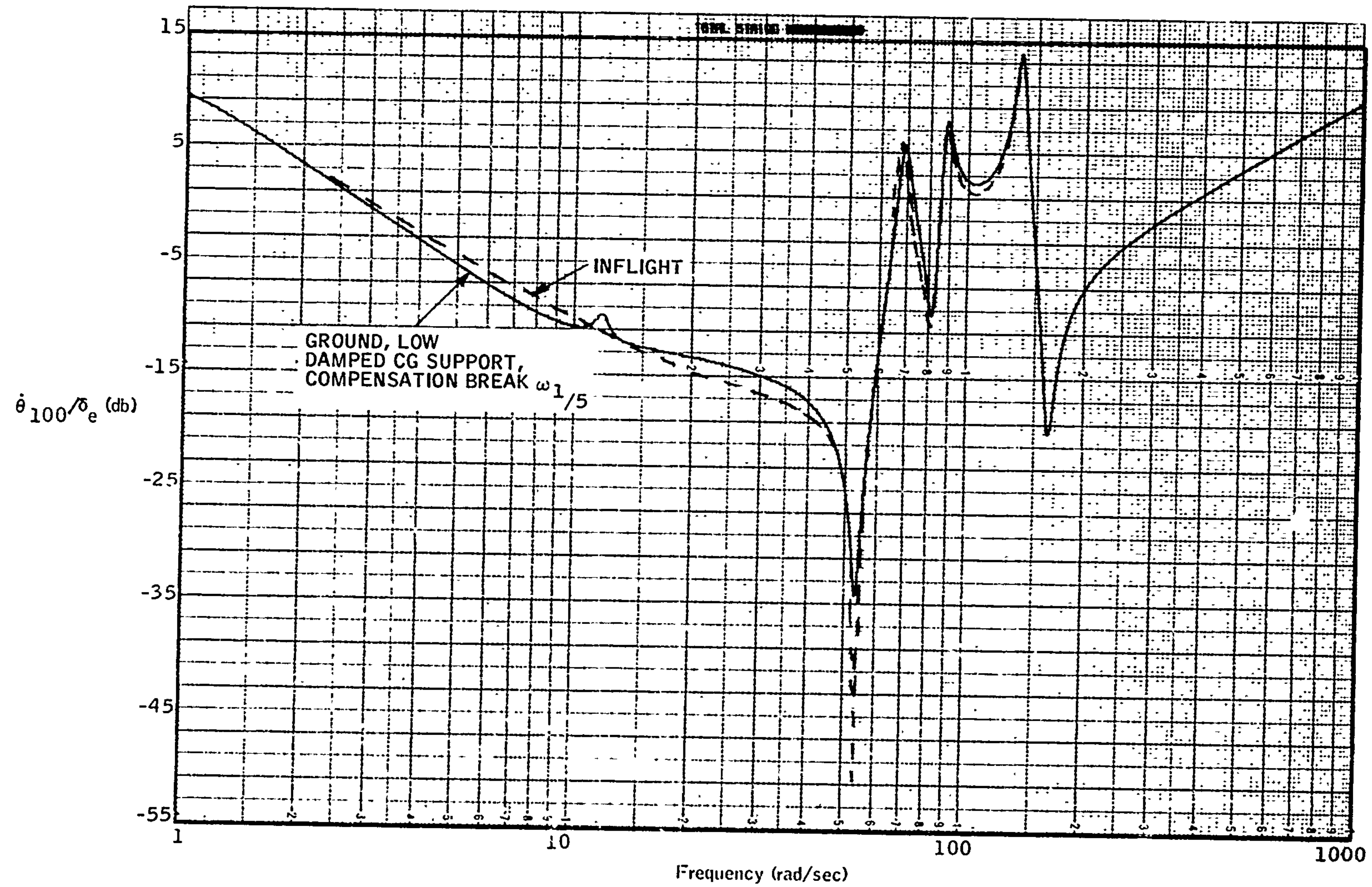


Figure 8-18. Amplitude Response Comparison, Total Ground Simulation versus in Flight

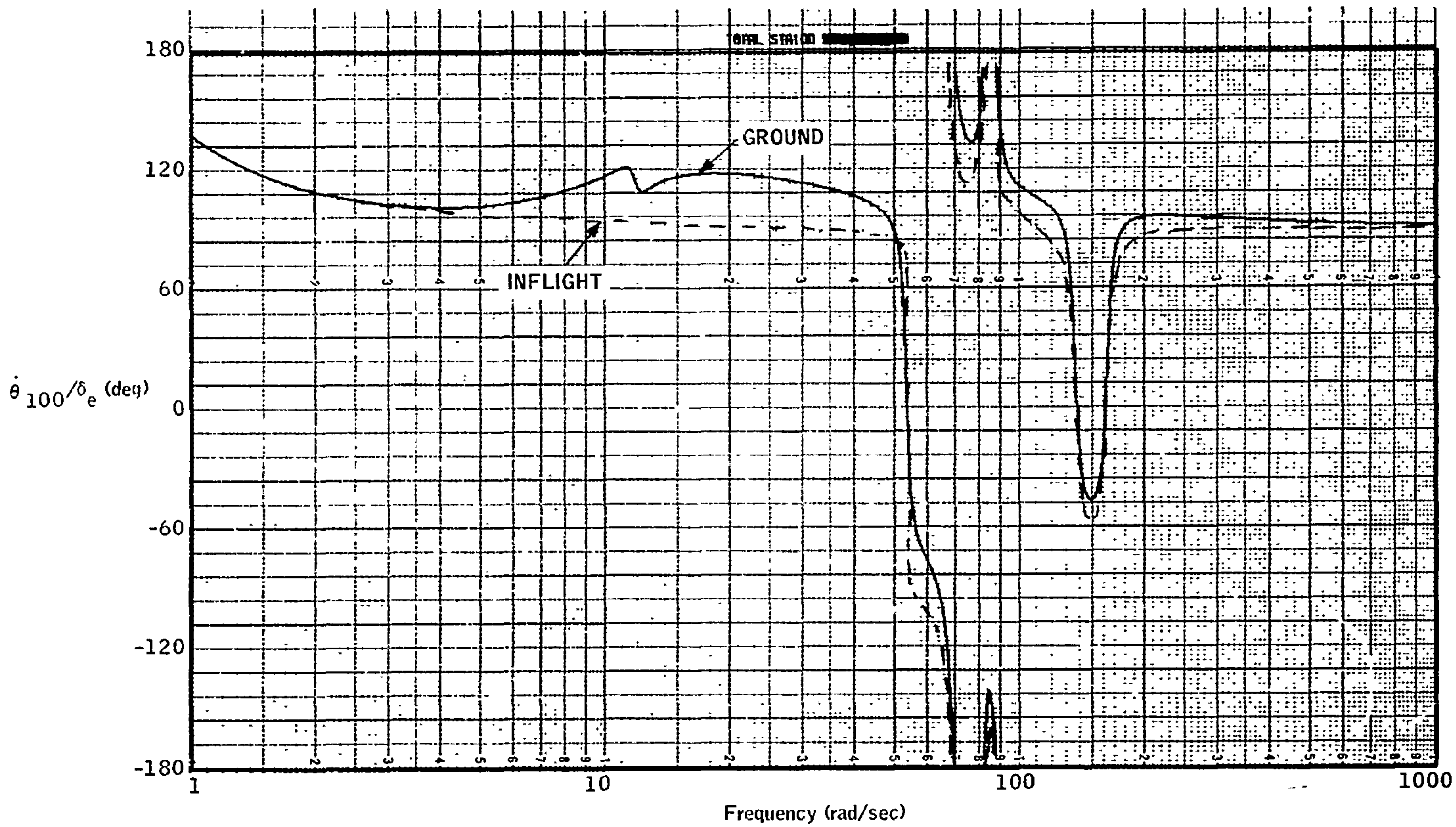


Figure 8-19. Phase Response Comparison, Total Ground Simulation versus in Flight

- Given proper means of airframe support and suitable correction for lack of surface aerodynamics, closed-loop system operation using actuators, sensors, and electronics in their flight configurations can be an effective and accurate procedure for verifying system stability in the presence of structural flexure.

- In general, the support provided by conventional landing gear contributes considerable damping to the lower-frequency bending modes, making structural stability tests under this condition of questionable value.

- Support means with low damping and located near the nodes of significant bending modes are desirable.

- Compensation for lack of surface aerodynamics must be provided in closed-loop structural testing.

- Total closed-loop testing for simultaneous evaluation of flexure and rigid performance is subject to considerable error around intermediate (between short-period and bending) frequencies unless particular support qualities and correction factors are employed.

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SECTION IX

CRITERIA FOR BUILT-IN TEST EQUIPMENT

BACKGROUND

The high-authority PFCS has assumed a role of importance to the basic airplane which strongly influences the question of built-in test equipment (BITE). A study of BITE always leads to questions dealing with system effectiveness, logistics, operational analysis, and other involved strategies. Past works have attempted to deal broadly with the subject as it applies to classes of equipment, for example, avionics as dealt with in Reference 9-1. Quantitative results are generally frustrated under such a generalization. In particular, application to extremely safety-oriented systems such as a PFCS is difficult, primarily because the vital connection between fault detection and operational reliability present in a redundant system is not sufficiently exercised. There is value in the broader studies, however, for direction in specific applications. Reference 9-1, for example, presents the following guidelines for performance monitoring:

(a) Monitor by function or mode;

(b) Safety items are prime candidates;

(c) Monitor that status which is not self evident;

(d) Consider probabilities of failure and components;

(e) Indicate failure modes to enhance mission decision making.

Items (b) and (d) have particular impact for the subject investigation. Most significant are the safety-of-flight implications of the system. Flight safety is of concern either:

- Because the aircraft is dangerous to fly without the system, or
- Because the aircraft is impossible to fly without the system.

The latter case would, of course, include the fly-by-wire system. For either case, using piece parts will be available over at least the next decade, it is expected that some degree of continued operation after failures will be necessary (i.e., redundancy). This in turn implies means of failure detection as a part of the basic system. Recognizing that the monitors, comparators, and signal-selection devices common to today's redundant flight controls are in themselves a form of test equipment, any study of BITE for the subject PFCS must of necessity include both ground and in-flight testing.

For the purposes of this study, the definition of BITE given by MIL-STD-1309 applies: "Any device permanently mounted in the prime equipment and used for the express purpose of testing the prime equipment, either independently or in association with external test equipment".

BITE therefore includes:

- Monitors which compare like channels;
- Monitors which detect tracer signals;
- Equipment which applies stimuli and detects the results.

Considered in this broad sense and applied to the high-authority PFCS, it is evident that the primary purpose of BITE and any other auxiliary testers or procedures is to contribute to the necessary system reliability, both from a flight-safety and mission-completion standpoint. A secondary purpose may be to enhance maintainability. In view of the primary purpose of BITE, therefore, its performance in terms of failure-detection capability and the relationship of this property to the system-failure and mission-abort probabilities allowed for the PFCS are the major concerns of this study.

The approach taken is to investigate the quality of testing necessary with little attention to the source of the tests (e.g., BITE, crew observation, AGE, etc.) other than their ground or inflight nature. In pursuing this question, it will be necessary to categorize the PFCS equipment in terms of both the nature of the tests and the consequences of the potential failures. The latter is necessary to avoid overspecifying tests of noncritical functions and vice versa. Current flight control detail specifications sometimes define a BITE "confidence factor" requirement -- a minimum probability of failure detection for the entire system. Numbers such as 95 percent may be extremely demanding in terms of BITE design, yet woefully inadequate for safety-critical functions. This dilemma would be alleviated by properly allocating test effort.

TOTAL SYSTEM FAILURE CRITERIA

The following subsections are concerned with total failure of a set of redundant channels applied as part of a PFCS. Hence, flight safety is involved. In this respect only equipment that can impact flight safety, either directly or indirectly, is of concern. Correspondingly, the derived criterion are applicable only to this class of equipment.

Flight safety/reliability is the probability of continued service of some function upon which flight safety is dependent. The requirement for this probability is determined by the consequences of a failure, as affected by:

- The availability of a backup system;
- The controllability of the aircraft with the backup system;
- Transients induced by the failure.

A numerical value for flight safety/reliability is often difficult to resolve, it being a catastrophic event in some cases. An example of such a figure suggested for the entire PFCS by Reference 9-2 is a probability of control loss of 2.3×10^{-7} per flight hour. This figure is derived from reports of those commercial aircraft accidents which were attributed to the flight control system for the years of 1952 to 1959. Since this figure is based on relatively simple mechanical flight controls, it seems probable that the more complex systems with fully powered surfaces would be less reliable. Considerable support for this proposition is offered by Section III of this report where the catastrophic failure rate due to the PFCS of the F-4 is reported at 3.8×10^{-6} failures per hour based on 3,000,000 flight hours. Additional support comes from Reference 9-3 which derives a failure rate of 1.14×10^{-6} failures per hour for the current F-4 longitudinal PFCS[1]. Allowing an additional amount for the lateral-directional control produces a close agreement with the data of Section III. These data suggest that the figure of Reference 9-2 may be an order of magnitude too low for current fighter aircraft.

Some judgment is required when analyzing a specific system to determine whether certain piece parts are flight-safety related. An elapsed-time meter obviously is not. An electronic filter may or may not be, depending on the consequences of its failure. Portions of BITE which are not used in flight or which cannot contribute latent test failures are not flight-safety related.

In the following analysis, equipment is categorized according to its contribution to single-point failures and according to the means by which it is tested. Common to all classes, however, is a finite impact on flight safety, either because it is essential to safe flight or because of its ability to disrupt other equipment which is essential to safe flight.

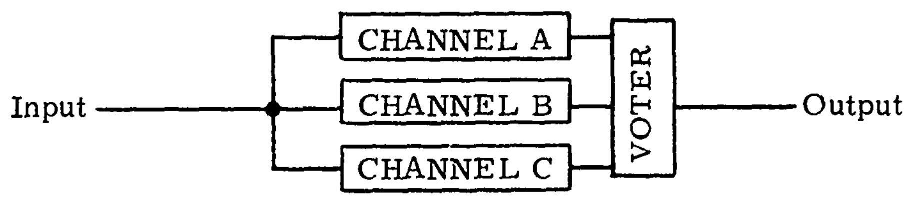
It should also be recognized that division of equipment to represent reliability effect may transcend physical possibility. For example, an open failure of a capacitor may have no impact on flight safety whereas a short-circuit failure may. The associated failure rates of each mode of failure then constitute the desired categorization.

BENEFITS OF REDUNDANCY

Redundancy is the means whereby a reliable system can be created using relatively unreliable parts. This is indeed common knowledge, but a brief review may be beneficial to establish terminology and analytical format. Consider, for example, a single control channel of a single control axis of a PFCS:

[1]This figure is based on a very inclusive representation of the PFCS, including the engines as the source of hydraulic pump power. It is interesting to note that the failure rate of nonredundant mechanical elements was estimated to be the dominant influence.

This element would, using state-of-the-art parts in electronics, sensors, and actuators, have a failure rate in the order of 3×10^{-4} per hour. Since this is grossly inadequate for very critical functions (perhaps by three orders of magnitude), redundancy is applied. One common technique is to use three channels:



Three are used to enable a voting process on the outputs. With this procedure, system failure occurs when two out of three fail, a probability of

$$Q_s = 3Q_c^2 = (\lambda_c t_m)^2 \qquad (9.1)$$[2]

where

Q_c = probability of an individual channel failing during the mission

Q_s = probability of the entire system failing during the mission

t_m = mission time

λ_c = individual channel failure rate, per unit time.

Here λ_c includes all the flight-safety-related piece parts as previously discussed, inclding those monitors which are designed such that their failure is treated as a failure of the associated channel. Equation (9.1) is defined, for purposes of this study, as the "basic" failure probability and will be referred to as such in subsequent subsections. This expression is, however, only a first approximation which may be grossly inaccurate for an actual system. and is based on a number of critical assumptions.

[2]The approximation $1 - e^{-\lambda t} = \lambda t$ can be used throughout this analyses with negligible error because of the small values of λt. Note also that the comparatively negligible probability of system failure due to all three channels failing is neglected.

If valid, however, a dramatic potential reliability gain is indicated. For example, using $\lambda_c = 3 \times 10^{-4}$, the triple system would have a failure probability for a one-hour mission of

$$Q_s = 3 \times (3 \times 10^{-4})^2 = 2.7 \times 10^{-7} \tag{9.2}$$

for a factor of 1000 improvement.

In the following paragraphs, the various contributions to system failure probability will be explored and the associated impact of BITE identified. As a means of quantifying the failure constituents, each will be compared to the so called "basic" failure probability as expressed by Equation (9.1). Even through the added sources are sometimes regarded as being "second-order" effects, it should be recognized that they often dominate in the overall system reliability. Criteria for key system parameters will be expressed relative to the basic probability, but inference should not be drawn that a good redundant system design is one which essentially achieves its basic reliability potential as expressed by (9.1). To do so, one may incur a large penalty in system complexity and associated costs. Rather, the basic influences should be recognized and properties selected to produce required reliability levels within cost constraints.

It should also be remembered in the following analyses that there are many possible variations to a triple system. Any number of crossfeeds and monitoring planes can be used to alter the reliability configuration and the associated failure potentialities. The basic format considered here, however, is the building block common to all triple systems currently contemplated for flight controls, and the relationships and guidelines developed should be generally applicable.

SINGLE-POINT FAILURES

First (and perhaps most important), the system was assumed in Equation (9.1) to be "truly triple". In most redundant flight controls, one can usually find an eventual "single element", even if it reduces to the use of a single control surface. Obviously, these must be very reliable elements, since their failure permits no second chance. Their presence adds a second term to the system failure rate:

$$Q_s = 3Q_c^2 + Q_{IC} = 3(\lambda_c t_m)^2 + \lambda_{IC} t_m \tag{9.3}$$

where Q_{IC} is the probability of failure of "individually critical" elements and λ_{IC} is the associated failure rate per unit time. Again considering a one-hour flight time and recalling the example of Equation (9.2), Q_{IC} would have to be less than 2.7×10^{-7} if the improvement provided by triple redundancy is not to be compromised. This corresponds to the failure rates of some single transistors. This suggests that the system failure rate can easily be dominated by single-point failures, and this is indeed the case for some redundant systems. Their presence is not necessarily bad design practice, but recognition of their impact by the designer is all-important.

It is also important that the designer recognize that individually critical failures may arise from redundant elements if the monitoring and failure correction capabilities of the system are unable to cope with them. For example, a capacitor failure in a filter might result in a limit cycle with an amplitude insufficient to cause a monitor trip. Such a limit cycle may, however, be destructive. Here an analysis is necessary, involving the probability of occurrence of the failure, the ability to detect it inflight, and the ability of ground testing to detect it.

LATENT FAILURES

The failure probability expressed by Equation (9.1) for a given operating time assumes that all equipment operates properly at the beginning of the period in question. If, in fact, a latent failure does exist such that the system will malfunction when the next failure occurs, the actual reliability of the redundant set is inferior to that of a single-channel system. Latent failures result from testing deficiencies, and it is this question which determines BITE capabilities.

In terms of their contribution to overall system failure probability, latent failures are ass ed (somewhat conservatively) to be potentially disruptive whenever they occur in conjunction with any other failure of one of the control channels. This joint-probability situation adds a third significant term to Equation (9.3):

$$Q_s = 3Q_c^2 + Q_{IC} + 3Q_L Q_c \qquad (9.4)$$

where

Q_L = probability of a latent failure existing at the start of a flight in one channel

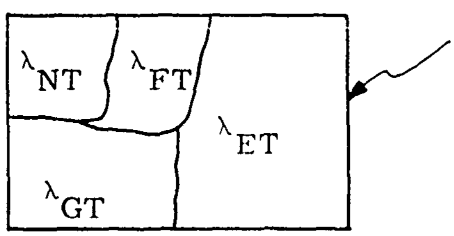
Q_c = probability of any failure occurring in a channel during the flight

The factor of 3 included in the latter term of (9.4) arises from the presumption that the latent failure existed in a channel which had not been used for active control. A second channel then failed, whereupon the latent channel was selected for control from the remaining two (a 50 percent chance). A conservative aspect of the added term is that for some types of latent failures, only particular types of faults in a second channel may produce difficulties.

One example is where a monitor has developed a latent failure, and a channel fails such that it must be disconnected if the set is to continue to function. These situations may be hypothesized and approximate analytical expressions derived having minor (e.g., factor of 2) differences from the latter term in Equation (9.4). These inaccuracies can, however, be largely compensated in a particular application by appropriate selection of failure rates, which are in themselves subject to even greater deviations.

Reasons why one channel can be failed at takeoff are illustrated by Figure 9-1. Note that all are associated with some deficiency in the ground tests, either because certain equipment is not tested by the ground tests or because a hardware failure has occurred in the ground test equipment which caused a failure to be missed. This assumes that should a failure be detected by ground testing, repair would be effected until the ground test was satisfied.

For purposes of this investigation, the total set of equipment that is essential to function in each channel is divided into four categories on the basis of the test means as illustrated in the following sketch. This equipment group excludes the "individually critical" elements, since the latter will not be latent failures.



All flight-essential equipment in one channel, failure rate of λ_c

$$\lambda_c = \lambda_{NT} + \lambda_{GT} + \lambda_{ET} + \lambda_{FT}$$

λ_{NT} = "not tested" equipment, on ground or in flight

λ_{FT} = "flight tested" equipment (<u>not</u> ground tested)

λ_{GT} = "ground tested" equipment (<u>not</u> flight tested)

λ_{ET} = "everywhere tested" equipment (on ground and in flight)

Note that these four categories are associated with the four contributions to latent channel failures illustrated by Figure 9-1:

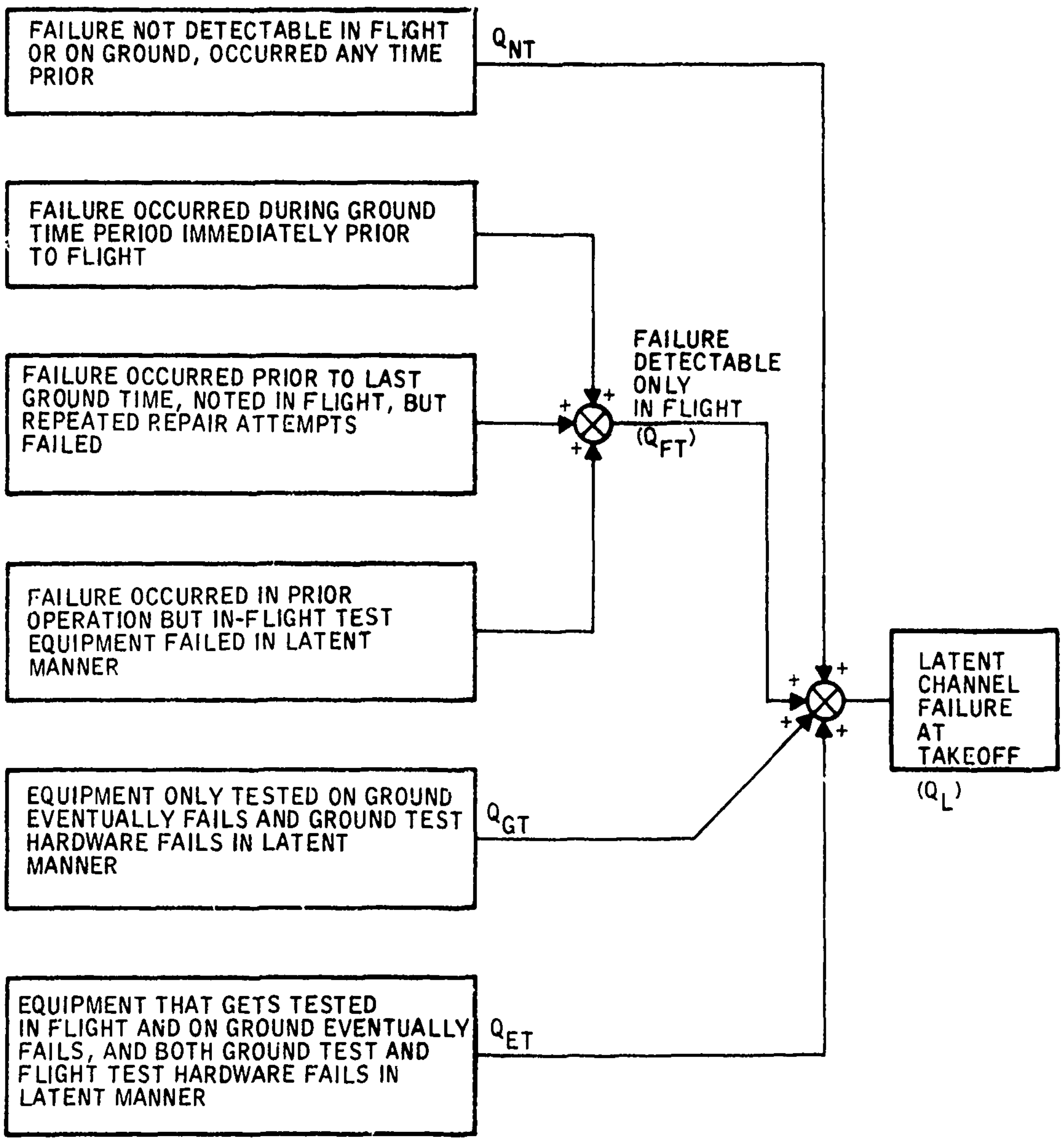


Figure 9-1. Contributions to a Failed Channel at Takeoff

$$Q_L = Q_{NT} + Q_{FT} + Q_{GT} + Q_{ET} \tag{9.5}$$

combining Equations (9.4) and (9.5) produces

$$Q_S = 3Q_c^2 + Q_{IC} + 3Q_C (Q_{NT} + Q_{FT} + Q_{GT} + Q_{ET}) \tag{9.6}$$

Each of the added "latent" terms are discussed in following paragraphs.

Latent Failures Due To Not-Tested Equipment

If equipment is not tested in a redundant system, there is a probability that a failure can exist such that when the affected channel is called upon to perform, it will be unable to do so. The probability of this source is expressable as

$$Q_{NT} = \lambda_{NT} t_t \tag{9.7}$$

where

λ_{NT} = failure rate of not tested equipment in each channel

t_t = total operation time since manufacture

In terms of the probability of a failure of a given channel during a specified mission length, this latent contribution represents an effective degradation in reliability as time accumulates on the system. Comparing the probability of being failed prior to the mission ($\lambda_{NT} t_t$) to the probability that a failure will occur during the mission

$$\frac{Q_{NT}}{Q_c} = \frac{\lambda_{NT} t_t}{\lambda_c t_m} \triangleq D_T \left(\frac{t_t}{t_m}\right) \tag{9.8}$$

where D_T is defined as the total test deficiency level of the equipment.

This relationship is plotted in Figure 9-2. It is notable that very thorough tests must be performed to avoid a significant reliability reduction after a relatively short operating time.

Extending these results to a triple system, the contribution of this failure source is $3Q_c Q_{NT}$ [see Equation (9.6)]. Hence, in terms of only this source and the basic triple-channel probability ($3Q_c^2$)

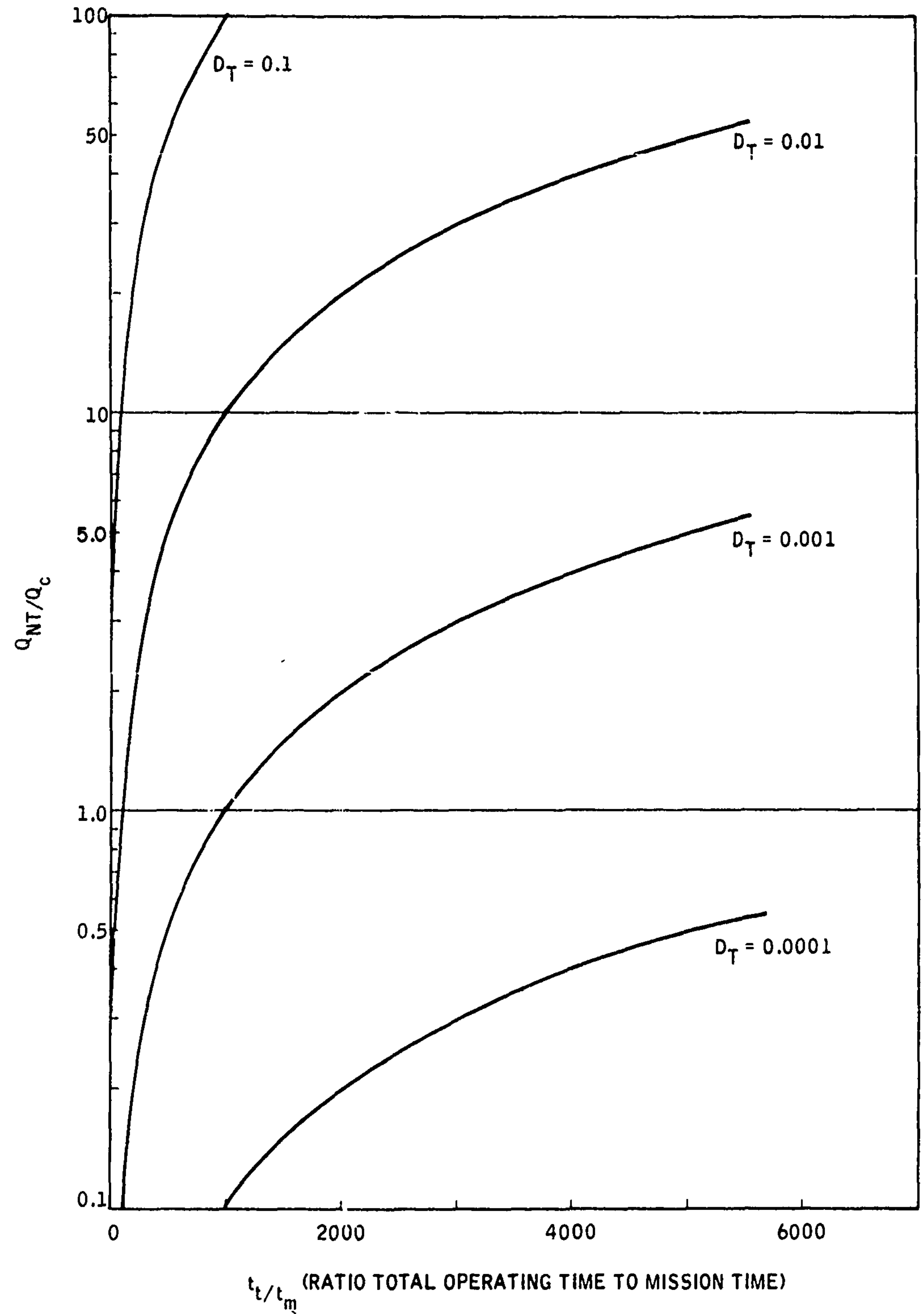


Figure 9-2. Unit Reliability Degradation with Operating Time

$$Q_s \triangleq 3Q_c^{\,2} + 3Q_c Q_{NT} \tag{9.9}$$

or

$$Q_s/3Q_c^{\,2} \triangleq 1 + \lambda_{NT}\, t_t / \lambda_c\, t_m \tag{9.10}$$

$$Q_s/3Q_c^{\,2} \triangleq 1 + D_T\, t_t / t_m \tag{9.11}$$

The ratio $Q_s/3Q_c^2$ measures the reliability degradation due to total test quality as time accumulates on the system. The function of (9.11) is plotted as Figure 9-3, again indicating that a high overall test quality (i.e., a low D_T) is necessary if significant degradation in triple-channel reliability is to be avoided. In terms of avoiding a dominant degradation of system reliability at end of life, it is evident from both Figure 9-2 and Figure 9-3 that $D_T < 0.0002$.

It is interesting to reflect at this point that, with the typical channel failure rate of 3×10^{-4} per hour quoted previously, a value of 0.0002 for D_T allows an "untested" failure rate of 6×10^{-8} per hour. This amount could be contributed by a simple electronic part (e.g., a diode).

Latent Failures In Equipment Tested Only in Flight

This category includes failures which the ground-test function does not test for. An example might be certain sensor qualities. Figure 9-1 identifies three situations applicable to this category:

(a) A failure could occur during the ground test interval (t_g) just prior to the current mission. This probability is expressed as

$$Q = \lambda_{FT}\, t_g \tag{9.12}$$

(b) A failure could occur sometime prior to the last ground time, be noted in one or more prior flights, and have repeated repair attempts fail. This probability is expressed as

$$Q = \lambda_{FT}\,(t_g + t_m)\,(Q_{ur} + Q_{ur}^{\,2} + Q_{ur}^{\,3} + \text{---}) \tag{9.13}$$

where Q_{ur} = probability of an unsuccessful repair

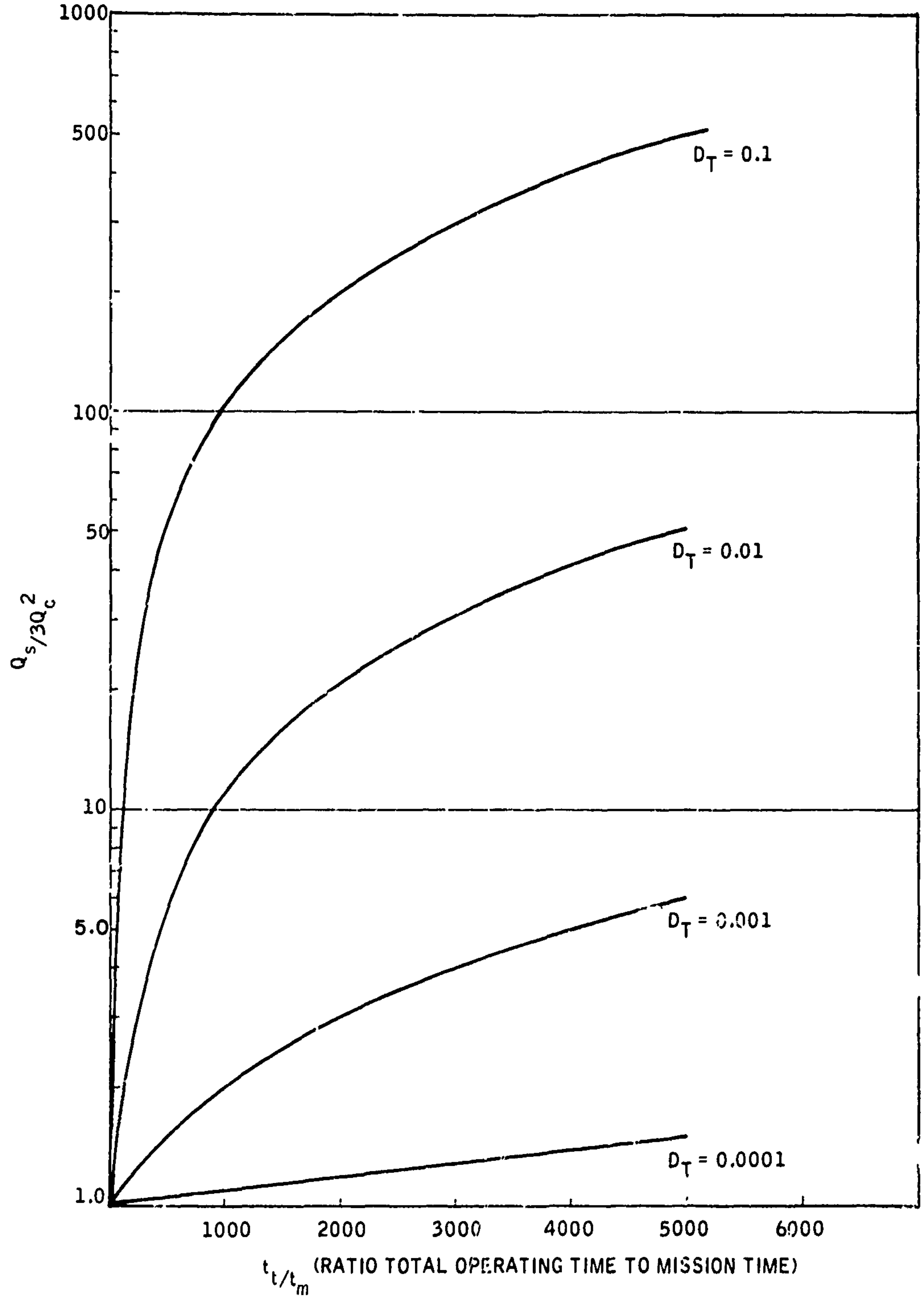


Figure 9-3. Effect of Non-Tested Equipment on Triple-Channel Reliability

The above expression is somewhat pessimistic because in-flight aborts due to the failure would tend to make the mission time shorter than usual.

(c) A failure occurred in prior operation but in-flight test equipment failed in a latent manner. This probability is expressed as

$$Q_c = \lambda_{FT} \Delta t_{PT} Q_{LFTF} \tag{9.14}$$

where Q_{LFTF} = probability of a latent flight test failure existing during the prior mission.

Δt_{PT} = time interval between perfect testing, i.e., auxiliary tests to determine absolute condition of either the test equipment or the other associated system equipment.

If, for example, no "perfect" tests were ever run on the system or its BITE after manufacture, Δt_{pt} could equal total operating time, t_t. Alternatively, if the BITE were made absolutely failsafe, $Q_{LFTF} = 0$.

Taken together, the above three sources determine Q_{FT}, the probability of a latent channel failure at takeoff in equipment that is tested only in flight:

$$Q_{FT} = \lambda_{FT} t_g + \lambda_{FT} (t_g + t_m)(Q_{ur} + Q_{ur}^2 + \ldots) + \lambda_{FT} \Delta t_{PT} Q_{LFTF} \tag{9.15}$$

In terms of these contributions to total system failure probability and the basic probability:

$$Q_s \triangleq 3Q_c^2 + 3Q_c Q_{FT} \tag{9.16}$$

Combining (9.15) amd (9.16):

$$Q_s = 3(\lambda_c t_m)^2 + 3(\lambda_c t_m) \lambda_{FT} [t_g + (t_g + t_m)(Q_{ur} + Q_{ur}^2 + \ldots) + \Delta t_{PT} Q_{LFTF}] \tag{9.17}$$

Equation (9.17) enables comparison of the above sources (a), (b), and (c) to the basic triple-channel failure probability $3Q_c^2$.

Considering first situation (a), note that since the system ground time t_g compares in magnitude to the mission flight time t_m, and since the amount of equipment tested only in flight (λ_{FT}) is probably small compared to the total equipment (λ_c), the contribution of (a) can be neglected. Furthermore, since the failure is notable in flight, the abort option would make its role as a safety threat questionable.

Considering situation (b), the probability of an unsuccessful repair is pessimistically placed at 0.5. Consequently, source (b) is comparable to (a) and can also be neglected. It too involves an abort option and is a questionable safety threat.

Situation (c) is more difficult. If (9.17) is reduced by neglecting sources (a) and (b)

$$Q_s \triangleq 3(\lambda_c t_m)^2 + 3(\lambda_c t_m)\, \lambda_{FT}\, \Delta t_{PT}\, Q_{LFTF} \tag{9.18}$$

$$\frac{Q_s}{3Q_c^{\ 2}} \triangleq 1 + \frac{\lambda_{FT}\, \Delta t_{PT}}{\lambda_c t_m} \tag{9.19}$$

The ratio $Q_s/3Q_c^{\ 2}$ in this instance measures the reliability degradation due to latent flight test failures. As shown by (9.19), the quality required in Q_{LFTF} depends on its importance to the overall test function, the latter being measured by the ratio λ_{FT}/λ_c.

To keep latent flight test failures as a negligible source of unreliability,

$$Q_{LFTF} < \left(\frac{\lambda_c}{\lambda_{FT}}\right)\left(\frac{t_m}{\Delta t_{PT}}\right) \tag{9.20}$$

<u>Latent Failures in Equipment Tested Only on Ground</u>

The probability of this source is

$$Q_{GT} = \lambda_{GT}\, \Delta t_{PT}\, Q_{LGTF} \tag{9.21}$$

where

Q_{LGTF} = probability of a latent failure <u>existing</u> in the ground test equipment during the checkout <u>prior</u> to the current mission

An example of equipment which falls into this category is a monitoring and failure correction function which is active only in the event of a failure. Since evaluation of proper action often requires an actual or simulated failure, the ground test may be the only way in which this element is exercised. The portion of test equipment which is subject to this limitation is made as small as possible by design.

All effort is applied to make "fail safe" monitors, i.e., monitors which disable their associated channel upon failure. An example of such a monitor is the circuit comparator which requires a continuous oscillation passage to verify performance. Most circuit failures within the monitor disrupt the test signal, causing failure indication.

Again temporarily reducing Equation (9.6) to only the basic triple-channel failure probability and the subject probability

$$Q_s \triangleq 3Q_c^{\,2} + 3Q_c\, Q_{GT} \tag{9.22}$$

With (9.21) and (9.22),

$$\frac{Q_s}{3Q_c^{\,2}} \triangleq 1 + \frac{Q_{GT}}{Q_c} = 1 + \frac{\lambda_{GT}\,\Delta t_{PT}\,Q_{LGTF}}{\lambda_c t_m} \tag{9.23}$$

Hence, to keep the ground test contribution low

$$Q_{LGTF} < \left(\frac{\lambda_c}{\lambda_{GT}}\right)\left(\frac{t_m}{\Delta t_{PT}}\right) \tag{9.24}$$

<u>Latent Failures in Equipment Tested Both on Ground and in Flight</u>

The probability of this source as

$$Q_{ET} = \lambda_{FT}\,\Delta t_{PT}\,Q_{LTF} \tag{9.25}$$

where

Q_{LTF} = probability of a latent failure of both ground test and flight test equipment <u>existing</u> during the current ground test period and the prior flight

From the definition of Q_{LTF} it might be concluded that a joint probability is involved, but in fact the same equipment will probably be involved for both tests. Comparing (9.25) to (9.14) or (9.21) demonstrates that a requirement for Q_{LTF} comparable to (9.20) or (9.24) is necessary:

$$Q_{LTF} < \left(\frac{\lambda_c}{\lambda_{ET}}\right) \left(\frac{t_m}{\Delta t_{PT}}\right) \qquad (9.26)$$

Observance of (9.26) will make the associated source of total system failure small compared to the basic triple-channel rate. Since in most redundant systems the majority of equipment is tested both on ground and in flight

$$\lambda_{ET} \approx \lambda_c$$

and (9.26) will reduce to

$$Q_{LTF} < \frac{t_m}{\Delta t_{PT}}$$

Assuming $\Delta t_{PT} = t_t$ and with t_m/t_t having a minimum value of about 1/5000

$$Q_{LTF} < 0.0002$$

for this source of failure to be negligible.

NUISANCE DISENGAGEMENTS

The signal comparison monitoring technique used in most redundant systems has the problem of separating normal tolerance deviations from failures over a relatively broad frequency range. The two phenomena may actually overlap in some cases. The system designer wants as low an allowed deviation ("trip level") as possible to catch low-amplitude failures, but as wide a deviation as possible to permit normal tolerances and avoid nuisance disengagements or "trips". The use of selective filtering sometimes enables improved distinction. The net result of these attempts is that the monitors are set to catch the failures, and high tolerances cause nuisance trips. Provision is then made for pilot reset of the disabled channel. This capability will ensure that the first registered failure is valid (supposing that repeated reset attempts have failed). For the second failure, however, total disengagement will occur for either a valid failure or a nuisance trip. Hence, a modification of (9.1) is in order:

$$Q_s = 3Q_c^2 \rightarrow 3Q_c^2 + 6Q_cQ_{CN} \qquad (9.27)$$

where

Q_{CN} = probability of a nuisance trip during the mission

Defining $F_I = Q_{CN}/Q_c$, (9.27) may be written

$$Q_s = 3Q_c^{\,2}\,(1 + 2F_I) \tag{9.28}$$

where

F_I = "false indication rate"

The treatment of nuisance trips as a random quantity for a specific vehicle is debatable, but for an entire fleet its origin in random equipment tolerances is evident. Specific vehicles will, consequently, demonstrate nuisance-trip problems far more than others and present maintenance problems. From experience with past operational redundant systems, F_I can approach 10, indicating a comparable increase in total failure probability.

CONCLUSIONS ON TRIPLE-SYSTEM TOTAL FAILURE

The preceding subsections analyzed the major sources of total system failure in a triple-redundant voted system and identified pertinent parameters. Collecting the major sources, the probability of total failure can be approximated by

$$Q_s = \lambda_{IC}t_m + 3(\lambda_c t_m)^2 \left[1 + \frac{D_T t_t}{t_m} + 2F_I + \frac{\Delta t_{PT}}{t_m}\left\{+\frac{\lambda_{GT}}{\lambda_c} Q_{LGTF} + \frac{\lambda_{FT}}{\lambda_c} Q_{LFTF} + \frac{\lambda_{ET}}{\lambda_c} Q_{LTF}\right\}\right] \tag{9.29}$$

The terms involving test nature and quality are seen to be functions which increase in importance as the system life increases, and it is evident that these can soon dominate the "basic" probability. The wisdom of allowing this is dependent on tradeoffs unique to each application. The total equipment failure rate λ_c also includes BITE to the extent that certain BITE failures may result in disablement of a channel. Hence, decreasing D_T generally increases λ_c, suggesting optimization studies for each case.

There are several significant conclusions notable from Equation (9.29):

- For the triple-channel system, total test deficiency (D_T) will become a dominant influence on total system failure for values greater than 2×10^{-4} (based on a maximum ratio of total time to mission time of 5000).

- The allowable probabilities of latent test failures are dependent on the portion of the equipment tested (e.g., λ_{FT}/λ_c) and on the time interval between "perfect" system tests (Δt_{PT}). Avoidance of scheduled maintenance tends to make $\Delta t_{PT} = t_t$ and indicates that latent test failure probabilities greater than 2×10^{-4} near the end of system life for the majority of BITE will cause their influence to be significant.

- The important thing for system safety is that the equipment be tested at regular intervals, either on ground or in flight, enabling various combinations of λ_{GT}, λ_{FT}, and λ_{ET} to prevail. To achieve maximum safety after the first failure has occurred, however, a timely knowledge of faults occurring inflight is desired so that their existence can be known to the pilot. This knowledge will enable action to minimize the hazard, perhaps by a prudent change in flight condition or an abort to minimize remaining flight time. Hence, an attempt to design with a minimum amount of equipment categorized as λ_{GT} is a desirable goal.

- Nuisance trips more numerous than 50 percent of the valid failures ($F_I > 0.5$) significantly increase failure probability.

The potential of flight aborts is of course a concern in overall system effectiveness. The influence of BITE on the abort question is explored in the following subsection.

MISSION SUCCESS CRITERIA

For the redundant PFCS, a secondary yet vital concern to flight safety is mission reliability as measured by the probability that an abort will be necessary due to partial system failure. It is entirely possible that a given redundant system design can provide adequate flight safety yet be deficient from the standpoint of mission reliability. Again using the simple three-channel voted system as a point of study, it is probable that, for most applications, failure of any one channel is cause for abort (currently true for the F-111 CAS). If this were not the case, it is questionable that the complexity of the triple-channel system could be justified. Assuming that the first failure constitutes grounds for an abort, then

$$Q_{sa} \approx 3Q_c = 3\lambda_c t_m \qquad (9.30)$$

where

Q_{sa} = probability of system abort

Q_c = probability of failure of any one channel during the mission time t_m

It is interesting to compare the potential performance of a triple fly-by-wire system in the areas of flight safety (total failure) and mission reliability (single failure) to experience with a current hydromechanical PFCS. Such a comparison is made in Figure 9-4. Data for the F-4 PFCS is taken from Table 3-11 of Section III. The total-failure and single-failure probabilities for a perfectly tested triple system (assumed to constitute one axis of a three-axis-pitch, roll, yaw-system) are plotted versus single-channel failure rate. Also indicated is the failure rate corresponding to a hypothetical fly-by-wire channel composed of state-of-the-art equipment in electronics, hydraulics, actuation, and sensing. Figure 9-4 indicates that current equipment may produce adequate safety levels, but it is a factor of six deficient in mission reliability. Consequently, this area merits attention in current developments, including study of the impact of BITE performance.

CONTRIBUTIONS TO MISSION ABORT

The preceding subsection defined the "basic" source of first-failure probability of a triple-channel system in Equation (9.30). To this must be added the "second-order" contributions. Consider the ways in which an abort can occur as illustrated in Figure 9-5.

Failures Occurring During Ground Time

During ground time various scheduled and unscheduled maintenance activities take place which directly or indirectly affect system availability for the next mission. Certainly substantial operating time on PFCS and AFCS equipment is accrued (about 50 percent of the flight time according to the operational survey). Of perhaps more importance are the multiple-system transients and other on-off cycles which prevail and which intuitively are significant failure generators. This source of failures is quantified by Figure 9-5 as being $3(\lambda_c + \lambda_{GTE})\, t_g$ where t_g is the ground operating time and λ_c is the flight-essential equipment failure rate as used previously. A new class of equipment is introduced at this point having a failure rate of λ_{GTE}, accounting for potential BITE which is used to conduct ground tests only and which has no effect after the ground test sequence is accomplished. A BITE sequencer could, for example, be in this category. Generally, these devices are fail safe such that a test sequence is stopped if a BITE failure occurs. With accomplishment of a "go" indication necessary for flight, such a failure will result in an abort.

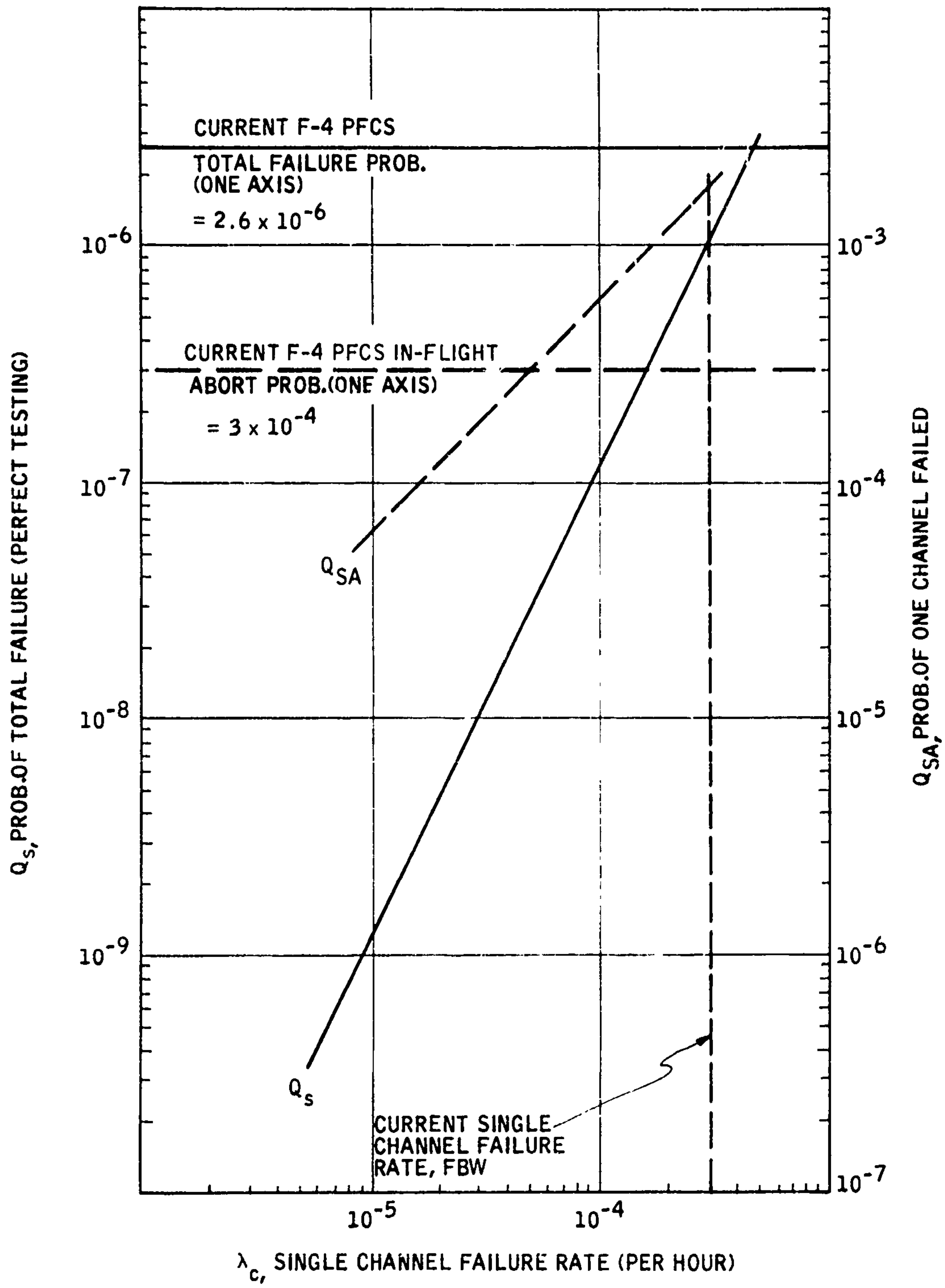


Figure 9-4. Triple-Channel Performance Relative to Current F-4 PFCS, Two-Hour Mission

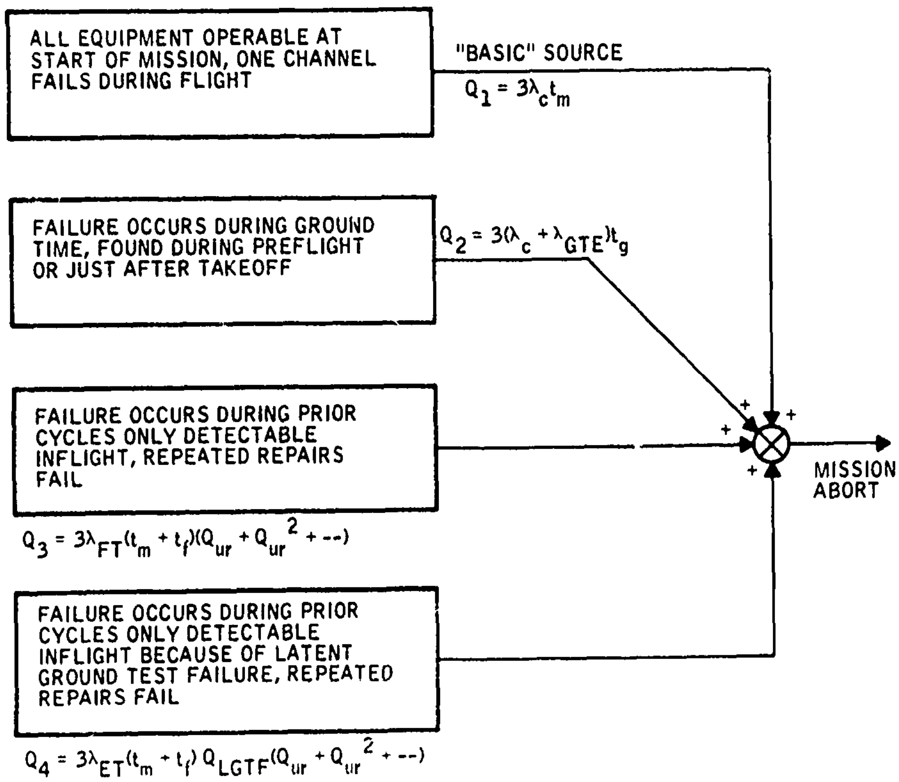


Figure 9-5. Contributions to Mission Abort, Triple System

An assumption involved in the subject contribution is that the time available after failure indication will be insufficient to enable repair. With proper design and failure-isolation capabilities, certain elements such as computer packages may be changeable during preflight check periods, thus reducing the "ground time" failure contribution to aborts. Certainly an "advanced checkout" followed by a lengthy shutdown prior to the preflight checkout would contribute little assuming that no failures were evident during the prior flight.

Comparing the ground time failure source to the basic abort source of $3\lambda_c t_m$, the same order of magnitude is evident unless a large amount of equipment in the λ_{GTE} category is used. Since the latter increases with the amount of BITE as does λ_c, it is evident that test equipment degrades mission reliability by both sources.

Failures Detectable Only in Flight

This source contributes an abort probability of

$$Q_3 = 3\lambda_{FT}(t_m + t_g)(Q_{ur} + Q_{ur}^2 + \ldots) \tag{9.31}$$

where terms are as previously defined. The rationale here is that a failure can occur in a prior ground/flight cycle, be subject to repeated repair attempts each having a probability of failure of Q_{ur}, and contribute according to the probability of a failure which is detectable sometime during the following flight. It is pessimistically assumed in (9.31) that each flight time has the full t_m, whereas an abort will probably reduce it. Regardless of this, Q_3 is negligible compared to Q_1 and Q_2 by virtue of λ_{FT} being small compared to λ_c and Q_{ur} probably less than 0.5.

Failures Passed by Latent Ground Test Failures

This source contributes an abort probability of

$$Q_4 = 3\lambda_{FT} (t_m + t_g) Q_{LGTF} (Q_{ur} + Q_{ur}^2 + \dots) \qquad (9.32)$$

where terms are again as previously defined. Here equipment which is tested both in flight and on ground fails in some cycle, the failure is missed by a ground-test-latent failure, subsequently noted in the next flight, and repair attempts repeatedly unsuccessful. Because of a low probability of the latent ground test failure (Q_{LGTF}) and because of $Q_{ur} < 0.5$, Q_4 will also be negligible compared to Q_1 and Q_2.

Conclusions on Triple-System Mission Reliability

Preceding subsections analyzed the major sources of single-system failures which determine the probability of mission abort for a three-channel system. It was shown that the abort probability could be approximated by

$$Q_{sa} = 3\lambda_c t_m + 3(\lambda_c + \lambda_{GTE}) t_g \qquad (9.33)$$

Since BITE contributes to increased failure rate of both flight-essential equipment (λ_c) and purely ground test equipment (λ_{GTF}), it has a degrading effect on mission reliability. Note that nuisance trips do not impact abort rate because of pilot reset capability.

TOTAL FAILURE PROBABILITIES FOR A BASIC QUAD SYSTEM

The preceding subsections studied the total failure probabilities and mission reliability for a triple system and related these to significant test qualities. This subsection investigates comparable quantities for a quad (four-channel) system. Such a system presents more complications than a triple system and is more difficult to treat on a general basis. Certain trends can be identified, however, which bear significantly on the BITE question.

Basic Quad Total Failure

The quad system considered here is as shown in Figure 9-6. Four signals are comparison-monitored to eliminate faulty channels. The concept assumes that all channels are initially good; and that, as failures occur, they are detected and eliminated from further consideration in the voting process. This arrangement continues until two channels are eliminated, after which a third failure shuts down the remainder of the set. This operation is commonly termed "two-fail-operate". The associated probability of three failures occurring during the mission (ABC, ABD, ACD, or BCD), is approximated by

$$Q_s = 4Q_c^3 \tag{9.34}$$

where

Q_c = probability of a given channel failing during mission as previously defined

Again

$$Q_c = \lambda_c t_m \tag{9.35}$$

where

λ_c = total failure rate of all flight-safety-related equipment in each channel, including monitors whose failures result in disablement of an assigned control channel

t_m = mission time

The "basic" probability given by (9.34) will now be augmented by additional contributions.

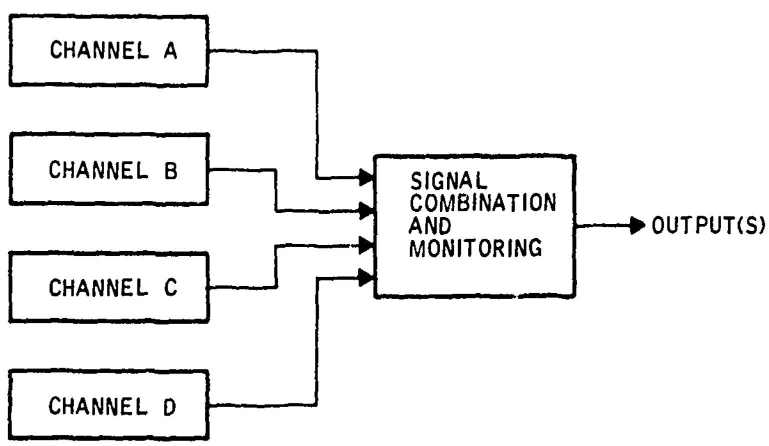


Figure 9-6. Quad System Redundant Set

Independently Critical Failures

The "single-point" failure potential must again be recognized and added to (9.34)

$$Q_s = 4Q_c^3 + Q_{IC} \tag{9.36}$$

Here the effect is even more significant than for the triple system because of the low values expected for $4Q_c^3$. For example, taking one-hour values for Q_c of 3×10^{-4} (a complete flight control channel) and for Q_{IC} of 1×10^{-8} (a single fixed composition resistor)

$$4Q_c^3 = 1.08 \times 10^{-10}$$

$$Q_{IC} = 1 \times 10^{-8}$$

The single-point probability is 100 times greater than the basic, suggesting that the basic reliability of the quad system may be an academic goal in an actual application.

Latent Failures at Takeoff

Latent failures are a potential source of severe reliability degradation in a quad system. The rationale applied for this source in a triple system was that if a latent existed in one channel at takeoff, a subsequent failure in another channel could result in failure of the entire set because the latent channel could be selected from the remaining two (a 50-50 chance).

Alternatively, a latent monitor failure could result in improper rejection of a faulty channel on its failure. It is evident that similar difficulties prevail in a quad system, and the presence of an added channel does little to alleviate the threat. The problem is simply that systems with few channels must recognize and reject failed members.

If the type of latent failure is considered which only causes difficulty if its associated channel is selected, the following cases exist:

(1) The latent channel might be selected after another channel has failed in a "conventional" manner.

(2) The latent channel might be selected after two other channels have failed. This probability is negligible compared to (1) and will be ignored.

Considering case (1) above, then, the probability of total system failure during the mission due to the latent at the start of the mission is

$$Q = \frac{(4Q_L)\times(3Q_c)}{3} = 4Q_L Q_c \qquad (9.37)$$

where

Q_L = probability of a latent failure.

The factor of 4 in (9.37) arises from the four channels capable of developing latent failures in a quad system, the factor of 3 in the numerator from the three channels without latent failures capable of having "conventional" ones, while the factor of 3 in the denominator accounts for the one-out-of-three chance that latent channel will be selected after the " conventional" failure of the other channel.

There is another type of latent failure which differs somewhat from the above type, one which involves an unknown failure in some monitoring function which becomes critical when its associated channel fails. This source would present a probability of total system failure of

$$Q = 4Q_L Q_c \qquad (9.38)$$

The factor of 4 here arises from the quad channels, producing a term identical to (9.37). Still other latent failure problems can be identified involving higher-order joint probabilities but which are comparatively negligible. Equation (9.37) adds the third term to the system failure expression of (9.36):

$$Q_s = 4Q_c^{\,3} + Q_{IC} + 4Q_L Q_c \qquad (9.39)$$

The sources of the latent failures will now be considered. In so doing, an equipment categorization according to the testing applied (not tested, flight tested only, ground tested only, or tested everywhere) as previously defined for the triple system will be used.

A review of the latent failure sources presented for the triple system indicates their continued validity for the quad system, and Equations (9.7), (9.14), (9.21), and (9.25) may be applied directly. Comparing (9.39) and (9.4) for the quad and triple systems, respectively, it is evident that the quad system can actually be inferior to the triple system from a total failure standpoint unless the total test deficiency (D_T) and probabilities of latent test failure (Q_{LFTF}, Q_{LGTF}, and Q_{LTF}) are made sufficiently small compared to the "basic" probability of failure of the triple system. A comparison of the total failure probabilities for the triple and quad systems is shown in Figure 9-7 for the assumed case of zero latent test failures ($Q_{LTF} = Q_{LFTF} = 0$), zero individually critical failures ($\lambda_{IC} = 0$), a channel failure rate of 3×10^{-4} (typical flight control channel) and a mission time of 2 hours. The variable

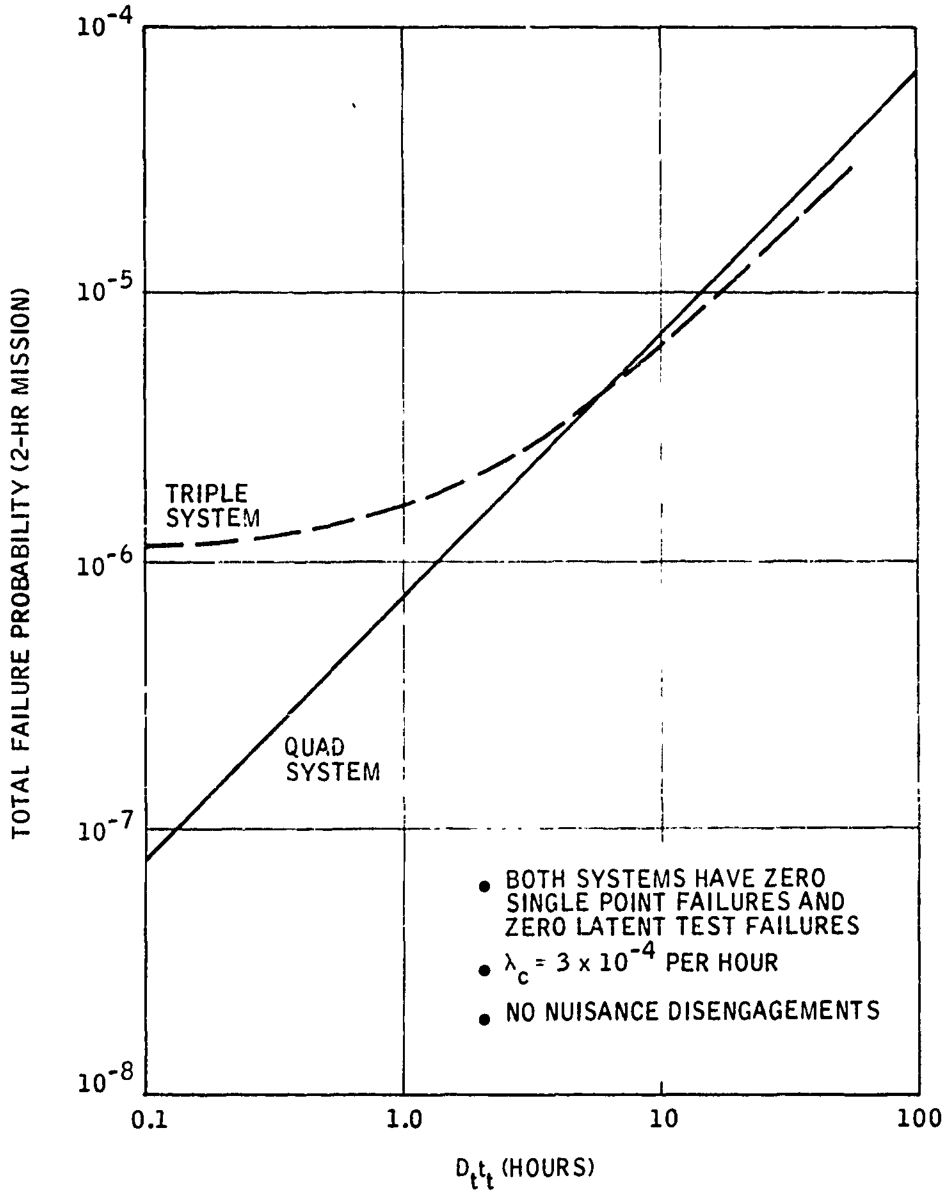


Figure 9-7. Total-System Failure Comparison

remaining is $D_T t_t$, the product of total test deficiency and total operating time. A maximum value of 50 is considered, corresponding to a total life of 5000 hours and a maximum test deficiency of 0.01. For this example, the triple system has superior flight safety after a value of 6 has been reached for $D_T t_t$.

Latent Failures During the Mission

The two-fail-operate quad system utilizes a voting concept which requires rejection of faulty channels as they fail. If, for example, two channels become "dead" with no monitor action (because of no stimulus, for example), the voter has a 50-50 chance of rejecting the two live channels. Such a difficulty also prevails with a triple system, but its failure with two bad channels is an accepted likelihood[3].

Certain elements of the system are prone to avoid a timely failure-detection process during flight. Equipment tested only on the ground (λ_{GT}) is in this category. Other devi .s may be significantly active only during portions of the flight, such as pitch rate gyros and bending filters. If this class of equipment is assigned a failure rate per channel of λ_{LF}, a total system failure probability during the mission is contributed of

$$Q = \frac{6Q_{LF}^2}{2} = 3(\lambda_{LF} t_m)^2 \tag{9.40}$$

where the factor of 6 accounts for the six pairs of channels possible and the factor of 2 accounts for the probability that the faulty pair will be selected by the logic. Hence, the total probability of (9.39) is expanded by an added term

$$Q_s = 4Q_c^3 + Q_{IC} + 4Q_L Q_c + 3Q_{LF}^2 \tag{9.41}$$

To minimize the contribution of the latter term, the amount of equipment not tested in flight on a continuous basis must be minimized. For it to be small compared to the basic failure probability

$$3Q_{LF}^2 < 4Q_c^3 \tag{9.42}$$

$$3(\lambda_{LF} t_m)^2 < 4(\lambda_c t_m)^3 \tag{9.43}$$

[3]There may be some preference for a dead total failure over an active total failure, but no distinction is made between the two in this study.

$$\frac{\lambda_{LF}}{\lambda_c} < \frac{4}{3} t_m \lambda_c \quad (9.44)$$

Using figures of 2 hours mission time and $\lambda_c = 3 \times 10^{-4}$

$$\frac{\lambda_{LF}}{\lambda_c} < 8 \times 10^{-4} \quad (9.45)$$

Hence, less than 0.08 percent of the total equipment failure rate must be in this category for the subject contribution to be obviously negligible.

Nuisance Disengagements

The possibility of nuisance disengagements presents a unique threat to a quad system in the form of a voting quandry, i.e., two bad channels against two good channels. This can arise if two similar failures occur and if the pilot is able (and does) reset both tripped channels.

If this is permitted by the system design, and, if it is assumed that the pilot will reset, a probability of total failure due to this source prevails as follows:

$$Q = \frac{6Q_{SF}^2}{2} = 3Q_{SF}^2 \quad (9.46)$$

where

Q_{SF} = probability of a given kind of failure occurring in one channel.

For example, taking similar failures to be either hardovers (Q_{HO}) or dead (Q_D),

$$Q = 3 [Q_{HO}^2 + Q_D^2] \quad (9.47)$$

Further assuming, if the total channel failure probability (Q_c) were either due to hardover or dead failures, and if $Q_{HO} = Q_D$, then

$$Q = 3[2Q_{HO}^2] = \frac{3}{2} Q_c^2 \quad (9.48)$$

Comparing the result of (9.48) with the basic probability for a triple system ($3Q_c^2$), it appears that the subject hazard degrades the quad flight safety to about half that of a basic triple system. Arguing that this is unquestionably unsatisfactory, it is concluded that pilot reset of two failed channels in a quad system cannot be allowed by design.

Without the reset ability for all failure levels, the probability of a nuisance trip affects the flight safety of the system.

Assume that the pilot is allowed to reset the first failure only. This capability will tend to ensure that the first channel is ultimately disabled for a valid failure only. After such has occurred, therefore, the second and third disablements can result from either nuisance trips or valid failures, contributing the following modification to the "basic" failure probability of a quad system:

$$4Q_c^3 \rightarrow 4Q_c^3 + 12Q_cQ_{CN}^2 \tag{9.49}$$

or

$$4Q_c^3 \rightarrow 4Q_c^3 \left(1 + 3\,\frac{Q_{CN}^2}{Q_c^2}\right) \tag{9.50}$$

where

Q_{CN} = probability of a nuisance trip during the mission.

The ratio of Q_{CN} to Q_c is problematical, but in past operational systems, it has approached 10. With this magnitude, Equation (9.49) indicates about a factor of 300 degradation in basic quad channel reliability.

Recalling that Q_{CN}/Q_c has been defined as F_I, the "false indication rate" Equation (9.50) may be written

$$4Q_c^3 \rightarrow 4Q_c^3 (1 + 3F_I^2) \tag{9.51}$$

To avoid more than a factor of two degradation in basic failure probability

$$F_I < 0.58$$

Conclusions on Quad System Total Failure

The preceding subsections contributed elements to define the probability of total failure of a four-channel system during a mission:

$$Q_s = Q_{IC} + 4Q_c(Q_{NT} + Q_{FT} + Q_{GT} + Q_{ET}) + 3Q_{LF}^3 + 4Q_c^3(1 + 3F_I^2) \tag{9.52}$$

The latter term assumes reset capability for the first failure only. Written in alternate terms, (9.52) becomes

$$Q_s = \lambda_{IC} t_m + 4(\lambda_c t_m)^2 \left(\frac{D_T t_t}{t_m} + \frac{\lambda_{GT}}{\lambda_c} \frac{\Delta t_{PT}}{t_m} Q_{LGTF} + \frac{\lambda_{FT}}{\lambda_c} \frac{\Delta t_{PT}}{t_m} Q_{LFTF} + \frac{\lambda_{ET}}{\lambda_c} \frac{\Delta t_{PT}}{t_m} Q_{LTF} + \frac{\lambda_{LF}^2}{\lambda_c^2} \right) \quad (9.53)$$

$$+ \; 4 (\lambda_c t_m)^3 (1 + 3F_I^2)$$

The following conclusions are drawn from (9.53):

- Total test deficiency (D_T) will become a dominant influence on total failure probability when

  $$D_T > \lambda_c t_m^2 / t_t$$

  For a typical flight control channel at 10,000 hours life and a 2-hour mission, this corresponds to $D_T = 1.2 \times 10^{-7}$. This very small value indicates that very thorough testing is required on a regular basis for a quad system.

- The requirements on probabilities of latent test failure (Q_{LFTF}, Q_{LGTF}, and Q_{LTF}) are comparable to that for D_T. These are alleviated by "perfect" testing at selected time intervals such that $\Delta t_{PT} < t_t$.

- The probability of two latent failures occurring inflight and causing voter confusion can be significant if excessive failure rates exist in this category. Since equipment only tested on the ground is applicable here, such should be minimized in a quad system.

- False failure indications in flight (nuisance trips) as measured by F_I, the ratio of nuisance trips to valid failures, will significantly degrade flight safety for $F_I < 0.58$

MISSION RELIABILITY FOR A QUAD SYSTEM

The mission reliability, as measured by the probability of abort, for a quad system is basically determined by the probability of incurring the second failure. No abort is executed after the first failure because the remaining three channels can generally provide an acceptable safety level under usual circumstances. Indeed, if an abort were specified after only one failure, the mission reliability for a quad system would be worse than that of a triple system, which is in itself generally deficient in this area. The basic probability of the second failure during the mission for the quad system of Figure 9-6 is expressed as

$$Q_{sa} = 6Q_c^2 \tag{9.54}$$

where the factor of 6 accounts for the six identifiable channel pairs. To this must be added a number of other sources.

Failures Occurring During Ground Time

Failures occurring during ground time (t_g) in either the flight-essential equipment (λ_c) or the BITE used only for ground testing (λ_{GTE}) can cause aborts. The probability of this occurring in two channels is

$$Q = 6(\lambda_c + \lambda_{GTE})^2 t_g^2 \tag{9.55}$$

With λ_{GTE} probably smaller than λ_c and mission time and ground time comparable, (9.54) and (9.55) indicate that the in flight and ground abort rates are comparable. It is interesting to note from the results of the operational survey reported in Section III that this comparison is also reasonably valid for current aircraft.

Failures Detectable Only in Flight or Missed by Latent Ground Test Faults

This again is the problem of finding a failure in flight (probably shortly after takeoff), aborting the mission, attempting a repair which is confirmable only in flight, again attempting a mission, and so on for one or more attempts. Such difficulties are conceivable in equipment that is only tested in flight (λ_{FT}) or in the case of a latent failure in the ground test equipment (Q_{LGTF}). These elements for one channel are expressed by

$$Q = \left[\lambda_{FT} + \lambda_{ET}\, Q_{LGTF}\right]\left[t_m + t_g\right]\left[Q_{ur} + Q_{ur}^2 + Q_{ur}^3\, t \ldots\right] \tag{9.56}$$

As with the triple system, it is pessimistically assumed in (9.56) that the entire mission time is expended prior to each abort.

Because the probability given by (9.56) will be substantially less than the sources of either (9.54) or (9.55), this effect is negligible.

Nuisance Disengagements

False failure indications can increase the probability of a second "failure" being registered if the reset capability is restricted to just the first failure as previously discussed. This is considered as primarily a flight problem because ground tests can be repeated if necessary and because ground testing is under a more controlled operational environment which tends to avoid the off-nominal conditions that aggravate tolerance accumulations.

With the second failure indication required for abort and with the first failure resettable (and hence a true failure), the six pairs in the quad system contribute

$$Q = 6Q_c^2 + 12\, Q_c\, Q_{CN} \tag{9.57}$$

Equation (9.57) is a revised form of (9.54). Using F_I as defined by (9.50)

$$Q = 6Q_c^2\,(1 + 2F_I) = 6(\lambda_c t_m)^2\,(1 + 2F_I) \tag{9.58}$$

Latent Failures

Latent failures can in theory degrade mission reliability if their effect contributes a probability of total failure which is comparable to (9.55) or (9.58). In this way testing (through its resolution of latent failures) can have a beneficial effect on mission reliability. For this situation to prevail, however, the mission reliability would approach the total failure level (a single-channel quality), and such would involve an unsatisfactory (low) level of safety.

Conclusions on Quad System Mission Reliability

Collecting the major sources of abort probability derived in previous subsections

$$Q_{sa} = 6\,(\lambda_c t_m)^2\,(1 + 2F_I) + 6\,(\lambda_c + \lambda_{GTF})^2\, t_g^2 \tag{9.59}$$

The following is concluded from (9.59):

- BITE has a generally degrading effect on mission reliability because of an increase in system equipment.
- False failure indications in flight will significantly degrade mission reliability when the nuisance trips exceed 50 percent of the true failures, ($F_I > 0.5$).

ASSESSMENT OF CURRENT QUAD CHANNEL POTENTIAL

It is of interest to compute some of the pertinent requirements for a quad system hypothesized as one axis of a fly-by-wire system and make comparisons to related qualities of a conventional PFCS. Such can be done using portions of Equations (9.53) and (9.59). The following added assumptions will be made:

- No individually critical failures;
- No latent test failures ($Q_{LGTF} = Q_{LTF} = Q_{LFTF} = 0$);
- No latent failures occurring during the subject mission ($\lambda_{LF}=0$);
- Only the in-flight abort rate will be compared.

With these assumptions, the equations reduce to

$$Q_{s4} = 4(\lambda_c t_m)^2 D_T\, t_t/t_m + 4\,(\lambda_c t_m)^3\,(1 + 3F_I^2) \qquad (9.60)$$

$$Q_{sa4} = 6\,(\lambda_c t_m)^2\,(1 + 2F_I) \qquad (9.61)$$

The comparisons will be made for a 2-hour mission ($t_m = 2$), a 5000-hour system total time ($t_t = 5000$), and a unit channel failure rate of $= 3 \times 10^{-4}$ per hour. These assumptions produce

$$Q_{s4} = 3.6 \times 10^{-3}\, D_T + 8.64 \times 10^{-10}\,(1 + 3F_I^2) \qquad (9.62)$$

$$Q_{sa4} = 2.16 \times 10^{-6}\,(1 + 2F_I) \qquad (9.63)$$

Equations (9.62) and (9.63) are plotted as Figure 9-8. Also shown are comparable qualities for the current F-4 PFCS as obtained from Section III of this report and the total failure probability of commercial PFCS as obtained from Reference 9-2. The following may be concluded from Figure 9-8:

- The quad system provides abort probabilities at least an order of magnitude lower than the current F-4 system, even for high nuisance-trip rates. Hence the major deficiency of the triple system as indicated by Figure 9-4 is corrected.

- The quad system provides flight safety in accordance with the percentage of flight-essential equipment tested, this being between 99.9 and 99.99 percent to match current F-4 capabilities. For this range of values, the effect of nuisance-trip rate is minor.

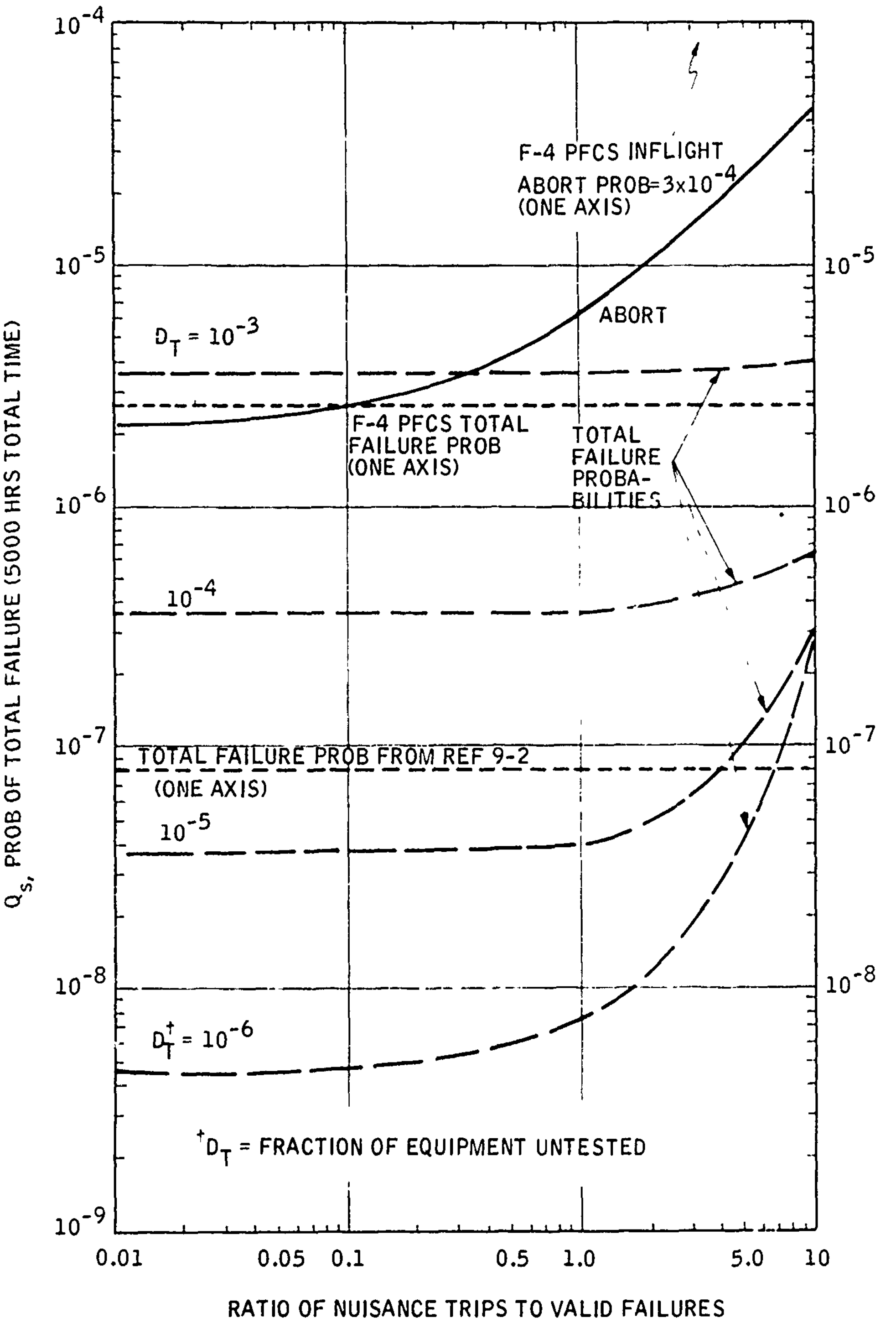


Figure 9-8. Quad Channel Performance Related to Test Deficiency and Nuisance Trips (Two-Hour Mission)

GENERAL CONCLUSIONS AND RECOMMENDATIONS ON BITE CRITERIA FOR FAIL-OPERATIVE PRIMARY FLIGHT CONTROLS

The study has produced a number of quantitative conclusions applicable to current configurations of redundant systems. Possible variations to the studied configurations are many, and the design criteria must be critically reviewed for each application. There are, however, certain qualitative generalities which have been deduced which are believed to be universally useful:

- All major failure sources must be considered when determining total and partial failure probabilities, including multiple-channel failures, single-point failures, latent failures in both prime equipment and the BITE, and nuisance disengagements.

- Requirements for test quality should reflect the objective of the tests (e. g., flight safety, maintenance, etc.) and should be imposed only on that equipment in the system which affects the objective.

- Test quality (i. e., thoroughness and elimination of latent failures) has a highly significant effect on total failure probability which becomes more critical as the number of channels of redundancy and the system life increase.

- Nuisance disengagements reduce flight safety and (except for the triple system) mission reliability. Their effect is more pronounced for the quad system than the triple.

- The reliability degradation due to latent failures can be avoided by testing at regular intervals (on ground or inflight). It is advantageous, however (particularly for the quad system), to test as thoroughly as possible in flight.

- BITE must be designed either with very low probability of having a latent failure (i. e., "fail-safe" qualities) or be tested at regular intervals by auxiliary equipment. This requirement becomes more vital with more channels of redundancy.

REFERENCES

9-1. Fostlethwaite, C. W., et al, A Guide to the Application of Built-In-Test to Navy Avionic Equipment, Publication 562-01-1-866 (AD837694), ARINC Research Corp., Annapolis, Maryland, 26 February 1968.

9-2. Rabinowitz, C. and Patti, S., Self-Contained Electronic Flight Control Systems, Final Report AD426327, Kaman Aircraft Corporation, Bloomfield, Connecticut, 1963.

9-3. Hooker, D. S., et al, Survivable Flight Control System Program - Simplex Actuator Package, AFFDL-TR-70-135, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, November 1970.

SECTION X

FLIGHT CONTROL ACTUATOR DESIGNS

This section reports the results of the flight control actuator survey and presents the associated design data catalog. The latter (Appendix V) is intended as a reference source for future aircraft flight control studies. It represents the contributions of several airframe and actuator manufacturers, including General Dynamics (Fort Worth and San Diego), Grumman, Lockheed (Georgia), LTV (Vought Aeronautics), McDonnell, and National Water Lift.

DATA CONTENT

Data was acquired in whole or part for the A-7, B-58, C-5A, C-141, F-4, F-111, F-14, F-15, and F-106 actuators. An attempt was made to cover the range of military fixed-wing aircraft. The following items are tabulated for each actuator:

(1) Manufacturer and model number;

(2) Hydraulic fluid type, pressure, and temperature range;

(3) A general description;

(4) Integrated series servo authority;

(5) Total piston areas;

(6) Numbers of pistons per actuator, actuators per surface, and hydraulic supplies per actuator;

(7) No load velocity;

(8) Loop gain;

(9) Output arm radius and surface deflection per unit stroke;

(10) Weight;

(11) Surface deflection range;

(12) Operating conditions determining no load velocity;

(13) Operating conditions determining maximum available surface hinge moment and its value;

(14) Maximum aerodynamic hinge moment and its origin;

(15) Result of one hydraulic-source failure;

(16) Static stiffness;

(17) Maximum no-load input horsepower;

(18) Horsepower output at maximum aerodynamic hinge moment;

(19) Fluid temperature rise at maximum hinge moment.

The nature of the above items and their significance are discussed further in Appendix V.

CONCLUSIONS

A study of the contents of Appendix V will disclose some fairly pertinent facts:

- Dual tandem actuators are the most popular configuration;
- Fully powered surfaces are practically universal;
- MIL-H-5606 continues to be the most popular fluid;
- 3000 psi is the supply pressure of all of the source aircraft;
- Significant achievements have been made in the matching of cylinder-type actuators to control-surface loads. The practice of achieving stiffness through an oversized force capability is less common, primarily due to the refinement of structures and actuator dynamics (e.g., hydrodynamic stabilization);
- Surface velocities, determined by low-q maneuver demands, have increased in proportion to the increases in aircraft performance;
- Electrically-controlled actuators are becoming more common;
- The use of constant-pressure, central hydraulic supplies continues to keep actuation deficiencies low;
- Redundant control surfaces are becoming more popular;
- Actuator response (as indicated by servo loop gains) has not been increased, except in unusual applications.

APPENDIX I

PERSONNEL INTERVIEWED AND ITEMS DISCUSSED FOR FLIGHT CONTROL SYSTEM SURVEY

The personnel contacted during this survey and a brief description of the items discussed are given in the following tables.

The personnel interviewed provided a wealth of technical information and prior experience in aircraft operation and maintenance. We received a cordial attitude from every contact and again offer our thanks for their cooperation and assistance.

Contact	Organization	Information Summary
Wright-Patterson AFB		
Jim Williams (ASNF) and Earl Parker (Engineers)	F-4 SPO, WPAFB	• PFCS problems and mods, stall warning, SAS failures, ADC changes
John Carpenter (Mechanic)	Flight Line, F-4 Maintenance, WPAFB	• PFCS maintenance, AFCS testing
John Etnyre (Technician)	Shop PFCS Maintenance, WPAFB	• Lubrication, rigging, TOs
John Smear (Technician)	AFCS Shop A7 Aircraft, WPAFB	• Electrical connectors, BITE indicators, component accessibility, nuisance trips under humidity, self-test benefits
Capt. Duane Zieg (Pilot - 200 hours on F-4, 1000 hours on F-105)	Fighter Test Group WPAFB	• Performance questionnaire, trim characteristics, coordination during target tracking, dive recovery
Hill AFB		
Lt. Col. J. Metcalf	Test pilot group - F-4 IRAM and Engineer Project Tests OOAMA, Hill AFB	• Had tour in SEA in F-4 • No detailed interview
Major G. F. Myers	Test pilot group - F-4 IRAM and Engineer Project Tests OOMA, Hill AFB	• 530 hours F-4, TAC • 1100 hours F-102, ADC • F-102 and F-106 handling versus F-4 • Spin potential impacts handling quality • Multimission workload

Contact	Organization	Information Summary
Major R. L. Benson	Test pilot group - F-4 IRAM and Engineer Project Tests OOMA, Hill AFB	• 400 hours F-4, SEA, photo recon. • In-flight refueling - probe geometry • SAS effects on response speed • Rudder usage at high angles of attack • Command signal limiter applicability
Capt. A. Sapyta	Test pilot group - F-4 IRAM and Engineer Project Tests OOMA, Hill AFB	• F-4 • Avionics reliability, aircraft simplicity • Spin recovery
Harold Haddock David Pratt	MMERA (Reliability) OOMA, Hill AFB	• Obtained USAF 66-1 data on F-4C, RF-4C, F-4D, F-4E, and RF-101G, H and F-101BF Aircraft. Information is from 6, 7, 15 - Log K261 Reports on Work Unit Codes 14 (Primary Flight Controls) 51, (Instruments), 52 (Automatic Flight Controls Systems).
Earl Brixey Norm Thedell	MMFTEA (F-4 Flight Controls) OOAMA-Hill AFB	• AFCS testing techniques • PFCS maintenance techniques • Gyro problems • Angle-of-attack system
Andrew Stokes Ren Mattson	MMEEA (Electronics) OOAMA Hill AFB	• AFCS testing • Gyro and amplifier failures • Air data computer and connector failures
Rick Holsman Major Parker Larry Raddle	OONEW-A (Engineering) OOAMA-Hill AFB	• Surface actuator specifications and problems • Aircraft safety, accident and incident information
	Ent AFB	
Lt. Col. Larry Graves, USAF	Chief, Flight Safety Division ADC Headquarters Ent AFB	• F-101B/F aircraft - accidents/problems • ADC aircraft loss of control incidents • Maintainability problems on fleet-wide basis

Contact	Organization	Information Summary
Roger Crewse	Civilian Chief, Analytical Division Office of Safety ADC Headquarters Ent AFB	• F-101B/F aircraft - accidents/ problems • ADC aircraft loss of control incidents • Maintainability problems on fleet-wide basis
	Tinker AFB	
Ray McInnes Fred Kraeer (MMPM)	OCAMA Data Services Tinker AFB, Oklahoma City	• Reliability data - USAF GG-1 data, 6 - Log K261-Failure data, 25 - Log K261-Maintenance manhours, 7 - Log K261-Abort summary, 15 - Log K261-Accident incident data
Al Bartels (MMNTA)	OCAMA Tech Services Hydraulic Actuators A-7	• Discussed original problems caused by high temperature of hydraulic system - seals - actuator response characteristics
John Anderson (MMNTA)	OCAMA Tech Services Hydraulic Actuators (F-106, F-111)	• Actuator integrated package design and operation were discussed. Average integrated package service life of 350 hours before wearout appears to be standard on both F-111 and F-106 which used a similar design
Glenn R. Dick (MMNTA)	OCAMA Tech Services Hydraulic Actuators (F-101, F-104)	• Discussed actuator frequency response characteristics. Fact that actuator wearout is caused by high frequency valve movements about the center position. Dither voltages (when used) increase that wearout factor.
Nobel Scott William Chandler (MMCTC)	OCAMA Tech Services Systems Managers Branch (A /D)	• PFCS - major problems are presently nonexistent • AFCS - no major problems, however, gyro wearout and a recently installed lateral accelerometer cause majority of complaints • Air Data Computer is not compatible with future AIMS requirements

Contact	Organization	Information Summary
William Kaiser J. Weems	OCAMA Engineering	• General PFCS and AFCS problem discussions. Suggested that the Navy might be able to provide more accurate reliability data based on A-7A, B, E flight time.
	Langley AFB	
Major James Sharp, USAF	Langley (TAC Headquarters) Aircraft Performance DRFA	• Primary responsibility is F-111, has some flight experience in F-111 • Recommended Nellis visit • AFCS in the F-111
Major Carl Young, USAF	Langley (TAC Headquarters) Aircraft Performance DRFA	• 2600 hours in F-100 SEA • 20 hours A-7D • Interview and questionnaire completed • Recommended Luke visit
Major Jerry Johnson, USAF	Langley (TAC Headquarters) Aircraft Performance DRFA	• Primary responsibility Harrier • 700 hours F-4 • Completed interview and questionnaire
Lt. Col. Dudley F. Nelson, USAF	Langley (TAC Headquarters) Chief Fighter Branch - Office of Safety	• TAC safety records • Maintenance support and effect on aircraft performance • BITE potential
	Tyndall AFB	
Major Tom Larson, USAF	Defense Weapons Center Tyndall AFB 4756 Combat Training Squad.	• 1900 hours F-106, primary duty instructor F-106 • Completed interview and questionnaire • F-106 spin problems, command signal limiter potential
Major W.G. Dolan, USAF	Defense Weapons Center Tyndall AFB 4750 Test Squad.	• 800 hours F-4, 800 hours F-102 • Completed interview and questionnaire

Contact	Organization	Information Summary
Major Richard Voehl, USAF	Defense Weapons Center Tyndall AFB 4750 Test Squad.	• 1500 hours F-100, 100 hours F-4 • General comments and discussion on ordnance delivery techniques air to ground
Major David Tucker, USAF	Defense Weapons Center, Tyndall, 4756 Combat Training Squadron	• 1900 hours F101B, 500 hours F105 (SEA), 500 hours F-102 • Command signal limiter benefits, ILS and fire control couplers • Coupler for use of guns to assist with gross maneuver • Completed interview and questionnaire
Fargo-Hector Field - Air National Guard		
Col. Alex MacDonald, Commanding Officer	189 FIS ANG Hector Field North Dakota	• Squadron operations with F-101B - just won "William Tell" meet at Tyndall AFB • Complexity and reliability of present and proposed aircraft systems
Major Steve Mallener, USAF	178 FIS ANG USAF Advisor	• Transition ANG to F-101B • Fire control coupler performance
1st Lt. T. M. Austin	178 FIS ANG Squad Pilot	• 80 hours F-101B, just checked out in A/C • Completed interview and questionnaire • Yaw damper and command signal limiter contributions
Capt. Gary Kaiser	178 FIS ANG Squad Pilot	• 105 hours F-101B • Completed interview and questionnaire • Fire control and ILS coupler
Major R. V. Hermanson	178 FIS ANG Standard Eval. Pilot	• 130 hours F-101B • Command signal limiter and coupler usage
Sgt. Rueben Richter	178 FIS ANG AFCS Shop Chief	• AFCS component accessibility, PFCS component accessibility • Tester size and usage criteria • Tech order organization and usage

Contact	Organization	Information Summary
Nellis AFB		
Sgt. Laurence Schauff	AFCS Shop Technician	• Requirement for ground hydraulic and electrical power • Requirement for large ground/ flight line testers • Connector and wiring experience and preferences • T.O. and tester preferences • Component accessibility, test point value and usage
Lt. Bob Scheel	Nellis AFB 430th TAC Fighter Squadron Avionics (F-111A)	• Organizational maintenance structure under TAC concept 66-31 • Avionics maintenance requirements on F-111A • Failure and time reporting to USAF 66-1 failure data system
Lt. Sucec Sgt. Buehler Sgt. Cooper Sgt. Hoffman	Nellis AFB F-111-Avionics Repair Shop	• Automatic testing procedures • Fault/Failure isolation procedures at LRU level • Device repair time and quality procedures
St. Thomas Bateman	Nellis AFB 430th TAC Fighter Squadron Flight Line - Avionics	• Primary and automatic flight control system operation • Use of BITE/Self Test on flight controls • Component accessibility • Failure rates, angle-of-attack system failures
Sgt. John Tinker M. Sgt. Eldridge	428th and 429th TAC Fighter Squadrons Flight Line - Avionics	• Fault isolation procedures and time requirements at the flight line • Rigging of manual flight controls • Aircraft aborts, accidents, incidents
M. Sgt. Monks	Nellis AFB 57th Fighter Weapons Wing (F-4C, D, E)	• Flight line and shop testing and troubleshooting • High-failure items • Temperature and dust effects on avionics • Major operational squawks

Contact	Organization	Information Summary
M. Sgt. Wilson Sgt. Spurlock	Nellis AFB 57th Fighter Weapons Wing (F-4C, D, E)	• Gyro failures - cause and possible maintenance actions • Flight line tester problems • Angle-of-attack transducer problems
Lt. Col. Calvin Broadway USAF	Nellis AFB 57th Wing 422 Fighter Weapons Squadron	• 300 hours F-111A and E - 422 FWS is an operational test and evaluation unit • 1000 hours F-105 SEA • Detailed interview conducted on F-111
Capt. Tom Sandford USAF	Nellis AFB 57 Wing 422 Fighter Weapons Squadron	• 280 hours F-111 A and F, 1200 hours F-100D SEA • Detailed interview conducted on F-111
Lt. Col. Allan Parks, USAF	Nellis AFB 57th Wing Commanding Officer, 422, FWS	• Commander of 422nd FWS • Discussed F-111 tactics and aircraft mission capability • Discussed general pilot acceptance,utilization of aircraft
Lt. Col. Cook USAF	Nellis AFB Operations Officer 422 FWS	• Operations Officer • General discussion on aircraft mission capability and pilot acceptance
Major Ray Carlson, USAF	Nellis AFB 57th Wing Assist. Operations Officer	• General discussion on the effectiveness of the aircraft as a weapons system. CAS essential
Lt. Col. Dale Nelson	Nellis AFB 414th Fighter Intercepter Squad Operations Officer	• Discussed Air Combat Maneuver Tactics in F-4
Lt. Col. Armstrong, USAF	Nellis AFB 414 FIS Commanding Officer	• General discussion F-4 capability

Contact	Organization	Information Summary
Capt. Sam Holmes, USAF	Nellis AFB 414 FIS, Squadron Pilot	• Completed exchange tour with US Navy in A-4 aircraft • A-4 outstanding aircraft for attack mission - high ordnance-carrying capability - very simple aircraft - no SAS required for mission accomplishment
Major R. M. Suter USAF	Nellis AFB 414 FIS ACM Expert	• Detailed interview conducted on the tactics used in air combat maneuvering of the F-4 aircraft • Abnormal maneuver pilot control techniques • PFCS and AFCS behavior affecting maneuver
Major James Sharp USAF	Langley AFB TAC Headquarters Directorate Fighter Requirements Project Officer	• Sharp was visiting from Langley to get flight time • 450 hours F-111 A and E • Detailed interview conducted F-111 • Escorted Bailey on a tour of the aircraft and gave a cockpit checkout
	Luke AFB	
Major Parks S. M. Sgt. Widing M. Sgt. Salsich	Luke AFB 58th Organizational Maintenance Squadron (A-7D)	• Obtained failure, maintenance, abort statistical data • Discussed internal failure reporting system (Maintenance concept at Luke not under TAC 66-31 concept)
M. Sgt. Larson Sgt. Buer	Luke AFB 58th Field Maintenance Squadron (A-7D)	• Flight line testing and troubleshooting • Use of self-test and flight line test equipment • Component accessibility and environmental problems
Sgt. McKay	Luke AFB 58th Avionic Maintenance Squadron (A-7D)	• Shop testing and troubleshooting procedures • Throwaway module concept • System packaging • Correlation of flight line/shop failure reports

Contact	Organization	Information Summar
Connie Granada Sal Richardson	Luke AFB LTV Technical Reps. (A-7D)	• AFCS self-test mechanization and effectiveness • PFCS rigging and actuator replacement • Hydraulic system mechanization and need for triple redundancy
Lt. Col. Charles McLarren, USAF	Luke AFB 58th TAC Fighter Training Wing, Test Detachment	• Detailed interview conducted covering AFCS and CAS in A-7 aircraft • CAS is essential to mission accomplishment • A-7D, A, B 800 hours - tour SEA A-7A U.S. Navy and SEA F105 • Escorted Bailey on a tour of the aircraft and gave a cockpit checkout
Major John Morhissey USAF	Luke AFB 58th TAC FTW Test Detachment	• Detailed interview conducted covering AFCS and CAS in A-7 aircraft • A7-D 440 hours, F-105 1450 hours SEA • Maintenance problems in control system
Major James O'Brien, USAF	Luke AFB 58th TAC FTW Test Detachment	• Detailed interview conducted covering CAS A-7 aircraft • Buffet margins broad, pedal shaker warning masked • CSL concept very interesting • A-7A, B and D 450 hours, A-4 300 hours Navy exchange, F-100 2200 hours SEA
Capt. Len Higgens, USAF	Luke AFB 58th TAC FTW Test Detachment	• General discussion on PFCS and CAS regarding frequency and types of discrepancies

Contact	Organization	Information Summary
	McClellan AFB	
Victor Feast (Engineer)	McClellan AFB F-111 AMA MMFTE	• F-111A Automatic Flight Controls - Mechanization - Depot testing concept - BITE/self test
Willy Reese (Technician)	McClellan AFB F-111 AMA MMFTE	• F-111A Automatic Flight Controls - Individual device reliability - Feel and trim computer - Angle-of-attack system
Luther Riley (Technician)	McClellan AFB F-111 AMA MMFTA	• F-111A Primary Flight Controls - Actuator wearout and time-change intervals - Periodic maintenance interval and tasks - Individual device failure rates
	Norton AFB	
Lt. Col. H. L. Seely	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• F-4, F-111 safety records • Discussion of flying hour data • Discussion of maintenance relationship to safety
Mr. Vernet Poupitch	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• AFM 66-1 and Norton Accident/Incident Data for A-7, F-101, F-4, and F-111 • Component failures and effects on flight safety • Detailed data on flight control system failures causing accidents
Lt. Col. E. Thrush	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• F-4 survivability, systems analysis (primarily missiles) • Thrush is involved in a new area of systems safety analysis which has to do with design criteria and systems study early in the development of an aircraft or missile

Contact	Organization	Information Summary
Lt. Col. Don Schmidt	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• Schmidt is in charge of the Systems Safety Engineering for aircraft. The C-5A is the first aircraft that has had this program applied. Schmidt and Thrush were very active in the F-15 systems finalization
Lt. Col. Nelson Allen	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• F-111, Project Safety Pilot • Discussion of the part flight controls have played in the accidents in this aircraft.
Lt. Col. Ray Ramsey	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• F-4, Project Safety Pilot • Discussion of 51 aircraft loss-of-control accidents
Major Bryant Heston	Inspector General Aircraft Flight Safety ICDSFR - Norton AFB	• F-105, Project Safety Pilot • Discussion of stability and controlability for delivery of ordnance. Also survivability characteristics of F-105 as compared to other aircraft in SEA

APPENDIX II

SURVEY DATA

The information on the following data sheets was obtained from the USAF Maintenance Reporting System defined by Air Force Manual 66-1 and commonly referred to as the 66-1 system.

The individual data sheets contain the time period for which the data is applicable. The data for each aircraft is given on two pages, the first page giving the Mean Time Between Failure (MTBF), the Mean Time Between Maintenance Action (MTBMA) and the Mean Time Between Abort (MTBA) and the second Maintenance Manhour per Flight Hour (MM/FH) values.

THIS SUMMARY IS FOR THE F-101B/F AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 14670.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER, 1977

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS
ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	27.2	7.0	232.1
PRIMARY FLIGHT CONTROLS	39.4	11.6	315.0
CONTROL STICK MECHANISM	341.2	142.4	14670.0
LATERAL CONTROL SYSTEM	153.5	39.0	2734.0
STABILATOR CONTROL SYSTEM	107.9	30.9	2734.0
RUDDER CONTROL SYSTEM	133.4	47.0	1333.7
SECONDARY FLIGHT CONTROLS	37.3	13.0	431.5
FLAP CONTROL SYSTEM	123.3	29.9	1333.7
SPEED BRAKE SYSTEM	117.5	47.9	733.5
TAKE OFF TRIM SYSTEM	1123.5	439.0	2445.0
AUTOMATIC FLIGHT CONTROLS	19.9	5.1	473.2
RATE GYROS (4)	431.5	257.4	7335.0
ACCELEROMETERS (3)	1047.9	439.0	14670.0
PILOT CONTROL PANELS	252.9	113.7	4890.0
COMPUTERS, SENSORS, SCHEDULERS	23.2	5.3	564.2
PFCS/AFCS ACTUATOR SUMMARY	32.9	37.5	1123.5
PFCS ACTUATORS ONLY (1)	473.2	157.7	14670.0
AFCS ACTUATORS ONLY (2)	315.0	431.5	7335.0
INTEGRATED ACTUATORS (3)	114.6	55.3	1467.0
AVERAGE ACTUATOR VALUES	595.9	225.7	5735.7
ELECTRICAL POWER SYSTEM	71.6	17.1	349.3
AC POWER SYSTEM	199.5	32.0	772.1
DC POWER SYSTEM	253.0	77.6	4890.0
AC GENERATORS	240.5	42.2	315.0
HYDRAULIC POWER SUPPLY	52.0	12.3	523.9
PRIMARY (#1) SUPPLY	172.6	53.9	2095.7
PRIMARY (#2) SUPPLY	115.5	37.7	978.0
GENERAL COMPONENTS	341.2	32.6	4890.0
AIRCRAFT INSTRUMENTS SYSTEM	18.4	3.2	473.2

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THIS SUMMARY IS FOR THE F-104A/F AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 14570.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER,1970

MM/FH = MAINTENANCE MANHOURS PER FLIGHT HOUR
SCHEDULED MM/FH = SCHEDULED PERIODIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHOP TASKS
TOTAL MM/FH = SUM OF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HOURS	UNSCHEDULED MM/FH HOURS	TOTAL MM/FH HOURS
MANUAL FLIGHT CONTROLS	.25746	.45849	.71595
PRIMARY FLIGHT CONTROLS	.16135	.28132	.44267
CONTROL STICK MECHANISM	.00273	.01091	.01363
LATERAL CONTROL SYSTEM	.06885	.09025	.15910
STABILATOR CONTROL SYSTEM	.05290	.11663	.16953
RUDDER CONTROL SYSTEM	.03688	.06353	.10041
SECONDARY FLIGHT CONTROLS	.09318	.15992	.25310
FLAP CONTROL SYSTEM	.06844	.10136	.16980
SPEED BRAKE SYSTEM	.02365	.05228	.07594
TAKE OFF TRIM SYSTEM	.00109	.00627	.00735
AUTOMATIC FLIGHT CONTROLS	.04090	1.12856	1.16946
RATE GYROS (4)	.00307	.03361	.03667
ACCELEROMETERS (3)	.00600	.01875	.02474
PILOT CONTROL PANELS	.00061	.03306	.03367
COMPUTERS,SENSORS,SCHEDULERS	.03122	1.04315	1.07437
PFCS/AFCS ACTUATOR SUMMARY	.04738	.11731	.16469
PFCS ACTUATORS ONLY (1)	.00927	.01956	.02883
AFCS ACTUATORS ONLY (2)	.00034	.01738	.01772
INTEGRATED ACTUATORS (3)	.03776	.08037	.11813
AVERAGE ACTUATOR VALUES	.00791	.01956	.02747
ELECTRICAL POWER SYSTEM	.02788	.24949	.27737
AC POWER SYSTEM	.00450	.04240	.04690
DC POWER SYSTEM	.00695	.06408	.07103
AC GENERATORS	.01050	.09925	.10975
HYDRAULIC POWER SUPPLY	.09598	.31247	.40845
PRIMARY (#1) SUPPLY	.01636	.05297	.06933
PRIMARY (#2) SUPPLY	.01322	.11663	.12986
GENERAL COMPONENTS	.05815	.07076	.12891
AIRCRAFT INSTRUMENTS SYSTEM	.01650	.57744	.59393

THIS SUMMARY IS FOR THE TOTAL F-4 AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 274946.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER,1970

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS

ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	42.0	5.1	676.1
PRIMARY FLIGHT CONTROLS	72.0	8.4	1447.1
CONTROL STICK MECHANISM	739.1	148.1	15274.0
LATERAL CONTROL SYSTEM	204.7	17.9	3273.2
STABILATOR CONTROL SYSTEM	205.0	22.9	4623.6
RUDDER CONTROL SYSTEM	359.9	76.7	6609.2
SECONDARY FLIGHT CONTROLS	100.5	14.5	1341.2
FLAP CONTROL SYSTEM	136.2	19.9	1447.1
SPEED BRAKE SYSTEM	474.0	71.2	22712.2
WING FOLD SYSTEM	1992.4	213.6	91646.7
AUTOMATIC FLIGHT CONTROLS	89.9	15.1	1205.9
RATE GYROS (3)	369.6	146.1	65736.5
ACCELEROMETERS (2)	803.9	569.2	45524.3
PILOT CONTROL PANELS	6066.6	1145.6	54967.2
COMPUTERS,SENSORS,SCHEDULERS	145.2	16.6	1536.0
PFCS/AFCS ACTUATOR SUMMARY	215.6	55.0	4623.6
PFCS ACTUATORS ONLY (6)	737.1	154.6	13747.[illegible]
AFCS ACTUATORS ONLY (2)	1303.1	366.6	66736.5
INTEGRATED ACTUATORS (2)	397.9	111.2	8331.7
AVERAGE ACTUATOR VALUES	2146.0	549.9	46236.1
ELECTRICAL POWER SYSTEM	66.6	21.1	723.5
AC POWER SYSTEM	446.3	130.7	2164.9
DC POWER SYSTEM	692.5	174.7	24995.1
AC GENERATORS	218.2	49.0	1636.6
HYDRAULIC POWER SUPPLY	156.3	16.9	1145.6
PRIMARY (#1) SUPPLY	662.2	61.5	6109.9
PRIMARY (#2) SUPPLY	605.6	82.0	4740.4
UTILITY (#3) SUPPLY	334.1	40.1	2115.0
AIRCRAFT INSTRUMENTS SYSTEM	55.5	13.5	1290.6

THIS SUMMARY IS FOR THE F-4 AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 274964.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER,1970

MM/FH = MAINTENANCE MANHOURS PER FLIGHT HOUR
SCHEDULED MM/FH = SCHEDULED PERIODIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHOP TASKS
TOTAL MM/FH = SUM OF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HOURS	UNSCHEDULED MM/FH HOURS	TOTAL MM/FH HOURS
MANUAL FLIGHT CONTROLS	.17290	.83705	1.00995
PRIMARY FLIGHT CONTROLS	.10661	.51304	.61965
CONTROL STICK MECHANISM	.00213	.02424	.02637
LATERAL CONTROL SYSTEM	.04439	.21503	.25942
STABILATOR CONTROL SYSTEM	.05088	.18754	.23841
RUDDER CONTROL SYSTEM	.00922	.07174	.08096
SECONDARY FLIGHT CONTROLS	.06625	.33763	.40388
FLAP CONTROL SYSTEM	.04677	.28266	.32943
SPEED BRAKE SYSTEM	.01499	.03457	.04956
WING FOLD SYSTEM	.00449	.02045	.02494
AUTOMATIC FLIGHT CONTROLS	.01752	.36358	.38110
RATE GYROS (3)	.00324	.03216	.03541
ACCELEROMETERS (2)	.00030	.01175	.01205
PILOT CONTROL PANELS	.00031	.00440	.00471
COMPUTERS,SENSORS,SCHEDULERS	.00831	.30426	.31257
PFCS/AFCS ACTUATOR SUMMARY	.01258	.11893	.13151
PFCS ACTUATORS ONLY (6)	.00540	.03754	.04294
AFCS ACTUATORS ONLY (2)	.00140	.01527	.01667
INTEGRATED ACTUATORS (2)	.00588	.06612	.07199
AVERAGE ACTUATOR VALUES	.00127	.01190	.01317
ELECTRICAL POWER SYSTEM	.01294	.39583	.40877
AC POWER SYSTEM	.00341	.04270	.04611
DC POWER SYSTEM	.00157	.10677	.10835
AC GENERATORS	.00337	.16713	.17050
HYDRAULIC POWER SUPPLY	.03906	.28350	.32256
PRIMARY (#1) SUPPLY	.00908	.06288	.07197
PRIMARY (#2) SUPPLY	.00809	.05544	.06353
UTILITY (#3) SUPPLY	.01963	.14338	.16301
AIRCRAFT INSTRUMENTS SYSTEM	.01007	.34393	.35400

THIS SUMMARY IS FOR THE F-111A AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 14149.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31,DECEMBER,1969

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS
ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	30.4	5.6	404.3
PRIMARY FLIGHT CONTROLS	76.9	13.6	1179.1
CONTROL STICK MECHANISM	1286.3	54.4	14149.0
LATERAL CONTROL SYSTEM	150.5	41.9	1768.6
STABILATOR CONTROL SYSTEM	208.1	43.1	7074.5
RUDDER CONTROL SYSTEM	1286.3	140.1	7074.5
SECONDARY FLIGHT CONTROLS	54.4	9.1	744.7
FLAP CONTROL SYSTEM	181.4	29.1	2829.8
SLAT SYSTEM	162.6	30.7	2358.2
WING SWEEP SYSTEM	2358.2	85.8	4716.3
OTHER GENERAL COMPONENTS	159.0	31.7	2829.8
AUTOMATIC FLIGHT CONTROLS	46.1	20.7	487.9
RATE GYROS (9)	1768.6	589.5	7074.5
ACCELEROMETERS (6)	3537.3	1179.1	14149.0
PILOT CONTROL PANELS	14149.0	1572.1	14149.0
COMPUTERS	48.1	22.2	566.0
PFCS/AFCS ACTUATOR SUMMARY	142.9	35.9	2829.8
PFCS SURFACE ACTUATORS (6)	179.1	41.6	3537.3
AFCS ACTUATORS (3)	707.5	262.0	14149.0
AVERAGE ACTUATOR VALUES	1286.3	314.4	25725.5
ELECTRICAL POWER SYSTEM	141.5	23.5	1010.6
AC POWER SYSTEM	176.9	27.9	1088.4
DC POWER SYSTEM	744.7	138.7	14149.0
HYDRAULIC POWER SUPPLY	62.1	18.0	673.8
PRIMARY (#1) SUPPLY	133.5	36.8	1286.3
PRIMARY (#2) SUPPLY	116.0	35.4	1414.9
AIRCRAFT INSTRUMENT SYSTEM	34.0	15.9	1414.9

THIS SUMMARY IS FØR THE F-111A AIRCRAFT

THE FLIGHT HØURS ASSØCIATED WITH THIS REPØRT ARE 14149.
THE FLIGHT HØURS WERE ACCUMULATED IN THE 6 MØ. PRECEDING 31,DECEMBER,1969

MM/FH = MAINTENANCE MANHØURS PER FLIGHT HØUR
SCHEDULED MM/FH = SCHEDULED PERIØDIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHØP TASKS
TØTAL MM/FH = SUM ØF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HØURS	UNSCHEDULED MM/FH HØURS	TØTAL MM/FH HØURS
MANUAL FLIGHT CØNTRØLS	.04255	1.47692	1.51947
PRIMARY FLIGHT CØNTRØLS	.00954	.47678	.48632
CØNTRØL STICK MECHANISM	.00021	.04636	.04658
LATERAL CØNTRØL SYSTEM	.00382	.22320	.22701
STABILATØR CØNTRØL SYSTEM	.00382	.15711	.16093
RUDDER CØNTRØL SYSTEM	.00170	.05166	.05336
SECØNDARY FLIGHT CØNTRØLS	.02975	.83858	.86833
FLAP CØNTRØL SYSTEM	.00219	.28030	.28249
SLAT SYSTEM	.01823	.30949	.32773
WING SWEEP SYSTEM	.00905	.11393	.12298
ØTHER GENERAL CØMPØNENTS	.00035	.12616	.12651
AUTØMATIC FLIGHT CØNTRØLS	.00021	.32971	.32992
RATE GYRØS (9)	.00000	.00968	.00968
ACCELERØMETERS (6)	.00000	.00615	.00615
PILØT CØNTRØL PANELS	.00000	.00283	.00283
CØMPUTERS	.00021	.30596	.30617
PFCS/AFCS ACTUATØR SUMMARY	.00636	.25309	.25945
PFCS SURFACE ACTUATØRS (6)	.00636	.22072	.22708
AFCS ACTUATØRS (3)	.00000	.03237	.03237
AVERAGE ACTUATØR VALUES	.00071	.02806	.02877
ELECTRICAL PØWER SYSTEM	.01128	.22072	.23200
AC PØWER SYSTEM	.01018	.17146	.18164
DC PØWER SYSTEM	.00106	.04566	.04672
HYDRAULIC PØWER SUPPLY	.01074	.33953	.35027
PRIMARY (#1) SUPPLY	.00551	.17499	.18051
PRIMARY (#2) SUPPLY	.00523	.15697	.16220
AIRCRAFT INSTRUMENT SYSTEM	.00057	.31712	.31769

THIS SUMMARY IS FOR THE A-7D AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 7358.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, DEC.,1970

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS

ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	65.1	11.1	459.9
PRIMARY FLIGHT CONTROLS	105.1	44.6	566.0
CONTROL STICK MECHANISM	7358.0	3679.0	7358.0
LATERAL CONTROL SYSTEM	175.2	131.4	735.8
STABILATOR CONTROL SYSTEM	306.6	86.6	3679.0
RUDDER CONTROL SYSTEM	1839.5	334.5	7358.0
SECONDARY FLIGHT CONTROLS	193.6	26.8	2452.7
FLAP CONTROL SYSTEM	262.8	38.7	7358.0
SPEED BRAKE SYSTEM	817.6	147.2	3679.0
SPOILER SYSTEM	7358.0	210.2	7358.0
AUTOMATIC FLIGHT CONTROLS	86.6	14.6	7358.0
RATE GYROS (6)	1226.3	817.6	7358.0
ACCELEROMETERS (4)	668.9	367.3	7358.0
PILOT CONTROL PANELS	7358.0	2452.7	7358.0
COMPUTERS	229.9	33.3	7358.0
AFCS SERVOS (3)	179.5	41.6	7358.0
PFCS/AFCS ACTUATOR SUMMARY	367.9	60.3	3679.0
PFCS SURFACE ACTUATORS (4)	613.2	87.6	3679.0
AFCS ACTUATORS (3)	919.8	193.6	7358.0
AVERAGE ACTUATOR VALUES	1098.2	432.8	25372.4
ELECTRICAL POWER SYSTEM	102.2	25.5	2452.7
HYDRAULIC POWER SUPPLY	70.1	16.3	525.6
PC NO. 1 SUPPLY	179.5	68.6	3679.0
PC NO. 2 SUPPLY	103.6	27.8	613.2
PC NO. 3 SUPPLY	735.8	109.8	7358.0
RAM AIR TURBINE POWER	525.6	124.7	7358.0
AIRCRAFT INSTRUMENT SYSTEM	57.5	19.5	7358.0

THIS SUMMARY IS FØR THE A-7D AIRCRAFT

THE FLIGHT HØURS ASSØCIATED WITH THIS REPØRT ARE 7358.
THE FLIGHT HØURS WERE ACCUMULATED IN THE 6 MØ. PRECEDING 31, DEC.,1970

MM/FH = MAINTENANCE MANHØURS PER FLIGHT HØUR
SCHEDULED MM/FH = SCHEDULED PERIØDIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHØP TASKS
TØTAL MM/FH = SUM ØF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HØURS	UNSCHEDULED MM/FH HØURS	TØTAL MM/FH HØURS
MANUAL FLIGHT CØNTRØLS	.04662	.47880	.52541
PRIMARY FLIGHT CØNTRØLS	.03248	.29655	.32903
CØNTRØL STICK MECHANISM	.00000	.00122	.00122
LATERAL CØNTRØL SYSTEM	.01984	.14596	.16581
STABILATØR CØNTRØL SYSTEM	.01114	.11185	.12300
RUDDER CØNTRØL SYSTEM	.00149	.03615	.03765
SECØNDARY FLIGHT CØNTRØLS	.01345	.13536	.14882
FLAP CØNTRØL SYSTEM	.01237	.11267	.12503
SPEED BRAKE SYSTEM	.00095	.04267	.04363
SPØILER SYSTEM	.00014	.00720	.00734
AUTØMATIC FLIGHT CØNTRØLS	.00394	.42552	.42946
RATE GYRØS (6)	.00000	.00584	.00584
ACCELERØMETERS (4)	.00000	.01264	.01264
PILØT CØNTRØL PANELS	.00000	.00027	.00027
CØMPUTERS	.00000	.19054	.19054
AFCS SERVØS (3)	.00394	.15303	.15697
PFCS/AFCS ACTUATØR SUMMARY	.01114	.08657	.09772
PFCS SURFACE ACTUATØRS (4)	.01114	.05681	.06795
AFCS ACTUATØRS (3)	.00000	.02976	.02976
AVERAGE ACTUATØR VALUES	.00159	.01237	.01396
ELECTRICAL PØWER SYSTEM	.00666	.24395	.25061
HYDRAULIC PØWER SUPPLY	.02569	.30796	.33365
PC NØ. 1 SUPPLY	.00489	.07271	.07760
PC NØ. 2 SUPPLY	.01509	.19802	.21310
PC NØ. 3 SUPPLY	.00421	.02963	.03384
RAM AIR TURBINE PØWER	.00462	.03384	.03846
AIRCRAFT INSTRUMENT SYSTEM	.00231	.22275	.22506

THIS SUMMARY IS FOR THE F-4C AIRCRAFT .

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 44180.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER, 1970

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS
ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	28.6	3.5	371.3
PRIMARY FLIGHT CONTROLS	52.6	6.1	712.6
CONTROL STICK MECHANISM	748.8	109.4	14726.7
LATERAL CONTROL SYSTEM	127.0	10.9	1523.4
STABILATOR CONTROL SYSTEM	164.2	22.3	2325.3
RUDDER CONTROL SYSTEM	269.4	52.3	4016.4
SECONDARY FLIGHT CONTROLS	62.5	8.1	775.1
FLAP CONTROL SYSTEM	86.0	11.2	803.3
SPEED BRAKE SYSTEM	285.0	41.1	44180.0
WING FOLD SYSTEM	1162.6	107.0	44180.0
AUTOMATIC FLIGHT CONTROLS	73.8	11.0	1194.1
RATE GYROS (3)	339.8	188.0	7363.3
ACCELEROMETERS (2)	631.1	412.9	44180.0
PILOT CONTROL PANELS	4908.9	496.4	44180.0
COMPUTERS, SENSORS, SCHEDULERS	113.3	12.3	1523.4
MFCS/AFCS ACTUATOR SUMMARY	175.3	45.9	3155.7
MFCS ACTUATORS ONLY (6)	455.5	106.7	7363.3
AFCS ACTUATORS ONLY (2)	1920.9	424.8	44180.0
INTEGRATED ACTUATORS (2)	334.7	99.5	6311.4
AVERAGE ACTUATOR VALUES	1767.2	460.2	31557.1
ELECTRICAL POWER SYSTEM	55.0	13.3	532.3
AC POWER SYSTEM	166.1	39.4	866.3
DC POWER SYSTEM	460.2	107.8	44180.0
[illegible]	311.1	46.5	2325.3
HYDRAULIC POWER SUPPLY	107.8	10.8	559.2
PRIMARY (#1) SUPPLY	460.2	49.4	3681.7
PRIMARY (#2) SUPPLY	420.8	48.0	2209.0
UTILITY (#3) SUPPLY	238.8	22.2	1027.4
AIRCRAFT INSTRUMENTS SYSTEM	28.0	9.0	428.9

THIS SUMMARY IS FØR THE F-4C AIRCRAFT

THE FLIGHT HØURS ASSØCIATED WITH THIS REPØRT ARE 44130.
THE FLIGHT HØURS WERE ACCUMULATED IN THE 6 MØ. PRECEDING 31, ØCTØBER, 197

MM/FH = MAINTENANCE MANHØURS PER FLIGHT HØUR
SCHEDULED MM/FH = SCHEDULED PERIØDIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHØP TASKS
TØTAL MM/FH = SUM ØF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HØURS	UNSCHEDULED MM/FH HØURS	TØTAL MM/FH HØURS
MANUAL FLIGHT CØNTRØLS	.23334	1.19176	1.42510
PRIMARY FLIGHT CØNTRØLS	.13402	.69092	.82494
CØNTRØL STICK MECHANISM	.00124	.03572	.03696
LATERAL CØNTRØL SYSTEM	.05656	.30704	.36360
STABILATØR CØNTRØL SYSTEM	.06720	.24814	.31534
RUDDER CØNTRØL SYSTEM	.00901	.09991	.10892
SECØNDARY FLIGHT CØNTRØLS	.09934	.50045	.59979
FLAP CØNTRØL SYSTEM	.07173	.40387	.47560
SPEED BRAKE SYSTEM	.02302	.05416	.07718
WING FØLD SYSTEM	.00459	.04242	.04701
AUTØMATIC FLIGHT CØNTRØLS	.00704	.42154	.42858
RATE GYRØS (3)	.00002	.02289	.02291
ACCELERØMETERS (2)	.00045	.01059	.01104
PILØT CØNTRØL PANELS	.00034	.00625	.00659
CØMPUTERS,SENSØRS,SCHEDULERS	.00622	.39414	.40036
PFCS/AFCS ACTUATØR SUMMARY	.01098	.18757	.19855
PFCS ACTUATØRS ØNLY (6)	.00600	.04613	.05213
AFCS ACTUATØRS ØNLY (2)	.00084	.01086	.01170
INTEGRATED ACTUATØRS (2)	.00414	.13058	.13472
AVERAGE ACTUATØR VALUES	.00111	.01376	.01487
ELECTRICAL PØWER SYSTEM	.01806	.54365	.56171
AC PØWER SYSTEM	.00774	.10792	.11566
DC PØWER SYSTEM	.00344	.10290	.10634
AC GENERATØRS	.00326	.22334	.22660
HYDRAULIC PØWER SUPPLY	.04371	.37709	.42080
PRIMARY (#1) SUPPLY	.01003	.06706	.07709
PRIMARY (#2) SUPPLY	.00887	.07529	.08416
UTILITY (#3) SUPPLY	.02028	.21281	.23309
AIRCRAFT INSTRUMENTS SYSTEM	.00994	.51478	.52472

THIS SUMMARY IS FOR THE F-4D AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 85606.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MØ. PRECEDING 31, ØCTØBER, 197

MM/FH = MAINTENANCE MANHØURS PER FLIGHT HØUR
SCHEDULED MM/FH = SCHEDULED PERIØDIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHØP TASKS
TOTAL MM/FH = SUM ØF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HØURS	UNSCHEDULED MM/FH HØURS	TØTAL MM/FH HØURS
MANUAL FLIGHT CØNTRØLS	.22143	.94672	1.16815
PRIMARY FLIGHT CØNTRØLS	.13345	.53289	.66634
CONTRØL STICK MECHANISM	.00241	.02231	.02472
LATERAL CØNTRØL SYSTEM	.06545	.25642	.32187
STABILATØR CØNTRØL SYSTEM	.05495	.17681	.23176
RUDDER CØNTRØL SYSTEM	.01064	.07735	.08800
SECONDARY FLIGHT CØNTRØLS	.08784	.41221	.50006
FLAP CØNTRØL SYSTEM	.05926	.35549	.41475
SPEED BRAKE SYSTEM	.02028	.03481	.05509
WING FØLD SYSTEM	.00831	.02191	.03022
AUTØMATIC FLIGHT CØNTRØLS	.02892	.35881	.38773
RATE GYRØS (3)	.01453	.03495	.04948
ACCELERØMETERS (2)	.00027	.01109	.01135
PILØT CØNTRØL PANELS	.00051	.00519	.00570
CØMPUTERS,SENSØRS,SCHEDULERS	.01361	.30758	.32119
PFCS/AFCS ACTUATØR SUMMARY	.01652	.12702	.14354
PFCS ACTUATØRS ØNLY (6)	.00680	.04663	.05343
AFCS ACTUATØRS ØNLY (2)	.00201	.02002	.02203
INTEGRATED ACTUATØRS (2)	.00771	.06037	.06808
AVERAGE ACTUATØR VALUES	.00165	.01271	.01436
ELECTRICAL PØWER SYSTEM	.01177	.42951	.44129
AC PØWER SYSTEM	.00078	.03481	.03559
DC PØWER SYSTEM	.00207	.12338	.12545
AC GENERATØRS	.00389	.19376	.19765
HYDRAULIC PØWER SUPPLY	.05758	.33480	.39238
PRIMARY (#1) SUPPLY	.01322	.07581	.08904
PRIMARY (#2) SUPPLY	.01204	.06507	.07711
UTILITY (#3) SUPPLY	.02932	.16584	.19516
AIRCRAFT INSTRUMENTS SYSTEM	.00739	.39962	.40702

THIS SUMMARY IS FOR THE F-4D AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 55606
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER, 197[illegible]

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS

ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	40.0	4.0	[illegible]
PRIMARY FLIGHT CONTROLS	70.1	7.2	1712.1
CONTROL STICK MECHANISM	713.4	145.1	21401.5
LATERAL CONTROL SYSTEM	190.2	14.0	[illegible]
STABILATOR CONTROL SYSTEM	208.8	21.6	[illegible]
RUDDER CONTROL SYSTEM	355.2	66.9	[illegible]
SECONDARY FLIGHT CONTROLS	94.0	12.0	[illegible]
FLAP CONTROL SYSTEM	135.0	17.0	[illegible]
SPEED BRAKE SYSTEM	411.8	54.9	[illegible]
WING FOLD SYSTEM	1240.7	152.9	[illegible]
AUTOMATIC FLIGHT CONTROLS	88.8	15.0	[illegible]
RATE GYROS (3)	303.6	95.3	[illegible]
ACCELEROMETERS (2)	1007.1	713.4	[illegible]
PILOT CONTROL PANELS	7133.8	1476.0	[illegible]
COMPUTERS, SENSORS, SCHEDULERS	146.3	16.4	[illegible]
PFCS/AFCS ACTUATOR SUMMARY	211.4	46.0	[illegible]
PFCS ACTUATORS ONLY (6)	785.4	135.0	[illegible]
AFCS ACTUATORS ONLY (2)	1019.1	262.5	[illegible]
INTEGRATED ACTUATORS (2)	403.8	92.8	[illegible]
AVERAGE ACTUATOR VALUES	2088.0	480.2	[illegible]
ELECTRICAL POWER SYSTEM	79.0	19.3	[illegible]
AC POWER SYSTEM	602.9	205.8	[illegible]
DC POWER SYSTEM	660.8	153.7	[illegible]
AC GENERATORS	155.6	36.2	[illegible]
HYDRAULIC POWER SUPPLY	150.7	17.3	1747.1
PRIMARY (#1) SUPPLY	653.5	[illegible]	[illegible]
PRIMARY (#2) SUPPLY	624.9	[illegible]	[illegible]
UTILITY (#3) SUPPLY	319.4	59.1	[illegible]
AIRCRAFT INSTRUMENTS SYSTEM	31.0	12.0	1551.9

THIS SUMMARY IS FOR THE F-4E AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 92367
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MØ. PRECEDING 31, OCTOBER, 1970

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTIØN
MTBA = MEAN TIME BETWEEN ABØRTS
ALL HØURS IN THIS SUMMARY ARE FLIGHT HØURS

	MTBF HØURS	MTBMA HØURS	MTBA HØURS
MANUAL FLIGHT CØNTRØLS	63.2	8.0	1061.7
PRIMARY FLIGHT CØNTRØLS	94.0	10.3	1924.3
CONTROL STICK MECHANISM	952.2	215.8	18473.4
LATERAL CONTRØL SYSTEM	316.3	28.0	4618.3
STABILATOR CONTROL SYSTEM	244.4	21.0	6597.6
RUDDER CONTROL SYSTEM	427.6	106.8	10263.0
SECONDARY FLIGHT CØNTRØLS	187.7	26.6	2366.4
FLAP CONTROL SYSTEM	249.0	38.2	2496.4
SPEED BRAKE SYSTEM	824.7	131.6	46183.5
WING FOLD SYSTEM	10263.0	839.7	92367.0
AUTOMATIC FLIGHT CONTROLS	114.0	21.0	2006.0
RATE GYROS (3)	488.7	173.6	8397.0
ACCELEROMETERS (2)	757.1	546.6	92367.0
PILOT CONTROL PANELS	18473.4	1565.5	92367.0
COMPUTERS, SENSØRS, SCHEDULERS	187.0	25.4	2716.7
MFCS/AFCS ACTUATOR SUMMARY	256.6	72.8	4618.3
MFCS ACTUATORS ØNLY (6)	888.1	194.9	10263.0
AFCS ACTUATORS ØNLY (2)	1338.7	450.6	92367.0
INTEGRATED ACTUATORS (2)	493.9	156.6	9236.7
AVERAGE ACTUATOR VALUES	2565.8	727.3	46183.5
ELECTRICAL POWER SYSTEM	157.4	34.9	1026.3
AC POWER SYSTEM	1126.4	504.7	8397.0
DC POWER SYSTEM	1004.0	251.7	13195.3
AC GENERATORS	303.8	72.1	1539.4
HYDRAULIC POWER SUPPLY	218.4	30.1	1399.5
PRIMARY (#1) SUPPLY	993.2	121.4	6157.8
PRIMARY (#2) SUPPLY	776.2	146.8	4398.4
UTILITY (#3) SUPPLY	488.7	65.6	3185.1
AIRCRAFT INSTRUMENTS SYSTEM	53.1	19.0	3298.8

THIS SUMMARY IS FOR THE F-4E AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 92367.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER, 197

MM/FH = MAINTENANCE MANHOURS PER FLIGHT HOUR
SCHEDULED MM/FH = SCHEDULED PERIODIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHOP TASKS
TOTAL MM/FH = SUM OF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HOURS	UNSCHEDULED MM/FH HOURS	TOTAL MM/FH HOURS
MANUAL FLIGHT CONTROLS	.10624	.61233	.71857
PRIMARY FLIGHT CONTROLS	.07470	.46203	.53673
CONTROL STICK MECHANISM	.00118	.01936	.02054
LATERAL CONTROL SYSTEM	.02289	.15641	.17930
STABILATOR CONTROL SYSTEM	.04346	.18559	.22904
RUDDER CONTROL SYSTEM	.00718	.05759	.06476
SECONDARY FLIGHT CONTROLS	.03155	.19354	.22509
FLAP CONTROL SYSTEM	.02431	.15666	.18096
SPEED BRAKE SYSTEM	.00613	.02708	.03320
WING FOLD SYSTEM	.00112	.00981	.01092
AUTOMATIC FLIGHT CONTROLS	.01590	.29376	.30967
RATE GYROS (3)	.01082	.02579	.03660
ACCELEROMETERS (2)	.00032	.01168	.01201
PILOT CONTROL PANELS	.00016	.00322	.00338
COMPUTERS,SENSORS,SCHEDULERS	.00460	.25308	.25768
PFCS/AFCS ACTUATOR SUMMARY	.00921	.08739	.09660
PFCS ACTUATORS ONLY (6)	.00357	.03196	.03553
AFCS ACTUATORS ONLY (2)	.00120	.01246	.01366
INTEGRATED ACTUATORS (2)	.00444	.04297	.04741
AVERAGE ACTUATOR VALUES	.00092	.00875	.00967
ELECTRICAL POWER SYSTEM	.00520	.28036	.28556
AC POWER SYSTEM	.00010	.01800	.01810
DC POWER SYSTEM	.00062	.10168	.10230
AC GENERATORS	.00146	.11697	.11843
HYDRAULIC POWER SUPPLY	.01894	.20068	.21961
PRIMARY (#1) SUPPLY	.00537	.04842	.05379
PRIMARY (#2) SUPPLY	.00358	.03713	.04072
UTILITY (#3) SUPPLY	.00899	.10365	.11264
AIRCRAFT INSTRUMENTS SYSTEM	.00848	.22872	.23720

THIS SUMMARY IS FOR THE RF-4C AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 52811
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER, 1970

MTBF = MEAN TIME BETWEEN FAILURE
MTBMA = MEAN TIME BETWEEN MAINTENANCE ACTION
MTBA = MEAN TIME BETWEEN ABORTS
ALL HOURS IN THIS SUMMARY ARE FLIGHT HOURS

	MTBF HOURS	MTBMA HOURS	MTBA HOURS
MANUAL FLIGHT CONTROLS	37.6	6.8	668.5
PRIMARY FLIGHT CONTROLS	68.1	11.4	1760.4
CONTROL STICK MECHANISM	550.1	121.4	8801.8
LATERAL CONTROL SYSTEM	208.7	27.1	8801.8
STABILATOR CONTROL SYSTEM	186.0	32.1	4801.0
RUDDER CONTROL SYSTEM	369.3	88.3	7544.4
SECONDARY FLIGHT CONTROLS	84.2	16.7	1077.8
FLAP CONTROL SYSTEM	105.6	21.8	1173.6
SPEED BRAKE SYSTEM	503.0	100.4	17603.7
WING FOLD SYSTEM	2400.5	258.9	52811.0
AUTOMATIC FLIGHT CONTROLS	77.0	13.0	1015.6
RATE GYROS (3)	369.3	276.5	8801.8
ACCELEROMETERS (2)	812.5	607.0	52811.0
PILOT CONTROL PANELS	6601.4	1553.3	26405.5
COMPUTERS, SENSORS, SCHEDULERS	124.6	18.9	1173.6
MFCS/AFCS ACTUATOR SUMMARY	204.7	58.0	6601.4
MFCS ACTUATORS ONLY (6)	838.3	207.9	52811.0
AFCS ACTUATORS ONLY (2)	1508.9	394.1	52811.0
INTEGRATED ACTUATORS (2)	330.1	102.1	8801.8
AVERAGE ACTUATOR VALUES	2031.2	580.3	66013.7
ELECTRICAL POWER SYSTEM	76.3	19.9	652.0
AC POWER SYSTEM	419.1	135.1	2296.1
DC POWER SYSTEM	621.3	220.0	52811.0
AC GENERATORS	200.0	46.1	1427.3
HYDRAULIC POWER SUPPLY	147.5	22.1	1148.1
PRIMARY (#1) SUPPLY	636.3	102.9	3772.2
PRIMARY (#2) SUPPLY	567.9	69.2	8801.8
UTILITY (#3) SUPPLY	291.8	41.6	2031.2
AIRCRAFT INSTRUMENTS SYSTEM	34.1	15.0	2112.4

THIS SUMMARY IS FOR THE RF-4C AIRCRAFT

THE FLIGHT HOURS ASSOCIATED WITH THIS REPORT ARE 52811.
THE FLIGHT HOURS WERE ACCUMULATED IN THE 6 MO. PRECEDING 31, OCTOBER,1970

MM/FH = MAINTENANCE MANHOURS PER FLIGHT HOUR
SCHEDULED MM/FH = SCHEDULED PERIODIC MAINTENANCE TASKS
UNSCHEDULED MM/FH = UNSCHEDULED FLIGHT LINE AND SHOP TASKS
TOTAL MM/FH = SUM OF SCHEDULED AND UNSCHEDULED TASKS

	SCHEDULED MM/FH HOURS	UNSCHEDULED MM/FH HOURS	TOTAL MM/FH HOURS
MANUAL FLIGHT CONTROLS	.16027	.75556	.91583
PRIMARY FLIGHT CONTROLS	.09600	.42126	.51726
CONTROL STICK MECHANISM	.00411	.02630	.03041
LATERAL CONTROL SYSTEM	.03766	.17349	.21115
STABILATOR CONTROL SYSTEM	.04359	.15764	.20123
RUDDER CONTROL SYSTEM	.01064	.06383	.07447
SECONDARY FLIGHT CONTROLS	.06427	.33281	.39708
FLAP CONTROL SYSTEM	.04495	.28360	.32855
SPEED BRAKE SYSTEM	.01521	.03090	.04611
WING FOLD SYSTEM	.00411	.01831	.02242
AUTOMATIC FLIGHT CONTROLS	.01060	.47089	.48149
RATE GYROS (3)	.00044	.02573	.02617
ACCELEROMETERS (2)	.00017	.01394	.01411
PILOT CONTROL PANELS	.00019	.00365	.00384
COMPUTERS,SENSORS,SCHEDULERS	.00793	.31321	.32115
PFCS/AFCS ACTUATOR SUMMARY	.01396	.10354	.11749
PFCS ACTUATORS ONLY (6)	.00585	.02535	.03121
AFCS ACTUATORS ONLY (2)	.00123	.01619	.01742
INTEGRATED ACTUATORS (2)	.00687	.06199	.06887
AVERAGE ACTUATOR VALUES	.00140	.01036	.01176
ELECTRICAL POWER SYSTEM	.02407	.41953	.44360
AC POWER SYSTEM	.00983	.04414	.05397
DC POWER SYSTEM	.00089	.09199	.09288
AC GENERATORS	.00595	.16493	.17087
HYDRAULIC POWER SUPPLY	.04037	.26690	.30727
PRIMARY (#1) SUPPLY	.00807	.06374	.07180
PRIMARY (#2) SUPPLY	.00892	.05523	.06415
UTILITY (#3) SUPPLY	.02223	.14438	.16661
AIRCRAFT INSTRUMENTS SYSTEM	.01733	.31223	.32955

APPENDIX III

RECOMMENDED REVISIONS TO MIL-F-9490C (USAF)

The subject specification was reviewed subsequent to completion of the PFCS Design Criteria Study so that the impact of study results could be assessed. Suggested changes and additions to the Specification were determined and documented in the following paragraphs. Each is numbered according to the respective paragraphs of the Specification.

1.2 Classification

(a) Current deficiency:

The categorization of primary and automatic flight controls now tends to be according to equipment type, with hydromechanical elements in the PFCS and electrical devices (in particular feedback controls) under AFCS. This arrangement is unsatisfactory for future systems, in particular, fly-by-wire. It seems that the key distinction is one of pilot participation rather than equipment type, and this concept forms the basis for the suggested revision.

(b) Proposed revision:

1.2.1 "Primary Flight Control Systems - Systems which, in conjunction with continuous pilot participation, control the flight path of the aircraft in accordance with prescribed handling and response qualities. Control forces and moments are supplied as functions of pilot input and necessary augmentive feedback signals. Control means could include aerodynamic control surfaces, helicopter rotor blades, reaction controls, thrust orientation, and inertial elements. The systems shall be defined as including, etc."

1.2.3 "Automatic Flight Control Systems - Systems which control aircraft attitude or flight path, in conjunction with elements of the PFCS, without necessity for continuous pilot participation."

3.1 System Design Requirements

(a) Current deficiency:

The simple concept of MTBF as reflecting the safety, mission accomplishment, and maintenance qualities of a flight control system is inadequate and leads to faulty design practices. For example, safety in redundant systems can be compromised to achieve fewer noncritical failures; and minimum maintenance cost is not synonymous with MTBF. In addition, certain remarks pertinent to redundant configurations appear necessary.

(b) Proposed revision:

"Flight control systems shall be the most simple, direct, and foolproof as possible, consistent with overall aircraft effectiveness, with respect to the design, operation, inspection, and maintenance. At the earliest stage in the design of the airplane, careful consideration shall be given to the overall system design of the controls in view of the type of airplane and its mission. Based on total weapon system qualities, allocations shall be made to the flight control system in areas of flight safety (total failure rate), mission reliability (abort rate), and maintainability (costs per flight hour). Such allocations must reflect the nature of the requirement (i.e., safety, mission performance, or maintenance) and be restricted to applicable equipment and functions.

In determining total or partial system reliability, all major failure sources must be considered, including multiple-channel failures, single-point failures, latent failures in both prime and built-in test equipment, and nuisance disengagements of redundant elements. The performance of built-in test equipment shall reflect the test objective (e.g., flight safety, maintenance, etc.) and be applicable only to that equipment which affects the objective. The influences of preflight and in-flight test qualities, test frequencies, and system life on redundant flight control reliability shall be established and satisfactorily resolved by the system design and operational concepts.

The detailed specification shall assign numerical values, methods of demonstration, and associated confidence levels."

3.1.1.1.7 Power Control Override Provisions

(a) Current deficiency:

A requirement for direct pilot effort to free jammed valves is stated. Such will not always be possible or practical.

(b) Proposed revision:

"Provisions shall be made to overcome power control failures, such as jammed control valves, as necessary to comply with total reliability requirements. These provisions shall be as simple and direct as possible, consistent with the basic control system design. In the case of mechanical control linkages, the application of direct pilot effort in conjunction with suitable load-relieving springs or shear elements shall be possible to free jams."

3.1.2.1.5 Trim Position Indicators

(a) Current deficiency:

The stated requirements for trim indication may be somewhat vague in situations where high-authority servos are employed or where certain control arrangements make interpretation of available control range difficult. Loss of an F-111 has been attributed to lack of pitch control authority information under an unusual cg situation.

(b) Proposed revision (add to subject paragraph):

"For control systems where multiple trim devices (e.g., series and parallel) may be used or where available control authority (e.g., net surface position) is not substantially indicated by the control stick or wheel, provision of suitable devices to indicate available control operating range is mandatory."

3.1.3.1.1 Augmentation

(a) Current deficiency:

It seems clear that the augmentation functions are evolving toward inclusion with the PFCS rather than the AFCS as previously mentioned. Indeed the PFCS has included such with bobweights and other dynamic influences. In addition, the current paragraph is restrictive in its inference that "damping" is the sole augmentation function.

As a general comment to the categorization of augmentation systems, if the latter is accepted as part of the PFCS, a general organization change of MIL-F-9490 appears appropriate in that there are now areas under AFCS which more generally apply to both AFCS and PFCS. Examples are 3.1.3.5 (Functional Requirements), 3.1.3.6 (General Tie-in Requirements), and 3.1.3.8 (General Requirements).

(b) Proposed revision:

Delete 3.1.3.1.1 and add the following to 3.1.1:

"Augmentation functions shall be provided as required to improve the stability and handling characteristics of the air vehicle. Unless the response qualities of the airframe-PFCS system are specifically given in the detail specification or mission requirements for the various operational functions, the performance shall be governed by MIL-F-8785. If automatic trim functions are included as part of a closed-loop PFCS, the associated dynamic effects must not degrade handling qualities appreciably or aggravate stall/spin tendencies. In the event of conflicting performance requirements for alternate mission tasks, provision of selectable control modes should be considered."

3.1.3.5.6 Automatic Trim

(a) Current deficiency:

Restrictions of automatic trim to the stated modes is no longer practical.

(b) Proposed revisions:

Delete the last sentence of the subject paragraph.

3.1.3.7.1 Augmentation

Current paragraph appears satisfactory but is considered part of 3.1.1.

3.1.3.7.2.1.2 Residual Oscillation During Steady-State Flight

These criteria appear applicable to both the PFCS and the AFCS.

3.1.3.7.2.4 Automatic Turn Coordination

(a) Current deficiency:

Potential deviations to the specified requirements to achieve superior mission performance is not addressed.

(b) Proposed revision (add to subject paragraph):

"Deviations in the specified performance may be made where superior mission accomplishment can be demonstrated. For example, improved gunnery may be achieved under certain weapon system configurations at the expense of miscoordination in terms of lateral acceleration or sideslip angle."

3.2.16 Electrical and Electronic Systems

(a) Current deficiency:

The phrase "to the greatest extent possible" is inappropriate.

(b) Proposed revision:

Replace the above phrase with "to the extent necessary to provide specified flight safety and mission reliability".

3.3.2 Bearings

While no specific recommendation appears appropriate at this time, it is noted that lubrication requirements on current mechanical control devices cause considerable complaint by Air Force maintenance crews. They question the need for relubrication in certain cases and the variety of the fittings provided.

4.1.2 Developmental Tests

(a) Although simulator or mockup testing is addressed, the use of closed-loop shipside testing is not.

(b) Proposed revision (add to subject paragraph):

"For feedback control systems which materially affect the airplane response qualities or have flight safety implications, closed-loop tests shall also be run using the actual aircraft and system components to the maximum practical extent mounted in their flight configurations."

Also add to 4.1.2.1.1:

"These tests, where possible, shall include closed-loop ground testing using the actual airplane, with aerodynamic effects simulated by a general-purpose computer. The latter will receive measured control surface positions from the airplane and compute associated sensed quantities. These will be applied in lieu of the normal sensor outputs to the flight control computer, which will complete the control loops by driving the relevant control actuators. The general-purpose computer will compute the aircraft response at various key flight conditions so that the flight control stability and performance can be evaluated accordingly. Compliance with the residual oscillation requirements of 3.1.3.7.2.1.2 will be demonstrated. In cases where structural flexure can appreciably affect loop stability, evaluation of this property shall also be accomplished where possible by closed-loop ground tests. This test will utilize all control elements in their flight configurations - sensors, computers, and actuators - with series correction as required to correct for lack of surface aerodynamics. Need for this correction must be determined by analysis of the inertial and aerodynamic properties of the control element (surface). Analysis must also be used to determine proper airframe support for the ground tests to avoid critical bending mode distortion. Support means as used for ground shake tests may be feasible. The closed-loop structural response tests will determine potential instabilities and limit cycles over the pertinent frequency ranges of the bending modes."

3.1.3.8.1 Gain Control

(a) Current deficiency

An addition appears advisable in the area of failure effects.

(b) Proposed revision (add to subject paragraph):

"The reliability and failure properties of the gain control shall be consistent with established safety and mission completion requirements."

APPENDIX IV
F-4 SPIN MODEL

This appendix contains the equations of motion and the approximations used to represent the aerodynamic data in the computer simulation. The basic data source and associated formulations are taken from McDonnell report A0005, Vol. I and II, "Model F-4 Spin Evaluation Program", 15 August 1969. Approximations to the aerodynamic coefficients given by Table IV-1 are unique to the subject study.

SYMBOLS AND NOMENCLATURE

The symbols and nomenclature used for the model are defined as follows: Figures IV-1 and IV-2 are included to illustrate certain variables:

C_x	axial-force coefficient $\frac{F_x}{1/2\ \rho V^2 S}$
C_y	side-force coefficient $\frac{F_y}{1/2 \rho V^2 S}$
C_z	normal-force coefficient $\frac{F_z}{1/2\ \rho V^2 S}$
C_ℓ	rolling-moment coefficient $\frac{M_x}{1/2 \rho V^2 Sb}$
C_m	pitching-moment coefficient $\frac{M_y}{1/2\ \rho V^2 S\bar{c}}$
C_n	yawing-moment coefficient $\frac{M_z}{1/2 \rho V^2 Sb}$
$C_{x_{\delta_s}}$	$\left(\frac{\partial C_x}{\partial \delta_s}\right)$

$C_{y_{\delta_r}}$	$\left(\frac{\partial C_y}{\partial \delta_r}\right)$
$C_{z_{\delta_s}}$	$\left(\frac{\partial C_z}{\partial \delta_s}\right)$
$C_{\ell_{\delta_a}}$	$\left(\frac{\partial C_\ell}{\partial \delta_a}\right)$
$C_{\ell_{\delta_r}}$	$\left(\frac{\partial C_\ell}{\partial \delta_r}\right)$
$C_{m_{\delta_s}}$	$\left(\frac{\partial C_m}{\partial \delta_s}\right)$
$C_{n_{\delta_r}}$	$\left(\frac{\partial C_n}{\partial \delta_r}\right)$
$C_{n_{\delta_a}}$	$\left(\frac{\partial C_n}{\partial \delta_a}\right)$
C_y	$\left(\frac{\partial C_y}{\partial \left(\frac{rb}{2V}\right)}\right)$
C_{y_p}	$\left(\frac{\partial C_y}{\partial \left(\frac{pb}{2V}\right)}\right)$
C_{ℓ_r}	$\left(\frac{\partial C_\ell}{\partial \left(\frac{rb}{2V}\right)}\right)$
C_{ℓ_p}	$\left(\frac{\partial C_\ell}{\partial \left(\frac{pb}{2V}\right)}\right)$

C_{n_r}	$\left(\dfrac{\partial C_n}{\partial\left(\frac{rb}{2V}\right)}\right)$
C_{n_p}	$\left(\dfrac{\partial C_n}{\partial\left(\frac{pb}{2V}\right)}\right)$
$C_{y_{\dot{\beta}}}$	$\left(\dfrac{\partial C_y}{\partial\left(\frac{\dot{\beta}b}{2V}\right)}\right)$
$C_{\ell_{\dot{\beta}}}$	$\left(\dfrac{\partial C_\ell}{\partial\left(\frac{\dot{\beta}b}{2V}\right)}\right)$
$C_{m_{\dot{\alpha}}}$	$\left(\dfrac{\partial C_m}{\partial\left(\frac{\dot{\alpha}\bar{c}}{V}\right)}\right)$
$C_{n_{\dot{\beta}}}$	$\left(\dfrac{\partial C_n}{\partial\left(\frac{\dot{\beta}b}{2V}\right)}\right)$
ΔC_x	change in C_x due to spin rate coefficient, $\frac{\Omega_v b}{2V}$
ΔC_y	change in C_y due to spin rate coefficient, $\frac{\Omega_v b}{2V}$
ΔC_z	change in C_z due to spin rate coefficient, $\frac{\Omega_v b}{2V}$
ΔC_ℓ	change in C_ℓ due to spin rate coefficient, $\frac{\Omega_v b}{2V}$
ΔC_m	change in C_r due to spin rate coefficient, $\frac{\Omega_v b}{2V}$
ΔC_n	change in C_n due to spin rate coefficient, $\frac{\Omega_v b}{2V}$

α_w	wing angle of attack	(deg)
α_b	body axis angle of attack (α_w - 1 deg) (Figure IV-2)	(deg)
β	sideslip angle (Figure IV-2)	(deg)
δ_a	aileron deflection ~ positive when aileron on right wing is deflected down	(deg)
δ_r	rudder deflection ~ trailing edge left is positive	(deg)
δ_e	stabilator deflection ~ trailing edge down is positive	(deg)
θ	Euler angle representing the total angular movement of X body axis from horizontal plane measured in a vertical plane, positive airplane nose up refer to Figure IV-I	(deg)
ϕ	Euler angle between Y body axis and horizontal plane measured in the YZ body plane, positive right wing down (Figure IV-1)	(deg)
ψ	Euler angle representing the horizontal component of total angular deflection of X body axis from reference position in horizontal plane, positive airplane nose right (Figure IV-1)	(deg)
Ω	total angular velocity	(rad/sec)
Ω_v	component of total aircraft angular velocity parallel to velocity vector, positive clockwise looking along velocity vector	(rad/sec)
Ω_e	engine rotation rate positive clockwise looking forward	(rpm, rad/sec)
ρ	air density	(slugs/ft^3)
b	wing span (38.67 ft)	(ft)
$\bar{c}$	mean aerodynamic chord (16.032 ft)	(ft)
c.g.	center of gravity	(ft)
F_x	force along X_B, positive forward (Figure IV-1)	(lb)

F_y	force along Y_B, positive to the right (Figure IV-1)	(lb)
F_z	force along Z_B, positive down (Figure IV-1)	(lb)
I_x	moment of inertia about X body axis	(slug-ft^2)
I_y	moment of inertia about Y body axis	(slug-ft^2)
I_z	moment of inertia about Z body axis	(slug-ft^2)
I_{xz}	product of inertia about X and Z body axes	(slug-ft^2)
$\bar{i}$	unit vector along X_B positive forward	
$\bar{j}$	unit vector along Y_B positive to the right	
$\bar{k}$	unit vector along Z_B positive down	
M_x	moment about X_B, positive clockwise looking forward (Figure IV-1)	(ft-lb)
M_y	moment about Y_B, positive clockwise looking right (Figure IV-1)	(ft-lb)
M_z	moment about Z_B, positive clockwise looking down (Figure IV-1)	(ft-lb)
m	mass of aircraft	(slugs)
p	roll rate, positive right wing down	(rad/sec)
q	pitch rate, positive nose up	(rad/sec)
r	yaw rate, positive nose right	(rad/sec)
p_o, q_o, r_o	oscillatory components of p, q and r	(rad/sec)
p_s, q_s, r_s	steady-state components of p, q and r	(rad/sec)
S	wing area (530 ft^2)	(ft^2)
T	thrust	(lb)
v, w, z	components of V along the X_B, Y_B and Z_B axes, respectively	(ft/sec)

V	total linear velocity	(ft/sec)
X_B	the body axis in the plane of symmetry which is parallel to the fuselage waterline. Positive forward (Figures IV-1 and IV-2).	
Y_B	the body axis perpendicular to the plane of symmetry. Positive toward the right wing. (Figures IV-1 and IV-2).	
Z_B	the body axis perpendicular to the X_B - Y_B axes. Positive toward the bottom of the aircraft. (Figures IV-1 and IV-2).	
X_E	horizontal earth axis which lies in a vertical plane containing the X_B axis at time zero (Figure IV-1)	
Y_E	horizontal earth axis perpendicular to the X_E - Z_E plane (Figure IV-1)	
Z_E	vertical earth axis pointing downward towards the center of the earth (Figure IV-1)	
X_w	axis in plane of symmetry which is parallel to the wing chord plane (Figure IV-2)	
Y_w	axis perpendicular to the plane of symmetry (Figure IV-2)	
Z_w	axis perpendicular to the X_w - Y_w axes (Figure IV-2)	

EQUATIONS OF MOTION

The equations of motion used in the simulation are as follows:

$$\dot{p} = \frac{I_y - I_z}{I_x} qr + \frac{I_{xz}}{I_x} \dot{r} + \frac{I_{xz}}{I_x} pq + \frac{M_x}{I_x} \qquad \text{(IV-1)}$$

$$\dot{q} = \frac{I_z - I_x}{I_y} pr + \frac{I_{xz}}{I_y} r^2 - \frac{I_{xz}}{I_y} p^2 - \frac{I_{eng}\,\Omega_{eng}}{I_y} r + \frac{M_y}{I_y} \qquad \text{(IV-2)}$$

$$\dot{r} = \frac{I_x - I_y}{I_z} pq + \frac{I_{xz}}{I_z} \dot{p} - \frac{I_{xz}}{I_z} qr + \frac{I_{eng}\,\Omega_{eng}}{I_z} q + \frac{M_z}{I_z} \qquad \text{(IV-3)}$$

$$\dot{u} = vr - wq + \frac{F_x}{m} \qquad \text{(IV-4)}$$

$$\dot{v} = wp - ur + \frac{F_y}{m} \qquad \text{(IV-5)}$$

$$\dot{w} = uq - vp + \frac{F_z}{m} \qquad \text{(IV-6)}$$

where

$$M_x = \frac{1}{2}\rho V^2 Sb \left[C_{\ell} + C_{\ell_{\delta r}} \delta r + C_{\ell_{\delta a}} \delta a + \frac{b}{2V}\left(C_{\ell_r} r_o + C_{\ell_p} p_o \right) - \Delta C_{\ell} \right]$$

$$M_y = \frac{1}{2}\rho V^2 S\bar{c} \left[C_{m_{\delta_e}} \delta_e + C_m + \frac{\bar{c}}{2V}\left(C_{m_q} q_o \right) + \Delta C_m \right]$$

$$M_z = \frac{1}{2}\rho V^2 Sb \left[C_n + C_{n_{\delta r}} \delta r + C_{n_{\delta a}} \delta_a + \frac{b}{2V}\left(C_{n_r} r_o + C_{n_p} p_o \right) + \Delta C_n \right]$$

and

$$F_x = -W\sin\theta + T + \frac{1}{2}\rho V^2 S \left[C_x + C_{x_{\delta_e}} \delta_e + \Delta C_x \right]$$

$$F_y = W\cos\theta\sin\phi + \frac{1}{2}\rho V^2 S \left[C_y + C_{y_{\delta_r}} \delta_r + \frac{b}{2V}\left(C_{y_r} r_o + C_{y_p} p_o \right) + \Delta C_y \right]$$

$$F_z = W\cos\theta\cos\phi + \frac{1}{2}\rho V^2 S \left[C_z + C_{z_{\delta_e}} \delta_e \right]$$

where

$$\Delta C_{\ell},\ \Delta C_{m},\ \Delta C_{n},\ \Delta C_{x} \text{ and } \Delta C_{y}$$

are the rotary balance derivatives which are functions of α. β, and $\Omega_V b/2V$.

The forced oscillation derivatives (e.g., $C\ell_r$) are applied only to the oscillatory components of rotation and the rotary balance data are applied as a function of the steady-state rotation. It is assumed that this steady-state rotation is the component of rotation parallel to the velocity vector.

The total rotational velocity is given by

$$\bar{\Omega} = p\bar{i} + q\bar{j} + r\bar{k} \tag{IV-7}$$

The component of $\bar{\Omega}$ parallel to the velocity vector is given by

$$\Omega_V = \bar{\Omega}\,\frac{\bar{V}}{|V|}$$

where

$$V = u\bar{i} + v\bar{j} + w\bar{k}$$

therefore

$$\Omega_V = \frac{up + vq + wr}{V} \tag{IV-8}$$

but

$$\frac{u}{V} = \cos\beta\cos\alpha$$

$$\frac{v}{V} = \sin\beta$$

$$\frac{w}{V} = \cos\beta\sin\alpha$$

substituting into Equation (IV-8) gives

$$\Omega_V = p\cos\beta\cos\alpha + q\sin\beta + r\cos\beta\sin\alpha \tag{IV-9}$$

In order to separate p, q and r into steady-state (parallel to V) and oscillatory (perpendicular to V) components, the components of Ω_V along the X, Y and Z axes are first determined:

$$\bar{\Omega}_V = \Omega_V \frac{\bar{V}}{V}$$

$$\bar{\Omega}_V = \Omega_V \cos\beta \cos\alpha\, \bar{i} + \Omega_V \sin\beta\, \bar{j} + \Omega_V \cos\beta \sin\alpha\, \bar{k} \qquad \text{(IV-10)}$$

Components of Ω_V in the $\bar{i}$, $\bar{j}$ and $\bar{k}$ directions are the steady-state components of p, q and r respectively.

Therefore

$$P_s = \Omega_V \cos\beta \cos\alpha$$

$$q_s = \Omega_V \sin\beta \qquad \text{(IV-11)}$$

$$r_s = \Omega_V \cos\beta \sin\alpha$$

The oscillatory components are

$$p_o = p - p_s$$

$$q_o = q - q_s \qquad \text{(IV-12)}$$

$$r_o = r - r_s$$

When using rotary balance data, only the oscillatory components of p, q, and r are applied to the forced oscillation derivatives. The changes in force and moment coefficients due to steady-state rotation rates are included as a function of $\Omega_V b/2V$.

AERODYNAMIC DATA

The nondimensional aerodynamic derivatives contained in A0005 were approximated wherever practical by analytic expressions. These expressions are shown in Table IV-1. Analytic expressions were not used to approximate the derivatives C_n and $C_{n_{\delta_r}}$. The values of these derivatives as functions of angle of attack and sideslip angle are as shown in Figures IV-3 and IV-4.

Table IV-1. Aero Derivatives

Derivative	α_w Range (deg)	Function (α and β in degrees)
C_x	≤ 15	$-0.03 + 0.002\ \alpha_w$
C_x	> 15 < 55	$[0.03 - 0.0011\ \lvert\beta\rvert\ ]\ \sin(4.5\ \alpha_w - 248)$
C_x	≥ 55	$0.03\ \sin(4.5\ \alpha_w - 248)$
C_y	< 20	-0.014β
C_y	≥ 20 ≤ 59	$(-0.0194 + 0.00027\ \alpha_w)\ \beta$
C_y	> 59	-0.0035β
C_z	< 20	$-0.12 - 0.05\ \alpha_w$
C_z	≥ 20 ≤ 60	$-0.84 - 0.014\ \alpha_w$
C_z	> 60	-1.68
$C_\ell \lvert\beta\rvert \leq 10$	< 40	$\{-0.0010 - 0.00102\ \sin(8.2\ \alpha_w)$ $+ 0.000433\ \cos(16.4\ \alpha_w)\}\beta$
$C_\ell \lvert\beta\rvert > 10$	< 40	$\{-0.0010 - 0.00102\ \sin(8.2\ \alpha_w)$ $+ 0.000433\ \cos(16.4\ \alpha_w)\}\ \beta$ $+ \{0.0030 - 1.6 \times 10^{-4}\alpha_w\}\ (\beta - 10)\ \dfrac{\beta}{\lvert\beta\rvert}$
C_ℓ	≥ 40	-0.0024β
C_m	≤ 90	$-0.000052\ \alpha_w^2 + 0.0002\ \beta^2 - 0.0055\ \lvert\beta\rvert$
C_n	≤ 90	See Figure 3, $C_n\ (\beta = x) = -\ C_n\ (B = x)$,

Table IV-1. Aero Derivatives (Continued)

Derivative	α_w Range (deg)	Function (α and β in degrees)
$C_{x_{\delta_e}}$ per deg	≤ 90	$0.002 - 0.000064\ \alpha_w$
C_{y_r}	> 38	$0.0096\ \alpha_w - 1.27$
C_{y_p}	< 30	$0.0233\ \alpha_w$
C_{y_p}	≥ 30 ≤ 52	$2.5 - 0.6\ \alpha_w$
C_{y_p}	> 52	-0.62
C_{ℓ_r}	< 25	$-0.148 + 0.0295\ \alpha_w$
C_{ℓ_r}	≥ 25 ≤ 55	$1.083 - 0.0197\ \alpha_w$
C_{ℓ_r}	> 55	0
C_{ℓ_p}	< 29	$-0.08 - 0.011\ \alpha_w$
C_{ℓ_p}	≥ 29 ≤ 52	$-0.803 + 0.0139\ \alpha_w$
C_{ℓ_p}	> 52	$0.33 - 0.0079\ \alpha_w$
C_{n_r}	≤ 90	$-0.7 + 0.0082\ \alpha_w$
C_{n_p}	< 25	0
C_{n_p}	≥ 25 ≤ 61	$0.146 - 0.0058\ \alpha_w$

Table IV-1. Aero Derivatives (Continued)

Derivative	α_w Range (deg)	Function (α and β in degrees)
C_{n_p}	> 61	$-1.34 + 0.019\ \alpha_w$
$C_{y_{\delta_r}}$ (deg^{-1})	< 47	$0.00246 - 0.0000523\ \alpha_w$
$C_{y_{\delta_r}}$ (deg^{-1})	≥ 47	0
$C_{z_{\delta_e}}$ (deg^{-1})	≤ 90	$-0.0077 + 0.00008\ \alpha_w$
$C_{\ell_{\delta_a}}$	≤ 10	-0.877
$C_{\ell_{\delta_a}}$[a] (deg^{-1})	> 10 ≤ 90	$[-10.72/\alpha_w + 0.195] \times 10^{-3}$
$C_{\ell_{\delta_r}}$ (deg^{-1})	< 30	$-[0.92 \times 10^{-5}]\ \alpha_w + 27.6 \times 10^{-5}$
$C_{\ell_{\delta_r}}$ (deg^{-1})	≥ 30	0
$C_{m_{\delta_e}}$ (deg^{-1})	< 25	-0.009
$C_{m_{\delta_e}}$ (deg^{-1})	≥ 25 ≤ 40	$0.00033\ \alpha_w - 0.017$

[a] δ_a is single aileron, + for right aileron down.

Table IV-1. Aero Derivatives (Continued)

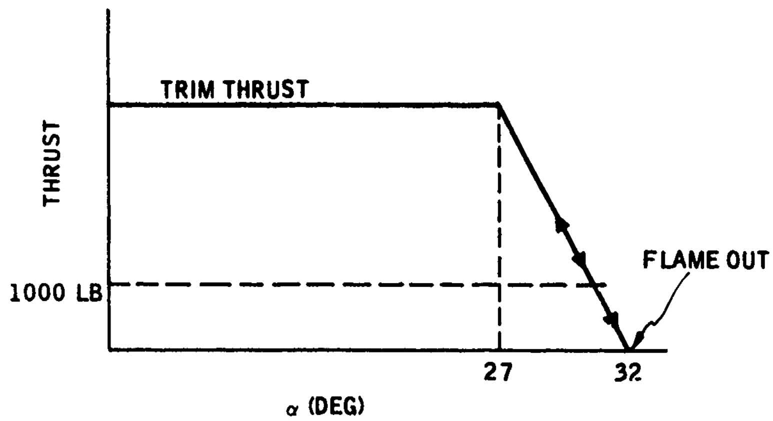
Derivative	α_w Range (deg)	Function (α and β in degrees)
$C_{m_{\delta_e}}$ (deg^{-1})	> 40	-0.004
$C_{n_{\delta_r}}$ (deg^{-1})	≤ 90	See Figure IV-4
$C_{n_{\delta_a}}$ (deg^{-1})	< 20	$(7.5 \times 10^{-6})\ \alpha_w$
$C_{n_{\delta_a}}$ (deg^{-1})	≥ 20	1.5×10^{-4}
C_{y_r}	< 20	0.700
C_{y_r}	≥ 20 ≤ 38	$-0.089\ \alpha_w + 2.48$
C_{m_q}	≤ 41	$-3.3 - 0.161\ \alpha_w$
C_{m_q}	> 41	$-13.5 + 0.088\ \alpha_w$
ΔC_x	< 40	$(0.1) \left(\frac{\lvert \Omega_V \rvert b}{2V} \right)$ (note absolute value)
ΔC_x	≥ 40	$[0.58 - 0.012\ \alpha_w] \left[\frac{\lvert \Omega_V \rvert b}{2V} \right]$
ΔC_y	< 30	0
ΔC_y	≥ 30 ≤ 50	$[0.99 - 0.033\ \alpha_w] \left[\frac{\Omega_V b}{2V} \right]$ (note no absolute value)

Table IV-1. Aero Derivatives (Concluded)

Derivative	α_w Range (deg)	Function (α and β in degrees)
ΔC_y	> 50	$[-1.36 + 0.014\, \alpha_w] \left[\frac{\Omega_v b}{2V}\right]$
ΔC_ℓ	≤ 90	$-0.0375 \left\lvert\frac{\Omega_v b}{2V}\right\rvert$
ΔC_m	< 30	$-0.05\, \beta \left\lvert\frac{\Omega_v b}{2V}\right\rvert$
ΔC_m	≥ 30	$[-0.3 + 0.01\, \alpha_w] \frac{\lvert\Omega_v\rvert b}{2V} - 0.05\, \beta \frac{\Omega_v b}{2V}$
ΔC_n	< 30	$-0.24\, \Omega_v b/2V$
ΔC_n	≥ 30 ≤ 54	$[-0.58 + 0.0113\, \alpha_w]\, \Omega_v b/2V$
ΔC_n	> 54	$0.03\, \Omega_v b/2V$

THRUST CONTROL

Engine thrust is assumed to vary only with α and the selected initial conditions:



After falling below 1000 pounds, a permanent flameout is assumed. The indicated α is in respect to body axes.

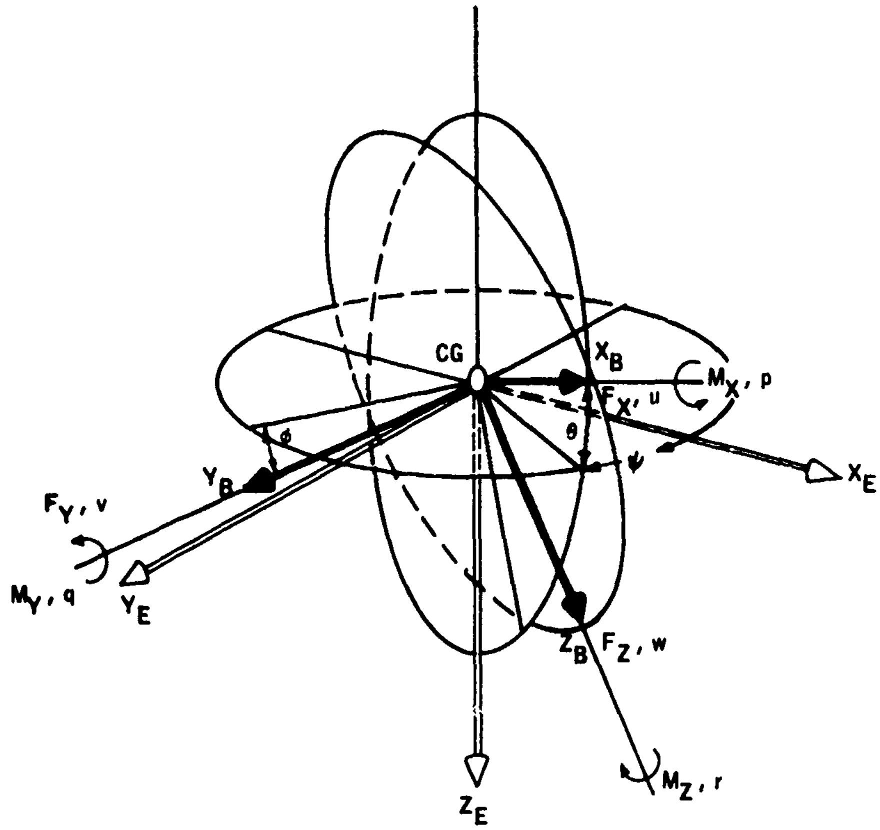


Figure IV-1. Sign Convention Used for Euler Angles, Forces, Moments and Velocities

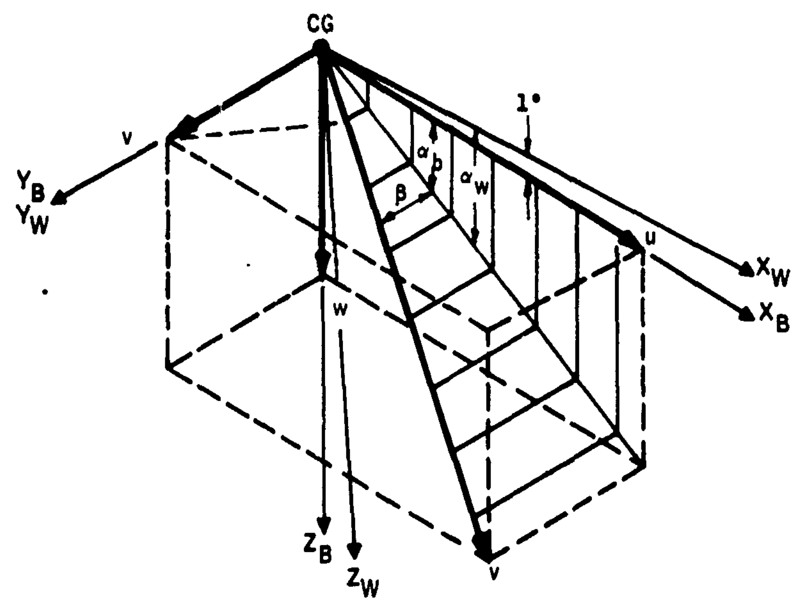


Figure IV-2. Reference Axes, Angle-of-Attack and Angle-of-Sideslip Definition

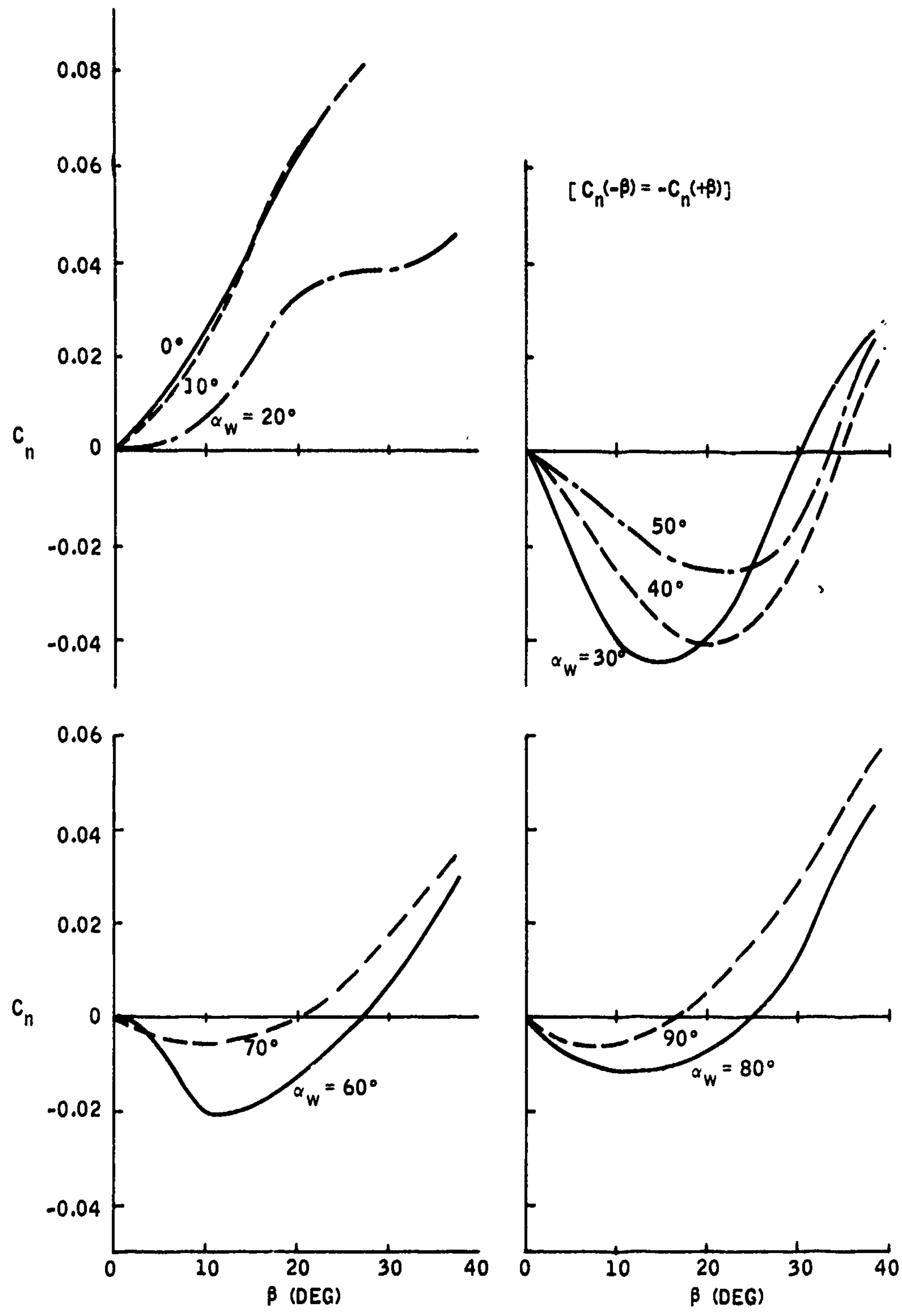


Figure IV-3. Variation of Yawing-Moment Coefficient with Sideslip at Constant Angle of Attack

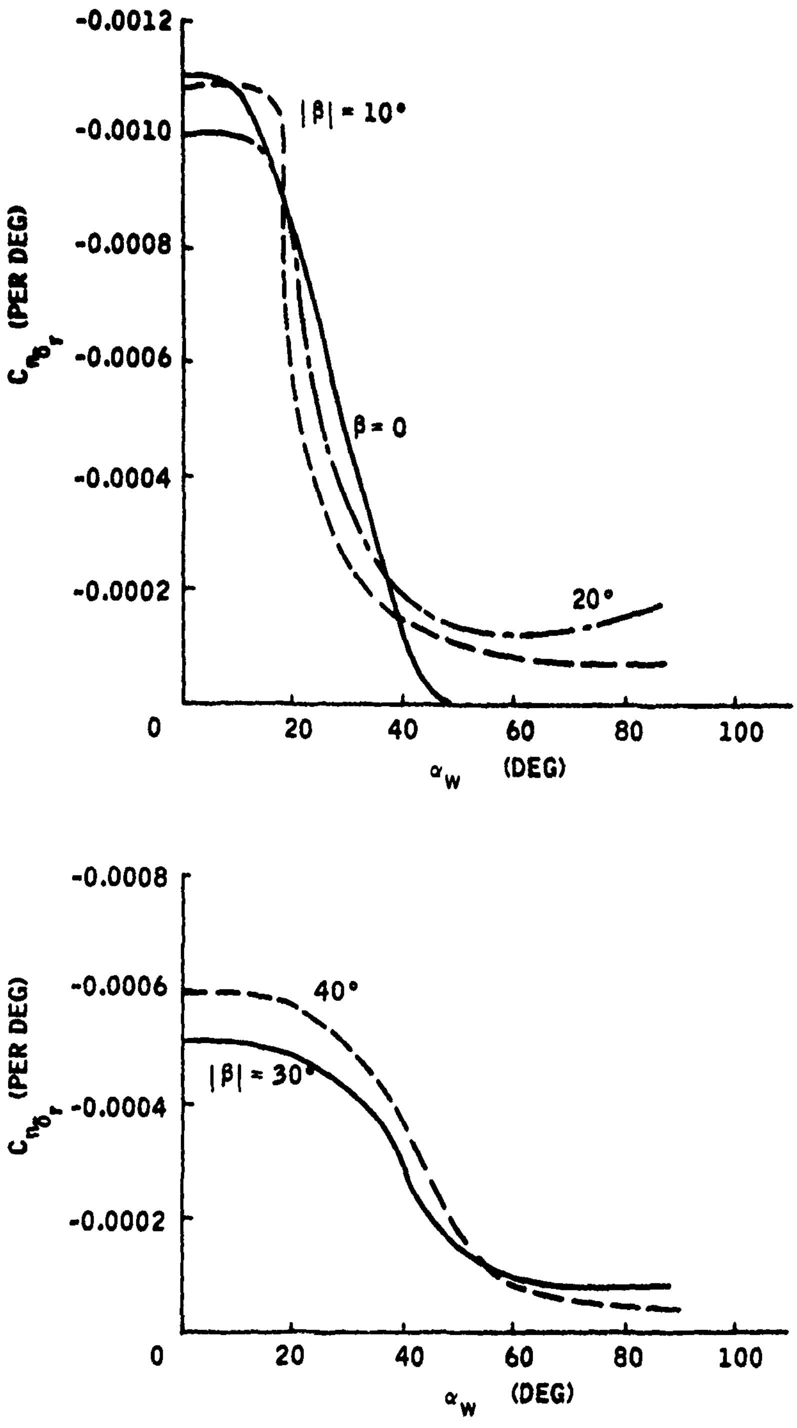


Figure IV-4. Yawing Moment Due to Rudder

APPENDIX V

ACTUATOR DESIGN DATA CATALOG

ACTUATOR DEFINITION

Throughout the tabular data presented in this appendix it will be assumed that an actuator consists of all of the elements necessary for an actuation servo loop, i.e.: the entire actuator consists of the cylinder, servovalve, feedback linkage, and other associated hydraulic elements. This sometimes results in an actuator with ten cylinders, but by this definition, it is still one actuator.

ACTUATOR DATA TABULATION

The subject data are organized according to individual actuators, one data sheet assigned to each. The set contains data for 34 actuators.

- Throughout the sheets, the following abbreviations are used:

T.E. up	Trailing edge up
T.E. dn	Trailing edge down
U	Data unavailable
N/A	Not applicable
rps	Radians per second
G	Gravitational constant
ft	Feet
psi	Pounds per square inch
°/sec	Degrees per second
SL	Sea level
kts	Knots

- The data is organized in three basic groups:

  (1) Actuator physical characteristics, items numbered 1 through 10, indicating the specific features and sizes of the individual actuator

(2) Load (control surface) data, items 11 - 16. This section attempts to show why and how the actuator was sized as it was.

(3) Derived relative performance data, items 17 - 20. These indicate the amount of power required for the actuator, and the power margin for handling the control surface load.

- The actuator characteristics listed are all quite obvious, but a few deserve comment:

  (1) Stroke: this may not always correspond to the deflection and arm radius angles because of end clearances, tolerances, etc.

  (2) Output Arm Radius: in some instances this may be a calculated value due to the unavailability of data.

  (3) Surface Deflection/Actuator inch; because of the geometry of the output arm, this number is not a constant, but an average value. As with the previous data, it also may be a computed figure.

- The load data shown is the result of research aimed at finding out "why" ---- in some cases, specific data was available, but in most cases, only a general statement could be offered. There is enough data to indicate the trends, however.

- Relative performance data was obtained as follows:

  (17) Max. (no load) hydraulic (input) horsepower=

$$\frac{\text{(max. available hinge moment) x (no load surface velocity)}}{\text{(appropriate constant)}}$$

  (18) To obtain horsepower output at maximum hinge moment, the pressure drop across the valve was calculated from the supply pressures and the ratio of aero hinge moment to available hinge moment.

  Knowing the pressure drop, the reduction in velocity can be computed, and the output HP is a product of:

  (max. input hp) x (valve flow factor) x (load factor)

where

$$\text{valve flow factor} = \frac{\text{(flow at max. aero hinge moment)}}{\text{(flow at max. no load velocity)}}$$

$$\text{load factor} = \frac{\text{(max. aero hinge moment)}}{\text{(max. available hinge moment)}}$$

(19) Efficiency in a positive displacement, constant supply pressure system is merely:

$$\frac{\text{aerodynamic hinge moment}}{\text{available max. hinge moment}} \times 100$$

(20) Temperature rise is directly proportional to the pressure drop across the valve, i.e.:

$\Delta T(°F) = 6.64$ deg/1000 psi,

and the pressure drop across the valve is readily obtained from

$$\frac{\left[\text{(max. available hinge moment)} - \text{(aero hinge moment)}\right] \times \begin{pmatrix}\text{supply}\\ \text{pressure}\end{pmatrix}}{\text{(max. available hinge moment)}}$$

These constants are for MIL-H-5606, and were also used for MIL-H-8446 because accurate data was not available. The relative error should be minor.

- Relative performance data should be useful in estimating peak horsepower requirements, as well as cooling system needs. Item (19) is included to show that <u>peak</u> efficiencies can only be achieved in rare circumstances.

ACTUATOR DATA: A-7D Aileron

1. Manufacturer: Vought Aeronautics Model No.: 215-82031
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3050 psi, Temp: -65 to +275°F
3. Description: Dual tandem cylinder (no tail rod) with separate servovalve and feedback linkage. Incorporates special structural feedback linkage.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 1.85 in^2 T.E.up, Stroke 2.65 in., T.E.up
   2.89 in^2 T.E.dn., 2.65 in., T.E.dn.
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 10.6 in. per sec T.E.up, 10.6 in. per sec T.E.dn.
8. Loop Gain: 25 radians per second
9. Output Arm Radius: 6.25 in.; Surface Deflection °/Actuator inch: 9.44
10. Actuator Weight: 10.1 lb
11. Surface Deflection: 25 ° T.E.up; 25 ° T.E.dn.
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed handling to meet MIL-F-8785B
    No Load Surface Velocity: 100 °/sec T.E.up, 100 °/sec T.E.dn.
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Maximum aerodynamic hinge moment.
    Max. Available Hinge Moment: 34,800 in-lb T.E.up, 54,200 in-lb T.E. dn.
14. Max. Aerodynamic Hinge Moment: 38,400 in-lb T.E.dn, ______ in-lb ___
    Occurs at Mach: 1.12, Altitude: 7000 ft, 3 rps roll date
15. Failure of One Hydraulic System Has the Following Result: Actuator can handle only half of full performance load.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 14.3 (T.E.dn.)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 5.48 (T.E.up)
19. Efficiency at Max. Aerodynamic Hinge Moment: 71 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 5.77 °F

ACTUATOR DATA: A-7D Horizontal Stabilizer

1. Manufacturer: Vought Aeronautics Model No.: 15-601051
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3050 psi, Temp: -65 to +275°F
3. Description: Dual tandem cylinder (no tail rod) with integral servovalve. External feedback linkage incorporates structural feedback.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 11.6 in^2 T.E.up, Stroke 4.47 in., T.E.up
   13.7 in^2 T.E.dn, 1.18 in., T.E.dn
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 4.25 in. per sec T.E.up, 4.25 in. per sec T.E.dn
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 10.0 in.; Surface Deflection °/Actuator inch: 5.89
10. Actuator Weight: 32.4 lb
11. Surface Deflection: 26.5° T.E.up; 6.75° T.E.dn
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed maneuvering specs.
    No Load Surface Velocity: 25 °/sec T.E.up, 25 °/sec T.E.dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Stiffness
    Max. Available Hinge Moment: 348,000 in-lb T.E.up, 411,000 in-lb T.E.dn
14. Max. Aerodynamic Hinge Moment: 201,500 in-lb T.E.up, ______ in-lb ___
    Occurs at Mach: 1.12, Altitude: 12,000 ft, 7G
15. Failure of One Hydraulic System Has the Following Result: Stiffness requirement can still be met with 1/2 actuator.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 27.3 (T.E.dn)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 8.35 (T.E.up)
19. Efficiency at Max. Aerodynamic Hinge Moment: 57.9%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 7.82 °F

ACTUATOR DATA: A-7D Rudder

1. Manufacturer: Vought Aeronautics Model No.: 15-151039
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3050 psi, Temp: -65 to +275°F
3. Description: Dual tandem cylinder (equal area) with separate servovalve and feedback linkage. Incorporates special structural feedback linkage. Stroke is limited to ±6° when flaps are in "up" position
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 3.54 in^2 right, Stroke 1.6 in., right

   3.54 in^2 left, 1.6 in., left
6. Pistons Per Actuator: 2 Actuators per Surface: 1

   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 9.35 in. per sec right, 9.35 in. per sec left
8. Loop Gain: 25 radians per second
9. Output Arm Radius: 3.5 in.; Surface Deflection °/Actuator inch: 15.0
10. Actuator Weight: 11.7 lb
11. Surface Deflection: 24 ° right; 24 ° left
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed maneuver specification

    No Load Surface Velocity: 140 °/sec right, 140 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic load and flutter

    Max. Available Hinge Moment: 37,100 in-lb 37,100, left in-lb
14. Max. Aerodynamic Hinge Moment: 24,300 in-lb right, 24,300 in-lb left

    Occurs at Mach: 1.12, Altitude: 7000 ft,
15. Failure of One Hydraulic System Has the Following Result: 1/2 actuator will meet flutter requirements.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 18.7
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 7.2
19. Efficiency at Max. Aerodynamic Hinge Moment: 65.5%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 6.9 °F

ACTUATOR DATA: A-7D Spoiler and Deflector

1. Manufacturer: Vought Aeronautics Model No.: 215-72031

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3050 psi, Temp: -65 to 275°F

3. Description: Dual tandem cylinder (no tail rod) with integral servovalve. Connected to spoiler and deflector with special variable-ratio linkage.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 1.85 in^2 T.E.up, Stroke 5.3 in., Total
   2.29 in^2 T.E.dn, ___ in., ___

6. Pistons Per Actuator: 2 Actuators per Surface: (2 surfaces per actuator)
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 17.3 in. per sec T.E.up, 17.3 in. per sec T.E.dn

8. Loop Gain: NA radians per second

9. Output Arm Radius: 5.3(min) in.; Spoiler Deflection °/Actuator inch: 11.52 min (spoiler)

10. Actuator Weight: 9.08 lb

11. Spoiler Deflection: 60 °T.E.up ; 0 ° T.E.dn

12. Operating Conditions Which Determined No Load Surface Velocity: Roll rate at low altitude

    No Load Surface Velocity: 200 °/sec T.E.up, 200 °/sec T.E.dn

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads

    Max. Available Hinge Moment: 29,400 in-lb T.E.up, 36,400 in-lb T.E.dn

14. Max. Aerodynamic Hinge Moment: 9,270 in-lb T.E.up, (spoiler)
    Occurs at Mach: 1.02, Altitude: 7000 ft, 7G

15. Failure of One Hydraulic System Has the Following Result: Single section capable of handling loads

16. Static Stiffness Capability: U

17. Max. (No Load) Hydraulic (Input) Horsepower: 19.3 (T.E.dn)

18. Horsepower Output at Max. Aerodynamic Hinge Moment: 4.06(spoiler only)

19. Efficiency at Max. Aerodynamic Hinge Moment: 31.5%

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 13.7 °F

ACTUATOR DATA: B-58 Elevon

1. Manufacturer: Bendix-Pacific Model No.: ______
2. Hydraulic Fluid: MIL-H-8446 Pressure: 3000 psi, Temp: -65° to +325°F
3. Description: Ten single cylinders (no tail rod) each of different area and stroke, coupled to elevon with different output arm radius. Separate feedback linkage and servovalve.

4. Cyl. No.	(No. used)	5. T.E. up area	(T.E. dn area)	6. Stroke	(Arm.R.)	Deg/In.	7. Vel. T.E. up	(T.E. dn)
1	(2)	8.43	(7.72)	3.46	(3.81)	16.2	1.50	(1.63)
2	(4)	9.73	(9.10)	4.08	(4.42)	13.6	1.78	(1.93)
3	(2)	12.58	(11.88)	5.17	(5.50)	10.8	2.21	(2.40)
4	(1)	12.58	(11.88)	5.17	(5.50)	10.8	2.21	(2.40)
6	(1)	12.58	(11.88)	5.17	(5.50)	10.8	2.21	(2.40)

8. Number of Hydraulic Supplies per Actuator: 2 (5 cyls on each)
9. Loop Gain: 31.5 Radians per second.
10. Actuator Weight: 123.82 lb (not incl. valve and linkage)
11. Surface Deflection: 35 ° T.E. up ; 20 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed maneuvering; Mach: 0.24 @ SL and limit load factor (Hinge moment less than 12,000 in-lbs)
    No Load Surface Velocity: 25 °/sec T.E. up , 25 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Maximum aerodynamic load; 14° T.E. u
    Max. Available Hinge Moment: 1,536,000 in-lb T.E. up, 1,440,000 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: 1,536,000 in-lb T.E. up, 1,440,000 in-lb T.E. down
    Occurs at Mach: 1.05, Altitude: 25,000 ft, 2.2 G
15. Failure of One Hydraulic System Has the Following Result: Maximum surface deflection reduced.
16. Static Stiffness Capability: ______
17. Max. (No Load) Hydraulic (Input) Horsepower: 101.4
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0 (actuator stall)
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: B-58 Rudder

1. Manufacturer: Bendix-Pacific Model No.:
2. Hydraulic Fluid: MIL-H-8446 Pressure: 3000 psi, Temp: -65 to +375°F
3. Description: Four single cylinders (no rod ends) arranged in pairs on either side of the hinge line, two upper and two lower. Separate feedback and servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 21.8 in^2 right, Stroke 1.75in. right (upper), 2.62in. left (lower)
   21.8 in^2 left, 1.75in. left (upper) 2.62in. left (lower)
6. Pistons Per Actuator: 1 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2 (2 cyls each)
7. No Load Velocity: 4.7 in. per sec (upper), 7.0 in. per sec (lower)
8. Loop Gain: 20.2 radians per second
9. Output Arm Radius: 3.39 upper 5.08 lower Surface Deflection °/Actuator inch: 11.45 upper 17.14 lower
10. Actuator Weight: 40 lb (less valve and linkage)
11. Surface Deflection: 30 ° right ; 30 ° left
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed maneuvering; Mach: 0.24 SL, 50 fps gust
    No Load Surface Velocity: 90 °/sec right, 90 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic and flutter
    Max. Available Hinge Moment: 277,600 in-lb right, 277,600 in-lb left
14. Max. Aerodynamic Hinge Moment: 228,000 in-lb right, 228,000 in-lb left
    Occurs at Mach: 1.36, Altitude: 25,000 ft. 150 fps gust
15. Failure of One Hydraulic System Has the Following Result: Remaining actuator will meet flutter and aerodynamic loads
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 66.1
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 9.47
19. Efficiency at Max. Aerodynamic Hinge Moment: 82.1%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 3.49 °F

ACTUATOR DATA: C-5 Aileron

1. Manufacturer: Bertea, Inc. Model No.: 92900

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +160°F

3. Description: Two single cylinders (no tail rods) with separate feedback linkages and servovalve. Series servo integral with servovalve assembly.

4. Integrated Series Servo Authority: ±15°

5. Total Piston Area per Surface: 15.754 in^2 T.E. up, Stroke 3.534 in., T.E. up
20.562 in^2 T.E. dn, 2.735 in., T.E. dn

6. Pistons Per Actuator: 2 Actuators per Surface: 1
Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 4.46 in. per sec T.E. up, 4.85 in. per sec T.E. dn

8. Loop Gain: NA radians per second

9. Output Arm Radius: 9.15 in.; Surface Deflection °/Actuator inch: 8.25

10. Actuator Weight: 73.76 lb

11. Surface Deflection: 25 ° T.E. up ; 15 ° T.E. dn

12. Operating Conditions Which Determined No Load Surface Velocity: 15°/sec roll rate at max. aerodynamic hinge moment; surface velocity needed = 21°/sec T.E. up and 19°/sec T.E. dn at max. surface deflection
No Load Surface Velocity: 32 °/sec T.E. up, 36.8°/sec T.E. dn

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic hinge moment for roll rate

Max. Available Hinge Moment: 433,000 in-lb T.E. up, 564,000 in-lb T.E. dn

14. Max. Aerodynamic Hinge Moment: 405,000 in-lb T.E. up, 350,000 in-lb T.E. dn
Occurs at: 402 kts; Altitude: SL ft,

15. Failure of One Hydraulic System Has the Following Result:
Reduced surface velocity and roll rate

16. Static Stiffness Capability: U

17. Max. (No Load) Hydraulic (Input) Horsepower: 54.9 T.E. dn

18. Horsepower Output at Max. Aerodynamic Hinge Moment: 20.9 T.E. up

19. Efficiency at Max. Aerodynamic Hinge Moment: 71.9 %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 5.61 °F

ACTUATOR DATA: C-5 Inboard Elevator

1. Manufacturer: Bertea, Inc. Model No.: 92800
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +160°F
3. Description: Two single cylinders (no tail rods) with separate feedback linkages and servovalve. Series servo integral with servovalve.
4. Integrated Series Servo Authority: ± 10°
5. Total Piston Area per Surface: 16.57 in^2 T.E. up, Stroke 4.170 in., T.E. up
   15.464 in^2 T.E. dn, 2.597 in., T.E. dn
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 4.80 in. per sec T.E. up, 4.08 in. per sec T.E. dn
8. Loop Gain: NA radians per second
9. Output Arm Radius: 10.0 in.; Surface Deflection °/Actuator inch: 5.92
10. Actuator Weight: 57.36 lb
11. Surface Deflection: 25 ° T.E. up; 15 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Surface velocity needed & max. aerodynamic hinge moment w/stabilizer nose down, 1.5 g's, velocity = 16.2°/sec T.E. up, 13.1°/sec T.E. dn, full surface deflection.
    No Load Surface Velocity: 30.1 °/sec T.E. up, 26.7 °/sec T.E. dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic hinge moment for load factor

    Max. Available Hinge Moment: 497,000 in-lb T.E. up, 464,000 in-lb T.E. dn
14. Max. Aerodynamic Hinge Moment: 412,500 in-lb T.E. up, 400,000 in-lb T.E. dn
    Occurs at: 392 kts, Altitude: 22,400 ft.
15. Failure of One Hydraulic System Has the Following Result: Reduced surface velocity
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 39.5 (T.E. up)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 13.5 (T.E. up)
19. Efficiency at Max. Aerodynamic Hinge Moment: 83.1 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 3.36 °F

ACTUATOR DATA: C-5 Outboard Elevator

1. Manufacturer: Bertea, Inc. Model No.: 93200
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Three single cylinders (no tail rod) with separate servovalve and feedback linkage.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 6.591 in^2 T.E. up, Stroke 3.214 in., T.E. up
   5.814 in^2 T.E. dn, 2.078 in., T.E. dn
6. Pistons Per Actuator: 3 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 3
7. No Load Velocity: 3.67 in. per sec T.E. up, 3.14 in. per sec T.E. dn
8. Loop Gain: NA radians per second
9. Output Arm Radius: 8.0 in.; Surface Deflection °/Actuator inch: 7.6
10. Actuator Weight: 36.04 lb
11. Surface Deflection: 25 ° T.E. up; 15 ° T.E. dn
12. Operating Conditions Which Determined No Load Surface Velocity: Surface velocity (17°/sec T.E. up, 13.5°/sec T.E. dn) needed at max. aerodynamic hinge moment with stabilizer nose down & 1.5G's, full surface deflection.
    No Load Surface Velocity: 31.8 °/sec T.E. up, 28.3 °/sec T.E. dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads for load factor
    Max. Available Hinge Moment: 158,000 in-lb T.E. up, 139,500 in-lb T.E. dn
14. Max. Aerodynamic Hinge Moment: 124,400 in-lb T.E. up, 122,800 in-lb T.E. dn
    Occurs at: 392 kts, Altitude: 22,400 ft, 1.5G., stab. nose down
15. Failure of One Hydraulic System Has the Following Result: Reduced surface velocity
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 13.3 (T.E. up)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 4.04 (T.E. up)
19. Efficiency at Max. Aerodynamic Hinge Moment: 78.8%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 2.97 °F

ACTUATOR DATA: C-5 Lower Rudder

1. Manufacturer: Bertea, Inc. Model No.: 92700
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Two single cylinders (no tail rod), one on either side of hinge line with separate feedback linkage and servovalve. Series servo integral with servovalve. Position limited to ±4° when q > 200 psf., & ±12° if q is 80-200 psf.
4. Integrated Series Servo Authority: ±20.5°
5. Total Piston Area per Surface: 8.40 in^2 right, Stroke 4.112 in., right
   8.40 in^2 left, 4.218 in., left
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 5.78 in. per sec right, 6.04 in. per sec left
8. Loop Gain: NA radians per second
9. Output Arm Radius: 7.38 in.; Surface Deflection °/Actuator inch: 8.4
10. Actuator Weight: 37.36 lb
11. Surface Deflection: 35 ° right ; 35 ° left
12. Operating Conditions Which Determined No Load Surface Velocity:

    No Load Surface Velocity: 55 °/sec right, 55 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads with one engine out.

    Max. Available Hinge Moment: 185,500 in-lb right, 185,500 in-lb left
14. Max. Aerodynamic Hinge Moment: 141,000 in-lb right, 191,000 in-lb left
    Occurs at: 402 kts, Altitude: SL ft,
15. Failure of One Hydraulic System Has the Following Result: U

16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 26.9 (T.E. right)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 10.0 (T.E. right)
19. Efficiency at Max. Aerodynamic Hinge Moment: 76 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 4.78 °F

ACTUATOR DATA: C-5 Upper Rudder

1. Manufacturer: Bertea, Inc. Model No.: 93300
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Two single cylinders (no tail rod), one on either side of hinge line, with separate feedback linkage and servovalve. Series servo integral with servovalve. Stroke limited (see lower rudder)
4. Integrated Series Servo Authority: ±20.5°
5. Total Piston Area per Surface: 6.182 in^2 right, Stroke 4.119 in., right
   6.182 in^2 right, 4.240 in., left
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 5.72 in. per sec right, 6.05 in. per sec left
8. Loop Gain: NA radians per second
9. Output Arm Radius: 7.38 in.; Surface Deflection °/Actuator inch: 8.37
10. Actuator Weight: 33.21 lb
11. Surface Deflection: 35 ° right ; 35° left
12. Operating Conditions Which Determined No Load Surface Velocity:
   No Load Surface Velocity: 54.9°/sec right, 54.9/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic hinge moment, one engine out.
   Max. Available Hinge Moment: 136,000 in-lb right, 136,000 in-lb left
14. Max. Aerodynamic Hinge Moment: 104,500 in-lb right, 104,500 in-lb left
   Occurs at: 402 kts., Altitude: SL ft,
15. Failure of One Hydraulic System Has the Following Result:
   Reduced hinge moment capability.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 19.7 (T.E. right)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 7.3 (T.E. right)
19. Efficiency at Max. Aerodynamic Hinge Moment: 76.8%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 4.62 °F

ACTUATOR DATA: C-5 Flight Spoiler

1. Manufacturer: Cadillac Controls Model No.: 27265

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +160°F

3. Description: Dual tandem cylinder, unequal area, integral valve.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 14.64 in^2 T.E. up, Stroke 4.08 in., T.E. up
   U in^2 T.E. down ____ in.. ____

6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 14.3 in. per sec T.E. up , 14.0 in. per sec ____

8. Loop Gain: U radians per second

9. Output Arm Radius: 10.5 in.; Surface Deflection °/Actuator inch: 5.75

10. Actuator Weight: 45.6 lb

11. Surface Deflection: 22.5° T.E. up; 0 ° T.E. down

12. Operating Conditions Which Determined No Load Surface Velocity: U

    No Load Surface Velocity: 82 °/sec T.E. up, 80.6°/sec T.E. down

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: U

    Max. Available Hinge Moment: 462,000 in-lb T.E. up, ____ in-lb ____

14. Max. Aerodynamic Hinge Moment: U in-lb T.E. up, ____ in-lb ____
    Occurs at Mach: U , Altitude: U ft, ____

15. Failure of One Hydraulic System Has the Following Result:
    Reduces maximum hinge moment capability

16. Static Stiffness Capability: U

17. Max. (No Load) Hydraulic (Input) Horsepower: 100 (T.E. up)

18. Horsepower Output at Max. Aerodynamic Hinge Moment: U

19. Efficiency at Max. Aerodynamic Hinge Moment: U %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: C-141 Aileron

1. Manufacturer: National Waterlift Model No.: 1867-13
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Two single cylinders (no tail rod) with separate feedback linkage and servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 5.52 in^2 T.E. up, Stroke 3.306 in., T.E. up

   6.90 in^2 T.E. dn, 2.063 in., T.E. dn
6. Pistons Per Actuator: 2 Actuators per Surface: 1

   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 4.55 in. per sec T.E. up, 4.78 in. per sec T.E. dn
8. Loop Gain: 17 radians per second
9. Output Arm Radius: 8.0 in.; Surface Deflection °/Actuator inch: 7.39
10. Actuator Weight: 60.4 lb
11. Surface Deflection: 25 ° T.E. up; 15 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: U

    No Load Surface Velocity: 34 °/sec T.E. up, 34 °/sec T.E. dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: U

    Max. Available Hinge Moment: 132,000 in-lb T.E. up, 165,000 in-lb T.E. dn
14. Max. Aerodynamic Hinge Moment: 111,275 in-lb T.E. up, 159,214 in-lb T.E. dn

    Occurs at Mach: 0.75, Altitude: SL ft,
15. Failure of One Hydraulic System Has the Following Result: U
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 14.9 T.E. dn
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 2.68 T.E. dn
19. Efficiency at Max. Aerodynamic Hinge Moment: 96.5%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0.70 °F

ACTUATOR DATA: C-141 Elevator

1. Manufacturer: National Waterlift Model No.: 1885-4

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +160°F

3. Description: Two single cylinders with separate servovalve and feedback linkage, plus an emergency actuator consisting of a single cylinder and separate servovalve.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 9.91 in^2 T.E. up, Stroke 5.2 in., T.E. up
6.52 in^2 T.E. dn, 3.43 in., T.E. dn

5a. Emergency Actuator: Piston area: 3.14 in^2 T.E. up, Stroke 3.16 in., T.E. up
2.531 in^2 T.E. dn, 1.84 in., T.E. dn

6. Pistons Per Actuator: 3 Actuators per Surface: 1
Number of Hydraulic Supplies per Actuator: 3

7. No Load Velocity: 6.26 in. per sec T.E. up, 6.13 in. per sec T.E. dn

8. Loop Gain: 27.9 radians per second (43.0 emergency actuator)

9. Output Arm Radius: 12.31 in.; Surface Deflection °/Actuator inch: 4.85 (8.38 emergency)

10. Actuator Weight: 112.5 lb

11. Surface Deflection: 23.5° T.E. up; 18° T.E. dn

12. Operating Conditions Which Determined No Load Surface Velocity: Landing approach
No Load Surface Velocity: 39 °/sec T.E. up, 34.5 °/sec T.E. dn

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
Max. Available Hinge Moment: 366,000 in-lb T.E. up, 240,100 in-lb T.E. dn

14. Max. Aerodynamic Hinge Moment: 333,200 in-lb T.E. up, 258,000 in-lb T.E. dn
Occurs at Mach: 0.825, Altitude: 24,800 ft, 2.5G. T.E. up
21,700 ft., 1.0G. T.E. dn

15. Failure of One Hydraulic System Has the Following Result: Automatic logic valves activate emergency actuator which can add 67,000 in-lbs (T.E. up) & 51,000 in-lbs (T.E. dn) to hinge line.

16. Static Stiffness Capability:

17. Max. (No Load) Hydraulic (Input) Horsepower: 37.8 T.E. up

18. Horsepower Output at Max. Aerodynamic Hinge Moment: 10.3 T.E. up

19. Efficiency at Max. Aerodynamic Hinge Moment: 91 %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 1.79 °F

ACTUATOR DATA: C-141 Rudder

1. Manufacturer: National Waterlift Model No.: 1868-13
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +160°F
3. Description: Two single cylinders, one on each side of the hinge line, with separate feedback linkage and servovalve. Electromechanical series servo mounted integral with servovalve. Pressure reduced to limit hinge moment when commanded electrically.
4. Integrated Series Servo Authority: ±10°
5. Total Piston Area per Surface: 4.724 in^2 right, Stroke 3.913 in., right
   4.724 in^2 right, 3.913 in., right
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 4.6 in. per sec right, 4.6 in. per sec left
8. Loop Gain: 18.85 radians per second
9. Output Arm Radius: ______ in.; Surface Deflection °/Actuator inch: 8.94
10. Actuator Weight: 66.8 lb
11. Surface Deflection: 35 ° right ; 35 ° left
12. Operating Conditions Which Determined No Load Surface Velocity: ______
    No Load Surface Velocity: 38 °/sec right, 38 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: ______
    Max. Available Hinge Moment: 79,000 in-lb right, 79,000 in-lb left at 2450 psi
    29,000 in-lb right, 29,000 in-lb left at 900 psi
14. Max. Aerodynamic Hinge Moment: 68,000 in-lb right, 68,000 in-lb left
    Occurs at Mach: 0.25, Altitude: 2000 ft,
15. Failure of One Hydraulic System Has the Following Result:
    Reduces maximum hinge moment capability
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 7.95 dn/up
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 2.55 dn/up
19. Efficiency at Max. Aerodynamic Hinge Moment: 86.1%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 2.78 °F

ACTUATOR DATA: F-4 Aileron

1. Manufacturer: National Water Lift Model No.: 2219-1,2
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Four single cylinders (no tail rods) in parallel integrated with feedback linkage and servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 4.88 in^2 T.E. up, Stroke .173 in., T.E. up
   15.92 in^2 T.E. dn, 2.025 in., T.E. dn
6. Pistons Per Actuator: 4 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 7.0 in. per sec T.E. up, 12.2 in. per sec T.E. dn
8. Loop Gain: 10 radians per second
9. Output Arm Radius: 3.67 in.; Surface Deflection °/Actuator inch: 15.8
10. Actuator Weight: 48 lb
11. Surface Deflection: 1 °T.E. up ; 30 ° T.E. dn
12. Operating Conditions Which Determined No Load Surface Velocity: Low q maneuvering (Mach 1.9 @ 40,000 ft, M: 0.8 @ SL)
    No Load Surface Velocity: 111 °/sec T.E. up, 193 °/sec T.E. dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
    Max. Available Hinge Moment: ______ in-lb T.E. up, 171,300 in-lb T.E. dn
14. Max. Aerodynamic Hinge Moment: 14,600 in-lb T.E. up, 171,300 in-lb T.E. dn
    Occurs at Mach: .98, Altitude: SL ft, T.E. 5° dn actuator stalled
15. Failure of One Hydraulic System Has the Following Result: Reduced hinge moment capability
16. Static Stiffness Capability: 982,029 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 24.4 (T.E. dn)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-4 Rudder

1. Manufacturer: Bertea Model No.: 26500-317
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Single equal area cylinder with integral servovalve arranged as a "boost" actuator, so pilot can manually overpower. Has integral series servo.
4. Integrated Series Servo Authority: ±5°
5. Total Piston Area per Surface: .53 in^2 right, Stroke 2.32 in., right
   .53 in^2 left, 2.32 in., left
6. Pistons Per Actuator: 1 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 1
7. No Load Velocity: 6.45 in. per sec right, 6.95 in. per sec left
8. Loop Gain: 18 radians per second
9. Output Arm Radius: 4.5 in.; Surface Deflection °/Actuator inch: 13.33
10. Actuator Weight: 10 lb
11. Surface Deflection: 30 ° right ; 30 ° left
12. Operating Conditions Which Determined No Load Surface Velocity: Maneuverability at low q (M: 0.75 @ 45,000 ft, M: 0.3 @ SL)
    No Load Surface Velocity: 34 °/sec right, 34 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
    Max. Available Hinge Moment: 8400 in-lb right, 8400 in-lb left
14. Max. Aerodynamic Hinge Moment: 8400 in-lb right, 8400 in-lb left
    Occurs at Mach: 1.1, Altitude: SL ft, ±0.5° actuator stalled
15. Failure of One Hydraulic System Has the Following Result: Reverts to manual
16. Static Stiffness Capability: 40,265 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: .644
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-4 Inboard Spoiler

1. Manufacturer: Bertea Model No.: 32-64517
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65 to +275°F
3. Description: Dual tandem cylinder (no tail rod) with separate servovalve
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 2.52 in^2 T.E. up, Stroke 3.94 in., ___
   2.0 in^2 T.E. dn ___ in., ___
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 17.5 in. per sec T.E. up, 16.6 in. per sec T.E. dn
8. Loop Gain: 10 radians per second
9. Output Arm Radius: 5.01 in.; Surface Deflection °/Actuator inch: 11.42
10. Actuator Weight: 3.5 lb
11. Surface Deflection: 45 ° T.E. up; 0 ° T.E. dn
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed handling
    No Load Surface Velocity: 150 °/sec T.E. up, 150 °/sec T.E. dn
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads for surface effectiveness
    Max. Available Hinge Moment: 38,900 in-lb T.E. up, 31,200 in-lb T.E. dn
14. Max. Aerodynamic Hinge Moment: 38,900 in-lb T.E. up, ___ in-lb ___
    Occurs at Mach: 1.05, Altitude: SL ft, T.E. 12° up actuator stalled
15. Failure of One Hydraulic System Has the Following Result: Reduced surface effectiveness
16. Static Stiffness Capability: 85,443 lb/in
17. Max. (No Load) Hydraulic (Input) Horsepower: 15.4
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-4 Outboard Spoiler

1. Manufacturer: Bertea Model No.: 32-69525
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder (no tail rod) with separate servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 3.82 in^2 T.E. up, Stroke 2.83 in., total
   2.82 in^2 T.E. down ___ in., ___
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 10.3 in. per sec T.E. up, 11.6 in. per sec T.E. down
8. Loop Gain: 10 radians per second
9. Output Arm Radius: 3.60 in.; Surface Deflection °/Actuator inch: 15.90
10. Actuator Weight: 5.0 lb
11. Surface Deflection: 45 ° T.E. up ; 0 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed roll rate

    No Load Surface Velocity: 150 °/sec T.E. up, 150 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads for surface effectiveness.

    Max. Available Hinge Moment: 42,600 in-lb T.E. up, 31,500 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: 42,600 in-lb T.E. up ___ in-lb ___
    Occurs at Mach: 2.0, Altitude: SL ft, 12° T.T. up, actuator stalled
15. Failure of One Hydraulic System Has the Following Result: Reduced surface effectiveness.
16. Static Stiffness Capability: 168,359 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 16.9
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-4 Stabilator

1. Manufacturer: Weston Model No.: 22930
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder (no tail rod) with integral servovalve and "Mod. Piston." Mod piston can serve as series servo or operate entire actuator in parallel mode.
4. Integrated Series Servo Authority: ±0.5°
5. Total Piston Area per Surface: 12.08 in^2 T.E. up, Stroke 6.956 in., T.E. up
   11.27 in^2 T.E. down 3.546 in., T.E. down
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 8.55 in. per sec T.E. up, 8.31 in. per sec T.E. down
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 20.4 in.; Surface Deflection °/Actuator inch: 2.87
10. Actuator Weight: 38 lb
11. Surface Deflection: 20 ° T.E. up ; 10 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low q maneuver response
    No Load Surface Velocity: 24.4 °/sec T.E. up, 24.0 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Stiffness and flutter
    Max. Available Hinge Moment: 739,000 in-lb T.E. up, 688,500 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: 190,000 in-lb T.E. up, 240,000 in-lb T.E. down
    Occurs at Mach: 0.95, Altitude: SL ft, 6.5g
15. Failure of One Hydraulic System Has the Following Result: 1/2 actuator meets stiffness requirement
16. Static Stiffness Capability: 296,298 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 44.2 (T.E. down)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 13.7
19. Efficiency at Max. Aerodynamic Hinge Moment: 40 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 11.9 °F

ACTUATOR DATA: F-14 Rudder

1. Manufacturer: Bendix Model No.: 318-6800

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F

3. Description: Dual tandem cylinder (no tail rod) with integral servovalve.

4. Integrated Series Servo Authority: ____

5. Total Piston Area per Surface: 7.28 in^2 right , Stroke 2.0 in., right
   8.28 in^2 left , 2.0 in., left

6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 7.08 in. per sec right , 7.08 in. per sec left

8. Loop Gain: 75 radians per second

9. Output Arm Radius: 4 in.; Surface Deflection °/Actuator inch: 15.0

10. Actuator Weight: 20 lb

11. Surface Deflection: 30 ° right ; 30 ° left

12. Operating Conditions Which Determined No Load Surface Velocity: Low q, single engine out

    No Load Surface Velocity: 106 °/sec right , 106 °/sec left

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads and flutter

    Max. Available Hinge Moment: 87,300 in-lb right , 99,200 in-lb left

14. Max. Aerodynamic Hinge Moment: U in-lb right , ____ in-lb ____
    Occurs at Mach: U , Altitude: U ft,

15. Failure of One Hydraulic System Has the Following Result: Standby electrically-powered pump will come on line.

16. Static Stiffness Capability: 300,000 lb/in.

17. Max. (No Load) Hydraulic (Input) Horsepower: 27.8

18. Horsepower Output at Max. Aerodynamic Hinge Moment: U

19. Efficiency at Max. Aerodynamic Hinge Moment: U %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-14 Inboard Spoiler

1. Manufacturer: National Waterlift Model No.: 3282

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F

3. Description: Single, equal-area cylinder with integral electrohydraulic servovalve and transducers for electrical positioning command.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 1.7 in^2 T.E.up, Stroke 2.06 in., T.E.up
1.7 in^2 T.E.down in.,

6. Pistons Per Actuator: 1 Actuators per Surface: 1
Number of Hydraulic Supplies per Actuator: 1

7. No Load Velocity: 8.4 in. per sec T.E.up, 8.4 in. per sec T.E.down

8. Loop Gain: 100 radians per second

9. Output Arm Radius: 2.2 in.; Surface Deflection °/Actuator inch: 15.6

10. Actuator Weight: 8.5 lb

11. Surface Deflection: 55 ° T.E.up; 0° T.E. down

12. Operating Conditions Which Determined No Load Surface Velocity: Low q maneuvering

No Load Surface Velocity: 220°/sec T.E.up, 220°/sec T.E. down

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads

Max. Available Hinge Moment: 11,200 in-lb T.E.up, 11,200 in-lb T.E. down

14. Max. Aerodynamic Hinge Moment: U in-lb T.E.up, in-lb
Occurs at Mach: U, Altitude: U ft,

15. Failure of One Hydraulic System Has the Following Result: Sets of four surfaces (two right, two left) are retracted.

16. Static Stiffness Capability: U

17. Max. (No Load) Hydraulic (Input) Horsepower: 6.5

18. Horsepower Output at Max. Aerodynamic Hinge Moment: U

19. Efficiency at Max. Aerodynamic Hinge Moment: U %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-14 Mid-Spoiler

1. Manufacturer: National Waterlift Model No.: 3283
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Single, equal-area clylinder with integral electrohydraulic servovalve and transducers for electrical positioning command.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 2.22 in^2 T.E.up, Stroke 1.56 in., T.E.up
   2.22 in^2 T.E.down ____ in., ____
6. Pistons Per Actuator: 1 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 1
7. No Load Velocity: 6.3 in. per sec T.E.up, 6.3 in. per sec T.E.down
8. Loop Gain: 100 radians per second
9. Output Arm Radius: 1.69 in.; Surface Deflection °/Actuator inch: 35.4
10. Actuator Weight: 10.55 lb
11. Surface Deflection: 55 ° T.E.up ; 0 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low q maneuvering
    No Load Surface Velocity: 220 °/sec T.E.up, 220 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
    Max. Available Hinge Moment: 11,200 in-lb T.E.up 11,200 in-lb T.E.down
14. Max. Aerodynamic Hinge Moment: U in-lb T.E.up, ____ in-lb ____
    Occurs at Mach: U, Altitude: U ft, ____
15. Failure of One Hydraulic System Has the Following Result: Sets of four surfaces (two right, two left) are retracted.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 6.5
18. Horsepower Output at Max. Aerodynamic Hinge Moment: U
19. Efficiency at Max. Aerodynamic Hinge Moment: U %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-14 Outboard Spoiler

1. Manufacturer: National Waterlift Model No.: 3284
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Single, equal-area cylinder with integral electrohydraulic servovalve and transducers for electrical positioning command.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 1.63 in^2 T.E. up, Stroke 1.56 in., T.E. up
   1.63 in^2 T.E. down ___ in., ___
6. Pistons Per Actuator: 1 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 1
7. No Load Velocity: 6.46 in. per sec T.E. up, 6.46 in. per sec T.E. down
8. Loop Gain: 140 radians per second
9. Output Arm Radius: 1.69 in.; Surface Deflection °/Actuator inch: 35.4
10. Actuator Weight: 8.5 lb
11. Surface Deflection: 55 ° T.E. up; 0° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low q maneuvering
    No Load Surface Velocity: 225 °/sec T.E. up, 225 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
    Max. Available Hinge Moment: 8,280 in-lb T.E. up, 8,280 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: U in-lb T.E. up, ___ in-lb ___
    Occurs at Mach: U, Altitude: U ft, ___
15. Failure of One Hydraulic System Has the Following Result: Sets of four surfaces (two right, two left) are retracted.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 4.92
18. Horsepower Output at Max. Aerodynamic Hinge Moment: U
19. Efficiency at Max. Aerodynamic Hinge Moment: U %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-14 Stabilator

1. Manufacturer: Bendix Model No.: 318-7000
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder (no tail rod) with integral servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 34.0 in^2 T.E. up, Stroke 2.49 in., T.E. up
   40.5 in^2 T.E. down 5.81 in., T.E. down
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 5.8 in. per sec T.E. up, 5.81 in. per sec T.E. down
8. Loop Gain: 25 radians per second
9. Output Arm Radius: 10 in.; Surface Deflection °/Actuator inch: 6.02
10. Actuator Weight: 82 lb
11. Surface Deflection: 15 ° T.E. up; 35 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: Low q handling
    No Load Surface Velocity: 35 °/sec T.E. up, 35 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Stiffness and flutter
    Max. Available Hinge Moment: 1,020,000 in-lb T.E. up, 1,213,000 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: U in-lb T.E. up, U in-lb T.E. down
    Occurs at Mach: U, Altitude: U ft,
15. Failure of One Hydraulic System Has the Following Result: Electrically powered standby is activated.
16. Static Stiffness Capability: 550,000 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 112.3 (T.E. down)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: U
19. Efficiency at Max. Aerodynamic Hinge Moment: U %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-15 Aileron

1. Manufacturer: Ozone Model No.: OMP-3829
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem (no rail rod) with integral servovalve.

4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 10.55 in^2 T.E. up, Stroke 0.695 in., T.E. up

   11.78 in^2 T.E. down 0.695 in., T.E. down
6. Pistons Per Actuator: 2 Actuators per Surface: 1

   Number of Hydraulic Supplies per Actuator: 1
7. No Load Velocity: 25.1 in. per sec T.E. up, 25.1 in. per sec T.E. down
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 1.9 in.; Surface Deflection °/Actuator inch: 29.8
10. Actuator Weight: 20 lb
11. Surface Deflection: 20 ° T.E. up ; 20 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: To meet high q roll rate

    No Load Surface Velocity: 75 °/sec T.E. up, 75 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Maximum possible in envelope - hinge moment limited.

    Max. Available Hinge Moment: 60,300 in-lb T.E. up 67,100 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: 60,300 in-lb T.E. up, 67,100 in-lb T.E. down
    Occurs at Mach: 500kt+ Altitude: SL ft,
15. Failure of One Hydraulic System Has the Following Result: Switches to utility supply or reverts to damper
16. Static Stiffness Capability: 280,000 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 13.3 (T.E. down)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-15 Rudder

1. Manufacturer: Ronson Model No.: 3U3151-2
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Rotary vane on surface hinge line (is lower hinge) integral servovalve and series servo.
4. Integrated Series Servo Authority: ±15°
5. Total Piston Area per Surface: NA in^2 ± , Stroke ±30° in., ____
   ____ in^2 ____ , ____ in., ____
6. Pistons Per Actuator: 3 vanes Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 1
7. No Load Velocity: 90° per sec right , 90° per sec left
8. Loop Gain: 20 radians per second
9. Output Arm Radius: NA in.; Surface Deflection °/Actuator inch: NA
10. Actuator Weight: 19.5 lb
11. Surface Deflection: 30 ° right ; 30° left
12. Operating Conditions Which Determined No Load Surface Velocity: Low speed maneuver with crosswind gust.
    No Load Surface Velocity: 90 °/sec right , 90 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Stiffness (actuator operates at reduced pressure)
    Max. Available Hinge Moment: 12,000 in-lb right , 12,000 in-lb left
14. Max. Aerodynamic Hinge Moment: 12,000 in-lb right , 12,000 in-lb left
    Occurs at: 300kt+; Altitude: SL ft, Hinge moment limited.
15. Failure of One Hydraulic System Has the Following Result: Switches to alternate supply or reverts to damper.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 2.86
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0
19. Efficiency at Max. Aerodynamic Hinge Moment: 100%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 °F

ACTUATOR DATA: F-15 Stabilator

1. Manufacturer: National Waterlift Model No.: 3831-2
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder (no tail rod) with integral servovalve and dual series servo.
4. Integrated Series Servo Authority: ±10°
5. Total Piston Area per Surface: 13.70 in^2 T.E. up, Stroke 5.12 in., T.E. up
   14.55 in^2 T.E. down 2.65 in., T.E. down
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 8.0 in. per sec T.E. up, 8.0 in. per sec T.E. down
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 10.5 in.; Surface Deflection °/Actuator inch: 5.67
10. Actuator Weight: 53 lb
11. Surface Deflection: 29 ° T.E. up; 15 ° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: NA
    No Load Surface Velocity: 45 °/sec T.E. up, 45 °/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic hinge moment at high q flight condition.
    Max. Available Hinge Moment: 431,000 in-lb T.E. up, 458,000 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: U in-lb, U in-lb
    Occurs at Mach: U, Altitude: U ft,
15. Failure of One Hydraulic System Has the Following Result: 1/2 Actuator meets stiffness requirement
16. Static Stiffness Capability: 230,000 lb/in.(single system)
17. Max. (No Load) Hydraulic (Input) Horsepower: 54.5 (T.E. down)
18. Horsepower Output at Max. Aerodynamic Hinge Moment: U
19. Efficiency at Max. Aerodynamic Hinge Moment: U %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: U °F

ACTUATOR DATA: F-106 Elevon

1. Manufacturer: O & M Machine/ Hydraulic Research Model No.: 10907

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F

3. Description: Two dual tandem cylinders (no rod ends) with separate feedback linkage and three-mode servovalve (manual, series servo and 1/3 rate mod. piston for AFCS)

4. Integrated Series Servo Authority: ±1°

5. Total Piston Area per Surface: 13.28 in^2 T.E. up, Stroke 3.91 in., T.E. up
   13.28 in^2 T.E. down 1.84 in., T.E. down

6. Pistons Per Actuator: 4 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 6.3 in per sec T.E. up, 6.3 in. per sec T.E. down

8. Loop Gain: 20 radians per second

9. Output Arm Radius: 7.25 in., Surface Deflection °/Actuator inch: 8.18

10. Actuator Weight: 22.4 lb

11. Surface Deflection: 32 ° T.E. up; 15 ° T.E. down

12. Operating Conditions Which Determined No Load Surface Velocity: 50 fps gust during landing
    No Load Surface Velocity: 50 °/sec T.E. up, 50 °/sec T.E. down

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads
    Max. Available Hinge Moment: 288,000 in-lb T.E. up, 288,000 in-lb T.E. down

14. Max. Aerodynamic Hinge Moment: 288,000 in-lb T.E. up, 288,000 in-lb T.E. down
    Occurs at Mach: 2.0, Altitude: U ft, 7.5 G's

15. Failure of One Hydraulic System Has the Following Result: Will accept degradation in performance.

16. Static Stiffness Capability: U

17. Max. (No Load) Hydraulic (Input) Horsepower: 38.2

18. Horsepower Output at Max. Aerodynamic Hinge Moment: 0

19. Efficiency at Max. Aerodynamic Hinge Moment: 100 %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0 F

ACTUATOR DATA: F-106 Rudder

1. Manufacturer: Hydraulic Research Model No.: 100700
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder, (equal area) with integral dual-mode servovalve (manual and series servo).
4. Integrated Series Servo Authority: ±6°
5. Total Piston Area per Surface: 6.18 in^2 right, Stroke 1.7 in., right
   6.18 in^2 left, 1.7 in., left
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 3.38 in. per sec right, 3.38 in. per sec left
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 4 in.; Surface Deflection °/Actuator inch: 14.75
10. Actuator Weight: 18.2 lb
11. Surface Deflection: 25 ° right ; 25° left
12. Operating Conditions Which Determined No Load Surface Velocity: 50 fps gust during landing
    No Load Surface Velocity: 50 °/sec right, 50 °/sec left
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: Aerodynamic loads and flutter
    Max. Available Hinge Moment: 74,200 in-lb right, 74,200 in-lb left
14. Max. Aerodynamic Hinge Moment: 72,000 in-lb right, 72,000 in-lb left
    Occurs at Mach: 2.0, Altitude: U ft,
15. Failure of One Hydraulic System Has the Following Result: Will accept degradation in performance.
16. Static Stiffness Capability: U
17. Max. (No Load) Hydraulic (Input) Horsepower: 9.78
18. Horsepower Output at Max. Aerodynamic Hinge Moment: 1.65
19. Efficiency at Max. Aerodynamic Hinge Moment: 97.2%
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: 0.604 °F

ACTUATOR DATA: F-111 Inboard Spoiler

1. Manufacturer: Bendix Electrodynamics Model No: 3157542-3

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F

3. Description: Two-piston actuator, one for proportional control and the other for positive retraction upon loss of control.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 7.0 in^2 T.E. up, Stroke 2.32 in., T.E. up
   4.0 in^2 T.E. down 0 in., T.E. down

6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 8.8 in. per sec T.E. up, ____ in. per sec ____

8. Loop Gain: 20 radians per second

9. Output Arm Radius: 3.03 in.; Surface Deflection °/Actuator inch: 19.4

10. Actuator Weight: 15 lb

11. Surface Deflection: 45 ° T.E. up; 0° T.E. down

12. Operating Conditions Which Determined No Load Surface Velocity: ____

    No Load Surface Velocity: 162 °/sec T.E. up ____ °/sec ____

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: ____

    Max. Available Hinge Moment: 63,600 in-lb T.E. up, 36,400 in-lb T.E. down

14. Max. Aerodynamic Hinge Moment: ____ in-lb ____, ____ in-lb ____
    Occurs at Mach: ____, Altitude: ____ ft, ____

15. Failure of One Hydraulic System Has the Following Result: ____

16. Static Stiffness Capability: ____

17. Max. (No Load) Hydraulic (Input) Horsepower: 27.2

18. Horsepower Output at Max. Aerodynamic Hinge Moment: ____

19. Efficiency at Max. Aerodynamic Hinge Moment: ____ %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: ____ °F

ACTUATOR DATA: F-111 Outboard Spoiler

1. Manufacturer: Bendix Electrodynamics Model No: 3157540-1

2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F

3. Description: Dual piston actuator, one for proportional position control and the other for positive retraction upon loss of control.

4. Integrated Series Servo Authority: None

5. Total Piston Area per Surface: 4.85 in^2 T.E. up, Stroke 1.85 in., T.E. up
   2.58 in^2 T.E. down 0 in., T.E. down

6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2

7. No Load Velocity: 7.0 in. per sec T.E. up, ____ in. per sec ____

8. Loop Gain: 20 radians per second

9. Output Arm Radius: 2.42 in.; Surface Deflection °/Actuator inch: 24.3

10. Actuator Weight: 12 lb

11. Surface Deflection: 45 °T.E. up ; 0 ° T.E. down

12. Operating Conditions Which Determined No Load Surface Velocity: ____

    No Load Surface Velocity: 169 °/sec T.E. up ____ °/sec ____

13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: ____

    Max. Available Hinge Moment: 35,200 in-lb T.E. up, 18,700 in-lb T.E. down

14. Max. Aerodynamic Hinge Moment: ____ in-lb ____ , ____ in-lb ____
    Occurs at Mach: ____ , Altitude: ____ ft, ____

15. Failure of One Hydraulic System Has the Following Result: ____

16. Static Stiffness Capability: ____

17. Max. (No Load) Hydraulic (Input) Horsepower: 15.7

18. Horsepower Output at Max. Aerodynamic Hinge Moment: ____

19. Efficiency at Max. Aerodynamic Hinge Moment: ____ %

20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: ____ °F

ACTUATOR DATA: F-111 Stabilator

1. Manufacturer: Bendix Electrodynamics Model No: 3157770
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem cylinder with integral servovalve. Equal (balanced) area pistons and enclosed tail rod.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 35.134 in^2 T.E. up, Stroke 4.50 in., T.E. up
   35.134 in^2 T.E. down 2.25 in., T.E. down
6. Pistons Per Actuator: 2 Actuators per Surface: 1 (LH and RH)
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 5.4 in. per sec T.E. up, 5.4 in. per sec T.E. down
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 8.8 in.; Surface Deflection °/Actuator inch: 6.67
10. Actuator Weight: 110 lb
11. Surface Deflection: 30 ° T.E. up ; 15° T.E. down
12. Operating Conditions Which Determined No Load Surface Velocity: ______
   No Load Surface Velocity: 36 °/sec T.E. up, 36°/sec T.E. down
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment: ______
   Max. Available Hinge Moment: 928,000 in-lb T.E. up, 928,000 in-lb T.E. down
14. Max. Aerodynamic Hinge Moment: ______ in-lb ______, ______ in-lb ___
   Occurs at Mach: ____, Altitude: _____ ft, ______
15. Failure of One Hydraulic System Has the Following Result: ______
16. Static Stiffness Capability: 1.13×10^6 lb/in.
17. Max. (No Load) Hydraulic (Input) Horsepower: 88.1
18. Horsepower Output at Max. Aerodynamic Hinge Moment: ______
19. Efficiency at Max. Aerodynamic Hinge Moment: ___ %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: ______ °F

ACTUATOR DATA: F-111 Rudder

1. Manufacturer: Bendix Electrodynamics Model No: 3144210
2. Hydraulic Fluid: MIL-H-5606 Pressure: 3000 psi, Temp: -65° to +275°F
3. Description: Dual tandem with integral servovalve.
4. Integrated Series Servo Authority: None
5. Total Piston Area per Surface: 16.75in^2 left , Stroke 2.408in., left
   16.75in^2 right , 2.408in., right
6. Pistons Per Actuator: 2 Actuators per Surface: 1
   Number of Hydraulic Supplies per Actuator: 2
7. No Load Velocity: 14.4 in. per sec left , 14.4 in. per sec right
8. Loop Gain: 20 radians per second
9. Output Arm Radius: 4.81 in.; Surface Deflection °/Actuator inch: 12.5
10. Actuator Weight: 38 lb
11. Surface Deflection: 30 ° left ; 30 ° right
12. Operating Conditions Which Determined No Load Surface Velocity:
    No Load Surface Velocity: 18 °/sec left , 18 °/sec right
13. Operating Conditions Which Determined Maximum Available Surface Hinge Moment:
    Max. Available Hinge Moment: 242,000 in-lb left , 242,000 in-lb right
14. Max. Aerodynamic Hinge Moment: in-lb , in-lb
    Occurs at Mach: , Altitude: ft,
15. Failure of One Hydraulic System Has the Following Result:
16. Static Stiffness Capability: 0.67 x 10^6 lb/in. (retracted)
17. Max. (No Load) Hydraulic (Input) Horsepower: 11.5
18. Horsepower Output at Max. Aerodynamic Hinge Moment:
19. Efficiency at Max. Aerodynamic Hinge Moment: %
20. Fluid Temp. Rise at Max. Aerodynamic Hinge Moment: °F

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13. ABSTRACT

A study to develop improved design criteria for primary flight controls which feature feedback techniques is reported. The study consisted of eight parts, including a survey of operational problems, a review of system-gain changing (requirements and techniques), stabilization criteria for high-frequency control modes, an analysis of stall/spin maneuvers, a catalog of dominant performance characteristics which affect flying qualities, an analysis of system/airframe compatibility testing, definition of criteria for built-in test equipment, and a catalog of flight control actuator designs. Operational problems include high angle-of-attack stability and potential control loss. A math model of a spinning F-4 was used to study basic effects and associated control criteria. Nominal control laws in pitch and yaw tended to be beneficial for departure inhibition; roll control degraded controllability. Spin recovery demands full surface deployment without detraction by feedback. The controllability limit for spin recovery was defined. The compatibility test analysis featured closed-loop simulation of structural response. Compensation for surface aerodynamics and special airframe support to avoid bending mode distortion were justified. Criteria for built-in test equipment in redundant flight controls to produce adequate flight safety and mission reliability were expressed in terms of test thoroughness, latent failure probabilities, and false indication rate. Test quality was shown to have a highly significant effect on system reliability that becomes more critical with the number of redundant channels and system life.

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stall limiting						
spin recovery						
spin preventions						
redundancy						
spin simulation						
compatibility tests						
stability margins						
structural stabilization						
bending stability						
structural coupling						
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stability criteria						
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