APPLICATION OF PILOT-CONTROLLER INTEGRATION TECHNIQUES TO A REPRESENTATIVE V/STOL AIRCRAFT

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FOREWORD

This document is a final report summarizing results obtained from a study of the "Application of Pilot-Controller Integration Techniques to a Representative V/STOL Aircraft". This program was sponsored by the Flight Control Division, Air Force Flight Dynamics Laboratory, Research and Technology Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio under Contract AF33(615)-1866 with Bell Aerosystems Company, Buffalo 5, New York.

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ABSTRACT

This report presents final results of a study of the application of Pilot-Controller Integration (PCI) design techniques to the flight control system of a representative V/STOL aircraft. Under this program the validity of the concept was established in the application to the X-22A V/STOL. In this application the PCI technique indicated the areas of the X-22A flight control system where modifications would result in the greatest improvement to the probability of mission accomplishment. Design modifications were made and an iteration using the technique was accomplished and the payoff was evaluated. The digital program which was developed and applied to the X-22A has general applicability to other aircraft. Several improvements to this program as well as to the details of technique application are suggested.

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LIST OF ABBREVIATIONS

AMSP	Analysis of mission success probability
CSFM	Catastrophic single failure mode
DBFA	Δ blade (propeller); forward and aft - pitch control
DBLR	Δ blade (propeller); left and right - roll and yaw control
DCFA	Δ flap (elevon); forward and aft - pitch control
DFLR	Δ flap (elevon); left and right - roll and yaw control
FCS	Flight control system
FMC	Failure mode combination
GW	Gross Weight
MS	Mission Segment
PCI	Pilot-controller integration
PR	Pilot Rating
PSAR	Probability of successful aircraft recovery
PSMA	Probability of successful mission accomplishment
SAS	Stability augmentation system
SD	Success diagram
STOL	Short takeoff and landing
VSS	Variable stability system
VTOL	Vertical Takeoff and landing
WL	Workload
ΔWL	Workload increment

LIST OF SYMBOLS

b	Wing reference span
$\bar{c}$	Wing reference cord
C_{ℓ_p}	Roll damping coefficient
C_{ℓ_r}	Coefficient of rolling moment due to yaw rate
C_{ℓ_s}	Rolling moment coefficient referenced q_T
C_{m_q}	Pitch damping coefficient
C_{m_s}	Pitching moment coefficient referenced to q_T
C_{n_p}	Coefficient of yawing moment due to roll rate
C_{n_r}	Yaw damping coefficient
C_{n_s}	Yawing moment coefficient referenced to q_T
C_{T_s}	Thrust coefficient referenced to q_T
C_{X_s}	Longitudinal force coefficient referenced to q_T; body axis
C_{y_p}	Coefficient of side force due to roll rate
C_{y_r}	Coefficient of side force due to yaw rate
C_{y_s}	Side force coefficient referenced to q_T; body axis
C_{z_s}	Vertical force coefficient referenced to q_T; body axis
g	Gravitational constant
h	Altitude
I_x	Moment of inertia about the x-body axis
I_y	Moment of inertia about the y-body axis
I_z	Moment of inertia about the z-body axis
I_{xz}	Product of inertia
L_c	Roll control power - rad/sec^2
L_p	Roll damping - 1/sec

m	Vehicle mass
M_c	Pitch control power - rad/sec^2
M_q	Pitch damping - 1/sec
M_μ	Pitching moment derivative due to horizontal velocity
M_α	Pitching moment derivative due to angle of attack
P	Body axis roll rate
q	Body axis pitch rate
q_e	Duct exit dynamic pressure
q_F	Free stream dynamic pressure
q_T	Thrust referenced dynamic pressure = q_F + T/S
r	Body axis yaw rate
S	Wing reference area
T	Thrust
u	Body axis longitudinal velocity
v	Body axis lateral velocity
V	Free stream total velocity
w	Body axis vertical velocity
X_{AXA}	Perpendicular distance between the aft ducts axial force and the c.g.
X_{AXF}	Perpendicular distance between the forward ducts axial force and the c.g.
$\dot{X}_I$	North referenced local inertial velocity component
X_{NA}	Perpendicular distance between the aft ducts normal force and the c.g.
X_{NF}	Perpendicular distance between the forward ducts normal force and the c.g.
y_A	Lateral distance between the aft duct centerline and the fuselage centerline
y_F	Lateral distance between the forward duct centerline and the fuselage centerline
$\dot{Y}_I$	East referenced local inertial velocity component
$\dot{Z}_I$	Vertical referenced local inertial velocity component
Z_w	Vertical damping
α	Angle of attack
β	Angle of side slip
β_{prop}	Propeller incremental blade angle command

γ	Vertical flight path angle
Δ	Denotes incremental value
δ_e	Elevon position
θ	Euler pitch angle
λ	Duct position - deg
ξ	Earth referenced azimuth angle
ϕ	Euler roll angle
ψ	Euler yaw angle
$(\dot{\ })$	Denotes differentation with respect to time
$(\ )_{DC}$	Denotes digital computation

SECTION I

INTRODUCTION AND SUMMARY

This is a final report presenting the results of a study of Pilot-Controller Integration techniques accomplished under Contract AF33(615)-1866. The primary objective of this program was to provide a detailed example of the application of pilot-controller integration (PCI) design techniques to the flight control system of a representative V/STOL aircraft - the X-22A. Also, the payoff in FCS operational performance resulting from PCI orientated component redesign was to be evaluated. Additional objectives were to evaluate the existing concept procedures [(1)] in terms of validity and utility, and to expand and improve these procedures (concept improvements) to better accomplish the primary objective. A secondary objective was the conduct of a control power-damping handling qualities analysis of the X-22A.

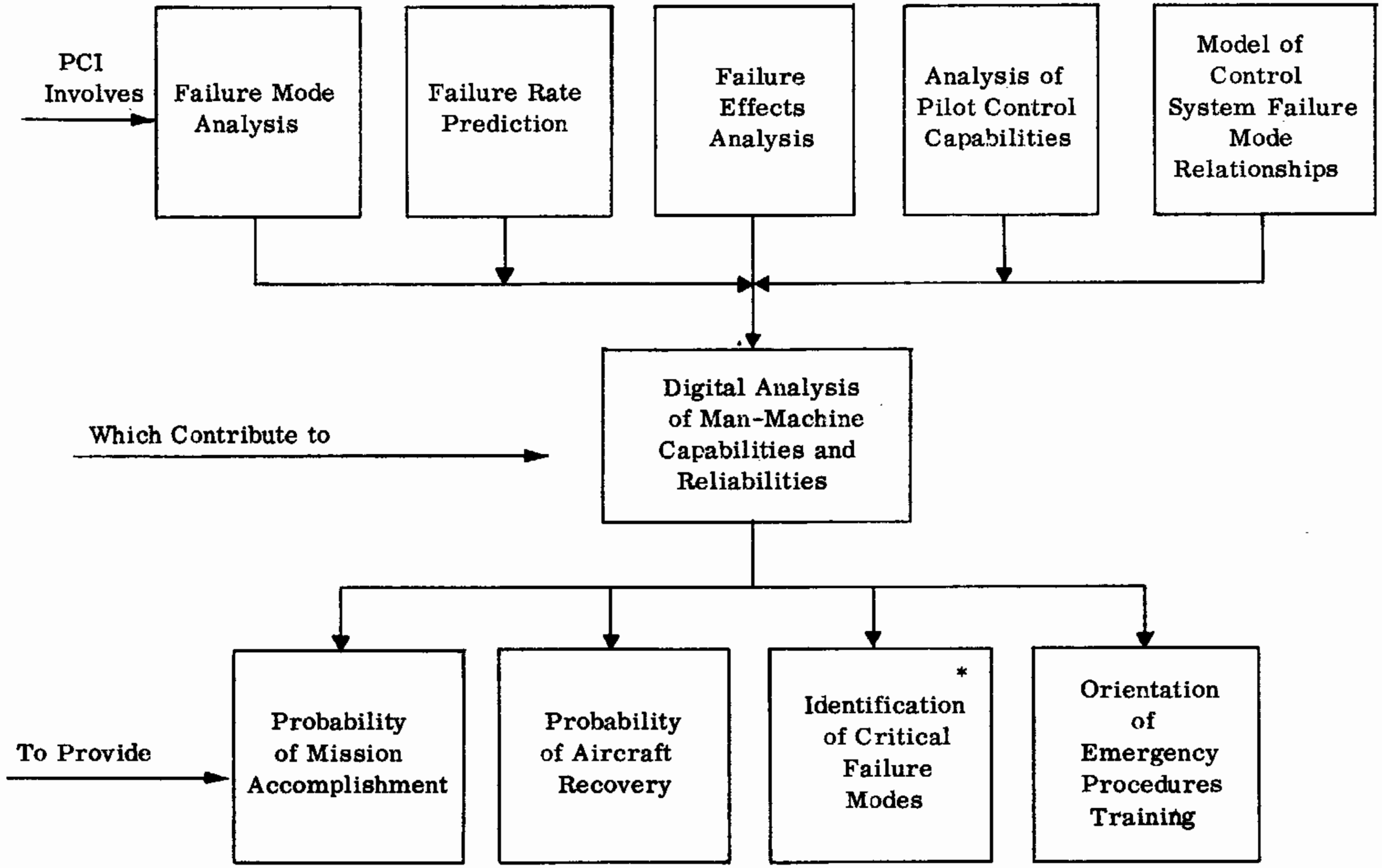
The increasing complexity of the interface relationships between the human operator and state-of-the-art aerospace vehicle control systems has resulted in severe limitations in the ability of man-machine systems to successfully accomplish their operational objectives. The analysis of potentially catastrophic failure conditions is currently treated in an essentially open loop manner, with each design specialty (or group) contributing its separate inputs to the design task, and with no fully coordinated approach to the efficient integration of the information generated by the participating groups.

The purpose of the PCI technique is to fill the need for a systematic approach to the quantitative prediction of the integrated capabilities and reliabilities of the man-machine system. PCI basically utilizes existing engineering talents in the areas of systems design, reliability analysis, and human factors, and effectively integrates them into a digital program for the analysis of mission success probability (AMSP) - refer to Figure 1.

In addition to the prediction of overall mission success probability, the AMSP program provides a listing - in ranked order of probability - of those failure mode combinations making the greatest contribution to the probability of mission failure. This data provides the basis for logical and efficient control systems redesign.

Additional payoff accruing from the application of PCI in the design procedure results from the applicability of PCI program output to operational and cost effectiveness studies.

[(1)] The PCI technique as originally developed by Minneapolis-Honeywell under Contract AF33(657)-7601 is defined in Technical Documentary Report No. RTD-TDR-63-4092.



*Basis for Intelligent Flight
Control Systems Redesign

Figure 1. Functional Definition of Pilot-Controller Integration Technique

The scope of the PCI application to the flight control system of the X-22A under this program dictated the specification of certain limiting conditions including the following:

(1) Automatic attitude control [2], height control and height damping [2], and the duct rotation subsystems were not included in the analysis.

(2) The prediction of component failure modes and rates was not as detailed and complete as would be required if the output from the PCI application were to be utilized in the actual vehicle design procedure. It is estimated that the 450 hours devoted to the reliability analysis under this program would be multiplied by a factor of 3 to 4 in the case of a prototype vehicle design.

(3) Detailed redesign of FCS components as dictated by PCI was carried only to a point sufficient to determine the validity of the proposed changes.

Inclusion of the above constraints in no way restricted the accomplishment of contract objectives in the application of PCI to the X-22A.

The payoff in man-machine systems performance was studied for the hypothetical redesign of the four elevon control actuator systems. The resulting payoff was as follows:

(1) The probability of successful mission accomplishment increased from 98.79 to 99.22% - a 35% reduction in the probability of not completing the mission.

(2) The probability of successful aircraft recovery (including the ability to successfully abort the defined mission) increased from 98.97% to 99.42%.

(3) Cost effectiveness studies indicate a saving of 4.4 aircraft (total value of $6,600,000) in flying 1000 sorties of a type similar to the evaluation mission.

A number of improvements to the basic PCI concept (listed in Section IV) evolved during the subject application; the following were the most significant:

(1) The construction and functional aspects of mission success diagrams were expanded.

(2) The problem of interference effects between pilot work load and/or transient control capability for certain concurrent failures was recognized, and methods of analysis were suggested.

(3) The structure of the digital program for the analysis of mission success probability was modified and expanded.

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[2] Not part of the X-22A primary control system. These functions are integral to the variable stability system (VSS).

(4) The utilization of the digital program for AMSP to also predict the probability of successful aircraft recovery was explained.

Longitudinal and lateral handling qualities studies for hover flight and transition were conducted as a separate task under this program. Pilot opinion ratings and measured pilot work loads were recorded at each control power-damping test condition. This data is discussed in Appendix I.

PCI provides a useful tool for systems design from the preliminary phase on through the design procedure and into post production - in service modification programs. The value of PCI in the design procedure is dependent upon the detail, accuracy, and completeness of the data required for input to the AMSP program; i.e., the prediction of component failure modes and rates; the modeling of failure mode dependencies in the mission success diagrams; and the analysis of pilot work loads and transient control capabilities.

The PCI technique is not restricted to aerospace vehicle flight control systems but may have application to the general area of integrated man-machine systems that have a mission or task reliability goal.

SECTION II

PILOT-CONTROLLER INTEGRATION CONCEPT PROCEDURES

1. INTRODUCTION

The pilot-controller integration technique is a systematic approach to the quantitative prediction of the integrated capabilities and reliabilities of the man-machine system. The PCI technique consists of twelve separate phases (Figure 2), each of which constitutes a unique task in the accomplishment of the overall concept objectives. The PCI phases as presently defined (3), are described below.

a. Familiarization with the Vehicle Systems

The background knowledge of vehicle operational tasks, performance capabilities, systems descriptions, and design goals required throughout the implementation of PCI will be provided during this phase.

b. Development of Success Diagrams

The development of success diagrams involves the construction of block diagrams defining the control continuity and pilot work load relationships between control system failure modes which affect mission success probability in each of the evaluation mission segments. The success diagrams provide the basis for the construction of the mathematical model required for the functional testing of control system failure modes in the digital analysis of mission success probability.

c. Prediction of Component Failure Mode

The prediction of flight control system failure modes provides a measure of the effects of component failure on systems and vehicle response. This information is basic to the construction of mission success diagrams, the prediction of component failure rates, and the analysis of pilot control capabilities.

d. Development of Vehicle-Controller Models

Mathematical models of the vehicle-controller system are formulated for use in the analysis of simple and complex failure conditions.

e. Establishment of Man-Machine Constraints

The physical limitations to the safe operating regimes for both the human operator and the machine systems are required as criteria in the specification of catastrophic pilot-vehicle response to systems failures.

(3) The concept improvements described in Section IV are incorporated into the PCI procedures as originally defined in Reference 1

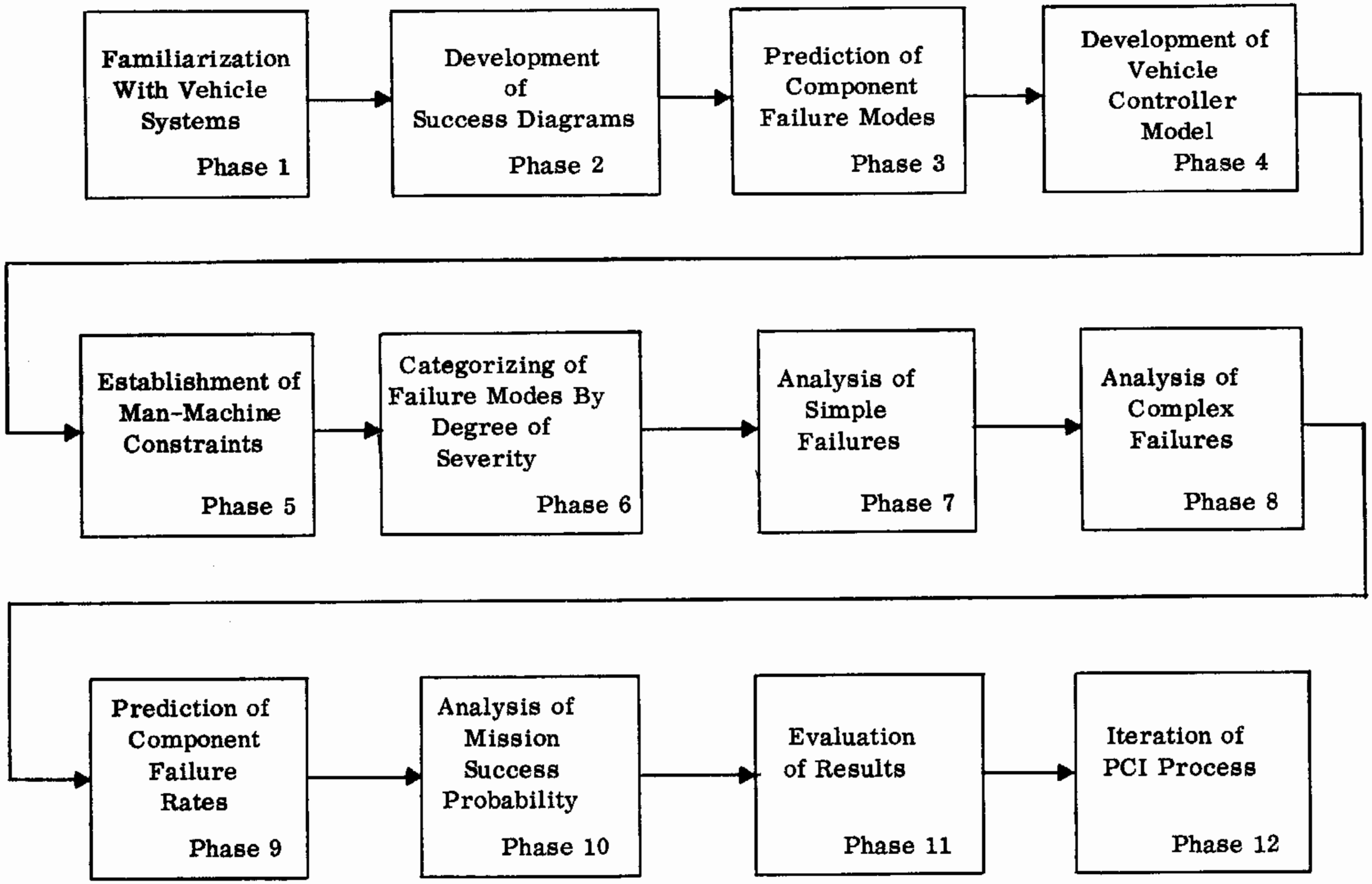


Figure 2. Phases of the Pilot-Controller Integration Technique

f. Categorizing of Failure Modes by Degree of Severity

Failure modes must be identified as being definitely catastrophic, potentially catastrophic, or trivial in their effects on vehicle response in order to establish the nature of succeeding failure effects analysis and treatment in the AMSP.

g. Analysis of Simple Failures

This analysis is limited to the failure effects analysis of those failure modes which result in readily definable (through direct mathematical means) pilot-controller-vehicle response. It is employed primarily in the preliminary design phase.

h. Analysis of Complex Failures

The analysis of complex failures is concerned with the failure effects and work load analysis of those potentially catastrophic failure modes resulting in highly nonlinear and/or interaction effects between vehicle-control system characteristics.

i. Prediction of Component Failure Rates

Failure rates of all those component modes of failure determined to be either definitely catastrophic or potentially catastrophic are predicted for input to the digital AMSP.

j. Digital Analysis of Mission Success Probability

The analysis of mission success probability is the focal point of the PCI procedure. Mission success diagrams (in the form of a mathematical model) failure mode, failure effect, pilot work load, and failure rate data generated in the previous phases is funneled into the digital program for the analysis of mission success probability. The AMSP program generates, in addition to overall mission success probability, the contributions of the individual failure mode combinations to mission success or failure. When the predicted mission success probability meets or exceeds the design goal, no further iteration through the PCI procedures is required.

k. Evaluation of Results

On first pass through the PCI procedure, the output from the AMSP program is analyzed to establish those failure modes having the greatest influence on mission failure.

On successive iterations through the PCI procedure, evaluations similar to that explained above are conducted; however, in addition, the application payoff [(4)] resulting from each successive systems modification or redesign is evaluated.

[(4)] Refer to Sections III and III-1 for further detail

1. Iteration of the PCI Process

Should the predicted mission success probability fall below the design goal, the most critical systems components should be examined from the standpoint of modifications to improve their reliability and/or to decrease their failure effects on pilot control capabilities. After system redesign, the PCI procedures should be reapplied. Several such cycles may be required until the mission success goal is attained.

Also, as the design progresses and more refined and accurate information becomes available, PCI should be reiterated to establish more realistic predictions of mission success.

2. FAMILIARIZATION WITH THE VEHICLE SYSTEMS - PCI PHASE 1

Application of the PCI technique to a given aerospace vehicle requires a knowledge of the vehicle and its system, the operational tasks, and vehicle performance capabilities as well as a definition of the control-display interface between the man and the machine. Thus, in this first phase of the PCI program, a background knowledge for use in the succeeding phases of the procedure is developed.

a. Vehicle and Control System Definition

(1) Aircraft General Description

The Bell X-22A (Figure 3) selected as the application vehicle for the PCI concept is a dual tandem-ducted propeller V/STOL research aircraft in the 15,000-pound gross weight class. Figure 4 shows the general arrangement of the aircraft and lists its dimensional data.

The aircraft carries a flight crew of two men and a 1200-pound payload in the cargo/passenger compartment. Four interconnected and rotatable ducted propeller propulsion units provide a high static thrust-to-weight ratio for vertical takeoff. The duct-propeller combination also provides a large part of the lift in level flight. The ducts contain seven-foot controllable pitch propellers which are powered by four T58-GE-8 turboshaft engines rated at 1250 shp each. The engines are mounted in pairs in nacelles on each wing adjacent to the fuselage at five degrees incidence relative to a fuselage waterline and are geared directly to the cross-shaft between the two aft ducts. A shaft carries the power forward from the rear cross-shaft to the front duct cross-shaft. Because the propellers are interconnected, the aircraft may be powered more efficiently during cruise by two or three engines with only a small drag penalty due to the shutdown engines which are declutched from the drive system. In the nominal cruise configuration, the forward ducts are locked at two degrees incidence with respect to the fuselage reference line and the aft ducts at minus three degrees. These duct incidence settings can be ground adjusted plus or minus two degrees from the nominal in one degree increments.

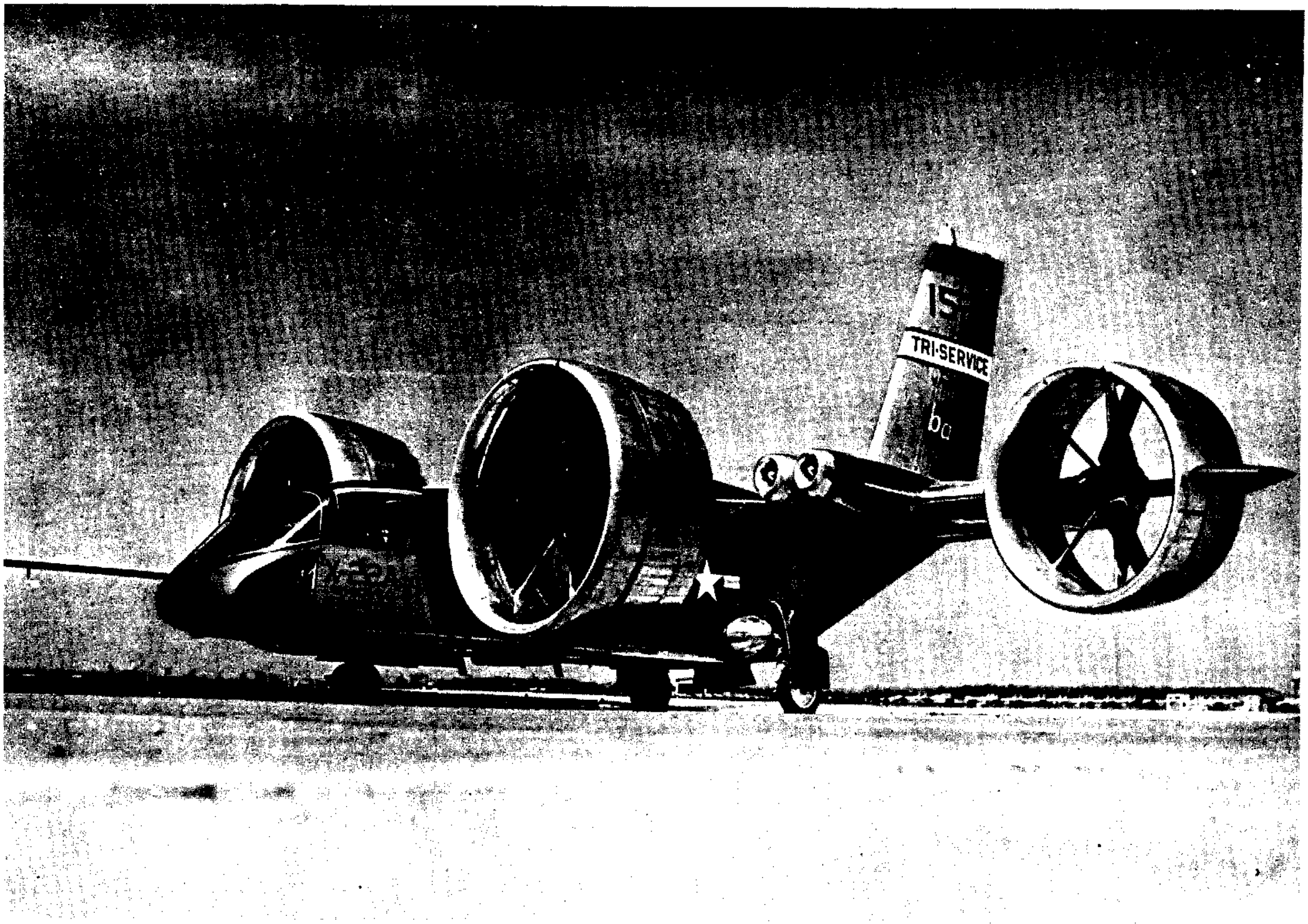


Figure 3. X-22A V/STOL Ducted Propeller Airplane

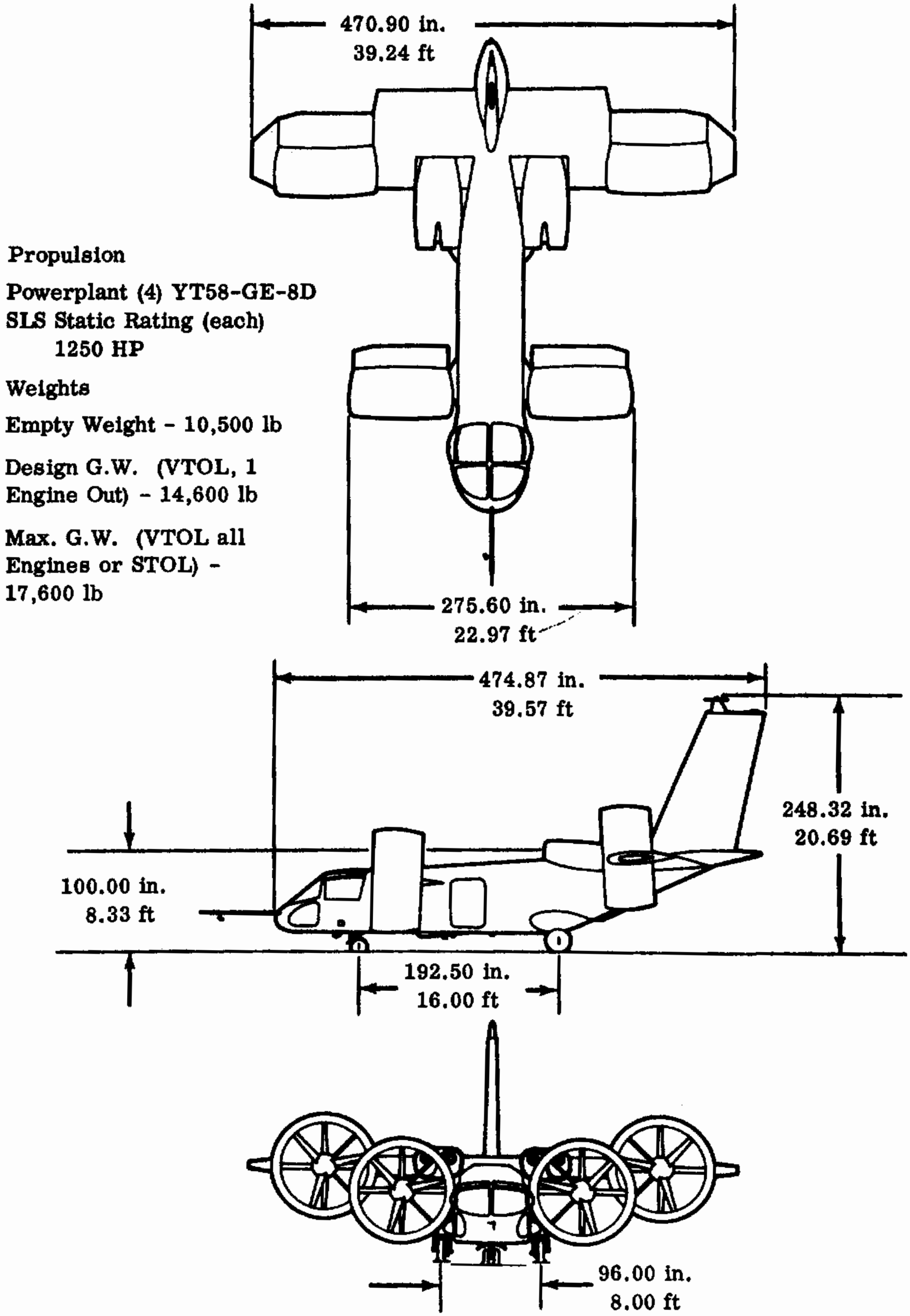


Figure 4. X-22A General Arrangement 3-View

For vertical takeoff and landing, the ducts rotate together at a maximum rate of 5 deg/sec. This rate drops in the last five degrees of travel to 1 deg/sec, one-half degree ahead of the mechanical stop. The maximum angular travel from the cruise duct incidence combination is 93° for the forward pair and 98.34° for the aft pair.

Longitudinal stability is provided by the rear lifting assembly which consists of the ducts, wing, and horizontal stabilizers. Directional stability is provided by a conventional fixed vertical stabilizer. The in-board wing is set at three degrees positive incidence relative to a fuselage waterline. The horizontal stabilizers are mounted at zero degrees incidence relative to the aft duct centerline. These stabilizers rotate with the aft ducts during transition.

Aircraft control is provided by conventional flap control surfaces located in the exit planes of all four ducts and by differential thrust variation between pairs of ducts through propeller blade pitch change. The elevon control surfaces provide pitch and roll control in level flight and yaw control in hover. Thrust variation between pairs of ducts is used to provide pitch and roll control in hovering; thrust variation between the left and right ducts provides yaw control in level flight. The elevon deflections and propeller blade angle travels used for control about each axis during transition are varied as a function of duct angle to minimize roll due to yaw control, and yaw due to roll control. The X-22A design provides full STOL as well as VTOL operating capabilities. The bounding conditions for a specified STOL configuration are shown in Figure 17, the transition flight envelope. All controls are operated by dual irreversible powered systems, which are designed to meet control specifications for normal operation and with a single soft failure.

Primary system artificial feel is provided by means of a q sensitive electrohydraulic system with a backup spring system.

An automatic stability augmentation system (SAS) is provided in all three axes by means of dual hydraulically powered systems and appropriate electronics. The SAS is included in the PCI evaluation.

A variable stability control system is provided in addition to the primary hydromechanical system. Its purpose is to provide the capability to simulate other V/STOL aircraft, to evaluate handling qualities requirements, define optimum and minimum VTOL handling qualities and simulate changes in the aircraft to evaluate different concepts and modifications. The variable stability system is not included in the PCI evaluation.

(2) Flight Control System

The primary flight control system is a type III power operated system designed to meet the requirements of MIL-F-18372. It is a dual hydraulic power system consisting of two basically independent single hydraulic systems simultaneously driven by a single mechanical system consisting of push-pull tubes, bellcranks, rotating shafts and gearboxes. Figure 5 is a single line schematic showing the basic system using symbolic representation. The control systems used for the various flight modes are:

	Hover	Transition	Level Flight
Pitch	Thrust Modulation (T.M.)	T.M. & Elevons	Elevons
Roll	T.M.	T.M. & Elevons	Elevons
Yaw	Elevons	T.M. & Elevons	T.M.

Change in control from T.M. to elevon for the various modes of flight is accomplished by means of variable ratio bellcranks and lockout bellcranks driven and positioned by duct angular position.

As shown in Figures 10, 11, and 12, both elevon and thrust modulation control are mixed in the correct proportion when the ducts are at angular locations between those used for hover and level flight. Mixing levers are used in the control system to provide the correct direction of control to the four ducts during all modes of flight. The propeller pitch and the elevon controls are operated by dual hydraulic power systems.

(a) Hydraulic System

The hydraulic system used on the X-22A VTOL airplane is a Type II, class 3000 psi system designed in accordance with MIL-H-5440B, using fluid conforming to MIL-H-5606.

Two separate and independent hydraulic systems are provided (see Figure 6). The primary system provides power to the flight control system, while the secondary system supplies redundant power to the flight control system plus power for landing gear actuation and for the variable stability system.

Each system is supplied hydraulic pressure by a variable volume, pressure compensated pump, driven off the accessory gearbox. The pumps are rotated at 6000 rpm when the engines are operated at full military power. At this engine speed, the rated output of each pump is 22.5 gpm and the rated pressure is 3000 psi.

(b) Electrical Systems

The electrohydraulic portion of the primary artificial feel and trim system and the dual stability augmentation systems is designed to operate from 28 volts d.c. and 115/200 volts 400 cps 3-phase power in accordance with MIL-STD-704 Category B. A block diagram of the electrical power system as it pertains to the feel and trim system is shown in Figure 7.

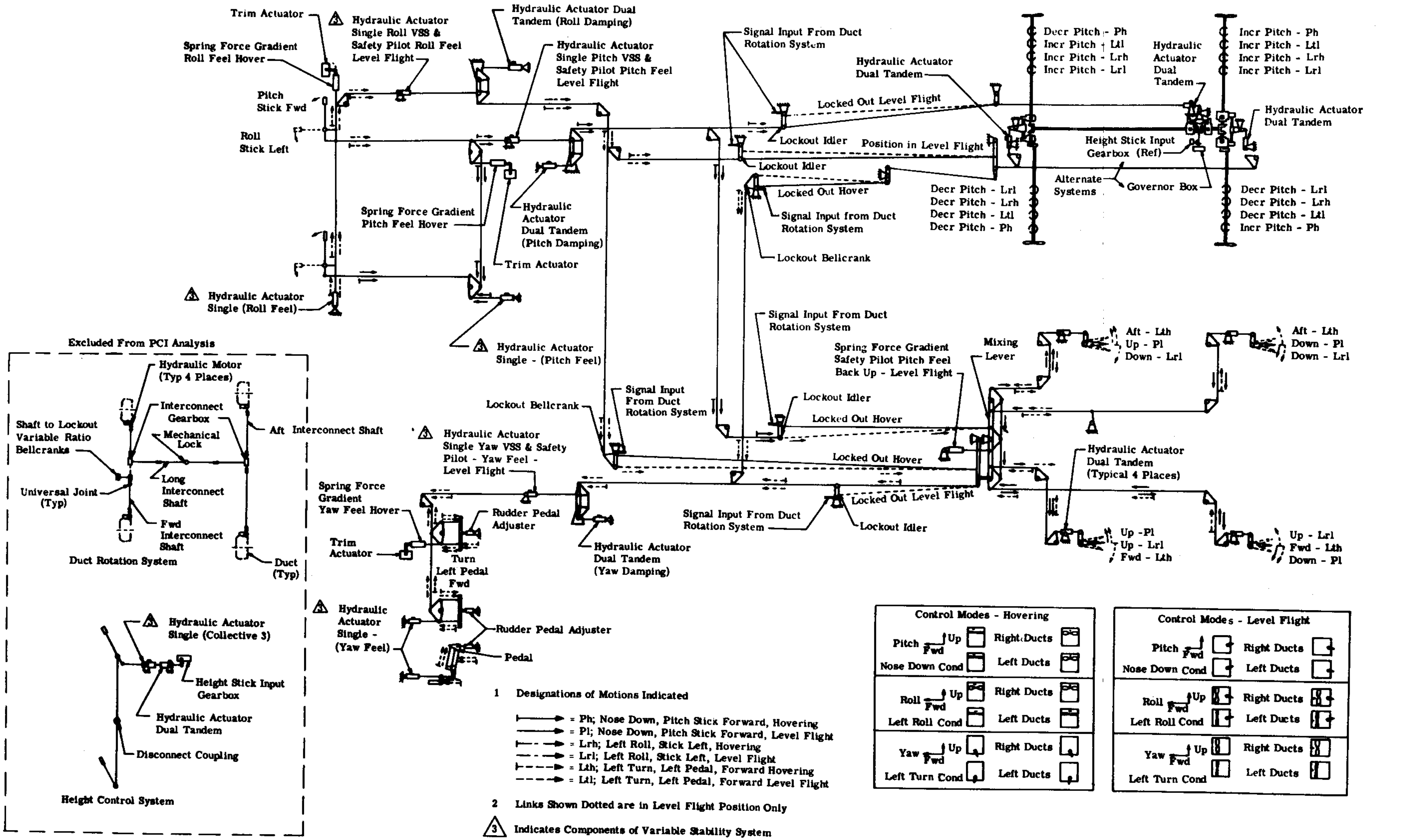


Figure 5. Flight Control System Layout

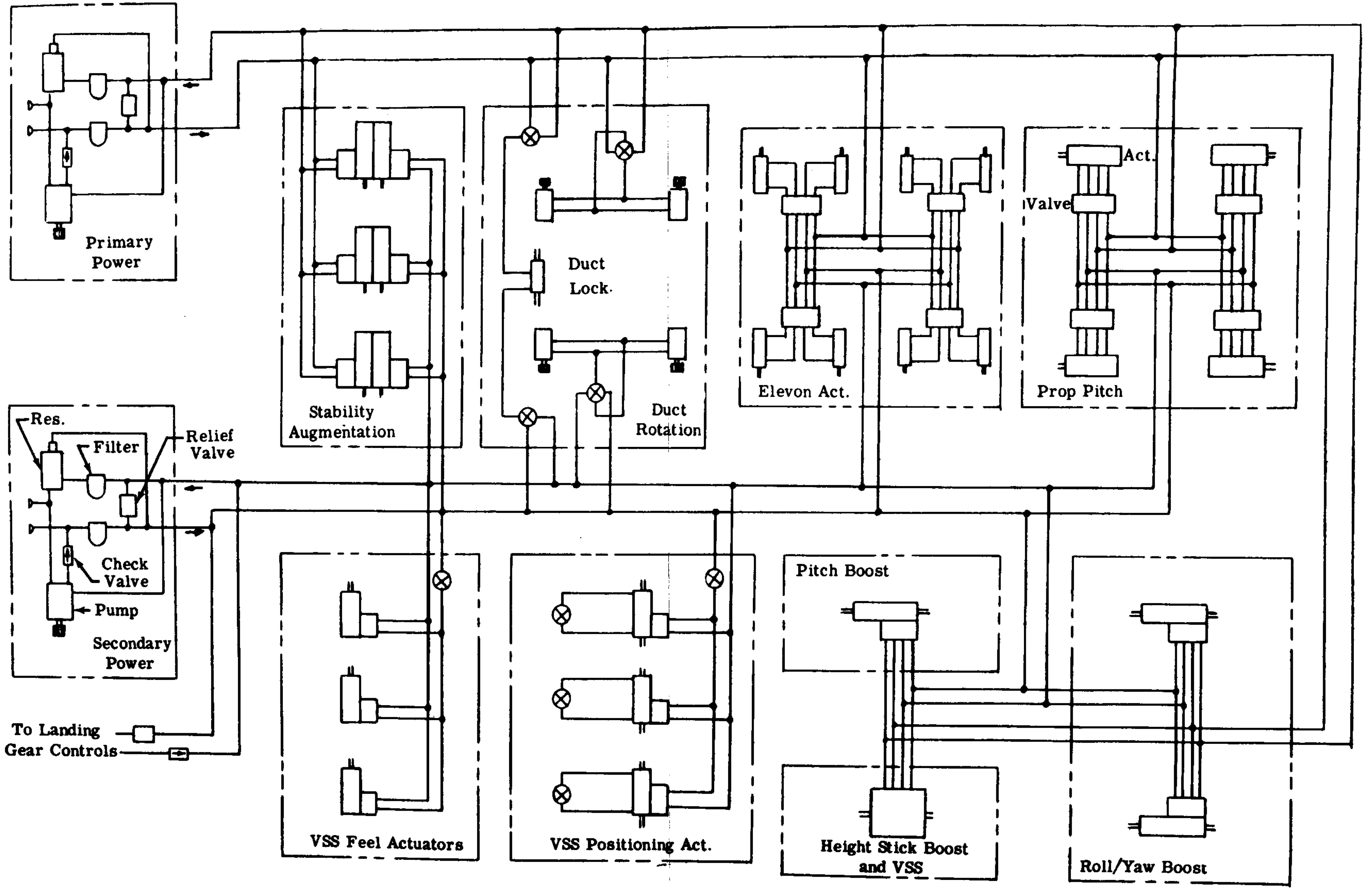


Figure 6. Hydraulic System Schematic

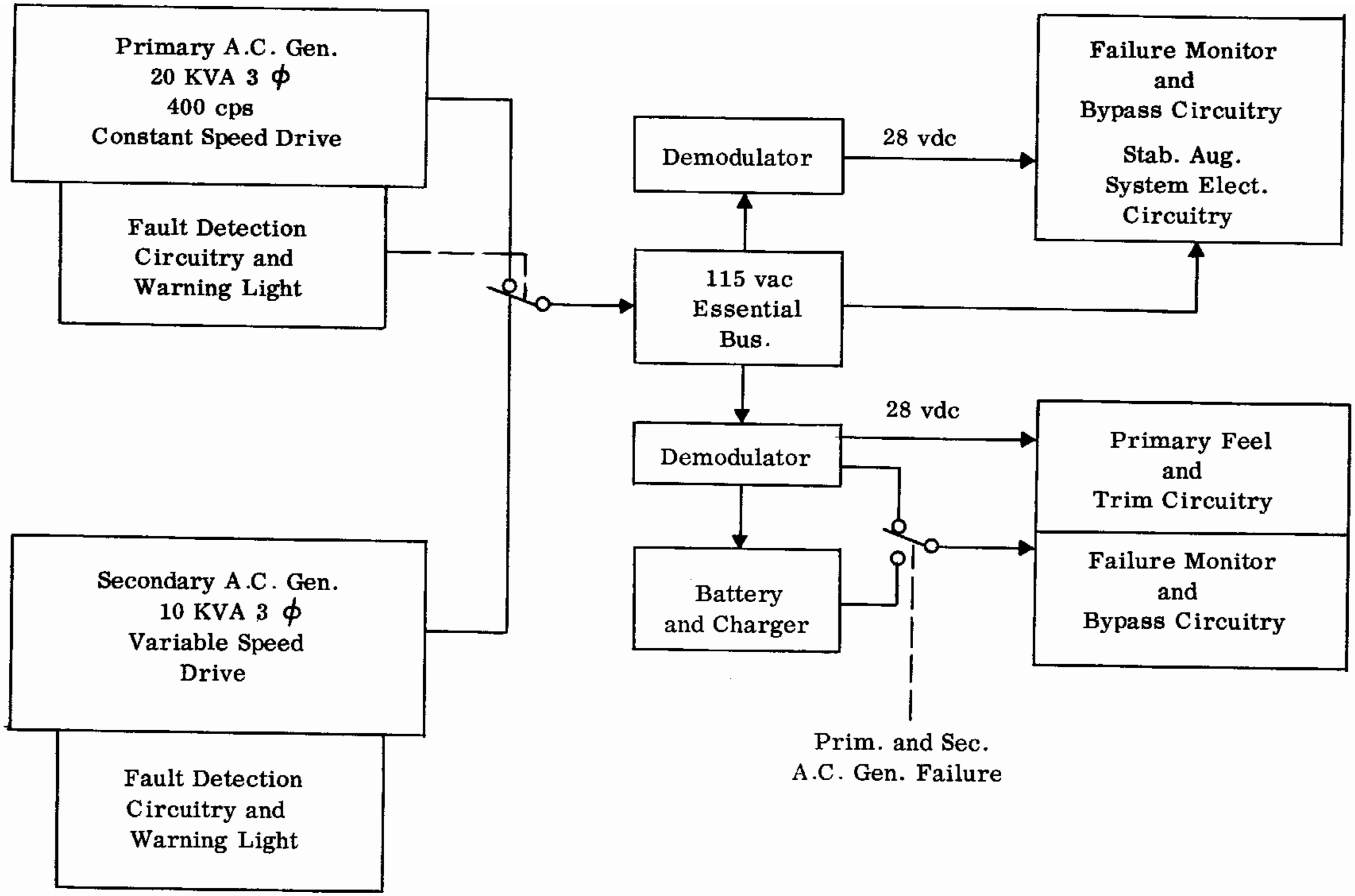


Figure 7. Electrical Power Generation System and Distribution to Feel Trim and Stability Augmentation Systems

The system contains a standby battery, which supplies emergency power to the primary feel/trim system and SAS pitch axis high speed lockouts for periods of at least sixty seconds without power from the aircraft primary electrical system.

(c) Feel and Trim System

The artificial feel and trim system has two modes of operation: an electrohydraulic primary mode, and a mechanical backup mode.

The electrohydraulic primary system, shown in Figure 8, utilizes strain gauges located on the safety pilot's stick and pedals to generate force command signals. These signals are transmitted through electronic amplification and gain shaping circuitry to power amplifiers which drive the variable stability system position actuators. These actuators in turn position the respective pilot controls to reflect the applied forces.

The feel forces in the pitch and roll axes are varied as functions of equivalent airspeed and feel forces in the yaw axis are varied as a function of duct angle.

Pitch and roll feel forces are trimmed at rates measured in pounds per second and yaw forces are trimmed in terms of pedal displacement. Fore-aft actuation of a "coolie hat" switch on the top of the stick trims out pitch feel forces while lateral actuation trims out roll forces. Rotation of a potentiometer on the pilot's left side panel trims yaw forces.

A monitor circuit is provided to automatically detect any electrical malfunction in the primary feel/trim system. Should a failure occur during hover or transition flight (ducts unlocked), the primary feel/trim system will "fail soft"; i.e., the VSS position actuators will automatically go into bypass, placing the system in the mechanical backup mode of operation. Should a failure occur in conventional flight with the ducts locked in the down position, the stick and pedals will be locked in the position they were in when the failure was detected. The purpose for locking the controls in the event of conventional flight failure is to give the pilot time to adjust to the greatly reduced force gradients of the backup system such that he will not prematurely overdrive the controls resulting in overstressing of the aircraft. After lockup, the pilot must actuate the emergency override switch to put the VSS position actuators in bypass.

Backup system feel is provided by hover springs in all three axes with instantaneous trim provided through mechanical clutches. Actuation of the "coolie hat" in any direction will trim out all three axes simultaneously with the single exception of the pitch standby feel spring. The standby feel spring - programmed with duct angle - is designed to provide a longitudinal stick force of 3 pounds per g at maximum forward velocity.

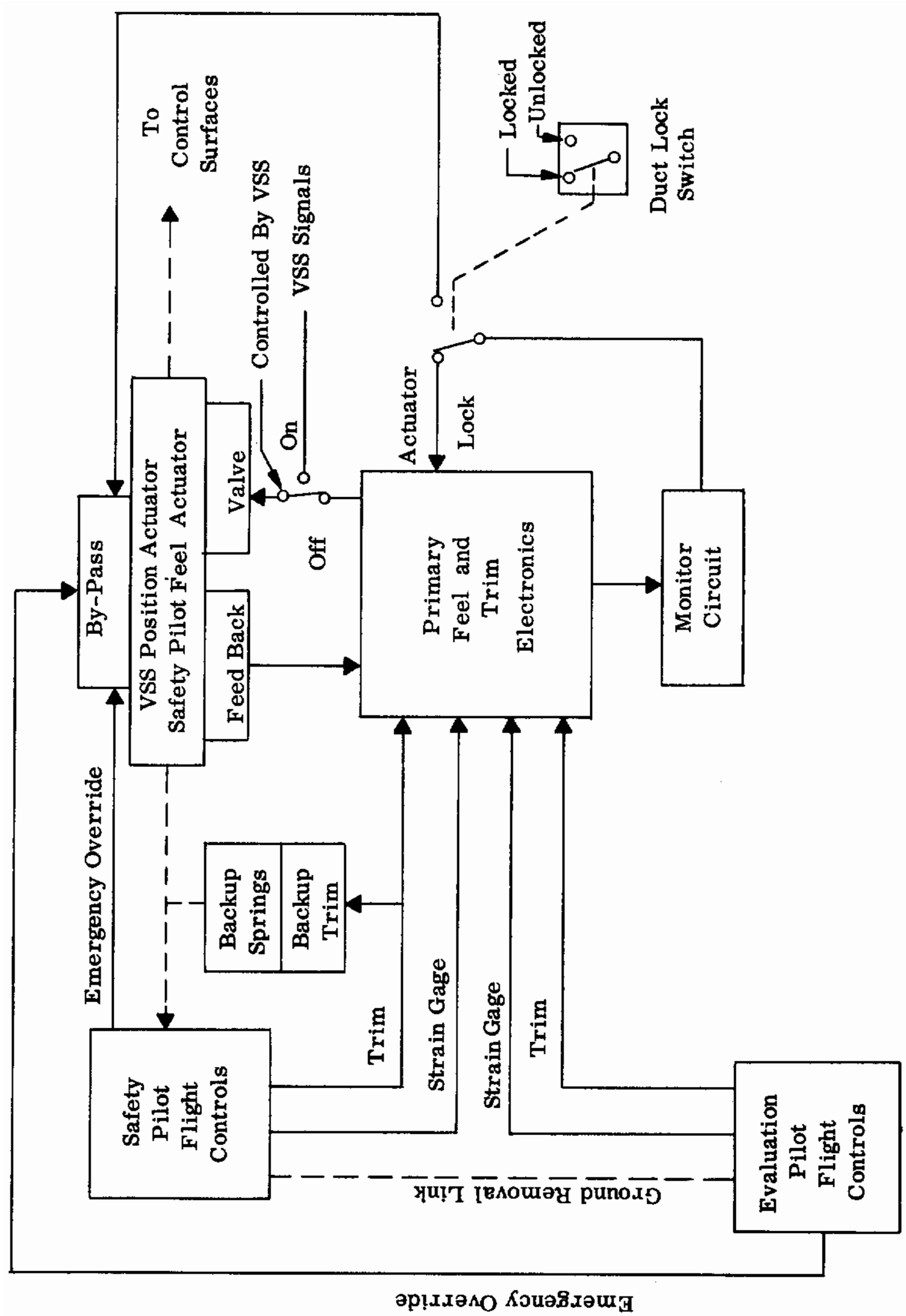


Figure 8. Primary Artificial Feel and Trim System Block Diagram

(d) Stability Augmentation System

Stability augmentation is provided in roll, yaw, and pitch. A representative block diagram of SAS operation is shown in Figure 9. In each axis the actuator assembly consists of a pair of actuators whose output positions are additive. The output of the actuators is connected to a common output linkage. Electrohydraulic servovalves control the positioning of the actuators. Electrical transducers, internally mounted, provide position feedback for each actuator. The actuators can be centered and locked individually by deenergizing an integral solenoid valve in the pressure line. Centering forces are provided by mechanical springs and locking is accomplished by a spring-loaded detent. Locking of one section of the assembly does not prevent the operation of the other section. A switch which is actuated by the locking mechanism provides an indication when the actuator is locked.

Fail-safe features are incorporated in the pitch, roll and yaw axes of the SAS such that a single malfunction or failure in any axis of the system will not affect safety of flight. A malfunction or failure of one channel in the pitch, roll, or yaw axes will not cause the other channel of the SAS to shut down or be rendered inoperative. Electrical signals from the two transducers in the dual servo actuator in the pitch, roll, and yaw axes are compared in a differential amplifier and detector. Any difference between transducers indicates impairment of the SAS function, which is signalled by a warning light. The warning light is located on the annunicator panel, providing visual warning to the pilot when a malfunction occurs in the SAS in any of the three axes.

An automatic SAS shutdown is incorporated in the pitch axis only to prevent a hardover electrical signal malfunction from causing a catastrophic failure to the aircraft when flying at speeds in excess of 200 knots. At such speeds, the maximum output (authority) of the SAS in the pitch axis may otherwise exceed the authority limitations of MIL-H-8501A.

(e) Mechanical System

Mechanical mixing levers are employed in the control system to integrate pitch, roll, and yaw commands for different combinations of signals from attitude stick and rudder pedals to the elevons and propeller pitch change control. The mixing levers take the pilot inputs and command the affected system or systems to move in the correct direction differentially in pairs, right and left and fore and aft for all modes of flight.

The variable ratio bellcranks are basically screw and nut type commanded by duct angle. Each varies the amount of output signal from the pilot control input from full command in some cases to lockout in other cases. These output signals go to the mixing levers which determine the direction the signal shall take or which system shall move.

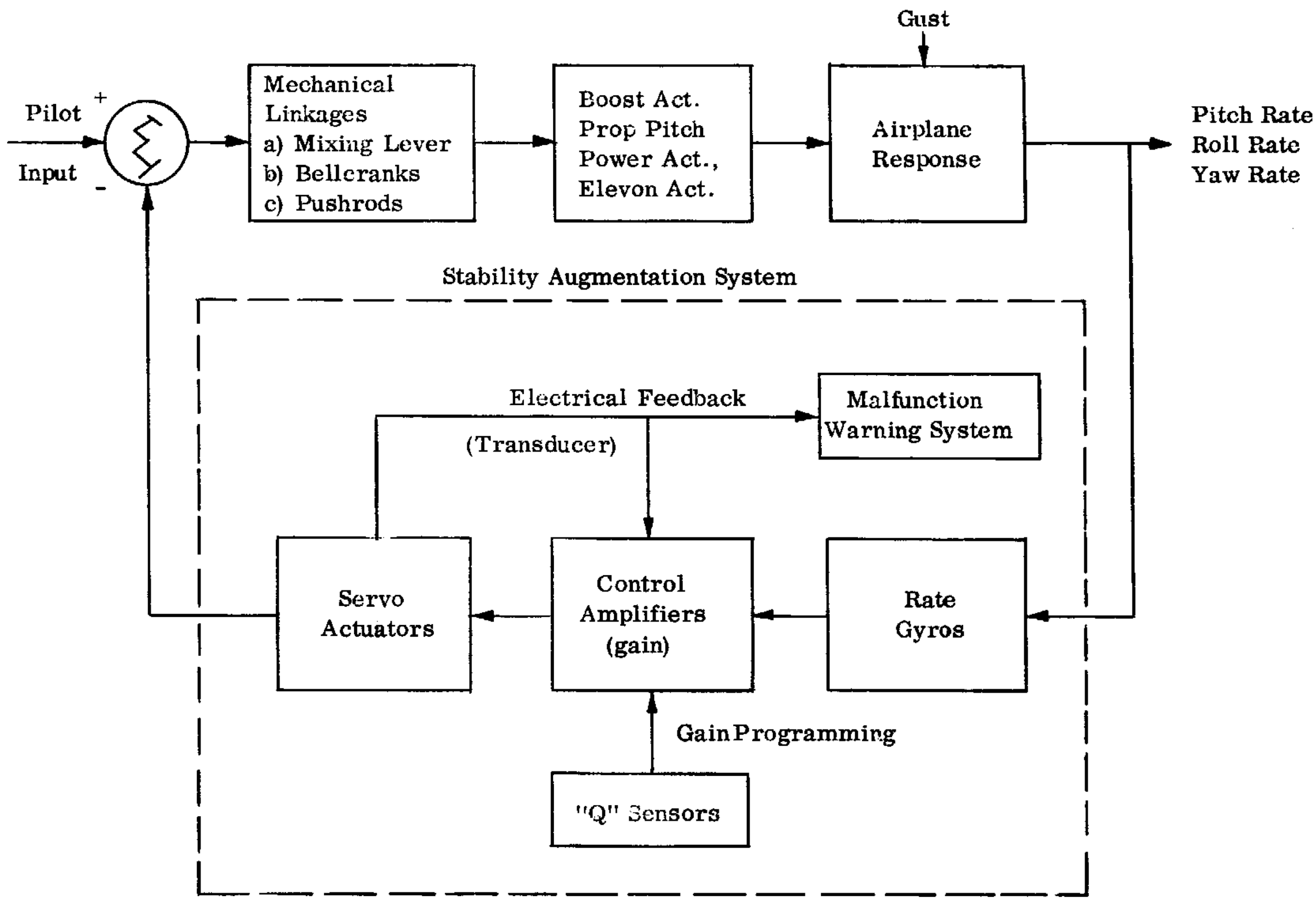


Figure 9. Stability Augmentation System Block Diagram

The variable ratio bellcrank ratio change is commanded by duct position by means of mechanical shafts and gearboxes driven from the duct rotation system.

The maximum elevon and propeller pitch angles for all duct positions are shown in Figures 10, 11, and 12. Maximum travel of all controls is established by stops at the sticks and rudder pedals in conjunction with the position of the variable ratio bellcranks.

(f) Propeller Control

Propeller pitch control is accomplished by means of a mechanical rotating drive shaft system which drives a signal converter located in each propeller hub resulting in blade pitch angle changes both collectively and differentially - between the four propeller systems.

Two types of signals can be introduced into this system: differential signals by means of attitude stick and rudder pedal; and collective signals by means of the master governor. The differential signals are introduced from the attitude stick and rudder pedals by means of a push-pull tube system going through variable ratio bellcranks and mixing levers which establish the correct signal and magnitude of signal to the appropriate boost actuator and gearbox. There are two gearboxes and boosters in the differential roll-yaw system. One is located in the aft end of the aircraft and the other in the forward fuselage. The differential pitch attitude input system requires one gearbox and booster located in the aft fuselage area. The boosters are dual hydraulic servo actuators.

Collective pitch signals are introduced directly into the propeller blade control system through a gearbox driven by the master governor. The master governor holds rpm of the propeller constant by change of blade pitch angle for varying power commands from the throttle controls in the cockpit.

(g) Elevon Control

The elevon actuation system is located in the aft portion of each propeller hub. The assembly consists of an input and feedback linkage, a tandem servovalve, and a pair of push-pull actuators. The mechanical inputs are fed into the tandem spool valve, which modulates and directs the flow of hydraulic fluid into the actuators according to the degree and direction of displacement of the spool from the neutral position. To eliminate structural compliance problems in a one-system-out condition, one hydraulic system supplies the retract side of both actuators and the other hydraulic system supplies the extend sides. The output motions of actuators rotate the elevon about its pivot point. Elevon motion is fed back through a set of linkages and is algebraically summed with the input to reposition the servovalve to neutral when the steady state elevon position is reached. The servovalve incorporates a load pressure feedback mechanism which acts in a direction to close the servovalve with increasing load pressure. This provides dynamic stability to the system.

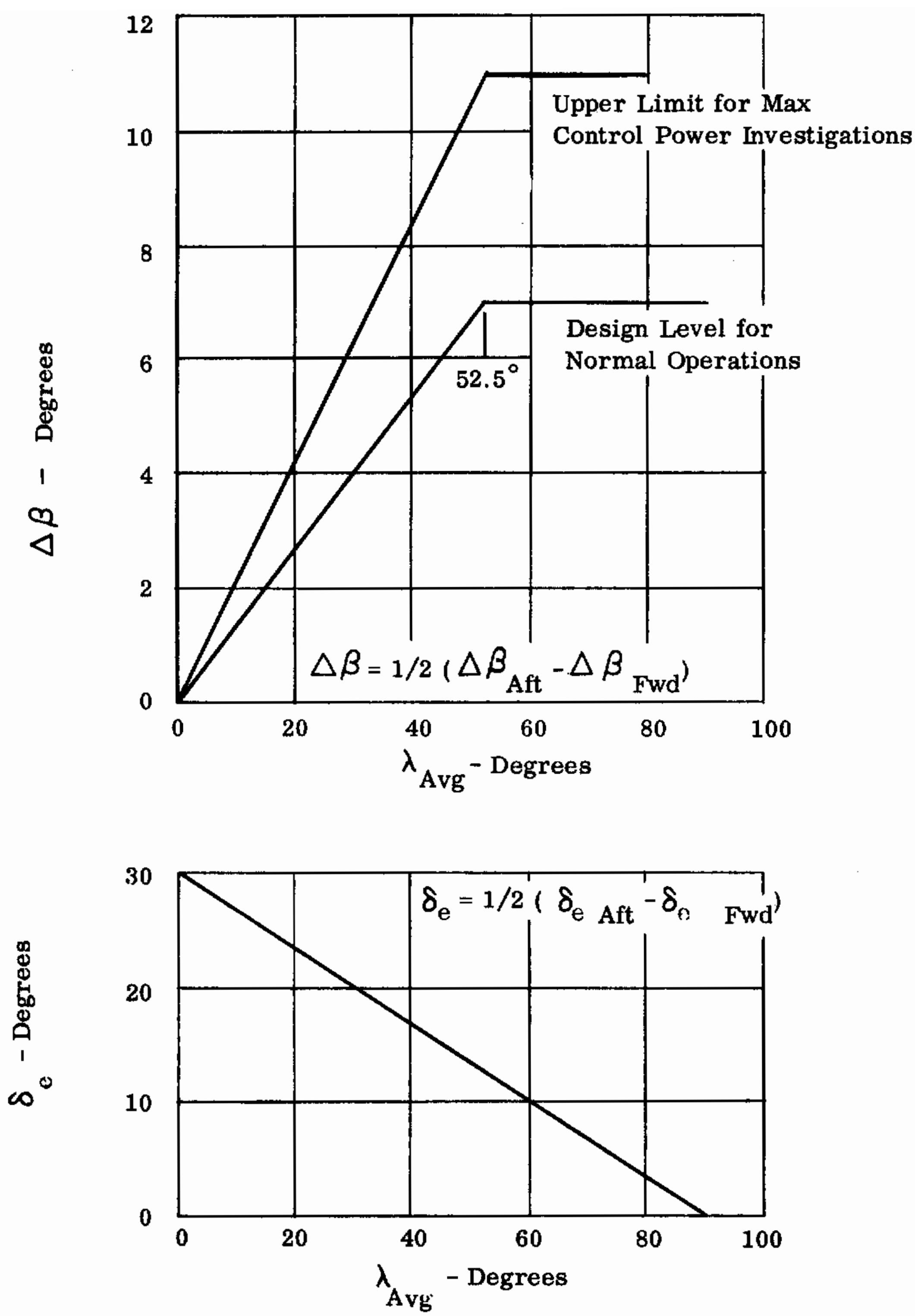


Figure 10. Maximum Pitch Control Deflections - Deflections Shown are Full Forward or Aft Stick (±5.6 in.)

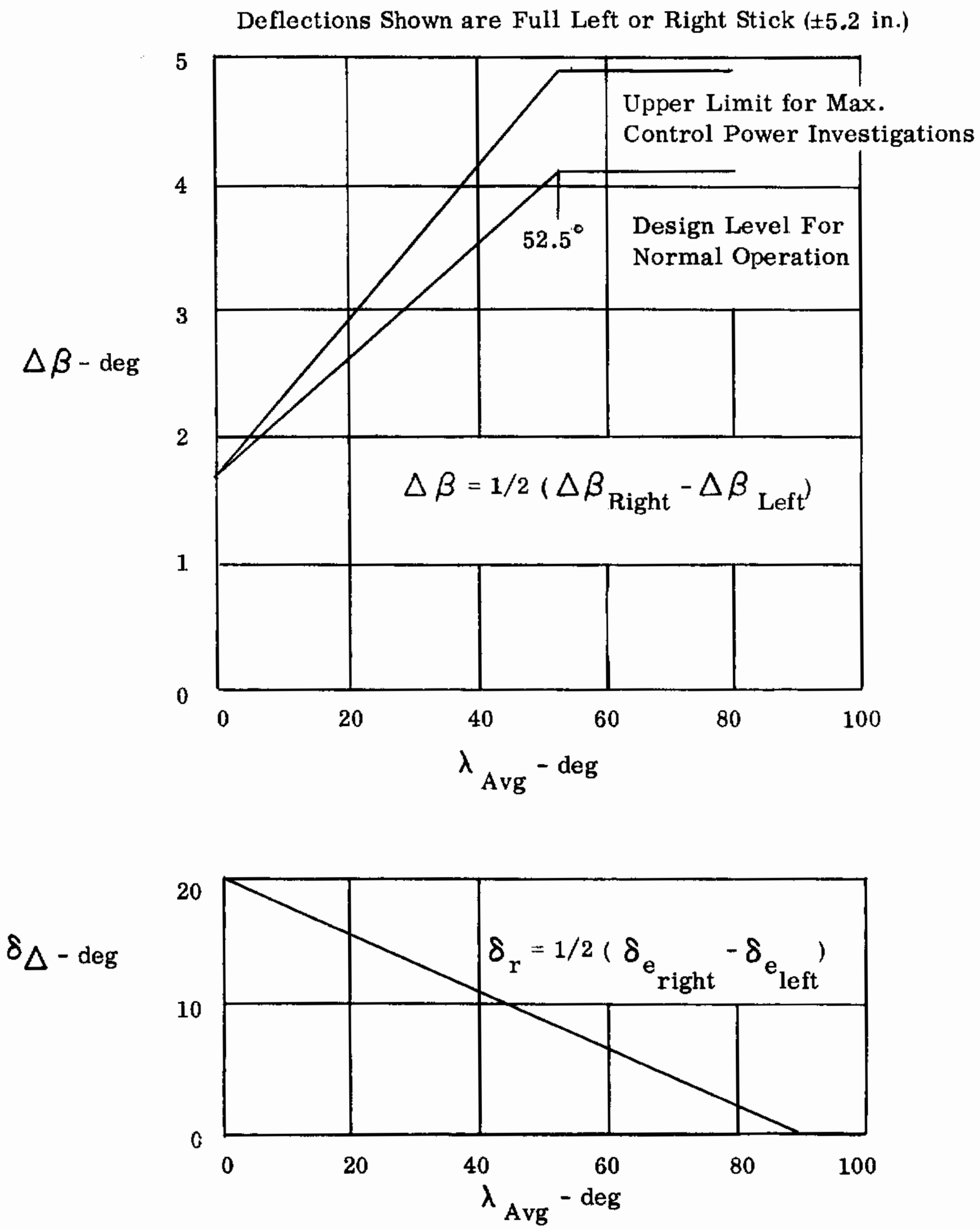


Figure 11. Maximum Roll Control Deflections

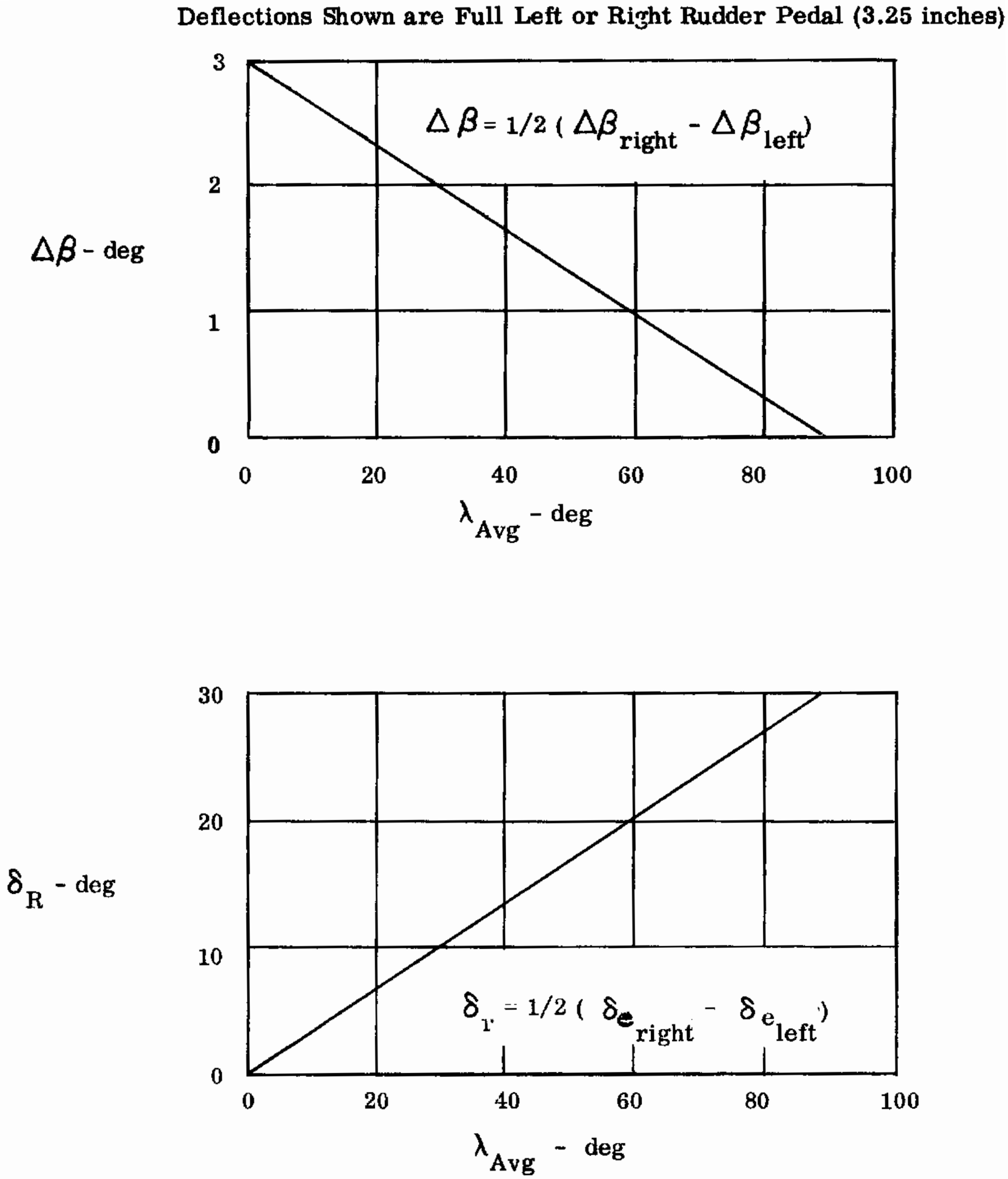


Figure 12. Maximum Yaw Control Deflections

b. Layout of Pilot Controls and Displays

Basic flight controls (stick, rudder pedals, throttle, rpm) have been located in the conventional manner (Figure 13). The pitch/roll trim - coolie hat switch - is located atop the stick and a directional trim position potentiometer is on the pilot's left hand control panel. The duct rotation switch is located on the right side of the in-board throttle. Emergency control switches have been located for easy pilot access.

Cockpit instrumentation is located as shown in Figure 13. Instruments are arranged in functional groups in a manner to best facilitate pilot scanning. The major functional groups are: primary flight displays; engine instruments, failure warning displays (annunciator panel); hydraulic system and landing gear displays; variable stability system control-display panel; and radio communications and navigation.

Instrument arrangement is based on the ground rule that the more important parameters shall be placed in the generally accepted (conventional) locations.

c. Mission Description

A VTOL utility transport, radius of action flight was selected as the basic mission model for the PCI analysis. The mission is flown under VFR conditions by a single pilot.

The mission model was broken down into five separate and unique mission segments (as shown in Figure 14) on the basis of the nature and anticipated degree of difficulty of the control tasks. Sequentially, the five mission segments are defined as follows:

MS1 - Vertical liftoff, climb to 50 feet, and momentary hover (preliminary to start of transition) in the presence of a steady 20/25 knot wind.

MS2 - Transition from hover to conventional flight with a climb to 500 feet cruise altitude.

MS3 - Cruise at constant altitude to vicinity of landing site.

MS4 - Descent from cruise altitude and transition to hover.

MS5 - Landing site positional maneuvering in steady wind and vertical descent to touchdown.

d. Vehicle Control and Performance Characteristics

The general functional requirements governing X-22A maneuverability, control, and stability are defined in the military specifications covering handling qualities (MIL-H-8501A and MIL-F-8785), flight control system design (MIL-F-18372) and flight demonstration (MIL-D-8708A). The more significant characteristics of the vehicle may be summarized as follows.

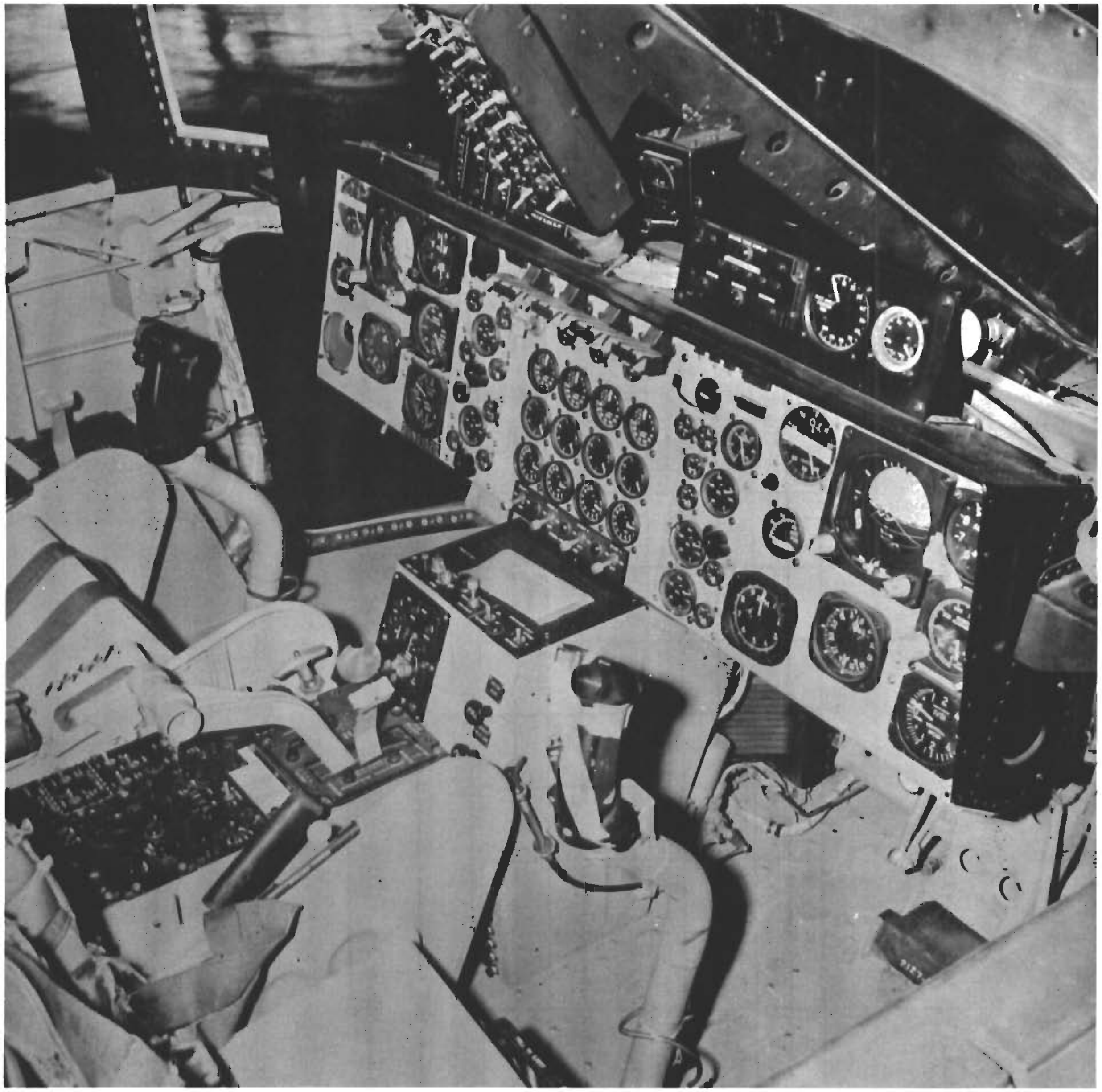


Figure 13. X-22A Cockpit Instrument Panels

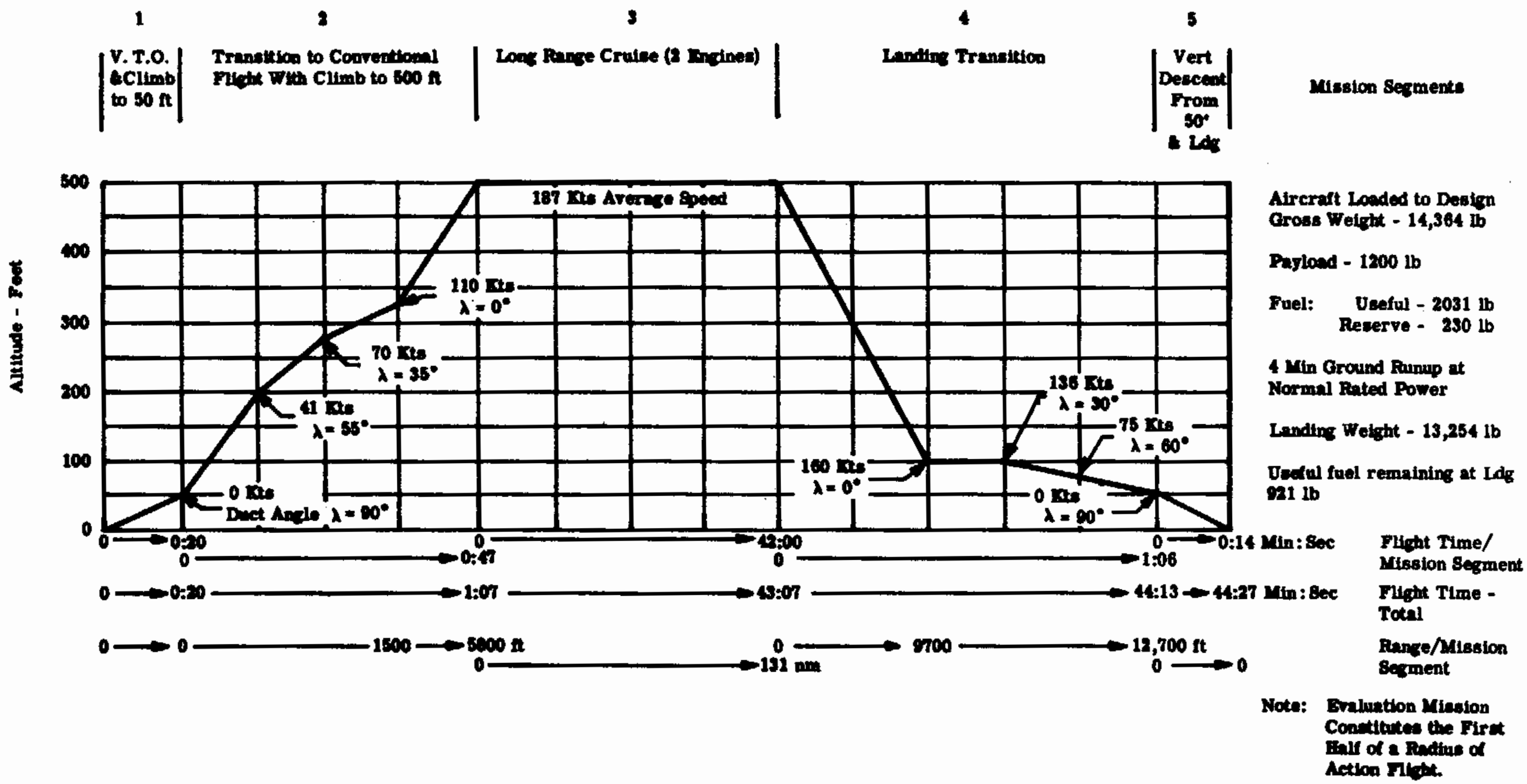


Figure 14. Evaluation(Utility Transport) Mission Profile

Throughout hover flight and while in the lower speed range of transition ($V < 60$ knots) the X-22A has very low natural stability both longitudinally and laterally/directionally. The natural stability gradually increases such that as velocity increases to 160 knots and higher the basic vehicle damping has reached satisfactory levels. The natural vehicle damping has been augmented by a three axis, dual channel SAS with gain levels programmed as functions of equivalent air speed in order to provide optimum damping with both channels functioning and acceptable levels of damping with single channel operation.

Considerable margins of control power, above the levels required for the most severe trim conditions, have been provided about all three axes. This capability was dictated by the V/STOL research requirements specified for the X-22A. Although this surplus of control authority may generally be appreciated by the pilots, it also raises some serious difficulties in regard to minimizing the effects of systems failures which tend to drive control surfaces (propeller blade and elevon) hardover. A number of monitor functions and component redundancies have been incorporated in the control systems design to prevent the occurrence of catastrophic single failures.

e. Preliminary Pilot-Controller Task Description for Normal and Emergency Operation

During vertical liftoff, hover and translation flight, the primary pilot task involves 3-axis maneuvering with respect to a terrain-restricted takeoff or landing site. Maneuver control is obtained by the modulation of vehicle attitudes (stick and rudder pedals) coupled with throttle control of engine power. Considerable pilot attention is required in this mode of control as attitude trim changes rapidly with translation velocities.

Throughout takeoff transition flight, the primary effort is one of longitudinal control. Modulation of duct attitude and engine power couples with the increasing aerodynamic effects to generate large changes in pitch trim (Figure 15) in an atmosphere of near neutral static stability as the vehicle accelerates. Landing transition presents a similar type of control problem with the trim requirement varying in a reverse order.

Throughout hover, translation and transition, pilot cues are obtained by out-of-the-window observation of surrounding terrain with a minimum of attention devoted to cockpit instruments and secondary tasks.

The flight control system has been designed to provide adequate margins of stability and control in the event of a single active electrical failure or an active failure plus a hidden failure. Thus, in the event of a primary feel/trim system failure, the monitor circuit automatically switches to the mechanical backup mode of control. This situation will result in a minor change in the pitch mode of control - as the pitch backup spring is not trimmable - and no change in roll or yaw.

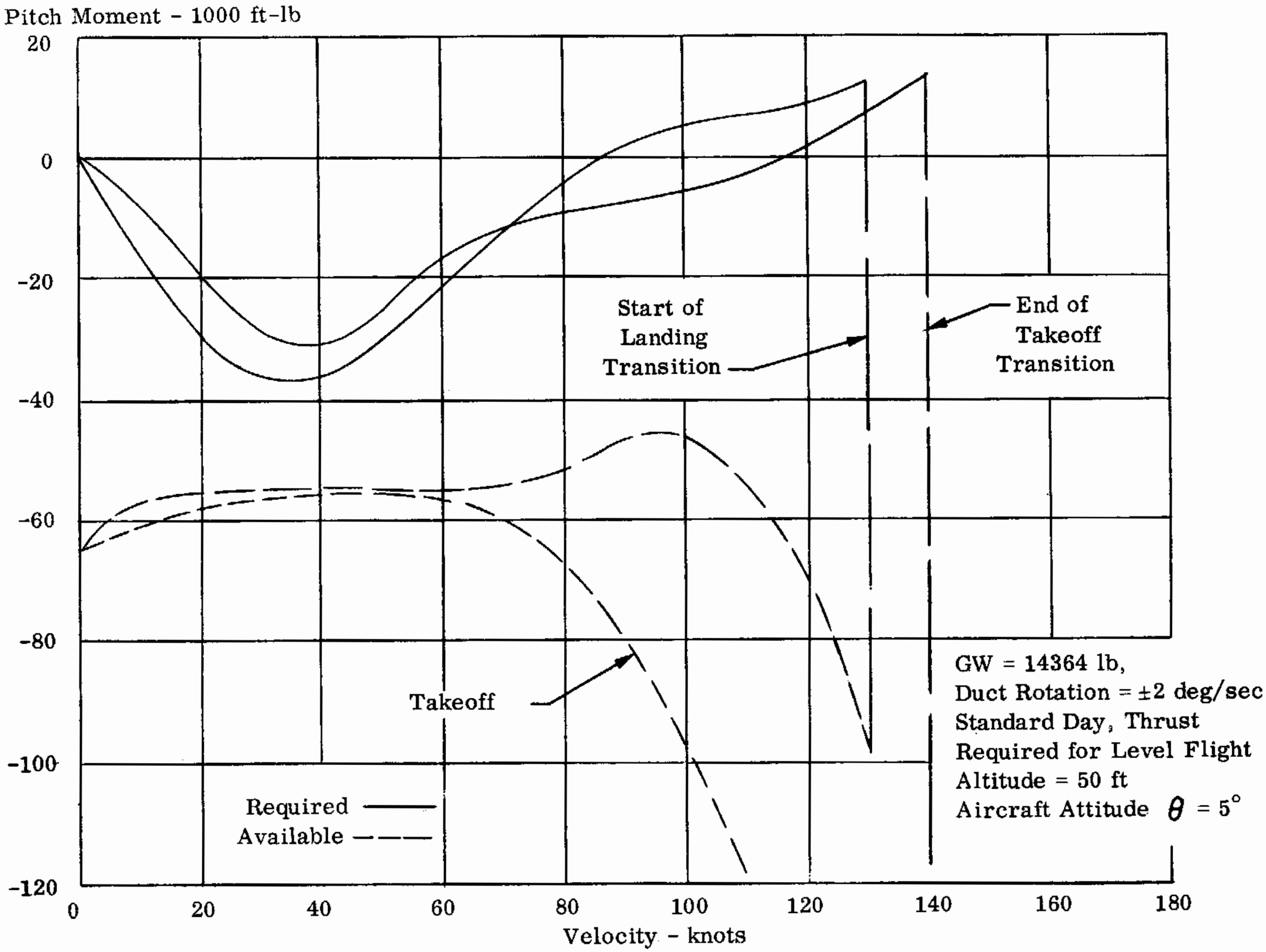


Figure 15. Pitching Moments Required and Available for Typical Landing and Takeoff Transitions

A single channel SAS failure will result in a 50% decrease in damping augmentation about the associated axis. Generally in the event of a single SAS failure, it would not be necessary to shutdown the respective system, thus reducing the augmentation in all axes by 50%. However, for the purposes of this study, it was assumed that a single failure would always result in pilot shutdown of the respective stability augmentation system. Although this effected a conservative bias in mission analysis results, it tended to simplify the mathematical model representation of the SAS in that a failure in one of the dual hydraulic power supplies or in one of the 1 psi dynamic pressure transducer-servo units (SAS gain change drive) would result in complete loss of one of the dual stability augmentation systems. Were this an actual "on line" design application of PCI, this simplifying restriction would be removed in the next iteration.

Flight control of the X-22A does not differ appreciably from normal transport category aircraft in the conventional mode of flight. However, as a result of increasingly high elevon control surface effectiveness as conventional flight velocity increases, steps must be taken to protect against overstressing the aircraft in the event of feel/trim or SAS failures.

In conventional flight, the feel/trim monitor system has been designed to lock pilot controls in their positions at the time of failure detection. After assessing the situation, the pilot actuates the emergency override switch placing the VSS position actuators in bypass and thus resorting to the mechanical backup system.

SAS hardover failures in pitch are protected against by dualized centering and lockout of the pitch axis SAS actuators at velocities above 200 knots (SAS gains are phased to zero by 160 knots).

In all modes of flight, single hydraulic failures are protected against by dualized power sources, parallel distribution system, and dual actuators for propeller boost, propeller control, and elevon control.

f. Establishment of Pilot-Controller Reliability Goals

For the purpose of this study, a reliability goal - probability of successful mission accomplishment - of 0.96 for one hour of flight was established for the X-22A flight control system. The setting of this goal has taken into account the fact that the aircraft under study is primarily a research vehicle even though the evaluation mission has been tailored along operational lines. Also, the simplifying assumptions made in limiting the scope of the failure modes and rates analyses resulted in considerable conservatism in estimating subsystems reliability.

As noted in Section VI, Concept Improvements, the PCI procedure has been expanded to include the analysis of probability of successful aircraft recovery. Should this additional analysis be desired, it will of course be necessary to define an equivalent goal.

3. DEVELOPMENT OF SUCCESS DIAGRAMS - PCI PHASE 2

The success diagram is a block diagram representing all the functional components of the flight control system - oriented to show the effect of concurrent failures on control continuity, catastrophic single failures, and pilot work load. Success diagram construction is based primarily on conventional reliability diagram rules with each failure mode represented by a separate block.

The primary purpose of the success diagram is to serve as a reference model for the development of a mathematical (model) representation of the flight control system required in the digital analysis of mission success probability. A more detailed discussion of success diagram philosophy, construction, and usage is included in Section IV.

Because the effects and interrelationships of component failures differ in the several mission segments, three separate success diagrams were constructed to represent the functional relationships within the flight control system of the X-22A throughout the defined mission (refer to Section II.2.a). Figure 16a represents mission segments 1 and 5 - vertical flight and hover. Figure 16b represents the takeoff and landing transitions, and Figure 16c represents the conventional flight portion of the defined mission.

Pilot reliabilities, which properly should be placed in series with the respective manual backup failure mode reliabilities, were assumed to be 100%. This assumption was dictated by the lack of a state-of-the-art means for the determination of pilot reliabilities in the performance of continuous tracking tasks.

Although development of success diagrams is listed as the second PCI phase, it is apparent that no more than a rough pass at SD construction can be made at this early stage in the PCI application. Construction of the final success diagrams fully and correctly showing the functional relationships between the various control system failure modes must await the completion of PCI phases 3, 5, and 6.

4. PREDICTION OF COMPONENT FAILURE MODES - PCI PHASE 3

Failure modes were determined for each hydromechanical subsystem of the X-22A flight control system. These subsystems were expanded to the component level, and it was at this level that the basic failure mode and effects analyses were conducted. Failure modes were not determined for hydraulic and electrical power generation, primary feel/trim, and stability augmentation, and worst case failures were therefore assumed to account for 100% of the respective failure rates.

For each subsystem component, the probable modes of failure and failure effects were predicted through analysis based on layout and schematic drawings reinforced by bench tests in the case of several of the more critical electronic components.

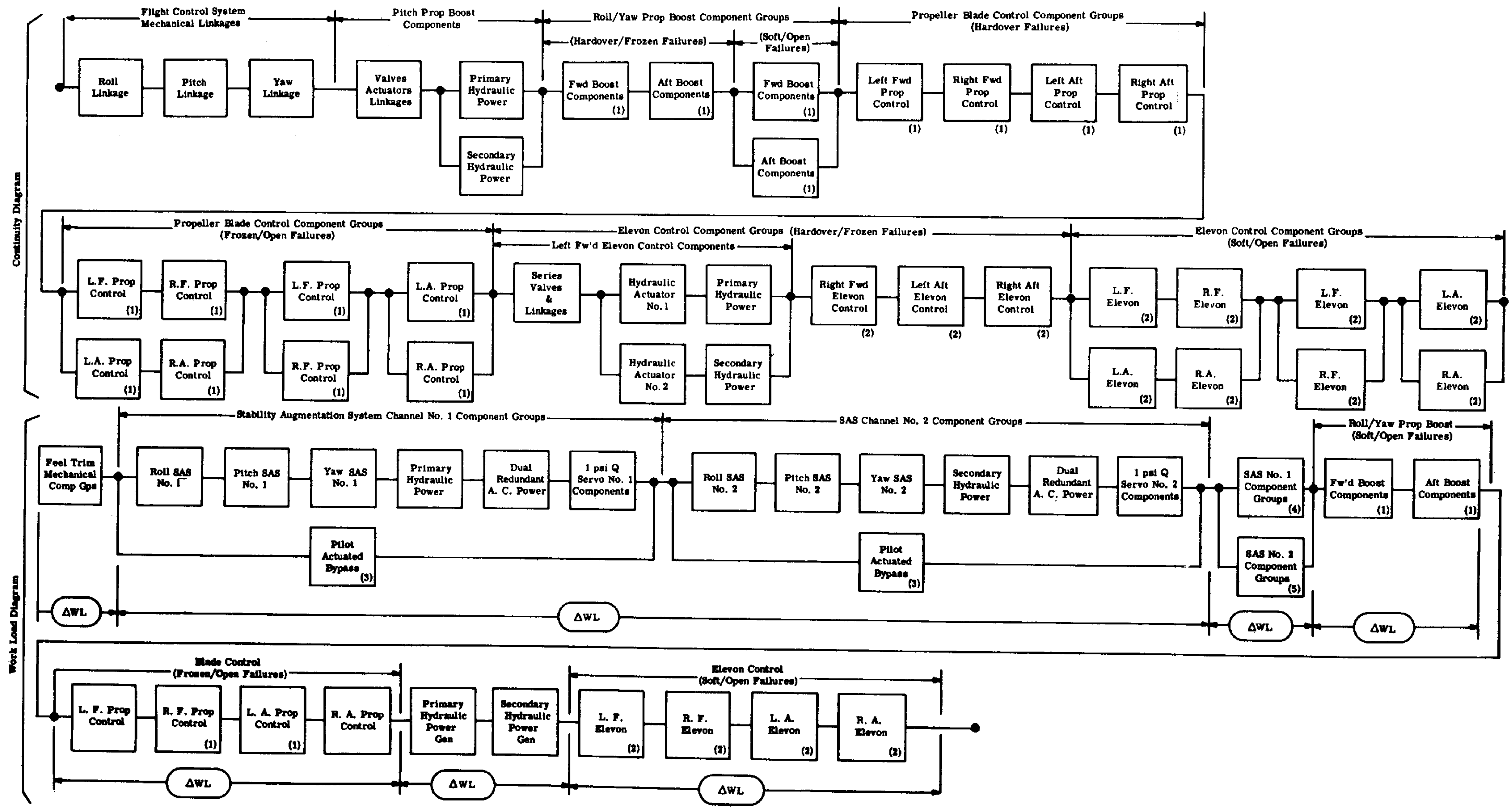


Figure 16a. Success Diagram - Mission Segments 1 and 5

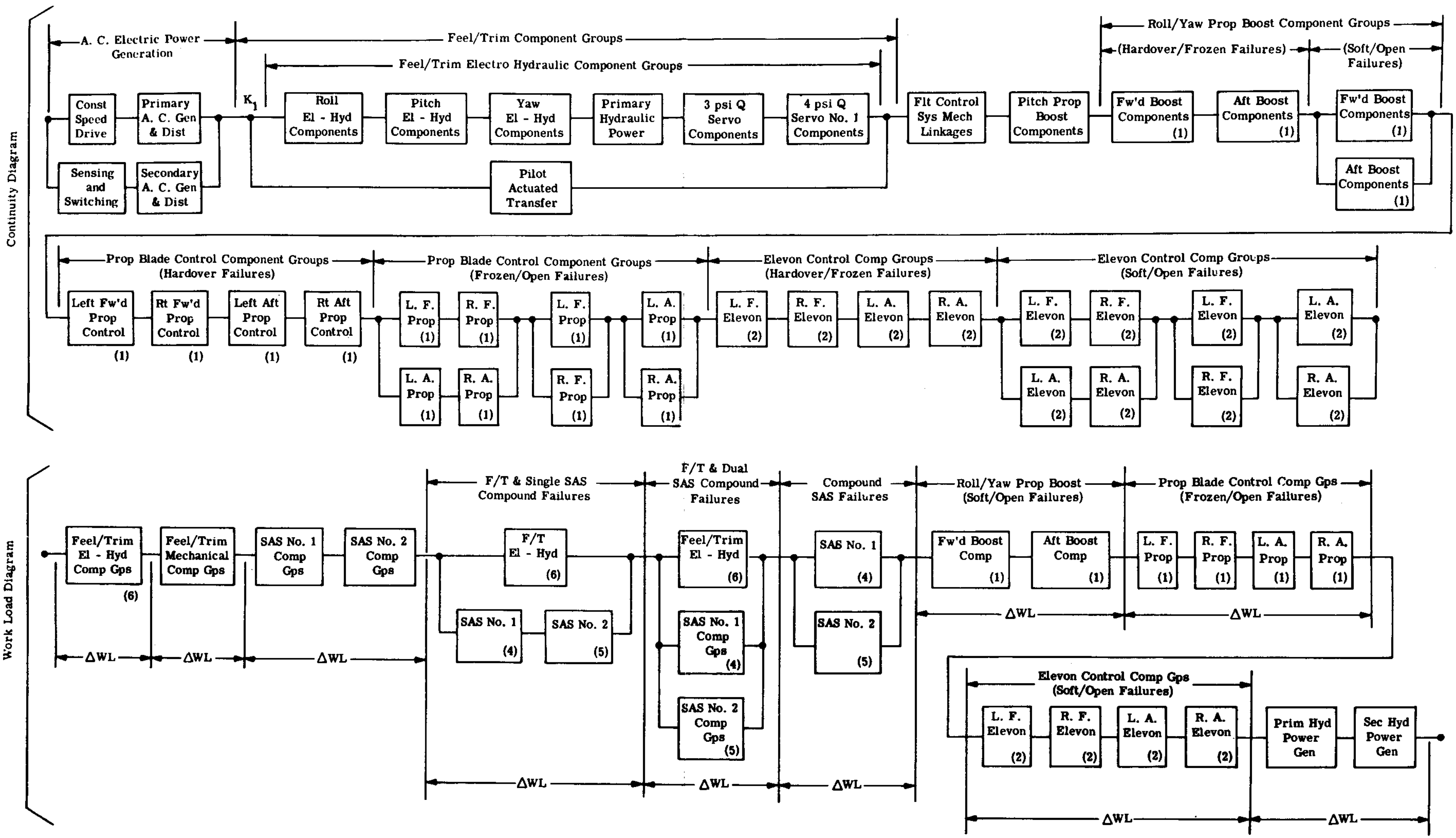
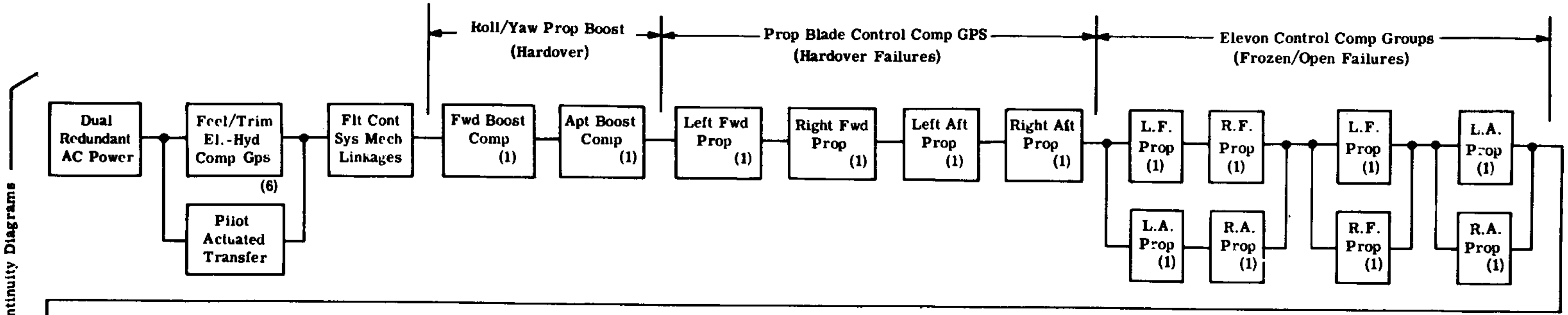
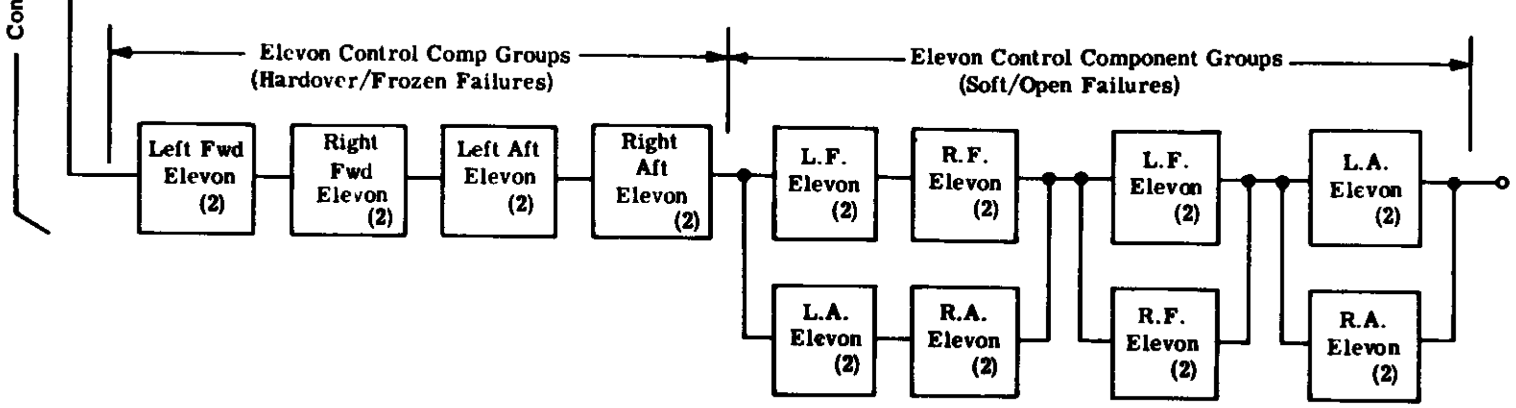


Figure 16b. Success Diagram - Mission Segments 2 and 4





Notes:

(1) Components identical to those shown for Pitch Prop Boost at top of page 33.
(2) Components identical to those shown for Left Fwd Elevon Control on page 33.
(3) Active path in the event of concurrent oscillatory SAS and pilot actuated bypass failures only.
(4) Identical to SAS Channel No. 1 Component Groups as expanded on page 33.
(5) Identical to SAS Channel No. 2 Component Groups as expanded on page 33.
(6) Identical to Feel/Trim El-Hyd Component Groups as expanded at top of page 33.

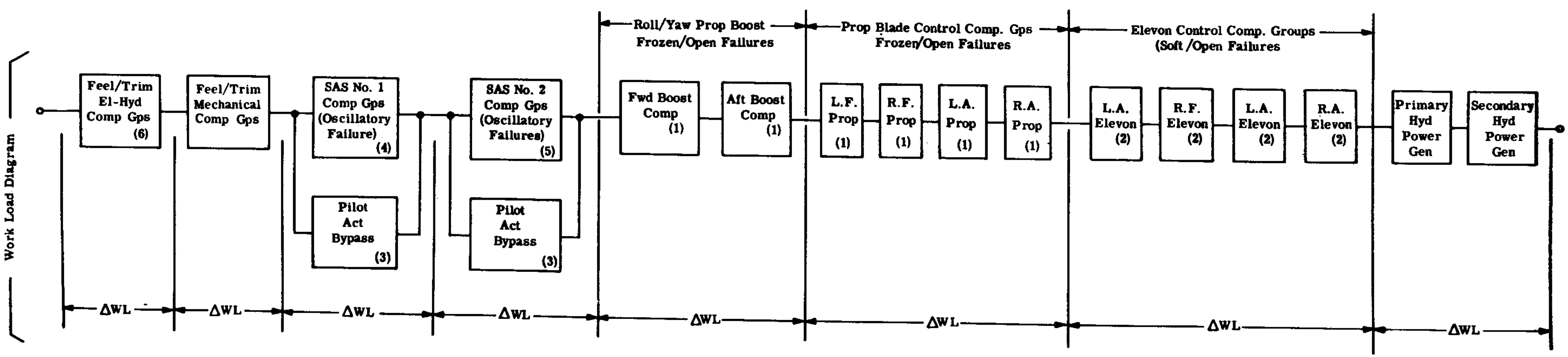


Figure 16c. Success Diagram - Mission Segment 3

Tables IA to II summarize the results of the above analyses of the flight control subsystems and components. The tables also include the predicted generic failure rates for each mode, with the rates distributed by degree of severity (5).

The generic failure rates have been included in these tables in order to present all of the essential reliability parameters in a concise, unified manner.

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(5) Failures classified as:

Catastrophic	-	definitely results in loss of ability to control the vehicle.
Major	-	definitely results in degradation of system's performance with probable effects on pilot's control capabilities.
Minor	-	maximum severity limited to a discernible effect on control response but not of sufficient significance to influence pilot control capabilities.
Trivial	-	exercises no appreciable effect on system operation and pilot response.

TABLE IA

PROPELLER BLADE ANGLE CONTROL ACTUATOR - FAILURE MODE AND EFFECTS ANALYSIS

Component	Units per Assembly	Malfunction	Effect on Component	Failure Effects* & Rates**								
				1 C	2 C	3 M	4 M	5 m	6 m	7 m	8 m	9 N
Pitch Change Linkage Trunion	3	One trunion breaks	One blade goes to flat pitch	5								
Pitch Change Linkage Roller	3	One roller wears excessively	Flutter possible						21			
Cap Dome 592093	1	Crack in dome	Lubrication					15				
O'Seal 69494824	1	Shell, or	Loss					5				
O'Seal 694940252	1	Seal leakage						5				
Nut, Retaining	1	Loss of preload	Excessive backlash				5		12			
Sleeve Rot. and Sta. Trans-Set	2	Bearing seizure	Will cause 592078 bolt to shear									10
Piston Rod	1	Crack at feedback link slot	Excessive oil leakage									10
Pin, Straight	2	Pin breaks	Loss of blade angle control		6							
Shaft, Feedback	1	Rod breaks	Prop. will go to high or low pitch stop	8								
Pin, Straight	1	Pin breaks	Prop. will go to high pitch	3								
Housing and Insert Transfer	1	Crack in housing	Excess oil leakage									15
Seal, Face	4	Cracked or jammed seal	Excess oil leakage									32

*Remarks - Page 39

**Per 10^6 hours

TABLE IA (CONT)

Component	Units per Assembly	Malfunction	Effect on Component	Failure Effects* & Rates**								
				1 C	2 C	3 M	4 M	5 m	6 m	7 m	8 m	9 N
Nut, Retaining	2	Loss of preload	Will continue to operate on one cylinder						24			
Link, Feedback	1	Linkage breaks	Prop. will go to high or low pitch stop	8								
Cap, End-Aft	1	Cracks at pin joint	Prop. will go to high pitch	2								
Piston and Plate, Actuator	1	Crack in shell	Excessive oil leakage					8				
		Break at yoke	Excess backlash, or flat pitch				4			8	4	
Bearing, Ball-Duplex	2	Spalling and incipient seizure	Assembly will continue to operate									100
Bearings	2	Spalling and incipient seizure	Assembly will continue to operate				20	100				
Input Shafts	2	Incipient thread seizure	Assembly will continue to operate				10	20				
Drive Shafts	2	Excessive thread wear	Excessive control hysteresis							20		
		Broken shaft	Will not transmit motion			16						
Housing	1	Distortion or fracture of housing	Possible binding action					12				

*Remarks - Page 39
**Per 10^6 hours

TABLE IA (CONT)

Component	Units per Assembly	Malfunction	Effect on Component	Failure Effects* & Rates**								
				1 C	2 C	3 M	4 M	5 m	6 m	7 m	8 m	9 N
Assembly Link	1	Breakage of assembly link or lever	Prop. will drive itself to low pitch		12							
Piston and Sleeve Valve	1	Breakage at outboard end of piston	Valve piston will not move on command		5							
		Chip between valve land and port	Valve spindle will jam		2			6				
Housing, Valve	1	Crack in housing	Oil leakage					8				
Tubes, Hydraulic	6	Crack in tubing	Oil leakage					12				
Actuator Cylinder Seals		Gradual loss of sealing capacity	Hydraulic oil leakage									15
Transfer Bearing Seals and Distributor Valve Seals		Loss of sealing capacity due to excess back pressure	Hydraulic oil leakage									20

*Remarks - Page 39
**Per 10^6 hours

TABLE IA (CONT)

Failure Mode	Failure Severity			
	Catastrophic	Major	Minor	Trivial
Hardover - High Pitch	26			
Hardover - Low Pitch	25			
No Control		16		
Unstable Control		39		
Poor Control			191	
Excess Vibration			57	
Flat Pitch			4	
Excess Hysteresis			28	
Trivial or Negligible Effect				202
Total $\sim$ Per 10^6 hours	51	55	280	202

Remarks

1 - Hardover High Pitch
2 - Hardover Low Pitch
3 - No Control
4 - Unstable Control
5 - Poor Control
6 - Excess Vibration
7 - Excess Hysteresis
8 - Flat Pitch
9 - Negligible Effect

C - Catastrophic
M - Major
m - Minor
N - Negligible

TABLE IB

FORWARD ROLL/YAW PROPELLER BOOST ACTUATOR - FAILURE MODE AND EFFECTS ANALYSIS

Vital Component Description	Units per Assembly	Malfunction	Effect on Component	Failure Rate/10^6 Hours		
				Catast.	Major	Minor
Dual-Tandem Hydraulic Actuator	1					
Valve (Dual)		chip between port and land	Valve spindle will jam		2	20
Valve or Feedback		worn linkage	excess hysteresis		6	18
Valve or Piston		oil seal leakage	sluggish response		8	16
		broke valve linkage	may go hardover	5		
Differential Gearbox	1					
Linear to Angle Converter		worn bearing	noticeable back lash		2	26
Gear Train Bearings		spalling and incipient seizure	assembly will continue to operate		6	18
Housing		distortion or fracture of housing	possible binding action			10
Failure Rate Totals				5	24	108
Aft Roll/Yaw Components (Failure modes and rates are identical to those of forward roll/yaw components)				5	24	108
Pitch Propeller Boost Actuator (Failure modes for the pitch boost are identical to those of the forward roll/yaw boost)				4	15	

TABLE IC

ELEVON ACTUATORS - FAILURE MODE AND EFFECTS ANALYSIS

Vital Component Description	Units per Assembly	Malfunction	Effect on Component	Failure Rate/10^6 Hours		
				Catast.	Major	Minor
Control Valve Assembly						
Dual Hydraulic Valve	1	chip between port and land	spindle will jam			18
Valve Piston Rod	1	broken valve piston rod	hardover elevon	4		
		oil seal leakage	sluggish response			15
Load Pressure (Dual) Feedback Cylinder	1	chip between port and land	spindle will jam			12
		broken piston rod		14		
		oil seal leakage as result of successive chip breakage	possible vibration		12	
Position Feedback/ Summation Linkage - 7 Link	1	broken linkage	hardover elevon down	3		
Hydraulic Actuator Assembly Components (per Actuator)	2					
Hydraulic Swivel Joints	2	leaky "O" rings	sluggish response			24
Hydraulic Piston	1	leaky piston rings	sluggish response			24
Hydraulic Cylinder	1	cracked cylinder				12
Piston Rod	1	worn rod end	hysteresis		1	20

TABLE IC (CONT)

Vital Component Description	Units per Assembly	Malfunction	Effect on Component	Failure Rate/10^6 Hours		
				Catast.	Major	Minor
Drag Link - Feedback	1	worn connection to piston rod	hysteresis		2	20
		broken link	possible hardover	1		
Total Non Redundant Failure Rate per Elevon Actuator Subsystem				22	22	145

TABLE ID

FAILURE RATE SUMMARY
(HYDRAULIC POWER GENERATION PLUS DISTRIBUTION SYSTEM)

Element Descriptions:	Failure Rate/10^6 hr	
	Common Elements	Single System Elements
Primary Hydraulic System:		
Hydraulic Pump		10.0
Check Valve		3.0
Ground Test Connections		3.0
(Pressure) Filter		1.5
Pressure Transducer		2.0
Relief Valve		3.0
(Return) Filter		1.5
Case Drain Line		1.0
Reservoir		5.0
Pressure Switch		2.0
Aft Prop Pitch and Elevon Manifolds		3.0
Aft Prop Pitch and Elevon Check Valves		4.0
Aft Prop Pitch and Elevon Tubing, Etc.		3.0
Fwd Prop Pitch and Elevon Manifolds		2.0
Fwd Prop Pitch and Elevon Check Valves		4.0
Fwd Prop Pitch and Elevon Tubing, Etc.		3.0
Aft Duct Rotation Valves and Plumbing		7.0
Fwd Duct Rotation Valves and Plumbing		7.0
Duct Lock Control Valves		3.0
Stability Augmentation Plumbing		4.0
Height Control Plumbing		3.0
Primary System Total		75.0
Secondary Hydraulic System:		
As Above for Primary System		75.0
V.S.S. Plumbing and Valves		3.0
Feel and Trim Plumbing and Valves		2.0
Landing Gear Plumbing and Valves		5.0
Secondary System Total		85.0
Gear Drive for Pumps	7.0	

TABLE IE

FAILURE RATE SUMMARY
(AC POWER GENERATION AND DISTRIBUTION SYSTEM)

Element Descriptions	Failure Rate/10^6 hr	
	Series Elements	Redundant Elements
Primary AC System:		
Constant Speed Drive		44.0
20 KVA 3ϕGenerator		35.0
Voltage Regulator		20.0
P.M. Generator and T.P. Unit		25.0
Fail Light and Contactor		5.0
3ϕGenerator Current Transformer		8.0
3 Single ϕ Load Current Transformers		12.0
Fault Detection Circuitry		16.0
AC Power, Line Contactor		12.0
AC Power, Slave Relay		12.0
Ext.Power, Relay Contacts		8.0
Power Transfer Relay Contacts		8.0
Total Primary AC System		a = 205.0
Secondary AC System:		
Variable Speed Drive		22.0
10 KVA 3ϕGenerator		35.0
Voltage Regulator		20.0
P.M. Generator and T.P. Unit		25.0
Fail Light and Contactor		5.0
3ϕGenerator Current Transformer		8.0
3 Single ϕ Load Current Transformers		12.0
Fault Detection Circuitry		16.0
AC Power, Line Contactor		12.0
AC Power, Slave Relay		12.0
Ext.Power, Relay Contacts		8.0
Power Transfer Relay Contacts		8.0
Secondary AC Bus Tie Relay		12.0
Total Secondary AC System		b = 195.0
115v - 3ϕAirframe Essential Bus	3	
115v - 3ϕCockpit Essential Bus	3	
26v - 1ϕAirframe Essential Bus	2	
26v - 1ϕCockpit Essential Bus	2	
115v/26v Transformer	4	
Auxiliary Instrumentation	4	
4 Circuit Breakers	6	
Series Elements Total Feeder Cables and Fuses	24	
Total AC Failure Rate Per Hour	24×10^{-6}	$a \times b \times 10^{-12} = 0.04 \times 10^{-6}$*

*Discarded Hereafter as Having Trivial Contribution

TABLE IF

FAILURE RATE SUMMARY

(3 psi Q-Pressure Transducer/Servo System)

Element		Failure Rate
Pressure Transducer		25
Servo Motor and Gear Train		31
Servo Amp and Quadrature Correction		42
Feedback Pot		7
	Total	105 per 10^6 hr

TABLE IG

FAILURE RATE SUMMARY

(1 psi Q-Pressure Transducer/Servo System No. 1)

Element		Failure Rate
Pressure Transducer		15
Servo Motor and Gear Train		21
Servo Amp and Quadrature Correction		42
Feedback Pot		4
	Total (System No. 1)	82 per 10^6 hr
	Total (System No. 2)	82 per 10^6 hr

TABLE IH

FAILURE RATE SUMMARY

(Primary Electrohydraulic Feel and Trim System)
(Pitch Channel)

Feel Components	Failure Rate
Block I Strain Gauge Bridge	8.0
Constant Current Source	6.8
A_1 Amplifier	3.2
Auxiliary Parts for A_1	8.0
A_3 and A_4	25.2
Limiter	5.6
Auxiliary Parts for A_3 and A_4	12.8
Servo Actuated Pots (2)	17.6
Total (Block I)	97.2

Trim Components	
Trim Button	3.4
Trim Clutches	36.0
Relays (2) and Suppressors (Redundant)	0
Relay and Suppressor	12.6
±15 -Volt Source	6.8
Input Network and Multipot	14.2
Limiter (1)	5.6
Amplifier with Chopper	38.0
Auxiliary Components	10.4
Feedback Multipot and Gain Adjust	10.4
Total	137.4

TABLE IH (CONT)

Composite		Failure Rate
A_1 and A_3 Amplifiers		25.2
Auxiliary Components and Limiter		18.0
Output Stage		8.4
Monitor Amp A_4		12.0
Input Network		5.0
Internal Feedback Network		3.8
VSS Relays (2)		25.2
Pitch Valve Coil and Actuator		28.0
Pitch Actuator Pot		2.4
	Total	128.0
	Total (Pitch Channel)	362.6 per 10^6 hr

(Yaw Channel)

Feel Components (Block I)		Failure Rate
Strain Gauge Bridge		8.0
Const. Current Source		6.8
A_1, A_3 and A_4 Amps		38.4
Auxiliary Comp A_1, A_3 and A_4		20.8
Limiter		5.6
Duct Angle Pots (2)		11.8
	Total	91.4

Trim Components (Block II)	Failure Rate
Trim Pot and Supply Resistors	8.4
±15-Volt Source	6.8
Input Network	5.6
Transfer Relay	12.4
Limiter	5.6

TABLE IH (CONT)

Trim Components (Block II) (cont)		Failure Rate
A_1 Amplifier		13.2
Output Network		8.2
Duct Pot and Assoc. Resistors		8.4
	Total	68.6

Composite (Block III)		Failure Rate
Same as Pitch Channel	Total	128.0
	Total (Yaw Channel)	288.0

Power Supply		Failure Rate
Transformers (2) and Capacitors (2)		18.0
Power Rectifiers (4)		3.0
Isolation Rectifiers, Resistors and Zeners		7.0
Trickle Charger		5.0
24V Batteries		3.0
Switching Relay and Filter Capacitors		14.0
22.5V Sources + and -		22.0
	Total	72.0

Totals	Pitch Channel	362.6
	Roll (Identical to Pitch)	362.6
	Yaw Channel	288.0

TABLE I-I

FAILURE RATE SUMMARY

(Stability Augmentation System No. 1)

Pitch Element		Failure Rate
1 Gyro		45.0
1 Actuator Pickoff		10.0
32 Fixed Resistors		10.0
4 Limiters		20.0
7 Capacitors		5.6
4 Amps		48.0
1 Master Pot		6.0
3 Minor Pot		3.0
2 Transformers		6.0
8 Demod. Diodes		4.0
2 Regul. Diodes		4.0
2 Driver Diodes		1.0
2 Power Transistors		2.4
Reset Contacts		2.0
	Total	173.0 per 10^6 hr
	Roll Axis (Identical to Pitch)	173.0
	Yaw Axis (Identical to Pitch)	173.0

Monitor Circuitry	Failure Rate
32 Fixed Resistors	16.0
2 Limiters	10.0
1 Regulator	2.0
1 Relay	10.0
2 Diodes	1.0

TABLE I-I (CONT)

Monitor Circuitry (cont)		Failure Rate
1 5V Zener		1.0
2 Delay Capacitors		1.6
2 Amps		26.4
	Total	68.0

Power Supply		Failure Rate
3 Power Transformers		9.0
8 Rectifiers		4.0
6.8-Volt Zeners		2.0
4 Power Supply Transistors (Lo)		2.8
4 Power Supply Transistors (Hi)		2.4
4 Capacitors .		4.0
2 Suppressor Capacitors		1.6
2 Potentiometers		2.2
	Total	28.0

Total for SAS No. 1	615 per 10^6 hr
Total for SAS No. 2 (Identical to No. 1)	615 per 10^6 hr

5. DEVELOPMENT OF VEHICLE-CONTROLLER MODEL - PCI PHASE 4

For this program, the primary means of failure effects and pilot work load has been a six degree of freedom hybrid simulation of the X-22A. The reasons for this decision are given in Section 2.8.

The mathematical model of the X-22A as defined in the hybrid simulation consists of the following equations.

Equations Mechanized on Analog Computers

(1) $\dot{u} = -qw + rv - g \sin\theta + (\dot{u})_{DC}$

(2) $\dot{w} = -pv + qu + g \cos\theta \cos\phi + (\dot{w})_{DC}$

(3) $\dot{v} = -ru + pw + g \cos\theta \sin\phi + (\dot{v})_{DC}$

(4) $\dot{p} = \frac{1}{I_x}\left\{I_{xz}(\dot{r} + qp) - (I_z - I_y)qr\right\} + (\dot{p})_{DC} + \Delta\dot{p}_c\,(B)_{DC}\left(\frac{DBLR}{DBLR_{max}}\right) +$

$\Delta\dot{p}_c\,(F)_{DC}\left(\frac{DFLR}{DFLR_{max}}\right)$

(5) $\dot{q} = \frac{1}{I_y}\left\{I_{xz}(r^2 - p^2) - (I_x - I_z)\,pr\right\} + (\dot{q})_{DC} +$

$\Delta\dot{q}_c\,(B)_{DC}\left(\frac{DBAF}{DBAF_{max}}\right) + \Delta\dot{q}_c\,(F)_{DC}\left(\frac{DFAF}{DFAF_{max}}\right)$

(6) $\dot{r} = \frac{1}{I_z}\left\{I_{xz}(\dot{p} - qr) - (I_y - I_x)\,pq\right\} + (\dot{r})_{DC} +$

$\Delta\dot{r}_c\,(B)_{DC}\left(\frac{DBLR}{DBLR_{max}}\right) + \Delta\dot{r}_c\,(F)_{DC}\left(\frac{DFLR}{DFLR_{max}}\right)$

(7) $\dot{\phi} = p + \dot{\psi}\sin\theta$

(8) $\dot{\theta} = q\cos\phi - r\sin\theta$

(9) $\dot{\psi} = (r\cos\phi + q\sin\phi)/\cos\theta$

Also mechanized on the analog computers were:

(1) Stability Augmentation System
(2) Attitude Control Phasing
(3) Power Control (Thrust)
} Refer to Section II.2

(4) Control System Failures; Introduction of - refer to Section II.9

Equations Mechanized on the IBM 7090 Digital Computer

(1) $(\dot{u})_{DC} = (C_{x_s} q_T S)/m$

(2) $(\dot{w})_{DC} = (C_{z_s} q_T S)/m$

(3) $(\dot{v})_{DC} = \frac{S}{M}\left[C_{y_s} q_T + \frac{q_F b}{2V}\left\{C_{y_r} r + C_{y_p} p\right\} - \frac{346\lambda^2}{8100S}^{*} p\right]$

(4) $(\dot{p})_{DC} = \frac{bS}{I_x}\left[C_{\ell_s} q_T + \frac{q_F b}{2V}\left\{C_{\ell_p} p + C_{\ell_r} r\right\} - \frac{4250\lambda^2}{8100bS}^{*} p + \frac{2650\lambda^2}{8100bS}^{*} r\right]$

(5) $(\dot{q})_{DC} = \frac{\bar{c}S}{I_y}\left[Cm_s q_T + \frac{q_F \bar{c}}{2V} Cm_q q - \frac{4130\lambda^2}{8100\bar{c}S}^{*} q\right]$

(6) $(\dot{r})_{DC} = \frac{bS}{I_z}\left[Cn_s q_T + \frac{q_F b}{2V}\left\{Cn_p r + Cn_p p\right\} - \frac{6670\lambda^2}{8100bS}^{*} r\right]$

(7) $V = \sqrt{u^2 + v^2 + w^2}$

(8) $q_F = \frac{1}{2}\rho V^2$

(9) $q_T = \frac{T}{S} + q_F$

(10) $C_{T_s} = \frac{T}{q_T S}$

(11) $\alpha = \tan^{-1}\left(\frac{w}{u}\right)$

(12) $\beta = \sin^{-1}\left(\frac{v}{V}\right)$

(13) $\gamma = \sin^{-1}\left(\frac{\dot{h}}{V}\right)$

(14) $\xi = \tan^{-1} (\dot{Y}_I)_{DC}/(\dot{X}_I)_{DC}$

* Artificial dimensional derivatives introduced to correct the rotary derivatives as $\lambda \to 90°$. The effects of these quantities wash out as λ decreases

(15) $(\dot{X}_I)_{DC} = u(\cos\psi\cos\theta) + v(\cos\psi\sin\theta\sin\phi - \sin\psi\cos\phi) + w(\sin\psi\sin\phi + \cos\psi\sin\theta\cos\phi)$

(16) $(\dot{Y}_I)_{DC} = u(\sin\psi\cos\theta) + v(\cos\psi\cos\phi + \sin\psi\sin\theta\sin\phi) + w(\sin\psi\sin\theta\cos\phi - \cos\psi\sin\phi)$

(17) $(\dot{Z}_I)_{DC} = -\dot{h} = -u\sin\theta + v(\cos\theta\sin\phi) + w(\cos\theta\cos\phi)$

Functions Stored on Digital (Aero, Propulsion, Control)

(1) $C_{x_s}\ (\alpha, \lambda, C_{T_s})$

(2) $C_{z_s}\ (\alpha, \lambda, C_{T_s})$

(3) $C_{m_s}\ (\alpha, \lambda, C_{T_s})$

(4) $C_{y_s}\ (\alpha, \beta, \lambda, C_{T_s})$

(5) $C_{\ell_s}\ (\alpha, \beta, \lambda, C_{T_s})$

(6) $C_{n_s}\ (\alpha, \beta, \lambda, C_{T_s})$

(7) $\left.\dfrac{\partial C_N}{\partial C_{T_s}}\right|_{\substack{\text{fwd} \\ \alpha = 0^\circ}} (\lambda, C_{T_s})$

(8) $\left.\dfrac{\partial C_N}{\partial C_{T_s}}\right|_{\substack{\text{aft} \\ \alpha = 0^\circ}} (\lambda, C_{T_s})$

(9) $\dfrac{X_{NF}}{\bar{c}}\ (\lambda, C_{T_s})$

(10) $\dfrac{X_{NA}}{\bar{c}}\ (\lambda, C_{T_s})$

(11) $C_{m_q}\ (\lambda, C_{T_s})$

(12) $T_{max} \left[(\alpha + \lambda),\ |u|\right]$

(13) $C_{y_p} \ (\alpha, \lambda)$

(14) $C_{\ell_p} \left[(\alpha + \lambda), C_{T_s}\right]$

(15) $C_{n_p} \ (\alpha, \lambda)$

(16) $C_{y_r} \ (\alpha, \lambda)$

(17) $C_{\ell_r} \ (\alpha, \lambda)$

(18) $C_{n_r} \ (\alpha, \lambda, C_{T_s})$

Control Equations Solved by 7090 and Sent to Analog

(1) Pitch Acceleration Due to Prop Blade

$$\Delta \dot{q}_c \text{ (B)} = \left(\frac{-1}{I_y}\right)\left(2\bar{c}\ \frac{\partial T}{\partial \beta_{PROP}}\right)\left[\left(\frac{X_{AXF} - X_{AXA}}{\bar{c}}\right) + 2\frac{\partial C_N}{\partial C_{T_s}}\Bigg|_{\substack{fwd \\ \alpha = 0^\circ}} \left(\frac{X_{NF}}{\bar{c}}\right) - 2\frac{\partial C_N}{\partial C_{T_s}}\Bigg|_{\substack{aft \\ \alpha = 0^\circ}} \left(\frac{X_{NA}}{\bar{c}}\right)\right] DBFA_{max} \sim \text{rad/sec}^2$$

where:

$$2\bar{c}\ \frac{\partial T}{\partial \beta_{PROP}} = (3080 - 3.1\ V) \sim \text{(ft-lb)/deg,}$$

$DBAF_{max} = +\ 7$ deg

$\bar{c}\ =\ 7$ ft

$$\frac{X_{AXF} - X_{AXA}}{\bar{c}} = (0.32 + 2.36 \sin \lambda)$$

(2) Pitch Acceleration Due to Flaps

$$\Delta \dot{q}_c\,(F) = \frac{-1}{I_y}\,(0.04 - 0.000444\,\lambda)\,(q_e\,S\bar{c})\;DFAF_{max} \sim rad/sec^2$$

where:

$$DFAF_{max} = +30\ deg$$

$$S = 225\ ft^2$$

$$\bar{c} = 7\ ft$$

$$q_e = f\left[(\alpha + \lambda),\, C_{T_s}\right]$$

(3) Roll Acceleration Due to Prop Blade

$$\Delta \dot{p}_c\,(B) = \left(\frac{-1}{I_x}\right)\left(2\,\frac{\partial T}{\partial \beta_{PROP}}\right)\left\{(y_F + y_A)\sin\lambda + 2\left(\frac{\partial C_N}{\partial C_T}\Bigg|_{\substack{fwd\\ \alpha = 0}} y_F + \frac{\partial C_N}{\partial C_T}\Bigg|_{\substack{aft\\ \alpha = 0}} y_A\right)\cos\lambda\right\}\;DBLR_{max} \sim rad/sec^2$$

where:

$y_F = 7.25$ ft; $y_A = 13.5$ ft; $DBLR_{max} = +6$ deg

(4) Roll Acceleration Due to Flap

$$\Delta \dot{p}_c\,(F) = (-0.005\cos\lambda)\,(q_e\,S\,b)\left(\frac{1}{I_x}\right)DFLR_{max} \sim rad/sec^2$$

where:

$b = 38.33$ ft; $DFLR_{max} = +30$ deg

(5) Yaw Acceleration Due to Blade

$$\Delta \dot{r}_c\,(B) = \left(\frac{1}{I_z}\right)\left(2\,\frac{\partial T}{\partial \beta_{PROP}}\right)\left\{(y_F + y_A)\cos\lambda - 2\left(\frac{\partial C_N}{\partial C_T}\Bigg|_{\substack{fwd\\ \alpha = 0}} y_f + \frac{\partial C_N}{\partial C_T}\Bigg|_{\substack{aft\\ \alpha = 0}} y_A\right)\sin\lambda\right\}\;DBLR_{max} \sim rad/sec^2$$

(6) Yaw Acceleration Due to Flap

$$\Delta \dot{r}_c\ (F) = \frac{A\ q_e\ S\ b}{I_z}\quad DFLR_{max} \sim rad/sec^2$$

where:

45 deg < λ < 90 deg	A = +0.0045
30 deg < λ < 45 deg	A = +(0.0018 + 0.00006 λ)
0 deg < λ < 30 deg	A = 0.00012 λ

6. ESTABLISHMENT OF MAN-MACHINE CONSTRAINTS - PHASE 5

The analysis of failure effects requires the establishment of criteria defining the boundaries between non-catastrophic and catastrophic man-machine response. These criteria are generally specified in terms of critical structural stress and performance limits.

Normal acceleration structural limits of the X-22A - at VTOL design gross weight of 14,364 pounds - are shown in Figure 17 for hover and transition flight and Figure 18 for conventional flight. Limiting stall conditions are shown in the same figures.

Certain performance limits such as α_{stall} while flying at extremely low altitude are autocratic by nature. However, other performance limits such as minimum control margins above trim requirements and minimum levels of static stability are dependent upon such factors as flight environment, maneuver tasks and pilot proficiency.

For the purposes of this program - constraint boundaries for these more subjective limits have been defined as follows:

(1) As the entire X-22A subject mission is flown at low altitude, h $\leq$ 500 ft, the flight trajectory was assumed to be catastrophic if the altitude deviation from reference altitude exceeded the following limits:

a. $\Delta h_{negative} \geq 0.5\ h_{ref}$

b. $\Delta h_{positive} \geq h_{ref}$

(2) Throughout transition and conventional flight, heading deviations $\geq 30^\circ$ were considered to result in catastrophic consequences.

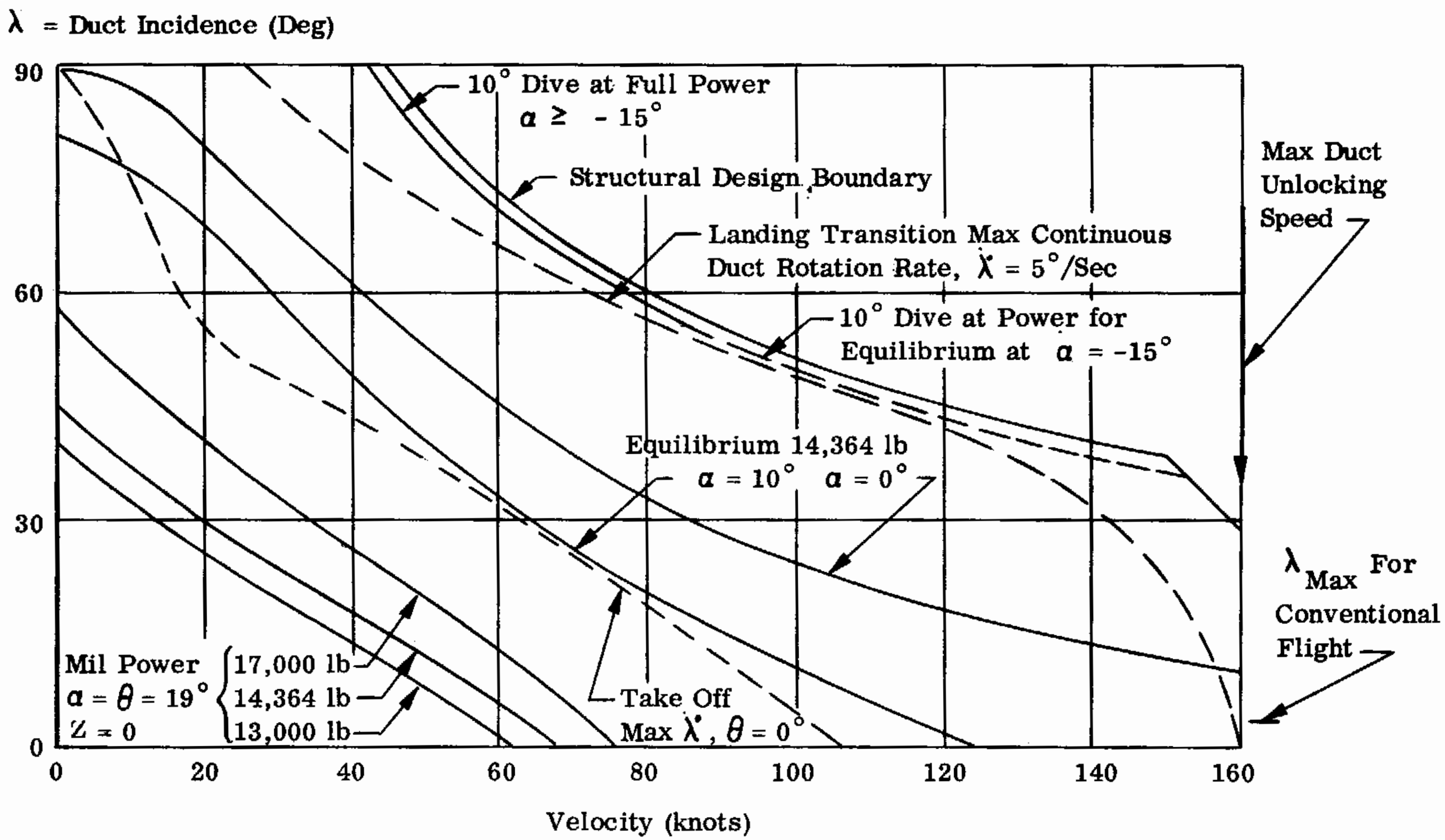

Figure 17. Operational Envelope — Transition Flight

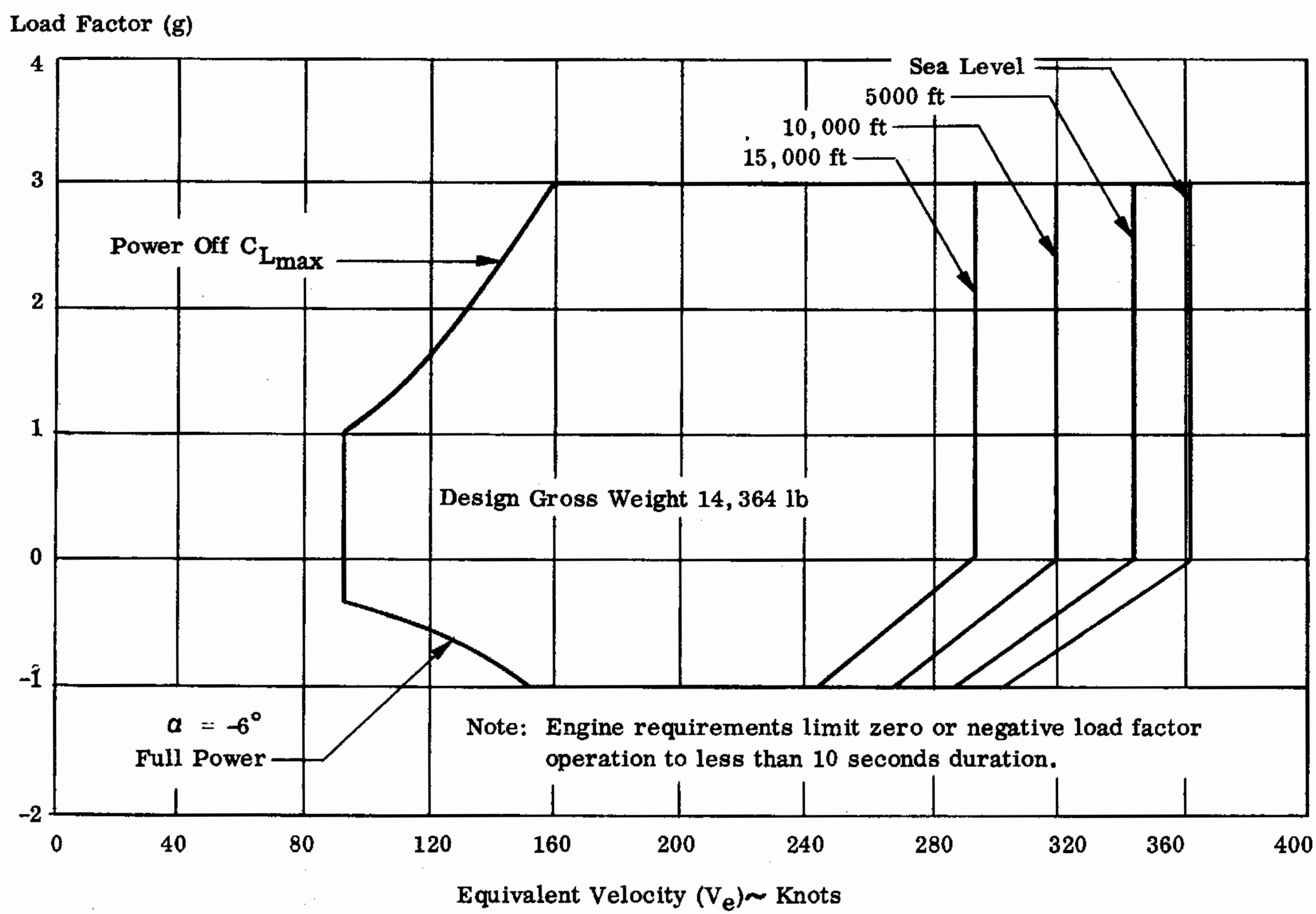

Figure 18. X-22A Operational Flight Envelope Conventional Flight Configuration (Front Ducts 2°, Aft Ducts -3°)

As with most transport type aircraft, the tolerance levels of the pilot are higher than those of the X-22A aircraft structures. In only one respect is the pilot performance limited, i.e., maximum work load, and this is a limitation of capability and not physical tolerance levels.

7. CATEGORIZING OF FAILURE MODES BY DEGREE OF SEVERITY - PCI PHASE 6

A detailed inspection was made of flight control subsystem failure modes in order to categorize them by degree of severity. The subsystem failure mode definition tables in Section 2.3 supplied the basic failure effects ingredients necessary to translate the consequences of specific failures into control surface motions and hence the prediction of vehicle-pilot responses.

Those failures determined to be definitely catastrophic are indicated in Table IIA, and the potentially catastrophic failures are listed in Table IIB. In both of the above tables the individual subsystem failure modes are identified by their respective numerical codings used in the digital program for the analysis of mission success probability.

The minor and trivial failures were eliminated from further analysis at this point in the program as a result of their insignificant influence on man-machine capabilities.

TABLE IIA

NUMERICAL IDENTIFICATION OF CATASTROPHIC SINGLE FAILURE MODES

01 Flight Control System Mechanical Linkages

02 Dual A.C. Electrical Power Generation System

03 Pitch Propeller Boost Actuator

04 Forward Roll/Yaw Propeller Boost Actuator - Hardover Failures

05 Aft Roll/Yaw Propeller Boost Actuator - Hardover Failures

06 Left Forward Propeller Blade Control - Hardover Failures

07 Right Forward Propeller Blade Control - Hardover Failures

08 Left Aft Propeller Blade Control - Hardover Failures

09 Right Aft Propeller Blade Control - Hardover Failures

10 Left Forward Elevon Control - Hardover/Frozen Failures

11 Right Forward Elevon Control - Hardover/Frozen Failures

12 Left Aft Elevon Control - Hardover/Frozen Failures

13 Right Aft Elevon Control - Hardover/Frozen Failures

14 Dual Hydraulic Power Generation Drive (Common Components)

TABLE IIB

NUMERICAL IDENTIFICATION OF POTENTIALLY CATASTROPHIC FAILURE MODES INVOLVED IN CONTROL CONTINUITY AND PILOT WORK LOAD TESTING

0100	Primary Hydraulic Power Generation System
0200	Secondary Hydraulic Power Generation System
0400	Q-Pressure Transducer/Servo System (3 psi)
0500	Q-Pressure Transducer/Servo System (1 psi), No. 1
0600	Q-Pressure Transducer/Servo System (1 psi), No. 2
0700	Backup Feel/Trim System (Mechanical)
0801	Primary Feel/Trim System (Electrohydraulic)
0901	Stability Augmentation System, No. 1
1001	Stability Augmentation System, No. 2
1100	Forward Roll/Yaw Propeller Boost Actuator - Soft Failures
1200	Aft Roll/Yaw Propeller Boost Actuator - Soft Failures
1300	Left Forward Propeller Blade Control - Soft/Frozen Failures
1400	Right Forward Propeller Blade Control - Soft/Frozen Failures
1500	Left Aft Propeller Blade Control - Soft/Frozen Failures
1600	Right Aft Propeller Blade Control - Soft/Frozen Failures
1700	Left Forward Elevon Control - Soft Failures
1800	Right Forward Elevon Control - Soft Failures
1900	Left Aft Elevon Control - Soft Failures
2000	Right Aft Elevon Control - Soft Failures

8. ANALYSIS OF SIMPLE FAILURES - PCI PHASE 7

For the reasons explained below, none of the potentially catastrophic failure modes of the X-22A flight control system were treated as simple failures throughout the failure effects and pilot work load analysis.

During vertical flight and transition, the aerodynamic characteristics of the X-22A are functionally dependent on thrust level and duct angle as well as velocity. The employment of differential propeller blade pitch and duct-exit elevons for three-axis attitude control results in control surface effectiveness being strongly affected by thrust and duct angle. Single-axis attitude control is phased between propeller pitch and elevon on the basis of duct angle. As a consequence of the above physical dependencies, all the significant X-22A control system failure modes in the potentially catastrophic category are complex in nature - requiring analysis by appropriate piloted simulation techniques.

The six-degree-of-freedom hybrid simulation of the X-22A was consequently made available for all of the PCI failure effects and pilot control capabilities studies.

9. ANALYSIS OF COMPLEX FAILURES - PCI PHASE 8

Potentially catastrophic, complex failures were analyzed through use of a hybrid simulation of the X-22A. The simulation model of the flight control system was modified to provide for the introduction of those active, non-catastrophic failures affecting pilot control capabilities.

The objectives of the simulator analysis were to:

(1) Determine steady state pilot control capabilities in the presence of existing failure conditions - by measurement of pilot work loads.

(2) Evaluate active failure transient control capabilities; i.e., can the pilot maintain control immediately following the introduction of an active [6] failure and bring the vehicle into a state of equilibrium without exceeding system constraints ?

The failure modes to be analyzed on the simulator are listed in Table III by mission segments and identified as to system and pilot response to failures.

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(6) For the purposes of this study, an active failure is defined as a failure which results in a discontinuity or transient in the response of one or more control surfaces of sufficient magnitude as to be apparent to the pilot.

TABLE III

CONTROL SYSTEM FAILURES ANALYZED ON SIMULATOR

Control System Components	Failure Mode	Mission Segments For Steady State Failures			Mission Segments For Transient Failures			System Response to Failure	Pilot Action
		1/5	2/4	3	1/5	2/4	3		
1. Electrohydraulic Feel/Trim	Electrical or Hydraulic Failure	-	x	x	-	x	x	Mission Segments: 2/4 Control actuators to bypass 3 Control Actuator	Switch to mechanical backup
2. Mechanical Feel/Trim	Broken Spring or Clutch	x	x	-	-	-	-	Limp stick or pedals with no trim	No specific emergency procedure
3. SAS No. 1 (or Primary Hyd. Pwr.) or SAS No. 2 (or 1 psi Q. System No. 2)	Hardover, Soft, or Oscillatory	x	x	-	x	x	-	Trim change or rotational oscillation	Put appropriate SAS in bypass
4. Forward Roll/Yaw Prop. Boost	Soft Failure	x	x	-	x	x	-	Failed booster limits effective range of good booster to extent of valve piston travel; beyond this point, opposing motion of failed booster cancels out good booster	Must exercise very limited roll/yaw control motions
5. Aft Roll/Yaw Prop. Boost	Soft Failure	x	x	-	x	x	-	No output until input exceeds limit of valve piston travel; beyond this point aft booster dragged by fwd booster - sluggish response	For initial 45% of pilot control displacement - response from fwd booster only
6. Prop. Blade Control, Single	Frozen or Soft	x	x	-	x	x	x	Erratic trim change with pilot control inputs	Reduce power; restrict control motions in freq. and displacement
7. Elevon Control, Single	Soft, Dragging	x	x	x	x	x	x	Similar in nature in prop. control failures	Reduce power; restrict control motions in freq. and displacement
8. El-hyd. Feel/Trim and SAS No. 1 (or 1 psi Q. System No. 1) or El-hyd. Feel/Trim and SAS No. 2 (or 2nd Hyd. Pwr.)	Hardover or short in Q System or Separate Failures	-	x	-	-	x	-	Feel/trim lockup and trim change or rotational oscillation	Switch F/T to mech. backup and bypass appropriate SAS
9. SAS No. 1 and SAS No. 2	Failures in both Systems	x	x	-	-	-	-	Trim change and/or rotational oscillation	Put both SAS in bypass

The block diagram for the work load measurement unit developed for the simulation studies is shown in Figure 19. The pilot's auxiliary work load task consists of resetting to zero a three-state, randomly generated error signal at a maximum tolerable rate without compromising performance of the primary flight control task. The error signal is displayed to the pilot on a 4 inch vertical edge reading meter which is installed to the right of the altimeters on the cockpit instrument panel (refer to Figure 20). The pilot responds to the displayed error by depressing a rocker switch, mounted on the front of the control stick grip, in the same direction as the observed error signal; i.e., depress top side of switch for "hi" errors and bottom side for "low" errors.

This task is similar in concept to the sixteen light-switch array utilized in the original PCI study program (Reference 1). However, the work load task employed in the X-22A simulator studies was restricted to a three-state task as a result of the continual occupation of both of the pilot's hands in the flight control task throughout vertical flight and transition; i.e., the right hand required for attitude control (pitch-roll stick) and left hand required for throttle control.

Two company pilots served as the subjects for these studies. Both men are experienced test pilots with a variety of aircraft qualifications as indicated in the following table.

	Pilot A	Pilot B
Total Flight Time	3100 hr	8200 hr
Conventional A/C Time (jet and recip. engine)	3000	4200
Helicopter Time	100	4000
Number of Conventional A/C Types	25+	50+
Number of Helicopter Types	5	12

Also, each pilot had 40 to 50 hours time "flying" the X-22 simulator prior to the beginning of these studies.

Simulated flight profiles - representative of the several segments of the subject evaluation mission - are outlined in Table IV.

The general procedure followed in the conduct of the simulation studies was:

(1) Familiarize the pilots with the operation of the work load measurement unit.

(2) Familiarization and warm up: Fly the simulated flight profiles under normal system operation and under degraded conditions (failure) - with and without auxiliary work load task.

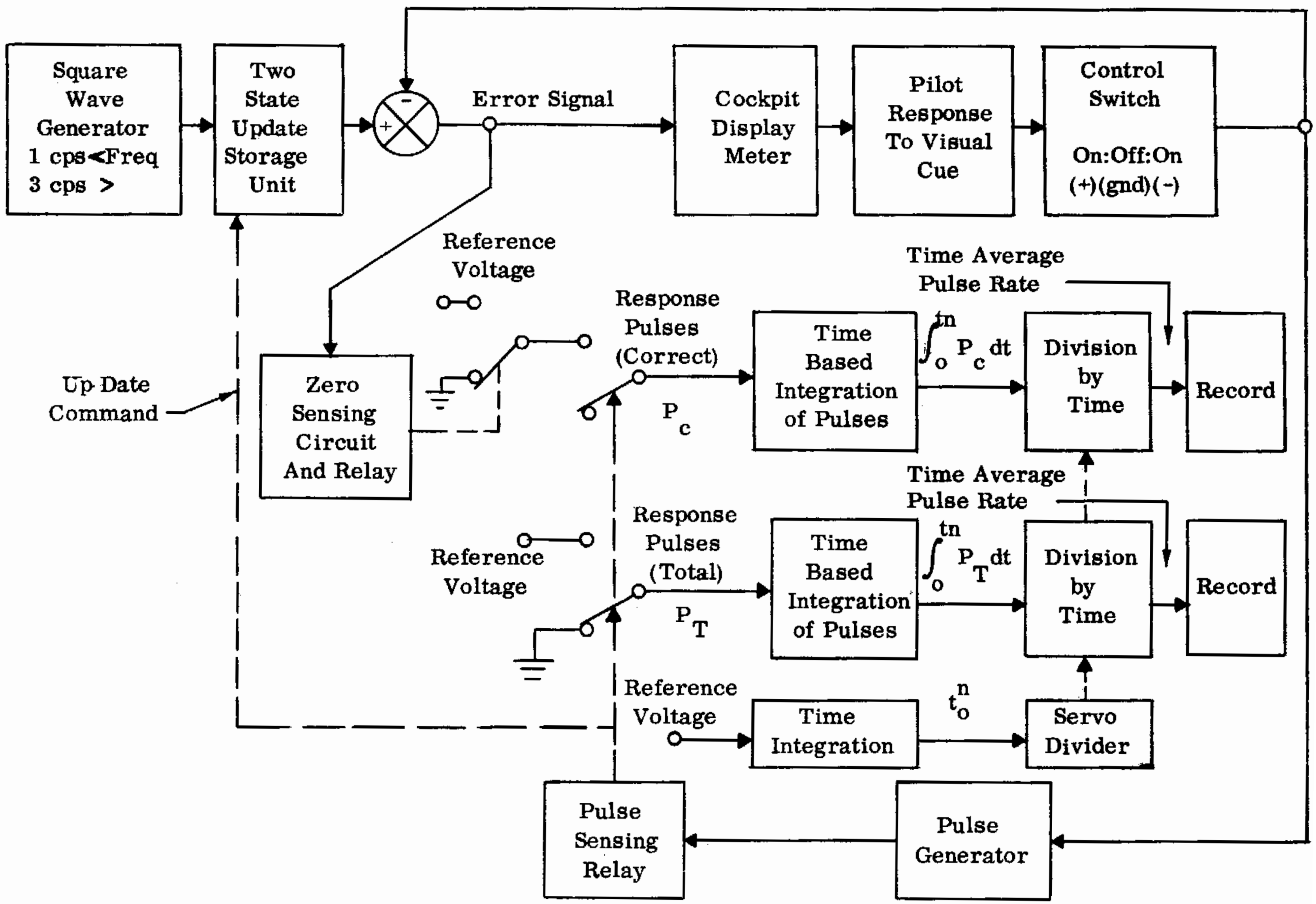

Figure 19. Pilot Workload Measurement Unit

Figure 20. X-22A Simulator Cockpit

TABLE IV

DEFINITION OF MISSION PROFILES - PILOTED SIMULATED PROGRAM

1. <u>Hover - Vertical Flight</u> $100 < t < 120$ sec

   a. Climb to 100 ft, momentary hover

   b. Translate forward to 25 knots, hold for 5 seconds

   c. Return to hover, momentary stabilize

   d. Climb to 150 ft, momentary hover

   e. Translate laterally (right or left) to 25 knots, and hold for 5 seconds

   f. Return to hover, momentary stabilize

   g. Translate backward to 25 knots, hold for 5 seconds

   h. Return to hover, momentary stabilize

   i. Descent to zero altitude - end of flight

2. <u>Transition Flight</u> $90 < t < 150$ sec

   a. Begin flight stabilized at V = 0 knots, h = 100 ft

   b. Transition at constant altitude to conventional flight $30 < \Delta t < 60$ seconds

   c. Accelerate to 150 knots and stabilize for 5-10 seconds

   d. Decelerate and transition to vertical flight at constant attitude $30 < \Delta t < 60$ sec

   e. Stabilize in hover - end of flight

3. <u>Conventional Flight</u> $140 < t < 180$ sec

   a. Start in level flight stabilized at V = 240 knots, h = 1000 ft

   b. Execute 90° (left or right) turn at standard rate and stabilize on new heading for 5-10 seconds

   c. Decelerate to 170 knots and return to original heading at standard rate

   d. Set up 1000 fpm rate of descent holding 170 knots

   e. Level off at 100 feet maintaining 170 knots - end of flight

(3) Evaluate steady state control capabilities (work load measurement) for normal system operation and for the specified failure conditions.

(4) Evaluate transient control capabilities with active failures introduced during a run. The evaluation runs were randomly mixed with handling qualities studies discussed in Appendix I to minimize pilot anticipation of failures. Transient failures were introduced in 20% to 25% of these runs.

The pilots each made 5 to 8 evaluation runs for each steady state failure condition, and pilot "A" made 3 to 5 runs for each transient control condition.

The averaged pilot work loads for the potentially catastrophic failure conditions are shown in Table V.

The basic work loads represent the average loading on the pilot throughout the most critical 10 second period of each mission segment with all systems operating normally. The work load increments - for the seven indicated failure modes - are added to the basic work load for a given mission segment to arrive at a total failure work load. In a case of multiple failure interference - characterized by the two failure mode combinations at the bottom of Table V - the interference increment is summed with the respective individual failure increments and the basic work load to arrive at a total failure work load.

The significance of interference effects is illustrated by the example of a dual stability augmentation system failure in hover/translation flight. As indicated in Table V, the failure of either SAS No. 1 or SAS No. 2 results in a ΔWL of 0.12. Summing the ΔWL's for failure of both SAS No. 1 and No. 2 with the basic (no failure) work load results in a total value of 0.84. However, the measured (simulated) total WL with both SAS's failed was 1.00. This indicates that a discrepancy of 16% would exist in the total WL if the interference effects had not been taken into account.

The relatively high basic pilot work loads - with normal system operation - appear to reflect the large pitch trim and normal force sensitivities to air speed, angle of attack, and thrust level as well as the somewhat marginal thrust response to throttle control inputs (characterized by an equivalent 0.65 second time lag).

The level of pilot work loads was also influenced by the inherent IFR nature of the simulated flight tasks with all visual cues derived from cockpit instruments; i.e., no visual scene display for simulated VFR. A conservative approximation to the reduction in work loads for VFR performance of the defined tasks can be attempted by correlating the difference between equivalent IFR and VFR pilot rating boundaries (Figure 25 of Appendix I) as established by Salmirs and Tapscott with the work load boundaries of Figure 30. This procedure indicates a reduction of 6% to 10% in work load for the visual flight task.

TABLE V

PILOT WORK LOADS FOR POTENTIALLY CATASTROPHIC FAILURE MODES: ESTABLISHED THROUGH SIMULATION ANALYSIS

Failure Mode	Mission Segments 1/5 (Vertical Flt/Hover and Translation)	Mission Segments 2/4 (Takeoff and Landing Transition)	Mission Segments 3 (conventional Flight)
Basic Work Loads (No Failures):	0.60	0.72	0.58
Single Failure Work Load Increments			
Primary Feel/Trim System (Electrohydraulic)	0	0.08	0.11
Backup Feel/Trim System (Mechanical)	0.04	0	0.05
Stability Augmentation System No. 1 or No. 2	0.12	0.08	0
Forward Roll/Yaw Prop. Boost	0.09	0.11	0
Aft Roll/Yaw Prop. Boost	0.04	0.05	0
Propeller Blade Control, Single	0.16	0.19	0
Elevon Control, Single, Soft	0.14	0.15	0.20 (1)
Multiple Failure Work Load Interference increments:			
Primary Feel/Trim and SAS No. 1 or No. 2	0	-0.06	0
SAS No. 1 and No. 2	0.16	0.10/0.95 (2)	0

Notes: (1) Catastrophic transient failure situation if moderate amount of pitch or roll control is being employed at time of failure, and pilot does not react instantaneously.

(2) Takeoff transition $\Delta WL = 0.10$; landing transition total work load exceeds 1.0 as duct angle increases beyond 65° to 70°

The only failure mode condition analyzed which resulted in a steady state work load greater than unity was a dual SAS failure during landing transition or vertical flight. It should be noted, that if the possibility of work load interference had not been taken into account in the simulator analysis, this pilot overload condition would have gone unrecognized. A procedure for the analysis of work load interference is discussed under the heading of Concept Improvements - Section IV.

Pilot opinion ratings were also recorded during the steady state failure studies. The analysis of this data is discussed under the Handling Qualities Studies in Appendix I.

Pilot transient control response to the introduction of active failures - under worst case conditions - is indicated in Table VI. The only transient situation which results in loss of control is an elevon control failure at conventional flight speeds of 220/240 knots or greater during mission segment 3. This failure mode will thus be treated as a catastrophic single failure in the digital analysis of mission success probability.

10. PREDICTION OF COMPONENT FAILURE RATES - PCI PHASE 9

The failure rates for certain of the electronic components and piece parts were established on the basis of in-house bench test data.

As the majority of the electrohydraulic and hydromechanical components are unique to the X-22A flight control system, no first hand field experience or laboratory test data was available. As a result of the above situation, the prediction of failure rates for these components was based primarily on data published for similar units - with the application of appropriate modification factors. The reference reliability publications are listed in the bibliography.

Component failures are generally grouped in three categories according to the major influences of failure:

(1) Initial or wear-in failures
(2) Intermediate chance failures
(3) Wear out failures

This study is concerned only with the second category - the intermediate or chance failures. The assumption is made that the aircraft has flown no less than 100 hours and that the influence of wear-in failures has thus been eliminated.

Chance failures are those which occur throughout the majority of the normal operating life of the system. Failure rates during this period are assumed to be essentially constant since failures are characterized by randomness with future probability being independent of the operational age of components.

TABLE VI

ACTIVE FAILURES - PILOT TRANSIENT CONTROL RESPONSE CAPABILITIES

Failure Mode	Mission Segments	Control Capability
1. Primary Feel/Trim (Electrohydraulic)	1/5	System automatically reverts to mechanical backup with identical gradients/breakout forces.
	2/4	Backup system gradients less in roll and yaw and moderately higher in pitch: slight transient at auto. transfer - not catastrophic.
	3	Potentially catastrophic in high speed pullout only - well beyond limits of defined mission envelope - not potentially catastrophic
2. Stability Augmentation System No. 1 or No. 2 (Hardover Failure)	1/5, 2/4	Disturbances easily controlled by Pilot in accordance with design philosophy; re MIL-H-8501A Section 3.5.9 (damper authority).
	3	Would require two hidden failures (dual pitch SAS high speed lockouts) in addition to the active failure - extremely remote. At V = 288 knots, $a_n > +4.0$ g's
3. SAS No. 1 or No. 2 (Oscillatory Failure)	1/5, 2/4	Limit cycle not perceptible by pilot for freq. > 2 cps; f < 2 cps maximum attitude excursions equivalent to mild gust.
4. Roll/Yaw Prop. Boost; Fwd or Aft (Soft Failure)	1/5, 2/4, 3	Consequent steady state control imposes much more stringent work load on pilot than initial transient condition.
5. Propeller Blade Control, Single (Soft Failure)	1/5, 2/4, 3	Consequent steady state control imposes much more stringent work load on pilot than initial transient conditions.
6. Elevon Control, Single (Soft Failure)	1/5	Mild yaw moment with slight coupling into pitch/roll.
	2/4	Failures introduced at $10^\circ < \lambda < 30^\circ$; mild bump in all axes - no problem for pilot.

TABLE VI (CONT)

Failure Mode	Mission Segments	Control Capability
	3	For $V \geq$ 220/240 knots CATASTROPHIC if not reacted to instantaneously
7. Primary Feel/Trim and SAS No. 1 or No. 2	1/5, 2/4, 3	Transient conditions somewhat more severe than primary feel/trim only - not potentially catastrophic.

The generic failure rates indicated in Tables VII and VIII are arrived at on the basis of ideal laboratory type operating conditions. These generic rates must be modified on the basis of stresses imposed by the operating environment of the subject system in order to arrive at "basic" failure rate values. After consulting the available literature, it was decided for the purposes of this study to use a basic application factor of 50.

The basic failure rates are thus considered appropriate for X-22A operation in the conventional flight mode. However, as the vehicle control system is exposed to more stringent operating conditions (cycle rates, vibration, temperature variations, g's) in hover and transition flight than in conventional flight, it was deemed appropriate to apply additional "mission segment application factors" to these modes of operation. Thus, the generic failure rates for the defined mission are modified as follows:

Mission Segment	Applied Failure Rate
1 and 5 (hover and vertical flight)	3 x 50 x Generic Failure Rate
2 and 4 (takeoff and landing transitions)	5 x 50 x Generic Failure Rate
3 (conventional flight)	1 x 50 x Generic Failure Rate

The applied failure rates for the potentially catastrophic and definitely catastrophic failure modes of the X-22A flight control system are listed in Tables VII and VIII by mission segments.

11. DIGITAL ANALYSIS OF MISSION SUCCESS PROBABILITY - PCI PHASE 10

A digital program was developed to provide a systematic method for analyzing the effect of control system failure combinations on mission success or failure [7].

In very simple systems involving a restricted number of failure mode combinations ($FMC_{max} \leq 100$), paper and pencil analysis of mission success probability is practical; however, because the analysis procedure expands rapidly as the number of failure modes and mission segments increase, digital processing becomes the only practical method of analysis.

The failure rates and pilot work loads for the several subsystem modes of failure - as developed in PCI Phases 8 and 9 - are inputs to the digital program along with the pilot work load and control continuity failure mode relationships as defined in the mission success diagrams - PCI Phase 2 - and shown in digital flow chart form in Figures 21 and 22.

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[7] A detailed description of the digital program is presented in Appendix III.

TABLE VII

FAILURE RATES FOR POTENTIALLY CATASTROPHIC FAILURES INVOLVED IN CONTROL CONTINUITY AND PILOT WORK LOAD TESTING

Element Identification*	Mission Segments				
	1	2	3	4	5
0100	11,250	18,750	3,750	18,750	11,250
0200	12,750	21,250	4,250	21,250	12,750
0400	15,750	26,250	5,250	26,250	15,750
0500	12,300	20,500	4,100	20,500	12,300
0600	12,300	20,500	4,100	20,500	12,300
0700	900	1,500	300	1,500	900
0801	162,750	271,250	54,250	271,250	162,750
0901	92,250	153,750	15,375	153,750	92,250
1001	92,250	153,750	15,375	153,750	92,250
1100	3,600	6,000	1,200	6,000	3,600
1200	3,600	6,000	1,200	6,000	3,600
1300	1,650	2,750	550	2,750	1,650
1400	1,650	2,750	550	2,750	1,650
1500	1,650	2,750	550	2,750	1,650
1600	1,650	2,750	550	2,750	1,650
1700	3,000	5,000	0	5,000	3,000
1800	3,000	5,000	0	5,000	3,000
1900	3,000	5,000	0	5,000	3,000
2000	3,000	5,000	0	5,000	3,000

* See Table IIB

Note: Failure rates are indicated for 10^6 hours

TABLE VIII

FAILURE RATES FOR CATASTROPHIC SINGLE FAILURE MODES

Section Identification*	Mission Segments				
	1	2	3	4	5
01	900	1500	300	1500	900
02	3600	6000	1200	6000	3600
03	2850	4750	200	4750	2850
04	750	1250	250	1250	750
05	750	1250	250	1250	750
06	1530	2550	510	2550	1530
07	1530	2550	510	2550	1530
08	1530	2550	510	2550	1530
09	1530	2550	510	2550	1530
10	3300	5500	2100	5500	3300
11	3300	5500	2100	5500	3300
12	3300	5500	2100	5500	3300
13	3300	5500	2100	5500	3300
14	1050	1750	350	1750	1050

* See Table IIA

Note: Failure rates are indicated for 10^6 hours

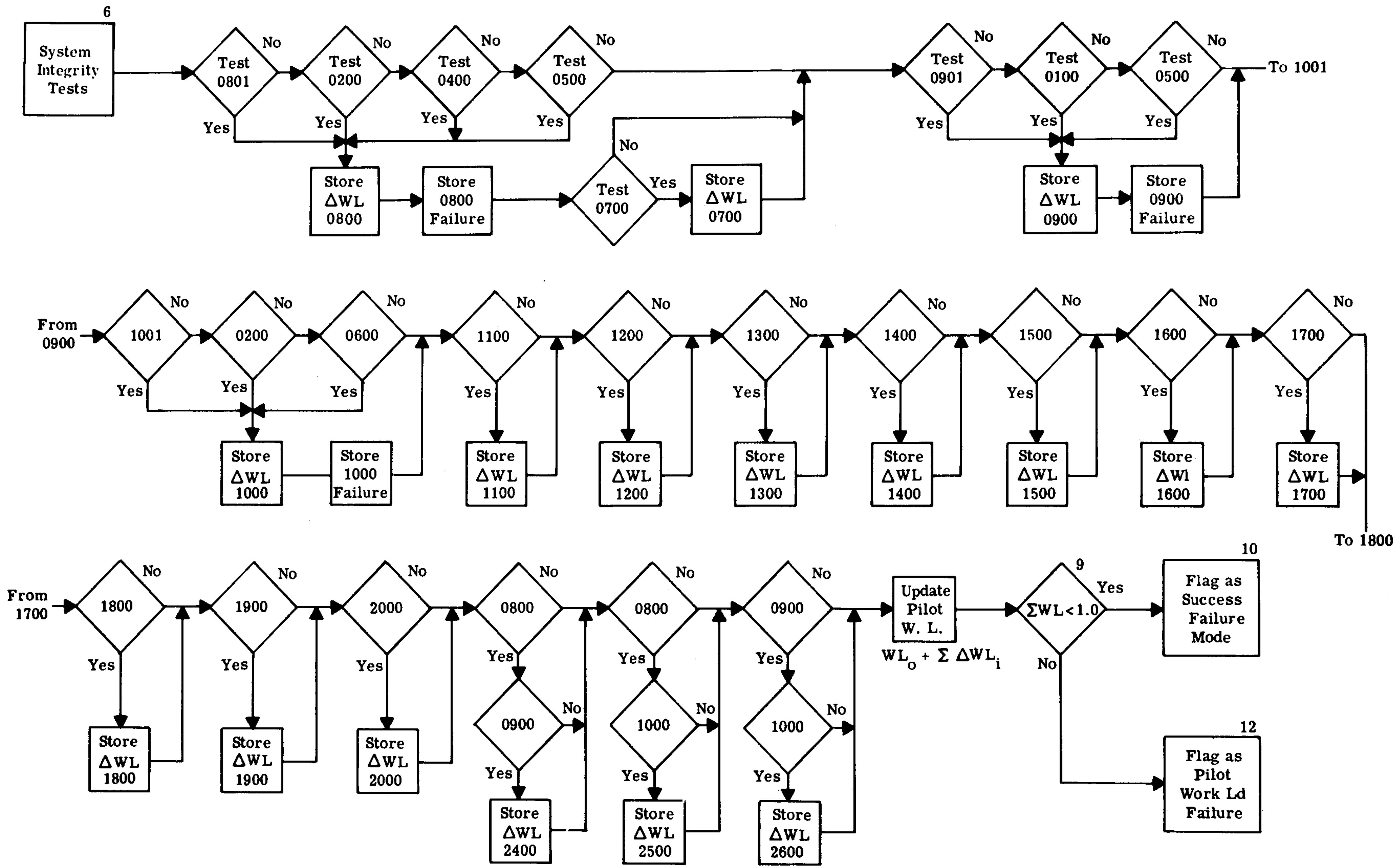

Figure 21. Logic for Pilot Work Load Bypass Testing

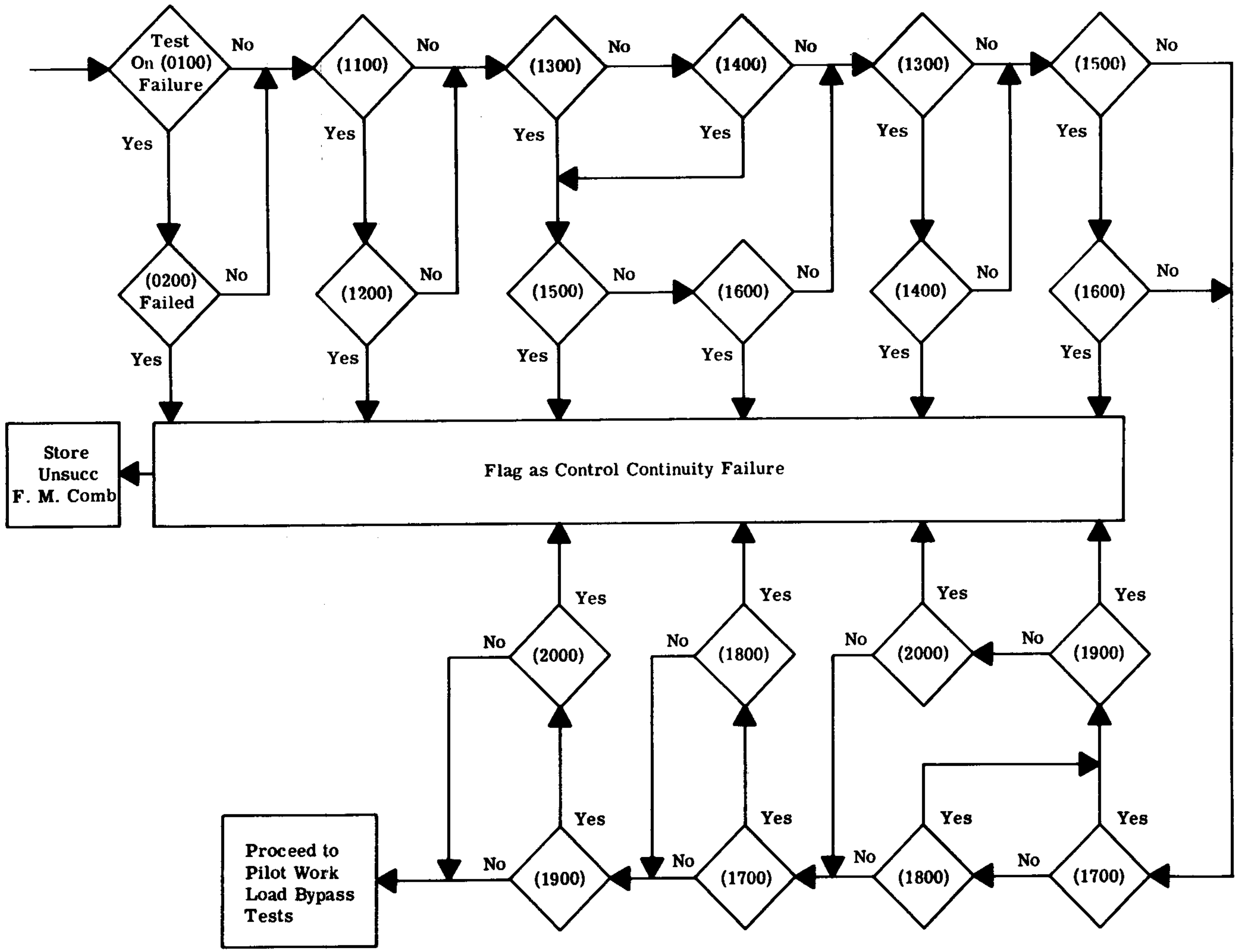

Figure 22. Logic for System Integrity Testing

As the flight control system designer is primarily interested in the effect of the more probable failures, capability was provided in the program to rank failure mode combinations in descending order of probability. Cutoff criteria are put into the program to exlude from analysis all those failure mode combinations with a probability less than a specified value and with a cumulative probability of occurrence less than a selected triviality total (8). This subject is further discussed in Appendix III.

The pilot work load and control continuity testing of the ranked failure mode combinations separate those failure conditions which result in mission failure from those which permit successful accomplishment of the defined mission.

Those component failure modes for which there is no backup path (catastrophic single failures) do not enter into the pilot work load and control continuity testing. The product of their reliabilities is instead multiplied by the summed success probability of the individual failure mode combinations to arrive at the overall probability of mission success.

Examples of the program output sheets listing the catastrophic (excessive pilot work load and backup control continuity) and noncatastrophic failure mode combinations are shown in Appendix III.

12. EVALUATION OF RESULTS - PCI PHASE 11

Evaluation of the output data from the digital analysis of mission success probability indicates that the flight control system reliability exceeds the goal of 0.96 per hour of flight. The computed probability of successful mission accomplishment for the evaluation mission with a duration of 44 minutes and 27 seconds is 0.988. Normalizing the computed reliability on the basis of one hour of flight reduces the value to 0.984. It is this last figure which is compared with the original reliability goal.

A breakdown of mission failure probability by failure categories indicates the following distribution:

Failure Category	Mission Failure Probability
(1) Catastrophic Single Failure Modes (CSFM)	1.08%
(2) Pilot Work Load Failures	0.04%
(3) Control Continuity Failures	One "significant" failure with a probability of 7.05×10^{-4}%
Total Mission Failure Probability	1.12%

(8) For the present study, all FMC's with a probability less than 2×10^{-6} were excluded, and the cumulative triviality total was set at 0.08%.

The above table indicates the predominant influence of catastrophic single failures on the probability of not successfully completing the defined mission.

A detailed analysis of the catastrophic single failure modes and pilot work load failures indicated the individual failure modes listed below to be the major contributors to mission failure.

Failure Mode	Mission Failure Probability
(1) Elevon Control - Hardover	0.39%
(2) Elevon Control - Soft (ranked with CSFM in mission segment 3)	0.28%
(3) Propeller Blade Control - Hardover	0.18%
(4) Dual a-c El. Power Generation	0.08%

The above table indicates the extremely large influence of failures in the four elevon control systems on the total probability of mission failure - contributing to 60% of the mission failures. It is therefore apparent that the greatest improvement in probability of successful mission accomplishment will result from redesign of the elevon control systems.

13. ITERATION OF PILOT-CONTROLLER INTEGRATION PROCESS - PCI PHASE 12

In the event that the probability of mission success goal is not met, suitable modifications must be made to those control system components identified in Phase 11 as being the major contributors to mission failure.

The dual objective in the redesign process is to reduce pilot work load to tolerable levels and to increase component reliabilities such that the mission success goal will be met or exceeded upon subsequent iteration through the PCI process.

As indicated in Phase 11, the success goal for the defined mission has been exceeded, and iteration is therefore not required. However, as the evaluation of digital program results indicated an extremely large contribution on the part of elevon control failures (4 units) to the total probability of mission failure, the decision was made to proceed with a hypothetical redesign of these units.

a. Redesign of Elevon Control Servo Actuator Systems

In the original Mark I [9] elevon design, a hardover or frozen failure of one of the four units results in a restriction in the freedom of control motions all the way

[9] The original and redesigned control systems will henceforth be referred to as the Mark I and Mark II systems, respectively.

back upstream to the pilot's stick and pedals. This failure condition was thus of the catastrophic single failure variety resulting directly in loss of control of the aircraft.

In order to eliminate the possibility of the hardover/frozen control condition from feeding upstream from the failed unit, load reliever springs were installed on the drive (command signal input) linkage of each elevon actuator.

In addition, the elevon actuator control valve assemblies were redesigned to reduce high stress points in the pressure feedback piston rod extension and the control valve piston rod in the event of need to break loose chips jamming the valves. The Mark I actuator system is shown in Figure 23. As indicated in Figure 24, the pressure feedback assembly was dualized with the two piston rods joined to the input linkage through a U-link rather than through the original L-shaped extension. The diameters of the control valve cylinders, pistons, and piston rod were increased to provide sufficient strength (>500 lb) to break free chips jamming the valve.

b. Prediction of Component Failure Modes

The installation of load relievers on the elevon input linkages lessens the effect of hardover/frozen failures in regard to aircraft control capabilities, but does not alter the basic elevon failure modes. Also, redesign of the control valve assemblies will reduce the rates of failure but will not alter the modes.

c. Categorizing of Failure Modes by Degree of Severity

The severity of single elevon hardover failures has been altered by the installation of the load relievers. The pilot will now have freedom of control and the capability to trim out the unbalance moments generated by a failed elevon (but with less control authority remaining in the direction of the failure) throughout vertical flight, translation, and transition. However, a hardover elevon will continue to be catastrophic in conventional flight due to the generation of excessive airframe structural stresses and large uncontrollable moments in the higher speed range.

d. Analysis of Complex Failures by Simulation

The simulation model of the elevon control system was modified to provide for the introduction of single elevon hardover failures.

The averaged pilot work loads for the steady state elevon hardover failure condition were:

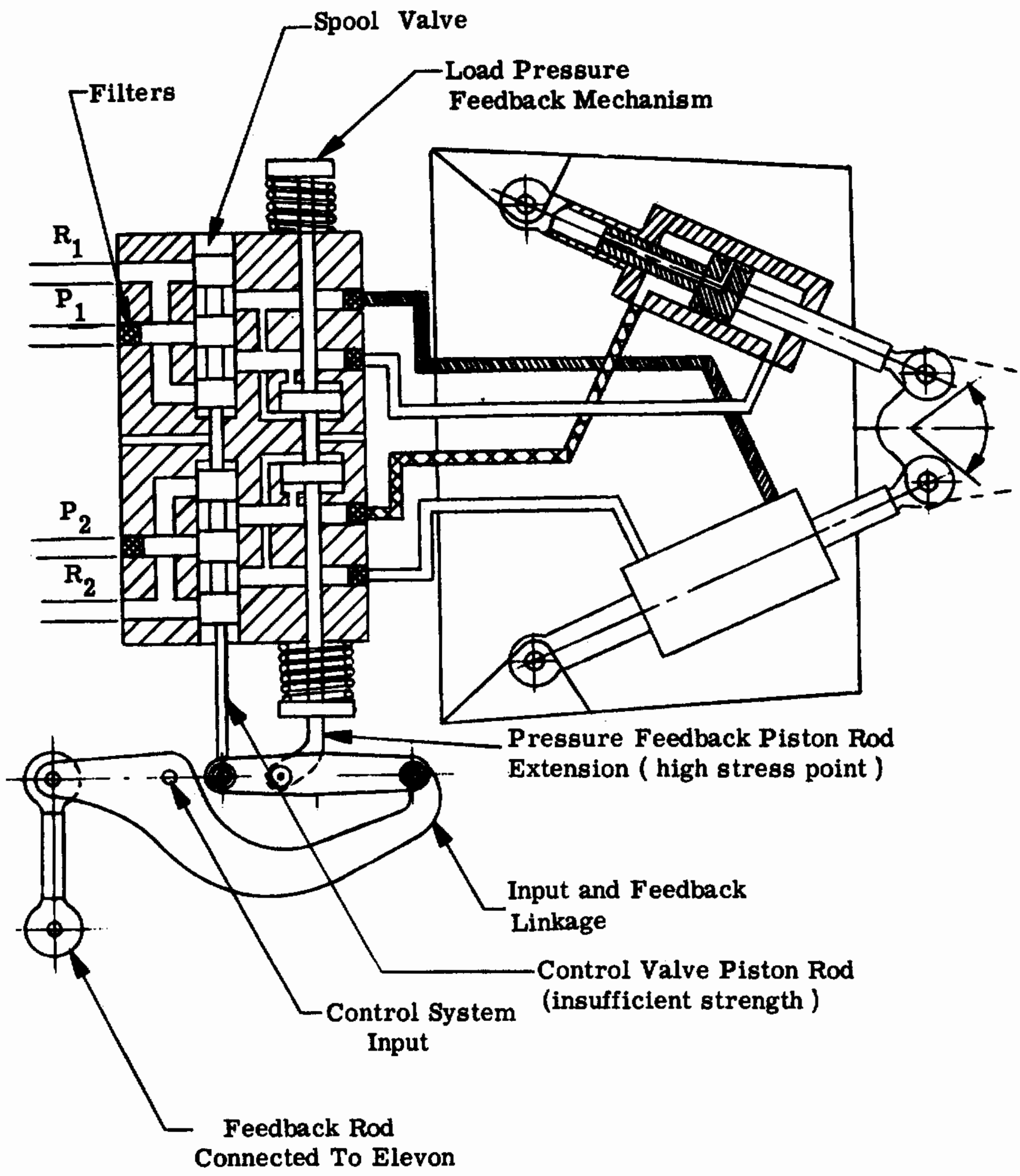

Figure 23. X-22A Elevon Actuation System - Mark I Design

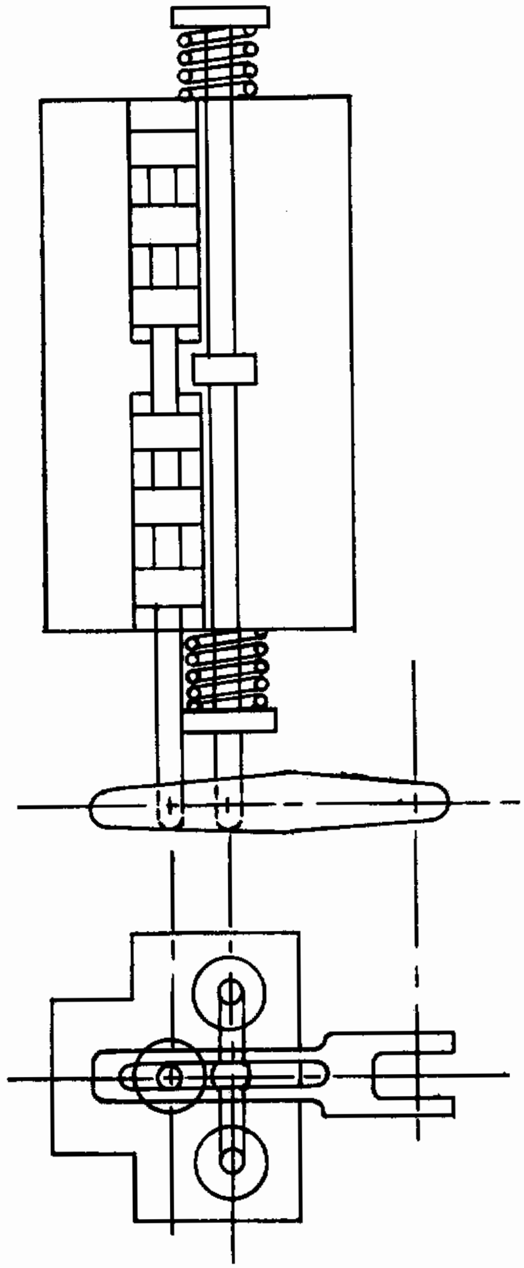

Figure 24. Elevon Actuator Valve Assembly - Mark II Design

	ΔWL
Vertical Flight and Translation (mission segments 1 and 5)	0.17
Takeoff and Landing Transitions (mission segments 2 and 4)	0.21
Conventional Flight (mission segment 3)	catastrophic single failure (not tested)

Pilot transient control response to the introduction of active hardover elevon failures in mission segments 1, 2, 4, and 5 indicated no situation where loss of control occurred [10].

e. Prediction of Component Failure Rates

New failure rates were predicted for the Mark II elevon control systems. The generic failure rates for the Mark II system are compared with those for the Mark I system in Table IX. The payoff resulting from the redesign indicates a 63% reduction in hardover failures and a 55% reduction in major failures.

f. Digital Analysis of Mission Success Probability

Modifications were made to the digital program input to account for the changes in elevon control system failure mode severity and failure rates as indicated in Paragraphs II.13.c and II.13.e.

g. Evaluation of Results

The results from the digital analysis of mission success probability for the redesigned Mark II control system have been evaluated and are compared with the Mark I system in the following two tables:

(10) The introduction of the transient failure situation at low duct angles ($\lambda < 30°$) during the landing transition resulted in loss of control in approximately one third of the test cases. As it was felt that the loss of control resulted primarily from inadequate presentation of vehicle attitude visual cues (to the pilot), this failure has not been treated as being catastrophic.

Failure Modes	Mission Failure Probability Mark I System	Mark II System
(1) Elevon Control - Hardover	0.390%	0.116%
(2) Elevon Control - Soft	0.280%	0.126%
Total Flight Control System	1.129%	0.702%
	Probability of Successful Mission Accomplishment	
Defined Mission	98.791%	99.218%
Per Hour of Flight	98.4%	98.9%
Total Trivial Probabilities (not included in digital analysis)	0.08%	0.08%

The payoff resulting from application of the PCI technique to the X-22A flight control system represents better than a 35% improvement in ability to successfully complete the defined mission [11].

(11) The improvement in probability of successful mission accomplishment was computed as follows: (99.22 - 98.79) x 100/(100.00 - 98.79) = 35%.

TABLE IX

ELEVON ACTUATOR CRITICAL FAILURE MODES: ORIGINAL AND REDESIGNED UNITS

Component Description	Malfunction	Effect on Unit Operation	Failure Modes* Catastrophic Orig.	Catastrophic Redes.**	Major Orig.	Major Redes.
Control Valve: Piston Rod	Broken Rod	Hardover Elevon	4	2	-	-
Load Pressure Feedback	Broken Piston Rod Extension	Hardover Elevon	14	2		
	Excessive oil seal leakage resulting from normal force loading on seal	Nonlinear elevon response and vibration			12	1
Summation Linkage (-7 Link)	Broken Linkage	Hardover Elevon	3	3		
	Worn Linkage	Excessive Vibration; hysteresis			5	5
Drag Link (Feedback)	Broken Link	Hardover Elevon	1	1		
	Worn Rod End	Excessive Vibration: hysteresis			2	2
Hydraulic Actuator: Piston Rod	Worn Rod End	Hysteresis	-	-	1	1
Total Generic Failure Rates per Elevon Actuator System:		Original Design	22		20	
		Redesigned		8		9

* Failure Rates per 10^6 Hours

** Treated as catastrophic single failures in conventional flight only. Major failures in vertical flight, translation, and transition.

SECTION III

APPLICATION PAYOFF

As a result of the hypothetical control systems redesign based on application of the PCI procedures to the X-22A, the following statements can be made in regard to improvements in man-machine systems performance.

1. MISSION RELIABILITY

The improvement in mission reliability (successful mission accomplishment) resulting from elevon control redesign is summarized in the discussion of the Iteration procedure rework efforts - paragraph II.13, Evaluation of Results. The improvement resulting from application of the PCI technique to the X-22A flight control system represents better than a 35% increase in ability to successfully complete the defined mission.

2. FLIGHT SAFETY

Flight safety has been enhanced by the installation of load relievers on the elevon input linkages. With load relievers installed, the pilot now has freedom of control and the capability to trim out the unbalance moments generated by a hardover elevon failure throughout vertical flight, translation, and transition. Such elevon failures will continue to be catastrophic in conventional flight due to the generation of excessive airframe structural stresses.

The combined effect of load reliever installation and redesign of the elevon actuator servo valve assemblies, in addition to the improvement in probability of successful mission accomplishment, has been to increase the probability for successful abort in the event of failure situations occurring which, although not immediately catastrophic, will result in loss of the aircraft at some future flight condition if the defined mission profile is adhered to.

Analyzing the output from the digital program on the basis of flight safety (including the ability to successfully abort the mission with safe recovery of the aircraft) indicates the following:

Control System Configuration	Probability of Successful A/C Recovery*	Probability of Successful Mission Accomplishment
Mark I	98.97%	98.79%
Mark II	99.42%	99.22%

* Computed on the basis of no catastrophic failure situations developing after the decision has been made to abort.

The above simplified analysis of successful aircraft recovery was predicated on the stated assumption. A more exact analysis would require extensive modification to the digital program which is beyond the scope of the present contract.

3. MODIFICATION COSTS

The total nonrecurring costs for design, development, test and checkout for the load relievers and redesigned elevon servo valve assemblies is estimated to be $7000 to 7500.

The fabrication and installation costs per aircraft are estimated to be: $13,200 if the redesigned elevon control systems are installed as original equipment; or $19,500 to 20,000 if the redesigned control systems are retrofitted to aircraft previously equipped with the original control systems.

There would be some decrease in the servicing and maintenance requirements/ costs for the Mark II system compared with the Mark I; however, as there is no field service information available on the subject systems, it is not at this time possible to quantitatively evaluate the cost savings.

It is not anticipated that there would be any appreciable difference in maintenance personnel or crew training requirements between the Mark I and Mark II systems.

4. OPERATIONAL EFFECTIVENESS

Studies have been conducted to determine the payoff in operational effectiveness resulting from the control systems redesign. For 1000 sorties of the type of the sample defined mission, the following savings would result from the Mark II redesign:

reduction in the number of aircraft lost in flying 1000 sorties is 4.4 with the Mark II system as opposed to Mark I

value of the difference in aircraft losses is 4.4 x $1,500,000** = $6,600,000

if the number of A/C used to fly the 1000 sorties is 20, the savings would be $6,600,000 - 20 A/C x $20,000 retrofit cost per A/C = $6,200,000

if 100 aircraft were used, the savings would still be $4,600,000

** Price per aircraft is based on a total purchase of 200 to 300 vehicles

5. COST EFFECTIVENESS OF PCI APPLICATION

The cost effectiveness payoff resulting from application of PCI to the X-22A is based on a comparison of the manpower required to implement PCI in the FCS design effort with the flight time equivalence in probability of aircraft loss without PCI.

The manpower required for the PCI application to the complete FCS of the X-22A was estimated on the basis of the limited application performed under this program (refer to Introduction and Summary) modified to provide complete systems coverage in sufficient detail for actual vehicle design.

Manpower Distribution	Hours
Systems Analysis	1900
Systems Design	200
Reliability Analysis	600
Human Factors	700
Test Pilots (Simulation)	400
Total	3800

The above estimate is based on the following conditions:

(1) Assumes a basic knowledge of the PCI technique including familiarity with the digital program for ASMP.

(2) Includes only that effort which is unique to PCI, and which would not normally be included in a "conventional design" program.

(3) Assumes two iterations of the PCI procedure needed to attain the required reliability design goal(s) (PSMA and PSAR).

(4) Does not include hours required for component redesign, as those were covered in paragraph III.3.

(5) Assumes 2000/2500 hours for basic reliability analysis and consequent systems redesign to be a part of the normal prototype vehicle design effort.

On the basis of the above manhour estimate, the cost of including PCI in the FCS design effort would be approximately $60,000.

From paragraph III.2, the increase in probability of successful aircraft recovery resulting from PCI application is 99.42-98.97 or 0.45% (assuming same improvement as in present application). Inverting this value results in an equivalent $\triangle$ MTBF of 222 hours.

The flight time equivalence to the cost of PCI is

$$\Delta t \cong \frac{\text{PCI Cost}}{\text{A/C unit cost}} \times \Delta\text{MTBF}$$

$$\cong \frac{60{,}000}{1{,}500{,}000} \times 220$$

$$\cong 8.8 \text{ hours of flight time}$$

The above computation indicates that only 8.8 aircraft hours would have to be flown before the manhour investment in PCI would be fully recovered.

SECTION IV

PILOT CONTROLLER INTEGRATION CONCEPT IMPROVEMENTS

The recommended concept improvements discussed in this section evolved as the PCI application to the X-22A aircraft progressed. These refinements were the result of objective evaluation of the existing techniques (as defined in Reference 1), with the aim of providing the necessary validity, utility, and ease of application throughout the PCI procedures to satisfy the primary contract objective; i.e., the application of pilot-controller integration techniques to a representative V/STOL aircraft.

1. PILOT-CONTROLLER INTEGRATION PHILOSOPHY

Application of PCI in the design procedure will provide - in addition to the previously established payoffs in the prediction of mission success capability and logical basis for systems redesign - the following:

(1) Measures of ability to successfully recover the aircraft by following selected abort procedures and flight profiles in the event of certain failure situations occurring which, although not immediately catastrophic, will result in loss of the aircraft at some future flight condition if the defined mission profile is adhered to.

(2) Factors essential to the prediction of control systems maintainability and logistics requirements.

(3) A basis for the estimation of systems modification costs.

(4) A basis for operational/cost effectiveness evaluations of systems redesign.

2. ESTABLISHMENT OF PILOT-CONTROLLER RELIABILITY GOALS

Reliability goals may be specified both on the basis of probability of successful mission accomplishment and probability of successful aircraft recovery - considering appropriate abort procedures and flight trajectories.

3. SUCCESS DIAGRAM PHILOSOPHY AND CONSTRUCTION

Success diagram (SD) layout is based primarily on conventional reliability diagram construction and employment.

In order to clearly and concisely define the failure effect relationships between the various subsystems and components of the flight control system, the SD - for each mission segment - has been divided into two separate parts:

(1) A system continuity diagram - includes only those failure modes which singly or in combination affect systems continuity.

(2) A work load diagram - includes only those failure modes which result in changes in total pilot work load.

The logic governing the continuity diagram remains the same as established in Reference 1; i.e., all components required in the primary mode of operation are in series, and backup/redundant components are shown in parallel with their respective primary components. Thus, any combination of failures which breaks the continuity through the diagram is a catastrophic condition.

The logic governing the pilot work load diagram is as follows:

(1) Test for continuity through the logic blocks representing a single component group or section failure.

(2) Successful passage indicates no work load increment for that component group or section.

(3) No passage indicates the addition of the associated work load increment.

(4) After all component groups/sections have been tested in the above manner, the indicated work load increments are summed with the basic work load to arrive at a total work load for the particular failure conditions being tested.

Workload diagram philosophy also provides for the incorporation of "interference effects" between certain failure modes (Refer to Sections II.9 and IV.6 for further explanation and discussion of workload interference). Workload interference is accounted for in the success diagrams by the inclusion of dummy sections with the contributing individual failure modes shown in parallel.

4. CATEGORIZING OF FAILURE MODES BY DEGREE OF SEVERITY

The scope of PCI Phase 6 has been expanded from "the identification of non-catastrophic failures" to "the categorizing of failure modes by degree of severity"; i.e., separating the failure modes into catastrophic single failures, potentially catastrophic failures, and noncatastrophic (trivial) failures.

5. REORIENTATION OF PCI PHASES 7 AND 8

The orientation of Phases 7 and 8 were redefined on the basis of failure mode complexity rather than by methods of analysis. Thus, Phase 7 becomes "the Analysis of Simple Failures" rather than "Failure Analysis by Paper and Pencil Methods" and Phase 8 becomes "Analysis of Complex Failures" rather than "Failure Analysis by Simulation".

The revised orientation of these phases is felt to better reflect the progression in degree of analysis sophistication commensurate with the increasing complexity and detail definition of the vehicle response representation as the design procedure progresses from the preliminary to the final detail stages.

6. PILOT WORKLOAD AND TRANSIENT CONTROL INTERFERENCE

The possibility of workload and/or transient control capability interactions between concurrent or excessive failures is not a new or unique situation. Multiple failures, each of which affects the dynamic response of a vehicle in the same degrees of freedom, are the most obvious example of interference - as represented by a dual damper failure.

An attempt to analyze through paper-and-pencil or simulation techniques all possible combinations of workload and transient control response interactions could become a ponderous job for a system with appreciable nonlinear variations in vehicle aerodynamic characteristics.

A more logical approach for the analysis of such situations is:

(1) On first pass through the PCI procedure, consider only single failure effects (including possible secondary failures) on workload and transient control capabilities

(2) Skip over PCI Phases 7 and 8, and proceed directly to Part I of the digital program for the analysis of mission success probability - the ranking of failure mode combinations in descending order of probability.

(3) On the basis of the ranked probabilities of occurrence, those failure mode combinations with a probability of occurrence of "significant magnitude" can be identified.

(4) The "Significant" failure mode combinations can now be analyzed to determine the degree of work load and/or transient control interference.

7. MEASUREMENT OF PILOT WORKLOAD

The quantitative accuracy of pilot work load data has a profound influence on the validity of AMSP results. The selection of techniques for the measurement of pilot work loads - especially where the primary task is one of multi-axis tracking - is deserving of considerable study. The nature of vehicle response characteristics, the pilot control-display interface, and simulation profiles are all factors which must be considered in the selection process.

8. PHILOSOPHY AND STRUCTURE OF THE DIGITAL PROGRAM FOR THE ANALYSIS OF MISSION SUCCESS PROBABILITY

Evaluation of the results from the AMSP requires, for intelligent system redesign, an ability to isolate those failure mode combinations having the major influence on the probability of mission failure. To enhance the accomplishment of the above objective, the digital program was provided with a capability to rank the failure mode combinations in descending order of probability.

A second significant feature of the ranking procedure is the provision to limit the extent of the table of the ranked failure modes by eliminating from further analysis all those FMC's with a probability less than a selected value. This feature results in a very significant saving in digital processing time and cost as opposed to analyzing all possible FMC's. In the case of the X-22A application, by eliminating all combinations with a probability less than 2×10^{-6}, the number of FMC's processes varied between 200 and 300. If all possible combinations had been processed they would have totalled $\approx 6 \times 10^{14}$. Moreover, all those combinations in excess of the largest 200 to 300 account for less than 0.08% of the total probabilities of occurrence.

The structure of the digital program for AMSP as stated in Reference 1 has been redefined and expanded upon in order to meet the requirements of the present PCI application and also to provide a basic program which will be readily interpreted and immediately usable in future applications of PCI.

A detailed description of the digital program is presented in Appendix III.

9. PREDICTED PROBABILITY OF SUCCESSFUL AIRCRAFT RECOVERY

The basic data required for predicting the probability of successful aircraft recovery are available in the output from the digital analysis of mission success probability. As the digital program is presently defined, it is necessary to manually scan through the listings of those failure mode combinations contributing to mission failure to determine which failure situations do not explicitly eliminate the potential for successful mission abort. The potential for mission abort should be based on a pilot's anticipated ability to analyze the failure situation and make the proper decisions. Pilot capabilities in such situations are dependent upon: failure warning and indication; knowledge of vehicle systems and emergency operating procedures; environmental conditions; availability of landing sites; etc.

Automating the prediction of successful aircraft recovery probabilities is discussed in Section V.1.

10. REORIENTATION OF PCI PHASE 11

To better reflect the importance of the dual tasks originally included in Phase 11, this phase has been split in two. Phase 11 now constitutes "Evaluation of Results" - including the payoffs in operational effectiveness resulting from systems redesign (refer to Section IV.1). "Iteration of the PCI Process" has been redesignated as Phase 12.

SECTION V

SUGGESTED AREAS FOR FUTURE DEVELOPMENT AND EXPANSION OF PCI DESIGN PROCEDURES

The subjects discussed in this section represent areas in which additional effort is required to bring the PCI procedures to a degree of utility sufficient to satisfy the demands of general usage and application.

(1) Develop methods to incorporate pilot ability to redefine missions downstream of systems failure based on foreknowledge of future effects of a given failure on ability to control. The pilot capability to analyze such situations may result in a decision to:

  (a) Modify the proposed mission profile but still anticipate successful completion of mission objectives.

  (b) Abort the mission and return to base (or alternate landing site).

  (c) Sustain controlled flight for a sufficient time to evacuate the aircraft or initiate an emergency landing.

(2) Investigation of techniques for the measurement of pilot workload for primary tasks involving multi-axis tracking and control response.

  (a) Auxiliary Task - Approach I

  (b) Measurement of Pilot Control Response (gain and frequency) - Approach II

  (c) Study correlations between pilot workload and pilot opinion rating.

  (d) Multiple crew participation in performance of the primary task.

(3) Expand the flexibilities and capabilities of the digital program for the analysis of mission success probability to include:

  (a) Analysis of items (1) (a) and (b) from above.

  (b) Provide greater flexibility in the inputting of subsystem component relationships.

  (c) Provide print out of failure combinations contributing to mission failure grouped according to individual failure modes.

  (d) Provide selectable reliability equations to compute proper probabilities of success and failure based on subsystem component configuration.

(4) Compile a handbook for use of the PCI procedures.

SECTION VI

CONCLUSIONS

The primary objective of this program has been to provide a detailed example of the application of the PCI technique with secondary objectives of evaluating existing procedures and expanding them as required to perform the primary objective. Accomplishment of these objectives within the scope of the program necessitated the exclusion of certain of the X-22A flight control subsystems from the sample application, and the simplified treatment of others (refer to the Introduction and Summary for more detail in this regard). The above restrictions are not regarded as being limiting factors in the accomplishment of contract objectives, as they resulted primarily in a reduction in the quantity of data to be analyzed and did not affect the quality of data generation and analysis.

The sample application of the PCI techniques to the X-22A V/STOL aircraft has demonstrated the general utility and effectiveness of the procedure. Primarily, PCI provided information essential to the systematic and logical design and development of flight control systems as follows:

(1) The prediction of mission reliability.

(2) An evaluation of the effects of individual subsystem/component failures on mission reliability - this evaluation provides the basis for systems modification and redesign.

(3) The basis for evaluating the payoff accruing from systems redesign in the areas of:

   (a) Mission reliability
   (b) Flight safety
   (c) Costs of modification, redesign, and training
   (d) Systems maintainability and logistics support requirements
   (e) Operational/cost effectiveness.

As the PCI application to the X-22A was not conducted with the level of effort required for the development of a prototype, especially in regard to the depth of the reliability analysis, incorporation of the Mark II systems modifications to the actual aircraft hardware were not proposed.

Throughout the sample application, the PCI procedures - as originally stated in Reference (1) - were evaluated and a number of improvements (Section IV) were incorporated.

Those limitations which still exist in the PCI procedure are felt to be basically oriented in several of the individual analytical techniques rather than in the concepts and methods of PCI itself. That is, more extensive field operational and test data are required for the prediction of failure modes and rates. Also, state-of-the-art advances are required in the accurate measurement of pilot workloads and in the determination of human operator reliabilities.

APPENDIX I

HANDLING QUALITIES STUDIES OF THE X-22A

1. INTRODUCTION AND SUMMARY

An additional objective of this program was to establish handling qualities criteria defining the limits of acceptable stability and control characteristics for the X-22A in the hover and transition modes of flight.

Handling qualities criteria are stated in terms of pilot ratings (Cooper Rating Scale) and pilot workloads as functions of longitudinal and lateral control powers and damping levels.

These studies were conducted in conjunction with the PCI failure workload and transient control capability studies (as described in Section II.9) utilizing the Bell X-22A hybrid simulator.

The X-22A pilot rating boundaries are compared with the data generated in several of the more frequently quoted handling qualities investigations (References 2 and 3).

Lateral pilot rating boundaries were found to agree quite well; however, X-22A longitudinal requirements for satisfactory handling qualities were found to be considerably higher in both control power and damping than the published data. The variation in longitudinal requirements is explained on the basis of X-22A aerodynamic (levels of static stability - M_α and M_u) and control system (feel force gradients and thrust response) characteristics.

A comparison of pilot rating boundaries with pilot workload contours for the X-22A shows good correlation in the areas of optimum control power-damping combinations.

Shifts in the quantitative correspondence of pilot rating and workload data are apparent as the definition of the control task and flight environment are varied.

The shift in pilot rating and pilot workload correspondence is accountable to the subjective nature of both pilot opinion rating and the technique of workload measurement.

It must be concluded that, based on the present state-of-the-art, the technique of measuring workloads has not bridged the gap in the quantitative assessment of operator performance loading and capability.

2. PROCEDURE FOR SIMULATOR STUDIES

Matrices of test conditions - in terms of control power-damping combinations - were defined for both the longitudinal and lateral degrees-of-freedom. The handling qualities evaluation of the above test conditions was arranged in a random manner to minimize the influence of bias in pilot ratings between consecutive test points.

The hover-vertical flight simulator task was identical to the mission profile defined in Table IV (Section II.9). The landing transition task involved a constant altitude transition from a level flight speed of 150 knots to a stabilized hover condition.

The piloting task was conducted under simulated IFR conditions with all visual cues derived from cockpit instrumentation; i.e., no presentation of an out-of-the-window visual field.

3. DISCUSSION OF STUDY RESULTS

The X-22A lateral 3.5 and 6.5 pilot ratings for hover flight correspond quite well with Salmirs and Tapscott's minimum acceptable IFR and VFR boundaries as shown in Figures 25 and 26.

The X-22A longitudinal 3.5 pilot rating contour bounds an area of higher control power ($\Delta M_c \cong 1.6$) and higher damping ($\Delta M_q \cong -1.0$) than the equivalent curves determined by Salmirs, Tapscott, and Faye - Figure 25. The primary reasons for the increased X-22A requirements are:

(1) Pitch trim control power requirements increase to 0.4/0.6 rad/sec^2 as velocity increases to 25 knots as opposed to the relative minor trim changes experienced in the referenced studies.

(2) X-22A static stability becomes destabilizing with increasing translation velocity (M_α = 0.8 rad/sec^2/rad at 25 knots).

(3) The high level of speed stability ($M_u \cong 0.0258$ 1/sec ft) results in increased control power and damping required to provide satisfactory handling qualities.

(4) The X-22A longitudinal and lateral feel force gradients (>1.0 lb/in.) have been adjudged by project pilots to be considerably higher than optimum for hover and low speed flight. This condition has some degrading influence on pilot ratings (approximately 0.5 units on the Cooper Scale).

(5) Thrust response to throttle control (represented by a 0.65 sec and time lag in the simulation) is somewhat marginal. A time lag of this magnitude has been found to degrade pilot rating of height control by 1.0 to 1.5 on the Cooper Scale with a lesser influence on attitude control.

(6) The low level of vertical damping (Z_w = 0.21 1/sec) also has some adverse effect on attitude control.

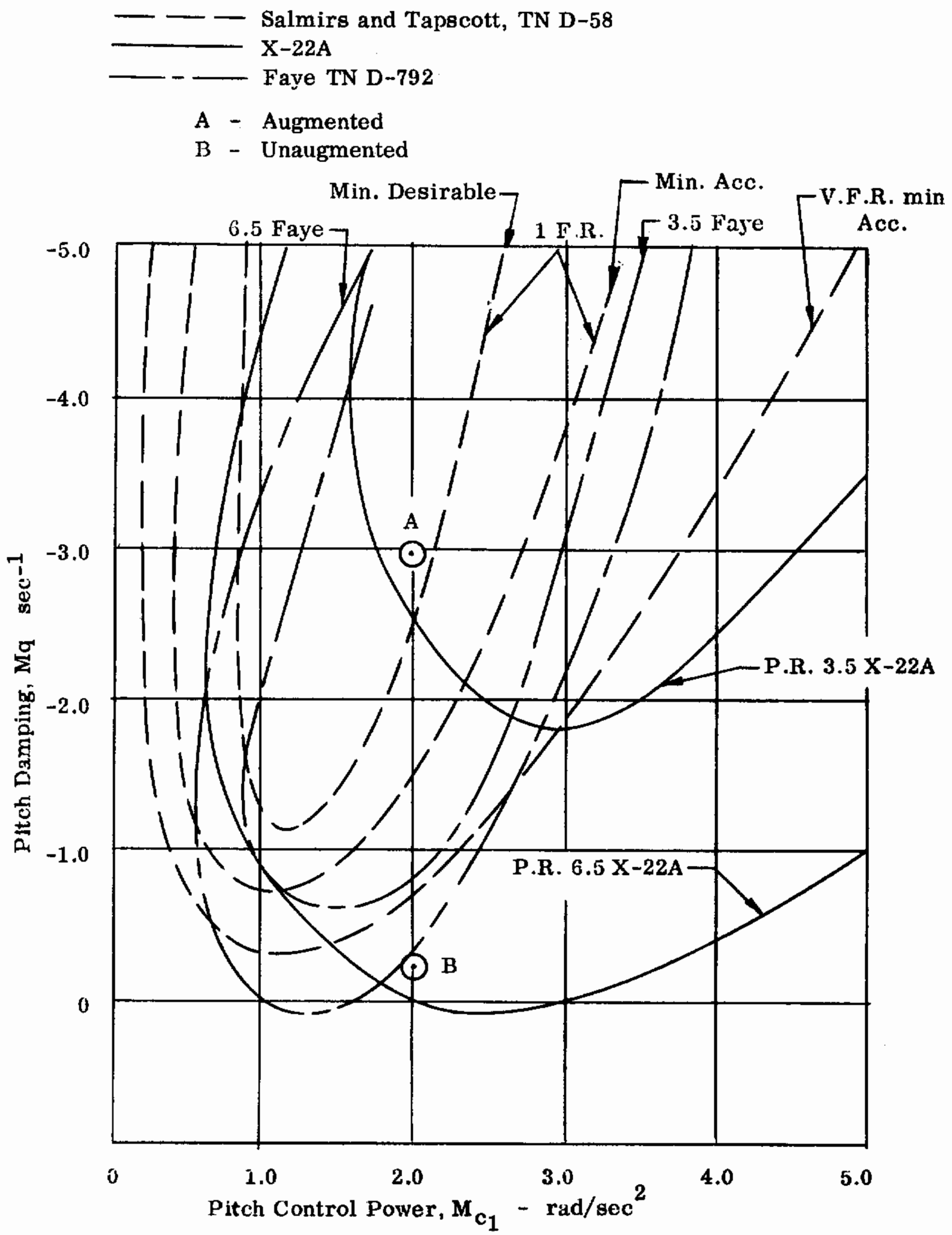

Figure 25. Longitudinal Hovering Handling Qualities Boundaries

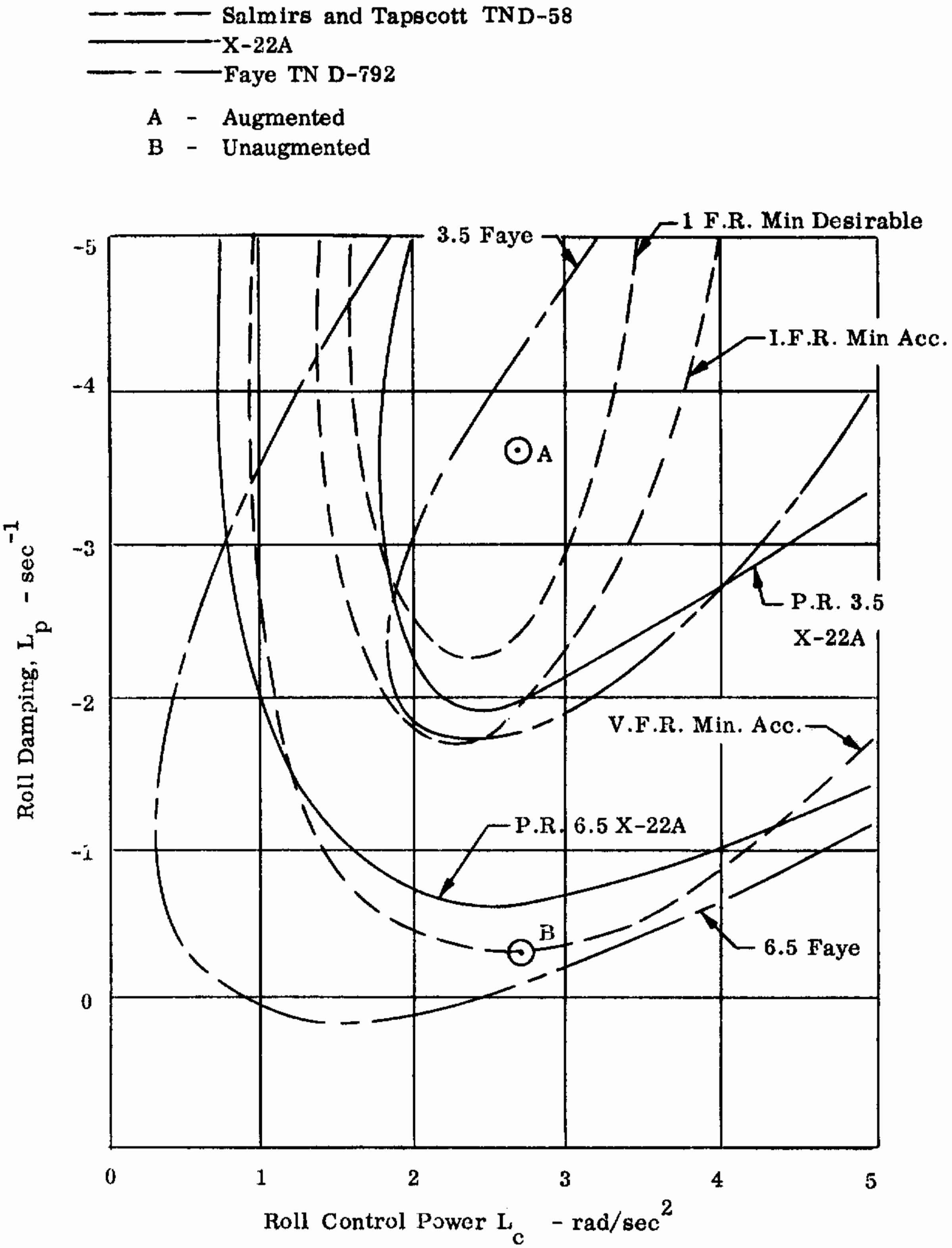

Figure 26. Lateral Hovering Handling Qualities Boundaries

It should be noted that the differences in the maneuver tasks between this and the referenced programs also contribute to the variations in pilot ratings.

The shapes of the pilot rating boundaries for landing transitions (Figure 27) are quite similar to those for hover mode flight with some upward shift in damping levels ($\Delta M_q \leq 0.5$ and $\Delta L_p \leq 0.6$).

The primary reasons for the increased damping requirements are felt to be the considerable variations in longitudinal static stability and control trim as influenced by M_α (Figure 28) and M_u (Figure 29).

Figure 30 shows the comparison between pilot workloads, PWL, and pilot ratings, PR, for the hover mode of flight. In both the pitch and roll axes a good correlation exists between PWL contours and PR boundaries, especially in the areas of the optimum control power-damping combinations. In both axes, the PWL difference between the 3.5 and 6.5 PR boundaries is on the order of 10%. This also holds true for transition flight.

Between hover and transition flight, the PR to PWL correspondence shifts in the direction of increased workload by approximately 15% in pitch and 10% in roll.

Similar shifts between mission segments are apparent in the cross plot (Figure 31) of PR's and PWL's recorded during the study of pilot control during established flight control systems failures - Section II.9 of this report.[12]

The above shifts in PR/PWL correspondence are accountable to the subjective factors implicit in the technique of pilot opinion ratings and in the state-of-the-art measurement of pilot workloads.

Workload measurement - especially where the primary task is one of continuous tracking - is exposed to subjective influences inherent in the pilot decision relative to the division of his time between performance of the primary task and operation of the measurement unit.

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[12] The scatter in workloads recorded at individual test points varied between ±4% and ±8% of maximum tolerable pilot loading.

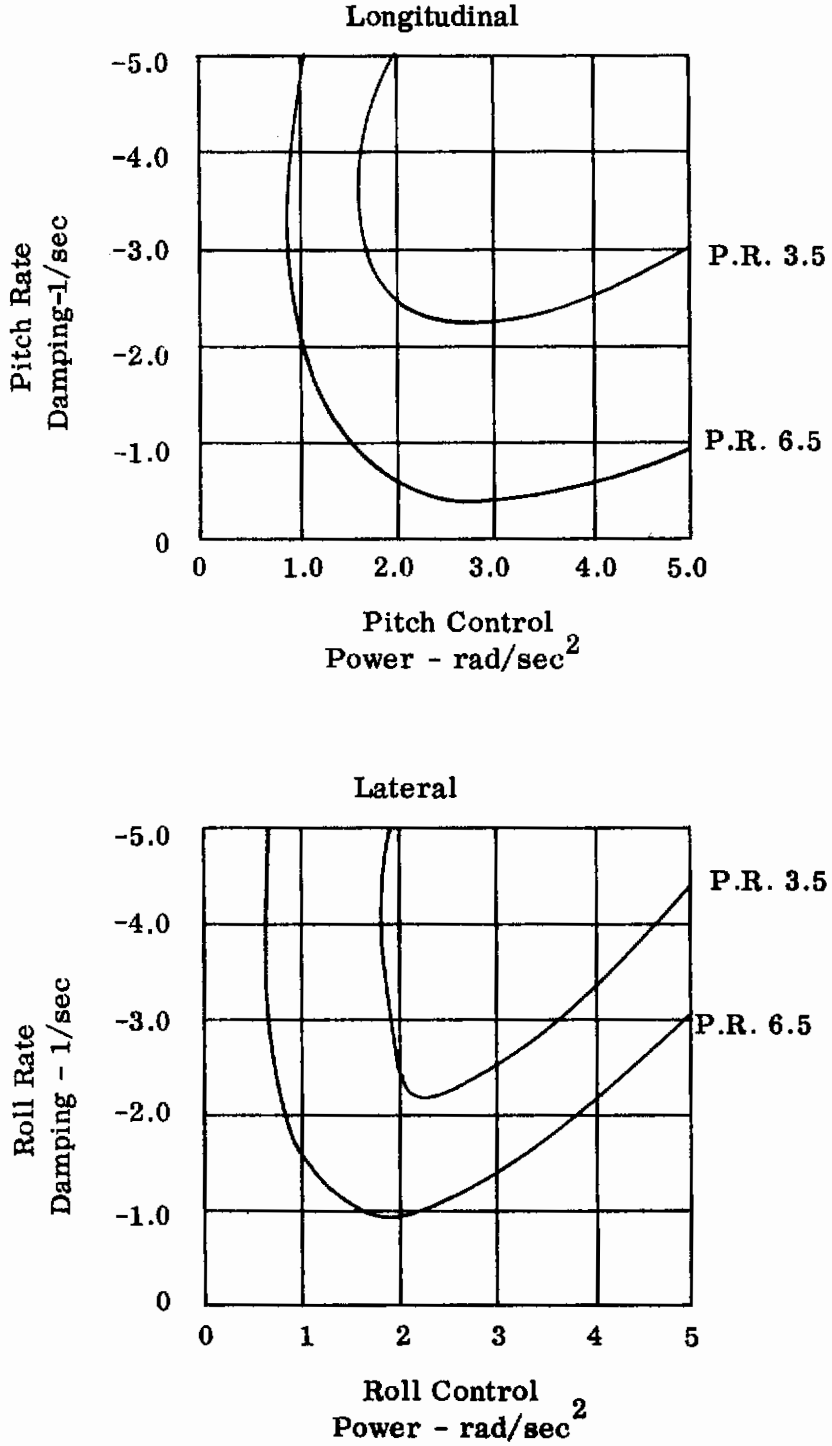

Figure 27. X-22A Handling Qualities Landing Transition

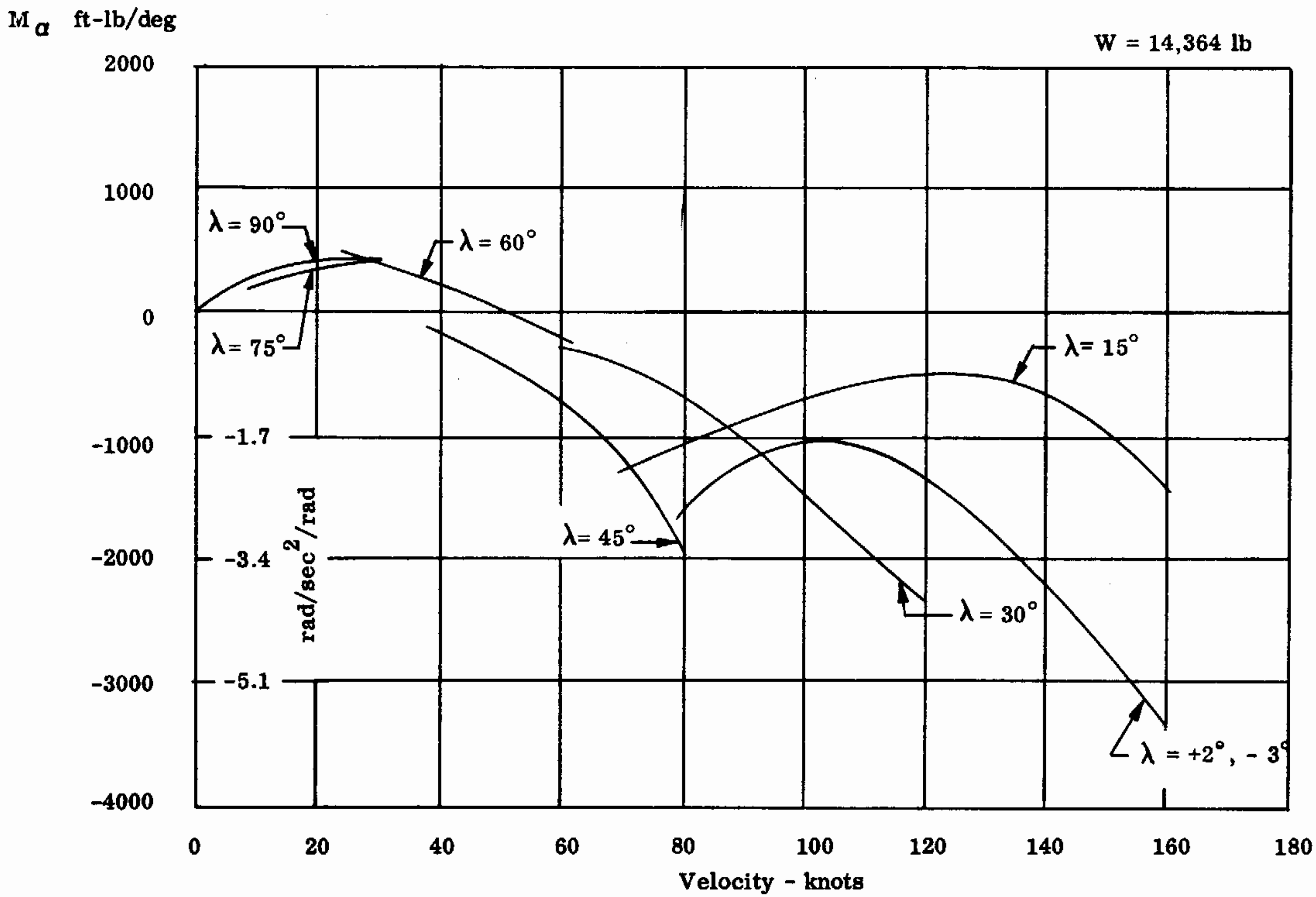

Figure 28. Linearized Derivative, M_α , for Equilibrium Level Flight in Transition

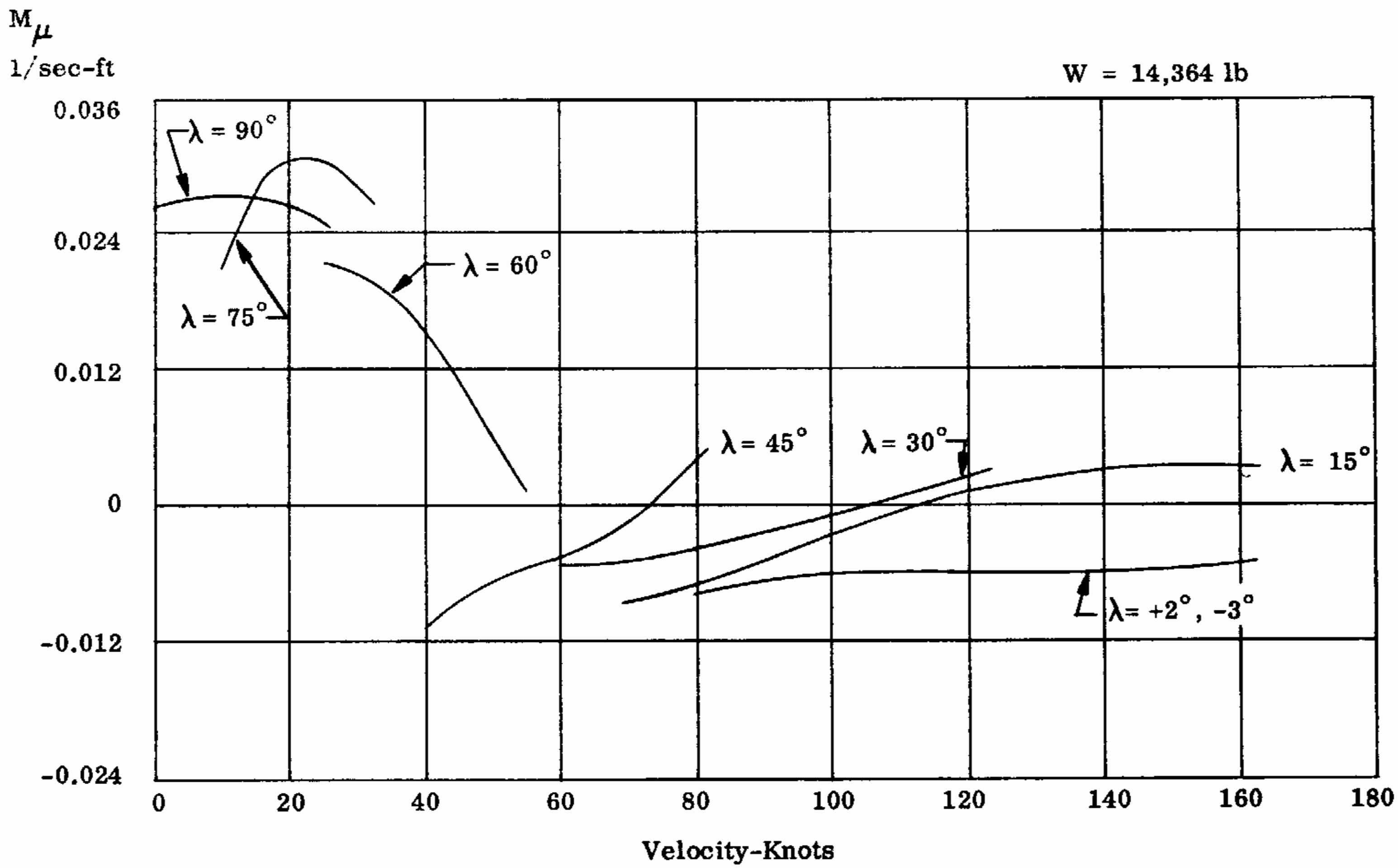

Figure 29. Perturbation Derivative, M_{μ}, for Equilibrium Level Flight in Transition

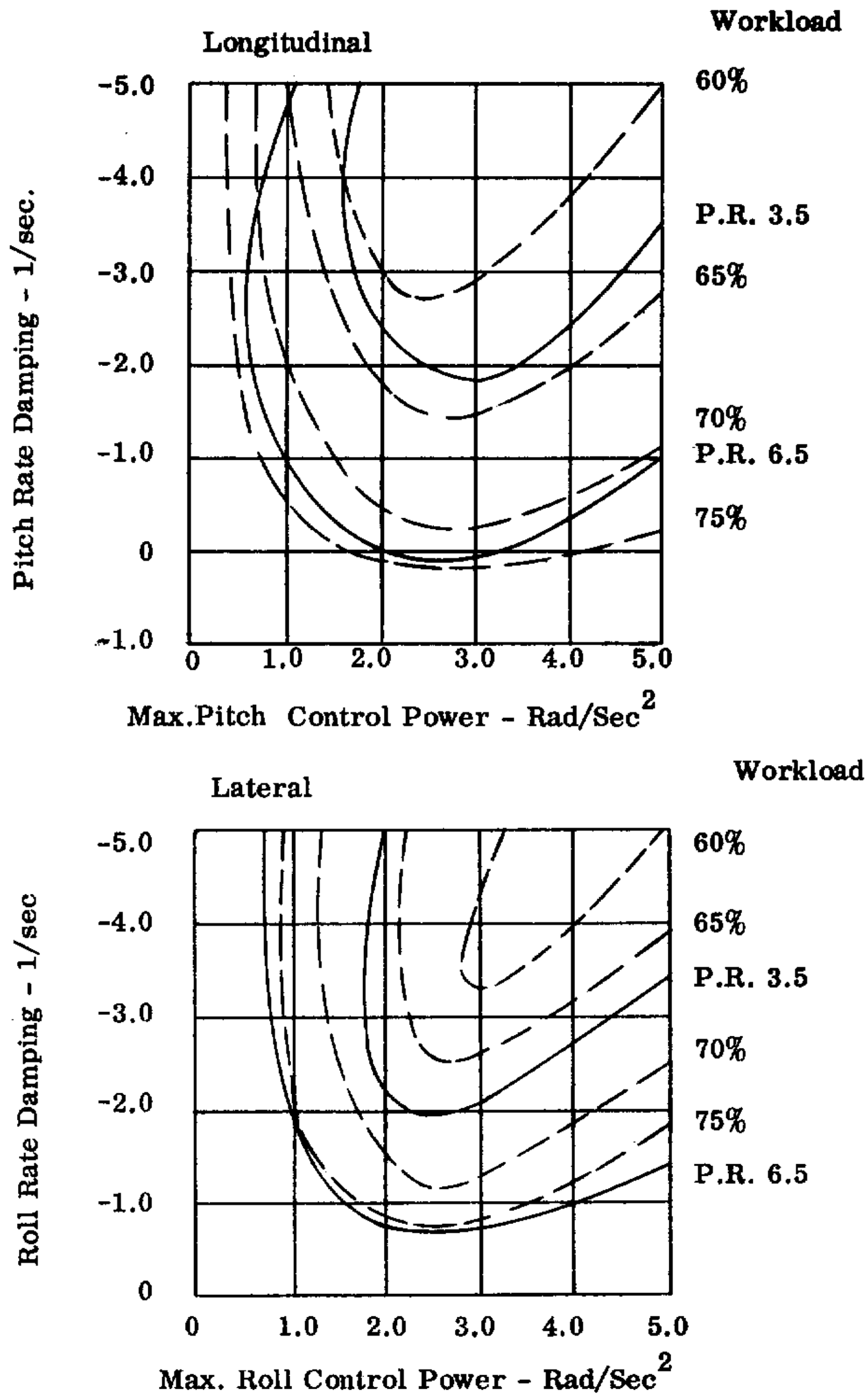

Figure 30. X-22A Pilot Workloads and Handling Qualities Hover Flight

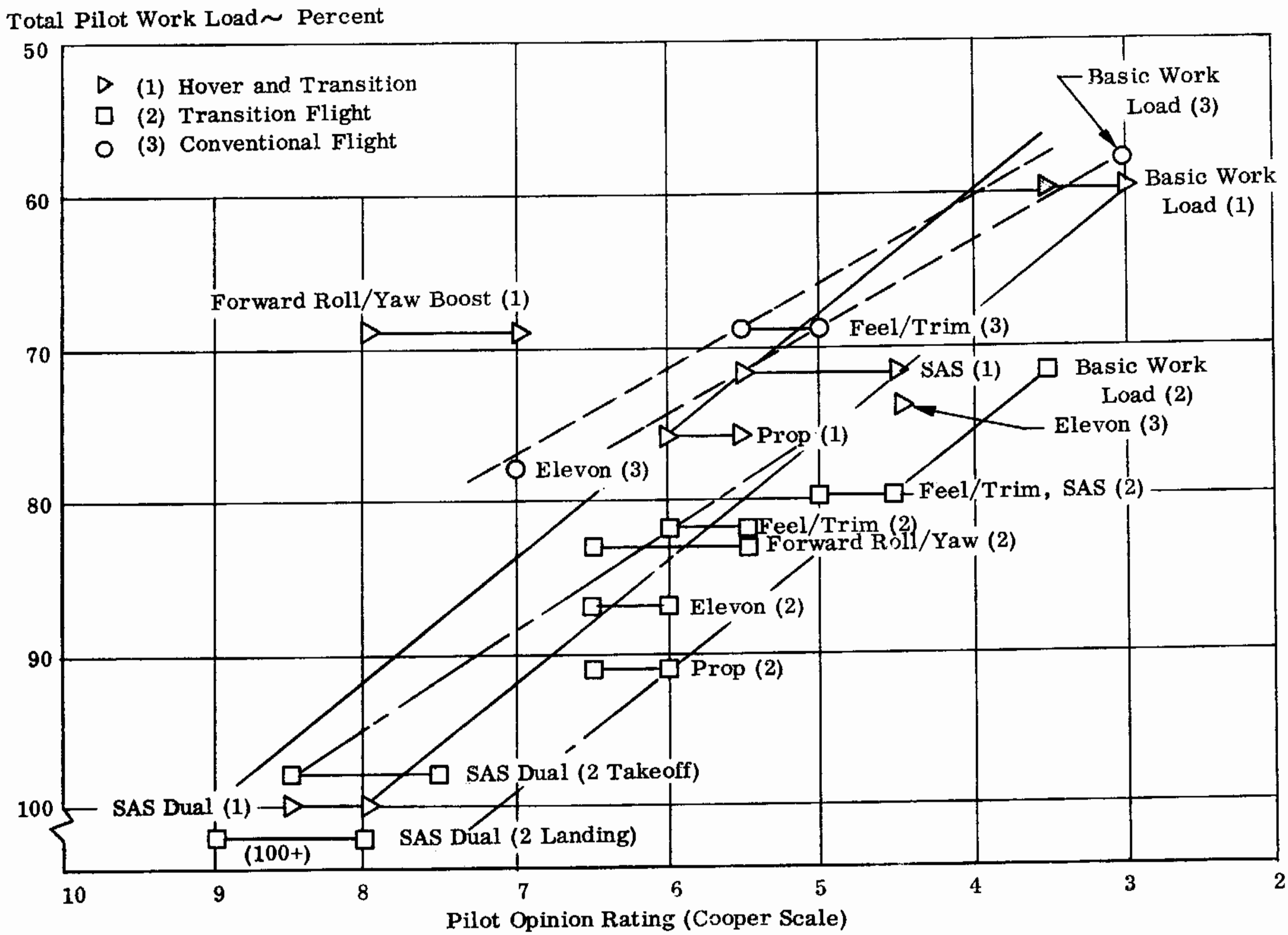

Figure 31. Pilot Ratings versus Pilot Workloads for Established Flight Control System Failures (Refer to Section II.9, Table III)

APPENDIX II

HYBRID SIMULATION OF THE X-22A

A six-degree-of-freedom (6 DOF), fixed base, hybrid simulation of the X-22A was developed by Bell Aerosystems Company under the Tri-Service V/STOL program.

This simulator has been used as a design tool in the evaluation of X-22A stability and control characteristics (handling qualities), feel/trim and flight control system design requirements, and control techniques throughout the V/STOL regime.

The simulation consists of four major operating components:

(1) Two PACE 231-R analog computers

(2) IBM Model 7090 digital computer

(3) Conversion system linking the analog and digital computers

(4) X-22A simulator cockpit

Program computation is divided between the analog and digital computers as shown in Figure 32. The mathematical model of the vehicle consisting of the aero-thrust parameters, flight control and propulsion systems representations, vehicle equations of motion, and various auxiliary equations is given in Section II.5 of this report.

The ADDA conversion system provides the interface between the analog and digital domains through 13 bit (plus sign) data conversion and logic level (sense line) transmission in both directions. The X-22A simulation utilized 15 channels of ADC and 20 channels of DAC. This system also provides the timing pulses required for real time operation of the digital computer.

The simulator cockpit (shown in Figure 20 - Section II.9) is configured in a manner similar to the actual X-22A aircraft. The simulator includes a main instrument panel, left and right side control consoles, and engine and flight controls.

The main instrument panel contains all of the active flight and power displays. These displays consist of two dual trace 5 inch oscilloscopes, eleven center pivot dial indicators, two edge reading meters, and a clock. The oscilloscopes - mounted on the pilot's visual center line - are programmed to represent various configurations of attitude and horizontal situation indicators.

The flight parameters displayed on the center pivot dial indicators and edge meters are listed in the following table along with their respective scale ranges.

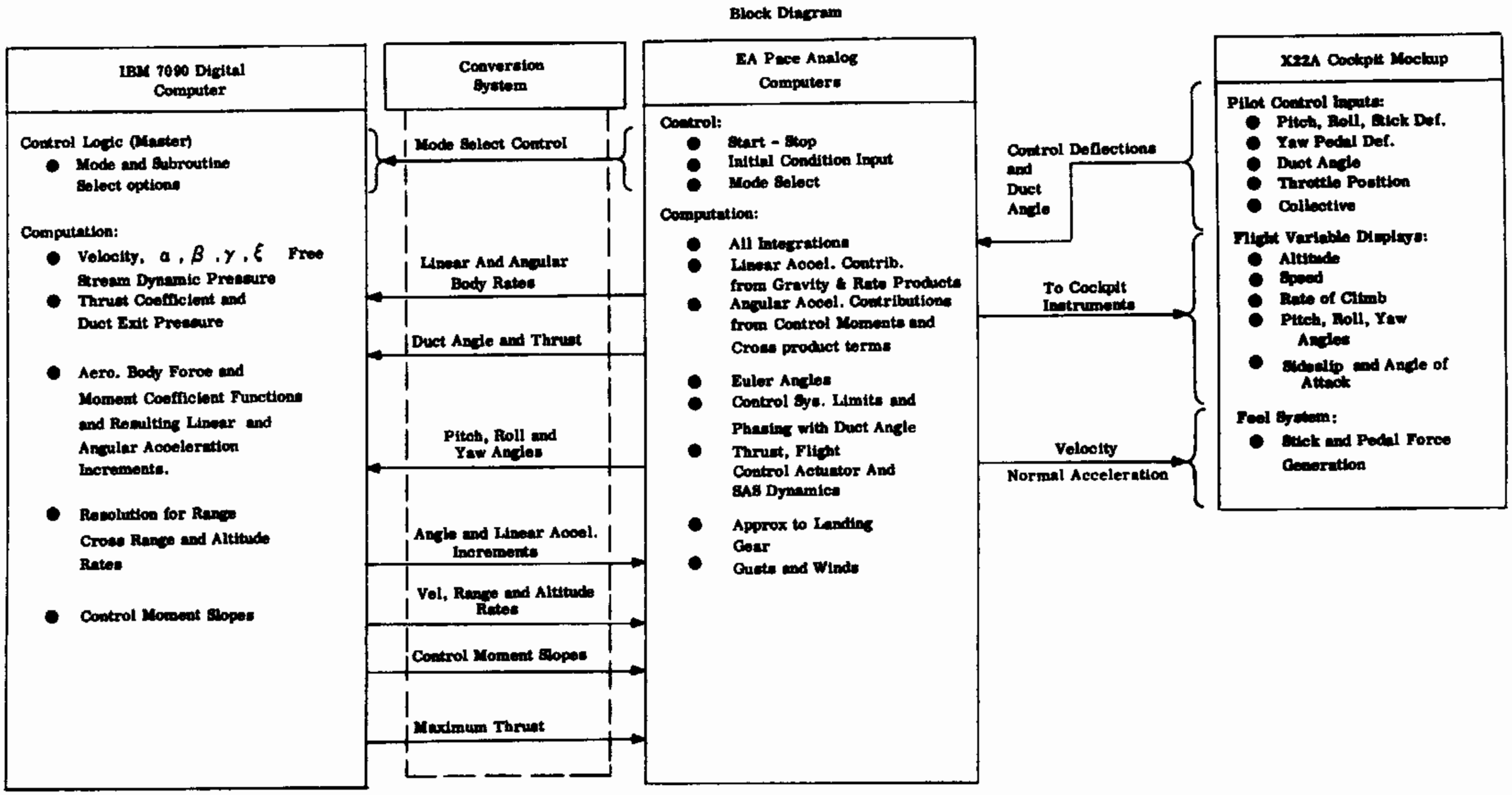

Figure 32. X-22A Analog/Digital Simulation

	Flight Parameter	Range
1.	Barometric Altitude	0 to 30,000 feet
2.	Radar Altitude	0 to 500 feet
3.	Vertical Speed	±4000 feet/minute
4.	Normal Acceleration	-3 to +6 g's
5.	Airspeed	10 to 350 knots
6.	Low Range Airspeed	-40 to 160 knots
7.	Flight Path Angle	-20 to +40 degrees
8.	Duct Angle	0 to 100 degrees
9.	Angle of Attack	-20 to +40 degrees
10.	Percent Thrust	0 to 160%
11.	Propeller rpm	0 to 3000 rpm
12.	Rate of Turn	Full left or right
13.	Workload Task	+ Error, Null, - Error

The side consoles contain the engine throttles (4), master rpm control, indicator lights and control mode selector switches.

A collective pitch control stick is located to the left side of the pilot's seat in the conventional helicopter manner.

The simulator is equipped with a conventional control stick and rudder pedals which are operated by means of an electrohydraulic feel/trim system. The feel/trim system is similar in configuration to the X-22A flight hardware. The programming of force gradients and trim rates on the basis of equivalent velocity (pitch and roll) and duct angle (yaw) is accomplished in an electronics rack adjacent to the cockpit. This rack also contains controls to make gross gain changes to the stick and pedal feel forces, trim rates and breakout forces.

APPENDIX III

DIGITAL PROGRAM FOR THE ANALYSIS OF MISSION SUCCESS PROBABILITY

1. INTRODUCTION

The purpose of the appendix is to provide a general description of the AMSP digital program capabilities and flexibilities as well as a detailed description of program operations and input/output data.

In programming the analysis of mission success probability on Bell Aerosystems IBM 7090 computer, primary considerations were given to the following requirements:

(1) Input Flexibilities - The program shall be structured such that all data defining flight control systems configurations, component failure rates, and pilot workloads are inputs to the program rather than being implicitly tied to the program structure. The input format must be defined in a manner such that engineering personnel can with a minimum of instruction prepare their own input sheets; i.e., program input written in FORTRAN IV or equivalent symbolic language. Modification of input must be accomplished with a minimum of effort; i.e., ability to change individual cards in the input deck is required.

(2) Output Requirements - Program output shall be oriented to provide that systems failure mode data which is essential to effective evaluation of mission success capability and to the determination of those failure modes which have the major influence on probability of mission failure. In order to best accomplish the above objectives, the program must be provided with the capability to rank the failure mode combinations in descending order of probability. The causes of mission failure must be identified for each FMC as to lack of control continuity or excessive pilot work load.

(3) Efficient Processing of Data - Failure mode combinations shall be ranked in descending order of probability to provide a logical basis for limiting the extent of control continuity and workload testing to those FMC's with a "significant" probability of occurrence; i.e., input a cutoff criterion to exclude from further analysis all those FMC's with a probability less than a selected value.

This approach also facilitates the second form of constraint, which involves the establishment of a cumulative triviality criteria whereby all those FMC's with a total probability less than a selected value are discarded.[13]

These features result in a significant saving in processing time and cost as opposed to analyzing all possible FMC's, and at no sacrifice in the quantity of "significant" output data.

It may at times be desirable to study the table of ranked FMC's before proceeding with the control continuity and workload testing. So that unnecessary computation may be avoided, the program should be defined as a two stage process. Part I of the program shall include all initial reliability computations, the ranking of FMC's in descending order of probability and outputting of prescribed data. Part II shall constitute the control continuity and workload testing of the table of ranked FMC's plus the processing and printing out of prescribed data. The program user shall have the option of executing Part I only or both Parts I and II.

(4) Adequate Processing Capabilities - Provide a program with a capability and flexibilities sufficient to meet the needs not only of the present PCI application to the X-22A, but also those of future program users.

The maximum numbers of key parameters which the program is presently able to process are shown in the following table. The program, as written for an IBM 7090 computer with a 32 K memory, could readily be expanded to process the numbers of parameters indicated in the second column of the following table.

	Present Program		Expanded Program
Component Failure Modes/ Rates	40	per mission segment	50
Mission Segments	5		10
Failure Mode Combinations	800	total per mission	2000
Maximum Number of Failure Modes per FMC	4		4

---

[13] In the case of the X-22A application, all individual FMC's with a probability less than 2×10^{-6} were eliminated and the cumulative triviality criteria was set at 0.08%. These conditions resulted in between 200 and 300 FMC's (accounting for 99.92% of the total probabilities) being included in the analysis. If all possible combinations had been processed they would have totalled $\approx 6 \times 10^{14}$ with storage requirements exceeding the combined capacity of all the digital computers ever built.

	Present Program		Expanded Program
Pilot Workload Increments (with failure mode ident.)	40	per mission segment	50
Control Continuity Groups	50	total per mission	200
Work Load Dummy Arrays	50	total per mission	200
Work Load Interference Groups	50	total per mission	50
Catastrophic Single Failure Modes/Rates	40	per mission segment	50

2. GENERAL DESCRIPTION OF PROGRAM OPERATIONS

Figure 33 illustrates the data flow through the various steps of the AMSP digital program.

Step 1 Program input includes:

1. Duration of mission segments
2. Criteria for limiting the extent of the table of ranked FMC's.
3. Component failure modes and respective failure rates for each mission segment.
4. Basic pilot workloads for each mission segment
5. Pilot workload increments
6. Pilot workload dummy arrays[14]
7. Pilot workload interference groups[15]
8. Control continuity failure groups[16]
9. Catastrophic single failure modes and their respective failure rates.

NOTE: Only items 1 through 3 need be input for execution of Part I of the Program.

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[14] Illustrated by the failure effect relationships within the primary feel/trim system (0800) as shown at the upper left of Figure 21.

[15] Illustrated by the failure effect relationships between 0800 and 0900 (bottom of Figure 21) to generate a 2400 workload increment.

[16] Illustrated by Figure 22.

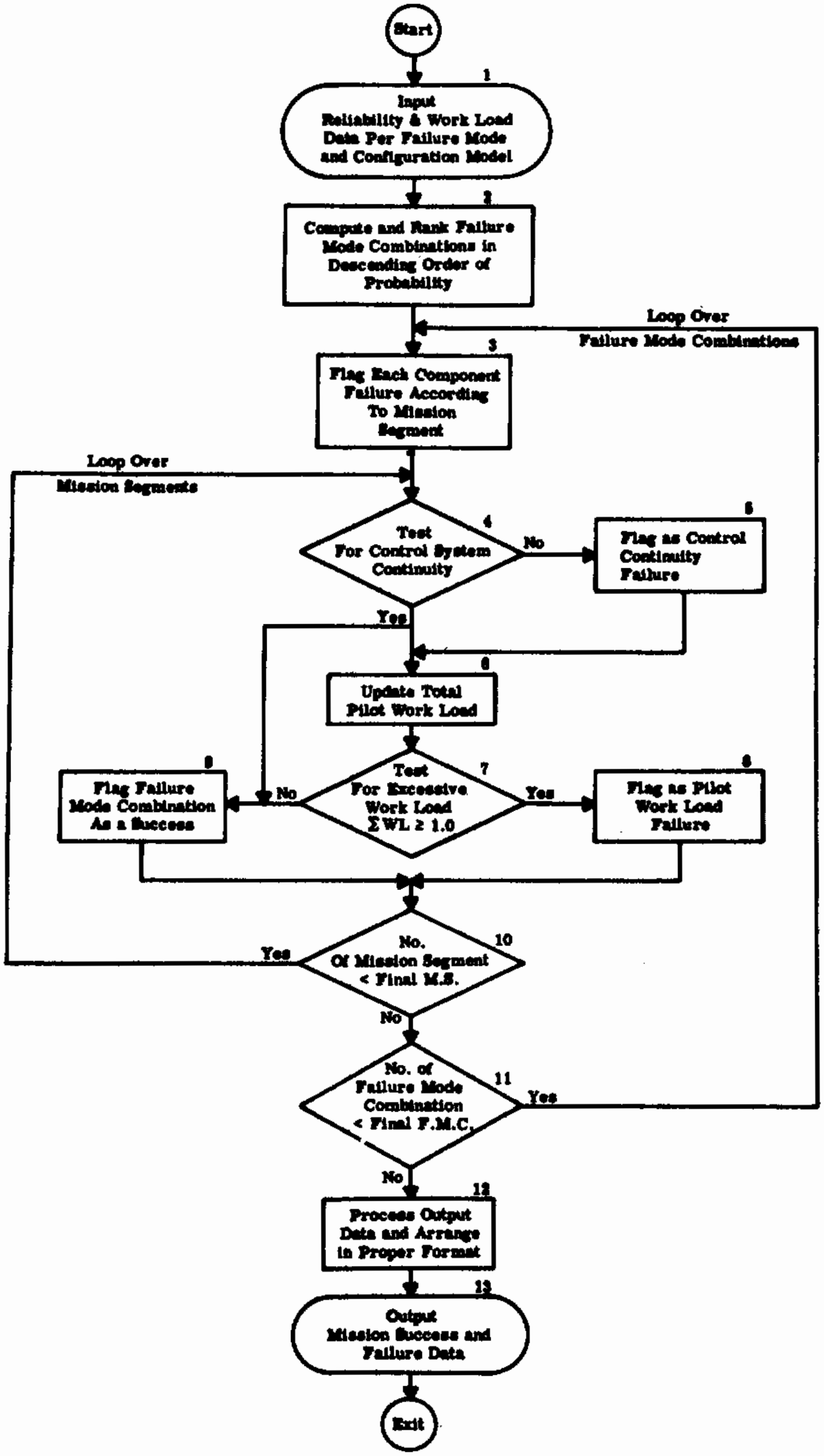

Figure 33. Flow Chart for Digital Analysis of Mission Success Probability

Step 2 The reliabilities and probabilities of failure are computed for each failure mode in each mission segment. The table of failure mode combinations ranked in descending order of probability is computed to the extent permitted by the triviality criteria.

The completion of Step 2 concludes Part I of the program.

Step 3 The particular failure mode combination being analyzed is tested to determine the lowest numbered mission segment in which a failure(s) is present. The inclusion of this step eliminates the unnecessary looping over of those mission segments prior to the occurrence of the first failure.

Step 4 The failure mode combination is tested to determine the presence or absence of a control continuity failure condition.

Step 5 If a control continuity failure condition exists, it is flagged as such for future use in output processing.

Step 6 The total pilot workload is computed for the FMC and mission segment being analyzed. The total workload is the sum of the basic workload and the additional increments resulting from individual failures and interference effects between certain concurrent failures.

Step 7 The total workload is tested to determine if a condition of excessive workload exists, $\sum WL \geq 1.00$.

Step 8 If the total workload exceeds unity, the FMC is flagged as a workload catastrophic failure.

Step 9 If the FMC has not been flagged as a control continuity or workload failure, flagged as a noncatastrophic, successful condition.

Step 10 Test the mission segment being analyzed against the final mission segment. If the present mission segment is less than final mission segment, loop over to the next mission segment. If mission segment is equal to final mission segment, proceed to Step 11.

Step 11 Test the rank order number of the FMC being analyzed against the total number of FMC's. If the present FMC identification is less than that of the final FMC, loop over to the analysis of the next lower ordered (probability) FMC in the ranked table. If FMC identification is equal to final FMC proceed to Step 12.

Step 12 Process the successful (noncatastrophic) and unsuccessful (catastrophic) failure mode analysis results and arrange according to output format.

Step 13 Output the tables of successful and unsuccessful FMC's in ranked order along with a summary of total mission success and failure probabilities.

3. DETAILED DESCRIPTION OF PROGRAM (Bell No. 3246)

a. Input Format

The entries on the IBM 7090 FORTRAN program forms which illustrate the program input format, are defined in the following discussion. The input format is shown on pages 120 to 124.

Program No. 3246 consists of two parts:

(1) Part 1 ranks failure mode combinations in descending order of probability of occurrence.

(2) Part 2 analyzes the ranked failure mode combinations generated in Part 1.

The data described in 1, 2, and 3 is required by Part 1. The rest of the data is required by Part 2.

(1) KP = 01, execute intermediate print-out

KP = 02, omit intermediate print-out

KDE = 01, execute Part 1 of Program No. 3246

KDE = 02, execute Parts 1 and 2 of Program No. 3246

(2) NS = number of sections in the configuration

NMS = number of mission segments

Q1 = limit imposed on individual intersections. If an intersection is less than Q1, it is considered a trivial intersection and it is omitted from the truth table.

Q2 = limit imposed on the summation of the current nontrivial intersections.

(3) TH(J) = number of hours in the mission segment J

TM(J) = number of minutes in the mission segment J

TS(J) = number of seconds in the mission segment J

IDS(I, J) = section identification number of section I during mission segment J

FR(I, J) = failure rate (per hour) associated with section I during mission segment J

for $1 \leq I \leq NS$ and $1 \leq J \leq NMS$

(4) BPWL(J) = basic pilot workload associated with mission segment J

$1 \le J \le NMS$

(5) NPWL = number of rows in the pilot workload array

IPWL(I) = section identification number of row I of the pilot workload array

$1 \le I \le NPWL$

PWL(I, J) = pilot workload associated with the section which appears in row I and mission segment J

$1 \le J \le NMS$

(6) NDMY = number of rows in the dummy array

IDMY(I) = section identification number of row I of the dummy section array

IDMY1(I) = section identification number of section 1 associated with IDMY(I) of the dummy section array

IDMY2(I) = section identification number of section 2 associated with IDMY(I) of the dummy section array

$1 \le I \le NDMY$

(7) NXG2 = number of groups of 2 elements which provide an extra pilot workload

IXG21(I) = identification number of element 1 of group I of 2 elements which provide an extra pilot workload

IXG22(I) = identification number of element 2 of group I of 2 elements which provide an extra pilot workload

IXPWL(I) = identification number of the section in the pilot workload array which provides the extra pilot workload for the group IXG21(I) and IXG22(I)

$1 \le I \le NXG2$

(8) NG2 = number of 2-element groups which cause control continuity failure

IG21(I) = identification number of element 1 of group I of the 2-element groups which cause control continuity failure

IG22(I) = identification number of element 2 of group I of the 2-element groups which cause control continuity failure

$1 \le I \le NG2$

(9) NAFM = number of rows per mission segment in the catastrophic single failure mode array

IAFM(I, J) = section identification number of section I during mission segment J

ALAMDA(I,J) = catastrophic single failure rate associated with section I during mission segment J

$1 \leq I \leq$ NAFM, $1 \leq J \leq$ NMS

(10) The following data NEVER have a decimal point:

KP, KDE, NS, NMS, IDS(I, J), NPWL, IPWL(I), NDMY, IDMY(I), IDMY1(I), IDMY2(I), NXG2, IXG21(I), IXG22(I), IXPWL(I), NG2, IG21(I), IG22(I), NAFM, IAFM(I, J).

(11) The following data ALWAYS have a decimal point:

(They are right adjusted in the field if the E notation is used, otherwise they are left adjusted.) Q1, Q2, TH(J), TM(J), TS(J), FR(I, J), BPWL(J), PWL(I, J), ALAMDA(I, J).

b. Program Operations

This section describes in detail the computer routine - defining the step by step execution of program operations.

(1) For every section i of the system during mission segment j, calculate $p_{i,j}$ and $q_{i,j}$

$p_{i,j}$ = prob (section i is a success during mission segment j)

$q_{i,j}$ = prob (section i is a failure during mission segment j)

Define $p_{i,j}$ and $q_{i,j}$ as follows:

$$p_{i,j} = e^{-(\lambda_{i,j})t_j}$$

$$q_{i,j} = 1.0 - p_{i,j}$$

where

$\lambda_{i,j}$ = given failure rate of section i during mission segment j

t_j = given duration of mission segment j

$1 \leq i \leq$ NS

$1 \leq j \leq$ NMS

NS = number of sections in the system

NMS = number of mission segments in the mission

INPUT FØRMAT
PRØG. #3246

7090 FORTRAN PROGRAM FORM

Form 0810 CUSTOMER ______ Page 1 of 5

PROGRAM | PROBLEM | ANALYST | PERMANENT NO. | DEPT. | W.O. | C.C. | DATE | 3246

STATEMENT NUMBER	FORTRAN STATEMENT				NO.
	CASE LABEL IN	COLUMNS	7-72	INCLUSIVE	
	KP in col. 7-8	KDE in col. 22-23			
	NS in col. 7-8	NHS in col. 22-23	Q1	Q2	
	TH(1)	TM(1)	TS(1)		
	IDS(1,1) col 7-10	FR(1,1)			
	IDS(2,1) col 7-10	FR(2,1)			
	.	.			
	.	.			
	IDS(I,1)	FR(I,1)			
	.	.			
	.	.			
	IDS(NS,1)	FR(NS,1)			
	.	.			
	.	.			
	.	.			
	.	.			
	TH(J)	TM(J)	TS(J)		
	IDS(1,J) col 7-10	FR(1,J)			
	IDS(2,J) col 7-10	FR(2,J)			
	.	.			
	.	.			
	IDS(I,J)	FR(I,J)			
	.	.			
	.	.			
	IDS(NS,J)	FR(NS,J)			

INPUT FØRMAT
PRØG. #3246

7090 FORTRAN PROGRAM FORM

Form 0810 CUSTOMER ______ Page 2 of 5

PROGRAM | PROBLEM | ANALYST | PERMANENT NO. | DEPT. | W.O. | C.C. | DATE | 3246

STATEMENT NUMBER | FORTRAN STATEMENT | NO.

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      .                                      .
      .                                      .
      .                                      .
      .                                      .
      TH(NMS)                                TM(NMS)                      TS(NMS)
      IDS(1,NMS)                             FR(1,NMS)
      IDS(2,NMS)                             FR(2,NMS)
      .                                      .
      .                                      .
      IDS(I,NMS)                             FR(I,NMS)
      .                                      .
      .                                      .
      IDS(NS,NMS)                            FR(NS,NMS)
      BPWL(1)                                BPWL(2)                      .              .
      .                                      .                            BPWL(J)        .
      .                                      BPWL(NMS)
      NPWL in col. 7-10
      IPWL(1) in col 7-10
      PWL(1,1)                               PWL(1,2)                     PWL(1,3)       .
      .                                      PWL(1,NMS)
      .                                      .                            .              .
      .                                      .                            .              .
      IPWL(I) in col 7-10
      PWL(I,1)                               PWL(I,2)                     PWL(I,3)       .
      .                                      PWL(I,NMS)
```

INPUT FORMAT
PROG. #3246

7090 FORTRAN PROGRAM FORM

Form 0810 CUSTOMER ______ Page 3 of 5

PROGRAM | PROBLEM | ANALYST | PERMANENT NO. | DEPT. | W.O. | C.C. | DATE | 3246

STATEMENT NUMBER | FORTRAN STATEMENT | NO.

```
       .                    .                    .                    .
       .                    .                    .                    .
IPWL(NPWL,1)
   PWL(NPWL,1)           PWL(NPWL,2)           PWL(NPWL,3)            .
       .                 PWL(NPWL,NMS)
NDMY in col. 7-10
IDMY(1) in col. 7-10 IDMY1(1) in col.22-25 IDMY2(1) in col.37-40
IDMY(2) in col. 7-10 IDMY1(1) in col.22-25 IDMY2(1) in col.37-40
       .                    .                    .
       .                    .                    .
       .                    .                    .
IDMY(I)              IDMY1(I)              IDMY2(I)
       .                    .                    .
       .                    .                    .
       .                    .                    .
IDMY(NDMY)           IDMY1(NDMY)           IDMY2(NDMY)
NXG2 in col. 7-10
IXG21(1) in col.7-10 IXG22(1) in col.22-25 IXPWL(1) in col.37-40
IXG21(2) in col.7-10 IXG22(2) in col.22-25 IXPWL(2) in col.37-40
       .                    .                    .
       .                    .                    .
IXG21(I)             IXG22(I)              IXPWL(I)
       .                    .                    .
       .                    .                    .
IXG21(NXG2)          IXG22(NXG2)           IXPWL(NXG2)
```

INPUT PRØG. FØRMAT #3246

7090 FORTRAN PROGRAM FORM

Form 0810 CUSTOMER ______ Page 4 of 5

PROGRAM | PROBLEM | ANALYST | PERMANENT NO. | DEPT. | W.O. | C.C. | DATE | 3246

STATEMENT NUMBER | FORTRAN STATEMENT | NO.

```
      NG2 in col. 7-10
      IG21(1) in col.7-10 IG22(1) in col. 22-25
      IG21(2) in col.7-10 IG22(2) in col. 22-25
      .              .
      .              .
      .              .
      IG21(I)        IG22(I)
      .              .
      .              .
      .              .
      IG21(NG2)      IG22(NG2)
      NAFM in col. 7-10
      IAFM(1,1) in col.7-10  ALAMDA(1,1)
      IAFM(2,1) in col.7-10  ALAMDA(2,1)
      .
      .
      IAFM(I,1) in col.7-10  ALAMDA(I,1)
      .
      .
      IAFM(NG2,1)     ALAMDA(NG2,1)
      .
      .
      .
      .
      .
```

INPUT FØRMAT
PRØG. #3246

7090 FORTRAN PROGRAM FORM

Form 0810 CUSTOMER ______ Page 5 of 5

PROGRAM | PROBLEM | ANALYST | PERMANENT NO. | DEPT. | W.O. | C.C. | DATE | 3246

STATEMENT NUMBER | FORTRAN STATEMENT | NO.

```
IAFM(1,J) in col.7-10   ALAMDA (1,J)
IAFM(2,J)               ALAMDA (2,J)
  .                       .
  .                       .
IAFM(I,J)               ALAMDA (I,J)
  .                       .
  .                       .
IAFM(NG2,J)             ALAMDA (NG2,J)
  .
  .
  .
  .
  .
IAFM(NAFM,1)            ALAMDA (NAFM,1)
IAFM(NAFM,2)            ALAMDA (NAFM,2)
  .                       .
  .                       .
IAFM(NAFM,J)            ALAMDA (NAFM,J)
  .                       .
  .                       .
IAFM(NAFM,NMS)          ALAMDA (NAFM,NMS)
```

(2) Arrange the $q_{i,j}$ in rank order

(3) Test every $q_{i,j}$ for triviality

If $q_{i,j} < Q1$, then $q_{i,j}$ is trivial

Q1 is a given constant

(4) Delete the trivial $q_{i,j}$ from the list

(5) Let q_k = nontrivial $q_{i,j}$

p_k = $p_{i,j}$ associated with q_k

IDS_k = section identification number associated with q_k

IDM_k = mission segment identification number associated with q_k

The table of q_k, p_k, IDS_k, and IDM_k shall be referred to as the initial probability table.

The range of k is, $1 \leq k \leq NQ$ where $NQ \leq (NS)\ (NMS)$

q_1 = maximum probability of failure

.

.

.

q_{NQ} = minimum nontrivial probability of failure

(6) Calculate Po using the initial probability table

Po = prob (0 failures occur during the mission)

$$Po = \prod_{k=1}^{NQ} p_k$$

(7) Calculate all possible nontrivial P1 using the initial probability table

P1 = prob (1 section fails during the mission)

$$P1_k = \frac{q_k}{p_k} \prod_{j=z}^{NQ} p_j$$

It is assumed that any section which fails during mission segment i remains in the failure state for the rest of the mission. Therefore,

$p_j = 1.0$ when 1. $IDS_k = IDS_j$

2. $IDM_k \leq IDM_j$

If $P1_k < Q1$, then $P1_k$ is trivial

Let NQ1 = number of nontrivial P1

(8) Arrange the nontrivial P1 in rank order (refer to page C-31 for detail of ranking procedure).

(9) Calculate SUM

$$SUM = Po + \sum_{k=1}^{NK} P1_k, \quad NK \leq NQ1$$

Subject to the constraint $SUM \leq Q2$, where Q2 is a given constant

(10) Refer to the list of $P1_k$ for $1 \leq k \leq NK$ as the current ranked table.

The $P1_k$ which cause SUM to exceed Q2 are deleted from the ranked table.

Refer to the probability combinations in the current ranked table as PC_k.

(11) Calculate all possible nontrivial P2 using the initial probability table.

P2 = prob (2 sections fail during the mission)

$$P2 = \frac{q_\ell q_m}{p_\ell p_m} \prod_{n=1}^{NQ} \qquad m \neq \ell$$

We assume that any section which fails during mission segment i remains in the failure state for the rest of the mission. Therefore,

$p_n = 1.0$ when

either $IDS_m = IDS_n$ and $IDM_m < IDM_n$

or $IDS_\ell = IDS_n$ and $IDM_\ell < IDM_n$

Therefore, $IDS_\ell \neq IDS_m$

If $P2_\ell < Q1$, then $P2_\ell$ is trivial

Let NQ2 = number of nontrivial P2

If NQ2 = 0, steps 12-22 are not executed

Proceed to step 23

(12) Append the list of $P2_\ell$ probabilities where $1 \le \ell \le NQ2$ to the current table

(13) Arrange the current table in rank order..

(14) Calculate SUM

$$SUM = Po + \sum_{k=1}^{NK} PC_k, \quad NK \le \text{old } NK + NQ2$$

subject to the constraint SUM $\le$ Q2, where Q2 is a given constant.

The PC_k which causes SUM to exceed Q2 are deleted from the ranked table.

At this point the possible probability combinations, PC_k, are of types P1 and P2.

(15) Calculate all possible nontrivial P3 using the initial probability table.

P3 = prob (3 sections fail during the mission)

$$P3 = \frac{q_\ell q_m q_n}{p_\ell p_m p_n} \prod_{i=1}^{NQ} \qquad \begin{matrix} m \ne \ell \\ n \ne \ell \\ n \ne m \end{matrix}$$

It is assumed that any section which fails during mission segment i remains in the failure state for the rest of the mission. Therefore,

$p_i = 1.0$ when

either $IDS_n = IDS_i$ and $IDM_n < IDM_i$

or $IDS_m = IDS_i$ and $IDM_m < IDM_i$

or $IDS_\ell = IDS_i$ and $IDM_\ell < IDM_i$

Therefore, $IDS_\ell \ne IDS_m$

$IDS_\ell \ne IDS_n$

$IDS_m \ne IDS_n$

If $p3_\ell < Q1$, then $P3_\ell$ is trivial

Let NQ3 = number of nontrivial P3

If NQ3 = 0, steps 16-22 are not executed.

Proceed to step 23

(16) Append the list of $P3_\ell$ probabilities, where $1 \leq \ell \leq NQ3$, to the current table.

(17) Arrange the current table in rank order.

(18) Calculate SUM

$$SUM = Po + \sum_{k=1}^{NK} PC_k, \; NK \leq \text{old } NK + NQ3$$

subject to the constraint SUM $\leq$ Q2, where Q2 is a given constant.

The PC_k which cause SUM to exceed Q2 are deleted from the ranked table.

At this point the possible probability combinations, PC_k, are of types P1, P2, P3

(19) Calculate all possible nontrivial P4 using the initial probability table.

P4 = prob (4 sections fail during the mission)

$$P4 = \frac{q_\ell q_m q_n q_i}{p_\ell p_m p_n p_i} \prod_{j=1}^{NQ} p_j \qquad \begin{matrix} m \neq & n \neq & i \neq \\ n \neq m & i \neq m & i \neq n \end{matrix}$$

We assume that any section which fails during mission segment i remains in the failure state for the rest of the mission. Therefore,

$p_j = 1.0$ when

either $IDS_i = IDS_j$ and $IDM_i < IDM_j$

$IDS_n = IDS_j$ and $IDM_n < IDM_j$

$IDS_m = IDS_j$ and $IDM_m < IDM_j$

$IDS_\ell = IDS_j$ and $IDM_\ell < IDM_j$

Therefore, $IDS_\ell \neq IDS_m$, $IDS_\ell \neq IDS_n$, $IDS_\ell \neq IDS_i$,

$IDS_m \neq IDS_n$, $IDS_m \neq IDS_i$, $IDS_n \neq IDS_i$

If P4 < Q1, then P4 is trivial

Let NQ4 = number of nontrivial P4

If NQ4 = 0, steps 20-22 are not executed. Proceed to step 23

(20) Append the list of P4 probabilities, where $1 \leq \ell \leq NQ4$, to the current table.

(21) Arrange the current table in rank order.

(22) Calculate SUM

$$SUM = Po + \sum_{k=1}^{NK} PC_k, \quad NK \leq \text{old } NK + NQ4$$

subject to the constraint $SUM \leq Q2$, where Q2 is a given constant.

The PC_k which cause SUM to exceed Q2 are deleted from the ranked table.

At this point the possible probability combinations, PC_k, are of types P1, P2, P3, P4, and the final ranked table of FMC's has been established.

(23) Determine the element in the pilot workload array which is associated with every dummy variable.

If there are no dummy variables associated with the system being studied, then step 23 is not executed.

(24) Determine the element in the pilot workload array which is associated with every group of 2 elements which give an extra pilot workload when both elements fail during a mission.

If there are no groups of 2 elements which give an extra pilot workload, then step 24 is not executed.

(25) Prepare a list of groups of 2 elements which cause continuity failure.

If there are no groups of 2 elements which cause continuity failure in the system being studied, then step 25 is not executed.

(26) For every section i of the system which is considered a catastrophic single failure mode during mission segment j, calculate $p^*_{i,j}$ and $q^*_{i,j}$

$p^*_{i,j}$ = prob (section i is a success during mission segment j)

$q^*_{i,j}$ = prob (section i is a failure during mission segment j)

Define $p^*_{i,j}$ and $q^*_{i,j}$ as follows:

$$p^*_{i,j} = e^{-(\lambda^*_{i,j})(t_j)}$$

$$q^*_{i,j} = 1.0 - p^*_{i,j}$$

Where

$\lambda^*_{i,j}$ = given failure rate of section i during mission segment j

t_j = given duration of mission segment j

$1 \leq i \leq NAFM$

$1 \leq j \leq NMS$

NAFM = number of catastrophic single failure modes

NMS = number of mission segments

(27) Calculate RF, the reliability factor for the system

$$RF = \prod_{j=1}^{NMS} \left(\prod_{i=1}^{NAFM} p^*_{i,j} \right)$$

(28) Arrange the $q^*_{i,j}$ in rank order.

If NAFM = 0, steps 26-28 are not executed and RF = 1.0

Analyze each entry of the final ranked table to determine whether the probability combination would result in a successful or a catastrophic mission. There are two conditions which result in a catastrophe. They are:

(1) Control continuity failure

(2) Excessive pilot workload

(29) Determine whether entry k of the final ranked table would result in a control continuity failure.

Procedure -

NG2 = number of groups of 2 elements which cause control continuity failure. Each group is unique

$(IG21)_i$ = identification number of element 1 of a group of 2 elements which cause control continuity failure

$(IG22)_i$ = identification number of element 2 of a group of 2 elements which cause control continuity failure

$1 \leq i \leq NG2$

NG2, the array IG21 and the array IG22 are input to the program.

Let NEL = number of elements in entry k

If NEL = 1, entry k cannot result in a control continuity failure and the rest of the section is not executed

If NEL > 1, proceed with the rest of the testing

Compare entry k with every combination $IG21_i$ and $IG22_i$ for $1 \leq i \leq NG2$

If entry k includes any group of 2 elements $IG21_i$ and $IG22_i$, then entry k will cause a control continuity failure.

If entry k is a continuity failure, record this fact.

(30) If entry k causes a control continuity failure, enter the data associated with entry k in the control continuity failure table.

(31) Determine the total pilot workload for each mission segment j for entry k of the final ranked table.

Total pilot workload = A + B + C + D for mission segment j

Where,

A = basic pilot workload for mission segment j

B = pilot workload required to compensate for the failure of the sections involved in the probability combination associated with entry k

C = pilot workload created by the dummy variables associated with entry k

D = extra pilot workload created by the failure of certain combinations of sections which may appear in entry k

(32) Determine whether entry k of the final ranked table causes a pilot workload failure.

Let $TPWL_j$ = total pilot workload for entry k

If any $TPWL_j \leq 0.995$, then entry k of the final ranked table causes a pilot workload failure.

If entry k causes a pilot workload failure, enter the data associated with entry k in the pilot workload failure table and proceed to analyze the next entry in the final ranked table.

If entry k results in a successful mission, then enter the data associated with entry k in the noncatastrophic outcomes table and proceed to analyze the next entry in the final ranked table.

(33) After the entire final ranked table has been analyzed, output

1. Table of noncatastrophic outcomes
2. Table of pilot workload failures
3. Table of control continuity failures

(34) Summarize the results of the analysis of the final ranked table.

1. Calculate the total probability of success
2. Calculate the probability of failure due to control continuity failure
3. Calculate the probability of failure due to excessive pilot workload
4. Calculate the total probability of failure
5. Calculate the probability of successful mission accomplishment

$$\begin{matrix}\text{Probability of} \\ \text{Successful Mission} \\ \text{Accomplishment}\end{matrix} = \begin{pmatrix}\text{total success} \\ \text{probability}\end{pmatrix} \times \begin{pmatrix}\text{Reliability of} \\ \text{Catastrophic Single} \\ \text{Failure Modes}\end{pmatrix}$$

Procedure for the Arrangement of P's in Rank Order -
Applies to Steps 8, 13, 17, and 21

The arrangement of an array of q_i in rank order when every q_i has an associated $y1_i, y2_i, \ldots\ldots\ldots yk_i, \ldots\ldots\ldots\ldots y_{n_i}$ (n $\leq$ 4 in the present program).

Let $1 \leq k \leq NQ$

1. N2 = NQ - 1 NQ = number of nontrivial P's in the current table
2. Set I = 0
3. I = I + 1
4. ID = I ID = numberical ident. of BIGQ, where
BIGQ = Q(I) BIGQ = currently largest identified P in the partially ranked table
N1 = I + 1
5. J = N1 - 1
6. J = J + 1
7. Test [BIGQ - Q(J)]
If BIGQ - Q(J) < 0, go to 8
If BIGQ - Q(J) ≥ 0, go to 9
8. ID = J
BIGQ = Q(J)

9. Test J = NQ

   If J $\neq$ NQ, go to 6
   If J = NQ, go to 10

10. Test ID = I

   If ID $\neq$ I, go to 11
   If ID = I, go to 12

11. Interchange

   Q(I) and Q(ID)

   y_1(I) and y_1(ID)

   .

   .

   y_k(I) and y_k(ID)

   .

   .

   y_n(I) and y_n(ID)

12. Test I = N2

   If I $\neq$ N2, go to 3
   If I = N2, we have arranged the array of q_i in rank order

4. PROGRAM PRINT OUT

Tables X and XI illustrate the print format for the data input to the No. 3246 program. It is desirable to include this data in the program print out to provide a check against possible input errors.

The top two lines of Table X contain general information describing the particular computer run.

The main body of Table X consists of a listing of the sections entered into the program (individual potentially catastrophic failure modes) and their respective failure rates (per hour) for the specified mission segment. The failure mode reliabilities (P) and probabilities of failure (Q) computed in the program on the basis of the mission segment duration are also listed.

Similar listings are printed for each mission segment.

The print format for catastrophic single failure modes, failure rates, reliabilities and probabilities of failure is similar to Table X.

Table XI indicates failed element workload increments per mission segment. The basic (no failure) workloads for each mission segment are listed across the bottom of the table.

Other input data such as pilot workload dummy arrays, workload interference combinations, and control continuity failure groups are also printed out.

Tables XII through XV illustrate the print format for the No. 3246 program output.

Table XII shows the nontrivial failure mode combinations arranged in descending order of probability. IDS's identify the individual failure modes and IDM's identify the mission segment in which the related failure is first present. PC is the probability of occurrence of the specified failure mode combination. Although this example shows only the 38 largest failure mode combinations in terms of probability of occurrence, the table includes all those FMC's determined by the program to be nontrivial.

Table XIII lists those failure mode combinations - in descending order of probability - which do not result in loss of the vehicle due either to lack of control continuity or excessive pilot workload. Columns 2 through 5 identify the individual failure modes, and their respective mission segments of occurrence. Column 6 lists the probabilities of occurrence of the failure mode combinations. The last five columns list the total pilot workloads in each mission segment.[17]

Table XIV lists those failure mode combinations resulting in catastrophic outcomes as a consequence of excessive pilot workload. This table is identical in format to Table XIII.

The failure mode combinations resulting in catastrophic outcomes as a consequence of loss of control continuity are listed in a format similar to Tables XIII and XIV with one change. A column has been added for the identification of the mission segment in which the control continuity failure occurs.

Table XV summarizes the results of the analysis of mission success probability. The summations of the successful (noncatastrophic) probabilities, failure (catastrophic) probabilities, and trivial probabilities are given. The probability of successful mission accomplishment - defined as the product of the total success probability and the reliability of the catastrophic single failure modes - is listed as the final entry in the program output.

---

[17] The pilot workloads for the noncatastrophic outcomes have been included in the print format as providing potentially valuable supporting information in the evaluation of pilot-controller systems performance.

TABLE X

AMSP DIGITAL PROGRAM INPUT DATA; FAILURE MODES (SECTIONS) AND RATES

X-22A   RUN NC.3A ,   PRCGRAM 3246   5-14-65

22 SECTICNS .   5 MISSICN SEGMENTS   Q1 = C.200CCCCCE-C5   Q2 = C.99919999E CC

THE CURATICN CF MISSICN SEGMENT 1 IS C. HCURS, C. MINUTES, 20. SECCNLS CR C.CC5556 HCURS

SECTICN	FAILURE RATE	P	C
100	C.11250CC0E-C1	C.99993750E CC	C.62495470E-C4
200	C.12750000E-01	C.99992917E OC	C.70825219E-C4
400	C.15749999E-C1	C.99991252E OC	C.87484717E-C4
500	C.1230000CE-C1	C.99993168E OC	C.68321824E-C4
600	C.12300000E-C1	C.99993168E OC	C.68321824E-C4
801	0.16275000E-00	C.99909624E CC	C.90375543E-C3
901	C.92249999E-C1	C.99948764E CC	C.51236153E-C3
1001	0.92249999E-C1	C.99948764E OC	C.51236153E-C3
1100	0.25999999E-02	C.99998CC0E CC	C.19997358E-C4
1200	C.35999999E-02	C.99998000E OC	C.19997358E-C4
1300	C.16500000E-C2	C.99999C84E OC	C.91642141E-C5
1400	C.1650000CE-02	C.99999084E OC	C.91642141E-C5
1500	C.16500000E-02	C.99999084E OC	C.91642141E-C5
1600	C.1650000CE-02	C.99999C84E OC	C.91642141E-C5
1700	C.1350000CE-C2	C.99999250E OC	C.74952841E-C5
1800	0.13500000E-02	C.99999250E OC	C.74952841E-C5
1900	0.13500000E-02	C.99999250E OC	C.74952841E-C5
2000	C.135000C0E-02	C.99999250E OC	C.74952841E-C5
1750	C.12000000E-C2	C.99999334E OC	C.66608191E-C5
1850	0.1200000CE-02	C.99999334E OC	C.66608191E-C5
1950	C.12000000E-02	C.99999334E OC	C.66608191E-C5
2050	C.1200000CE-02	C.99999334E OC	C.66608191E-C5

TABLE XI

AMSP DIGITAL PROGRAM INPUT DATA; FAILED ELEMENT WORK LOAD INCREMENTS PER MISSION SEGMENT

FAILED ELEMENT WORK LOAD INCREMENTS PER MISSION SEGMENT

SECTION	M.S. 1	M.S. 2	M.S. 3	M.S. 4	M.S. 5
800	0.	C.79999999E-01	C.11COCCCOE-CO	C.79999999E-C1	C.
900	C.12000000E-CO	C.79999999E-01	C.	C.79999999E-C1	C.12CCCCCCE-CC
1000	0.12000000E-CO	C.79999999E-01	C.	C.79999999E-C1	C.12CCCCCCE-CC
1100	C.5000C000E-C1	C.11CCCOCOE-OC	C.	C.11C000COE-CC	C.5CCOCCCOE-C1
1200	0.39999999E-C1	C.49999999E-01	C.	C.49999999E-01	C.39999999E-C1
1300	C.16000000E-CO	C.19CCCCCOE-CC	C.	C.15C00CCCE-CC	C.16CCCCCCE-CO
1400	0.16000000E-CO	C.19CCCOCOE-OC	C.	C.15C0C0CCE-CC	C.16CCCCCCE-CC
1500	0.16000000E-CO	C.19CCCCCOE-CC	C.	C.19C0C0COE-00	0.16CCCCCOE-CC
1600	C.16000000E-CO	C.19COCCCOE-OC	C.	C.15C000CCE-CC	C.16CCCCCCE-CC
1700	C.13999999E-CC	C.15CCCOCOE-OC	C.2CCCCCCOE-CC	C.15C00CCCE-CC	C.13999999E-OC
1800	0.13999999E-CO	C.15CCCCCOE-OC	C.2CCOCCCOE-CC	C.15C000CCE-CC	C.13999999E-CC
1900	0.13999999E-CO	C.15CCC000E-OC	C.2CCOCOCOE-CO	C.15C000COE-CC	C.13999999E-CC
2000	0.13999999E-00	C.15COC000E-OC	C.2CCCCOCOE-CO	C.15C000CCE-OC	C.13999999E-CO
1750	C.16999999E-CO	C.20999999E-OC	C.55CCCCCOE CC	C.2C999999E-CC	0.16999999E-CO
1850	0.16999999E-CO	C.20999999E-CC	C.55COCCCOE CC	C.2C999999E-CC	C.16999999E-CC
1950	0.16999999E-00	C.20999999E-OC	C.55C0000OE CC	C.2C999999E-CC	C.16999999E-CC
2050	C.16999999E-00	C.20999999E-OC	0.55COCOCOE CC	C.2C999999E-CC	C.16999999E-CC
1799	-C.13999999E-CO	-C.15COCCCOE-OC	-C.2CCOCCCCE-CC	-C.15C000COE-CC	-C.13999999E-CO
1899	-0.13999999E-CO	-C.15COCCCOE-CC	-C.2CCOC000E-CC	-C.15C000CCE-CC	-C.13999999E-CC
1999	-0.13999999E-CO	-C.15COCCOOE-OC	-C.2CCCCCCOE-CC	-C.15C0C0CCE-CC	-C.13999999E-00
2C99	-C.13999999E-00	-C.15CCC0COE-OC	-C.2CCOCCCOE-CC	-C.15C000CCE-CC	-C.13999999E-CC
2400	C.	-C.59999999E-C1	C.	-C.59999999E-C1	C.
2500	0.	-C.59999999E-01	C.	-C.59999999E-C1	C.
2600	0.16000000E-00	C.09999999E-OC	C.	C.95C000COE OC	C.16CCCCCCE-CC
BASIC	C.59999999E CO	C.72CCC000E OC	C.58COCCCOE CC	C.72C00CCCE CC	0.59999999E CC

TABLE XII

AMSP DIGITAL PROGRAM OUTPUT; FINAL TABLE OF NONTRIVIAL INTERSECTIONS ARRANGED IN RANK ORDER

THE FINAL TABLE OF NON-TRIVIAL INTERSECTIONS ARRANGED IN RANK ORDER

K	IDS1	IDM1	IDS2	IDM2	IDS3	IDM3	IDS4	IDM4	PC
	PROB(0 FAILURES)								0.89964326E 00
1	801	3	0	0	0	0	0	0	0.35016659E-01
2	901	3	0	0	0	0	0	0	0.97656753E-02
3	1001	3	0	0	0	0	0	0	0.97656753E-02
4	801	4	0	0	0	0	0	0	0.44878266E-02
5	601	2	0	0	0	0	0	0	0.33337297E-02
6	400	3	0	0	0	0	0	0	0.33140627E-02
7	200	3	0	0	0	0	0	0	0.26815983E-02
8	500	3	0	0	0	0	0	0	0.25867756E-02
9	600	3	0	0	0	0	0	0	0.25867756E-02
10	901	4	0	0	0	0	0	0	0.25403471E-02
11	1001	4	0	0	0	0	0	0	0.25403471E-02
12	100	3	0	0	0	0	0	0	0.23655715E-02
13	901	2	0	0	0	0	0	0	0.18330222E-02
14	1001	2	0	0	0	0	0	0	0.18330222E-02
15	801	1	0	0	0	0	0	0	0.85305845E-03
16	1100	3	0	0	0	0	0	0	0.75610106E-03
17	1200	3	0	0	0	0	0	0	0.75610106E-03
18	801	5	0	0	0	0	0	0	0.56956745E-03
19	901	1	0	0	0	0	0	0	0.46859238E-03
20	1001	1	0	0	0	0	0	0	0.46859238E-03
21	400	4	0	0	0	0	0	0	0.43307906E-03
22	801	3	901	3	0	0	0	0	0.38010778E-03
23	801	3	1001	3	0	0	0	0	0.38010778E-03
24	200	4	0	0	0	0	0	0	0.35056622E-03
25	1500	3	0	0	0	0	0	0	0.34643636E-03
26	1600	3	0	0	0	0	0	0	0.34643636E-03
27	1400	3	0	0	0	0	0	0	0.34643636E-03
28	1300	3	0	0	0	0	0	0	0.34643636E-03
29	500	4	0	0	0	0	0	0	0.33819548E-03
30	600	4	0	0	0	0	0	0	0.33819548E-03
31	901	5	0	0	0	0	0	0	0.32279161E-03
32	1001	5	0	0	0	0	0	0	0.32279161E-03
33	400	2	0	0	0	0	0	0	0.30966013E-03
34	100	4	0	0	0	0	0	0	0.30930951E-03
35	1950	3	0	0	0	0	0	0	0.25193427E-03
36	2050	3	0	0	0	0	0	0	0.25193427E-03
37	1850	3	0	0	0	0	0	0	0.25193427E-03
38	1750	3	0	0	0	0	0	0	0.25193427E-03
39	200	2	0	0	0	0	0	0	0.25047047E-03
40	600	2	0	0	0	0	0	0	0.24159695E-03
41	500	2	0	0	0	0	0	0	0.24159695E-03

Contrails

TABLE XIII

AMSP DIGITAL PROGRAM OUTPUT; NONCATASTROPHIC OUTCOMES

PRCBABILITY PRECICTICN CF PILCT CCNTRCLLER MISSICN SLCCESS

NCN-CATASTRCPHIC CUTCCMES

K	FAILEC SECTICNS				PRCBABILITY CF OCCURRENCE	TCTAL PILCT WCRK LCACS / MISSICN SEGMENT 1	2	3	4	5
	NC FAILURES				C.859643E CC	C.6C	C.72	0.58	C.72	0.60
1	8C1 3	0 0	C C	C C	C.350167E-C1	C.6C	C.72	0.69	C.8C	0.60
2	5C1 3	0 0	C C	C C	0.976568E-C2	C.6C	C.72	0.58	C.8C	0.72
3	1CC1 3	0 0	C C	C C	C.976568E-C2	C.6C	C.72	0.58	C.8C	0.72
4	8C1 4	0 0	C C	C C	C.448783E-C2	C.6C	C.72	C.58	C.8C	0.60
5	8C1 2	0 0	C C	C C	C.333273E-C2	C.60	C.8C	0.69	C.8C	0.60
6	4C0 3	0 0	0 C	C 0	0.331406E-C2	C.6C	C.72	C.69	C.8C	0.60
7	2C0 3	0 0	C C	C C	C.268160E-C2	C.60	C.72	0.69	C.8E	0.72
8	5C0 3	0 0	C 0	C C	0.258678E-C2	C.6C	C.72	C.69	C.8E	0.72
9	6C0 3	0 0	C C	C C	C.258678E-C2	C.6C	C.72	C.58	C.8C	0.72
10	5C1 4	0 0	0 C	C C	C.254C35E-C2	C.6C	0.72	C.58	C.8C	0.72
11	1CC1 4	0 C	C C	C 0	C.254C35E-C2	C.6C	0.72	C.58	C.8C	0.72
12	1C0 3	0 C	C C	C C	C.236557E-C2	C.6C	0.72	C.58	0.80	0.72

TABLE XIV

AMSP DIGITAL PROGRAM OUTPUT; CATASTROPHIC OUTCOMES - PILOT WORKLOAD FAILURES

PROBABILITY PRECICTICN CF PILCT CCNTRCLLER MISSICN SUCCESS

CATASTRCPHIC CUTCCMES - PILCT WORK LOAC FAILURES   TABLE C.6

K	FAILEC SECTICNS				PRCBABILITY CF OCCURRENCE	TCTAL PILCT WCRK LCACS / MISSICN SEGMENT 1	2	3	4	5
1	1550	0	C	C	C.251S34E-C3	C.6C	0.72	1.53	C.S3	C.77
	3	0	C	C						
2	2C50	0	C	C	C.251S34E-C3	C.6C	C.72	1.53	C.S3	C.77
	3	0	C	C						
3	1850	0	C	C	C.251S34E-C3	C.6C	C.72	1.53	0.S3	C.77
	3	0	C	C						
4	1750	0	C	C	C.251S34E-C3	C.6C	C.72	1.53	0.S3	C.77
	3	0	C	C						
5	SC1	1001	C	C	0.1C6CC7E-C3	C.6C	0.72	0.58	1.83	1.CC
	3	3	0	C						
6	SC1	2C0	C	C	C.291C89E-04	C.6C	C.72	0.69	1.7S	1.CC
	3	3	C	C						
7	SC1	600	C	C	C.280756E-C4	C.6C	C.72	0.58	1.83	1.CC
	3	3	C	0						
8	1CC1	500	C	C	C.280756E-C4	C.6C	C.72	0.69	1.7S	1.CC
	3	3	0	C						
9	SC1	1001	0	0	0.275756E-C4	C.6C	0.72	0.58	1.83	1.CC
	3	4	C	C						
10	1CC1	901	C	C	0.275756E-C4	C.6C	0.72	0.58	1.83	1.CC
	3	4	0	0						
11	1CC1	100	0	C	C.256784E-04	C.60	0.72	0.58	1.83	1.CC
	3	3	C	C						
12	1750	0	0	0	0.234950E-C4	C.6C	C.93	1.53	C.S3	C.77
	2	0	C	C						
13	18FC	0	C	C	C.234S5CE-C4	C.6C	0.93	1.53	C.S3	C.77

TABLE XV

AMSP DIGITAL PROGRAM OUTPUT; SUMMARY

PROBABILITY PRECICTICN CF PILCT CCNTRCLLER MISSICN SUCCESS

SUMMARY

SUCCESS PROBABILITY	FAILURE PRCBABILITY TOTAL	FAILURE PROBABILITY CCNTRCL CCNTINUITY	FAILURE PRCBABILITY PILCT WCRK LCAC	TCTAL TRIVIAL PRCBABILITIES
C.95762257	0.0015783E	C.CCCCC7C5	C.0C157123	C.CCC79SC4

PROBABILITY OF SUCCESSFUL MISSION ACCOMPLISHMENT = SUCCESS PROBABILITY X RELIABILITY OF CATASTROPHIC SINGLE FAILURE MODES

0.99218825 = 0.99762257 X 0.99455273

APPENDIX IV

PILOT WORK LOAD MEASUREMENT TECHNIQUES

GENERAL

Work load may be defined as that portion of a subject's normal working capacity required in the performance of an assigned task. The measurement of pilot work loads under normal vehicle operating conditions as well as under emergency (systems failure) conditions constitutes an essential element in the PCI technique. The most generally accepted approach to the measurement of work loads involves an auxiliary task to be performed concurrently with the primary task under evaluation. The effort devoted to this auxiliary task then is a measure of the subject's work capacity in excess of that required by the primary task.

Requirements for an auxiliary work load measurement task are as follows:

(a) It should be self-pacing; i.e., it should be able to absorb as much (or as little) of the pilot's effort as he is able to devote to it.

(b) It should have a minimal disruptive influence on the performance of the primary task.

(c) It should be basically simple in nature, and should require very little learning.

(d) Assuming the simulation to be a part task analysis, consideration should be given to structuring of the auxiliary work load task to simulate the nature of actual flight secondary tasks.

(e) It should permit convenient and accurate measurement of the amount of pilot effort devoted to it.

The auxiliary work load task developed for the X-22A simulation studies required the pilot to set to zero a two-state (positive or negative) discrete value signal. A block diagram of this auxiliary task measurement unit is shown in Figure 19 and the simulator mechanization is shown in Figure 34. The error between the discrete command signal and the pilot response was displayed on a four-inch vertical meter installed directly to the right of the main instrument panel as shown in Figure 20. This display was located as close as practical to the primary flight instruments in order to minimize the shift in the pilot's normal scan pattern and hence exercise a minimum disruptive influence on primary task performance.

As both of the X-22A pilots hands are continually occupied in the primary flight control task throughout vertical flight and transition[18], a means of response to the auxiliary work load task had to be defined which would not conflict with the above commitment. A three-position rocker switch, mounted in the front of the control stick grip, was evaluated and found adequate for this function.

(18)Right hand on attitude control stick; left hand on throttles.

Operation of the auxiliary work load unit is as follows:

(1) A randomly selected plus or minus error is displayed on the meter.

(2) Upon recognition of the error state, the pilot depresses the rocker switch on the corresponding side (top or bottom). This instantaneously drives the error to zero (center of the meter) where it is held until the switch is released.

(3) Immediately upon switch release, a new random error is displayed which is held until the pilot again operates the response switch.

(4) Should the pilot actuate the switch in the wrong direction, he is immediately made aware of the situation as the amplitude of the error is doubled.

The unit as defined will cycle as rapidly as commanded by the pilot.

The pilot's performance is automatically recorded continuously throughout each run by the monitoring of four parameters. The outputs of comparators M4J and M4K (Figure 34) indicate each pilot response (switch operation) and each correct pilot response, respectively.

For each pilot response, the timed pulse generated on the arm of comparator M4J is integrated through amplifier 40. The output of amplifier 40 indicates the total number of pulses. For convenience in data reduction, the output of amplifier 40 is divided by time to generate pulse rate per minute on the output of amplifier 41 which is then recorded. The circuitry including comparator M4K and amplifiers 45 and 44 is setup to process the number of correct pilot response pulses in a similar manner.

The reference base for work load scoring is determined for each pilot at the beginning of each run period by having the subject operate the measurement unit as rapidly as possible for a one to two-minute period in the absence of the primary task (no load condition). After a fair amount of practice, the reference base stabilized between 80 to 90 pulses per minute for each pilot. The measured work load for a given simulated flight is determined by dividing the run response (pulse) rate per minute by the base reference (no load) response rate, and subtracting the dividend from one; i.e.,

$$\text{Measure Work Load} = 1 - \frac{\text{Pulses per minute for simulated flight}}{\text{Pulses per minute base reference}}$$

In scaning the work loads, the incorrect responses were taken into account by introducing into the numerator of the above equation the average of the total pulse rate and the correct pulse rate.

If the pulse rate varies significantly during a run, that condition should be rerun, and the integrators should be allowed to operate only during that portion the simulated flight where the output of comparator M4J has indicated the pulse rate to be lowest (highest work load). This technique will result in the recording of the peak work load with minimum bias.

The auxiliary task described above was the second scheme which was evaluated for use with the X-22A simulation program. The first scheme, which is briefly described below, did not satisfy the requirements for an acceptable work load measurement unit.

The first work load measurement scheme called upon the pilot to:

(1) Monitor the difference between a randomly changing function and the pilot response function displayed on the four-inch vertical meter mounted on the instrument panel.

(2) As the primary task permitted, the pilot attempted to keep the excursions of the random function nulled out through bi-directional operation of the stick grip rocker switch mechanized to provide a hi-gain rate response signal.

(3) Work load measure was based on the ratio of the integrated error signal to the integrated value of the random tracking function.

After a period of evaluation, it became apparent that this task as basically defined did not satisfy the requirements for an acceptable work load measurement unit.

One deficiency in the operation of this unit was that the loading imposed on the pilot by the random function was not constant, but was time varient over a considerable range. This resulted in some degree of interference with the primary flight task. Also, for the short duration of the simulation runs (90 to 180 seconds), it was not possible to adequately calibrate the unit as a result of time varient shifts in the base reference.

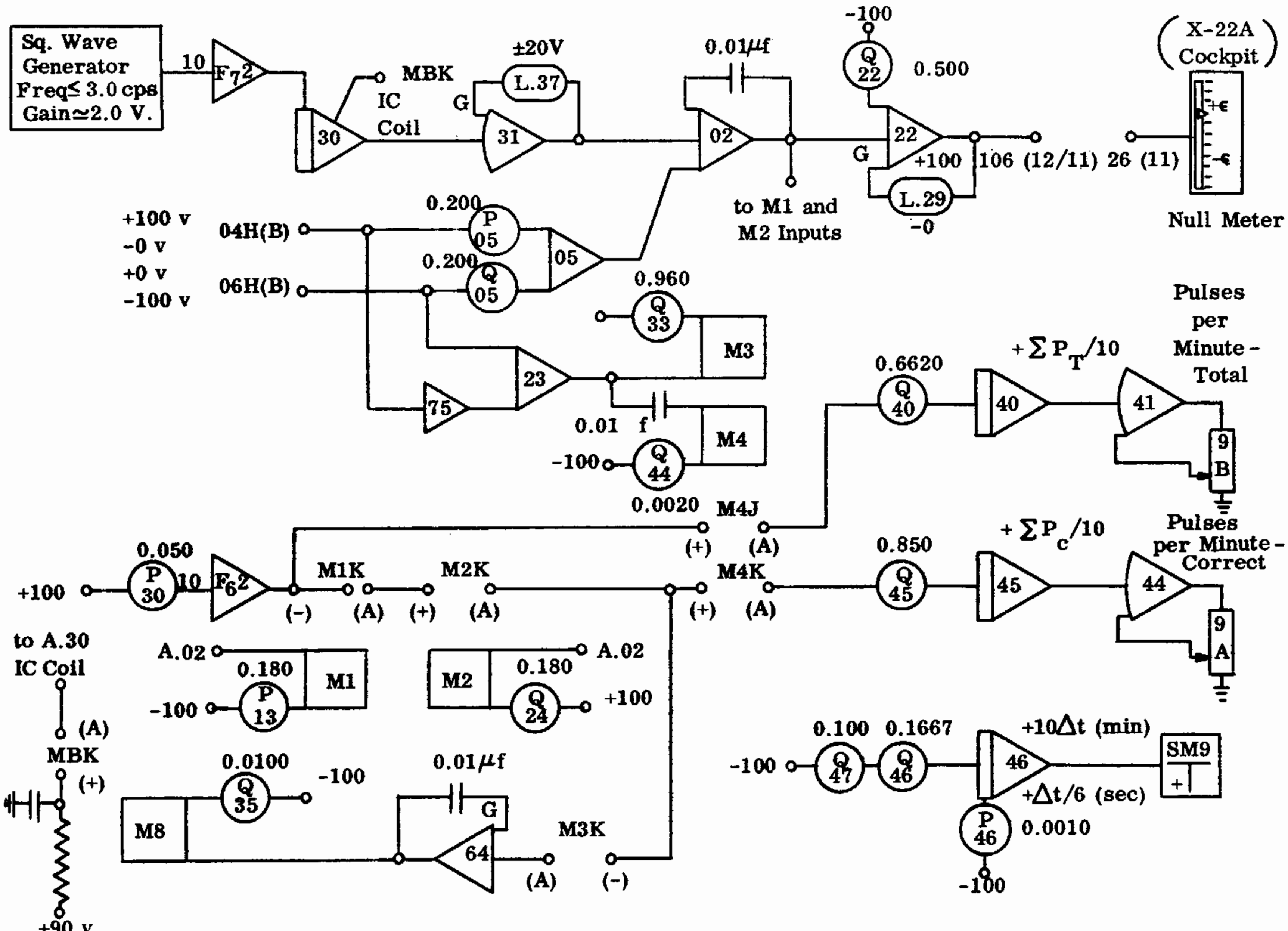

Figure 34. Schematic of Work Load Measurement Unit

REFERENCES

1. Cole, G., et al: "Study of Pilot-Controller Integration for Emergency Conditions", Tech. Documentary Report No. RTD-TDR-63-4092, Prepared by Honeywell Regulator Co., January 1964.

2. Salmirs, S. and Tapscott, R.J.: "The Effects of Various Combinations of Damping and Control Power on Helicopter Handling Qualities During Both Instrument and Visual Flight", NASA TN D-58, 1959.

3. Faye, A.E., Jr.: "Attitude Control Requirements for Hovering Determined Through the Use of a Piloted Flight Simulator", NASA TN D-792, April 1961.

BIBLIOGRAPHY

1. Anon.: "Proposal to Study the Application of Pilot-Controller Integration Techniques to a Representative V/STOL Aircraft", Bell Aerosystems Report No. D2226-953001, January 1964.

2. Brenl, H.T.: "A Simulator Study of Tilt-Wing Handling Qualities", paper presented at AIAA Simulation for Aerospace Flight Conference, August 1963.

3. Clark, J.W. and Miller, D.P.: "Research on Factors Influencing Handling Qualities for Precision Hovering and Gun Platform Tasks", United Aircraft Research Labs, Report UAR-D55, April 1965.

4. "Composite Detail Specification for Model X-22A Tri-Service VTOL Research Aircraft", Bell Aerosystems Report No. 2127-947001, September 1964.

5. "Demonstration Requirements for Airplanes; MIL-D-8708A(WEPS)", September 1960.

6. Dooley, M.R.: "Specification for Primary Artificial Feel and Trim System X-22A V/STOL Research Aircraft", Revision A, Bell Aerosystems Report No. 2127-947002, August 1964.

7. "Failure Rates and Analytical Techniques for Mechanical and Electromechanical Space Craft and Satellite Equipment", Lockheed Missile and Space Co. Document A048101-No. 6-16, July 1962.

8. "Flying Qualities of Piloted Airplanes; MIL-F-8785 (ASG)", November 1957.

9. Fromm, P.A.: "Hydraulic and Pneumatic System Design Report X-22A", Bell Aerosystems Report No. 2127-904001, October 1964.

10. Gampp, M.: "Flight Control System X-22A Aircraft - Schematics, Description, and Analysis", Bell Aerosystems Report No. 2127-903001, January 1965.

11. "Helicopter Flying and Ground Handling Qualities; General Requirements for; MIL-H-8501A", September 1961.

12. Knowles, W.B.: "Operator Loading Tasks", Human Factors, April 1963.

13. Martin, J.F.: "Investigation of Longitudinal Handling Qualities Requirements and the Development of Piloting Techniques for Tilt-Wing V/STOL Aircraft in Normal and Emergency Conditions", paper presented at AIAA Simulation for Aerospace Flight Conference, August 1963.

14. Michaels, J.L. and Hesby, A.T.: "Aerodynamic Stability and Control and Flying Qualities, X-22A", Bell Aerosystems Report No. 2127-917003, December 1962 - Revision A, December 1964.

15. "Operational Reliability and Maintenance Summary on the C-130B Airplane" Lockheed Georgia Co., October 1960.

16. "Reliability Control in Aerospace Equipment Development", Society of Automotive Engineers, 1963.

17. "Reliability Data Book; Estimating Reliability", Martin Co., ER 12123A, June 1962.

18. "Reliability Engineering Data Series - Failure Rates", AVCO Corp., April 1962.

19. "Reliability Stress Analysis for Electronic Equipment", MIL-HDBK-217 (WEPS), December 1961.

20. Seekel, E., Traybar, J.J. and Miller, G.E.: "Longitudinal Handling Qualities for Hovering", Princeton University Report No. 594, December 1961.

21. Tremont, R.A.: "Operational Experiences and Characteristics of the X-15 Flight Control System", NASA TN D-1402.

22. "X-22A Propeller Malfunction Analysis Report", Hamilton Standard Engineering Report HSER 2947, Rev. A., May 1964.

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13. ABSTRACT

This report presents final results of a study of the application of Pilot-Controller integration design techniques to the flight control system of a representative V/STOL aircraft. Under this program the validity of the concept was established in the application to the X-22A V/STOL. In this application the PCI technique indicated the areas of the X-22A flight control system where modifications would result in the greatest improvement to the probability of mission accomplishment. Design modifications were made and an iteration using the technique was accomplished and the payoff was evaluated. The digital program which was developed and applied to the X-22A has general applicability to other aircraft. Several improvements to this program as well as to the details of technique application are suggested.

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V/STOL flight control						

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