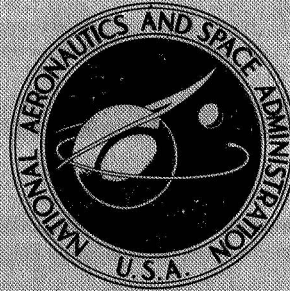NASA TECHNICAL
MEMORANDUM

NASA TM X-3409

NASA TM X-3409

ADVANCED CONTROL TECHNOLOGY AND ITS POTENTIAL FOR FUTURE TRANSPORT AIRCRAFT

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16. Abstract

This document is a compilation of papers presented at a symposium on advanced control technology held in Los Angeles, Calif., July 9 to 11, 1974. Papers were presented by representatives of industry, the Air Force, and various NASA Centers. The topics covered included fly by wire, digital control, control-configured vehicles, applications to advanced flight vehicles, advanced propulsion control systems, and the benefits of applying active control technology to transport aircraft.

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FOREWORD

Flight control technology passed several milestones in 1973 with the successful flight testing of two pure fly-by-wire systems—the NASA F-8 digital fly-by-wire system and the Air Force F-4 survivable flight control system. The emergence of these highly reliable fly-by-wire systems makes it possible to consider a stronger reliance on automatic control systems for the design optimization of future air transports. By including specialists in flight control systems in the preliminary design process, as well as specialists in aerodynamics, structures, and propulsion, the synergistic effect of the integrated design can be exploited to an extent not possible previously. This design philosophy has been referred to as the control-configured vehicle approach or the application of active control technology. Significant payoffs can be expected in terms of improved performance, longer aircraft life, reduced fuel consumption, reduced noise, and greater passenger comfort. Several studies and flight tests sponsored by the Air Force Flight Dynamics Laboratory and NASA demonstrated the potential of control-configured vehicles and active control technology.

A symposium on advanced control technology was sponsored by NASA to discuss recent advances in control technology and the impact they should have on future transport aircraft. The technical papers presented discussed work performed by government and industry. The topics covered included the flight test results of advanced control technology programs, such as fly by wire, digital control, and control-configured vehicles; the application of advanced control systems to such vehicles as the space shuttle orbiter, the Lockheed C-5A airplane, and the Boeing B-747 airplane; advanced and integrated propulsion control systems; and case studies of the benefits of applying active control technology to transport aircraft. Also included are papers on the design, testing, and reliability of advanced control systems, which are directed primarily toward the technical specialist.

A high point in the symposium was a panel discussion concerning the application of active controls to future transport aircraft, in which representatives of NASA, the Air Force, the FAA, the airlines, and aircraft manufacturers participated.

Herman A. Rediess
Symposium Technical Chairman
July 11, 1974

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SESSION III: ADVANCED CONTROL SYSTEMS APPLICATIONS

Chairman: Kenneth J. Szalai, NASA Flight Research Center

SESSION IV: ADVANCED CONTROL OF PROPULSION SYSTEMS

Chairman: Frank W. Burcham, Jr., NASA Flight Research Center

SESSION V: ADVANCED CONTROL/NAVIGATION SYSTEMS

Chairman: Brent Y. Creer, NASA Ames Research Center

SESSION VI: CASE STUDIES OF ADVANCED CONTROL TECHNOLOGY BENEFITS

Chairman: Laurence K. Loftin, Jr., NASA Langley Research Center

SESSION XI: RELIABLE CONTROL SYSTEMS TECHNOLOGY

Chairman: Remus N. Bretoi, NASA Ames Research Center

NASA ADVANCED CONTROL TECHNOLOGY:

AN OVERVIEW

Peter R. Kurzhals
NASA Headquarters

ABSTRACT

NASA's current and projected advanced control technology programs for future transport aircraft include the design and verification of full-flight envelope autopilots, the development and flight test of all-digital fly-by-wire systems, the evolution of low-cost innovative avionics concepts such as split surface stability augmentation systems, the evaluation of integrated propulsion control and cooperative autopilot/propulsion control systems, the application of active control systems to short-haul and long-haul transports, and the demonstration of reconfigured active-control aircraft. Key technical features and anticipated contributions of these technologies are outlined.

INTRODUCTION

With the advent of digital microelectronics, practical design of a new class of transport aircraft with significant performance gains and weight savings through active controls has become feasible. NASA is conducting several research and development programs specifically aimed at generation of the critical technology for such

advanced control aircraft and for the associated digital
flight control systems. This paper provides an overview
of these programs.

Other ongoing NASA efforts, such as the development
of highly-reliable, easily-maintainable computer systems
or automatic landing systems - essential but not unique
to active controls - are not treated here. NASA-sponsored
aircraft design studies concerned primarily with the
definition of advanced control technology requirements
and benefits for future transports - but not including
control research and development efforts - are also
omitted.

TECHNOLOGY OVERVIEW

NASA advanced control programs can be naturally
broken into the development of improved control systems
and into the application of the resultant control concepts
for the design of more efficient transport aircraft.
Figure 1 introduces the related activities. Work on
advanced control systems is addressed by programs on
full flight envelope autopilots (FFEAP), digital fly-by-
wire systems, innovative avionics systems, and propulsion
control systems. The extension of these control capabil-
ities to the definition and validation of advanced aircraft
designs will be addressed by the active control aircraft
and the proposed active-control-configured transport
programs.

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Major design and flight test milestones, indicated on the figure, for the NASA programs, will be outlined in the specific summary of each program. Completion of these design and verification tasks by the early 1980's is planned to permit the incorporation of advanced control concepts in the next generation of transport aircraft - currently projected to enter service in the mid to late 1980's.

Full-Flight-Envelope Autopilots

Under the first program on improved controls, the Ames Research Center is investigating the application of optimal control theory to the design of full flight envelope autopilots (reference 1). The associated design approach is illustrated in figure 2. The aircraft is calibrated over the entire flight regime by trim maps which tabulate lift, drag, and moment coefficients versus critical aircraft variables such as angle of attack and flap angle. The control deflections required to obtain a commanded acceleration can then be calculated from these trim maps, with feedback used to compensate for mismatch between the trim maps and the actual aircraft characteristics. The result is a linear acceleration command system, and linear optimization theory can be applied to define desirable overall trajectory and attitude control algorithms.

Detailed design of this full-flight-envelope auto-pilot (FFEAP) is presently underway and is expected to be completed in mid 1975. Following FFEAP validation during six-degree-of-freedom simulations in 1976, the first implementation of the FFEAP is planned as an experiment, the FFEAP algorithms will be programmed on the onboard STOLAND system, and evaluated during representative flight operations. If proven successful, follow-on flight tests of the FFEAP system will be conducted on short-haul powered-lift aircraft, such as a tilt-rotor configuration, in late 1978.

Preliminary FFEAP results indicate that the optimized controller mechanization could significantly increase transport aircraft performance over the entire flight envelope, and could minimize delays and fuel consumption during terminal area operations.

Digital Fly-By-Wire Systems

A companion NASA control system program, conducted jointly by the Flight Research Center and the Langley Research Center, involves the development and flight verification of digital fly-by-wire (DFBW) systems. The basic phases of this program are represented in figure 3. Phase I (references 2-4) has demonstrated the feasibility and performance of DFBW control using

Apollo hardware in a single-channel primary system with
a triplex analog backup system (reference 5) installed in
an F-8 aircraft. Direct, stability augmentation, and
command augmentation system modes were successfully
evaluated during approximately 18 months of flight testing.
For Phase II, the Phase I fly-by-wire systems will be
replaced by a triplex all DFBW system using aircraft-
compatible computers and sensors. The all-DFBW system
will then serve as a test bed for early verification
of critical Space Shuttle software concepts and for
flight implementation of several advanced control law
concepts. The first of these, summarized in figure 4,
will investigate performance improvements obtainable
through synthesis of selected control configured
vehicle (CCV) concepts. Specific CCV systems considered
include static stability augmentation, maneuver and gust
load control, and envelope limiting. The associated
control algorithms were designed through an iterative
quadratic optimization process (reference 6), and are
being validated during laboratory simulations at the
Langley Research Center. After software coding and
iron-bird checkout of the CCV algorithms, first flight
tests of the CCV system are scheduled for mid 1976.

The second advanced control study, illustrated in
figure 5, addresses the mechanization of an adaptive
control system compatible with potential transport

applications. Candidate concepts under investigation are an implicit identification scheme (reference 7) involving multiple-model hypothesis testing; and two explicit identification schemes based on different techniques for parameter identification and control optimization. The first uses a recursive, weighted-least-squares identifier, and an algebraic equation to determine the control changes from the previous commands (reference 8). The second uses a modified Newton-Raphson technique for identification. Other potential advanced control approaches being considered for flight test include self-organizing systems (references 9-10), which can automatically restructure themselves to accommodate sensor and actuator failures with considerable attendant reliability increases; and learning control systems with the capability to evolve improved aircraft modeling and estimation techniques during flight. The most promising of these control concepts will be selected for flight implementation in 1976. In-flight tests on the F-8 will then be conducted in 1977, following mechanization and ground verification of the resultant flight control systems.

Current Flight Research Center plans for the total redundant DFBW systems tests call for a 30 month flight test program beginning in 1976. Approximately six months will be devoted to validation of the basic system.

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configuration and to inflight verification of Space
Shuttle software designs. The remainder of the program
will be available for the advanced control law tests.

Innovative Avionics Systems

Besides these efforts on the exploitation of digital
control, work at the Flight and Ames Research Center is
concerned with the design and mechanization of innovative
avionics systems which could reduce avionics cost
through simplification and modularization. While the
primary users of such concepts will be general aviation
aircraft, many of the associated design philosophies
may be applicable to transports as well.

One of these concepts, depicted in figure 6,
involves the development and flight demonstration of a
separate surface stability augmentation system (SSSAS)
on a Beech 99 commuter airlines under a contract managed
by the Flight Research Center. With this approach
(reference 11), the aircraft control surfaces are split
into primary and secondary segments, and the separate
secondary surfaces are incorporated in a limited-
authority ride smoothing and gust alleviation system.
Since the primary control system can overide the
secondary system in case of a hard-over failure, the
SSSAS may be mechanized with single-string, low-cost
components with considerable associated system cost

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savings. Major improvements in ride quality are expected through this approach, which will be validated during extensive flight tests in 1975.

Another low-cost avionics program, conducted by the Ames Research Center, focuses on the design of integrated avionics systems which take maximum advantage of recent advances in microelectronics and digital circuit technology. The design philosophy for this system, illustrated in figure 7, will be initiated with subsystem concept studies and 1980 technology and air traffic projections. The resultant specifications and requirements will be used to define candidate modular avionics systems. The most cost-effective of these systems will be carried through subsystem development and final design by 1979; and will be evaluated through piloted flight simulations in 1980.

Propulsion Control Systems

The application of advanced control techniques to the optimization of aircraft propulsion systems performance can also result in large improvement in engine thrust and fuel economy. Two related NASA programs, conducted jointly by the Flight and Lewis Research Centers, are concerned with the development of integrated propulsion control systems (IPCS) and with cooperative aircraft and propulsion control.

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For the first of these efforts, the Air Force and NASA have undertaken a joint program (reference 12) to demonstrate inflight the benefits obtainable from an integrated propulsion control system in an F-111 aircraft. The associated design philosophy, indicated in figure 8, utilizes a high-response control system which rapidly senses changes in flow conditions and uses a digital controller to command engine inlet geometry configurations needed for optimal propulsion performance. Such an IPCS can minimize stall margin throughout the flight environment, and could permit significant reduction in current engine safety margins, with attendant increases in range projected as large as 10 percent. The F-111 IPCS is slated for flight tests in 1975.

A second NASA effort on propulsion control involves the integration of the propulsion and aircraft control systems (reference 13) for the YF-12 research vehicle. The analysis of supersonic flight tests on the XB-70 and YF-12 indicate that airframe/propulsion system interactions are the primary cause of altitude fluctuations in supersonic cruise, of poor lateral-directional characteristics, and of severe transients during inlet unstarts. It is clear from these flight results that the propulsion system cannot be treated independently from the aircraft control system. A proposed integrated

airframe/propulsion control system, shown in figure 9, thus incorporates a digital control system which combines the inlet, engine and airplane flight controls. The longitudinal phase of this cooperative control system will be flight tested on a YF-12 in 1975, followed by YF-12 flight tests of the lateral directional phase in 1976. Design specifications for a total cooperative control system, based on these interim test results, are expected to be available by late 1977.

The most significant payoff of the advanced control approaches discussed so far requires consideration of their capabilities in the selection of the initial aircraft configuration through a new aircraft design approach which permits full tradeoffs between aerodynamics, structures, and control for the designated mission requirements. With this active control design approach, reductions in the aircraft natural aerodynamic stability and structural loads could be obtained through reliance on the damping and load control capabilities of a flight-critical automatic control system. These reductions in turn can permit large savings in aircraft gross weight and fuel. NASA is conducting two programs to provide and verify the critical technology required for early application of such active control designs in future civil transports.

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Active Control Aircraft

The Active Control Aircraft (ACA) program, carried out by the Langley, Flight and Ames Research Centers, concentrates on development of the integrated active control system and aircraft design technology to meet the needs of new short-haul and long-haul transport designs in the early 1980's. Initial work will focus on the formulation of an adequate modeling and analysis base for ACA design. Specific associated tasks include the generation and validation of transonic aerodynamic pressure distributions for deflected and oscillating control surfaces, of aeroelastic design programs for flutter suppression, of prediction techniques for aircraft structural dynamics and static deformations, and of insensitive control techniques which allow for uncertainties in the aircraft aerodynamic and structural parameters. An integrated conceptual design program incorporating these modeling and analytical procedures for ACA will be derived to permit incorporation of all the active control functions into a workable system, and selection of the most cost-effective aircraft configuration for a given mission. One of the approaches under consideration for the conceptual design process is outlined in figure 10. After specification of general configuration guidelines and mission requirements, this

computer-aided design program defines the initial
aircraft geometry and uses a quadratic optimization
procedure to converge on suitable final configurations.
An economic assessment subroutine is then employed to
determine the best of these alternate configurations,
and to select the final active control aircraft and
system designs. To provide the necessary system and
aircraft inputs for this approach, wind tunnel tests
and validation flights, using DHC-6 and subsonic
transport "test beds", are planned in 1976 and 1977.

The next phase of the program involves the
extension of this initial work into specific short-haul
and long-haul transport applications. For the short-
haul application, depicted in figure 11, a ride quality
and precise trajectory tracking system will be designed
and installed on a DHC-6 Twin Otter aircraft. Representa-
tive operational flights of the modified DHC-6 will be
conducted in 1978 to demonstrate the active control
system performance and benefits. The system is
expected to significantly improve ride quality for
low-wing-loading STOL aircraft, as indicated in the
figure. Following completion of these tests, a more
extensive short-haul active control design for powered-
lift aircraft incorporating envelope limiting, ride
quality control, gust load alleviation, maneuver load
control, and flight path control will be designed and

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evaluated for a Tilt Rotor vehicle. Completion of these evaluations is scheduled for the mid 1979 time frame.

For the long-haul application, represented in figure 12, a series of contracted active control aircraft designs considering reduced static stability, gust and maneuver load alleviation, ride quality and fatigue-life control, envelope limiting, and flutter and structural mode suppression will be conducted for representative subsonic, freighter, transonic, and supersonic missions; and the results will be compared with conventional aircraft designs. The most promising of these designs will then be evaluated in the 1980-1981 time period through design, fabrication and flight tests of a scaled research vehicle which will concentrate on the demonstration of the high-risk technologies essential to validation of the ACA design techniques.

Completion of the ACA program should provide a comprehensive design base for the application of active control.

Active Control Configured Transport

In addition to NASA's work on active control design procedures and systems, a companion program which would carry this technology into practice through actual redesign of a jet transport is under consideration for initiation in mid 1975. This Active Control Configured

Transport (ACCT) program, to be managed by the Flight
Research Center, would make direct use of the digital-fly-
by wire and active control technology program outputs to
redesign a small jet transport, such as a Jetstar or
B-737, to evaluate the resultant benefits and penalties
in a realistic operational environment. Such a recon-
figured aircraft could offer major performance improve-
ments (reference 14) through synergism of active controls
and advanced aerodynamic technologies. Figure 13
illustrates some of these potential benefits in terms
of relative fuel consumption. While individual
contributions of either control or aerodynamic technologies
are relatively small, the combination of a fly-by-wire
active control system with a high-aspect-ratio super-
critical wing design made possible through maneuver-
load and gust alleviation can yield appreciable fuel
savings. Noise footprints for such an ACCT design
could also be reduced by as much as 90%, based on early
engineering estimates.

The ACCT program, represented in figure 14, could
include an actively-controlled supercritical wing
located for optimum static margin and a corresponding
new horizontal and vertical tail. To minimize
demonstration costs, the associated active control
system could be mechanized using the all DFBW system
proven during the F-8 program. Extensive ACCT flight

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tests would verify the fuel savings and performance and ride quality improvements obtainable with an integrated active control transport design, and would provide invaluable experience on the active control system and aircraft operations in a representative flight environment.

If warranted by the initial conceptual designs and cost/benefit studies, an ACCT test aircraft would be selected in 1976. Design of an active control configuration for this transport could be completed in 1977, and the test aircraft could be modified by 1979 for operational flights in the 1980-1981 time span. By involving potential users throughout the design, implementation, and flight test phases of such a demonstrator, airline, industry, and FAA acceptance of active controls could be significantly accelerated and the associated technology could be made widely available for future transport applications.

CONCLUDING REMARKS

The successful completion of the NASA programs touched on in this brief overview should permit a major step forward in the application of advanced control concepts by providing a better understanding of the associated system and aircraft design problems and benefits. Maximum participation by industry in the

definition and implementation of these programs, and wide dissemination of the resultant design, development, and flight test data will - we hope - be instrumental in bringing about the early realization of the potential of active controls.

We stand on the threshold of a revolution in aircraft design, if we can learn to practically harness the capability of digital avionics and advanced controls. With the increased emphasis on cost-effectiveness and fuel-economy, we must take full advantage of this capability for the development of more efficient and competitive future transports to maintain our leadership role in the marketplace - and in the air.

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NASA ADVANCED CONTROL OVERVIEW

FULL FLIGHT ENVELOPE AUTO PILOTS (FFEAP)

DESIGN C-8 FFEAP TILT ROTOR FFEAP

DIGITAL FLY-BY-WIRE SYSTEMS

F-8 APOLLO SYSTEM

F-8 ALL DIGITAL SYSTEM

BEECH 99 SSSAS

INTEG. GA SYSTEM DESIGN

INNOVATIVE AVIONICS SYSTEMS

F-111 IPCS

YF-12 COOP. CONTROL DESIGN

PROPULSION CONTROL SYSTEMS

VALIDATION FLIGHT TESTS

SHORT HAUL ACA SYSTEM

LONG HAUL ACA SYSTEM

ACTIVE CONTROL AIRCRAFT

DESIGN A/C MOD FLT TESTS

ACTIVE CONTROL-CONFIGURED TRANSPORT

72	73	74	75	76	77	78	79	80	81	82

CALENDAR YEARS

FIGURE 1

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FULL FLIGHT ENVELOPE AUTO PILOT

SENSORS

ESTIMATION SYSTEM

COMMAND GENERATOR

PATH REGULATOR

TRIM MAP

ATTITUDE REGULATOR

ATTITUDE TRIM MAP

CONTROL SURFACE COMMANDS

ACTUATORS

ENGINE COMMANDS

ESTIMATION

CONTROL

ATT. COMMAND

LEGEND

X	STATE VECTOR
Ω	ANGULAR RATE VECTOR
ϕ, θ, ψ	EULER ANGLES
v	LINEAR VELOCITY
a	LINEAR ACCELERATION
ε	ERROR
c	COMMAND

FIGURE 2

DIGITAL FLY-BY-WIRE SYSTEM

PHASE I
DEMONSTRATE
FEASIBILITY
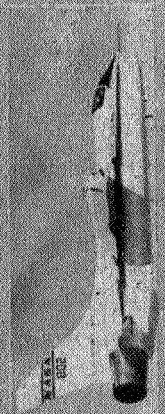
PHASE II
ESTABLISH
DESIGN
BASE

APOLLO COMPUTER

APOLLO SENSORS
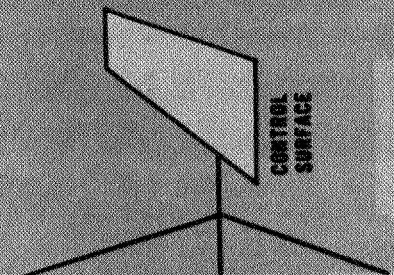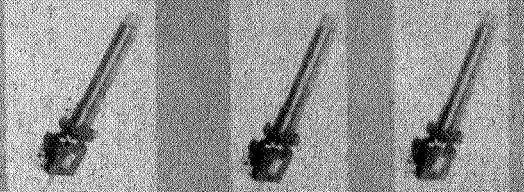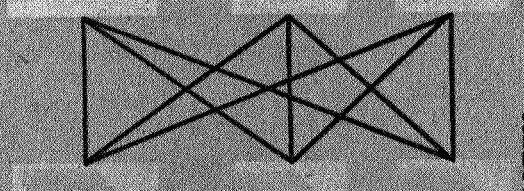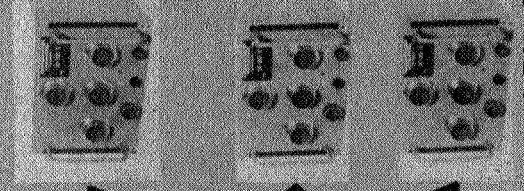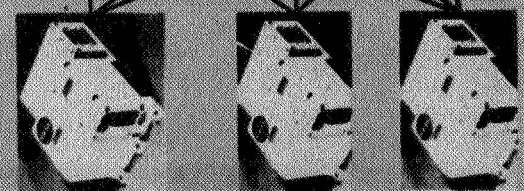
CONTROL SURFACE

ALL DIGITAL SYSTEM

FIGURE 3

24

DIGITAL FLY-BY-WIRE FLIGHT TESTS
CCV CONCEPTS

FIGURE 4

25
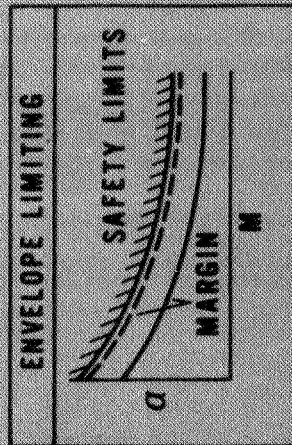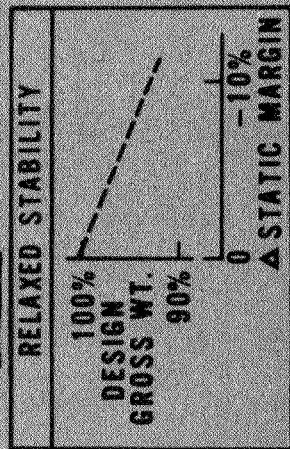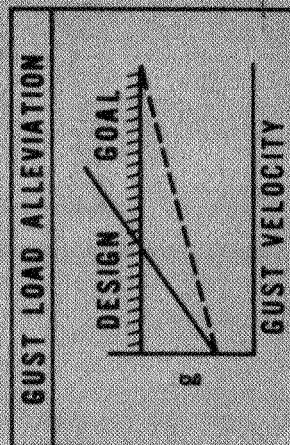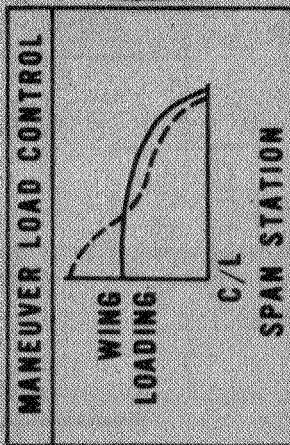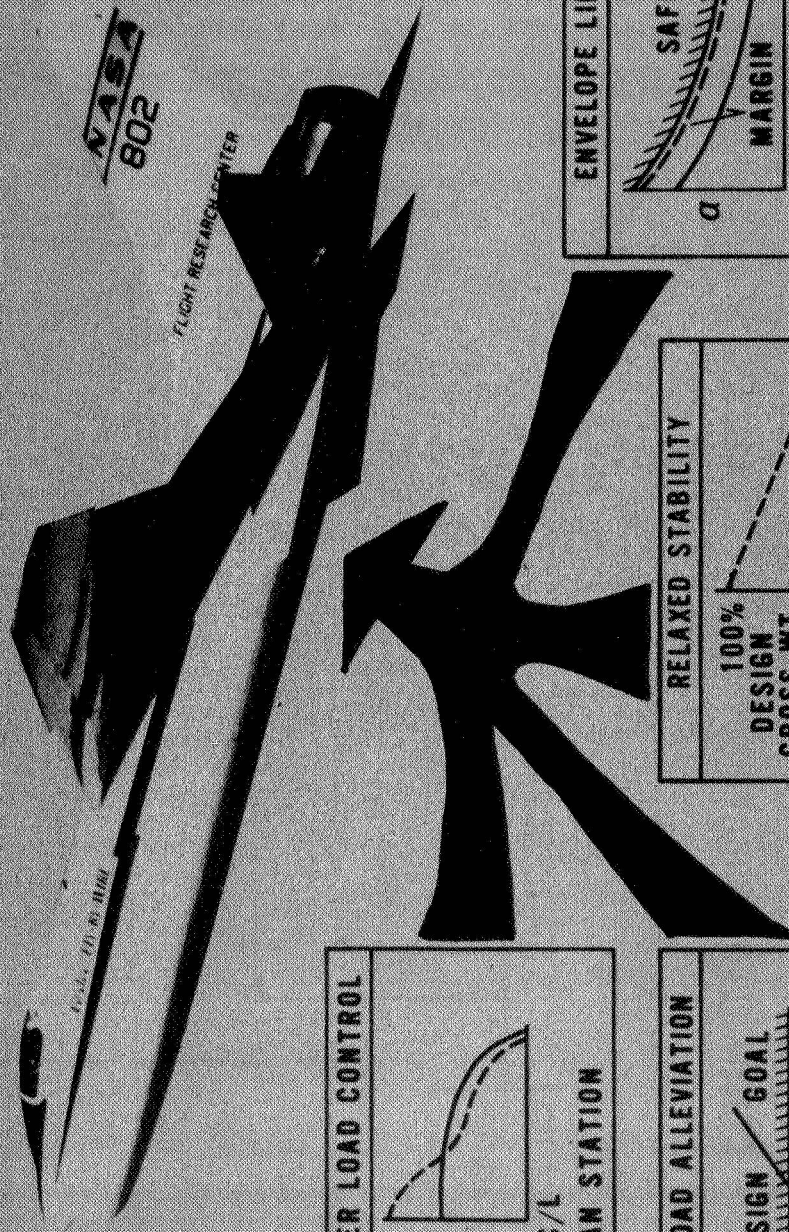
DIGITAL FLY-BY-WIRE FLIGHT TESTS

ADAPTIVE CONTROL CONCEPTS

CONTROLLER STRUCTURE
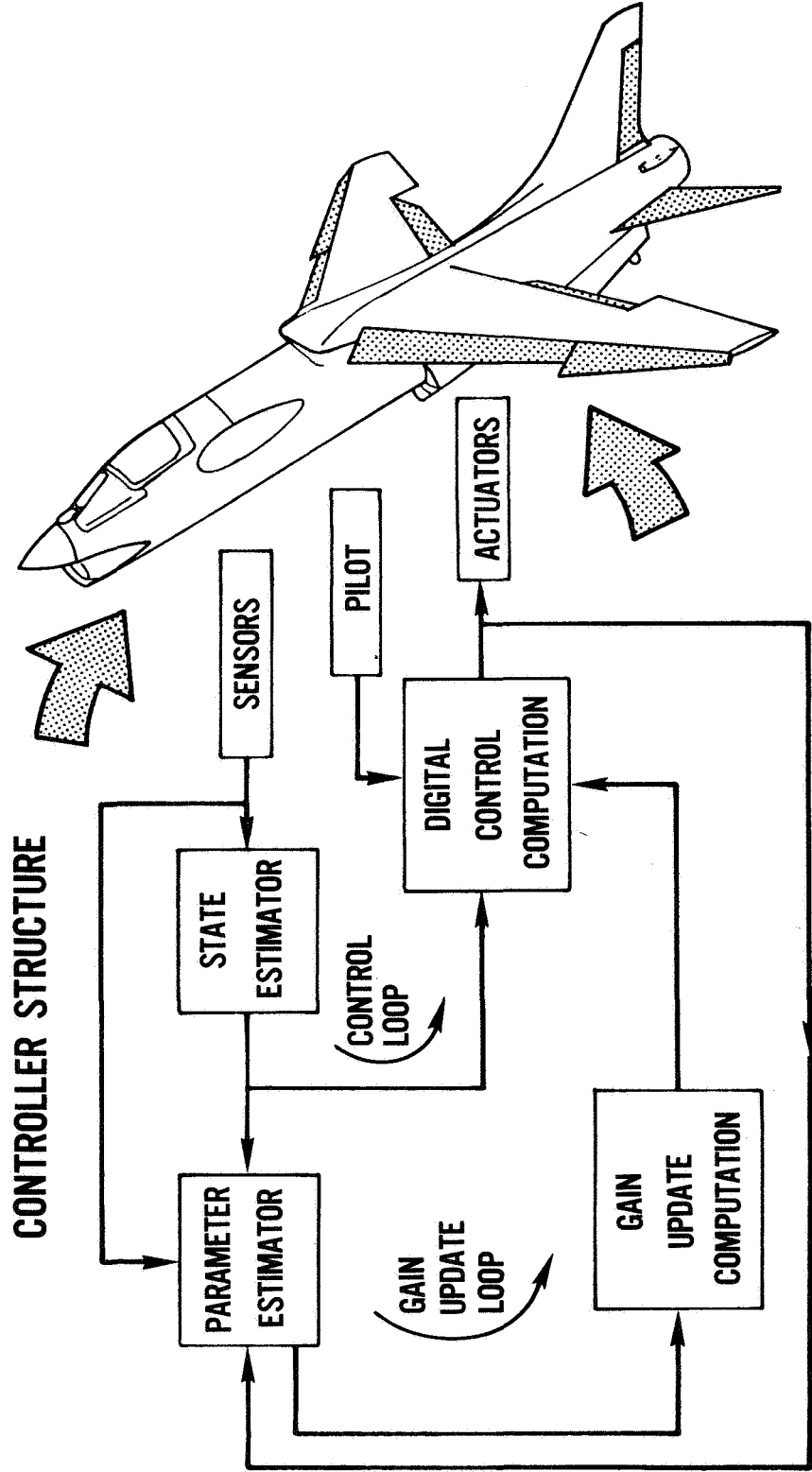
FIGURE 5

SEPARATE SURFACE STABILITY AUGMENTATION

FIGURE 6

27
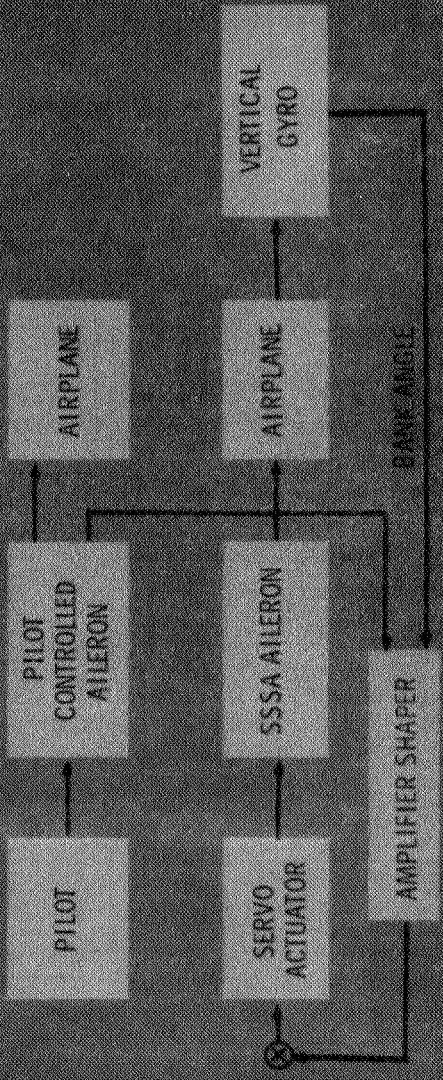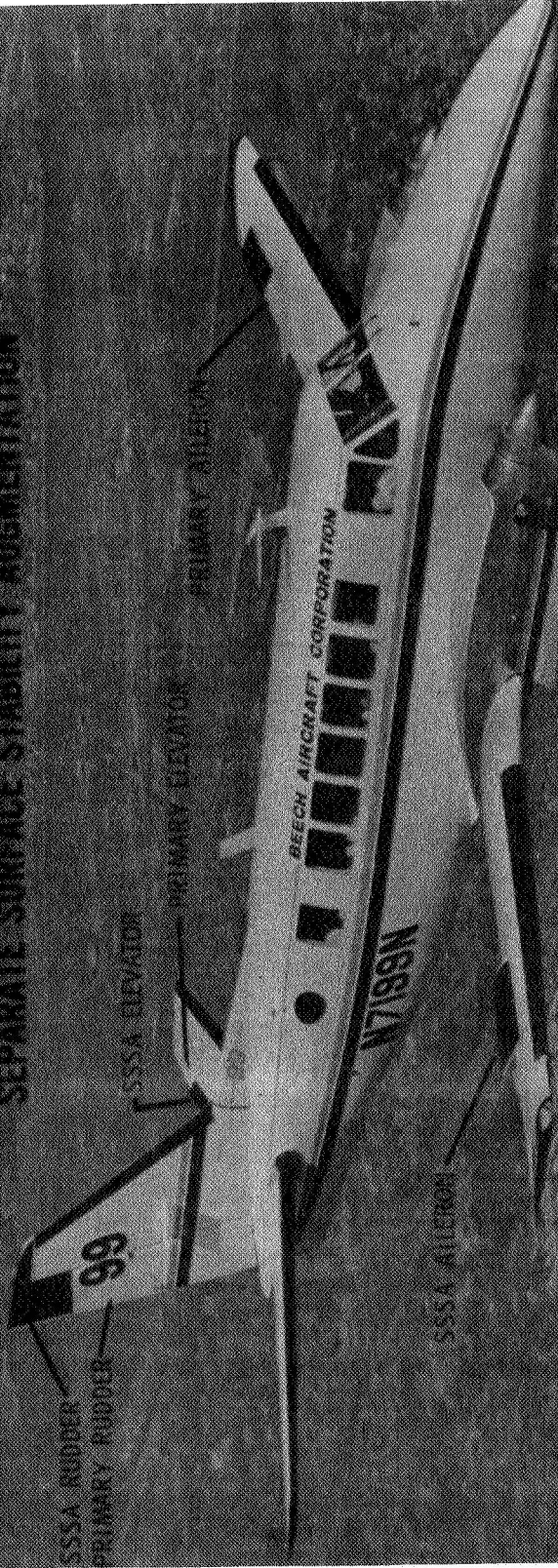
INTEGRATED AVIONICS SYSTEM

FIGURE 7

28
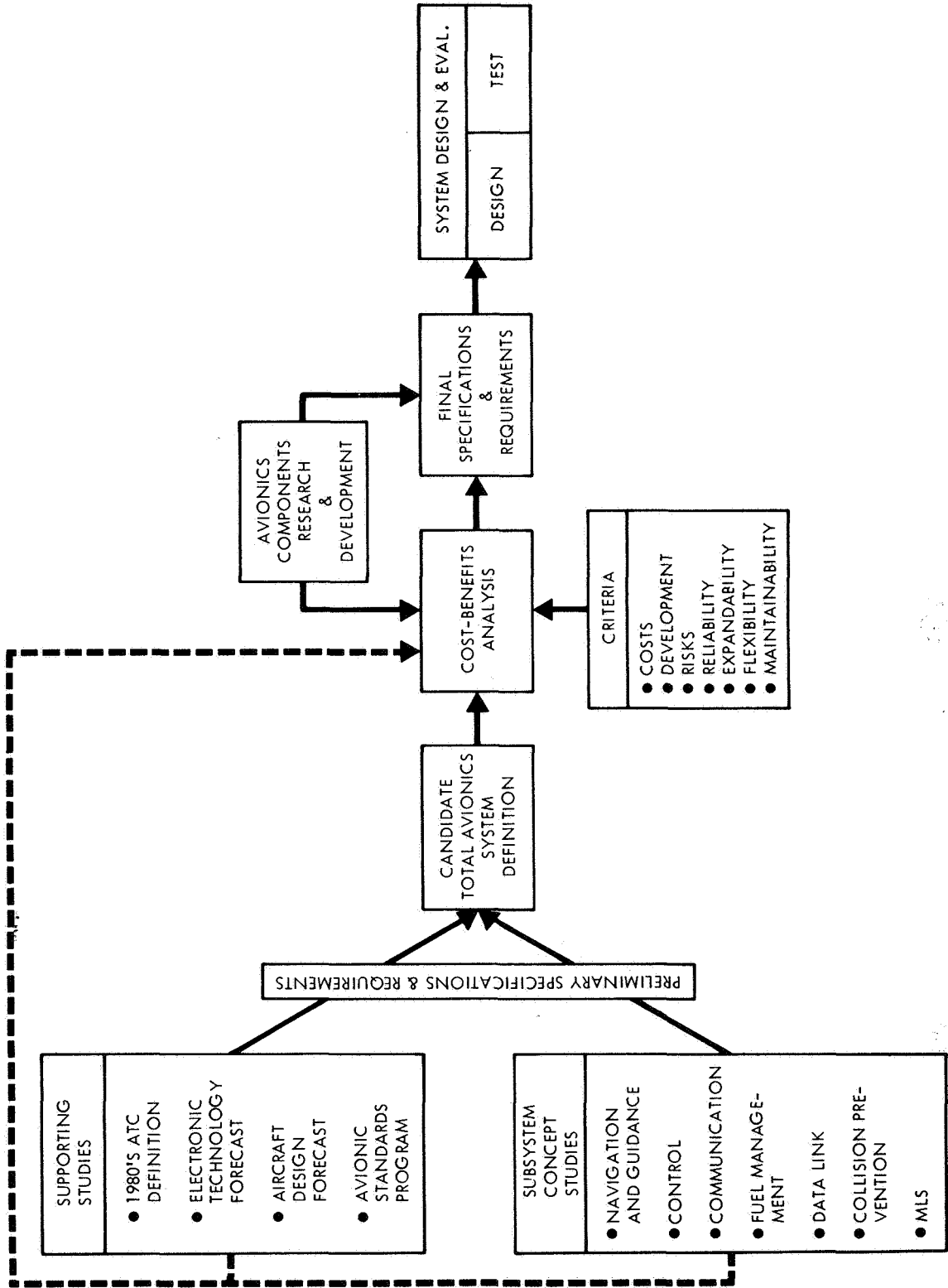
INTEGRATED PROPULSION CONTROL SYSTEM
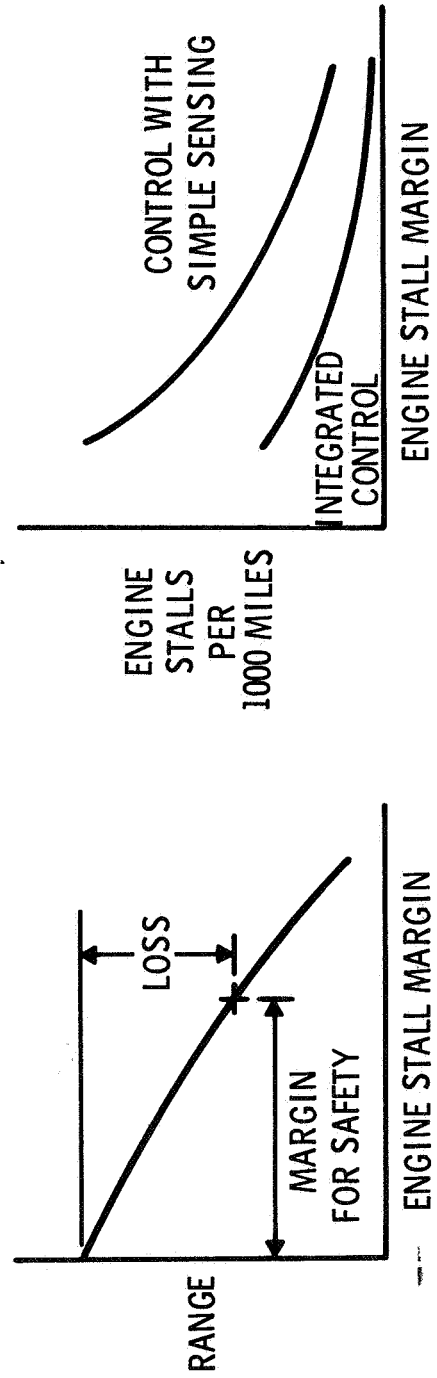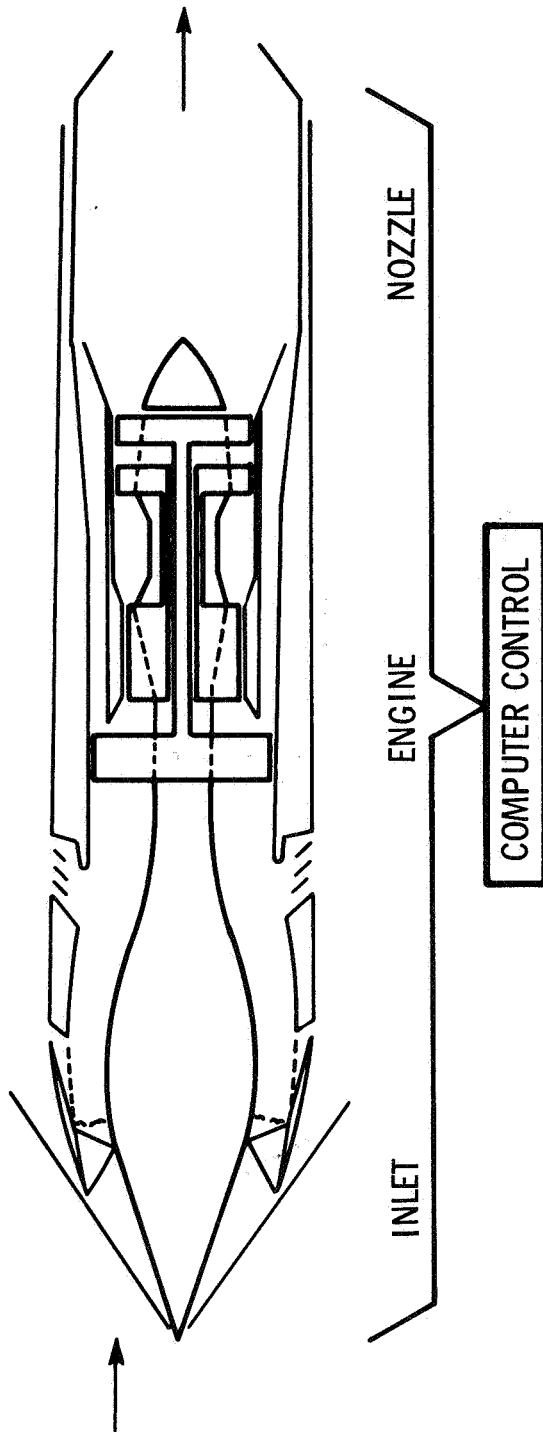
FIGURE 8

COOP AUTOPILOT/SAS/PROPULSION CONTROL SYSTEM

AUTOTHROTTLE

SPIKE POSITION SERVO
BYPASS DOOR SERVO

DUCT PRESSURE RATIO
SPIKE POSITION
BYPASS DOOR POSITION

COMPUTER

RATE
ACCELERATIONS

FLIGHT
CONTROL
SERVOS

SAS
AUTOPILOT

α
β

FIGURE 9

30
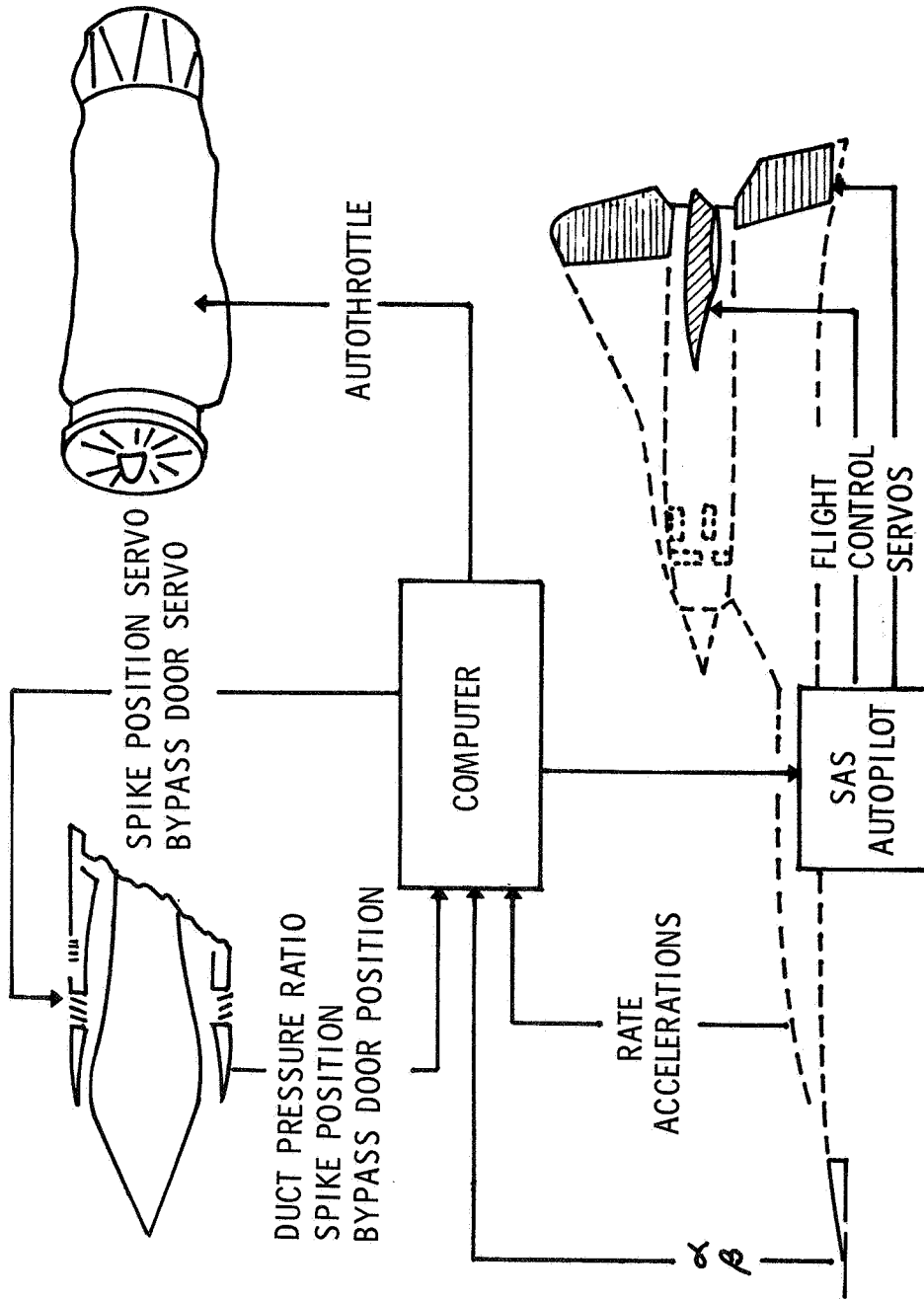
COMPUTER-AIDED DESIGN APPROACH
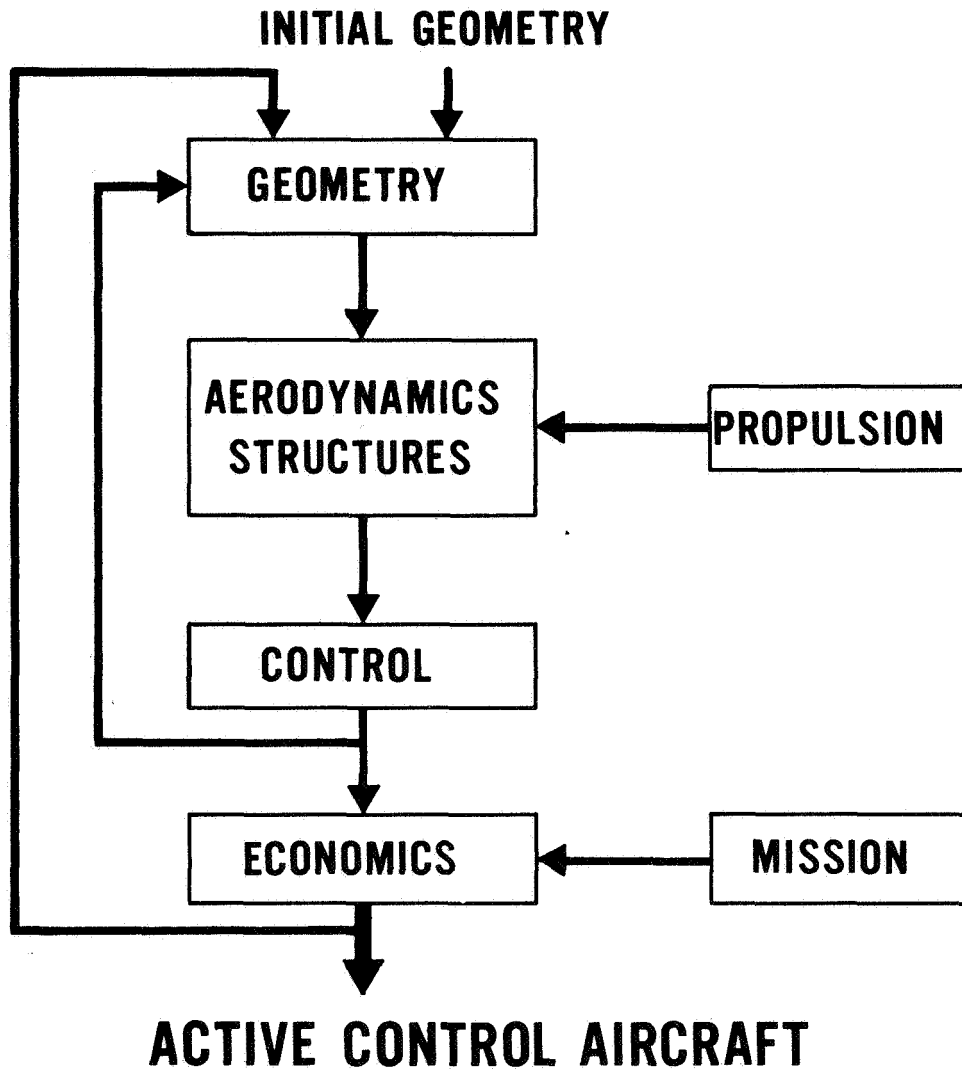
FIGURE 10

ACTIVE CONTROL AIRCRAFT

SYSTEM DESIGN & VALIDATION

OBJECTIVES

- ACTIVE CONTROL SYSTEM SYNTHESIS

- IMPROVED RIDE QUALITY & TRAJECTORY CONTROL

- OPERATIONAL FLIGHT EXPERIENCE

CONVENTIONAL DESIGN

CRITERIA

ACA DESIGN

VERTICAL RMS ACCELERATION ~ g's

.3 .2 .1 0

FWD   MID   AFT

PASSENGER COMPARTMENT LOCATION
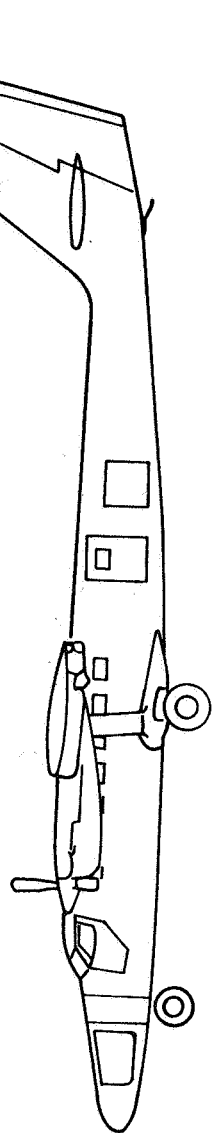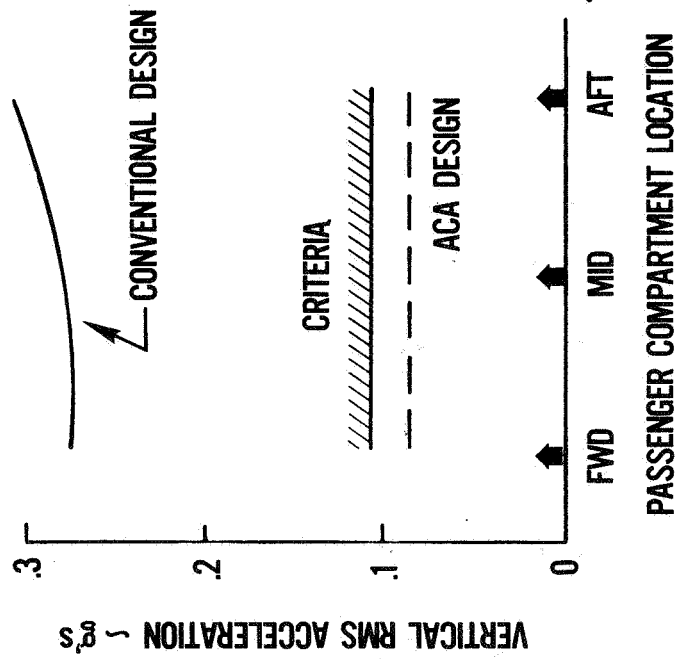
DHC-6 TWIN OTTER AIRCRAFT

FIGURE 11

ACTIVE CONTROL AIRCRAFT
HIGH-RISK CONCEPT VERIFICATION

CONTROL

PROPULSION

AERODYNAMICS

STRUCTURES

ACT

INTEGRATED CONCEPTUAL DESIGNS

SUBSONIC

FREIGHTER

TRANSONIC

SUPERSONIC

SPECIAL RESEARCH VEHICLE
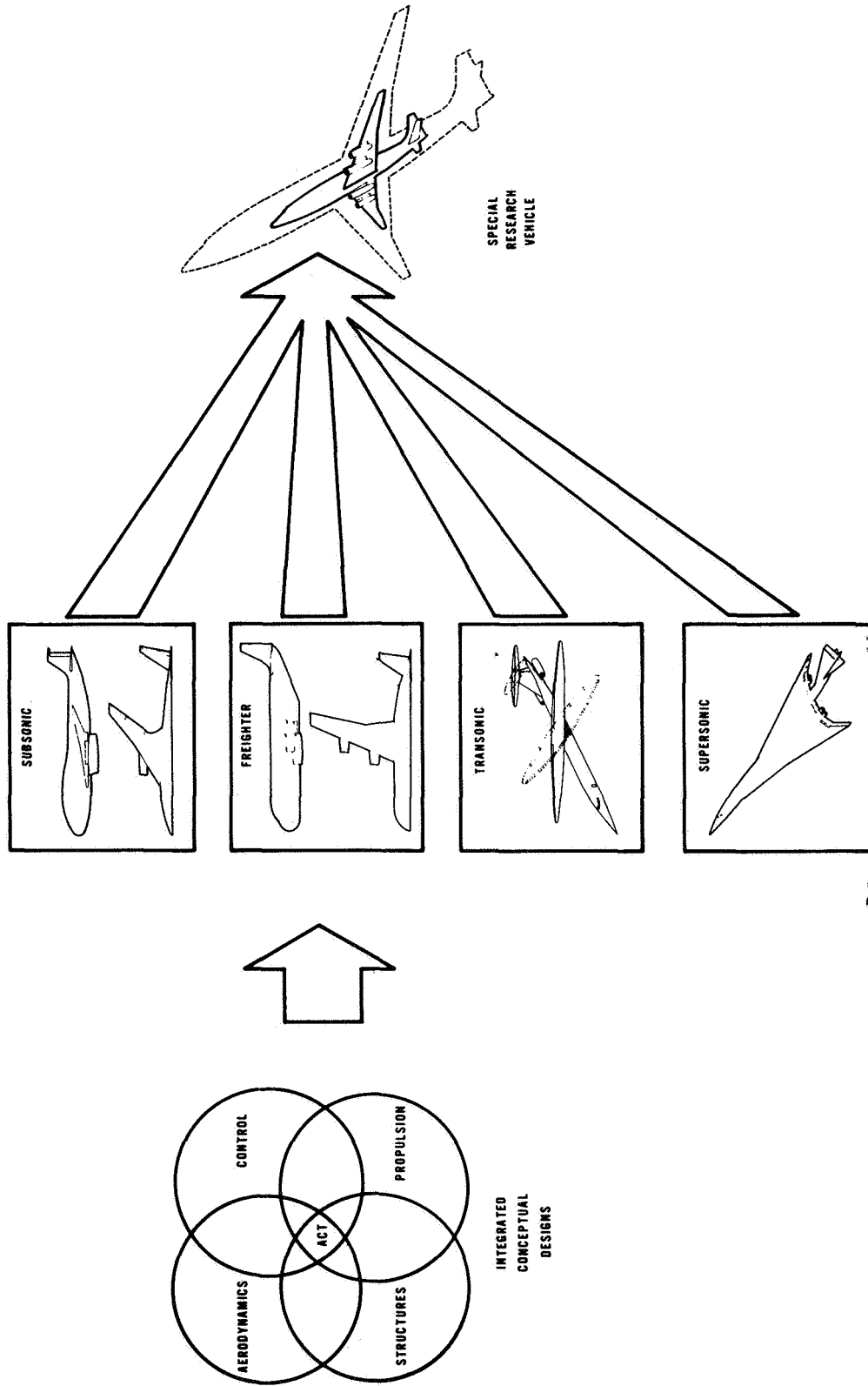
FIGURE 12

ACTIVE CONTROL CONFIGURED TRANSPORT
POTENTIAL BENEFITS

BASIC JETSTAR

REDUCED STABILITY
FLY-BY-WIRE

SUPERCRITICAL
WING

ACCT REDESIGN
SUPERCRITICAL WING
FLY-BY-WIRE

RELATIVE REDUCTION IN FUEL CONSUMPTION

FIGURE 13

34
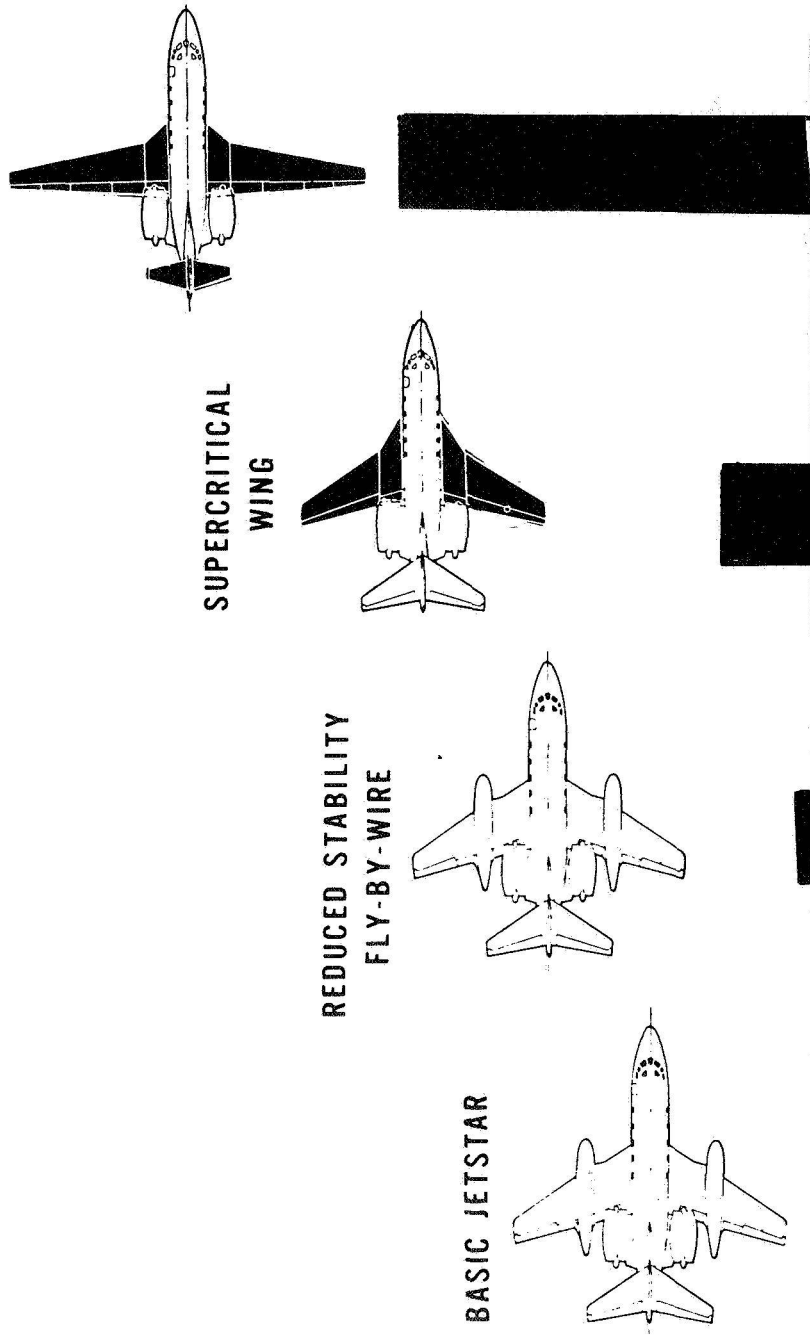
ACTIVE CONTROL CONFIGURED TRANSPORT
PROGRAM OBJECTIVES

ENCOURAGE PARTICIPATION BY

- INDUSTRY
- FAA
- AIRLINE PILOTS
- AIRLINES

APPLY DIGITAL FLY-BY-WIRE

COMPUTER
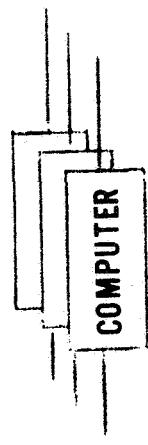
GAIN EXPERIENCE IN

- RELIABILITY
- PRACTICALITY
- MAINTAINABILITY

IMPLEMENT INTEGRATED DESIGN

STRUCT.

FLT. CONT.

AERO.

PROP.
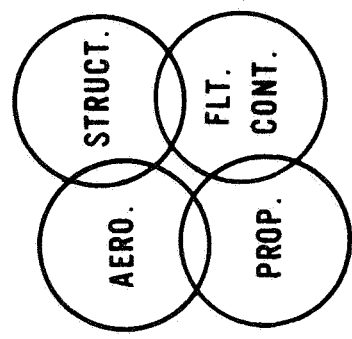
GAIN

- CONFIDENCE
- EXPERTISE
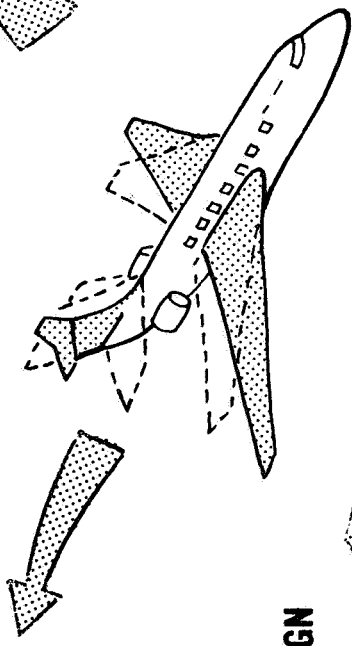
PERFORMS ORIGINAL MISSION

ORIGINAL FUEL

20% LESS FUEL

DEMONSTRATE

- FUEL SAVINGS
- IMPROVED PERFORMANCE
- RIDE QUALITIES

FIGURE 14

35
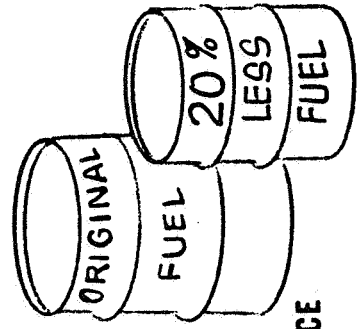
AFFDL ADVANCED CONTROL TECHNOLOGY PROGRAMS —

AN OVERVIEW

William E. Lamar
Air Force Flight Dynamics Laboratory

ABSTRACT

Over the past years the Air Force Flight Dynamics Laboratory has been active in developing a broad technology base for advanced flight control systems. This has permitted timely implementation and continued progressive evaluation of Control Configured Vehicle (CCV) concepts. The recent fly-by-wire and CCV B-52 successes have led to increased application of the concept in such aircraft as the B-1, C-5A, YF-16, and YF-17. The new Advanced Fighter Technology Integration (AFTI) program of the Laboratory represents a more extensive embodiment of the concept of a control configured aircraft.

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SURVIVABLE FLIGHT CONTROL SYSTEM
FLY-BY-WIRE DEVELOPMENT AND FLIGHT TEST

James E. Hunter
Air Force Flight Dynamics Laboratory

SUMMARY

The United States Air Force initiated the Survivable Flight Control System (SFCS) Advanced Development Program in July 1969 in which one of the major objectives was to establish the practicality of the Fly-by-Wire (FBW) concept for use in military fighter aircraft. This advanced development program provided for the design, fabrication, qualification, and highly successful flight test evaluation of a quad-redundant FBW primary flight control system in an F-4 aircraft without conventional mechanical controls.

Results from this intensive FBW advanced development effort indicate significant improvements in overall flight control system performance, reliability, safety and maintainability. Additionally, the strong and credible FBW technology base developed as a result of this program has paved the way for further exploitation through the application of advanced concepts such as Control Configured Vehicles and Multi-Mode Controls.

INTRODUCTION

The Air Force has accomplished a significant major milestone in advanced fighter flight control design and reliability with successful completion of its fly-by-wire flight test under the Survivable Flight Control System (SFCS) Program. This portion of the SFCS Program has consisted of a four year, $16.5 million development program with primary objective for developing and flight testing control elements to improve combat survivability of aircraft weapon systems. The development program was performed by McDonnell Douglas under contract to the Air Force Flight Dynamics Laboratory, with Sperry Rand, General Electric, and Lear Siegler as the principal equipment suppliers. Although the primary purpose of this program was to develop a highly reliable flight control system designed to improve survivability, major improvements in handling qualities, stability and performance, and weapon delivery accuracy were also achieved goals.

The SFCS mechanized in a YF-4E test aircraft uses a quadruply redundant (two-fail operate), dispersed, three-axis, analog, fly-by-wire (FBW) primary flight control system allowing the pilot to command aircraft motion rather than the conventional control surface position. This is the first high performance fighter aircraft ever to fly using a futuristic "all electric" system.

The potential and advantages of FBW had been demonstrated in several

exploratory development programs beginning in 1959 at the Flight Dynamics Laboratory. FBW acceptance was dependent upon answering the question of whether electrical control systems, such as implemented in the SFCS, could be made as reliable as conventional mechanical control systems. The task, undertaken during the SFCS flight test, was to demonstrate that the system developed during this program performs significantly better with greater reliability than currently operational flight control systems.

TECHNICAL APPROACH

Simple direct mechanical linkages, cables, and feel springs for manual control can no longer cope with many of the control system problems associated with modern high performance aircraft and aerospace vehicles. In an effort to meet the greater demands of these advanced control system requirements, the flight control designer has been forced to increase the complexity of the mechanical system with a resulting increase in weight, volume, cost, and a decrease in flexibility and reliability. Often he is forced to compromise between the desired performance and design requirements and a practical mechanization. FBW not only meets the demands of these advanced control system design requirements but does so with a decrease in complexity, weight, volume and cost. It also provides an increase in flexibility, reliability, and by utilizing redundancy and dispersion increased survivability.

While providing improved survivability, it was felt and later verified that the use of FBW primary flight control would enable major improvements in the tactical capability of the vehicle. The design of the system was therefore based upon studies to define a set of control laws which provided nearly optimum aircraft response characteristics desired by pilots for use during the various mission tasks assigned to the test aircraft in its normal operational use.

Traditional performance criteria, such as many of those presented in MIL-F-8785B, were directed toward bounding the values of various short period response parameters such as frequency, damping, time constants, etc., which pilots have felt are consistent with the precision and control needed during maneuvering flight. Many of these parameters which are easily defined in terms of the basic aircraft dynamics are often masked by the forced response of multiloop high gain control systems. The newer control performance criteria express short period response in terms of a response envelope in the time domain. These criteria are applicable to both the high gain multiloop controlled aircraft response and the basic aircraft response, and are a supplement to the traditional forms of control performance criteria.

The new criteria used in the SFCS program consist of three basic time history performance criteria with boundaries on both the basic parameter and the time rate of change of that parameter. These parameters are a normalized blend of pitch rate and normal acceleration (C^*) for the pitch axis, a normalized roll rate (P_N) for the roll axis, and a normalized blend of lateral acceleration and sideslip (D^*) for the directional axis. A C^* criterion has been available for some time as documented in Reference 5. The definition of the C^* expression in equation form as used in the SFCS program is:

$$C^* = \Delta n_{z_p} + K_2 q$$

Δn_{z_p} = Incremental normal load factor at pilot station (g's)

q = Pitch rate (rad/sec)

K_2 = $C_2 V_{c_o}$ = Pitch rate gain constant

C_2 = Dimensional constant $(1/32.2 \frac{g's - sec^2}{ft})$

V_{c_o} = Crossover velocity (assume 400 ft/sec)

The Δn_{z_p} is the total incremental load factor at the pilot station and includes normal effective acceleration at the aircraft center of gravity (c.g.) and the additional normal acceleration at the pilot station due to pitch acceleration ($\dot{q}$), multiplied by the moment arm from the vehicle c.g. to the pilot station. The studies reported in Reference 2 proposed modifications to the C^* boundary presented in Reference 5. Figure 1 shows the proposed normalized C^* envelope and a $\dot{C}^*$, or normalized rate of change of C^*, envelope. The $\dot{C}^*$ envelope is required to control higher order effects such as low damped low amplitude oscillations which could be accommodated by the C^* envelope but still be undesirable.

The P_N response envelope is shown in Figure 2. Included also is a roll acceleration ($\dot{P}_N$) response envelope. These envelopes restrict the roll mode response time constant, overshoot and oscillations.

The D^* criteria define the transient response characteristics in the directional axis due to a lateral step command input from the pilot. The D^* expression combines sideslip, which is considered the principal low speed handling quality parameter, and lateral acceleration, which is a more important consideration during high speed flight. The definition of the D^* expression in equation form as used in the SFCS program is:

$$D^* = \Delta n_{y_p} + K_3 \beta$$

$$D_1^* = D^*/K_3 = \beta + \Delta n_{y_p}/K_3$$

Δn_{y_p} = Incremental lateral load factor at pilot station (g's)

β = Sideslip angle (rad)

K_3 = $C_3 q_{co}$ = Slideslip gain constant

C_3 = Dimensional constant $(-9.91 \times 10^{-3} \frac{g's - ft^2}{lb})$

q_{co} = Crossover Dynamic Pressure (assume 350 lb/ft^2)

The Δn_{y_p} is the total incremental lateral acceleration at the pilot station. It includes lateral acceleration at the aircraft c.g. and the additional lateral acceleration at the pilot station due to roll acceleration

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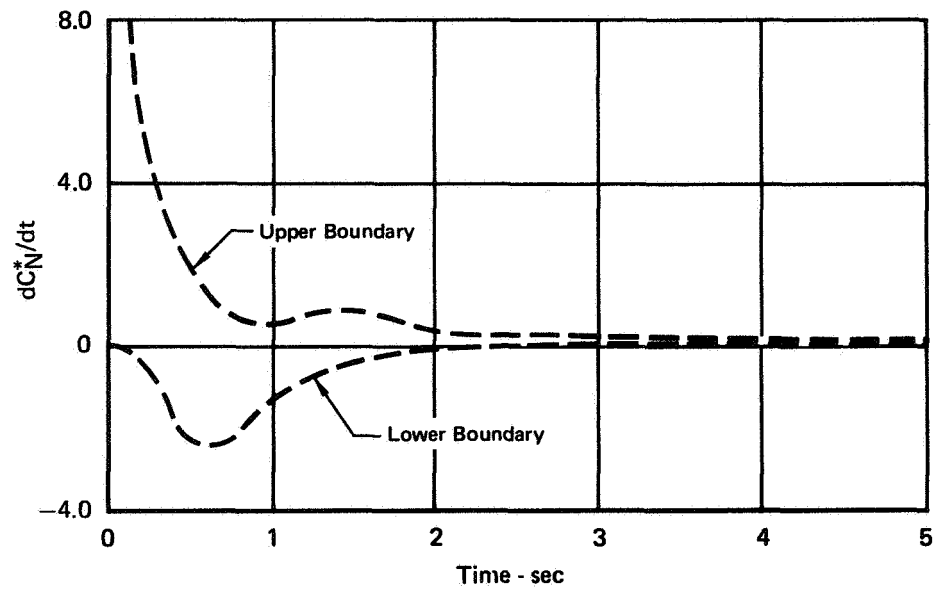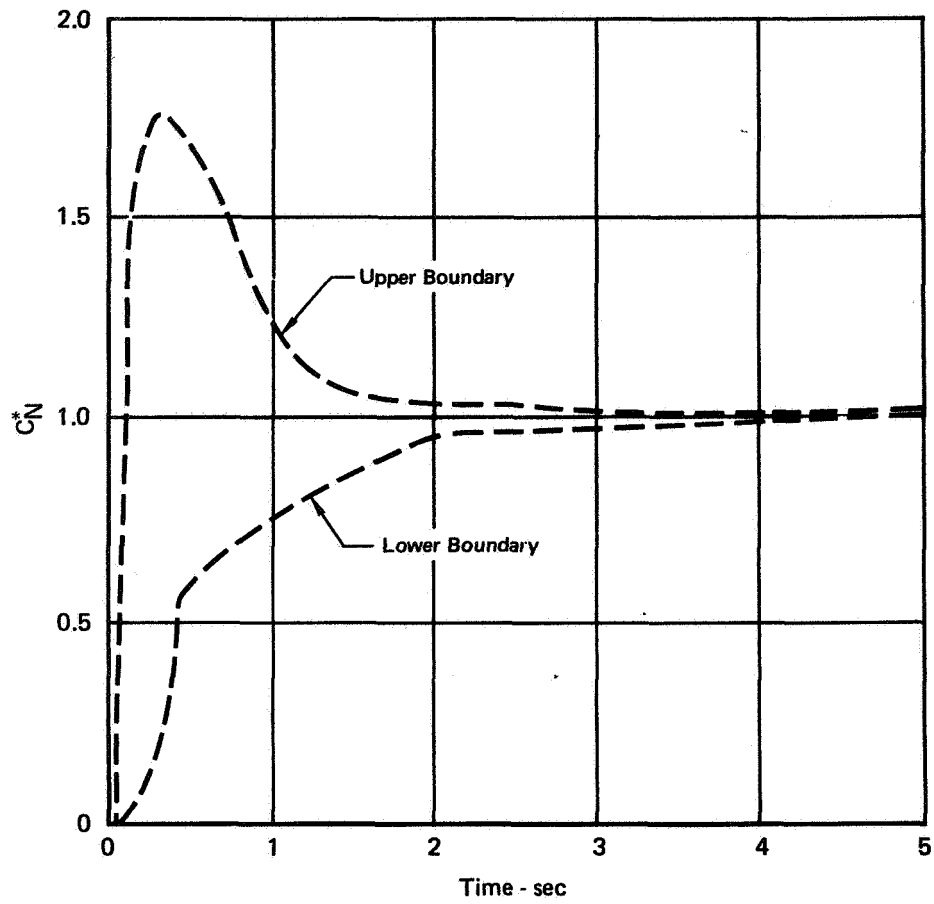
FIGURE 1
SFCS PITCH AXIS TIME HISTORY CRITERION
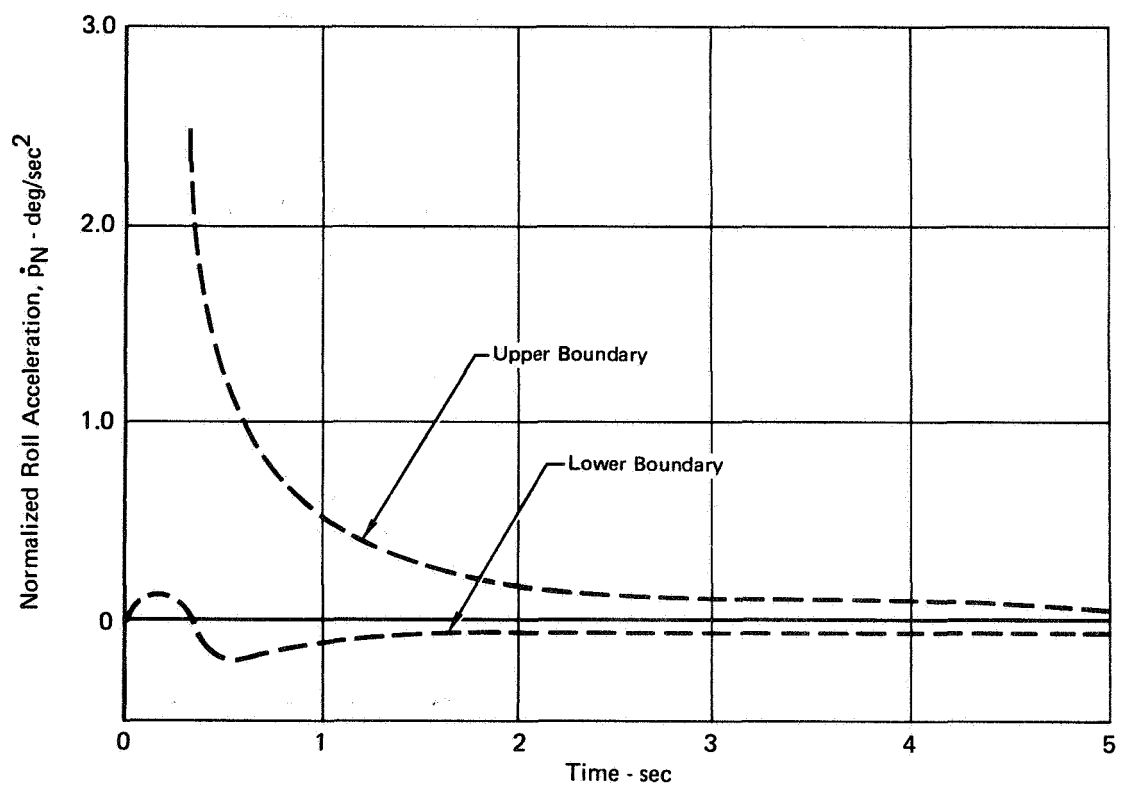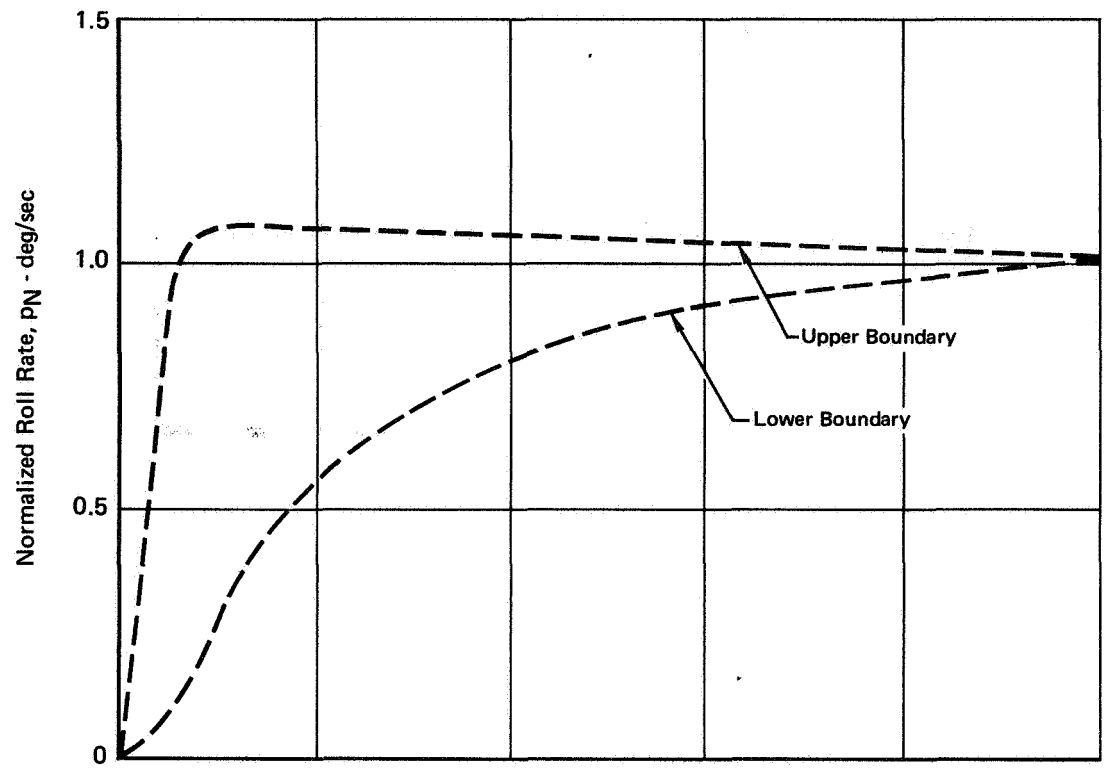
FIGURE 2
SFCS ROLL AXIS TIME HISTORY CRITERION

$(\dot{p})$ and yaw acceleration $(\dot{r})$ multiplied by the respective moment arms. The roll moment arm is the distance from the roll axis to the pilot station. The yaw moment arm is the distance from the aircraft c.g. to the pilot station.

In contrast to the C^* concept where the equivalent gain constant is a function of velocity, the D^* equation employs a crossover dynamic pressure to establish when low and high speed flying qualities are rated equally. The D^* equation can be modified to yield an expression which is more in harmony with the traditional lateral-directional handling qualities specifications on sideslip excursion limitations. This expression, D_1^*, has the sideslip units of degrees or radians. Figure 3 shows the D_1^* and $\dot{D}_1^*$, the rate of change of D_1^*, boundaries used in the SFCS program. The boundaries are expressed in terms of the factor "K", where "K" is the ratio of "commanded roll performance" to "applicable roll performance requirement" as defined in MIL-F-8785B.

Having now established the performance criterion a six-degree-of-freedom, man-in-the-loop simulation program was conducted to evaluate the control law implementation. This simulation included the capability to maneuver the aircraft throughout the F-4 flight envelope including stall and post stall conditions. As a result of this simulation, several design modifications were identified, evaluated, and subsequently implemented into the SFCS design. A series of simulations were used during the development and test of the SFCS to assist not only in the design, but to verify equipment performance, train pilots, and correlate flight test data. This test program has shown the importance of compatibility testing with a manned simulator in preparing for a flight test program. Reference 1 describes completely the thorough simulation effort which resulted in savings of time, money, and most importantly accelerated progression to three-axis FBW control of the aircraft in the very early stages of the flight test program.

SYSTEM DESCRIPTION

The SFCS is a three-axis, analog, fly-by-wire, primary flight control system using secondary actuators to provide position commands to the surface actuators.

The system functions in a closed loop as a direct function of pilot applied inputs to command aircraft motion, instead of surface position. In addition to conventional controls, a sidestick controller (SSC) located on the pilot's right-hand console is included for SFCS control. The final configuration of the SFCS as mechanized had no mechanical control of control surface position in any of the three axes. In addition to a normal mode of operation which commands aircraft motion, electrical backup modes command surface positions in the event of malfunctions of the normal mode. A capability for reversion to a mechanical backup mode, provided in the pitch and yaw axes for the early phase of the flight testing, was removed following flight test validation of the SFCS modes and functions.

The normal mode shown in Figure 4 utilizes rate and acceleration feedbacks to control aircraft motion. The three levels of closed-loop gain for the pitch and yaw normal modes may be selected either by the pilot or automatically by the adaptive gain computer. This variable gain system provides

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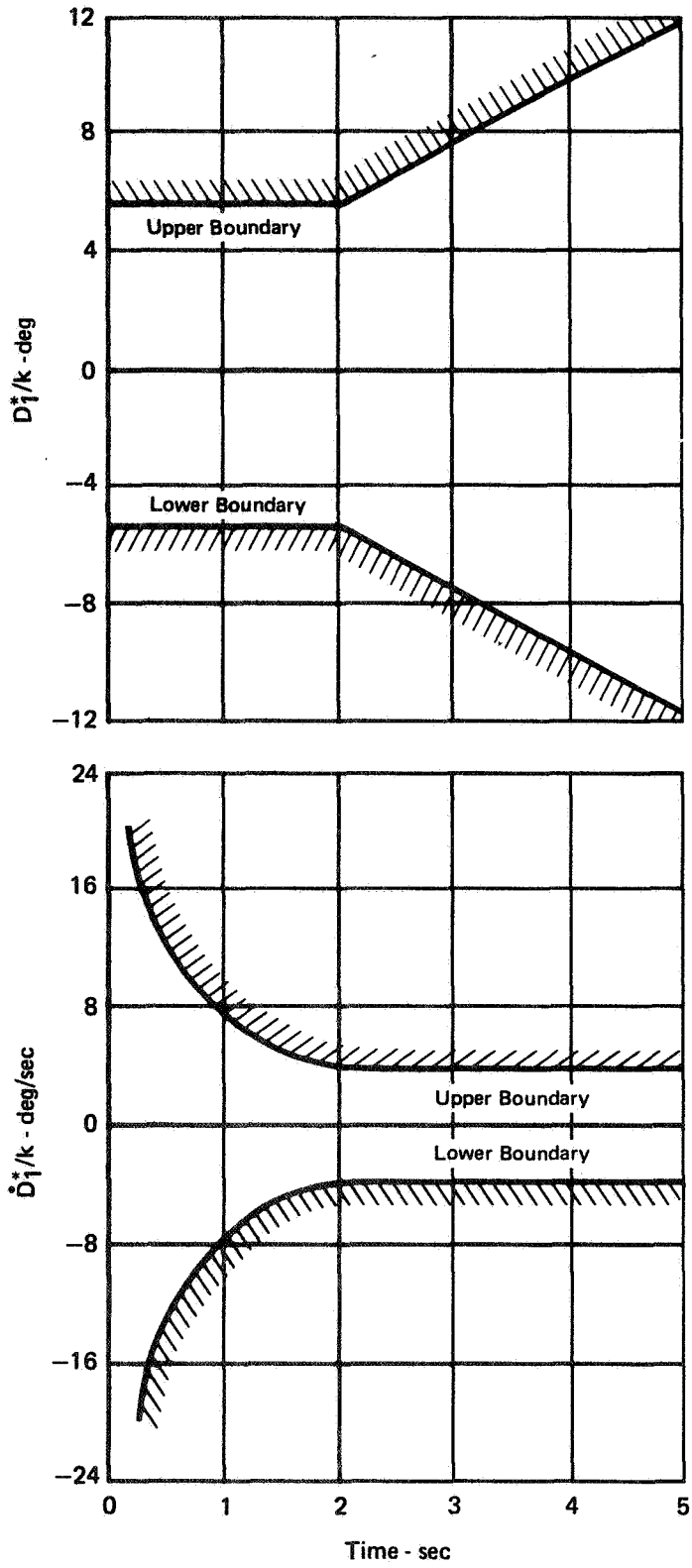
FIGURE 3
SFCS DIRECTIONAL TIME HISTORY CRITERIA
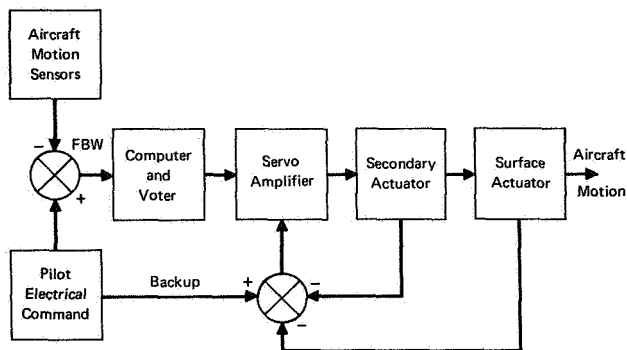
Figure 4 Survivable Flight Control System Mechanization

an almost unchanging aircraft dynamic response for varying flight conditions. The pitch axis normal mode provides a neutral speed stability (NSS) auto-trim function with the landing gear retracted. The auto-trim is provided by integration implemented in the forward loop. A stall warning function is provided through a blend of angle-of-attack and lagged pitch rate. Nose down pitch rates are rejected in the control law so that pilot push-recovery from a stall condition is not impeded. The stall warning function is to reduce the command gain in the longitudinal axis, effectively increasing stick force per g, and to remove the roll rate feedback from the roll axis; both changes occur linearly as the stall region is penetrated.

Quarduplex (four channel) redundancy is used in all system components to provide improved system reliability and to achieve a two-fail operate system. Four transformer rectifiers (TR), each shunted by a battery, are independent power supplies. Two engine-driven ac generators provide the primary source of power to the four TRs. In the event of a single generator failure, the remaining unit can power all four TRs. If both generators fail, the batteries can power the SFCS for approximately one hour of flight time, sufficient to allow for a return to base. Three hydraulic pressure sources are normally available in the F-4 aircraft; PC-1, PC-2 and utility. A fourth hydraulic system, required to maintain quadruplex redundancy for the test aircraft, was an auxiliary power unit containing an electric motor driven hydraulic pump.

The necessary computations for the three control axes are performed by four analog computer voter units (CVU), one for each of the four channels. The quadruplex electrical signals in each of the three axes are processed by signal selection devices, and the selected signals are applied to electro-hydraulic secondary actuators. Four secondary actuators provide mechanical inputs to the rudder, stabilator and the two aileron and spoilor surface actuators. Each secondary actuator (SA) is quadruplex in that four identical units are integrated side by side and their output rams are force-summed providing a single point command. The CVUs monitor the status of each SA and will shut down any individual unit which is determined to have a significantly different differential hydraulic pressure than the differential hydraulic pressures of the other units of that SA. Reset switches with integral status indicator lights are installed on the main instrument panel of the test aircraft to provide continuous system status information. Momentary or inadvertent failures can be reset using these switches.

An extensive ground Built-In-Test (BIT) capability is included in the SFCS. The system automatically tests the SFCS with several hundred separate functional tests and subsequently indicates a GO or NO GO condition to the pilot and ground crew. Most detected failures are automatically isolated to a specific Line Replaceable Unit (LRU). LRU failure indications are

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displayed on a maintenance test panel. In addition, the test number of any failed test is indicated to further help locate where in an LRU the failure occurred. The ground BIT requires approximately four minutes, and is positively deactivated during flight.

INSTRUMENTATION AND DATA ACQUISITION SYSTEM

The data acquisition system consisted of various instrumentation components located in the nose area of the test aircraft. The system included an Ampex AR 200 14 track magnetic tape recorder, PDM multicoders, proportional NBFM multiplexing equipment, power supplies, and signal conditioning electronics. An L-Band UHF telemetry system was located in the center fuselage upper equipment bay. Approximately 275 data measurands were recorded during the initial SFCS flights. Certain measurands such as component temperatures and multichannel SFCS performance monitoring were deleted from the instrumentation once adequate data had been accumulated.

FLIGHT TEST APPROACH

Flight testing of the SFCS was initiated on 29 April 1972 from the contractor's facility in St. Louis and, on 5 July 1972, transferred to their Edwards AFB facility for further flight envelope expansion. Flight testing was structured into four progressive phases.

The first phase consisted of 27 flights, providing 23 hours of FBW flight time, and used to develop and evaluate the FBW flight control for all three axes while retaining a mechanical backup (MBU) system for the pitch and directional axes. During this phase the flight envelope and maneuvering boundaries were progressively expanded to cover the normal operating flight regime of the F-4. Conventional flight test techniques were used to examine the longitudinal stability and control as well as the lateral directional mode characteristics. It must be re-emphasized here that the program objectives were to develop and demonstrate a functional SFCS, not to optimize such a system for a particular aircraft such as the F-4. For this reason a minimal amount of effort was expended in axes optimization. Data was taken to investigate the system-component's operational environment as well as the effects of simulated equipment failures on the SFCS. Testing of the MBU system was limited to only that which was required to assure aircraft controllability when reverting to this mode. When confidence in the functional operation of the FBW system had been established the MBU was removed and the second phase of testing initiated.

The second phase of testing required 19 flights, providing 18 hours of FBW flight time. This period was used to continue evaluation of the SFCS performance with use of the Normal/Adaptive gain mode of operation. Aircraft flying qualities for gross maneuvering and precision of flying with the center-stick and sidestick controller and vernier control were evaluated as well as simulated combat maneuvering, instrument flying, and various other mission oriented tasks.

The third phase of system evaluation was conducted by the AF Flight Test Center. A team of Air Force test pilots flew 15 flights with emphasis on mission-oriented tasks. Testing included evaluations of stability and control, clean and with external tanks, electrical back-up control, air-to-air and air-to-ground tracking, gross maneuvering, and precision flying. Detailed test results of this portion of testing have been documented in Reference 7.

The fourth phase consisted of a total of 21 flights used for system demonstration, training and technology transition. The 21 flights were made by thirteen Air Force, Marine, and NASA pilots. All participants received back seat flights and three demonstration pilots flew two flights each from the front seat. The flights were generally designed to demonstrate SFCS performance and functional features, supersonic and transonic handling characteristics and maneuvering and precision flying.

SUMMARY OF AIRCRAFT FLIGHT CHARACTERISTICS

Longitudinal Stability and Control

When operating in the Normal FBW mode, pitch control was generally improved over the basic F-4. The pitch axis was better damped than the F-4 yet the aircraft still had adequate short period response. Pitch short period damping ranged from dead-beat to slightly over-damped throughout the flight envelope. The SFCS reduced the tendency to couple with the short period motion. Stick centering was greatly improved compared to the F-4 resulting in better PA configuration speed stability stick force cues.

The Neutral Speed Stability (NSS) function enhanced the longitudinal control characteristics by providing automatic pitch trim to maintain 1 g flight throughout the flight envelope with landing gear up. NSS tends to reduce the pilot work load during maneuvers involving rapid airspeed or altitude changes since manual trimming is not required. Consequently, pitch control is improved as only the constant maneuvering stick forces are required. Effectiveness of the NSS was very obvious during the decelerating wind-up turn maneuver through the transonic area. The normal F-4 nose rise was not present and manual trimming was not required.

The centerstick maneuvering force gradients for the longitudinal SFCS are compared to the basic F-4 for several flight conditions as shown in Figure 4. The data substantiates pilot comments of improved maneuvering pitch control over the F-4. The SFCS provides a more comfortable stick force gradient and stick displacement throughout the flight envelope allowing more precise control of pitch rate and g. The improved pitch control at the high g values is also attributed to the overall linearity of the Fs/g gradient versus g. The SFCS stick force per g characteristics obtained from flight test data compare favorably with the predicted values determined from earlier studies, analyses and simulations.

Pitch MED gain was determined to be optimum for takeoff and landing with manual gain selected in the FBW Normal mode. Low gain was then selected at 275-300 knots after takeoff. Takeoff in FBW requires only a small aft stick force to obtain the stabilator position for rotation at liftoff. The application of aft stick forces greater than required for full stabilator can delay subsequent nose down stabilator response. Takeoff control in

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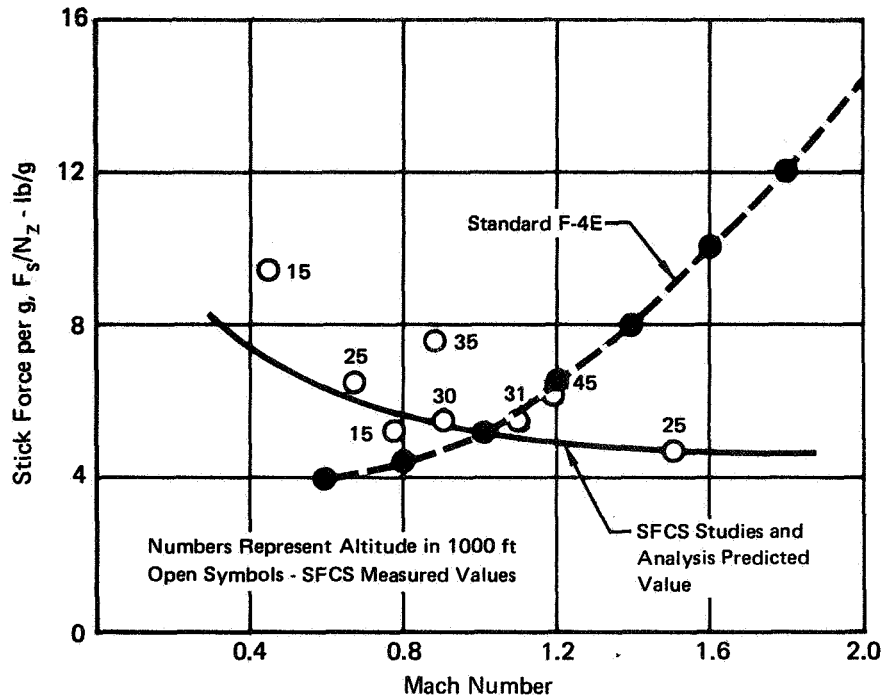
FIGURE 4
STICK FORCE PER G vs MACH NUMBER
Adaptive Gain - Normal Mode

Normal mode is good. Pitch control is excellent for landing in Normal mode. Touch and go landings exhibited superior handling qualities and the presence of ground effect was not detectable.

The aircraft with external wing tanks installed was also well damped and exhibited no noticeable change to in-flight characteristics as compared to the clean SFCS aircraft.

Additional differences were noted in longitudinal response characteristics between the SFCS and a production F-4 aircraft. The SFCS aircraft has a tendency to maintain 1 g during stall approaches, necessitating pilot action to push the nose down for recovery. The SFCS also attempts to hold zero pitch rate at the top of a loop and the nose must be pulled down to complete the maneuver. Pilots adapted readily to these differences which were not considered deficiencies.

Lateral Stability and Control

During initial flight tests, the lateral control system was reported to be oversensitive around neutral at airspeeds above 250 knots in the cruise configuration. The control produced a sharp, abrupt first motion which was quite objectionable; however, the response was not considered objectionable for the steady command inputs required for gross maneuvering. In an effort to reduce this first motion characteristic, the roll rate to lateral force gain was substantially reduced in the first 1/3 of stick force, the aileron over travel was eliminated and the spoiler deadband increased. This resulted in very sluggish roll response and was considered unsatisfactory.

Various flight tests were conducted in an attempt to define roll power for small inputs around neutral for application to analog and simulation studies. Higher than anticipated roll power for small deflections was identified as a major cause of the high lateral sensitivity. The SFCS Flight Simulator was utilized extensively to evaluate design change candidates which would improve the lateral mechanization. Results indicated that a three gradient roll rate command would provide the most desirable lateral characteristics.

Since the three gradient roll rate command modification would not be available until Phase two, an interim modification was incorporated to improve the lateral axis sensitivity characteristics. The modification retained the original two gradient command, but provided decreased cruise configuration sensitivity and increased PA configuration sensitivity. The interm modification, installed for Flights 23 through 27, was reported to be a significant improvement over the previous configurations.

The three gradient roll rate command modification was installed during the Second Phase layup and subsequently evaluated throughout the flight envelope. It was concluded that the modified roll command was an improvement over the two gradient shaping, but that the lateral control had not been completely optimized.

Lateral response was well damped at all flight conditions except for small amplitude oscillations at high $\bar{q}$ subsonic conditions and there is no tendency for control free divergence. Lateral response in the landing configuration is classified as good. In the clean configuration above 250 knots, the response is uncomfortably sharp and becomes objectionable in any tight control task such as tracking or formation flight. The problem was not one of roll rate attained but of high entry and recovery roll accelerations and was referred to by the pilots as "hard starts and stops".

The high control application rates possible with FBW technology permit reduction in the variation of roll time constant normally experienced as a function of flight condition. The design aim of a nearly constant roll rate time constant was achieved as shown in Figure 5. These data show that the SFCS roll time constant throughout much of the flight envelope closely matches that of the basic F-4 at high speed, low altitude flight. This fast roll response together with such other factors as stick torquing, linkage nonlinearities and high roll power at small surface deflections contributed to the high roll accelerations.

Roll rate to stick force ratios were also more uniform as shown in Figure 6. As illustrated in the flight test data, the variation in roll rate to stick force is substantially reduced as compared to the basic F-4 with yaw SAS. The three gradient roll command shaping provides higher ratios for larger commands to allow maximum roll rates to be obtained without excessive force.

Directional Stability and Control

Directional control was generally satisfactory throughout the flight envelope. Short period damping was essentially deadbeat to slightly overdamped. The roll-yaw crossfeed was initially weak in the PA configuration. The PA configuration roll-yaw crossfeed gains were modified with incorporation of the three-gradient roll command in the second phase. Roll-yaw crossfeed was improved for subsequent testing.

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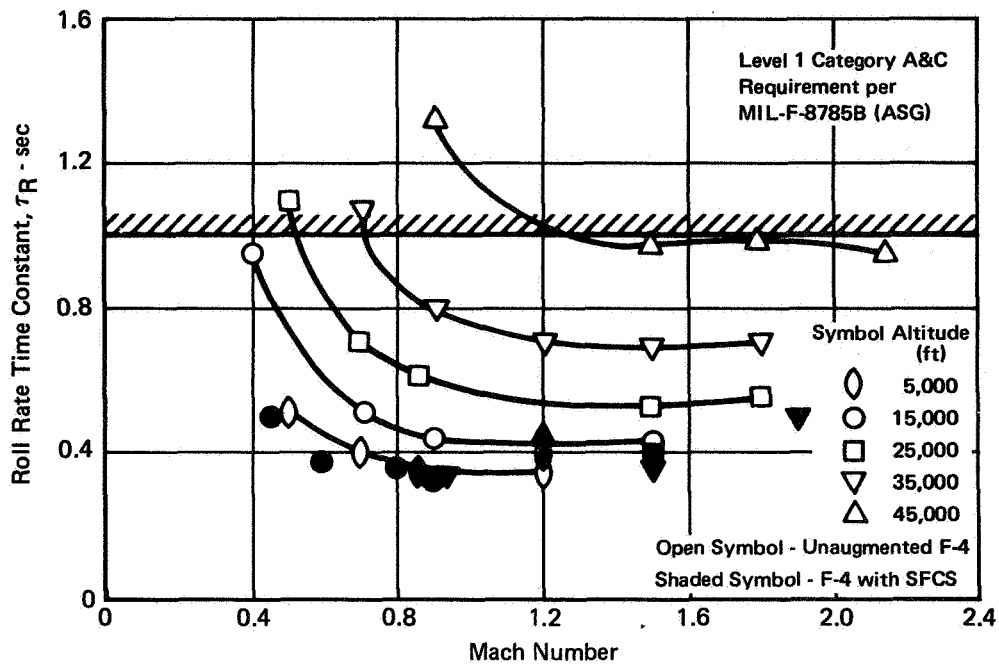
FIGURE 5
ROLL RATE TIME CONSTANT SFCS vs F-4
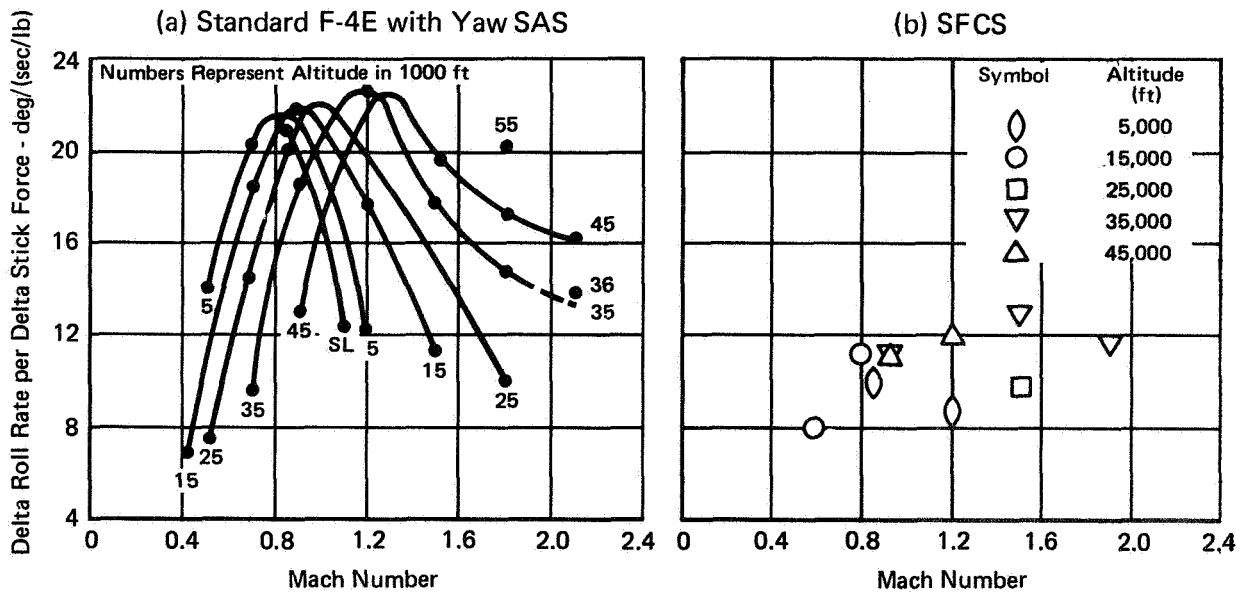
FIGURE 6
ROLL RATE PER STICK FORCE SFCS vs F-4

General Maneuvering

Various maneuvering tests were accomplished to evaluate FBW control for large command inputs. Maneuvering tasks included 360° rolls, Immelmann turns, transonic decelerating wind-up turns and 1/2 Cuban Eights. Overall, the maneuvers exhibited control charactersitics which were improved over the basic F-4 aircraft. A more positive control of load factor was apparent during evaluation of wind-up turns. The deceleration through the transonic region while holding load factor is significantly improved since the normal F-4 pitch transient is eliminated by the blended rate and acceleration feedback and by the NSS trim function. The full rolls are more uniform due to the use of roll rate feedback. The Immelmann turn was reported to be more comfortable with the FBW control due to the roll-to-yaw crossfeed effectiveness.

Air-to-Air and Air-to-Ground Tracking

Air-to-air and air-to-ground tracking tasks were evaluated by three MCAIR pilots, two USAF evaluation pilots, and three USAF demonstration pilots. Qualitative pilot comments varied slightly; however, certain characteristics were noted by all three MCAIR pilots. The CSC was generally preferred for tracking by a majority of the pilots. The sharp lateral response discussed previously was irritating with either controller for the tracking task. When using the CSC, the normal tendency to "tighten up" resulted in torquing the grip, producing jerky lateral commands. The SSC provided better lateral control in this respect. However, there was a tendency to overcontrol in pitch with the SSC when making small corrections. A learning curve of two or three flights was required before effective tracking ability was attained. Air-to-ground tracking was satisfactory with either controller. Mild sideslip excursions or drift were noted on occasions; however, the ground target was regained easily.

SUMMARY OF SFCS SYSTEM OPERATION

Adaptive Gain Changer

This function operated satisfactorily throughout the program but it is felt that the complication of the design due to its inclusion was excessive for benefits achieved. A less complicated device, using a highly reliable air data system, would probably be sufficient for most vehicles.

Stall Warning Computer

The stall warning function was activated for evaluation on Flight No. 24 and subsequent flights during the first phase of flight testing. Functional operation of stall warning was verified during 1 g stall approaches and wind-up turns. Pilots commented that the pitch stall warning is effective in wind-up turns where significant pilot commands are being applied. Its effectiveness is severely limited in situations where only small pitch commands are being applied. For instance, the NSS function during a 1 g deceleration can stall the aircraft with no pilot command and consequently

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no FBW stall warning stick force cues. The normal F-4 audio stall warning cues were retained to supply protection in this region. The environment of heavy wing rock was not explored to assess the total effectiveness of stall warning function in the lateral feedback loop. Flight data however, verified that the lateral function was operating satisfactorily.

Sidestick Controller

Although not optimized for this aircraft, the SSC provided an acceptable means of control for all tasks performed during the test program. It was inherently more sensitive to pilot inputs than the centerstick, but a relatively brief exposure was necessary for various pilots to become accustomed to it. The controller's mounting on the right console was not an optimum position for precision tasks such as landing. The input pivot was below the grip, and coordinated maneuvers were difficult to accomplish at high load factors.

Reliability and Maintainability

During 88.5 total program flying hours, only 5 equipment malfunctions were reported. Four of these failures were detected by BIT prior to flight. Only one non-resettable in-flight failure, a yaw rate gyro which does not effect safety of flight, occurred during the entire flight test program. The calculated probability of flight control failure, which is improved over the basic F-4, is 10.685×10^{-7}. This figure does not consider the improvement provided by EBU. Figures on maintenance manhours per flight hour also show improvement in the SFCS system when compared to the F-4 mechanical system.

In the last two months or 44 working days of flight testing the test aircraft flew 31 days, and in the last month of 23 working days the aircraft flew 21 days. Of a program total of 84 flights and 88.5 flight hours flown during 10 months, 41 flights and 42.8 flight hours were flown in the last two months. These last two months were without delays due to maintenance.

CONCLUSION

Future designs envisioned throughout the industry include such Configuration Controlled Vehicle features as Relaxed Static Stability, Direct Lift Control, Direct Side Force Control, Maneuver Load Control, etc., which may employ canards, movable tails and movable wing tips. These features provide significant maneuvering performance improvements and control techniques not possible with a mechanical control system. This makes mandatory the transition from a mechanical control system to FBW a basic necessity.

This program's flight testing has provided design criteria, reliability, cost and maintainability data, specification requirements, and most importantly, the confidence level required for installation of advanced flight control systems of this type in future aircraft.

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DEVELOPMENT OF AN ACTIVE FLY-BY-WIRE FLIGHT CONTROL SYSTEM

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SUMMARY

This paper presents a summary of the YF-16 flight control system. The basic functions of the flight control system are discussed, as well as the unique features such as Relaxed Static Longitudinal Stability (RSS), Fly-By-Wire (FBW), and Side-Stick Pilot's Controller (SSC). In addition, the basic philosophy behind the selection of the flight control system functions and unique features is discussed.

INTRODUCTION

The YF-16 is the first aircraft developed in which an Active Flight Control System was incorporated from its inception. In the past, the design of a flight control system was undertaken after the basic aircraft aerodynamic design was set and was used mainly to improve handling qualities. This usually involved little more than augmenting pitch and lateral-directional damping. As aircraft handling and performance requirements increased, so did the complexity of the flight control system. The desire to obtain uniform aircraft response to pilot commands results in command augmentation systems being used in the flight control system. Since these systems required large authority surface commands to achieve the desired response, the requirement for highly reliable electronic systems was generated and achieved. The achievement of this reliability has allowed the application of an Active Control System in the YF-16.

SYMBOLS

A.C.   aerodynamic center

An   normal acceleration

C_D	drag coefficient
C_L	lift coefficient
$L_{\alpha_{WB}}$	lift of the wing body due to angle of attack
L_{WBT}	total lift of the wing-body-tail
L_{α_T}	lift of the tail due to angle of attack
L_{δ_T}	lift of the tail due to deflection
LH	left-hand
$M < 1$	Mach less than one
$M > 1$	Mach greater than one
MAC	mean aerodynamic chord
P_T	total pressure
P_S	static pressure
RH	right-hand
RSS	relaxed static longitudinal stability
SM	static margin
T.E.	trailing edge
W	weight
α	angle of attack
β	sideslip angle
$\dot{\theta}$	pitch rate
δ_e, δ_H	horizontal tail deflection

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DISCUSSION

The design of flight control systems has evolved from purely mechanical to active over the past two decades, as depicted in Figure 1. The advent of high-performance airplanes in the mid-1950's that were required to operate over larger performance envelopes necessitated the development of three-axis electronic stability augmentation systems. Originally, the B-58 utilized single-branch electronics in its three-axis augmentation system. The following generation of airplanes, e.g., the F-111, employed triple-redundant electronics in stability and command augmentation system due to the larger authority requirements. However, pilot mechanical controls were retained so that the aircraft could be flown safely in the event of electronic failures.

Limited FBW functions were incorporated into control system such as the spoilers, terrain following radar capability and low speed trim compensator on the F-111. In addition, several specialized airplane research and test programs have used dual, triple and quadruple redundant electronics in their control systems. These include the F-4 SFCS, C-141, NASA F-8, and TWeaD programs. Since only single-failure protection is provided with triple-redundant electronic systems, an active control system must employ quadruple-redundant electronics to provide the two-failure protection that is required. The development of a quadruple-redundant system has been a straightforward and low-risk extension of the 10 years of highly successful triple-redundant electronic application experience on the F-111 program and the quadruple-redundant experience gained during the F-4 SFCS program.

The YF-16 Control System

The functions of the YF-16 flight control system are very similar to those of most other new high performance aircraft. The basic functions of the flight control system that are common are air data scheduled gains, stability augmentation (dynamic), interconnects between roll and yaw axis and command augmentation. The unique features and functions of the flight control system are static longitudinal stability augmentation (RSS), minimum displacement side-stick controller (SSC), total Fly-By-Wire implementation (FBW) and angle-of-attack and normal acceleration limiting.

Why Relaxed Static Stability

For the primary design mission of the YF-16 - air superiority - the importance of maneuverability and range results in the RSS concept providing sufficient benefits to justify its incorporation. The basic RSS concept can be stated in a very simple way:

1. Balance the airplane for optimum performance

2. Rely on the flight control system to provide the desired level of static stability as well as dynamic characteristics.

Illustrations of the differences between a conventionally-balanced airplane and an airplane with relaxed static stability are given in Figures 2 and 3.

In the subsonic flight regime (Figure 2) the conventionally-balanced airplane is shown to have its wing-body lift acting forward of the center of gravity and the total lift acting aft of the center of gravity. Since in a stable system the moment produced by the wing-body lift as a function of angle of attack must be less than that produced by the tail, the tail must be deflected in a direction to reduce the total tail lift in order to trim the system. Therefore, the total trimmed lift available at a given angle of attack is reduced for a conventionally-balanced aircraft. The RSS-balanced aircraft has both the wing-body and the total lift acting forward of the center of gravity. In this case the moment produced by the wing-body lift as a function of angle of attack is greater than that produced by the tail and the tail must be deflected in a direction to increase the total tail lift in order to trim the system. Therefore, the total trimmed lift available at a given angle of attack is increased for an RSS configuration

In Figure 3, the same information is shown for a supersonic flight condition. In this case, both the conventionally-balanced and RSS airplanes have both the wing-body and total lift acting aft of the center of gravity. Because the RSS airplane has a farther aft center of gravity than the conventionally-balanced airplane, the down load on the tail required to trim the system is much smaller. Therefore, the RSS aircraft has a higher total lift available than a conventional balanced aircraft at the same angle of attack.

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Now what this all means is improved maneuverability and range. Representative trim requirements for the conventionally-balanced and RSS configurations are shown in Figure 4 for both subsonic and supersonic Mach numbers. The benefits that are obvious from this illustration are: (1) higher trimmable lift coefficient, and (2) lower trim deflections with attendant drag reduction and lower tail loads.

The trimmed drag polars shown in Figure 5 are illustrative of the trim drag reduction attributable to the RSS balance. The reduced trim drag results in higher sustained load factors and increased range. Note that the benefits are most pronounced at the higher lift coefficients, which is an extremely important region for the YF-16. A secondary benefit of the RSS balance is a somewhat reduced weight because of reduced tail loads.

Why-Fly-By-Wire

The decision to employ the CCV concept of relaxed static stability (RSS) for the YF-16 brought with it the responsibility for providing a reliable, full-time-operating, three-axis stability and command augmentation system. Since a reliable stability and command augmentation system is required, adequate electronic redundancy is necessary to fulfill this requirement. Therefore, the decision to be made is whether pilot commands should be transmitted via mechanical components (linkage, bellcranks, etc.) or electrical signal paths. If mechanical components are chosen, electrical components are still involved to implement the command augmentation system. It follows then that the retention of mechanical components for transmission of pilot stick commands is unjustifiable, since an unstable airplane cannot be controlled in flight without the benefit of a full-time-operating stability and command augmentation system. Therefore, fly-by-wire (FBW) is a natural outgrowth of a redundant electronic control system required for an augmentation system in an unstable (i.e., RSS) airplane.

An active control system offers four benefits which the YF-16 airplane enjoys: (1) precision control and optimum response; (2) design flexibility, offering growth capability and easy acceptance of design changes; (3) improvements in a maintainability and survivability as a result of simplified equipment installations; and (4) improved airplane performance, since the introduction of CCV concepts is compatible with FBW.

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How The Flight Control System Basically Works

The YF-16 quadruple-redundant system employs four independent signal branches, i.e., each input signal source (pilot, inertial sensors, etc.) originates as four signals, designated Branches A, B, C, and D. This redundancy concept is depicted for the pitch axis only in Figure 6. Each of the four branches are processed independently in the Flight Control Computer. This computer contains various functions which modify input signals from each of the three control axes, e.g., control dynamics, structural filters, gain-scheduling, selectors, power monitors, and various interconnecting electronic circuitry between the three control axes. Once the input signals have been gain-adjusted, filtered, and amplified, the resulting output signals are sent to each of the five large-authority, high-response, command servos. Each servo, in turn, drives its respective surface power actuator, as shown in Figure 6. The basic location of the hardware components of the flight control system is shown in Figure 7.

Flight path control is achieved through the actuation of an all-movable, differential horizontal tail for pitch and roll control, wing-mounted flaperons for roll control, and a conventional rudder for yaw control. Maneuver capability at high angles of attack is enhanced by automatic positioning of the full-span leading edge flap.

Important Design Considerations

The decision to employ an active control system in lieu of a conventional control system required the addressing of several important design considerations peculiar to these systems. These include: electronic circuit failure monitoring, electrical power failures, engine failures, command servos, surface actuators, and branch separation.

When employing redundant electronic systems, consideration must be given to the problem of proper signal selection and failure monitoring. The F-111 airplane utilizes triple-redundant electronics with middle-value signal selection. With more than 350,000 aircraft flight hours, there has been only one known dual electronic failure experienced. (The pilot landed the airplane without incident). With reliance on demonstrated operational service, the YF-16, quadruple-redundant system likewise utilizes middle-value signal selection on the processed input commands

(which result from the four separate electronic branches) that are ready for outputs to the command servos. To illustrate, signal Branches, A, B, and C are compared. The middle value is selected and then quadrupled so that four identical signals are available as output commands. If, for example, signal Branch B varies a predetermined amount from the other two, then Branch D is substituted instantaneously for B. If one of these three subsequently fails, say A, then the minimum value signal of C or D is chosen. By using this type of failure monitoring and signal selection, the control system is protected against dual failures.

The system is fully protected against power losses. Multiple electrical power sources are provided by an engine gear box-driven generator, a standby hydraulically-driven generator, and from multiple battery power as a last source. The standby generator, hydraulically driven by either the engine or emergency power unit (EPU), is automatically activated in the event of improper generator voltage or frequency. If both generators are lost, the batteries provide approximately 10 minutes of power. The end result is that the system receives uninterrupted regulated power with automatic or manual power switching capability. In addition to the above normal electrical protection, further protection relative to engine failure is provided by the EPU which automatically protects against low hydraulic system pressure.

Another consideration which is absolutely essential to the successful operation of an active control system is the conversion of electrical command signals to mechanical signals for commanding each surface power actuator. Each control surface is powered by a tandem valve-on-ram power actuator. In conventional airplanes, pilot stick and pedal inputs are summed mechanically with trim actuator and damper (stability-augmentation) servo inputs to command each power actuator's valve through conventional linkage. In the YF-16 active control system, the inputs are summed electrically and fed to a command (secondary) servo which provides a mechanical input to a power actuator's valve through a very short linkage run, as indicated in Figures 6 and 7.

Why Side Stick Controller

When the decision was made to adopt the fly-by-wire feature of the control system, the door was opened for simple implementation of any one of a number of new pilot-controller concepts. Should the control stick be retained in the conventional center

location or would it be more effective on the side? Should it be a displacement stick or a force-sensing stick? With these questions in mind, several studies and research programs were undertaken to determine the best solution for the YF-16.

After researching SSC installations that had previously been tested on such aircraft as the B-47, B-26, B-58, F-4, F-8, F-104, F-105, F-106, A-4, A-6, A-7, X-15 and others, General Dynamics built a flight control simulator to check out ideas and designs. A number of center-stick and side-stick hand controller designs were evaluated in a flight control simulator. Included in these were finger-type controllers, palm controllers, conventional grips with unconventional axes of rotation, and force-sensing controllers with both low and high feel-forces. The studies and evaluations showed that the force-sensing, side-stick controller was superior to all of the other approaches, including displacement and force-sensing center sticks and displacement-type side sticks.

The most widely recognized advantages of the force-sensing side-stick controller are: (1) improved high g tracking (based on results from the NASA Langley dual-mode simulator and the NASA bly-by-wire F-8 aircraft), (2) improved access to the instrument panel and increased panel area, (3) ease of implementation of pilot inputs in the computer (electrical signals proportional to stick force), and (4) pitch and roll axes better oriented to the pilot's arm and shoulder muscles.

The fly-by-wire aspect of the flight control system is particularly compatible with a force-sensing controller. Advantages of this combination include: (1) no linkage dynamics or friction felt at the controller, (2) no linkage balancing problems, (3) enhanced system survivability, (4) greater freedom in airframe design (including ease of change), and (5) potential for weight and cost reduction.

The pilot's controller shown in Figure 8 is a force-sensing (minimum deflection), side stick, mounted on and extending above the right-hand console. The location was developed to ensure easy access for the 5th through 95th percentile pilot. An adjustable arm support is provided to enhance pilot control. The arm support adjustments are vertical, fore and aft, and tilt. The force-sensing element, which contains quadrex transducers in both the pitch and roll axes is identical to the stick-sensing unit employed in the A-7 aircraft, except for the level of redundancy

since there is also mechanical linkage. The sensing element has been adapted to an F-111 grip.

The pilot introduces pitch and roll commands by applying appropriate forces to the stick. The forces imparted to the stick by the pilot cause electrical signals to be produced by the transducers located in the lower portion of the stick; these signals are input to the flight control computer. The trim button on the top of the stick grip allows the convenient and conventional input of pitch and roll trim commands. Other stick grip switches are provided to control elements of the armament system, nose-wheel steering, and aerial refueling.

Why Angle-of-Attack and Normal Acceleration Limiting

Since by definition an air superiority aircraft is highly maneuverable over its entire operting envelope, there are areas in which it is easy to obtain large values of angle-of-attack or normal acceleration. There are several ways that the pilot can be protected against such occurrences rather than requiring him to spend his time looking at cockpit instruments. One of these ways is to build in the required protection during aircraft design by putting on large enough aerodynamic surfaces (i.e., big vertical tail) and enough structural weight to assure that the pilot cannot spin or break the aircraft, no matter what he does with the stick. As you might surmise, this approach would severely penalize the aircraft's basic performance from a weight and drag standpoint.

Another method to protect the pilot is to build in enough aerodynamic resistance to stall throughout the usable angle-of-attack range and enough structural weight to obtain the required "g" plus a 1.5 safety factor and depend on the pilot to keep the aircraft within limits. The third method is to use the flight control system to limit angle-of-attack and normal acceleration which results in the lightest, best performing aircraft, but a very complex control system.

For the YF-16 we chose to use a combination of methods two and three which resulted in an aircraft with excellent performance characteristics with a minimum of complication in the flight control system. Using the above approach, i.e., minimum size surfaces and structural weight combined with angle-of-attack and normal acceleration limiting, has resulted in a high performance

fighter type aircraft which the pilot may truly maneuver with
"Complete Abandon."

YF-16 Flight Test Status

Thirty-one flights have been made by YF-16 No. 1 between
2 February and 13 April 1974 accruing 33:45 total flight time
with 1:39 being supersonic. Six pilots (2 contractor, 2 AFFTC
and 2 TAC) have flown to date with USAF pilots making their first
flights on flight Nos. 4, 12, 16 and 28.

Pilot acceptance of the advanced technology items, such as
side stick control with force inputs, fly-by-wire flight controls
with relaxed longitudinal aerodynamic stability and maneuvering
leading edge flaps, has been enthusiastic. Typical comments are
"performance and agility exceptional, easily and precisely con-
trollable, impressive roll response with almost immediate stop at
release of stick, comfortable and enjoyable to fly immediately,
no difficulty experienced in adapting to the side stick controller."

Confidence in the redundant active control system had
been so firmly established during simulation, ground tests and
checkouts, that all flights (including takeoff and landing) have
been made in a statically unstable configuration with the normal
c.g. for all flights to date being 36½% MAC (aircraft aerodynami-
cally unstable in pitch at subsonic and transonic conditions).

Some of the significant items demonstrated to date include:

1. Level flight acceleration to Mach numbers in
   excess of 1.6
2. Wind-up turns to 7+ g's at subsonic and
   supersonic speeds
3. Flight to angles of attack of 22° at low sub-
   sonic speeds and 18° at high subsonic speeds,
   and 9° sideslip.

Conclusions and Remarks

Although the YF-16 flight control system represents another
in a long line of advanced control system concepts, its implemen-
tation has been accomplished using current state of the art tech-
niques and hardware. The reliability of the hardware to date has

been exceptional as well as the pilot's acceptance of the system. The flying qualities and performance of the flight control system have been outstanding and we feel have provided the Air Force with an outstanding air superiority fighter prototype.
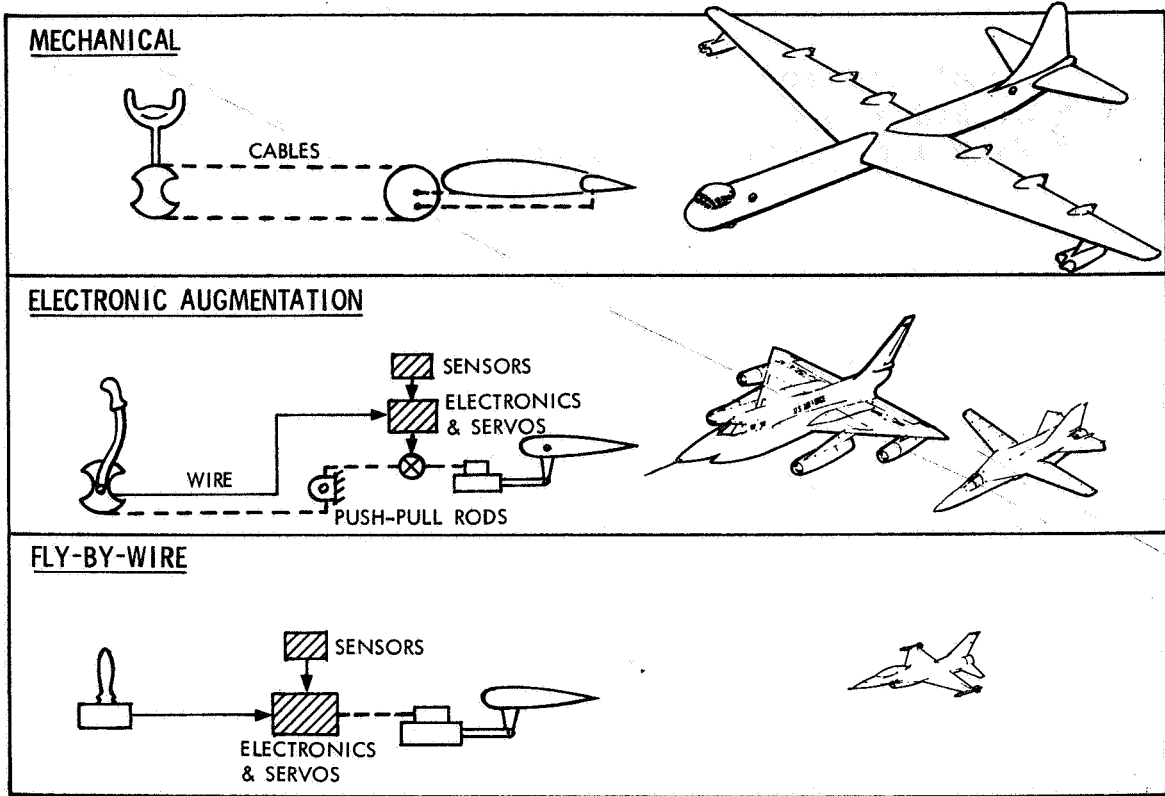
Figure 1   FLIGHT CONTROL SYSTEM EVOLUTION
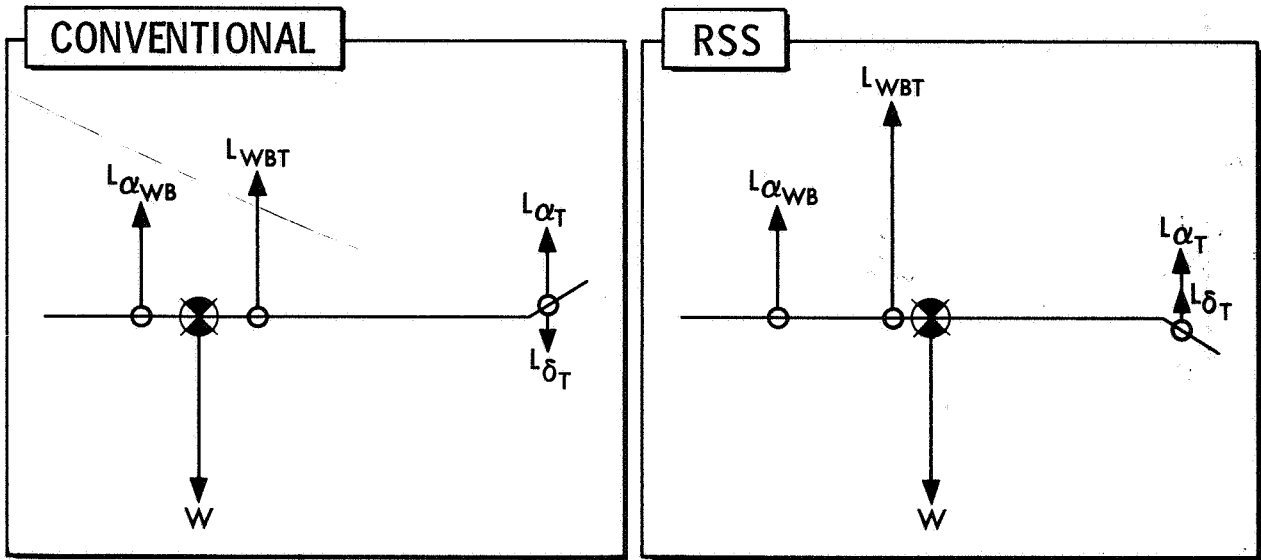
Figure 2   SUBSONIC BALANCE COMPARISON
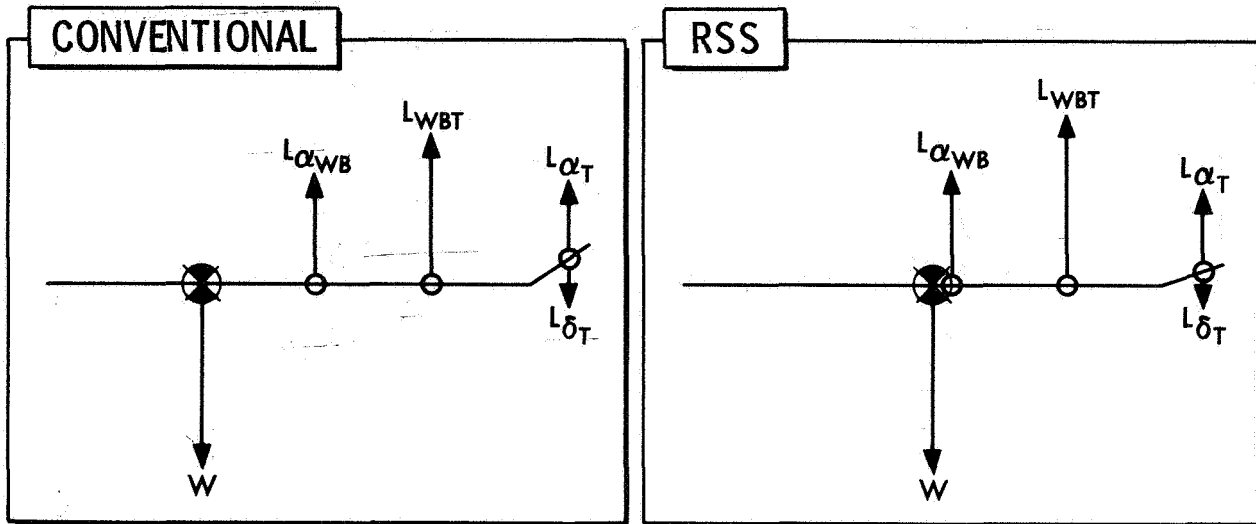
Figure 3   SUPERSONIC BALANCE COMPARISON
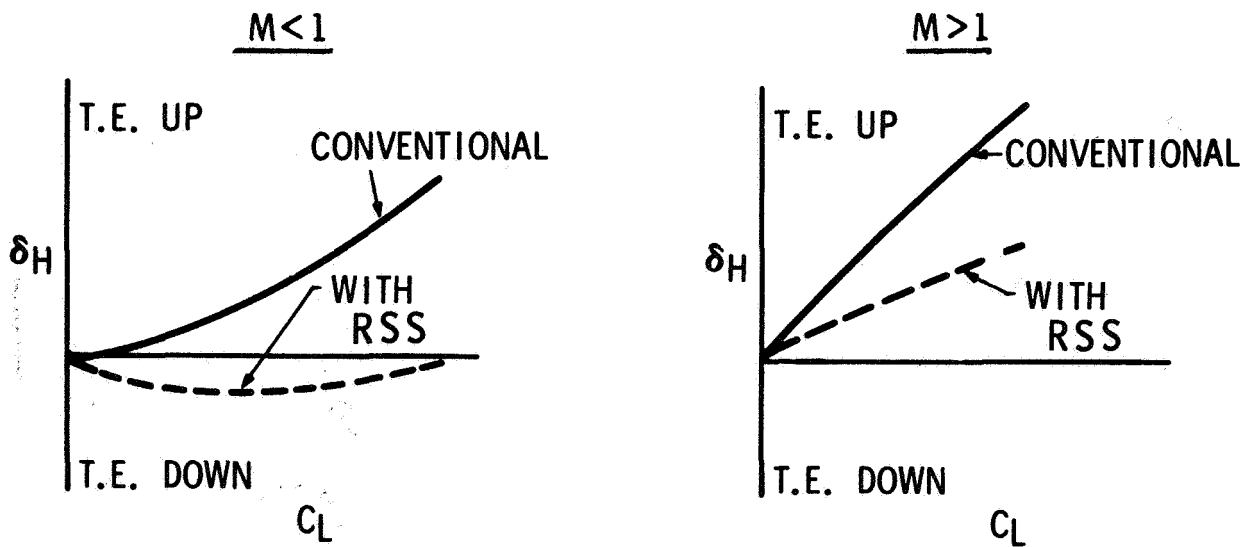
Figure 4   REDUCED TRIM REQUIREMENTS
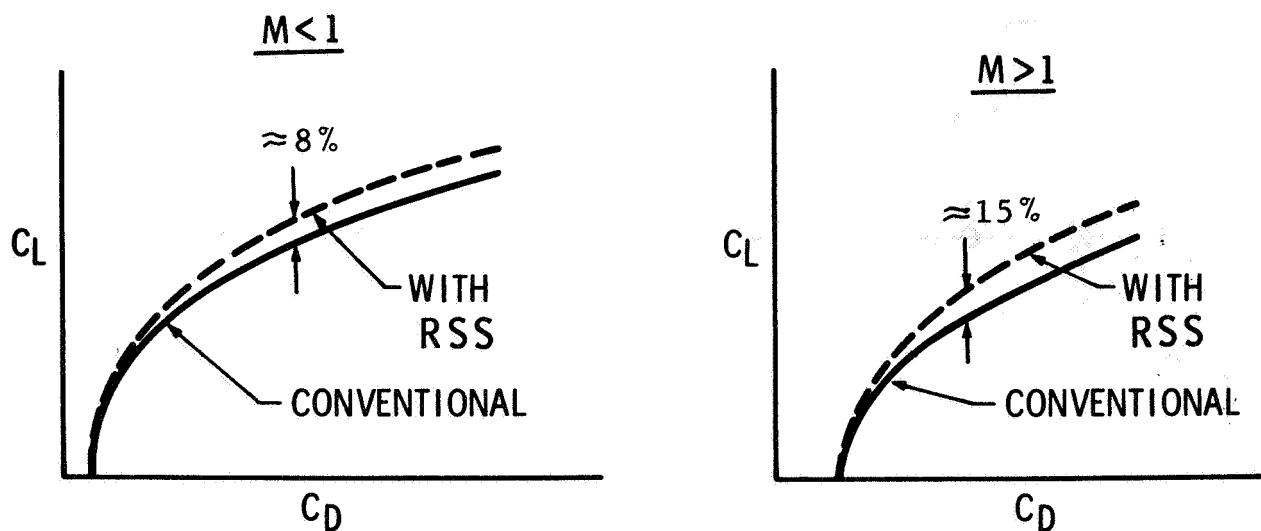
Figure 5   MANEUVERABILITY IMPROVEMENT
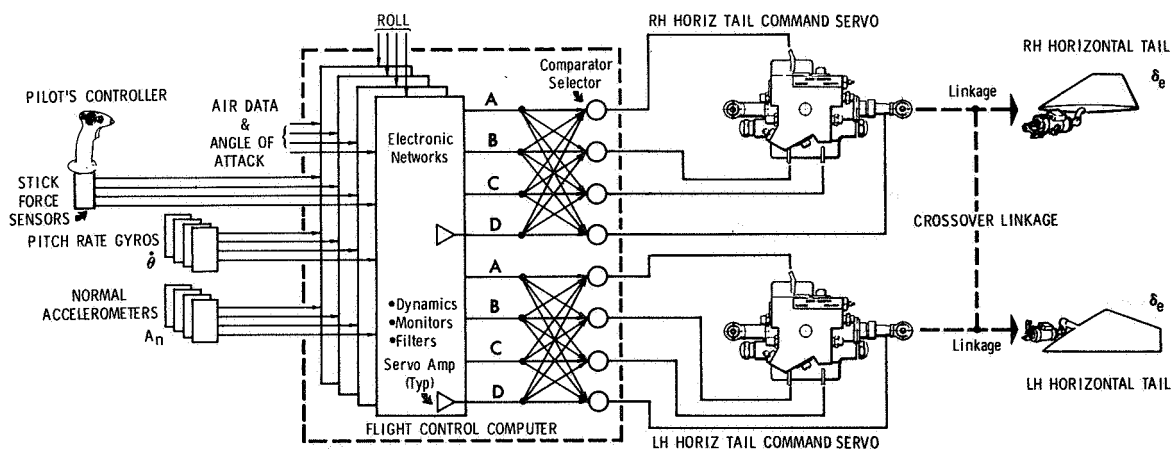
Figure 6   PITCH AXIS REDUNDANCY CONCEPT
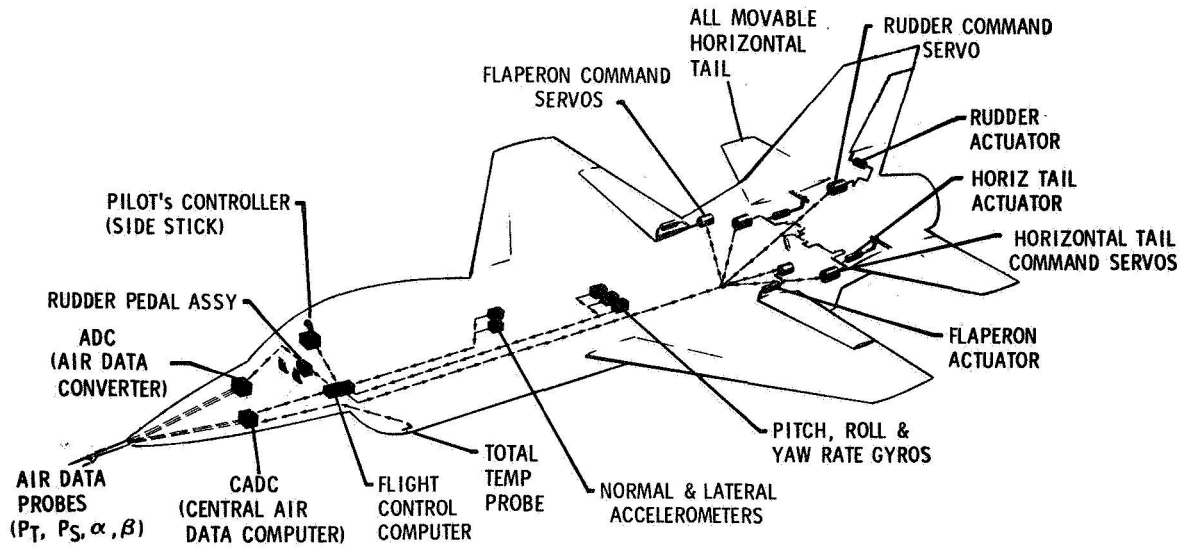
Figure 7 FLY-BY-WIRE FLIGHT CONTROLS
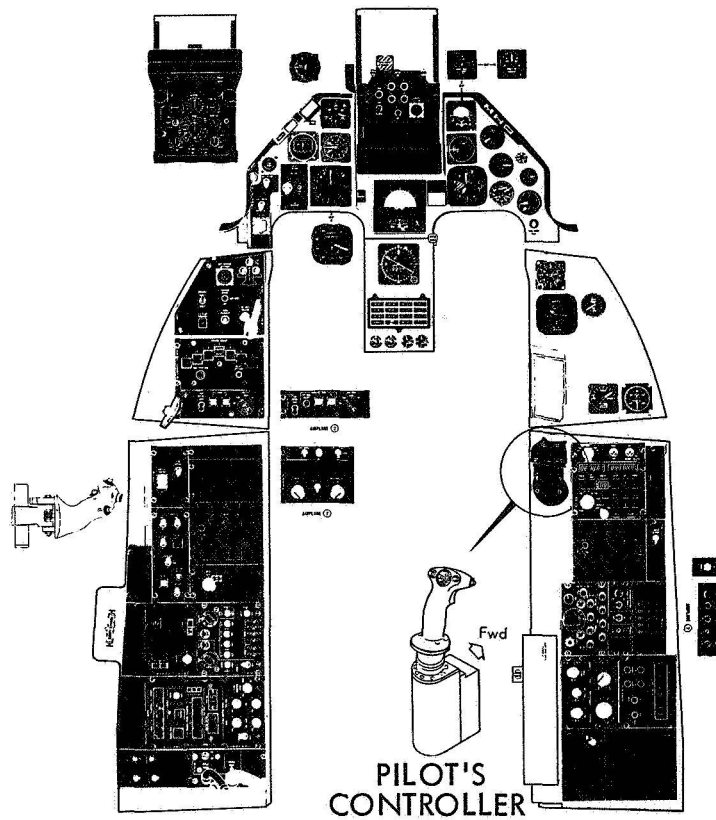
Figure 8 CREW STATION ARRANGEMENT

DIGITAL AND FLY-BY-WIRE SYSTEMS ON THE

YF-17 LIGHTWEIGHT FIGHTER

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ABSTRACT

The maneuvering flap control system on the YF-17, providing active control of lift-to-drag ratio and complementing longitudinal and lateral directional stability, is a digital fly-by-wire system. The roll control and directional interconnect system are fly-by-wire systems, digitally scheduled. Since the basic control system on the airplane is a mechanical/hydraulic system, it has been possible to use digital, fly-by-wire, and control configured vehicle technology in a permissive environment. This paper will discuss the design concept, the design and analysis involved in the mechanization, and the ground and flight testing of each of the systems.

B-52 CONTROL CONFIGURED VEHICLES:

FLIGHT TEST RESULTS

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SUMMARY

This paper summarizes recently completed B-52 Control Configured Vehicles (CCV) flight testing, and compares results to analytical predictions. Results are presented for five CCV system concepts: Ride Control, Maneuver Load Control, Flutter Mode Control, Augmented Stability, and Fatigue Reduction. Test results confirm analytical predictions and show that CCV system concepts achieve performance goals when operated individually or collectively.

INTRODUCTION

In July 1971 the Air Force Flight Dynamics Laboratory (AFFDL) initiated the B-52 phase of the Control Configured Vehicles (CCV) program in conjunction with The Boeing Company, Wichita Division. The program objective was to validate that the CCV concept is operationally practical and results in significant performance benefits on large flexible aircraft. The program was conducted under Contract F33615-71-C-1926 and included analysis, development, and flight validation of four new CCV system concepts. The systems developed were: Ride Control (RC), Flutter Mode Control (FMC), Maneuver Load Control (MLC) and Augmented Stability (AS). The Fatigue Reduction (FR) system, developed during the Load Alleviation and Mode Stabilization (LAMS) Program, Reference 1, was also evaluated during flight testing to validate compatibility with the four new CCV systems.

The Air Force participated in the performance of the program by conducting the analysis and development of the Ride Control concept at the AFFDL Advanced Development Project Office.

This paper summarizes the flight test portion of the program. The CCV tests, completed in November 1973, validate for the first time the CCV system performance and compatibility of multiple CCV systems. Actual benefits obtained by flight test are compared to the analytical predictions, thereby validating both the system performance and the analytical design techniques.

FLIGHT TEST SCOPE

Flight testing was conducted in two time periods. The Ride Control system was tested from 8 January through 9 February 1973. The remaining CCV systems were tested between 18 July and 11 November 1973. A total of 35 flights were flown, comprising 122 flight hours.

System Performance Goals

The CCV System performance goals outlined below were validated during the flight test program:

- A 30 percent reduction in vertical and lateral RMS acceleration in turbulence with a Ride Control system

- Meet MIL-A-8870 flutter damping criteria (g = .03) at 10 knots above the basic airplane flutter speed with a Flutter Mode Control system

- Reduce wing root bending moments during maneuvers with a Maneuver Load Control system by 8.2×10^6 inch-pounds, which is equivalent to a 10 percent reduction in maximum design load

- Provide acceptable flying qualities at a flight condition with neutral static stability with an Augmented Stability system

- Reduce fatigue damage rates at critical wing and fuselage locations with a Fatigue Reduction system

- Meet performance goals of each individual system with multiple CCV systems operating

Test System Configuration

Analytical studies were conducted to determine surface placement and size for each CCV concept and to evaluate the potential of various configurations to meet performance objectives. Existing B-52 control surfaces used for CCV functions are elevators and rudder. New additional surfaces consist of three segment flaperons, outboard ailerons, horizontal and a vertical canard. Figure 1 shows the surface arrangement and usage for each concept. The three segment flaperon replaces the existing inboard flaps.

The CCV systems were individually designed to achieve the specified performance objectives. Various system combinations were then analyzed and parameters were adjusted as necessary to meet objectives. A block diagram of the five B-52 CCV systems in presented in Figure 2. The angular rate and linear acceleration sensors associated with these systems are illustrated in Figure 3.

All new systems except the FMC were implemented on two onboard TR-48 analog computers. The FMC was hardwired. The FR system employed system hardware developed during the LAMS program. The fly-by-wire (FBW) system, also developed during the LAMS program, was used for pilot maneuver and flying qualities

evaluations. Figure 4 shows the modified test aircraft.

Validation Plan

The flight validation plan was structured around the types of flight test generally required in any large flexible aircraft test program. In addition, specific flight tests for math model accuracy determination were conducted. Five distinct categories of tests were accomplished: (1) flutter evaluations to determine the character of an artificially generated flutter mode and flutter mode control system validation, (2) control effectiveness evaluations to determine the aerodynamic characteristics of the new control surfaces, (3) in-flight dynamic response evaluations to determine the accuracy of the math model, (4) maneuver testing to determine flying qualities of the CCV systems and validation of the maneuver load control and augmented stability systems, and (5) low-level turbulence response evaluation to validate the ride control system and CCV systems compatibility with critical airframe loads and ride quality. Comparisons of actual test data and analysis predictions were made in all categories.

The matrix of test conditions developed to evaluate and validate system performance is shown in Figure 5. The three different fuel configurations are representative of a light weight B-52 with normal center-of-gravity (c.g.), a medium weight B-52 with a c.g. 7 percent aft of the current aft limit, and a heavy weight B-52 with normal c.g. Selected CCV systems were evaluated at various fuel configurations, test altitudes and airspeeds which best represent the true operational environment on the B-52 aircraft.

FMC SYSTEM TESTS

To evaluate the FMC system, a flutter mode (within the speed capabilities of the B-52 test vehicle) was created by adverse ballasting of the wing drop tanks. The left and right tanks, which normally carry 19,500 pounds of fuel each, were modified to carry 2000 pounds of lead in the forward end of each tank. At the 21,000 foot test altitude, the ballasted tanks were predicted to produce flutter at 330 knots calibrated airspeed for the light weight test configuration and 315 knots calibrated airspeed at the heavy weight configuration. Flutter was predicted to be a symmetric second wing bending and torsion mode at 2.4 Hz. Figure 6 compares actual speed versus damping (V-g) test results with analysis predictions for the light weight 260,000 pound baseline airplane. Baseline flutter was found to be approximately seven percent higher than predicted for both the light weight and heavy weight configurations.

Figure 7 shows the effects of FMC on speed versus damping characteristics and the compatibility of other CCV systems with the FMC. The test objective of flying 10 knots past flutter was achieved at both gross weights, and the FMC met or exceeded minimum damping requirements of g = .03 at all speeds. The addition of other CCV systems to the FMC further improved minimum damping at all speeds, thus validating the operational capability of the FMC with multiple CCV systems operating. A comparison of theoretical and flight test speed-

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damping results with the FMC on is shown in Figure 8. The FMC generally produced greater damping than predicted by analysis. In order to achieve these performance goals, the FMC system gains were increased up to twice nominal.

RIDE CONTROL SYSTEM TESTS

The Ride Control (RC) system was validated in low level turbulence at approximately 500 feet above the local terrain. Ten minute data samples were recorded for the baseline airplane and for the RC "on". Power spectral density analyses were accomplished on the random data samples to obtain gust response parameters. Figure 9 illustrates the effect of the RC on RMS vertical acceleration along the aircraft fuselage. Results are also compared to analytical predictions. The goal of 30 percent reduction was achieved at the crew station as predicted. Test results showed less improvement than predicted at the mid body, and a greater increase than predicted at the tail. However, the proper trend was predicted. RC effects on lateral acceleration are shown in Figure 10. The goal of 30 percent reduction at the crew station was also achieved in the lateral axis. Improvements were greater than predicted at both the mid body and aft body locations.

Figure 11 shows the change in aircraft acceleration with multiple CCV systems operating. A 30 percent acceleration reduction is still achieved with all systems operating. The addition of multiple CCV systems to the RC generally produced a further reduction in aircraft acceleration. An increase in the airplane gross weight by 100,000 pounds had no significant effects on the RC operation. No changes were required to the system, and performance goals were achieved in the vertical axis, which was the only axis tested at the heavy weight condition.

During the test program, it became necessary to increase the RC system gains by a factor of two in order to achieve the performance goals.

MANEUVER LOAD CONTROL SYSTEM TESTS

The MLC was flight tested to validate performance and compatibility at the light weight and heavy weight airplane configurations. The reduction in wing loads was determined from simulated pilot electrical inputs introduced to the MLC system through the onboard TR-48 analog computers. Flying qualities were evaluated for various pilot maneuvers. Although tests were not conducted at the B-52 design load condition (maximum gross weight, low speed configuration), the MLC goal was to reduce the maximum design wing root bending moment by 10 percent, or 8.2×10^{-6} inch-pounds. Figure 12 shows a comparison of theoretical and flight test results at the light weight low speed condition. The goal of 10 percent reduction in maximum design loads was achieved as predicted.

Comparison of theoretical and flight test results for the MLC are shown in Figure 13 over a speed range representative of B-52 maneuver operation. Maneuver loads were significantly reduced over the speed range.

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Compatibility of the MLC with other CCV systems is illustrated in Figure 14 for the lightweight medium speed condition. The addition of other CCV systems did not degrade MLC performance for any condition tested. No changes were required to the MLC to meet performance goals.

AUGMENTED STABILITY TESTS

The Augmented Stability (AS) system was tested to evaluate flying qualities of the medium weight airplane configuration with the c.g. shifted aft to the neutral point. The c.g. was shifted aft to 41.6 percent mean aerodynamic chord (MAC) by adverse fuel distribution. This c.g. location is 7 percent aft of the normal B-52 aft limit. The flying qualities were evaluated for various types of pilot maneuvers. Figure 15 shows a comparison of flight test and theoretical normalized pitch rate response to a step column input. The actual test data indicates good time constant correlation with less overshoot than analytically predicted.

Figure 16 indicates the decrease in stick force gradient as the c.g. was progressively shifted aft. The airplane without the AS system engaged shows very light stick forces, even at the normal aft limit of 35 percent MAC c.g. location, indicating a lower than normal artificial stick force gradient was mechanized on the FBW system. Even with these lower unaugmented airplane force gradients, the AS concept increased the force gradient a significant amount. These forces could have easily been made to meet the criteria by a FBW force gradient change and a gain change within the pilot command augmentation portion of the AS mechanization. Compatibility of AS and MLC is also shown.

FATIGUE REDUCTION SYSTEM COMPATIBILITY TESTS

The Fatigue Reduction (FR) system, validated singly during the LAMS program, was flight tested to validate compatibility with the remaining CCV concepts. The FR system was evaluated alone in low level turbulence with the light weight airplane configuration at approximately 500 feet above the local terrain. Once again, as during the RC tests, ten minute data samples were recorded for the baseline airplane and for the FR system "on". Power spectral density analyses were accomplished on the random data samples to obtain the gust response parameters. Reduction in RMS bending moments at critical wing and aft fuselage stations is shown in Figure 17 for the FR only, as well as with all systems "on" compared to the baseline airplane. With all systems "on", a slight increase in bending moment is shown at the aft fuselage location compared to the results obtained with FR "only". However, the bending moments are significantly reduced over the baseline airplane data.

The analytical predictions for bending moment reductions with all systems "on" at the same wing and fuselage locations are compared with actual data in Figure 18. The FR compatibility tests generally produced results greater than the analytical predictions. No changes were required in the FR system to enable achievement of the compatibility goals.

CONCLUSIONS

The flight test results from the B-52 CCV program have validated, for the first time, that significant performance benefits are achievable when the CCV concept is utilized.

The CCV systems proved to be operationally practical, both individually and collectively, at the gross weights, airspeeds, and altitudes tested.

The baseline mathematical models and theoretical predictions differed, in some cases, from the actual flight test data. Even with these differences between the math model and the actual airplane, the CCV systems met their individual and collective performance goals without system redesign. This result indicates that math model inaccuracies, which are inevitable in any airplane design program, can be compensated for by careful and deliberate design of the CCV systems. Simple gain changes, such as those required during the FMC and RC flight tests, to enable a system or combination of systems to meet the performance goal are considered to be a minor modification.

The results of the B-52 CCV program indicate that existing analysis techniques and performance prediction methods are indeed sufficiently accurate to permit incorporation of CCV concepts into future large aircraft designs.

FUTURE RESEARCH

As pointed out in Reference 2, the basic criteria for establishing acceptance of a new technology such as CCV is that: (1) the system meet predicted performance, (2) the system be operationally practical, (3) the system be reliable and safe, and (4) that it be cost effective. The B-52 CCV program has contributed significantly in establishing acceptance of CCV for large military aircraft by validating that predicted performance can be achieved over a limited operational range.

Future research efforts should primarily be concentrated in the two remaining areas. Since CCV technology is dependent on the concept of fly-by-wire, efforts should be focused on development of a highly reliable fly-by-wire system for large flexible aircraft. To validate that the technology is safe and cost effective, a technology demonstrator aircraft is needed which incorporates the full concept of CCV in the preliminary design. This test vehicle should be configured to demonstrate total dependence of the structural and aerodynamic design on the CCV concept.

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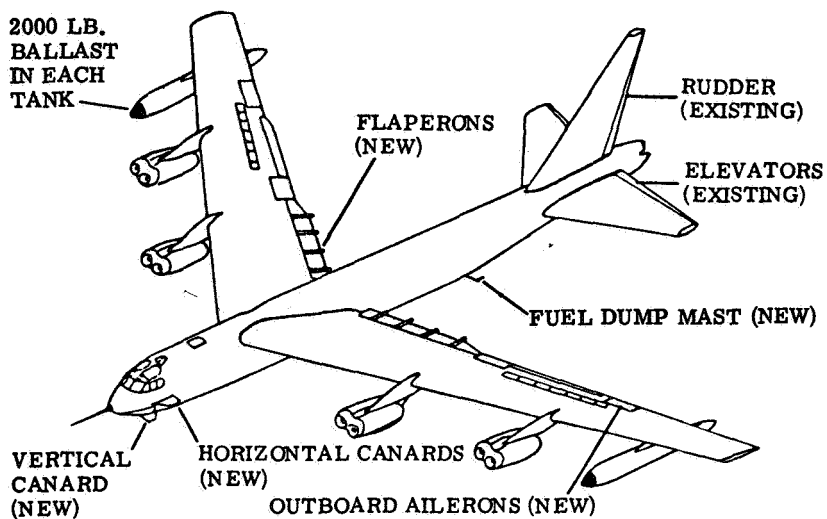
SURFACES	CCV CONCEPTS				
	RCS	FMC	MLC	AS	FR
RUDDER				X	
ELEVATOR			X	X	X
FLAPERON		1 SEGMENT	X		
OUTBOARD AILERON		X	X		X
HORIZONTAL CANARD	X				
VERTICAL CANARD	X				

Figure 1.- B-52 Test Vehicle Configuration
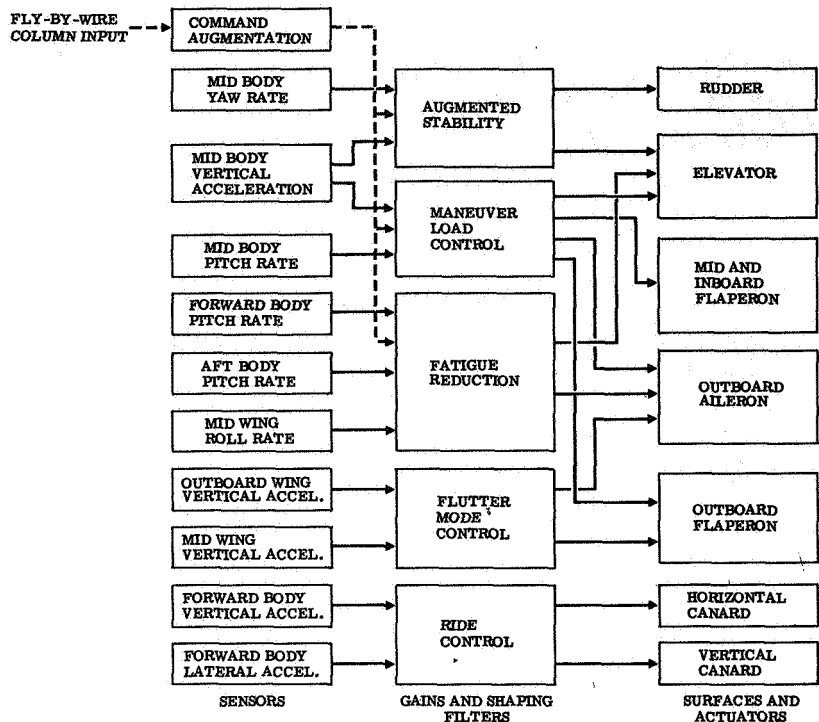
Figure 2.- B-52 CCV Systems Block Diagram
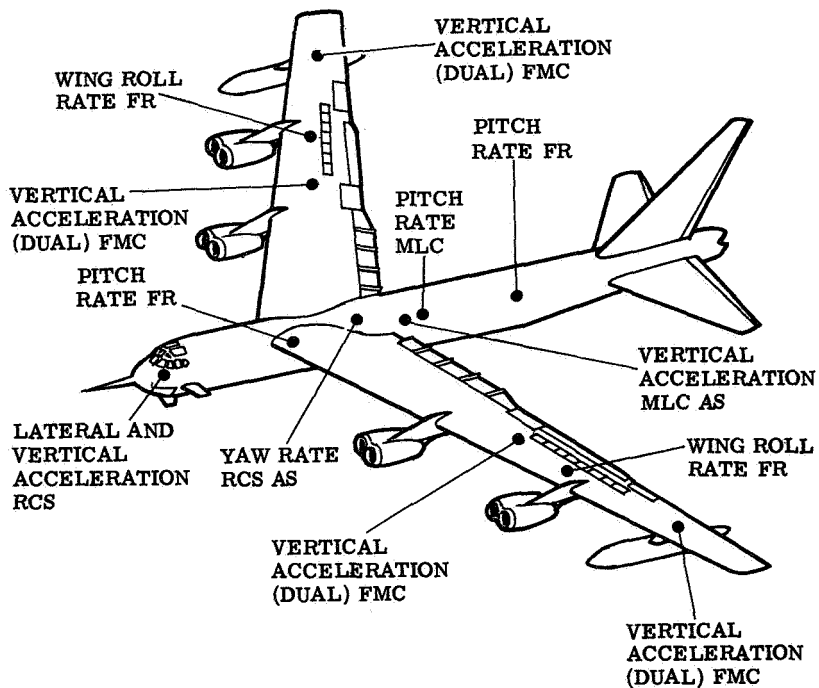
Figure 3.- B-52 CCV Systems Sensors

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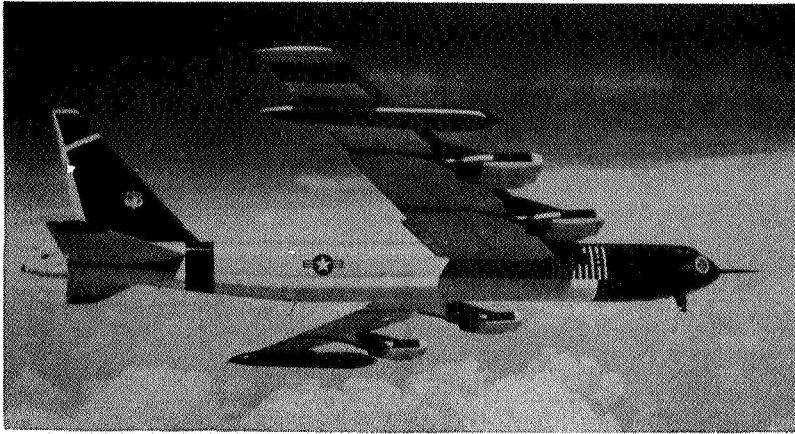
Figure 4.- B-52 CCV Test Aircraft

FUEL CONFIGURATION		ALTITUDES, FT.	AIRSPEEDS, KCAS
LIGHTWEIGHT, NORMAL C.G. 260,000 LB.	1	500 AGL, 6000, 21000	225 —→ 390
MEDIUM WEIGHT, C.G. AFT OF LIMIT 300,000 LB.	2	21,000	225 —→ 363
HEAVYWEIGHT, NORMAL C.G. 360,000 LB.	3	500 AGL, 6000, 21000	225 —→ 383

Figure 5.- B-52 CCV Test Conditions

FMC OFF

FUEL CONFIGURATION 1
21,000 FT.

- - - - - THEORETICAL
———— FLIGHT TEST
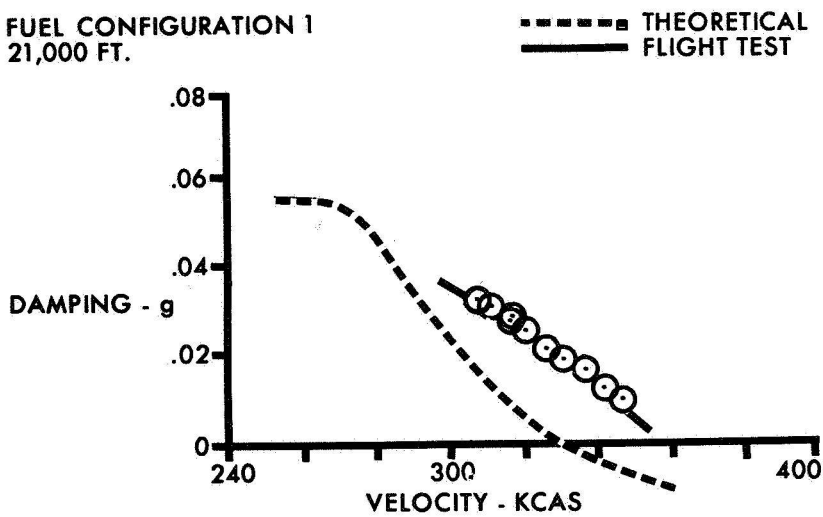
Figure 6.- V-g Comparison, FMC "Off"
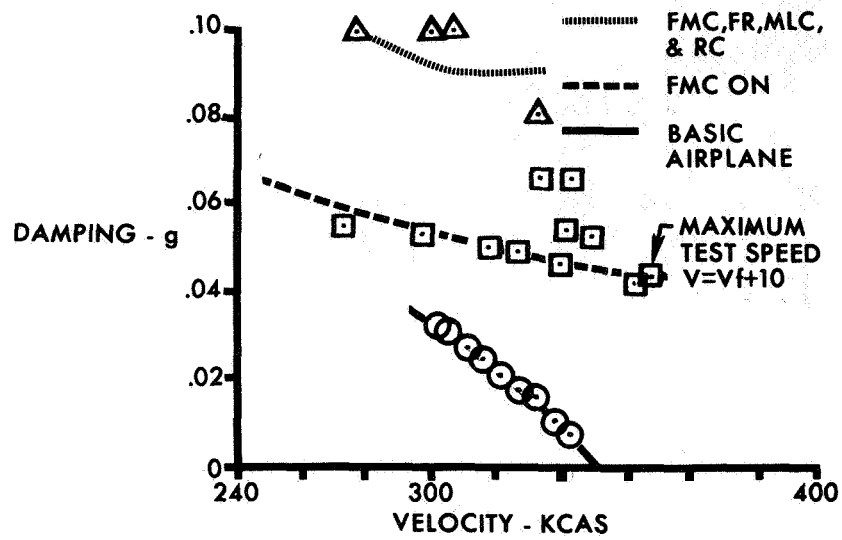
Figure 7.- Flight Test Flutter Results

FMC ON

FUEL CONFIGURATION 1
21,000 FT
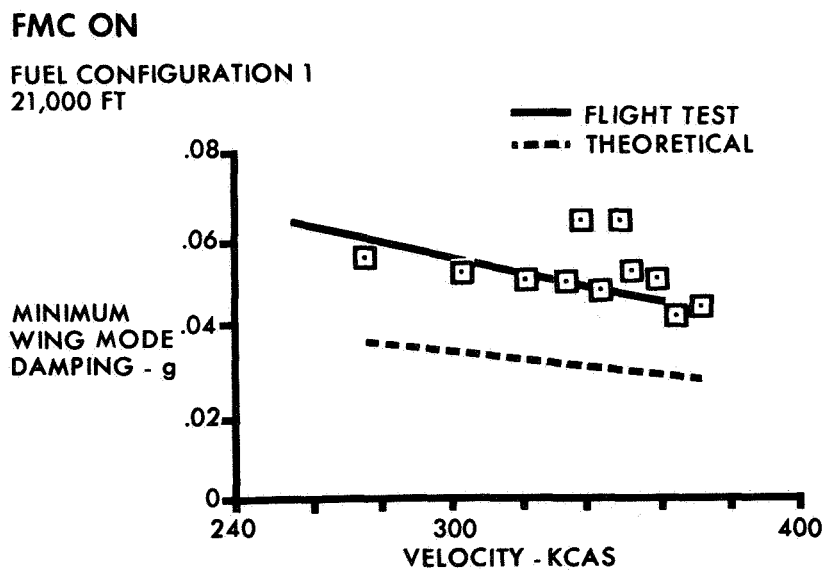
Figure 8.- V-g Comparison, FMC "On"
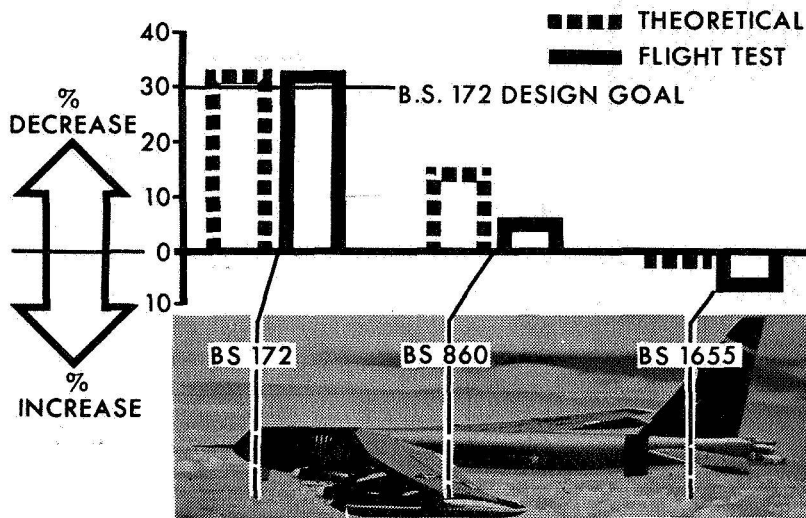
Figure 9.- RC Effect on Vertical Acceleration
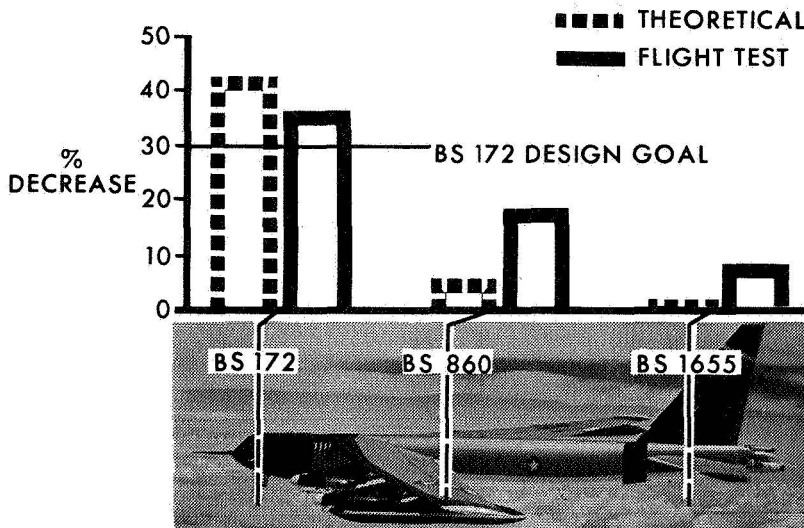
Figure 10.- RC Effect on Lateral Acceleration

FLIGHT TEST RESULTS

Figure 11.- Ride Quality Compatibility

Figure 12.- MLC Performance Comparison

WS 222

FUEL CONFIGURATION 1
RAMP HOLD INPUT

- - - THEORETICAL
— FLIGHT TEST

NORMALIZED
BENDING
MOMENT
IN-LBS/g

BENDING MOMENT
REDUCTION CRITERIA

MLC OFF

MLC ON

VELOCITY - KCAS
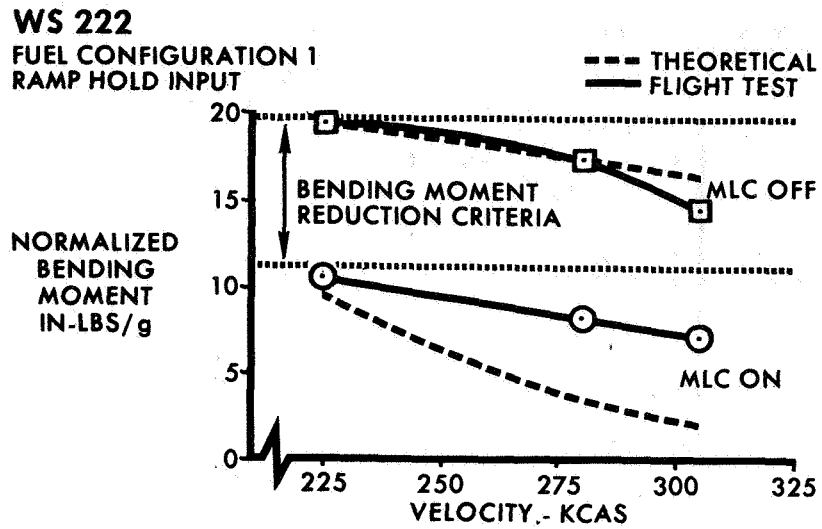
Figure 13.- Wing Root Vertical Bending Moment
Reduction Versus Airspeed

FUEL CONFIGURATION 1
305 KCAS
21,000 FT.
RAMP HOLD INPUT
$\Delta g=1$

VERTICAL
WING
ROOT
BENDING
MOMENT
WS 222
$(10^6$ IN-LB)

MLC OFF

MLC ON

FMC, FR, MLC & RCS ON

TIME SECONDS
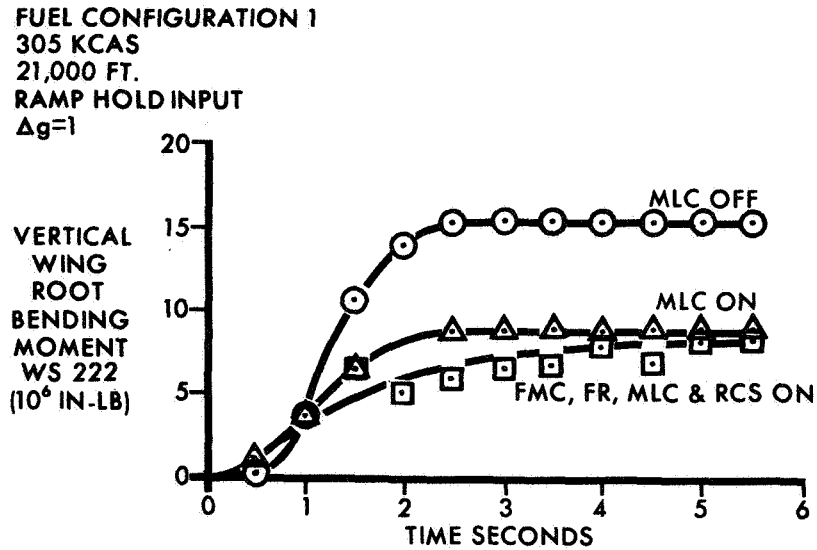
Figure 14.- MLC Compatibility

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FUEL CONFIGURATION 2
225 KCAS
21,000 FT.

STEP COLUMN INPUT
CG=41.6% MAC
LAS & PAS ON

- - - - THEORETICAL
G.W. = 298,500 LB.

——— FLIGHT TEST
G.W. = 287,700 LB.

$\dot{\theta}/\dot{\theta}_{ss}$

TIME - SECONDS
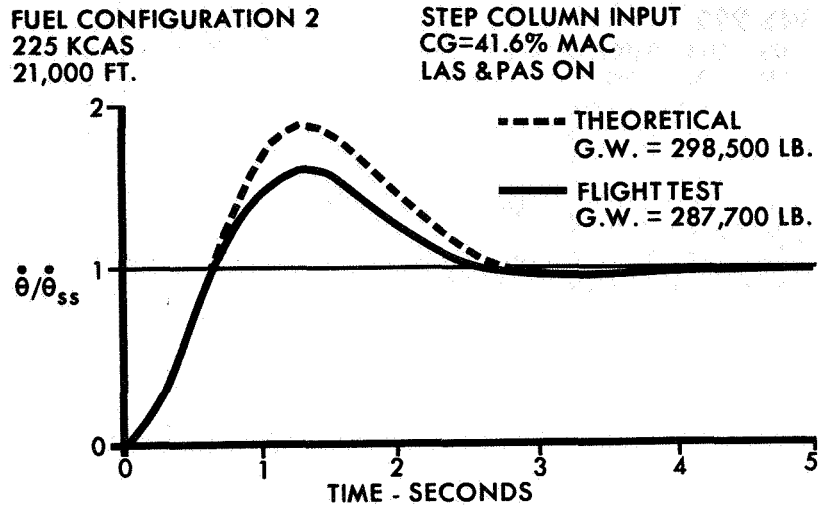
Figure 15.- AS Performance Comparison
at 41.6 Percent MAC C.G.

FUEL CONFIGURATION 2
21,000 FT
280 KCAS
⊙ BASIC AIRPLANE
⊡ AS ON
◇ AS + MLC ON

POUNDS/g

DESIGN GOAL
26.0 #/g

CG ~ % MAC
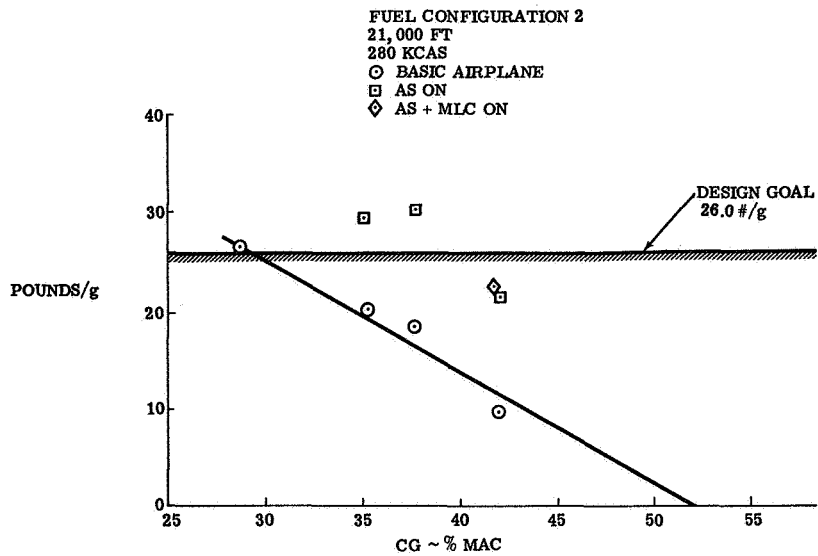
Figure 16.- Effect of C.G. on Stick Forces

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FLIGHT TEST RESULTS

- FUEL CONFIGURATION 1
  330 KCAS
  5,400 FT.
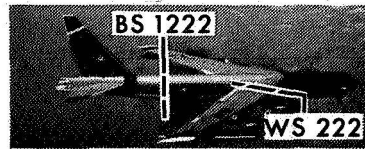
■■■ BASIC AIRPLANE
▥▥▥ FR
■ ■ ■ MLC, FMC, FR & RCS
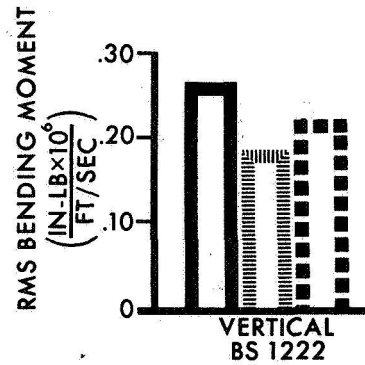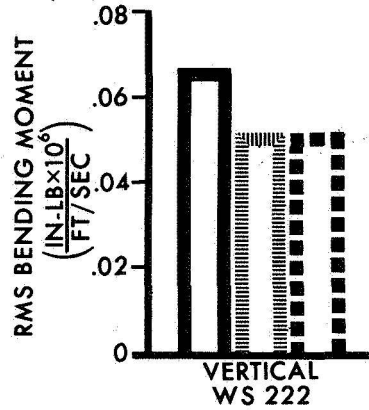
Figure 17.- FR Bending Moment Compatibility

- FUEL CONFIGURATION 1
  330 KCAS
  5,400 FT.
- FMC, FR, MLC & RC

■■■ FLIGHT TEST
■ ■ ■ THEORETICAL
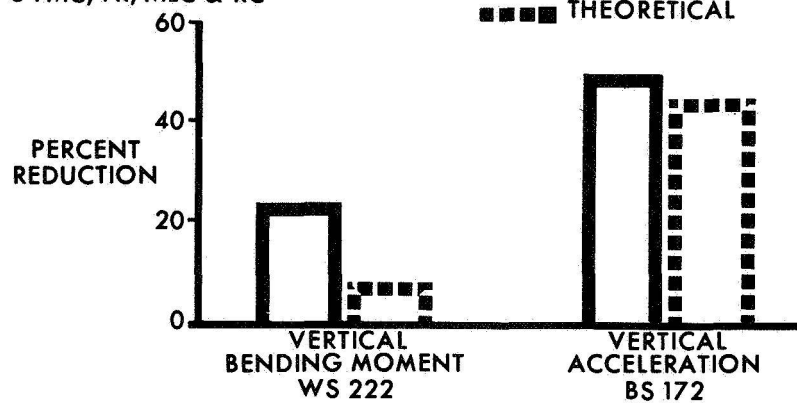
Figure 18.- FR Compatibility Comparison

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AN OVERVIEW OF NASA'S DIGITAL FLY-BY-WIRE

TECHNOLOGY DEVELOPMENT PROGRAM

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SUMMARY

The feasibility of using digital fly-by-wire systems to control aircraft was demonstrated by developing and flight testing a single channel system, which used Apollo hardware, in an F-8C test airplane. This is the first airplane to fly with a digital fly-by-wire system as its primary means of control and with no mechanical reversion capability. The development and flight test of a triplex digital fly-by-wire system, which will serve as an experimental prototype for future operational digital fly-by-wire systems, is underway.

INTRODUCTION

The advantages of digital fly-by-wire (DFBW) systems in terms of control system flexibility and reliability were demonstrated for spacecraft applications in NASA's manned space program. However, the transfer of this technology from spacecraft to aircraft is not direct and will require the identification and solution of many problems.

DFBW technology, when fully utilized in the flight control system of an aircraft, can provide significant advantages over conventional control systems in terms of reduced costs, weight, and volume, and in improved performance. A redundant digital system, which can identify in-flight system failures and reconfigure itself, offers a potential reliability comparable to that of the basic aircraft structure as well as the advantages of automatic control techniques.

Although these benefits cannot be easily quantified for all classes of aircraft, design studies do indicate major rewards in terms of more effective flight control systems and, thus, more effective aircraft. But even more important, these systems lay the ground work for active control technology, and it is the active-control-configured aircraft that offers the greatest potential in economic gains and performance advancements.

The overall objective of NASA's digital fly-by-wire program is to provide the foundation for this technology, in terms of design criteria and operational experience, which will lead to the development of practical digital fly-by-wire systems for future aircraft. To accomplish this objective, the program was separated into two phases, with an F-8C airplane (fig. 1) used as the test vehicle.

The goal of Phase I, which has been accomplished, was to demonstrate the feasibility of using a DFBW system as the primary flight control system of an aircraft. To accomplish this goal, a single channel DFBW primary flight control system was flight tested, using an analog backup control system for fail/safe redundancy.

The goal of Phase II is to establish a design base for the development of practical DFBW systems. This will involve the development and flight test of a triplex DFBW system using redundancy management and data bus concepts.

Figure 2 shows the schedule for Phases I and II. The major aspects of each phase are discussed in the following sections.

SINGLE CHANNEL SYSTEM DEVELOPMENT

To establish the feasibility of the DFBW concept, a system was designed to replace the basic mechanical primary flight control system of the F-8C test airplane in all three control axes. All mechanical connections linking the pilot's control stick and rudder pedals to the control surfaces were removed. To be compatible with fly-by-wire design philosophy and development practice, no mechanical reversion capability was provided even during the first part of the flight-test program. This is particularly significant because it required that satisfactory design and test techniques be demonstrated before the first flight. A single channel digital system concept was selected as the most straightforward approach to establishing system feasibility.

To minimize cost and development time, digital hardware and software originally developed for the Apollo program were used as the heart of the digital system. An Apollo guidance and navigation system was used which consisted of a digital guidance computer, an inertial measurement unit, and associated interface elements. Use of this hardware also made available highly trained Apollo support teams. Another factor leading to the selection of the Apollo computer was its demonstrated 70,000-hour mean-time-before-failure record. This factor overrode shortcomings of the hardware which resulted in some operational constraints.

A more complete description and discussion of the digital system is presented in reference 1. Pertinent aspects of man-rated software are covered in reference 2.

To provide redundancy if the primary digital system failed, an analog flight control system from a lifting body research vehicle was modified extensively and installed in the F-8C airplane as a triplex analog backup control system (ref. 3).

Phase I began in January 1971. During the following 15 months, five major contractors took part in the development and flight qualification of the Phase I system. These contractors and their areas of responsibility were:

Delco Electronics . . . . . . Digital system hardware

The Charles Stark Draper . . . Digital system software
   Laboratory, Inc. (MIT)

Sperry Flight Systems Division . . . Analog backup control system

Hydraulic Research and . . . . Secondary actuators
   Manufacturing Company

Ling-Temco-Vought, Inc. . . . . Aircraft and electrical systems

In addition to control law design and contractor coordination, NASA was responsible for specifying the Phase I system baseline configuration and interface requirements, verifying the final software and hardware flight readiness, and conducting the flight tests.

The Phase I system was first used in flight on May 25, 1972. This was the first flight of an aircraft in which a digital fly-by-wire flight control system was the primary means of control. As noted previously, no mechanical reversion capability was provided. Confidence in the reliability of the digital system was demonstrated by using it on the first takeoff and landing.

Forty-two flights were made before the flight program was completed in November 1973. The total flight time was 58 hours. The pilot controlled the airplane most of this time using the primary digital system. Approximately 14 hours were flown using the analog backup system for evaluation purposes, inasmuch as no digital system failures were experienced during flight. The flight-test results are presented in references 3 and 4.

Phase I established the feasibility of DFBW systems for primary aircraft control and provided flight data related to control law design, software verification, and operational procedures for DFBW systems.

MULTICHANNEL SYSTEM DEVELOPMENT

The goal of Phase II is to establish a design base for the development and implementation of future practical DFBW systems. To accomplish this goal a multichannel system is being developed which will provide redundancy management flight-test experience and verify other concepts of particular concern to the space shuttle orbiter development.

The Phase II system configuration and major tasks are discussed in the following sections.

System Configuration

A simplified diagram of the fully redundant triplex DFBW system is shown in figure 3. The principal elements of the system are to be installed on a removable pallet assembly, as the single channel system was in Phase I. Major components developed for Phase I, such as the analog backup control system, redundant secondary actuators, electrical power system, and instrumentation system, are to be retained for use during Phase II.

Dedicated, redundant sensors will be used to measure airplane angular rate, attitude, acceleration, and air data. Sensor inputs will be cross-strapped to each computer and synchronized on a bit-by-bit basis. Surface command outputs will be voted for fault detection and supplied to the triplex, force-summed, secondary actuator servo valves. Differential pressure equalization will be used to minimize nonlinear secondary actuator effects. A two channel (active and monitor) analog backup control system will be provided for use if the primary system fails.

The system will be designed to minimize ground operational and preflight support requirements. All system status testing will be automated and will be done onboard the airplane.

The digital processor selected for Phase II is a state-of-the-art, off-the-shelf, general-purpose computer with floating-point and microprograming features. The computer is an order of magnitude faster than the Apollo computer used in Phase I. The main storage memory is fully programable, which provides greater software flexibility. This increase in computer capability is of particular importance in carrying out the objectives of the Phase II program.

Evaluation of Space Shuttle Orbiter DFBW Concepts

An important aspect of Phase II is coordination with the shuttle orbiter flight control system development. In addition to being the first application of DFBW in an aerodynamic vehicle, the orbiter will contribute significantly to digital system technology by addressing the problems of redundancy management (reliability) and overall mechanization.

The shuttle flight control system will use the same digital processors as those being used in Phase II of the F-8 DFBW program. The Phase II triplex processor/ sensor configuration will thus make it possible to evaluate certain aspects of the shuttle system by using the F-8C airplane as a test-bed.

Redundancy management. — The redundancy management concept developed for the orbiter system to detect and isolate digital processor and control system sensor failures will be implemented and flight tested during Phase II. Because a reliable means of achieving failure detection and isolation is a major problem in the design of redundant DFBW systems, flight-test verification of the concepts in Phase II will establish a significant data base for future applications.

Data bus. — The data bus concept of reducing cabling and connector requirements for redundant systems by compressing data from several sensors onto

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redundant transmission lines is important in the development of DFBW technology. The discrete format of signals in DFBW systems makes the data bus a natural solution to the complex cabling problem. In Phase II the technique proposed for the shuttle system will be used to process trim commands and mode panel information (e.g., status lights, mode change commands) and to transmit the information from the airplane cockpit to the palletized system in the equipment bay. This will greatly reduce the number of wires and will verify data bus utility for shuttle as well as future system applications.

Computer synchronization. — Of major concern in the design of any redundant DFBW system is whether or not to synchronize the computer operations and, if so, the best way to do it. The Phase II system will be designed with enough flexibility to permit the use of various synchronization approaches as well as asynchronous operation. Included will be the baseline approach selected for the orbiter system.

Control laws. — The first control laws to be evaluated in flight during Phase II will be similar to those developed for the F-8C airplane during Phase I and similar in format to those being developed for the shuttle orbiter. These include C* and rate command modes for pitch and roll as well as direct control modes for each axis. Control law software required for moding and initialization will therefore be similar for both programs, which will permit some system verification.

Higher order programing language. — A higher order programing language, called Higher Aerospace Language (HAL), is being developed in support of shuttle software requirements. Use of this language in developing certain elements of the control laws for the Phase II system will make it possible to debug and verify it before it is actually applied to the shuttle orbiter.

Backup control system. — The present shuttle system configuration will require a dissimilar single channel digital backup control system during initial horizontal flight tests to override possible primary system generic failures. The executive structure for the shuttle backup system will be implemented in the Phase II system and flight-qualified through flight-test verification.

Advanced Control Law Development

To assess the capability of a digital system to perform the functions necessary for future active control applications, additional control laws will be programed and evaluated during Phase II. A specific task is the investigation of improvements that can be made in aircraft control law implementation as a result of the rapidly advancing digital fly-by-wire system capability. The availability of a powerful onboard digital computer system that can process sophisticated flight control laws in real time has added a new dimension to realizable control law development. Control laws previously too complex and unwieldy for analog system applications can now be considered prime candidates for digital applications.

Initial Phase II control law research is directed toward the use of active control for maneuver load control, possible improvement in ride quality, suppression of turbulence effects, flight envelope limiting techniques, and operation at reduced

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static-stability margins. The basic elements of such a control law now being developed for the longitudinal axis are illustrated in figure 4. The structure consists of a boundary controller for angle-of-attack limiting, a normal controller for longitudinal commands, a direct-lift controller for commanding symmetric ailerons, a load controller, and autopilot modes. A proportional flap-to-elevator crossfeed is planned to compensate for the pitching moment produced by symmetric aileron deflection.

The design objective for the longitudinal axis is to achieve good handling qualities by matching desired response criteria for both positive and negative static stability margins. Gust load alleviation is provided by additional damping of short-period dynamics using the elevator surface. Angle-of-attack limiting is provided throughout the flight envelope.

Direct lift of the symmetric ailerons is combined with the elevator to minimize drag during maneuvers and to enhance gust load alleviation during cruise. The three autopilot modes are the conventional attitude hold, altitude hold, and Mach hold.

Other advanced control law prospects, in which adaptive techniques and optimal control theory are used, are being studied for possible flight-test evaluation during Phase II.

Remotely Augmented Vehicle Facility

As part of Phase II, a unique remotely augmented vehicle facility is to be developed to support advanced control law research and flight-test evaluation (ref. 5). A diagram of the proposed facility is shown in figure 5. During a test flight, a special remotely augmented vehicle test mode may be selected by the pilot that will divert his control commands to a ground computer facility, via a telemetry down-link, on which a particular advanced control law to be evaluated is programed. Control surface commands are determined by the ground computer on the basis of the pilot's airborne commands, the airplane's response, and the programed control law. The surface commands are then transmitted, via a telemetry up-link, to the airplane system and the corresponding control surface. The pilot flies the airplane through the control laws programed on the remotely located ground computer. Fail safety will be maintained through the use of reasonability tests built into the ground computer facility and safety networks in the telemetry equipment. This approach will permit a great deal of flexibility for control law evaluation without compromising the basic airborne system verification requirements.

CONCLUDING REMARKS

The full realization of the benefits of active control technology and the benefits predicted by its application to aircraft design depends on the development of practical, reliable, and versatile digital fly-by-wire (DFBW) control systems. The feasibility of such systems and confidence in their reliability and integrity were

established in Phase I of the F-8 DFBW program. The goal of Phase II of the program is to establish a design base from which practical, reliable systems can be developed. This will be accomplished by developing and flight testing a fully redundant triplex DFBW system.

The multichannel system development carried out during Phase II will establish techniques for validating redundant system software and hardware interfaces and for establishing operating procedures unique to DFBW systems. Flight-test evaluation of orbiter control system concepts using the F-8C airplane will result in verification of redundancy management software for digital processor and sensor fault detection and reduced generic failure probabilities for the orbiter system.

Flight-test evaluation of advanced control laws during Phase II will provide an opportunity to assess the capability of DFBW systems to perform the complex control tasks associated with active control applications.

The NASA DFBW program, although complementary to other fly-by-wire activities, is aimed specifically at providing the technology for practical digital flight controls for civil aircraft. As such, it represents the first step toward a new generation of active-control-configured aircraft which will offer significant economic advantages.

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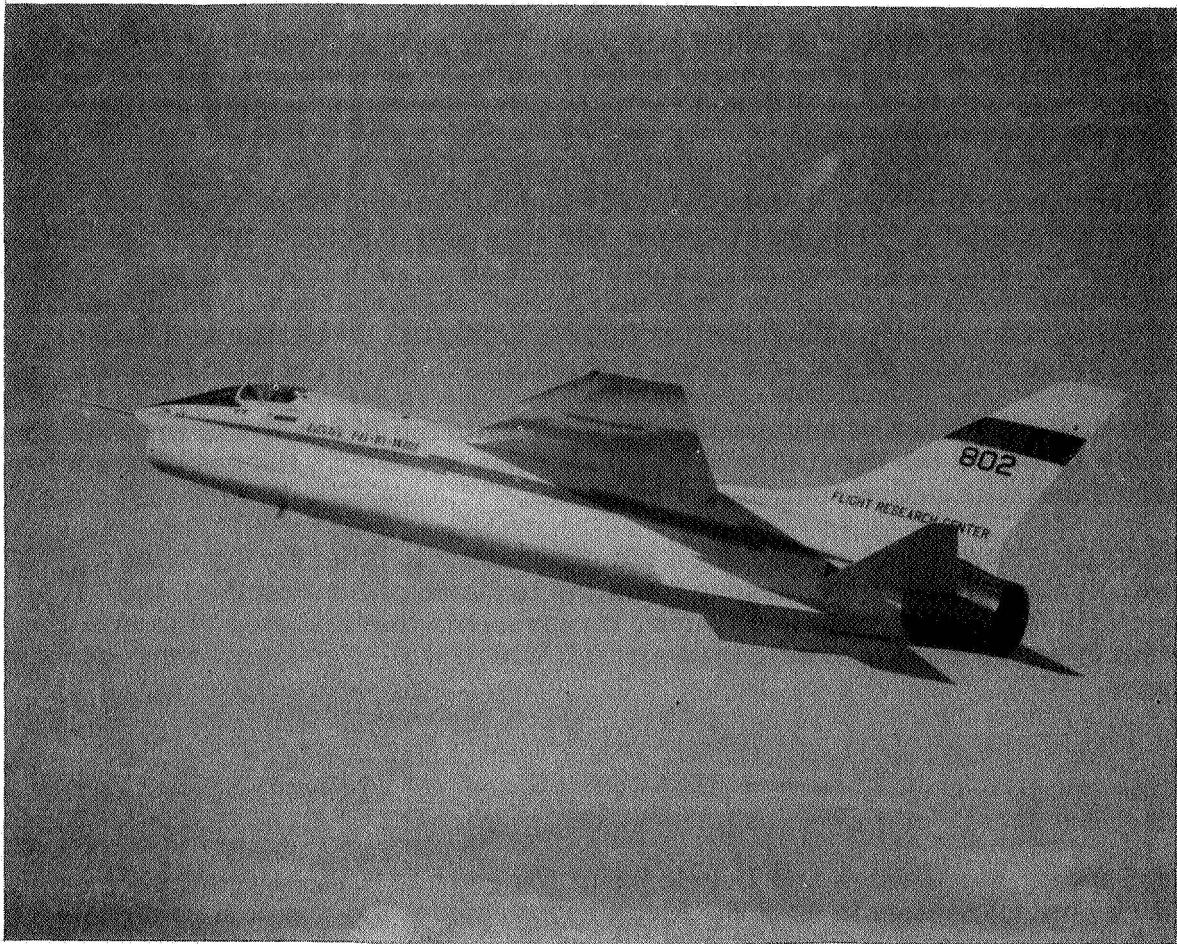
Figure 1. F-8C test airplane.
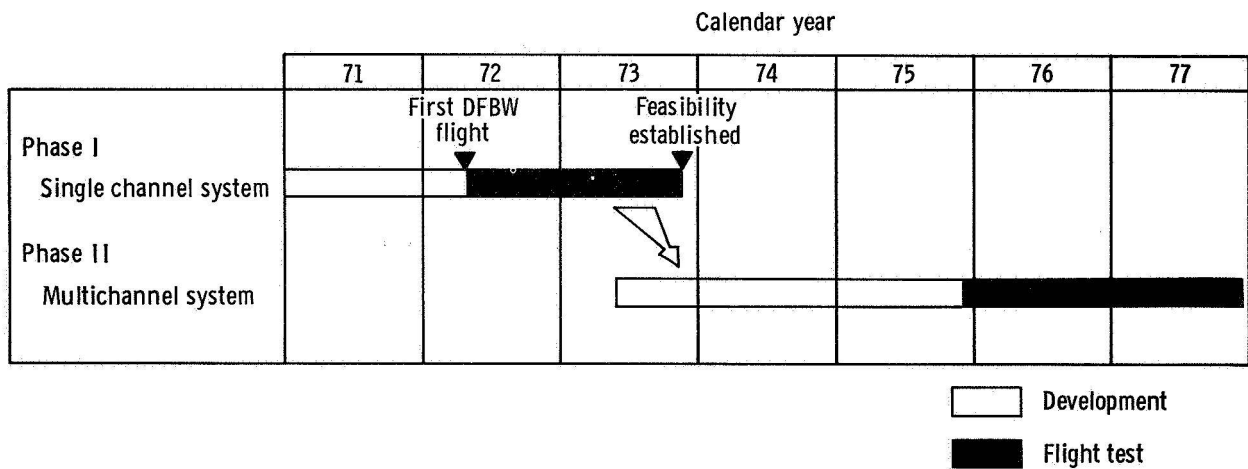
Figure 2. F-8 DFBW program schedule.
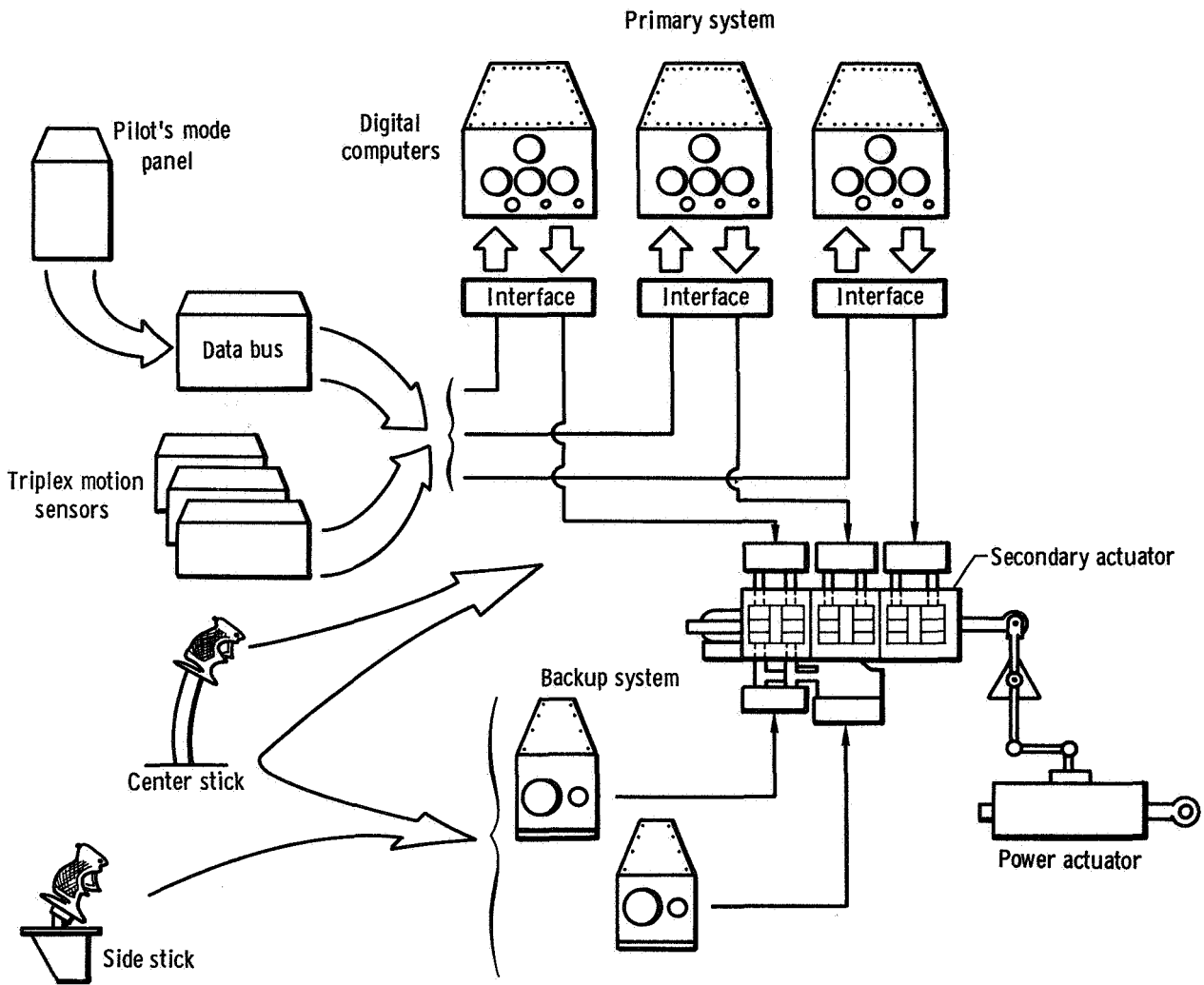
Figure 3. Phase II system configuration.
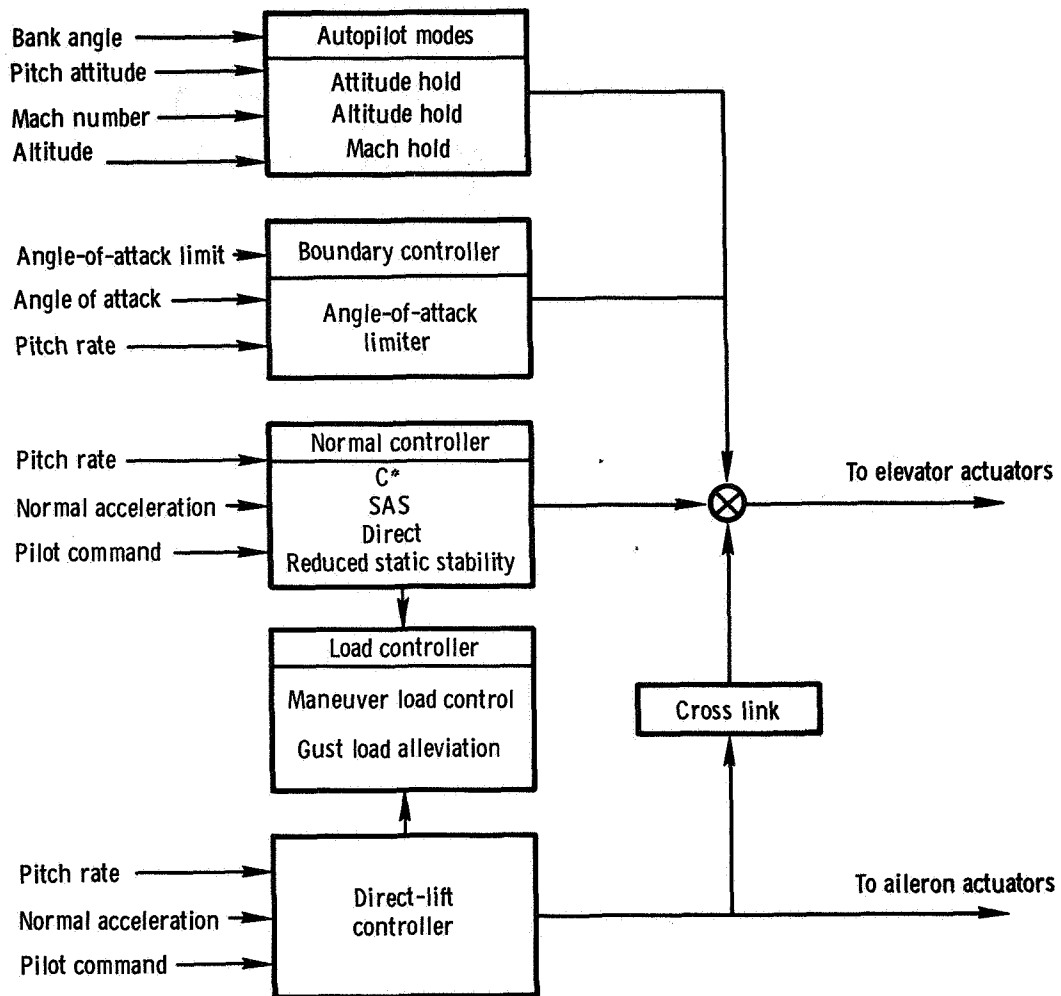
Figure 4. Active control law diagram for longitudinal axis.
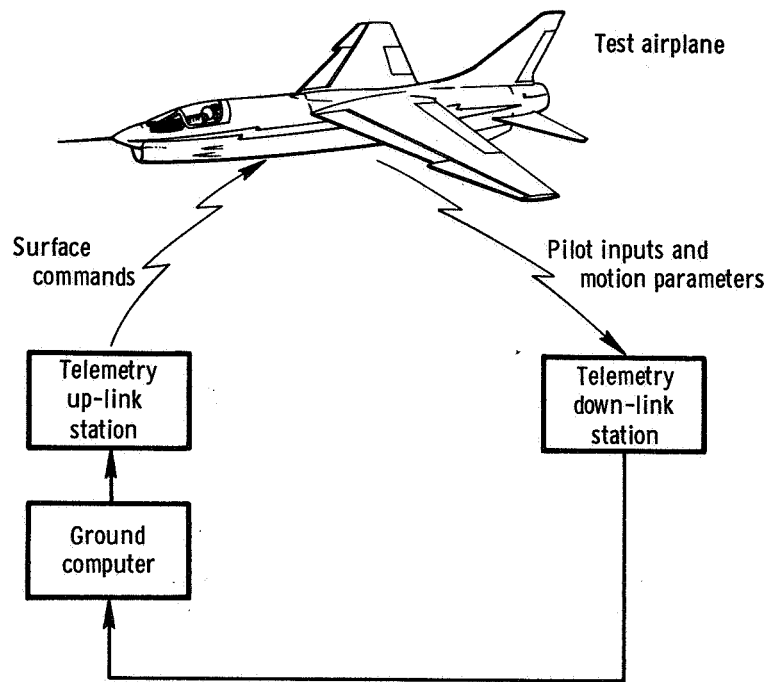
Figure 5. Remotely augmented vehicle facility.

DESIGN AND DEVELOPMENT EXPERIENCE

WITH A DIGITAL FLY-BY-WIRE CONTROL SYSTEM

IN AN F-8C AIRPLANE

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SUMMARY

To assess the feasibility of a digital fly-by-wire system, the mechanical flight -control system of an F-8C airplane was replaced with a digital primary system and an analog backup system. The Apollo computer was used as the heart of the primary system. This paper discusses the experience gained during the design and development of the system and relates it to active control systems that are anticipated for future civil transport applications.

INTRODUCTION

A major deterrent to the application of active controls to transport aircraft has been a lack of experience in designing highly reliable flight control augmentation systems and verifying them in flight. Digital fly-by-wire technology has the potential for providing the necessary reliability while still offering design flexibility. To assess the feasibility of a digital fly-by-wire system, the NASA Flight Research Center conducted a flight research program in which the mechanical flight control system of an F-8C airplane was replaced with a digital primary system and an electrical analog backup system.

This paper describes the fly-by-wire system and the design and development of the digital primary system. The system and the design procedures are assessed in light of similar applications being contemplated for future transport aircraft.

The paper was written in conjunction with references 1 to 3, which discuss the backup control system, software management, and results from the flight tests.

SYMBOLS

K	proportionality constant
KAZ	normal acceleration feedback gain to stabilizer, deg/g
KG	stick or rudder pedal gearing constant, deg/m
KP	roll rate feedback gain to ailerons, deg/deg/sec
KQ	pitch rate feedback gain to stabilizer, deg/deg/sec
KR	yaw rate feedback gain to rudder, deg/deg/sec
q_k	pitch rate at k^{th} sample, deg/sec
s	Laplace transform variable
T	sample period, sec
z	complex variable, e^{sT}
θ_k	pitch angle at k^{th} sample

TEST AIRPLANE

An F-8C airplane (fig. 1) was selected for use in flight testing a digital fly-by-wire system. Several characteristics of the airplane made it suitable for this test program. The handling qualities without control augmentation were acceptable for emergency operation, thus backup control could be provided through a relatively simple system. In addition, the airplane had enough space for the system's components, and the capacity of the hydraulic systems was adequate.

Some features of the F-8C airframe had an impact on the fly-by-wire system design. The variable-incidence wing moves up 8° for low-speed flight. This rotates the fuselage nose down relative to the free airstream, improving the pilot's visibility during the approach. Several functions within the flight control system are programed as a function of wing position. For example, the horizontal stabilizer is driven 5° leading edge up when the wing is raised.

The F-8C airplane does not have independent flap surfaces, so the ailerons are driven collectively to serve as flaps through a mechanical linkage independent of the primary control system. For this program the linkage was disconnected and the ailerons were driven to the drooped flap position through the fly-by-wire system.

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The original F-8C flight control system had a direct linkage in pitch (no augmentation), and roll and yaw stability augmentation systems (SAS) (figs. 2(a) and 2(b)). The yaw SAS included an aileron-to-rudder interconnect with a gain programed as a function of stabilizer position.

DATA ACQUISITION SYSTEM

A standard NASA pulse code modulation (PCM) system was installed to record airplane motion, pilot input, and fly-by-wire system parameters external to the computer. Seventy-seven channels of 9-bit data were recorded on an onboard tape and telemetered to a ground station for real-time monitoring. Eight of the channels were digital words indicating the state of 57 discrete values from the fly-by-wire system. Although excellent for automated data reduction, the PCM system proved to be unsuitable for investigating the effects of analog-to-digital and digital-to-analog quantization. The resolution of the PCM system was on the same order as that of the digital control system quantization, which made it difficult to isolate the effects of the control system quantization.

Internal digital computer parameters were recorded on the onboard PCM tape recorder. One-hundred-word pairs were strobed out every 2 seconds and recorded for postflight analysis. The word lists were resident in the software onboard the airplane and could be reprogramed during the flight tests. This recording system was used primarily to determine the gross status of the computed parameters within the flight control system; the strobe rate was inadequate for tracing individual parameters each control computational cycle.

IRON BIRD SIMULATOR

An iron bird simulator played an important role in the development of the fly-by-wire system. The simulator consisted of another F-8C airplane, in which all the digital fly-by-wire flight control hardware was installed, tied in with a hybrid computer and appropriate interface equipment (fig. 3). The digital fly-by-wire hardware was flight qualified and served as spares for the flight vehicle. The F-8C aerodynamics and bending modes were modeled by using the digital portion of the hybrid computer for the aerodynamics and the analog portion for the bending modes.

FLY-BY-WIRE SYSTEM

The fly-by-wire system had a digital primary control system and an electrical analog backup control system. Components of the fly-by-wire system are shown in figure 4, and the location of the components in the F-8C airplane is illustrated in figure 5.

A simplex digital primary system and a triplex electrical analog backup system

provided enough reliability that the mechanical system could be completely removed. The digital primary system consisted of a computer, inertial measurement unit, coupling data unit, and display and keyboard, all taken from the Apollo guidance and navigation system. Reference 4 provides details on this equipment relative to the Apollo application. The triplex backup control system consisted of only surface position command electronics. Specially designed electrohydraulic secondary actuators interfaced the primary and backup electronic commands with the conventional F-8C control surface power actuators.

Components of the fly-by-wire system were part of the primary or backup system and, in some instances, were shared between the two systems. Individual components are described in the following sections according to function.

Computational

The Apollo computer performed all flight control computations in the primary control system. Characteristics of this computer are summarized in the following tabulation:

Read-only memory	36,864 words
Scratch pad memory	2,048 words
Word length	14 bits plus sign and parity
Number system	Fixed point, ones complement
Memory cycle time	11.7 microseconds
Computation time —	
Add	23.4 microseconds
Multiply	46.8 microseconds
Divide	81.9 microseconds

Although slow by today's standards, the computer could perform all flight control functions within 30 milliseconds. Flight control laws for the F-8C airplane were programed for the computer's hardwired memory and could not be changed after the memory was manufactured. However, flexibility was achieved by placing feedback gains, logic flags, digital filter coefficients, and other gain variables in the computer's scratch pad memory; 105 of these variables associated with the flight control system could be changed.

Control laws for the backup system were mechanized in triplex control electronic boxes, which can be considered to be special-purpose analog computers. Each electronic box contained 67 operational amplifiers. Requirements for high reliability in the flight environment dictated the use of ruggedized packaging and hardwired circuits. Consequently, flexibility for changing control laws was limited to gain and nonlinear constant changes; even these changes required replacement of hardwired resistors and diodes. Although the control system is considered to be an analog system, more than half of each electronic box containing individual channels of the analog electronics was devoted to logic elements, such as comparators between the redundant channels.

Pilot Interface

Vehicle control and trim inputs.—The center stick was the primary means of control for both the primary and the backup systems. It was connected to the basic F-8C mechanical feel system. Disconnecting the mechanical links to the control surface caused undesirable looseness in the lateral stick, so a viscous damper was added. Stick position was sensed through two separate linear variable differential transformers (LVDT) in each axis. Each LVDT had triplex windings. Two of the windings from one LVDT in each axis were used by the primary system; the third winding was for instrumentation. The second LVDT provided triplex inputs to the backup control system.

A minimum displacement, two-axis side stick was used as an alternate controller input to the backup system. This side stick was not part of the original fly-by-wire system; it was used only as a means of evaluating a "force type" side stick controller in an actual aircraft environment (ref. 1).

The rudder pedals provided inputs to both the primary and the backup systems. Similar to the center stick, they were connected to the F-8C mechanical feel system and used LVDT's for sensing rudder pedal position.

Trim commands for primary pitch and roll were made through a two-axis beeper switch on the center stick. Backup trim inputs in all three axes were made through separate spring-loaded toggle switches on the pilot's left-hand console. These toggle switches were also used to provide primary yaw trim and primary pitch and roll trim inputs if the center stick trim switch failed. When the side stick was being used, trimming was accomplished through a beeper switch on the side stick.

Fly-by-wire functional control.—Figure 6 shows the mode and power panel, which was the pilot's means of communicating with the primary system. The pilot was able to choose between several different control system modes simply by depressing the appropriate button. Additionally, he was able to change system gains according to the logic loaded in the software before the flight. Several primary system failure status lights were located across the top of the panel, and power switches and power status lights were located across the bottom. Individual axes could be transferred to backup through the backup control system (BCS) switches. The pilot could also transfer all axes to backup simultaneously by using a "paddle switch" on the center stick.

A servo engage panel on the left-hand console permitted the pilot to selectively engage or disengage each channel of each servo actuator. The panel provided control over both the backup and the primary systems. The status of the actuation system and the backup electronics was displayed on this panel.

Motion Sensing and Interface

Another component from the Apollo guidance and navigation system was the inertial measurement unit. Although angular body rates and linear accelerations

were not measured directly in the Apollo application, the substitution of aircraft gyros and accelerometers for the fly-by-wire tests would have required a major system modification and would have sacrificed the integrity of the total Apollo system. An alternate approach was adopted: Body rates and linear accelerations were computed from the gimbal angles and the digital incremental velocity vector information the inertial measurement unit provided to the Apollo computer for use in the primary system.

A coupling data unit provided the interface between the inertial measurement unit and the Apollo computer in the Apollo guidance system and provided the same function for the primary system in the F-8C application. The angular resolution was 0.011°; however, the rate resolution was of more importance and was nonlinear with gimbal angle rate. Body angular rate estimation was directly related to gimbal angle rate resolution. Gimbal rate resolution was ±0.183 deg/sec for rates less than 4.4 deg/sec, and ±2.74 deg/sec for rates between 4.4 deg/sec and 70 deg/sec. The error was manifested as a random noise band of 2.74 deg/sec peak to peak for rates greater than 4.4 deg/sec. Acceleration was sensed by using pulse integrating pendulous accelerometers. The quantization level for the normal acceleration signals was 0.2g.

The coupling data unit contained several digital-to-analog converter channels, which made it possible to send the necessary primary system commands to the control surfaces. To protect against undetected failures between the computer and the surface actuators, dual signals were generated in each axis, beginning with dualized commands to dual digital-to-analog converters in the coupling data unit in each axis. The control surface drive signals were quantized to ±384 levels, which is somewhat less than a full 9-bit word.

Control Surface Actuation

Similar actuation systems were used in each axis. Each actuation system had a secondary actuator and a power actuator. Separate sets of actuators were used for the left and right horizontal stabilizers and ailerons. A single set of actuators was used for the rudder.

The hydraulic power actuators from the basic F-8C airplane were used without modification. Electrohydraulic secondary actuators were installed to drive the metering valves of each of the five power actuators. The secondary actuators acted as three-chamber force summing devices when driven from the backup system. The primary system drove the secondary actuators through active monitor servo valves.

When the secondary actuators were driven through the primary system, they were stabilized through the active servo valve in the primary system electronics box. Analog 5-hertz low-pass filters were included in the primary system electronics. When driven through the backup servo valves, the secondary actuators were stabilized in the backup system electronics packages, one for each of the three backup channels.

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Reference 1 describes the actuation systems in more detail.

Primary/Backup System Interface

A functional diagram of the fly-by-wire system is shown in figure 7. The Apollo computer received inputs from the pilot's stick together with aircraft motion information from the inertial measurement unit. Surface commands were computed according to the programed control laws.

The two drive signals for each surface represented commands to the secondary actuator position loop, which was closed with analog stabilization electronics outside the Apollo computer. As shown in figure 7, there was an active and a monitor servo path. If a failure occurred in either path, a hydraulic comparator would sense the differential pressure between the active and the monitor servo valve and transfer control to the backup control system. As long as the primary control system was operating normally, the backup control system would track the active channel by way of the synchronization network. Only the hydraulic pressure was bypassed at the secondary actuator, so that the backup system was ready to take over at any time. If a transfer to the backup system was requested, the bypass was removed and the synchronization network was disabled, resulting in immediate proportional control from the pilot's stick. In the backup mode, the active servo valve was blocked and the secondary actuator operated as a force summer for the three backup channels. The digital computer continued to operate, computing the control laws which gave the best estimate of what the backup system commanded. If a transfer to the primary control system was attempted, the transient was small as long as the computer was tracking the backup system. If the error was excessive between the primary control system and the backup control system, a cross-channel comparator prevented transfer to the primary control system.

Fault Detection

Although built-in fault detection was extremely important for both the primary and the backup systems, it was of particular importance in the primary system. Because the primary system was full authority as well as single channel, its responses could have been hazardous if failures were not handled properly. Therefore, it had to be established that no digital computer system hardware failure could cause a hardover or otherwise hazardous signal. Figure 8 shows the type of digital system failure detection used. The Apollo computer had an extensive and proved fault detection and reporting system which was built into the computer hardware (item 1 in the figure). This system, modified slightly for application to the F-8C airplane, was the most significant portion of the failure detection system. Some of the types of failures detected were:

> Logic circuits —
> > Parity failed
> > Program entered loop and did not exit
> > Program attempted to access unused read-only memory
> > Program failed to check in occasionally

Analog circuits —
Voltage went out of limits
Oscillator failed
Timing pulse generator failed

Each of the failures caused a restart, that is, a hardware-forced transfer out of the control law program to a software routine which performed several clearing and initialization steps in an attempt to correct the cause of the restart before allowing control law computations to continue. For some restart conditions, a signal was issued which caused a transfer to the backup control system.

The Apollo computer also monitored the performance of the inertial measurement unit (item 2, fig.8). Written into the software were decisions either to transfer the system to the backup control system for serious failures or to select the direct mode in the primary system for situations such as an inertial measurement unit accelerometer failure, which would affect only certain augmented modes.

Analysis of primary system failures showed the need for additional hardware failure detection circuitry (item 3, fig 8). The failure of certain channel outbits not monitored by the Apollo computer, in combination with normal pilot reactions, could have led to hazardous situations. These conditions first became apparent in piloted, closed-loop simulations using the iron bird simulator. The necessary hardware modifications were made and implemented in the system to circumvent these failure conditions or to cause a transfer to the backup control system when prevention was not possible.

Built-in test equipment for the backup system and primary electronics was provided in the pilot's side console. This self-test equipment could be activated only during preflight tests (ref. 1).

FLIGHT CONTROL SOFTWARE

Software flexibility made it possible to investigate a multimode F-8 digital flight control system using hardware that was designed for an entirely different purpose — guidance and navigation in space. The structure of the primary system control laws, which were implemented through software, and the associated logic functions are described.

Control Law Modes

Control in each axis was provided in the control laws. The simplest form in each axis was the direct mode, illustrated in figure 9. The control law structure and gain settings were selected to be as close as possible to those of the backup control system. The first level of augmentation was rate feedback in the pitch and roll axes. Figure 10 illustrates these SAS modes as they were during flight tests. In the yaw axis, an aileron-to-rudder interconnect was included in addition to the yaw rate feedback (fig. 11). The most advanced type of control law was a blended

pitch rate and normal acceleration command augmentation mode, illustrated in figure 12.

The sampled-data aspect of the digital fly-by-wire system significantly affected the implementation of the software. The pitch SAS mode, illustrated in figure 13, was representative of the other augmented modes. A multirate sampling system was used, with a major cycle sample time, T, of 30 milliseconds and a minor cycle sample time, 3T, of 90 milliseconds. Gimbal angles were transformed to body angles by using sines and cosines of gimbal angles updated once every minor cycle. A second-order rate estimator operating on these gimbal angles provided body rates, which were then filtered. General-purpose digital filters were programed so that different characteristics could be selected independently for each feedback parameter. Proper selection of the difference equation coefficients could provide a wide range of filter characteristics. A first-order prefilter, a dead band, and a parabolic nonlinear gradient were available to shape the pilot's input.

Logic Functions

An important capability made possible through software was the integration of logic statements in the control law code. Logic statements, even though complex, were easily written into the software. Had the system been analog rather than digital, special-purpose hardware would have been necessary to perform the same logic functions. One mode logic function associated with the yaw axis is discussed in reference 5.

Another type of logic function was the software reasonability test which was applied to each surface command before it was sent to the digital-to-analog converter. If the new command differed from the previous command by more than a predetermined amount, the affected axis would have transferred to the direct mode. This down mode philosophy was based on the assumption that a reasonability limit would be exceeded because of generic failures in the augmentation control laws rather than because of a hardware failure which would have affected the direct mode as well. It was assumed that a hardware failure would have been detected by the built-in Apollo computer fault detection logic.

Trim inputs were also tested for reasonability before the trim value was updated. If a combination of primary trim commands was sensed that corresponded to an impossible situation for an unfailed system, a failure was assumed, the primary trim was deactivated, and an auxiliary trim system was activated. A test for runaway trim was included which disabled trim updates if the trim command persisted for more than 3 seconds.

DIGITAL FLY-BY-WIRE DESIGN

Design Ground Rules

Several ground rules were established in order to meet the objectives of the

program. First, the airplane was to fly from the first flight without mechanical reversion capability. This forced the designers to take the care necessary to establish as much confidence in the system, including the software, as would be required for future active control applications. Second, the primary digital system was to utilize the hardware from the Apollo guidance and navigation system. The system was to remain intact; only software changes were allowed. The primary reason for this requirement was to retain the system's high degree of built-in integrity and reliability. A third basic design ground rule established early in the program was to make the pilot's interface with the computer as simple as possible. As a result, the pilot was given control over flight control functions rather than a direct communication with the computer. All functional changes (for example, a mode change) desired by the pilot were to be made through single switch actions. This allowed the pilot to perform functional changes rapidly and eliminated the possibility of incorrect entry or improper addressing which could have had dire consequences close to the ground or at high dynamic pressure. The last ground rule was to provide handling qualities that would be judged satisfactory by the pilots. A criterion based on C* response to a step pitch stick command (ref. 6) was used as a guide during the design of the longitudinal control system. Military Specification MIL-F-8785B, level 1, was used during the lateral-directional control system design.

These ground rules had further implications. For example, the interface equipment associated with the Apollo hardware established limits on the flight envelope for satisfactory operation. The analog-to-digital converter used for pilot stick inputs had only 45 usable discrete levels between zero and full stick. In the pitch axis each discrete level resulted in a specific level of aircraft normal acceleration, depending on stick gearing and dynamic pressure. In this instance, the acceleration increment became objectionable to the pilot within the basic flight envelope. Thus a new flight envelope limit was established at the dynamic pressure at which the stick quantization effect was not objectionable. Because of the design characteristics of the inertial measurement unit, some additional restrictions were placed on the airplane's maneuverability. These included a roll angular rate limit of 70 deg/sec and a pitch attitude limit of 70°.

Design Synthesis and Analysis

The closed-loop primary system was synthesized and analyzed by using two methods. The first was an analog sample and hold simulation which was useful in the learning process in that it pointed out the more general aspects of the digital control problem. For example, the acceptable range of sample rate, 25 to 50 samples per second, was defined. The effect of the folding phenomenon of sampled-data systems on the structural mode frequencies and the influence of common nonlinearities were also studied in this design phase. The second method used a digital synthesis program which provided linear analysis as a cross check and a background for the sample and hold simulation. Basic control laws, compensation, and logic were established by using these two methods. A specification for the control law software was then formulated, thus providing the basis for coding the flight software.

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PART I

ADVANCED CONTROL TECHNOLOGY AND ITS POTENTIAL
FOR FUTURE TRANSPORT AIRCRAFT

DRYDEN FLIGHT RESEARCH CENTER
EDWARDS, CALIFORNIA

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Design Verification and Refinement

With the delivery of portions of the flight hardware, including the Apollo computer and the coupling data unit, and early releases of flight software, design verification and refinement was started. A six-degree-of-freedom digital aerodynamic model of the F-8C airplane was used in conjunction with the flight hardware to form a partial hardware hybrid simulation.

The first two analysis methods did not consider any pilot interface, but the partial hardware hybrid simulation included a lunar module hand controller with which the F-8C model could be crudely flown. Coarse input quantization, a problem of importance later, did not become evident in this simulation because it was completely masked by the characteristics of the hand controller. This is one of the major disadvantages of any simulation which does not include major hardware elements.

Another important tool in the design verification and refinement was the batch process all-digital simulation. This simulation included the software being verified, an Apollo computer emulator, and a program representing the F-8C aerodynamics, all run on a large host computer. Powerful plotting routines made the internal computer parameters visible during each run. All the control system parameters were examined for reasonableness, particularly as they responded to mode and gain changes. One of the most useful plots was duty cycle versus run time. A typical variation of duty cycle during a maneuver is shown in figure 14 for a roll step. Because of some additional code for computation of stick nonlinearities when the stick was displaced from zero, additional computational time was required during this maneuver. This was reflected in an increase in the duty cycle, as shown in the figure. Other contributions to duty cycle were the interrupts from the motion sensors. The increase in roll rate produced a loss in available computation time roughly proportional to the roll rate. This loss of computation time effectively increased the duty cycle.

The last step in the design verification used the iron bird simulator. One problem — the coarse quantization of the pilot's stick inputs — was uncovered immediately. The problem became obvious once the hand controller was replaced with the actual center stick. The staircase shape of the computer output commands produced sharp responses at the secondary actuators which were unacceptable from the standpoint of mechanical motion and structural element excitation. Low-pass filtering of the computer output was undesirable because of its adverse effect on closed-loop performance. This suggested the use of a digital pilot prefilter that had not been anticipated in the control law specifications. The flight software had already been substantially verified, but fortunately the read-only memory had not yet been manufactured. The prefilter was quickly programed in software and the code was reverified. Consequently, there was essentially no effect on the overall schedule. This points out one of the significant advantages of a digital flight control system: Necessary changes can be made late in the design without affecting hardware procurement, packaging, or requalification. Although additional software verification will be required, it will not have the adverse effect on program schedules that is typical of a hardware redesign of an analog system.

Looking back on the various design and analysis tools, it is apparent that they

complemented one another. Confidence in the system grew each time an independent simulation or analysis gave results comparable to those obtained previously. The importance of having the pilot in the loop with as much actual hardware as possible was demonstrated vividly. In terms of time spent on verifying the various aspects of the design, the largest proportion was devoted to systematically verifying each logic function and mode transfer and the effects of failures. Another time-consuming aspect was the refinement of stick gradients and nonlinearities near zero stick. A much smaller proportion of time was spent on closed-loop augmentation character-istics, probably because of the good agreement generally found between the results from sampled-data analysis methods and simulation results.

RELATIONSHIP TO FUTURE APPLICATIONS

The configurations of future fly-by-wire systems will probably be strongly dependent on the specific missions for which they are designed. As such, each system will be unique in some respects, but will have a large degree of commonality with other fly-by-wire systems. The F-8 fly-by-wire system was unique in that it consisted of a simplex digital primary system, a triplex analog backup system, and no mechanical reversion capability. However, in this unique system were several features that will be relevant to the systems that will be required to achieve the advantages that active control offers. These features were, basically, dissimi-lar redundancy, single string software, and the experience associated with the digital system design.

Dissimilar Redundancy

The F-8 fly-by-wire system experience with two dissimilar systems provides information applicable to future systems which are likely to have dissimilar redundancy. Most of the problems were concerned with the synchronization of the two systems. Transfers from one system to another were handled differently, but the goal was to minimize transients caused by the transfer. In each instance, the system in control was tracked by the other system so that transients would be minimized. However, the primary system tracked the backup system by estimating the surface command of the backup system based on the pilot's control commands and trim inputs only. In transfers from the primary system to the backup system, the backup system tracked the output of the primary system. Although this eliminated the need to reconstruct the primary system signal propagation in the backup system, it did open the possibility for unusual initialization conditions when the transfer occurred during an abrupt maneuver. Another factor was that a transfer from the primary system to the backup system could have been initiated automatically as a result of a failure, thus the failure analysis had to consider all possible failures that could have resulted in a transfer. The timing of this transfer was critical in some instances when it could have coupled with the pilot's normal response to cause unacceptable conditions.

Some aspects of the dissimilar redundant system gave insight into redundancy management problems which may be expected in the future. The backup system

mechanized the trim function using a digital integrator to reduce drift. Because of differences in the sampling mechanisms between the two systems, large errors between the two trim signals were noted after extended flight with the backup system in control in which numerous trim inputs were made. Figure 15 illustrates the two sampling mechanisms. The primary system sampled trim commands every 90 milliseconds. If trim was being commanded at the time of the sample, the trim value was updated in the software. The backup control system did not update its digital trim integrator until a capacitor was charged up to a prescribed threshold. Although the capacitor began charging the instant the trim button was pushed, approximately 175 milliseconds were required before the first update of trim. As a result of these two sampling mechanisms, trim inputs of less than 175 milliseconds, but greater than 90 milliseconds, caused the primary system, but not the backup system, to update trim. To correct the problem, there would have had to be either some exchange of actual trim value information between the two systems or some form of verification that one system received the trim command before the other system updated the trim value. Each of these possible solutions would have required additional connections between the two systems, which would have been undesirable because they would have created new failure possibilities. For this particular research application, a procedural change in conjunction with close monitoring of telemetered data in the control room made modification of the system unnecessary.

Single String Software

Because a simplex digital system can have only a single program in control at one time, it can be described as a system with single string software. However, redundant digital systems with the same program in each computer also effectively have single string software. The experiences with the F-8 digital system software are closely related, then, to the multichannel digital systems expected in future civil transports. Generic software failures would have equivalent effects on any system with single string software, regardless of the system's redundancy. The software controls described in reference 2 suggest that careful verification will always be necessary, but that the confidence necessary for man-rating the software can be established.

Another factor that emphasized the importance of man-rated software was that the single string software had full-authority control over the control surfaces; thus it was obviously flight critical. Digital systems will be called on to perform more and more flight-critical functions and, on the basis of our experience, can be depended on to perform with high integrity.

Removal of all mechanical reversion capability before the first flight had a significant effect on the entire design and verification process. It forced an approach that would establish complete confidence in the system on the basis of simulation alone. If the alternate approach had been taken, that of retaining a mechanical link, the most probable flight-test procedure would have been to fly to a safe altitude using the mechanical system and then engage the fly-by-wire system. After confidence was gained at altitude, the more critical flight safety functions, such as takeoff and landing, would have been encompassed gradually.

Design Experience

On the basis of the F-8 digital fly-by-wire design experience, several recommendations can be made regarding the design of digital control systems for future civil transports. Many of these recommendations correspond simply to good design practice. Analyzing closed-loop performance using standard sampled-data analysis techniques such as z-plane root locus can be relied on to give good agreement with more complete simulations. Several forms of simulation and analysis should be used to build confidence in the system before the first flight. A simulation that includes as much actual hardware as possible is important in correctly assessing system performance. The interface with the pilot is particularly important.

CONCLUDING REMARKS

The feasibility of a digital fly-by-wire system was assessed by replacing the mechanical flight control system of an F-8C airplane with a digital primary and an analog backup fly-by-wire system. The design and verification procedures which will be necessary if flight-critical active control is to be used in future aircraft were established and successfully applied as part of the flight program. Careful application of standard sampled-data design methods and systematic verification of control system hardware and software using complete simulations resulted in a digital fly-by-wire system with extremely high integrity. The successful use of single string software in a full-authority flight control system demonstrated the high level of confidence which can be placed in digital flight control.

The experience with the F-8 digital fly-by-wire system pointed up several factors that will be important in the successful design of future full-time, flight-critical digital control systems:

(1) Batch process all-digital simulation was extremely helpful in tracing internal computer variables and in providing visibility to system response during mode changes.

(2) A complete piloted simulation with actual flight control system hardware provided important results relative to the pilot/stick interface that had not been obtained in earlier simulations which did not include the actual control stick.

(3) The largest portion of the design and verification effort was devoted to logic functions, such as mode transfers, and the effects of failures.

(4) Software changes made late in the design to correct hardware-related problems had a negligible effect on the program schedule.

A major aspect of the F-8 digital fly-by-wire system which will have application to future systems was its dissimilar redundancy. Failure isolation between the primary and the backup systems was achieved as desired, although some problems were encountered with intersystem synchronization.

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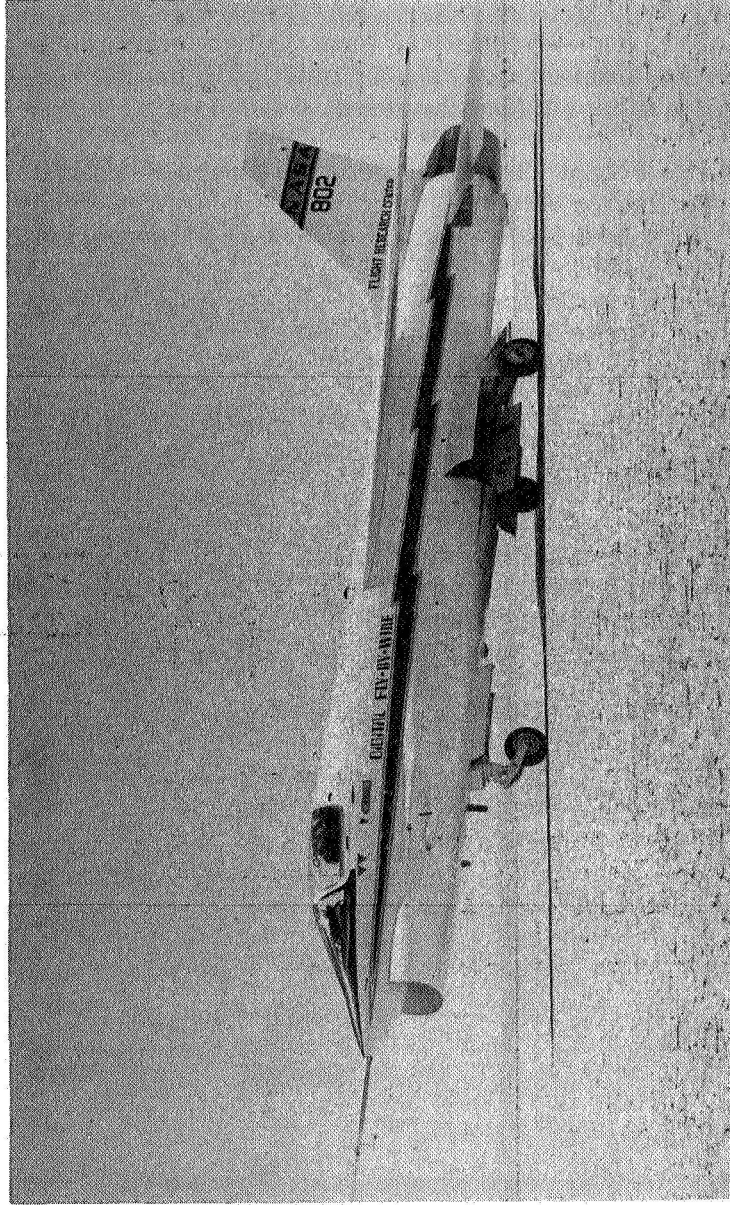
Figure 1. F-8 digital fly-by-wire airplane.
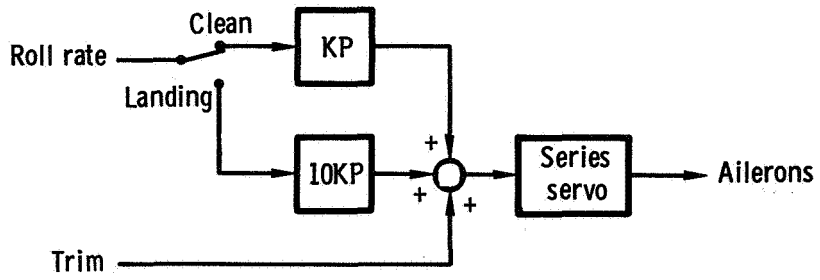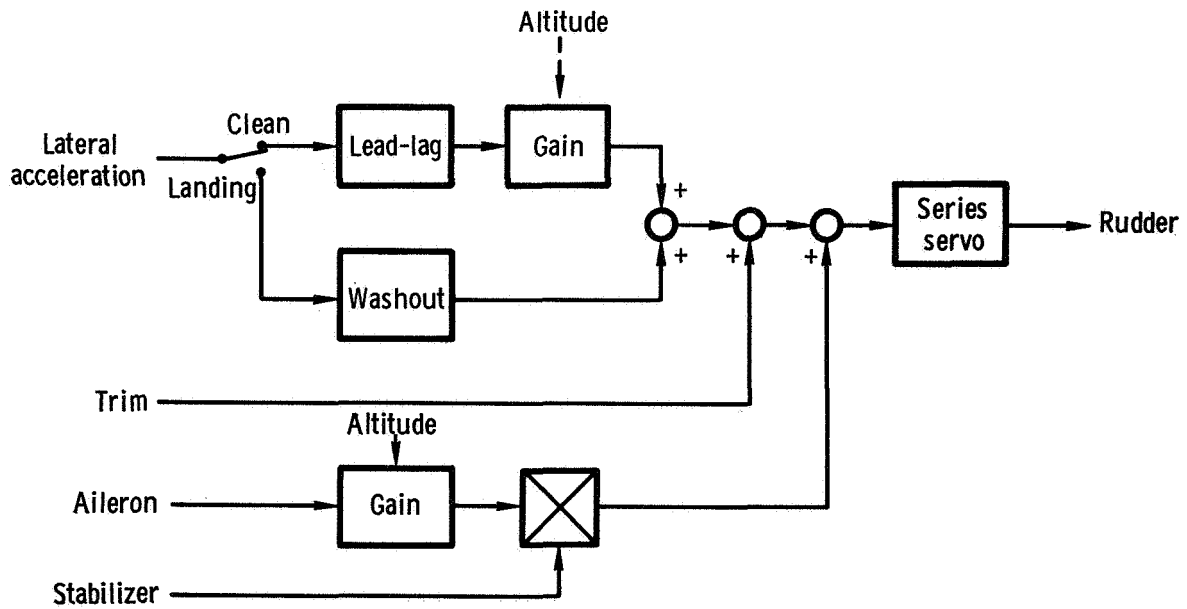
(a) Roll-axis damper.

(b) Yaw-axis damper.

Figure 2. Standard F-8C roll and yaw stability augmentation systems.
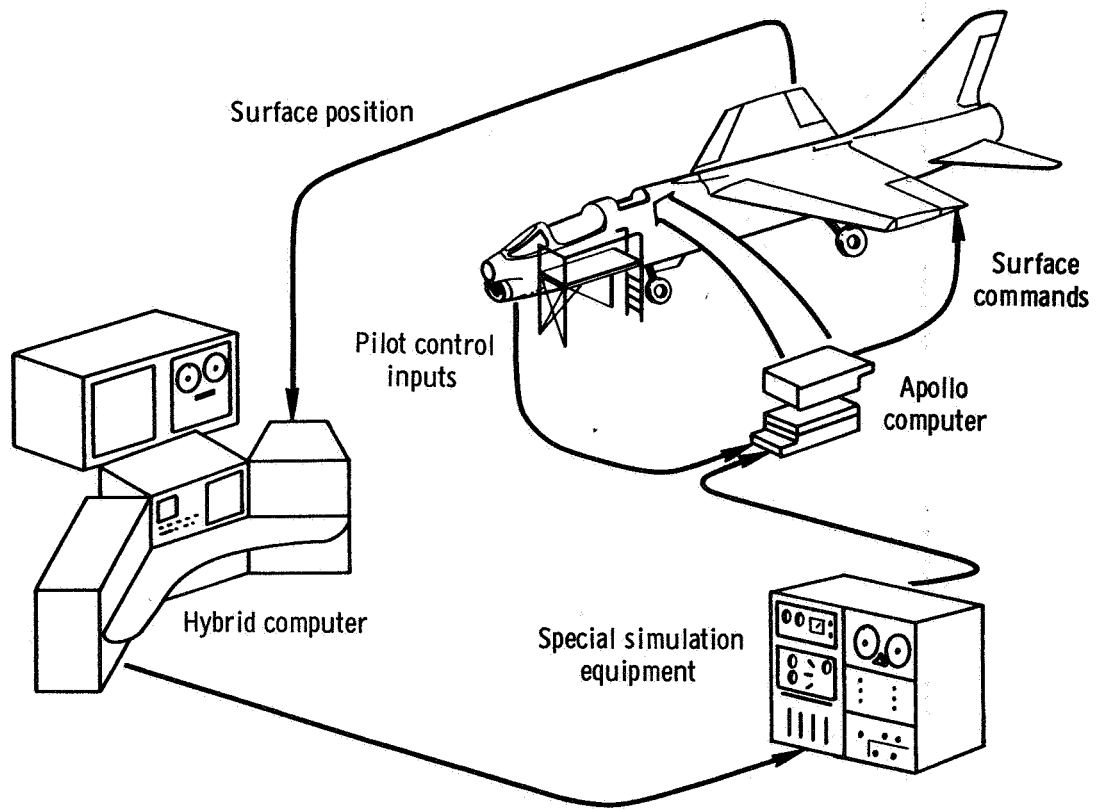
Figure 3. F-8C iron bird simulator.
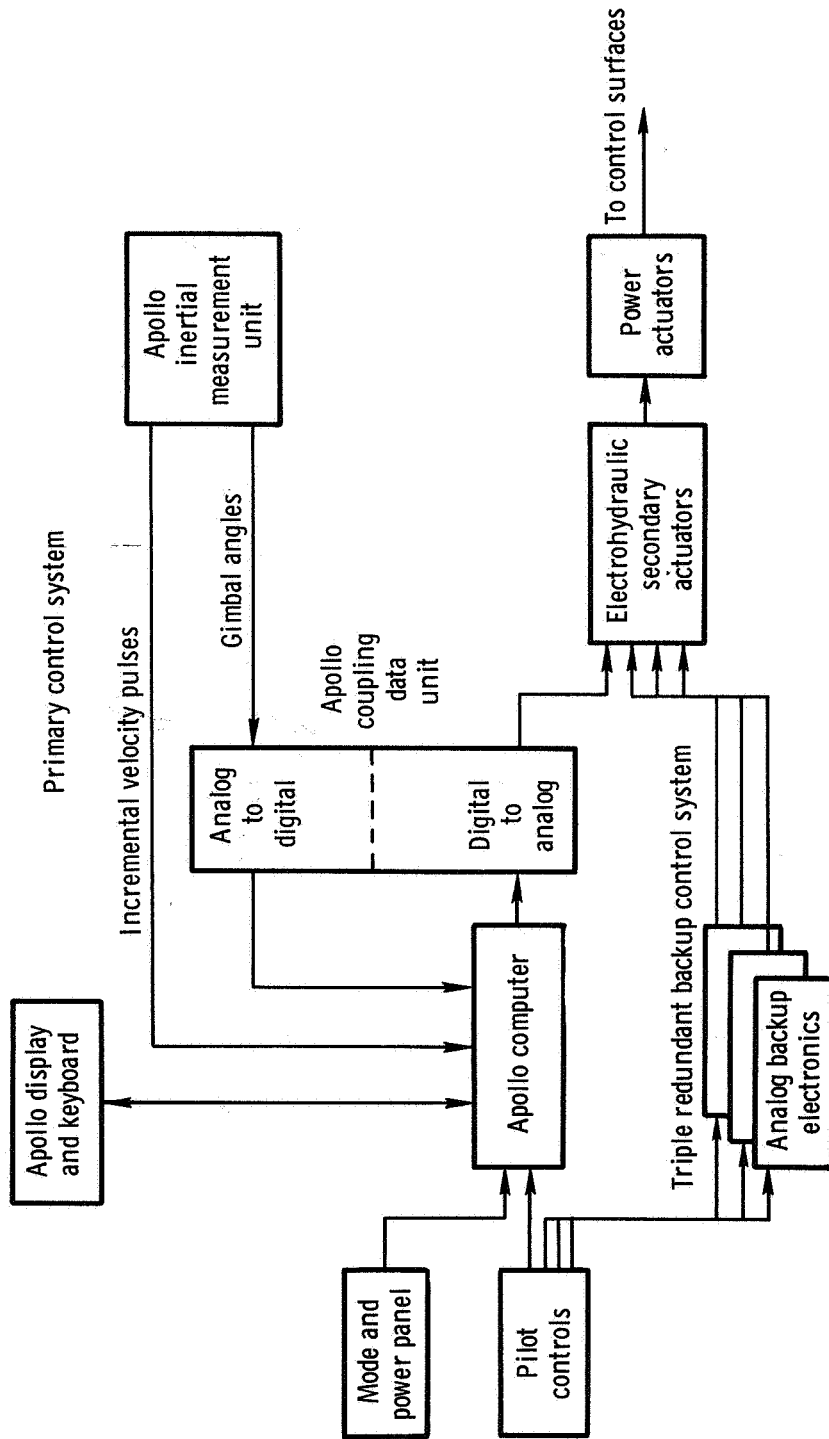
Figure 4. F-8 digital fly-by-wire control system components.

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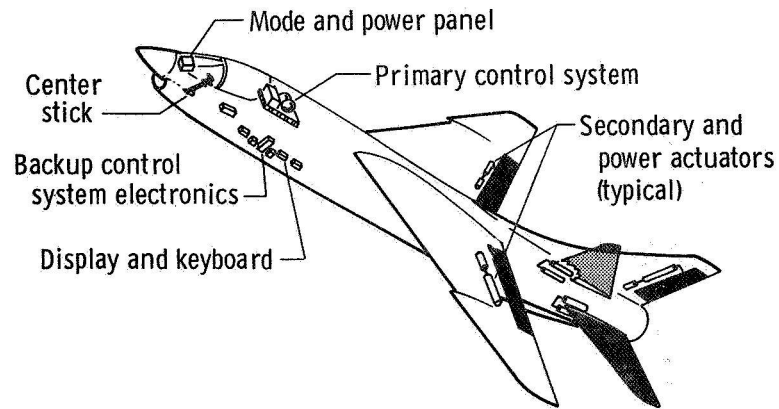
Figure 5. Location of fly-by-wire control system in F-8C airplane.
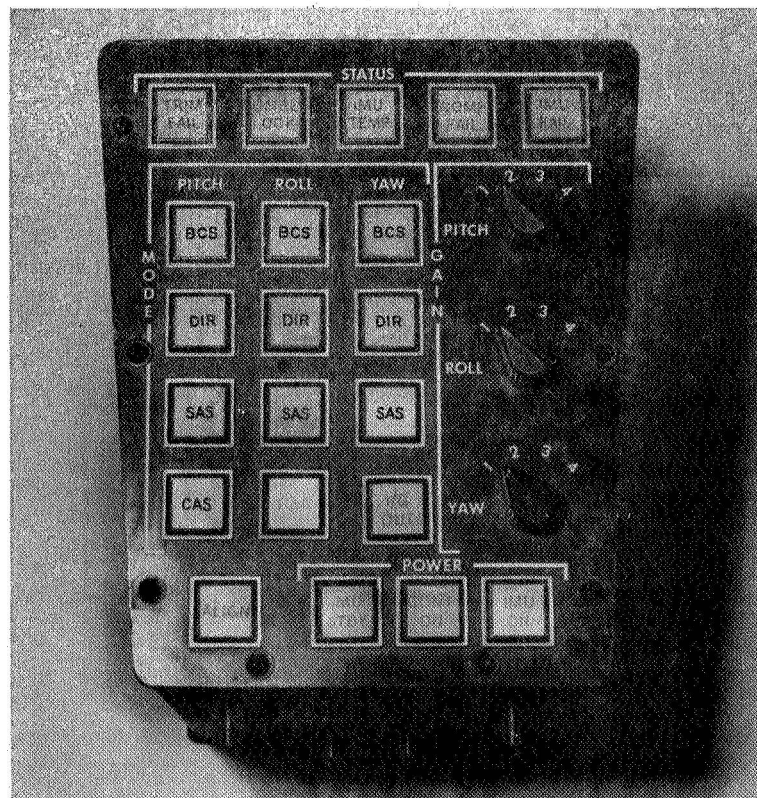
Figure 6. Mode and power panel.

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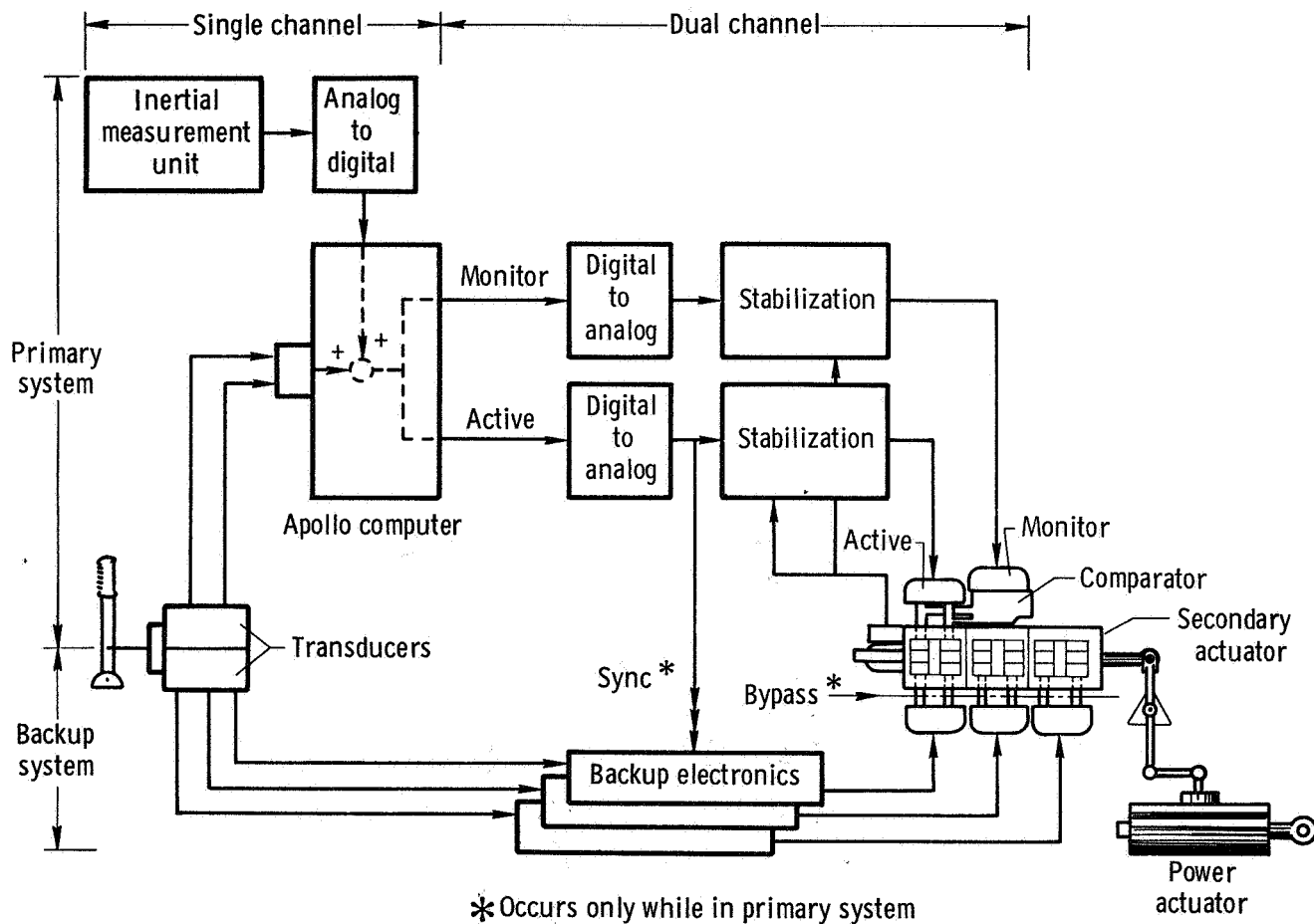
Figure 7. F-8 digital fly-by-wire system mechanization.
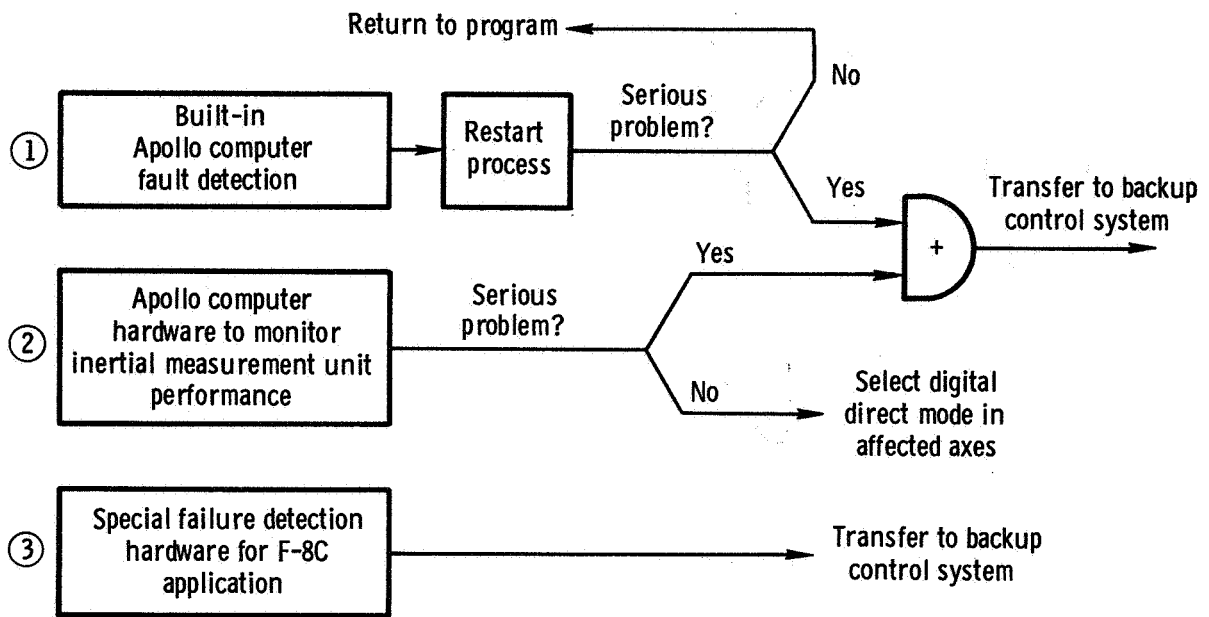
Figure 8. Digital system failure detection and reporting system.
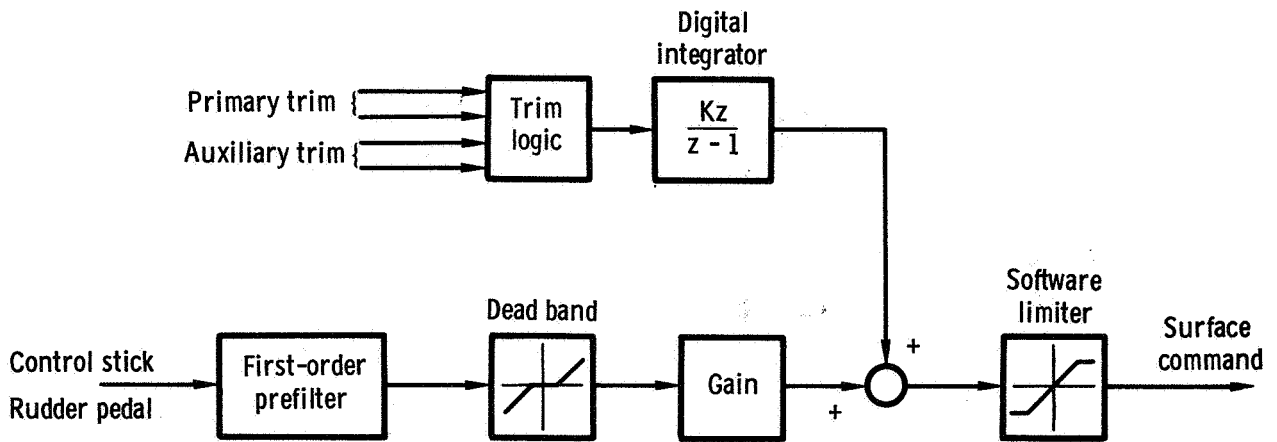
Figure 9. Direct modes for pitch, roll, and yaw axes.

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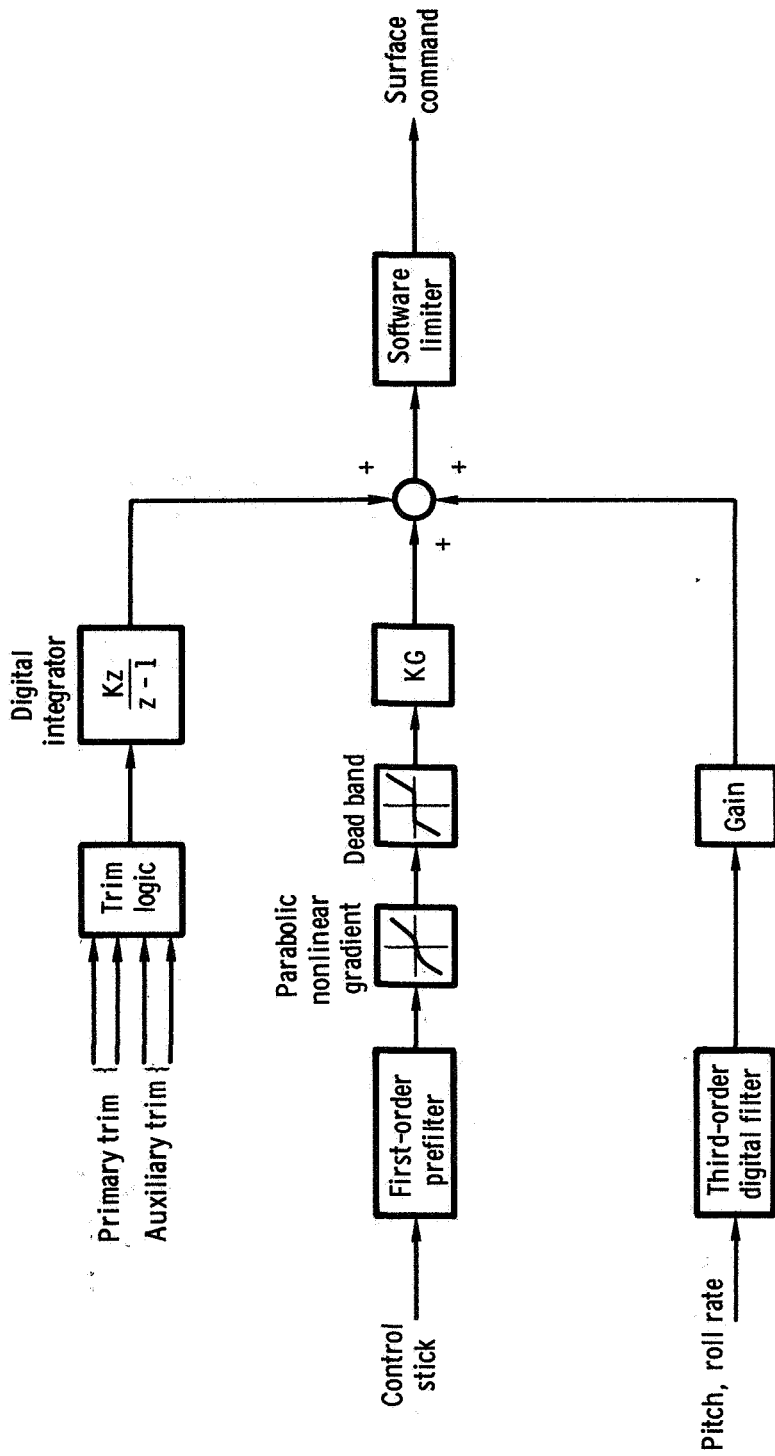
Figure 10. Pitch and roll stability augmentation systems.
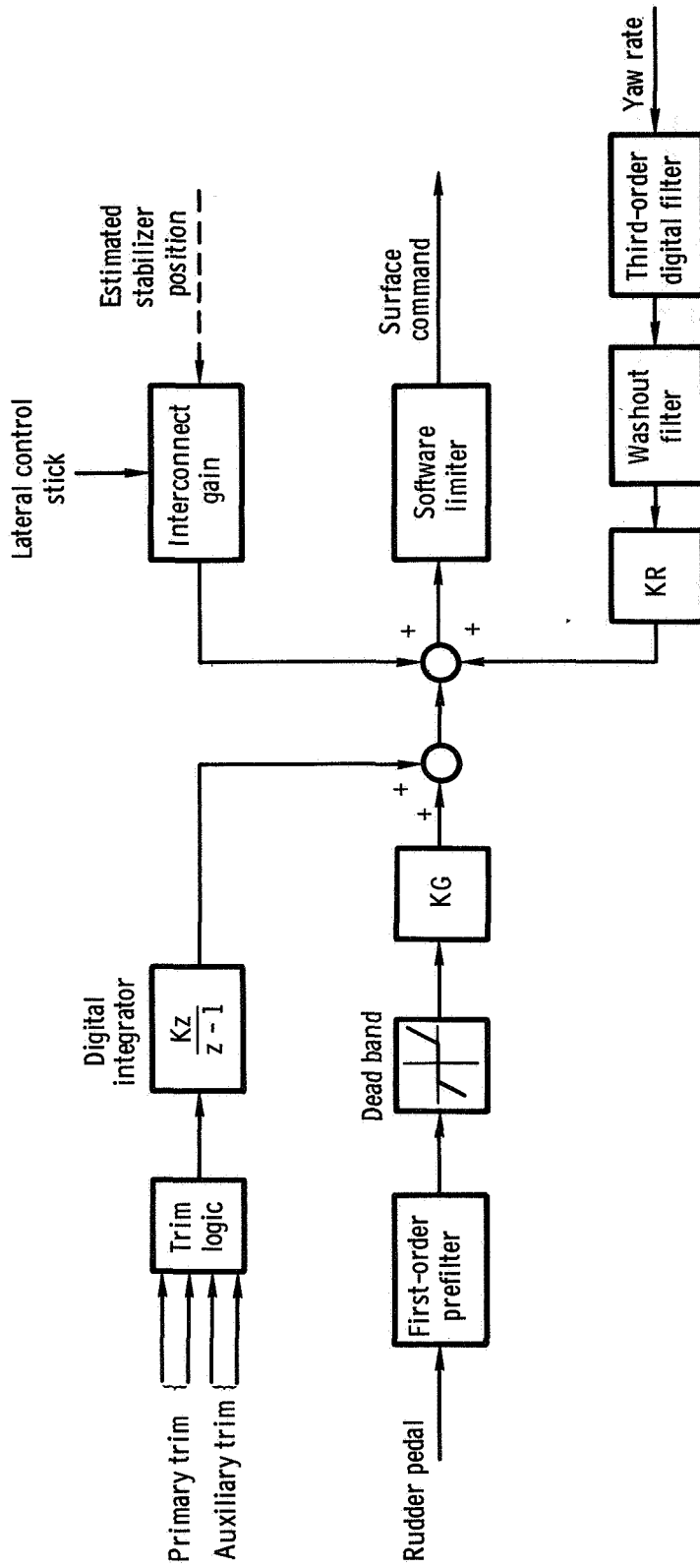
Figure 11. Yaw stability augmentation system.
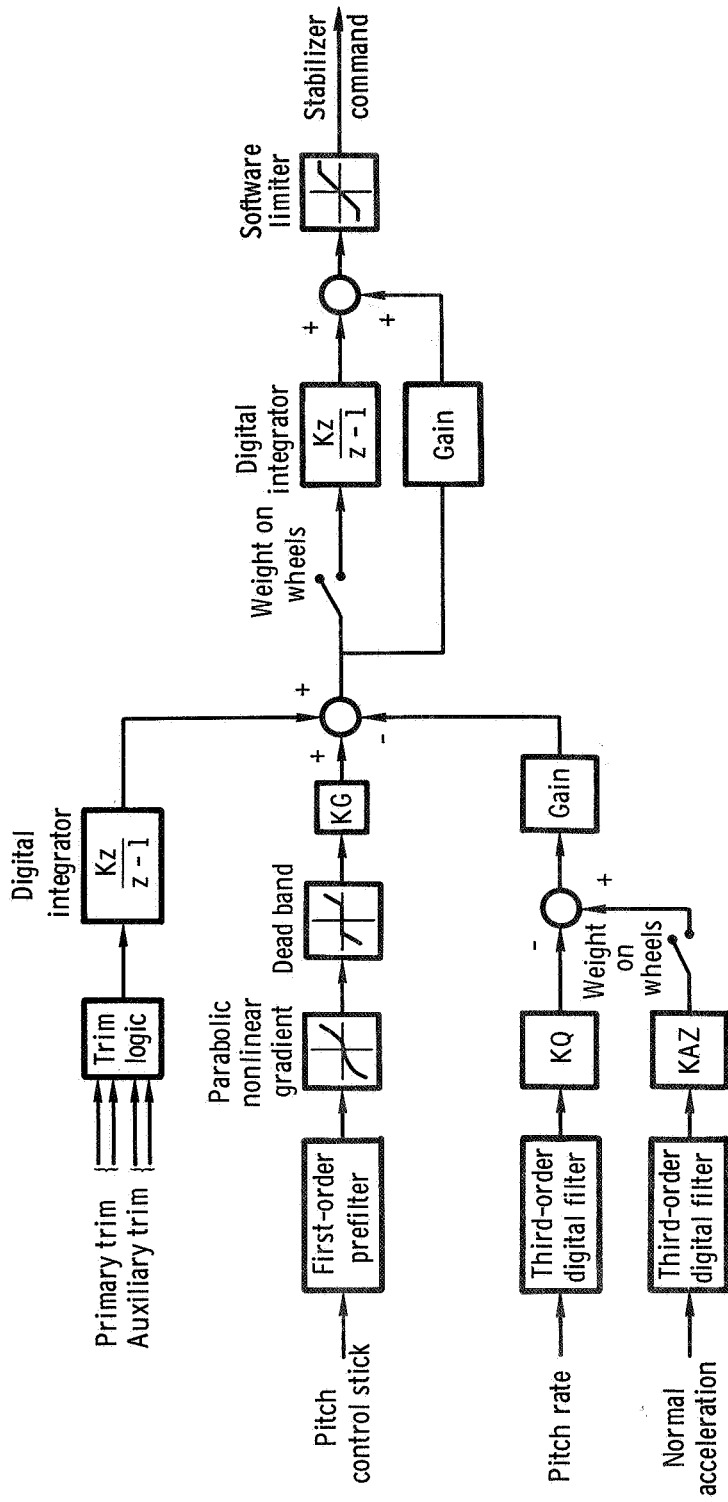
Figure 12. Pitch command augmentation system.

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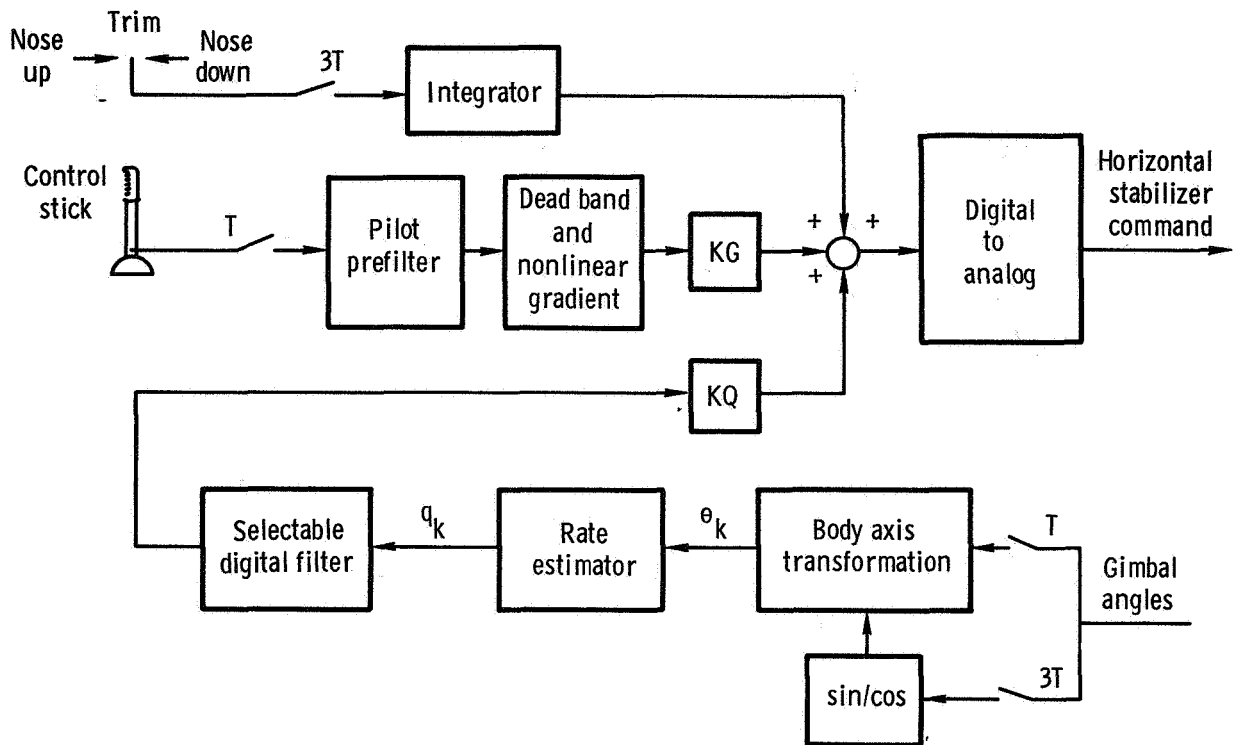
Figure 13. Pitch stability augmentation system mode as a sampled-data system.
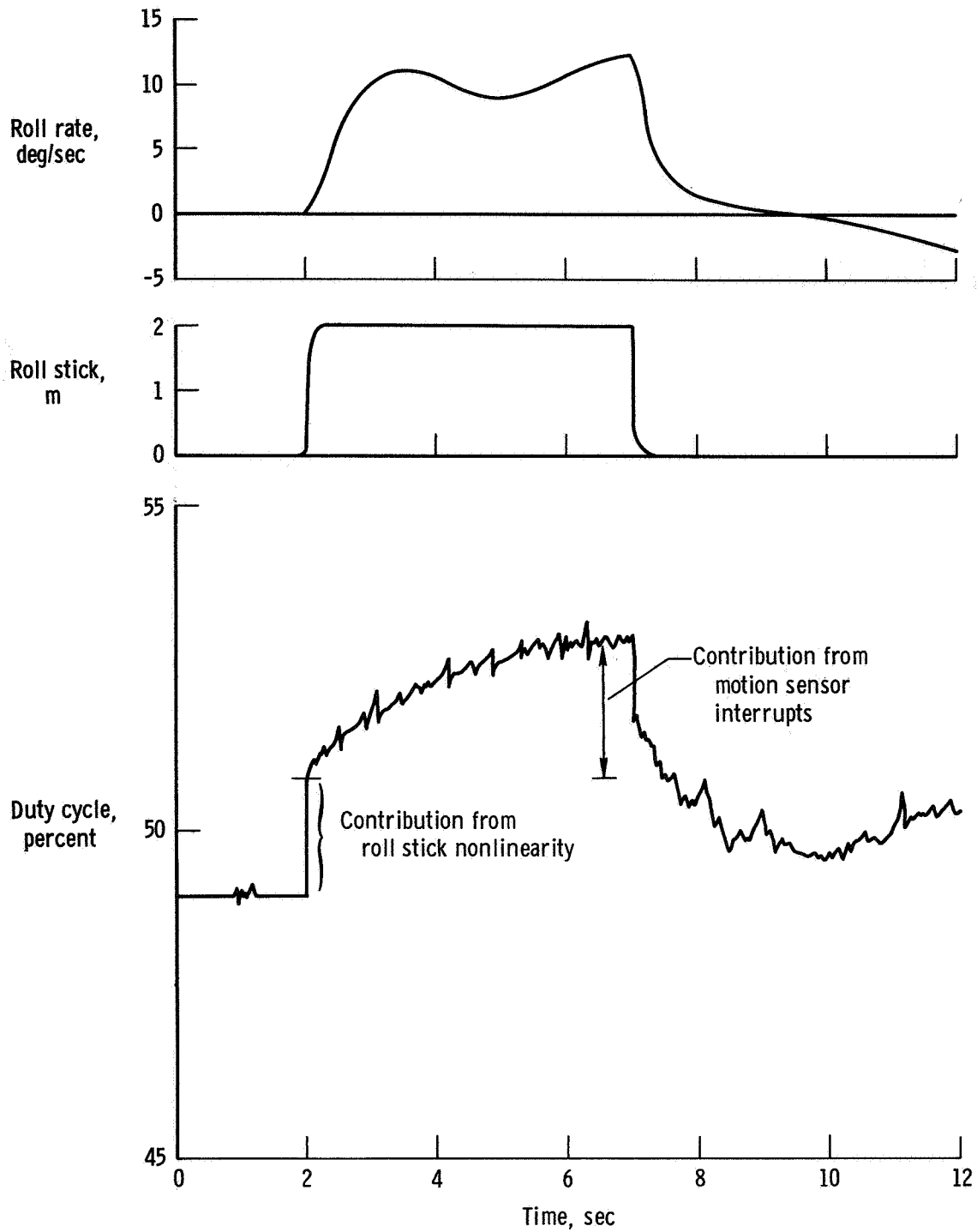
Figure 14. Duty cycle variation during roll step maneuver.
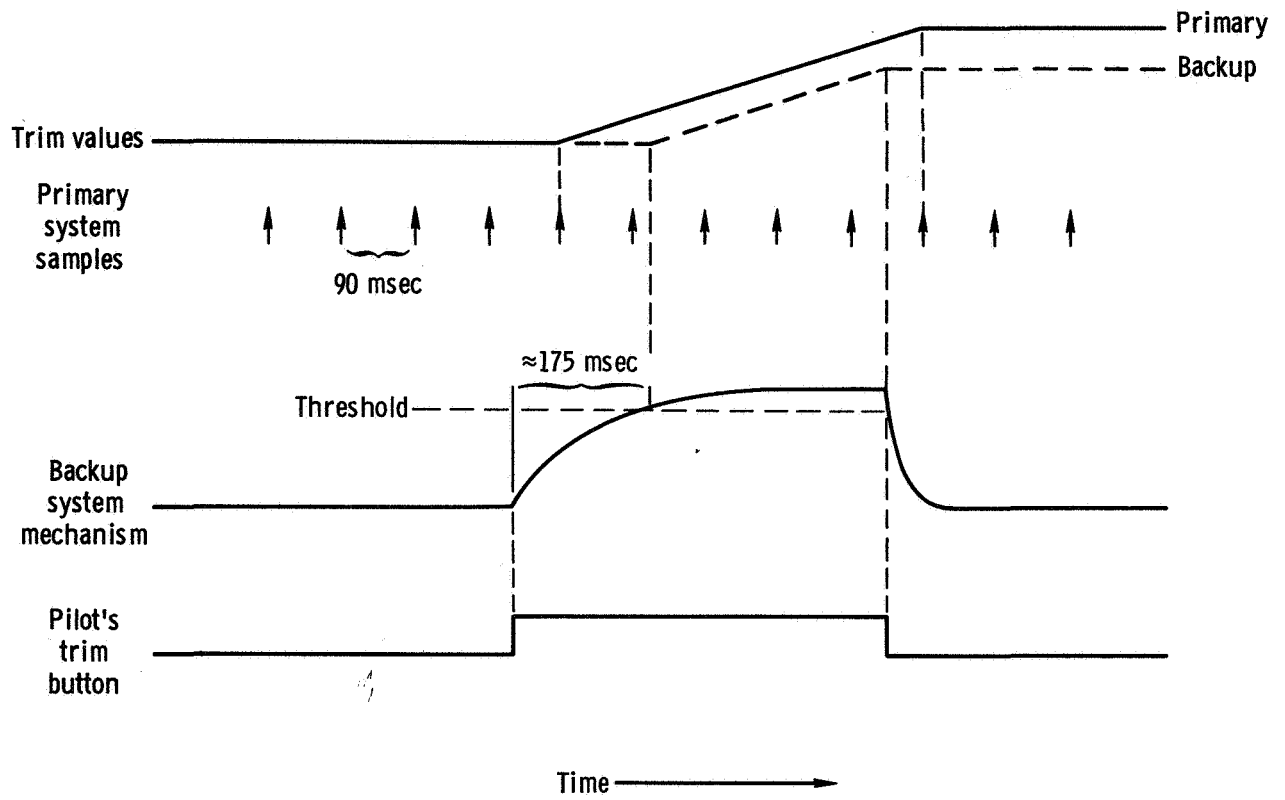
Figure 15. Trim sampling mechanisms of the primary and backup systems.

MECHANIZATION OF AND EXPERIENCE WITH A

TRIPLEX FLY-BY-WIRE BACKUP CONTROL SYSTEM

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and

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SUMMARY

A redundant three-axis analog control system was designed and developed to back up a digital fly-by-wire control system for an F-8C airplane. Forty-two flights, involving 58 hours of flight time, were flown by six pilots. The mechanization and operational experience with the backup control system, the problems involved in synchronizing it with the primary system, and the reliability of the system are discussed.

The backup control system was dissimilar to the primary system, and it provided satisfactory handling through the flight envelope evaluated. Limited flight tests of a variety of control tasks showed that control was also satisfactory when the backup control system was controlled by a minimum-displacement (force) side stick.

The operational reliability of the F-8 digital fly-by-wire control system was satisfactory, with no unintentional downmodes to the backup control system in flight. The ground and flight reliability of the system's components is discussed.

INTRODUCTION

A control system consisting of a primary digital fly-by-wire system and a dissimilar triplex analog backup system was flight tested in an F-8C airplane by the NASA Flight Research Center. The mechanical linkages of the original F-8C control system were removed except for cockpit stick and pedal centering and feel. A single channel digital computer, the associated electronics, a power-generating system, and electrohydraulic secondary actuators made up the primary control

system. A triplex backup control system provided the redundancy required for manned flight and gave the total system two-failure-operate reliability.

The main components of the backup control system were the sensors, the electronics, and the secondary actuators. The system was analog for signal processing, had no feedback for stability augmentation, and was designed to provide emergency return-home capability with airplane handling qualities equal to those of the basic F-8C airplane. The F-8C airplane can be flown through most of its flight envelope without augmentation.

This paper describes the backup control system and its integration with the primary control system, which is described in detail in reference 1. The mechanization of and operational experience with the primary and backup control systems are discussed. Some aspects of the primary and backup control system design were unique; however, many of the design features would apply to fly-by-wire control systems in active control aircraft. The reliability of the total system during the program is described.

A limited flight test evaluation of the backup control system was conducted using a minimum-displacement (force) side stick controller for pitch and roll control. These evaluations represent most of the maneuvering experience with the backup control system.

DESIGN FEATURES

The backup control system was designed to provide redundancy for the F-8 digital fly-by-wire control system. It was a triplex analog fly-by-wire control-stick-to-control-surface system in which the electronic trim, sensor and electronics equalization, primary control system synchronization, and servo and electronics monitoring were independent of the primary control system. The system incorporated several innovations that are common in modern electronics equipment but not as common in airplane control system hardware. These design features are described in the following sections.

A functional block diagram of the F-8 digital fly-by-wire control system is presented in figure 1. The upper portion of the figure is the primary control system, and the lower portion is the backup control system. The secondary actuators are shared between the primary and backup systems, and the primary control electronics provide the interface between the digital-to-analog converters of the primary system. The secondary actuators and the synchronization between the primary and backup systems are also discussed in this paper.

Triplex Channels

The backup control system consisted of three identical computing channels, one for each airplane control axis. The system provided an interface between the

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triplex control stick and pedal position sensors and the triplex servovalves. In addition, the three backup control system channels, including the sensors, electronics, and servos, were powered by three isolated power busses that were connected to a common power source.

The servo commands, which consisted of stick and pedal commands that were summed with the trim and equalization signals, were processed by voters in each channel to insure that the three backup channels tracked. The voter selected the middle value of the three channel commands to drive the control valve in each channel. For certain types of failures in the voter, actuator, or servo electronics, the failed channel would be detected and the servovalve associated with the failed channel would be disengaged. Therefore, the backup control system was operational after one or more failures.

Synchronization

An integrator in each axis of the three backup control system channels provided electronic trim, equalization, and synchronization. When a primary channel was engaged, the backup control system servo commands were synchronized with the primary servo commands with these integrators. These inputs to the backup control system voters tracked the primary channel servo commands, even though variations in control sensor outputs and in intersystem control laws existed. Continuous synchronization of the backup with the primary control system was necessary to minimize control surface transients during the switchover from the primary to the backup control system. Switchover occurred if there was a failure in the primary system or if disengagement was commanded by the pilot. The synchronization network had a bandwidth of approximately 2.5 hertz.

Equalization and Trim

When the backup control system was engaged, the integrators performed the backup control system trim and equalization function. Trim was accomplished by applying a fixed reference to the integrator, changing trim at a fixed rate. The integrator output was then summed with the control stick or pedal position inputs to form the total surface command. Since the trim inputs, sensor position inputs, and electronic gains were not necessarily the same in each backup control system channel, equalization was included to reduce errors between channels. Limited equalization, combined with the voters, produced essentially identical channel servo commands to the three backup control system servovalves and minimized the force fight between the secondary actuator pistons.

The trim and equalization functions required a low or zero drift integrator. The backup control system integrator design, which was classified as having zero drift, used digital techniques to accomplish the zero drift or memory function and analog techniques for the integration function.

Backup Control System Monitoring

Electronic and servo signals were monitored at two points within the backup control system. The channel voter output was compared with the voter input. If the signal difference was greater than the set threshold, the monitor was latched and the electronic channel was reported failed.

The other monitoring point was the backup control system servos. Backup control system servo monitoring was accomplished by cross-channel comparison of the differential pressure signals from each of the three servovalves. This detected excessive force fights within the backup control system servos. A preset difference in two of the three differential pressure signals resulted in latching the common servo monitor, disengaging the failed servo, and reporting the failure to the pilot. When a failure was detected and the failed servo disengaged, the resulting surface transient was minimized by the operating characteristics of the force-summed actuator.

Self-Test Procedure

Preflight testing was accomplished by an automatic self-test procedure that provided a pseudo-end-to-end testing of the system. The self-test involved the introduction of a logic-controlled stimulus and the disabling of circuit functions and used in-flight monitors to indicate the response. The use of the in-flight monitors as the self-test feedback elements served to check the channel signal paths and the operation of the in-flight monitors. This resulted in a "bang-bang" type of test with no indication of system degradation.

A block diagram of the self-test unit is shown in figure 2. The power for the self-test was routed to the computing electronic assemblies only after the self-test power switch was closed and the self-test start switch was depressed. A counter started to count and addressed the read-only memory, which was preprogramed for each particular test to activate certain stimuli and disable certain circuits in the electronic assemblies. The test results were compared with the predicted results, which were stored in the read-only memories in the diagnostic analysis circuitry. The self-test automatically stepped to the next test if the test results were as predicted. This procedure was repeated with different combinations of stimuli and disabling circuits active until the test was complete and a GO signal was reported in each airplane control axis.

If the test results from the electronic assemblies were not as predicted, the test sequence was stopped and a diagnostic routine was initiated. The diagnostic analysis circuitry analyzed the test results with respect to the predicted results to determine where the failure occurred. The diagnosis was indicated on the self-test diagnostic readout.

Status Engage Panel

The status engage panel was in the left cockpit console. It housed all the servo engage switches and servo status lights and indicated the status of the

backup control system electronics. The panel also contained the self-test program, power switch, and diagnostic readout to indicate a failed self-test condition or satisfactory completion of the self-test. The servo switches were three-position positive-action switches labeled Auto-Off and Manual. Even though five secondary actuators were utilized for the three control system axes, only three switches and servo status lights were used for the three primary control channels, whereas individual channel switches and status lights were mechanized for each backup control valve. The lights indicated when the various comparator networks had exceeded preset values. The light was also a reset switch that sent a reset pulse to its comparator. The servo status lights for backup servo systems 2, 3, and 4 lit up after any two common comparators tripped. That is, the left pitch 2-3 comparator and the left pitch 4-2 comparator lit the left pitch number 2 light when both comparators tripped. The logic for the primary control system pitch servos was that if either the left or the right pitch channel indicated failure, the number 1 pitch status light lit, and control was switched from the primary to the backup control system. The servo system logic was designed to provide a manual override capability for any channel per actuator regardless of the remaining servo system switch positions.

SYSTEM COMPONENTS

The approximate locations of the control system components in the F-8 digital fly-by-wire test airplane are shown in figure 3. As might be expected, using the F-8C airplane as a test-bed resulted in some design problems that were unique to the F-8C configuration. A major problem was the requirement for different control gearing for the wing-up (approach and landing) and wing-down (cruise) positions. A pair of dual wing potentiometers was mounted to provide an electrical signal proportional to wing position to droop the ailerons for flaps and to provide automatic trim of the horizontal tail. Other system components that provided control, signal conditioning, and actuation are described below.

Control

Stick and pedal transducers.— Two transducers that each contained triplex redundant linear variable differential transformers (LVDT's) were connected to the existing F-8C flight control linkage to provide electrical signals as functions of the pilot's stick and rudder commands. One transducer was provided for the primary control system, and one was provided for the backup control system for each airplane axis of control. The pitch transducers were on the right and the roll transducers were on the left side of the airplane underneath the primary flight pallet. Because of rudder cable stretch, the two rudder transducers were installed in the base of the vertical tail.

Each transducer assembly contained isolated sensors for excitation and signal output to drive as many as three separate control paths. All the transducers were linear, except for the pitch transducers for the backup control system, and all had an electrical stroke of ±1.5 centimeters. The pitch transducer for the backup control system transducer had a special winding to provide parabolic stick shaping.

Stick and pedal gearing.— An attempt was made to duplicate the control authority and gearing of the original F-8C airplane in the F-8 digital fly-by-wire airplane. The final gearings are shown in figures 4(a) to 4(c) for the pitch, roll, and yaw axes.

The data presented in figure 4(a) indicate that there was reasonable agreement between the pitch stick gearing of the fly-by-wire airplane and that of a conventional F-8C airplane. Only the wing-down data are presented for zero trim command. With the wing in the up position, the horizontal stabilizer surface was biased 5° from the wing-down position, and the zero stick position corresponded to zero surface position.

Figure 4(b) shows the left aileron position as a function of lateral stick position for a wing-down and a wing-up configuration with zero trim command. The gradients are nearly the same for all backup control system commands except for the wing-up right stick command, where the gradient is higher than in the conventional F-8C airplane. The fly-by-wire gradients were symmetrical for both wing positions, whereas the wing-up gradient was not symmetrical (differential aileron) for the conventional F-8C airplane. The aileron did not move down as far as it moved up for a given stick command.

Figure 4(c) shows rudder displacement as a function of pedal force. Gradients are shown for the wing-up and the wing-down configurations. The higher gradient was used with the wing-down configuration. The pedal forces were provided by the existing F-8C mechanism. The gradients show good agreement for both wing positions. The backup control system deadband was slightly larger.

Side stick.— The side stick sensor flight tested during the program was a two-axis, four-channel, minimum-displacement transducer. The principal of operation for the transducer was that an applied force at the stick grip caused a flexure-supported tube assembly to move quadruplex LVDT's that generated a voltage proportional to the applied force. The side stick transducer was recessed in the right cockpit console to allow the pilot to sit comfortably in the seat with his arm in a natural position.

Side stick gradients.— The side stick gradients flight tested are shown in figures 5(a) and 5(b) for the pitch and roll axes, respectively. Figure 5(a) shows the pitch stick force as a function of elevator surface position for both wing positions. The circuit mechanization consisted of a deadband, a low gradient, and a high gradient for both a pull and a push force. The variable high gradient was mechanized to function only with the wing down, and it was controlled by a switch in the cockpit. In figures 5(a) and 5(b), switch positions increase with increasing stick gradient. The side stick authority was always less than the center stick authority.

Electronics

Backup control system.— Three identical backup control system electronics boxes were the heart of the backup control system. Each box contained all the

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signal processing, engage logic, monitoring, and dc power necessary for a single backup control system channel in the pitch, roll, and yaw axes. A block diagram of a single channel in the roll axis (channel 3) is shown in figure 6. Except for scaling, trim rate, and the gearing change with wing position, the network for the pitch axis was basically the same. The yaw axis differed in scaling and trim rate, and a limiter was added just after the voter. The voter output drove only one actuator network.

Figure 6 shows one electrical comparator across the voter and a single comparator across each backup valve. In total, there were eight comparators per backup control electronics box. The trip level of the electrical comparators was set at 3.0 volts, which was approximately one-third the maximum voltage for each axis. This corresponds to a stick displacement of approximately 2.5 centimeters for roll and 5 centimeters for pitch. The differential pressure comparators were set to trip at 2.4 volts, which represents a differential pressure error of 8273 kN/m^2.

Primary control electronics. — The primary control system electronics box contained the signal interface between the computer's digital-to-analog converter outputs and each secondary actuator for the airplane's pitch, roll, and yaw axes. A simplified block diagram of a typical primary signal circuit is shown in figure 7. For each control axis, there were two identical signal paths, the active and monitor channels, from the computer to the control valves of the respective secondary actuator. The primary control electronics box contained two 5-hertz second-order smoothing filters in each of the three axes. Follow-up signals from the secondary actuator were biased with the wing position voltage for the pitch and roll actuators. The signal was then divided for summing and sent directly to the monitor servo amplifier or quadruplex voter and processed with the three comparable signals from the backup control system. In conjunction with the hydrologic comparator, this provided hard-over protection from open servo follow-up signals.

The primary control electronics box also contained engage logic, monitoring, and the dc power supply for the box and the primary secondary actuators. A separate return comparator was used to monitor the difference between comparable points in each axis of the primary and backup control systems. When the error was greater than 3°, 4°, and 3° for the elevator, aileron, and rudder, respectively, the primary control system could not be engaged. However, the backup control system could always be selected.

Side stick electronics. — The installation of a side stick required additional electronics that could not be readily added to the backup control electronics boxes. Therefore, the additional electronic networks needed to provide demodulation, deadband, shaping, and gradient (fig. 8) were mechanized to interface between the side stick transducer and the backup control electronics boxes. The triplex electronics concept was maintained from sensor output to the appropriate channel sum points in the backup control electronics boxes.

Secondary Actuators

The secondary actuator (fig. 9) was a four-channel electrohydraulic actuator designed to convert electrical signals to surface motion and to have two-fail-operate

capability. There were five secondary actuators: two for roll, two for pitch, and one for yaw. The mechanization of the secondary actuator was an active/standby configuration which consisted of two valves in the active configuration and three valves in the standby configuration. The secondary actuator was designed to be controlled by any of the four electrohydraulic control channels. Each primary channel commanded one active valve to position the actuator; the second valve, in conjunction with the hydrologic failure detection network, was used for self-monitoring. The actuator standby or backup channels commanded by the backup control system consisted of three force-summed channels with electronic failure detection.

Two-stage flapper nozzle servovalves were used for the primary system active and monitor valves. During normal operation, these valves received separate commands, and the active valve positioned the actuator ram as required. The failure detection for the primary control system was provided by a hydraulic comparator network. A comparator spool was balanced between the force exerted by two springs and the output pressures from the active and monitor spools. If a pressure difference beyond a predetermined threshold existed, motion of the comparator spool dumped the supply pressure to the return line, which caused the primary engage valve to reposition and block the commands from the active servovalve. Errors that could cause the hydrologic comparator to trip were measured in terms of either single control surface deflection or commanded current. These were 4°, 4°, and 1.5° for the elevator, aileron, and rudder control surfaces, respectively, or one-half the maximum valve current.

A dual pressure switch was installed in the primary hydraulic circuit of each secondary actuator to sense minimum pressure. The switch caused the primary servo system to disengage at 4137 kN/m^2, and a pressure of at least 5516 kN/m^2 was required for manual reengagement. When the primary channel tripped, the pressure switch opened, which caused the engage logic to automatically energize the three solenoids in the backup control system and to transfer control to the three single-stage jet pipe servovalves (servo systems 2, 3, and 4).

The backup system servos were monitored by differential pressure transducers that were installed across the output legs of each jet pipe servovalve. Each differential pressure signal was compared with the other two differential pressure signals for each actuator. The comparison was made in the backup control electronics boxes.

The secondary actuators were modular in construction and were designed around three tandem pistons on a common shaft. The primary channel and one backup control system channel shared one of the piston networks, and the remaining pistons were controlled by the other two backup systems. Each secondary actuator was supplied by two separate hydraulic systems. Figure 9 shows the secondary actuator mechanization in the primary configuration.

The figure shows that the valve outputs of backup channels 2 and 4 were blocked by separate hydraulic engage valves and that the cylinders bypassed fluid as the ram moved. Backup channel 3 was blocked by an engage valve with a slightly different design.

The servo position loop was closed electrically for each channel in the elec-

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tronics boxes. The electrical signal utilized for the servo ram position came from the quadruplex redundant LVDT in each servo actuator shaft. The stroke of all the secondary actuators was 5 centimeters, and by utilizing the necessary mechanical linkage, the desired control surface rotation was obtained for all five surfaces.

Response characteristics. — Ground test data were taken for each actuator with different valve combinations. The performance of each secondary actuator was a function of the engaged servovalves. The primary valves had much higher response than the valves used in the backup servo systems (systems 2, 3, 4), but because of hardware problems (ref. 2) the primary servo amplifier gain was lowered.

A typical frequency response curve of the elevator secondary actuator with the primary servovalve in control is shown in figure 10. The figure compares the flight tested servo amplifier gain, 5 milliamperes per volt, with the designed servo amplifier gain of 22 milliamperes per volt. Even though the pitch servo bandwidth flight tested was 6 hertz, the addition of a second-order filter reduced the effective servo bandwidth to 2.5 hertz. The total bandwidth of the filter, secondary actuator, and elevator power actuator was 1.5 hertz for an elevator surface amplitude of 1° peak to peak, normalized at 0.5 hertz.

The frequency response of the same pitch secondary actuator when controlled by the backup control system valves is shown in figure 11. Data are compared for two valve drive configurations. One data set was obtained with a single backup control system channel valve in control of the secondary actuator. The other data set was obtained with all three backup valves in control. The single backup control system channel bandwidth was 7 hertz, and the bandwidth of the three backup control system channels was 13 hertz. All three backup control system channels per airplane control axis had the same servo loop gain, which indicates that the performance increase was a result of the force summing of the secondary actuator pistons.

Hysteresis. — Hysteresis measurements were also taken for each secondary actuator for the various valve drive combinations. The data were obtained by driving the appropriate servovalves with a signal generator set at 0.01 hertz. For example, the hysteresis of the elevator secondary actuator for the primary channel (fig. 10) was 0.44°. By increasing the loop gain, this value could be reduced to 0.13°. The equivalent measurements for the two backup control system conditions presented in figure 11 are 1.10° for the single-channel drive configuration and 0.47° for the three-channel drive configuration.

A minor item of interest pertaining to the secondary actuators was observed during single channel operation with the backup control system. Even though the electrical commands to each paired surface, such as the aileron and elevator, were the same, the control surfaces did not track each other during large control cycles. This was caused by the component offset characteristics in the servo loop as well as by the seal friction of the respective actuator channel. A given servo system took more current to retract the ram for the left control surface than the right control surface and less current to extend the left than the right. From outside the airplane the control surfaces did not appear to track. This was most noticeable with the elevator surfaces. This condition existed with every actuator, and there was no way to adjust the offset. When additional servo systems were engaged, the condition was

minimized and the agreement between the deflections of the paired surfaces was good.

The condition was not apparent with the primary control system engaged because of the higher bandwidth servovalve and pressure gain.

Electrical Power

The electrical power for the operation of the F-8C aircraft was supplied by the main generator power package. This unit was comprised of ac and dc brushless generators that were mounted on a common shaft, regulators for the generators, an air turbine motor, and the necessary reduction gears. Energy for the turbine was supplied by high pressure bleed air from the engine. The ac generator was rated at 12 kilovolt amperes at 115 volts and 400 hertz. The dc capacity was 68 amperes at 30 volts. An emergency power package supplied backup electrical power as well as a hydraulic pump driven from a ram air turbine. The capacity of this unit was 30 amperes of dc and 4.2 kilovolt amperes of 400-hertz power. Figure 12 shows the power distribution system of the F-8 digital fly-by-wire airplane.

The power requirements of the fly-by-wire system were determined by the characteristics of the Apollo equipment. This equipment limited the ripple, spike, and surge voltages on the nominal 28-volt bus to a maximum of 32.5 volts and a minimum of 24.5 volts, with a peak current demand of 60 amperes. These requirements, in addition to a requirement for an estimated 30 amperes for the backup control system, made it necessary to install an additional power source in the airplane. Therefore, a direct-drive, 100-ampere, 32-volt dc flight control system generator was mounted in the nose cone of the engine. The voltage regulator was set to provide 28-volt power at the primary (number 1) bus. To give the additional protection required by the Apollo equipment, zener diodes and a 55,000-microfarad capacitor were placed on the number 1 bus. Flight control system power was controlled from the cockpit through normally closed power relay contacts. A warning indicator informed the pilot of loss of generator power.

To provide the necessary redundancy, 28-volt power was divided into four separate busses by isolation diodes and circuit breakers (fig. 12). Each bus, one for the primary system and one each for the triply redundant backup control system, had a 24-volt, 11-ampere-hour nickel cadmium battery as an alternate source of power. Backup control system batteries were always on the line, and they were kept fully charged by a constant trickle charge. They could provide power for a minimum of 1 hour after the loss of the flight control system generator. For additional protection, it was made possible for the pilot to place the main dc generator on the backup control system busses with normal loads reduced. To assist the pilot in monitoring the condition of the backup control system battery, a battery capacity meter was installed in the cockpit. This device measured current flowing into or out of the battery in terms of percent of full charge. It was not intended for the number 1 battery to supply the primary system with power for more than a few minutes. Its sole purpose was to aid in the stabilization of the bus voltage and to allow operation during temporary power interrupts like those that occurred during bus switching. For the protection of the number 1 battery, a circuit was installed to remove the battery from the bus whenever voltage dropped below 20 volts.

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OPERATIONAL EXPERIENCE

The first operational experience with the F-8 digital fly-by-wire control system was acquired during the integration and checkout of control system components in an iron bird simulator (ref. 1). The simulator was used to fine tune the control system to give it the necessary authority, trim rates, servo loop gains, and comparator trip levels. Before the first flight, the entire flight control system was subjected to an extensive ground test program that lasted 7 months. During this period, two major hardware changes were made. Because of the nonlinear characteristics of the Apollo hardware (ref. 2) unacceptable noise was transmitted to the secondary actuators. A second-order filter network was installed to smooth the primary system electronics. The backup control system integrators were changed to digital from analog because of drift.

Backup Control System Flight Evaluation

Before the first flight, the backup control system was tailored to the primary channel gearing and trim rates in each airplane axis. The flight controllability of the primary control system and the backup control system was evaluated on the simulator. Since the sole purpose of the backup control system was to provide an emergency return-home capability if the primary system became inoperative, the flight testing of the backup system was minimal. The testing did insure that the backup system would provide acceptable controllability, and at least once per flight the F-8 digital fly-by-wire control system was downmoded to the backup control system to perform an inertial measurement unit alinement. This was done in level flight.

Center stick. — The piloting tasks used to evaluate the backup control system with the center stick paralleled those used to evaluate the primary control system in the direct mode. The evaluation maneuvers included routine flying while evaluating gross and fine control maneuvers, formation flight, and gunsight tracking. The low-speed evaluations included ground control approaches. The first flight evaluation took place at speeds between 275 and 300 knots indicated airspeed with routine flying maneuvers. The pilot comments indicated that roll response was adequate and pitch control was good at these flight conditions. The airplane also exhibited satisfactory handling qualities and control power in the landing approach. During subsequent flights, the airplane seemed sensitive in the roll axis, and in a more demanding control task, that is, formation flight, the pilot indicated that airplane roll response became too oscillatory. He assigned the task a pilot rating of 6 on the Cooper-Harper scale (ref. 3). The lateral sensitivity problem was reduced by adding electrical deadband to the roll stick command signals. The modification yielded the roll gearing shown in figure 4(b). Even though the backup control system roll gearing was approximately the same as that in a conventional F-8C airplane, some pilots commented that the airplane rolled a little faster than they liked for a given stick displacement at 300 knots indicated airspeed. However, they felt that the roll response was not overly sensitive. A viscous damper was added to the aileron stick linkage to improve the dynamic stick characteristics for both the primary and the backup control systems.

For the first eight flights of the F-8 digital fly-by-wire airplane, a linear trans-
ducer was used in the pitch axis of the backup control system. As flight speeds
increased, a longitudinal sensitivity problem was observed by the pilot in both the
primary and the backup control systems. This problem was solved by reducing the
slope of the curve around zero but maintaining the previous control authority. Be-
cause of the inflexibility of the design of the backup control electronics boxes, non-
linear characteristics were obtained by having a stick transducer manufactured that
was similar to the original but gave the desired curve shape. The pitch modification
and appropriate scaling change in the backup control electronics boxes resulted in
the backup control system pitch gearing presented in figure 4(a). Subsequent flight
evaluations indicated that control was satisfactory in cruise as well as in the landing
approach. In normal flight, the airplane's control characteristics with the backup
control system were similar to those in the primary control system's direct (unaug-
mented) mode. For maneuvers that required large changes in pitch, however, such
as gunsight tracking during windup turns, the pilots preferred the backup control
system to the primary control system because of its smoother pitch response. The
characteristics of the primary channel were poorer because of stick quantization
(ref. 4).

The trim switches for the backup control system pitch and roll axes were on the
left cockpit console just forward of the throttle control. During the evaluation of
the backup control system, it became apparent that the location of the trim switches
was undesirable. One pilot rating was at least one number higher (poorer) because
of the additional workload due to this location. Beginning with the side stick evalua-
tion phase of the flight testing, the backup trim was activated from the conventional
center stick trim switch.

Side Stick.— The side stick was evaluated primarily by two pilots during six
flights. Six other flights were flown by four pilots who were evaluating other fea-
tures of the control system. Although the side stick gradients were not optimized,
the side stick controller was considered to be of interest in the overall control sys-
tem evaluation. Side stick evaluation tasks included formation flight, gunsight
tracking, mild aerobatics, ground control approaches, landing, and takeoff. Since
takeoff was considered the most uncertain phase of flight, it was performed only
after side stick control was evaluated in a high pilot gain task during up and away
flight. During the 12 evaluation flights, three takeoffs and seven landings utilizing
the side stick controller were made.

The stick gradients selected for flight test were based upon the six-degree-of-
freedom simulation results obtained with the iron bird simulator. The stick-to-
surface gradients were selectable, as shown in figures 5(a) and 5(b). The wing-
down gradients selected by most of the pilots were position 1 in pitch and position 3
in roll. The roll gradients were not changed during any of the flights, whereas a
slight change was made in the pitch axis. The original transition, or knee, of the
curve between the low and high gradients was at approximately 36 newtons, and
this value was increased to approximately 57 newtons for the last three flights.

All the pilots adapted easily to the side stick controller in flight after practice
on the simulator. They all commented on the sensitivity of the pitch axis, particu-
larly in high pilot gain tasks like formation flight. The center stick was also some-
what sensitive, but the excursions were lower in amplitude. Some of the pilots

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tended to fly both pitch and roll with a pulsing type of input. Most pilots tended to hold a nose-up stick force during the various maneuvers. The value they used was approximately 23 newtons, which was outside the stick deadband. One of the six pilots noted arm fatigue after a flight in which he evaluated side stick control. Several pilots rated the formation flight control task 3 to 5.

As discussed in reference 5, gunsight tracking was typified by good to excellent control over the lateral-directional axis and continuous pitch oscillations caused by pitch commands that were too abrupt. Crosstalk was absent in the tracking task. A comparison between a side stick-controlled and a center stick-controlled tracking run showed a higher frequency output from the force side stick, indicating a higher pilot workload.

The wing-up stick force gradients were evaluated in the power approach configuration for pitch out maneuvers and ground control approach patterns. Many of the approaches were flown in light turbulence, which seemed to have little adverse effect on control. Pitch and roll control was adequate, and pilot ratings ranged from 2 to 4 for the landing approach task.

Synchronization Performance

An important design requirement for a backup control system is that it track the primary system closely to minimize the switching transients. Therefore, synchronization networks were used to keep the systems synchronized. During every flight, the primary system was downmoded to the backup control system at least once to aline the inertial measurement unit in level flight.

Thus, downmoding to the backup control system was checked approximately 40 times. The surface transients were always less than 1°. The transients observed during these downmodes were caused primarily by the differences in null between the primary and the three backup servovalves of each secondary actuator. Overall, the system's static performance was good.

Simulation studies on the iron bird simulator showed that the synchronization network bandwidth of 2.5 hertz provided satisfactory backup control system tracking of the primary system for all except abrupt stick commands. The simulator studies also indicated that the synchronization/trim network characteristics could produce a large out-of-trim condition during a dynamic downmode if stick or pedal commands were being applied. The corrective action was to trim out the stick or pedal signal present at the time of the downmode.

Trim

The backup control system was mechanized with a digital integrator for trimming the backup control system and for synchronizing the backup control system with the engaged primary system. Since the control systems had to be synchronized over the full authority of the control surfaces, the integrator had to be scaled for full control authority. This resulted in an integrator resolution of 0.18°, 0.30°, and 0.20° for

elevator, aileron, and rudder, respectively. Trimming a control surface became a stepping operation and was not precise.

Hydraulics

The conventional F-8C hydraulic systems were not changed except for the addition of the F-8 digital fly-by-wire secondary actuators. Two hydraulic pumps each delivered a maximum of 45 liters per minute at a nominal pressure of $20,684$ kN/m^2. This capacity was marginal during two operations. At idle power, the hydraulic flow was inadequate to support preflight self-tests. A power setting of 80 percent proved to be satisfactory and was used for airplane ground checks. The self-tests were designed to operate in all three axes or one axis at a time. The latter procedure was used most often, although the three-axis tests were completed in approximately 4 minutes. During landing at idle power, high control surface activity caused the hydraulic pressure to drop, which caused the secondary actuator pressure switches to downmode the F-8 digital fly-by-wire system from the primary to the backup control system. This occurred during two landing rollouts, but no control system transients were observed by the pilot.

CONTROL SYSTEM RELIABILITY

The primary and backup control systems operated approximately 2500 hours during the fly-by-wire program, including both aircraft and iron bird operation, without any major problems. Six evaluation pilots flew the F-8 digital fly-by-wire airplane 42 times for a total flight time of 58 hours. Because of its length, the program was not expected to establish a level of confidence in fly-by-wire control systems, but it did constitute a first step toward developing such confidence. From the first flight, the airplane was flown with a control system that had no mechanical backup or reversion capability. During the evaluation flights, there were no system failures that could be attributed to the fly-by-wire aspect of the digital flight control system. There were no electronic failures in flight in either the digital primary system or the backup control system. There was one hydraulic line failure that reduced the total system redundancy level from four channels to two channels, but flight was no more critical than it would have been if a similar failure had occurred in a standard F-8C airplane. This is discussed in more detail below.

In addition to the reliability of the total system, it is important to discuss the reliability of the elements of the system. Table 1 summarizes the discrepancies that occurred in the F-8 digital fly-by-wire control system. The table includes the discrepancies experienced with the iron bird simulator as well as those experienced with the F-8 digital fly-by-wire airplane. Discrepancies observed during ground operation, preflight testing, and in flight are listed by major system component. A discrepancy was any system operation that appeared to be abnormal. Some were minor transient effects that did not affect the system's performance or reliability. The number of discrepancies that required a repair or replacement action is indicated. Even if no repair was required, extensive tests were made to insure that the component in question performed as designed.

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The discrepancies listed for the computer and related hardware, which are discussed in reference 4, are listed here to present an overview of the operational problems encountered during the program. A coolant system designed specifically for the Apollo system used on the F-8 digital fly-by-wire airplane caused one flight to be canceled before takeoff and one flight to be terminated early. The coolant system problem was attributed to lines that were frozen and did not permit the coolant to flow through the cold plates. When this occurred in flight, the coolant system was being monitored and the flight was terminated before it affected the control system. The pilot continued to fly on the digital primary system, and four channel redundancy was retained through landing. This problem was unique to the Apollo equipment and therefore would not be expected in production fly-by-wire systems.

Three power turn-on problems were observed, two with the backup electronics and one with the primary electronics. On one occasion, measurements indicated that the voltage supply for the primary electronics was not present. Recycling the power switch brought the power supply on line, and during subsequent testing the problem did not reappear. Laboratory testing did not reveal the cause of the problem, but a similar power turn-on indication was obtained by grounding either the plus or minus power supply.

Six failures due to open buffer resistors were recorded in the primary and backup electronics early in the program. It was discovered that the resistance wire in these resistors was affected by chemical or electrolytic corrosion. All the buffer resistors were replaced by a different type of resistor, and no other problems of this type were encountered. The other component failures listed were caused by an intermittently functioning capacitor, a failed zener diode, and an open transistor. None of these occurred in flight, and all were detected through normal testing procedures. During the flight program there were 12 backup electronic comparator tripouts, but the redundancy level of the total system was not affected. Ground checkout indicated that there were no failed components.

The secondary actuator discrepancies consisted of component failures, problems related to differential pressure, and differential pressure comparator tripouts. With 25 servovalves, 20 engage solenoids, and 20 differential pressure transducers in the airplane, occasional problems were expected. The servovalve was the only secondary actuator component to fail. Three such failures occurred in the aircraft system. They were detected during ground tests and repaired. If such a failure had occurred in flight it would have caused the loss of one of the four actuator channels.

As the table shows, the largest number of discrepancies occurred in the secondary actuator differential pressure network. Four aborted takeoffs were charged against the differential pressure network, as well as four in-flight and 26 preflight differential pressure comparator tripouts. Most of these discrepancies were classified as nuisance tripouts and occurred during control cycles whenever the primary system was engaged. All comparator tripouts were resettable by the pilot, and the total system's redundancy was not affected. Generally speaking, most of the differential pressure problems experienced were caused by a tracking error between the various differential pressure signals, which caused the servo comparators to trip. This frequently occurred at the maximum travel of the actuator, where the differential pressure signals were the highest. These nuisance tripouts were caused by a

combination of the various components' tolerances and valve nulls, and were predictable for certain stick motions. The problem could be resolved by adding nulling capability to the servo loop to balance the various differential pressure signals.

Another problem associated with the differential pressure monitoring system was the inability to detect some of the open failures. Unless the ram was stationary, it was difficult to develop the differential pressure necessary to disengage the faulty servo channel. As a result, a latent channel failure could occur in flight in the backup control system and not be indicated to the pilot. However, no such failures occurred during the program.

Six discrepancies were attributed to system wiring and aircraft power distribution. Four involved, respectively, a pin that was pushed back in a connector, a short-circuited cable clamp, a defective latching relay, and a faulty battery capacity meter. The faulty items were identified and repaired during the regular airplane preflight. Two flights were aborted because of aircraft power problems. One was due to a checklist error that allowed the flight control system generator to remain off, causing a low-voltage shutdown of the computer, and the other was due to a main generator failure. All those discrepancies were considered to be typical airplane operating problems and not unique to fly-by-wire control systems.

Four discrepancies that affected or would have affected the digital fly-by-wire system occurred in the aircraft hydraulics systems, and all required repair action. Hydraulic leaks that caused two flights to be cancelled were detected in the secondary actuators. During one flight, hydraulic oil was seen streaming along the outside of the airplane, and as a precautionary measure the flight was terminated and the airplane returned for a normal landing. During another flight, a hydraulic line ruptured, causing a loss of hydraulic pressure to backup channels 2 and 4. The hydraulic line was part of the basic F-8C hydraulic system that was not modified for the program. The loss of hydraulic pressure was detected by the pilot from the conventional F-8C hydraulic pressure gages and warning lights. The pilot terminated the flight and landed the airplane with the primary control system. Hydraulic line failures are rare but serious for flight control systems that depend on irreversible hydraulic actuators, such as those being used in all high-performance fighter and bomber aircraft and many new transport aircraft. Protection against hydraulic system failure is provided by using dual or triple hydraulic systems. Experience with aircraft that use irreversible actuators has shown the protection provided by this practice to be adequate.

As the table shows, similar operating problems were experienced with the iron bird control system. All the simulation systems were flight qualified and could be flown on the airplane except the mechanizations of the primary and backup electronics, which were not maintained with flight system quality control. The experience obtained during the almost 2500 hours of operating time on the iron bird and the F-8 digital fly-by-wire airplane is indicative of what could be expected of a similar period on the aircraft system.

Although many component discrepancies occurred during the program, they were detected by the monitoring system and testing procedures, and the reliability of the total system was maintained throughout the program.

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CONCLUDING REMARKS

A digital fly-by-wire control system with a triplex analog backup control system was flight tested in an F-8C airplane. Six pilots logged 58 flight hours during 42 flights. The backup control system operated well in conjunction with the digital primary system and provided satisfactory handling qualities throughout the flight envelope evaluated. This experience showed that a dissimilar control system can be made to synchronize with the primary flight control system and provide satisfactory control during normal flight maneuvers.

A limited flight test program was flown to evaluate airplane handling qualities with a force side stick controller through the backup control system. Even though side stick force gradients were not optimized, the control of the airplane in a variety of control tasks, including takeoff, landing, and formation flight, was satisfactory.

The operational reliability of the digital fly-by-wire system, both primary and backup, was excellent. There were no downmodes from the digital primary control system to the backup control system in flight due to real or apparent system failures. Several component discrepancies occurred within the redundant system, but they did not affect the reliability of the total system. Most of the discrepancies were in the secondary actuator differential pressure network and were nuisance tripouts (capable of being reset) within the backup control system during large control inputs to the digital primary control system.

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TABLE 1.— F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM DISCREPANCIES

	Iron bird simulator		F-8 digital fly-by-wire airplane			
	Total discrepancies	Repair action required	Total discrepancies	Repair action required	Preflight	In flight
Primary system						
Computer (program characteristics)	4	0	6	0	0	0
Related hardware – mode panel, stick transducer, pallet signal boxes, inertial measurement unit	1	1	4	3	0	0
Coolant system	2	0	2	2	a_1	a_1
Primary control electronics						
Power	0	0	1	0	0	0
Components	0	0	2	2	0	0
Backup control electronics						
Power	0	0	2	0	0	0
Components	0	0	7	7	a_1	0
Electronic comparator (tripouts)	0	0	12	0	0	12
Secondary actuators						
Components	6	5	3	3	0	0
Differential pressure related	2	0	5	4	a_4	0
Differential pressure comparator (tripouts)	0	0	30	0	26	4
Aircraft and system wiring	0	0	2	2	0	0
Aircraft power	0	0	4	3	a_2	0
Aircraft hydraulics	0	0	4	4	a_2	a,b_2

[a] Flight terminated.
[b] System redundancy reduced to two channels on one flight.

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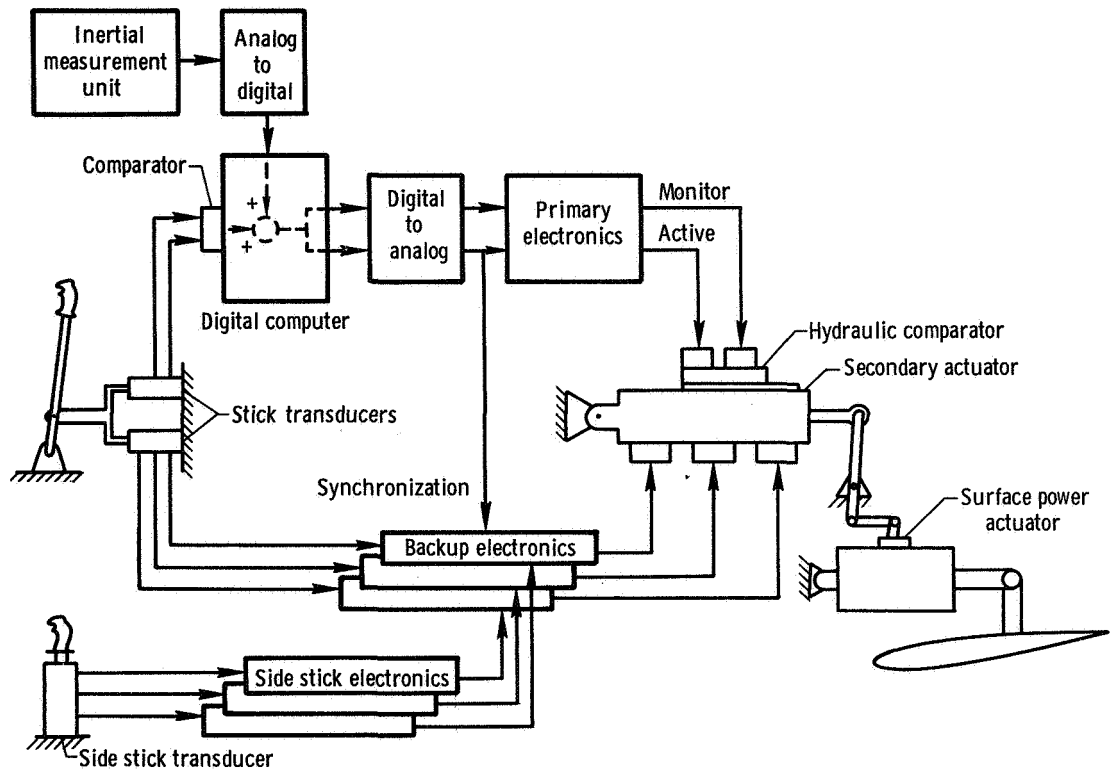
Figure 1. F-8 digital fly-by-wire control system.
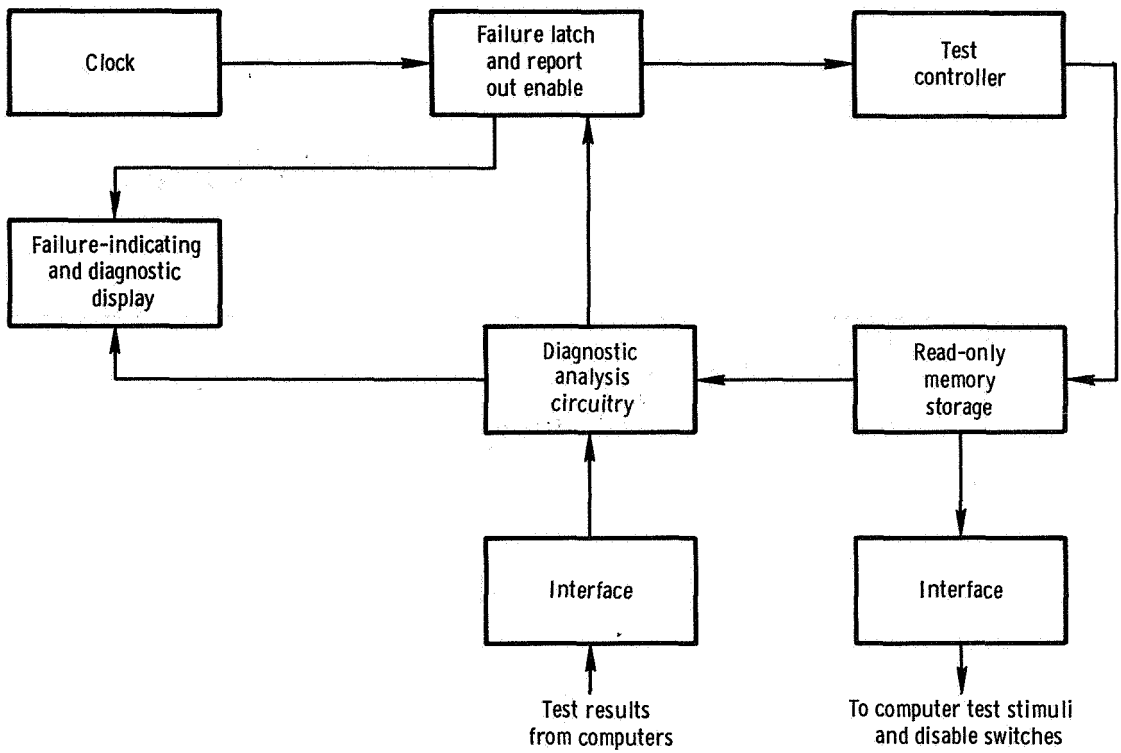
Figure 2. Self-test unit.
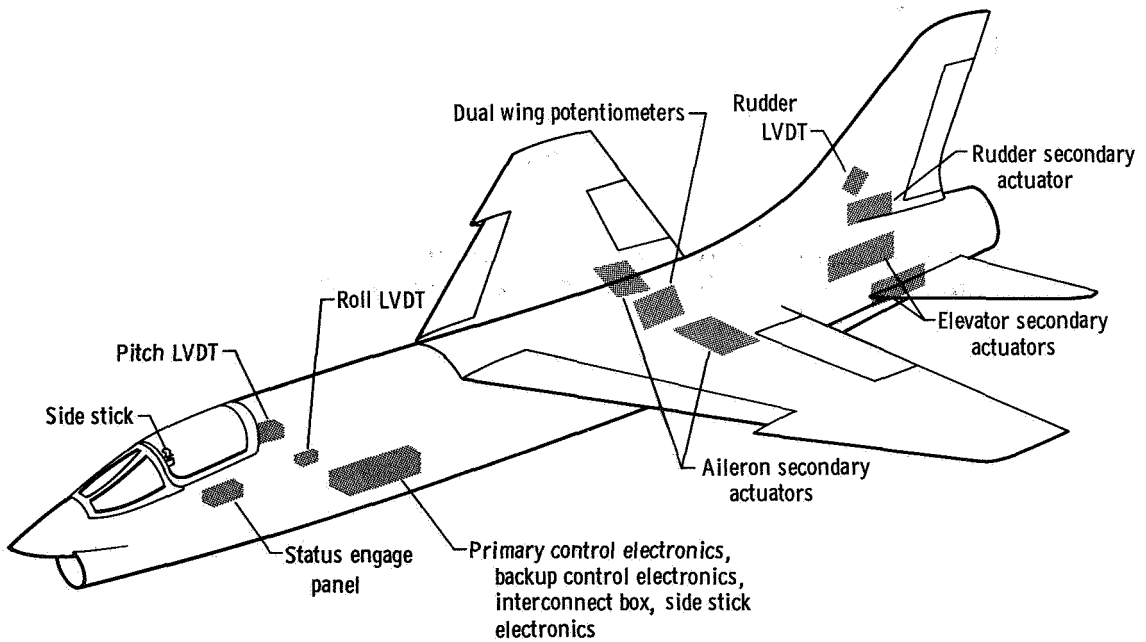
Figure 3. Components of F-8 digital fly-by-wire control system.
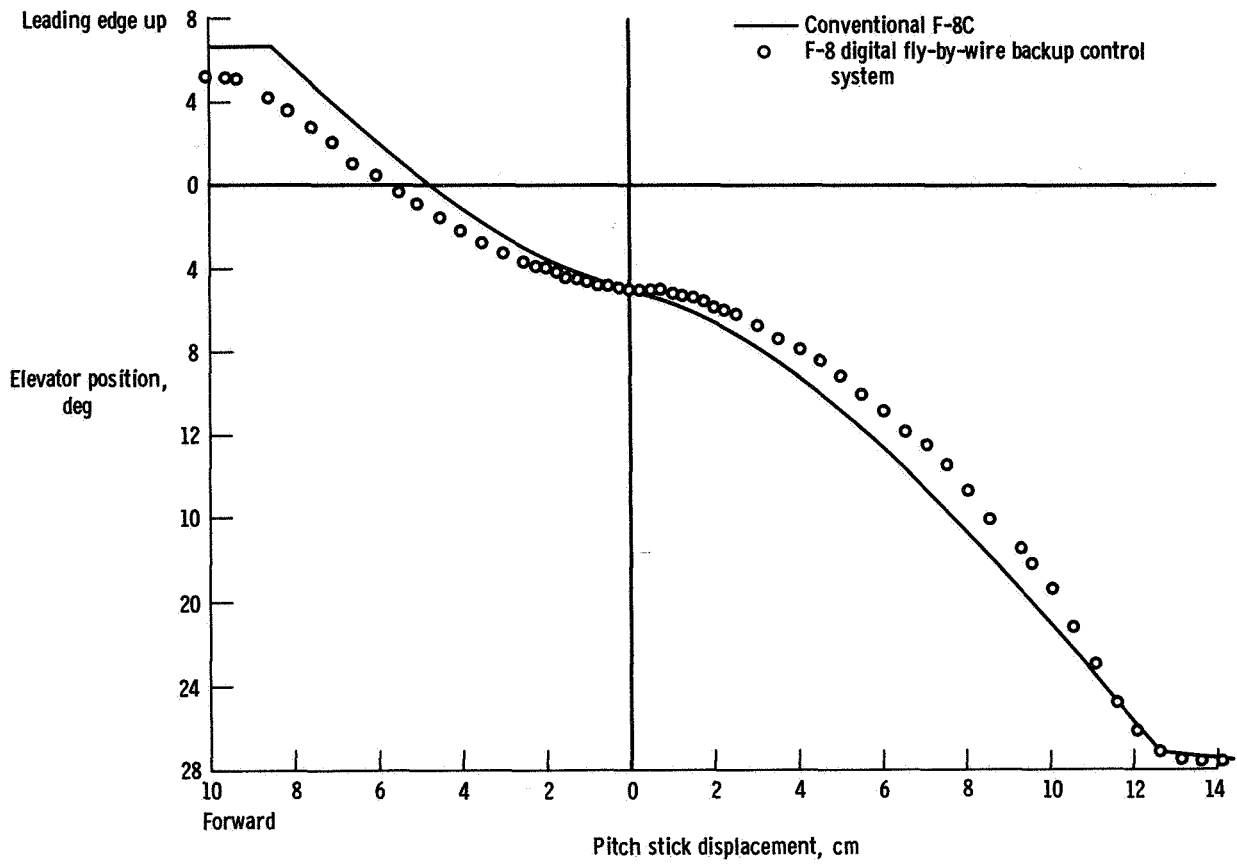
(a) Pitch.

Figure 4. Comparison of conventional F-8C and F-8 digital fly-by-wire backup control system pitch, roll, and yaw axis gearing.

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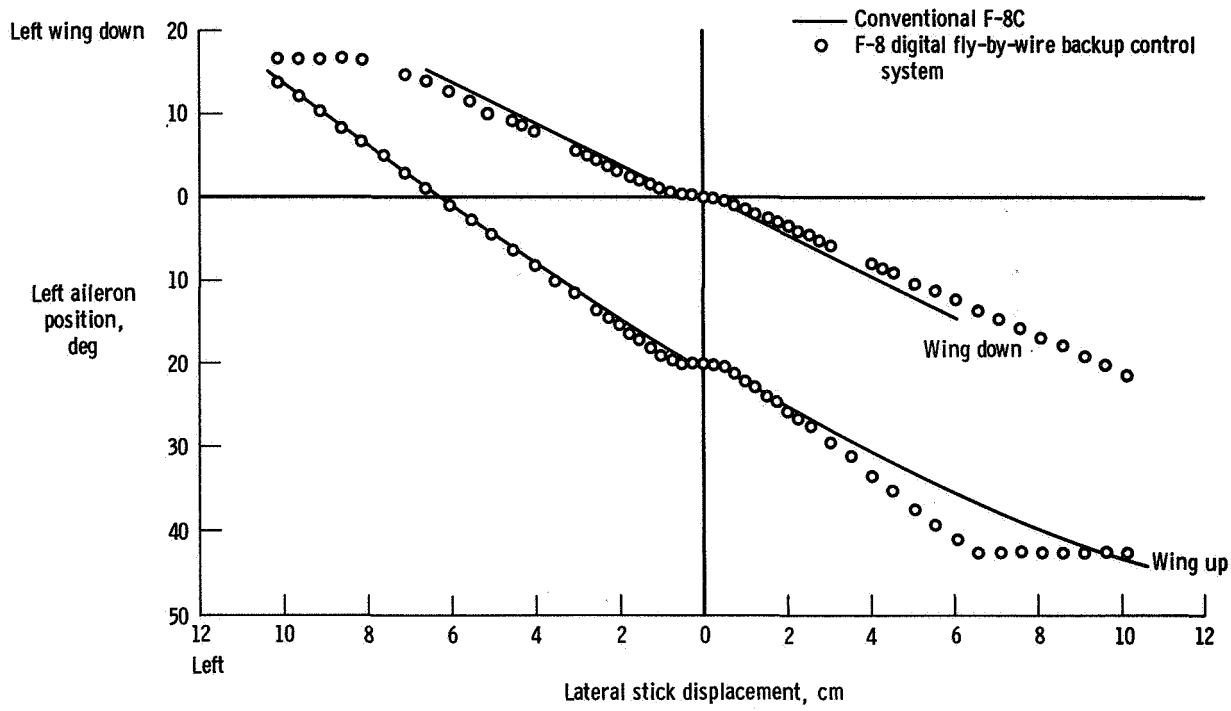
(b) Roll.

Figure 4. Continued.
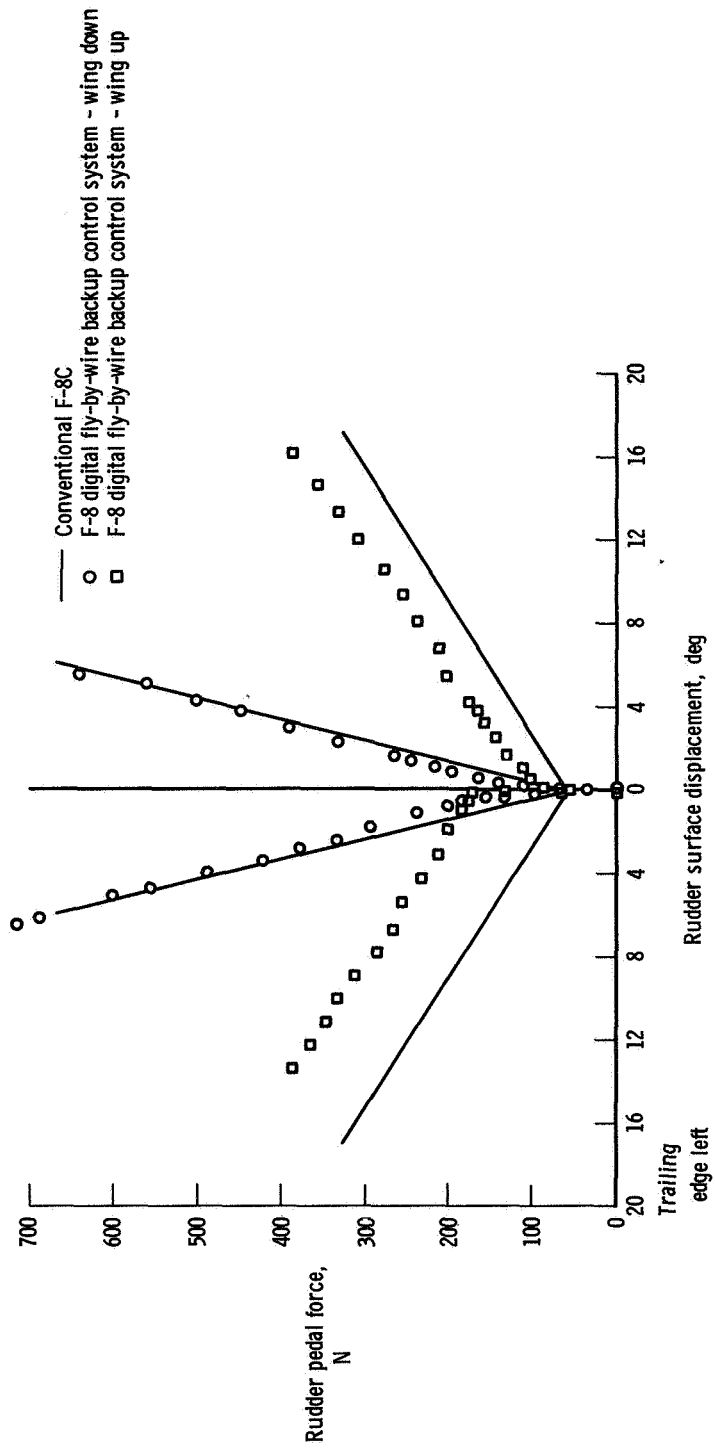
(c) Yaw.

Figure 4. Concluded.
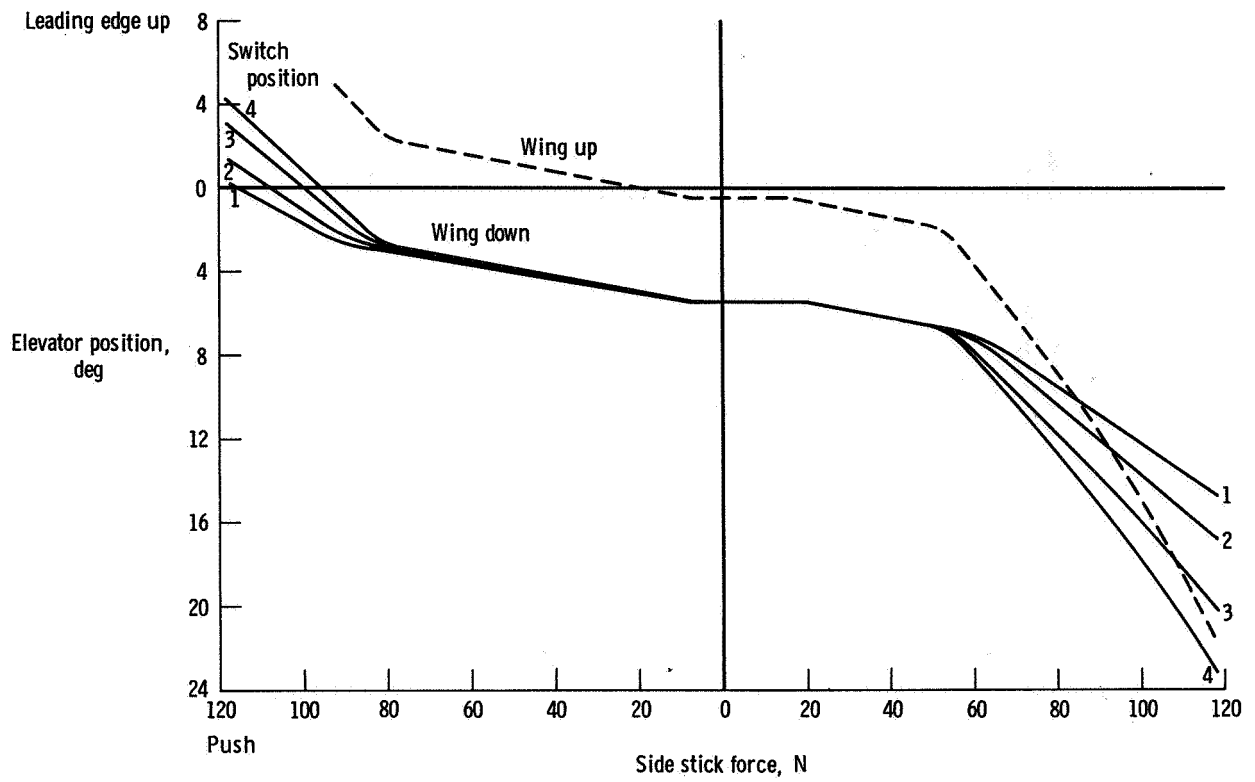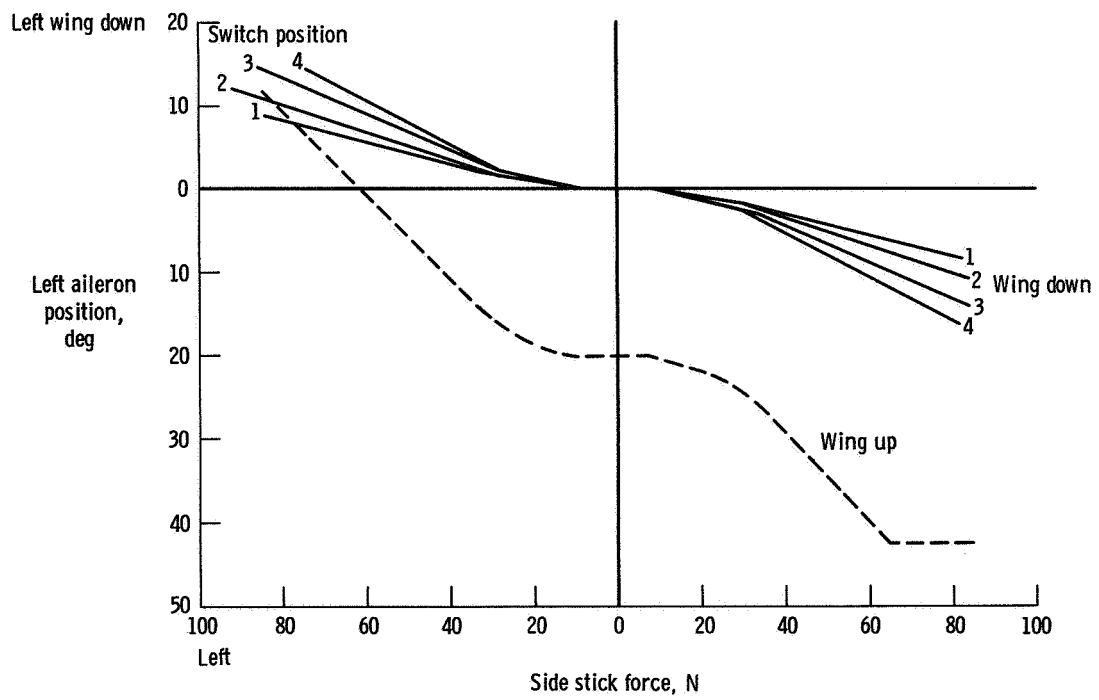
(a) Pitch.

(b) Roll.

Figure 5. Side stick gearing in pitch and roll axes.
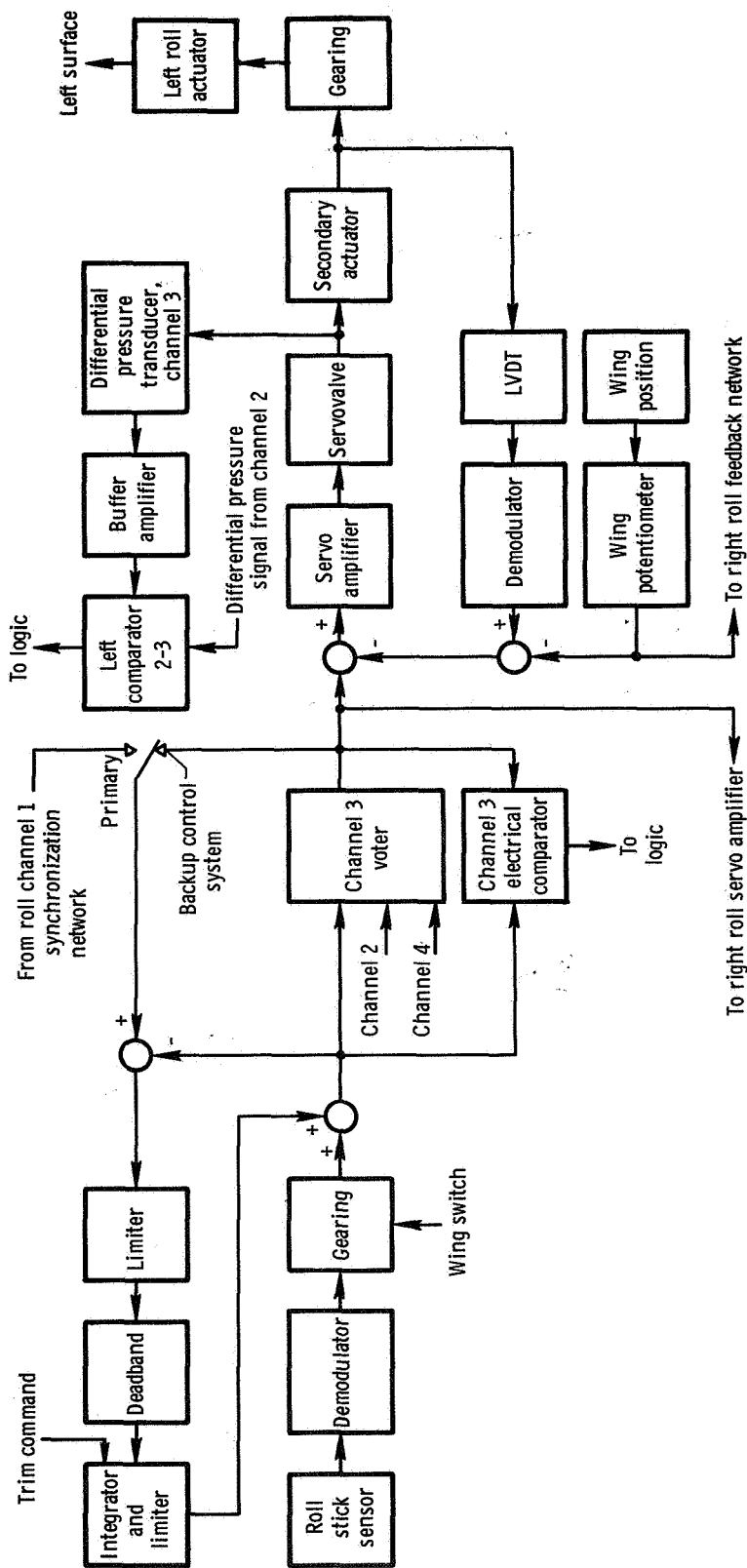
Figure 6. Roll backup control system channel 3.
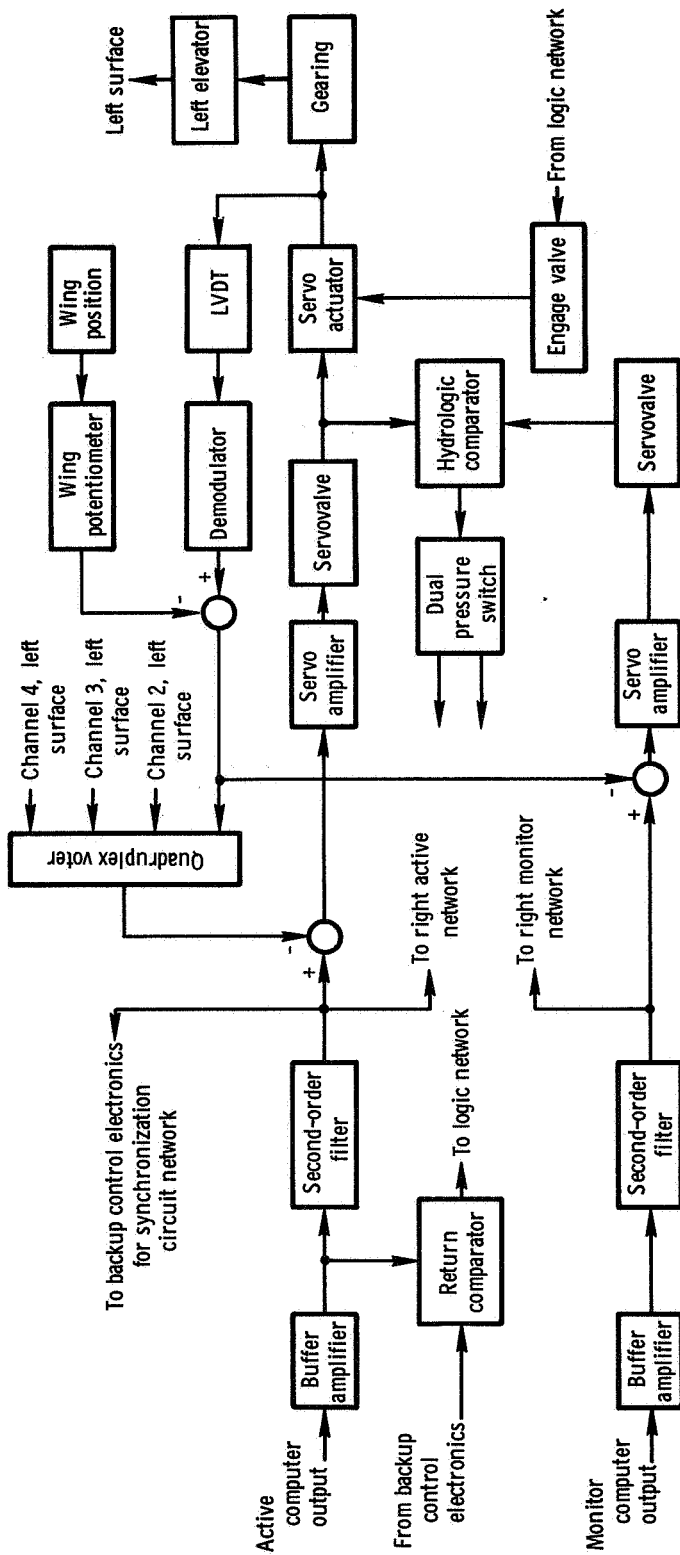
Figure 7. Typical primary control system channel.
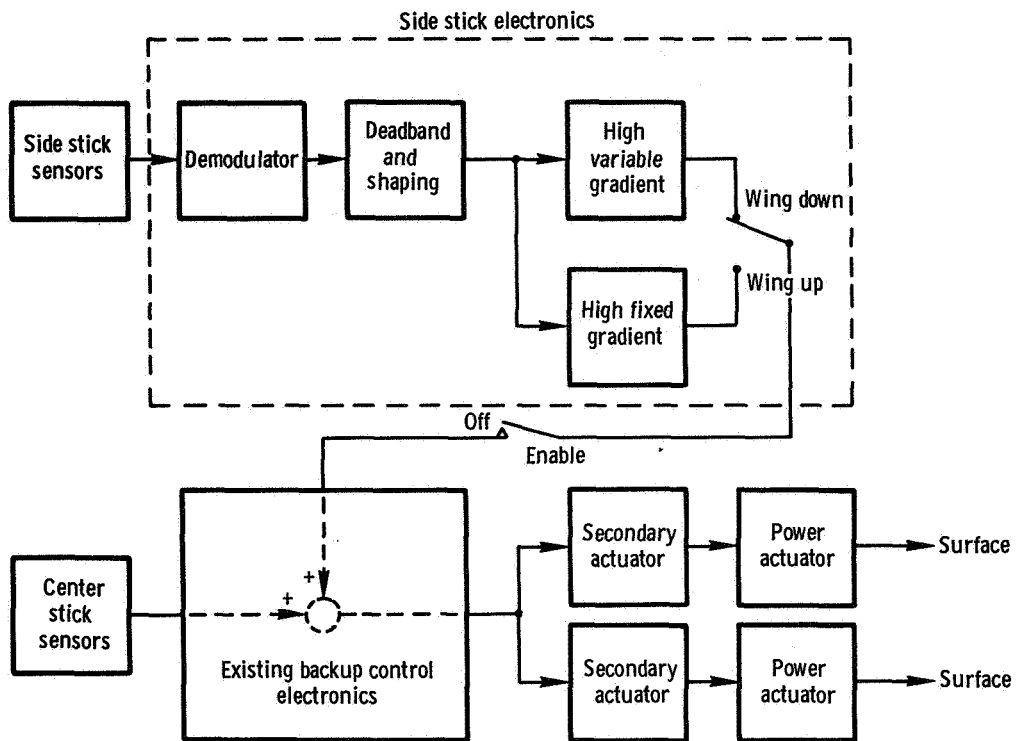
Figure 8. Side stick block diagram.
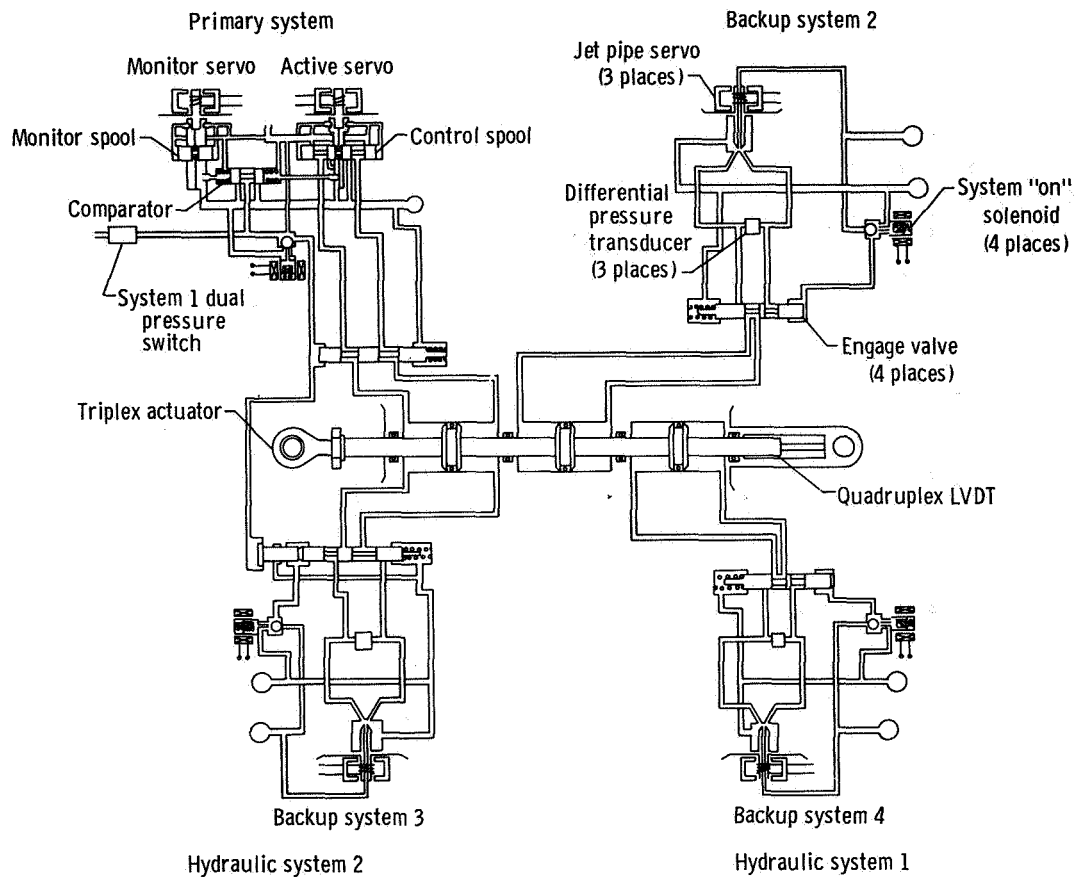
Figure 9. Hydraulics of secondary actuator with primary channel in control and channels 2, 3, and 4 in standby configuration.

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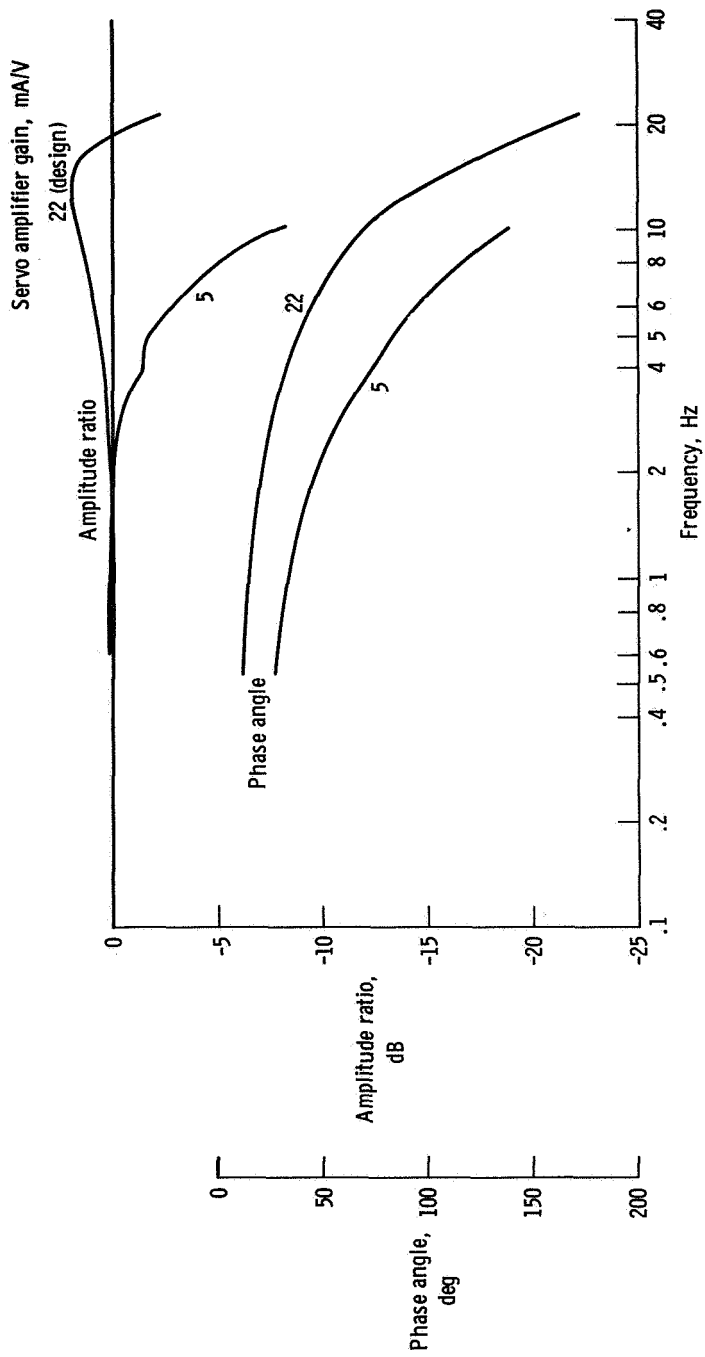
Figure 10. Comparison of elevator secondary actuator frequency response controlled with off-design and design gains of primary servo amplifier.
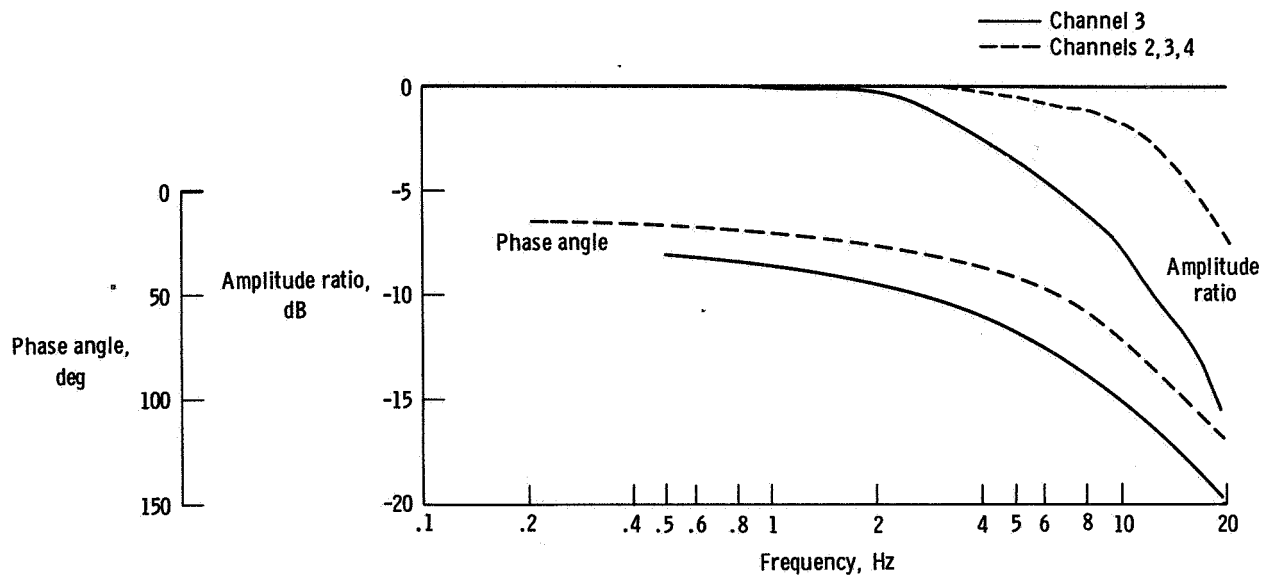
Figure 11. Comparison of elevator secondary actuator frequency response controlled with backup control system valves.
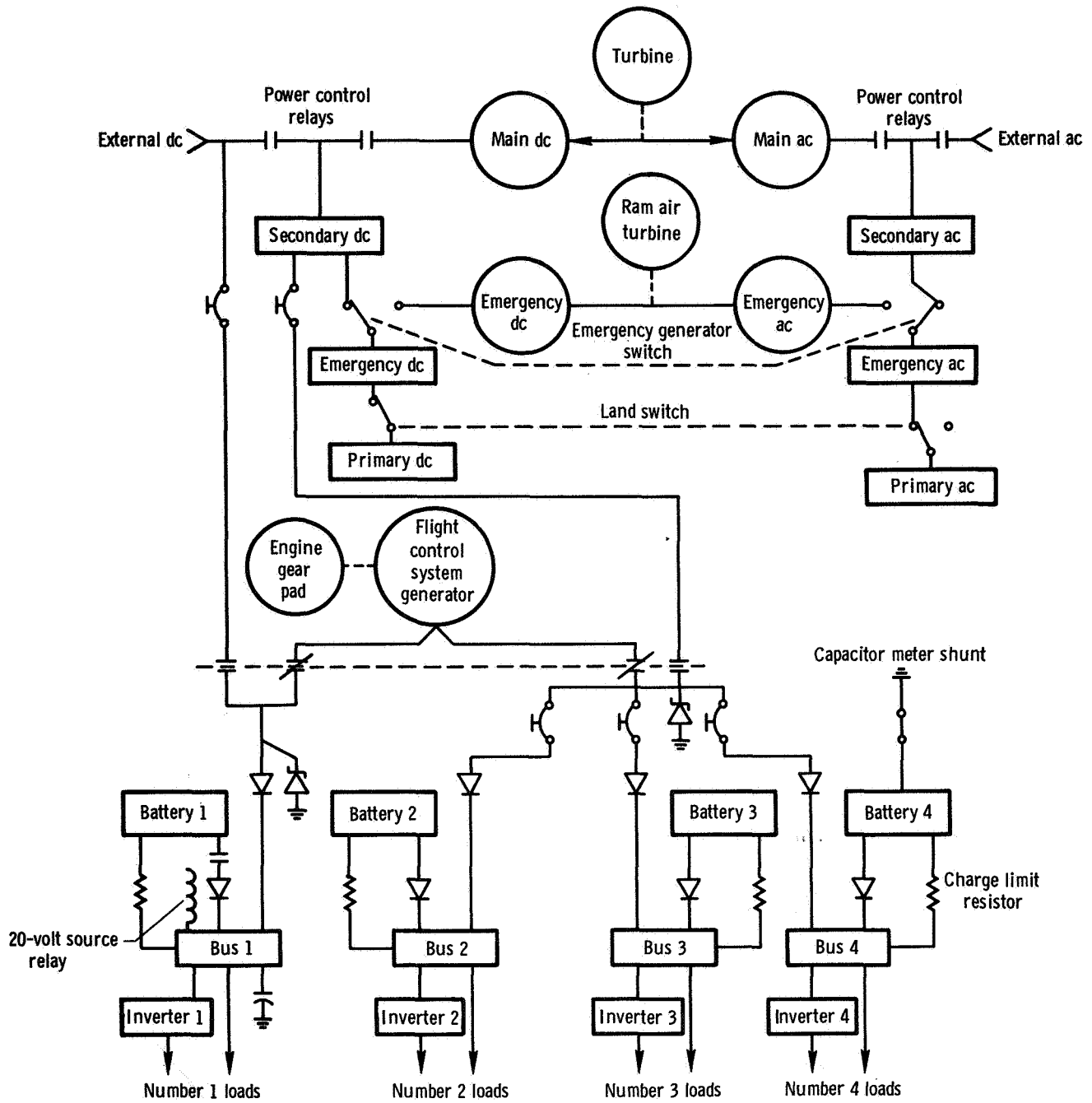
Figure 12. Simplified diagram of F-8 DFBW power distribution system.

MAN-RATED FLIGHT SOFTWARE FOR THE F-8 DFBW PROGRAM

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SUMMARY

The design, implementation, and verification of the flight control software used in the F-8 DFBW program are discussed. Since the DFBW utilizes an Apollo computer and hardware, the procedures, controls, and basic management techniques employed are based on those developed for the Apollo software system. Program Assembly Control, simulator configuration control, erasable-memory load generation, change procedures and anomaly reporting are discussed. The primary verification tools—the all-digital simulator, the hybrid simulator, and the Iron Bird simulator—are described, as well as the program test plans and their implementation on the various simulators. Failure-effects analysis and the creation of special failure-generating software for testing purposes are described. The quality of the end product is evidenced by the F-8 DFBW flight test program in which 42 flights, totaling 58 hours of flight time, were successfully made without any DFCS inflight software, or hardware, failures or surprises.

INTRODUCTION

From early 1971, CSDL participated in Phase 1 of the Digital Fly-by-Wire program being administered by NASA Flight Research Center (NASA/FRC). Overall program effort was directed toward a series of demonstration Fly-by-Wire (FBW) aircraft flights. A triply redundant Analog Fly-by-Wire (AFBW) Backup Control System (BCS), employing a simple open-loop control algorithm, is coupled with the primary flight control system to provide the two-fail-operate/fail-safe reliability necessary for severing mechanical linkages. The simplex Digital Fly-by-Wire (DFBW) Primary Control System (PCS) has both software and hardware failure-detection capability in the digital computer. There are also independent monitoring and failure-detection modules operating on PCS control commands, power supplies, pilot input devices, and other critical areas. Finally, there is the capability for pilot-initiated downmoding to BCS via several independent paths. There are seven selectable PCS flight control modes available. Three Direct (DIR) modes consist of pilot stick/pedal plus trim applied directly to the control surfaces. Three Stability Augmented System (SAS) modes incorporate body-axis angular rates (and lateral acceleration) as feedback variables. The Command Augmented System (CAS) mode is basically pitch SAS with normal acceleration feedback and forward-loop integral bypass. The only BCS mode, Direct, is also selectable by axis.

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The first Fly-by-Wire flight was made on 25 May 1972, in the high performance F-8C fighter assigned to the DFBW program. Takeoff and landing were made in PCS/DIR. Basic performance and handling qualities were demonstrated at several flight conditions, both in BCS and PCS/DIR. Closed-loop PCS/SAS was first flown on 18 August 1972 with subsequent flights building toward full system capability. The demonstration flight test program continued through late 1973.

The CSDL role in the F-8 DFBW program has been directed at the PCS software, hardware, and peripherals. Specific tasks have been: the hardware design, development, and testing of the uplink and downlink converters, the PIPA Simulator, and the Gimbal Angle Simulator; and software design, implementation, and verification of the NASA/FRC three-axis Primary Control System algorithms; the functional design, software design, production, and verification of the mode and gain change routines, miscellaneous ground test programs, and open-loop inflight earth-rate torquing routine; the interface design including failure analysis; simulation support; the review and verification of preflight erasable loads.

The F-8 DFBW System

Aircraft—The F-8C Crusader, a carrier-based U.S.Navy fighter of mid-50's vintage, is a high-performance single-seat aircraft capable of Mach 1.8 flight at altitudes of 60,000 feet. NASA/FRC obtained several surplus aircraft of the F-8 series. Two of them are involved in the F-8 DFBW program, one as the flight article and one as the Iron Bird Simulator test article. Figure 1 depicts the F-8C aircraft, showing the physical distribution of key F-8 DFBW hardware. Descriptions of the hardware are given in Table 1 and Table 2.

Digital System—The digital computer used by the PCS is the general purpose Apollo/LM Guidance Computer (LGC). An Apollo Inertial Measurement Unit (IMU) provides attitude angles, angular rates, and linear accelerations for feedback control. Major considerations for using the Apollo hardware were that it possessed a demonstrated reliability and flexibility. Moreover, surplus LM hardware was available from cancelled Apollo missions. Experienced teams of software and hardware specialists were also available, for software and systems integration tasks, at CSDL and Delco Electronics. A functioning Operating System software existed for the LGC, in addition to the supporting facilities of the powerful Assembler software, the All-Digital Simulator, and two hardware-integrated simulators at CSDL. Starting with this framework meant that a significant portion of the development task was already completed. There were some disadvantages, the most significant being the July 1972 scheduled shutdown of the core-rope manufacturing facilities for the LGC fixed memory. Another disadvantage, although not recognized immediately, was that the F-8C performance envelope exceeded the design capabilities of some Apollo hardware items. This influenced the digital flight control system (DFCS) performance, and required a reduced performance envelope, which, while less than F-8C capabilities, was nevertheless acceptable for an experimental digital fly-by-wire testbed.

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Computer—The LGC contains two distinct memories, fixed and erasable, as well as hardware logic circuits. The fixed memory is stored in a wire braid which is manufactured and installed in the computer. This memory cannot be changed after manufacture and it can only be read by the computer. Fixed memory contains 36,864 words of memory grouped into 36 banks. Each word contains 15 bits of information, plus a parity bit. The erasable memory makes use of ferrite cores which can be both read and changed. It consists of 2048 words divided into 8 banks. Erasable memory is used to store such data as may change up to or during a mission, and is also used for temporary storage by the programs operating in the computer. The memory cycle time (MCT) in the LGC is 11.7 μs. Most single-precision instructions are completed in two MCTs; most double-precision machine instructions are completed in three MCTs.

SOFTWARE DEVELOPMENT

The software control procedures employed for F-8 DFBW selectively follow those developed and successfully applied during the generation of software program assemblies for the Apollo command and lunar module computers. A continuation of useful procedures, made necessary because the F-8C uses the same Apollo hardware, and desirable because of schedule limitations, was easily imposed by the CSDL personnel connected with F-8, all of whom were contributors to the Apollo effort. The limited scope of F-8 dictated some changes in procedure, but these were basically simplifications commensurate with the level of effort. After all, approximately 400 man-months/month were expended in Apollo by CSDL programming and engineering groups just prior to the first lunar landing, while F-8 DFBW peaked at about 9 man-months/month. The critical time span was from Control Law Specification delivery in March of 1971 until program release for fixed-memory core-rope manufacture in mid-December of 1971. Since that date, CSDL has supported Preflight Erasable Load generation, failure analysis, preflight procedure preparation, and Erasable Memory Program development and verification. The timely development and excellent flight-test performance of DFBW software attest to the effectiveness of the control procedures employed. It is worth emphasizing that we now have more modern software techniques, but that Phase 1 of F-8 DFBW was a basic evaluation program, and utilized off-the-shelf software as well as hardware. Approximately 85 man-months and 95 hours of IBM 360/74 computer time were required for the Phase 1 software design, implementation, and verification tasks. The F-8 chronology is shown in Fig. 2.

Operational Software

The operational software for F-8 DFBW consists of two basic categories: the DFCS Program Assembly, and the Preflight Erasable Load Assembly. In the finished product, the DFCS Program Assembly is embodied in the core rope and comprises the computer's fixed memory. At this stage, it has become hardware and is effectively a breadboard autopilot in that the structure is invariant while most parameter values and switch words are variable. For F-8 DFBW, there is only one final Program Assembly, from which the flight rope and an identical

spare are manufactured. The Preflight Erasable Load Assembly is embodied in a tape and comprises the computer's Initial Data Load. The tape, KSTART, contains parameter values and switch settings required by the program, and the computer receives it as a part of each power-up sequence. A new Preflight Erasable Load Assembly is made whenever a flight test requires new parameter values. To ensure the high degree of reliability and safety that is necessary for man-rated flight software, both assembly processes are carefully controlled.

Program Assembly

The Program Assembly has two main functional areas: Systems and Applications. Grouped under Systems are Executive, Restart, and Service. Applications covers Flight Control, and Miscellaneous. The Executive code includes the priority job-queue processor, the time task-queue processor, the time-dependent interrupt processor, the idle-job routine. The Restart code includes the hardware restart interrupt processor, computer initialization routine, the program alarm processor, the restart-group phase-control routines. The Service code includes the list-processing interpreter, the IMU monitor, the computer self-test routines, the man-machine interface routines, the interrupt processors. The Flight Control code includes the autopilot initialization routine, the mainline processor, the filter pushdown and wrap-up processor, the input discrete processor, the Mode and Gain change processor, the body transformation matrix processor. The miscellaneous code includes the ground test programs, and special-purpose applications routines.

In several areas, the flight control requirements and the LGC characteristics posed interesting problems. Some of these are singled out.

Duty Cyle—Early in the development process it became clear that the Flight Control system would create a relatively high duty cycle in the LGC due to several causes: LGC instruction time (24 µs/instr), the flight control sample period (30 ms) and the generalized nature of the control system. Since the entire LGC is devoted to the DFCS, words of code could be traded for increased time efficiency wherever possible; that is, code is designed for minimum execution time rather than for minimum storage. Time savings are also realized for control parameters, where combinable multiple parameters are replaced by an equivalent single parameter in a working register, whose value is generated only once by program initialization.

Restart Protection—A hardware restart is a special interrupt that takes precedence over all other interrupts, and that cannot be inhibited. The hardware restart is triggered by circuitry in event of selected computer malfunctions. On completion of the restart, all output channel discretes are cleared, and computer control is transferred to a specific memory location, i.e., to the Restart Routine. The Restart software rapidly reestablishes the channel output interfaces because F-8C control surface commands and the PCS primary-enable signals depend on a viable interface. The restart software next restores the program flow by reestablishing the job-queue and time-queue, and by causing the program whose execution was interrupted to resume at the latest restart

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point. Restart points are entry points, breaking program flow into separate blocks, such that a properly restart-protected program will reproduce the same values after a restart as before.

In general, a repetition of code execution is involved following a restart because the nature of the LGC requires software recovery procedures. However, the repetition requires that special care be taken during code generation to avoid creating situations where a restart will cause a multiple update of a variable. For example, if the operation A+B → A occurs between two restart points, then A is updated at each pass through the code. This violates the rule that the values generated by code repetition after a restart must be the same as before. The situation of multiple updates is avoided by a copy cycle, which involves an intermediate variable and an additional restart point. For the example we have A+B → C, followed by the new restart point, followed by C → A. Clearly, the final value of cell A is unaffected by code repetition. Copy cycles are common in Apollo code and have the advantage of economy of erasable memory usage although they are expensive in terms of execution time. Note that cell C is intermediate and can be used by many copy cycles.

Rather than use copy cycles, F-8 DFBW prefers a method that, because of the high DFCS duty cycle, is conservative of time but is expensive in fixed and erasable memory cells, doubling the number. Two functionally identical strings of code, a J-branch and a K-branch, are required with processing alternating from one to the other. Two equivalent sets of erasables are required, also J-branch and K-branch. The J-branch code uses K-branch (past value) outputs plus J-branch (present value) inputs to compute J-branch (present value) outputs. No special copy cycles are required, and computations are efficiently performed. Copy cycles would likely have pushed DFCS duty cycle dangerously close to 100%. It reaches 91% even with time-efficient restart protection.

Indirect Transfer—At sixteen critical points in F-8 DFBW program flow, and at one point in the downlink program, a capability is provided for erasable indirect transfer of control. In application the program flow of the hardware core-rope fixed memory program is determined by the address contained in a specific erasable cell at the time the cell is accessed by the program. Erasable cells used in this manner fall into two classes. There is the class of cells whose contents (the destination address) is changed regularly under program control, say every 20 ms or 30 ms. These cells, although erasable, form an integral part of the core-rope. The second class consists of cells whose contents are in general established only once, either by an initialization pass or by the Initial Data Load (KSTART tape). It is this second class of erasable cells that provides the powerful capability of altering the program flow after core-rope manufacture by means of Erasable Memory Programs.

Generalized Filters—Inasmuch as F-8 DFBW is a flying breadboard, the feedback sensor quantities are each provided with a generalized filter. The five filters, three for body rates and two for linear accelerations, allow flexibility of filter choice: bypass, first order, second order, and third order. An alternate third order is obtained by cascading the first and second

order sections to obtain control over individual poles and zeros. The filter coefficients are parameters in the KSTART tape. The filters are active at all times, even in BCS/DIR.

The computations are divided into two phases, the main phase which incorporates the current input with past values to update the output, and the pushdown or wrap-up phase which updates the other filter quantities in preparation for the next cycle. In this manner the control surface commands which use the filter outputs are generated with the shortest delay. The time-consuming filter wrap-up calculations are not performed until after closing the aircraft control loop, and so do not contribute to the delay. The saving is significant because the wrap-up can represent as much as 92% of the total filter load.

Gain Change—Manual gain changing is provided in lieu of automatic gain changing as a function of, say, dynamic pressure. Separate pitch, roll, and yaw gain-select switches on the MAPP, each with four positions, comprise the pilot interface. Selection of a specific gain (or coefficient) parameter is made from a fixed list of 105 candidates, serially numbered from 1 to 105. Each gain is associated (by axis) with a particular gain-select switch, and a maximum of 9 gains can be designated for a given flight. Each gain chosen, with its serial number and four values, becomes part of the PEL. When a gain-select switch is changed by the pilot, the program recognizes the change and the PEL-designated gains associated with that switch axis are changed. For each gain in turn, a small routine implements the change, performing all necessary scaling, recomputing all working registers using that gain, and initializing any filter using that gain.

Erasable Memory Programming—Erasable memory programming provides the only means of modifying the program once the core rope is manufactured. Modification can sometimes be accomplished by breaking into the program flow at a suitable erasable branch point, which must be of the second class as defined above. The procedure is to change the erasable cell contents to point to an unused block of erasable memory and to load executable code into that area (called an Erasable Memory Program or EMP). The final instruction of the EMP returns control to the fixed memory program. The EMP allows some unanticipated problems to be solved by shoehorning suitable code into the program flow.

Erasable Downlist—In Apollo, the identification and specification of telemetered data was done by means of address tables built into the core rope. For a mature design such as Apollo, quantities of interest are well known, and properly can be built into the rope. F-8 DFBW, on the other hand, must offer flexibility for experimental design. Variables and quantities of interest can change from day to day depending on a given flight plan. To accomplish this end, erasable specification of the downlist quantities by means of KSTART tape is incorporated into the Downlink program.

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Preflight Erasable Load Assembly

Flexibility is achieved in the F-8 DFBW despite the hardware status of the core-rope program by providing for a large number of erasable parameters. The aggregate, called the Preflight Erasable Load, consists of three categories: Data words, Downlist Words, and Erasable Memory Program words. The Data words are constants and include loop gains, filter coefficients, nonlinearity parameters, IMU compensation parameters, branch control parameters, and branch control address constants. The Downlist words are address constants that define the quantities to be telemetered. The EMP words are executable code and associated constants.

Early in the program the Preflight Erasable Load and the KSTART tape consisted only of Data words and Downlist words, and were generated by CSDL. But the responsibility for the data values resided with FRC, so generation of the Preflight Erasable Load and KSTART shifted to FRC as the software capability was developed there. However, Erasable Memory Program development was a CSDL function, and the verified and accepted EMP code was incorporated into the KSTART by FRC.

Several unique or extremely helpful features characterize the F-8 Preflight Erasable Load (PEL), and the generation of its KSTART uplink tape, specifically:

(1) PEL parameters are expressed in conveniently scaled, physically significant engineering units.

(2) A DFCS initialization routine translates each PEL parameter (units and scaling) into DFCS operational parameters. Factored or ratioed parameters are combined into single operational parameters at this time.

(3) Comprehensive error checking and diagnostic indicators are built into the KSTART tape generating programs.

Parameters—The basic DFCS parameters are expressed in conveniently scaled engineering units and constitute the erasable load. The DFCS working registers (gains, limit levels, coefficients) are defined so as to minimize computation time where possible. This usually results in unusual scaling, e.g., number of DFCS samples instead of seconds, or DAC bits instead of surface degrees. Other working registers are functions of basic parameters, such as a simple product, or a limit level that is computed from intercept/slope/breakpoint values. Also a working register might contain an address constant, selected from a table in accordance with certain rules. To accomplish the interface between working registers and erasable load parameters, F-8 DFBW utilizes an initialization routine. By having an initialization routine available to translate the working registers, the engineer preparing KSTART tapes, or changing parameters manually via the DSKY during a simulation, can continue to think in basic engineering terms. This is especially important in F-8 DFBW, since much of the development is performed on hybrid simulators

where the DSKY interface is the only practical interface for changing DFCS parameters. By keeping PEL specifications simple and by formulating them in engineering terms for both physical feel and visibility, the possibility for error is greatly reduced. Since programmed and verified initialization software is involved, reliable and complete changes are made quickly by single-parameter data entries even though that parameter exhibits multiple usage.

KSTART Generation—Two off-line diagnostic programs, DOWNDIAG and SHERLOCK, developed by NASA/FRC, contribute significantly to the generation of a highly reliable PEL and its KSTART tape. Operational use of these programs is shown schematically in Fig. 3.

DOWNDIAG checks the erasable downlink list specification against format, opcode, address, and keypunch errors. It punches the Erasable Downlist (EDL) and Downlink Processor (DLP) decks only after error-free input is provided. The DLP deck is used for post-flight or post-simulation downlink processing. The EDL deck is integrated with the DFCS parameter deck for input to SHERLOCK.

SHERLOCK likewise checks against keystroke, octal, and address errors, but more significantly performs comprehensive reasonability checks, e.g., minimum/maximum range or compatibility between related elements. SHERLOCK also extracts filter polynomial roots, checks the stability of poles, and checks zeroes against minimum/maximum ranges. Diagnostic printouts must be answered by corrections to the SHERLOCK inputs, or by signed waivers, before output decks are punched, one for the F-8 All-Digital Simulator at CSDL, and the other for input to KPUNCH, the KSTART tape diagnostic and punch program.

KPUNCH calculates the initialization values for the uplink summation (UPSUM) registers such that with a proper uplinking of the KSTART tape, the UPSUM registers equal 77777 77777 when displayed on the DSKY. Errors generated during uplinking will leave numbers other than 7s. KPUNCH also performs limited diagnostic checking and ultimately punches the KSTART tape, ready for uplinking to the LGC prior to flight.

F-8 DFBW Software Package

The F-8 DFBW software package can be broken down as in Table 3 (Fixed Memory Allocation), and Table 4 (Erasable Memory Allocation). The DFCS code is by far the largest single item. Extensive fixed memory is used by Display Interfaces (DSKY processing), Interpreter/Executive, and IMU Alignment. Most of this code was transferred directly or with minor change from the LM program for Apollo 14. The Self-Test Self-Check code came from Apollo preflight erasable code. Roughly half (696) of the erasables used are DFCS related, and a significant number (389) belong to the Preflight Erasable Load.

SOFTWARE PROGRAM CONTROL

The flight software for F-8 DFBW program leans heavily on the experience developed for Apollo. The main difference between Apollo software and other

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(previous) software is that the Apollo software had to work perfectly the first time it was used in its real environment. Apollo manned missions had a one-shot nature that required guaranteed performance. To achieve such reliability, management and supervision controls were set up, and have evolved over several years into a system to monitor and check software progress very closely and yet not to create an environment that is oppressive to the creativity, persever-ance, and dedication of engineers. The system thus created has been proven in both developmental and incremental phases of software. Man-rated flight soft-ware depends on reliability and confidence built up by careful management and supervision controls supported by thorough software verification using real hardware and high-fidelity models in simulation.

Software Management

A successfully managed software effort must provide:

(1) Realistic estimates of requirements including manpower, assembly and simulation budgets, memory allocations.

(2) Efficiency in the development and verification process including non-overlapping testing, effective use of man and machine re-sources.

(3) Achievement of milestones on schedule.

(4) Visibility of the product including developmental status, trouble spots, user-oriented operations and interfaces.

(5) Flexible and efficient response to design change requests.

(6) Systematic verification procedures at all module interface levels of testing and performance.

(7) Reliability of final products.

(8) Quality performance of final products.

The software management and control system developed for Apollo provided such capability. Its selection for F-8 DFBW was a natural outgrowth of success-ful prior experience with it. Changes were made, but only when the differing situations indicated a modified approach.

The management and control of flight software is directed toward the timely preparation of two end items: a software program assembly from which the read-only core-rope memory is manufactured, and a software preflight eras-able-load assembly from which a KSTART tape is manufactured to initialize the erasable read-write memory. Operational efficiency, performance capability, operational flexibility, and overall reliability are demanded of both the fixed and the erasable-memory assemblies, since they complement each other in terms

of overall performance. Timely availability is likewise a requirement in terms of schedule milestones. Changes and additions to the baseline design must be implemented with the same quality and timely control.

Organization and Controls

The software organization used by F-8 DFBW is relatively simple. The Project Manager is the customer's contact point. The Project Manager interfaces with the Software Manager, who interfaces with the engineers doing the software design, coding, and verification. Both of the latter interface with Assembly Control, which is responsible for the assembly process. The types of control machinery available to the Project Manager and the Software Manager are as follows:

(1) Software Specification Document is the product specification to which the software must conform.

(2) PCR—a Program Change Request, that officially changes the Software Specification (must be signed off by customer, Project Manager, and Software Manager).

(3) PCN—a Program Change Notice, similar to a PCR but deemed imperative by CSDL (must be signed off by Project Manager and Software Manager).

(4) Anomaly—a request to fix an error in the program (must be signed off by Project Manager and Software Manager).

(5) ACB—an Assembly Control Board request, identifies a necessary program change that is not a specification change (must be signed off by Software Manager).

Under Configuration Control, all coding changes and additions must be covered by one of the above forms of approval before the Assembly Control Supervisor will incorporate the code into the assembly.

Assembly Control

The Assembly Control functions in Apollo were highly structured and very formal for the mainline program assemblies. There was an Applications Programming Development and Testing Group for the two major assemblies. A System Integration Programming Group served for all assemblies, but the major assemblies had separate Assembly Control Supervisors. Finally, the Assembly Control Service Group served all needs.

The software generation process is illustratively simplified in Fig. 4. A coding task is routed to the appropriate programming group for code design. Discussions with the other groups might follow. Completed code is submitted to Assembly Control where it is either accepted for the next revision or returned

for corrections. At appropriate times, the assembly update deck is submitted to make the new revision. The Assembler output is examined by Assembly Control and errors are either fixed or referred back to the coder for rectification. Notification of a good assembly is given to coder/testers who submit simulation test runs. If tests do not work correctly, corrected code is submitted for the next revision. On receipt of good results, a new coding task is begun.

In F-8 DFBW, with a total programming team of about nine people, such structuring was not practical or necessary. Nevertheless the spirit of the Assembly Control process was maintained. One member of the DFBW team was designated Assembly Control Supervisor, but his activities spanned all four of the structured areas as time permitted and activity made necessary. For example, he monitored, coordinated and submitted all assembly changes, maintained the Simulator test packages, published the assembly documentation, maintained and verified IGC System software, coded and verified some Applications code, and participated in Level 4/Level 5 testing. The other team members likewise found their activities spanning the four groups as specific needs came and went, each contributing in areas of greatest interest and ability.

Controllable Items

In addition to the main program assembly, there are also other areas where control procedures must apply. These are the Preflight Erasable Load Assembly, Simulator Test Packages, Off-line Program Assemblies, and Erasable Memory Programs.

A Preflight Erasable Load Assembly is associated with each mainline program revision, and consists of data constants, branch-control constants, and address constants that are defined in the mainline revision. The Preflight Erasable Load Assembly is used to generate data and address decks for Simulator test runs and it is essential that these decks be error free.

The Simulator Test Package supports the software testing and verification by providing a common library of test case decks. Functionally the decks cover three categories: program initialization, simulation control, and edit control. Operationally the decks are invoked in suitable configurations at run time by single cards in the user's test deck.

Off-line Assemblies—As the mainline program matures, off-line versions are useful to check out code prior to updating the mainline assembly. Once the design and coding is checked out, a simple transfer of appropriate code is made to the mainline assembly. In F-8 DFBW two examples occurred; one was to check out a major design modification in the BCS downmode logic just prior to Configuration Control, and the other was to create a testing and training tool capable of failing input/output discretes via DSKY commands.

Erasable Memory Programs—Erasable-memory programming is a tool enabling a limited flexibility for modifying core-rope program flow. A block of code is designed to reside in and operate from erasable memory, and a way is devised to access the code from the existing rope.

Assembly Control Tools

Assembler—Since the software was not written in a Higher Order Language, a sophisticated assembler was of utmost importance. The Assembler is by far the most powerful tool in the Assembly Control process. The lengthy evolutionary period of Apollo has generated many fine features.

Diagnostic Package—The Assembler diagnoses faulty coding in both basic and interpretive languages. It issues diagnostic messages about references to non-existent variables, multiple definitions, illegal sequences of instructions, improper erasable-bank or fixed-bank references, and many others.

Basic and Interpretive Language—The Assembler recognizes two languages: basic language, and a list-processing interpretive language. The latter permits vector and matrix as well as double and triple precision operations; these are processed by the Interpreter software routines in the LGC. The Assembler recognizes data constants, noun and verb constants, downlink list specification constants, and address constants.

Flexibility of Memory Allocation—Blocks of fixed-memory programming can be referenced to each other so that if a block expands, another block need not be moved to make room for it. Overlapping of program memory is flagged if it occurs. Overlapping of erasable storage (time-sharing), on the other hand, is facilitated by the Assembler.

Program Visibility —The Assembler provides complete mnemonic cross-reference tables, a summary of erasable memory assignments, and maps of both erasable- and fixed-memory storage. All operand references are threaded, allowing rapid eyeball debugging even when the relevant passages are scattered through hundreds of pages. Word count, including a breakdown by functional area, is provided.

Modularity—The Assembler provides the ability to separately assemble and partially diagnose sections of the full program. These can be coded separately and brought together into full programs for verification.

Interface with All-Digital Simulator—The Assembler output includes input information for the All-Digital Simulator, which is useful for simulator initializations, and for simulator run-time diagnostic error detection. The Symbol Table enables the addressing of erasable cells and fixed locations by name, rather than by number which tends to vary from revision to revision as memory layout is modified. Tapes for fixed-memory loading of core-rope simulator can be generated. Constants, bad words (assembler-detected errors), unused words, and coding instructions are distinctively flagged to permit detection of such run-time errors as 'executing a constant' or 'executing from unused fixed memory'. KSTART tapes can be punched directly from the Preflight Erasable Load Assembly as a feature of the Assembler.

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Erasable Memory Map

The limited erasable-memory size of the LGC forced a policy of cell sharing as a means of extending memory capability in Apollo; extensive cell sharing was necessary, more than doubling the erasable complement and resulting in as many as seven distinct usages. An erasable-memory map was used as a bookkeeping and planning tool. The map was looked on as a short-lived necessity, otherwise the cell-sharing process would have been automated. In F-8 DFBW, even though memory cell sharing is limited, the Erasable Memory Map is an especially useful document. A separate map is prepared for each erasable bank by the Assembly Control Supervisor. The primary allocation is identified in the first column, with the overlays defined in the subsequent columns. The map simplifies the problem of assigning multiple use to cells or blocks of cells and minimizes the problem of run-time conflicts between LGC programs. The maps are extremely valuable to the programmer preparing erasable memory code by identifying unused blocks of cells and by aiding in the time-sharing usage of cells.

Software Development Activity

The software development process, involving all phases of software activity, can be summarized in Fig. 5. All software design is based on written specification. In Apollo, the specification was the seven volume Guidance System Operations Plan. In F-8 DFBW, the Control Laws, backup interface requirements, pilot interface requirements, and data retrieval requirements are prescribed in the Software Specification. The LGC executive hierarchy, service routines, interrupt processors, restart routines, downlink, and all others that came from Apollo are specified by inference as being the same as Apollo. The few changes in this category by rights should be documented by PCRs or ACBs. However the ultimate documentation in this area, as was similarly true in Apollo, is the detailed flowchart. Nevertheless, in the software development, authorization must exist in one of the forms: Software Specification, Program Change Request, Program Change Notice, or Assembly Control Board direction.

Another class of input to the Software Development, shown in Fig. 5, is the Initial Data Load which becomes the Preflight Erasable Load. The load is the cumulative array of values for control law parameters and for other routines' parameters and, as such, is jointly specified by FRC and CSDL. The load is revised and updated to keep pace with the software development.

A third class of input to the software development is the test plans, the most important one being the Level 4 Test Plan. Test plans exist at all levels and are the basis for the level testing. At the lower levels, the plans are informal tools to ensure thorough unit testing by individual programmers. The Level 3 Test Plan and the Level 4/5 Test Plan are carefully documented compendiums of specific tests, and cover all areas of the software. The test plan is reviewed and updated by all concerned; it can be added to at any time to include any overlooked areas.

Continuing in Fig. 5, the software is designed in blocks or units with each being tested before proceeding to the next. Testing at the unit level (Level 1/2) is generally bit-by-bit digital simulation. When a sufficient number of units are completed, the hardware and alarm interfaces are tested as appropriate. These tests generally involve all three simulators: the Digital, Hybrid, and System Test Laboratory. Modular Testing (Level 3) commences in any given area when all units in a given program function are completed, for example, the DFCS Direct Mode in the pitch axis. This level of testing continues until all DFCS modes and capabilities are completed. Since several program areas are developed in parallel, but not all at the same rate, testing at several levels takes place during any given time frame.

When all major programs appear to be essentially completed, Configuration Control is instituted, officially designating the start of Level 4, although limited Interface testing can take place earlier. Subsequent to Configuration Control, all program changes require the careful scrutiny and approval of one or more of the software supervisors, as well as the coding experts in the areas affected. Software Specification changes require a PCR. Level 4 tests are based on the Test Plan, and all incorrect, or unexpected, or incomplete, or anomalous behavior is documented in an anomaly report or a discrepancy report. Discrepancies are software errors detected after Configuration Control, but prior to release-for-manufacture. Anomalies are software errors detected after release-for-manufacture. Verification at Level 4 and above involves exercising the program on the three CSDL simulators, as well as the FRC Iron Bird System. All documented anomalies and discrepancies must be resolved. In some cases resolution of a Hybrid or Iron Bird item requires an attempt to reproduce the behavior on another simulator, or perhaps the Digital, in order to pinpoint the cause. When the cause of a discrepancy or anomaly is identified, an assessment is made to determine: (1) the operational impact when the problem is encountered if the program is left as is, (2) the procedures necessary to avoid or to work around the problem, (3) the coding change necessary to eliminate the problem, (4) the schedule impact of implementing and verifying the coding change. The assessment is documented as a PCR, PCN, or ACB which, if approved, is implemented as a fixed-coding change. Erasable coding is not used at this level for permanent changes. Disapproved PCRs, PCNs, and ACBs become program Notes. Sometimes it turns out that what was thought to be an anomaly, or discrepancy, was caused by a simulator bug, or a test deck error; in which case the problem is fixed and the test is rerun.

When all pending program changes are incorporated and tested at Level 4, and when no unresolved problems remain, the program is ready for release and is declared frozen. A technical review of the Level 4 testing is held (pre-FACI). If, in any areas the testing appears to need reinforcement, then new or additional Level 4 tests are defined. The Level 5 testing consists of rerunning all of the Level 4 test decks on the final version. If any new anomalies or discrepancies turn up and are serious enough to require a PCR, the Erasable Memory Program option is weighted heavily against a manufacturing schedule slip. The First Article Configuration Inspection (FACI) is a formal review of all Level 5 testing results, anomaly reports, change requests, program notes, and operational restrictions. The end action of the FACI is the granting of approval to release the rope assembly for manufacturing.

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Flight Support Activity

The Flight Support Activity takes place after delivery of the Manufactured rope modules and centers around Level 6 testing as shown in Fig. 6. The KSTART tape is generated from the Preflight Erasable Load involving the Initial Data Load and any existing Erasable Memory Programs. Evaluation involves careful scrutiny of all parameters, by computer Program and by eyeball, to identify and assess changes from the previous KSTART tape. Additionally, the CSDL evaluation utilizes the Hybrid Simulator, the All-Digital Simulator, and the Systems Test Laboratory hardware installation. The testing is complemented by extensive mission-sequence testing on the Iron Bird Simulator at FRC, and involves pilot training, pilot procedures, and system performance. The test results are presented at the Flight Readiness Review (FRR), and any anomalies resolved, perhaps by modifying the operational envelope. FRR approval is required for flight go-ahead. Following a successful flight to test one DFCS capability, the Initial Data Load can be modified to test another capability, or to change the downlink coverage, and the procedure of Fig. 6 is repeated.

Alternatively, the flight test results can indicate a serious need for a DFCS capability that does not exist in the rope. In this case, a PCR is submitted to request that the capability be developed as an EMP. After assessment, if the PCR is approved, the development and test of the EMP is undertaken as was shown in the previous figure, Fig. 5. When completed, the verified EMP is incorporated into the KSTART tape for Level 6 testing.

Software Milestones

The development activity is tracked by milestones. Schedule milestones were not treated with the level of formality accorded their Apollo counterparts. Small meetings of one or two technical personnel with management personnel marked many F-8 DFBW events. Nevertheless, schedule milestones were vital to a timely development and verification process. The major milestones are indicated in Fig. 2.

The Preliminary Design Review (PDR) for F-8 consisted of several meetings, each covering a specific area of interest. These were preliminary in the sense that changes were expected as subcontractors and customer had the opportunity to review carefully each other's needs, plans, and suggestions.

The Critical Design Review (CDR) also consisted of several meetings, each covering a specific area in minute detail. The CDRs for the Control System Specification and the Interface Control Document are specific examples.

Level 1, 2, 3 Testing (Unit and Modular testing) allows tracking of units of software in the early stages of development when coding and verification are relatively independent of tight controls.

Configuration Control marks the transition to tightly controlled software configuration and testing procedures.

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Level 4 Testing (Interface testing) allows tracking of interfaces between modules of software. Program changes require written approval and all anomalous simulation behavior requires documentation, analysis, and resolution.

Level 5 (Formal testing) allows tracking of software prototype.

First Article Configuration Inspection (FACI) is a formal review of all aspects of prototype software. The final action is the approval of the final assembly for manufacture.

Release-for-Manufacture—Following FACI approval, a weaving tape is generated from the final assembly to be used for core-rope manufacture.

Level 6 Testing (Mission Performance testing) is based on the KSTART tape for the particular flight. Evaluation consists of exercising the KSTART tape on the three CSDL Simulators and on the FRC Iron Bird System.

A Flight Readiness Review (FRR) is conducted prior to each flight. A statement from CSDL is required concerning its review on the Preflight Erasable Load and KSTART tape. The initial FRR had the longest agenda. The review assessed the flight readiness of the primary control system, the backup control system, the flight vehicle subsystems, to name a few. Known anomalies and their avoidance or work-around procedures were discussed. Erasable Memory Programs were explained, both functionally and operationally. The failure analysis studie were reviewed, as well as available documentation. Flight readiness reviews subsequent to the initial flight generally consider the current KSTART tape and any newly applicable areas.

SOFTWARE VERIFICATION

The software verification process is vital to the preparation of reliable high-quality software. A screening process is employed, whereby code is subjected to many tests representing many different situations. This approach to testing is one of diminishing returns: early tests show up most of the coding errors, but the later tests build confidence in the overall quality of the program assembly. Establishing the proper balance between insufficient and excessive verification testing is a critical task. Indeed, the verification process does not terminate with release-for-manufacture; it continues, in the hope of catching any remaining errors before they show up operationally with unexpected and perhaps dangerous consequences.

The verification process cannot be separated from the assembly control process, at least prior to release-for-manufacture. The ultimate quality and reliability of code depends heavily on the verification process. The attainment of the verification goals involves far more than the execution of high quality object code available near the end of the software development cycle. Facilities are required in the early stages of program development when the code available is of low quality and may not even be executable. In the early stages a benign and cooperative environment is required; it must provide a detailed

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visibility into the execution of code. Simplified, but fast-operating environment algorithms are desirable. Extensive diagnostic capability is mandatory, involving both run-time and post-run software packages. As code quality is refined, the environment quality can be updated to include such factors as sensor errors and higher order effects. Ultimately the code should be exercised in a highly realistic environment including as much real hardware as possible.

Software Verification Facilities

Several distinct facilities were utilized during the DFCS verification process. The complementary nature of their unique capabilities is significant. Each has contributed to the DFCS quality, and by its absence would have affected the development adversely, mainly in terms of schedule, but perhaps even in terms of operational performance. CSDL has utilized the All-Digital Simulator, the Hybrid Simulator, and the System Test Laboratory facilities for the software development and verification activities. NASA/FRC has utilized the analog Stage 1 engineering simulation, the bench lashup Stage 2 hardware integration simulation, and the Stage 3 Iron Bird Simulator for the systems design, hardware integration, design verification, and pilot training/evaluation activities.

Each of these facilities has contributed to the overall success of F-8 DFBW, but certainly the significant contributions to system integration have come from the Stage 3 Iron Bird Simulator. It was on this facility that significant hardware integration problems were first encountered. The Stage 3 piloted simulations gave insight for design-change evaluation. Stage 3 permitted real-time demonstration of failure effects, and permitted engineering preliminary and final design. Stage 3 was used for much supportive software verification and essentially all of the system design verification. For the flight testing, where CSDL's verification role was supportive, the Stage 3 simulation was especially important as the primary design, verification, and training tool.

The All-Digital Simulator (ADS) at CSDL played the significant role in F-8 software design, development, and verification, primarily because of the powerful run-time diagnostic and post-run edit capability, as well as features such as repeatability and snapshot/rollback. Rigidly controlled simulator software provided a stable environment and ensured repeatability.

The Hybrid Simulator at CSDL was a very useful tool during preliminary verification, primarily because of its real-time interactive capabilities. Its role was somewhat diminished because CSDL did not have DFCS design responsibility, which is where the real-time interactive aspects of hybrid simulation can vastly improve the control-system designer's efficiency. However, on two separate occasions, one being the time-critical development period between Stage 2 and Stage 3 simulation, NASA/FRC came to CSDL and conducted basic and detailed design on our Hybrid facilities.

Piloted simulations early in the development phases can improve the overall quality of the end item, especially when schedules are tight. Pilot contributions cover a wide range of experience including such items as human

factors suggestions, functional change requests, performance and handling qualities evaluation, and safety considerations.

The complementary nature of all-digital, hybrid, and hardware integration facilities is important. The ADS provides diagnostic and edit capability plus detailed hard-copy for documentation. The Hybrid Simulator is unmatched in its real-time interactive capabilities for preliminary design, parameter-variation, and sensitivity studies. The hardware integration facility represents the ultimate interface verification tool short of flight test. Here, interfaces are actually mated, often for the first time. Failures can be studied and pilot-in-the-loop evaluations based on a maximum hardware complement can be performed. Each of the design, development, verification, and training tools can play a key non-overlapping role. It is the complementary nature of each facility which should be emphasized and utilized for greatest program efficiency and end-item quality.

A brief description of each of these facilities follows.

CSDL All-Digital Simulator —The Apollo Digital Simulator is a basic tool developed and employed primarily to support the design, development, and verification of Apollo Guidance Computer (AGC) programs. The simulator is entirely digital and consists of a number of programs implemented on a general purpose digital computer. It simulates the operation of the AGC in storage layout, and in detailed arithmetic and logical operation. Consistent with one's objectives, mathematical and logical models ranging from rudimentary to comprehensive may be selected to simulate the hardware and flight environment within which the AGC and its coding operate. For the F-8C, only the rigid body degrees of freedom are mechanized and there is no takeoff or landing capability. The BCS flight control system is not simulated, so controlled flight is possible only in the DFCS modes. The Pilot Action Simulator provides open-loop actions such as stick and rudder deflections, push button and trim switch activity, and DSKY operations. In addition, the simulator has numerous on-line diagnostic features, a snapshop/rollback capability, and extensive post-run edit capability available. The edit package provides for flexible run-time data storage and for post-run data retrieval. The user has the choice of using standard edit programs or of writing his own. Extensive edit programs for plotting, computational verification, and formatting were developed for F-8 formal verification. Summary printing includes data on DFCS mode changes, timing, and computational delays. Plot variables include numerous DFCS and environmental quantities. Timing data indicating duty cycle and job activity is plotted. A downlink processor edit was prepared to verify proper downlink operation. The simulation system is illustrated schematically in Fig. 7.

The CSDL Hybrid Simulator—The Hybrid Simulator is a combination of selected flight hardware used in concert with analog and digital computers to provide real-time simulated flight. The flight hardware consists of an LGC computer, a DSKY, and the coupling data units. The LGC memory is replaced by a Core Rope Simulator (CRS), which provides a complete erasable memory as well as helpful features, such as the ability to monitor and change location contents, to stop at a location address, or to single-step the program. The IMU

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is simulated with special-purpose electronics. Elements needing precision of storage, as the trajectory dynamics, the aerodynamics, and the rotational transformations, are simulated in an XDS 9300 digital computer. The high-frequency actuator dynamics, the BCS loops, and some discrete logic are simulated on the analog computer. The algorithms for BCS control and BCS downmode-trim initialization are simulated, but the cross-channel comparator and the hydrologic subsystems of the F-8C are not modelled. Also, provision is not made for a parking, landing, or takeoff capability. A minimal cockpit uses the Apollo three-axis rotational hand controller in place of stick/pedal controls. Cockpit instrumentation includes artificial horizon, altitude, airspeed, rate-of-climb, % thrust, g, angle of attack, and a mockup Mode And Power Panel for real-time man-in-the-loop simulations. Strip-chart recordings and initialization printout are the only hard-copy output. The Hybrid Simulator runs in real time to allow man-in-the-loop testing, on-line debugging, and flexibility in verification procedures. The LGC can function alone or with the Simulator providing an environment; in the former mode it is available independently of the availablity of the hybrid facility. Reproducibility is not in general possible, but this is an advantage in that a realistic randomness is introduced into the testing.

CSDL System Test Laboratory—The System Test Laboratory (STL) is an Apollo hardware integration facility. A real IMU interfaces with the LGC, CRS, and DSKY. Uplink and downlink are operational. Channel inbit discretes can be set or cleared manually and independently. The aircraft and BCS systems are not simulated. A trace capability is available via the Apollo CORONER and off-line processing; this is the only hard-copy output from this facility.

NASA/FRC Stage 1 Simulator—The Stage 1 Simulator was a preliminary de-sign tool used to develop the flight control system specification equations. Simple analog models and sample-and-hold networks were utilized. Linear analysis based on continuous and sample-data control system design, using root locus and w-plane techniques, provided backup for the simulation effort.

NASA/FRC Stage 2 Simulator—The Stage 2 Simulator was a hardware inte-gration and preliminary design evaluation facility. Breadboard lashup of major hardware components was first performed here. The LGC, the Program Analyzer Console (PAC, equivalent to the CRS), DSKY, IMU Gimbal Angle Simulator (GAS), and CDU package were involved. Aircraft and aero-surface servo actuator dynamics were modelled on a small analog computer. A rudimentary version of the DFCS and Operating System software participated.

NASA/FRC Stage 3 Simulator—The Stage 3 (or Iron Bird) Simulator is an F-8C airframe that includes all key hardware in the configuration of the flight article, including the pallet mounting of the LGC computer, IMU, and CDUs. The BCS electronics, power supplies, and hydraulics are flight-article type systems. The manufactured core-rope or PAC software can be used as the LGC memory. Simu-lated trajectory dynamics and aerodynamics permit closed-loop simulations using the GAS. Simple external visuals, sideslip angle and horizon line with sky/earth differentation, are provided on a TV screen mounted on the aircraft nose. Access to LGC and flight control system variables is by means of downlink with post-run editing or by DSKY display.

Software Verification Testing

.It is difficult to separate software development and software verification since both go hand in hand throughout the development phase. To consider software verification it is necessary to consider software development. Generally speaking, there are two categories of software design changes that contribute to program construction.

(1) Developmental changes - these are creation of a new program or a new routine, or extensive changes within an existing program or routine.

(2) Incremental changes - these are modifications to existing code that cause small alterations and repercussions.

Clearly, a Developmental change has a major impact on the existing program and requires an extensive testing approach to assure that the new code works properly and does not interfere with other existing coding. It is equally clear that an Incremental change has a minor impact on the existing code and requires a localized testing approach. This is sort of by definition. However, it is not always clear into which of the two categories a given software change should be placed. Classification is a difficult problem and requires experience and thorough knowledge of the programs. For example, a one word change could require extensive testing if that word were, say, a sample period affecting event timing. On the other hand, the replacement of one Boolean relationship by another, involving perhaps 30 words, could be local in effect and require only local testing. Thus, the full arsenal of testing is brought to bear on Developmental software, while a subset is used for Incremental software.

Developmental Software Testing—In order to test out developmental changes, the six official levels of testing are normally performed. These are Unit testing (Levels 1 and 2), Modular testing (Level 3), Interface testing (Level 4), Formal testing (Level 5), and Mission Performance testing (Level 6). The majority of the F-8 DFBW programming effort falls into the developmental category, as exemplified by the flight control coding, input/output processing, ground test programs, and special routines. Design changes that occur late in the development cycle are often accorded the Developmental treatment. Erasable Memory Program design is also in this category, although there have been exceptions.

Incremental Software Testing—Incremental changes require adequate testing to assure that all paths in the program affected by the change are exercised. This may necessitate designing new tests for specific code changes. Incremental testing involves some combination of Unit testing, Modular testing, and Interface testing. Since all incremental changes become part of the program rope, they are automatically subjected to Level 5 and Level 6 testing.

There have been a number of incremental changes in F-8 DFBW. Initially, much of the software (about 60%) came from the Apollo Lunar Module Program. Many areas of the code required minor incremental changes to meet F-8C requiremen(Late in the development cycle, especially as the release-for-manufacture date approached, changes even to flight control code can often be treated as incremental, especially if significant Level 4 interface testing has already been completed.

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Some Erasable Memory Programs have been classified as Incremental. In one case, two lines of code were added to an existing EMP to create the one-pulse rudder pedal deadband. The other case was a preflight checkout program. These have received minimal Level 4/5 testing. Conversely, other EMPs involved significant design changes deeply imbedded in interface or systems code: parabolic shaping of stick inputs, or restart-triggering of BCS downmoding. These have received significant Level 4/5 testing, being developmental in nature.

Special Testing—There are a number of special tests deserving of mention that establish confidence in the flight software mainly by failing to find a fault rather than by exhaustively proving every possibility. This approach is in general true when the number of ways to exercise the code becomes unwieldy. The fact that interaction between the Executive, interrupt processors, and service routines falls into this category can be overlooked. A specific example is restart testing where a large number of artificially generated asynchronous time-triggered and location-triggered interrupts exercise the restart protection mechanism. Stress testing involves testing operational sequences under abnormal conditions. Potential anomaly testing attempts to duplicate the event sequences which led to questionable behavior on another hybrid facility. Hybrid testing occasionally encounters unexpected behavior that is usually a hardware problem, but can be a software problem. If a problem is found, digital testing gives conclusive evidence. Alternatively, if no problem is found, a measure of confidence is restored.

An 'eyeballing' effort was performed on the F-8 DBFW assembly just prior to release. Experienced Apollo programmers were assigned sections of the code to eyeball for errors, based on their accumulated experience. Several errors were uncovered, although off-nominal operational procedures would have been needed to encounter difficulties. The fact that errors were found gave weight to the effort as a worthwhile task. The absence of any serious errors, and the minimal number of errors encountered, added to the confidence level being built by the verification process.

Input and Output Discrete Failure Effects

A formal failure effects investigation was conducted late in the development cycle by CSDL and by other systems contractors. All interfaces were studied for fail-on and fail-off effects. Engineering analysis was the primary investigative tool, but simulated failures were utilized whenever pilot-in-loop problems were expected. To this end, a special version of the mainline program was created for the Iron Bird and was given the capability to fail any selected input/output discrete in the off-state or on-state. Failures were introduced during Iron Bird piloted simulations by a test engineer at the DSKY. The capability enabled pilot training in recognition and recovery procedures.

An important conclusion of the failure analysis is that such studies should be initiated early in the preliminary design phase so that failure effects can be recognized and avoided by careful design of hardware, software, and interfaces. Early recognition leads to design changes that often can be incorporated at no additional cost, whereas late recognition can be quite expensive.

Erasable Memory Programs

The concept of an Erasable Memory Program only has application in reference to a fixed memory computer when the capability to manufacture a new fixed memory is no longer available. Certainly, as long as the capability does exist, the redesign of a portion of the program or the inclusion of a new portion poses no particular problem even in a relatively mature program. In F-8 DFBW for example, the result of early Iron Bird simulations uncovered a hardware interface problem in that the anti-dropout filter in the CDU error counters interfered with restart recovery. Since the software was still under development, a straightforward redesign of the restart recovery routine was undertaken, including redevelopment and verification. On the other hand, when the ability to remanufacture the rope memory is gone, it is necessary to resort to an artifice, like erasable memory programming, if any change is to be incorporated into the program flow. If, however, one is dealing with a programmable memory computer, then post-release software changes are treated the same as pre-release software changes. The purpose of this section on EMPs then is to illustrate by example that sufficient cause for software changes can and will arise after program release, and to describe the F-8 DFBW experience.

Some of the late Stage 3 Iron Bird discoveries were not compatible with software development schedules, bound as they were by the anticipated shutdown of the core-rope manufacturing facilities. Erasable memory programming and major hardware changes were required instead. For example, piloted simulations in early 1972 indicated pilot-response problems with certain computer failures. The work-around concept was straightforward and a software change could have been made, except that the DFCS was no longer software; core-rope manufacture was under way. Fortunately, an Erasable Memory Program (EMP-001, Restart Downmoding to BCS) could do the job, so remanufacture was not necessary. However, the design and especially the verification tasks were much tougher for the EMP than they would have been for the fixed-memory equivalent, a characteristic of most erasable memory programming. Nevertheless, the flexibility provided by last-minute software changes represents a major selling point for digital flight control.

Design changes to minimize the effects of stick/pedal input quantization were not formalized until after the first flight. Hardware changes had been made earlier, prior to core-rope manufacture, but these proved to be inadequate. Again, an Erasable Memory Program (EMP-004, Parabolic Stick Shaping) provided an acceptable approach, but the fixed-memory equivalent would have been easier to design, develop, and verify. Also, the DFCS computational burden would have been lower with the equivalent fixed memory code, and operational aspects would have been simpler.

Problems do not always show up during the systems analysis and preliminary design phases, no matter how detailed the activity, but instead crop up during the hardware integration phase, or even worse, conceal their identity until the flight test phase. F-8C, during high-q flight for example, encountered a single-pulse null shift in the output from the pedal LVDT, which supplies the rudder pilot commands to the DFCS. The phenomenon apparently has something to do with

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airframe distortion at high-q flight conditions. Neither the Stage 3 Iron Bird Simulator nor preliminary analysis models could indicate such a phenomenon. In this case, the hardware problem of rudder bias shift was eliminated by software, by inserting a one-pulse deadband (EMP-007, Single-pulse Pedal Deadband). There is a real motivation for a flight test phase, however brief, between the proto-type and production software.

CONCLUDING REMARKS

The F-8 DFBW is an experimental digital fly-by-wire testbed flight control system, implemented with Apollo off-the-shelf hardware. Existing off-the-shelf software and software control techniques were dictated by hardware as well as manufacturing schedule limitations. Software design was bottom-up. Time-efficient code was important because of LGC speed. (Some of the techniques discussed would not be applicable for a modern, faster, all core computer.) Despite the LGC fixed memory, post-manufacturing design changes to the Specifica-tion were possible through Erasable Memory Programs. Proof of the benefits that accrue from good software control and from careful and thorough verification testing is evidenced by the F-8 DFBW flight test program results: In a year and a half, 42 flights, totaling 58 hours of flight time, were made successfully without any DFCS inflight software failures or performance surprises.

REFERENCES

The author has made generous use of References 1 and 2. Apollo hard-ware details are not included but can be found in Reference 2 and Reference 3. Supportive use was made of References 4, 5, and 6.

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TABLE 1

APOLLO HARDWARE USED IN F-8 DFBW

LGC   —   LM Guidance Computer (approximately 2k of erasable and 36k of
              programmable fixed core-rope memory; programmable hardware-
              interrupt and software-executive systems; hardware restart
              logic, etc.).

DSKY   —   (LM) Display and Keyboard (three 5-digit-plus-sign display
              windows; miscellaneous warning lights; keyboard including 0
              through 9, +, -, PRO (proceed), ENTR, CLR (clear), VERB, NOUN,
              etc; the DSKY is the computer/astronaut or computer/ground
              crew interface).

IMU   —   Inertial Measurement Unit (a three-gimballed gyroscopically-
              stabilized platform for the PIPA accelerometers; gimbal angle
              resolver and PIPA signals ultimately interface with the LGC;
              several platform alignment techniques are under LGC software
              control).

CDU   —   Coupling Data Unit (for analog-to-digital conversion of IMU
              gimbal angle indications; for digital-to-analog conversion
              of LGC computer outputs; for control of IMU moding; includes
              failure detection; used to derive body axis angular rates).

PIPA   —   Pulsed Integrating Pendulous Accelerometer (three mutually-
              perpendicular contact-acceleration-sensing and incremental-
              velocity-indicating devices located on the IMU stable member,
              with a direct LGC interface; used to derive body axis normal
              and lateral acceleration).

PSA   —   Power and Servo Assembly (power supplies, amplifiers, etc.,
              for inertial subsystem).

PTA   —   Pulse Torque Assembly (input/output processing for inertial
              subsystem).

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TABLE 2

HARDWARE UNIQUE TO F-8 DFBW

MAPP   —   Mode and Power Panel (computer and IMU power control, auto-pilot gain and mode select/indicators, warning indicators, etc.

IFB   —   Interface Box (junction box containing an Apollo DAC stick/pedal comparators, special amplifiers, etc.).

BCS   —   Backup Control System (triply-redundant stick/pedal to aero-surface open-loop control, with trim, hydrologic comparator; cross-channel comparator; etc.).

DLC/IFR   —   Downlink Converter/Inflight Recorder (100 word-pair list every 2 seconds on a 20ms interrupt; recording on FM tape for post-flight processing/review).

GSE   —   Ground Support Equipment (the Apollo Program Analyzer Console (PAC) for simulating LGC hard-wire rope memory; the Uplink Converter (ULC) for preflight erasable loading and for DSKY-type program control via tape; the Ground Test Cart containing downlink converter/ground recorder, miscellaneous switches and indicators; etc.).

SPCC   —   Servo Pressure Control Console (PRI select/indicators for each axis; servo pressure switches and indicators for each BCS servo-valve and for PCS servo-valve pairs; each switch has three positions: OFF which disables that valve, AUTO which enables that valve, and MAN which overrides any auto-matic moding and locks that valve into the active state).

CCS   —   Coolant Control System (coolant for IMU, computer, etc.).

TABLE 3

F-8 DFBW FIXED-MEMORY ALLOCATION

F-8 DFBW Flight Control System (total)		5586
Body Rate/Acceleration Feedback	320	
Generalized Feedback Filters	1930	
Pilot Stick/Pedal Processing	168	
Control Loop Equations	1178	
Channel Monitor Routine	523	
Gain/Mode Change Routine	985	
Initialization/Restarts/Miscellaneous	482	
Ground Test Programs/Extended Verbs		768
Self Test/Check		1436
Fresh Start/Restart/V37/etc.		853
Display Interfaces/Pinball/etc.		3578
Interpreter/Executive/Waitlist/Downlink/Uplink/etc.		3830
IMU Alignment, Compensation, and Tests/T4RUPT		3263
TOTAL F-8 DFBW FIXED-MEMORY USED		19314
TOTAL LGC FIXED-MEMORY (36 FBANKS AT 1024)		36864

TABLE 4

F-8 DFBW ERASABLE-MEMORY ALLOCATION

Preflight Erasable Load (total)		389
F-8 DFBW Flight Control System	169	
IMU Compensation/Alignment	33	
Erasable Downlink List	100	
Erasable Memory Programming (EMP-001,4,7)	87	
F-8 DFBW Flight Control System Working Registers		321
Extended Verbs/Ground Test Prog/Miscellaneous		50
Self Test/Check		263
IMU Alignment/Perf Test/Ops Test		17
Uplink/Downlink		32
Display Interfaces/Pinball/etc.		56
Executive/Waitlist/Service/Centrals/etc.		468
TOTAL F-8 DFBW ERASABLE-MEMORY USED		1596
TOTAL LGC ERASABLE-MEMORY (8 EBANKS AT 256)		2048
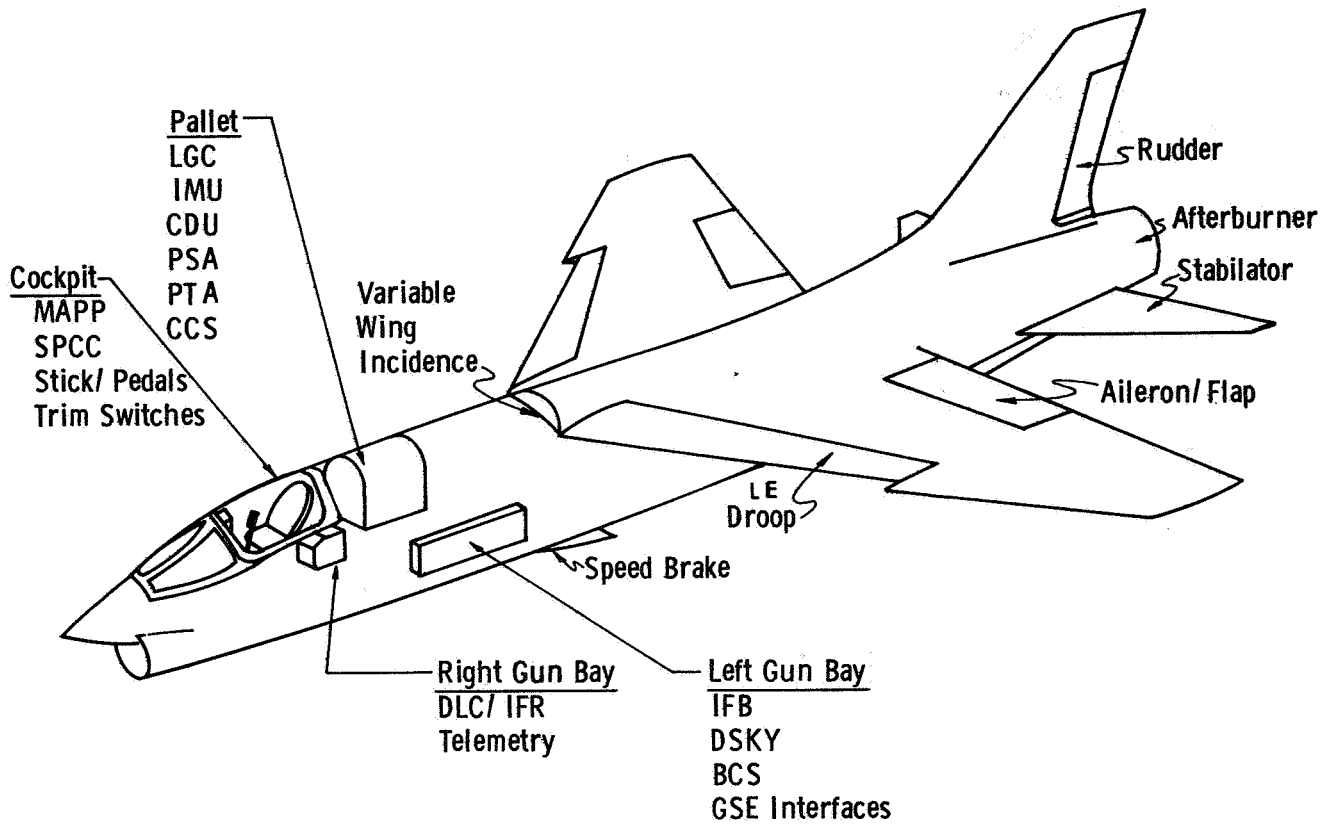
Fig. 1. F-8C DFBW Aircraft and Hardware
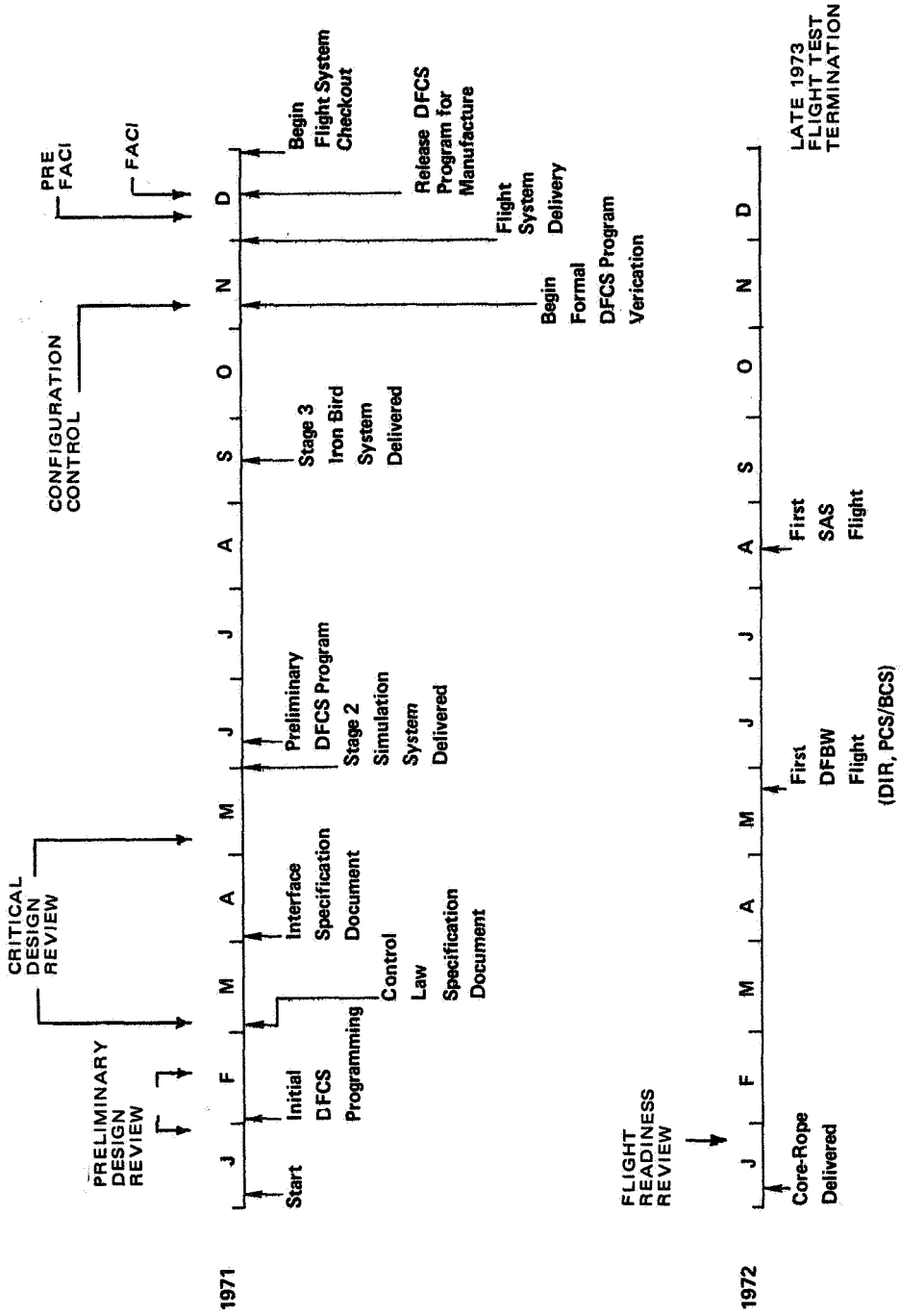
Fig. 2. F-8 DFBW Chronology
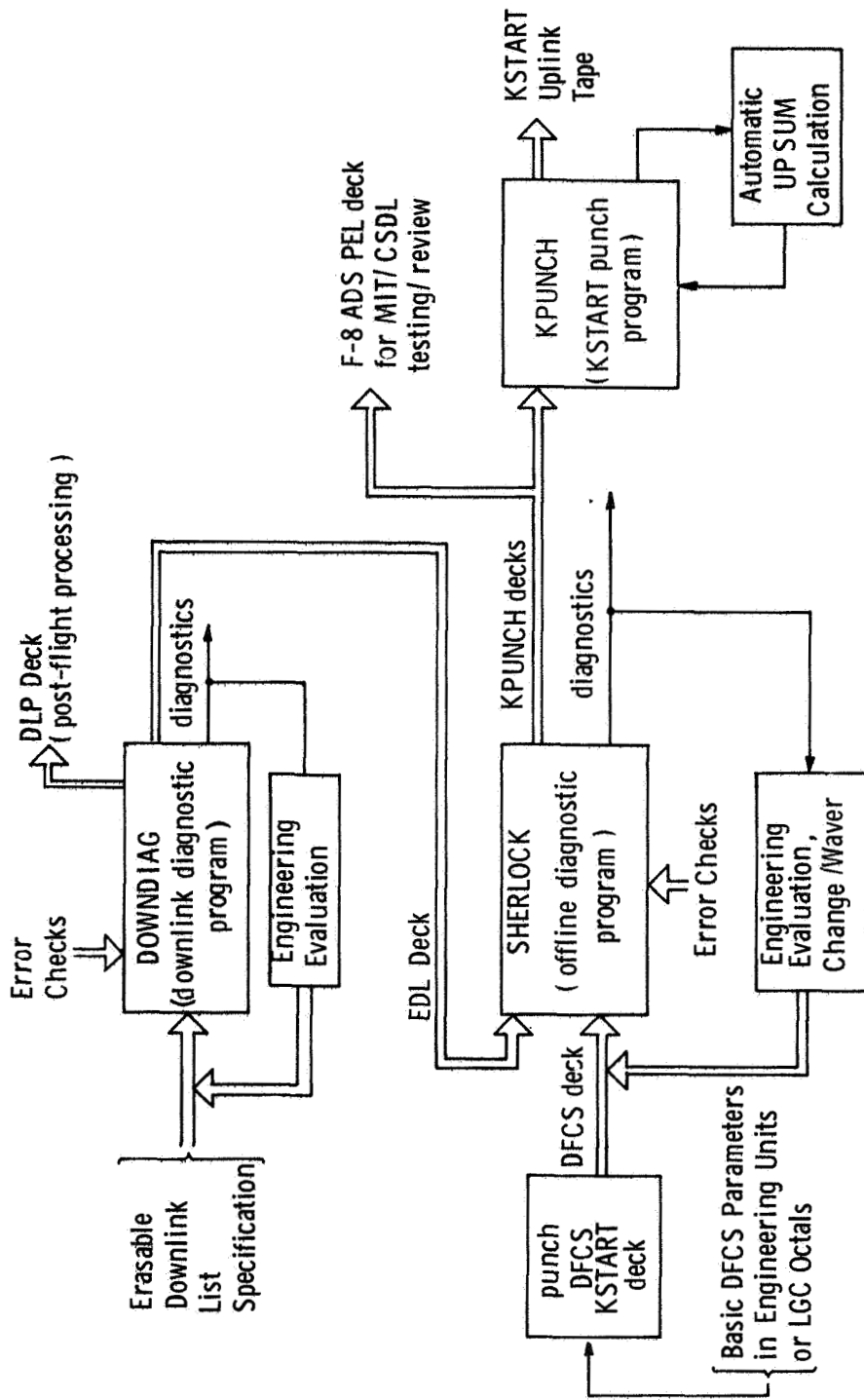
Fig. 3. KSTART Preparation

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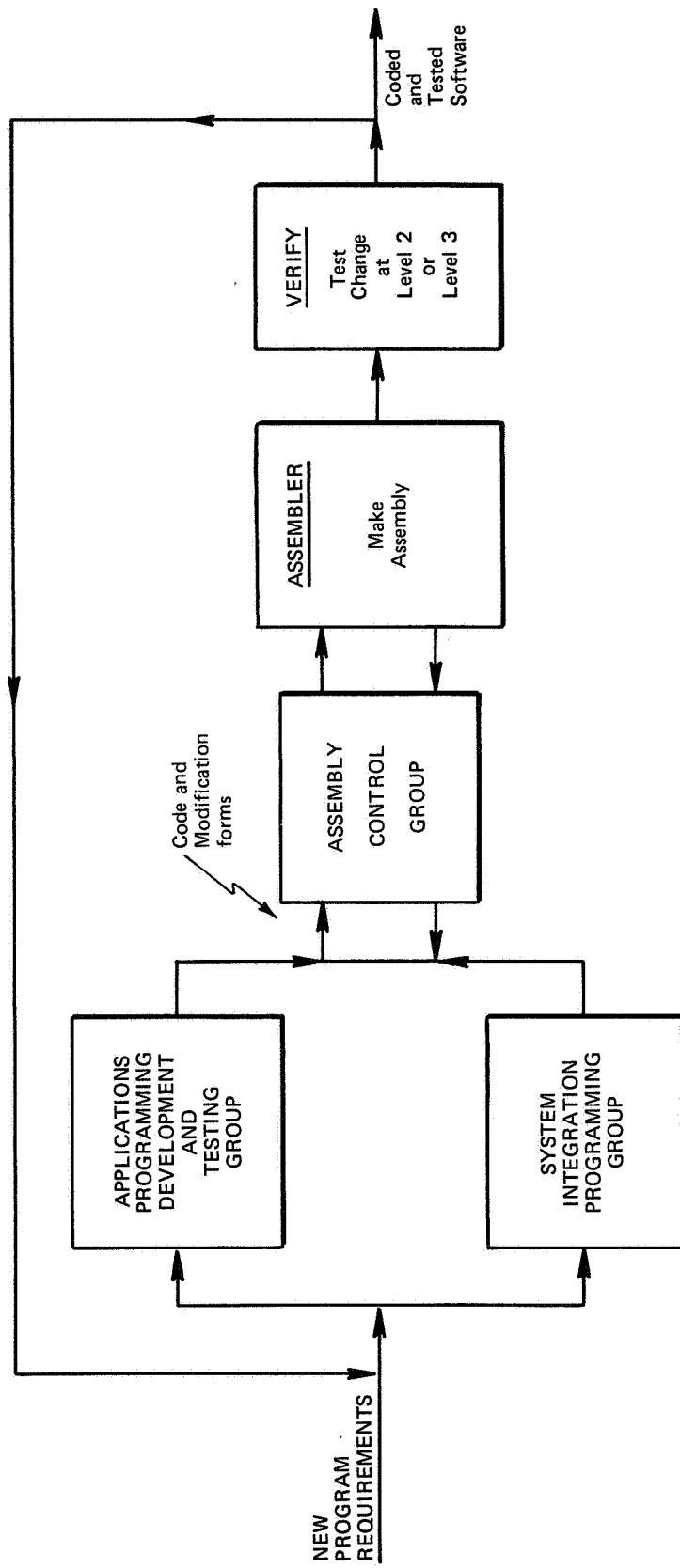
Fig. 4. Program & Assembly Control Activity

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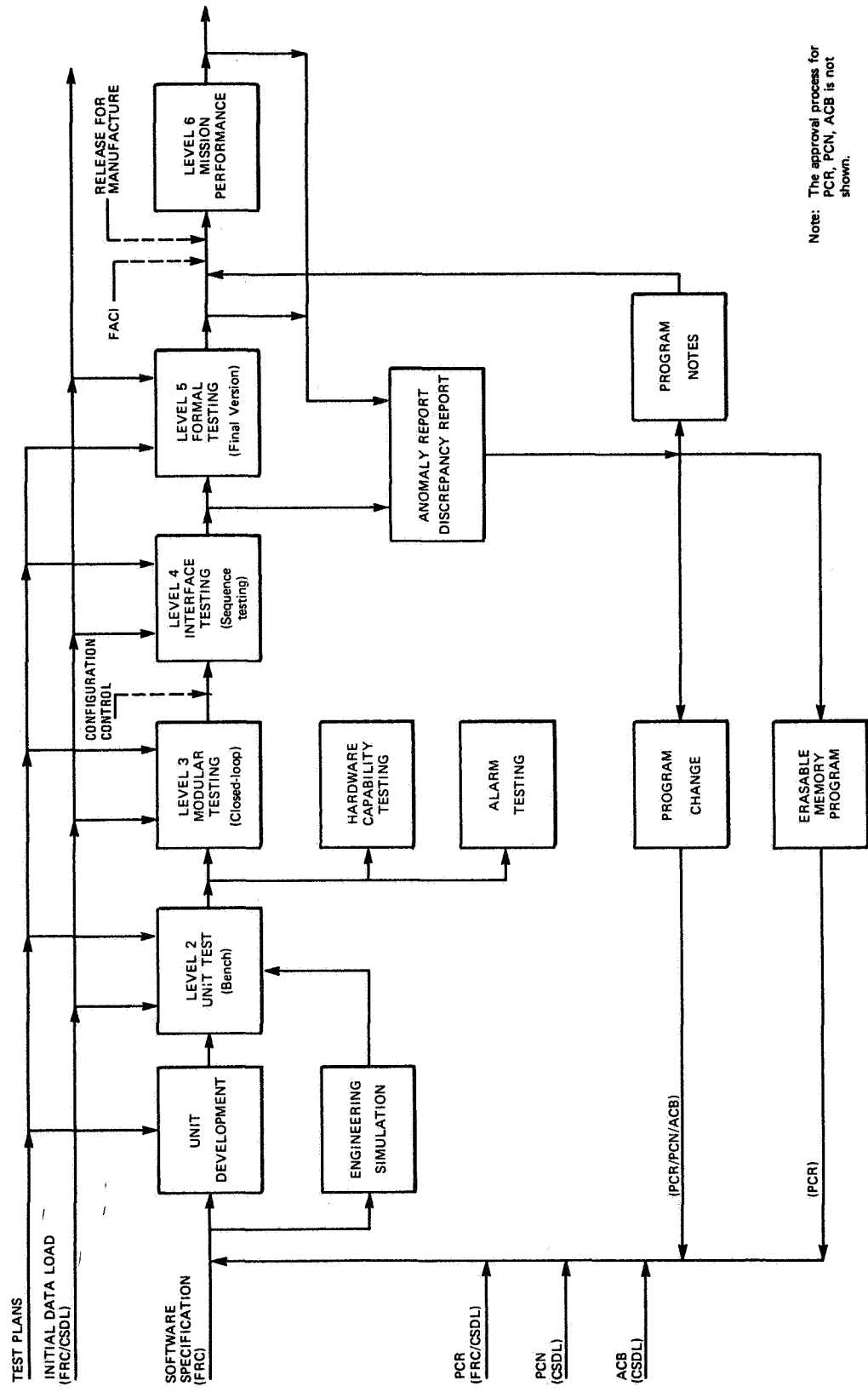
Fig. 5. F-8 DFBW Software Development Activity
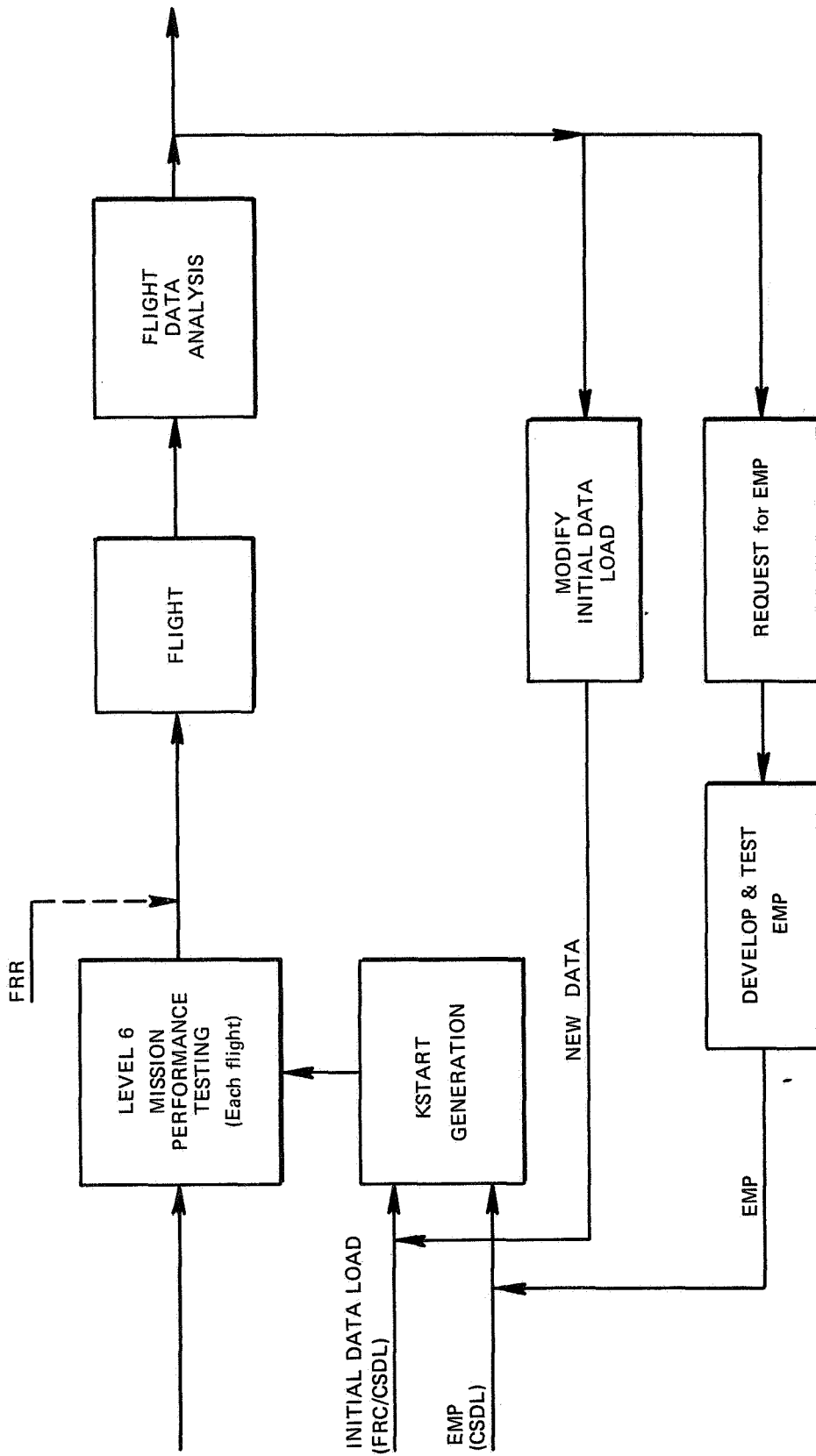
Fig. 6. Flight Support Activity

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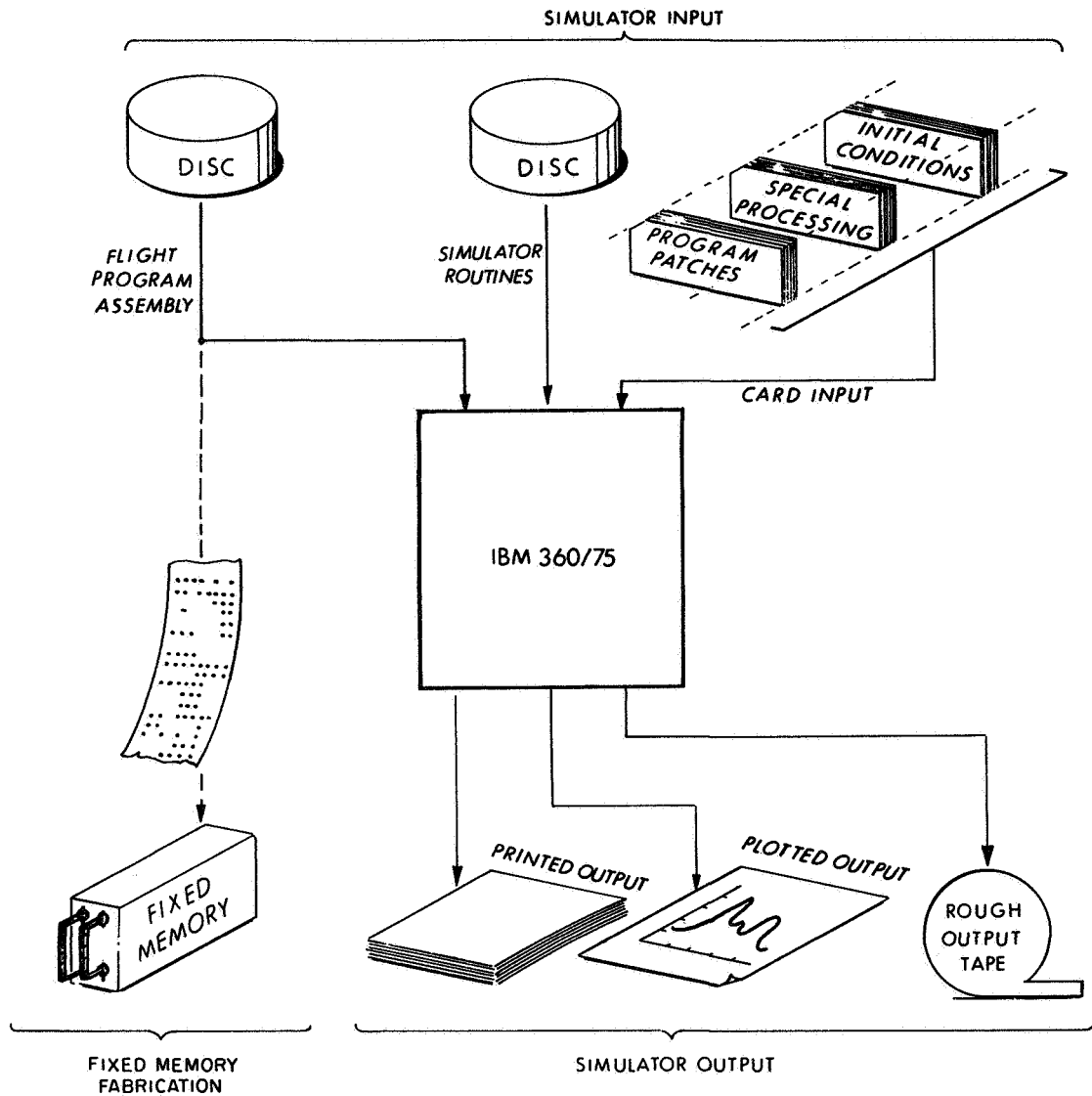
Fig. 7. Simulator System Schematic

FLIGHT TEST EXPERIENCE WITH THE F-8

DIGITAL FLY-BY-WIRE SYSTEM

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SUMMARY

Flight test results of the F-8 digital fly-by-wire (DFBW) control system are presented and the implications for application to active control technology (ACT) are discussed. The F-8 DFBW system has several of the attributes of proposed ACT systems, so the flight test experience is helpful in assessing the capabilities of those systems. Topics of discussion include the predicted and actual flight performance of the control system, assessments of aircraft flying qualities and other piloting factors, software management and control, and operational experience.

INTRODUCTION

In May 1972 the flight testing of the F-8 DFBW aircraft began. This aircraft, which used Apollo guidance and navigation system hardware, was the first to rely on a DFBW system for primary flight control. The design and development of the F-8 DFBW control system are described in references 1 to 3. This paper presents the major flight test results for the control system. A detailed description of the system's software development and verification is given in reference 4, and the backup control actuation systems are described in reference 5.

The primary objectives of the flight tests were to evaluate the performance of the digital flight control system and to acquire operating experience with it. The program also served to determine whether the long-advertised advantages and capabilities of DFBW control systems could be realized. Many of these advantages, such as software flexibility, system reliability, and computational ability, make a DFBW system a logical candidate for active control technology applications. The F-8 DFBW control system had characteristics in common with systems proposed for ACT applications. Specifically, it was a highly reliable, full authority system that was committed for use from the first takeoff and landing. An analog control system was the only backup to the DFBW system. The mechanical controls of the basic F-8C airplane were removed before the first flight.

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This approach parallels that taken toward the development of an active control system, both in terms of the importance attributed to the design of the control system and the reliability and management of hardware and software, and in terms of the requirement for detailed preflight testing. This paper emphasizes the aspects of the flight test program that relate to the broader considerations of an active control system.

SYMBOLS

a_1, a_2, a_3, b_1, b_2, b_3, digital filter coefficients

$C^* = n_Z - \dfrac{V_{co}}{57.3g}\, q$, g

$G(s)$ general s-plane filter

$G(w)$ general w-plane filter

$G(z)$ general digital filter

K general gain constant

K_{C^*} C^* feedback gain, deg/g

K_p roll rate feedback gain, deg/deg/sec

K_q pitch rate feedback gain, deg/deg/sec

K_r yaw rate feedback gain, deg/deg/sec

L_{δ_a} roll acceleration due to aileron deflection, $\mathrm{deg/sec^2/deg}$

M Mach number

M_{δ_e} pitch acceleration due to elevon deflection, $\mathrm{deg/sec^2/deg}$

N_{δ_r} yaw acceleration due to rudder deflection, $\mathrm{deg/sec^2/deg}$

n_Z acceleration along positive Z-body axis, g

p roll rate, deg/sec

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q	pitch rate, deg/sec
r	yaw rate, deg/sec
s	Laplace transform variable
T	sample period, sec
V	velocity, KIAS
V_{co}	crossover velocity, m/sec
w	sampled-data system frequency domain variable
z	sampled-data domain transform variable
Δ	incremental change
δ	general surface command, deg
δ_{a_p}	pilot roll stick deflection, cm
δ_e	horizontal stabilizer deflection, deg
ζ	damping ratio
θ	pitch attitude, deg
$\tau_{r_{eff}}$	effective roll mode time constant, sec
φ	roll attitude, deg
ψ	heading angle, deg
ω	natural frequency, Hz

Subscripts:

d	Dutch roll mode
n	current sample
$n-1$	last sample
p	pilot
sp	longitudinal short period mode

ss	steady state
Z	component along aircraft Z-body axis in positive (down) direction
$(\hat{\ })$	derived quantity ( )

ABBREVIATIONS

ACT	active control technology
A/D	analog to digital
CAS	command augmentation system
D/A	digital to analog
DFBW	digital fly-by-wire
DSKY	display and keyboard
KIAS	knots indicated airspeed
PCM	pulse code modulation
PIO	pilot-induced oscillation
SAS	stability augmentation system

CONDUCT OF FLIGHT TEST PROGRAM

Figure 1 illustrates the nature and sequence of the phases of the flight test program. The first three flights were made by using the proportional control, or direct, digital mode. The fourth flight culminated in a landing during which three-axis DFBW stability augmentation was used. The evaluation of the DFBW control system progressed rapidly from then on, and by the eighth flight all modes had been flown. The airplane was then evaluated in a variety of tasks, including ground-controlled approaches, gunsight tracking, mild aerobatics, and formation flight. The latter portion of the flight program concentrated on flying qualities assessments by additional pilots and on an evaluation of a minimum-displacement side stick that operated through the backup control system only (ref. 5). In total, 58 hours were accumulated by six pilots during 42 flights.

The F-8 DFBW system was flight tested within the flight envelope shown in figure 2. Most of the closed-loop evaluations were made at speeds between 250 knots indicated airspeed (KIAS) and 400 KIAS and altitudes from 6000 meters to

10,700 meters. Tests at low speeds (below 200 KIAS) were made with the variable-incidence wing of the F-8C airplane in the up position. Pilot ratings were given in accordance with the Cooper-Harper scale (ref. 6).

All flights were conducted during the daytime under VFR conditions. They averaged 80 minutes in duration. Each flight was monitored in a control room in which 36 airplane parameters were displayed. In addition, duplicates of the pilot's mode panel and servo status panel showed the state of the fly-by-wire control system. All parameters were telemetered from the aircraft's pulse code modulation (PCM) data acquisition system.

CONTROL SYSTEM PERFORMANCE

The digital flight control system consisted of pilot-selectable modes in each axis. The mode panel layout is described in reference 3. The available modes are shown by axis in the table below:

	Axis		
	Pitch	Roll	Yaw
Modes available	Direct	Direct	Direct
	SAS	SAS	SAS
	CAS	Test	– – – –

The direct mode, which had no augmentation, and a stability augmentation system (SAS) mode were provided in each axis. A command augmentation system (CAS) mode was also available in the pitch axis. The roll test mode was used to facilitate comparisons between various SAS mode configurations. Block diagrams of the digital control modes are shown in figures 3(a) to 3(c).

Direct Mode

The direct mode provided proportional control with no augmentation. Figure 3(a) shows the direct mode mechanization, which was similar in all axes. Analog-to-digital (A/D) quantization of the stick outputs, effective quantization on trim due to sample rate, and digital-to-analog (D/A) output quantization are aspects of digital flight control that were apparent in this mode. Linear and nonlinear stick shaping

were used during the flight program. In the pitch axis, linear and parabolic shaping were used (fig. 4). The Apollo A/D interface allowed a maximum of 45 quantization levels for full stick or pedal deflection in one direction. The Apollo computer D/A converter output quantization, which had ±384 levels, was approximately an order of magnitude finer than the stick A/D converter. The linear gearing mechanization resulted in a quantization level of 0.59° of horizontal stabilizer deflection when full pitch control authority was retained. During early flights, various linear gearing gains were evaluated. Table 1 summarizes the pitch quantization effects found with linear gearing. The threshold of quantization detection appeared to be from 0.15g to 0.2g and 1.2 degrees per second to 1.5 degrees per second of peak pitch rate. Figure 5 shows an example of the thumping that the pilot detected at 365 KIAS as he attempted to increase pitch rate smoothly. This small airplane excitation was characteristic of the quantization effect in the pitch and roll axes resulting from control surface actuator response to staircase commands.

The parabolic stick shaping resulted in a nonlinear quantization. The step size is shown in table 2. This shaping greatly improved the fine pitch control of the airplane, while retaining nearly full stabilizer authority. With this mechanization, pilots reported that quantization was not apparent at speeds up to approximately 400 KIAS. In the roll axis, stick quantization had to be reduced by changing the linear gearing about the center stick position. The initial value of 1.04° of total aileron command was changed to 0.36°. This reduced the minimum commanded roll rate from 8.32 degrees per second to 2.90 degrees per second at 250 KIAS and yielded acceptable roll control around trim. The only noticeable effect of quantization in the yaw axis was in random 1-bit commands that were observed at 400 KIAS. Lateral acceleration peaks of 0.03g due to 1-bit or 0.38° rudder surface commands were observed. This problem was corrected by writing software in erasable memory to allow a 1-bit deadband in the rudder pedal command. No other rudder pedal quantization effects were seen.

It should be noted that the ±45 quantization steps available represented less than a 6-bit A/D conversion. A 12-bit (11 bits plus sign) A/D capability is available today. This yields a resolution nearly 50 times as fine as that in the F-8 DFBW system. At the most sensitive F-8C flight condition, which was Mach 0.86 at sea level, a 12-bit A/D interface would have allowed digital commands as small as 0.001g, assuming linear gearing and full surface authority. Therefore it is safe to assume that the quantization effects of a modern A/D interface would be negligible and undetectable by the pilot.

Quantization of pilot trim inputs due to sample rate also became apparent in the flight program. In the F-8 DFBW mechanization, trim command discretes were sampled every 90 milliseconds. Based on the pitch trim rate value of 1.25 degrees per second, the minimum software command was 0.11°. This command is nearly twice as coarse as the D/A converter quantization steps of 0.069° for the horizontal stabilizer. This effective trim quantization was a factor in making precise trim of the F-8 DFBW aircraft difficult at a target speed and altitude.

The pitch trim discrete inputs should have been sampled at the major cycle sample period of 30 milliseconds, which would have resulted in a trim quantization of 0.0375°. This would have taken full advantage of the output D/A quantization. This

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points out the need to sample beep trim discrete inputs at a high enough rate to yield acceptable output quantization. In some cases, trim discretes may have to be sampled at rates higher than the major cycle sample rate, if fine trim resolution is required.

Stability Augmentation System Mode

The nominal SAS configurations flown are shown in figure 3(b). Body axis rate was estimated by filtering the transformed inertial attitude from the Apollo inertial platform. Compensation filtering and gain were placed in the feedback path. There was an aileron-to-rudder interconnect in the yaw SAS mode only. The stick and trim processing were identical to those in the direct mode. A rate reasonability check was applied to the final command, and an automatic transfer to the direct mode resulted if the reasonability threshold was exceeded.

The digital SAS modes operated as expected. This is important from the point of view of the sampled-data design process. The acceptance of digital control systems depends in large part on the ability to predict system performance accurately.

The digital SAS loops were designed by using sampled-data analysis methods, especially the z-plane root locus method. The linear system model used in the pitch axis is shown in figure 6. An ideal pitch rate signal was assumed. At first, the rate estimation filter that acted on pitch attitude was used in the model, but the resulting pitch rate signal was found to be nearly identical to that for the ideal case at the F-8C short period frequencies. Neither the highly nonlinear A/D conversion of gimbal angles nor the axis transformation steps were modeled. Four symmetrical bending modes were included in the analysis.

The z-plane root locus for the pitch SAS mode without lead-lag compensation is shown in figure 7(a). A lead-lag filter was designed to improve the performance of the pitch rate loop in increasing the short period damping ratio. A w-plane frequency response was used to select the compensation root locations. The w-plane compensation,

$$G(w) = \frac{w/0.1 + 1}{w^2/0.16 + w/0.286 + 1}$$

was transformed to the z-plane by $w = \frac{z - 1}{z + 1}$ and yielded a discrete filter,

$$G(z) = \frac{1.023(1 + z^{-1})(1 - 0.818z^{-1})}{1.0 - 0.976z^{-1} + 0.349z^{-2}}$$

The root locus for the compensated system is shown in figure 7(b). Higher short period damping ratios were achieved by using the lead-lag filter, as one would expect in a continuous system. A comparison between the predicted effects of the

compensation filter and those measured in flight is shown in figure 8, where the increment in short period damping ratio is shown for three flight conditions. The sampled-data system prediction is good.

The improvement in airplane response with the pitch SAS is evident in the flight time histories in figure 9. Figures 10(a) to 10(c) show a comparison of predicted with measured damping in the three airplane axes. Agreement is good for the longitudinal short period (fig. 10(a)) and Dutch roll (fig. 10(b)) modes. At low gains, rate estimation quantization and actuator friction restricted surface motion at the angular rates tested, and, as a result, the SAS loop was less effective.

The flight performance of the digital roll SAS mode is illustrated in figure 10(c). Since the roll rate response that resulted from a step lateral stick command was contaminated slightly by the Dutch roll, an effective roll mode time constant corresponding to the time between the initial roll rate response and the time when 63 percent of steady state was achieved was used. Yaw SAS was engaged on all runs to reduce the Dutch roll contamination. The predicted trend, which was for decreasing roll mode time constant with increasing roll SAS gain, is clear, although a bias of approximately 0.05 second is apparent. One factor that contributed to this bias was the nonideal pilot step input, which resembled a rapid ramp. This resulted in a slightly higher than predicted effective time constant, since the predicted value was based on a perfect step input.

To further evaluate the sampled-data analysis method, the pitch rate feedback gain was increased in flight until the compensation root approached neutral stability. Figure 11 shows the z-plane root locus prediction of the neutral stability point to be in good agreement with the flight-measured results.

The SAS modes also operated well at low speeds. Pitch SAS results are shown in figure 12(a). A washout filter was designed for low speed operation in the s-plane as

$$G(s) = \frac{s}{s+1}$$

The discrete washout filter formed by using the bilinear transformation for real roots was

$$G(z) = \frac{0.98522(1 - z^{-1})}{1 - 0.9704z^{-1}}$$

The results of the washout filter addition to the feedback loop on aircraft response was as expected (fig. 12(b)). The highest loop gains used in flight were $|K_q M_{\delta_e}| = 3.8$ in pitch, $|K_p L_{\delta_a}| = 3.2$ in roll, and $|K_r N_{\delta_r}| = 1.2$ in yaw. One further observation is appropriate. The Apollo inertial platform was designed for precise navigation. It had an A/D interface, the coupling data unit, that was not designed to facilitate rate estimation. Even so, the derived body rate provided a signal that could be used satisfactorily for the F-8 DFBW damper modes.

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Command Augmentation System Mode

The pitch CAS mode block diagram is shown in figure 3(c). Derived normal acceleration is blended with derived pitch rate to form the feedback signal, C* (ref. 7). A forward loop integrator and bypass path provided zero steady state error and resulted in neutral aircraft speed stability. The cos θ correction term eliminated acceleration feedback in a steady climb or descent. The pilot stick and trim interface with this mode was the same as in the direct and SAS modes.

As was the case in the pitch SAS mode, the performance of the digital CAS mode was essentially as predicted by linear sampled-data systems analysis. However, gain values selected for the C* feedback gain during the preliminary design could not be used in flight. The reasons for this are traceable to the noise problems associated with using rates and accelerations derived from the Apollo inertial measurement unit and interface hardware. These problems are not inherent in a digital mechanization. For acceptable noise levels at the horizontal stabilizer, the C* feedback gain was too low for optimum response. The flight performance of the CAS mode was reasonable at low speeds, however. Figures 13(a) and 13(b) compare the F-8 DFBW C* response in the direct and CAS modes at 180 KIAS and 250 KIAS, respectively. These responses, normalized to the final value, are shown with respect to the C* power approach and cruise design envelopes, respectively. The improvement in airplane response is substantial. The 250-KIAS response illustrates the problem encountered in CAS with insufficient loop gain. The short period response was satisfactory, but the aircraft exhibited drift in the 3- to 8-second time period that was actually the first-order mode resulting from the forward loop integrator. This effect was apparent to the pilots.

The CAS mode provided the expected neutral speed stability. Figures 14(a) and 14(b) show the phugoid response of the F-8 DFBW aircraft in the direct and CAS modes, respectively. The aircraft, trimmed at 180 KIAS, was slowed approximately 10 KIAS, where the stick was again centered. The CAS mode held zero pitch rate while the aircraft slowed to a new steady state speed of approximately 138 KIAS. Normal acceleration (not shown) remained constant at nearly 1g during the maneuver, while angle of attack (not shown), which started at 3.5°, stabilized at 10°.

The effectiveness of the CAS mode in suppressing transient effects is shown in figure 15, where the response of the F-8C airplane is compared in the direct and CAS modes during a wing transition (wing incidence changes from -1° to 7°). Both responses were without pilot inputs.

Although the performance of the CAS mode was degraded by the limitations of the Apollo hardware, the control system design was relatively straightforward, and flight results again matched predictions quite closely.

Implications of Digital Fly-By-Wire Design for Active Control Systems

The flight verification of the F-8 DFBW control system design was encouraging from an active control technology standpoint. First, the body of continuous control system design experience is largely applicable. In fact, if there is a

reasonable separation between the half sample frequency and modes of interest, the design can be accomplished in the continuous domain and then exactly transformed to the discrete domain by using the bilinear transform. Furthermore, direct z-plane design is also possible. The most serious difficulty about using the latter approach is lack of experience with direct digital design.

The entire F-8 DFBW three-axis digital flight control system problem could be solved by the Apollo computer in less than a 30-millisecond major cycle time period. The capabilities of a current high performance computer and those of the Apollo computer are:

	Apollo computer	Current computer
Memory cycle time, μsec	11.7	1.0
Add time, μsec	23.4	2.5
Multiply time, μsec	46.8	6.0

The table shows that a state-of-the-art computer can be expected to be an order of magnitude faster than the Apollo computer. This suggests a sample rate or job capacity increase of the same magnitude. Although computer sizing must await a specific ACT configuration, the capability of today's computers would appear to be more than adequate for the control system tasks envisioned.

PILOTING FACTORS

Considered in conjunction with the control system performance reported in the previous section, the handling qualities results confirmed the feasibility and utility of a digital fly-by-wire control system.

Handling Qualities Summary

The flying qualities of the F-8 DFBW were evaluated by the pilots in a variety of tasks, including simulated instrument cruise, large or abrupt maneuvers, ground-controlled approaches, gunsight tracking, and close formation flight (ref. 8).

Figure 16(a) summarizes the longitudinal handling qualities results for small instrument maneuvers, and figure 16(b) summarizes the results for large maneuvers. The piloting tasks and the comment guide used for these evaluations are given in the appendix. In figure 16(a) the comments and ratings are typical of the findings of pilots at low-to-moderate cruise speeds (less than 350 KIAS). For large maneuvers the pilot rating improvement with control system sophistication was evident. Pilot acceptance of the SAS and CAS modes was expected on the basis of the control system and vehicle response characteristics reported in the previous section. Some pilots did report a long period overshooting tendency in the CAS mode for certain maneuvers where steady state pitch rates had to be arrested. This correlated with the first-order integrator mode present in the CAS step response.

Figure 17 is characteristic of the improvement in pitch control with digital SAS as seen by the pilots in a wind-up turn. In the direct mode, the F-8C airplane displays its undesirable short period damping. The same maneuver could be performed easily and precisely in the pitch SAS mode.

Ground-controlled approaches were flown down to approximately 60 meters under simulated instrument flight conditions in the various digital modes. Figures 18(a) and 18(b) show typical pilot comments and ratings in the lateral-directional and longitudinal axes. The pilot ratings reflect the improvement in Dutch roll damping provided by the yaw SAS mode. In figure 18(b) pilot A objected to a slight long-term overshooting tendency in the CAS mode.

The tracking performance of the F-8C airplane with the digital control system was degraded by stick quantization problems in both the pitch and roll axes. The parabolic pitch stick shaping resulted in unacceptable quantization steps at large aft stick positions (table 2). This degraded the pitch control of the airplane so much that even augmentation did not significantly improve the tracking performance. Some improvement with roll and yaw SAS was evident in a 2g gunsight tracking maneuver, as the time histories in figure 19 and the associated pilot comments and ratings in figure 20 show. The augmented time histories in figure 19 correspond to a yaw SAS gain, K_r, of 0.4 deg/deg/sec.

Close formation flight revealed deficiencies in the flying qualities that were often not apparent in maneuvers where the pilot was not required to be "in the loop" as tightly. The improvement shown in figure 21 of the longitudinal flying qualities with digital augmentation is typical. Pilot comments reflected the decreased workload evident in the time history. Barrel rolls, aileron rolls, and wingovers were performed in all control modes. Pilots noted little difference in their ability to perform these maneuvers between the direct and augmented modes, perhaps because these maneuvers tended to be more open loop in nature.

Except in maneuvers where the coarse stick quantization problem was overriding, as in the gunsight tracking maneuver, the DFBW control system markedly improved the flying qualities of the unaugmented F-8C aircraft. Because of the control system performance described in the previous section, this was not unexpected. One pilot who flew F-8C airplanes regularly found the F-8 DFBW vehicle superior even to a standard F-8C airplane with normal augmentation. He noted in particular the lack of the usual mechanical control slop.

The results of the flying qualities evaluations, coupled with the control system performance previously described, indicate that a DFBW control system can perform as well as or better than a conventional control system. The only serious problems encountered were due to the limitations of the Apollo system hardware, which would not be factors in a current design.

Pilot Interface With the Digital-Fly-by-Wire System

The F-8 DFBW system was designed to permit a simple, yet flexible, interface with the pilot. The normal astronaut interface with the Apollo guidance and navigation system was a display and keyboard device (DSKY) that allowed the operator to display memory contents, load erasable memory, or initiate special programs. The versatility of this interface was important to the design and test engineers during the development and flight test program, but it was not made available to the pilot because of its complexity for a single place aircraft. The pilot's only interface with the digital computer was through a mode and gain panel, which is described in reference 3. The pilot's gain switch mechanization in software contributed to the rapid, safe flight checkout of the digital flight control system. Table 3 lists the different digital control system parameters that were tied to the gain switches during the flight test program. In all, 105 parameters could be connected via software to the three gain switches.

With this gain mechanization, different control system parameters could rapidly be selected and optimized during the research program. More important, the gain switches allowed the designer to make use of the pilot's capabilities. Nominal values of critical gains that were established during the simulation phase were placed on the gain switches along with larger and smaller values. The pilot could change the gain values at any time. For example, one of the gain switches was for pitch gearing. During the first flight, when the effects of the pitch quantization and sensitivity had not yet been established, the pilot took off in the nominal gain position. By 13 minutes after takeoff at 300 KIAS, he had reduced the gearing 10 percent because of pitch control sensitivity. Before landing he evaluated three gain positions, finally selecting the nominal gain value 2 1/2 minutes before touchdown. Apart from its research value, this type of gain selection and evaluation gave the pilot an important degree of freedom. Switch arrangements like this are not unique to digital flight control systems, but the ability to designate such a large number (105) of parameters for this use with virtually no hardware impact is unique to a digital system.

This kind of flexibility can be carried in a digital computer with only a small increase in software complexity. This mechanization approach would also be advantageous in an active control system design, because the F-8 DFBW experience showed that the pilot could rapidly and safely assess open- and closed-loop gain parameter variations about the nominal design point during flight.

Flight experience also showed the multimode digital flight control system to be safe and valuable for both research and proof testing phases of the flight program. The low mode of control in the primary digital system (direct) provided a fallback position for both the pilot and the system. Since the direct and augmented modes were fully synchronized, they could be switched manually or automatically under any dynamic conditions with a minimum and safe aircraft response transient. The pilots took advantage of this multimode mechanization to diagnose the cause of flying qualities deficiencies by comparing airplane response in each mode.

Like the gain switch arrangement, the multimode mechanization makes use of the online monitoring capabilities of the pilot. It too is a good candidate for active

control mechanization, especially for the first few flights. One problem was encountered with this approach. Mode changes could occur without being commanded by the pilot due, for example, to a reasonability test. The mode panel display light configuration would change, but this was not easily detected by the pilot. A master caution and annunciator warning of any uncommanded mode change should have been incorporated.

In summary, software flexibility allowed the test pilot to use his real-time diagnostic capability and to make control system alterations. The alterations could be made with almost no hardware impact and with minimum additional software complexity. These concepts are applicable to early flight testing of full time active control systems.

MANAGEMENT OF FLIGHT SOFTWARE

The flexibility and versatility of digital flight control system software carries with it the need for software management and control. Perhaps no other area of digital fly-by-wire control raises as many questions and doubts as software reliability. The concern centers on whether it is possible to achieve reliable man-rated flight control software at a reasonable cost and whether software flexibility is compatible with software reliability in a practical application. The F-8 DFBW experience indicates that both questions can be answered yes.

Two aspects of the F-8 DFBW flight test program are of significance to full authority, man-rated digital flight control software. First, not a single software programing error was discovered during the flight test program. Much of the credit for this is due to the thorough verification procedures and facilities developed for the Apollo software, which were also used during the F-8 DFBW program, although on a smaller scale. The procedures are described in detail in reference 4. Secondly, not a single incorrect erasable memory constant propagated to a flight tape that was used to load the Apollo computer. These results are significant because an active control system must achieve the same level of reliability as the basic airframe. The software, in turn, is central to the active control system's reliability, because even though an active control system would have redundant digital channels, the software would be common to all, as it was in the F-8 DFBW system. For this reason, it is worthwhile to examine the software management procedures used in the F-8 DFBW program.

Figure 22(a) outlines the procedures established to control software programing changes during the flight program. These procedures were used three times after the hardwired memory was manufactured and before the first flight. The three special purpose programs written into the erasable memory consisted of pitch and roll parabolic stick shaping, yaw pedal deadband, and a special failure mode monitor.

The software control board in figure 22(a) consisted of representatives from control system engineering, project management, operations, and the pilots' office. Step 7 in figure 22(a) consisted not only of checking out the new code but rerunning former, documented tests on related code to insure proper program

interaction, if any. Extensive files of detailed all-digital simulation runs generated during the initial verification phase were kept for comparison with identical runs with the modified code. This permitted short turnaround time for new additions to the code.

Figure 22(b) shows the steps taken in the alteration of control system constants in the erasable memory. In total, 394 erasable memory locations had to be loaded for each flight. Table 4 gives a breakdown of these constants. Sum checks and built-in data transmission checks in the Apollo computer made it possible to insure that the desired octal numbers were loaded into the computer.

Making sure that the 168 control system values loaded were those actually desired was less straightforward. A punched tape was used to load the computer. During the flight program six tapes were manufactured, each of which represented a different flight control system configuration.

Because the Apollo digital computer is a fixed-point machine, there were magnitude restrictions due to program scaling on most parameters. A variety of other restrictions combined to create a formidable set of rules for the set of control system constants.

An off-line diagnostic digital program (step 3 of fig. 22(b)), which ran on a data processing computer, was developed to ease the burden of verifying the correct content of the master load list, which was kept on standard punch cards. One task performed by the diagnostic program was to check each of the 394 constants against a previously drawn list of reasonable values. This reasonability list was constructed after considerable experience was gained from iron bird simulation, but before the first flight tape was made. The limits were set to encompass the expected or allowable operating range of each variable. Deviations from reasonability limits were flagged by the program as major errors and had to be corrected or signed off by the responsible engineer.

The program also reconstructed digital filter forms from their coefficients and computed their vital characteristics, such as root location, steady state gain, and absolute root magnitude in the z-plane. This was helpful in the case of digital filters, the characteristics of which are not as obvious as those of continuous filters.

One aspect of software control became apparent during the ground testing and simulation of various control system gain configurations. When many gain changes had been made and the precise configuration was in doubt, it was only necessary to dump the contents of erasable memory on magnetic tape to create a complete description of any given configuration. This capability proved to be extremely valuable in the control system refinement stage, and it is unique to a digital mechanization. It was also possible to revert to the baseline configuration merely by reloading memory with the baseline punched tape. This required approximately 3 minutes on the Apollo computer.

In summary, the F-8 DFBW flight experience indicates that highly reliable flight software can be generated and maintained, but that it requires thorough control.

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Because the F-8 DFBW program was intended for research, the software program was made more flexible than would be necessary for a production airplane. Even with this flexibility, the software was easily managed with diagnostic digital programs, resulting in high overall system reliability. In fact, changes were made to the digital system more confidently than they were to the airplane's analog systems because there was no hardware impact.

Partly because of the built-in flexibility of the control system mechanization, only minor changes had to be made to the basic program during the flight test program. More program changes would be expected in a prototype system development, thus increasing the need for strict configuration management for software.

The F-8 DFBW flight results confirmed that a DFBW control system could be used in an active control application from the standpoint of software reliability and system flexibility.

OPERATIONAL FACTORS

Reasonability Checks

The software reasonability checks used in all augmented modes are surface command rate checks made over one sample period (30 milliseconds). Exceeding the threshold value in any axis resulted in an automatic downmode to the direct mode in that axis. The threshold values per sample period that were found to be usable in flight were 4.5° in pitch, 13.0° in roll, and 8.0° in yaw. These were the smallest values that allowed nearly any pilot input. Ten downmodes occurred in flight. All except four were directly related to sharp pilot step inputs that were made for test purposes. The other four were due to noise peaks that resulted from the angular rate derivation. At least one of these occurred in each axis.

The reasonability check was designed to detect abrupt command changes due to sensor failures or major software faults. Experience with the F-8 DFBW system indicated that the threshold rate limit could be reduced by at least 50 percent in all axes for an operational fighter. If unreasonable commands were allowed to exist for 100 milliseconds (approximately three sample periods), nuisance down-modes would be eliminated without sacrificing protection.

Digital System Reliability

The F-8 DFBW digital control system utilized a single highly reliable digital computer. This configuration would probably not be used in an active control system. However, the reliability requirements of the F-8 digital system are representative of the requirements of an ACT application. First, no single failure was permitted that would have resulted in the generation of a hazardous control surface command. Second, any serious failure within the digital system had to be detected. In the F-8 DFBW airplane, the failure warning signals were used to transfer control to the analog backup control system. In a redundant digital control system,

operation would continue on the remaining good digital channels after a component failure. F-8 DFBW reliability experience is nevertheless applicable to active control technology in terms of failure detection and also in terms of the features of the digital mechanization that led to a high level of confidence in this system.

No hardware failures occurred in the primary digital flight control system on any flight. This is not surprising in view of the demonstrated in-service reliability of the Apollo guidance and navigation equipment. The discrepancies noted in the DFBW flight system, excluding the actuators and their drive electronics, are listed in table 5. Three component failures occurred in two systems during the 2500 hours of operation (items 4, 5, and 10). Item 4 would have had no impact on normal flight operation. The failure monitor in item 5 was added to the system during the flight program to protect against a potentially hazardous single-point failure mode in the Apollo computer output interface hardware. The monitor box failed before its first use in flight, although it failed in the proper "safe" mode (transfer to the backup control system). The roll stick circuit failure (item 10) would have caused a downmode to the backup control system in flight, as it did on the ground. There were no unresolved anomalies.

Preflight Procedures

Two preflight test procedures were used for the digital system. The first was a 1-hour test done on the system in the hangar the day before flight. Electrical and hydraulic power were external. The second procedure was part of the total aircraft preflight immediately before flight, with engine-supplied electrical and hydraulic power. The elements of the hangar and flight line preflight tests are listed in table 6. Virtually all the hangar tests except the specialized inertial measurement unit checks and the detailed surface deflection measurements were repeated. Although the digital system's flight line preflight was not optimized in the built-in software, it took only 10 to 15 minutes.

One sensitive preflight test was the computer activity check. A program in the erasable memory was used to measure computer duty cycle indirectly, by detecting idle time over a several second interval. In a given configuration, the duty cycle was consistent within a few percent over several time intervals. This test confirmed proper software operation to a high level of confidence.

During the investigation of the anomalies that occurred on both the iron bird and the F-8 DFBW airplane, it became apparent that it was possible to determine the health of the digital control system rapidly and confidently. The state of the digital control system could be determined in less than 5 minutes by running a self-test and by monitoring the internal control system parameters on the DSKY in the flight control modes. The monitor feature was indispensable during the flight test program. With half a dozen keystrokes, three control system parameters could be displayed in engineering units and in decimal format. The display was updated every second, so even under dynamic conditions the display was intelligible. This monitor format permitted the immediate checkout of virtually any part of the control system. Any future digital flight control system should incorporate such display software capability.

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The ability to quickly and confidently assure proper control system performance is of paramount importance to active control systems. The repeatability of the test results of the F-8 DFBW program inspired enormous confidence in the operational readiness of the system before flight. Even personnel not thoroughly familiar with the digital control system were able to perform detailed tests of the system because of the well-designed display and monitor software. The component failures that did occur during ground operation were all detected by the system itself.

CONCLUDING REMARKS

The F-8 digital fly-by-wire (DFBW) flight program showed the feasibility and advantages of DFBW control for aircraft. Even with hardware designed a decade ago for space applications, an Apollo computer easily handled the F-8 DFBW flight control computation task. This demonstrated the inherent flexibility of a digital system.

The following conclusions can be drawn on the basis of the F-8 DFBW flight test program.

(1) Existing design tools, such as the w-plane frequency response and the z-plane root locus, are suitable for the synthesis of digital flight control.

(2) Flight performance of the digital flight control system verified the accuracy of the sampled-data design results for contemporary command and stability augmentation system modes.

(3) Pilot opinion correlated with that expected on the basis of the control system performance.

(4) A modern digital control system design would display no quantization effects noticeable to the pilot.

(5) The flexibility of the digital control system permits effective use of the pilot in configuration optimization in early flight test stages.

(6) Man-rated software can be safely managed while retaining a high degree of flexibility. The use of off-line diagnostic programs greatly reduced the engineering burden of software management.

(7) Digital system integrity can be rapidly and confidently determined in pre-flight tests by using flexible and extensive engineering interfaces.

The implications of these results for an active control application can be broadly stated as follows:

(1) A DFBW control system possesses the computational ability and flexibility necessary for advanced active control applications. Computer hardware advances are leading control system applications.

(2) Reliable software can be produced and is not an obstacle to an active control application.

(3) The fault detection and preflight test technology necessary for digital control systems exists. Full realization of DFBW potential awaits the successful demonstration of reconfiguration and normal operation after component failures in a practical redundant system.

There was no flight or ground experience that would indicate that a DFBW system could not be used in an active control technology application. In fact, the F-8 DFBW flight program achieved in practice the advantages so long attributed to a DFBW control system and confirmed the suitability of digital control for active control technology.

APPENDIX

PILOT COMMENT GUIDE FOR LONGITUDINAL HANDLING QUALITIES

Instrument Flight Maneuvering

(1) Trim the aircraft to desired speed at a zero rate of climb
(2) Make small heading changes of less than 30°
(3) Make air traffic control altitude changes
(4) Make air traffic control speed changes

Comment on:

(1) The ability to fine trim the aircraft
(2) The need to monitor the pitch axis during lateral-directional tasks
(3) The ability to make accurate changes in attitude
(4) Stick breakout and deadband forces
(5) The acceptability of these aircraft characteristics for fighter aircraft
(6) Overall longitudinal pilot rating

Large or Abrupt Maneuvers

(1) From trimmed flight, quickly establish a 1.5g to 2.5g turn
(2) Recover to trimmed, level flight
(3) Quickly set up a constant speed high performance climb by selecting a target pitch attitude and throttle setting
(4) Recover to trimmed, level flight at target altitude
(5) Increase speed 50 KIAS, and retrim

Comment on:

(1) The ability to control attitude and g. Tendencies to overshoot or for pilot-induced oscillations
(2) The ability to restore the aircraft to trimmed flight
(3) Stick breakout and deadband forces
(4) The lag in aircraft response to stick inputs
(5) Residual small-amplitude oscillations
(6) The acceptability of these characteristics for fighter aircraft
(7) Overall pilot rating for the large or abrupt maneuvers

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TABLE 1.—PITCH STICK QUANTIZATION EFFECTS WITH LINEAR GEARING

V, KIAS	Altitude, m	Dynamic pressure, N/m^2	Stabilizer quantization, deg	Response to quantization		Pilot comments
				n_Z peak, g	q peak, deg/sec	
300	5,500	13,885	0.53	0.2	1.5	Stepping noticeable, but easy to make small corrections.
365	6,100	19,870	0.5	0.3	2.0	Feel thumping in pitch control. Can see quantization.
400	6,100	22,980	0.3	0.15	1.2	Cannot see quantization.
[a]180	3,050	5,030	0.53	0.05	0.5	Cannot see any quantization. Smooth control.

[a]Wing up.

TABLE 2.—QUANTIZATION MAGNITUDE FOR FULL AUTHORITY PARABOLIC PITCH STICK SHAPING

Nominal stick position, cm	Quantization size, deg
0	0.1
5	0.3
10	0.7
15	1.2

TABLE 3.—DIGITAL CONTROL SYSTEM PARAMETERS TIED TO GAIN SWITCHES

Axis	Mode	Description
Pitch	Direct	Stick gearing
Pitch	SAS	Pitch rate feedback gain
Pitch	SAS	Type of digital filter
Pitch	CAS	Forward loop integrator gain
Pitch	CAS	C* feedback gain
Pitch	CAS	Pitch rate blending gain
Roll	Direct	Stick gearing — wing down
Roll	Direct	Stick gearing — wing up
Roll	SAS	Stick gearing
Roll	SAS	Nonlinear stick shaping
Roll	SAS	Roll rate feedback gain
Yaw	SAS	Yaw rate feedback gain
Yaw	SAS	Interconnect function slope
Yaw	SAS	Interconnect function intercept

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TABLE 4.—ERASABLE MEMORY CONSTANTS LOADED
FOR EACH F-8 DFBW FLIGHT

Description	Number
Control system constants	168
Computer downlink identity tags	100
Inertial subsystem	29
Erasable memory program (parabolic stick shaping)	87
Miscellaneous	10
	Total: 394

TABLE 5.—DIGITAL SYSTEM DISCREPANCIES DURING GROUND OPERATION

(a) Discrepancies.

Item	Discrepancy	Reason for discrepancy
1	Computer restarts	Procedural error
2	Computer time-of-day wrong	Procedural error
3	Inertial measurement unit test result out of specification	Inertial measurement unit degradation for navigation
[a]4	Yaw direct light cycling on-off	Failed transistor in mode panel
[a]5	Backup control system down-mode for rudder inputs	Failure in relay in external fail monitor
6	Computer locked in loop	Procedural error
7	Failure of preflight test	Damage to punched tape
8	Aileron offset	Procedural error
9	Roll D/A drift during backup control system self-test	Truncation during repeated primary/backup control system moding
[a]10	Backup control system down-mode for aileron inputs	Failed resistor in external stick electronics

[a]Primary electronics failures.

(b) Summary.

Component	Failures
Apollo hardware	0
Primary electronics	3

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TABLE 6.—ELEMENTS OF F-8 DFBW PREFLIGHT TESTS

Element	Hangar	Flight line
Verify correct memory load	Yes	Yes
Computer self-test	Yes	Yes
Inertial measurement unit fail discretes	Yes	No
Inertial measurement unit turn-on sequence	Yes	Yes
Proper aline	Yes	Yes
Pilot gimbal angle indicator	Yes	Yes
Inertial measurement unit operational test (12 minutes)	Yes	No
Primary/backup control system moding	Yes	Yes
Gain switch discretes	Yes	Yes
Wing position discrete	Yes	Yes
Forced computer restart	Yes	Yes
Inertial measurement unit interface zero and reset	Yes	No
Forced computer fail discrete	Yes	Yes
Mode panel warning lights	Yes	Yes
Differential D/A output — backup control system downmode	Yes	No
Trim rate and trim fail detection	Yes	Yes
Stick-to-surface gearing measurements	Yes	No
Computer activity	Yes	Yes
Check failure monitor box	Yes	Yes
Maximum surface deflections	Yes	Yes
Load time-of-day	No	Yes
Load computer for flight	No	Yes
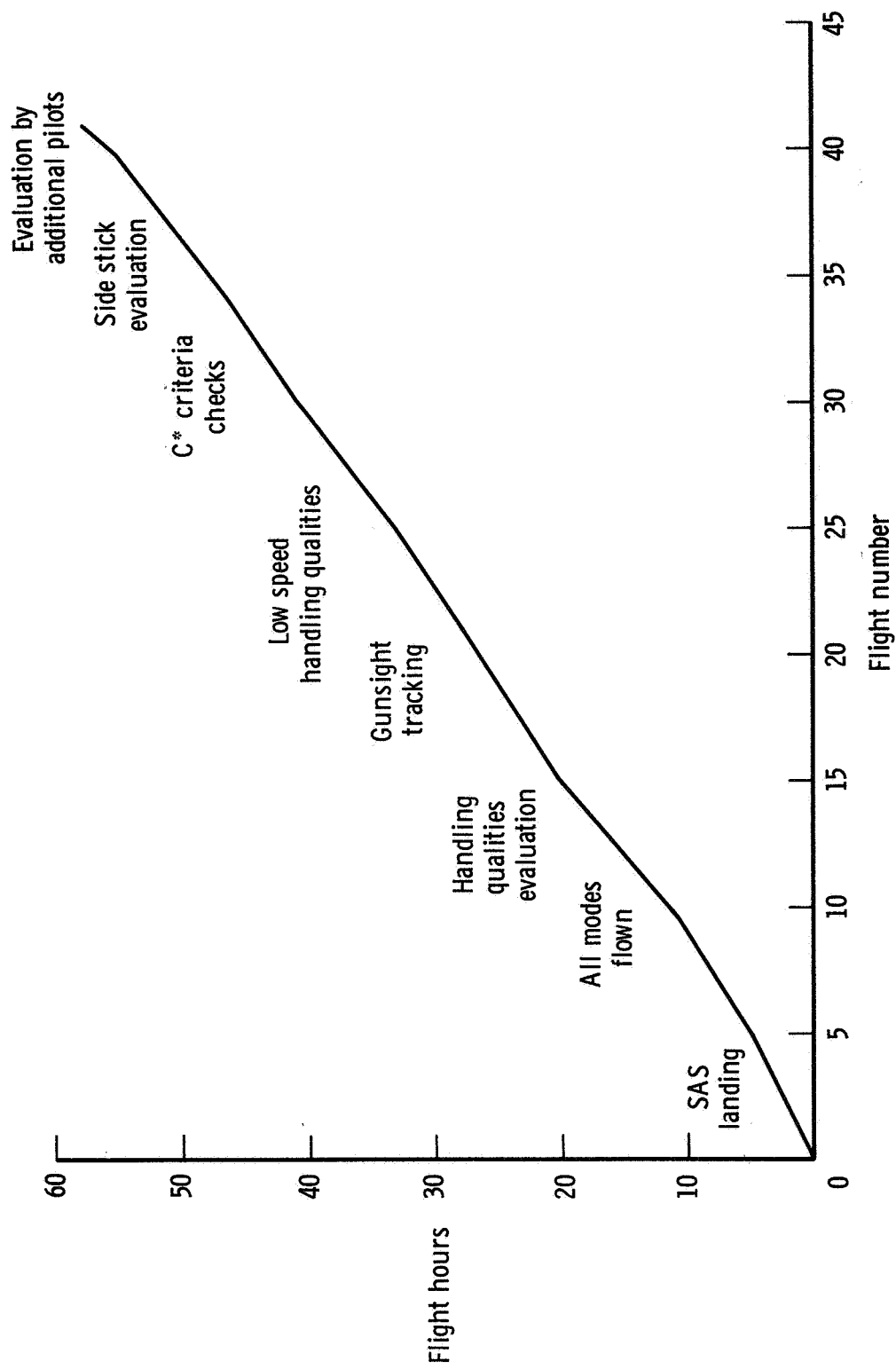
Figure 1. F-8 DFBW flight test summary.
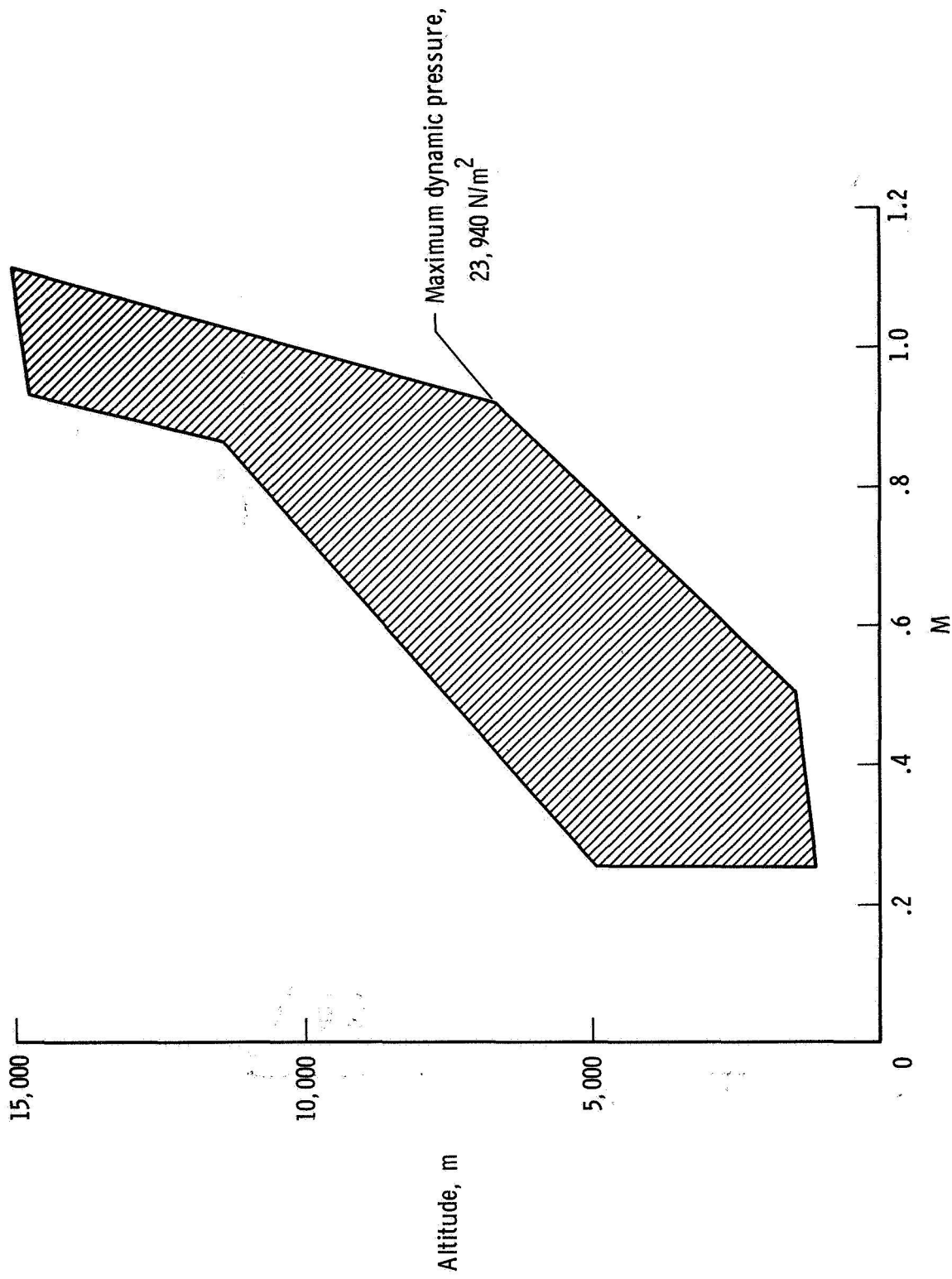
Figure 2. F-8 DFBW flight test envelope.

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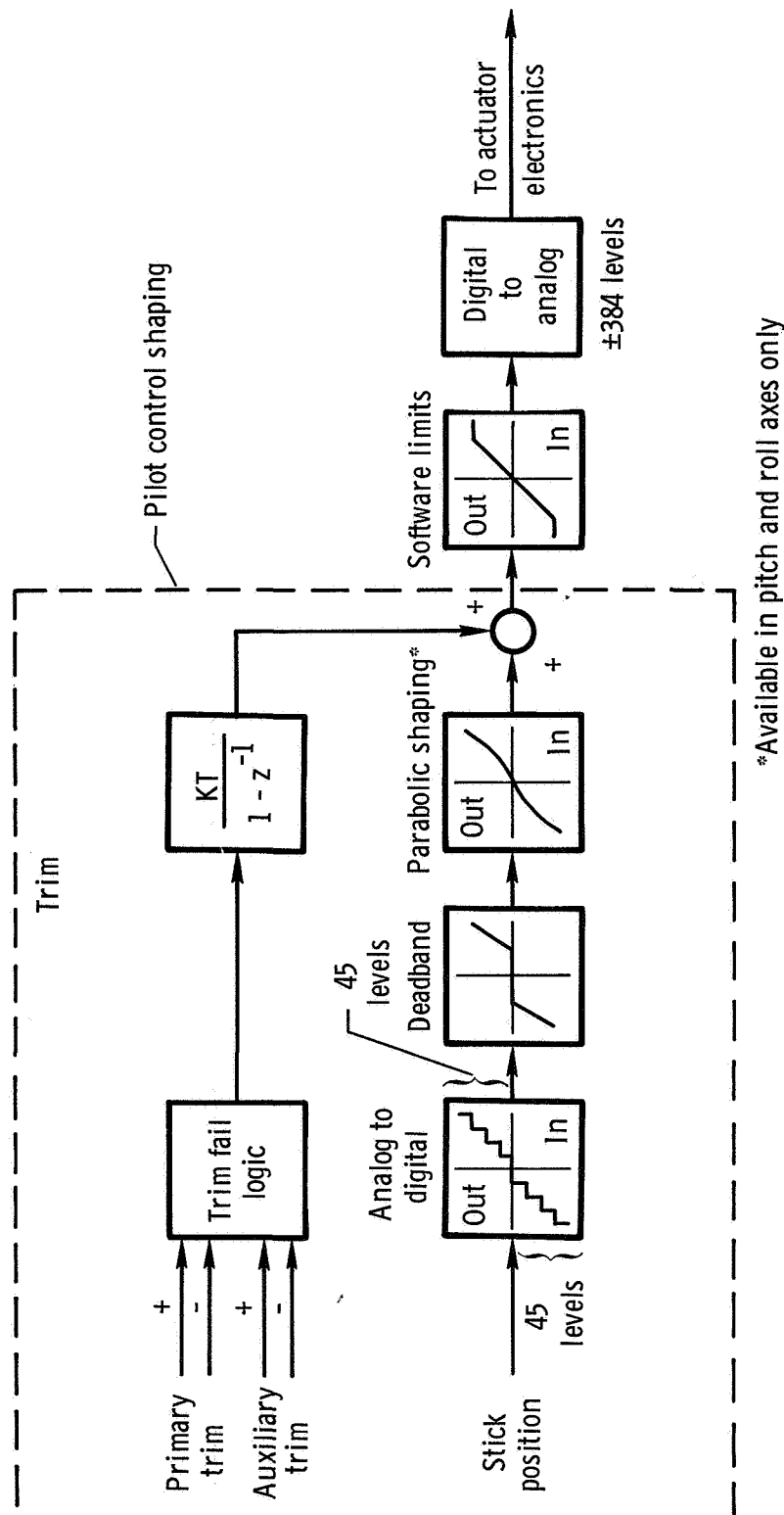
Figure 3. Digital flight control law diagram.

(a) Direct modes.

*Available in pitch and roll axes only

Yaw SAS: aileron-to-rudder interconnect

Software limits

Reasonability check

Transfer to direct if $|\delta_n - \delta_{n-1}| > K$

Digital to analog

To actuator electronics

Pilot control shaping, trim (fig. 3(a))

Feedback gains

K_p, K_r, K_q

General filter

$$\frac{a_1 + a_2 z^{-1} + a_3 z^{-2}}{b_1 + b_2 z^{-1} + b_3 z^{-2}}$$

Pitch: lead lag
Yaw: washout

$\hat{p}, \hat{r}, \hat{q}$
Derived body angular rates

Rate derivation

$$\frac{\frac{1}{2T}(3 - 4z^{-1} + z^{-2})}{z^2}$$

φ, ψ, θ
Body axis attitude

Inertial to body axis

Nonlinear analog to digital

Inertial measurement unit gimbal angles

(b) SAS modes.

Figure 3. Continued.

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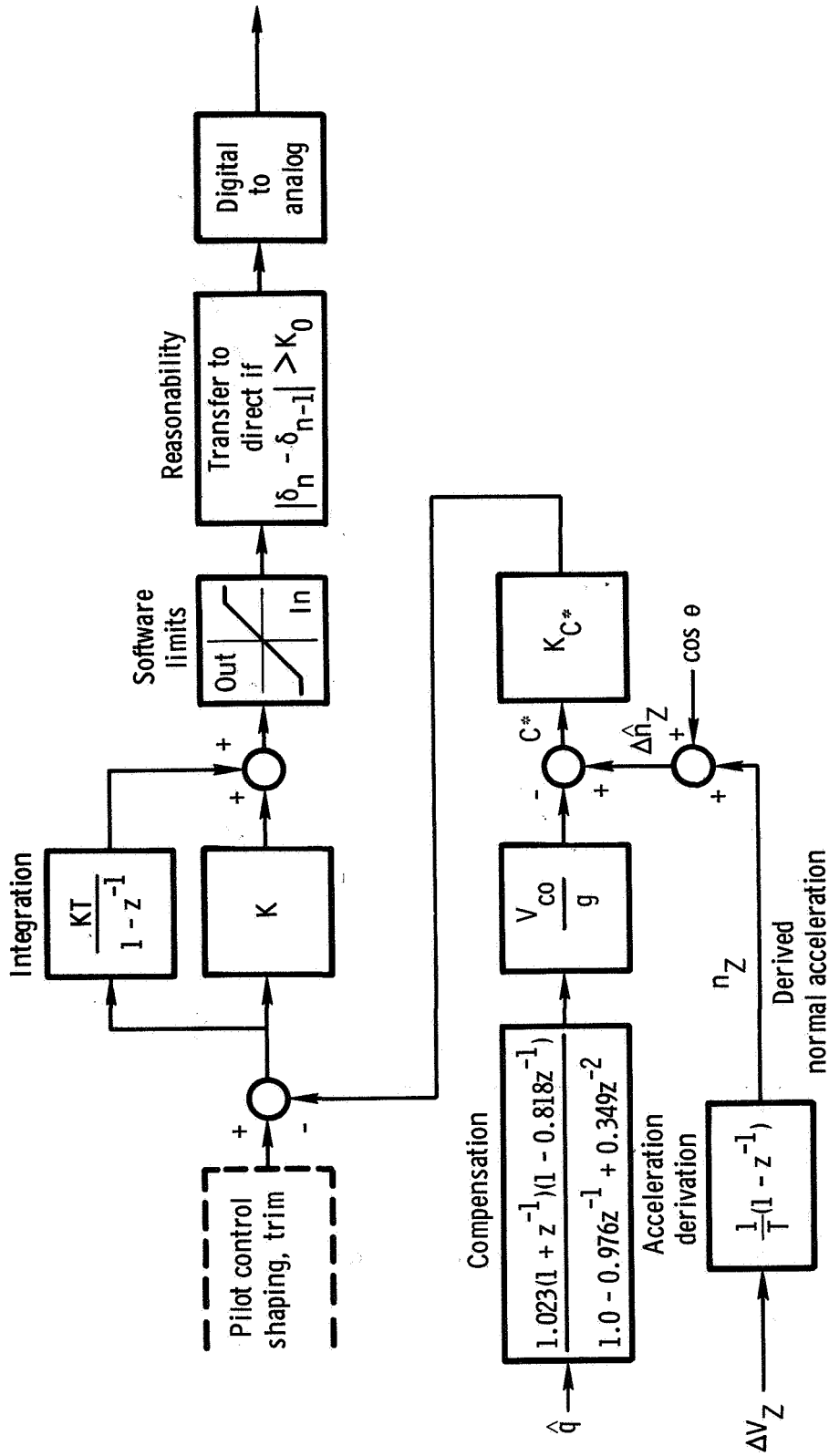
(c) Pitch CAS mode.

Figure 3. Concluded.

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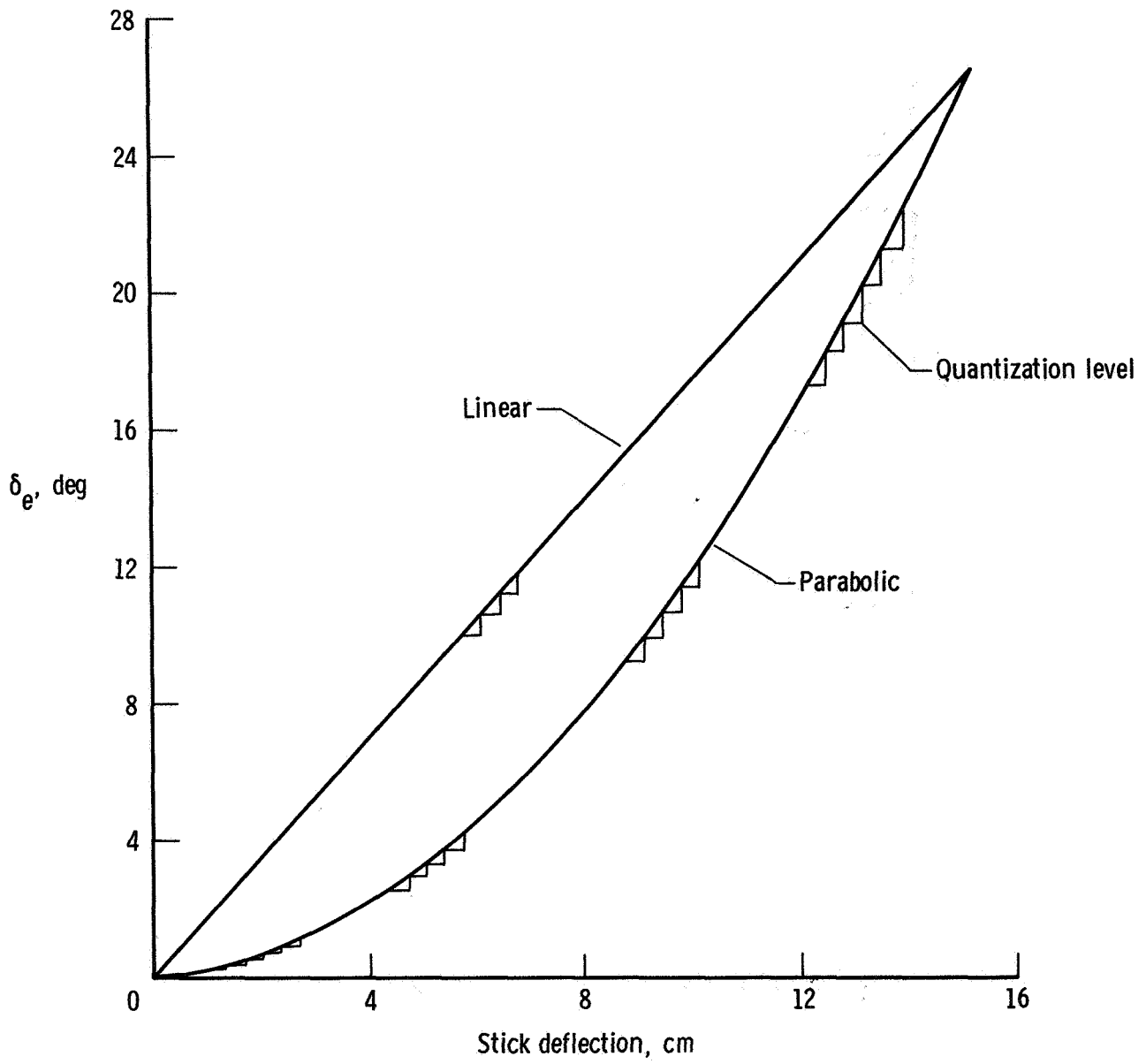
Figure 4. Pitch gearing comparison.
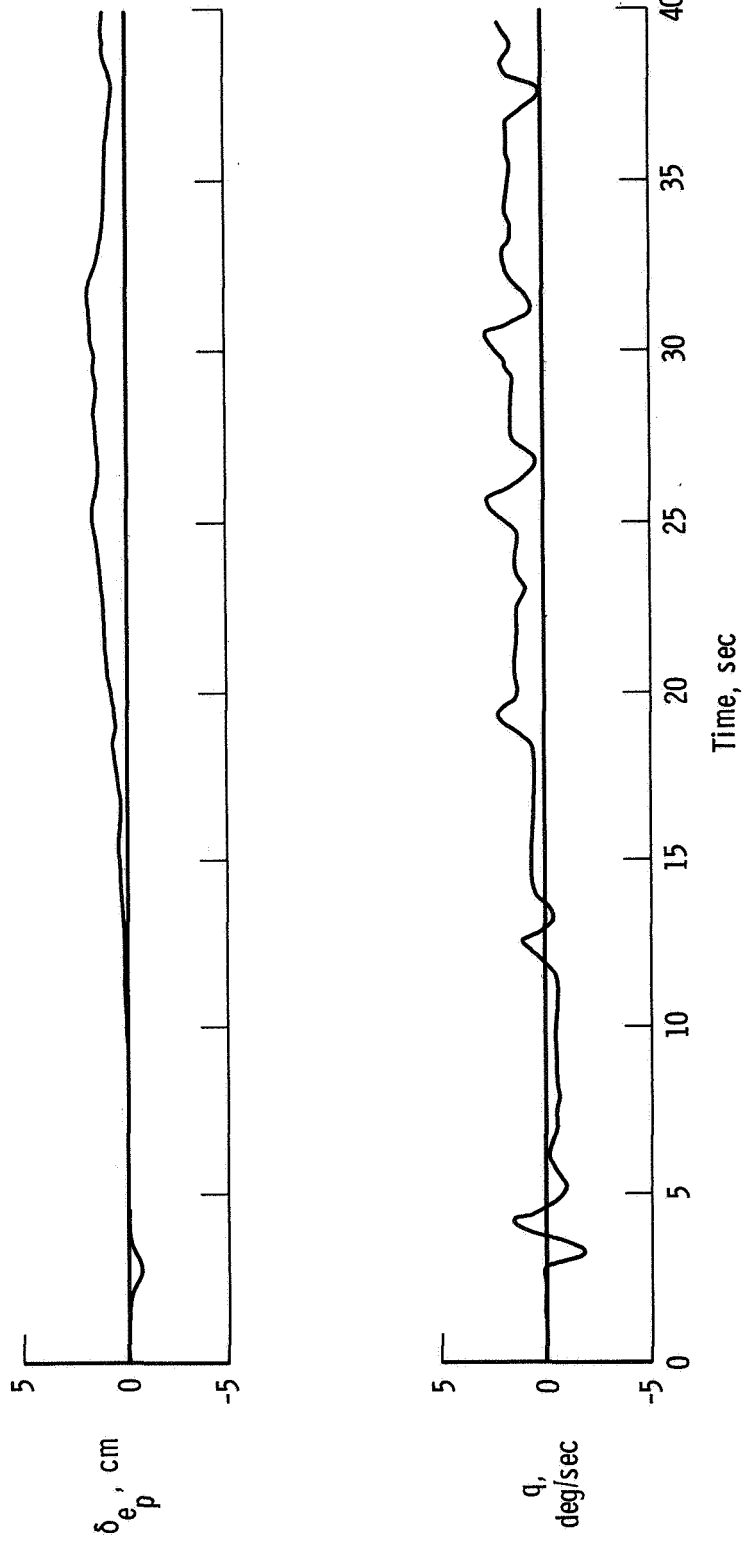
Figure 5. Pitch quantization with linear gearing. 365 KIAS; 6100 m.

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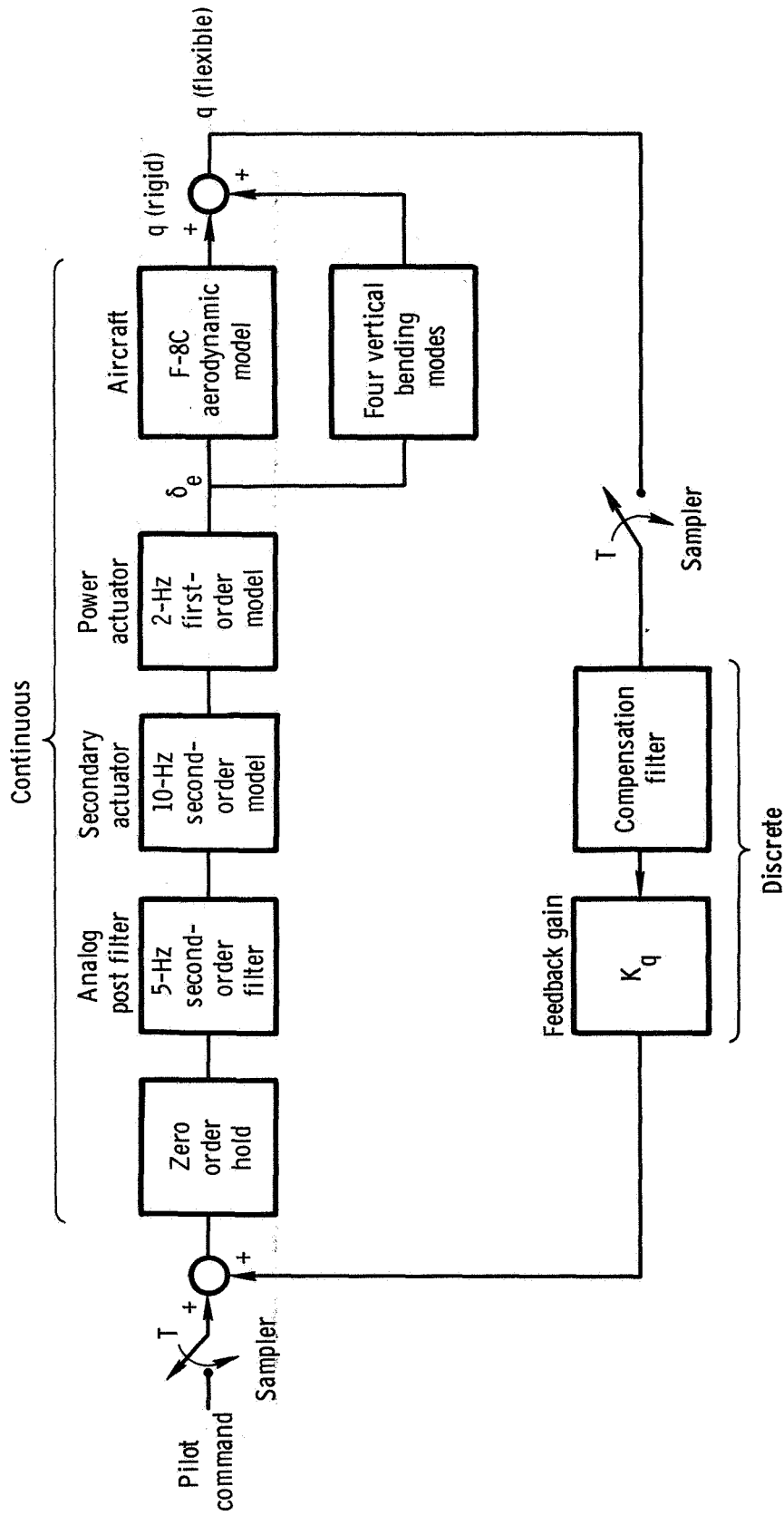
Figure 6. Linear system model for pitch SAS analysis.
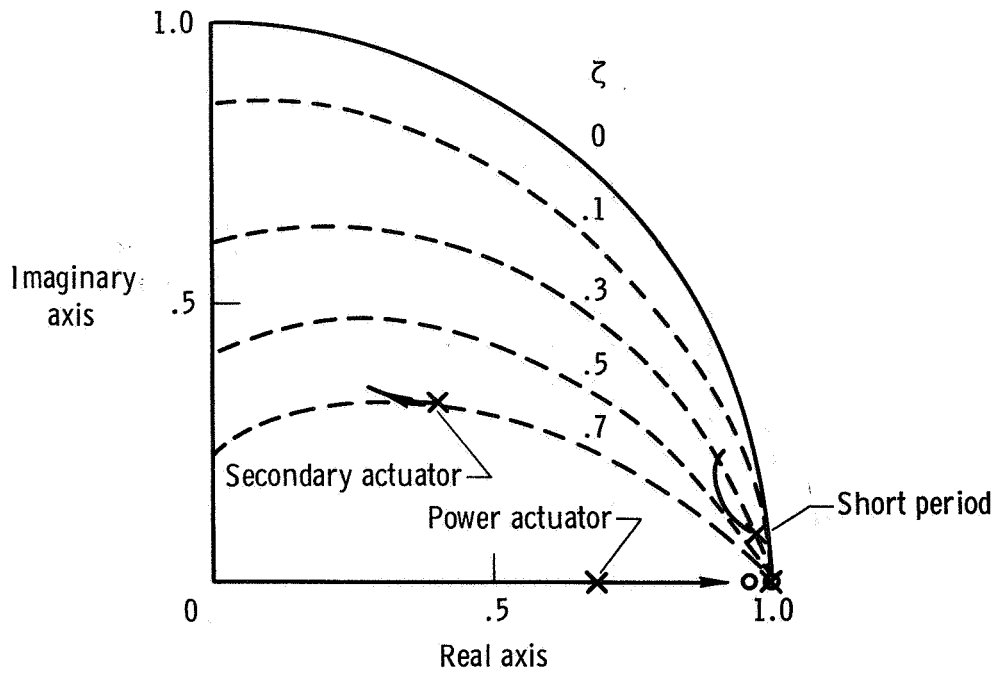
(a) Uncompensated system locus.
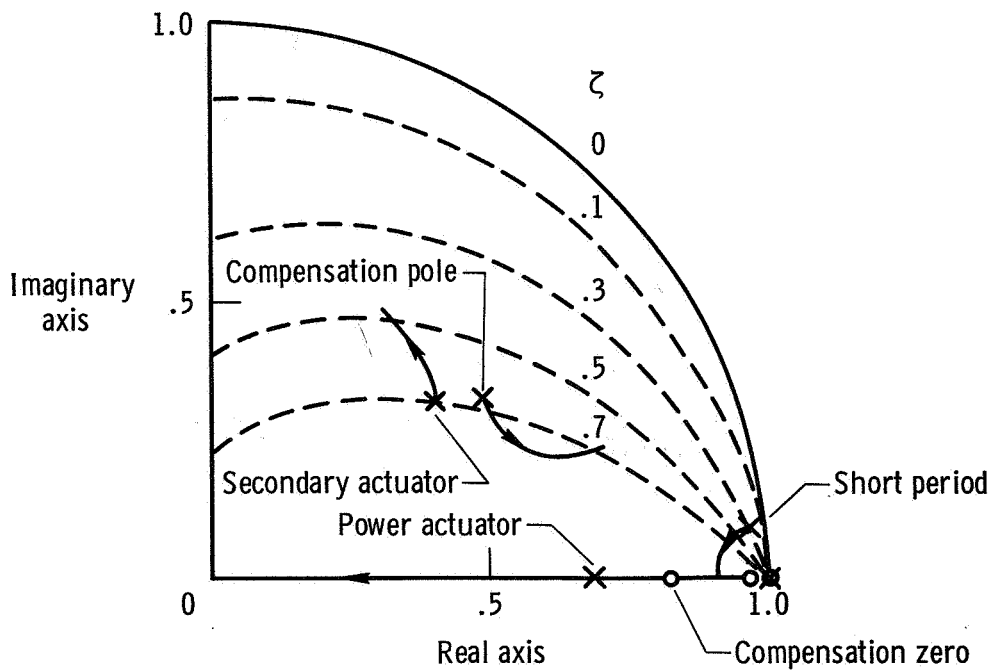
(b) Locus with lead-lag compensation.

Figure 7. Sampled-data system design.
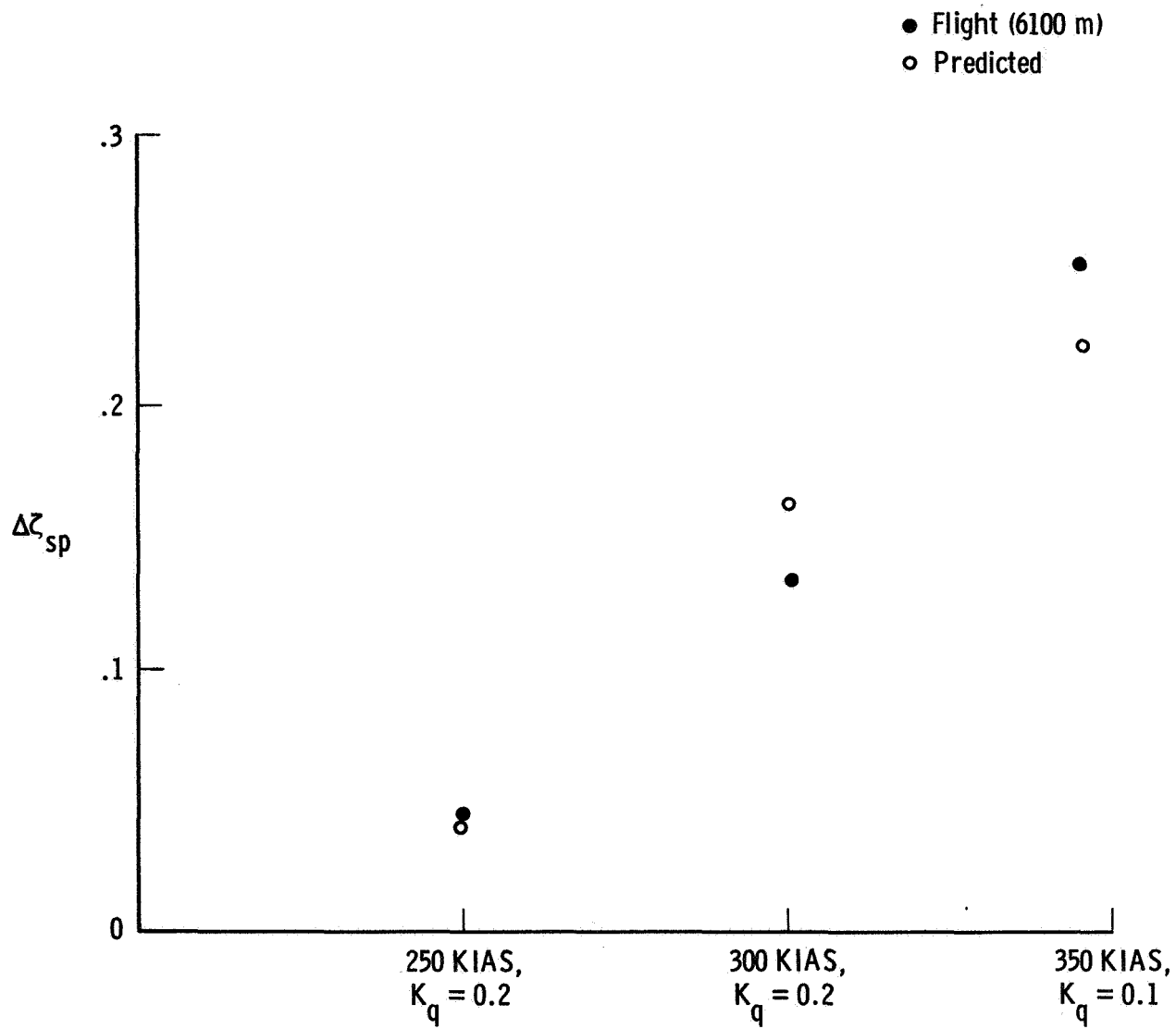
Figure 8. Improvement in short period damping ratio for lead-lag filter.
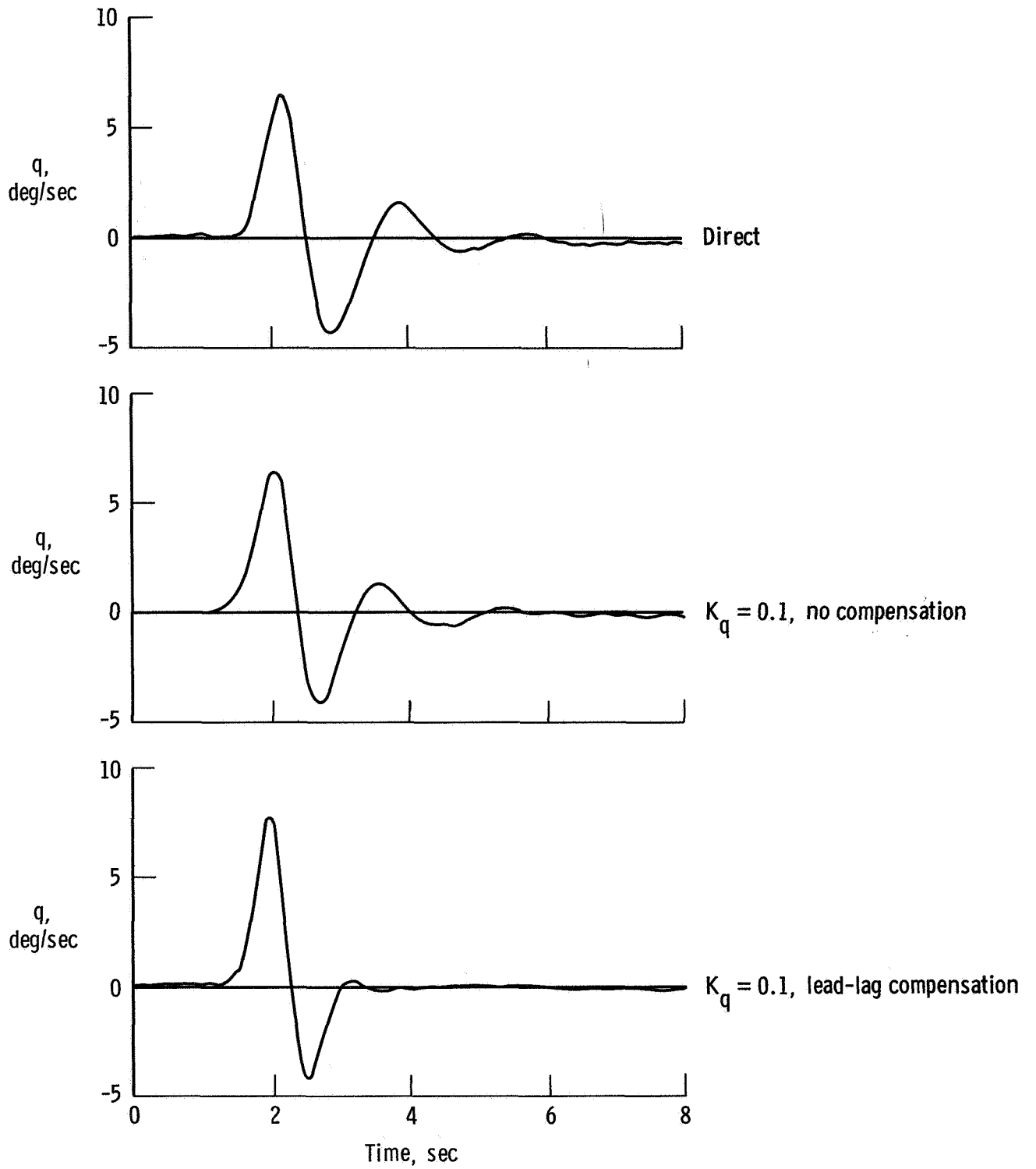
Figure 9. Effect of digital stability augmentation. Pitch SAS; 350 KIAS; 6100 m.

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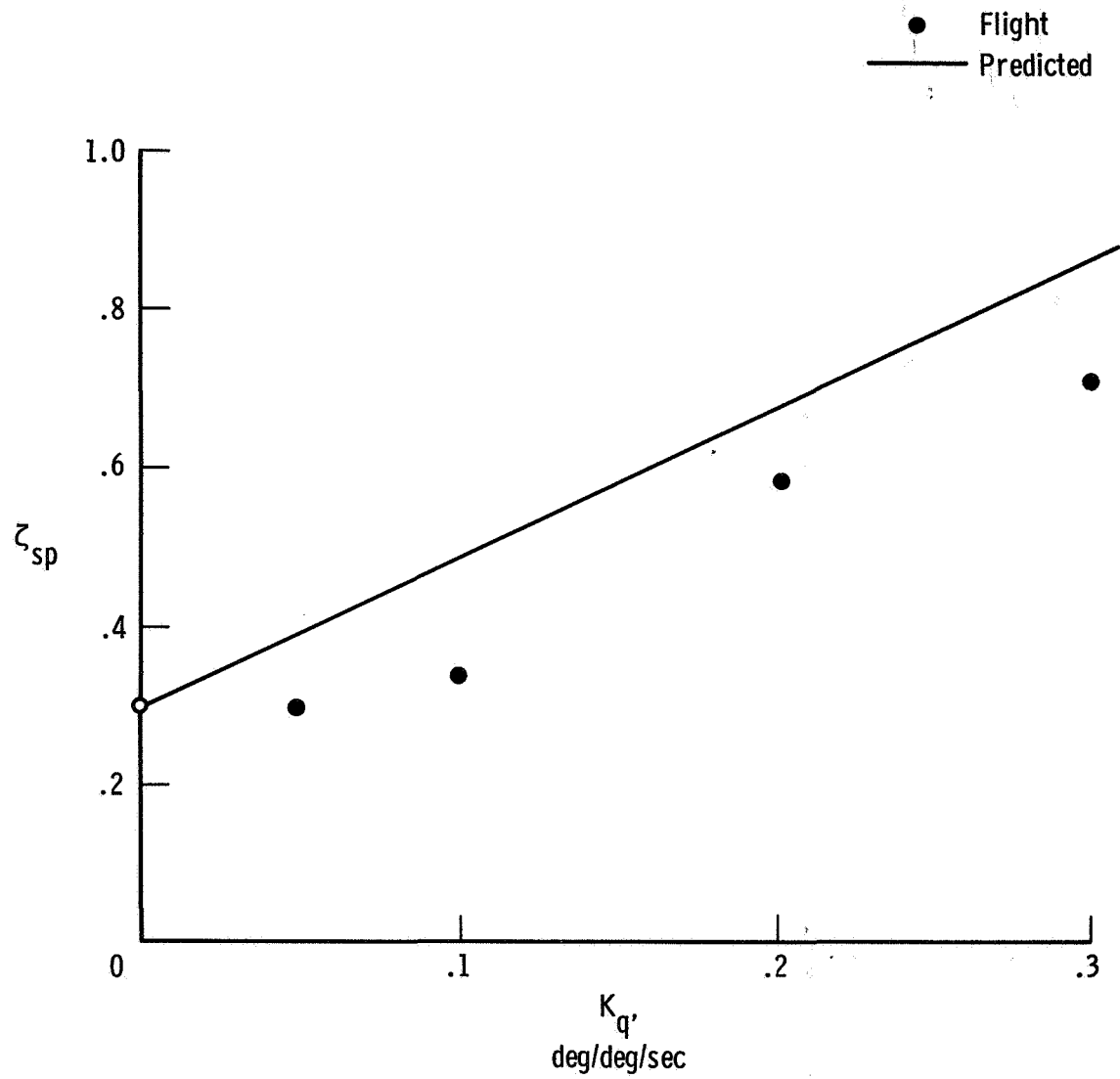
(a) Pitch SAS, 300 KIAS, 6100 m, 10,890 kg.

Figure 10. Comparison of in-flight and predicted digital SAS performance.
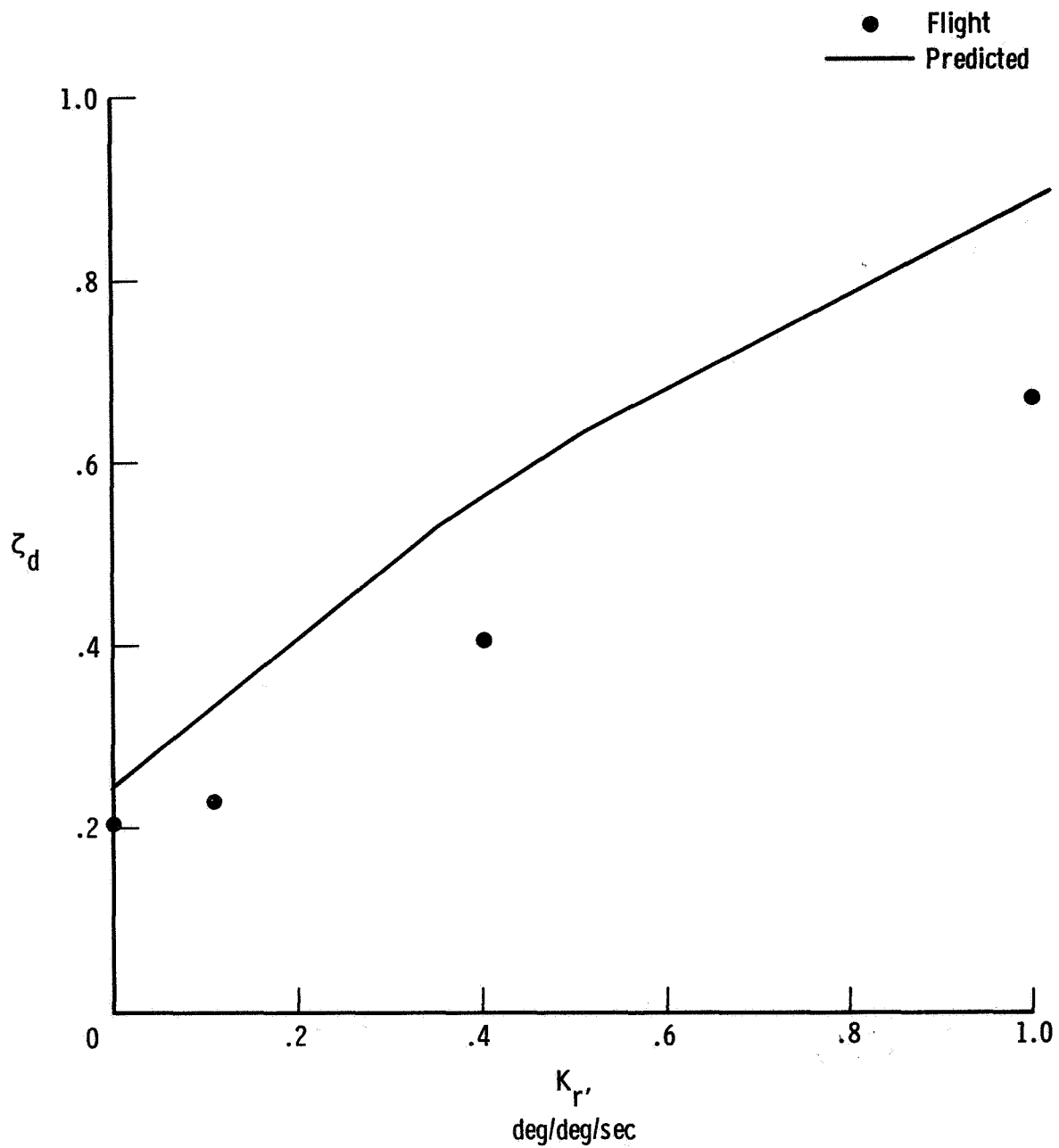
(b) Improvement in Dutch-roll damping for increase in yaw rate gain. 250 KIAS, 6100 m.

Figure 10. Continued.
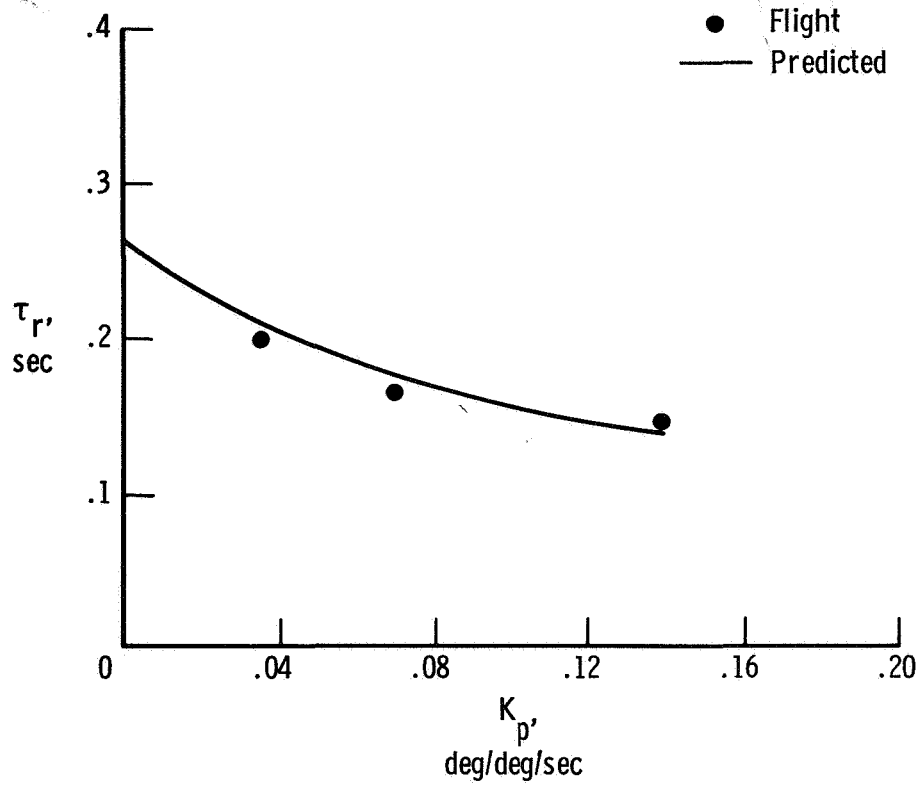
(c) Improvement in roll mode time constant for increase in roll damper gain. 250 KIAS.

Figure 10. Concluded.
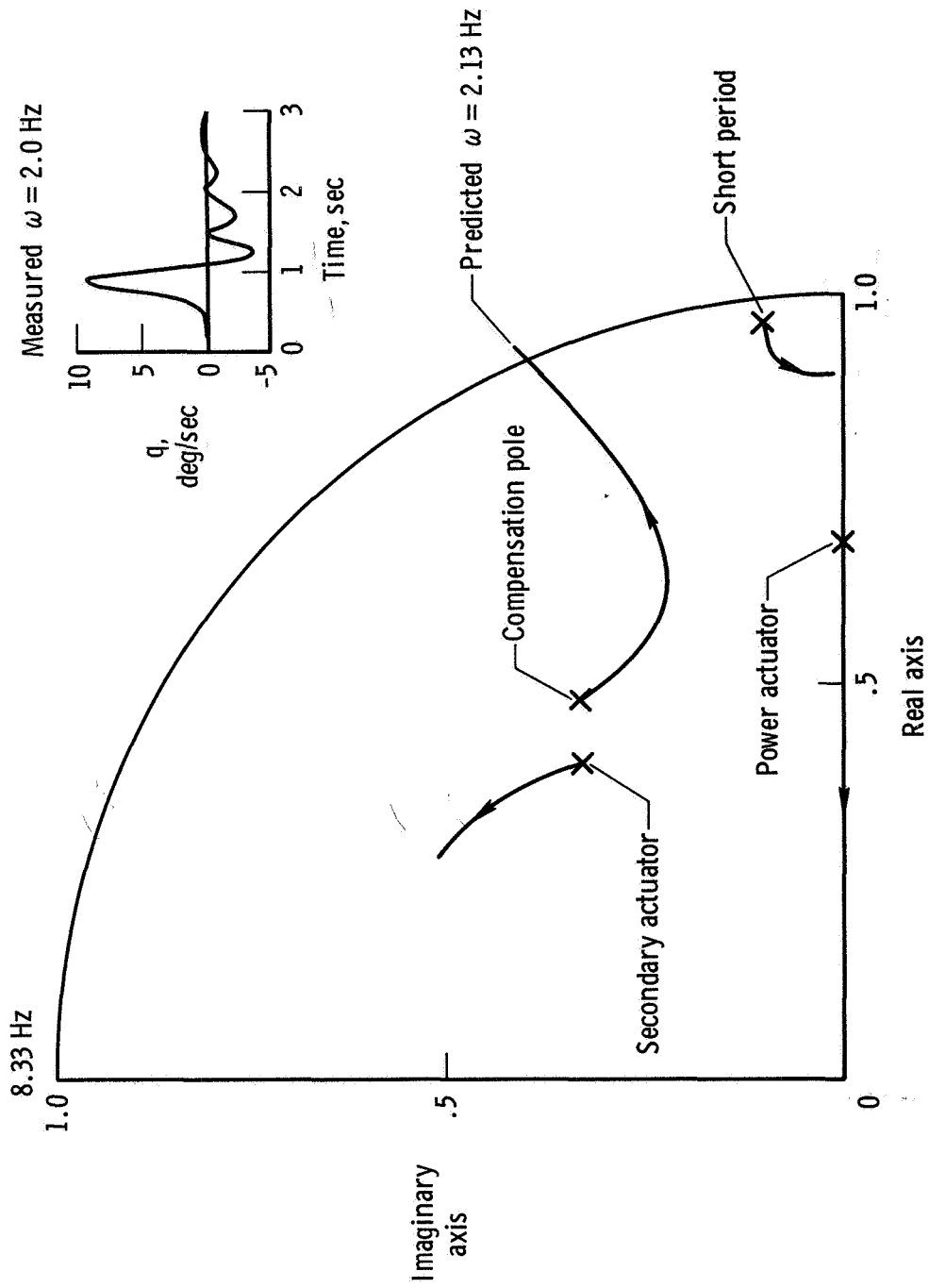
Figure 11. Prediction of system instability at high gain. 350 KIAS; 6100 m.

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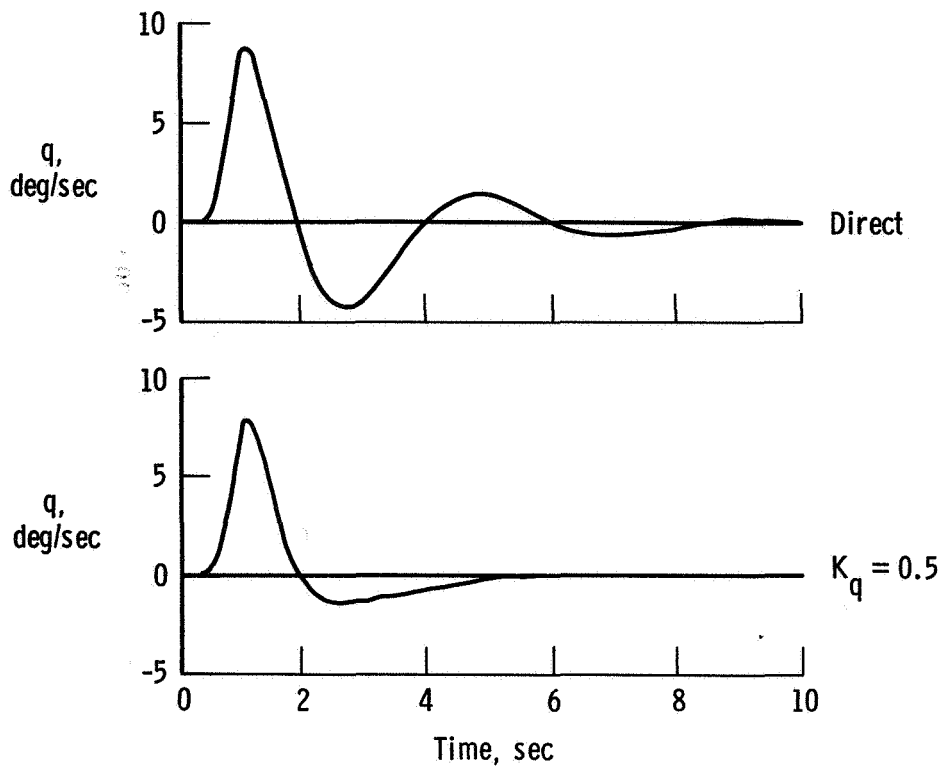
(a) Effect of SAS on damping ratio. Pulse pilot input.
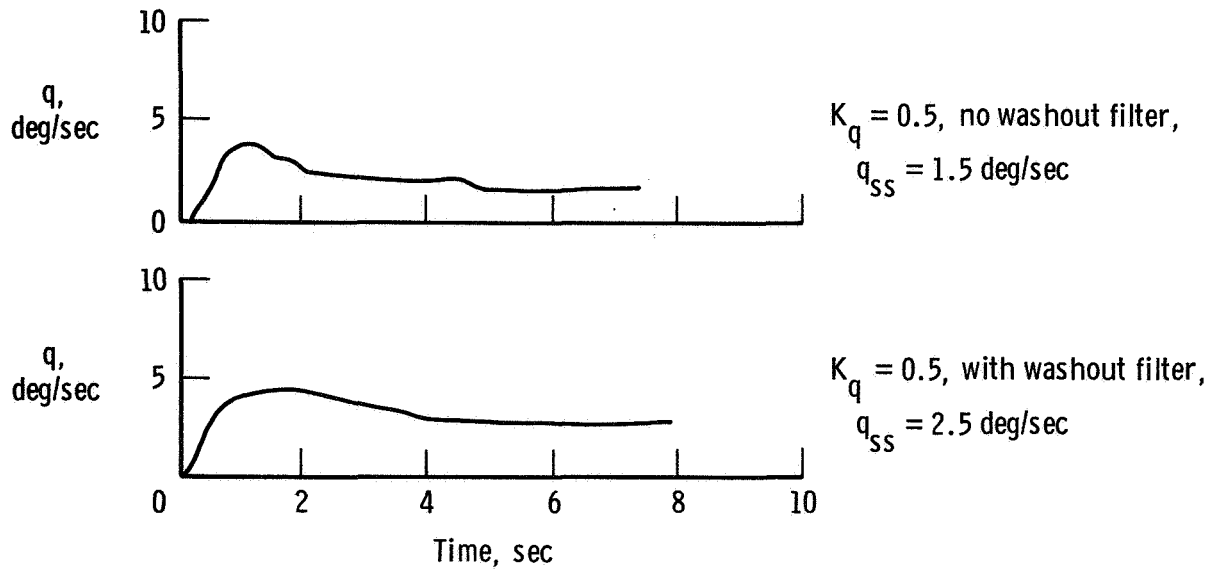
(b) Effect of washout filter on steady state response. Step pilot input.

Figure 12. Pitch SAS operation at low speed. 180 KIAS; wing up.
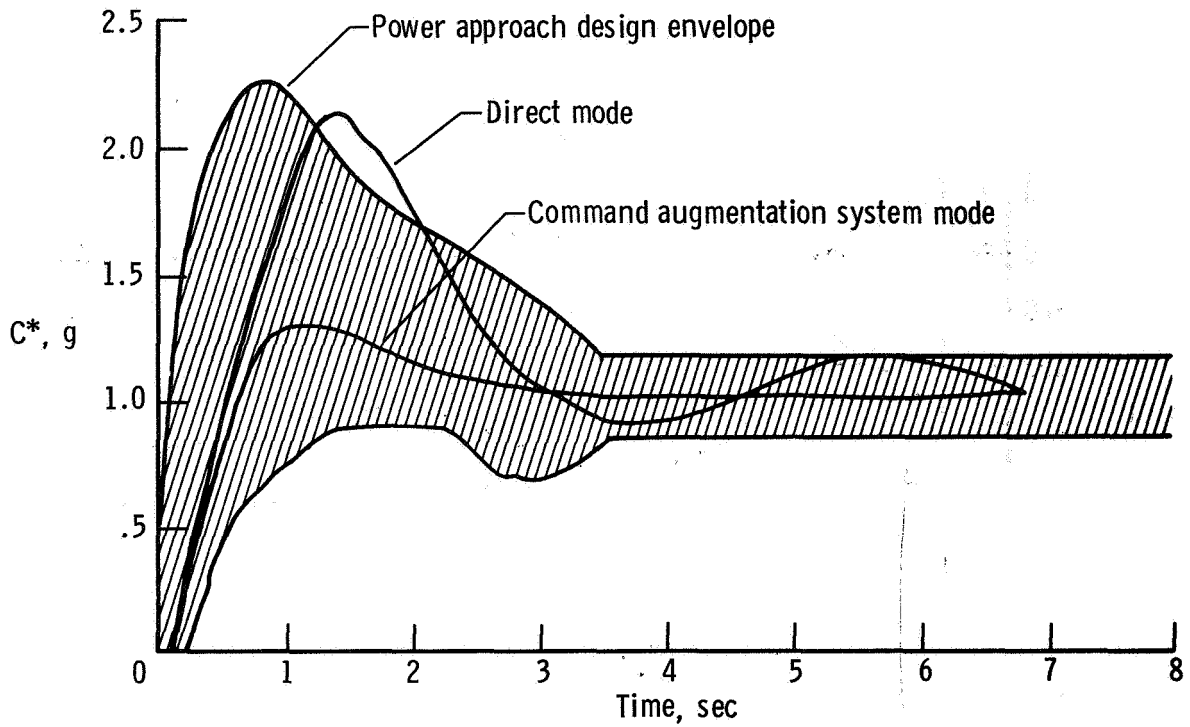
(a) 180 KIAS, 4570 m, $K_{C*} = 2.0$ deg/g.
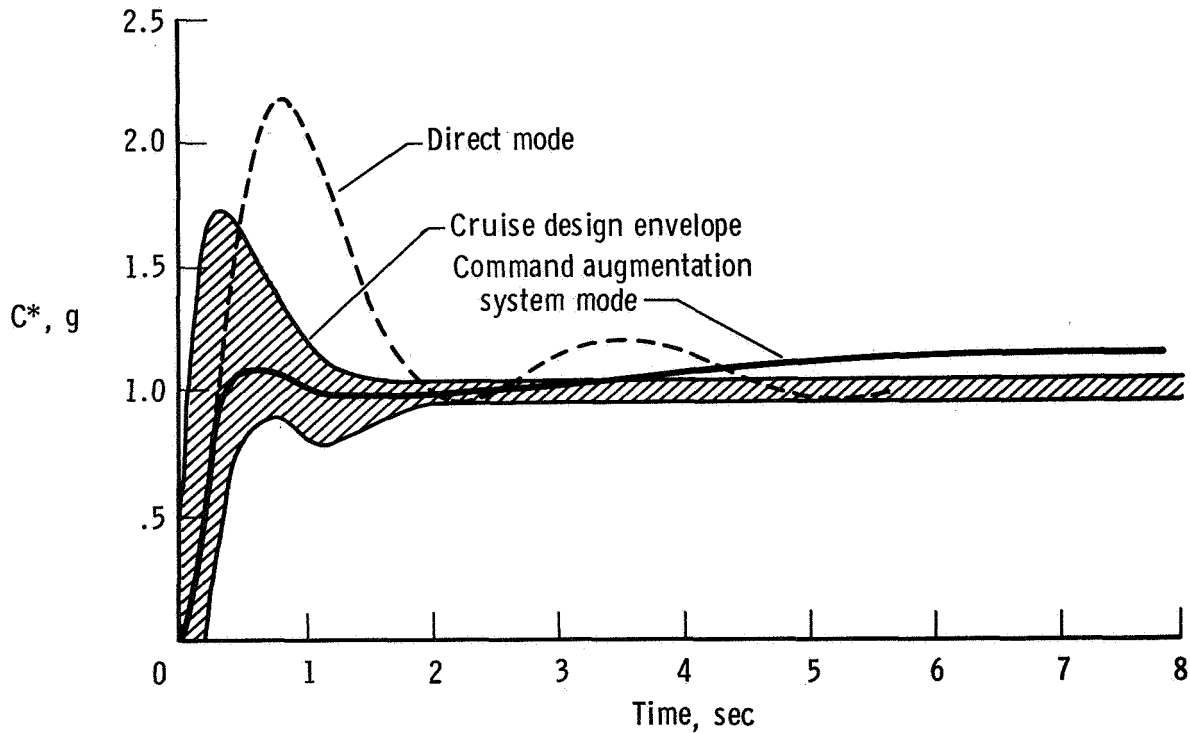
(b) 250 KIAS, 4570 m, $K_{C*} = 1.5$ deg/g.

Figure 13. C* response of F-8 DFBW aircraft.
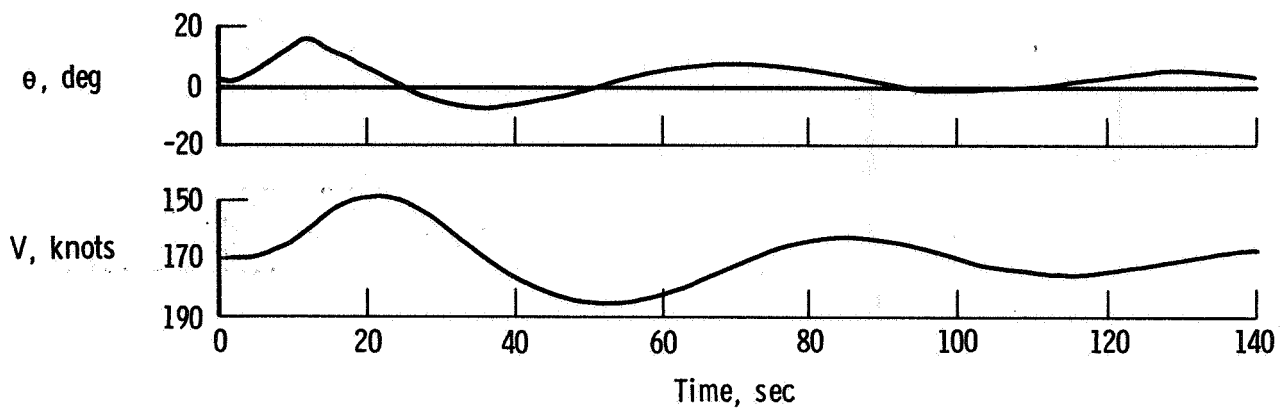
(a) Direct mode, hands off.
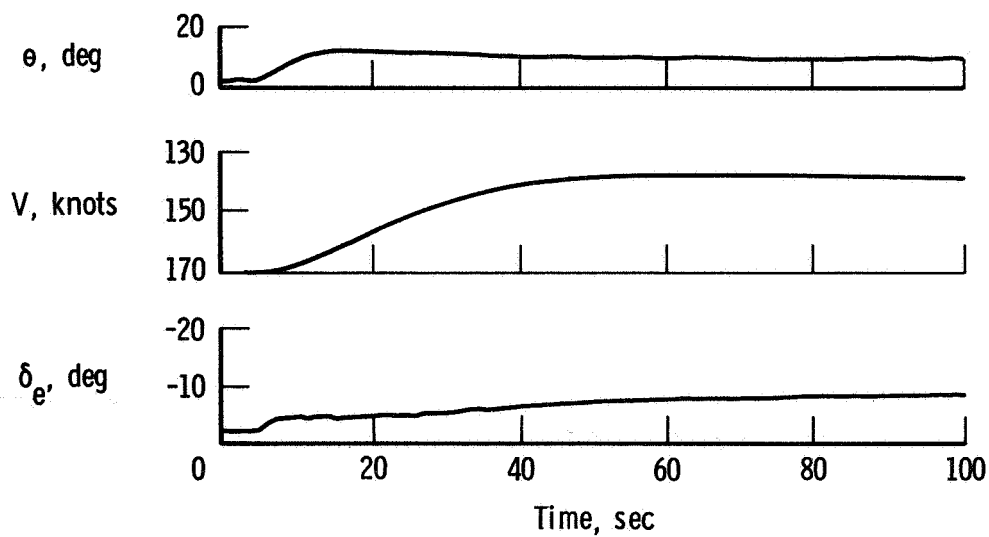
(b) CAS mode, hands off, K_{C*} = 2.0 deg/g.

Figure 14. F-8 DFBW phugoid response.
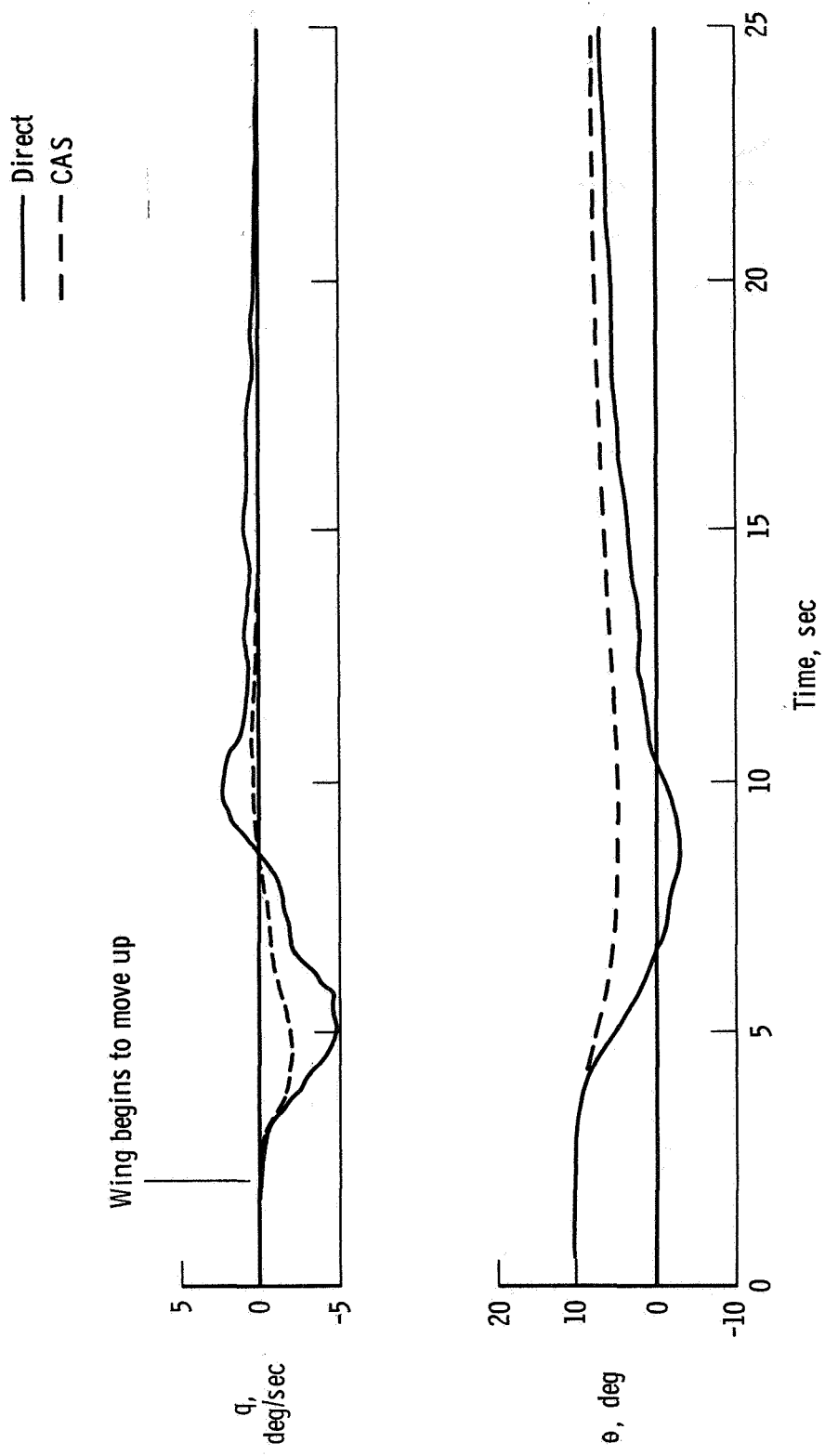
Figure 15. Effectiveness of CAS mode in reducing transient due to down-to-up wing transition.

Pilot rating

1

2 — Does not require much attention, can hold altitude well, workload half that in direct ●

3

Satisfactory / Unsatisfactory

4 — A little oscillation in level-offs, workload above normal, some short period evident in response ●

5

6

Low workload, easy to hold altitude, excellent trimmability ●

Direct   SAS   CAS
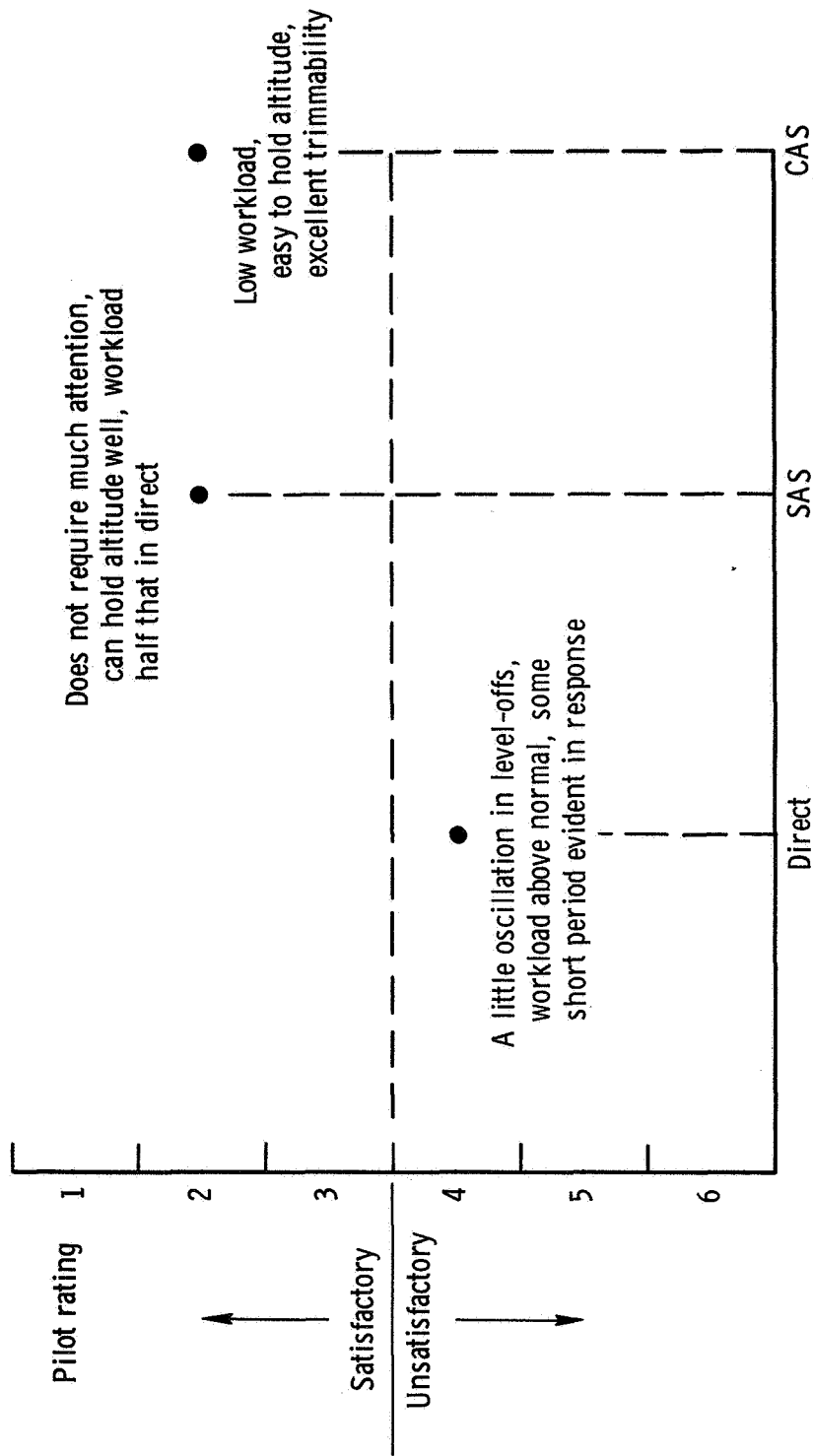
(a) Instrument maneuvers. 250 KIAS , 4570 m .

Figure 16. Longitudinal handling qualities summary — pitch mode.
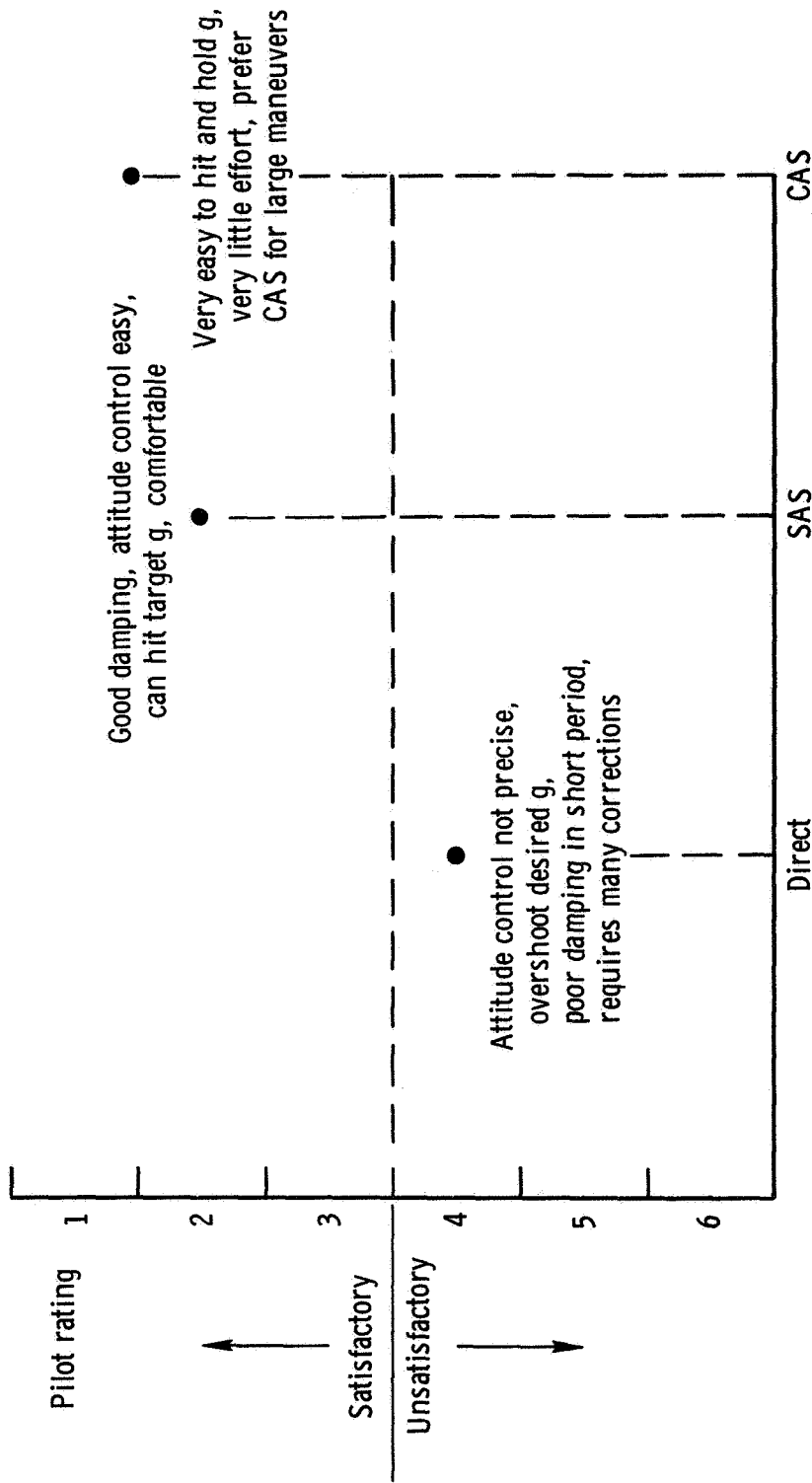
(b) Large maneuvers. 250 KIAS, 4570 m.

Figure 16. Concluded.
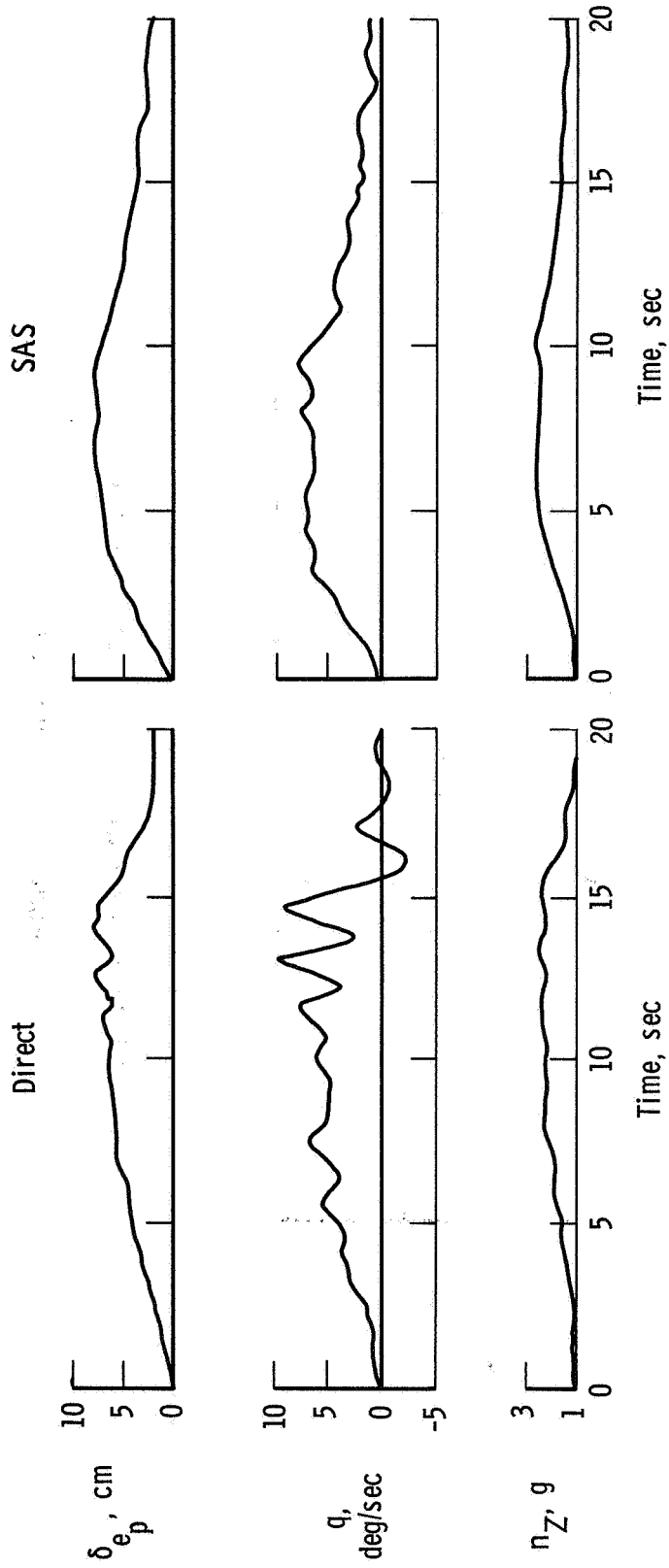
Figure 17. Effect of pitch SAS in wind-up turn. 300 KIAS; 6100 m.
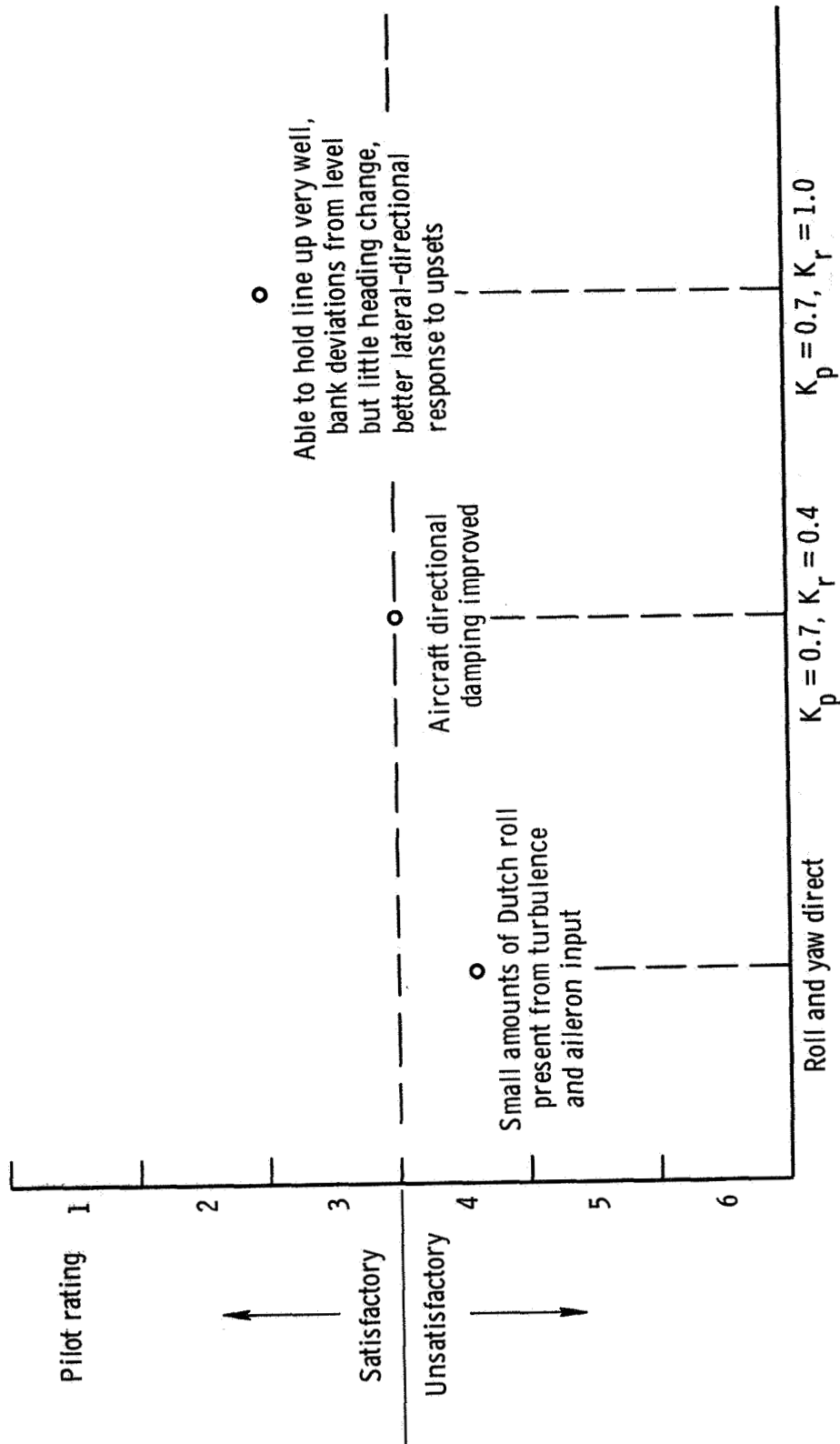
(a) Lateral-directional.

Figure 18. Ground-controlled approach handling qualities summary. Light-to-moderate turbulence.
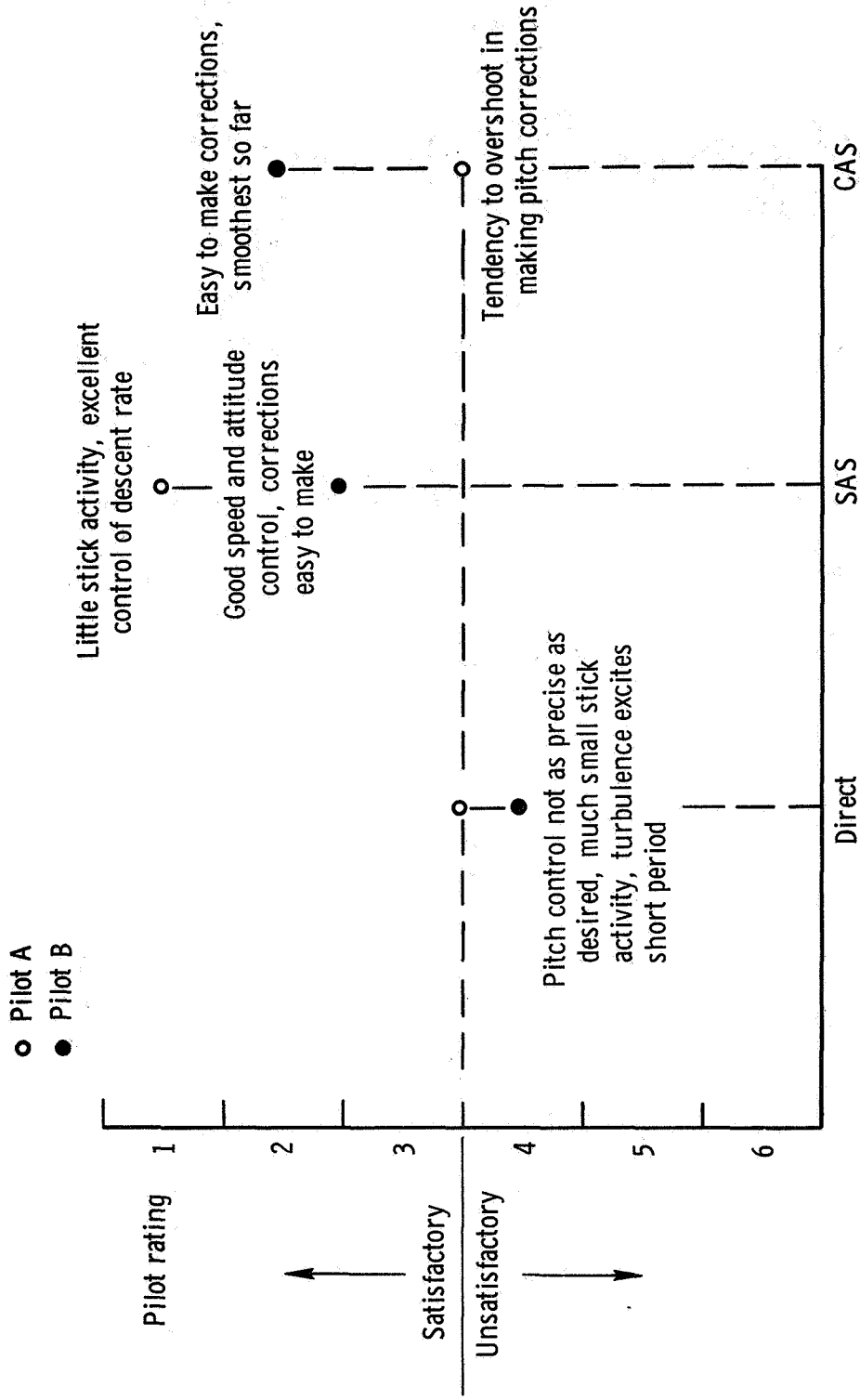
(b) Longitudinal, pitch mode.

Figure 18. Concluded.
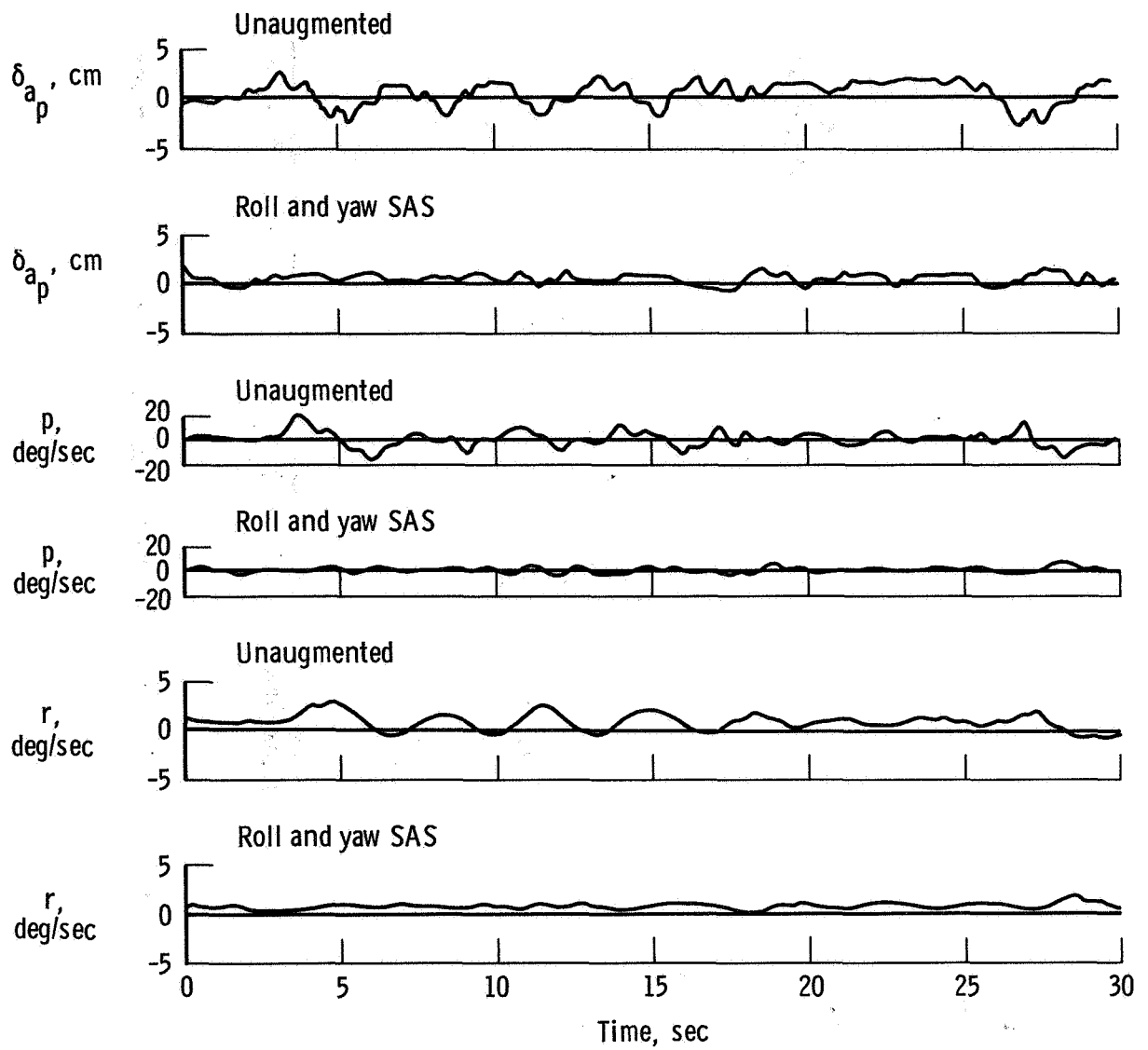
Figure 19. Improvement in lateral-directional flying qualities in gunsight tracking maneuver.
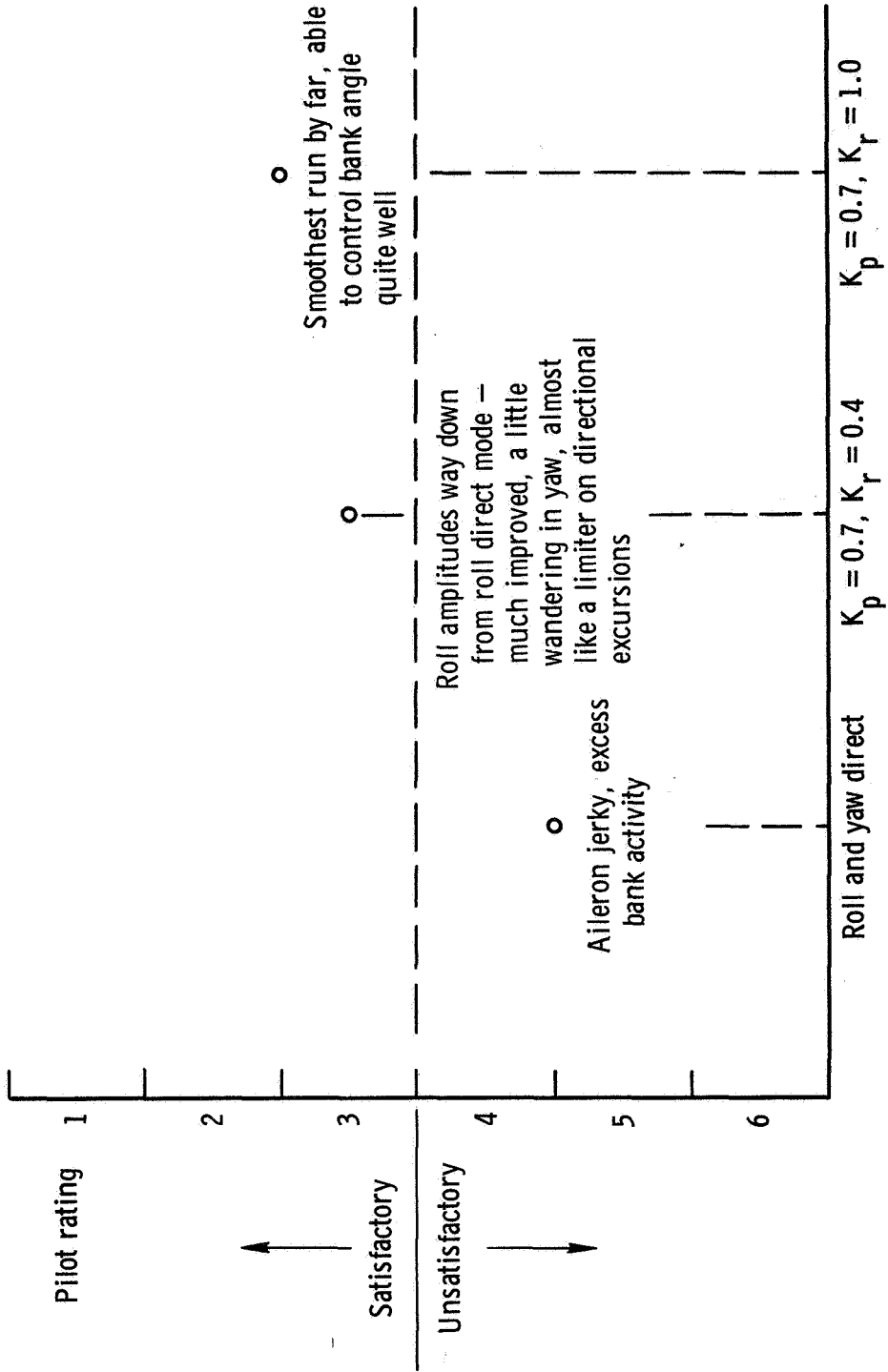
Figure 20. Pilot opinion of lateral-directional flying qualities in gunsight tracking maneuver.
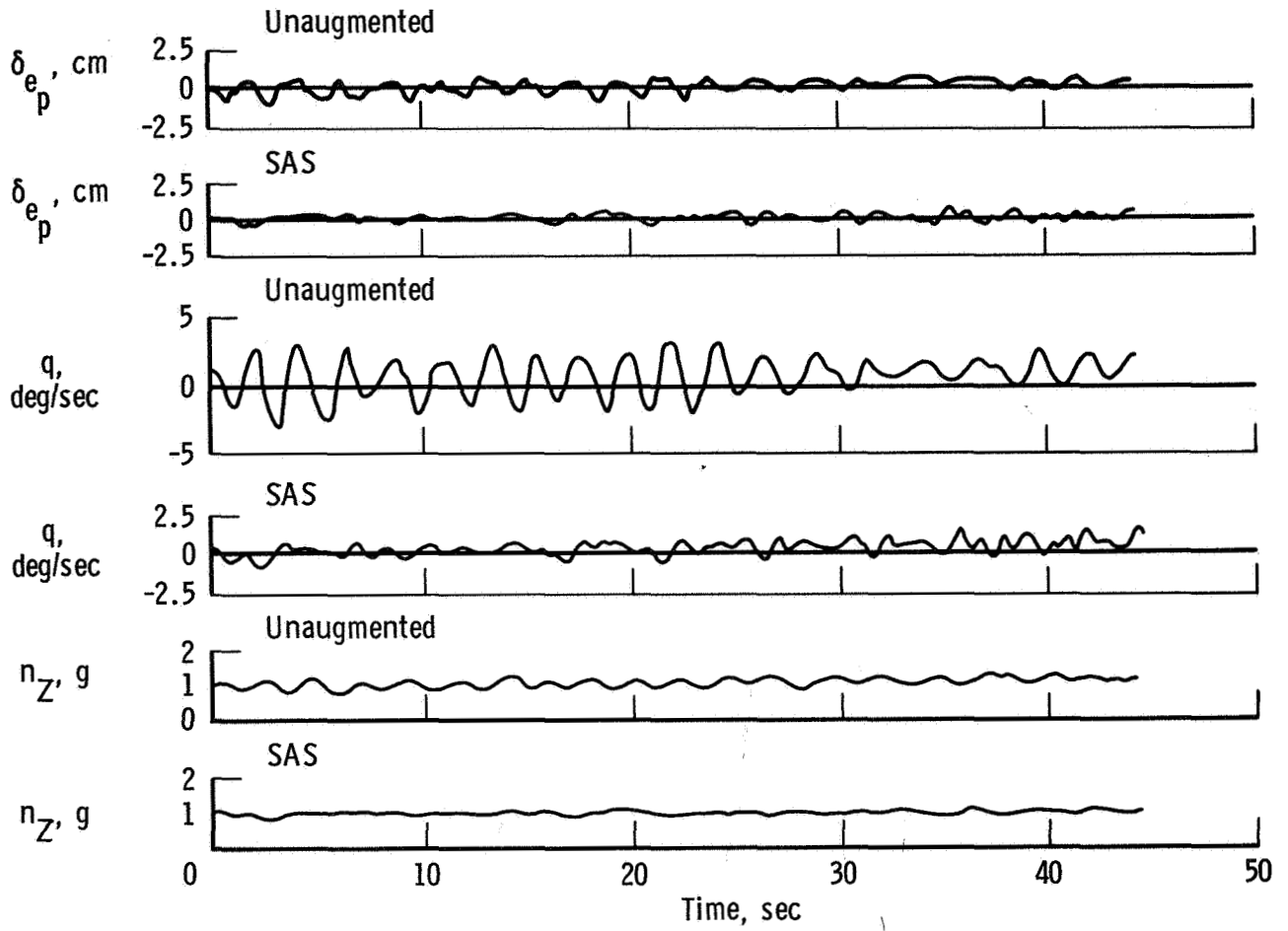
Figure 21. Pilot activity and airplane response in formation flight.

1	Initiator requests change
2	Software control board concurrence
3	Formal program change request to software contractor
4	Contractor estimates of core timing and cost
5	NASA go-ahead
6	Design and coding

NASA verification · Contractor verification

Piloted iron bird simulation

All-digital simulation

7

Hybrid simulation

8	Formal program release
9	Software control board approval
10	Implement in flight software

(a) Program change procedure.

Figure 22. Software management.

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1   Engineering requirements

2   Construct deck with desired memory load

3   Diagnostic computer program

4   Sequential duplicate deck punched

5   Examine listing for any errors detected

6   Punch tape program

7   Paper tape punched

8   Compare with predicted memory sums

     Load computer and display memory sums

9   Iron bird simulation

10   Flight ready
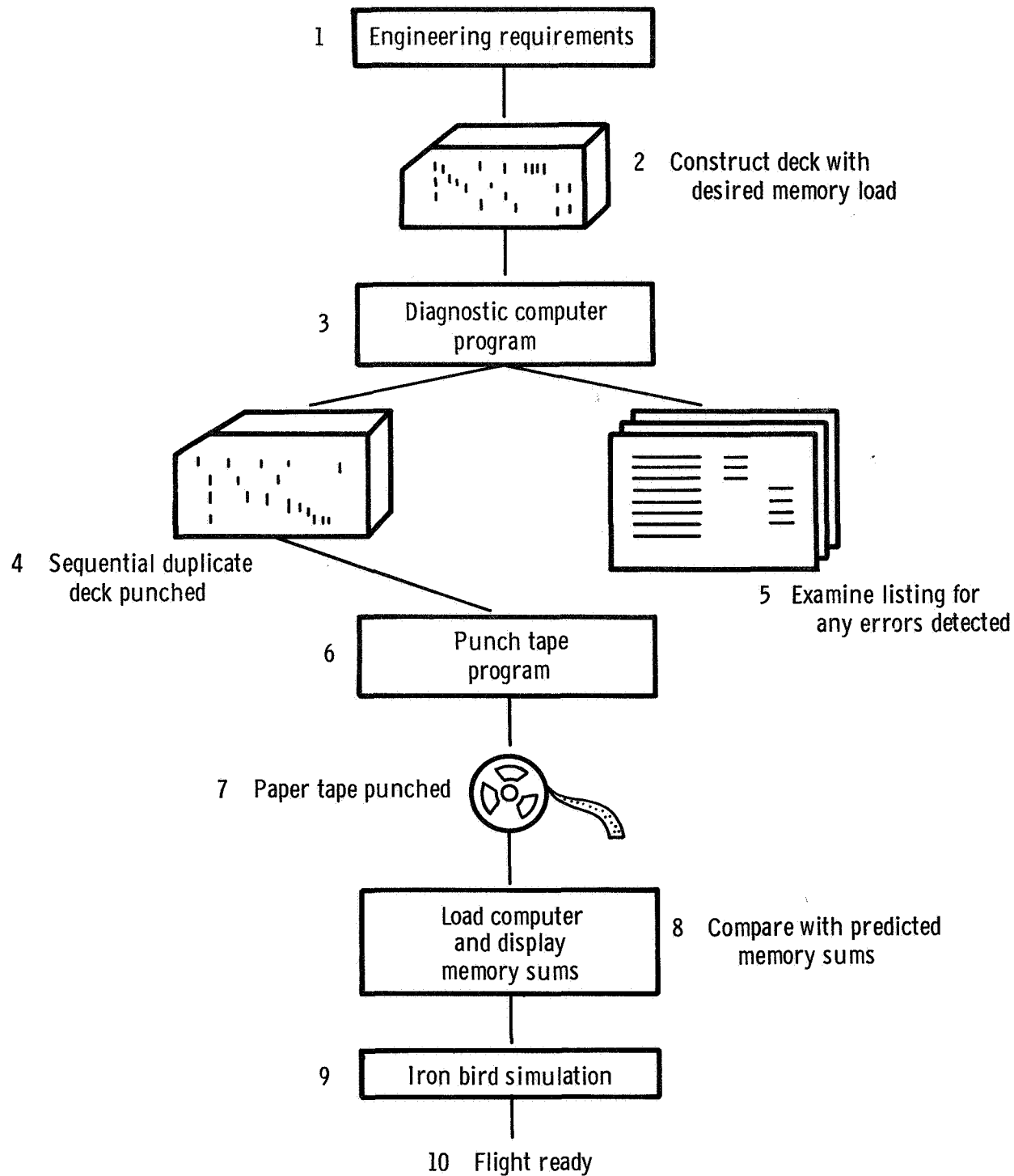
(b)   Procedure for new erasable memory load.

Figure 22. Concluded.

A PILOT'S OPINION OF THE F-8 DIGITAL FLY-BY-WIRE AIRPLANE

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SUMMARY

The handling qualities of the F-8 digital fly-by-wire airplane are evaluated by using the Cooper-Harper rating scale. The reasons for the ratings are given, as well as a short description of the flying tasks. It was concluded that the handling qualities of the airplane were good in most situations, although occasional ratings of unsatisfactory were given.

INTRODUCTION

A standard F-8C aircraft was equipped with a roll damper, a yaw damper, and an aileron-to-rudder interconnect. The airplane had no pitch damper. Handling qualities were satisfactory throughout a large portion of the flight envelope.

This paper evaluates the airplane's handling qualities on the basis of the Cooper-Harper rating scale (ref. 1 and fig. 1) after the removal of the mechanical control links and the addition of the Apollo hardware digital fly-by-wire control system.

A force side stick controller was mechanized in the analog backup control system and was evaluated by using the same tasks as those used to evaluate the digital primary control system.

The yaw axis was not extensively evaluated, so results are not reported in this paper.

The primary purpose of the program was to expeditiously demonstrate the feasibility and reliability of a digital fly-by-wire control system for an airplane. The space-proved Apollo system was adapted to the airplane, forcing compromises that did not allow optimization of the airplane's handling qualities. Nevertheless, the handling qualities were mostly satisfactory.

SYMBOLS AND ABBREVIATIONS

BCS   analog backup control system

CAS   command augmentation system

DIR   direct mode of control

q   dynamic pressure

SAS   stability augmentation system

x-plane from wingtip to wingtip of a target aircraft

y-plane from nose to tail of a target aircraft

CENTER STICK HANDLING QUALITIES

Takeoff

Takeoffs with the F-8 digital fly-by-wire airplane were normally made using the stability augmentation system (SAS) in all axes. This gave a well damped aircraft that handled turbulence effectively. Bank angle control was good and could be set quickly and relatively precisely. A pilot rating of 2 on the Cooper-Harper scale was given for the takeoff and climbout (figs. 2 and 3).

Cruise

Control for cruising flight was easily adequate and is not further discussed in this paper.

Gross Maneuvering and Aerobatics

Pitch and roll control for any moderate to high rate maneuver was similar in each flight control system configuration. Maneuvers performed with the backup control system (BCS), direct mode (DIR), SAS, and command augmentation system (CAS) appeared very much alike to the pilot, which suggests that these were not good tasks for an evaluation.

Formation Flight

The ability to fly good wing and trail formation (fig. 4) is a requisite for fighter aircraft. It is also a task that rapidly exposes deficiencies in the flight control system. Poor control harmony between pitch and roll, poor damping, incorrect time constants, undesired force gradients, and other problems are all

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revealed when the aircraft is put to the formation task. With a good formation-flight aircraft, vertical position can be held consistently within 30 centimeters and lateral displacement can be held as desired. The task rated with the F-8 digital fly-by-wire airplane was the ability to hold a close wing position and to assess the workload required to do it.

While the airplane was in the backup control system, pitch sinusoidal oscillations of ±60 centimeters from a base position were caused by the slight delay in response to pitch stick inputs. Considerable pilot compensation was required to achieve even that amount of control. The response in the stability augmentation system was satisfactory but slightly sluggish because we were operating in the flat portion of the stick curve (fig. 5). Control in the direct mode was inferior to control with the stability augmentation system because of underdamped short period oscillations.

By far the most difficulty was encountered in trying to conquer the roll axis. Considerable attention was required on the part of the pilot any time formation was attempted in the roll backup control system or the direct mode. Response was objectionable because of small control deflections when low stick displacements were used and fast response when the apparent lag was overcome by using larger stick displacements. Using the roll stability augmentation system markedly improved the ability to hold close position, possibly because the stability augmentation system tended to initially oppose a rapid response to a pilot input. The stability augmentation system made the aircraft well behaved up to speeds where quantization became a factor.

Tracking

Gunsight tracking with a fixed reticle (fig. 6) was flown because it was an excellent way to assess the response of the airplane to pilot commands. The film was analyzed frame by frame to determine the miss distance, which was referenced to the plane running through the target airplane's wingtips (x-plane) or to the plane running from the target airplane's nose to its tail (y-plane). This allowed control difficulties to be classified as either a lateral-directional or a pitch problem (figs. 2 and 3).

The pilot's ability to keep the gunsight aiming point (pipper) on the tailpipe of the target airplane in a dynamic, tight loop situation was the task rated.

Tracking in the pitch stability augmentation system was unsatisfactory unless considerable trim was used to return the stick to the flatter portion of the parabolic deflection curve. If the trimming was omitted, quantization and its accompanying short period oscillations caused pipper oscillation in the pitch plane. Tracking in the stability augmentation system with a trimmed stick was good enough to perform the mission without improvement. The same problems arose in the direct mode, but this mode was without pitch rate damping, and was thus rated moderately objectionable.

The pitch backup control system was by far the smoothest of the modes tested and afforded good pitch steering at all angles of attack. Some short period oscillations occurred, but they were not significant.

The difficulties were considerable in the roll axis. There was a definite tendency toward pilot-induced oscillations whenever precise, rapid corrections were required. This was evident in both the backup control system and the direct mode. The roll stability augmentation system reduced the magnitude of the problem, but its sensitivity degraded the airplane's ability to track precisely.

The fixed-ratio aileron-to-rudder interconnect produced slight proverse yaw during roll-in. This was considered desirable, since it provided a slight lead in the direction of the target.

Ground-Controlled Approach

Ground-controlled approaches were flown using radar for positioning. This was an excellent task for the evaluation of precision control during tight loop instrument flight. Deviations from a preset position and altitude were radioed to the pilot, who then maneuvered the airplane back toward zero deviation. The response of the airplane to the pilot-initiated corrections was rated.

Pitch control was fair in the backup control system and the direct mode because of the short period oscillations generated by pitch corrections. Pitch response in the stability augmentation system was excellent, in that 30-meter-per-minute changes could be made in the rate of descent. Corrections in the pitch command augmentation system were initiated satisfactorily, but a distracting tendency to overshoot was noted that increased the pilot workload and therefore worsened the pilot rating.

Lateral control with low damping gains showed some deficiencies because of continuous low amplitude oscillations up to ±6° of bank. No attempt was made to correct this deficiency during the flight test program.

Landing

A portion of several flights was devoted to the assessment of the aircraft in various control modes in the landing pattern. The pitch backup control system was relatively smooth, and there was little tendency for the pilot to couple with the aircraft. In the direct mode, however, there was a tendency toward a pilot-induced oscillation during wing and gear transients. Sink rate control was fair with both of these modes. The stability augmentation system offered good pitch control throughout the pattern, with reduced transients and good flare control. The pitch command augmentation system was the best mode evaluated, but it masked the speed stability, which tended to lead the pilot to believe that changing stick force meant changing aerodynamic conditions; that was not always true.

Flare and touchdown control were satisfactory as long as a slight amount of back stick pressure was held to keep the airplane off the flat portion of the parabolic pitch curve. If this was not done, the delay in response caused firm landings or ballooning.

Lateral control in the landing pattern was characterized by low damping, over-responsiveness, and some periods of continuous low amplitude bank excursions.

The effects of these characteristics were reduced somewhat by consciously lowering the pilot's response and having him accept 1° to 2° deviations from the bank angle desired. This was considered moderately objectionable in itself, and coupled with a strong crosswind it became unacceptable.

The stability augmentation system reduced the airplane's response to gusts and small inputs and therefore it was rated better than the simpler control modes.

SIDE STICK HANDLING QUALITIES

The side stick in the F-8 digital fly-by-wire airplane (fig. 7) was installed to ascertain whether a force side stick could be used to control an airplane during most phases of flight, especially takeoff and landing. No attempt was made to optimize the control parameters, although some changes were made for the flights near the end of the program. The side stick was mechanized in the analog backup control system, which had no dampers. A side stick takeoff was considered the most uncertain phase of flight and was therefore performed only after side stick control was evaluated in up and away flight.

Takeoff

During side stick takeoffs, the pilot applied nosewheel steering (with the center stick) until rudder power was sufficient and then moved his right hand to the side stick. He made no inputs until lift-off speed was reached, when he applied a smoothly increasing pitch force to the stick. No lateral force was used near the ground to reduce the tendency for pilot-induced oscillations. Lift-off was smooth and similar to center stick takeoffs except that the pilot did not know the elevator and aileron positions through stick position (figs. 8 and 9).

Gross Maneuvering

Gross maneuvering was easy with the side stick. Maneuvers such as large pitch attitude changes, wind-up turns, wingovers, and aileron rolls were performed without difficulty. Crosstalk between pitch and roll was not apparent.

Formation Flight

Formation flight, a high pilot gain task, was enlightening during the early development of the F-8 digital fly-by-wire control system, when it exposed the severity of the task. Formation flight was also difficult with the side stick.

Loose wing formation flight could be satisfactorily performed with the side stick, although there were occasional random force pulses in pitch or roll. As the distance between the two aircraft diminished, the pulsing became more frequent and pronounced, indicating the tightening of the pilot in the loop. This resulted in a tendency for pilot-induced oscillations in pitch or roll or both with the system as it was mechanized, that is, without dampers and without attempts at optimization.

Some crosstalk (force interaction) was apparent during formation flight. Although its effect was not severe, it did start a disturbance in one axis while the pilot was trying to control the other axis.

Tracking

Side stick tracking was typified by good to excellent control over the lateral-directional axis and continuous oscillations in pitch caused by pitch commands that were too abrupt and could not be smoothed. Crosstalk was absent in the tracking task.

Ground-Controlled Approach and Landing

Power approaches from both pitch out and ground-controlled approach patterns were flown easily with the side stick. Roll control was good with respect to bank angle itself, but continuous left and right lateral force inputs had to be made. This did not degrade bank control, but it did drive the workload up quite a bit. Pitch control was precise.

Many of the approaches were flown in turbulence, which had little adverse effect on control.

Landings were characterized by final approaches that were well controlled down to the flare point. The flare was easy to initiate, and control was good almost to touchdown. Just before touchdown on every flight, the flightpath was stairstep-like. This was caused by pulsing pitch inputs from the pilot.

No large extraneous motion was generated by a simulated go-around if the trim kept the forces down to low levels.

CONCLUDING REMARKS

The F-8 digital fly-by-wire airplane was generally well behaved throughout the flight envelope tested. Most of the handling qualities deficiencies encountered were a result of the original compromises made to adapt the Apollo system to the airplane. No extensive attempt to improve the Apollo-related deficiencies was made.

REFERENCE

1. Cooper, George E.; and Harper, Robert P., Jr.: The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities. NASA TN D-5153, 1969.

Controllable	Acceptable	Satisfactory	Excellent	1
			Good	2
			Fair	3
		Unsatisfactory	Some minor but annoying deficiencies	4
			Moderately objectionable deficiencies	5
			Very objectionable deficiencies	6
	Unacceptable		Major deficiencies which require mandatory improvement	7
			Controllable with difficulty	8
			Marginally controllable in mission	9
Uncontrollable Control will be lost during some portion of mission			Uncontrollable in mission	10

Figure 1. Cooper-Harper rating scale (ref. 1).
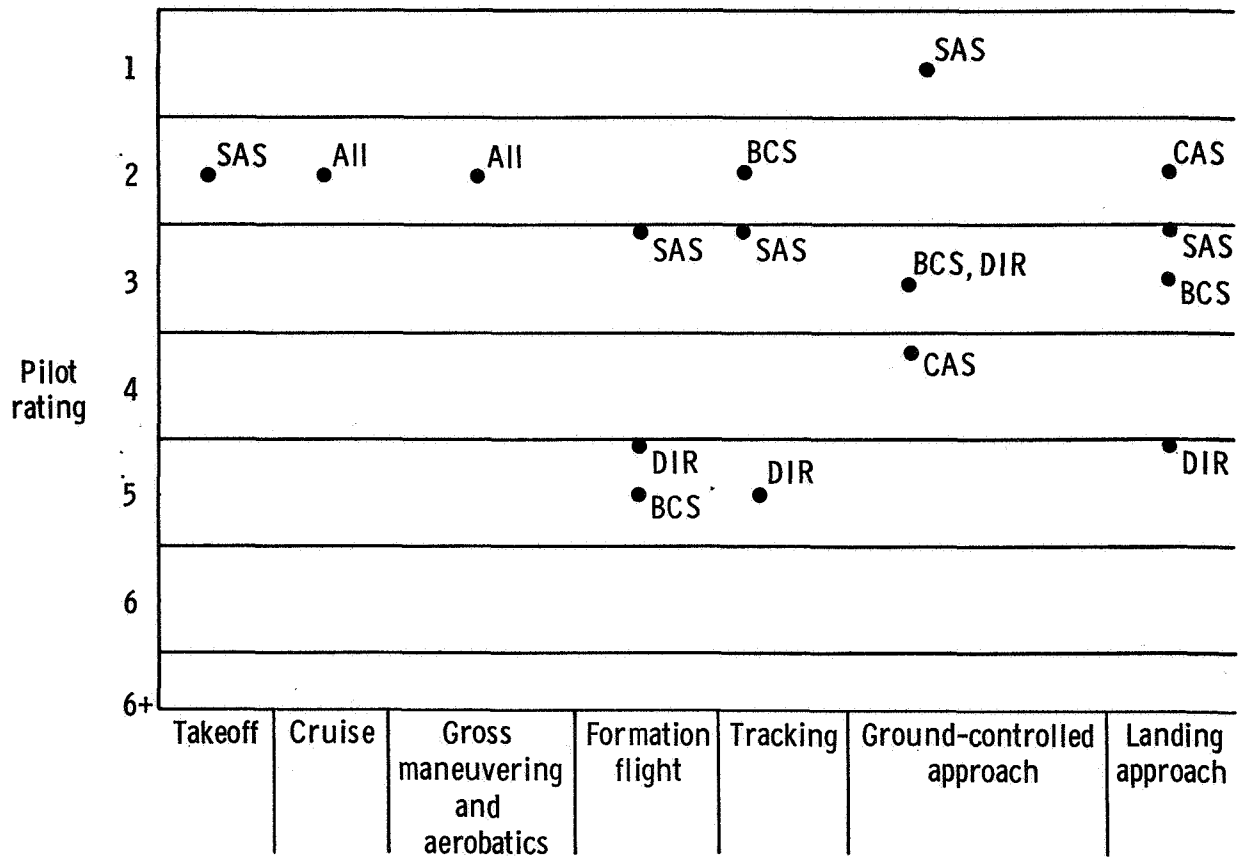
Figure 2. Center stick pilot ratings in pitch.
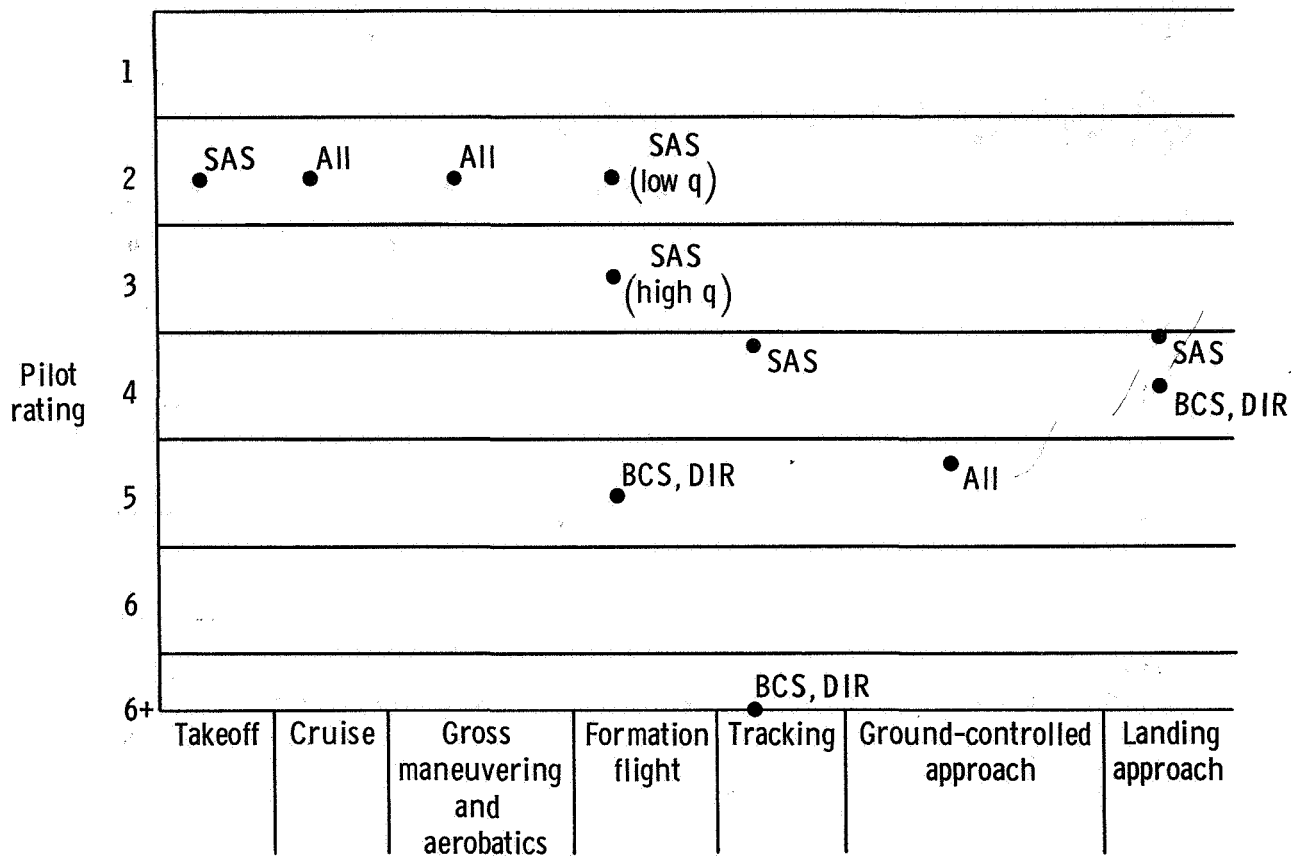
Figure 3. Center stick pilot ratings in roll.
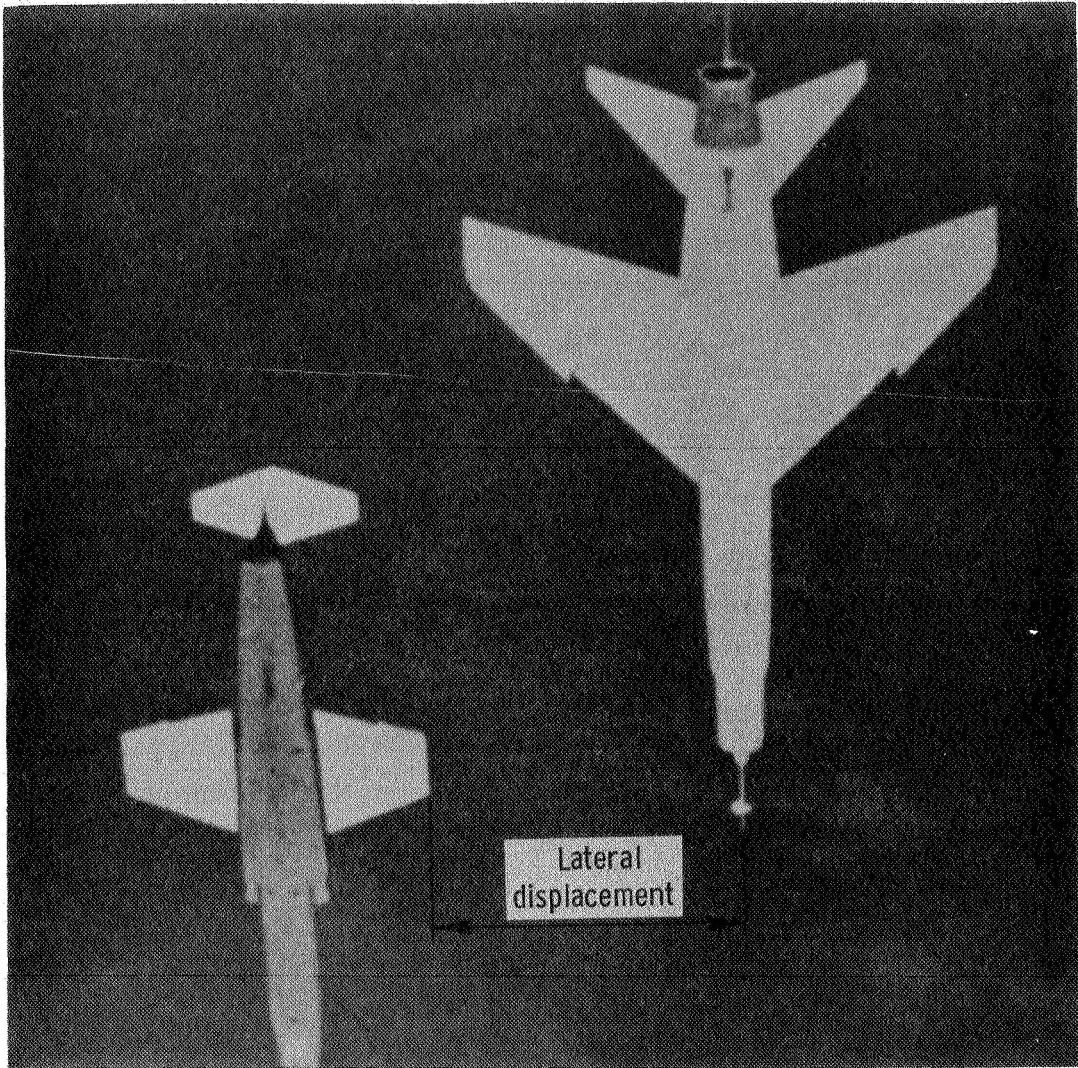
Figure 4. Formation flight.
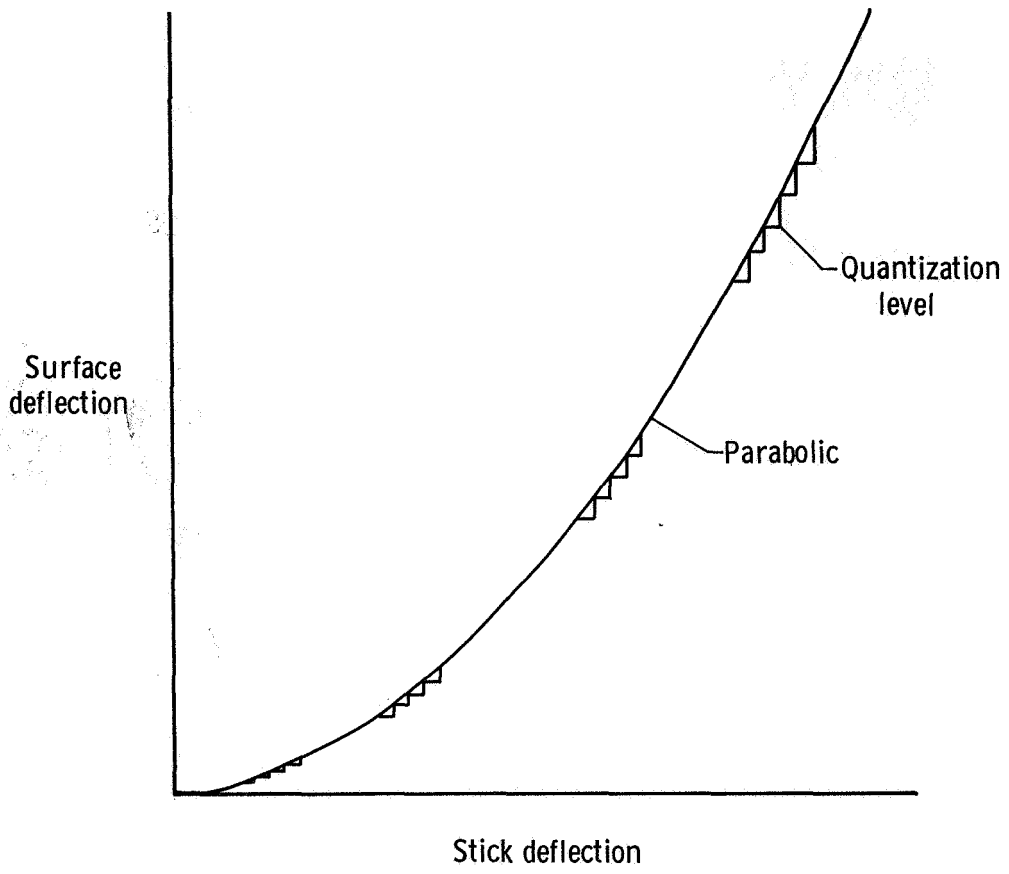
Figure 5. Control gearing.
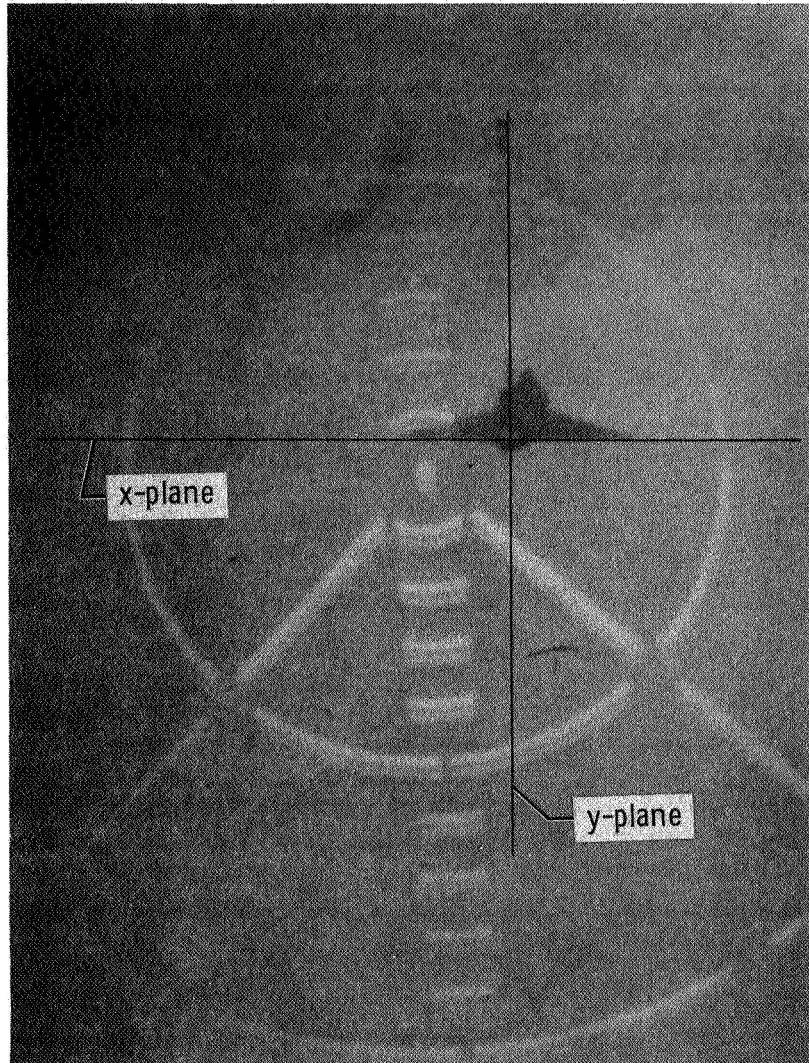
x-plane

y-plane

Figure 6. Gunsight tracking display.
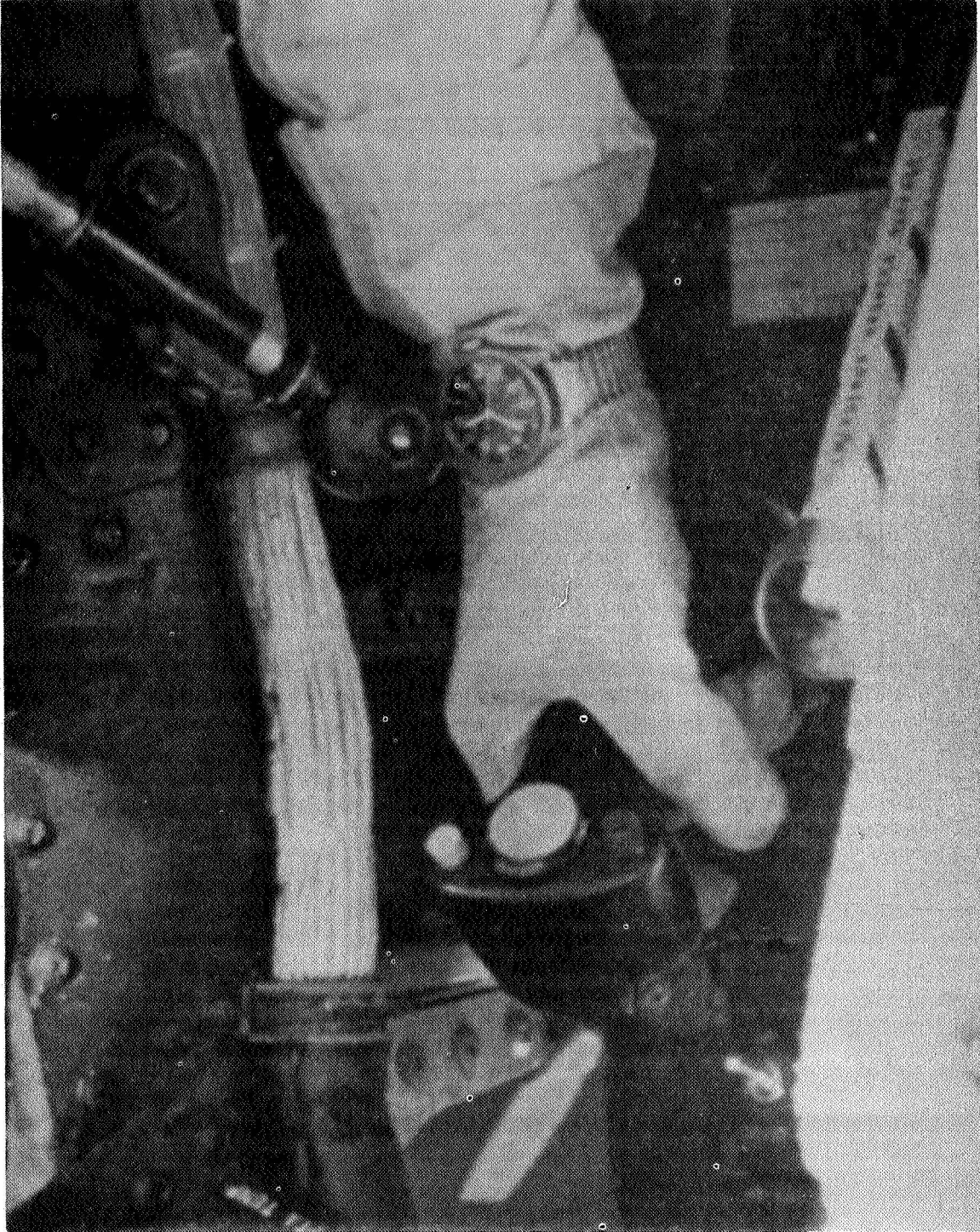
Figure 7. Force side stick in F-8 digital fly-by-wire airplane.
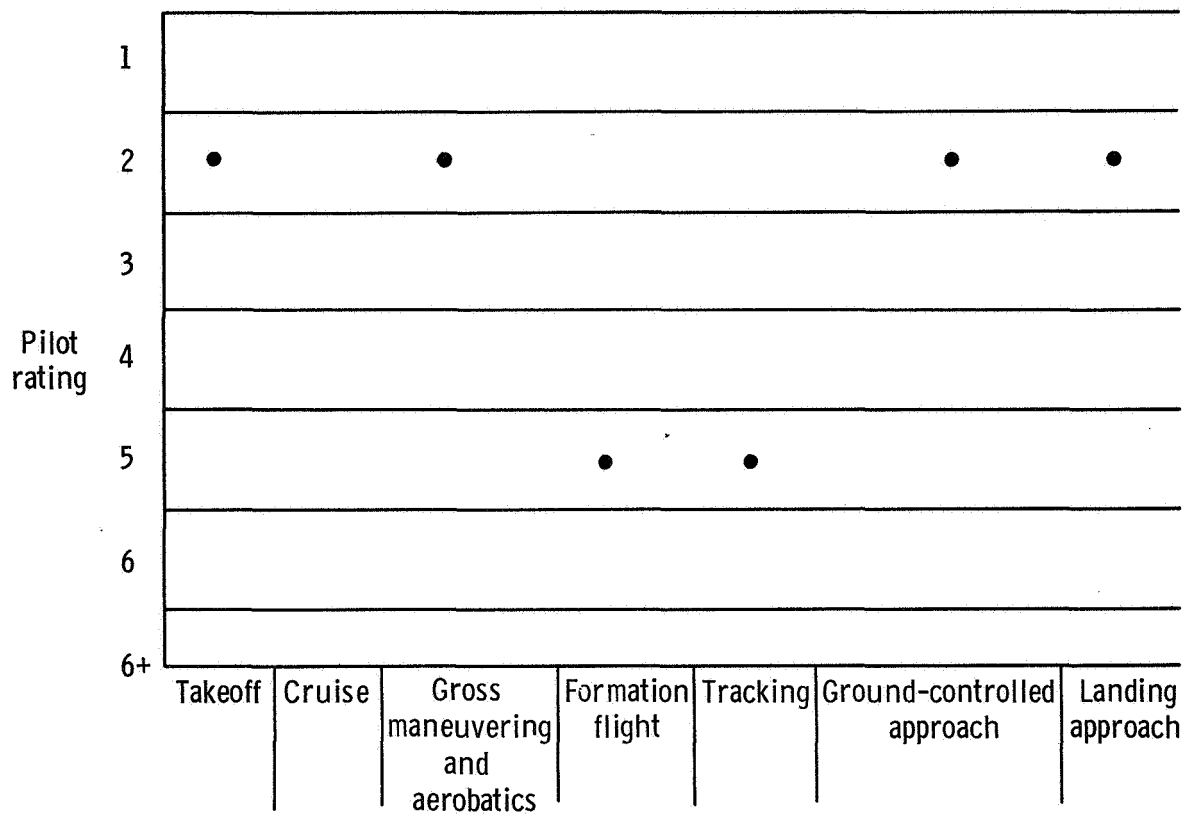
Figure 8. Side stick pilot ratings in pitch.
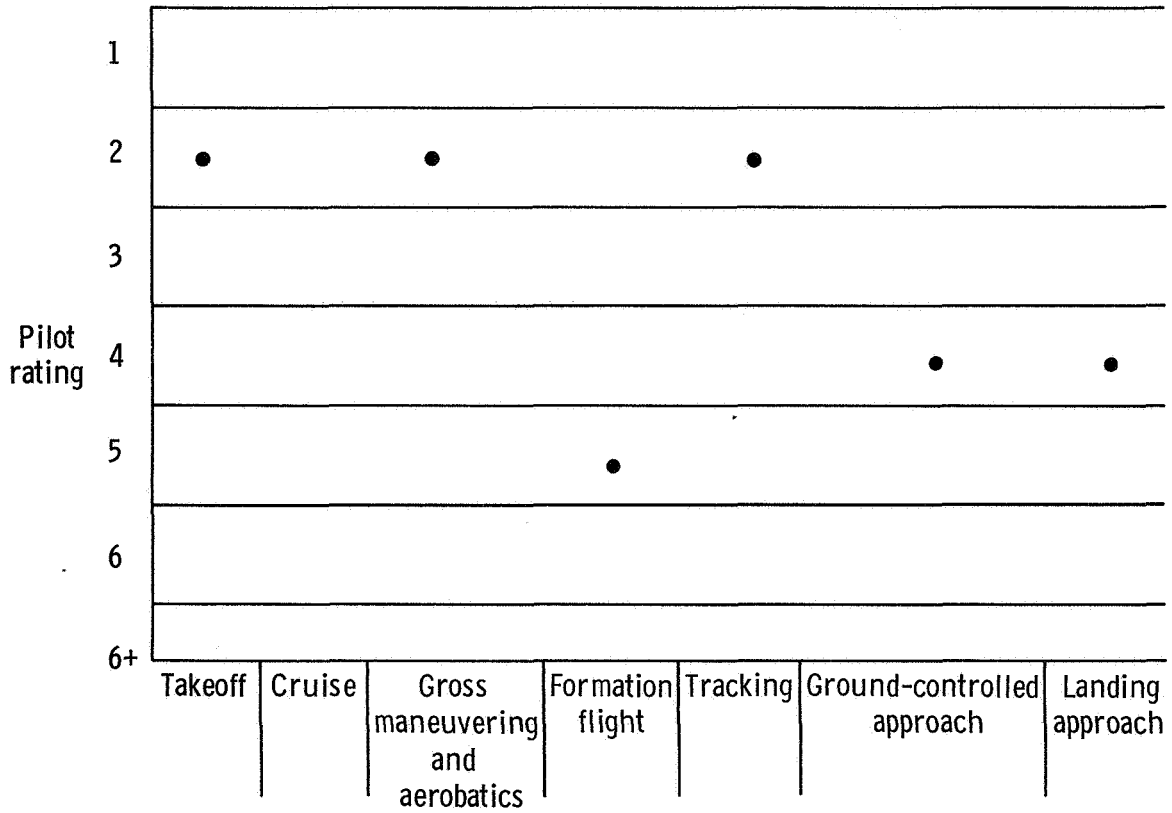
Figure 9. Side stick pilot ratings in roll.

SESSION III

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SPACE SHUTTLE DIGITAL FLIGHT CONTROL SYSTEM

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and

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SUMMARY

The Space Shuttle digital, fly-by-wire, flight control system (FCS) presents an interesting challenge in avionics system design. In residence in each of four redundant general purpose computers (GPC's) at lift-off are the guidance, navigation, and control algorithms for the entire flight. (A fifth GPC houses a backup FCS.) The mission is divided into several flight segments: first-stage ascent, second-stage ascent; abort to launch site, abort once around; on-orbit operations, entry, terminal area energy management (TAEM); and approach and landing. The FCS is complicated in that it must perform the functions to fly the Shuttle as a boost vehicle, as a spacecraft, as a reentry vehicle, and as a conventional aircraft. The crew is provided with both manual and automatic modes of operations in all flight phases including touchdown and rollout.

INTRODUCTION

The Shuttle vehicle configuration is shown in Figure 1. It consists of the orbiter vehicle, the orbiter external tank (ET), and two solid rocket boosters (SRB's). During Shuttle ascent, control authority is provided by thrust vector control (TVC) of the three orbiter main engines and each of the two SRB's. Orbit insertion and on-orbit control are accomplished by combinations of 46 reaction control jets plus two gimbaled orbit maneuvering engines (orbit maneuvering system or OMS). A blend of reaction control system (RCS) jets and the aerosurfaces is used during entry; all aerosurface control is used during TAEM and approach and landing. The aerosurfaces (Figure 2) include the elevons, used in unison for pitch control and differentially for roll control; rudder panels, used in unison for rudder control and differentially as a speed brake; and a body flap. Primarily the body flap protects the main engines from entry heating. However, it also supplements the elevons for pitch trim.

The orbiter is a first step in design of a control-configured vehicle. It is statically unstable in both pitch and yaw over a large percentage of the flight envelope (up to 2 and 1/2 percent of the body length in pitch). This design philosophy has permitted extensive weight (and hence cost) savings because it has allowed wing, tail, and aerosurface sizes to be minimized.

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DESCRIPTION OF THE SPACE SHUTTLE FLIGHT CONTROL PROBLEM

During Shuttle mated ascent, the FCS consists of a three-axis attitude command system (Figures 3 through 5). Five seconds after lift-off, commands are issued to accomplish the pitch-over and roll-to-flight-azimuth maneuvers. During regions of high dynamic pressure, a load relief system in both pitch and yaw minimizes air loads on the vehicle. The system is optimized with respect to weight savings (due to load reductions) versus weight penalties (due to added propellant caused by flight path dispersions arising from the use of the load relief system). The load relief function is accomplished by lateral and normal accelerometer feedbacks blended into the attitude command system starting at 25 seconds into the flight. After the region of high dynamic pressure passes, the load relief function is blended out (95 seconds). The guidance system commands an open loop pitch program versus time. The trajectory is shaped to minimize gimbal angle requirements and to balance the weight penalties associated with positive and negative air loads due to winds and gusts.

During SRB tail-off, which is sensed as an acceleration decay, the system is commanded to fly a pitch program versus time for proper SRB separation conditions. At staging, the control system is switched to the second-stage mechanization, which is a standard three-axis attitude command system (Figure 6). At a given time, which corresponds to a predicted dynamic pressure of 25 psf, the guidance loop is closed, and a form of linear tangent steering is used to guide the vehicle to the orbit insertion point. In addition to the automatic modes described, an augmented manual capability is provided in both first and second stages of flight.

The abort modes are not discussed in this paper.

In the on-orbit flight phase the crew is provided with the 13 manual and automatic control modes listed in Table 1. Two gimbaled, 6000-pound-thrust (OMS) engines are used for large delta-V maneuvers. Various combinations of forty 900-pound-thrust reaction jets are used for attitude control and small delta-V translation maneuvers. In addition, six 25-pound-thrust RCS jets are provided for high accuracy vehicle pointing. The jet select logic provides attitude and translational capability with less than 7 and 1/2 percent cross coupling into adjacent axes. The number of jets is predicated upon the requirement for a fail operational, fail safe capability throughout a mission.

The entry flight control system (Figures 7 through 9) is a blend of RCS and aerosurface control effectors. During the early portions of entry an all-RCS control system is used (Figure 10). When a dynamic pressure of 2 psf is reached (sensed from vehicle accelerations), the elevons are activated to provide a pitch and roll trim supplement to the RCS system. When a dynamic pressure of 10 psf is reached, the elevons provide sufficient authority for roll control, and the roll jets are inhibited. When a dynamic

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Table 1. On-Orbit Control Modes

Mode
Manual direct rotation acceleration command
Manual direct translation acceleration command
Manual direct translation pulse command
Manual direct rotation pulse command
Three-axis manual proportional rate command augmentation
Manual RCS rotation discrete rate command augmentation
Three-axis attitude hold mode
Three-axis automatic attitude command
Automatic RCS local-vertical barbecue attitude command
Three-axis automatic inertial barbecue command
Two-axis automatic RCS translation command
Automatic OMS thrust-vector control
Manual OMS thrust vector control command augmentation

pressure of 20 psf is reached, the pitch jets are inhibited, the yaw jets being retained for yaw stabilization and control. During the majority of entry the vehicle is statically unstable in yaw. However, the stick-fixed dutch roll mode is dynamically stable. The control system takes advantage of this stability in that the vehicle is permitted to oscillate within course dead bands in roll and yaw, thus avoiding an excess usage of RCS for yaw stabilization.

The heating rates and total heating load to the vehicle are minimized by flying the high-speed portion of entry (down to 8000 feet per second) at high angles of attack (approximately 30 degrees). At Mach 8, an angle-of-attack transition is initiated, ending at an angle of attack of approximately 10 degrees (roughly the maximum lift-to-drag condition) and a velocity of approximately 1500 feet per second. During this transition, the vertical tail and rudder become effective. At an angle of attack of 18 degrees (Mach 5) the rudder control is activated. By the time the vehicle reaches an angle of attack of 10 degrees the rudder is fully effective, and the yaw jets are thereafter inhibited. The FCS is switched to conventional aircraft control mode for the TAEM phase of flight. Manual and automatic modes during entry are similar, the only difference being the substitution of a guidance steering command in the auto system instead of the rotation hand controller output in the manual system.

The TAEM flight phase is initiated at a velocity of about 1500 feet per second during entry with a corresponding altitude of approximately 70,000 feet. This flight phase extends to the approach and landing interface at approximately 10,000 feet. During this period, the guidance system issues commands to control the dynamic pressure and energy state of the vehicle and to provide steering commands to arrive at the approach and landing interface in alignment with the runway (Figure 11). Three basic control modes are

provided to the crew: manual direct (MD), control stick steering (CSS), and automatic. The manual direct mode (Figures 12 through 14) is strictly a backup in which no augmentation is used (i.e., all feedback loops are open).

When the crew selects control stick steering, the basic mode of operation becomes a command augmentation system (CAS). It is implemented as an N_z command (normal load factor) mechanization in pitch (yaw is similar), and roll rate is commanded into the roll channel (Figures 15 through 19). Two submodes to CSS are available. One is attitude hold in pitch and/or roll (Figures 20 and 21). When the stick is out of detent, the CAS mode is operational; when the stick is returned to detent, the attitude function is initiated at the attitude existing at the time the stick was returned to detent. A second submode to CSS is an indicated air speed (IAS) hold (Figure 22). In this mode, the speed brakes are commanded to maintain the air speed commanded by the crew. When the IAS is not selected, speed brake control is a manual function.

In the auto-TAEM mode, N_z commands are issued from the guidance system to the pitch and yaw channel, and roll commands are issued to the roll channel. This is shown in Figure 23 for the pitch axis.

After the vehicle exits blackout during entry, a TACAN (tactical air navigation) navigation aid is acquired by the communication system for navigation update. The guidance system steers the vehicle to intercept a heading alignment circle to bring the vehicle to the approach and landing interface. As the vehicle rounds the heading alignment circle (Figure 24), its orientation becomes such that the antennas capture a microwave scan beam landing system (MSBLS) navigation aid. This will occur at an altitude of roughly 14,000 feet. When lock-on is verified, the flight phase switches from TAEM to the approach and landing. The same three basic modes are available to the crew as discussed for TAEM (i.e., manual direct, CSS, and auto). The manual direct and CSS modes are as described for the TAEM phase. In the autoland mode (Figure 18), the guidance system issues attitude commands to the vehicle to fly down a steep glide slope, which varies from 21 to 24 degrees depending upon the payload weight (Figure 25). The speed brakes are modulated to hold an air speed of 290 knots. At an altitude of 1800 to 2000 feet, depending on payload weight, a preflare maneuver is commanded to bring the vehicle exponentially to a 3-degree glide slope. A final flare is commanded at approximately 200 feet altitude, and the vehicle nominally lands with a sink rate of about 2 and 1/2 feet per second and about 4000 feet down the runway. When main gear touchdown is detected (by a squat switch), the normal and lateral acceleration feedbacks (in CSS) and integrator loops (in auto) are opened, and a pitchdown command is issued. Roll commands are driven to zero. Lateral steering is initially accomplished with the rudder. After the nose gear slapdown has been verified, and after the velocity is reduced to approximately 110 knots, nose wheel steering is engaged. The rest of the rollout is accomplished with the rudder and nose wheel steering. The autoland function is totally automatic with the exception of gear extension (h = 500 ft) and runway braking, which are done manually.

DESCRIPTION OF THE SPACE SHUTTLE FLIGHT CONTROL SYSTEM MECHANIZATION

The flight control problem just described is essentially controlling a large number of quite different flight phases, some of which include unstable vehicle dynamics. Thus, flight control is a flight safety-critical function that must have great flexibility. The concept chosen for flight control is an all-digital, fly-by-wire implementation that uses several general-purpose computers connected by serial digital data buses to remotely located multiplexer/demultiplexer units (MDM's). The MDM's in turn, are connected to the flight control sensors, effectors, and controls. The guidance and navigation problems are solved by this same mechanization (with the appropriate additional sensors). It is used for all flight phases and elements, including control of the SRB's during ascent. The block diagram of this configuration is shown in Figure 26.

Efficiency of presentation requires that the computer complex be described first, including the MDM's and data buses. Then the operating configuration of the flight control equipment will be described.

Figure 27 illustrates the internal configuration of the computer and associated elements of the central digital elements (collectively denoted as the digital processing subsystem or DPS). At the core of the DPS are five general purpose computers. Each GPC is a modified IBM AP-101 central processor unit and core memory with a special input/output processor (IOP) that interfaces with 27 serial digital data buses. The memory contains 64,000 32-bit words with a nominal one-microsecond cycle time. The IOP contains a master sequencer and 27 data bus control elements. Under overall control of the master sequencer, each data bus control element has the capability to send and receive data over its particular data bus. In addition to data, the transmittals include commands to other equipment connected to the bus. In addition to the ability to request and subsequently receive data on the bus, each data bus control element can monitor data on the bus resulting from other data bus control elements (associated with other GPC's). This monitoring capability is fundamental to the processing of flight control sensor data, as will be described.

Of all the data buses, those central to this discussion are the eight dedicated to guidance, navigation, and flight control and the five intercomputer data buses. Each of these buses is connected to all of the GPC's. Also, those eight buses dedicated to the guidance, navigation, and flight control functions are connected to four MDM's located in the forward end of the vehicle, another four located in the aft end of the vehicle, and various devices to interface with controls, displays, event controllers, and the main engines. Each flight control sensor and effector is connected to one of the aforementioned eight MDM's, and communication between all flight control elements is via these buses.

Data transmittal over this bus network is by time division multiplex techniques at a one-megabit data rate; each word is 28 bits with the first

three bits used for synchronization and distinguishing between command and data formats. The next five bits identify the address of the word destination or source, as appropriate. The rest of the word is devoted to command or data information, except for the last bit, which is a parity bit. Each bus operates in a half-duplex mode.

The function of each MDM is to interface between the serial data streams on the bus and the several elements connected to it. The interfaces between the MDM and the several elements may be analog, digital, or discrete and may generally be in either direction. Several hundred elements can typically be connected to the MDM, the exact number being dependent upon the specific mix of analog, digital, and discrete interfaces.

Data for use in a GPC are obtained by a request (under GPC software control) being issued through the IOP, over a data bus, to a specific MDM (or other interfacing element), and then to the particular device. The reply (usually data) follows the reverse path. Because of the monitoring capability of each data bus control element, each GPC can receive the data, even though only one requested it. This feature is used to advantage in the flight control system, as will be described. The intercomputer buses are used for exchange of data between GPC's.

The five computers are synchronized only at each minor cycle (40 milliseconds) and are then only synchronized close enough to ensure "sequence synchronization" between machines; i.e., all machines are on the same minor cycle except for a small interval near the beginning or end of a minor cycle.

With this summary description of the central digital processing subsystem, it is now possible to describe the mechanization of the flight control system. The system uses both rate gyros and lateral accelerometers as basic stabilization sensors; inertial measuring unit (IMU) gimbal data are used for certain attitude hold and attitude command modes. Three sets of rate gyros are used, two on the SRB's and one set on the orbiter. Each set consists of three axes, and each axis is triply redundant. There are two sets of lateral accelerometers, one forward and one aft on the orbiter. Each set senses the two axes orthogonal to the vehicle roll axis, and each axis is triply redundant. Each redundant instrument in a set is connected to a different MDM. Controls are generally triply redundant within each set, and most controls are duplicated between the left and right seats in the cockpit.

The effectors vary for each flight phase. Of primary concern are the effectors used during the so-called "critical flight phases," those during which a hard-over failure of an axis would lead to vehicle loss before the fault could be rectified by crew actions. Most of the nonorbital portions of flight, both ascent and return, fall into this category. The flight control system must be designed to tolerate any two failures and still permit safe vehicle and crew recovery. The flight control effectors used during these flight periods have multiple input ports at the hydraulic secondary-valve level. These multiple inputs are normally used in a force-fight mode. The multiple inputs are also compared, and any deviation of

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one input from the others of a specified amount for a specified time results in that input channel being "kicked out" by the actuator itself. These "pseudo-voting" actuators protect against two different input failures. Upon occurrence of a first failure, the action is the same for both the ascent and return flight phases: the bad channel is removed, and operation continues with the remaining channels. Upon occurrence of a second failure, the response varies according to the flight phase. The thrust vector control actuators used during ascent have only three input ports, and, upon occurrence of a second failure, the actuator simply centers that axis on that engine. Because there are multiple thrusters (three orbiter main engines and two SRB's), loss of thrust vector control in one axis is not flight critical. The mission can continue after the loss of thrust vector control from one axis. During the return portions of flight, things are different. There is essentially only one of each basic flight control surface, and centering of a surface would generally lead to loss of the vehicle. Consequently, the flight control effectors during these portions of flight have four input channels. After the first failure, operation continues with the remaining three. After failure of the second, failure detection and isolation of the bad input channel is easily detected by conventional comparison techniques, and operation continues with the remaining two channels.

During these critical flight phases, the flight control system is configured as shown in Figure 28. At the beginning of each flight control computational cycle (denoted as a minor cycle and equal to 40 milliseconds), the ith GPC requests data from the ith sensor. Because the involved data buses are connected to all GPC's, each GPC receives the data from the ith sensor, either directly (in the case of the ith GPC) or via the monitoring capabilities of the other GPC's. GPC's 1 through 3 are involved in data requests; GPC's receive data nominally only through the monitoring of bus traffic, but can assume the data request function of a failed GPC. The result is that all GPC's have all of the sensor data.

Each computer selects a single set of data for use in the flight control computations. Until an instrument failure is detected, the computer simply selects the middle value of the three data values for each measurement. Upon detection and isolation of an instrument failure (by combined hardware and software tests), the computer simply averages the data from the remaining two instruments. Upon detection and isolation of a second failure (again by a combination of hardware and software tests), the computer uses the data from the remaining good sensors. Since all computers have the same input data and use the same data selection algorithms, all computers use the same specific sensed values in the flight control computations.

On the downstream side of the computer, each computer is assigned to a specific input port of each flight control effector. Thus, if a computer should fail, the effect at the system level is simply a small transient because of the momentary incorrect force fight within the actuator. If such a failure occurs during ascent, GPC 4 assumes the role of the failed machine, and the failure tolerance of the system is returned to the same as it had at

launch. During entry through landing, four machines are nominally connected to the four input ports of each control effector. However, during these return phases, no GPC is brought in to replace a failed machine, since the ability to tolerate two failures exists without such replacement.

Because all GPC's are connected to all buses, and thus to all MDM's, it would be possible to operate the flight control equipment in a "master-slave" configuration; that is, one GPC would command all actuator ports until detection of its failure, at which time another GPC would assume command. This concept was rejected because of difficulty in proving that the master GPC would be made to relinquish control in all possible failure conditions.

A difficulty of operating flight control equipment in the parallel string configuration just described is possible divergence between the parallel control channels. This problem manifests itself by the commanded control signal from each computer differing from those issued by other computers by ever increasing amounts. In the Shuttle configuration, this would quickly cause various control channels to "kick out," even though no failures had occurred. The problem is caused by "noise" getting into the control channel that contains integrators. The noise can be from traditional amplitude variations (which could result from different channels using different sensors) or it could be a result of timing differences between channels. With "noise" and integrators in the control channels (almost always present for reasons of stiffening control loops or providing automatic vehicle trim capabilities), the problem will occur even when the loops are closed by a common set of vehicle dynamics.

There are two basic ways of solving this problem. One is to provide appropriate channel coupling to stabilize the divergence; the second is to suppress data and timing variations between channels. For the Space Shuttle, the latter has been chosen. As mentioned, common sensor data are selected by identical algorithms operating on identical data sets in all GPC's, and the GPC's are sequence-synchronized to prohibit divergence from timing variations.

During noncritical flight phases, mostly on orbit, the flight control is operated as an active-standby system, with one GPC in entire control and a second available for takeover should the first fail. The effectors on the OMS engines are also operated in an active-standby mode. RCS jets are operated with somewhat conventional jet-select logic.

During all flight phases, the fundamental flight control computational cycle is 40 milliseconds. A number of flight control computations take place only every 80 milliseconds, and some take place approximately once per second. In addition to the basic 40-millisecond minor cycle requirement, the total delay between sampling of the flight control sensors and transmittal of the resulting command to the effectors is constrained to be no greater than 20 milliseconds.

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Software is provided for a number of flight control modes, including manual direct (MD) (totally unaugmented), control stick steering (CSS) (significantly augmented), and automatic (guidance loops closed). On-orbit operations entail many more modes. Details of the digital control laws for the multiple flight phases and multiple flight control modes within each flight phase cannot be discussed within the space limitations of this paper.

CONCLUDING REMARKS

As stated, the Space Shuttle digital fly-by-wire, flight control system is a challenge in avionics system design. It is not because of the sophistication of the algorithms being implemented; rather it is because of the number of flight phases requiring completely different control algorithms and control effectors, the large number of control modes, both manual and automatic, and because of the sophisticated and complex techniques required for management of redundant systems. It is one of the most interesting development programs ever undertaken and because of that is one of the most rewarding.

SYMBOLS

α angle of attack

β sideslip angle

γ flight path angle

θ pitch attitude

θ_c pitch attitude command from guidance

h altitude

K_t time variable gain

K_y lateral acceleration gain

K_z normal acceleration gain

N_z normal load factor

N_y lateral load factor

P roll rate

q pitch rate

$\bar{q}$ dynamic pressure

$\dot{\bar{q}}$ dynamic pressure rate

r yaw rate

R range

t time

Y cross range

ϕ roll attitude

ϕ_c roll attitude command from guidance

ψ heading angle

ψ_ϵ heading angle from computer
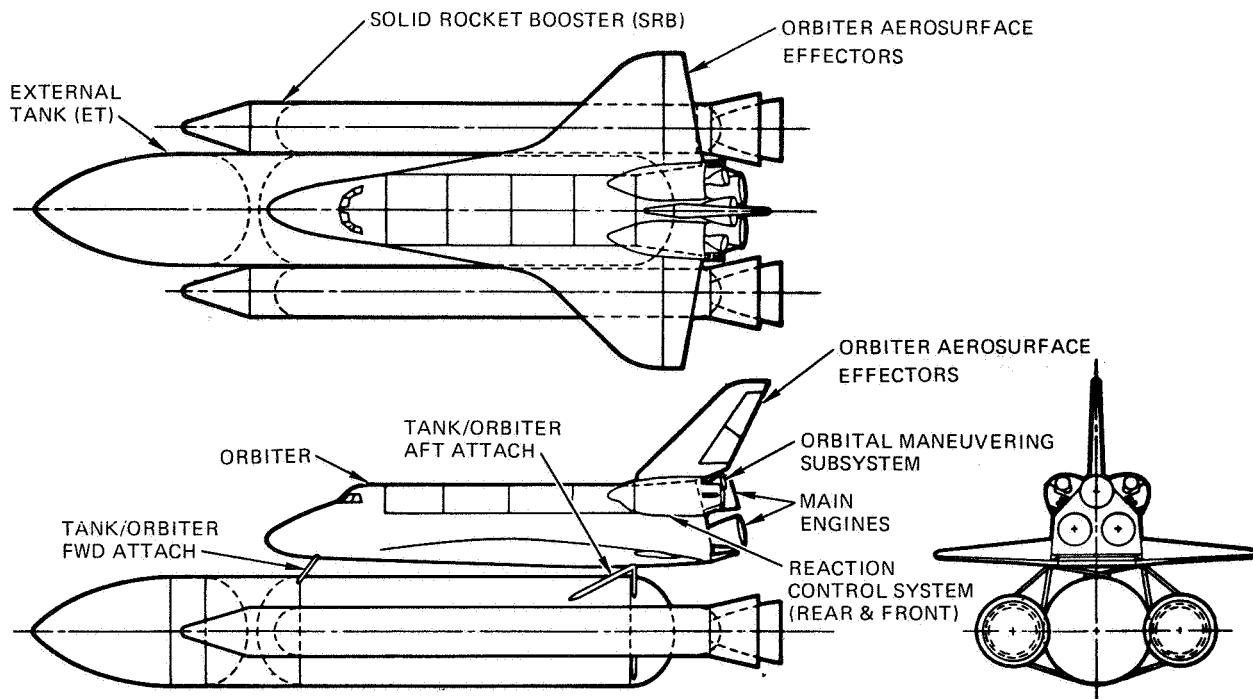
Figure 1. Shuttle Vehicle
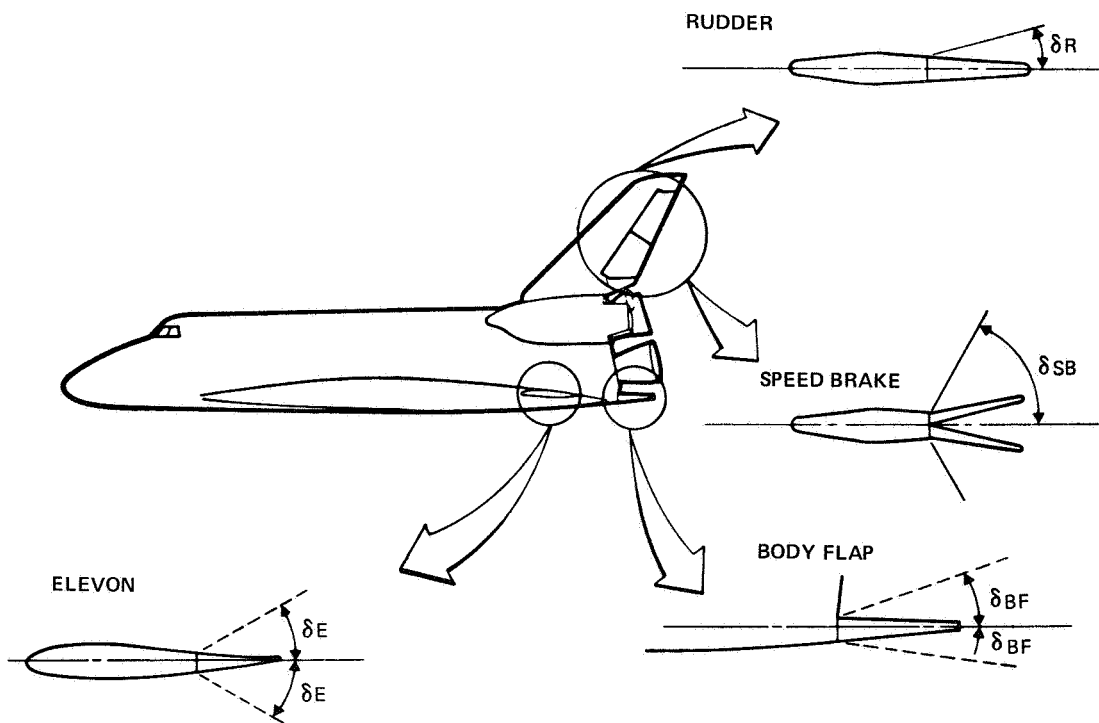
Figure 2. Aerosurface Configuration
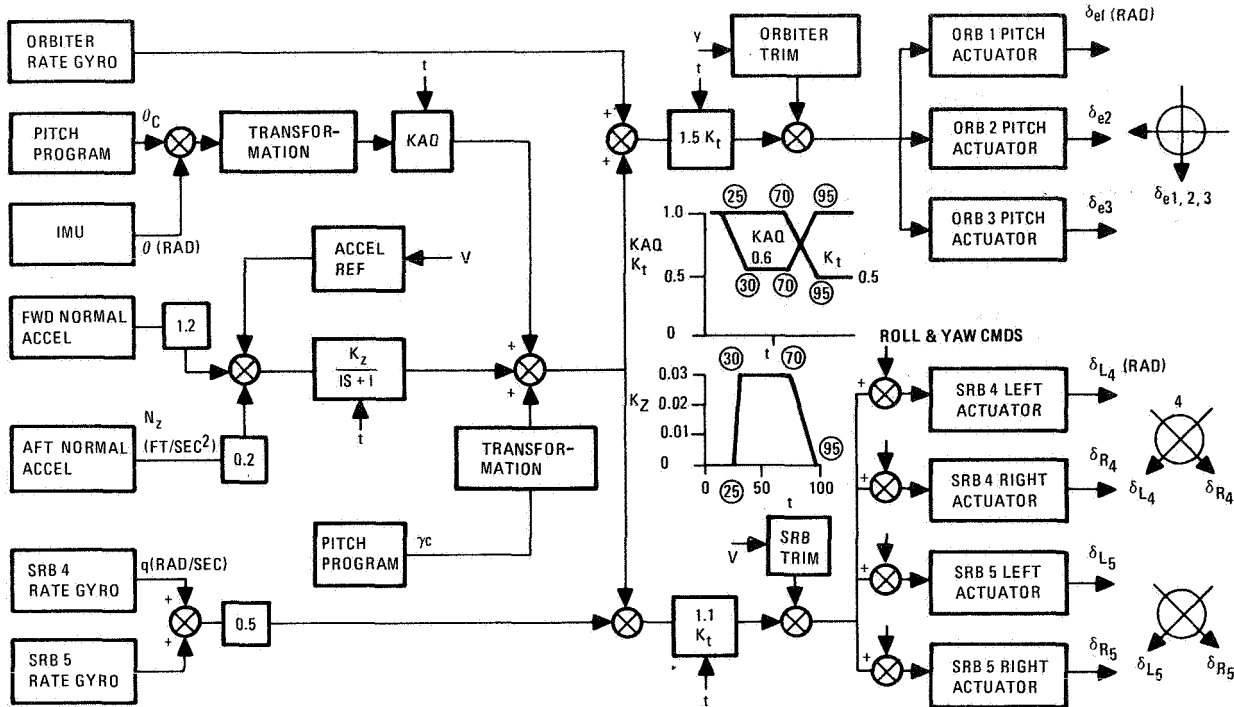
Figure 3. Mated-Ascent Pitch Axis Control
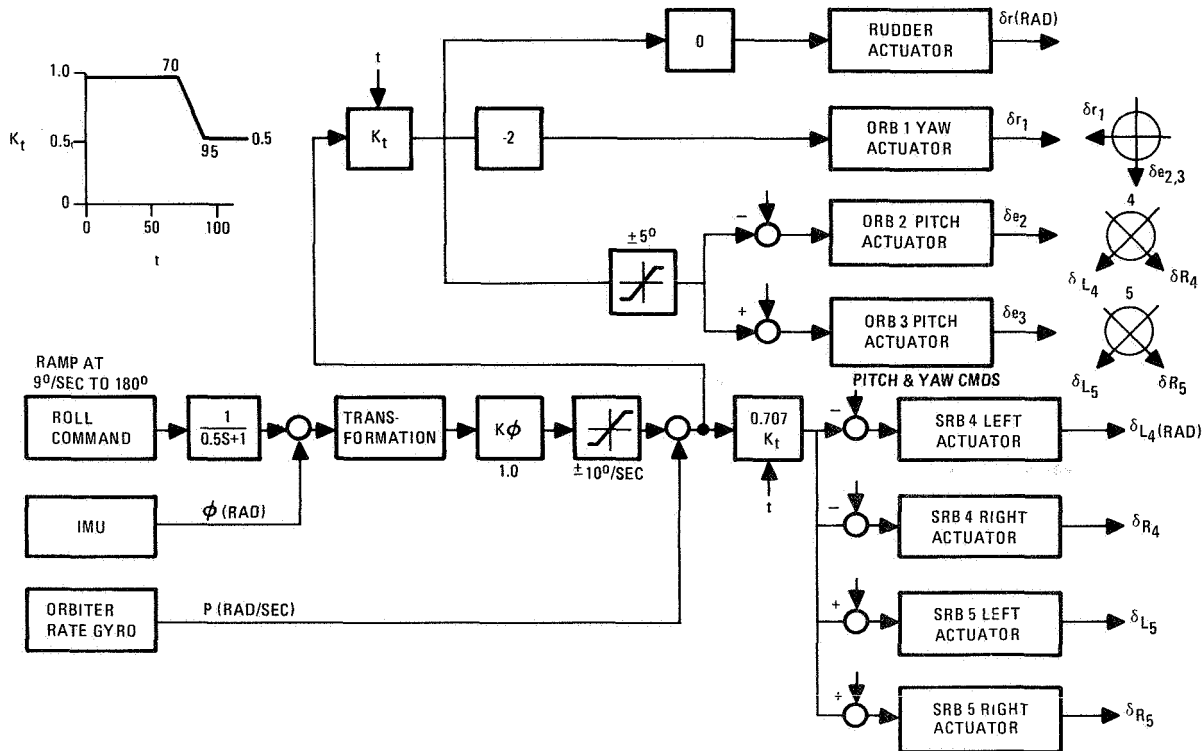
Figure 4. Mated-Ascent Roll Axis Control

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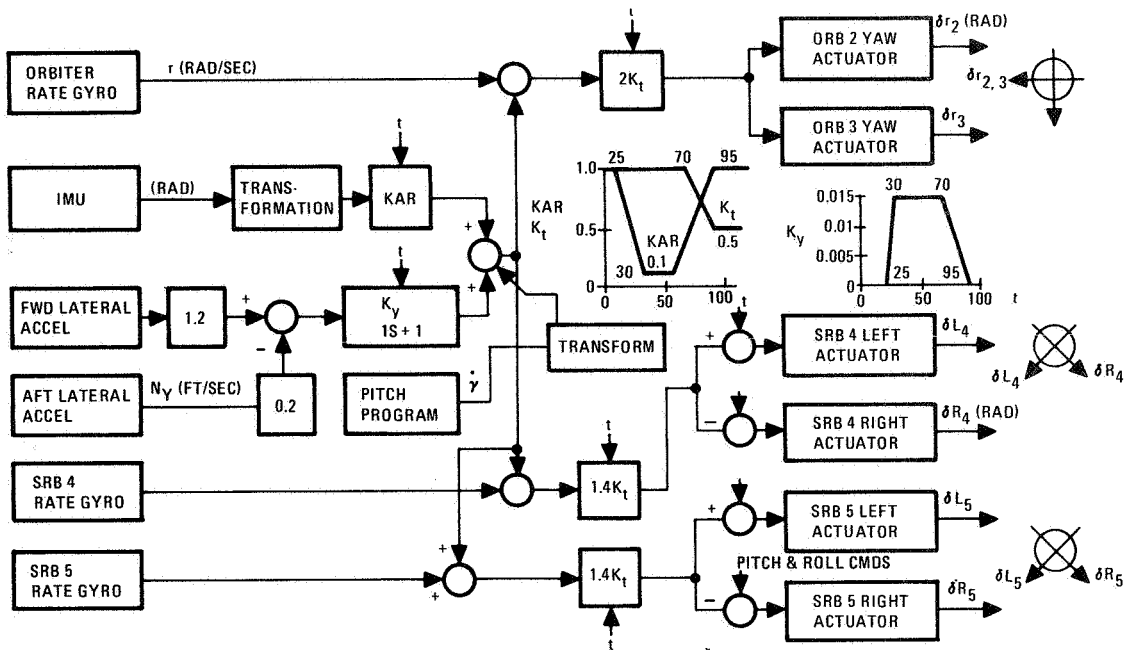
Figure 5. Mated-Ascent Yaw Axis Control
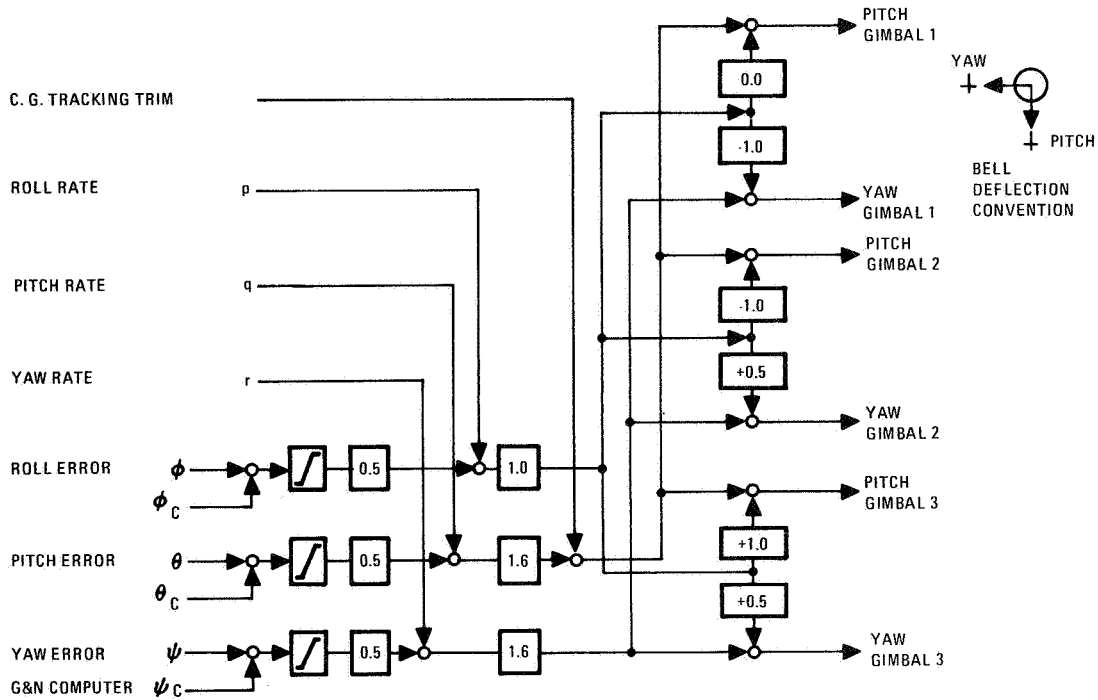
Figure 6. Second-Stage Flight Control Mechanization
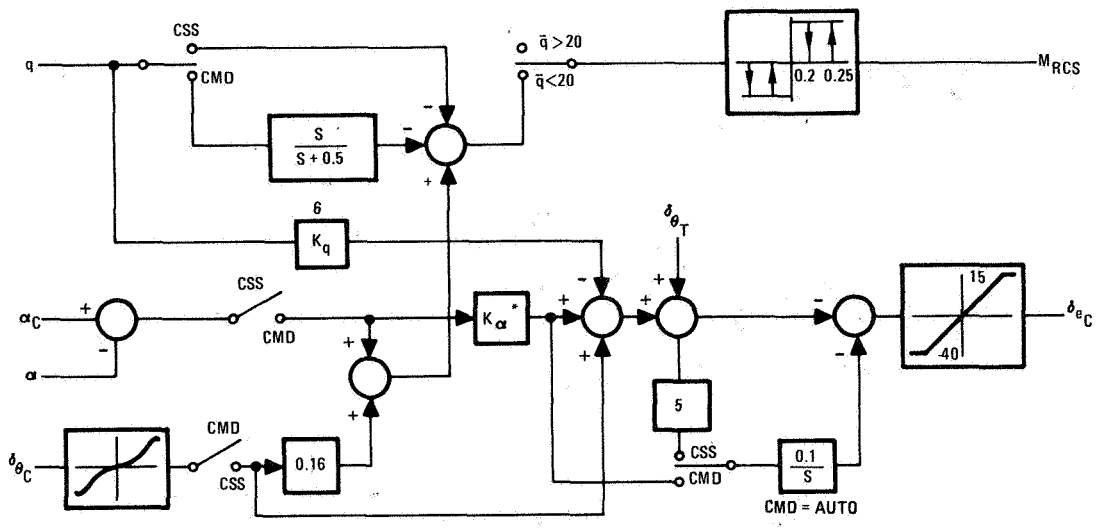
Figure 7. Entry Longitudinal Flight Control System
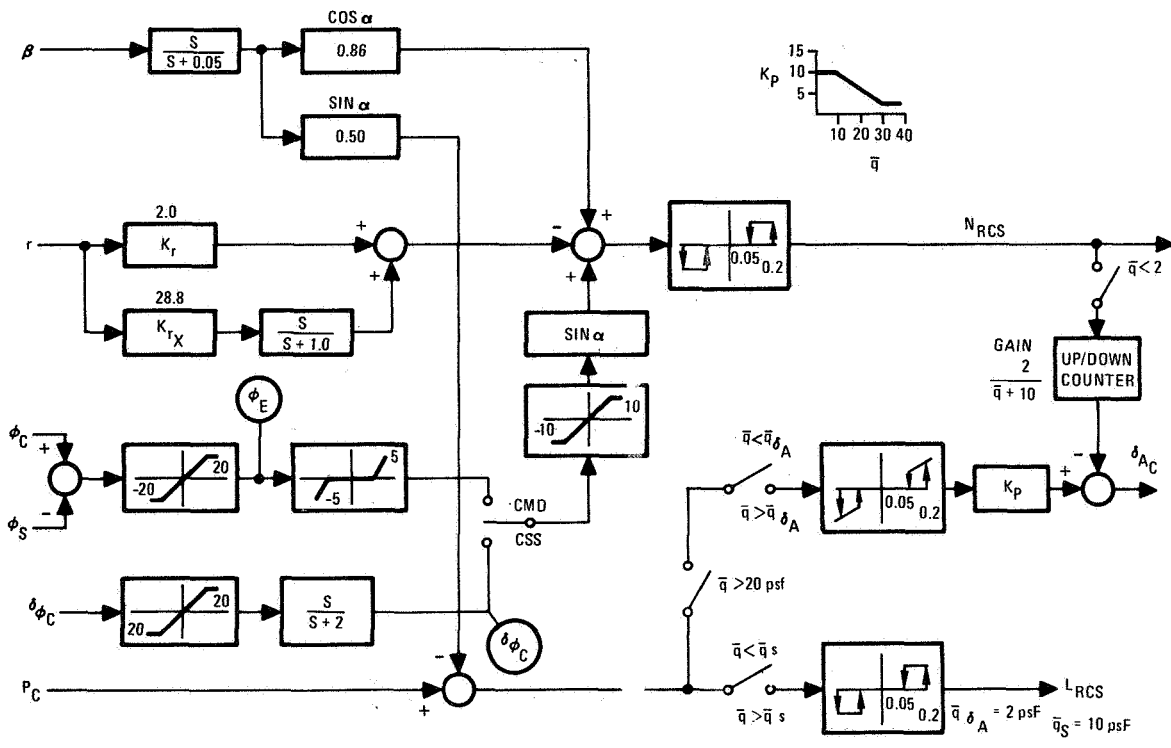
Figure 8. Early Entry Lateral Axis
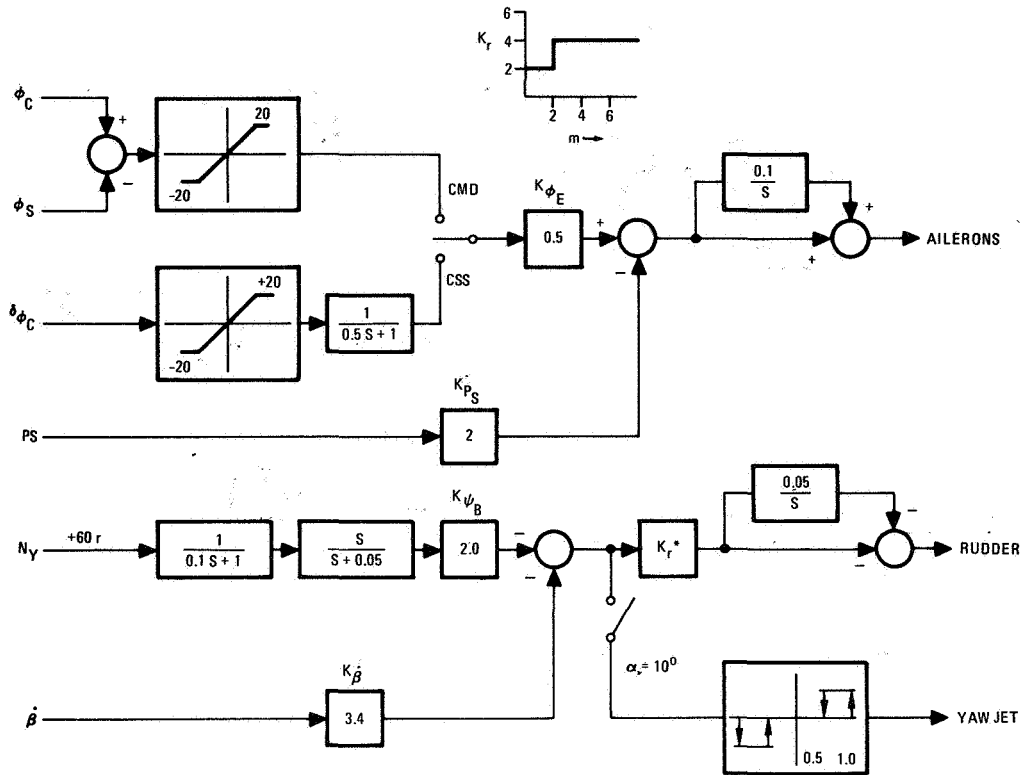
Figure 9. Final Entry Lateral Axis

Figure 10. Entry Profile

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TAEM GEOMETRY

HEADING
ALIGMENT
CIRCLE

20K FT

ENTRY/TAEM INTERFACE
● h = 70K FT
● q̄ = 150 PSF
● MACH = 1.5
● γ =-11°
● R = 34 NM

RUNWAY

FINAL APPROACH PLANE

TAEM/AUTOLAND INTERFACE
● h = 10K FT
● q̄ = 285 PSF
● γ =-21°
● R = 5 NM

ENERGY MANAGEMENT TECHNIQUE
(ENERGY GRADIENT METHOD)

DESIRED
PROFILE

ACTUAL
SLOPE

ENERGY
WEIGHT

DESIRED
SLOPE

2.5 NM

$$\dot{\delta}_{SBC} = K \left[\left(\frac{\Delta \frac{E}{W}}{\Delta R} \right)_D - \left(\frac{\Delta \frac{E}{W}}{\Delta R} \right)_A \right]$$

$$\bar{q}_c = f(\delta_{SBC})$$

RANGE

δ_{SBC}

$\bar{q}_c$

300

220

140

0   35   65   87

δ_{SBC}
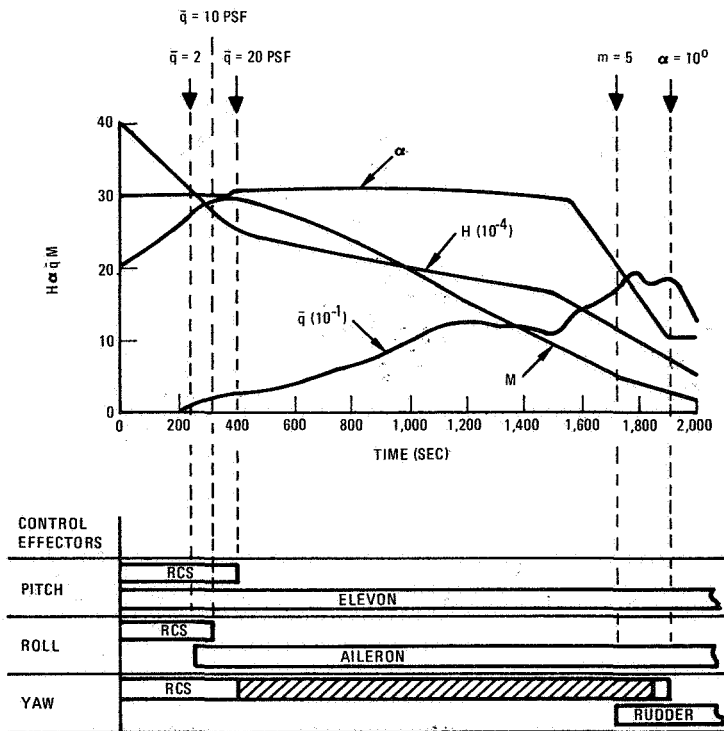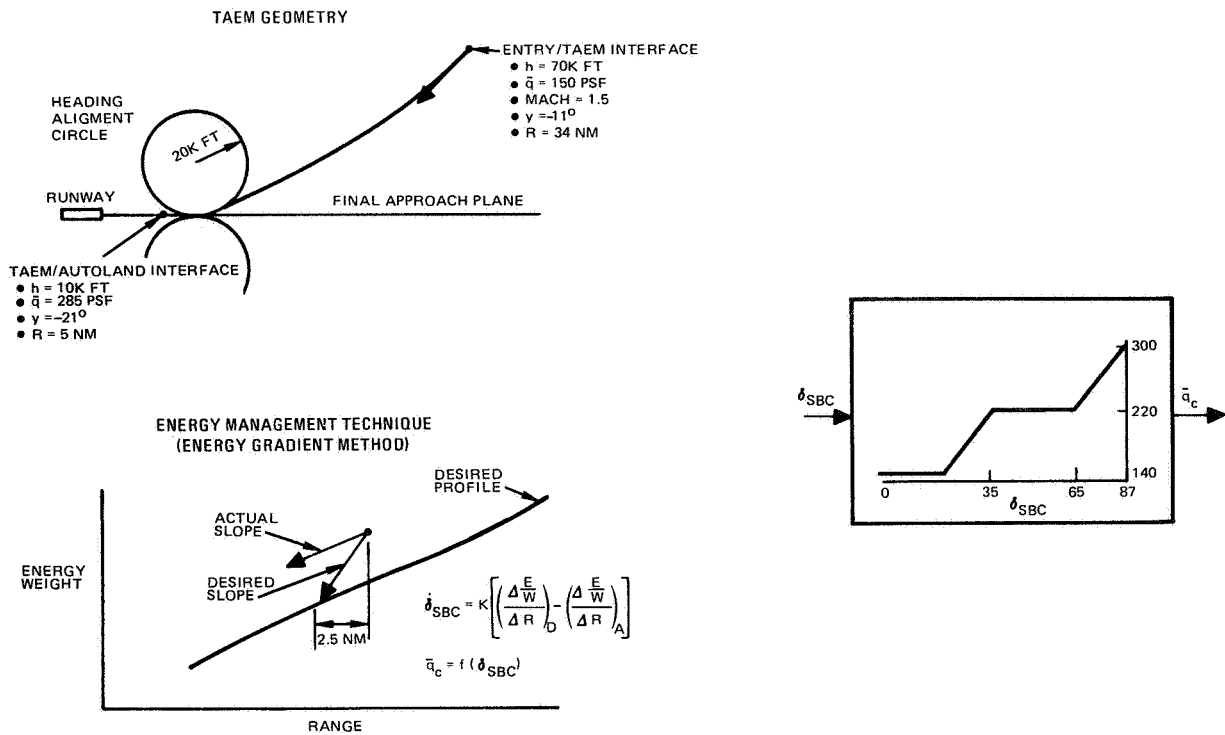
Figure 11. Terminal Area Energy Management (TAEM) Guidance System

RCH

BREAKOUTS
LEVEL DECTECTOR
QUADRATIC
SHAPING LIMITS

BEEP
TRIM

RECT
INTG

LIMITS

ACTUATOR

SPEED BRAKE
CROSS FEED
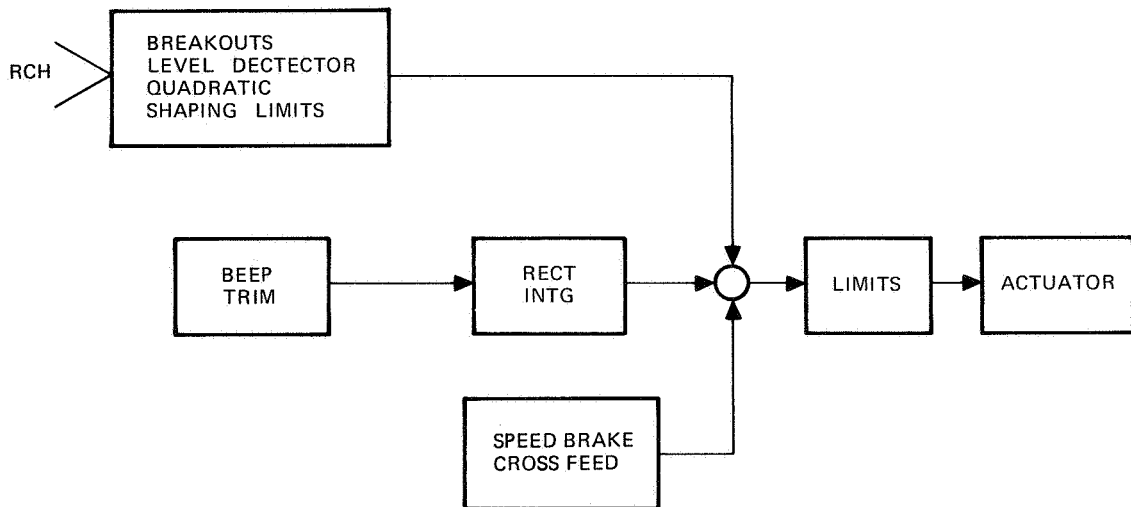
Figure 12. Pitch Axis—Manual Direct

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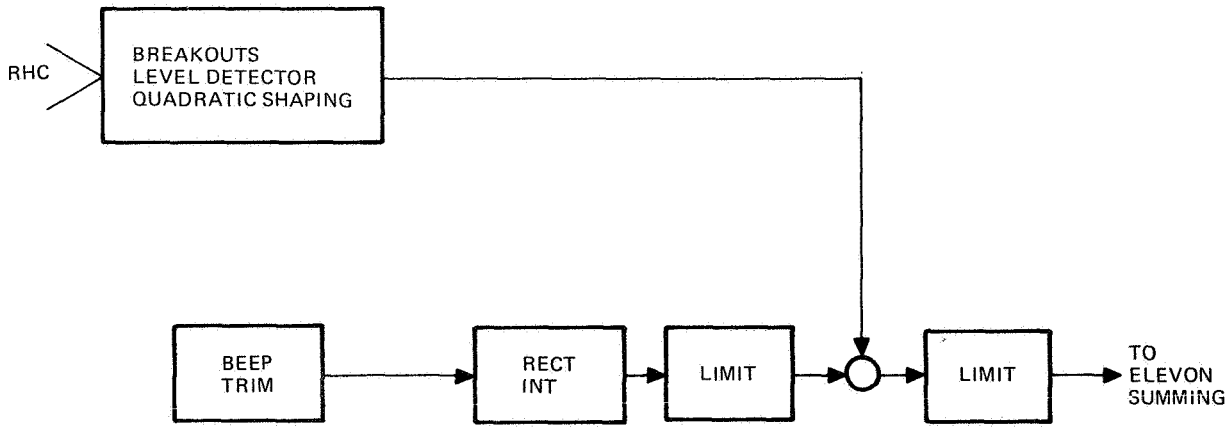
Figure 13. Roll Axis—Manual Direct
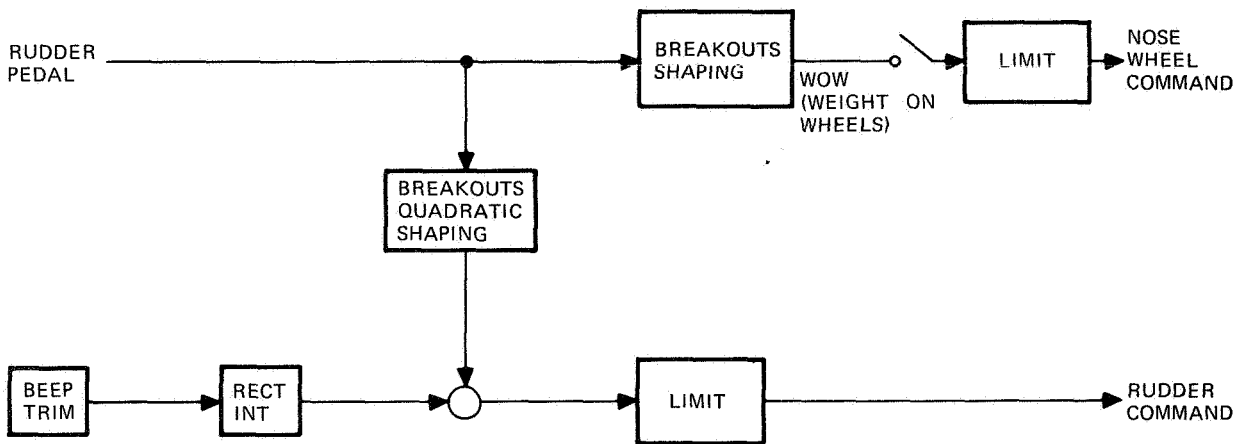
Figure 14. Yaw Axis—Manual Direct
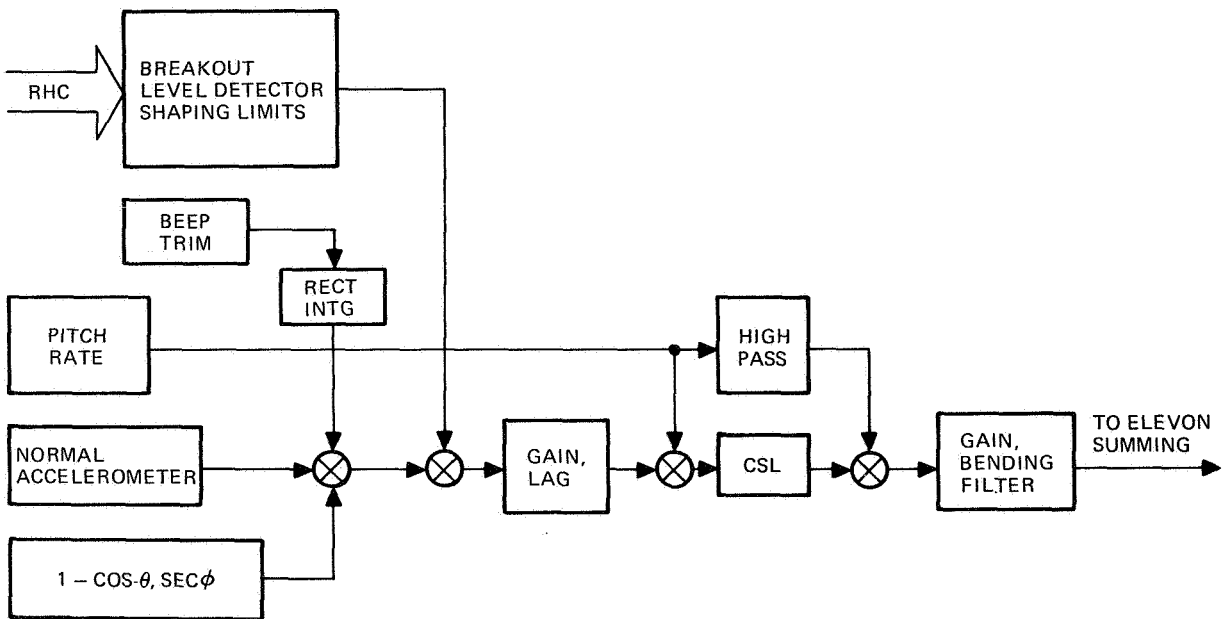
Figure 15. Pitch Command Augmentation

WOW = WEIGHT ON WHEELS
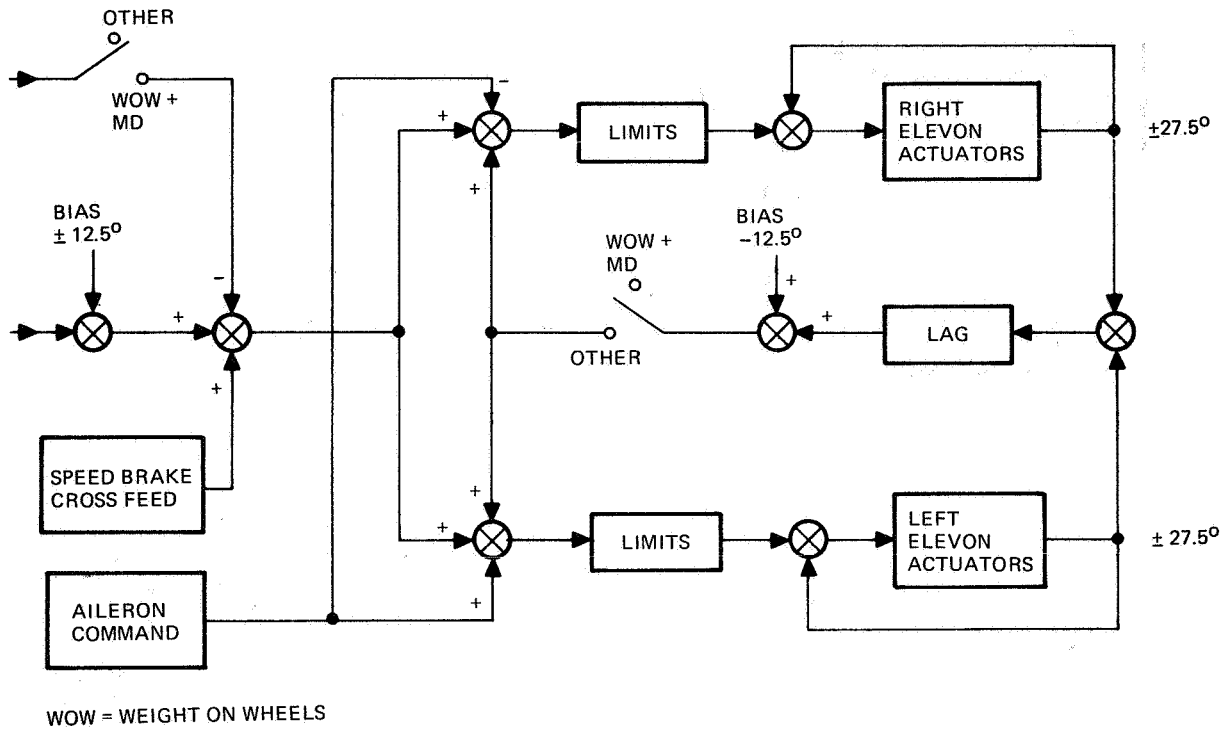
Figure 16. Elevon Summing
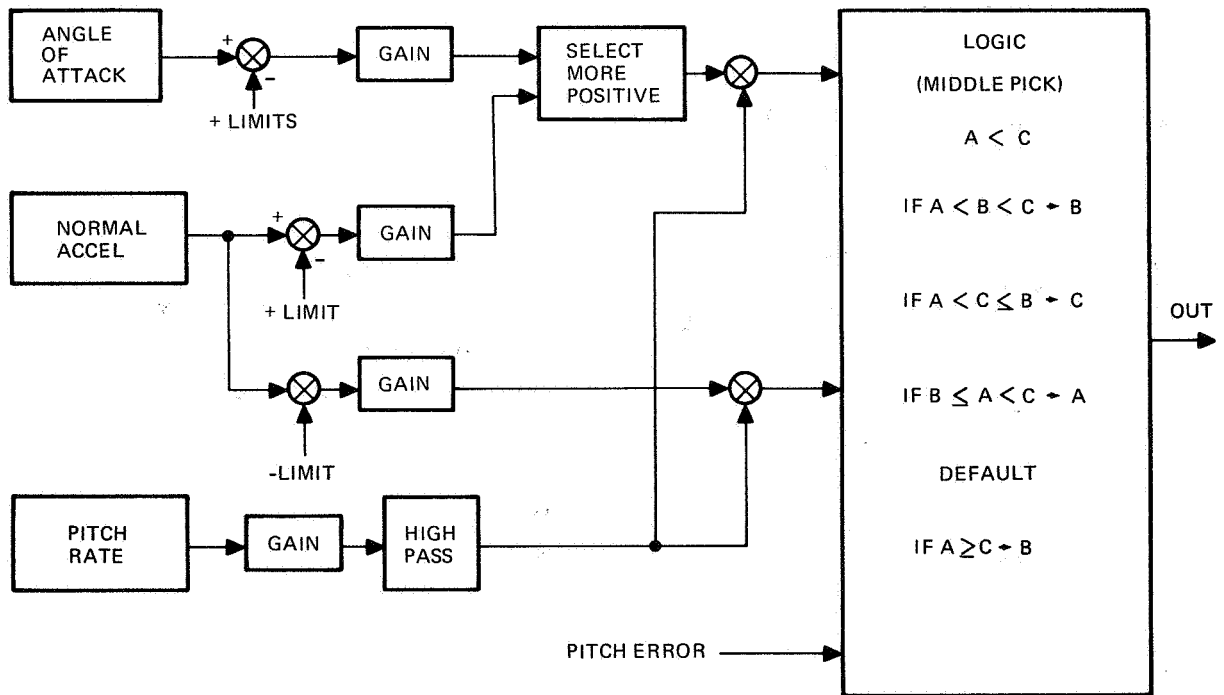
Figure 17. Command Signal Limiting
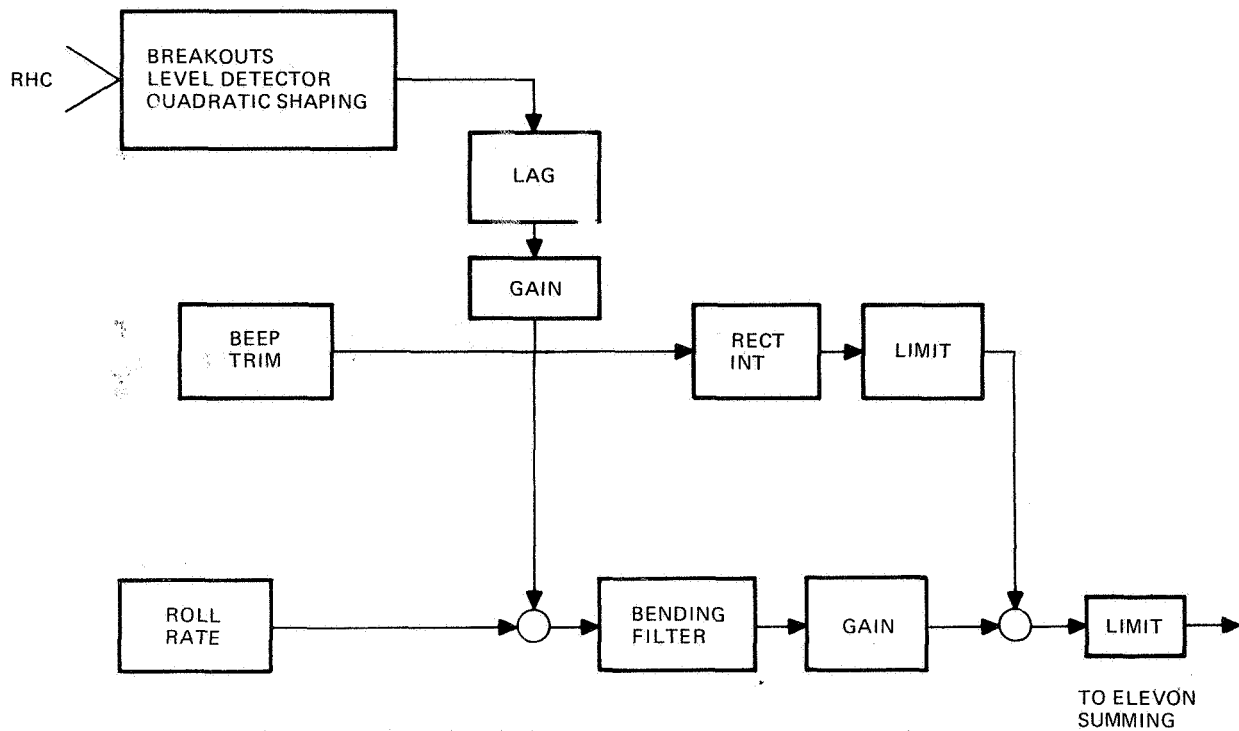
Figure 18.   Roll Axis—Computed Air Speed
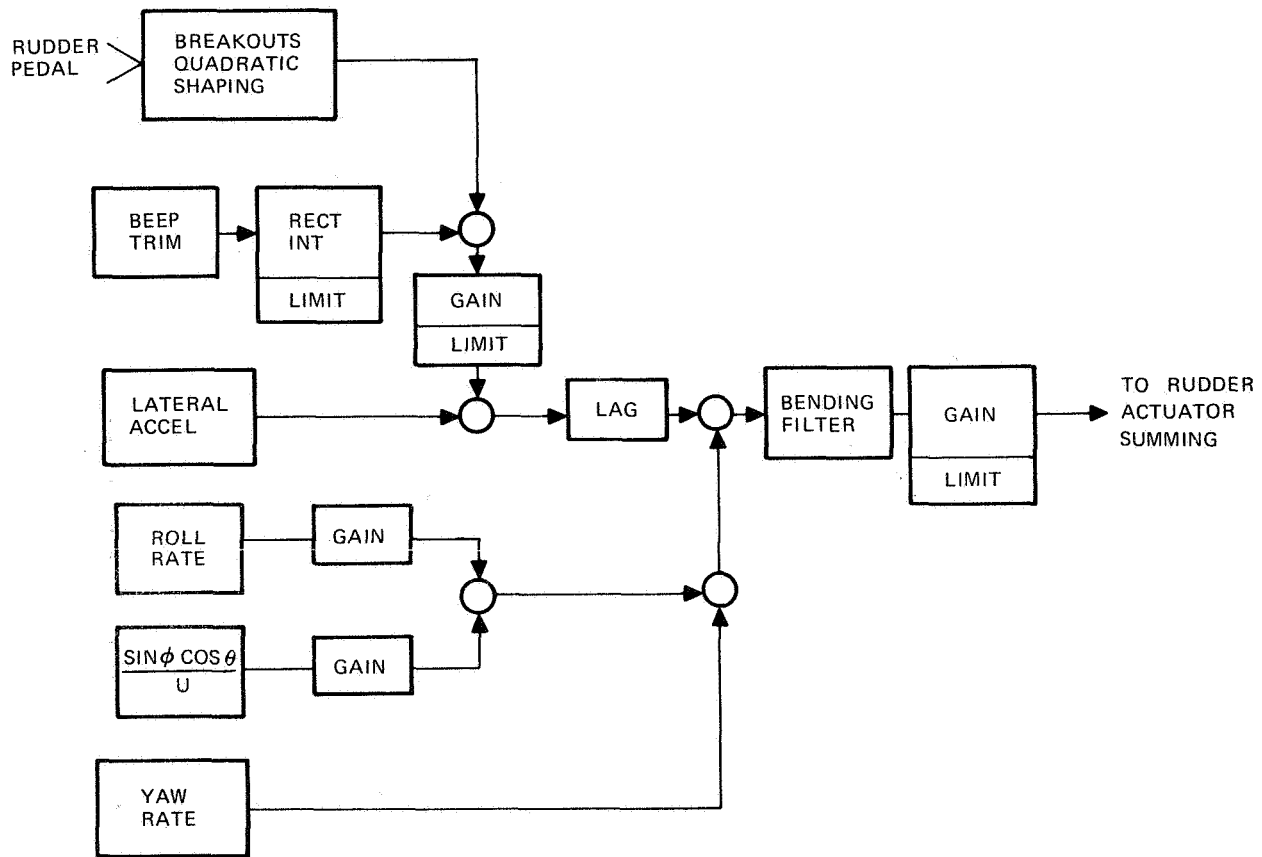
Figure 19.   Yaw Axis—Computed Air Speed
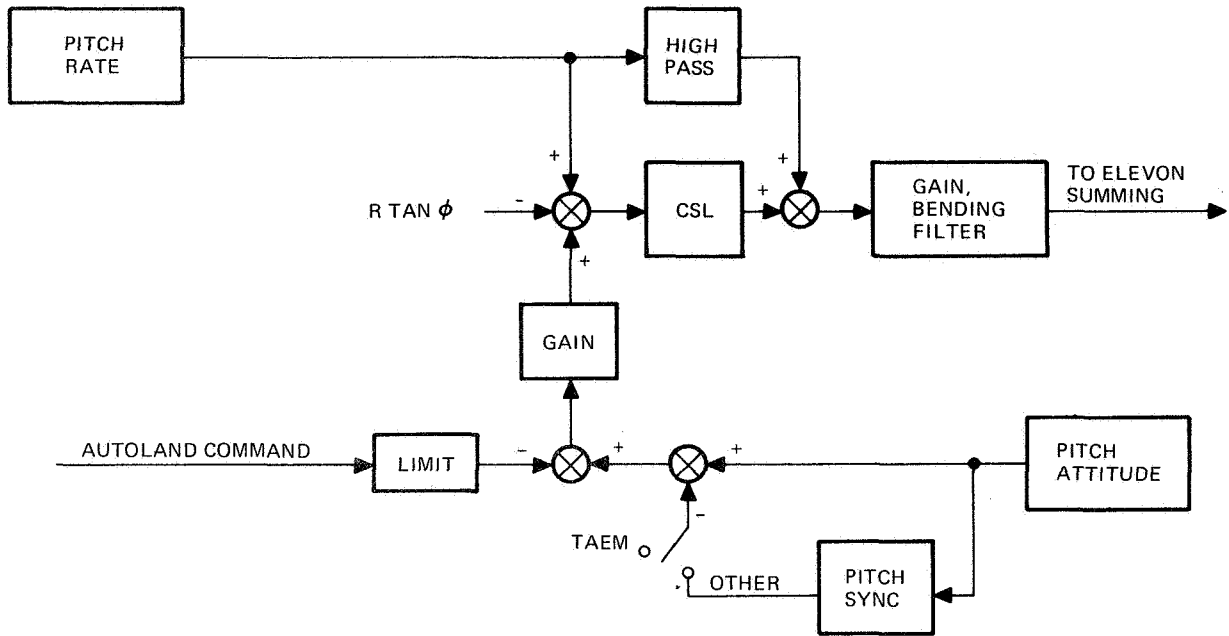
Figure 20.   Pitch Attitude Hold and Autoland
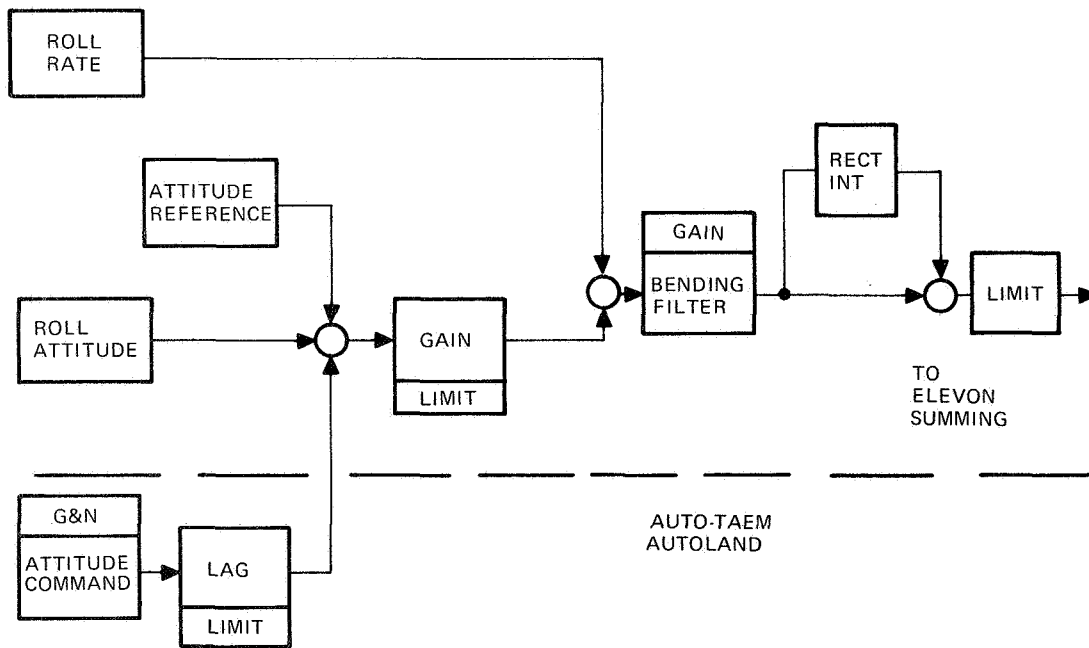
Figure 21.   Roll Attitude Hold

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C-4
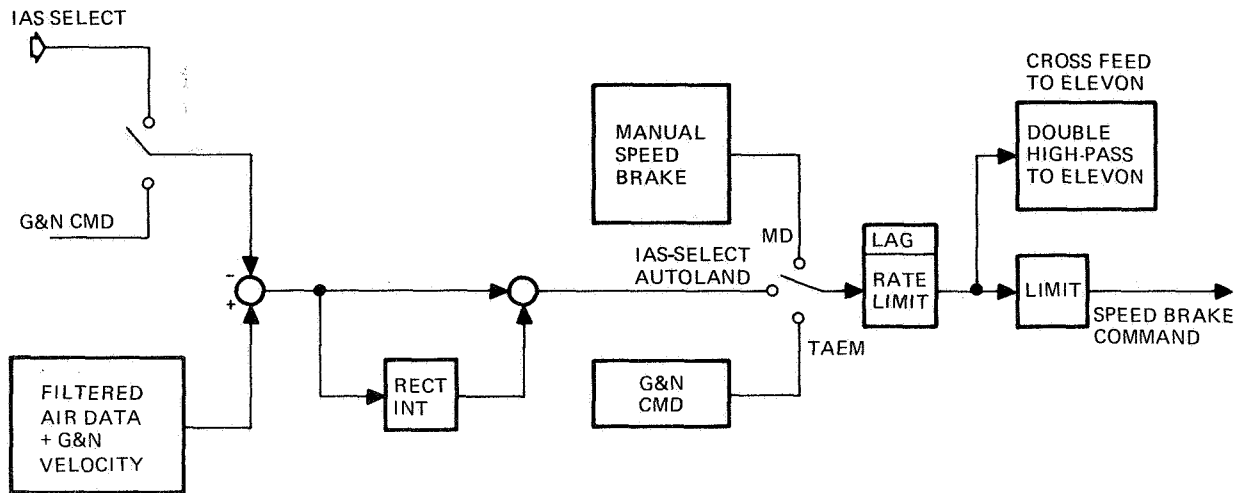
Figure 22. Indicated Air Speed (IAS) Hold
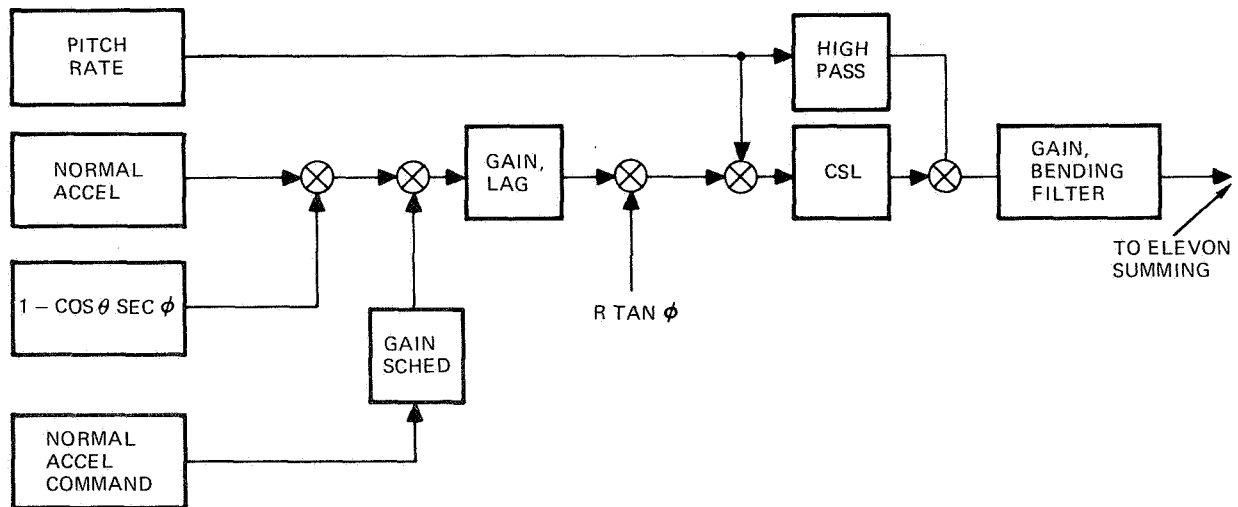
Figure 23. Pitch Auto-TAEM

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14,000 FT

12,000 FT

10,000 FT

20,000 FT

TAEM HEADING
ALIGNMENT
CYLINDER

GLIDE SLOPE = 21 - 24°

4,000 FT

4,000 FT

ELEVATION TRANSMITTER
200 FT LEFT OF CENTER LINE

DME/AZ

15,000 FT

1,000 FT

ACQUISITION PHASE ~18 SECOND
PERMITS G&N TRANSITION FROM
TACAN TO MSBLS DATA
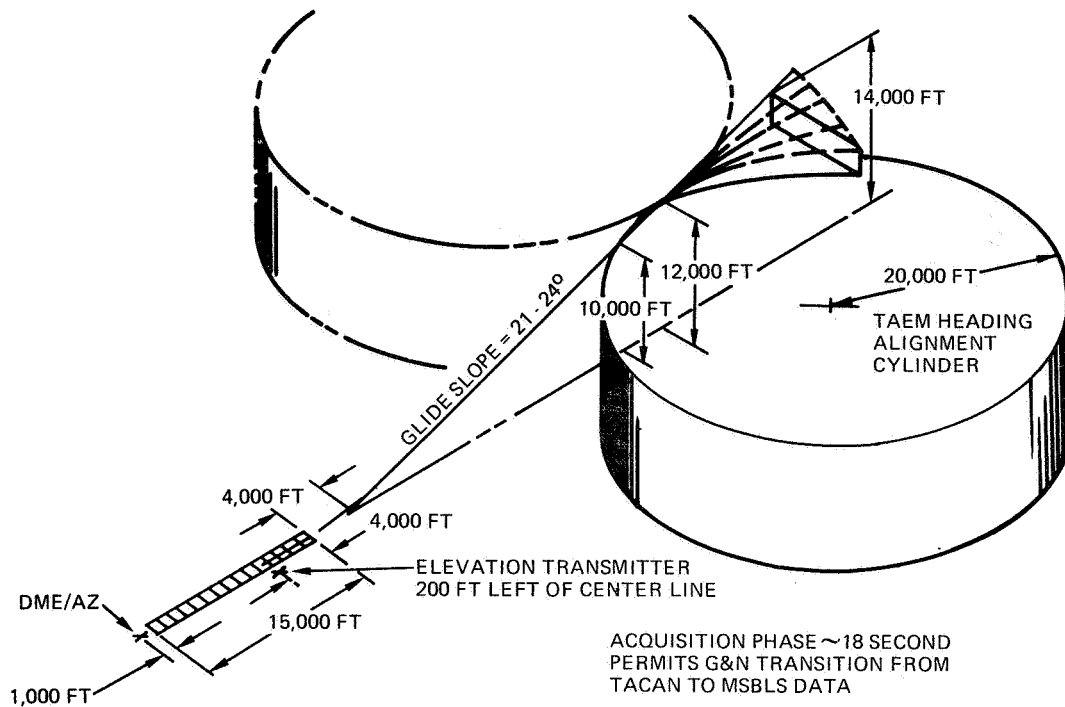
Figure 24. Microwave Scan Beam Landing System (MSBLS)

15,000

INITIATE AUTOLAND

10,000

ALTITUDE (FT)

INITIATE PULL UP &
SHALLOW GLIDE SLOPE

CIRCULARIZE

5,000

EXPONENTIAL CAPTURE

21 - 24°

FINAL FLARE

TOUCHDOWN

STOP

0

RUNWAY

-20,000   -10,000   0   10,000   20,000   30,000   40,000   50,000

DISTANCE FROM AIM POINT (FT)

0   10   20   30   40   50   60   70   80   90

TIME TO TOUCHDOWN (SEC)
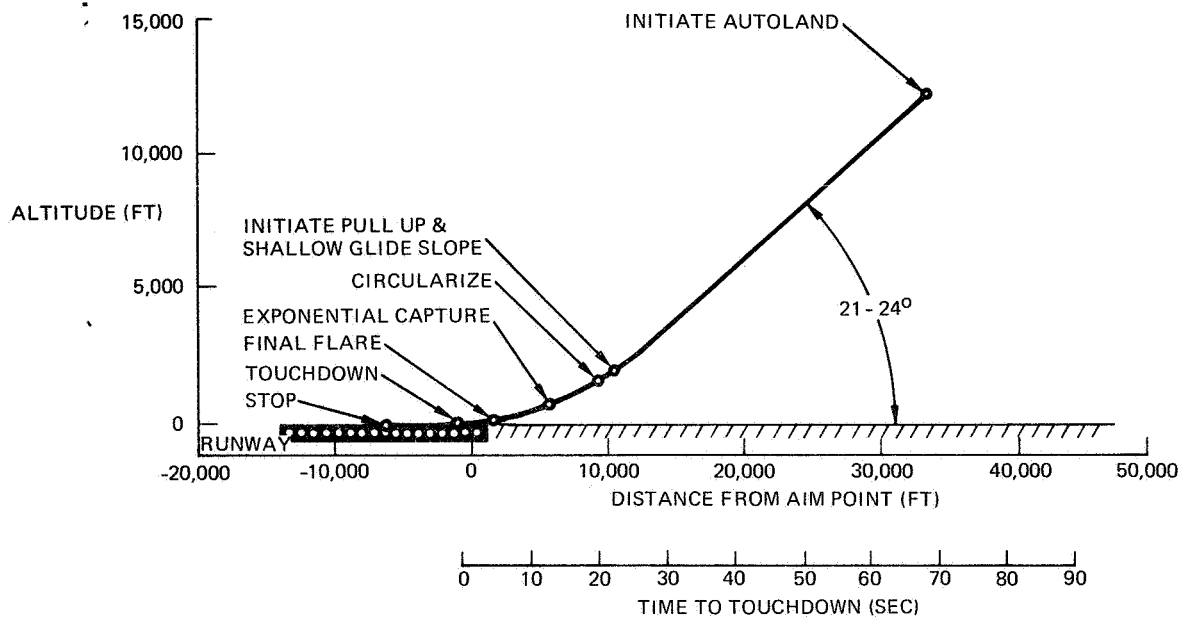
Figure 25. Autoland Trajectory

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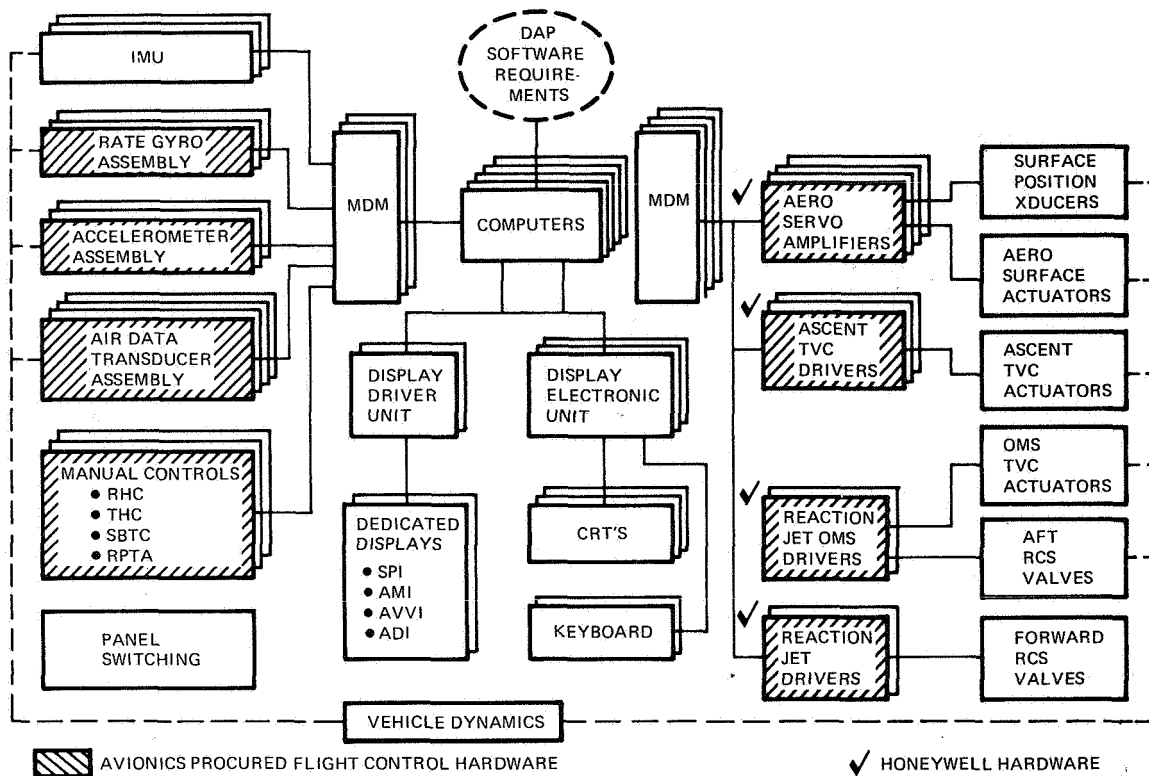
Figure 26. Flight Control System
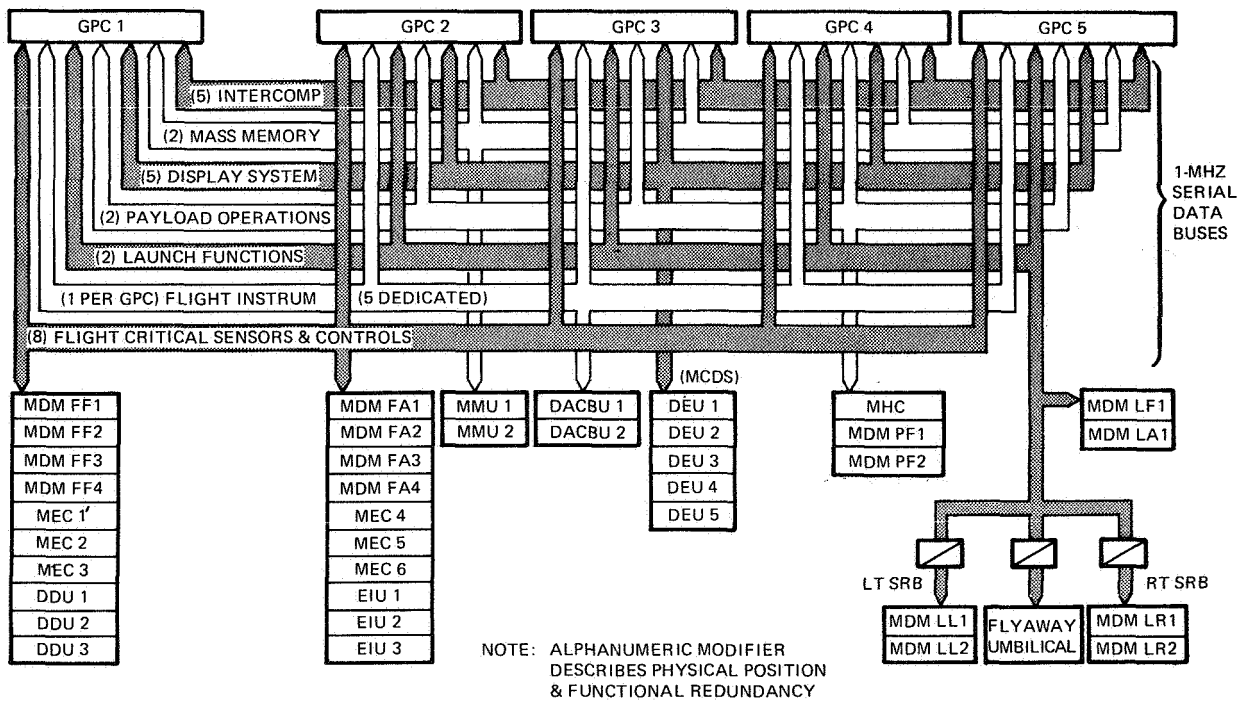
Figure 27. Digital Processing Subsystem Interface
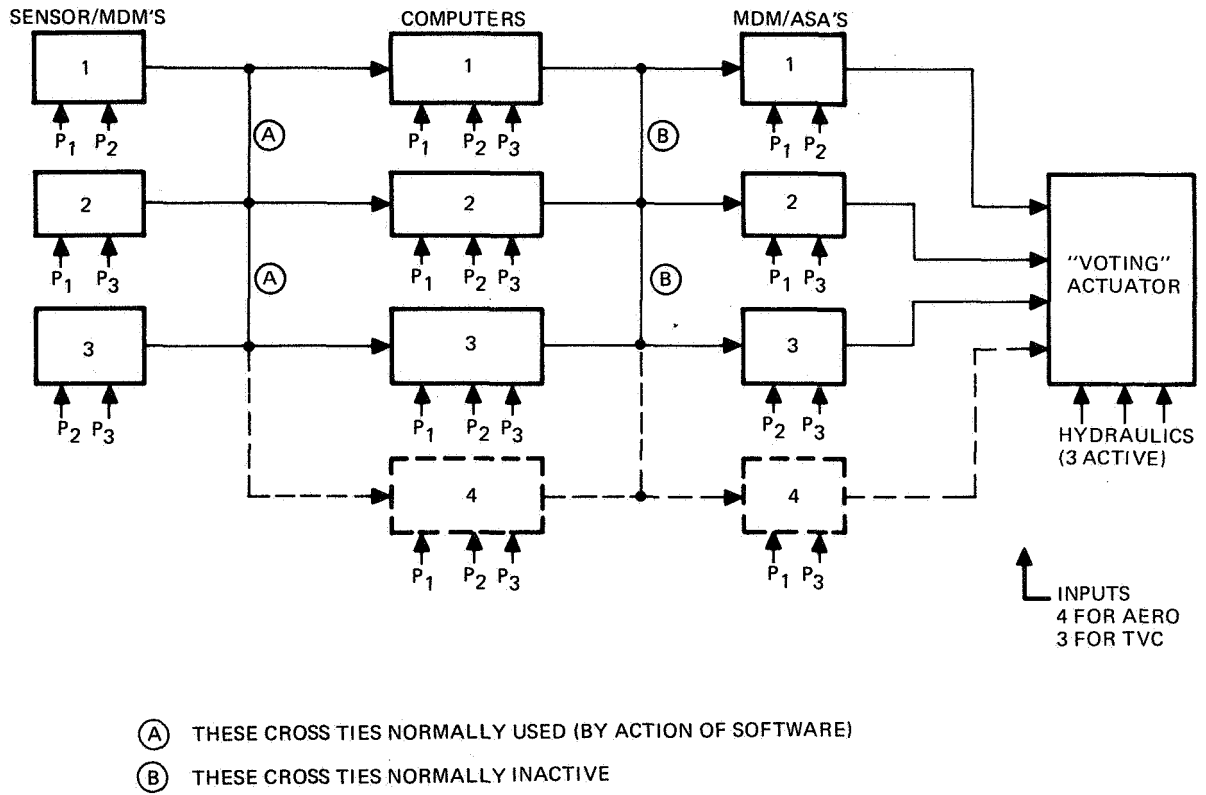
Figure 28. Basic Avionics Hardware Configuration for Critical Functions

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HISTORICAL REVIEW OF C-5A

LIFT DISTRIBUTION CONTROL SYSTEMS

T. E. Disney
Lockheed-Georgia Company

and

D. C. Eckholdt
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SUMMARY

Analytical and experimental development work on various load alleviation systems for the C-5A is reviewed to trace the development of the technical and hardware concepts to the present time. Variations in system objectives, means of implementation and effects on loads and airplane performance, stability and control are discussed.

This paper provides a logical lead in and introduction to the present system - the details of which are contained in the papers entitled "The C-5A Active Lift Distribution Control System" by W. J. Hargrove and "Some Experiences using Wind Tunnel Models in Active Control Studies" by R. V. Doggett, Jr., I. Abel, and C. L. Ruhlin.

INTRODUCTION

The work on load reduction systems for the C-5A at the Lockheed-Georgia Company began in 1967 and has progressed through several system variations to the present major effort on development of an Active Lift Distribution Control System (ALDCS). Figure 1 shows the chronological evolution of these efforts.

The Aircraft Load Alleviation and Mode Stabilization (LAMS) Program conducted by Boeing Wichita and Honeywell under contract to the Air Force Flight Dynamics Lab involved the C-5A to a small degree. The Lockheed-Georgia Company participated by providing C-5A data to demonstrate the applicability of the analysis methods and techniques to another large flexible airframe. Although the LAMS C-5A System Analysis and Synthesis was based on a single flight condition, the study results concluded that a LAMS type control system could reduce structural fatigue damage rates during flight through turbulence without significant degradation of basic aircraft stability and handling qualities.

During the conduct of the C-5A static test program in mid 1969, it became apparent that some form of wing maneuver load reduction system was

highly desirable for the purpose of reducing maximum wing upbending loads - a "strength design" load reduction rather than a fatigue load reduction system. The subsequent design and development effort involved analyses and test programs on a system which used symmetrical aileron deflections as a means of altering the spanwise airload distribution as a function of load factor, hence the name - Lift Distribution Control System or LDCS. The desire to reduce maximum wing upbending loads during maneuvers with minimum effect on performance and handling qualities led to an active system having a dead band below a load factor of 1.5 such that no system activity resulted until the load factor exceeded that magnitude. An additional selling point of this system was this "dead band" characteristic which resulted in no "black box" inputs during normal operation. This latter point is mentioned because of the natural reluctance on the part of flight crews to relinquish direct control of the aircraft to automatic flight controls. This system was developed and flight tested during late 1969 and early 1970 and is referred to as the maneuver LDCS (MLDCS) system.

A simplified version of the MLDCS known as the Passive LDCS (PLDCS) - fixed aileron uprig position selectable by the flight crew - was selected for fleet incorporation because it: a. Provided the desired maximum wing upbending moment reductions, b. Provided a reduction in 1.0g wing bending moments and thus a significant improvement in analytical fatigue life, c. Was attainable with a minimum hardware change and d. Did not involve "black box" control inputs independent of flight crew commands. The major detriment of this system is an increased drag due to the fixed aileron uprig resulting in significant takeoff performance, climb, and cruise drag penalties.

The results of the C-5 wing fatigue test program during the 1970-1972 time period, indicated a need for further wing load reductions or more appropriately, wing stress reductions, both during turbulence and during low load factor maneuvering. This need resulted in the present Active Lift Distribution Control System (ALDCS) Program which was initially explored by the C-5A Independent Structural Review Team (IRT) and recommended for development and fleet incorporation by the IRT in its report to the Air Force.

Subsequent sections of this paper discuss the objectives, means of implementation, load reductions and effects on performance and handling qualities of each of these systems. A comparison of these systems is made in the concluding section.

SYMBOLS

M'_x	Bending moment (Wing Swept Axis System)
M'_y	Torsional moment (Wing Swept Axis System)
N_o	Characteristic Frequency (Cycles per Second)
V_e	Equivalent Airspeed (Knots)
g	Gravitational acceleration constant (32.2 ft/sec^2)

i_T Stabilizer incidence angle

$N_{z_{cg}}$ Vertical load factor at c.g.

rms   root mean square

σ_s rms stress

LOAD ALLEVIATION AND MODE STABILIZATION - LAMS

The C-5A LAMS work was conducted by the Boeing Company and their technical partner, Honeywell, Inc., under contract with the Air Force Flight Dynamics Lab. The Lockheed-Georgia Company provided the math model and supported the analysis effort with their design background and baseline comparative data during these studies.

The purpose of the C-5A LAMS work was to demonstrate that the LAMS technology was applicable to aircraft other than the B-52 and to establish the potential benefits that such a system may offer on the C-5A. Selection of the C-5A to provide an additional aircraft on which to evaluate the LAMS technology was an excellent choice since the C-5 possesses relatively powerful-fully powered flight controls and three axis stability augmentation systems.

The major objective of this study was to develop a system having acceptable stability margins, retaining or improving existing aircraft handling qualities and providing a measurable improvement in fatigue damage rate and ride quality.

The resulting C-5A LAMS study is well documented in Reference 1. For comparative purposes, only the pitch axis portion of this system will be addressed in this paper.

The pitch axis mechanization of the C-5A LAMS Flight Control System is shown by the block diagram of Figure 2. The aileron and spoiler control loops provide a direct load reduction source through alteration of the lift distribution magnitude and shape, primarily as a function of vertical acceleration, while the inboard elevator loop provides an indirect wing load reduction by increasing the pitch damping to reduce pitch response in turbulence. In addition, it provides a pitch compensation effect to counter the pitching moment increments introduced by the ailerons and spoilers such that handling qualities remain relatively unaffected. The control column feed foward inputs provide cancelling signals to the normal acceleration and pitch rate feedback signals which would otherwise oppose a pilot command.

The aileron loop provides the required phasing for control of the first and second wing bending modes and additional gain attenuation for suppressing of undesirable higher order mode effects.

System performance as reflected by calculated stress values at selected airframe control points is summarized by Figure 3. It should be noted that

these stress values represent analysis of the gust source only and that total stress changes for all load sources (gust, maneuver, landing impact, taxi, etc.) were not evaluated during this study.

System performance relative to changes in flying qualities is summarized by Figure 4. In general, the response to pitch rate commands exhibits an increase in the time to reach a desired pitch attitude change with the response being overdamped. Addition of a normal acceleration signal to the inboard elevator channel would provide faster pitch response to input commands and would result in the comparative numbers shown under Modified LAMS FCS.

MANEUVER LOAD CONTROL - MLDCS

During late 1969 and early 1970, a study was conducted of various means of reducing maximum wing upbending moments on the C-5A. Figure 5 illustrates the various load reduction techniques evaluated and provides summary type trade-off information relative to load reduction magnitudes, hardware changes, development complexity, etc. The uprigged aileron concept was selected as the most practical means of obtaining significant wing bending moment reductions with minimum hardware change/least performance penalty.

A development program was initiated to design, develop and flight test an active load reduction system. The primary objectives of the system were:

o   Reduce positive maneuver maximum wing root bending moments by 10%
o   Minimize effects on handling qualities
o   Minimize effects on aircraft performance
o   Utilize existing hardware with minimum new components
o   Provide "full time - fail operative" system.

Since it was desirable to reduce the maximum upbending moments for "static strength" purposes only, the concept evolved into a system having a dead band below 1.5g with the system becoming active at higher load factors. This resulted in no drag penalty during takeoff, climb, cruise, etc., except during infrequent maneuvering to load factors above 1.5.

System implementation utilized existing, modified, and new hardware as shown by Figure 6. Normal accelerometers located at the wing first bending node line provided "rigid body" motion intelligence with minimum gain and phase effects for higher frequency responses. The existing pitch and yaw/ lateral Stability Augmentation System (SAS) computers provided the means of introducing desired commands to the ailerons and pitch compensation inputs to the inboard elevators. The breadboard MLDCS computer was designed to accept inputs from the accelerometers, a Mach signal from the Central Air Data Computer (CADC) for gain scheduling purposes, a flap position signal to deactivate the system in flaps extended configurations and a touchdown signal to deactivate the system during landing impact and ground operations. Outputs were provided to the yaw/lateral and pitch SAS computers, through which aileron and inboard elevator deflections are commanded, and to flight crew monitoring and control hardware. Triple channel redundancies and fail

safe features were incorporated in the system to fulfill the full time fail operative requirement.

A functional block diagram of the system is shown by Figure 7.

Structural load improvement attained with this system is illustrated by Figure 8. The MLDCS affects only maneuver loads at load factors above 1.5 thus there is no significant effect on fatigue loads resulting from the maneuver source. Gust loads are likewise not significantly affected due to both the rather high "g" onset level and the limited frequency response range of the system. During the development program, a compromise was made on aileron deflection magnitude due to the undesirable increase in positive wing torsion along with the desirable reduction in wing bending moment. Desirable bending moment reductions which reduced wing lower surface axial stress levels had to be limited since wing front beam web shear flow increased significantly due to the increased torsion loads as illustrated in Figure 9. The final scheduled maximum aileron deflection was set at ten degrees.

The development program included simulator testing and flight testing in addition to the analytical investigations. The flight test program evaluated handling qualities and provided substantiating data for structural load reductions. Figure 10 shows a comparison of analytical and flight test measured bending moments as function of load factor for a representative flight condition.

The effects of this system on aircraft performance and handling qualities are negligible. During flight testing it was difficult, if not impossible, to determine when this active system was operating. A more detailed discussion of this system is contained in reference 2.

PASSIVE LIFT DISTRIBUTION CONTROL SYSTEM - PLDCS

During the MLDCS development program, it became clear that some form of fatigue loads reduction was highly desirable. Moreover, it was desired to simplify the MLDCS from the standpoint of reduced new hardware in order to obtain early fleet incorporation of a load reduction system - thus the passive LDCS program was instituted.

The primary objectives of this system were:

o   Reduce positive maneuver maximum wing root bending moments by 10%,
o   Provide service life improvement by reduced 1.0g mean bending moments,
o   Minimize effects on aircraft performance,
o   Utilize existing hardware with minimum new components.

The PLDCS concept evolved into a fixed aileron uprig system with specific amounts of uprig as a function of airplane configuration and flight condition. Studies indicated that the "static" load reduction objective could be attained with a two position system having 5 degrees of uprig above 20,000 feet and 10 degrees of uprig below 20,000 feet. The objective to

attain a service life improvement required that the 5 degree setting be utilized in the takeoff and landing configuration in order to provide the reduced mean load benefit throughout the flight profile.

System implementation, as shown by Figure 11, then became a rather simple matter of using the existing individual aileron trim capability as an interim measure until the equally simple production changes could be incorporated by field level kit installation. The C-5 fleet has been using the PLDCS, interim and/or production systems, since November 1971.

The structural loads improvement attained with this system is illustrated in Figure 12. Note that the mean bending moment is reduced significantly along with the maximum bending moment.

This system results in significant effects on airplane performance as summarized by Figure 13. No change in aircraft handling qualities is generated since the system involves a fixed configuration change only which is compensated for in trim by use of slightly more airplane nose down stabilizer trim setting.

ACTIVE LIFT DISTRIBUTION CONTROL SYSTEM - ALDCS

In late 1972, the C-5A Independent Structural Review Team (IRT) included the development of an active LDCS in the list of options available to the Air Force as a means of extending the service life of the C-5A primary wing structure. Air Force review of the IRT options resulted in a decision to proceed with an ALDCS development program in mid 1973. This program involved the Lockheed-Georgia Company as prime contractor with participation of The Boeing Company (Wichita Division) and Honeywell as sub-contractors. The C-5 System Project Office was the contracting authority having technical and management control of the program with the Air Force Flight Dynamics Lab providing technical assistance and program review functions.

A unique aspect of this development effort was the use of a dynamically and elastically scaled model having an onboard hydraulic system to provide power for activation of the ailerons and horizontal stabilizer. The control system was operated by a console mounted analog computer simulation of the ALDCS computer using inputs from the onboard ALDCS sensors. This model provided an experimental dynamic loads/flutter data acquisition tool with which to gain confidence in the analytical methods used in development of the ALDCS mechanization. The model wind tunnel test program was accomplished at the NASA Transonic Dynamic Variable Density Tunnel at Langley AFB and involved a test team consisting of personnel from Lockheed, Boeing, NASA, and The Air Force.

The objectives of the ALDCS being developed in this program are as follows:

o   Reduce gust RMS wing root bending moments by 30%,

o   Limit gust RMS wing root torsional moment increases to not more than 5%,

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o   Reduce maneuver incremental wing root bending moments by 30%,
o   No increase in discrete gust wing loads,
o   No significant changes in existing performance and handling
     qualities,
o   Provide "full time - fail safe" system,
o   Interface with existing systems and use existing hardware
     where possible,
o   No significant degradation in flutter margins.

System mechanization was derived using the proposed IRT schematic as a baseline system. This system in itself had its beginnings in the C-5A LAMS pitch axis mechanization. System implementation includes PLDCS and involves use of existing control surfaces, actuators and servos, modified SAS and CADC computers and new hardware as shown in Figure 14. A functional diagram of the system is shown in Figure 15. This system, as was the MLDCS, is designed to interface with existing SAS and autopilot systems. It should be noted that the basic C-5A autopilot provides a significant reduction in continuous turbulence induced wing loads by means of the increased pitch damping effect attained when in the attitude hold mode.

The effects of the system on wing load improvement during maneuvering flight are represented by the plots of Figure 16. The bending-torsion plot illustrates the effect of the system on maneuvering loads for a typical strength design case. The 1.0g shift is due to the PLDCS static aileron up-rig. The significant slope change between 1.0 and 1.9g is the result of the ALDCS incremental aileron deflection. For load factors in excess of 1.9 the ALDCS incremental aileron deflection is removed such that at design limit load factor of 2.5 the system is again in the PLDCS configuration. This is necessary to prevent the generation of a wing front beam shear flow problem as discussed in the MLDCS section.

The effect of the ALDCS on the fatigue load spectra for maneuvering flight is shown by the right hand portion of Figure 16. Note that at high incremental load levels (load factors greater than 1.9) the two spectra are equal. The large number of maneuvers at load factors below 1.9 results in a significant reduction in the magnitude of the low and intermediate load levels. This is the area in which the majority of the maneuver source fatigue damage occurs; thus a significant improvement in the maneuver source damage is realized.

Loads improvement for the continuous gust source is illustrated by Figure 17. A typical wing root bending moment gust output spectrum is shown for the baseline and the ALDCS configurations. The effect of the system on the incremental gust load spectra is illustrated by the curves on the right side of Figure 17. The increase in characteristic frequency (N_O) is relatively unimportant from a fatigue damage standpoint since the load reduction effects are far more significant. As is the case with the maneuver spectra, the baseline and ALDCS curves become one at load levels corresponding with c.g. load factors greater than 1.9.

The aircraft performance and handling qualities effects introduced by this system are summarized in Figure 18.

COMPARISON OF C-5A LDCS SYSTEMS

The three systems which have been/are being developed and flight tested are compared in Figure 19 relative to major objectives, means of implementation, loads improvement magnitudes and aircraft performance/handling qualities effects.

It should be emphasized that the paramount objective in each of these systems was some form of wing bending moment reduction - either strength or fatigue related - with secondary objectives of system simplicity and minimum effects on aircraft performance/handling qualities. No attempt was made to provide a "mode stabilization/control" function for purposes of flutter boundary extension or ride control improvement.

Some of the trade-offs or compromises between conflicting objectives are apparent from the comparison chart. Note specifically that the price of obtaining reduced mean bending moments, as provided by the Passive System, is an aircraft performance penalty. An offsetting benefit on this system was the ability to attain an almost immediate incorporation with a minimum hardware impact.

The next variation - to provide reductions in maneuver and gust incremental bending moments while retaining the reduced mean loads generated a significantly larger hardware design/development problem than that of the original maneuver load control MLDCS and in addition retained the performance penalties of the passive system.

A comparison of the effects of each of the three systems on wing root loads is shown by Figure 20. The flight condition selected for this illustration was chosen to depict the initial objective of reducing maximum up-bending moment by approximately 10% (actually attained about 9% due to bending torsion trade-off effects). The reduction in the 1.0g bending moment is about 25% for the PLDCS and ALDCS while the incremental bending moment is reduced approximately 40% by ALDCS for this condition. Similar load reductions exist for other flight conditions.

CONCLUDING REMARKS

The work done over the past five years on the various LDCS systems has demonstrated the practicality of using existing flight control surfaces and systems to affect specific changes in structural load distributions and magnitudes and/or aerodynamic characteristics of the C-5A.

The attainment of desired primary objectives has resulted in certain compromises in one or more of the many diverse requirements of such a complex system as the C-5A.

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This work illustrates an application of active/passive control technology to the solution of one type of problem on an existing aircraft. Application of the same engineering principles during the design stage of a new aircraft could have significant effects on the overall "design compromise".

At this point a word of caution is deemed necessary. The success of the LDCS systems on the C-5A has been evaluated on the basis of attaining specific load reductions (primarily wing bending moments). The significance of these load reductions on the structural integrity and service life of the airframe has only been evaluated by existing state-of-the-art structural analysis and test methods. Since conventional fatigue analysis methods treat only axial stresses in a system based on constant amplitude cyclic test data, little is known about combined axial and shear stress effects on fatigue. The message here is to proceed slowly and don't commit to a design or a design fix on the basis of a partial evaluation.

REFERENCES

1.   The Boeing Company, Wichita Division and Honeywell, Inc., Aerospace Division: Aircraft Load Alleviation and Mode Stabilization (LAMS) C-5A System Analysis and Synthesis. AFFDL D3-7901-2, September 1968

2.   Silvers, C. L. and Phillips, J. W.: Documentation of C-5A MLDCS Testing on Ship 0003 during 1969-1970. Lockheed-Georgia Report LG74ER0037, April 1974
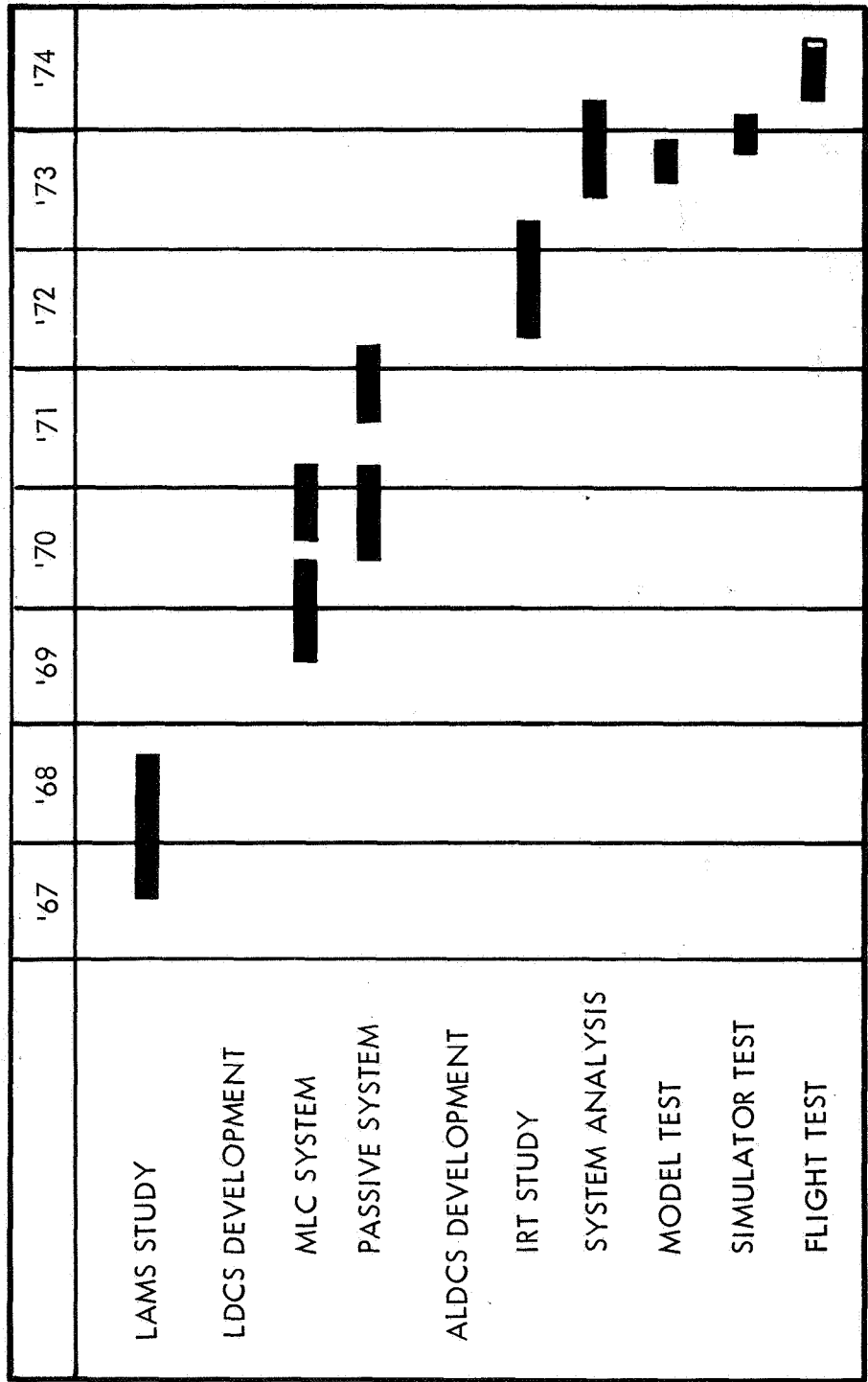
C-5A LIFT DISTRIBUTION CONTROL SYSTEM EVOLUTION

FIGURE 1

304
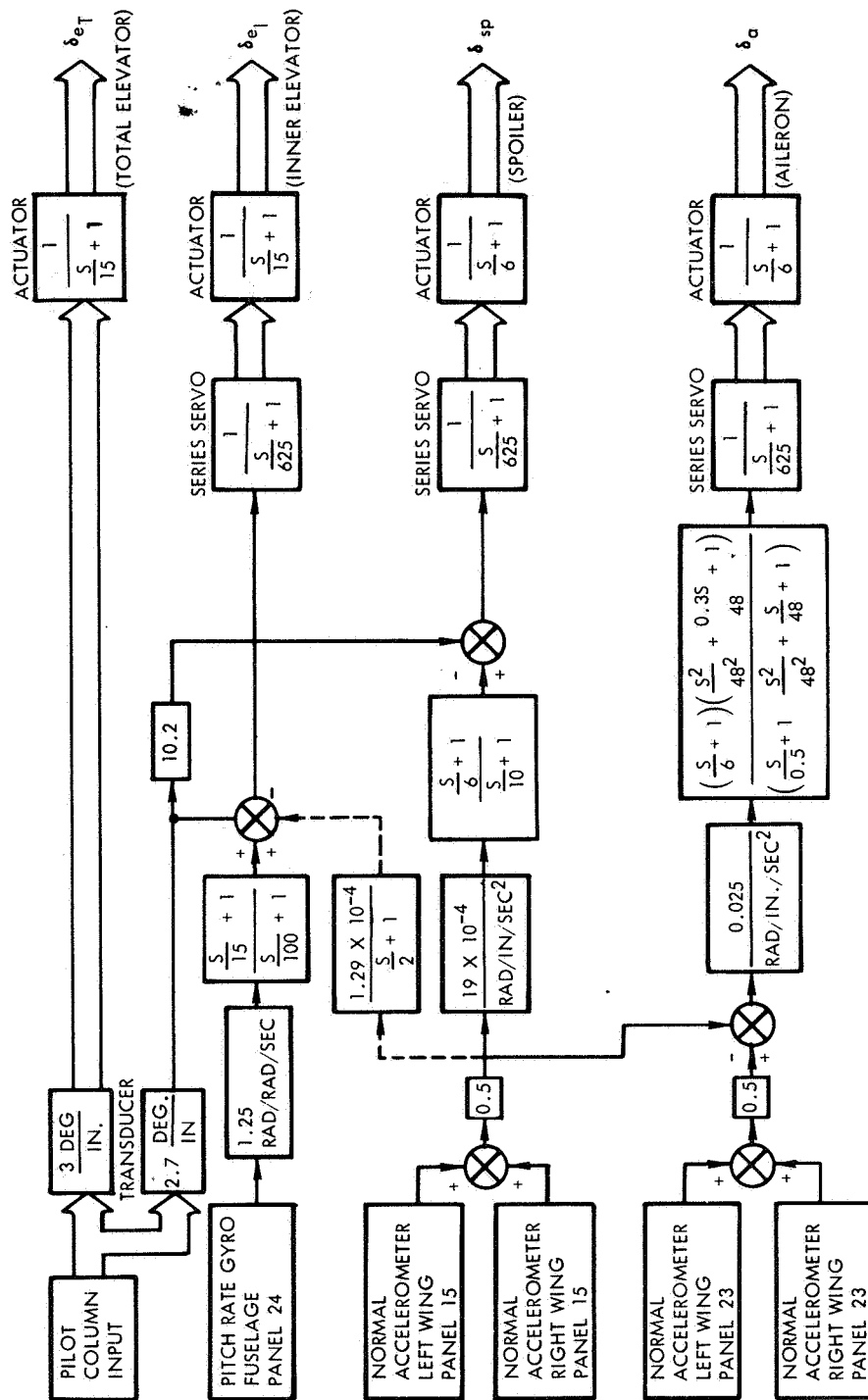
C-5A LAMS PITCH AXIS SYSTEM MECHANIZATION

FIGURE 2

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ANALYTICAL STRESS $\sim \sigma_S$

STRUCTURAL LOCATION	BASELINE AIRCRAFT	LAMS
WING STATION 120	207.1	48.6
WING STATION 746	230.2	82.2
HORIZONTAL TAIL ROOT	79.1	110.7
FUSELAGE STATION 1804	42.9	70.2
FUSELAGE STATION 1106	39.9	24.0

NOTES: 1. GUST INPUT OF 1 FT/SEC RMS

2. STRESS LEVELS ARE PSI

3. STRESS LEVELS CALCULATED USING ANALYTICAL BENDING MOMENTS AND STRESS TO LOAD RATIOS

C-5A LAMS STRUCTURAL PERFORMANCE IMPROVEMENT
PITCH AXIS

FIGURE 3

PITCH ATTITUDE RESPONSE CHARACTERISTICS -
ELEVATOR SQUARE WAVE INPUT

PARAMETER	BASELINE (SAS)	LAMS FCS	MODIFIED LAMS*
TIME TO 90% (SECONDS)	1.5	2.1	1.7
PERCENT OVERSHOOT	15.0	OVERDAMPED	10
ATTITUDE CHANGE	1.92	2.61	2.61

NOTE: THE USE OF UPRIGGED SPOILERS AS IN THE LAMS MECHANI-
ZATION WOULD GENERATE A DRAG PENALTY THUS A PAY
LOAD RANGE EFFECT WHICH WAS NOT EVALUATED DURING
THE STUDY

C-5A LAMS FLYING QUALITIES AND PERFORMANCE EFFECTS

FIGURE 4

C-5A WING LOAD REDUCTION TECHNIQUES EVALUATION

#	SYSTEM	PERCENT MAX BENDING MOMENT REDUCTION	NEW HARDWARE		MOD. EFFORT		R & D EFFORT			PERFORMANCE EFFECTS		
			SURFACE	CONTROLS	MINOR	MAJOR	SMALL	MEDIUM	LARGE	NEG.	MINOR	LARGE
1	INB'D AUX. SPLIT MECHANICAL FLAP 10° / 25°	2 / 10		•	•			•		•		
2	AILERONS 10° / 25°	7 / 11		•	•		•				•	
3	SPOILERS 25°	7		•	•		•				•	
4	SPOILERS & AILERONS	10+		•	•		•				•	
5	LOWER SURFACE FORWARD SPOILER	10	•		•				•	•		
6	BLOWN FLAPS	10	•			•			•	•		
7	ADVERSE JETS OUTBOARD	10	•			•			•	•		
8	RESERVE FUEL/DISTRIBUTION	2			•		•			•		
9	CLIPPED WING TIP	5			•		•					•
10	AERODYNAMIC FENCE	1			•			•		•		

FIGURE 5

NEW HARDWARE

o LDCS COMPUTER
o WING MOUNTED ACCELEROMETERS
o CONTROL PANEL
o ANNUNCIATOR LIGHTS

MODIFIED EXISTING HARDWARE

o YAW/LATERAL STABILITY AUGMENTATION COMPUTER
o PITCH STABILITY AUGMENTATION COMPUTER

EXISTING HARDWARE

o CENTRAL AIR DATA COMPUTERS
o FLAP POSITION SWITCHES
o TOUCHDOWN SWITCHES
o MADAR SUBSYSTEM

C-5A MLDCS SYSTEM IMPLEMENTATION
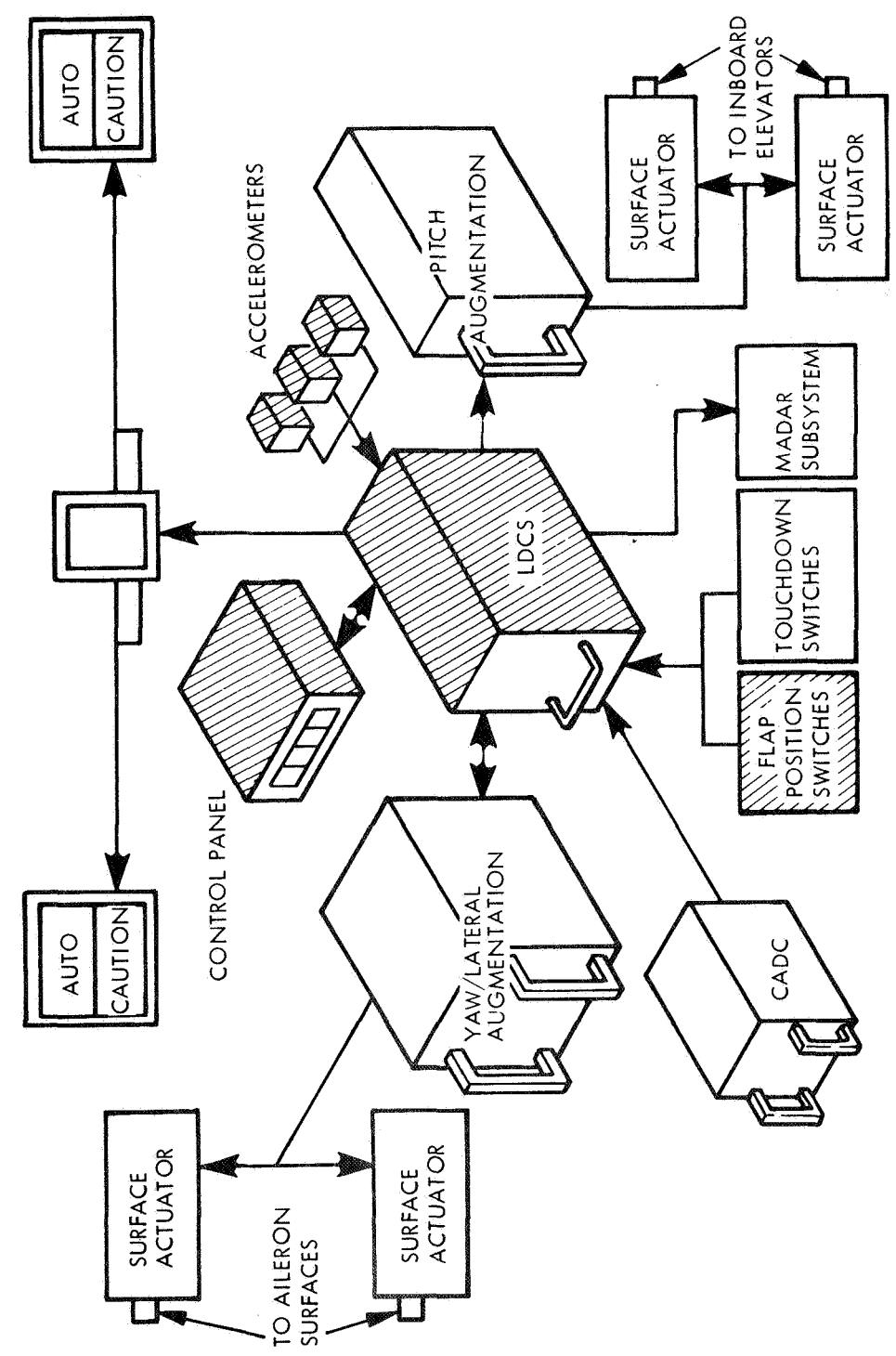
FIGURE 6

C-5 MLDCS FUNCTIONAL BLOCK DIAGRAM

FIGURE 7
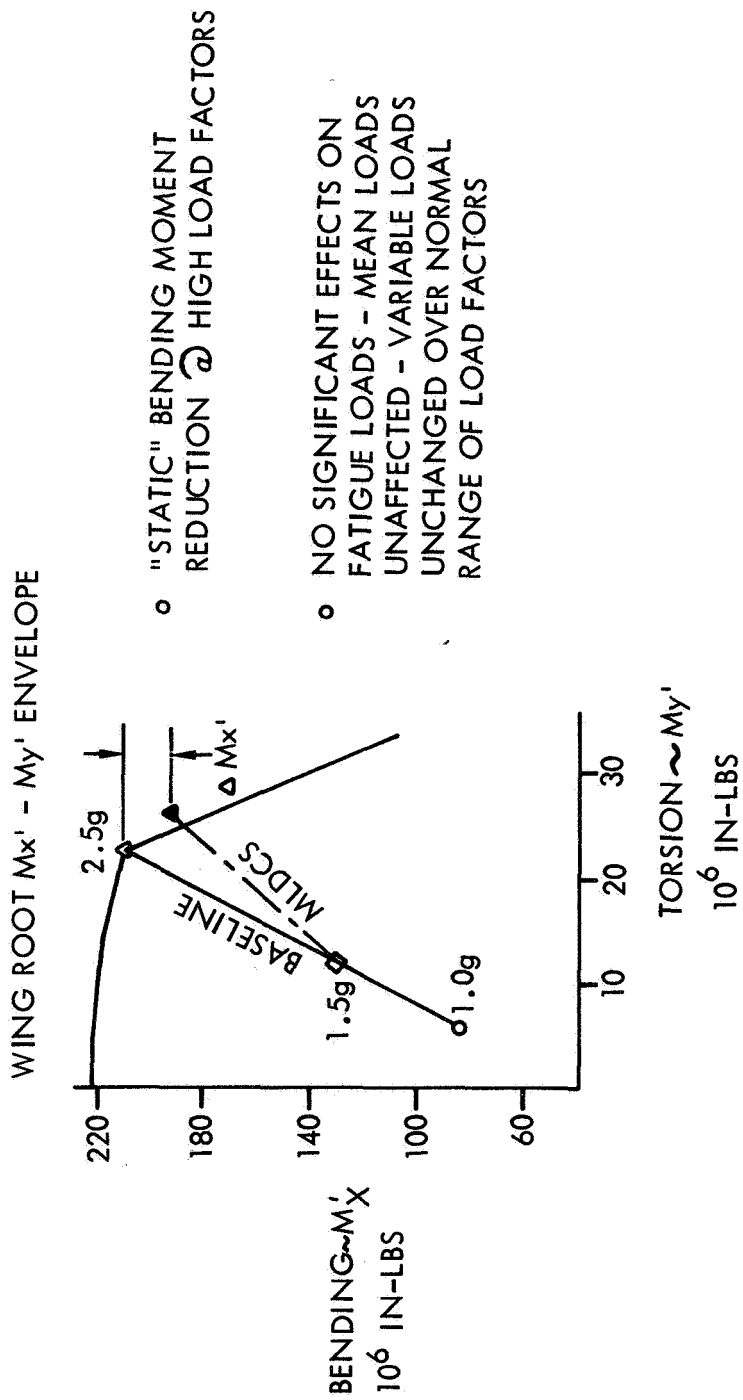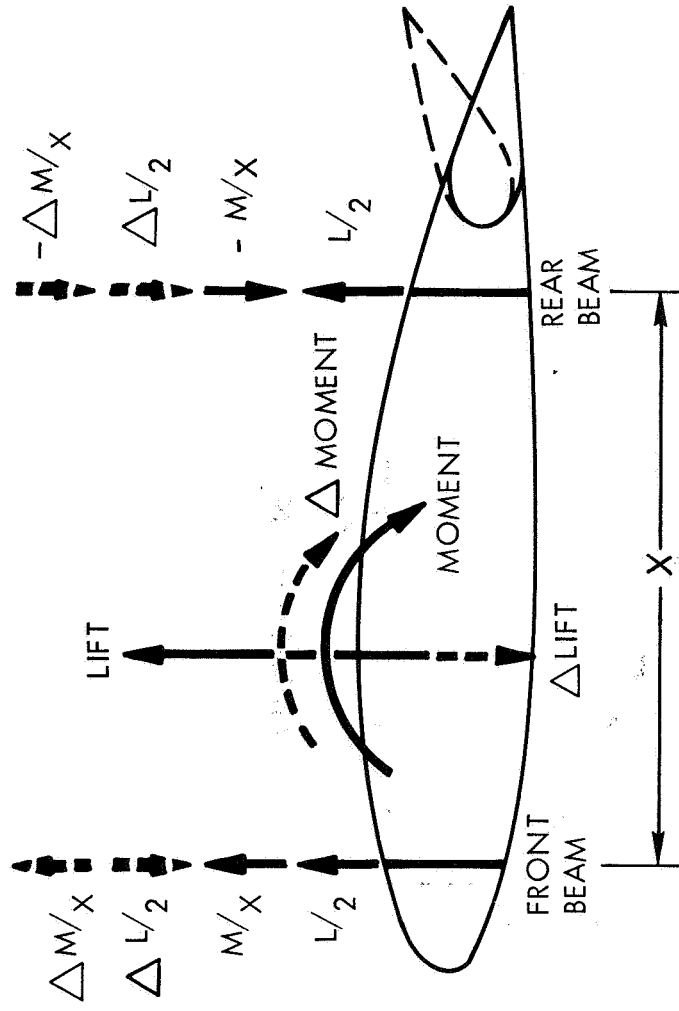
WING ROOT Mx' – My' ENVELOPE

o "STATIC" BENDING MOMENT REDUCTION @ HIGH LOAD FACTORS

o NO SIGNIFICANT EFFECTS ON FATIGUE LOADS – MEAN LOADS UNAFFECTED – VARIABLE LOADS UNCHANGED OVER NORMAL RANGE OF LOAD FACTORS

C-5 MLDCS STRUCTURAL LOAD IMPROVEMENT

FIGURE 8

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EFFECT OF UPRIGGED AILERON ON FRONT BEAM SHEAR

FIGURE 9

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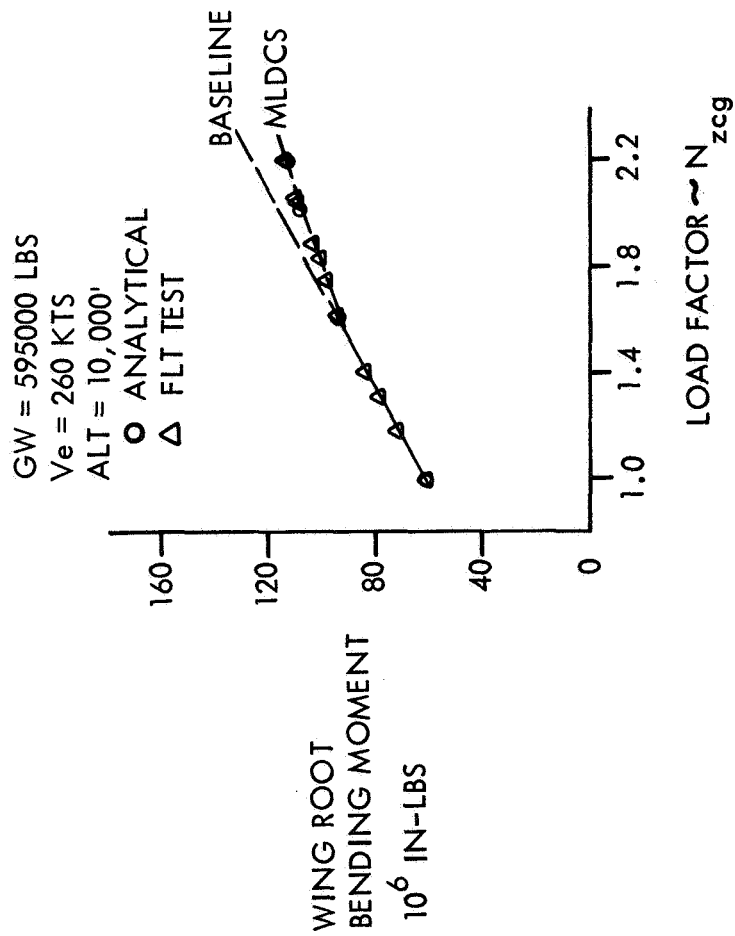
C-5 MLDCS COMPARISON OF ANALYTICAL AND FLIGHT TEST LOADS

FIGURE 10

INTERIM SYSTEM

o USE EXISTING INDIVIDUAL AILERON TRIM CAPABILITY

o ADD INSTRUCTIONS TO FLIGHT HANDBOOK

PRODUCTION SYSTEM

o INCREASE POSITIVE PITCH TRIM ACTUATOR STOP FROM 1. 5
   TO 2. 7 DEGREES

o INSTALL SHORTENED AILERON FEED BACK ROD - 6 DEGREES
   UPRIG NEUTRAL

o ADD LDCS ARM SWITCH AND MOMENTARY ON UPRIG AND
   DOWNRIG SWITCH

o ADD INDEX MARKS ON AILERON TRIM INDICATOR

C-5A PLDCS SYSTEM IMPLEMENTATION

FIGURE 11

WING ROOT Mx' – My' ENVELOPE

o "STATIC" BENDING MOMENT REDUCTION @ ALL LOAD FACTORS

o REDUCED MEAN LOADS – IMPROVED FATIGUE LIFE – NO CHANGE IN VARIABLE LOADS
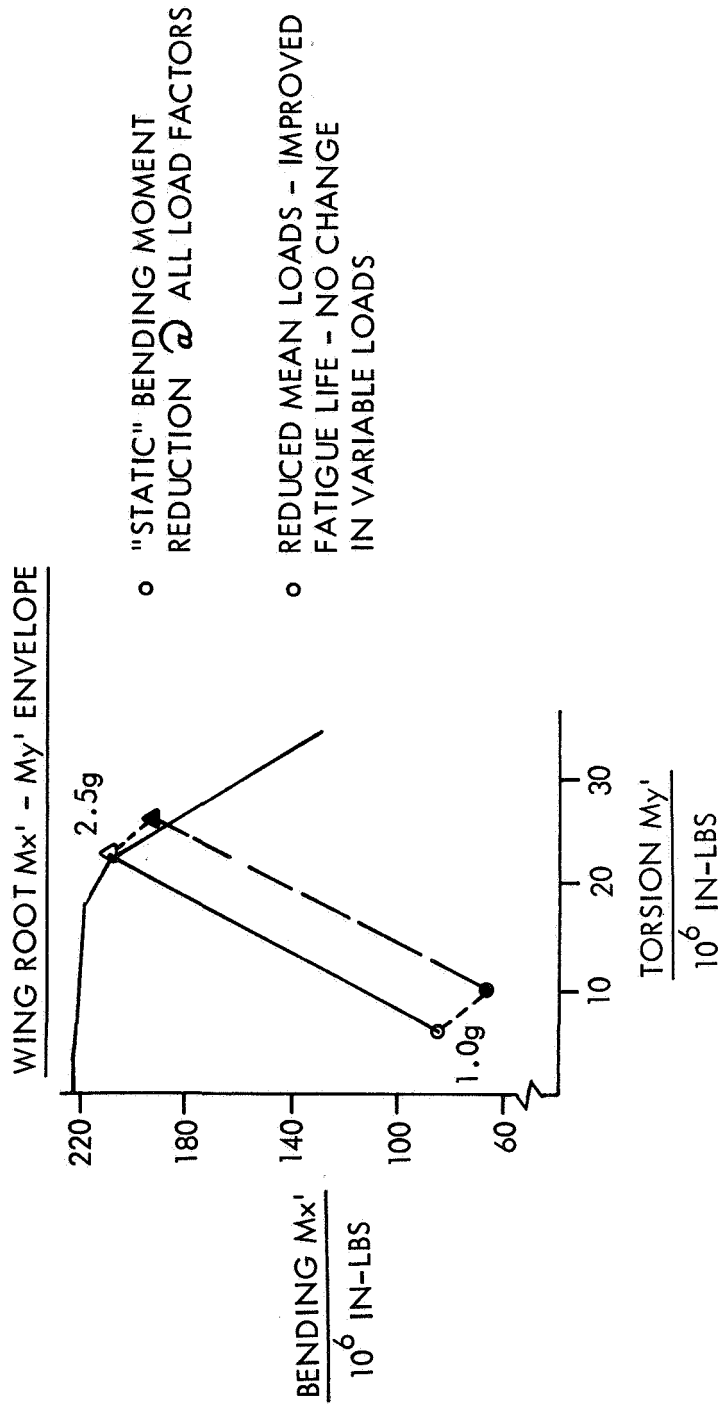
C-5A PLDCS STRUCTURAL LOAD IMPROVEMENT

FIGURE 12

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EFFECTS ON AIRCRAFT PERFORMANCE

o INCREASED T. O. FIELD LENGTH OR REDUCED T. O. G. W. OR
   INCREASED ROTATION SPEED

o REDUCED CLIMB PERFORMANCE (GRADIENT REDUCED . 23%)

o PAYLOAD RANGE REDUCTION (150 - 300 NM)

EFFECTS ON HANDLING QUALITIES

o NO SIGNIFICANT CHANGE

C-5A PLDCS EFFECTS ON PERFORMANCE AND
HANDLING QUALITIES

FIGURE 13

NEW HARDWARE

o ALDCS COMPUTER

o WING MOUNTED ACCELEROMETERS

o CONTROL PANEL

o ANNUNCIATOR LIGHTS

MODIFIED EXISTING HARDWARE

o YAW/LATERAL STABILITY AUGMENTATION COMPUTER

o PITCH STABILITY AUGMENTATION COMPUTER

EXISTING HARDWARE

o CENTRAL AIR DATA COMPUTERS

o AUTOPILOT NORMAL ACCELEROMETER

o SAS PITCH RATE GYRO

o CONTROL COLUMN POSITION SENSOR

o FLAP POSITION SWITCHES

C-5A ALDCS SYSTEM IMPLEMENTATION

FIGURE 14

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C-5A ALDCS FUNCTIONAL BLOCK DIAGRAM

FIGURE 15

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WING ROOT LOADS

MANEUVER SPECTRA
14 MISSION PROFILES

LIMIT LOAD ENVELOPE
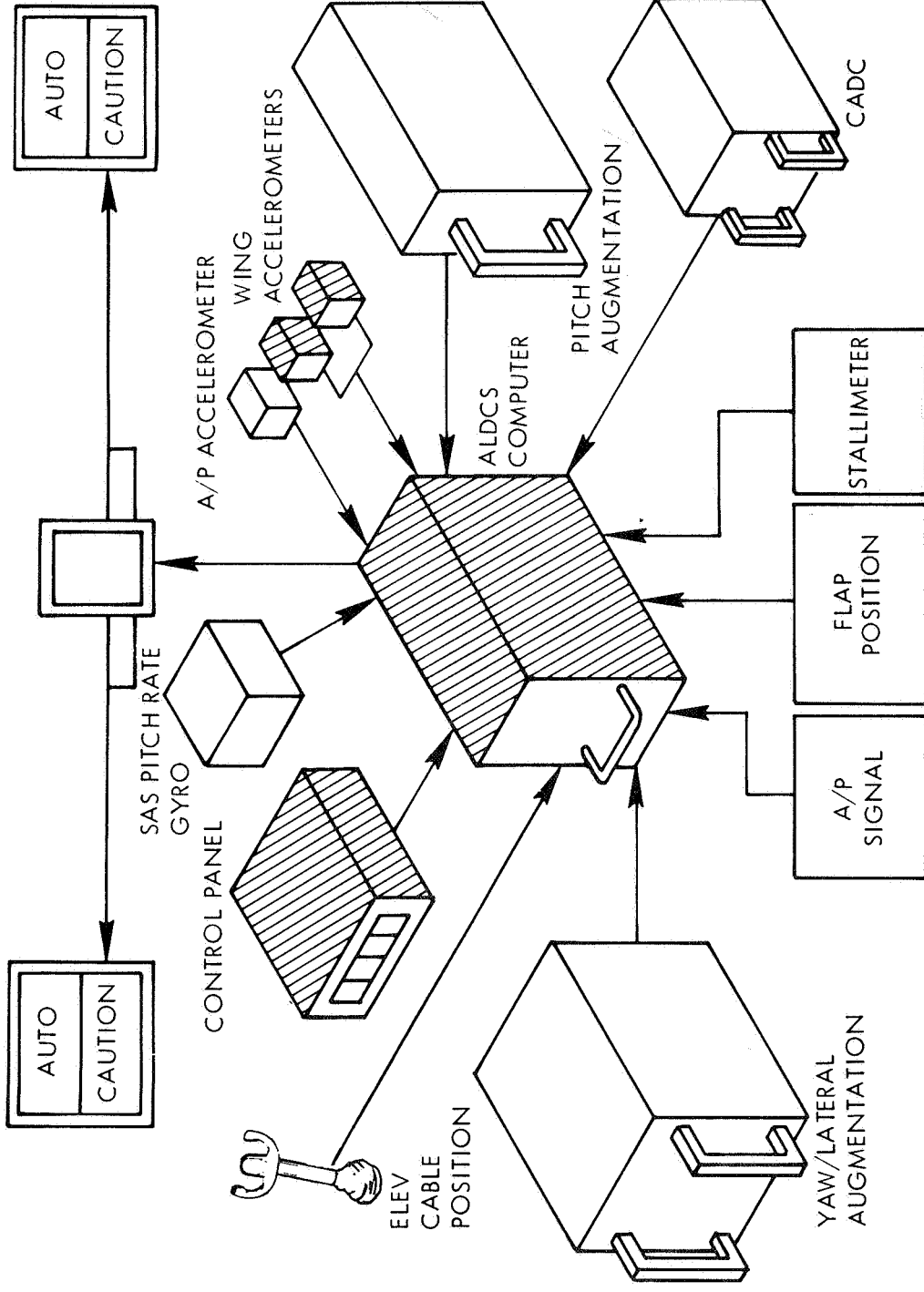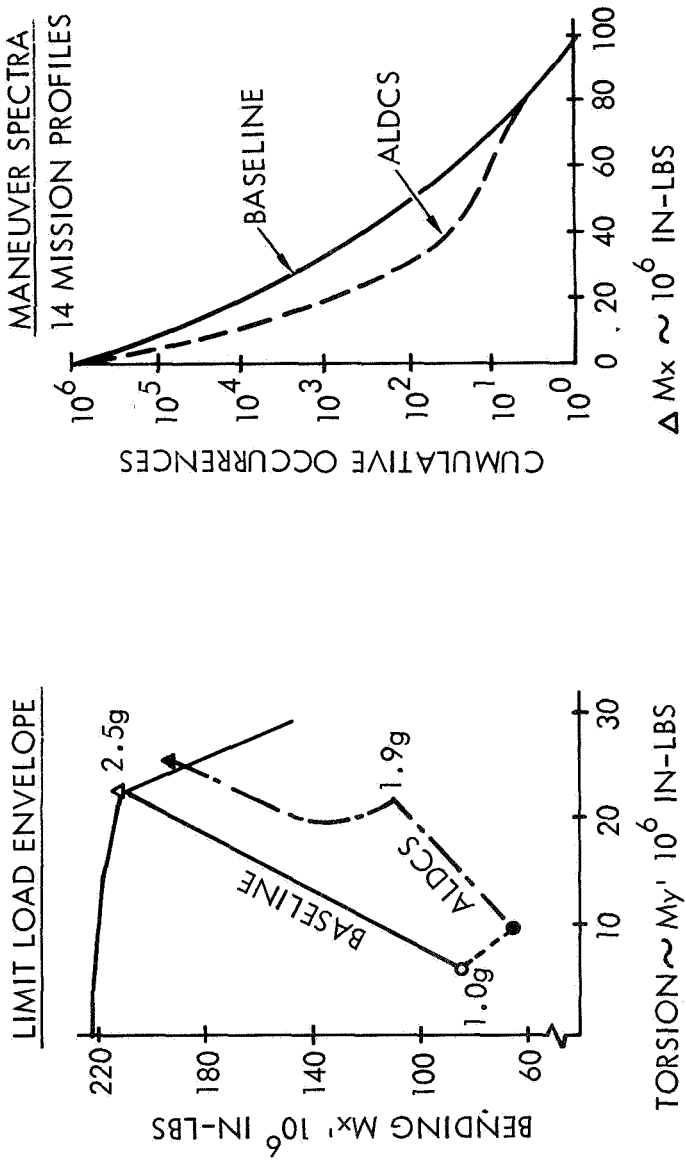
C-5A ALDCS MANEUVER LOADS IMPROVEMENT

FIGURE 16

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WING ROOT BENDING MOMENT

GUST SPECTRA
14 MISSION PROFILES

TYPICAL GUST
OUTPUT SPECTRUM
1 fps rms
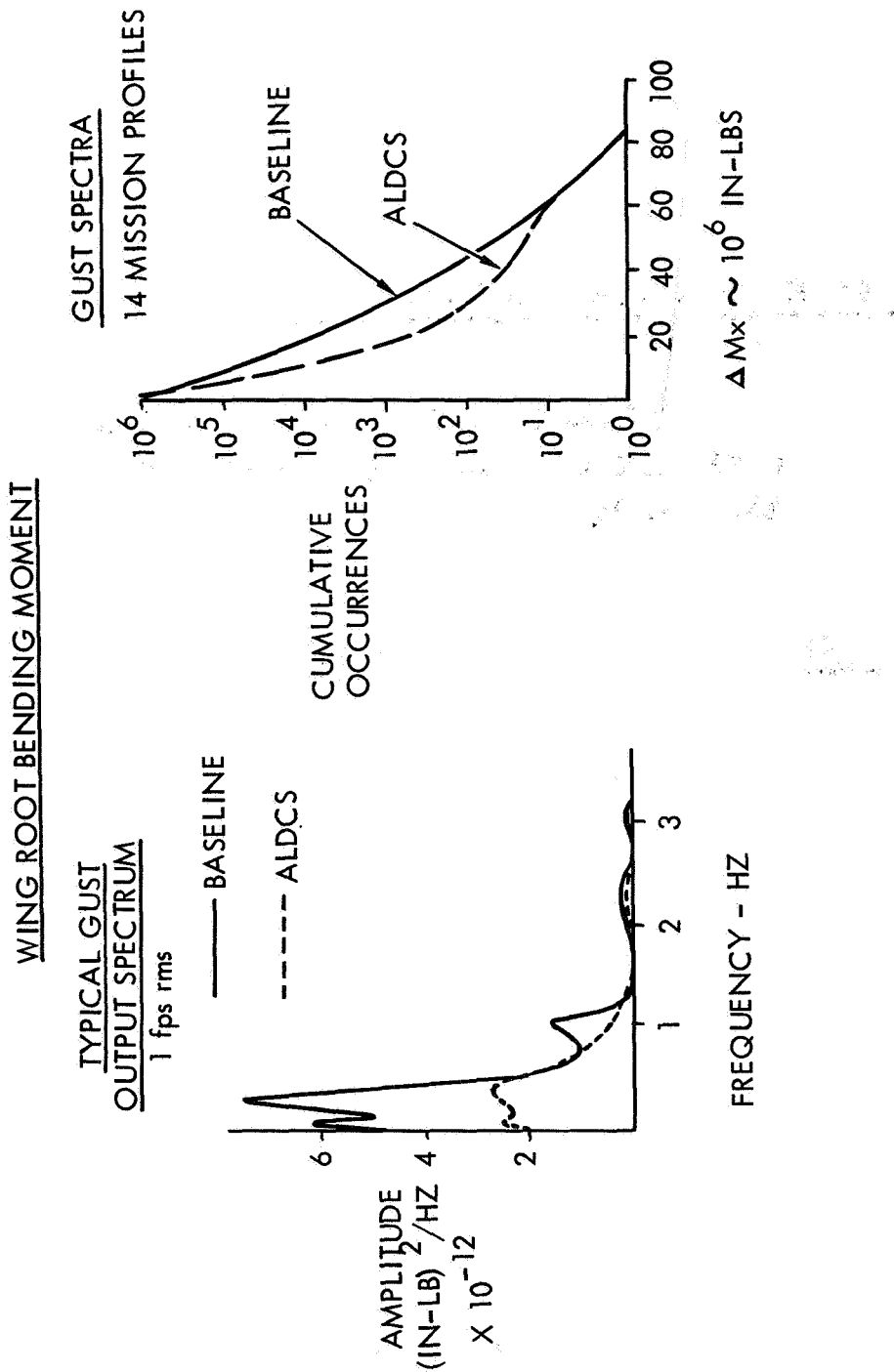
C-5A ALDCS GUST LOADS IMPROVEMENT

FIGURE 17

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EFFECTS ON AIRCRAFT PERFORMANCE

o INCREASED T. O. FIELD LENGTH - (SAME AS PLDCS)
o REDUCED CLIMB PERFORMANCE - (SAME AS PLDCS)
o PAYLOAD RANGE REDUCTION - (SAME AS PLDCS)

EFFECTS ON HANDLING QUALITIES

o NO SIGNIFICANT CHANGES

EFFECTS OF ALDCS ON PERFORMANCE AND HANDLING QUALITIES

FIGURE 18

SYSTEM	MAJOR OBJECTIVES	MEANS OF IMPLEMENTATION	LOADS IMPROVEMENT			PERF. & HAND. QUAL. EFFECTS
			MAX STATIC M_x'	FATIGUE M_x		
				MEAN	INC.	
MLDCS 1969 – 70	REDUCE MAX UPBENDING (M_x') @ DESIGN LOAD FACTOR	EXISTING SAS & CONTROL SYS. PLUS NEW COMPUTER & ACCEL'S	≈-9%	-	-	NONE
PLDCS 1970 – 72	SAME AS MLDCS PLUS REDUCED FATIGUE MEAN M_x'	EXISTING TRIM SYSTEM – INTERIM NEW CONTROL BOX & FOLLOW UP LINK AILERONS OPEN UP +i_T STOP	≈-9%	-10 TO -30%	-	T.O. CLIMB & CRUISE DRAG PENALTY NO F.Q. EFFECTS
ALDCS 1973 – 74	SAME AS PLDCS PLUS REDUCED FATIGUE INCREMENTAL	EXISTING SAS A/P & CONTROLS NEW COMPUTER & ACCEL'S	≈-9%	-10% TO -30%	-30 TO -50%	SAME AS PLDCS

COMPARISON OF C-5 LDCS SYSTEMS

FIGURE 19

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WING ROOT Mx' – My' ENVELOPE
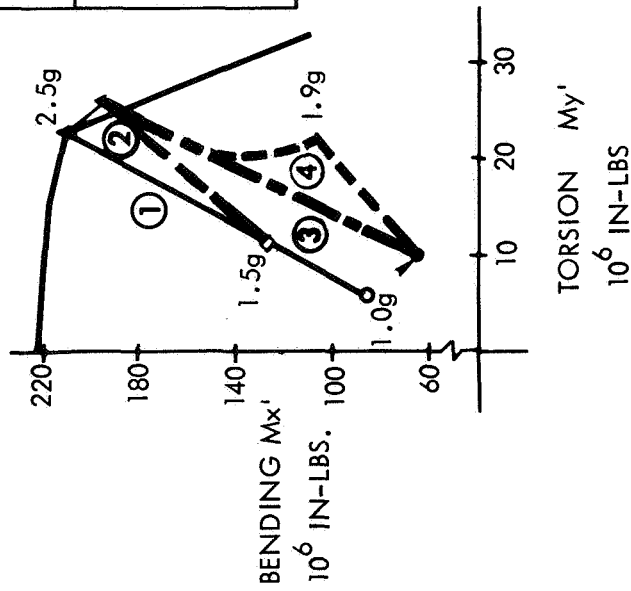
LINE	SYSTEM	MAX. Mx' 10^6 IN-LBS	MEAN Mx' 10^6 IN-LBS	VARIABLE Mx'/nz 10^6 IN-LBS
①	BASELINE	212	85.2	84.4
②	MLDCS	194	85.2	84.4/67.8
③	PLDCS	194	64.2	84.4
④	ALDCS	194	64.2	50.9

FLIGHT CONDITION

CLEAN CONFIG.
265 KNOTS @ S.L.
728000 LBS. GW.

BENDING Mx' 10^6 IN-LBS.

TORSION My' 10^6 IN-LBS

COMPARISON OF STRUCTURAL LOAD EFFECTS FOR LDCS SYSTEMS

FIGURE 20

THE C-5A ACTIVE LIFT DISTRIBUTION CONTROL SYSTEM

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SUMMARY

An Active Lift Distribution Control System (ALDCS) has been developed for the C-5A as a means to reduce wing fatigue damage due to maneuver and gust load sources. The Lockheed-Georgia Company proposed a four phase program; the development and design of a prototype system, flight test evaluation, production system fabrication, and airplane fleet installation of this subsystem.

This paper describes the ALDCS development and design tasks, ALDCS functional configuration, and resulting challenges encountered while accomplishing the first phase of the program. These tasks are establishing system requirements and criteria and synthesizing a system mechanization to meet the desired load alleviation, stability margins, flight safety, and flying qualities performance. Results of the ALDCS development and prototype system flight simulation programs, and control law optimization including system stability, handling qualities and structural load analyses are presented, along with concluding remarks relative to the system design integration.

INTRODUCTION

An Active Lift Distribution Control System (ALDCS) has been developed by Lockheed-Georgia Company under the direction of the USAF C-5 System Project Office to reduce wing fatigue damage due to incremental maneuver and gust load sources.

The ALDCS is an automatic flight control subsystem which provides redistribution of the wing spanwise lift through symmetrical deflection of the ailerons by inclusion of control inputs to the existing lateral augmentation subsystem. The net aileron control effect, as illustrated in figure 1, is to shift the wing spanwise center of pressure inboard, thus reducing the incremental wing root bending moments. Control input signals from the ALDCS are also provided to the inboard elevator surfaces through the existing pitch augmentation subsystem for reduction of gust induced loads and to compensate for the resulting degradation in airplane handling qualities.

Although the primary objective of the ALDCS is to reduce wing loads, minimizing the effects on the basic aircraft stability and handling qualities and

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minimizing changes to existing hardware while utilizing existing control surfaces were also basic design goals.

SYMBOLS AND SUBSCRIPTS

N_Z	Normal acceleration load factor.
$\dot{\theta}$	Pitch rate.
δ_F	Flap position.
M_X	Bending moment.
A/C	Aircraft
$\bar{q}$	Equivalent dynamic pressure.
g	Acceleration constant (32.2 ft/sec^2)
M	Mach number
Ve	Equivalent Airspeed
CADC	Central Air Data Computer.
C.G.	Center of gravity.
db, DB	Decibel
ECP	Elevator cable position.
H.Q.	Handling qualities.
H_Z	Hertz.
K	One thousand.
KCAS	Knots calibrated airspeed.
M_H	Maximum horizontal flight Mach number.
PLDCS	Passive Lift Distribution Control System.
PSF	Pounds per square foot.
PSD	Power spectrum density.
RMS	Root mean square.

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SYMBOLS AND SUBSCRIPTS (CONT'D)

V_D Maximum dive flight airspeed

SL   Sea level.

V_H Maximum horizontal flight airspeed.

VSS   Vehicle systems simulator.

W.S.   Wing station.

BACKGROUND

In 1969 the Lockheed-Georgia Company conducted a program to establish the feasibility of reducing the maximum C-5 wing upbending loads during accelerated flight maneuvers. This effort consisted of development, fabrication and flight test of a prototype subsystem referred to as the Maneuver LDCS (MLDCS). This subsystem successfully reduced the inner wing bending moments for positive accelerations above 1.5g without degrading airplane handling qualities. A simplified version of this system known as Passive LDCS (PLDCS) that involves manual aileron uprig through the trim system was selected for the C-5 fleet incorporation.

In 1972 a survey conducted by the C-5 Structural Independent Review Team (IRT) of the possible methods to improve the C-5 wing fatigue life characteristics included a recommendation to consider an active control system to improve fatigue life. A decision was made jointly by the USAF C-5 Systems Project Office and Lockheed-Georgia Company to develop and test such a subsystem which was to be called an Active Lift Distribution Control System. This subsystem was to be incorporated in addition to the PLDCS. In May of 1973 the ALDCS program was initiated for the development and test of a prototype subsystem with flight testing to be completed in July of 1974. The results of this program will affect a decision to produce the ALDCS for C-5 fleet retrofit.

DEVELOPMENT METHODS

A flow chart of the tasks required in the ALDCS development are shown in figure 2. Each task required direct involvement of a number of engineering disciplines to insure adequate assimilation of design requirements and data and proper maintenance of development results and the status of the subsystem mechanization. One of the paramount challenges was the integration of the affected design disciplines into a total design team since the functioning of this active subsystem had such interwoven influences on loads, handling qualities, stability, structural dynamics, and existing C-5 flight control systems. Fortunately, the experience of the earlier LDCS program provided an excellent design example.

Requirements and Criteria

Prior to synthesizing the ALDCS, design requirements and criteria were carefully established as a design base in the areas of structural loads, flight control subsystems, stability, and handling qualities. These requirements are:

Structural Loads -

° Continuous turbulence loads analysis shall result in RMS bending moments at the wing root (wing station 120) not exceeding 70% of the free airplane values.

° The continuous turbulence RMS torsion at the wing root shall not exceed the free aircraft values by more than 5%.

° The ALDCS shall not increase discrete gust loads.

° The incremental root bending moments load per g shall not exceed 70% of the free aircraft values during steady maneuvers, within the normal climb, cruise, and descent regime of the aircraft.

° The ALDCS shall produce no aileron input when the aircraft reaches the design positive maneuver load factor of 2.5.

° The system shall not be required to operate in the flaps down configurations.

° The ALDCS shall operate in the required speed/altitude flight envelope as defined in figure 3 for flaps up configurations.

Flight Control Subsystems -

° The ALDCS shall be designed to "fail-safe" concepts.

° The system shall be a dual channel analog design.

° Active operation of ailerons and inboard elevators through existing augmentation and primary control actuators are required.

° ALDCS will interface with existing C-5 sensors to the extent possible and will be compatible with existing C-5 automatic flight control subsystems.

° No ALDCS malfunction will affect normal pitch and lateral augmentation subsystem operations.

° The existing C-5 hydraulic servoactuators for the aileron and inboard elevators will be used without modifications.

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° The ALDCS will be required to operate on a "full-time basis" within the desired flight envelope and design criteria boundaries.

Stability -

The incorporation of the ALDCS shall not:

° Induce adverse structural mode coupling.

° Change significantly the existing maneuvering flight handling qualities.

° Induce significant degradation of existing flutter margins.

° Induce adverse coupling with existing flight control systems.

° Induce limit cycle tendencies.

The following ALDCS minimum stability margin and attentuation goals for each primary control surface feedback loop were established to meet the above system stability requirements. These goals were considered to be realistic and attainable throughout the ALDCS flight envelope.

°° Ground Test - 6 db gain margin and 45 degree phase margin.

°° Flight modes through control mode natural frequencies - 6 db gain margin and 45 degree phase margin.

°° Flight modes above control mode natural frequencies - 6 db gain margin and infinite phase margin. There was also a system attenuation goal of 60 db/decade established for these modes.

Handling Qualities -

° There shall be no significant change in the existing C-5 handling qualities.

° The ALDCS shall be disengaged prior to the aircraft stall event.

° Criteria for the C-5 handling qualities will be those characteristics established during previous flight test programs which concluded the C-5A flying qualities to be acceptable in all cases.

° Evaluation pilot comments will be utilized to obtain satisfactory results.

Design Data Acquisition

The task of acquiring necessary design data was simplified by the existence of airplane math model data, flight control subsystem mechanizations,

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and flight test response correlation data from the original C-5 design programs. The major void in design information existed in the characteristics of the aileron and elevator hydraulic servoactuators. This void existed due to the C-5 actuators being designed and tested primarily for handling qualities evaluations and automatic stabilization of aircraft low frequency short period and dutch roll modes, whereas the ALDCS would encompass the sensing and active control of higher frequency aeroelastic mode dynamics, potentially up to a factor of 15 above the short period frequency.

These missing actuator characteristics not only included frequency response but hysteresis, surface rates and tolerance bands in unloaded and loaded conditions. They were desired for actuators of various ages up to an expected full life. These data were obtained by tests on the C-5 Vehicle Systems Simulator of new and worn (over one life span) servoactuators, by tests performed by Bertea Corporation (the servoactuator manufacturer), and by frequency response flight tests on the C-5 aircraft.

A definite "design risk" was associated with the attempt to utilize existing C-5 servoactuators without bandwidth or authority limit modifications.

Computer Programs

Various computer programs were prepared and correlated with flight test data to provide analytical techniques for development of the ALDCS mechanization. These programs using hybrid and digital computation were:

° Stability - Eigenvalues and Frequency Response

° Dynamic Time History - Loads and Handling Qualities

° Accelerated Stability - Stick Force per 'g'

° PSD Loads

The following airplane and control system analytical models were used for the above programs.

° Three degrees-of-freedom quasi-elastic longitudinal axis dynamic models.

° Six degrees-of-freedom quasi-elastic longitudinal and lateral-directional axes dynamic models.

° Eighteen mode aeroelastic symmetric axis dynamic models, with first 15 flexible modes and Wagner and Kussner functions and gust penetration effects.

° Two degrees-of-freedom quasi-elastic steady-state maneuver model.

° Eight mode aeroelastic symmetric axis dynamic model with six most significant flexible modes.

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° Linear and non-linear flight control system servoactuator models.

Analysis and Synthesis Tasks

The analysis and synthesis tasks involved the development of an ALDCS mechanization to meet the load alleviation requirements and the determination of its effects on stability, handling qualities and existing flight control subsystem performance. Feedback control laws were synthesized to attain these requirements while minimizing system coupling effects with undesirable structural modes and rigid body dynamics.

Development of a realistic mechanization that could potentially be utilized as a guide for production design required indepth studies to establish the system's total flight envelope functional characteristics, sensor tolerance and response specifications, and prototype parameter adjust capabilities. Also involved were the analyses to determine effects of subsystem failures, component tolerance build-up, and servoactuator response characteristics. Other major analytical studies were accomplished to determine the impact of the ALDCS on handling qualities in the following areas:

° Dynamic Stability

° Maneuverability (Attitude Control)

° Accelerated Stability (Stick Force per 'g')

° Roll Control Performance

° Development of an ALDCS Handling Qualities Command Model

The interaction coupling effects of the flexible bending and rigid body response with the flight control system was thoroughly analyzed. This insured proper control law compensation for those flight conditions during which structural modes and handling qualities tend to degrade each other.

Flight Simulation Tasks

Tasks accomplished on the C-5 Developmental Handling Qualities Cockpit Simulator provided pilot evaluations of the ALDCS effect on the C-5 handling characteristics. The inflight tasks performed by the evaluating pilot consisted of the following:

° Symmetric 'g' pull-ups

° Stabilized bank turns and roll-outs

° Landing approach and flare

° Constant 'g' rolling pull-out maneuvers

° Take-off rotations

° Attitude tracking maneuvers during turbulence

° Air traffic control maneuvering (speed,
  altitude and heading changes)

The C-5 Developmental Handling Qualities Cockpit Simulator is real-time
six degrees-of-freedom simulation with an all digital computation and a termi-
nal area terrain model visual system.

Vehicle System Simulator (VSS) Tasks

Simulation afforded the capability to verify the prototype design and
system safety aspects in functional operation checkout and flight control sub-
system hardware integration. This technique also provided final pilot evalua-
tions utilizing the prototype subsystem. Pilot tasks were similar to those
used on the C-5 Developmental Handling Qualities Cockpit Simulation discussed
previously.

The VSS incorporates actual C-5 mechanical and hydraulic flight control
systems, moving surfaces and interfacing automatic flight control subsystems.

SCHEDULE MILESTONES

The accomplishment of the analysis, synthesis, simulation and design tasks
to meet a restrictive schedule was paramount. Flight test evaluations of the
prototype ALDCS were to begin within eleven months from contractual go-ahead.
Figure 4 illustrates the criticality of the design program schedule. With
go-ahead occurring on 7 May 1973, the subsystem design met the 90 percent func-
tional release date of 21 September 1973. The final mechanization was released
on the scheduled date of 7 November 1973 and the first prototype subsystem was
made available for flight simulation evaluation on 7 January 1974. Inflight
system evaluations began on 15 March 1974, approximately ten months after go-
ahead.

SYSTEM MECHANIZATION

The ALDCS has been mechanized to meet the demanding requirements placed
on it and to interface with existing C-5 sensors, augmentation and servo-
actuation subsystems.

Figure 5 provides a simplified interface diagram indicating the integration of the ALDCS computer with the existing C-5 flight control subsystems. The dual channel redundancy design ALDCS computer provides signals to both the lateral augmentation series servo to control the aileron actuators symmetrically and the pitch augmentation series servo to actuate the inboard elevator control surfaces. Aileron actuators also receive commands from the pilots, autopilot, and passive LDCS. The pilots and autopilot command inboard as well as outboard elevators. Figure 6 shows the C-5 airplane locations of the ALDCS sensors and interfacing computers and affected control surfaces. The wing mounted accelerometers are the only additional C-5 sensors required for ALDCS integration.

The ALDCS mechanization consists of an array of sensors, gains, and filters. Figure 7 is a block diagram of the ALDCS simplified mechanization to be used as a roadmap during the insuing discussion of the individual components and system development changes. The aileron and elevator channels will be discussed separately.

Aileron Channel

The aileron control channel commands the right and left ailerons symmetrically to accomplish the maneuver load relief function. The feedback sensors utilized for the aileron channel are provided by two vertical accelerometer locations per wing, one located on the forward main beam (W.S. 1186) and the other on the rear beam (W.S. 1152) both at an outer wing location. The signals from these accelerometers are averaged and compensated by smoothing filters that attenuate sensor noise and aid in the elimination of higher frequency wing vibration modes beyond the ALDCS control bandwidth.

The Stability and Load Control Gain and Filtering portion of the aileron channel provides the necessary compensation to adequately phase the feedback accelerometer signals for control of the inner wing bending moments and to attain the design goal stability margins.

A pilot's feedforward command, acquired from the existing C-5 elevator cable position (ECP) transducer, is summed with the compensated acceleration control signal to provide abrupt maneuver load control. The feedforward signal is filtered for proper abrupt load alleviation aileron command phase.

These control signals are then gain scheduled by aircraft dynamic pressure from the Central Air Data Computer (CADC) to provide proper stability and load relief schedules and to minimize handling qualities degradations throughout the aircraft speed envelope. Cut-off filters are provided to preclude adverse coupling with higher frequency uncontrolled modes. The ALDCS aileron command signal is controlled by boundary control logic which contains the circuitry to disengage the signal when exceeding flight boundaries where the ALDCS is not required. These operational boundary conditions are when the flaps are lowered, the Stallimiter subsystem is activated, the airplane exceeds maximum

horizontal airspeed/Mach (350 KCAS /M = 0.825), and when the airplane load factor exceeds 1.9 g's. These logic control signals are obtained from existing aircraft subsystems with the exception of load factor. This signal is derived from ALDCS wing and fuselage accelerometers to closely represent aircraft C.G. acceleration. The system is automatically re-engaged as the aircraft re-enters the ALDCS operational envelope. The aileron command signal is then limited and interfaced with the lateral SAS aileron series servoactuators.

Elevator Channel

The elevator channel contains three sensors, two active feedback parameters and one feedforward command. Airplane pitch rate, as provided by the pitch SAS rate gyro, is utilized to augment the airplane short period damping and thereby alleviate the excitation of short period induced gust loads and to restore the handling qualities degraded by the aileron pitching moment effects.

An existing C-5 autopilot subsystem vertical accelerometer mounted in the forward fuselage provides additional gust load control and compensates the airplane pitch response characteristics.

A feedforward signal, pilot's elevator input command, is required to restore the airplane maneuverability and accelerated stability (stick force per 'g') characteristics that are significantly degraded by the load control signals. This signal is scheduled as a function of airplane dynamic pressure and compensated by a command model filter to provide the proper system handling qualities throughout the operational envelope.

These three signals, pitch rate, normal acceleration and pilot elevator command input are summed and again scheduled with dynamic pressure and passed through system cut-off filters for stability and gust load control phasing.

The elevator signal is provided to a boundary control logic network that disengages the signal under the same conditions as the aileron channel. This circuit includes a fade-out filter to minimize acceleration transients resulting from abrupt surface disengagement. The command signal is then limited and interfaced with the pitch augmentation subsystem.

System Changes

The functional development of the ALDCS provided the usual subsystem changes which caused agonizing perturbations in the design of the prototype subsystem hardware. These modifications of the mechanization fall into the following major areas:

° Wing accelerometer location

° Operational flight envelope

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° Subsystem stability - filter compensation

Wing Accelerometer Location -

Trade studies were accomplished to determine the number and locations of the wing mounted accelerometers. The C-5 wing locations acceptable to sensor installation are essentially limited to the front and rear beams due to fuel tank locations. Original studies of the wing accelerometer location indicated the need for two sensors per wing, one on the mid-wing aft main beam and one in the outer wing to be mounted on the front main beam. These sensors were to provide "high gain" feedback control of the first and second wing flexible bending modes. Additional studies proved the "high gain" system design to be impractical and that the second wing mode did not contribute significantly to gust loads, thus the mid-wing sensor locations were eliminated. This removal and relocation of the outer wing front beam accelerometer to the rear beam, caused a favorable influence on subsystem stability and allowed the maneuver and gust load control functions to be simply combined with reduced gains in the aileron channel.

Later a second accelerometer was placed in its present location on the front beam to minimize a 48 radian per second outer wing coupling mode that, in turn, increased the stability margins and eliminated an original need for complex notch filtering. Figure 8 indicates the effect of single and blended multiple accelerometer locations on the ALDCS aileron closed loop frequency response. The rear beam sensor permits an amplitude gain peak of 7 db at 48 radians per second. The addition of the front beam accelerometer adequately blended with the rear accelerometer to simulate the critical 48 radians per second node location, reduces this peak to approximately one db. An external wing accelerometer installation was considered; however, the additional cost and associated design risks eliminated this design.

Operational Flight Envelope -

To insure proper functioning of the ALDCS throughout the required flight envelope, gain scheduling and subsystem disengagement are necessary. The original subsystem mechanization required complex nonlinear scheduling interfaces with the central air data computer. As the development progressed these schedules were simplified to linear functions. Also an original ALDCS requirement for flaps down operation was deleted, thereby eliminating the need for flap gain schedules and automatic landing interfaces. These functions were replaced by a flaps down boundary logic control disengagement signal. Another change necessitated by flight envelope requirements was the development of a fader to smoothly disengage the subsystem when the airplane exceeds the boundary condition of normal acceleration, stall approach, and speed/Mach. Acceptable handling qualities were attained at these boundary conditions with a simple track and fade-out circuit in the elevator channel.

Subsystem Stability - Filter Compensation -

The problem of subsystem stability followed the mechanization development throughout the program in both the aileron and elevator channels. Perturbations in the mechanization occurred continually with the altering of filter compensation. Major modifications were the elimination of original design notch filtering and the additions of simple first order stability filters to improve a 2.4 Hertz stability margin in the aileron channel and the inclusion of a low pass stability and fuselage load control phasing filter in the elevator channel.

SUBSYSTEM PERFORMANCE

The ALDCS as mechanized has provided the load alleviation requirements without significantly interfering with airplane stability, handling qualities, autopilot performance or flight safety. The performance, as discussed in the following paragraphs, has been obtained utilizing existing C-5 aileron and inboard elevator control surfaces, without modification to the primary servo-actuators.

Maneuver and Gust Loads

The resulting ALDCS maneuver and gust loads performance data are summarized in figures 9 through 12. These performance results indicate that the incremental load relief meets the design criterion of attaining 30 percent bending moment reduction at the wing root, while not exceeding five percent torsional increase during continuous turbulence flight.

The steady maneuver incremental wing root load per 'g' ratios of ALDCS on to the basic aircraft are presented in figure 9. This summary covers a typical cruise payload configuration of 160,000 pounds and 94,250 pounds of fuel for a variation of Mach number and altitude. With ALDCS operative, these results indicate inner wing load reductions of 32 to 52 percent. The basic design goal ratio of 0.70 was achieved for all configurations within the normal C-5 operational speed, altitude and payload flight envelopes.

A typical wing root bending moment gust frequency response and PSD output spectrum are shown for the airplane with and without ALDCS in figure 10. The ALDCS gust output spectrum is significantly reduced from that of the free airplane. The transfer function shows that the first vertical wing bending mode amplitude at 0.9 H_z is reduced to approximately one-half with ALDCS operative. ALDCS control bandwidth encompasses primarily the short period and first wing bending airplane modes through the frequency of approximately one H_z.

Wing root RMS bending and torsional moment ratios of ALDCS on to ALDCS off, for a variation of altitude and Mach numbers, are given in figures 11 and 12. The ALDCS reduces the RMS wing root bending moments by 30 to 50 percent of the free airplane without increasing the torsional moment by more than the design goal of 5 percent for any case. The torsional moment is less than that of

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the basic airplane for the majority of flight cases investigated.

Loads criteria for discrete gust were only specified to the extent that the ALDCS shall not increase the basic airplane discrete gust loads. Seven flight cases, similar to those presented in figure 9, were analyzed for the "1-cosine" discrete gust model. The wing root bending moment peaks, with ALDCS on, were reduced to values ranging from 78 to 52 percent of the free airplane for the critical gust frequency wavelengths.

Although no criteria were established for abrupt maneuver load control, analyses were conducted to evaluate the effect of ALDCS on abrupt maneuver load control characteristics. These analyses, conducted for seven selected flight conditions, revealed that the load reduction was from one to seventeen percent depending upon the particular flight case response characteristics. In an effort to improve this performance, a feedforward pitch control command signal was provided to the aileron channel. Results of analysis with the aileron feedforward signal for a selected number of cruise flight conditions indicated that the wing root bending moments could be reduced by 30 percent of the basic airplane. This feedforward signal mechanization was then incorporated in the ALDCS prototype system for flight test evaluation.

Fuselage loads performance was monitored during the continuous turbulence analysis to evaluate the effects of ALDCS. Results indicated that the aft fuselage bending moments were being increased up to 15% over the free airplane. A low-pass filter was added to the elevator channel that increased stability margins and decreased the aft body fuselage bending moments below those of the basic airplane for all cases.

Stability

The concern that the ALDCS possess adequate stability gain and phase margins caused considerable design optimization attention. This requirement was accomplished as indicated in figures 13 and 14. These gain and phase margins represent a series of reserve fuel loading cases that inherently possess the minimum aileron loop stability. The elevator loop stability is minimum with a high fuselage cargo loading, but in no cases were the phase margins less than 64 degrees or the gain margins less than 10 db.

The gain margins for both aileron and inboard elevator channels are well above the minimum requirement of 6 db for all cases.

The only flight case found to have the minimum phase margin of 45 degrees was that of a high altitude, reserve fuel and maximum ALDCS operational Mach number of 0.825. As fuel weight is added to this configuration, the aileron gain and phase margins are increased. A fuel capacity of approximately 30

percent for this case has a gain margin of 16.5 db and a phase margin of 62 degrees.

Minimum aileron gain and phase margins for all configurations occur at frequencies between 33 to 53 radians per second and between 6 and 17 radians per second, respectively. The minimum elevator gain margins for all configurations occur at frequencies between 6 and 8.6 radians per second with the phase margin frequencies ranging from 0.6 to 3.41 radians per second.

Handling Qualities

A basic ALDCS design goal was that there would be no significant degradation of the existing C-5 handling qualities. Extensive analysis and pilot-in-the-loop flight simulation evaluations were accomplished to insure that the ALDCS was compatible with the C-5 flying characteristics.

The handling quality areas of most concern that could be altered or significantly degraded by the ALDCS were:

° Maneuver response

° Accelerated stability-stick force per 'g'

° Short period stability

° Phugoid stability

° Roll performance

Development of an ALDCS elevator channel pilot command model filter was essential to retain the C-5 maneuver response and stick force per 'g' characteristics. ALDCS short period and phugoid stability effects were compensated by appropriate system gain and filter parameter optimization. The roll performance effect was greatly reduced by using the minimum aileron channel gain schedule required for maneuver load control.

The time histories shown in figure 15 present the effects of ALDCS on airplane normal C.G. acceleration and pitch rate responses for a typical pull-up maneuver. The input forcing function for this maneuver is a constant control force rate and hold after 3 seconds. This figure shows that the time to obtain steady-state maneuver values are practically the same with ALDCS off or on. The only difference with ALDCS on is that of a slight undershoot in peak pitch rate and a slight rise time improvement to acquire the steady state response. Simulator pilot evaluations of these type maneuvers indicated no degradation in airplane handling quality performance.

The longitudinal axis accelerated maneuvering stability, as shown in figure 16, was not significantly impaired by the ALDCS. The ALDCS stick force per 'g' values are well within the demonstrated boundaries of previously extracted

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flight test data without ALDCS. The steady-state elevator command model gain was optimized to provide identical stick force per 'g' characteristics for mid C.G. flight configurations with ALDCS on or off. Pitch column force required to hold a given acceleration for forward and aft C.G. with ALDCS on are slightly decreased and increased, respectively from the basic airplane. The simulator pilots were unable to distinguish these ALDCS characteristics from those of the basic airplane.

No short period and phugoid stability damping degradation was noticed during the development flight simulation program and analytical results, as presented in figures 17 and 18, confirm the pilot evaluations. The original basic C-5 short period damping requirement for the cruise configuration was that it shall damp to one-tenth amplitude within one cycle. This requirement has been exceeded by the basic airplane and is slightly more damped with ALDCS operative.

The phugoid mode, as shown in figure 18 exhibits sufficient stability, although the frequency is slightly reduced from that obtained from previous flight test data correlation studies. The original C-5 phugoid stability requirement was that if the period is less than 15 seconds, then this mode shall be at least neutrally stable. Data shown in figure 18 does not indicate any frequencies with periods less than approximately 65 seconds with ALDCS on.

There was a concern early in the development program, that the ALDCS would reduce the C-5 roll performance. This concern arose primarily due to symmetrical control of ailerons with high acceleration gains that may cause actuator saturation. Theoretically, there is a slight decrease in available roll power due to aileron saturation; however, flight simulation evaluations determined that the pilots could not detect this degradation. For maximum roll rate maneuvers, the simulation pilots would mask ALDCS effects by commanding ailerons for a slight additional amount of time to perform the same maneuver.

The following handling qualities pilot opinions were attained during the ALDCS development and prototype Vehicle System Simulation Program.

○ Ease of trimming to new speed - no degradation.

○ Phugoid and short period damping - no degradation.

○ Roll power - no noticeable degradation.

○ Stick force per 'g' characteristics - no degradation.

○ ALDCS fails to switch off - no degradation with flap extension.

A total of six pilots, including two from the Air Force, flew the development simulator with ALDCS on and off.

The effect of ALDCS on the C-5 handling qualities can be summarized by the fact that the simulation pilots were unable to detect whether the ALDCS was on or off during evaluations within the normal flight envelope.

Autopilot Compatibility

The ALDCS is designed to be engaged during autopilot operation, thus considerable design attention was directed to subsystem compatibility. This development was concentrated on autopilot interface stability, response performance and flight safety. It was found necessary that the ALDCS elevator channel control signals of pitch rate and pilot's feedforward command be disengaged during autopilot operation. Elimination of these control signals during autopilot operation improved the stability margins and minimized control wheel steering sensitivity, and airplane acceleration response due to an autopilot hardover failure.

Results indicate no apparent degradation in either stability or response of the autopilot attitude, altitude hold or control wheel steering modes. The effect of ALDCS on autopilot altitude hold and roll performance was insignificant with the airplane achieving limit bank angle with minimum altitude loss. Pitch autopilot hardover failures, with ALDCS engaged, yield a normal acceleration response slightly below that of the basic airplane and autopilot.

Flight Safety

To insure that ALDCS faults would not affect the C-5 flight safety, failure effects analysis and prototype vehicle system simulation evaluations were accomplished. These failures involved loss of ALDCS sensor signals, loss of ALDCS, hardovers of sensors and channel loop commands, gain schedule failures, and various stability augmentation subsystem (SAS) failures that could be effected by the ALDCS.

The analysis and simulator testing indicates that the ALDCS adequately meets the safety requirements and criteria. There is sufficient subsystem stability should any one sensor or channel in the ALDCS be lost. Neither of the various SAS failures were worse than those of the existing system; however, some failure detection and airplane transient improvement was exhibited with ALDCS operative.

Results of these studies indicated that there were no single ALDCS or automatic flight control interface failures that caused pilot concern. Adequate fault detection and annunciation of these failures was apparent to the pilot. The ALDCS has met the basic safety criteria and is acceptable for prototype development flight testing.

Ride Control

No real attempt was made during the ALDCS development program to improve the C-5 ride control characteristics. The pilot's station acceleration levels were monitored throughout the continuous turbulence analysis however, to insure that the ride quality was not adversely affected by the ALDCS.

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Results of these analyses revealed that the pilot's acceleration levels were reduced by 7 to 35 percent throughout the C-5 ALDCS flight envelope.

CONCLUDING REMARKS

A prototype maneuver and gust load alleviation control system has been successfully developed, fabricated and simulator tested meeting demanding schedules and functional requirements. It is felt that a major airplane active control subsystem integration accomplishment has been achieved by integrating the ALDCS into the total C-5 Vehicle System while maintaining compatibility with existing airplane stability, handling qualities, and flight control subsystems. While no specific requirements were established, it is noteworthy that the ALDCS has favorably influenced the pilot station accelerations (ride control), abrupt maneuver load control, aft fuselage gust loads, and some failure detection levels of interfacing automatic flight control subsystems.

Now as the Active Lift Distribution Control Subsystem enters development flight test evaluations the development engineers and the design personnel from the affected disciplines confidently feel that the subsystem will continue to meet its design objectives. These design engineers have integrated their experience, development techniques, and computer programs to meet a very restrictive schedule. The success of this development program can largely be attributed to the fact that the prototype systems were primarily designed and fabricated within the structure of one company.

It is planned, if successful in flight test, that the ALDCS be produced and retrofitted to the C-5 fleet. This ALDCS development program, even though it is not a true preliminary design application of active control technology, has provided an understanding of the problems facing the designer and the experience and design techniques needed to apply active controls to aircraft of the future.
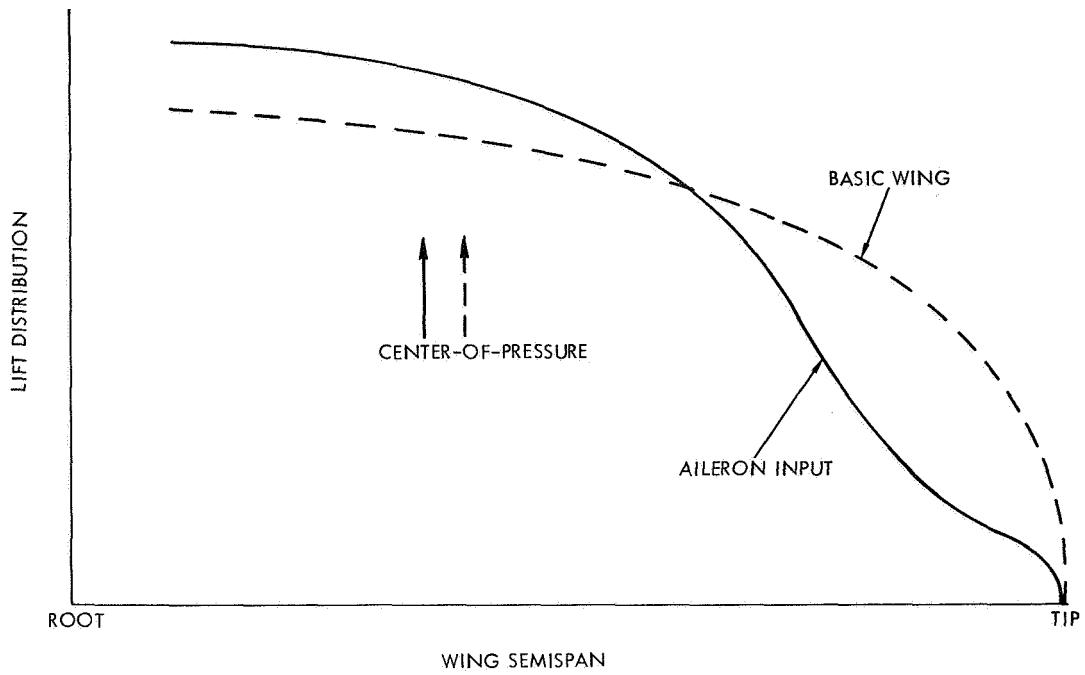
Figure 1.- Effect of aileron control on C-5
wing lift distribution.
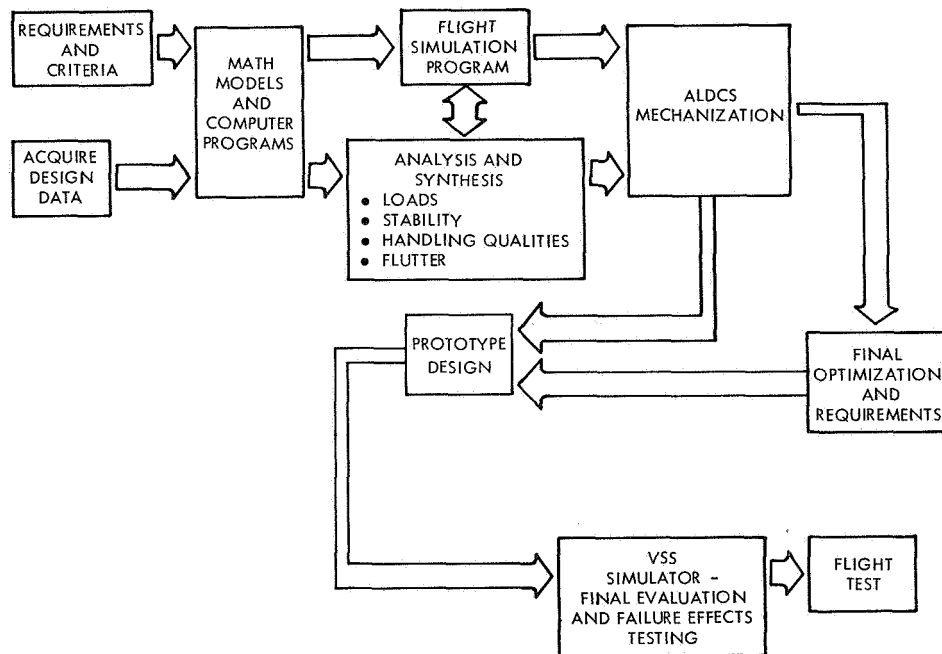
Figure 2.- C-5 ALDCS development program
flow diagram.

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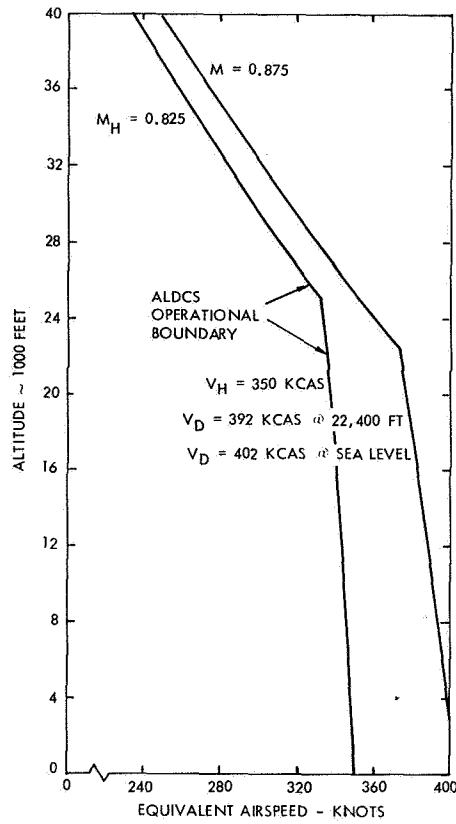
Figure 3.- C-5 ALDCS speed altitude envelope.

Figure 4.- C-5 ALDCS development program
schedule milestones.
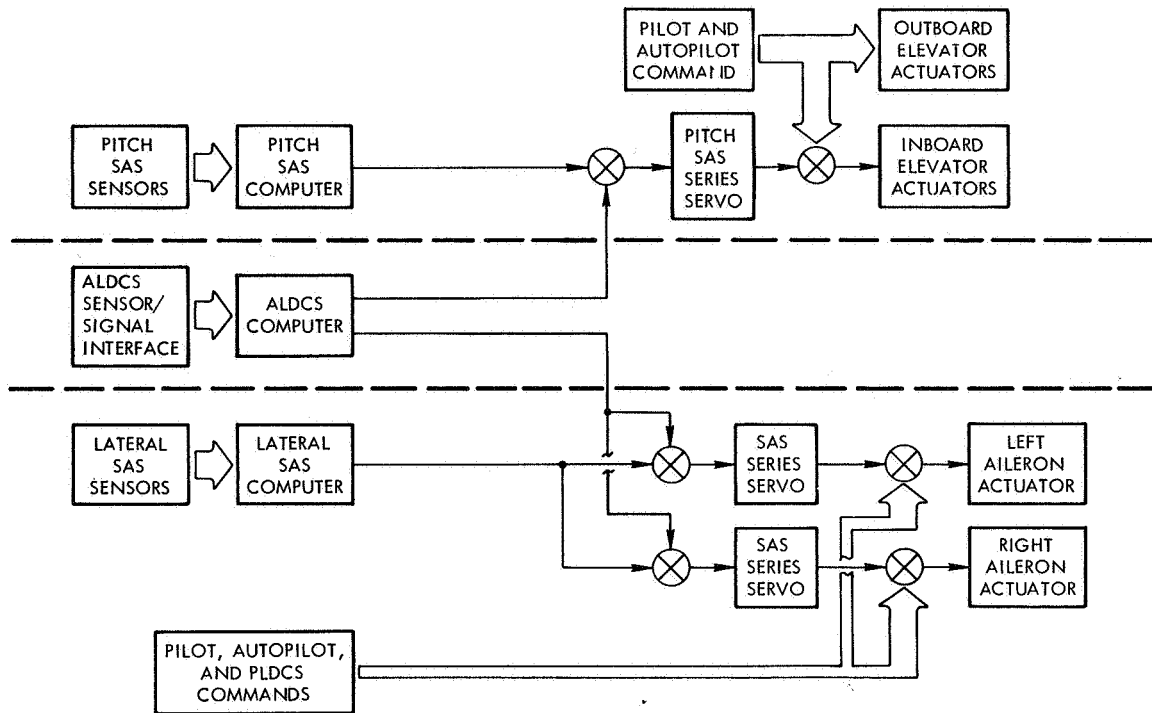
Figure 5.- ALDCS flight control system interface diagram.

MAJOR AIRCRAFT INTERFACE SUBSYSTEMS

A   ALDCS COMPUTER
B   CADC COMPUTER
C   PITCH SAS COMPUTER
D   YAW/LATERAL SAS COMPUTER
E   STALLIMITER
F   AUTOPILOT
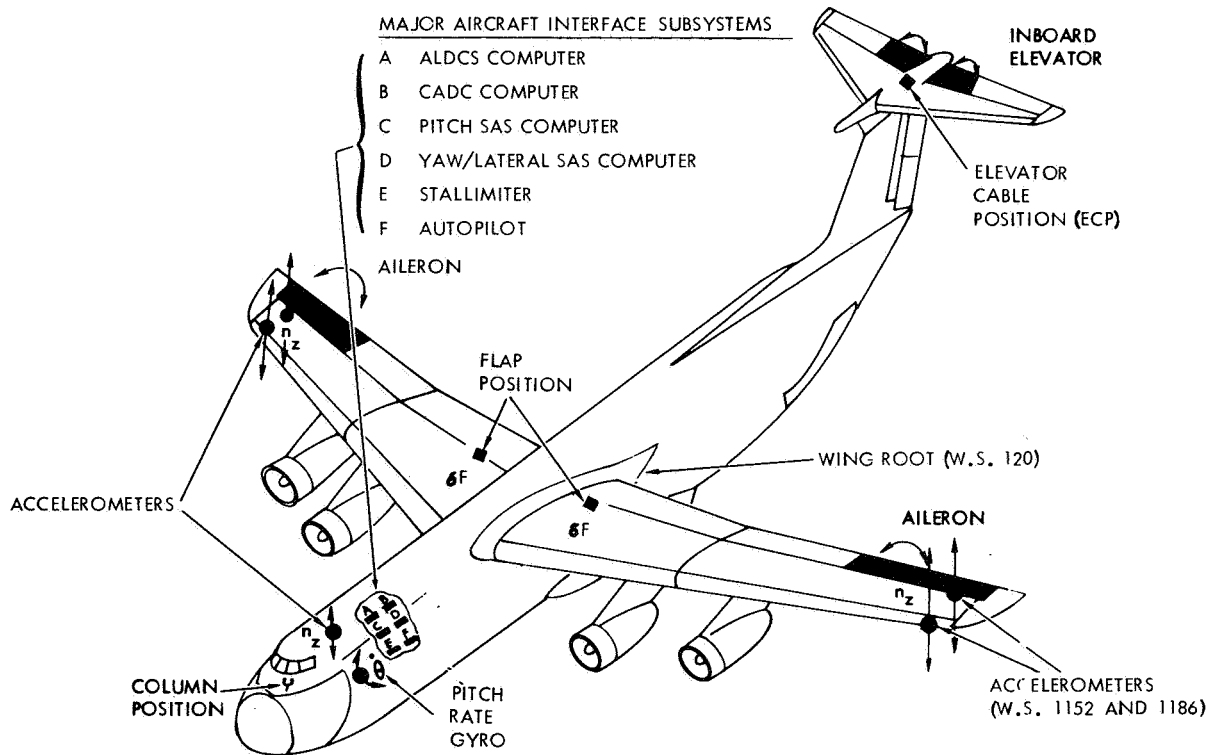
Figure 6.- C-5 ALDCS major airplane components interface.

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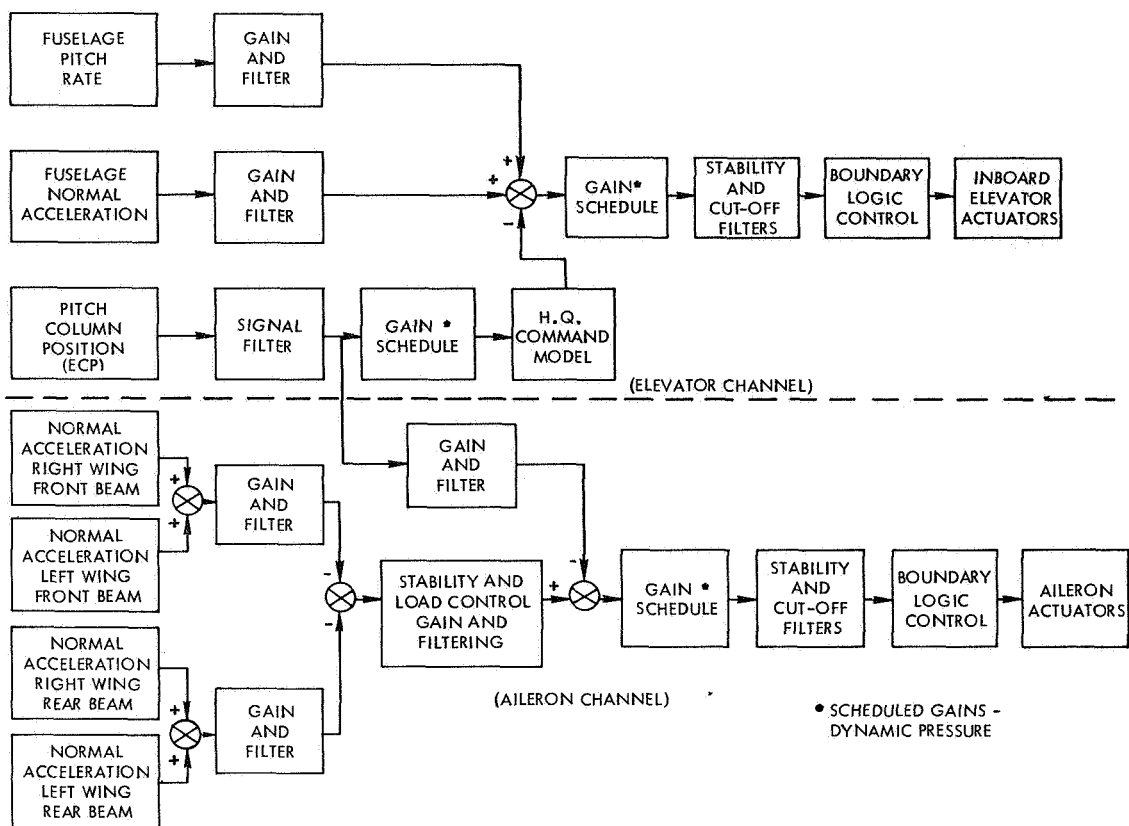
Figure 7.- C-5 ALDCS simplified functional
block diagram.
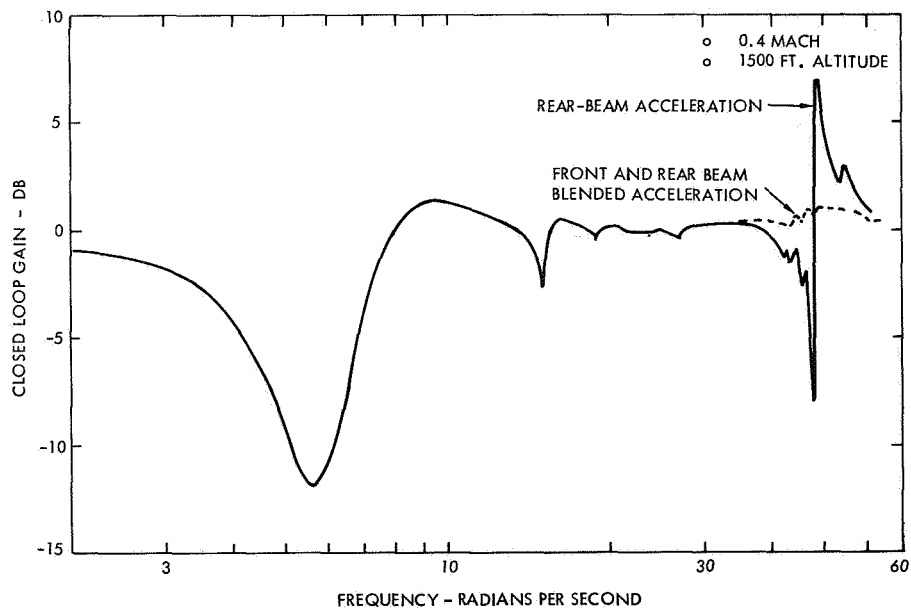
Figure 8.- C-5 ALDCS aileron closed-loop
frequency response.

FUEL WEIGHT 94,250 LB CARGO WEIGHT 160,000 LB

ALTITUDE (FEET)	MACH	$\dfrac{\Delta M_X/G \text{ (ALDCS)}}{\Delta M_X/G \text{ (NO ALDCS)}}$
SEA LEVEL	0.30	0.55
SEA LEVEL	0.40	0.48
SEA LEVEL	0.50	0.52
12,000	0.40	0.64
12,000	0.50	0.58
12,000	0.60	0.55
26,000	0.60	0.67
26,000	0.70	0.66
26,000	0.75	0.65
40,000	0.72	0.68
40,000	0.77	0.63
40,000	0.82	0.61

FIGURE 9. — C-5 ALDCS WING ROOT BENDING MOMENT RATIOS — STEADY MANEUVER

o FUEL WT = 94,250 LB
o CARGO WT = 160,000 LB
o MACH = 0.400
o ALT = 1500 FT

FREE AIRCRAFT
— — — ALDCS
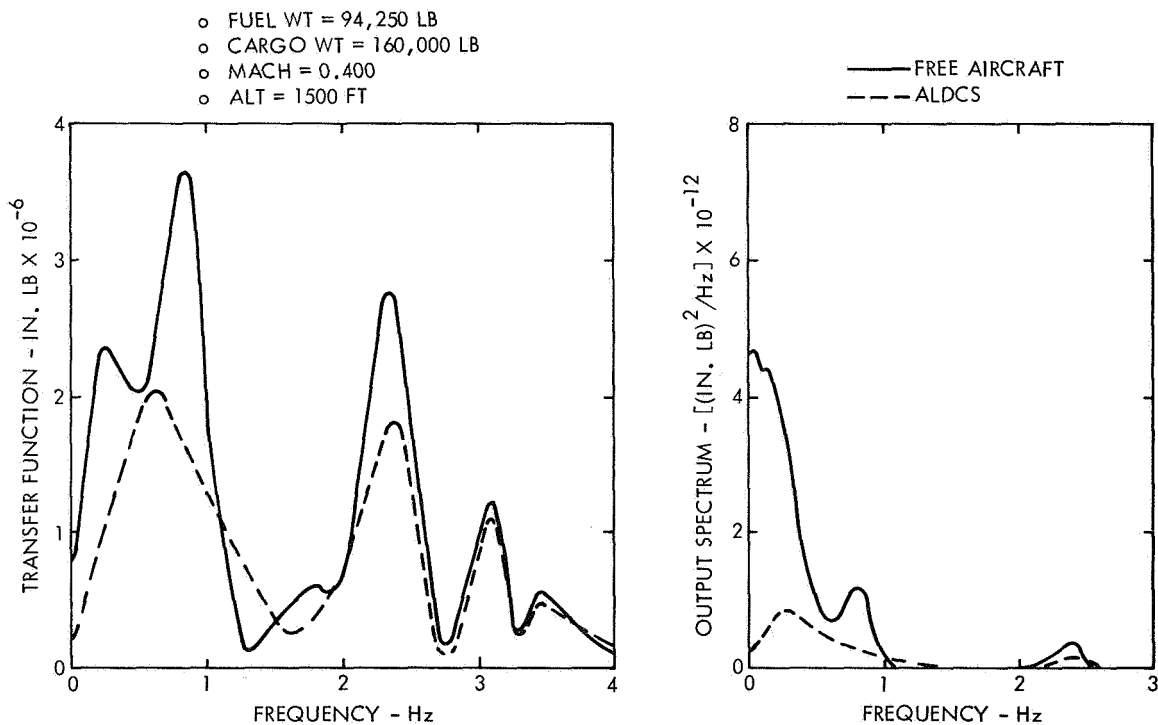
Figure 10.— C-5 ALDCS wing root bending moment — 1 fps RMS vertical gust.

FUEL WEIGHT 94,250 LB

CARGO WEIGHT 160,000 LB
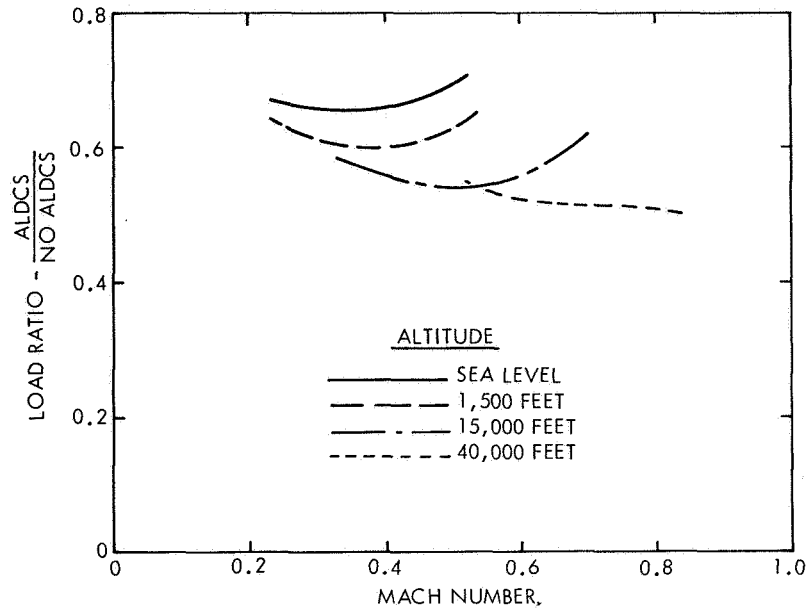
Figure 11. - C-5 ALDCS wing root gust RMS
bending moment ratio.

FUEL WEIGHT 94,250 LB

CARGO WEIGHT 160,000 LB
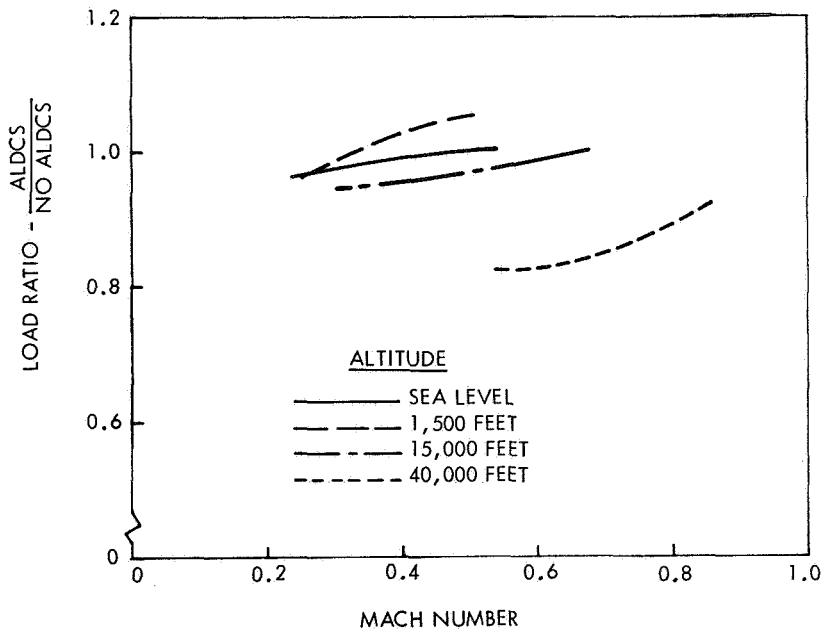
Figure 12. - C-5 ALDCS wing root gust RMS
torsion moment ratio

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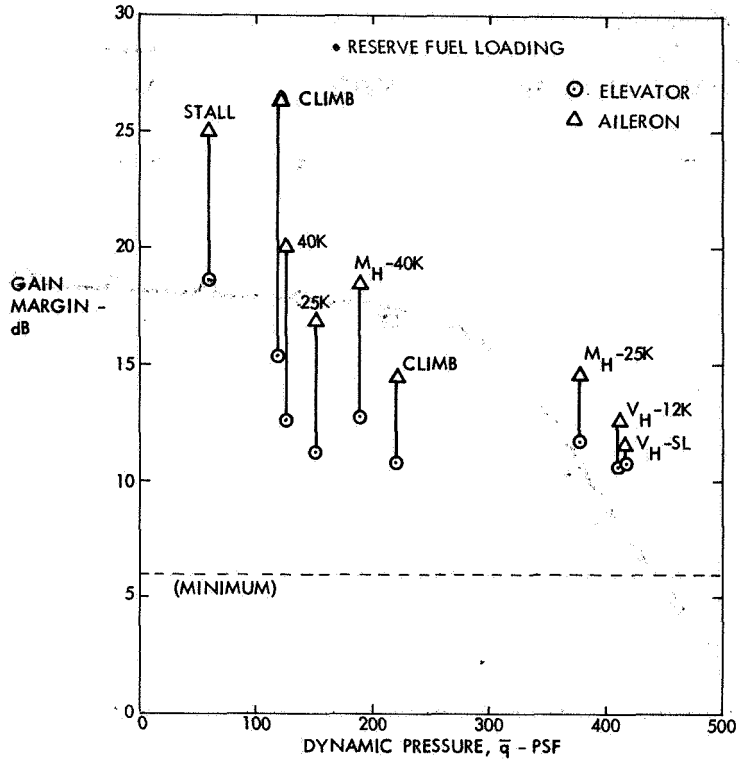
Figure 13.- C-5 ALDCS stability gain margins.
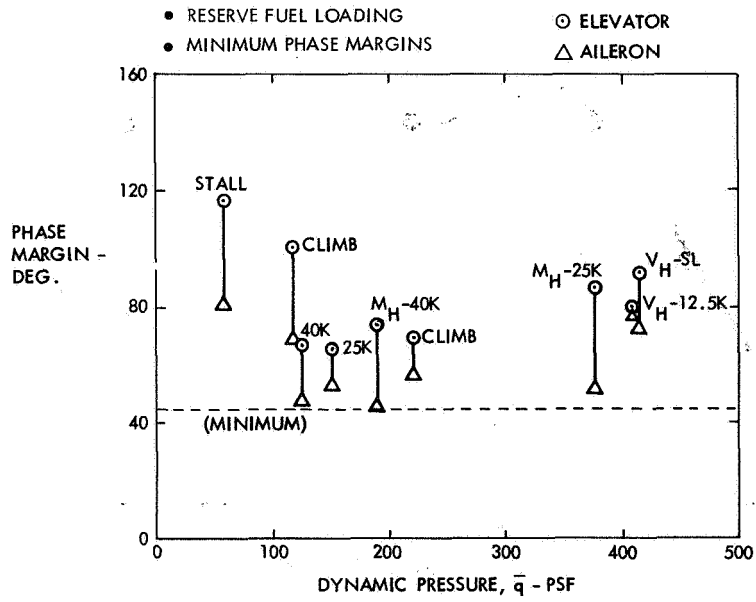
Figure 14.- C-5 ALDCS stability phase margins.
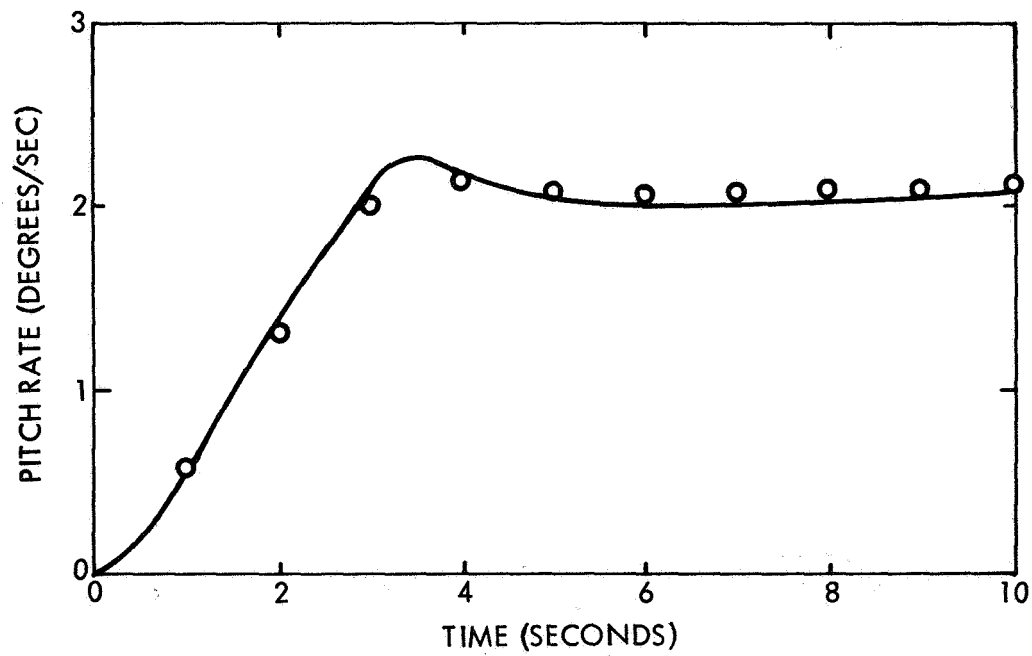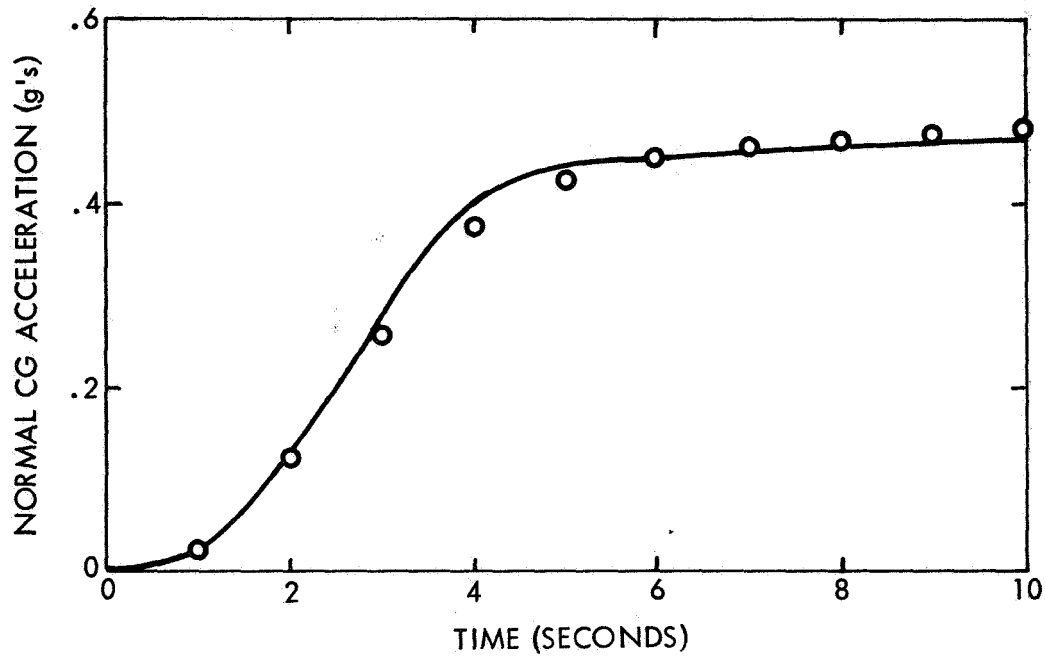
Figure 15.- C-5 ALDCS symmetric pull-up time history.
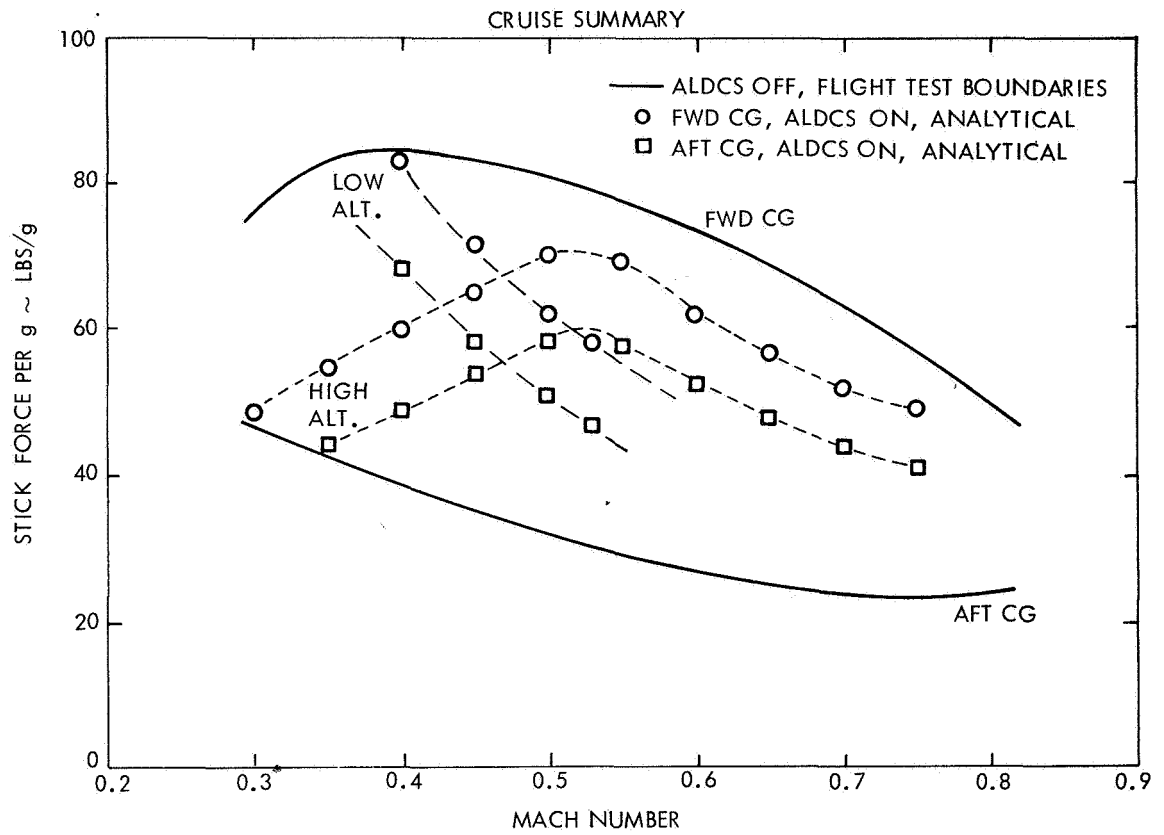
Figure 16.- C-5 ALDCS maneuvering longitudinal axis
stability - Stick force per g.
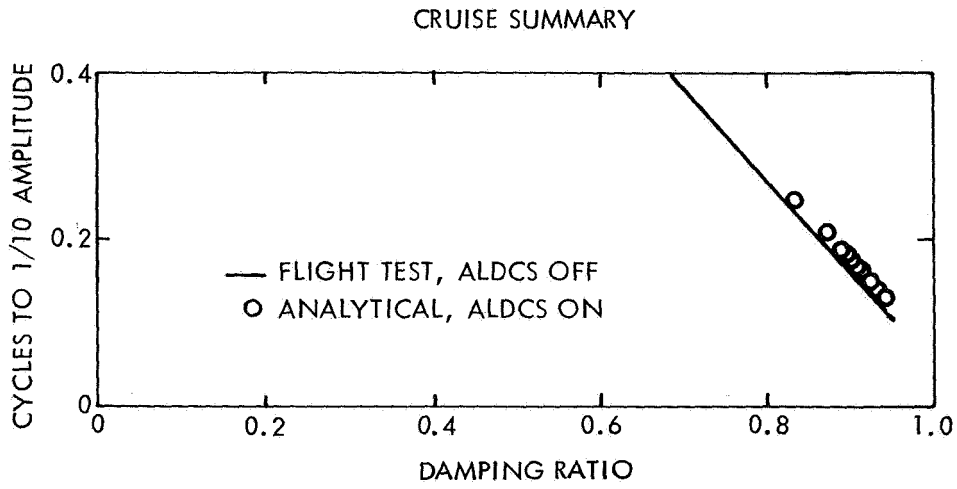
Figure 17.- C-5 ALDCS short period stability.
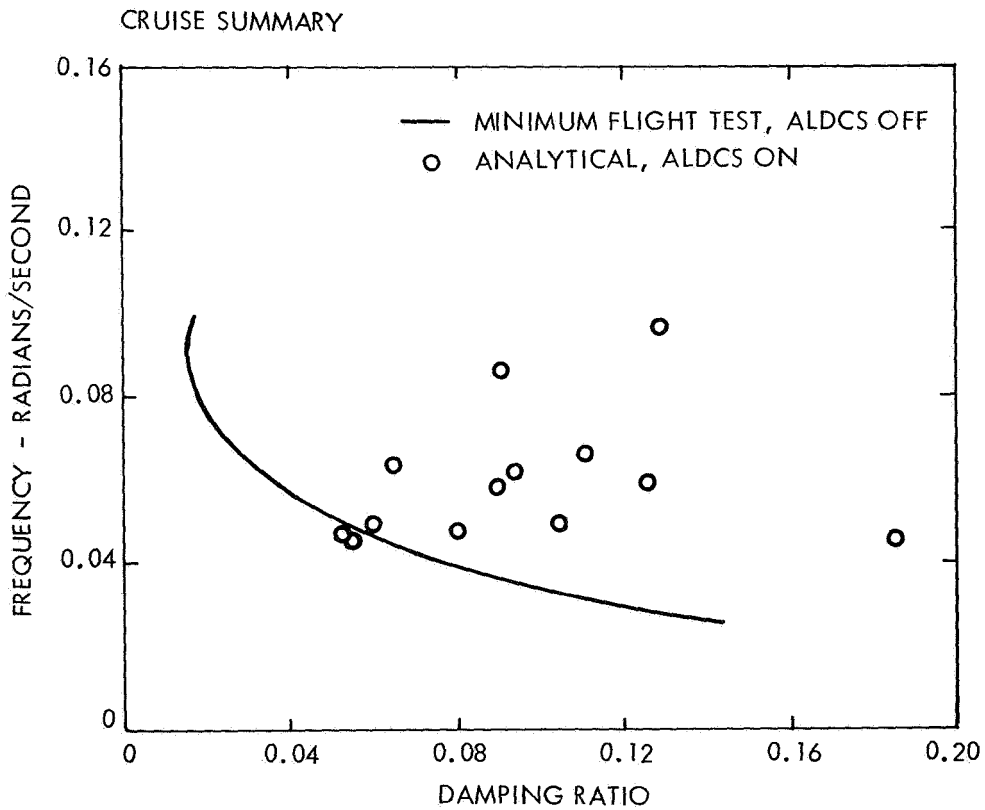
Figure 18.- C-5 ALDCS phugoid stability.

ACTIVE CONTROLS FOR RIDE SMOOTHING

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and

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INTRODUCTION

Active controls technology offers great promise for significantly smoothing the ride, and thus improving public and air carrier acceptance, of certain types of transport aircraft. Recent findings which support this promise will be presented in the following three pertinent areas:

1. Ride quality versus degree of traveler satisfaction
2. Significant findings from a feasibility study of a ride smoothing system
3. Potential ride problems identified for several advanced transport concepts

RIDE QUALITY AND TRAVELER SATISFACTION

Aircraft Motion Characteristics

Large differences in ride smoothness can exist for transport aircraft as illustrated in figure 1, where levels of vertical acceleration are presented for three vehicles as a function of percent time that acceleration levels are exceeded. The data shown for airplanes A and B are averaged values of measurements obtained in the passenger compartment about every 2 minutes between takeoff and landing during many flights onboard scheduled passenger service in the eastern seaboard region of the United States.

For airplane C, data from which averaged values were obtained are much more limited, but are considered representative of cruise flight conditions for present-day large jet transports. Table I lists approximate values of several factors for the aircraft believed to influence the levels of vertical response. Acceleration levels for airplane C are favorably minimized by high wing loading, by wing sweep, by low tail volume coefficient, and by high cruise altitude. For the two smaller aircraft which have somewhat similar properties, the vertical acceleration levels for airplane B are significantly lower than for airplane A, probably because of the higher cruise altitude of these studies. The question arises as to how to interpret data such as presented in figure 1 in terms of passenger satisfaction. Before design goals can be established for application of active controls to ride smoothing, information is needed concerning the influence of ride quality on traveler acceptance and use of vehicles.

Comfort Factors and Criteria

Subjective response to motion has been studied in some detail to establish tolerance-limit criteria (e.g., ability to perform a specific task under adverse environmental conditions, exposure-time limit allowable in a high-vibration environment, etc.). In the area of the ride comfort, which involves much lower magnitude motions, meaningful information is limited and criteria are not well established. To fill a need in this area, NASA has under way considerable research concerning ride quality and traveler satisfaction. The effort described in reference 1 involves both field measurements to identify important factors (e.g., motion, vibration, etc.) and to develop approximate criteria, and laboratory and research aircraft experiments under closely controlled conditions to establish a good understanding of all factors involved. Much of the field measurement effort has been carried out as part of a traveler acceptance study by the University of Virginia under NASA grant. The study with some of the findings is decribed in reference 2. Information from that part of the study which addresses motion environment and passenger response will be used in the next few figures to illustrate how evaluation can be made of the ride quality. The study also provided the data for figure 1.

During flights on air carriers and research aircraft, simultaneous recordings were made of reactions of test subjects as well as of aircraft motion environment in all six degrees of freedom. Correlation of extensive data from a number of differenct aircraft indicated that ride comfort, while influenced by many factors, is particularly affected by vertical and lateral accelerations. Based on just these two factors, initial criteria have been developed of passenger ride comfort response. These criteria are presented in figure 2. Lines of approximately equal comfort rating are shown as a function of lateral and vertical acceleration. A five-unit descriptive scale of comfort rating was employed with terms ranging from Very Comfortable to Very Uncomfortable. Test subjects were about twice as sensitive to lateral accelerations as to vertical accelerations. In figure 3, acceleration values measured onboard airplane A are superimposed on the same scale. Of a total of 409 points, 25 points, which correspond to 6 percent, lie in the zone between Very Uncomfortable and Uncomfortable. An additional 46 percent falls in the region between Uncomfortable and Neutral. Thus, more than one-half the time, passengers could be expected to rate the ride as significantly less than Comfortable. Ride rating, however, does not tell the whole story. In surveys of passengers made at the end of trips, many passengers indicated, even after a ride rated as Uncomfortable, willingness to repeat the same trip. Figure 4 presents these results expressed as the variation in overall trip ride comfort rating with percent of passengers satisfied. For this figure, the word "satisfied" is defined as willingness expressed by the passenger to buy another ticket on the same aircraft and to experience the same ride. As would be expected, passenger satisfaction decreases substantially as the ride becomes progressively less comfortable, until only 25 percent were satisfied for a ride rated as Very Uncomfortable. Thus, passenger satisfaction can be related to ride comfort, which in turn can be related to the vertical and lateral acceleration environment.

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Satisfaction Assessment of Aircraft

Figure 5 presents estimated traveler satisfaction characteristics derived from vertical and lateral acceleration data for the three aircraft discussed previously. Satisfaction is expressed in terms of percent travelers satisfied as a function of flight time percentile ranked by ride smoothness, with the smoothest periods of flight occurring at 0 percentile, and the roughest periods at 100 percentile. The term "traveler" is used rather than "passenger" to point out that about 5 percent of all travelers will not be satisfied in riding an aircraft no matter how smooth the ride may be. For this reason, airplane C, which is considered to have excellent ride characteristics when cruising in smooth air, is shown to be satisfactory under the best of conditions to only 95 percent of all travelers. For this aircraft, the ride quality continues to be quite favorable to the 90-percentile time point where about 90 percent of all travelers would be satisfied. In contrast, airplane A would be satisfactory to only 50 percent of all travelers at the 90-percentile time point and to slightly less than 80 percent of all travelers at the 50-percentile time point. While the trends indicated are considered significant, a note of caution needs to be interjected concerning the simplistic approach used to estimate traveler satisfaction. Actually, there are a number of factors other than vertical and lateral acceleration known to influence ride quality to some degree. Examples include disturbances in rolling motion, terminal-area maneuvers, visual cues, cabin temperature, and seat size. As more is learned in studies concerning these factors, the approach just described for estimating satisfaction can be refined to provide more precise evaluations.

Considerations for Application of Ride-Smoothing System

The trends shown in figure 5 indicate that, in terms of traveler satisfaction, the relative improvement possible by addition of an active-control system would be more modest for airplane C than for either airplane A or B. In addition to information as shown above, a decision to incorporate a ride-smoothing system into an aircraft involves a number of other considerations. Questions such as the following three are examples:

What is the ride-environment conditioning of the passengers who will be using the aircraft? For residents in undeveloped regions, the ride of a DHC-6 could be a big improvement over the ride of an off-road mode of transportation, while for residents of a metropolitan area, seasoned by smooth rides on long-range, heavy aircraft, equally good rides could be expected of smaller short-haul aircraft used by the connecting feeder lines.

Will increase in revenue from additional travelers gained by ride smoothing offset the increased costs of the active-control system? Carriers serving low-density markets may generate little, if any, additional business by ride smoothing, whereas air carriers serving high-density markets may generate considerable extra revenue by attracting customers from competitors whose aircraft have a poorer ride.

<u>Is there a public responsibility to make the ride acceptable to</u>
<u>the greatest possible number of travelers?</u> Perhaps carriers
serving the public should be obliged to conform to minimum
comfort standards as well as to requirements concerning safety
or to the amount of service given cities on their route structure.

Answers to the above questions will depend to a significant degree on detailed
information on the active-control systems required for ride smoothing.

RIDE-SMOOTHING SYSTEM FEASIBILITY STUDY

Concurrent with subjective studies of ride quality, a feasibility study
was carried out of an active-control system for the de Havilland DHC-6 aircraft
for NASA by the Wichita Division of The Boeing Company assisted by de Havilland
Aircraft of Canada, Limited. The objective was to examine the feasibility of
developing and certificating a ride-smoothing-control system for a typical
small feeder line aircraft known to have a ride environment not equal to that
found on larger, high-wing-loading jet transports. The DHC-6 was selected for
study not only because it has a low wing loading and is oftentimes operated
extensively in low-altitude turbulence, but also because it is the only STOL
vehicle presently certificated and extensively used by air carriers in this
country. Its capability to carry out steep-angle climbouts and descents and
to perform short-radius, terminal-area maneuvers makes suitable the study of
ride-quality situations reasonably typical of those which may be encountered
by subsequent advanced STOL/RTOL transports. An example application of this
nature is the Canadian STOL Demonstration Program between Ottawa and Montreal,
where modified DHC-6 aircraft are being used to obtain passenger acceptance
data as well as to study and refine systems operations in advance of intro-
duction of the new and larger DHC-7 STOL transport aircraft now being built
for such service.

Description of System Studied

Quite a bit of information having general application to ride-smoothing
systems was obtained from the feasibility study. Highlights of this general
information will be presented herein; detailed description of the study and
findings are presented in reference 3. Investigation of active controls was
limited to only vertical and lateral ride smoothing, as preliminary study
indicated response to turbulence to be acceptably low for the other degrees of
freedom. The aerodynamic surfaces considered in the system are shown in
figure 6 and include portions of the existing ailerons, elevators, and rudder
as well as all-new spoilers. Consideration of additional surfaces could not
be accommodated within the scope of the study. Ride control of each degree of
freedom was treated independently. Simplified block diagrams showing feedback
loops are presented in figure 7 for the vertical control system and in figure 8
for the lateral control system. Details such as transfer functions are not
shown. System effectiveness was determined as reduction of acceleration
response to a random turbulence intensity with an exceedance probability

of 0.01 which was established as a gust velocity of 2.1 meters per second (rms) for the design flight conditions.

Ride-Control Effectiveness

In the area of effectiveness, the most important finding was the requirement for relatively large direct-lift and direct-side-force surfaces located near the airplane center of gravity. As shown by the bar charts of figure 9, significant reductions in vertical acceleration response were obtained with wing flaps retracted during both climb and cruise conditions. The elevator surfaces contributed only a modest amount to this reduction. For the landing approach condition, new spoilers had to be employed to even achieve the less-than-adequate reductions shown. Design techniques need to be developed for integrating large, direct-lift surfaces for ride smoothing into wing-flap systems. Use of rudder surfaces for reducing lateral response was somewhat effective in the aft section of the passenger cabin, but was ineffective ahead of the cabin midpoint. Efficient (high side-force/drag) direct side-force surface configurations need to be provided at a fore-and-aft location near the airplane center of gravity. Some technology for such surfaces was generated in the development of the General Purpose Airborne Simulator (GPAS) and the Total In-Flight Simulator (TIFS) research aircraft.

Aircraft Stability, Control, and Handling Qualities

In this area, a ride-smoothing system can be designed which is satisfactory. Considerable attention must be given, however, to various potential problems in order that the system be tailored to minimize adverse effects. In the feasibility study, problems which had to be resolved involved the aircraft low-frequency longitudinal mode, the very-low-frequency phugoid mode, the Dutch-roll mode, and the lateral-directional spiral mode. A detailed control system synthesis and performance analysis is required to examine various trade offs. During the study, problems also had to be resolved in aircraft handling qualities such as one where adding the active-control system caused a loss of effectiveness of the elevator to relatively sharp inputs. In this case, satisfactory short-period handling quality was achieved by introducing a crossfeed signal to the system to initially cancel the ride-control signal which opposed the acceleration, and then to wash out at the same rate as the ride-control signal. Use of ground based simulators is appropriate to study and help resolve handling problems.

Reliability and Safety

No major problems in reliability are anticipated for the ride-smoothing system. Since use of the system is not critical to the well being of the aircraft, the system can be deactivated if malfunctions occur. The main concern involves transient problems which could arise at the time of any malfunction. The worst problem envisioned would be hard-over deflection of an aerodynamic surface used in the active-control system. If sufficient authority

is provided by the aircraft control system to control vehicle motions·caused by such a deflection, safety can be maintained. Such authority would be a reasonable requirement for system .certification. A fail-soft design control system, such as devised in the feasibility study, can also be incorporated for additional protection. The particular system studied contained dual signal channels with two stages of monitoring between channels for failure detection. An unfavorable comparison of channel signals would switch off the ride control signals.

System Components

Ride-smoothing hardware requirements are not considered to tax the present state of technology. Appropriate sensors, electronic elements, servosubsystems, and actuators are in production. The size and capacities of these components are not necessarily matched to detailed requirements, and modifications of existing designs may be required to obtain appropriately tailored articles. Aerodynamic requirements do require innovation, as discussed earlier, to develop configurations to efficiently produce aerodynamic forces through the center of gravity in both vertical and lateral directions.

Weight, Power, and Volume Requirements

Weight and power demands of a ride-smoothing system should not seriously burden the aircraft. Findings of the feasibility study indicated the total additional weight would amount to less than 2 percent of the aircraft gross weight. Additional power requirements of the system would amount to no more than 0.3 percent of the aircraft total engine power. Requirements for larger aircraft would not be expected to exceed these percentage values. Only a small additional volume is needed, but volume requirements in local regions near aerodynamic control surfaces may require special consideration, particularly if an existing aircraft is being retrofitted with a ride-smoothing system.

System Costs

Cost information is lacking because no detailed cost analysis has been carried out. Based on the findings presented above, system development and certification will require considerable effort which will be somewhat independent of aircraft size. Where the system is incorporated into the initial design of an all-new aircraft, the additional costs estimated for the system design through prototype flight tests and certification could range from 2 to 5 percent of the total costs. The additional cost would be expected to be higher if a system were to be designed and retrofitted into an existing vehicle. These higher costs result because of the probability of significant modification, requalification, and retesting of existing systems and structures. Estimated production costs for the system in terms of aircraft production cost could range from about 1 percent for large jumbo transports to as much as 4 or 5 percent for very small transports. Ride smoothing may be included as a feature of a multipurpose active-control system which performs other functions

as well, such as gust-load alleviation. Design and checkout of an appropriate multipurpose system would require considerable effort, possibly greater than the sum of efforts required for individual systems.

Maintenance and Repair

Specific maintenance information is lacking until a ride-smoothing system is put into service. Considerable experience has been obtained, however, on a closely related active-control, fatigue-reduction system, described in reference 4, which was applied to the United States Air Force B-52G and B-52H fleet of 280 aircraft. For this application, system performance and maintenance experience has been excellent and well within guideline limits. Since an active-control, ride-smoothing system is essentially a state-of-the-art system competitive with control systems used on modern transport aircraft, maintenance should be similar to that required for current control systems.

Time Required for System Implementation

Little, if any, additional time would be needed if the decision to proceed is made at the beginning of an all-new aircraft project. For retrofit of a ride-smoothing system into an existing aircraft, the total time required is estimated to range between 2 and 3 years.

POTENTIAL RIDE PROBLEMS FOR ADVANCED TRANSPORT CONCEPTS

A number of advanced transport concepts are in various stages of technology development. Sufficient information is presently available to identify potential problems in ride quality for some of these concepts. Since only a qualitative assessment can be made of each problem, the exact role to be played by ride-smoothing systems cannot be exactly defined at this time. A description of potential problems is given for six vehicle concepts.

Large, Low-Wing-Loading Aircraft

One attractive concept, described in reference 5, for achieving STOL/RTOL capability in transports for medium- to high-density market short-haul use, involves the combination of low-wing-loading, mechanical-flap configurations with an active-control, gust-load alleviation system to minimize structural weight. Because of the relatively large wing area, response to vertical gusts can be expected to produce a ride which is less than satisfactory. Use of an active-control system will probably be required not only for gust-load alleviation, but for ride smoothing as well.

Powered-Lift Aircraft

Powered-lift concepts, which involve internally or externally blown flaps, can produce usable maximum-lift coefficients of two to three times those for current transports as described in references 6 and 7. Such high-lift capability is attractive for providing STOL/RTOL performance with high-wing-loading transports. For such configurations, engine-out control requirements will probably dictate the need for a relatively large vertical tail surface. Use of a large tail introduces a potential problem of uncomfortably large responses of the passenger compartment to lateral gusts. Use of an active-control system to reduce this lateral response is anticipated.

Terminally Configured Vehicles

Technology is being developed in the form of advanced display guidance and control systems together with new flight paths and operating techniques which can be applied to advanced aircraft specifically configured to more efficiently use the airspace in terminal areas and, thus, help relieve airside traffic congestion (ref. 8). The flight maneuver techniques are anticipated to involve relatively tight turns and abrupt decelerations which could introduce ride-quality problems, particularly if aggravated by oscillating motions of the aircraft due to air turbulence. In order that the maximum degree of planned flight maneuvers can be utilized, use of active-control systems may be required to minimize the random motion environment.

Supersonic Aircraft

The need to achieve efficient operations in supersonic-cruise flight leads to a configuration requirement for long, slender, and relatively limber fuselage configurations. Use of such configurations is anticipated to introduce problems of motion response in the passenger compartment to aeroelastic inputs during high-speed descent from cruise altitude, and to runway roughness inputs during taxi, take-off, and landing rollout. The magnitude of motion responses will depend on the fuselage structural dynamic characteristics and can be expected to vary considerably down the length of the passenger compartment. Problems may be of sufficient magnitude to warrant use of active-control systems to minimize motion. Solution to problems could lead to the need for a system of somewhat unconventional design.

Civil Helicopters

Significant effort is being directed toward providing advanced technology for large civil helicopter transports suitable for short-haul operations. Based on experience with large military vehicles, ride-quality problems can be anticipated from oscillating aerodynamic inputs associated with the rotating blades. These inputs result in vertical and lateral responses at discrete frequencies of the passenger compartment. The need for an active-control, rotor-feedback system to reduce responses is anticipated.

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CONCLUDING REMARKS

A review has been given of the potential use of active-control systems for ride smoothing. Substantial differences in ride quality which can exist between transport aircraft have been illustrated, and a technique has been described for assessing these differences and the need for ride smoothing in terms of traveler satisfaction. Results from a ride smoothing feasibility study have been used to provide a generalized assessment of active-control systems for this purpose. The assessment, which includes effectiveness, reliability, maintainability, and costs, indicates that no major technical problems exist and that significant ride smoothing can be achieved within the present state of the art. Evaluation has been made of six advanced transport concepts to identify potential ride-quality problems and possible requirements for active controls. The next major step indicated for advancing ride smoothing technology is system application, demonstration, and evaluation for an aircraft in regular service.

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TABLE I - AIRCRAFT PROPERTIES AFFECTING RIDE QUALITY

	Airplane A (20 passengers)	Airplane B (29 passengers)	Airplane C (219 passengers)
Maximum take-off weight, N (lb)	55,600 (12,500)	104,000 (23,400)	1,490,000 (334,000)
Maximum wing loading, N/m^2 (lb/ft^2)	1,400 (30)	1,920 (40)	5,270 (110)
Wing sweep angle, degrees	0	0	35
Horizontal tail volume coefficient	0.94	1.02	0.63
Approximate cruise altitude of study, m ft	900 (3,000)	1,800 (6,000)	9,000 (30,000)
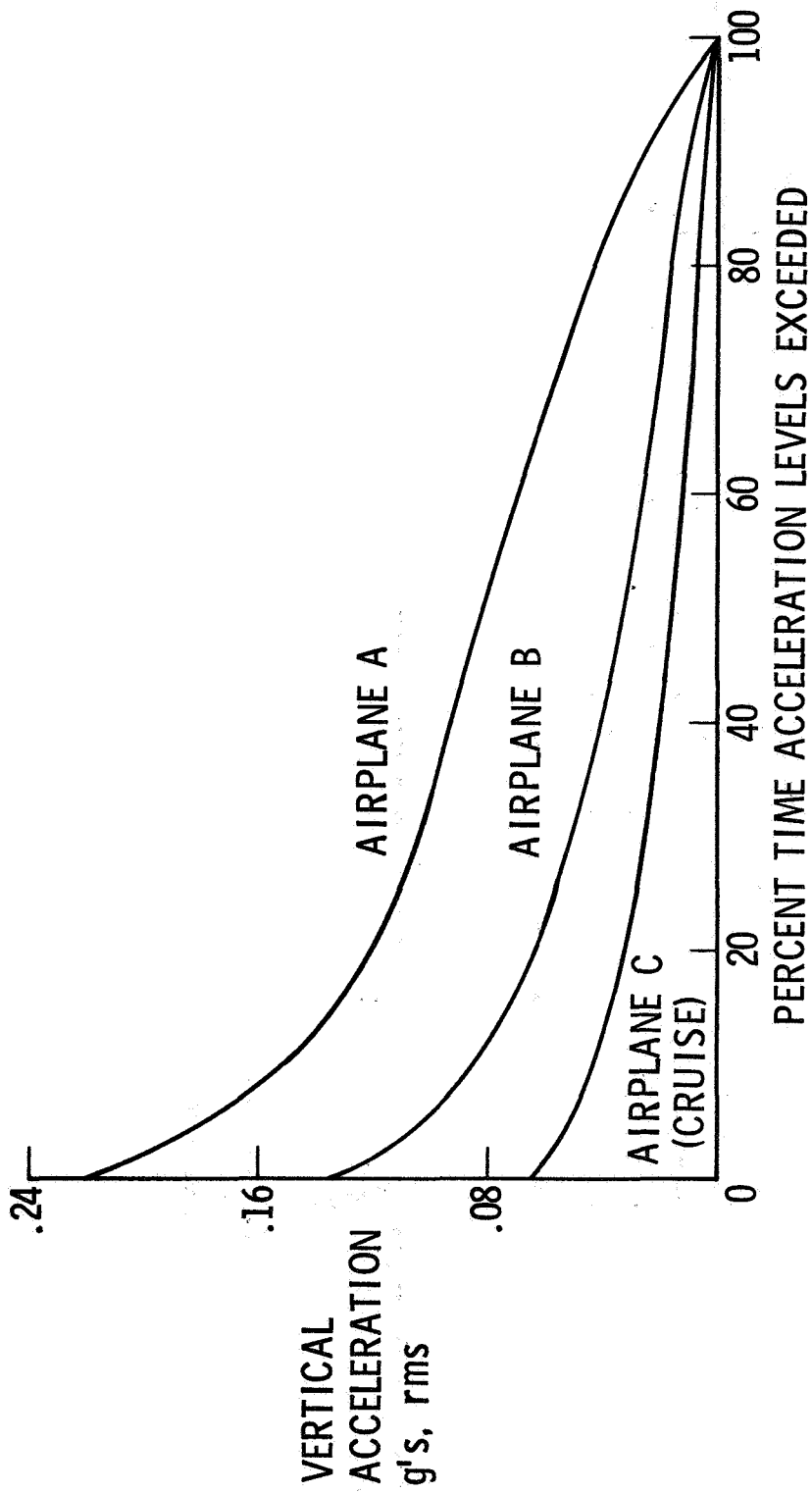
Figure 1. Example differences in aircraft motion characteristics for vertical acceleration.
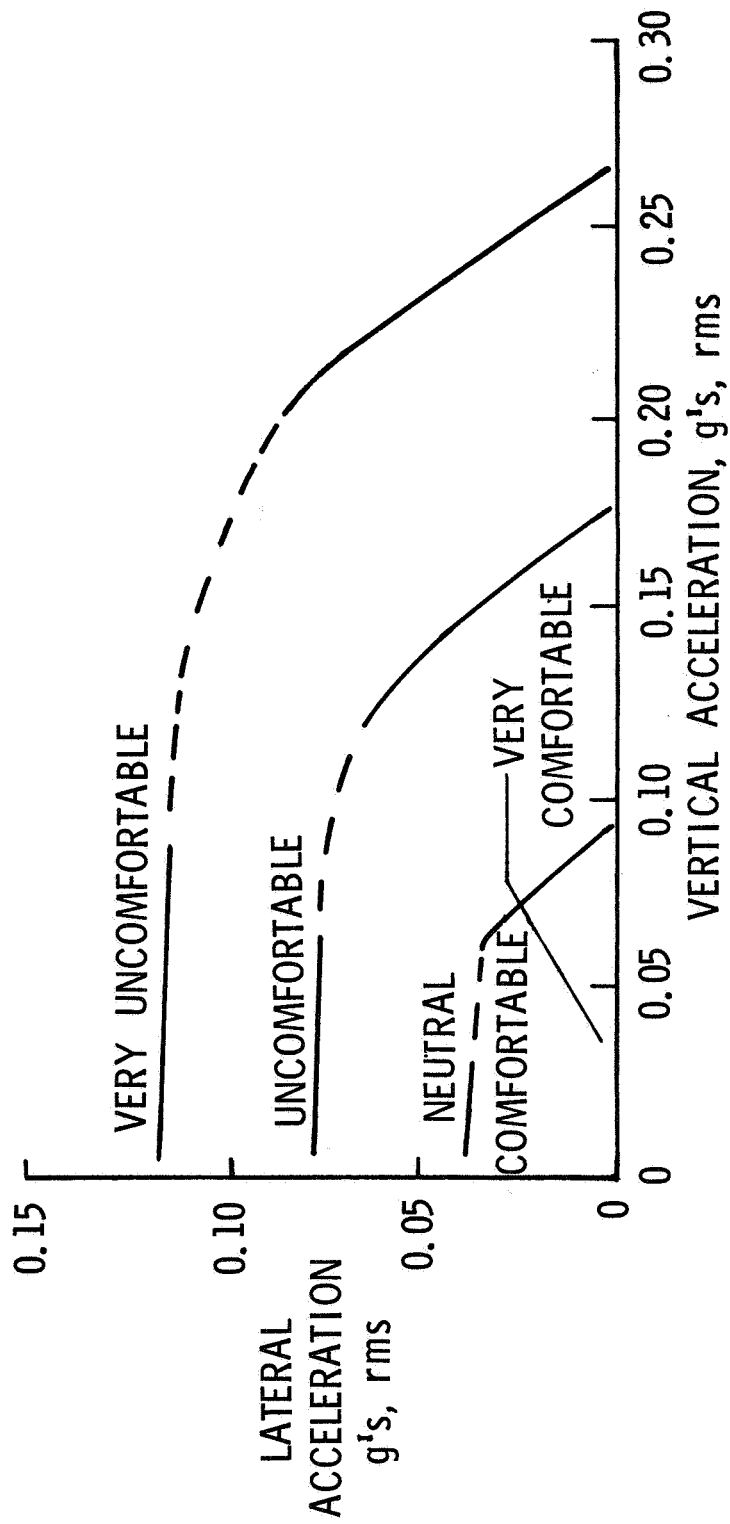
Figure 2. Significant comfort factors and initial criteria from airline and research aircraft.
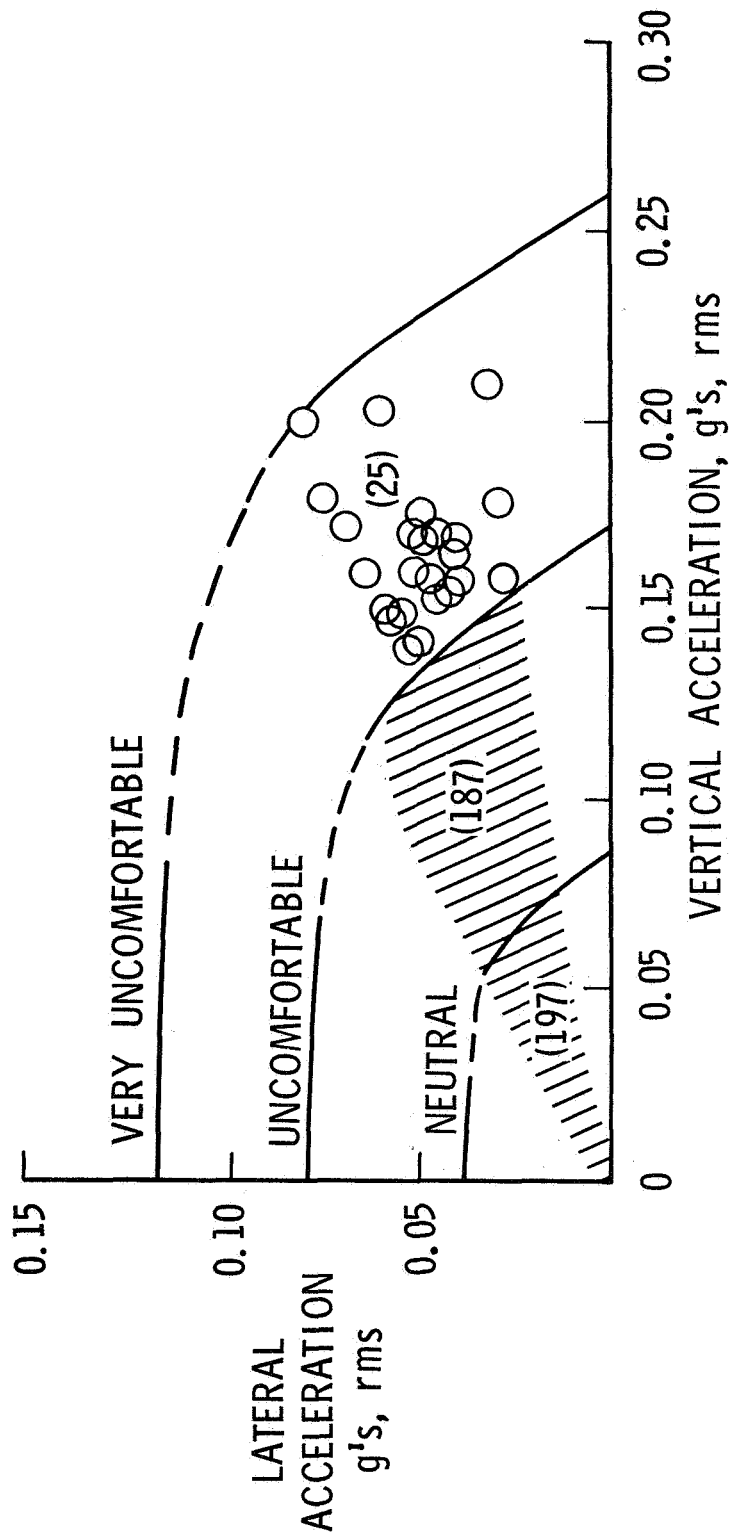
Figure 3. Comfort criteria applied to airplane A measurements (409 data points).
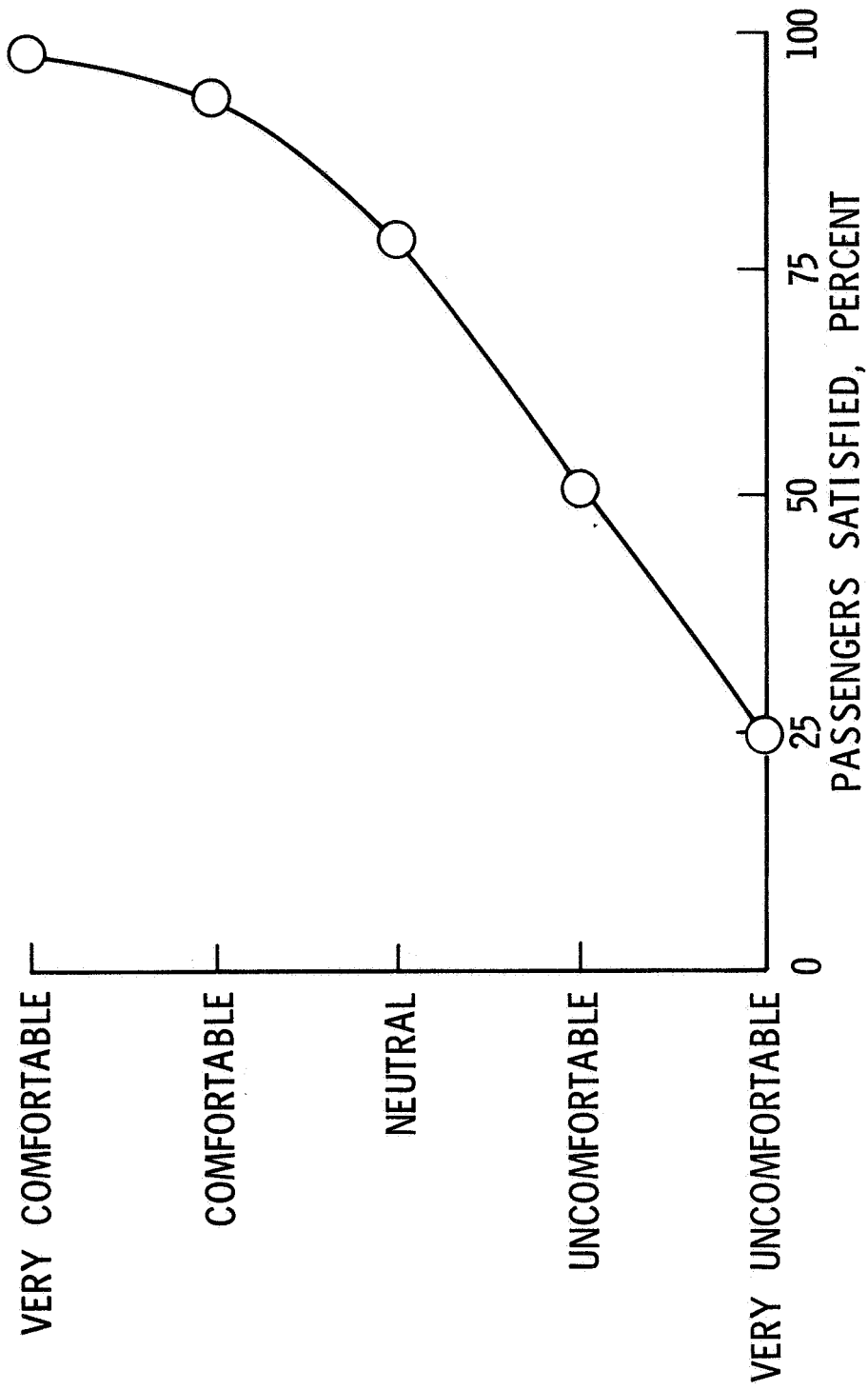
Figure 4. Relationship between comfort rating and passenger satisfaction.
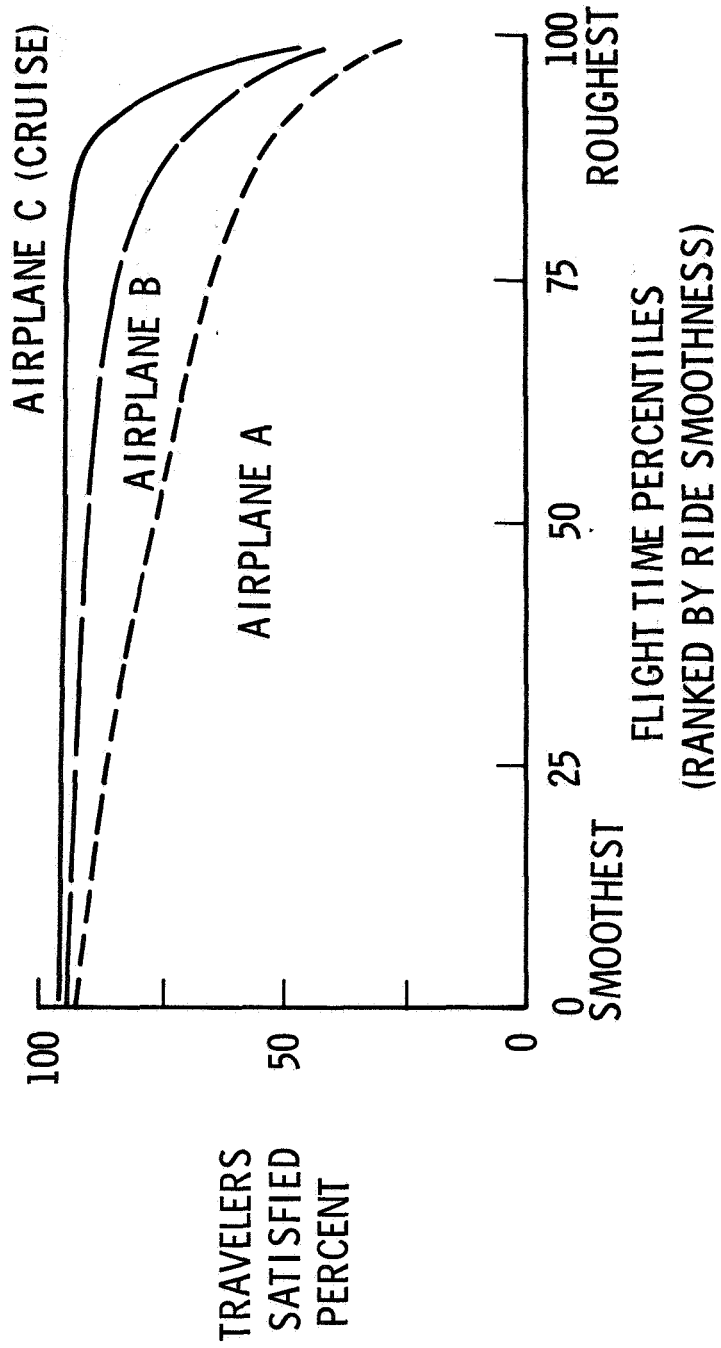
Figure 5. Estimated traveler satisfaction of three aircraft.
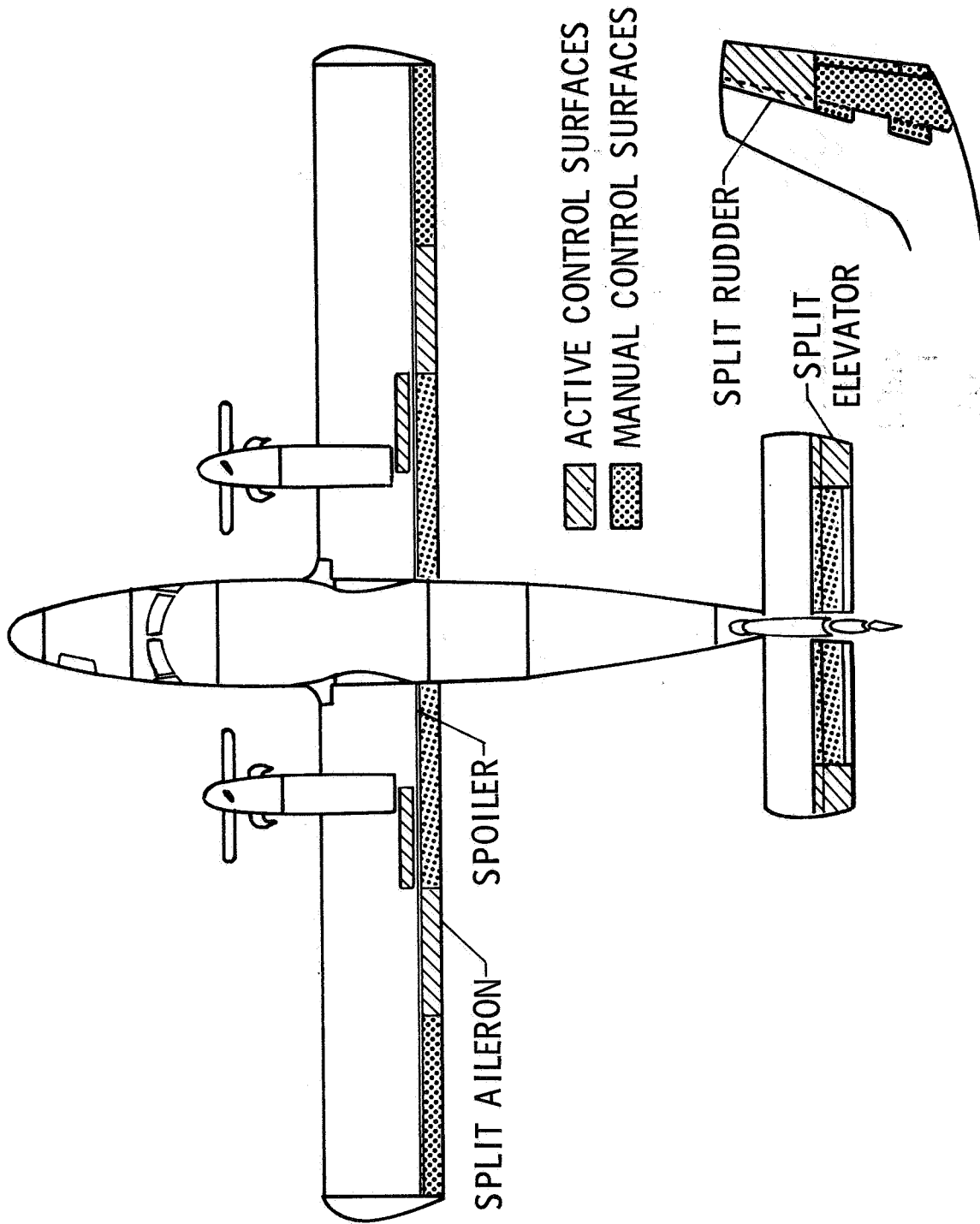
Figure 6. Active control surfaces studied on DHC-6 aircraft.

ACTIVE CONTROL SURFACES
MANUAL CONTROL SURFACES

SPLIT RUDDER

SPLIT ELEVATOR

SPOILER

SPLIT AILERON
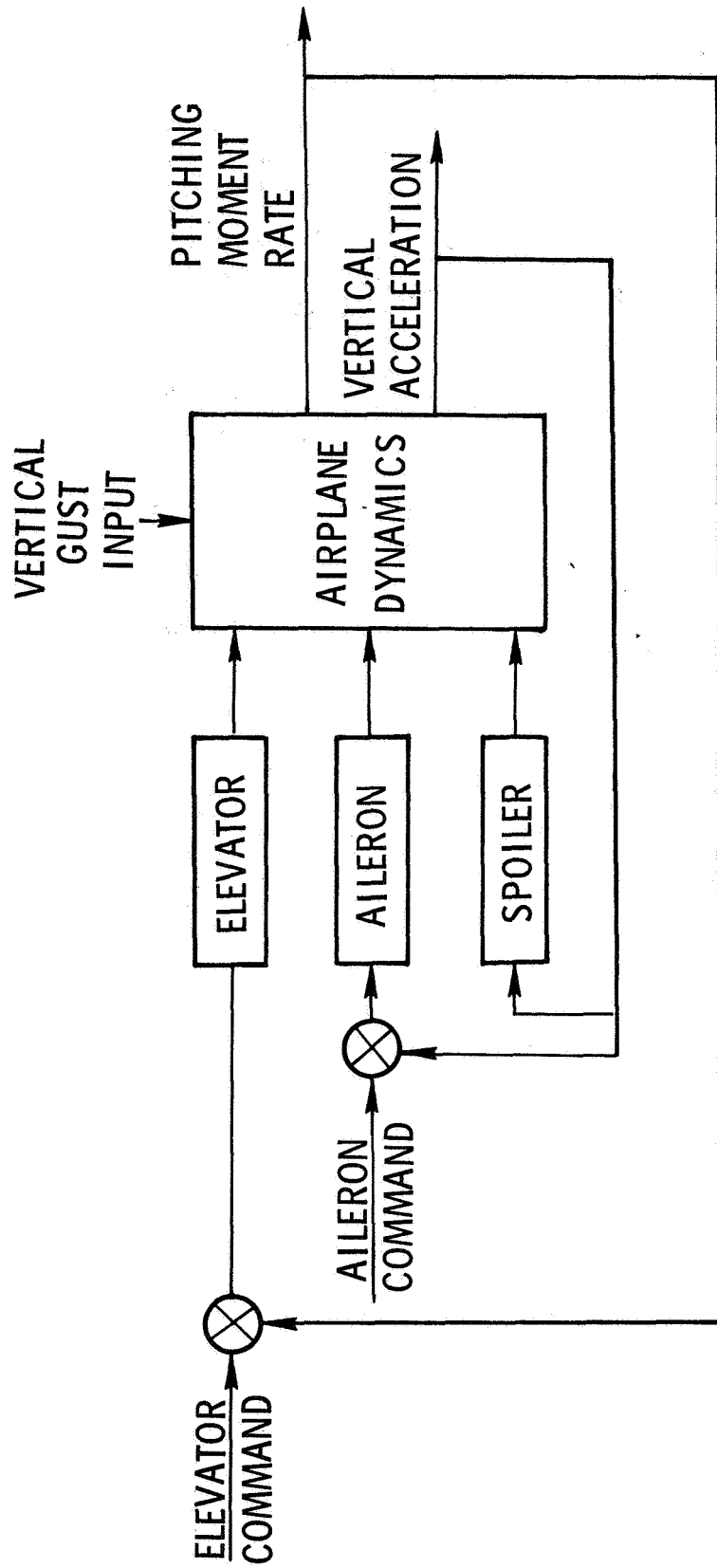
Figure 7. Block diagram of vertical ride-control system from DHC-6 study.
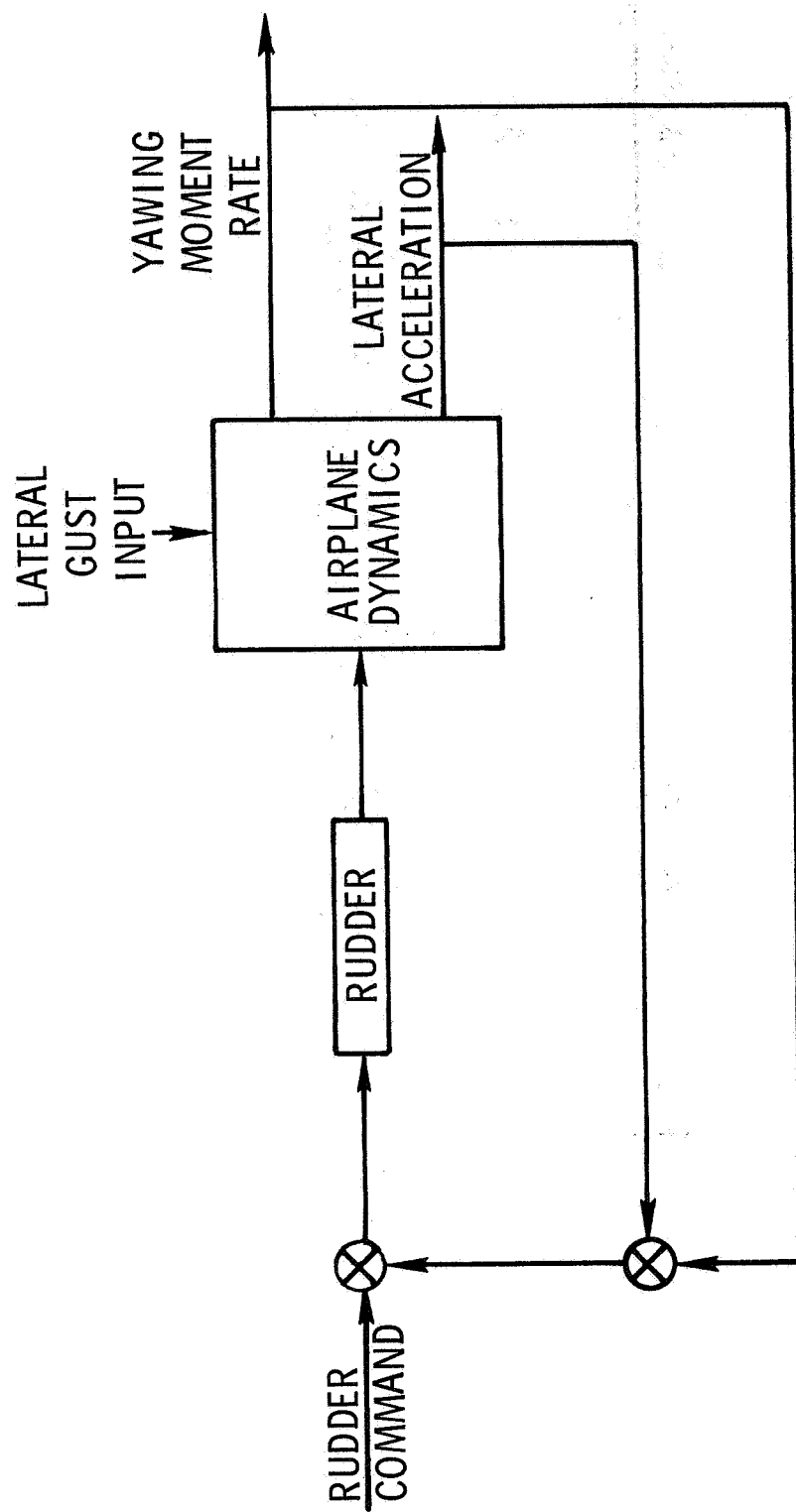
Figure 8. Block diagram of lateral ride-control system from DHC-6 study.
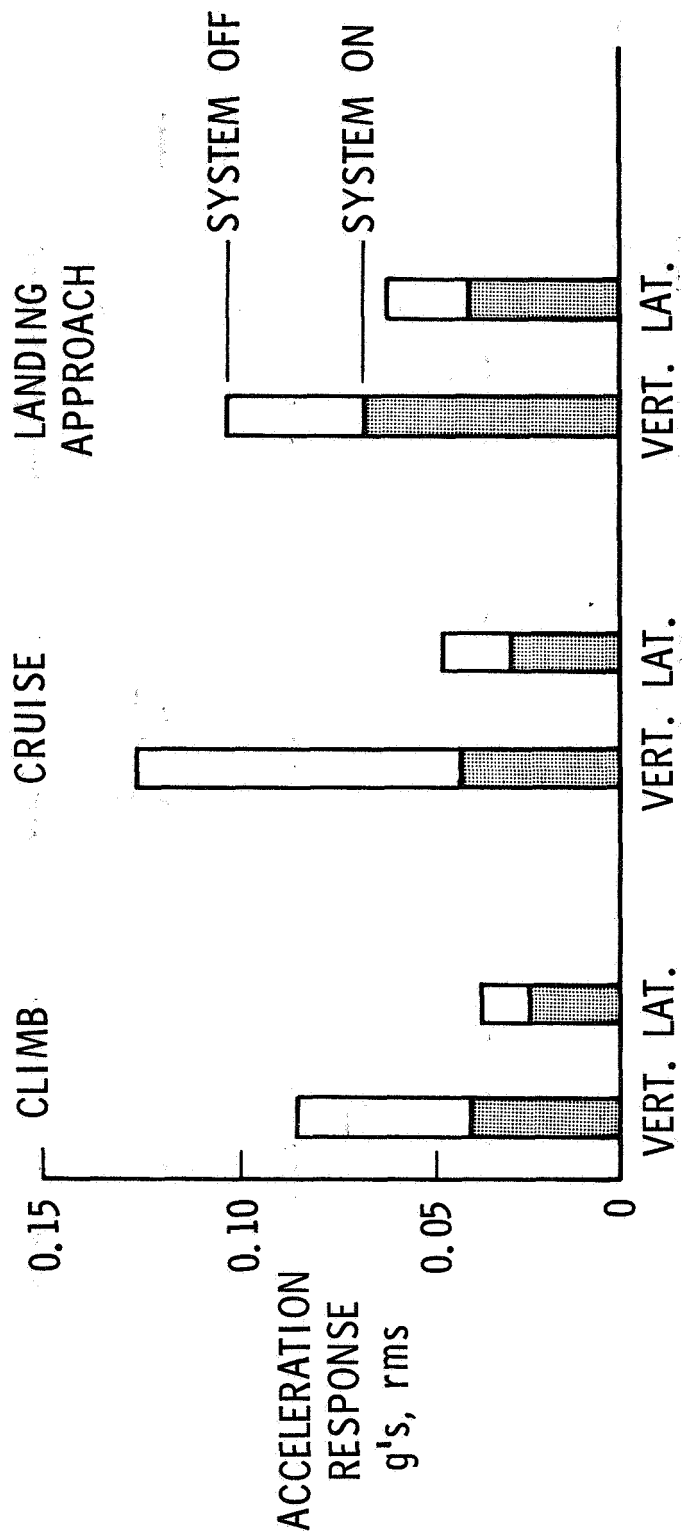
Figure 9. Effectiveness of ride-control system in terms of aft cabin response to 2.1 m/sec gusts.

USE OF ACTIVE CONTROL TECHNOLOGY TO IMPROVE

RIDE QUALITIES OF LARGE TRANSPORT AIRCRAFT

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SUMMARY

This paper describes the analyses, construction and flight testing of two systems, "Beta-vane" and modal suppression augmentation system (MSAS), which were developed to suppress gust induced lateral accelerations of large aircraft. The Boeing 747 transport was used as the test vehicle. The purpose of the Beta-vane system is to reduce acceleration levels at the "dutch roll" frequency whereas the function of the MSAS system is to reduce accelerations due to flexible body motions caused by turbulence. Data from flight test, with both systems engaged shows a 50-70 percent reduction in lateral aft body acceleration levels. Furthermore, this paper suggests that present day techniques used for developing dynamic equations of motion in the flexible mode region are limited. These techniques produce results which are satisfactory for analyzing dynamic loads and stability problems, but may be insufficient for development of active control systems operating in the same frequency region.

INTRODUCTION

The aft fuselage section of long slender airplanes is a position of relatively high lateral acceleration levels in moderate to heavy turbulence. These accelerations can be considered as being due to contributions from a rigid airplane with the elastic effects superimposed. Initially, because of the experimental nature of the program, two different approaches to gust alleviation were undertaken. One system worked the flexible body frequencies (MSAS system - Section I) whereas the second system worked primarily rigid body frequencies (Beta-vane system - Section II).

MODAL SUPPRESSION AUGMENTATION SYSTEM (MSAS)

INTRODUCTION

This section of the paper will describe in detail the analysis, construction and flight testing of a modal suppression augmentation system. This system was designed to reduce aft body lateral accelerations in the 1-3 Hz region when flexible body motions are perturbed by turbulence. Due to the problems associated with the lateral dynamic equations of motions as discussed in the following section (that is, comparison of analytical and measured transfer functions showed a variation in the flexible mode region), a technique was developed which involved 'curve fitting' transfer functions to experimental data. This method then allowed a modal suppression system to be developed without dependence on the analytical equations. Furthermore, by including the yaw damper actuator with the experimental data that was analyzed via the curve fit method, the problem associated with precise mathematical modeling of the structural compliance feedback-actuator system was avoided.

COMPARISON OF ANALYTICAL AND EXPERIMENTAL DATA

Results from 747 flight testing in turbulence indicated that aft end lateral motion was composed of the following two components:

(1) Rigid Airplane (dutch roll)   --- 0.2 Hz

  (50% contribution)

(2) Elastic effects   --- 1.0 - 3.0 Hz

  (50% contribution)

Within the 1.0 - 3.0 Hz band of frequencies the analytical equations predict five free-free modes, all of which are composed to some degree of wing, nacelle and body motions. These modes (corrected with results from

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the ground vibration test) are shown in Figure 1.
Flight test data reduced via the 'curve fitting'
technique (explained in the following section) is also
shown in Figure 1.

Based on their composition the modes are identified
as (1) outboard nacelle vertical bending, (2) fundamental
wing bending, (3) inboard nacelle side bending, (4) aft
body bending, and (5) outboard nacelle side bending. In
addition to the above set of modes, a stabilizer mode
at 3.16 Hz, a fore body mode at 4 Hz and a vertical fin
bending mode at 6 Hz are of concern in conjunction with
the development of the MSAS filter.

Although the analytical equations were reasonably
close to measured values and thus sufficient for flutter
studies, the development of an active control system,
however, requires not only that the characteristic equation
be correct but also that the residues of the transfer
function (the zeros shown in Figure 1) be properly
described.

From Figure 1, it is seen that even though the roots
of the system (poles) are identified and reasonably close
to those obtained via flight test data, it is obvious that
the associated zeros are misaligned. Various attempts
in the form of refinement in both structural and aero-
dynamic representation did not succeed in changing the
general picture. Further work along these lines still
remains to be pursued.

As a practical solution to the problem, a curve
fitting technique was applied to the measured transfer-
functions to identify the zeros and poles of the system
to be controlled.

CURVE FITTING TECHNIQUE

From initial experimental data, the aft body was
found to resonate at 1.8 and 2.4 Hz whereas the fundamental
frequency of the fore body was 4 Hz. Furthermore, the
aft body could be perturbed by gusts striking the fin or
gusts exciting the engine nacelles producing wing-body
coupling suggesting that either the ailerons or rudders
could be used for the active control system. Due to the
complexity associated with developing a system in
conjunction with the ailerons, a rudder suppression

system was chosen. Though the command signal to the rudder is rate limited at 13 deg/sec compared to a 50 deg/sec requirement for the load alleviation system developed for the B-52, it was determined that this lower rate limit would satisfy the requirement.

The experimental data was obtained by excitation of the lateral airframe degrees of freedom in the 1-7 Hz region via the upper and lower yaw damper servos and their respective rudders. The forcing function itself was a continuously changing constant amplitude sine wave frequency sweep, in the 1-7 Hz range, which was produced on a computer and stored on magnetic tape. Using the experimental data in conjunction with a Fast Fourier transform data reduction package, Bode plots for various sensor locations on the aircraft could be obtained.

The curve fitting' technique is based on the 2 papers given in references 1 and 2. These algorithms were programmed on the CDC 6600 during the development of the Boeing SST and were used in the design of 3rd and 4th order prefilters in conjunction with the Horowitz Circle technique. After a few attempts at deriving transfer functions from the experimental data, the following deficiencies in the computer program were observed:

1. The program could not handle 14th order systems.

2. The small non-linearities associated with the amplitude and phase curves were sufficient to make the computer program limit cycle.

3. The program was very sensitive to end point conditions.

Transfer functions that matched the experimental data were obtained by incorporating the following procedures:

1. The transfer functions were assumed to be of minimum phase (no right half plane zeros). Therefore, only the amplitude was input to the program.

2. A pole or pole-zero combination is always included on either side of the band of frequencies that is of interest.

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The transfer functions showed clearly that although the analytic equations could be manipulated so that the modes would have the correct frequencies, the zeros associated with these analytic equations (and therefore the phase) were not correct for the 2.1 and 2.4 Hz modes. The effects of the different zero locations on a control system will now be shown.

A root locus diagram of an accelerometer control system based on the analytical equations is shown in Figure 2. The control system adds approximately twice the damping to the 2nd and 4th modes; these two modes contribute 80% of the flexible energy. This system was flight tested and results showed that the 4th mode was destabilized and the 2nd mode increased in frequency as the gain of the control system was increased. This same control system based on the airplane transfer function obtained via the curve fit computer program has the root locus diagram shown in Figure 3. Notice that the loci are almost the same as those obtained in flight. This experimental verification of the curve fit technique showed that this method could be used with confidence.

The complete design technique in the development of the MSAS is the following:

1. EXCITE AIRPLANE VIA RUDDERS - CONSTANT AMPLITUDE SINE WAVE 1.0 to 7.0 HZ.

2. CURVE FIT TRANSFER FUNCTION TO AFT BODY SENSORS.

3. ROOT LOCUS METHODS TO DESIGN FILTER.

4. EXCITE AIRPLANE VIA RUDDERS, WITH/WITHOUT MSAS, TO VERIFY SUPPRESSION OF MODES.

5. FLY MSAS IN TURBULENCE TO VERIFY CONTROL SYSTEM.

Notice that this procedure does not allow analytical verification of gust suppression; it only substantiates analytically whether the control system adds damping to the modes.

Two control systems were designed and flight tested using the above procedure. The first system used an aft body mounted lateral accelerometer sensor whereas the second system used two yaw rate gyros, one aft body and one at the cg. Figure 4 shows the reduction in aft body acceleration (Body Station 2300) for the two systems when

the sine wave forcing function is fed to the lower rudder
and the control systems are commanding the upper rudder.
The accelerometer system was not chosen because the 2.4
Hz mode destabilized at high 'q' conditions. In addition,
to obtain equal reduction in acceleration levels during
turbulence, the accelerometer system required more rudder
than the gyro system suggesting that the gust zeros for
the two systems were quite different.

DESCRIPTION OF FINAL MSAS SYSTEM

The MSAS system is a single channel augmentation
system working via the lower yaw damper servo. A block
diagram of the control system is shown in Figure 5. The
augmentation system provides damping to the 1.8, 2.1, and
2.4 Hz aft body lateral modes without disturbing the
dutch roll mode. The salient features of the system
are the following:

1.   Two lateral yaw rate gyros.

2.   Single channel 'real time' monitoring.

3.   Scheduling of filter gain with calibrated
      air speed (CAS).

4.   Output of system limited to + 0.8 degrees
      of rudder (yaw damper authority is + 3.5
      degrees of rudder).

5.   Operation of system limited to flaps "up"
      condition.

Figure 6 represents a functional block diagram of
the computational path.

1.   MSAS Damping Signal

      The MSAS signal is derived from the subtraction of
      two yaw rate signals. The location of the sensors
      are the following:

      a. Aft End Gyro:

          Body Station 2280, WL190, RBL20

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b. CG Gyro:

   Body Station 1307, WL195, RBL5

   Due to the placement, the aft end gyro is sensitive
to dutch roll and flexible mode frequencies whereas
the cg gyro is sensitive only to dutch roll frequencies.
Upon subtraction of the two yaw rate signals, the remaining
signal contains only flexible mode frequencies.

2. Band Pass Filter

   At flaps up condition, the yaw rate signal passes
   through a band pass filter into the yaw damper servo
   amplifier. The band pass filter is composed of R-C
   components, operational amplifiers and multipliers.
   The transfer function of the filter can be expressed in
   Laplace form as the following:

$$\left(\frac{750\ K_{CAS}}{S+50}\right)\left(\frac{18\,S^2}{(S+6)^2}\right)\left(\frac{14}{S+14}\right)^2\left(\frac{1.56\,S^2+9.4\,S+320}{S^2+15.9\,S+320}\right)\left(\frac{.726\,S}{S+1}\right)$$

   A Bode plot of the filter is shown in Figure 7.

   The functions of the band pass filter are:

   a. To wash out the steady-state yaw rate signals
       and to eliminate null offset of sensors.

   b. To reduce high frequency signal amplitudes so
       as to minimize coupling with the higher struc-
       tural modes.

   c. To obtain the proper phasing between yaw rate
       signal and lower rudder so as to add damping to
       the aft body lateral flexible modes.

   Figure 8 represents the transfer function of yaw
rate/lower rudder at BS-2300 whereas Figure 9 shows the
effects of the MSAS filter on the above dynamics. The
reason for the complexity of the filter is that the 1.8

mode required 'lag' and the 2.4 mode 'lead' in order for the system to add the maximum damping to these modes. Although various body stations were investigated, sensor positions aft of the cg, along the floor 'water line' showed that there was no change in the phase relationship between the 1.8 and 2.4 cps mode.

Figure 10 represents a functional block diagram of the monitoring system and pre-engage mode. The function of the monitor system is the following:

a. Checks the principal gains and phase character-istics of the filter.

b. Detects failure of either gyro.

c. Detects failure of the limiter.

d. Detects failure of gain scheduler.

The purpose of the pre-engage mode is to verify that the MSAS electronic unit, including monitor, is operating correctly.

TEST RESULTS

A system corresponding to the filter shown in Figure 7 was flight tested (no monitor system, etc.). After initial calibration and stability criteria were satisfied (6 db gain margin and 60° phase shift), the system was flown in turbulence. Figure 11 shows one of the many time histories obtained. Figure 12 represents the cumulative accelerations for the time history plots of Figure 11. The MSAS system reduces the aft body flexible mode content by approximately 50% (although Figure 12 shows a 66% reduction). Figure 13 shows the cumulative acceleration at the pilot station. It may be noted that there is very little 1.8 and 2.4 Hz content at the pilot station and very little 4 Hz content in the aft end.

A production type unit has recently been flown (including monitor system, etc.) and the next step will be to certify the system together with the Beta-vane system. The combined systems will then be installed on a production airplane for in-service evaluation.

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BETA-VANE SYSTEM

INTRODUCTION

This section discusses a method devised for the 747 airplane of reducing those accelerations due to gust induced rigid airplane motions. As was pointed out in the previous section, the level of RMS accelerations due to turbulence is approximately 50% due to rigid body motions and 50% due to flexible motions (Figure 14). Consequently, a system designed to reduce the rigid body accelerations offers only half the potential reduction in the total level.

SYMBOLS

δ vane rotation

U_B longitudinal body axis velocity

V_B lateral body axis velocity

V_P total velocity

W_B vertical body axis velocity

P_B body axis roll rate

R_B body axis yaw rate

L_1 longitudinal distance from C.G. to vane station

H_1 waterline distance from airplane principal axis to vane station

A_y accelerometer output

Θ pitch angle

ϕ roll angle

METHOD OF SOLUTION

The method used for gust alleviation on the 747 in the frequency range 0 - 1 Hz is snown in Figure 15. The basic sensor is a relative wind vane which is used to sense lateral gusts; the output of the vane is used to drive the 747 upper rudder in a sense that reduces the airplane tendency to turn into the gust. The wind vane output signal is composed of the rapid change due to the lateral gust plus changes due to airplane motion from past disturbances. An approximate separation of these signals is accomplished through deriving airplane motion from lateral acceleration, yaw rate and roll attitude as shown in Figure 15. The resulting signal which is proportional to the lateral gust input is put tnrough a band pass filter before being summed with the existing yaw damper signal to drive the upper rudder. The purpose of this filter is to remove steady-state sensor errors and to prevent excitation of the flexible body modes. The approximate location of the wind vane and other system components on the 747 airplane is shown in Figure 16.

ANALYSIS

For the purposes of the analysis, it was assumed that the lateral dynamics could be considered independently and that only lateral gusts were present. The assumed form of these gusts was the typical Von Karman spectrum.

The vane output can be described as:

$$\delta = \frac{GUST}{V_p} + \frac{V_B}{V_p} + \frac{L_1 R_B}{V_p} + \frac{H_1 P_B}{V_p} \qquad (1)$$

where the last three terms give the sideslip angle at the vane location.

To derive a signal proportional to the gust input use is made of a lateral accelerometer mounted at the vane station. The accelerometer output is:

$$A_y = \dot{V}_B + U_B R_B - W_B P_B + L_1 \dot{R}_B + H_1 \dot{P}_B - g \cos\theta \sin\phi \qquad (2)$$

consequently,

$$\frac{V_B}{V_P} + \frac{L_1 R_B}{V_P} + \frac{H_1 P_B}{V_P} - \frac{1}{V_P}\int W_B P_B \, dt = \frac{1}{V_P}\int (A_y + g\phi - V_B R_B)\, dt \qquad (3)$$

or approximately,

$$\frac{V_B}{V_P} + \frac{L_1 R_B}{V_P} + \frac{H_1 P_B}{V_P} \simeq \frac{1}{V_P}\int (A_y + g\phi - V_P R_B)\, dt \qquad (4)$$

It is therefore possible to rewrite (1) as:

$$\delta - \frac{1}{V_P}\int (A_y + g\phi - V_P R_B)\, dt = \frac{GUST}{V_P} \qquad (5)$$

all the left side terms of (5) are available and this equation is the basis for mechanization of the system as shown in Figure 15.

The analysis was made using a Boeing derived computer program which accepts matrix inputs. This program provides root locus plots of the system and power spectral densities of designated parameters in response to given forcing functions. The complete analysis included consideration of the lateral airplane dynamics, roll, autopilot, yaw damper and gust suppression system. The performance of the gust suppression system was investigated throughout the flight envelope of the airplane with the intent of determining

optimum system gain for reduction of the rear fuselage lateral acceleration and also to determine system stability. Some particular results of the analysis are shown in Figure 17. Figure 17 shows a root locus plot for different gains of the gust suppression system with the corresponding RMS 'g' levels at an aft body station shown in Figure 18. It can be seen that a reduction of about 30% in the RMS 'g' level can be obtained at the bucket of the curve shown in Figure 18. This particular gain affects the airplane stability very slightly as can be seen in Figure 17. Similar results were obtained for various airplane altitudes and speeds, the value of gust suppression gain remaining essentially the same for minimum 'g' levels.

The reason for the change in airplane stability is the approximate form adopted for compensating the vane output for airplane motion. Theoretically, this signal could be perfect, in which case, the root locus shown in Figure 19 results for all system gains. The approximate method of compensation was chosen for practical implementation.

TEST RESULTS

A system corresponding to that shown in Figure 15 was constructed and test flown in the 747 airplane. Initial flights were made to calibrate the wind vane sensor and to investigate airplane handling with the system engaged in calm air. Pilot comments were that the operation of the system had undetectable effect on handling characteristics in either normal or emergency maneuvers. Subsequently, several flights to investigate performance during turbulence were made. Typical data from one such flight is shown in Figures 20 and 21. Figure 20 shows a typical gust signal command measured at the input to the Beta filter summing amplifier while Figure 21 shows the RMS 'g' levels recorded at the aft body station with the system ON then OFF in sequence. The reduction in acceleration levels with the system ON is of the same magnitude as that predicted.

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SERVICE EVALUATION

To obtain more data on the system, it has been installed on a commercial carrier airplane with a limited instrumentation package. Because this installation operates at a reduced gain while information is being collected, the results do not show such a large reduction in acceleration levels as those obtained during Boeing tests. A typical example of some of this data is shown in Figure 22, where a comparison of the acceleration levels at an aft body station during turbulence is shown with the system ON and OFF.

CONCLUSION

The development and testing of the Beta-vane and MSAS systems have been described. Data from flight test have indicated that a 50-70 percent reduction in aft body lateral acceleration levels can be achieved with the above systems. Non-linear filtering and different sensors will be the subject of future research.

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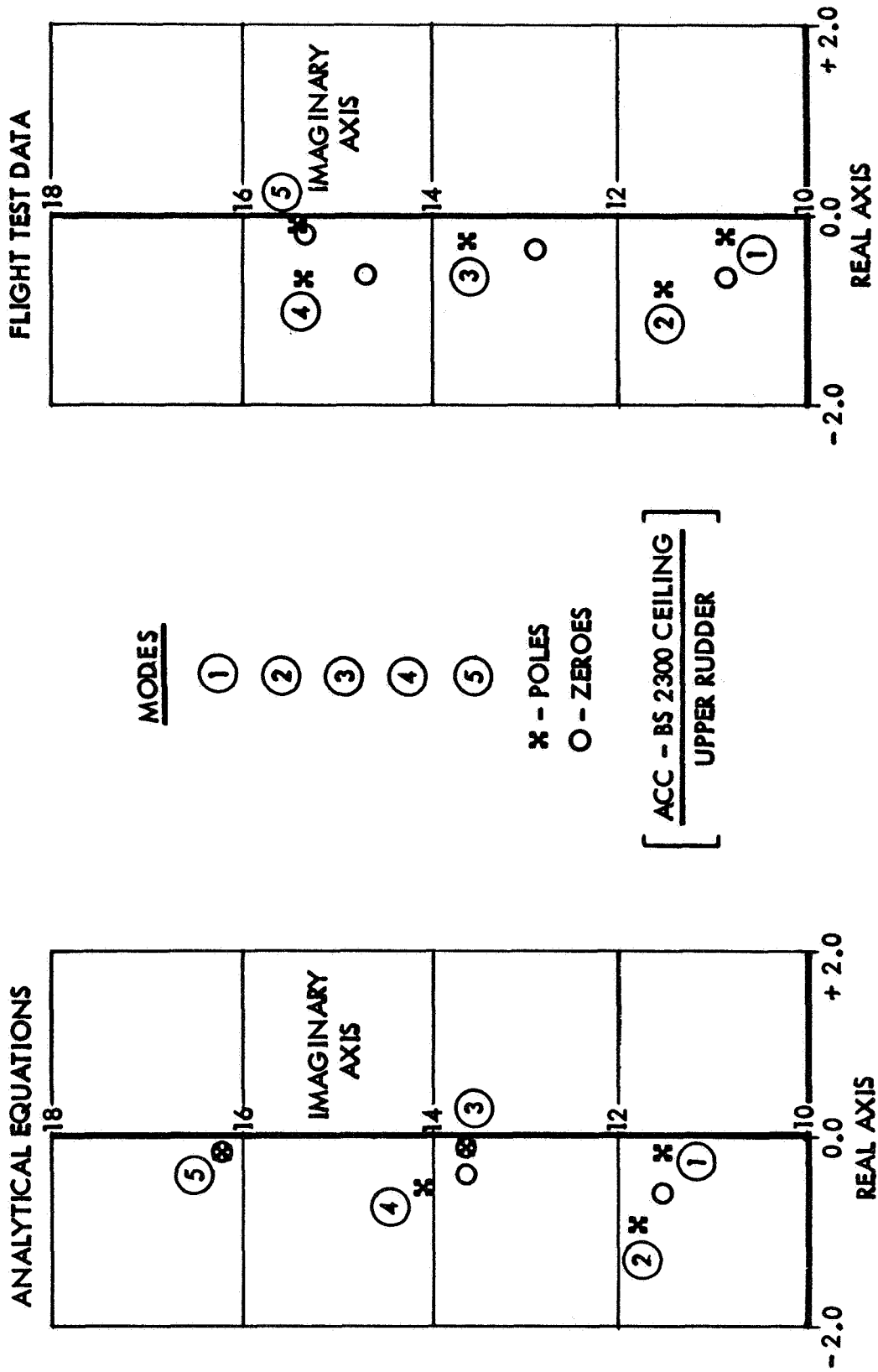
COMPARISON OF ANALYTICAL AND FLIGHT TEST DATA

FIGURE 1

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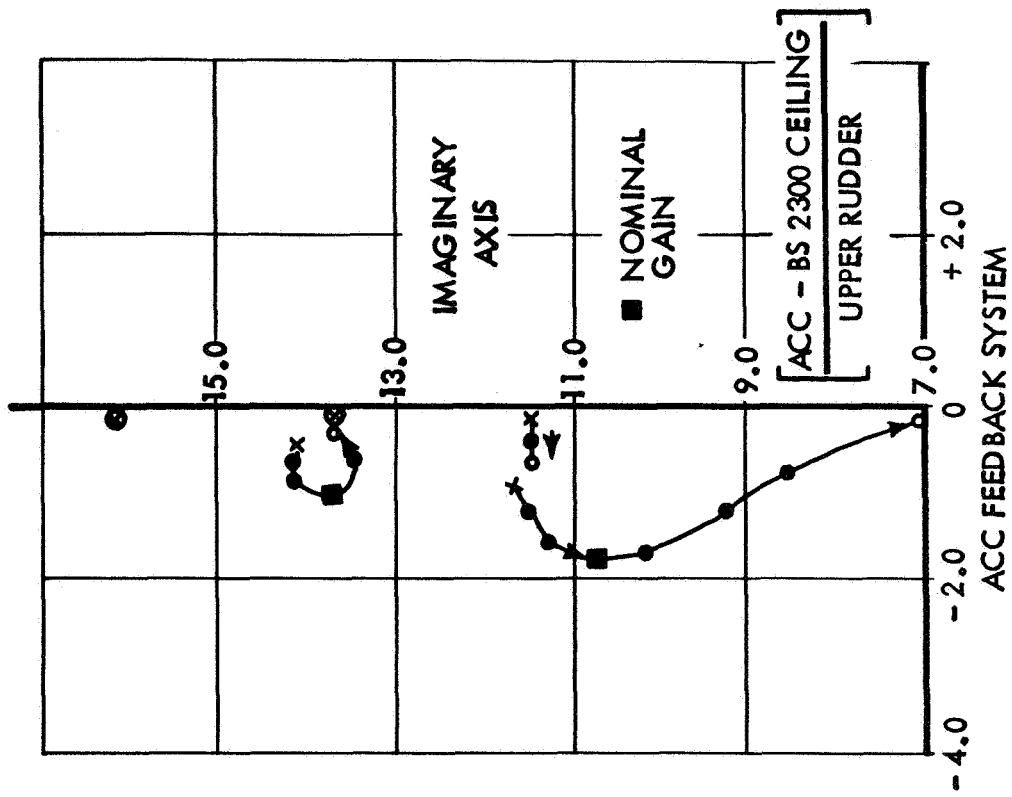
ROOT LOCUS PLOT OF CONTROL SYSTEM BASED ON ANALYTICAL EQUATIONS

FIGURE 2
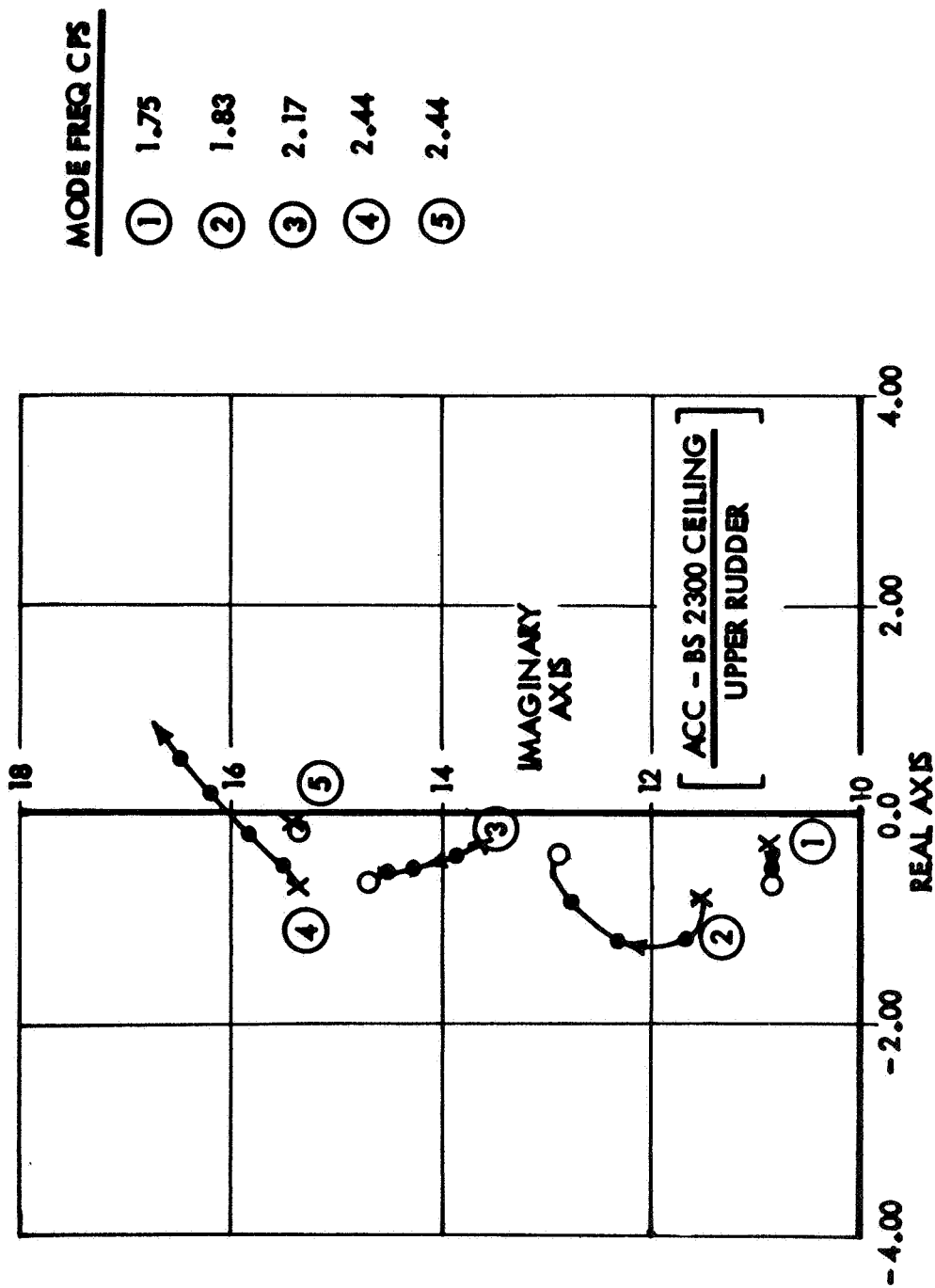
ROOT LOCUS OF PLOT CONTROL SYSTEM BASED ON CURVE FIT COMPUTER PROGRAM

FIGURE 3

388
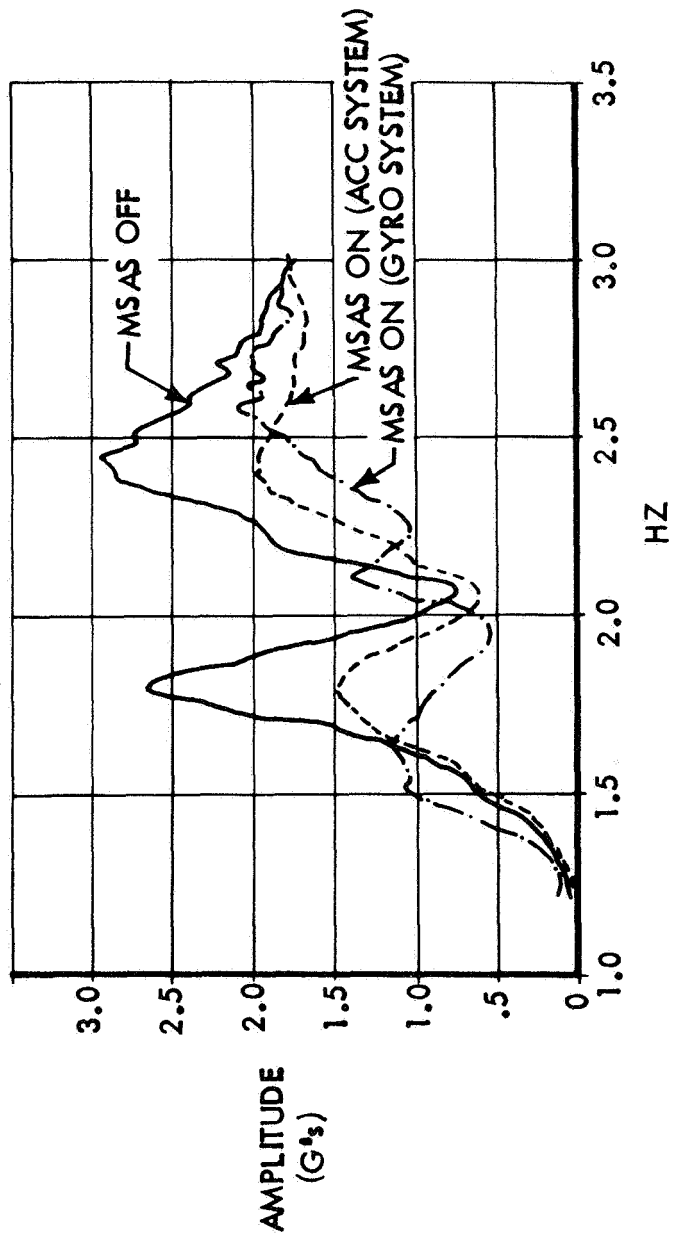
TRANSFER FUNCTION

LATERAL ACCELERATION (AFT BODY)
RUDDER (LOWER)

- ALTITUDE = 31,000 FT
- MACH = .86

FIGURE 4

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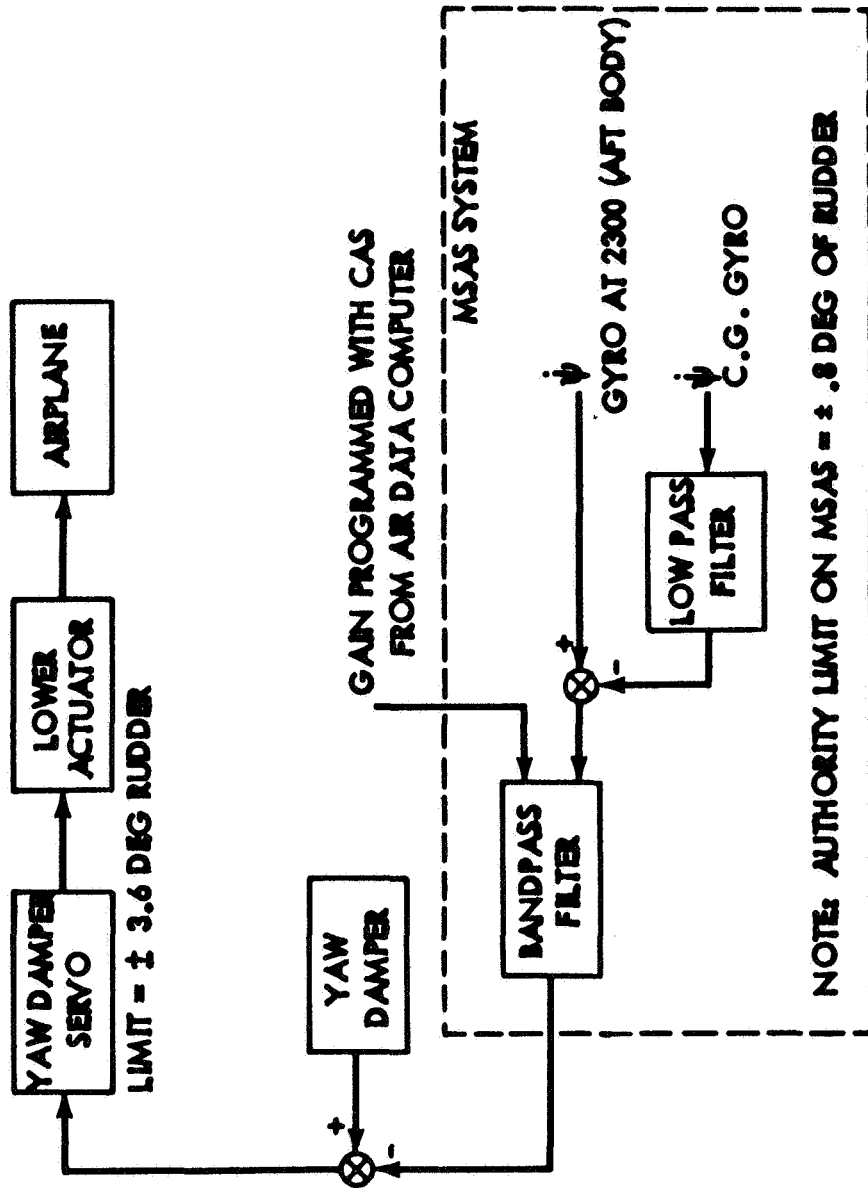
BLOCK DIAGRAM OF MSAS CONTROL SYSTEM

FIGURE 5
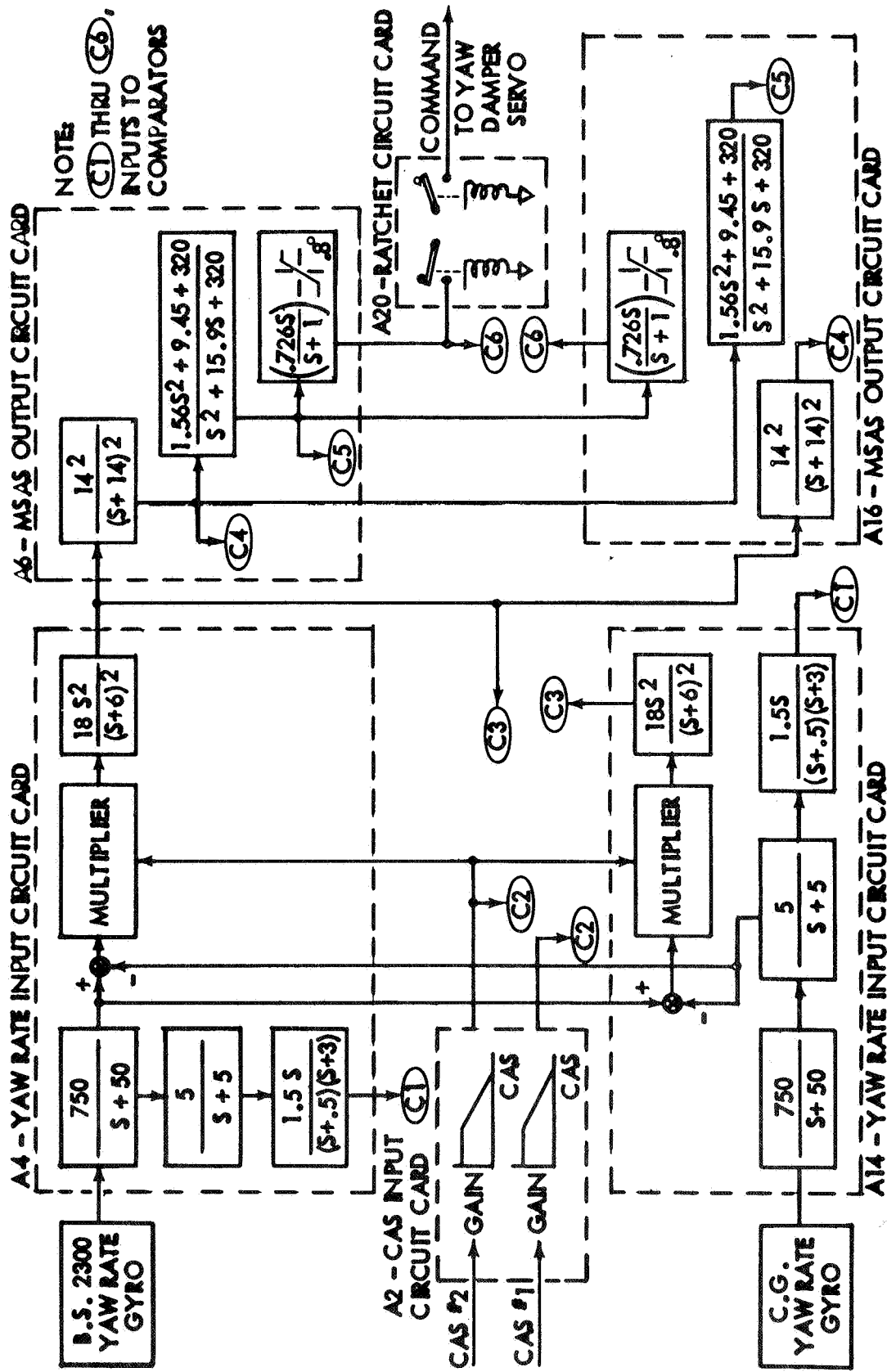
SYSTEM DIAGRAM MODAL SUPPRESSION AUGMENTATION SYSTEM

FIGURE 6

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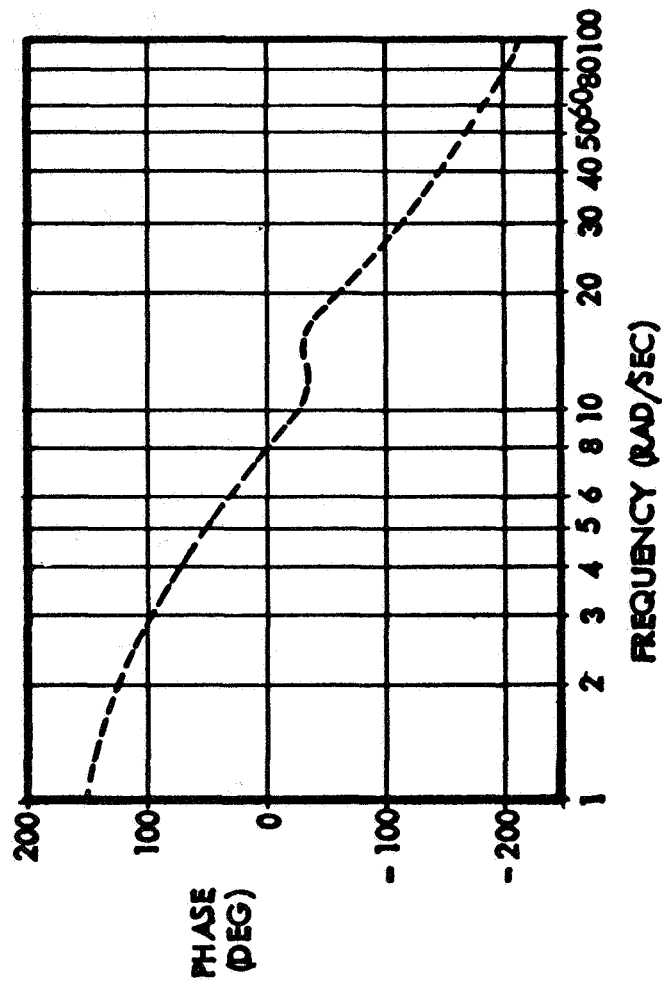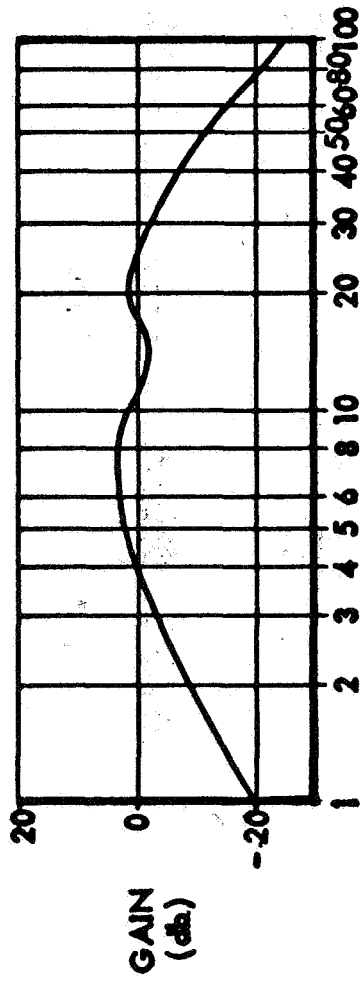
GAIN AND PHASE PLOT OF MSAS FILTER

FIGURE 7

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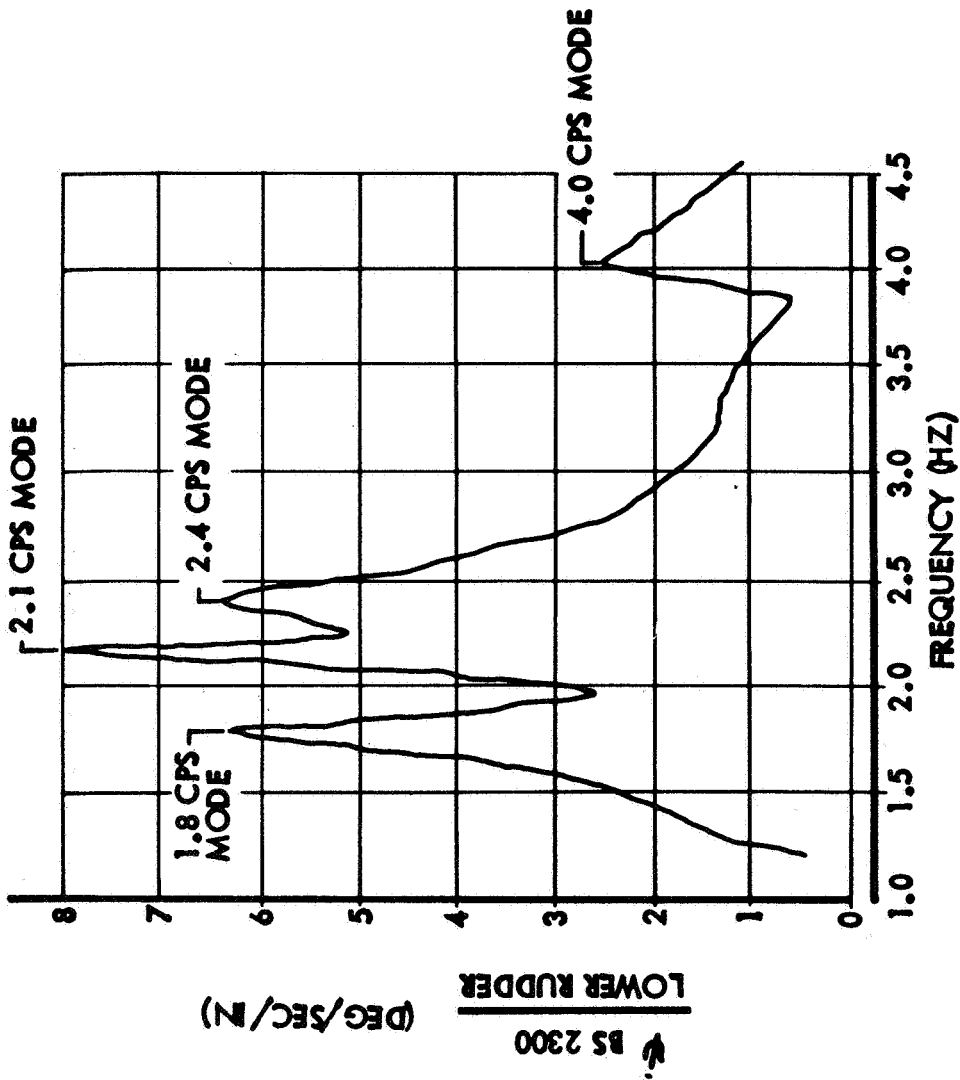
AMPLITUDE RESPONSE − $\dot{\psi}_{BS\,2300}/$ LOWER RUDDER INPUT

FIGURE 8

X OPEN LOOP POLES
O OPEN LOOP ZEROS
▲ NOMINAL GAIN

2.4 CPS

FILTER ZERO

2.1 CPS

IMAGINARY
AXIS

1.8 CPS

α 1.65 CPS

REAL AXIS
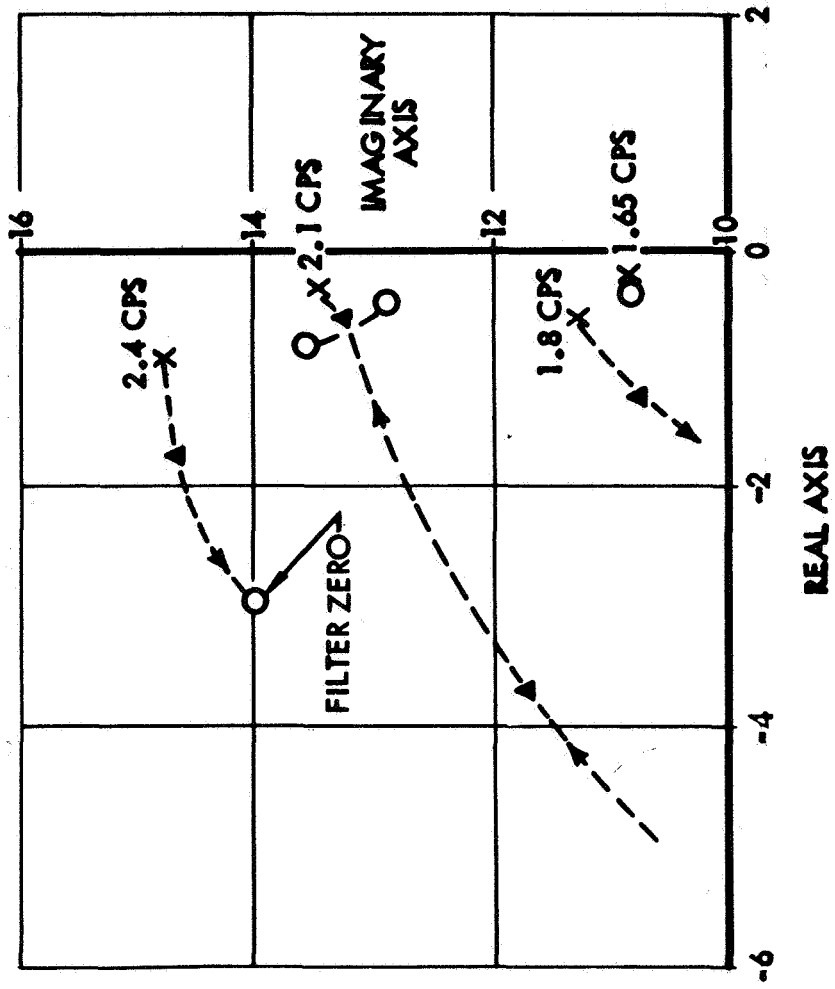
ROOT LOCUS PLOT OF CONTROL SYSTEM

FIGURE 9

394
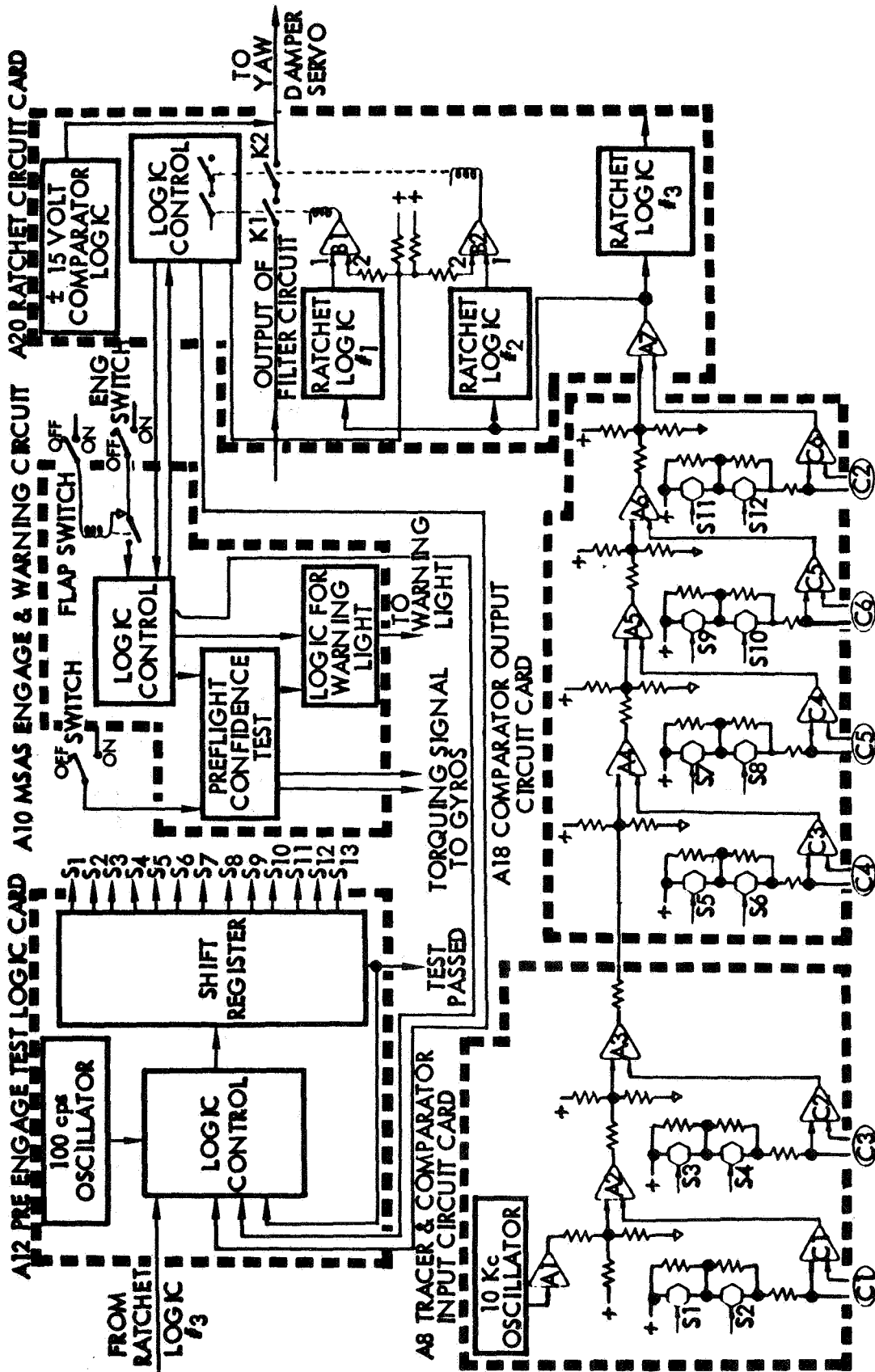
FUNCTIONAL BLOCK DIAGRAM OF MSAS MONITORING SYSTEM

FIGURE 10

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SYSTEM ON

SYSTEM OFF

SYSTEM ON
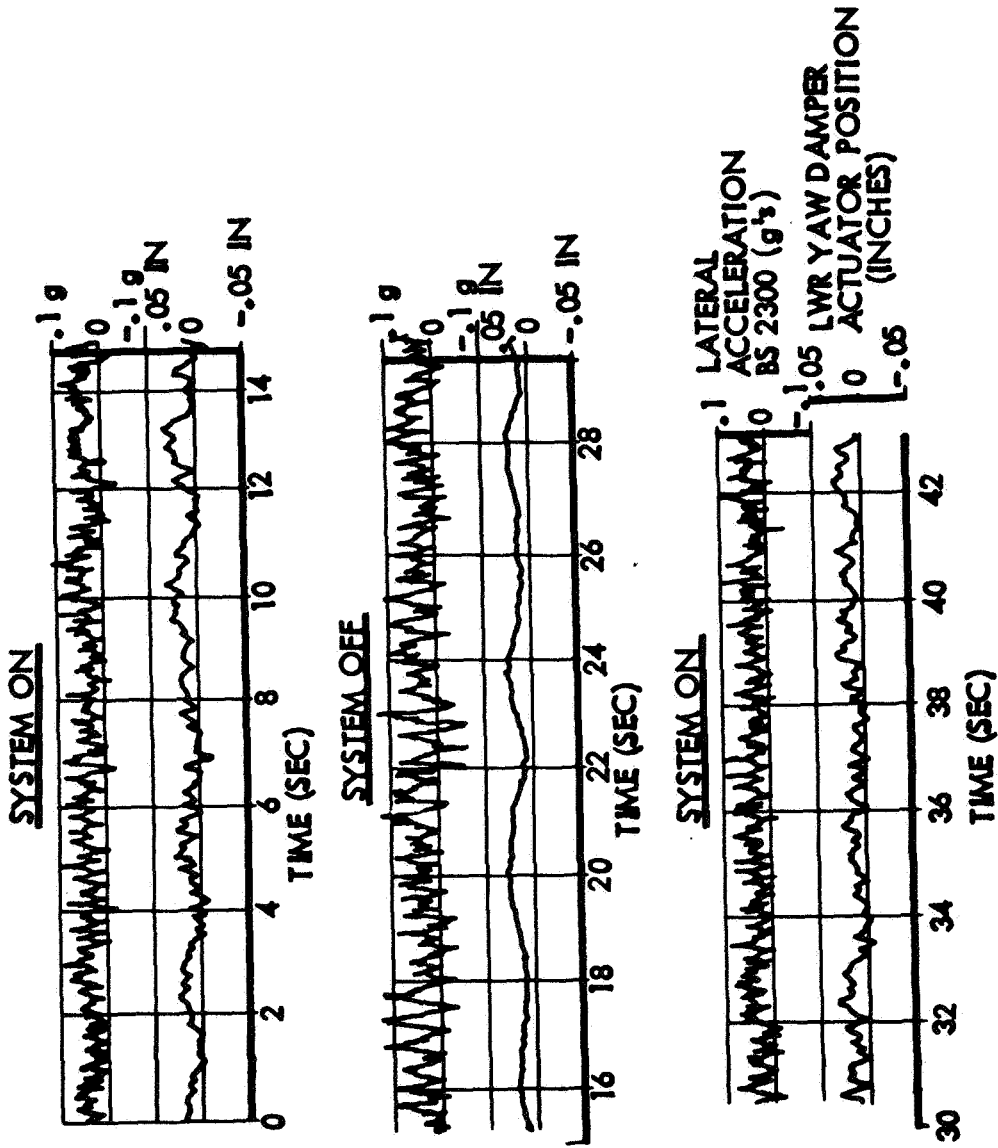
AIRPLANE RESPONSE IN TURBULENCE WITH AND WITHOUT MSAS

FIGURE 11

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ACCUMULATIVE RMS – TURBULENCE

FIGURE 12

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MSAS ON

MSAS OFF

MSAS ON

FREQUENCY (HZ)

LAT ACC MEASURED AT PILOT'S STATION - G's
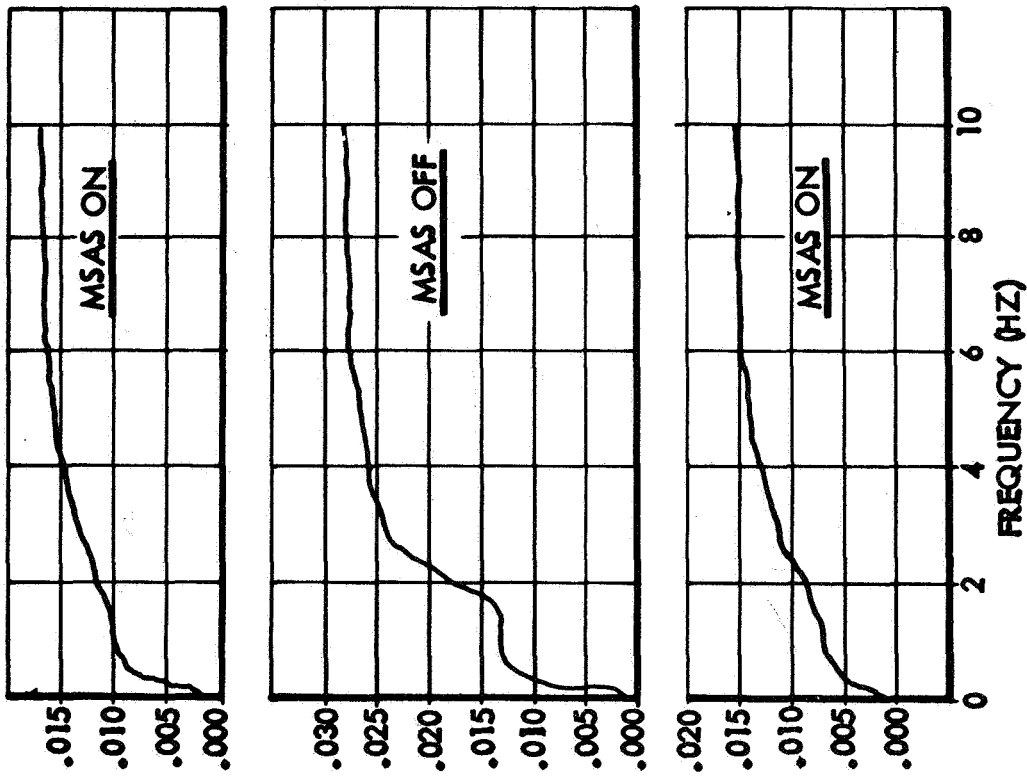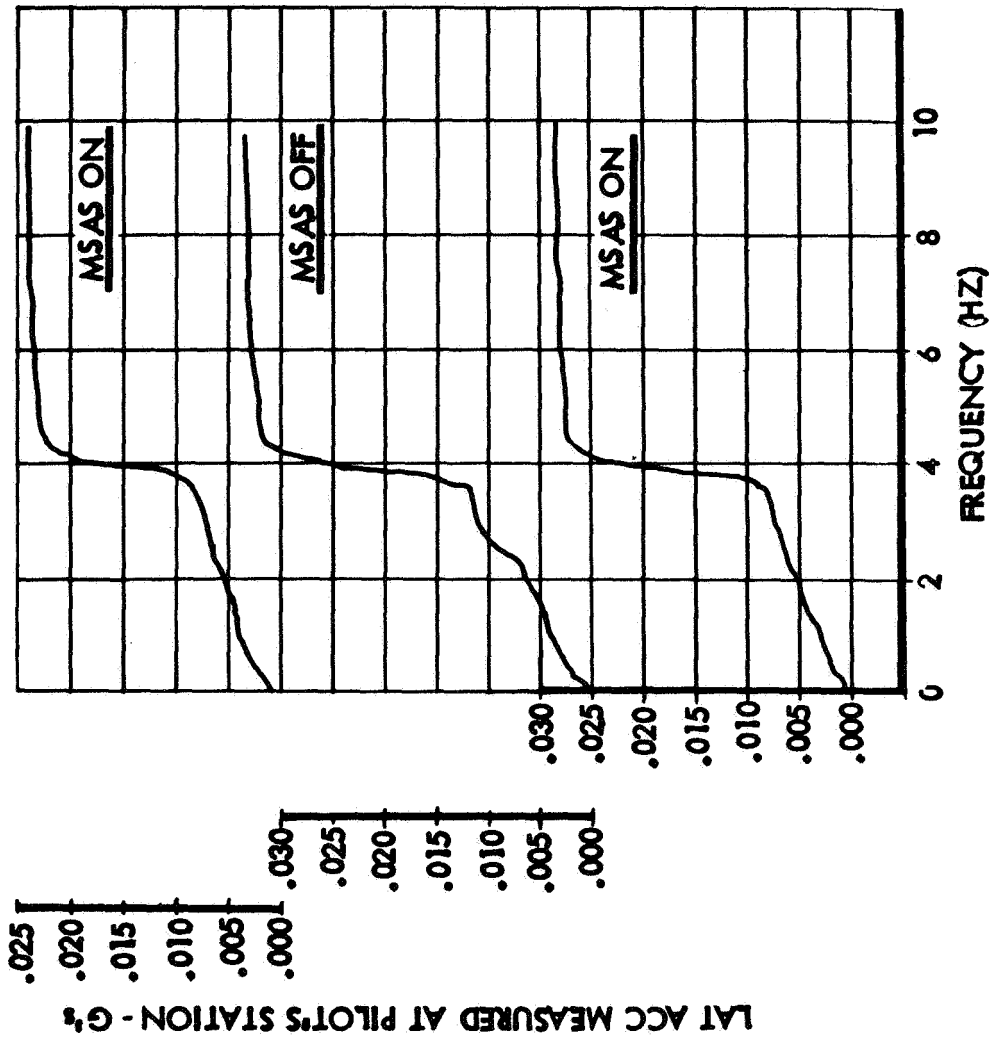
ACCUMULATIVE RMS - TURBULENCE

FIGURE 13

398

FREQUENCY (HZ)
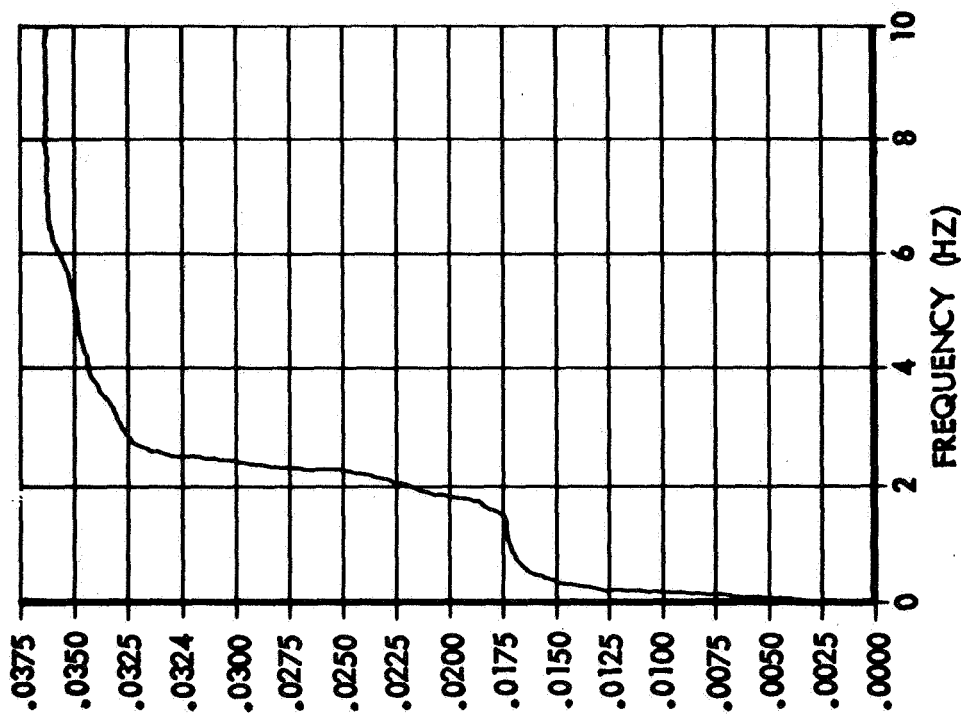
CUMULATIVE RMS – QANTAS BETA

SYSTEM EVALUATION IN TURBULENCE

FIGURE 14

399
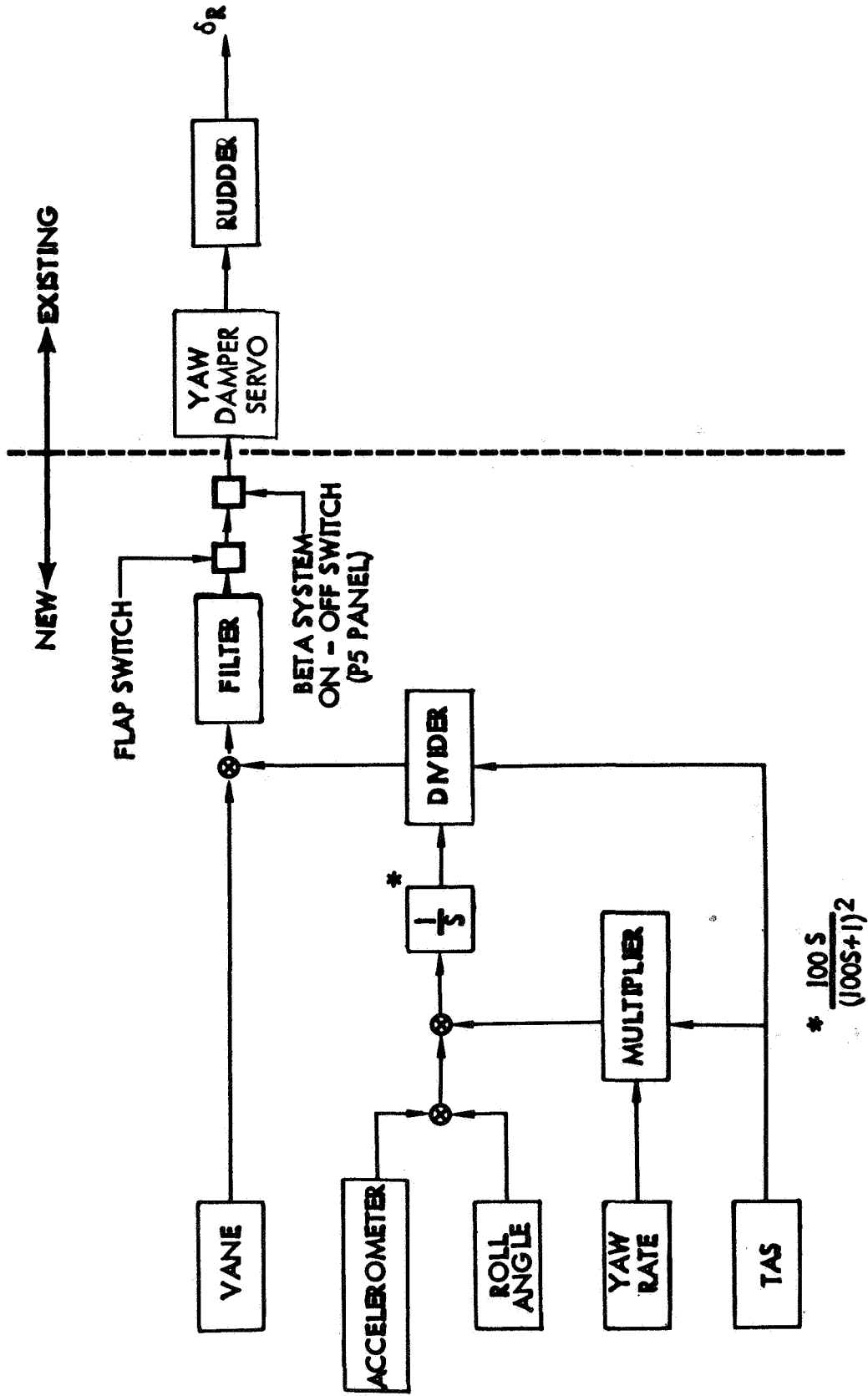
BETA MODIFIED YAW DAMPER BLOCK DIAGRAM

FIGURE 15

400

BETA VANE SERVO DRIVE SCHEMATIC

FWD

PROBE SECTION

PLENUM

BELLOWS

+ REF

- REF

SERVO MOTOR AND
GEAR TRAIN

B.S.
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(VANE LOCATED APPROX 5 IN. TO
RIGHT OF A/P ₵)

ACCELEROMETER

BETA VANE
(ORIGINAL
LOCATION)

ACCELEROMETER   B.S. 426

YAW DAMPER

ACCELEROMETER

B.S. 210

BETA VANE
(FINAL LOCATION)
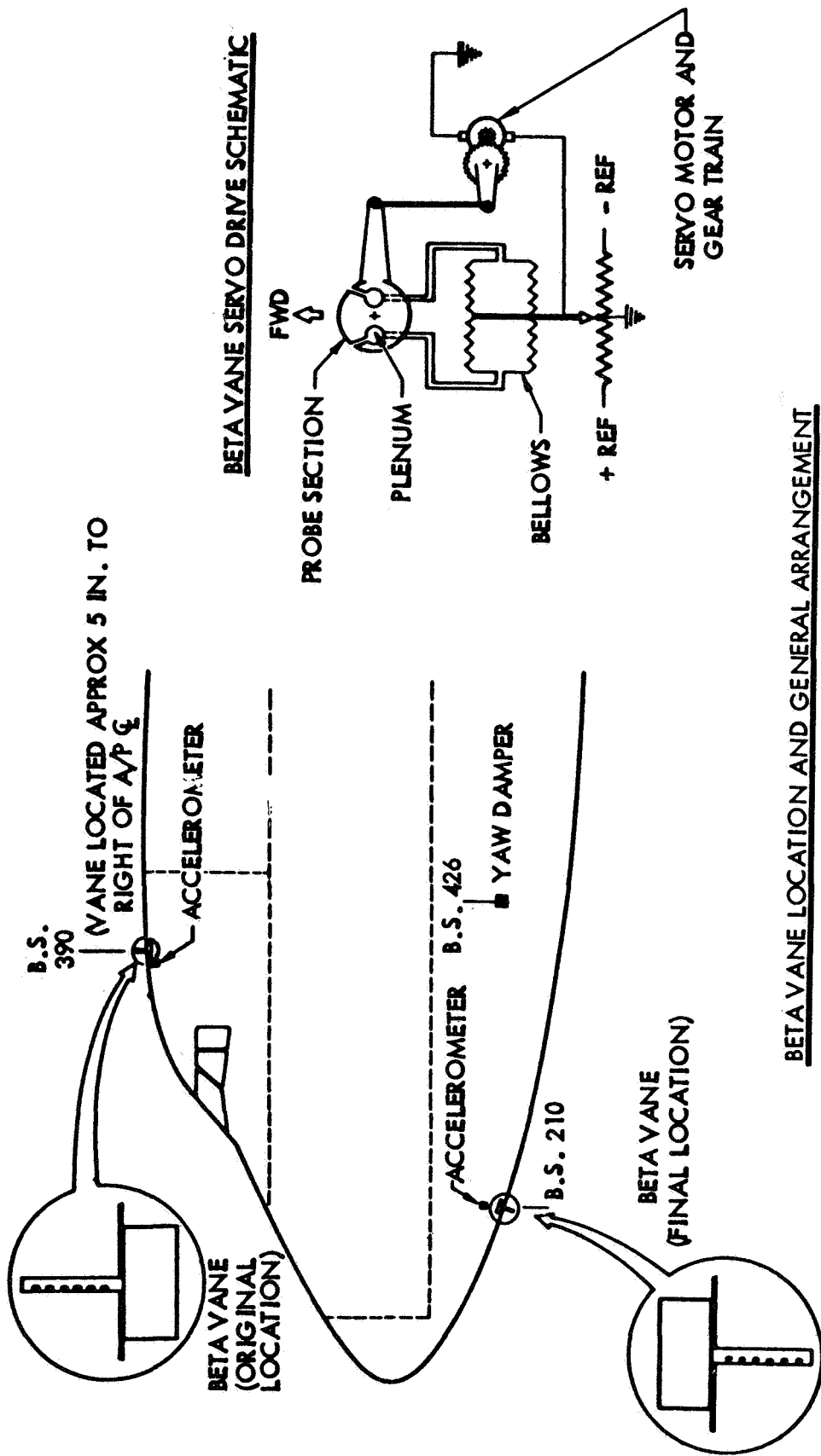
BETA VANE LOCATION AND GENERAL ARRANGEMENT

FIGURE 16

401

ROOT LOCUS FOR PARTIALLY COMPENSATED SYSTEM WITH NOMINAL YAW DAMPER GAIN

FIGURE 17

402
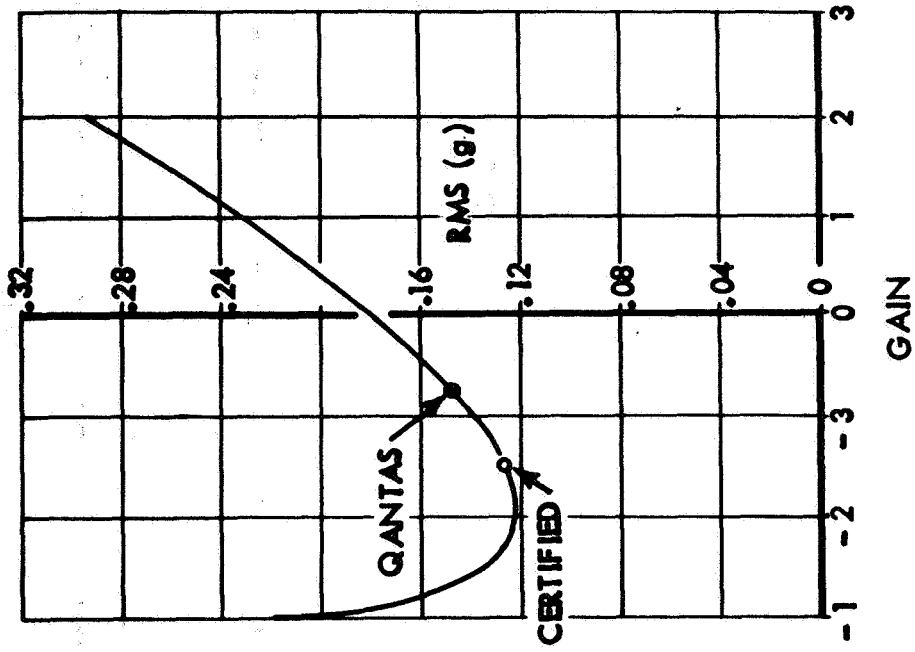
CUMULATIVE ROOT MEAN SQUARE ACCELERATION (BS 2300) VERSUS SYSTEM GAIN

Actual Compensation (Beta System)

FIGURE 18

IMAGINARY AXIS

△ SYMBOL FOR MULTIPLE POLE OR ZERO COMBINATIONS

DUTCH ROLL ROOT

REAL AXIS
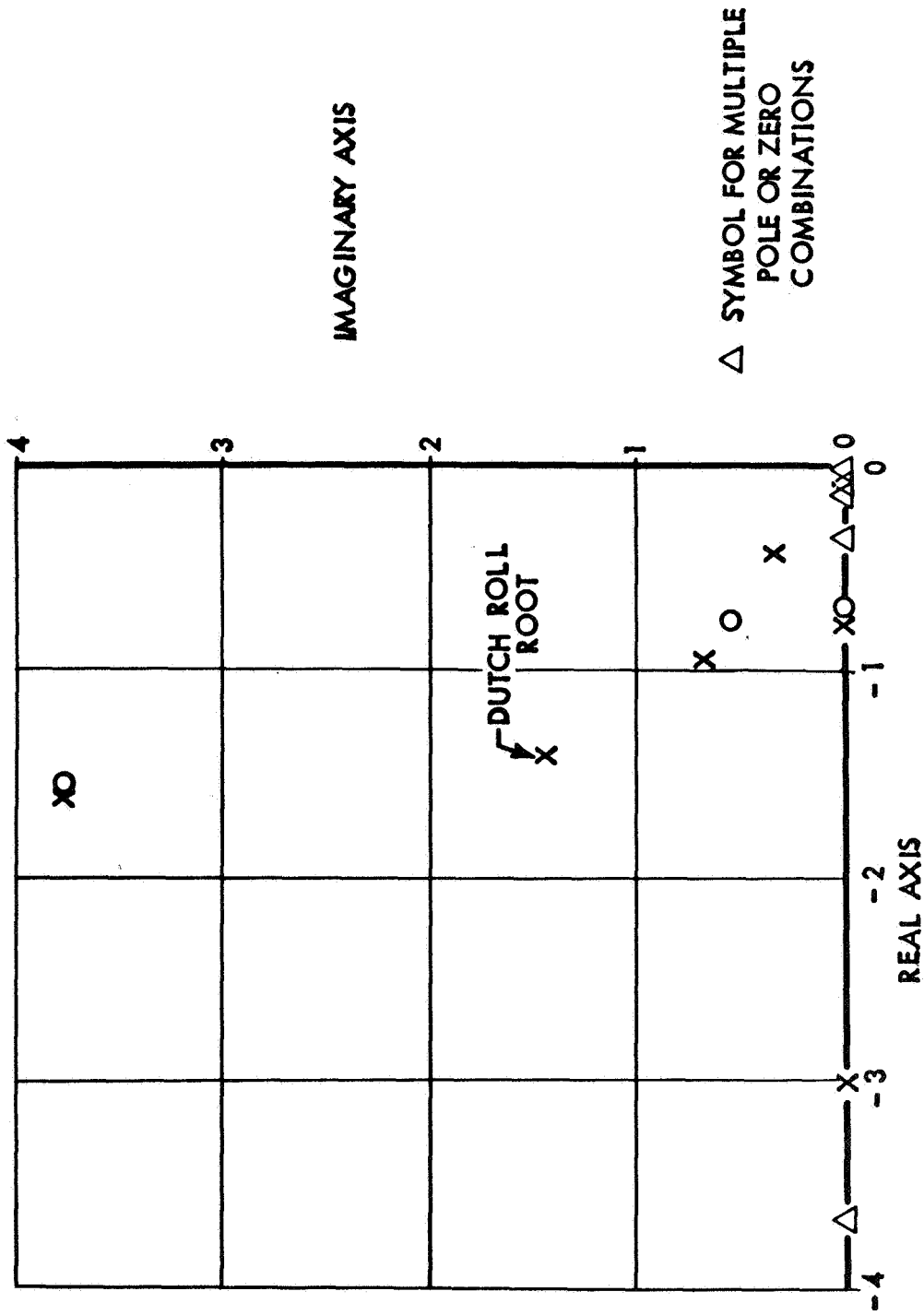
ROOT LOCUS FOR PERFECTLY COMPENSATED SYSTEM

Reduced to Airplane Roots Only

FIGURE 19

404
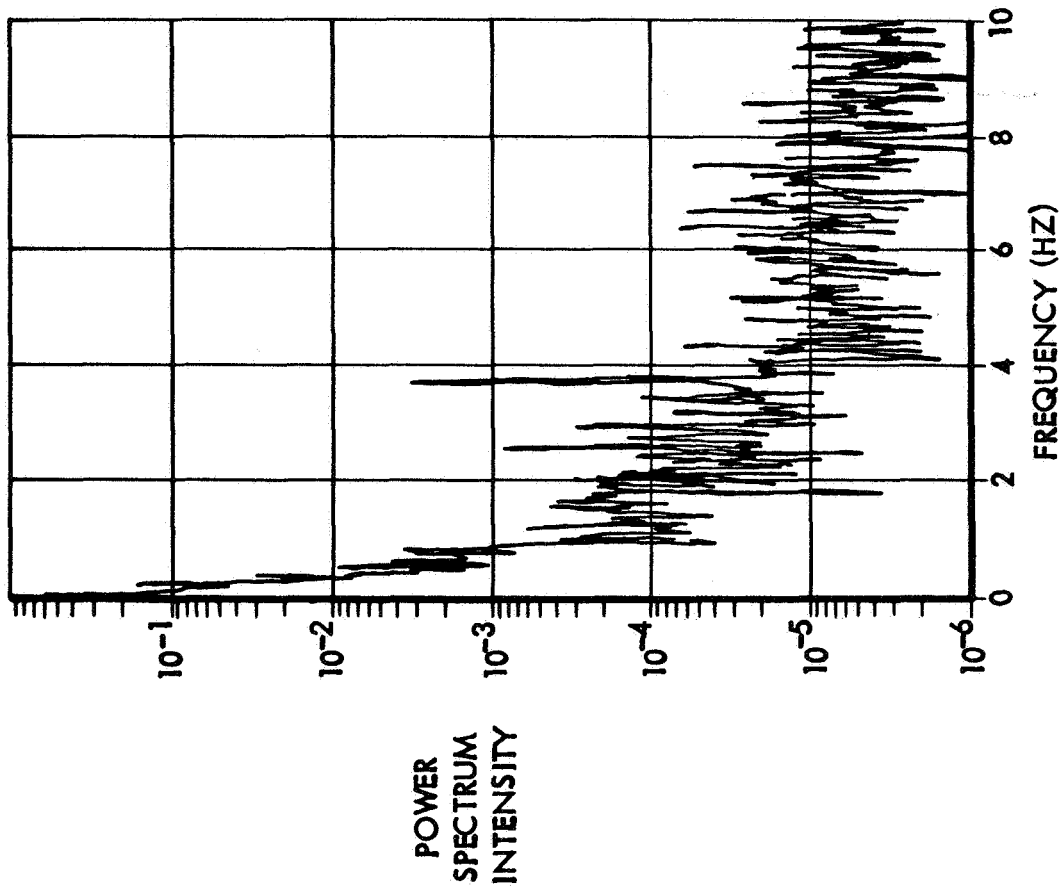
POWER SPECTRUM OF COMPENSATED BETA SIGNAL MEASURED IN TURBULENCE

FIGURE 20
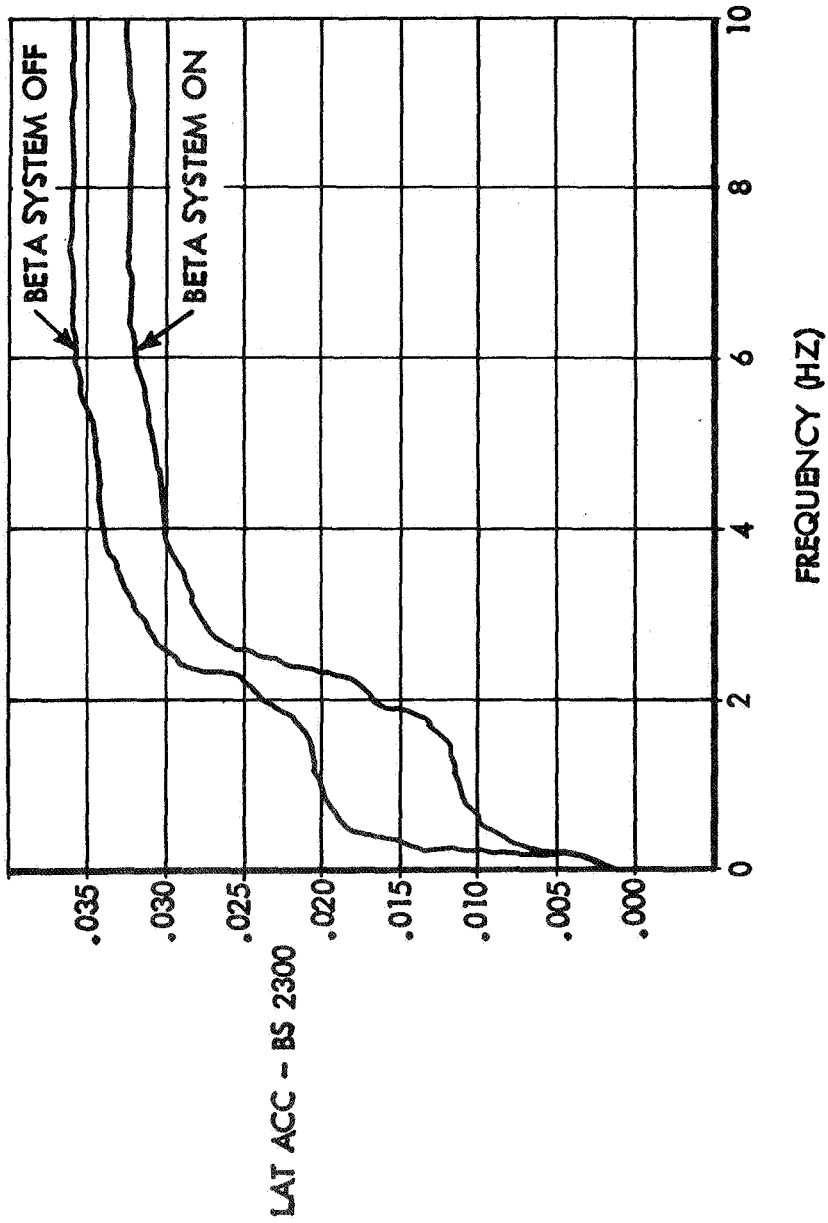
CUMULATIVE RMS

Beta System In Turbulence

FIGURE 21

406
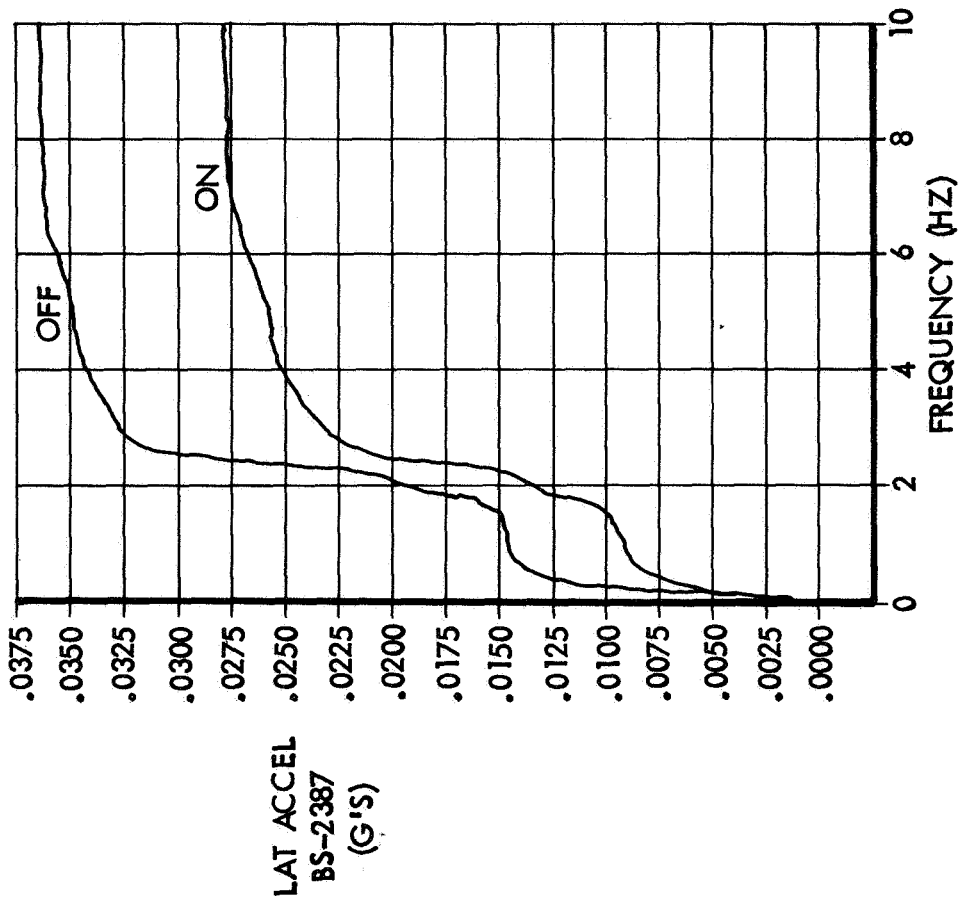
CUMULATIVE RMS QANTAS BETA
SYSTEM EVALUATION IN TURBULENCE

FIGURE 22

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DECOUPLING CONTROL TECHNOLOGY FOR MEDIUM STOL TRANSPORTS

Daniel K. Bird and Terry L. Neighbor
Air Force Flight Dynamics Laboratory

SUMMARY

This paper discusses the advanced control technology necessary to cope with the Medium STOL Transport landing problem and, in particular, the necessity to decouple with active control techniques. It will be shown that the need to decouple is independent of the powered-lift concept but that the provisioning for decoupling is most greatly dependent on the preassumed piloting technique. The implications of "decoupling" and "active control techniques" with respect to pilot technique options, handling quality criteria, flight control mechanization, and the use of piloted simulation as a design tool, will also be discussed.

INTRODUCTION

The Medium STOL Transport (MST) flight control system must play a major role in combining good up-and-away transport performance with good STOL capability. This STOL capability entails routine operation from a 2000 x 60 ft. strip. The use of powered-lift to provide the satisfactory low speed performance directly adds the need for active control technology while introducing many new unknowns, and aggravating the problem of engine failures. The basic foundation for this paper is derived from the results of recently completed AFFDL studies to develop the necessary MST technology. These studies involved three contractors, Boeing, General Dynamics and North American under AF Contracts F33615-71-C-1757, F33615-71-C-1754 and F33615-71-C-1760 respectively and included a collective total of approximately 500 hours of direct piloted simulation evaluations. (Refs. 1, 2, 3, 4, 5)

When attacking a problem area as large as the flight control system development for an MST, we quite often lose sight on the key points and tend to get bogged down in minor intracacies. It is easy to get involved in trivial arguments concerning the "hardware" in the kitchen before a suitable "foundation" for the house has been established. A stand-back-and-survey

view will therefore be presented with the hope that it will be enlightening. This stand-back position is the authors' main advantage. Because of our exposure to all three design efforts by the contractors, we saw certain patterns and restrictions occurring that have more meaning collectively than individually. A useful interpretation of these patterns and restrictions (real or self-imposed) is the main contribution we seek to present.

SYMBOLS

AFFDL	Air Force Flight Dynamics Laboratory
AFCS	Automatic Flight Control System
MST	Medium STOL Transport
STOL	Short Take-off and Landing
C_ℓ	Rolling moment coefficient
C_n	Yaw moment coefficient
V	Velocity, knots
n_z	Normal acceleration, g's
n_z/α	The steady-state normal acceleration change per unit change in angle of attack for an incremental elevator deflection at constant speed, g's/rad
α	Angle of attack, degs
β	Sideslip angle at the center of gravity, degs
γ	Flight path angle $= \sin^{-1} \dfrac{\text{vertical speed}}{\text{true speed}}$, positive for climb, degs
r	Yaw rate, deg/sec
θ	Pitch angle, degs
ψ	Airplane heading, degs

MST FLIGHT CONTROL PROBLEM

Mission Origin

The Medium STOL Transport (MST) flight control problem has its origin in the mission goals. Briefly, these mission goals seek a capability of delivering a 28,000 pound payload into a short, narrow (2000' x 60') austere landing strip, in addition to having a cruise Mach number of 0.75, an operation radius

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of 500 N.M. and a ferry range of 2600 N.M. The five fundamental phases of the flight control problem are:

1. Take-off

2. Cruise

3. Transition from Cruise to STOL configuration

4. STOL Approach

5. Transition from STOL Approach to Ground Roll

Of these phases, the "STOL Approach" receives the first, and justifiably, the most attention. This emphasis is due to the relative impact of this phase on the flight control system, both in defining requirements and limitations. While, in this paper, we will concentrate on the STOL Approach, the problems which arise from these other phases cannot be ignored.

It is appropriate to discuss the take-off briefly. There is a tendency to refer to "Take-off and Landing" as a joint lumping of a common problem area for flight control design. For the MST's, particularly, the take-off is a performance dominated ground-to-air problem whose influence on the flight control system is almost trivial compared to the air-to-ground landing problem.

STOL Approach Problem

The performance of a STOL landing on a 2000' x 60' runway necessitates low touchdown energy and reduced touchdown dispersions. To achieve these, a MST approaches at a low speed and steep flight path angle.

This low speed and steep approach angle leads to operation on the "backside" of the power-required curve and employment of active powered lift capability. The most prominent powered-lift systems under consideration are shown in Fig. 1. These powered-lift systems present coupling problems by their nature. The propulsive power that is now used directly to increase lift, also influences other force and moment generation. The "backside" area of the power curve effect is shown in Fig. 2. Note that for any velocity in this area, an increase in thrust setting at constant attitude results in an increased flight path angle with an accompanying decrease in airspeed. Similarly, if attitude is increased (with a fixed thrust setting) the aircraft responds to decrease velocity and increase rate of descent. This adverse coupling of attitude, airspeed and flight path, therefore, replaces the

favorable coupling associated with operation on the "frontside" of the power-required curve. The problems associated with a large angle between the flight path vector and the airplane body axis extend into the lateral-directional axes and further aggravate the normal coupling in these axes.

The large control surfaces required for low "q" operation add forces that are unfavorable in addition to the moments they are designed to produce. For example, a large elevator, in addition to producing a pitching moment, may produce a lift loss of significant magnitude.

A particular consequence of the low speed required is the increased sensitivity to atmospheric disturbances, i.e., wind shears, gusts, and turbulence, as they relate to both airborne controllability and landing precision. Internal disturbances such as engine failures and perhaps in combination with the "external" disturbances must be coped with.

Finally, in terms of conventional "flying quality" parameters, the dynamic characteristics of the "bare airframe" during landing approach can be generally characterized (Refs. 1, 2, 3, 4, 5) as having:

     (1)   Strongly divergent spiral modes

     (2)   Low Dutch Roll damping

     (3)   Long roll mode time constants

     (4)   Low n_z/α sensitivity

     (5)   Strongly coupled "short period" and "phugoid" modes

Item 5 stems directly from the attitude-speed-flight path coupling mentioned earlier and is worthy of special comment. The classical simplification to separately identify the "short period" mode as an oscillation of (α) at constant speed and the phugoid mode as an oscillation of (V) and (γ) at constant angle of attack is not valid. (Ref. 6) "Short term" and "long-term" response are as important as ever but they cannot be satisfactorily developed within the context of the classical "short-period" and "phugoid" modes.

FLIGHT CONTROL SYSTEM DEVELOPMENT

The control of a "powered-lift" airplane for the short field capability desired for MST's must, therefore, take a far more basic approach to the flight control system development than would normally be required. The suggested plan of attack is indicated by an article which appeared in Aerospace

Engineering, September 1962, entitled, "Control Response Requirements", by Waldemer O. Breuhaus and William F. Milliken, Jr. The authors made the simple but significant point that all flight control can be broken down into three basic types: (1) Up-Down, (2) Right-Left, and (3) Fast-Slow. Although the article was written within the context of conventional controls, i.e., elevators, ailerons, rudders and throttles, the present day correlation with (1) Direct Lift, (2) Direct Side Force, and (3) Direct Drag, is obvious. The presentation that follows is designed to show that the "decoupling" philosophy referred to in 1962 as an interesting area to investigate has become a basic consideration in the design of flight control systems for the MST's.

A typical landing approach portion of the MST landing flight task is shown in Fig. 3. This part of the landing flight task will be emphasized because it received the most attention in the completed studies. It is hard to over-emphasize, however, that the transition from up-and-away flight to the desired landing approach speed-flight path profile and the transition from the landing approach to actual landing and deceleration to a stop, or a go-around, must receive careful attention in the final control system development.

Longitudinal Control

"The two principal quantities that need to be controlled in symmetric flight are the speed and the flight path angle, that is to say, the vehicle's velocity vector. To achieve this obviously entails the ability to apply control forces both parallel and perpendicular to the flight path." (Ref. 6) This enlargement of what longitudinal control really is, as compared to the too often made assumption that longitudinal control is limited to "elevator control", is one of the important messages of flight control development for the MST's. The second is the supposition that the pilot and/or autopilot must be able to make commands for speed changes without materially affecting flight path and conversely the ability to make flight path angle changes without materially affecting speed. Fig. 4 illustrates the design approach indicated for the longitudinal control provisions. There is no commitment at this point as to what input device will be used to command speed change or flight path change or what force or moment generators will be used for control. There is no direct control of (θ) or consequently (α). The assumption is made that the airplane is "trimmed" for a given speed-flight path profile and that detection

of changes in this profile, (V) speed and (γ) flight-path, can be detected and acted upon by the pilot and/or autopilot. Fig. 4 serves as a common starting point to discuss the development of two MST longitudinal flight control systems by two separate contractors. One of the contractors featured an Externally Blown Flap (EBF) version for his study model and the other used a Mechanical Flaps plus Vectored Thrust version.

EBF Version

The key to the manner in which this would develop was this early statement, "In addition to the elevator, throttle and flaps are available for flight path control. The literature and experience indicates that the pilot would like to control flight path with the throttle on a power approach, where significant lift is due to the throttle. However, the coupling of airspeed and flight path through each of these controls makes bare airframe control deficient."

In effect, this philosophy indicates a strong preference to change the general form of Fig. 4 to assign a throttle level as the flight path command device. The same contractor goes on to say, "A direct lift system via throttle control gives the pilot two distinct means of controlling flight path:

1. Heave control with the throttle, with minor pitch changes.
2. Pitch control with the elevator, which depends on an adequate (n_z/α) to minimize (α) changes and make pitch changes result in flight path changes.

The equation $(\gamma = \theta - \alpha)$ expresses the two techniques, the heave control corresponding to changing (γ) with (α) and pitch control changing (γ) with (θ).* The linear derivatives indicate the throttle to be a better direct

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*This statement is particularly interesting and significant for powered lift MST's. The general definition of $(\alpha) = \tan^{-1}w/u$ does not exclude this concept of (γ) change with (α). It is a change in the conventional sense of (α), however, that must be carefully recognized in the application of many existing parameters. n_z/α for example, is defined in MIL-F-83300 as "the steady state normal acceleration change per unit change in angle of attack for an incremental pitch control deflection at constant speed", and in MIL-F-8785 as "the steady-state normal acceleration change per unit change in angle of attack for an incremental elevator deflection at constant speed (airspeed and Mach number)".

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lift control and the flaps as better speed control as seen by:

$$\frac{Z_{thrust}}{X_{thrust}} = \frac{-0.1316}{+0.0351} = -3.75 \qquad \frac{Z_{flaps}}{X_{flaps}} = \frac{23.48}{14.89} = 1.575''$$

where

$$\frac{Z_{thrust}}{X_{thrust}} =$$ Ratio of change in vertical force (Z-axis) per change in horizontal force (X-axis) for a given change in engine thrust.

and

$$\frac{Z_{flaps}}{X_{flaps}} =$$ Ratio of change in vertical force (Z-axis) per change in horizontal force (X-axis) for a given change in flaps deflections.

As a result of this reasoning, the contractor used decoupling crossfeeds to the trailing edge flaps to minimize speed changes due to flight path commands through the throttle levers. He then constrained the control column to command elevator deflections only. Further refinements included an auto-speed mode which controlled to the selected speed directly by using the trailing edge flaps as the primary speed correcting output. An attitude-hold mode was also used to minimize attitude coupling from flight path commands through the throttle lever.

The functional operation of this type of system is generally illustrated in Fig. 5. In essence, they provided "direct lift control" as direct control of engine thrust magnitude and as commanded through the throttle levers as

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Both of these definitions constrain the generality of (α) for the purpose of applying the specification criterion n_Z/α. Further, the equation ($\gamma = \theta-\alpha$) is, in itself, a severe constraint on the vantage point that must be attained to fully cope with the MST landing problem. (γ) is defined in MIL-F-8785 as $\sin^{-1}$ vertical speed/true airspeed. The distinction between this (γ), defined with respect to the "air mass", and a (γ) defined with respect to the ground $\tan^{-1}$ vertical speed/ground speed is too significant to ignore for the landing speeds and touchdown precision required for the MST's. Related criteria such as $\partial\gamma/\partial V$ must also be carefully reviewed for application on MST's. The use of this criteria within the context of MIL-F-83300 and MIL-F-8785 is not only purposely limited to the "air mass" reference but also requires that it must be measured with constant thrust in both direction and magnitude and with a perturbation of the airplane solely by an "elevator" or equivalent (θ) change producing device.

their main flight path control provision. They went on to conclude:

"In regard to piloting techniques for STOL terminal area flight
operations, there is clearly a preference for the STOL mode of
flight path control, i.e., power level adjustments for flight
path error corrections with relatively constant pitch attitude
maintained by a pitch-attitude-hold mode and airspeed regulated
by the autospeed function."

Mechanical Flaps Plus Vectored Thrust Version

The key to this contractor's philosophy with respect to Fig. 4 is
indicated by this statement:

"Pilots confirmed that they could use 'conventional' techniques for
controlling flight-path angle and airspeed. The 'conventional'
technique implies that flight path angle is controlled with the
column and that thrust vector angle is used for controlling air-
speed. The other control technique often used for STOL approaches
involves controlling airspeed with the control column and flight
path angle with thrust magnitude."

A control law structure that evolved from this concept is shown in Fig.
6. The use of this control during approach assumes that thrust vector trail-
ing edge flaps, thrust level and spoiler deflection have all been set to
satisfy "trim" for the desired flight path-speed profile. Flight path angle
deviations are controlled through the control column which commands elevator
and spoiler deflections about "trim" and speed deviations are controlled by
closing an automatic speedloop which varied the thrust vector angle around the
trim point, approximately 70° with the horizontal. A closed-loop decoupling
crossfeed was found to be necessary for flight path angle-to-speed changes.
The use of thrust vector angle changes to control speed reduced the speed-to-
flight path (open loop) coupling to a point where cancelling by closed-loop
decoupling was not considered necessary.

Summary of the Two Contractor Approaches

Each contractor recognizes the flight path angle-speed coupling problem.
Each contractor made an, a priori, assumption as to what input device would be
used to correct flight path deviations; in one case a throttle lever, and in
the other, a control column and then suppressed the pilot effort associated

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with speed corrections by using an automatic speed control loop. In the first case, the closure of the automatic speed loop was accomplished by deflecting the flaps about the trim position and thus vectoring the thrust indirectly with the flaps. In the second case, the efflux of the engine was vectored directly. Each arrived at a method of exerting forces for speed control, X-axis forces, that minimized the coupling of Z-axis (lift) forces.

Each used direct-lift to minimize speed changes caused by flight path change commands. In one case the "direct-lift" was in the form of thrust magnitude modulation about the trim position and in the other case symmetrical spoiler deflection about a trim position. Each was able to demonstrate within a reasonable degree of validation that the control of their respective study models, EBF and Mechanical Flaps Plus Vectored Thrust was generally satisfactory for an MST landing approach. Each contractor described his results as vindication of (1) the "STOL technique in one case and (2) the "Conventional" technique in the other. Substantiation arguments included the observation that when the "Conventional" technique was used with the "STOL" technique system designed for the EBF version, its performance was poorly rated by the simulation pilots. On the other side of the coin, it was pointed out that the "Conventional" technique was preferred for the Mechanical Flaps Plus Vectored Thrust version because:

"(1) With the thrust vector set at approximately $70°$, changes in vector angle primarily produce axial acceleration, with a small change in normal acceleration.

(2) In the nominal approach condition and with the power set at 75% of maximum, the aerodynamics and propulsive normal acceleration capability is $\Delta n_{Z \text{ aero}}$ (with DLC) = .45g and $\Delta n_{Z \text{ thrust}}$ = .1g. With a single engine failure, $\Delta n_{Z \text{ thrust}}$ = 0 if the thrust to weight ratio is maintained."

The comments quoted in support of either the "Conventional" technique or the "STOL" technique are true statements. Their relevance to supporting either "technique" and to MST longitudinal control provisioning in general, however, needs examination.

"STOL" versus "Conventional" Technique?

Many papers have been written that discuss this choice. The overwhelming

majority of these papers make the often non-stated assumption that a control column or stick command is synonomous with elevator deflection and the throttle lever is synonomous with engine thrust modulation. Under this constraint, there isn't much left to close the coupled flight path angle-speed control loops, other than "pilot-technique". Augmentation and automation techniques that are restricted to operating through only the elevators or engine thrust commands will also be of dubious help because of the inherent coupling. For airplanes already built, the pilot-technique issue has validity because it is a case of doing the best you can with the only variable left to analyze, the pilot himself. For the "powered lift" MST's however, the so-called "STOL" versus "Conventional" landing technique issue, as it is normally presented, is of extremely doubtful validity.

The argument that the system designed for the "STOL" technique would not perform well when the "Conventional" technique was applied or vice versa, is not really a supporting argument. The pilot is no longer commanding an "elevator" or "throttle" or selecting a technique to use them; he, or the AFCS, is commanding flight path corrections or speed corrections through whatever input device was assigned. Any attempt to interchange the use of these assigned devices, which now command a set of force and moment generators through a control law structure deemed most suitable to make flight path or speed corrections separately, is obviously going to be difficult. The after-the-fact pilot option has been removed, the real issue is the basis on which the cockpit control assignment is made to best serve the MST mission.

Other Control Considerations

Fig. 4, as stated previously, makes the assumption that the airplane is in "trim" during landing approach. As a part of these same studies, at least one contractor found that the transition from up-and-away flight to the configuration required for landing approach, plus capture and "trim" to the required flight path-speed profile is difficult. It should be obvious that the assignment of cockpit controls cannot be made without careful consideration of how they can best serve these transition needs. Fig. 4 makes it clear that "attitude" during the landing approach is not necessarily the dominating control parameter. It is only important to restrain "attitude" changes within certain limits. When the transition is made from landing approach (airborne

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flight) to touchdown (ground control) however, attitude must be reconciled along with a probable change in flight path from the "trimmed" condition.

The effects of engine failures and/or the need for a go-around must also be considered. The pilot must be given a control system that minimizes his workload in dealing with these emergencies. Finally, harmony with up-and-away flight where 95% of the mission time will be spent must be considered.

It is in this up-and-away flight regime where a more fundamental sense of what "Conventional" flight control really consists of can be more clearly illustrated. A recent paper (Ref. 7) states,

"The pilot must control the aircraft velocity vector in a three dimensional space. In a conventional airplane, the two vector angles (γ, ψ) usually are tracked using column and wheel inputs, and the vector magnitude (V) is controlled in essentially open loop or discontinuous fashion using throttle inputs".

The relegation of vector magnitude (Fast-Slow) control to an "essentially open-loop or discontinuous" manner is a key element of conventional flight control. For the MST's this type of control is no longer satisfactory during landing approach because of the severe coupling problem. Further, the control of speed, vector magnitude (V), cannot necessarily be limited to thrust magnitude modulation, and finally, the manual closure of this control loop by the pilot, in addition to closing his (γ, ψ) (Up-Down) (Right-Left) loops does not appear desirable from a pilot workload basis. The weight of the evidence indicates that (V) must be controlled independent of the pilot, i.e., automatically. Still another factor is worth emphasizing in the selection of the longitudinal control provisioning for the MST's.

The landing of an MST and subsequent deceleration to a stop, or a go-around are obviously "energy-control" problems. (Ref. 8) The rate of energy consumed (fuel) as it affects the total airplane energy state, potential (height) plus kinetic (speed), is directly changed by the engine thrust lever. On the other hand, control forces used to change flight path generally only transfer potential energy (height) to kinetic (speed) or visa versa.

This energy concept does not lead to some easily perceived longitudinal control provisioning concept for the MST's. It is fundamentally significant, however, and far more relevant than trying to justify the system on a

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preconceived "STOL" or "Conventional" technique basis. The merits of the longitudinal control provisioning must stand on its own feet.

Lateral-Directional Control

A vantage point for the lateral-directional control problems is illustrated by this quotation:

> "This simplicity is lost (the author is referring to longitudinal control) when we go to lateral motions, for then the rotation takes place about two axes (x) and (z). The moments associated with these rotations are cross coupled, i.e., (p) produces yawing moments C_N) as well as rolling moment C_ℓ, and yaw displacements (β) and rate (r) both produce rolling and yawing moments. Furthermore, the roll and yaw controls are also often cross-coupled, deflection of the ailerons can produce significant yawing moments, and deflection of the rudder can produce significant rolling moments."
> (Ref. 6)

In view of the previous discussion under longitudinal control, the reader is certainly entitled to question the "simplicity-comparison". The comparison is thought significant however. The need to provide "decoupling" in some degree has long been recognized for Right-Left control, while the need to do so for good (Up-Down) (Fast-Slow) is only fully appreciated when control must be provided near the minimum speed possible with "powered lift" techniques.

The treatment of lateral-directional control and its relationship to "decoupling" will therefore be less emphasized in this paper than longitudinal control, although this is not intended to suppress its importance. Landing on a 60 ft wide strip in the presence of "disturbances" is a demanding flight task.

None of the contractors involved in the studies investigated using direct side force for better Right-Left control although it has attractive possibilities. The decoupling approach, therefore, was immediately reduced from the generality presented in the longitudinal to that shown in Fig. 7, i.e., the control surfaces are conventional moment generators. The control law

development implied by Fig. 7 stems directly from the Etkins quote and the "decoupling" concept that the control wheel will command (p) without inducing (β) and the rudder pedals will command (β) without inducing (p).

Fig. 8 illustrates typical control laws that can develop from this premise. There can be no doubt that many of the symptoms of poor Right-Left control are removed by these active techniques. The initial development of undesired (β) during entry into the turn can be largely cancelled out by feedforward into the rudder, and the remainder well suppressed by feedback techniques. "Feedforward is really a very old trick to cancel out the effects of disturbances before they have altered the output" (Ref. 9). In this case, the "disturbance" is an unwanted coupling of the outputs.

The Dutch Roll modes can be damped reasonably well and perhaps at least as important, the tendency to "stir-them-up" with roll rate commands can be largely removed. The spiral mode can be made essentially neutral such that the bank angle tends to neither increase nor bleed off during the turn. The effective roll time constant can be decreased such that the small precise heading changes associated with landing on a minimal width runway can be enhanced.

The decoupling techniques used for lateral-directional control are not as sensitive to the remainder of the total flight regime as the longitudinal provisioning. The transition from landing approach to ground-roll, however, has a similar problem in that "attitude" must be reconciled and particularly so when landing in a cross wind. If the crab angle is accepted during the landing approach (zero β), then decoupling is desired to change the heading "attitude" of the airplane to that of the runway just prior to touchdown without changing flight path. The removal of yaw-to-roll coupling goes a long way towards achieving this type of flight path-to-attitude decoupling. If the forward-slip maneuver is executed, then the purposeful coupling must be "unwound" and the roll attitude of the airplane reconciled with the ground, again without materially affecting flight path.

There is one more aspect of "decoupling" that deserves mention, the coupling of lateral-directional or Right-Left control into (up-Down) (Fast-Slow) control. If heading changes are to be made without change of flight path or speed in the XZ-plane, then compensation must be provided for the loss of

lift due to bank angle. A relatively simple crossfeed of lift compensation per unit bank angle can be established if the previous "decoupling" of flight-path and speed has been accomplished.

IMPLICATIONS FOR DESIGN

The emphasis on decoupling just presented, is not an argument that complete "decoupling" must be provided for landing powered lift MST's. For many sound and substantial reasons, this is not likely to be either completely possible or desirable. This presentation is an argument, however, that the principles involved in decoupling must be thoroughly understood before the trades involved in backing-off can be justified. One of the most significant trades will be discussed briefly.

Flight Path-Speed-Attitude

The maturity of attitude sensors is well established whereas the ability to sense absolute flight path angle with respect to the ground involves finding the arctan vertical speed/ground speed. The latter quantity can be substantially different from airspeed and is not easy to obtain. A good attitude-hold loop, in itself, does a great deal to minimize the coupling between flight path and speed. On the other hand, the importance of controlling to an absolute ground referenced flight path for MST's can be appreciated by reading the article, "Effects of Wind Shear on Approach", by Captain W. W. Melvis, Delta Airlines, in the June, 1971 issue of Interceptor Magazine. The article discusses the problems of flight path and speed control in the context of 120 kt landing approach speeds. The increased concern at the MST landing approach speeds should be obvious.

Handling Qualities Criteria

There are many things that could be said about this controversial aspect of the MST's and as it relates to MIL-F-83300 and MIL-F-8785. A few of the observations considered most significant are listed here:

        1. The concept of pilot workload as it relates to the MST landing task performance and as set forth in MIL-F-83300 and MIL-F-8785 is a sound and valid measure of "goodness" for MST "flying-qualities" or perhaps more aptly titled Flight Control Performance.

        2. The interpretation of the pilot workload concept into

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mutually verifiable, necessary and sufficient, contractor-customer "flying-quality" requirements is not presently satisfied by either MIL-F-8785 or MIL-F-83300. In general, MIL-F-8785 purposely excludes applicability to powered-lift, direct-lift, direct-drag and those active techniques directly associated with satisfactory MST landing capability, while MIL-F-83300 is too strongly oriented towards STOL as a transition to or from VTOL instead of an extra-polation from CTOL.

   3. The main effect of this vacuum of applicability is to put a far higher premium on the use of piloted simulation as a "tool" for both contractor design development and customer assessment.

Piloted Simulation

It is difficult to make judgment as to which aspect of MST landing simulation was violated the most, the fidelity of the simulation required to be representative of what the pilot will actually experience, or the manner in which the simulation experiments were conducted. The representation of all forces acting on the airplane for the powered-lift MST's is, at least, an order of magnitude more complex than for a conventional airplane. Further, the quality of the vehicle dynamics data is susceptible to poor predictive techniques and the system design evaluations must recognize the need to consider variations from those assumed, even with extensive wind tunnel data.

The quality of the visual (outside world) presentation to the pilot has been troublesome. Unless the pilot is convinced that the representation is realistic, particularly in the landing touchdown transition area, the validity of the simulation data for design purposes is tenuous.

Motion can be required, particularly for investigation of engine failures, however, it is easy to overrate as a critical simulation parameter. Follow up experiments from fixed base to moving base during the MST studies revealed lateral acceleration as perhaps the most significant external force cue.

A large problem in accepting the piloting simulation data from these MST studies was the promiscuous use of Cooper Rating. The use of Cooper Rating as an after-the-fact evaluation is one thing. The use of Cooper Rating for design feedback without an active questioning of how and why the evaluations were given denies the needed use of piloted simulation as a design tool.

Flight Control System Mechanization

The "hardware" implementation is necessarily discussed last because this choice must first of all be based on ability to satisfy the control-laws found necessary and as they encompass "decoupling" with active techniques. It is short-sighted to be in a hurry to discuss safety, reliability, and maintainability until the mechanized capability to perform the job can be established. The mechanizing job shares a common facet with other parts of MST development, i.e., the need to avoid premature commitments based on past mechanizing practices. A short saga of the MST mechanizing problem unfolds in the following manner.

Pure mechanical systems cannot provide sufficient performance. Pure Fly-by-Wire systems have sufficient performance but invite risks at this time that do not seem justifiable when compared to the performance attainable with a hybrid mechanical-electrical system. The number one issue, therefore, is how to design this hybrid mechanical-electrical system in a fashion that makes the best possible integrated use of these two types of signal transmission and which recognizes in particular the overriding electromechanical interface problem.

CONCLUSIONS

1. The need for "decoupling" by active control techniques, i.e., separate non-interacting Up-Down, Right-Left, Fast-Slow control, is an essential part of MST flight control system design.

2. Cockpit controllers must be distinguished from the force and moment generators they control.

3. Piloted simulation must be used more extensively as a design tool.

4. Although the "landing approach" area is significant, the MST flight control system must fully recognize the total mission.

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INTERNALLY BLOWN FLAP (IBF)

EXTERNALLY BLOWN FLAP (EBF)

UPPER SURFACE BLOWING (USB)

AUGMENTOR WING

VECTORED THRUST (VT)
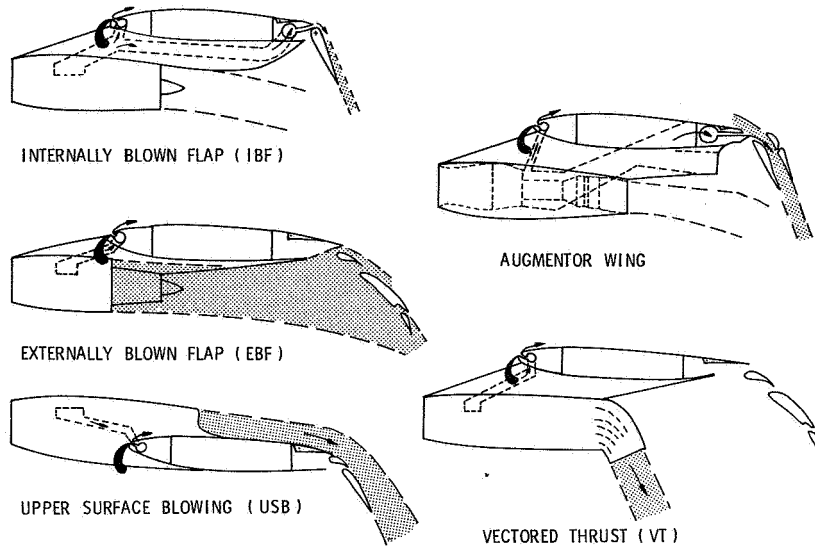
FIGURE 1.   POWERED – LIFT STOL CONCEPTS

FLAPS 95 DEG
WEIGHT 45,000 LB
BACKSIDE ←→ FRONTSIDE

GAS GENERATOR RPM
(PERCENT)
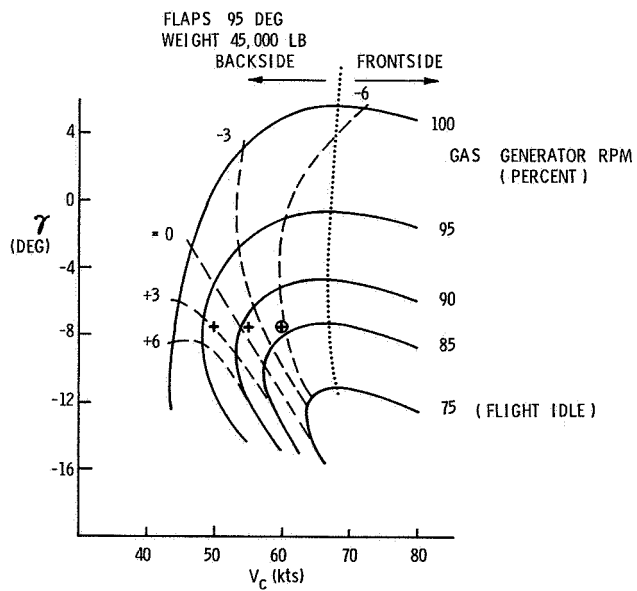
FIGURE 2. FLIGHT PATH VERSUS AIRSPEED FOR BREGUET 941

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FLIGHT TASK DESCRIPTION

AB : Approach Localizer with 45° cut at Constant Airspeed and Altitude with landing config.

BC: Capture Localizer and Glideslope and Reduce Airspeed by 10 Kts to 75 Kts.

CD: Track Glideslope and Localizer at 75 Kts.

D : Reduce Vertical Velocity to Less than 600 FPM.
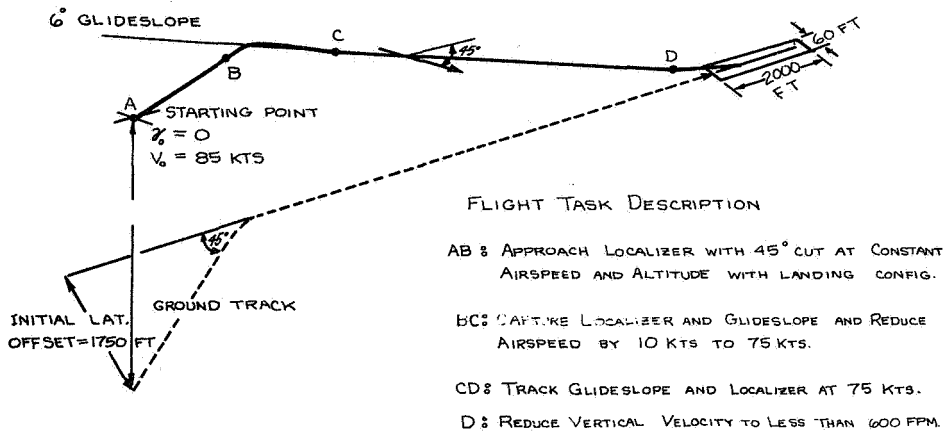
FIG. 3   TYPICAL LANDING APPROACH TASK
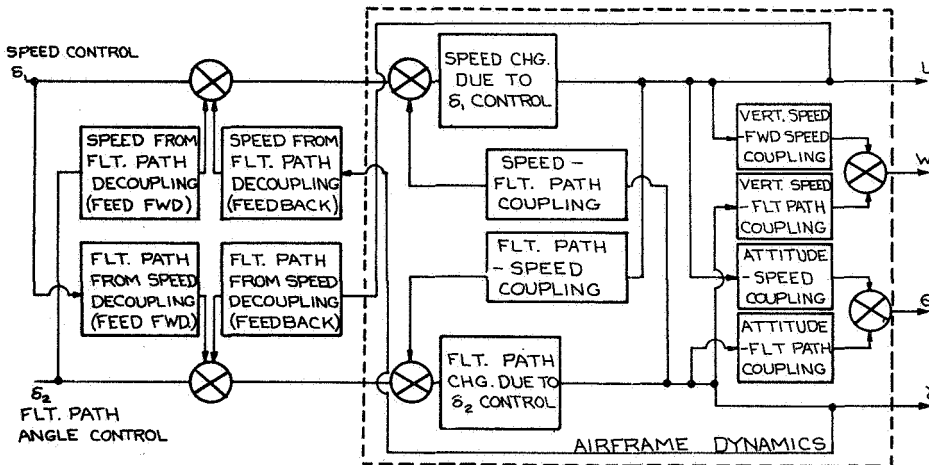
FIG. 4   DECOUPLED-LONGITUDINAL CONTROL

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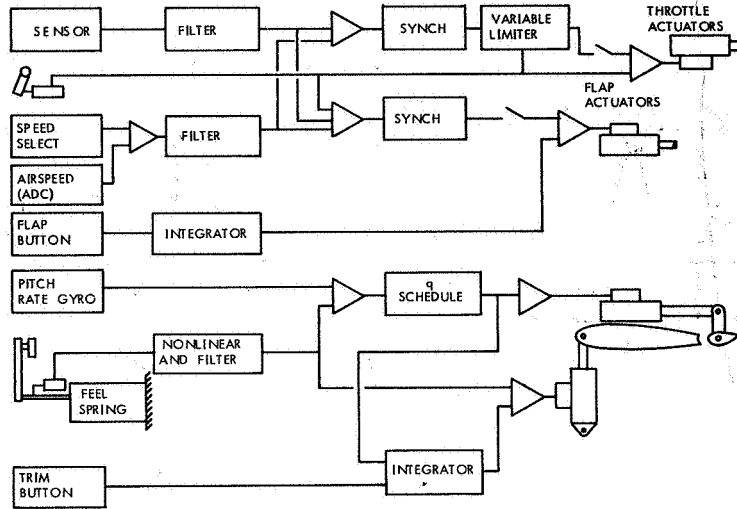
FIGURE 5 LONGITUDINAL CONTROL SYSTEM (EBF)
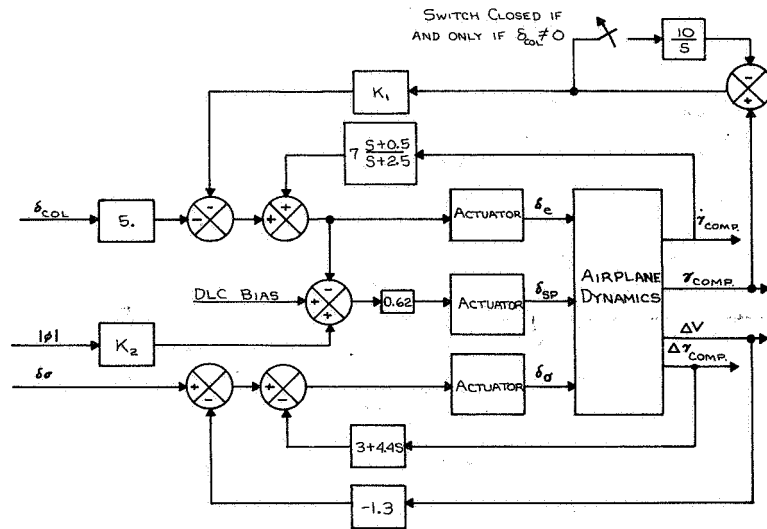
FIG. 6 LONGITUDINAL CONTROL SYSTEM (VT)

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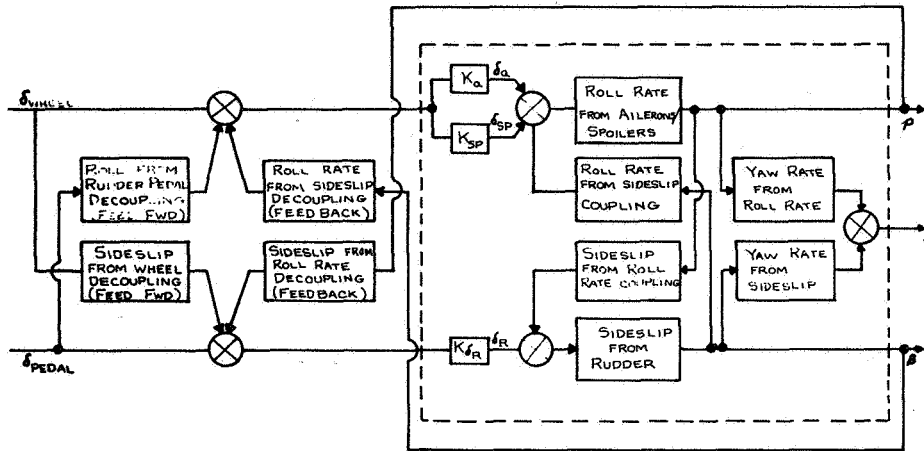
FIG. 7. DECOUPLED LATERAL-DIRECTIONAL CONTROL
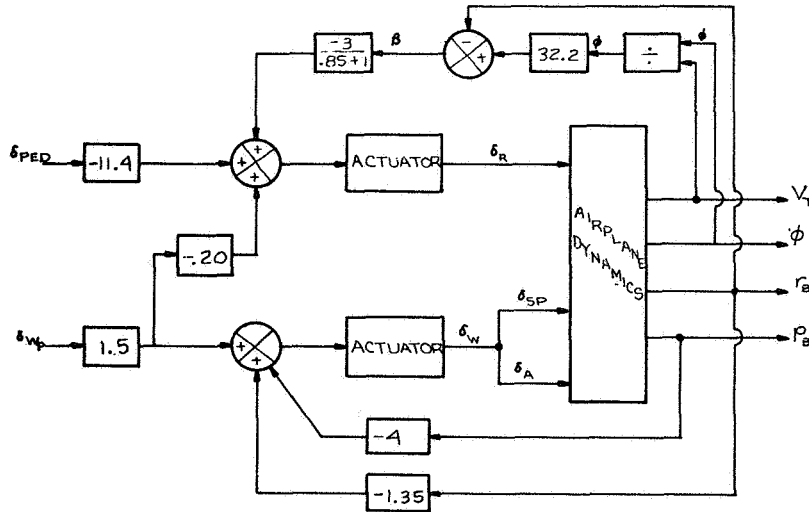
FIG. 8. LATERAL-DIRECTIONAL CONTROL SYSTEM (TYPICAL)

SESSION IV

POTENTIAL BENEFITS OF PROPULSION AND FLIGHT CONTROL

INTEGRATION FOR SUPERSONIC CRUISE VEHICLES*

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SUMMARY

Supersonic cruise aircraft can exhibit strong interactions between the propulsion system and the airframe. These interactions can be aggravated or improved by the behavior of the propulsion control system and the flight control system. When these controls are designed independently, they tend to affect the interactions adversely. When the propulsion and flight controls are integrated, however, the benefits can be synergistic.

This paper reviews typical airframe/propulsion interactions such as Mach/altitude excursions and inlet unstarts. The improvements in airplane performance and flight control that can be achieved by improving the interfaces between propulsion and flight control are estimated. A research program at the NASA Flight Research Center to determine the feasibility of integrating propulsion and flight control is described. This program includes analytical studies and YF-12 flight tests.

INTRODUCTION

Interactions between airframes and propulsion systems go back to the earliest history of powered aircraft. Along with the stories of daring aviators in open cockpits, we also heard of large rolling moments due to rotary engine torque and yawing moments induced by propeller slipstream. Interactions such as these were handled in a straightforward manner by applying large amounts of lateral stick and rudder control. The introduction of jet engines at first alleviated these interactions. However, as flight speeds increased, propulsion systems became more complex and sophisticated. A typical supersonic cruise aircraft has an inlet with variable geometry features programed by engine, inlet, and airframe variables. These propulsion system features influence the thrust, drag, performance, stability, and control of the entire vehicle. Efficient utilization of these interactive effects could greatly enhance the overall effectiveness of a supersonic cruise vehicle. To accomplish this, the engine, inlet, and flight controls must be integrated so that they work cooperatively for optimum vehicle performance.

This paper describes the principal types of interaction phenomena that have been encountered in NASA flight research (refs. 1 and 2) and proposes approaches and solutions to interaction problems. It discusses the potential benefits of integrating propulsion and flight controls into a cooperative airframe/propulsion control system and describes a research program to determine the feasibility of the system and to demonstrate it in an operational environment.

SYMBOLS

BPD   bypass door opening, percent of full open

g acceleration due to gravity, m/sec^2

I_X, I_Z moment of inertia about the X- and Z-body axes, respectively, $kg-m^2$

$$L = 57.3 \frac{\text{Rolling moment}}{I_X}, \text{deg/sec}^2$$

$$L_{unstart} = 57.3 \frac{\text{Rolling moment due to unstart}}{I_X}, \text{deg/sec}^2$$

$$N = 57.3 \frac{\text{Yawing moment}}{I_Z}, \text{deg/sec}^2$$

$$N_{unstart} = 57.3 \frac{\text{Yawing moment due to unstart}}{I_Z}, \text{deg/sec}^2$$

δ_a aileron deflection, percent of maximum deflection

δ_r rudder deflection, percent of maximum deflection

ζ Dutch roll damping

ω_d Dutch roll damped natural frequency, rad/sec

ω_n Dutch roll natural frequency, rad/sec

Subscripts:

BPD, δ_a, δ_r partial derivatives with respect to subscripted variable

max   maximum

DESCRIPTION OF INTERACTIONS

The types of interactions to be discussed are shown in figure 1, in which the flight vehicle is considered to consist of three elements: airframe, engine, and inlet. Many interactions are possible between these elements, and all possible combinations have probably occurred, at least to a minor extent. The figure illustrates three typical types of interactions that have been observed during NASA flight research: (1) the F-104 airplane interactions primarily involve the airframe and the inlet; (2) the F-111 airplane interactions are primarily between the engine and the inlet; and (3) the XB-70 and YF-12 airplane interactions, which are typical of supersonic cruise aircraft with mixed-compression inlets, primarily involve the airframe, inlet, and engine.

A prime example of an airframe/inlet interaction, shown in figure 2, was observed during the development of an F-104 airplane. Uncontrolled airplane motion began when the pilot initiated a left roll at Mach 1.87 (time = 0.8 sec), which caused the airplane to sideslip. This precipitated an engine surge at time = 2.5 seconds, which resulted in an engine mass flow reduction. A detailed analysis (ref. 3) showed that this reduction in mass flow forced the inlet shock forward on the lee side of the fuselage, creating a higher yawing moment in the opposite direction. The phase relationship to the natural frequency of the airplane was such that the vehicle's oscillations were divergent. After one-half cycle of the oscillation, the throttle was retarded to prevent an engine overtemperature which could have resulted from the surge. This power change further aggravated the yawing motion by reducing the mass flow through the inlet and causing the sideslip to exceed the 2° limit of the airplane.

The angle-of-attack excursions shown in figure 2 represent a pitching oscillation of 1.5g to 2.0g. The left and right side inlet recovery indicates the magnitude of the inlets' active participation in the motion. The interaction was eliminated on subsequent flights by extending the splitter plate between the left and right side inlets back to the compressor face, as shown in the sketch. This reduced the cross-flow between the two inlets that had caused the shock motions.

The F-111 airplane is an example of an interaction primarily between the engine and the inlet (ref. 4). A time history of a dynamic interaction on the F-111A airplane is shown in figure 3. These data were obtained during stabilized flight and constant power setting at a Mach number of 2.17. The dynamic distortion of the inlet initially oscillated within the stall limits but finally peaked above the boundary, resulting in an engine stall and an aborted flight. No significant airframe interactions induced by the engine or the inlet were noted during the NASA flight tests of the F-111 airplane in which more than 100 engine stalls were experienced throughout the flight envelope.

For maximum efficiency, supersonic cruise vehicles usually have a mixed-compression inlet, that is, an inlet in which the normal shock is in the throat rather than outside the cowl lip. This provides the highest inlet recovery and the best range for a point design aircraft. However, if the normal shock is disturbed and moves to a position forward of the throat, it can become unstable and "pop" out of

the inlet. This phenomenon is called an unstart. High pressure air from the inlet is suddenly discharged, causing massive flow disturbances over the external surfaces of the aircraft as well as inside the inlet. This results in strong interactions between the engine, inlet, and airframe. Figure 4 is a time history of a double unstart that occurred during a turn at Mach 3 with the XB-70 airplane. The unstart was believed to have been initiated by a minor disturbance in the left inlet. The right duct unstarted approximately 11 seconds after the left duct as a result of intervening airplane motions. The change in pressure under the left wing, caused by the expulsion of the normal shock forward of the inlet lip, increased the normal acceleration. The normal acceleration was further increased by the opening of the bypass doors, which acted essentially as elevons. The pilot countered this pitching motion with a longitudinal control input of approximately 3° nose-down elevon. The unstart and door movements also affected lateral control, causing the airplane to roll toward the side that had unstarted. The pilot's corrective action prevented the roll rate from becoming large, but bank angle changed noticeably. From the magnitude of the pilot's inputs to prevent the pitching and rolling motions, it was estimated that the unstart pitching and rolling moments would have produced a 2.5g steady-state acceleration and a 30-degree-per-second roll rate. Similarly, loss of thrust, increased spillage drag, and the opening of the bypass doors during the restart cycle caused a longitudinal deceleration of approximately 0.1g. Perhaps even more significant to a passenger on a supersonic transport would be the rate of onset of acceleration, which was nearly a 0.1g step function.

Additional appreciation for these interactive forces is provided by the following YF-12 data (ref. 2) which show the relative magnitudes of the accelerations produced by an unstart and the aerodynamic controls:

$$L_{unstart} = 3.3 \text{ deg/sec}^2 \qquad\qquad N_{unstart} = 6.4 \text{ deg/sec}^2$$

$$L_{\delta_a} \delta_{a_{max}} = 30.4 \text{ deg/sec}^2 \qquad\qquad N_{\delta_r} \delta_{r_{max}} = -7.3 \text{ deg/sec}^2$$

The effectiveness of the bypass doors in producing yawing and rolling accelerations during normal inlet operation at Mach 3 is shown by the following derivative equations (ref. 2):

$$L_{BPD} = 0.35 \frac{\text{deg/sec}^2}{\text{percent BPD}_{max}} \qquad\qquad N_{BPD} = 0.11 \frac{\text{deg/sec}^2}{\text{percent BPD}_{max}}$$

$$L_{\delta_a} = 0.295 \frac{\text{deg/sec}^2}{\text{percent } \delta_{a_{max}}} \qquad\qquad N_{\delta_r} = 0.073 \frac{\text{deg/sec}^2}{\text{percent } \delta_{r_{max}}}$$

The propulsion system is as effective as the aerodynamic control surfaces in producing angular accelerations. Also, the significant rolling accelerations produced by the bypass door operation indicate that the moments are not produced only by thrust changes, because the YF-12 airplane has no thrust moment arm about the roll axis.

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It is important to recognize that the interaction problem is not just one of stability and control. Interactions can also seriously affect the drag and range performance of an airplane. Figure 5 shows the effect of asymmetric bypass door opening at Mach 3 on the YF-12 airplane drag increment expressed as a percentage of the basic airplane drag, the vertical fin deflection from the trimmed condition, and the mass flow out of the bypass doors. Fully opened bypass doors cause a 25-percent increase in drag (per engine) and require 15 percent of the rudder authority to maintain zero sideslip. At smaller door openings, a 10-percent change in mass flow out of a single bypass door causes a 2.5-percent increase in drag. As the bypass doors open beyond 40 percent, the mass flow out of the doors levels off because of a flow choking effect. The similarity of the drag and rudder deflection curves to the airflow curve indicates that bypass airflow is the primary cause of the interactions.

The coupling discussed has been primarily the result of direct or open-loop interactions. A modern aircraft, however, has numerous artificial sensing and feedback loops to implement a variety of control tasks. Consequently, closed-loop interaction paths can be formed that magnify the open-loop effects or create new coupling effects. An example is shown in figure 6. The YF-12 inlet computer modulates the bypass door movement as a function of sideslip (among other parameters) to minimize unstarts. Because of the influence of the fuselage, the flow at each inlet is not the same at a given sideslip angle. Consequently, the bypass doors are modulated asymmetrically, which produces yawing moments. As the block diagram indicates, these yawing moments cause the aircraft to sideslip. The sideslip is sensed by the inlet computer, which commands bypass door changes that produce further yawing moments. Thus a closed-loop path is formed that couples the propulsion system and the airframe. Because of lags in the inlet computer sensing system, this coupling is unstable (ref. 5), and when the stability augmentation system (SAS) is turned off while the inlets are operating automatically, an unstable Dutch roll motion results. As illustrated in figure 7, when the inlets are fixed, the Dutch roll motion damps out, but when the inlets are operating automatically, the Dutch roll motion diverges.

Another example of closed-loop airframe/propulsion coupling is inlet control as a function of Mach number. As Mach number increases, the YF-12 inlet computer closes the bypass doors, decreasing drag and increasing thrust; however, this changes the variation of excess thrust with Mach number. The long-period longitudinal motion, or phugoid, is sensitive to variations of excess thrust with Mach number. Increases in excess thrust with Mach number reduce phugoid damping, as illustrated in figure 8, which shows the controls-free altitude response of the YF-12 airplane to drag disturbances with the inlets fixed and the inlets operating automatically. The decreased damping of the motion with the inlets operating automatically, in response to Mach number, is apparent. The large overshoot and oscillations make flightpath control difficult.

PREDICTION

As the previous discussion indicates, the nature and magnitude of airframe propulsion interactions were learned from flight tests; they were not predicted. To

achieve a basic solution to these problems, however, we must be able to predict the interaction effects so that they can be considered from the beginning of the vehicle design. As part of the YF-12 research program, wind tunnel tests were made to determine how detailed the model inlet geometry and airflow would have to be to provide data from which the interaction phenomena could be predicted adequately.

Our first effort in evaluating prediction techniques was to qualitatively assess the similarities of the local flow in the wind tunnel and in flight. In the wind tunnel, oil was placed on a 1/12-scale model of the YF-12 airplane which had been modified to simulate the bleed and bypass exits. The exits were slotted so that the flow was expelled at a 15° angle relative to the nacelle surface, and the bypass exits were fitted with screens to meter the flow. The mass flow out of the bleed and bypass exits was varied by changing the position of a butterfly valve in the inlet. The results of the oil flow tests are shown in figure 9, which indicates large areas of separated flow forward of the bleed and bypass exits on the nacelle and extending to the wing. Because the bleed and bypass exit simulation was not exact, it was questioned whether this represented the flow on the airplane. The exit louvers and the surrounding area of the nacelle and wing on the flight vehicle were tufted, and cameras for photographing the tufts were installed in the fuselage. Bleed and bypass mass flow ratios similar to those used in the wind tunnel were then evaluated in flight.

Figure 10 is a sketch of the flow field shown by the tuft pictures. The separated regions indicated by the wind tunnel oil flows are verified by the reversed flow forward of the bypass exits and the vertical standing tufts at the forward edge of the separated regions and on the bleed exit louvers. Thus it is expected that when all the wind tunnel data have been analyzed, the results will agree reasonably well with the flight-test data even though the exit simulation was not precise. Force and moment tests were also made on a 1/12-scale model with simulated inlet airflow. The results of these tests indicate that the forces and moments due to the propulsion system can be adequately predicted if the propulsion system is represented in sufficient detail.

Although it appears that wind tunnel data can adequately predict full-scale flight results, a general theoretical approach for predicting these aerodynamic effects is lacking. Nevertheless, by using wind tunnel tests and analytic techniques, mathematical models can be formulated for simulating and analyzing airframe/propulsion system coupling problems. Care must be taken to include all the elements that contribute to the interactive effects.

POTENTIAL BENEFITS

By using adequate simulation or analytical models, or both, that represent the entire system in the frequency range of interest, design trade-off studies can determine the advantages of integrated or cooperative controls. Many aspects must be considered in such a trade-off. For example:

(1) Should the vehicle be designed to eliminate interactions? What would be the penalty?

(2) Can the interactions be made favorable?

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(3) Is it more efficient to control the interactions with systems than to redesign the vehicle configuration?

Although these considerations are only a few of the many that must be taken into account, they are typical and will be discussed briefly to provide some insight into the problems.

Should interactions be designed out of the vehicle? One way to reduce interactions is to bypass air entirely within the nacelle. However, this requires a larger nacelle diameter which, for the YF-12 airplane, would increase the nacelle drag by approximately 25 percent. Therefore it appears that it would be better to control the interactions with cooperative engine/inlet flight controls. This might mean increased demands on systems in terms of reliability and complexity; however, the penalties in range, payload, and performance would be much less than those resulting from increasing the size of the nacelle.

Ideally, the interactions would be arranged to be complementary. This could perhaps be done by careful placement of bypass exits or by means of the control laws in a system approach. As previously discussed, a time lag in the sideslip sensor for the inlet computer resulted in a decrease in Dutch roll damping; however, the basic interaction was favorable, in that it increased Dutch roll static stability, that is, increased frequency.

Figure 11 shows the variation of Dutch roll frequency and damping as a function of sideslip sensor lag and inlet-induced yawing moment for a YF-12 type of configuration. It can be seen that Dutch roll stability can be improved by increased lead in sensing sideslip and increased yaw due to bypass door deflection. This illustrates that the potential exists for using airframe/propulsion control integration to augment the stability of the airplane, reduce the need for more redundant and complex systems, and even reduce the size of the aerodynamic stabilizing surfaces. The increased frequency and damping would make the airplane more resistant to sideslip excursions and allow the inlets to be designed with lower sideslip margins and thus higher efficiency. Also, performance degradation due to turbulence might be reduced, since increased airframe frequency and damping would minimize gust response. These benefits could be gained without increasing the tail size or the control system complexity.

The critical design factor that determines the size of the vertical tail on a supersonic cruise vehicle is usually control of the aircraft in response to the moments induced during an inlet unstart at maximum Mach number. An integrated control system that would reduce unstart transients through propulsion control as well as aerodynamic control could result in significant reductions in tail size and commensurate weight and drag savings. Automatic spike, bypass, and throttle activity on the other nacelles and fast unstart recovery could greatly reduce the yawing and rolling moments and longitudinal decelerations associated with an unstart.

Difficulties are often experienced with conventional autopilots in the Mach hold mode when an atmospheric temperature disturbance is encountered (ref. 6). The temperature change induces an immediate Mach number change, and the autopilot commands large normal acceleration or altitude changes, or both, in an attempt to

hold Mach number. Recent studies have shown that simple cooperation between the propulsion and flight controls through an autothrottle provides much smoother and more accurate response. This is illustrated in figure 12 which shows the altitude, Mach, and dynamic pressure excursions induced by a Mach hold autopilot with and without an autothrottle in response to a 4° C atmospheric temperature change. Shown is the response of a conventional Mach hold system in which pitch angle and Mach number are fed back to the elevons, and the response of a system with an auto-throttle in which pitch angle is fed to the elevons and Mach number is fed to the throttles. The significant reduction in the altitude excursions with the autothrottle system is evident, whereas Mach control is essentially equivalent. The autothrottle system shows the potential for a 0.60-kilometer reduction in altitude separation for air traffic control purposes.

The altitude excursions in figure 12 are accompanied by overshoots in dynamic pressure. A supersonic airplane usually cruises most efficiently at the highest dynamic pressure. The maximum dynamic pressure allowable for normal operation is based on the dynamic pressure limit of the airplane (for structural reasons) plus a suitable margin to allow for unintentional overshoots. The figure shows that the autothrottle reduces the dynamic pressure overshoot by 3200 N/m^2. This implies that the airplane could be operated safely at a correspondingly higher dynamic pressure, which amounts to approximately a 1-percent increase in cruise range.

Performance gains that may be realized by using a cooperative control system in a vehicle similar to the YF-12 airplane are summarized in the following table:

	Payload gain, percent of airplane gross weight
Margin reduction —	
Inlet stability	1.8
Engine temperature	2.0
Altitude control	1.0
Drag reduction —	
Propulsion system	1.25
Trim	0.70
Structural weight reduction —	
Ventral fin	0.40

If the inlet could be operated with minimum unstart margins (that is, with the shock at the throat rather than downstream), as much as a 5-percent increase in thrust could be realized. This translates into a 1.8-percent improvement in payload in terms of airplane gross weight. Similarly, improved sensing and control of the turbine inlet temperature rather than the low response turbine discharge temperature could produce more than a 5-percent increase in thrust or 2.0 percent in payload. Studies have indicated that the elimination of ±600-meter altitude excursions would allow approximately 1.0 percent increase in payload.

Drag reductions could be realized by better matching of the inlet and engine flows through use of engine speed control to vary the airflow at off-design operating conditions of atmospheric temperature and aircraft speed. Reduced unstart transients and improved flight control could make possible reduced aircraft stability

margins, with a resultant payload benefit of approximately 0.70 percent for trim drag reduction and 0.40 percent for decreased vertical fin weight. Although the individual gains listed may not be directly additive, they represent approximately 7 percent of the gross weight of a typical supersonic cruise airplane. If cooperative control concepts were incorporated into the original design of an airplane, the benefits could be even greater because of the synergistic savings in structural weight which have not been considered in this analysis.

DESIGN APPROACH

The magnitude of the problem of integrating the autopilot, stability augmentation, inlet, and engine can be illustrated by the matrix of control options shown in figure 13. State variables of the airplane, inlet, and engine can be fed back to each control. Typical state variables include:

Airplane – angular and linear velocities and accelerations, Mach number, altitude, angle of attack, angle of sideslip

Inlet – shock position, recovery, distortion

Engine – rpm, compressor face pressure and temperature, turbine discharge pressure and temperature

Typical controls include:

Autopilot – elevons, rudders, servo positions

Inlet – bypass door and spike position

Engine – power lever angle, exhaust nozzle position, fuel metering valve

A fully integrated control system would include at least one state variable feedback to each control, as indicated by an X in each square of figure 13(a). In contrast, figure 13(b) represents a system with no integration; that is, there is no communication or cooperation between the airplane, inlet, and engine controls. Between these extremes, varying degrees of integration are possible, as illustrated in figures 13(c) and 13(d). Figure 13(c) is representative of the existing YF-12 airplane, in that some airplane states such as angle of attack, angle of sideslip, and Mach number are used to control the inlet. Figure 13(d) could represent a YF-12 airplane with an autothrottle that used Mach number to control the power lever angle. Just how far to go in the integration process will depend on many practical as well as theoretical considerations.

Integrating all these diverse and complex factors is a formidable task. Classical approaches based on experience and engineering judgment have been used. If there is a high degree of interdisciplinary coordination, classical feedback techniques may be adequate. The most promising approach, however, may be based on optimal control techniques. This approach generally involves feeding back all state variables

and computing the control system gains required to minimize an appropriate performance penalty function.

When both classical and optimal control approaches have been applied to the same problem, the results have usually been the same. It should be kept in mind, however, that the classical techniques depend on analysts and designers with many years of applicable experience. When dealing with new phenomena involving complex interdisciplinary effects such as airframe/propulsion coupling, it may be difficult or impossible to find people with adequate backgrounds and practical experience to handle a classical approach. Conversely, the optimal control technique provides a systematic approach that can be used when there is little insight into the problem.

ONGOING RESEARCH

To explore and validate the benefits that could result from a cooperative control system, analytical and flight research is underway at the NASA Lewis and Flight Research Centers. The objectives of this effort are to determine the feasibility and advantages of a cooperative autopilot/SAS/propulsion control system and to verify and demonstrate the benefits of such a system in an operational environment.

The results of the basic YF-12 flight research program are being used in the cooperative control program. The pertinent elements of the basic program include investigations of the effect of airframe/propulsion system interactions on flightpath control, measurement of high-speed propulsion system performance, and comparisons of flight test, wind tunnel, and simulator results. Specifically, wind tunnel tests to determine steady-state and dynamic characteristics and to evaluate new inlet control concepts have been made at Lewis Research Center on a full-scale YF-12 inlet. Wind tunnel testing of a 1/3-scale inlet has been conducted by Lockheed Advanced Development Projects at NASA Ames Research Center to investigate scale effects. Tests have also been made at Ames on a 1/12-scale model to measure forces and moments induced by inlet airflow. Several studies have been conducted by Honeywell Inc. and Pratt & Whitney to update existing control systems and explore new control concepts.

The cooperative control program itself consists of two phases. The first phase is concerned with longitudinal flightpath control, that is, altitude and Mach excursions. The influence of atmospheric disturbances such as temperature and pressure changes and airframe propulsion interactions on longitudinal flightpath control is being studied. Control laws for autopilots and stability augmentation systems that are less sensitive to atmospheric changes are being explored. Both classical and optimal control techniques are being used to define the control laws. A first step toward airframe/propulsion control integration will be taken by implementing an autothrottle. Figure 14 shows the schedule for the cooperative control program. The analytical work in Phase I was completed in January, and an autothrottle is being fabricated. The first flight is planned for early 1975.

Phase II will consider lateral-directional interactions such as reduced Dutch roll damping and unstarts. Advanced propulsion and control integration concepts such

as optimum cruise control and unstart control utilizing a digital computer will be investigated.

The analytical portion of Phase II began recently. Flight tests of the more promising concepts are expected to begin in late 1975.

A conceptual diagram of the cooperative control system is shown in figure 15. The digital computer is used to compute, coordinate, and command the functions of the inlet, engine, and airframe in response to inputs such as those shown.

CONCLUDING REMARKS

Airframe/propulsion system interactions have been shown to significantly affect aircraft performance, stability, and control. Changes in drag as large as 25 percent (per engine) of the total drag can be involved. Forces and moments as powerful as those produced by the aerodynamic controls have been observed. If not accounted for, these effects can lead to large performance degradations, large flightpath excursions, and increased pilot workload.

Cooperative or integrated operation of the propulsion and flight controls may provide a solution to these problems. Control integration has the potential to not only eliminate the adverse effects of interactions but to significantly improve performance through synergistic effects such as less airframe weight, improved flightpath control, less overall system complexity, and more efficient operating limits. Analytical and flight research programs are underway at the NASA Flight Research Center to investigate the benefits of such a system in an operational environment.

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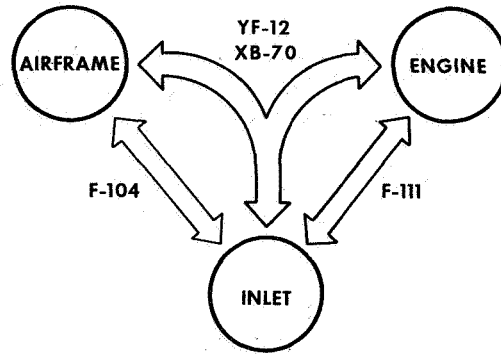
Figure 1. Functional nature of interactions.
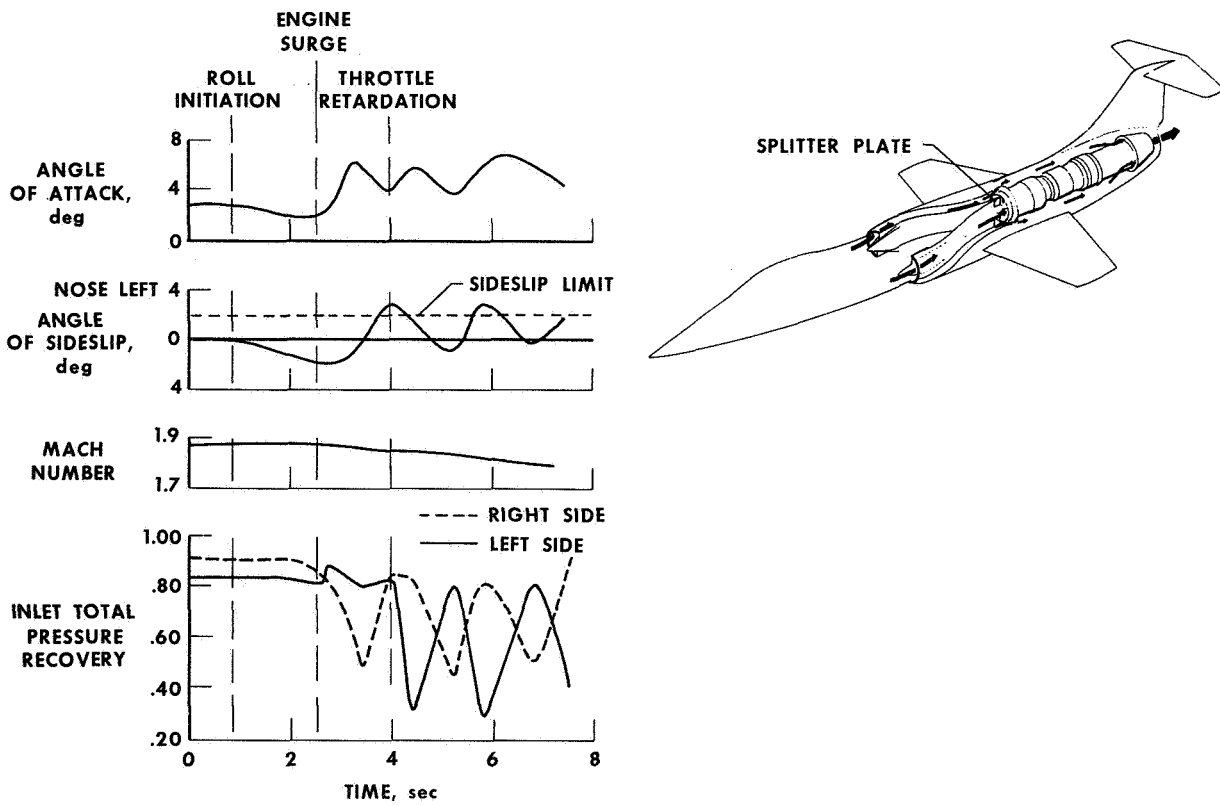
Figure 2. F-104 airframe/inlet interaction. Yaw damper off.
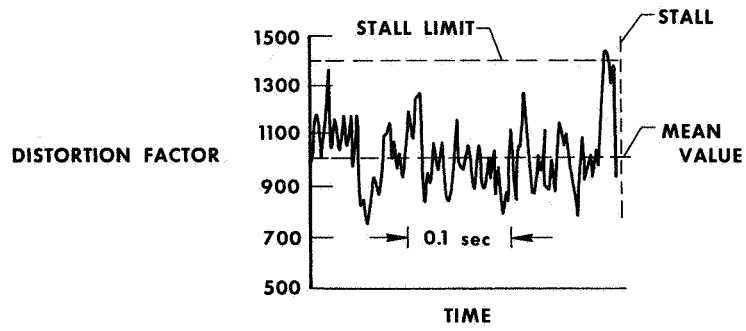
Figure 3. F-111 engine/inlet interaction. Mach 2.17.
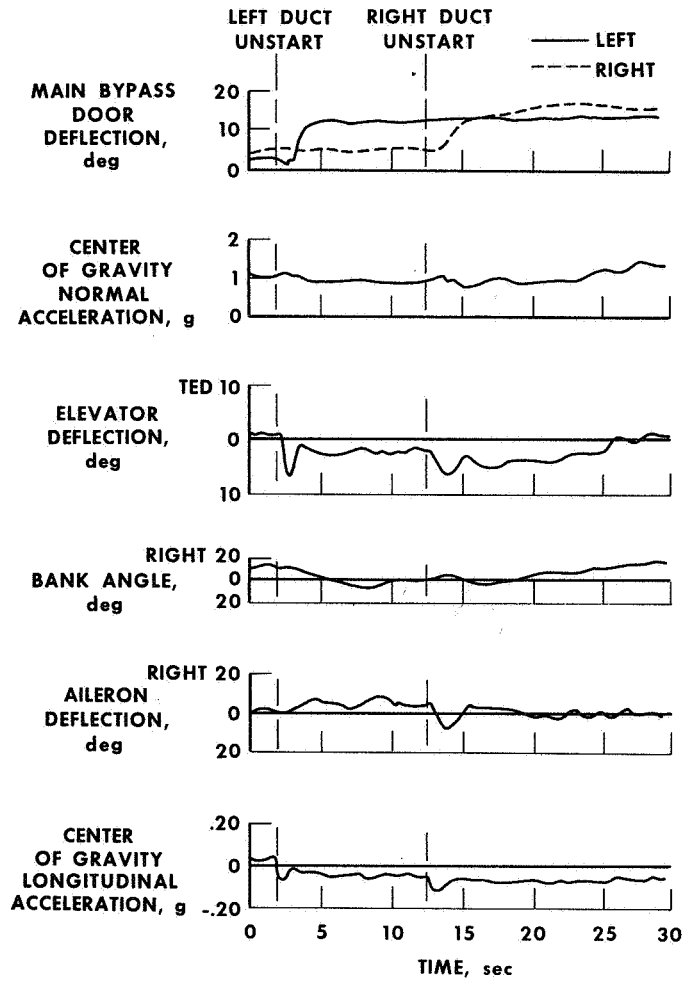
Figure 4. XB-70 engine/inlet/airframe interaction. Mach 3.

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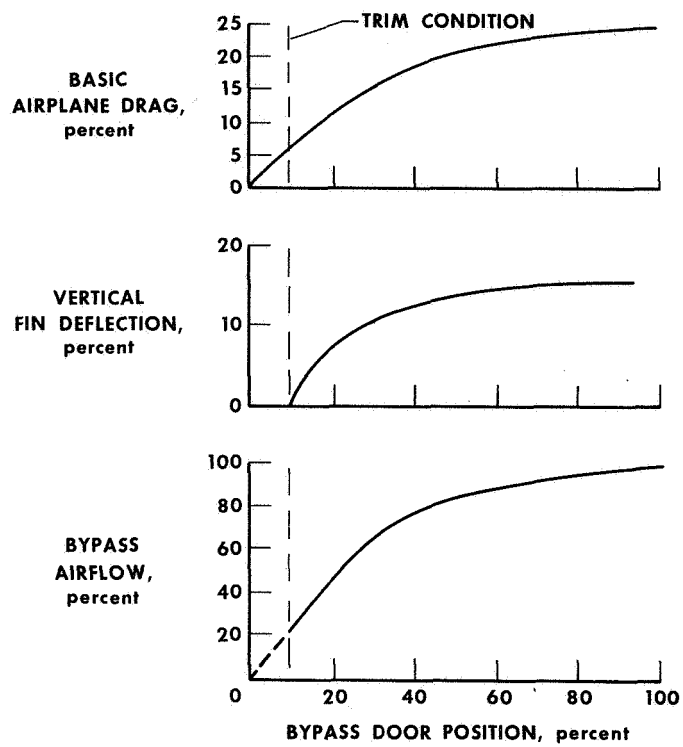
Figure 5. Effect of asymmetric bypass door opening on drag and directional control at Mach 3.
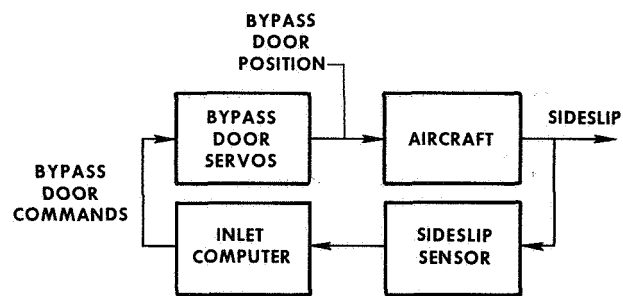
Figure 6. Sideslip coupling due to automatic inlet operation.
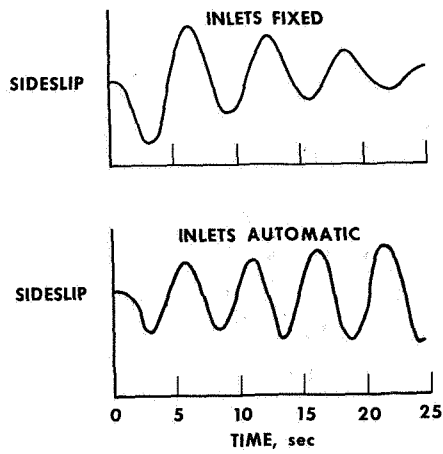
Figure 7. SAS-off rudder pulse response.
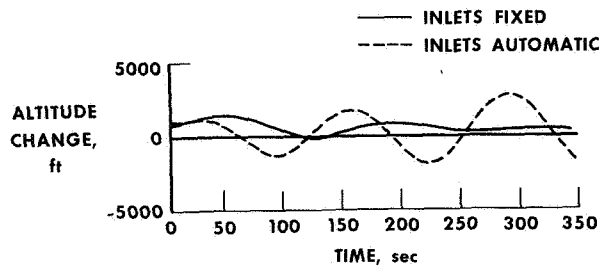
Figure 8. YF-12 controls-free altitude response to a drag pulse.
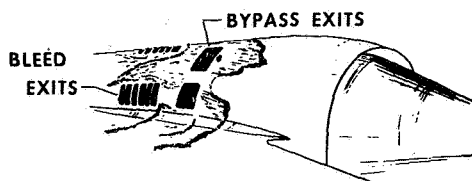
Figure 9. Wind tunnel surface oil flow study. Supersonic cruise Mach number; forward bypass and bleed open.
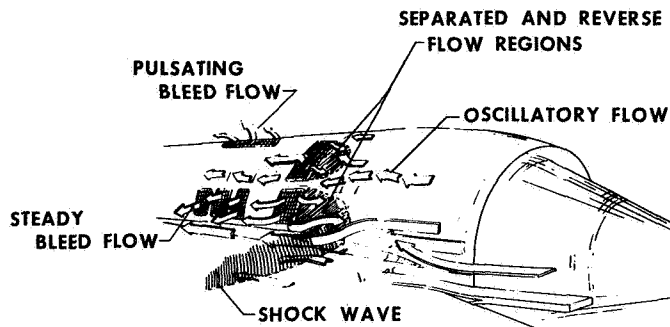
Figure 10. Flight tuft study. Supersonic cruise Mach number; forward bypass doors and bleed open.
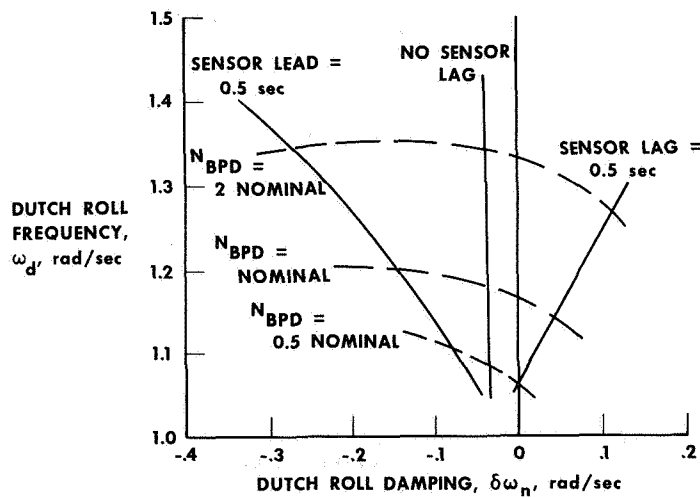
Figure 11. Effect of sideslip sensor lag and bypass door yawing moment on Dutch roll frequency and damping. Automatic inlet operation.
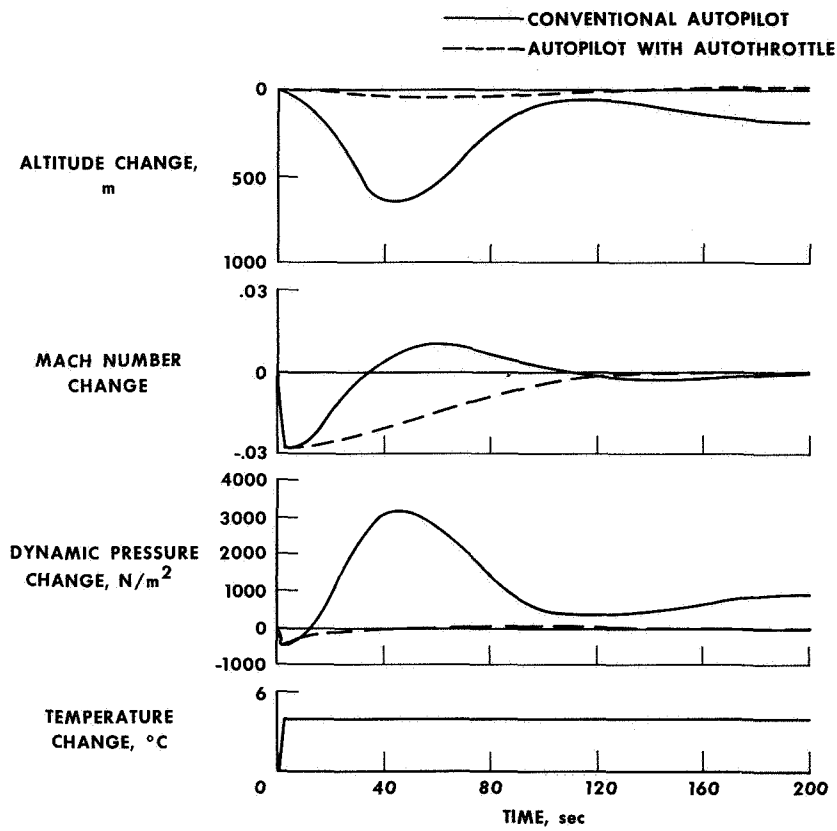
Figure 12. Mach hold autopilot response. YF-12 simulator; Mach 3.

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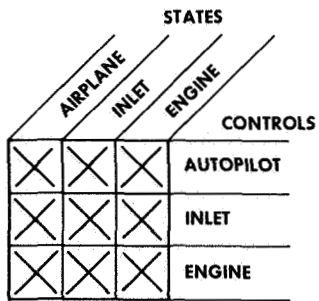
(a) Fully integrated.
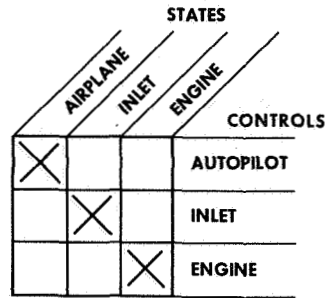
(b) No integration.
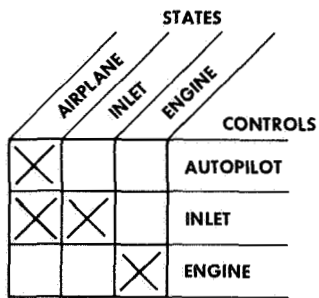
(c) Partial integration
(airplane-inlet).
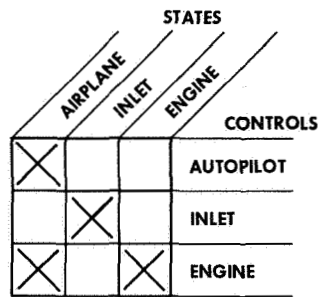
(d) Partial integration
(autothrottle).

Figure 13. Options for control integration.

	1973	1974	1975	1976	1977

CURRENT YF-12 WORK RELATED TO
COOPERATIVE CONTROL SYSTEM

PHASE I — LONGITUDINAL FLIGHT-
PATH CONTROL
   ANALYSIS
   HARDWARE DEVELOPMENT
   FLIGHT TEST

PHASE II — LATERAL-DIRECTIONAL
INTERACTION
   ANALYSIS
   HARDWARE DEVELOPMENT
   FLIGHT TEST
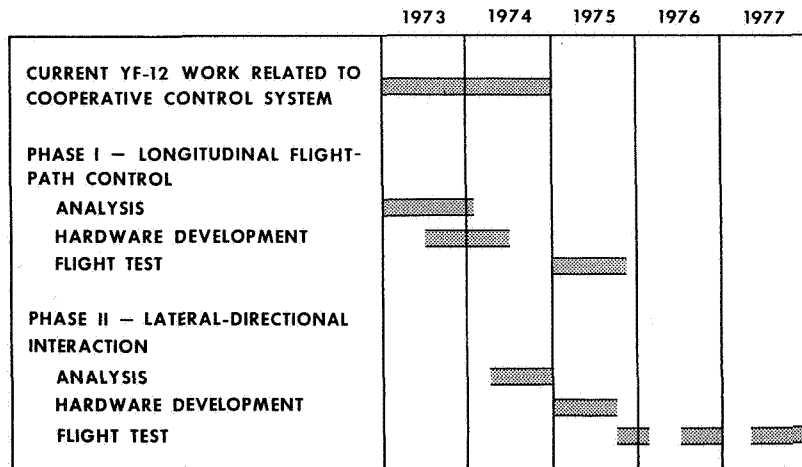
Figure 14. YF-12 cooperative control system schedule.
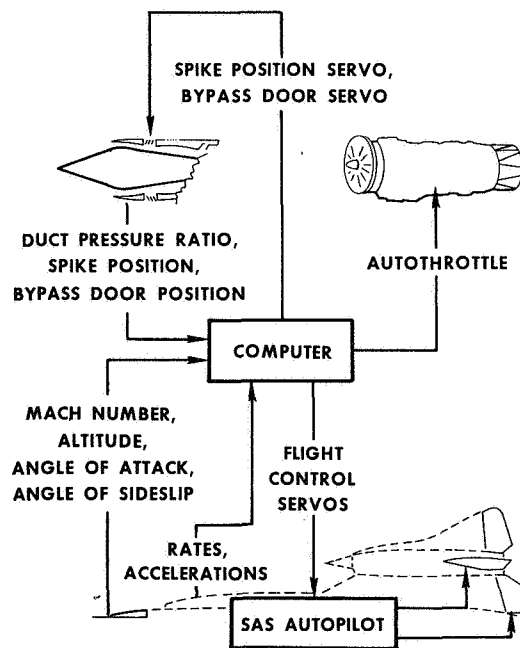
Figure 15. Cooperative autopilot/SAS/propulsion control system.

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IPCS IMPLICATIONS FOR FUTURE SUPERSONIC TRANSPORT AIRCRAFT

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SUMMARY

The Integrated Propulsion Control System (IPCS) will demonstrate control of an entire supersonic propulsion module - inlet, engine afterburner, and nozzle - with an HDC 601 digital computer. The program encompasses the design, build, qualification, and flight testing of control modes, software, and hardware. The flight test vehicle will be an F-111E airplane owned by the government. The L.H. inlet and engine will be operated under control of a digital computer mounted in the weapons bay. A general description and the current status of the IPCS program are given.

INTRODUCTION

The historical trend of controls development has been toward greater functional integration to maximize aircraft mission capability. This trend will undoubtedly continue as analytical techniques are refined and flight-worthy hardware becomes more readily available. The eventual result may be the integration of propulsion and flight control subsystems as diagrammed in figure 1. Until then, integrated control of propulsion system components must stand on its own merits. SST experience convinced Boeing that the classical approach to propulsion control is inadequate, expensive, and even hazardous when applied to high performance aircraft. New engineering techniques must be developed to obtain the required control coordination. New management techniques must be devised to permit simultaneous development by various manufacturers of subsystems that will share and use in an optimum fashion the information available to the total system.

We are confident that the Integrated Propulsion Control System is technically and economically reasonable. The IPCS program will demonstrate this feasibility in flight tests and lay the groundwork for its incorporation into future aircraft.

A discussion of some key aspects of the IPCS Program is given in this paper. Since many forms of technology are represented in the IPCS activity, a complete description would be very lengthy. Discussion of some activities has been deliberately omitted in this paper so that more space and time could be devoted to those features that may be relevant to future supersonic transport aircraft. This is consistent with the goals of the National Aeronautics and Space Administration in conducting the Symposium.

OVERVIEW

The Integrated Propulsion Control System (IPCS) Program encompasses the design, build, flight qualification, and flight testing of propulsion control modes, software, and hardware. The flight test vehicle will be an F-111E airplane owned by the government. The L-H inlet and TF30-P-9 engine will be modified to operate under control of an HDC-601 digital computer mounted in the aircraft weapons bay. The layout of the IPCS on the aircraft is shown in Figure 2.

The IPCS is one of the Exploratory Research Programs funded by the Air Force Aero Propulsion Laboratory*. Technical support is being provided by NASA; the Flight Research Center (FRC) and the Lewis Research Center (LeRC). Major contractors are Boeing Aerospace Company, Honeywell, Inc., G&AP Division, and Pratt and Whitney Division of United Aircraft (P&WA^{TM}). A diagram showing organizational responsibilities is given on figure 3.

The goals of the Air Force in funding the IPCS program are twofold:

1. Improve aircraft systems performance through technological advances.

2. Reduce the cost and risk of future development programs through an expanded technical data base and demonstrated management methodology.

Specific goals established for the IPCS program pursue the goals of the Air Force Exploratory Development Programs. The first of these is to develop, demonstrate, and evaluate in a flight environment, certain advanced technical features that have to date been explored only under very restricted conditions. These are listed in Table 1.

The second major goal is the development of an intercompany management approach applicable to the design and development of integrated systems. The IPCS management methodology addresses three areas of potential concern;

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*   Air Force Aero Propulsion Laboratory
     Air Force Systems Command
     United States Air Force
     Wright-Patterson AFB, Ohio

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TABLE 1

IPCS ADVANCED TECHNICAL FEATURES

o   Full authority digital propulsion control with hydromechanical backup.
    This will permit control law changes without hardware modification.

o   Closed loop control on turbine-inlet gas temperature (TIGT).

o   Use of compressor discharge Mach number for surge protection during
    engine transients.

o   Automatic detection and suppression of inlet buzz so that engine air-
    flow may be reduced during airplane deceleration.

o   Continuous monitoring of distortion to extend the operating envelope
    with the compressor surge bleeds closed.

o   Fuel manifold prefill logic to smooth afterburner transients.

TABLE 2

SALIENT FEATURES OF IPCS INTERCOMPANY MANAGEMENT APPROACH

o   Horizontal division of responsibility – each organization exercises its
    own area of expertise over the entire range of the program.

o   Direct communication at the working level is stressed.

o   Regular (monthly) coordination meetings are attended by representatives
    of the prime and major subcontractors.

o   Periodic working sessions are conducted with attendance by technical
    personnel of each of the three firms. These meeting sites are rotated.

o   Progressive step-by-step hardware test sequence.

o   Final decisions impacting program costs or schedule are made by the
    prime contractor.

division of responsibilities, communication and coordination between geograph-
ically remote organizations, and minimization of technical risk and cost
through a timely test sequence. The salient features of the IPCS management
approach are listed in Table 2.

Achievement of these program goals will identify potential development
problem areas. It will generate a body of technical data upon which to base
further development work and will provide a basis for estimating the time and
cost of development of an operational IPCS.

IPCS DEVELOPMENT SEQUENCE

Major IPCS activities are shown in figure 4. Contract date was 1 March
1973. The Air Force has determined that a 36 month program is compatible
with the scope of the program; hence flight test completion is scheduled for
29 February 1976. (An additional four months are allowed for data reduction
and preparation of the final report.) The IPCS schedule was developed to fit
these constraints.

It will be noted that about half of the total program period is devoted
to an extensive test program. This required careful scheduling of the analy-
sis, design, and fabrication of hardware and software to meet the test dates.
This requirement influenced the design procedure to a great extent, as will
be discussed later in this paper.

DATA MANAGEMENT

There are four classes of data involved in a program such as IPCS:

> Design data
> Hardware and software checkout data
> Data for test planning and test monitoring
> Test evaluation data (results)

Activity was initiated immediately after contract to compile all avail-
able data on the characteristics of the P&WA TF30-P-9 engine and the F-111E
inlet. In addition to published Air Force and NASA data, a substantial amount
of unpublished information was obtained under subcontract from P&WA and
General Dynamics/Convair Aerospace Division. These data were incorporated
into a document that will be updated at 6-month intervals as necessary through-
out the program. Much of the design work was based on the data compiled under
this task.

The data compilation discussed above is being supplemented by baseline
tests of the IPCS engines and aircraft. These tests also serve as development
vehicles for the data acquisition/reduction hardware, software, and procedures
to be used during the IPCS flight evaluation program. The baseline engine

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tests were conducted by NASA/LeRC in their altitude facility. This test series was completed in February, 1974. The baseline flight tests, to be conducted by NASA/FRC are scheduled to begin in July, 1974. The baseline test program is described later under the test program heading. The handling of the data is described below. It is anticipated that similar procedures will be used in subsequent system-level tests.

Instrumentation

The intent in selecting instrumentation for the IPCS program was to measure engine and inlet operating parameters with minimum disturbance to the gas flow. Thus, it was decided that the only rakes to be added to the flow path would be to measure compressor face distortion and new control signals. The remaining instrumentation is either production sensors or measurements which can be made at the wall or in control system lines, etc. To the extent possible the same or similar instrumentation will be used throughout the test program to facilitiate data comparisons from one test to another. Table 3 lists the instrumentation for the baseline and IPCS tests.

For the IPCS control mode, total pressure and temperature measurements are required at the exits of the high and low pressure compressors and total temperature is needed at the turbine inlet. Probe designs and the required engine case modifications for these probes were not available prior to the start of the baseline engine test. Total pressure and temperature measurements were made at the low pressure compressor exit using probes similar to the IPCS design that could be inserted through an existing hole in the engine case. These compressor exit measurements will not be made during the baseline flight test since the IPCS engines will not be used.

An unavoidable difference in instrumentation systems exists between LeRC and FRC. In the flight tests there will be no steady-state instrumentation equivalent to the DAMPR system at LeRC during the IPCS flight test, therefore data from the Digital Propulsion Control Unit (DPCU) will be used. Where possible the equivalent test instrumentation will be eliminated to avoid duplication in sensors and data processing. To a degree the same approach can be used during the IPCS altitude test, however, during this test it will be important to retain sufficient instrumentation to demonstrate the validity of the control sensors.

Recording

Both digital and analog recording systems will be used - digital for low frequency response data (DC-50HZ) and analog for high frequency response data (∿500 HZ). The NASA/LeRC digital system consists of a steady-state system for performance measurement, and a 200 channel low-to-medium frequency system used for recording transients. NASA/FRC uses a PCM digital system for both steady-state and low frequency transient data. At both facilities the high frequency data are recorded on FM analog systems.

TABLE 3

INSTRUMENTATION

VARIABLE	ENG TEST		FLT TEST	
	B/L	IPCS	B/L	IPCS
AIRPLANE/TEST CELL CONDITIONS				
Freestream Total Pressure	X	X	X	X
Freestream Static Pressure	X	X	X	X
CG Long. Accel.			X	X
Freestream Total Temperature	X	X	X	X
Wing Sweep Angle			X	X
Angle of Attack			X	X
Angle of Sideslip			X	X
ENGINE VARIABLES				
Total Fuel Flow. Wft	X	X	X	X
Engine Fuel Flow, Wfe	X	X	X	X
Engine Fuel Temperature	X	X	X	X
Throttle Position, (PLA)	X	X^1	X	X^1
RPM (N1)	X	X^1	X	X^1
RPM (N2)	X	X^1	X	X^1
Engine Hub Total Pressure, Pt2n			X	X^1
Fan Exit Static Pressure, PS13	X	X	X	X
LPC Exit Static Pressure, PS22	X	X^1	X	X^1
LPC Exit Total Pressure, P22	X	X^1		X^1
LPC Exit Pressure Differential (P-PS)22	X	X^1		X^1
HPC Exit Static Pressure, PS3	X	X^1	X	X^1
HPC Exit Total Pressure, P3		X^1		X^1
HPC Exit Pressure Differential, (P-PS)3		X^1		X^1
LP Turbine Exhaust Pressure (P6M)	X	X^1	X	X^1
Engine Pressure Ratio (EP2)			X	X
Compressor Face Temperature (T2)	X	X^1	X	X^1
HPC Inlet Temperature (T22)		X^1		X^1
HPC Exit Temperature (T3)	X	X^1	X	X^1
Turbine Inlet Temperature-measured (T4)		X^1		X^1
Turbine Inlet Temperature-harness (T4H)	X	X^1	X	X^1
Turbine Discharge Temperature (T5)	X	X^1	X	X^1
Nozzle Area (AJ)	X	X^1	X	X^1
Compressor Bleed Switch Positions	X	X^1	X	X^1
Engine Fuel Pressure (2)	X	X	X	X
A/B Fuel Pressure (5)	X	X	X	X
Main Fuel Valve Position		X^1		X^1
A/B Metering Valve Position (5)		X^1		X^1
Computer Calculated Parameters		X^1		X^1

1   Available from the DPCU

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TABLE 3 (Continued)

VARIABLE	ENG TEST B/L	ENG TEST IPCS	FLT TEST B/L	FLT TEST IPCS
INLET VARIABLES				
40-Probe Compressor Face Rake	X	X	X	X
Rake Zero Switch	X	X	X	X
Reference Pressure	X	X	X	X
Distortion Computer Output	X	X	X	X
Nulling Rake Pressure			X	X
Spike Position			X	X
Cone Angle			X	X_1
Local Mach Pressure Ratio			X	X^1
Duct Mach Pressure Ratio			X	X^1
Shock Position Signal			X	
Distortion Signal				X^1
MISC.				
Tape Motion Switch			X	X
Event Marker	X	X	X	X

1   Available from the DPCU

During flight tests the PCM data will be telemetered to the ground for use in monitoring the progress of the test flight. Approximately 80 digital-to-analog converters and Sanborn recorders are available to convert the data into time histories during the flight. There is no requirement to telemeter any of the analog data during the flight.

Data Processing

NASA/LeRC provided on-line capability to process much of the steady-state data during baseline engine tests. The on-line program, which uses a remote terminal on the IBM 360, had the capability of calculating any of several separate sets of variables. A complete run of the program was made overnight to provide the remainder of the data by early the morning after the run. In addition, selected data were recorded on oscillograph for use in running the test.

The PCM data will be telemetered during the flight tests. These data will be demultiplexed and up to 80 channels will be displayed on Sanborn recorders on-line, in real time. Digital tapes of the data from the PCM system will be prepared by NASA. These data will be calibrated and will be in engineering units. Printouts of these tapes will be available within a week of the flight for use at FRC.

The FM data are demultiplexed and digitized by the Boeing Test Data Processing Center. The typical data sample consists of a 200 millisecond interval centered about an event such as a period of high distortion or compressor surge. The pressure signals are low-pass filtered (-3Db at 160 Hz) to retain only the frequency range of significance to the engine. Data are digitized at a rate of 1,000 samples per second per channel. The output digital data tape is converted to a format compatible with the CDC 6600 for the remainder of the processing. The digitized data are then processed through the distortion routine used with the steady-state data. Figure 5 presents a typical distortion time history for a stall event from the baseline test.

The major differences between the engine and flight test data processing programs are in the input and output routines due to the different data systems and variables being recorded, absence in the flight programs of some engine calculations, and the addition of inlet and airplane computations in the flight data program. Data from steady flight conditions will be averaged to produce steady-state data.

DYNAMIC SIMULATIONS

Dynamic simulations of the F-111 propulsion systems have formed the foundation for the IPCS control system development and software validation. Two types of simulations have been generated. The first is an entirely digital simulation developed for use on a large digital computer such as a

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CDC 6600. The second is a comprehensive hybrid simulation, based on the digital simulation, that was developed by Honeywell. They incorporate much of the system definition data and hence form a compact and convenient repository for masses of detailed information. Linear state models extracted from the digital simulation have been used to study control system stability and response. The digital simulation has been the principal test bed for evaluating new control modes. The hybrid simulation is being used to evaluate the response of the system to selected failures and will be used to check out both the digital propulsion control unit (DPCU) and its software prior to shipment.

Digital Simulation

The digital simulation employs the SOAPP system developed by P&WA. With this modular system, most of the simulation is created from routines drawn from the SOAPP library. This library is a major reason for the development of SOAPP. It forms a repository for up-to-date versions of those utility routines that determine the speed and accuracy of the simulation.

The SOAPP program generates both steady-state and transient engine performance data. This feature is made possible by the application of a technique called SMITE, originally conceived by the Air Force AeroPropulsion Laboratory. It uses the solution to a set of linearized adjunct equations to obtain an iterative solution to the complex nonlinear equations in the simulation. Steady-state solutions are obtained merely by setting all the temporal derivatives to zero. Figure 6 illustrates the simulation adjustment. Data generated by the digital program are compared to corresponding baseline engine test data obtained at NASA/LeRC. Adjustment improved the fidelity of the simulation significantly.

Hybrid Simulation

The hybrid simulation of the propulsion system has been prepared using two 781 EAI analog computers, two 231R EAI analog computers, a PACER 16k digital computer, and a SIGMA 5 40k digital computer. The PACER is used solely for generating bivariate functions, for on-line analysis, and for problem setup. The Σ5 computer is used to generate the control functions and to drive a scope display. The system has been designed to run ten times slower than real time when under control of the Σ5.

Check-out of the DPCU hardware and software will be accomplished by replacing the Σ5 by the HDC 601 flight computer with its interface unit (IFU). A custom built simulation interface adapter (SIA) will condition signals from the analog computers to simulate the outputs from the flight transducers. In this service, the simulation will run in real time. All time-dependent functions are performed in the EAI 781 computers to facilitate time scale switching.

CONTROL MODE IDENTIFICATION

The DPCU will exercise control over six variables:

Gas generator fuel flow
Compressor bleeds - 7th and 12th stage
Afterburner fuel flow
Exhaust nozzle area
Inlet spike position
Inlet cone position

These variables must be adjusted and coordinated to provide engine thrust in response to power level setting while maintaining safe, stall free operation. Development of appropriate control modes was a major IPCS activity.

Gas Generator Fuel Flow

Isochronous governing of N2 is the primary gas generator fuel control mode. Steady state high rotor speed is a function of power level angle (PLA) and fan face conditions; total pressure (P2) and total temperature (T2). Isochronous control holds thrust more nearly constant during bleed and shaft power extraction than does droop control. It also provides better thrust response during part power excursions. Limiting loops are provided to override the N2 loop when required to protect engine integrity or operating stability. Direct measurements are used for limiting where available; otherwise correlations are used. The limiting loops are listed in Table 4.

TABLE 4

IPCS GAS GENERATOR LIMITING LOOPS

Limited Variable	Signal Source	Purpose
Low rotor speed	Tachometer	Structural Limitation
High rotor speed	Tachometer	Structural Limitation
Burner Pressure	Pressure transducer	Structural Limitation
Compressor exit Mach No.	$\Delta P/P$	Stall Prevention
Airflow	$f(N1/ \sqrt{\theta 2},$ EPR)	Engine/Inlet Compatibility
Turbine Inlet Temp.	Fluidic transducer	Turbine Overtemp Protection

Compressor Bleed Control

The distortion tolerance of the TF30-P-9 engine is a strong function of low rotor speed and compressor bleed position as shown by figure 7. The IPCS test aircraft will be equipped with four pressure probes in the inlet duct. Distortion will be inferred from the output of these four probes plus the output of a high-response (Kulite) transducer installed in the NASA test instrumentation rake. The distortion correlation is shown in figure 8.

In operation the engine distortion tolerance will be compared to the sensed distortion. Bleed positions will be selected as required to protect the engine. The compressor bleeds are also opened under certain conditions at low power settings and during engine deceleration to provide greater stall margin.

Afterburner Fuel Control

The IPCS modulates afterburner fuel flow in the afterburning region as limited by engine requirements and the need to maintain engine/inlet compatibility. The design approach was to use direct (or synthesized) measurements to schedule fuel flow and maintain fan suppression limits. The IPCS schedules engine stream and duct stream afterburner fuel-air (f/a) ratio as a function of a rate limited PLA. This signal is also used to schedule base exhaust nozzle area. Engine stream airflow is calculated as a function of HPC discharge pressures and temperature (P3, PS3, and T3). Duct stream airflow is obtained from the difference between total calculated airflow and engine stream airflow.

Calculation of the zone fill valve timing to permit prefill of manifolds is performed as a function of a rate limited power lever angle signal. Zone fill time, using flow rates and manifold volumes, determines the rate limited PLA signal at which the zone fill valve is opened. Transient performance improvement obtained through use of the A/B prefill logic is shown in figure 9. The IPCS will consistently achieve maximum thrust in the period of time shown in figure 9.

The normal mode for maintaining fan suppression is with the exhaust nozzle area. This mode is discussed under Exhaust Nozzle Area Control. There are, however, certain regions in the flight envelope where the exhaust nozzle area cannot be opened further. In this case control is transferred to the afterburner fuel control loop to permit fuel cutback to maintain the fan match.

Exhaust Nozzle Area Control

The rate limited PLA that schedules after burner fuel-air ratio is also used to schedule a nominal exhaust nozzle area. This schedule is set to minimize airflow trim requirements. The schedule is also designed to force the area open faster when fuel is added, and close slower when the fuel is

decreased. This provides a fan operating point during A/B transients that is farther away from the stall line, resulting in a slight undersuppression.

The IPCS fan suppression control for the TF30-P-9 engine uses the fan match line as a reference schedule for trimming about the base area setting. An airflow reference is balanced against the airflow correlation measurement to trim area until the fan match is satisfied. If the fan match cannot be satisfied due to area being at the maximum limit, trim authority is transferred to the afterburner fuel module. The main fuel module is also biased with a trim signal received from the inlet module, to improve the off design engine/inlet airflow match.

Inlet Control

A sketch of the F-111 inlet installation is shown in figure 10. The controllable aerodynamic surface is the spike, which translates fore and aft. The spike surface consists of two cones; the second cone may be expanded or contracted over the range of 8.5° to 26° included angle.

In the bill-of-materials (BOM) inlet control, both the spike and cone positions are scheduled as functions of local Mach number and duct exit Mach number. The BOM inlet control schedules have been retained for the IPCS. They are supplemented by an anticipation function that momentarily resets the surfaces for smoothing the afterburner light-off or shut-down transient. A buzz detector, based on that developed for the SST, is provided. It repositions the surfaces for more efficient supersonic air spillage when buzz is sensed. Engine/inlet compatibility is enhanced by an airflow loop that shifts both engine and inlet operating points slightly to control the inlet throat Mach number.

PERFORMANCE/STABILITY TRADES

There is usually a stability penalty associated with each performance improvement. The standard procedure during a development program is to establish a formal trade study to determine the optimum balance between performance and stability. A system such as IPCS, with the flexibility to change software schedules and set points well into the development cycle, lends confidence to the trade results since it can be based on actual, rather than projected system performance. For example, protection of compressor stability by sensing compressor exit Mach number has been discussed earlier. The program timing does not permit a test series to develop pressure probes to sense internal Mach number. Hence, a tentative compressor exit Mach schedule will be programmed into the software; the schedule will be modified as necessary during the engine test program.

One of the major IPCS goals is the sensing of incipient instability or conditions indicative of instability so that operating points may be shifted as required for duration of the disturbance. Three disturbances in particular are thus addressed for the first time in a flight test program:

Inlet buzz
Inlet flow distortion
Afterburner rumble

Steady-state inlet distortion was discussed in the previous section. The unsteady component of distortion, inlet buzz, and afterburner rumble are each sensed by circuits tuned to respond to pressure fluctions in the frequency ranges of interest. The circuit output increases gradually (with, e.g., a 0.5-second time constant) when pressure fluctions are sensed and decays to zero when the distribution disappears. This approach is based on the buzz detector/suppressor, developed for the SST, which was very successful in closed-loop wind tunnel tests.

STABILITY ANALYTICAL MODELS

This section describes in broad terms the methods used in the analytical design process. The fundamental procedure uses small-perturbation methods to design to a series of operating points in the flight envelope. The designs thus developed will be programmed for computer simulation. The simulation will then be subjected to gross transients over the entire flight placard to verify the design.

The procedure is diagrammed in Figure 11. It was designed to make maximum use of existing computer programs and has been developed to provide the greatest feasible degree of automation, both to save time and expense and to assure consistency in the application of design criteria. The procedure is as follows:

o Small-perturbation methods are applied to the digital simulation of the propulsion system to linearize about the desired operating points.
State models of the form

$$\underline{x} = A\underline{x} + B\underline{u}; \ \underline{Y} = C\underline{x} + D\underline{u}$$

are generated.

o Transfer functions required for loop-by-loop analysis are calculated from the state-matrix model.

o Classical linear, constant coefficient design methods are used to develop compensation on a loop-by-loop basis.

o   The above steps are performed for a series of operating points
    to obtain the relationships between compensation parameters and
    engine burner pressure. Polynominals are generated to describe
    the relationships.

o   The compensation polynominals are programmed into the nonlinear
    (SOAPP) simulation and subjected to standard disturbances over
    the flight envelope.

o   Adjustment is made and the process is repeated if necessary.

    In lieu of classical design specs, the following criteria have been
established as golas to be used in controller compensation design:

o   Phase margin of all loops shall be at least 65°.

o   Gain margin of all loops shall be at least 6db.

o   Loops designated as limiting (maximum or minimum) shall have
    no overshoot when subjected to a step input.

o   Where overshoot is permitted, overshoot shall not exceed the
    value attained under BOM control as predicted by the SOAPP
    simulation.

o   Rise time of each loop shall be as fast as the value attained
    under BOM control.

o   Settling times of variables shall not exceed settling times
    attained by the SOAPP simulation under BOM control.

    For some of the IPCS control loops there are no analogous loops in the
BOM controller. Time domain specs for these loops are based on engineering
judgement of what is required to attain good servo response. All loop com-
pensations designed by linear single input/single output methods are verified
using the non-linear SOAPP simulation.

SOFTWARE DEVELOPMENT

    Efficient software is crucial to the success of a program, such as IPCS,
that employs a digital computer as the central element of a control system.
Furthermore, it is a large-budget, long-lead-time item. Esoteric by nature
and unspectacular compared to high-technology hardware, software has been a
source of much grief to the unwary. In view of these factors, a significant
portion of the IPCS effort has been devoted to software development.

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Two sets of software are being developed for the IPCS program; a digital representation of the bill-of-material hydromechanical control (BØMDIG) and the computer implementation of the control modes discussed earlier in this paper (IPCS). The software is organized in modular form, which is consistent with the requirements of this program; since both sets of software will drive the same engine hardware. Since most of the sensors are common, many of the subroutines are common to the two programs. The software is being programmed for the Honeywell HDC 601 computer, which is a 16-bit machine with a 16k military core. Characteristics of this machine may be found in the literature.

Memory and timing estimates for the two programs are listed in Table 5. The factor that may appear unique to the engine control is the large number of functional relationships that are stored as tables. Table 5 indicates that 42% of the BØMDIG memory requirement is devoted to data storage. Deleting that portion of the data base that deals with initialization, input, output, etc., it is found that BOMDIG requires about 3600 locations for tables while the IPCS algorithm requires about 6700 locations.

TABLE 5 .

MEMORY & TIMING ESTIMATE OF CONTROLLERS FOR THE IPCS PROGRAM

	BOMDIG		IPCS	
SUBROUTINE	MEMORY (WORDS)	TIME (MSEC)	MEMORY (WORDS)	TIME (MSEC)
EXECUTIVE	1400	2.84	1400	2.84
SENSOR PROCESSING	1300	4.74	1300	4.74
CONTROL SUBROUTINE	6500	8.25	8300	14.66
OUTPUT PROCESSING	450	1.48	450	1.48
COMPUTER PROGRAM DATA BASE	750	—	750	—
	10400	17.31	12200	23.72
	+550	+0.86	+600	+1.17
	(42% Data)		(61% Data)	

Since the IPCS is an R&D effort, tasks such as BITE and redundancy do not comprise a large portion of the computing effort. There is a sample problem within the executive subroutine to check the computer function. The outputs of sensors critical to flight safety are tested to determine whether the signals are within the normal operating range. If a failure that might result in damage to the engine is sensed in this manner, control is transferred to the hydromechanical fuel control, which is retained as a backup in this program. Figure 12 diagrams the IPCS Fail Safe provisions. There are in addition some synthesized signals that will be used as replacements if the input goes out of the normal operating range.

Sampling periods of 20 milliseconds and 30 milliseconds for the BOMDIG and IPCS algorithms, respectively, are based upon the timing estimates shown in Table 5. Because program schedule limitations, no particular effort has been made to simplify the IPCS control functions. Neither has any effort been devoted to determining which functions could be sampled at intervals longer than the basic sampling period. In view of this, it is estimated that an optimized IPCS control of the future, without BITE and engine health monitoring added, would require about 11,300 words of memory and have a computational time requirement of 20.6 milliseconds.

In other digitally controlled systems where reliability is a major consideration (Space Shuttle Engine Control and some flight control systems), a rule of thumb has been that the control subroutine time requirement should be increased by a factor of three to include reliability needs. Under this assumption, the control subroutine computational time would be 44 milliseconds. This number suggests that additional computer capability is required to cope with the expanded work load.

The most attractive solution appears to be to apportion the control tasks to a number of parallel processors that are essentially identical to achieve reduced production costs through higher volume. This option can be exercised only if care is taken to provide adequate communication between subcontrollers so that true control integration can be achieved.

TEST PROGRAM

A sequence of hardware tests will be conducted to evaluate the IPCS. This series is progressing from baseline evaluation of the existing system in a low risk, step-by-step manner through flight evaluation of the IPCS. The test flow is diagrammed in figure 13. This test program provides high confidence of success at each phase due to the gradually increasing complexity of the tests.

Baseline Tests

Baseline engine and flight tests are designed to document the performance of the F-111E/TF30-P-9 system prior to the IPCS modifications. The baseline engine test has been completed. During this test the two engines to be modified for IPCS were tested over a range of flight conditions to establish steady-state and transient performance and distortion tolerance. Data from the test have been analyzed and used to update the dynamic simulation. The baseline flight test will provide similar data for the airplane and inlet.

Subsystem Tests

Individual component performance and physical integrity will be demonstrated, where necessary, through component and subsystem tests. Individual components will be subjected environmental tests, temperature cycling and vibration in particular, as required by the NASA specifications. The DPCC hardware and software will be thoroughly checked out prior to shipment from the Honeywell facility as indicated earlier under "Hybrid Simulation."
The control software will be loaded into the HDC flight computer and tested in real time with the loop closed by the hybrid simulation. The flight conditions to be explored are sea level static and three Mach numbers at 45,000 feet: 0.9, 1.6, and 2.1. A full complement of power transients will be executed. Typical flight disturbances will be presented to the system. The effect of transducer failure will be evaluated by disconnecting the signal lines to simulate failure. This extensive in-house test program will drastically reduce the number of "bugs" encountered during subsequent system-level testing and will thereby effect significant savings in both cost and calendar time.

Closed-Loop Bench Test

A comprehensive closed-loop bench test will follow the component tests. The TF30 fuel controls, modified to incorporate electrical interfaces, will be installed in the P&WA fuel bench test facility. A schematic of the test set-up is shown in figure 14. The flight DPCU will be connected to the fuel controls through electrical cables of length chosen to simulate the aircraft installation. The inlet actuators, with their position feedback transducers will be installed in a jig, supplied with hydraulic power, and connected to the DPCU. Analog simulations of the engine and the inlet aerodynamics will be provided to close the loops and generate the signals that would be sensed by transducers in the aircraft. In some cases, simulation interface adapters will be provided to simulate the transducer output format. This test will provide a functional check out of the modified fuel control unit and will establish compatibility between the DPCU and its software and the engine and inlet control hardware.

Sea Level Static Test

The second test of the series will be a sea level static (SLS) engine test, also conducted at the P&WA facility. The modified fuel controls will be installed on the modified TF-30 engines which will be mounted on a test stand. This test will provide the first opportunity to demonstrate IPCS operation with the engine and will permit the necessary fine tuning prior to the altitude test. The SLS test will also serve as the acceptance test for the IPCS and the modified engines.

Altitude Facility Test

The altitude test at NASA/LeRC will duplicate most of the operating conditions scheduled for investigation during the IPCS flight tests. Operation of the IPCS will be refined at points throughout the flight envelope, again using an analog inlet simulation. The NASA/LeRC "puff-jet" distortion generator will be used to create disturbances to check operation of the IPCS buzz suppression and distortion loops. Following the altitude test, the modified engine and DPCU will be installed in the airplane for a flight evaluation of the IPCS operation.

The installation of the IPCS on the flight-test aircraft will be performed by NASA/FRC. All electrical cables on the aircraft will be fabricated by FRC to drawings supplied by the contractor. Following installation, a thorough check-out and ground test will be conducted.

Flight Test

A six-month Flight test evaluation of the IPCS is scheduled. The tentative flight test points are shown figure 15. Test planning has not been completed at this writing. It is anticipated, however, that the projected 26 flights (approximately 50 flight hours) will provide sufficient time to evaluate all of the IPCS features under a variety of conditions.

CONCLUSIONS AND RECOMMENDATIONS

Since the IPCS program is not quite half completed as this is written, firm conclusions are premature. It is possible, however, to submit tentative conclusions based on experience to date. These are offered below subject to the qualification that they may be modified as further experience is gained.

1. Basing the bulk of the controls analysis on a detailed digital simulation of the propulsion system was a sound approach. A digital computer tape is a convenient and compact way to transmit masses of technical detail. This approach also assured consistency between analytical work performed by each of the three major contractors.

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2. Many delays were experienced during baseline engine testing due to shortages of electrical power to operate the facility. If energy shortages prove to be persistent, new approaches to test operation may be necessary. In particular, more rapid methods of establishing test conditions, instrumentation that minimizes time spent on condition, and data reduction methods that correlate data collected under slightly different test conditions would alleviate the problem significantly.

3. Computer software is an item whose importance can scarcely be overstated. There appears to be a tendency in the industry to underestimate lead time and overestimate the flexibility of software. Once constructed and checked out, software is almost as difficult to change as hardware. The principal difference is that software does not have to be vibrated to demonstrate mechanical integrity.

4. Standardized methods for transmitting information between airborne digital machines are essential if the full potential of digital electronics is to be realized. The centralized super computer that performs all calculations aboard the aircraft appears to be unfeasible throughout the foreseeable future. Sets of small machines operating in parallel are practical and economical, provided the communication problem is solved.

Although IPCS hardware has not been discussed in depth, some comments are in order:

1. Electromechanical and electrohydraulic interface devices must be selected very carefully to minimize electrical and electronic problems. From an electronics standpoint, for example, torque motors are preferable to stepper motors, linear variable differential transformers (LVDTs) are preferable to resolvers for position sensing.

2. Transducers will continue to present problems throughout the foreseeable future. Controls engineers must make a determined effort to design out of the system the requirement for high accuracy and/or high response.
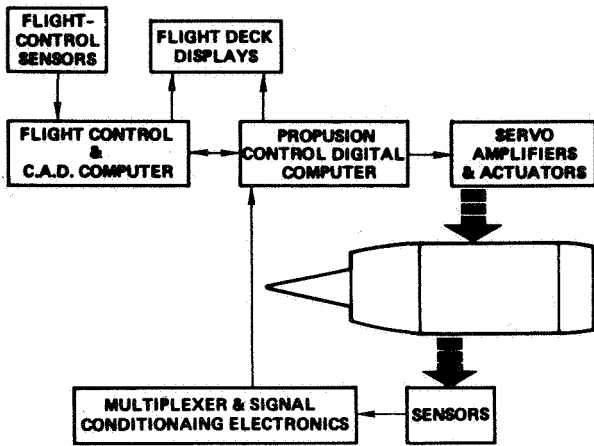
Figure 1: Integrated Flight/Propulsion Control Concept

Figure 2: Integrated Propulsion Control System

Figure 3: IPCS Organization
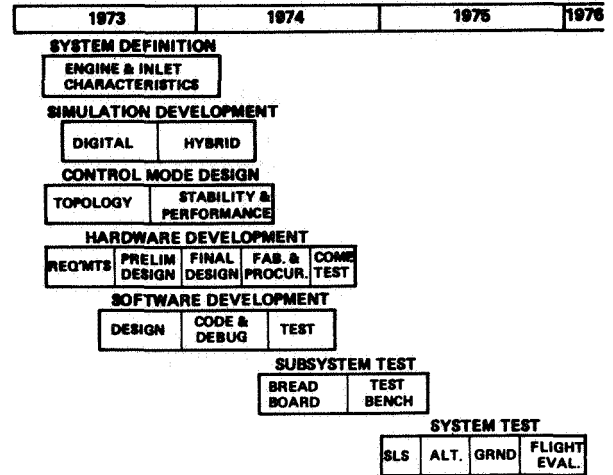
Figure 4: IPCS Development Sequence
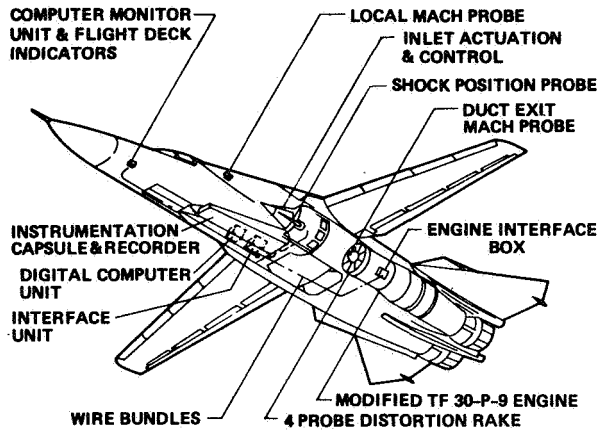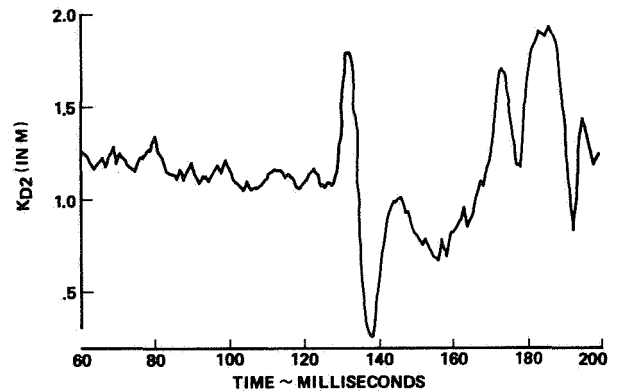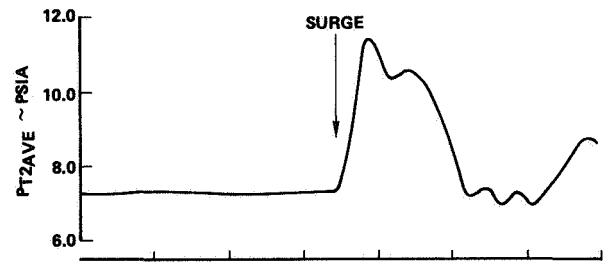
Figure 5: Distortion Time History, Baseline Engine Test

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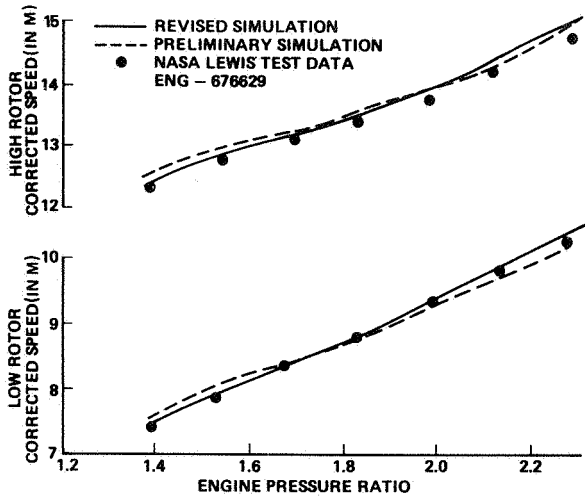
Figure 6: Comparison of Baseline Engine Data with Simulation at 30,000 Ft. Mach 0.8.
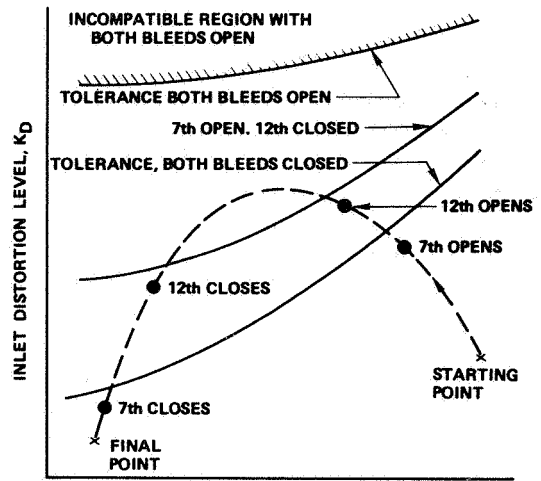
Figure 7: Compressor Bleed Operation

$$P/P = \frac{P_{MAX} - P_{MIN}}{P_{MIN}} \quad \text{FROM 4 PROBE RAKE}$$
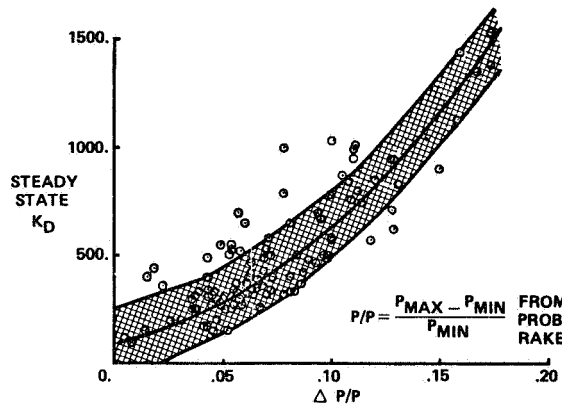
Figure 8: Steady State Distortion Correlation

TF30P-9 ENGINE/FIIIE INLET
MIL to MAX THRUST TRANSITION
45000 FT., MACH 0.9

IPCS T3738
B.O.M. T 9591
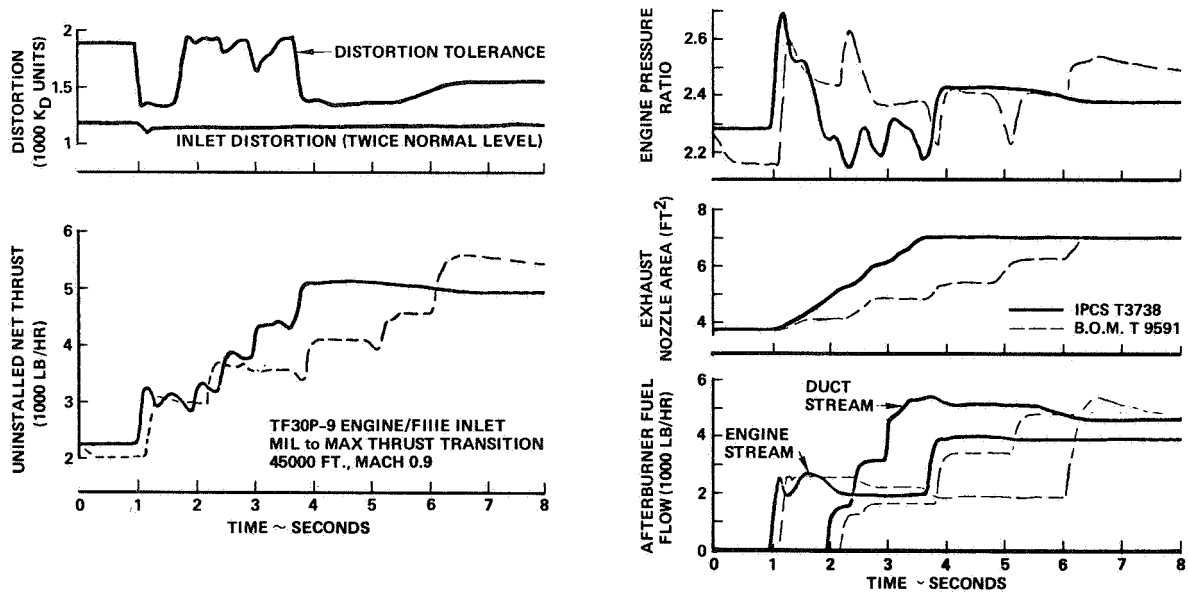
Figure 9: Military to Maximum Afterburner Transient

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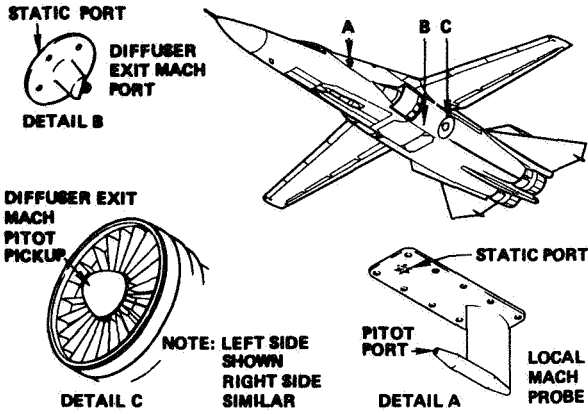
Figure 10:   F-111 Inlet Installation
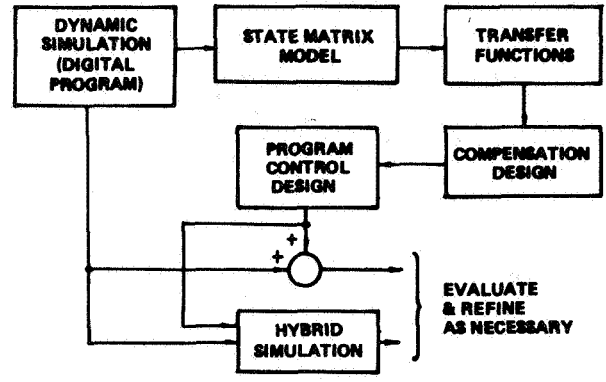
Figure 11:   Analytical Design Procedure
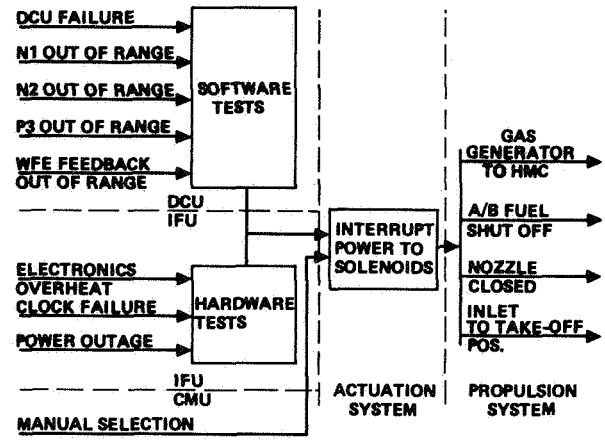
Figure 12:   IPCS Fail-Safe Provisions
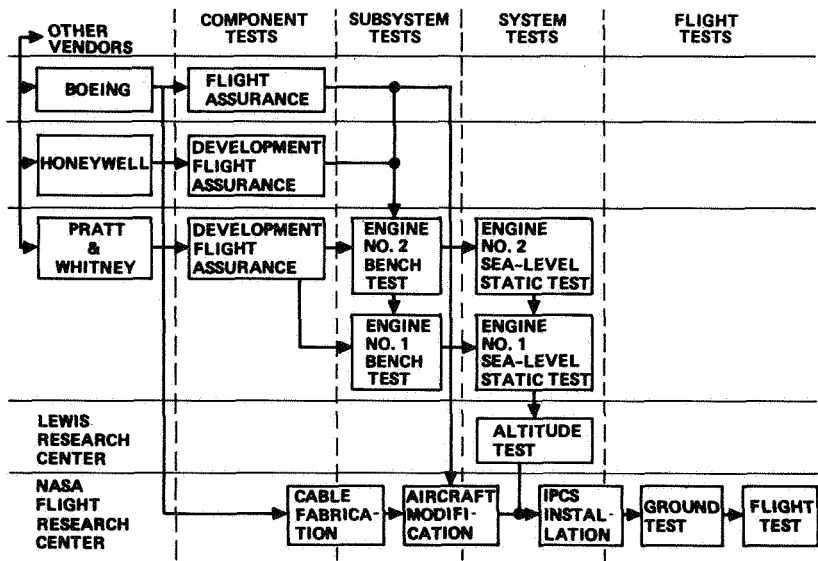
Figure 13:   Test Flow
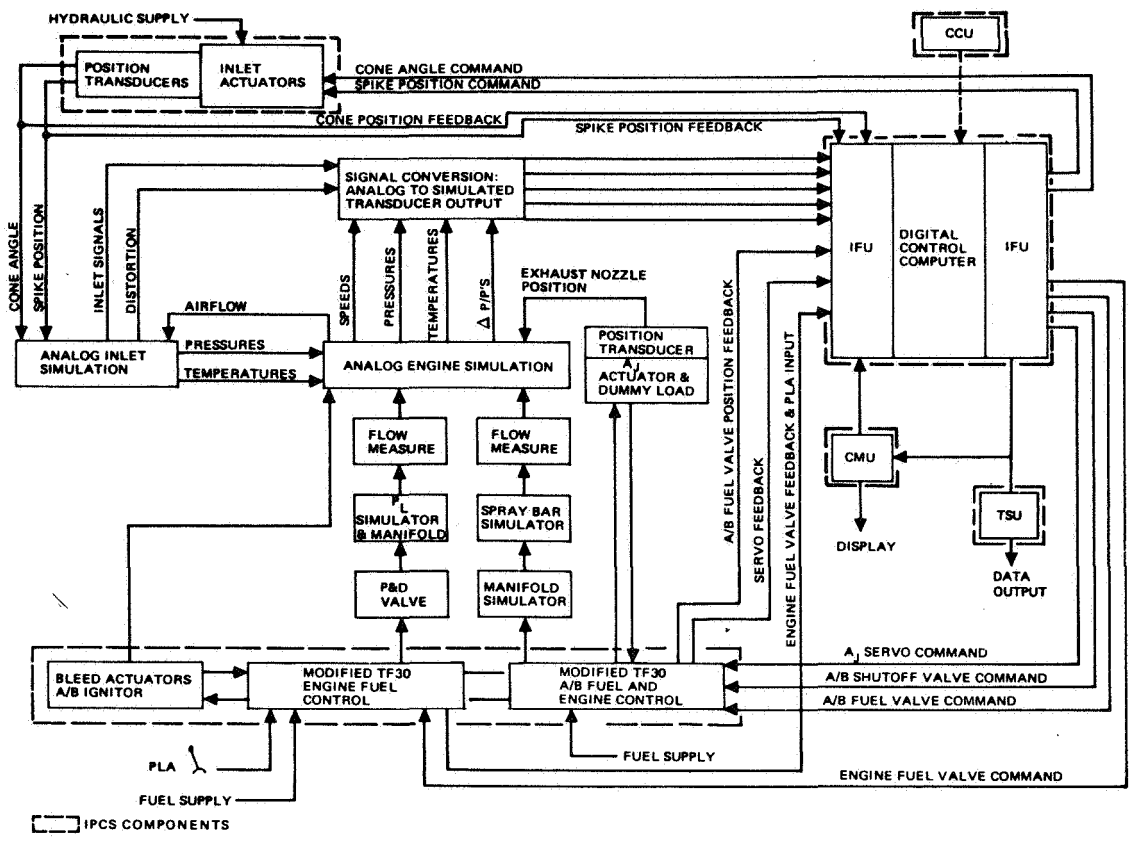
Figure 14:   IPCS Closed-Loop Bench Test Schematic
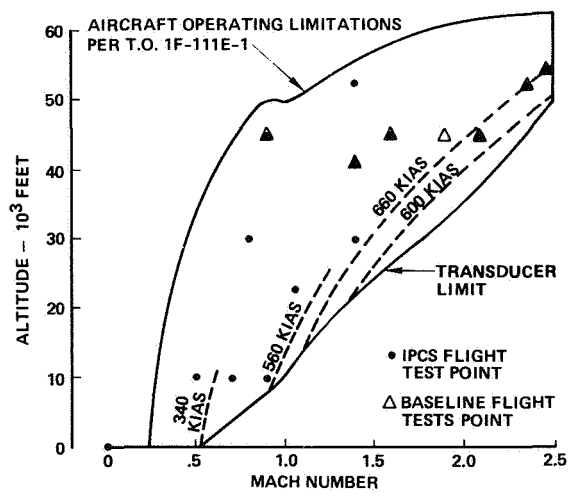
Figure 15:   IPCS Flight Test Envelope

A DIGITAL COMPUTER PROPULSION CONTROL FACILITY -

DESCRIPTION OF CAPABILITIES AND SUMMARY OF

EXPERIMENTAL PROGRAM RESULTS

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SUMMARY

Flight-weight digital computers are being used today to carry out many of the pro-
pulsion system control functions previously delegated exclusively to hydromechanical
controllers. An operational digital computer facility for propulsion control mode studies
has been used successfully in several experimental programs at the Lewis Research
Center. This paper describes the system and some of the results thus far obtained.
These results are concerned with engine control, inlet control, and inlet-engine integrat-
ed control. Analytical designs for the digital propulsion control modes include both
classical and modern/optimal techniques.

INTRODUCTION

With each advancement in integrated-circuit technology, the reliability of electronic
digital computers designed for use in severe aircraft environments improves. As a re-
sult, flight-weight digital computers will be used to carry out more and more of the pro-
pulsion control functions now being handled by continuous hydromechanical controllers.
Flight-weight digital computer controllers have already been selected for operational
flight applications as a full-authority supersonic inlet control and as a supervisory con-
trol on an afterburning turbofan engine. In addition to operational flight applications, a
general-purpose electronic digital computer controller can greatly simplify the develop-
ment of control modes for advanced airbreathing propulsion systems. The many schedul-
ing and logical manipulations necessary in the control of complex high-performance
propulsion systems are well suited to the capabilities of a digital computer. Control
modes can be easily implemented in software and checked out without actual flight hard-
ware development.

Because of the benefits of a general-purpose digital computer for propulsion control
mode development, the Lewis Research Center put into operation several years ago the

Digital Computer Propulsion Control Facility for developing advanced propulsion control modes. It is designed to permit real-time, on-line implementation of controls for various configurations of airbreathing propulsion systems operating in sea-level, altitude, and wind tunnel test facilities.

This paper describes the capabilities of this facility as they relate to the requirements of propulsion system control mode research and discusses various test programs in which this digital facility was used successfully. First, the Digital Computer Propulsion Control Facility in use at the Lewis Research Center is described in detail. Next, the various programs in which the facility was employed as an active control system are briefly described. The results obtained from the various programs are then summarized. These results include data related to such things as inlet shock-position control and engine fail-operational control. Finally, some of the present activities in this area are discussed and future areas of investigation for digital propulsion control recommended.

DIGITAL COMPUTER PROPULSION CONTROL FACILITY

Design Considerations

The Digital Computer Propulsion Control Facility was designed to provide versatile control of airbreathing propulsion systems including (1) inlet control, such as shock-position regulation and restart scheduling; (2) engine control to provide thrust and specific-fuel-consumption optimization under operational restrictions; and (3) combined inlet-engine interaction optimization.

The prime considerations in the design of the computer control facility were overall signal processing speed and computational capacity. The system must be capable of accepting the necessary system inputs, processing them, and outputting the commands within the frequency range of the propulsion system dynamics. For example, inlet control generally requires a high rate of control command updates but relatively few measurements and calculations. On the other hand, engine control does not generally require a high rate of control command updates but could entail the measurement of many engine parameters and could require extensive control computations.

System processing speed is a function of computer computational speed and the versatility of the input-output structure. Computational speed is maximized through the use of an efficient programming language and a fast-response-time computer. Computer response time is denoted by memory cycle time, which is the time it takes to read and restore a computer word in memory. Typical memory cycle times of real-time computers presently available range from a fraction of a microsecond to a few microseconds. Programming languages become less efficient as they are removed from the basic machine

language. The most efficient language offers a one-to-one correspondence with machine language. This language is generally called assembly language. The versatility of the assembly instruction set is of prime importance in the selection of a digital computer for purposes of control.

The input-output structure should require a minimum of machine computation time. Direct automatic transfer of blocks of data to and from computer memory on a cycle-stealing basis is necessary. This method causes disruption of the computation process for only one machine cycle per word of data transferred.

In addition to these internal equipment considerations, external characteristics of the digital control equipment were also a consideration. Control development is best done with the aid of comprehensive analog or hybrid simulations of the propulsion system. Since long-line communication between the Lewis simulation laboratory and all the planned propulsion test areas did not originally exist, portability of the digital control equipment was a requirement.

General Facility Description

The digital computer controller is made up of several distinct units:
(1) A digital computer designed for real-time control applications
(2) A digital interface capable of converting both analog and frequency signals to computer-compatible digital words and converting computer-generated words to analog and logical outputs
(3) Programming peripherals consisting of a high-speed, paper-tape reader and punch and a teletype
(4) A signal processing unit (SPU) which provides signal conditioning and monitoring, as well as some analog computation capability, between the digital interface and the propulsion system to be controlled

The digital computer, the digital interface, and the programming peripherals were supplied as a system from a digital computer manufacturer. The signal processing unit was assembled at Lewis from purchased components.

The system, excluding the teletype, is housed in five distinct racks (fig. 1) requiring approximately 3.4 running meters of floor space. Intercabinet cabling is accomplished in the rear and allows a maximum of 3.05 meters of spacing between adjacent cabinets. The system was designed to be portable. Typical teardown and setup time is 1 day, and complete system checkout requires 1 week

The block diagram of figure 2 illustrates the basic units and interconnection of the digital control system. All signals, to and from the propulsion system, pass through the signal processing unit (SPU).

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The SPU will accept high level (±10 V range) analog signals already amplified and signal conditioned from standard pressure transducers and thermocouples. It will accept frequency signals directly from flowmeters and magnetic speed transducers. System outputs may be directed to proportional electrohydraulic servosystems or on-off types of devices. These serve as the control inputs to the propulsion system manipulated variables. The SPU was designed to increase flexibility in the calibration and operation of the control system. In particular, the SPU provides

    (1) Ground isolation between the facility and the control unit

    (2) Signal filtering

    (3) Analog computation for propulsion system simulation or generation of time-dependent control functions

    (4) Flexibility in signal routing between the facility and the control unit

    (5) Calibration of the system

    (6) Comparators and signal conditioners for use with priority interrupts

    (7) Signal monitoring

Figure 3 illustrates the SPU cabinet layout and its equipment complement.

The digital interface consists of a high-level, analog signal acquisition unit; a frequency signal acquisition unit; an analog signal output unit; a logical output unit; and an external priority interrupt processor. The digital interface communicates with the computer on either a single-word or a block-data-transfer basis. The programming peripherals communicate only by single-character transfer. The signal-processing capability of the system is given in table I. Table II contains a complete list of specifications for the computer and digital interface equipment.

The computer itself is programmed through the use of paper tape. The system includes a high-speed, paper-tape reader and punch. The reader operates at 300 characters per second and punches at 110 characters per second. One character consists of eight binary bits. Paper tapes may be generated on an ASR 35 teletype and may also be read into the computer by this unit.

A more detailed description of the complete digital computer facility is given in reference 1.

<center>Digital Propulsion Control Programs</center>

The Digital Computer Propulsion Control Facility just described has been in operation at Lewis for approximately $4\frac{1}{2}$ years. The facility has been used continuously throughout that period. The experimental programs in which it has been utilized for propulsion control mode research are summarized as follows:

    (1) Mixed-compression experimental inlet in the 10- by 10-Foot Supersonic Wind Tunnel: High-performance shock-position and restart control studies using both

classical and modern control design techniques

(2) Full-authority digital computer control of a turbojet engine in a sea-level test stand: Bill-of-material control modes with prediction techniques

(3) Full-scale symmetric, mixed-compression inlet in the 10- by 10-Foot Supersonic Wind Tunnel: Digital implementation of bill-of-material control modes as well as research control modes

(4) Fail-operational type of turbojet engine controller: Evaluation in sea-level test stand

(5) Integrated engine-inlet control: Mixed-compression inlet and afterburning turbofan engine in the 10- by 10-Foot Supersonic Wind Tunnel

In each of these experimental programs, the software implementations of the digital propulsion control laws were checked out and debugged with real-time analog simulations of the inlet and/or engine. This activity was carried out with the computer equipment located in the simulation laboratory. In the early experimental programs, the equipment would then be physically moved to the control rooms of the experimental facilities. The most recent programs, though, have employed a central location (simulation laboratory) and communicated with the process being controlled via underground long-lines. This approach, using appropriate line-driving electronics, has been highly successful even for distances of some 450 meters. Cable communication is available at Lewis between the computer facility location and the 10- by 10-Foot Supersonic Wind Tunnel and the four altitude tanks. Since programs planned for the near future involve only these experimental facilities, the central location approach will be in effect for some time. Such an approach permits double duty for the facility, with simulation evaluation of controls taking place on one shift and experimental evaluation taking place on another shift.

In the following section, some of the results obtained in the digital propulsion control studies are discussed.

DIGITAL PROPULSION CONTROL RESULTS

Digital Inlet Control

The Digital Computer Propulsion Control Facility was first used for direct digital control of an experimental mixed-compression inlet in the 10- by 10-Foot Supersonic Wind Tunnel. The function of the controls research was to evaluate shock-position control techniques as well as restart control concepts. A complete description of the test program and its results is contained in reference 2. A brief summary of the results is included in this paper.

The inlet was equipped with a translating centerbody and high-response overboard bypass doors as the control inputs. The shock-position controller was configured as

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shown in the block diagram of figure 4. The purpose of the control design was to mini-mize shock motion caused by downstream airflow disturbances. Thus, it was to function as a shock-position regulator. Classical control design techniques (root locus analysis) were used to arrive at an acceptable inlet-shock-position-regulator control law. The continuous control law arrived at was integral in nature, with some additional lead-lag compensation. The integral control law was first implemented with electronic analog computer components, and experimental frequency response performance was obtained.

Figure 5 shows the open- and closed-loop frequency response of the normalized amplitude ratio of inlet shock position (as measured by a static pressure downstream of the throat) to an airflow disturbance as a function of the frequency of the downstream disturbance. (For brevity, only the amplitude responses are shown.) The solid curve is the open-loop or uncontrolled amplitude characteristic. The amplitude ratio of the shock motion to a downstream airflow disturbance for this and all future frequency re-sponse curves has been normalized to the steady-state, open-loop amplitude ratio. As shown in figure 5, the amplitude ratio responds about 1:1 to about 5 hertz. Beyond this, it starts to attenuate but does display a resonance at 50 to 60 hertz. The closed-loop performance of the continuous integral controller is shown by the dashed curve of fig-ure 5. Low-frequency shock motion is greatly attenuated by the integral control action. Thus, low-frequency downstream airflow disturbances have little effect on shock position. The various system phase lags, however, cause the control action to quit at about 5 hertz. In fact, with the gain selected, the controller actually amplifies shock motion above that of the open-loop or uncontrolled case from 5 to 20 hertz. Beyond 20 hertz, response behaves as if the control had no effect. Assuming most large-magnitude air-flow disturbances to be low frequency in nature, this control behavior is acceptable.

In order to evaluate the effects of using the digital computer system for direct con-trol, the integral-shock-regulator control law was converted to a discrete-time equiva-lent by using Z-transform techniques. The resulting algorithm was programmed into the digital computer control system, and the experimental closed-loop results of fig-ure 6 were obtained. Two different sample rates or control update intervals (1000 sam-ples/sec and 100 samples/sec) were evaluated. The results as shown in figure 6 were not too different from those of the continuous controller, but some slight degradation in response did occur when using only 100 samples per second.

In the future, the advantages of having a digital computer within the control loop may lead to the use of adaptive control techniques. Here the control algorithms may be such that controls gains will be determined on line as the process varies, and the com-plexity of the control gain computation will become of importance. Therefore, the per-formance of a simple finite difference approximation (backward difference method of ref. 2) to the continuous control law was compared with the performance of the more complicated Z-transform algorithm. A frequency response comparison as shown in

figure 7 was made at the slower rate of 100 samples per second. Some degradation does occur with the backward difference approximation, but performance is still tolerable.

Digital Inlet Control (Modern Control)

During the test program just described, efforts were made to study the merits of modern or optimal control theory when applied to the shock-position-regulator problem. This study and the experimental results obtained therein are described in detail in reference 3. A brief summary of the approach taken, as well as some selected results, is contained in this paper.

The design of the shock regulator was begun with the selection of a quadratic performance index which minimized the expected frequency of inlet unstarts created by a random downstream (compressor face) airflow disturbance. The spectral density of this disturbance assumed the majority of the energy to be at low frequencies. A noisy measurement of the sensed shock position was also assumed. The controller structure, therefore, had the optimal regulator - state estimator configuration described by the block diagram of figure 8. The problem was formulated as a continuous controller, and thus a discrete equivalent had to be generated for use with the digital computer controller. The technique by which this was done is discussed in detail in an appendix to reference 3 and is not repeated herein.

The block diagram of figure 9 shows the various elements which comprise the digital computer implementation of the modern or optimal shock-position regulator. The discrete optimal regulator - state algorithm did not permit the system to be sampled less frequently than 1000 samples per second. The sampled-data system became unstable if sampling less than once every 1 millisecond was attempted.

Figure 10 compares the closed-loop frequency response performance for the discrete optimal control with the continuous version implemented with analog computer components. The curves show the normalized amplitude ratio of shock position to the disturbance airflow against frequency. As shown in figure 10, there is very little difference between the analog and digital control performance. The curves show that the low-frequency shock motion is attenuated, which is similar to the integral control action of the classical inlet control design. The optimal regulator has been forced to this type of response by the nature of the spectral density of the disturbance (most of the energy at the low frequencies).

It should be emphasized at this point that the frequency response results of figure 10 are included only to show that a complicated, continuous, optimal regulator - state estimator control law could be discretized for use in a digital computer sampled-data system. The exact discretization demanded the 1000-sample-per-second rate. No effort

was made to develop control law simplifications which would reduce the required sampling rates.

<p style="text-align:center">Digital Turbojet Engine Control</p>

The capabilities of digital computers for turbojet engine control were investigated with a J85-GE-13 engine in a Lewis sea-level test stand. The computer system was programmed to implement the continuous bill-of-material control laws in a discrete fashion. Figure 11 compares the time responses of several engine variables for a throttle step from idle to military using the continuous intact hydromechanical controller with those using the discrete digital computer control. Although only an update interval of 2 milliseconds (500 samples/sec) is shown in the figure, identical transient performance was obtained at update intervals to 25 milliseconds (40 samples/sec). Beyond 25 milliseconds, the speed response began to become oscillatory. In figure 11 it can be seen that the digital control (solid curves) responds slightly faster than the hydromechanical controller (dashed curves). This slight difference was determined to be due to some small differences between the nominal control schedules programmed in the computer and the actual cam schedules in the specific hydromechanical controller used.

Using the Lewis digital system, the controller could sample measured control variables, compute the control algorithm using these sampled measurements, and output commands in an elapsed time of about 1.4 milliseconds (1400 μsec). The sequence of operations is diagramed in figure 12. The sequence was initialized with a priority interrupt from an interval timer. As shown in the figure, the total control computation takes approximately 1.408 milliseconds (1408 μsec). If the system is updating every 2 milliseconds (2000 μsec), there will be 0.6 millisecond (600 μsec) of idle time available between interrupts. At a 25-millisecond update interval though, the computer would be busy only 6 percent of the time. This "spare time" might be necessary if the computer would also be required to compute a complicated inlet control law and to update the inlet every 5 milliseconds or so.

Looking ahead then, to the time when the computer might be asked to do many more on-line, real-time tasks other than inlet and engine control, methods for reliably extending control update intervals were investigated. The curves \of figure 13 show some of the results of this investigation. A prediction algorithm was selected and applied to the sampled measurements. This technique permitted engine transient performance at 150-millisecond updates to closely match the 2-millisecond performance without prediction. Essentially, the prediction algorithm uses the present measurement and past measurements to determine the trend or direction in which particular measured variables are headed. It can then predict what the variable might be at some time during the

interval. It then uses the predicted value at some selected instant within the interval to compute the controller inputs to the engine. The complete details of this engine digital controls research activity are documented in reference 4. Also included is a complete description of the experimental equipment needed to accomplish electronic engine control.

Fail-Operational Digital Engine Control

A digital engine control study was carried out in the sea-level test stand to utilize the extensive computational and decision making potential of the digital computer to perform new control functions not attainable with state-of-the-art hydromechanical controllers. The concept studied was termed fail-operational control. Its purpose was to develop a controller able (1) to detect failures in certain specific sensed engine measurements, (2) to adapt to these failures, and (3) to continue to provide engine operation with as little performance degradation as possible.

In this first attempt at implementing a fail-operational control, only the sensed measurements of engine rotor speed and compressor-discharge static pressure were considered as candidates for possible failure. These are two primary measurements used in the J85-GE-13 bill-of-material control law. The fail-operational system was designed to operate with either or both of the two sensors failed. This investigation is described in reference 5. A brief description of the system and the experimental results obtained using it are contained in the next few paragraphs.

The basis of the fail-operational control is the fact that the compressor-discharge static pressure p_3 and engine rotor speed N are very strongly dependent on one another because of the inherent cycle characteristics of the turbojet engine. Figure 14 is a plot of this relation during normal steady-state operation as well as during accelerations and decelerations. The data are for speeds from idle to military (full 100 percent speed) and were taken at sea-level static conditions. The computer was programmed to store a representation of this characteristic in memory for use during a fail-operational control condition.

A generalized block diagram of the fail-operational control is presented in figure 15. The sensor measurements of engine rotor speed N and compressor-discharge static pressure p_3 are brought into the computer controller through its normal sampling mechanism. Before the sampled measurement is used in the normal engine control algorithm, however, a failure-detection algorithm is applied to each. If a failed sensor is detected, for pressure p_3 for instance, the control logic will switch from the incorrect measured value of p_3 to a stored value of p_3 representative of compressor-discharge exit static pressure at the speed at which the engine is operating. The normal engine

control algorithm will then be exercised, using the unfailed speed measurement and the modeled p_3 pressure value. The control will also use the throttle-input and compressor-face temperature and pressure measurements.

As shown in figure 14, the speed-pressure characteristic is double and even triple valued at the high end. This characteristic is due to exhaust nozzle motion caused by the turbine-discharge-temperature override control loop, which is standard on the J85-GE-13 engine. In order to avoid this multivalued condition and to be able to put realizable characteristic functions into the computer memory, the normal bill-of-material control was modified slightly. A limit was imposed on the minimum allowable exhaust nozzle area such that the temperature override would not be activated. Also, a limit was placed on the maximum throttle position that the control would accept. Admittedly, these limits sacrificed some thrust capability, but they did permit a straightforward approach to computer modeling and storage of the engine speed-pressure characteristic. The actual data that were tabulated in memory for use in the fail-operational control are shown on figure 16. Data of speed against pressure and pressure against speed are redundant information, but both are stored in the computer to simplify the retrieval from memory.

One of the innovations of the computer algorithms developed in this fail-operational investigation is that a self-teaching feature was developed for modeling the speed-pressure characteristics. The control was designed to start from a crude generalized engine characteristic for a J85-GE-13. Then, with all sensors operating, the control could teach itself the exact data for the engine being controlled. The system would require both a slow and a fast throttle transient to generate, in memory, data similar to those in figure 16. In this way the system operated with actual engine characteristics rather than precalculated nominal or average values for the whole family of J85-GE-13 engines. A detailed discussion of the fail-operational control is given in reference 5.

Figure 17 shows the response of the engine rotor speed for a throttle step from idle to military power setting under both normal control and fail-operational control. Cases for either a compressor pressure or engine speed sensor failure are shown. (Note that speed information for this data is obtained from speed instrumentation distinct from the control speed sensor whose failure is being simulated.) In these curves it was assumed that the individual sensor failures were detected prior to the start of the transient. Transient responses in which the individual sensor failed during the transient were also taken and operation was identical to figure 17. For either type of failure, correspondence to normal control is good. However, because throttle limits were built into the algorithm, the speed under fail-operation control does not quite reach full military speed. Likewise, the thrust response curves of figure 18 show that thrust also is limited by a small amount at either condition of fail-operational control. This limitation is due to lower speeds and the fact that the exhaust nozzle area was prevented from going fully

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closed to avoid the turbine-discharge-temperature override which would modulate the area.

Simultaneous failures of both the speed and pressure sensors were accommodated in the fail-operational control. Detection of a double sensor failure put the normal engine control into a throttle-rate-limit mode. In this mode, regardless of the throttle input from the outside, the controller would schedule engine operation per a selected rate of change of throttle position until the final throttle input was achieved. Figure 19 shows the response under the double-sensor-failure condition. Speed response from idle to military is about 30 seconds. The throttle rate limit was selected conservatively to demonstrate computer control capability. No attempt was made to optimize the response under double failures.

Inlet-Engine Digital Integrated Control

The Lewis Digital Computer Propulsion Control Facility was most recently employed in the control integration of an experimental supersonic mixed-compression inlet and a TF-30/P-3 afterburning turbofan engine. This experimental program is the subject of a paper to be presented at this session by Mr. P. Batterton and therefore is not discussed herein. A detailed discussion of that work is contained in reference 6.

Present Activities

At the present time the computer facility is being employed to study digital control of advanced turbofan engines. The engines presently are simulated in real time on the Lewis hybrid computer system, and new modes of control using a digital computer are being evaluated. These are complete wide-range simulations of operation at many altitude and Mach number conditions. In conjunction with this effort, experimental programs in the altitude test facilities are being planned to verify the control concepts being studied.

CONCLUDING REMARKS

If we consider the Lewis Digital Computer Propulsion Control Facility as a tool for propulsion control mode studies regardless of the future type of hardware implementation of the control, the past 4 years have been highly successful. The ability to assess control concepts and then simply modify software to investigate other control approaches has greatly expedited our research activities.

If we consider the digital facility as a predecessor of the type of control hardware that will actually be available for future operational propulsion systems, much valuable information has been learned. First, a modern digital control computer operating as a sampled-data system can definitely perform the control task for a complete supersonic airbreathing propulsion system. Modern computers are certainly fast enough and, equipped with sufficient memory capacity, can perform tasks previously considered impracticable.

Much more work remains to be done to ensure that new digital control laws can be reliably executed under the ever-changing requirements and conditions that a propulsion system encounters in an operational flight application. Some of these potential problems can, with careful planning, be attacked in ground test facilities and with sophisticated simulations. The utilization of the Lewis Digital Computer Propulsion Control Facility is directed toward this end. Other problems will have to be solved with flight programs such as the F-111 IPCS and YF-12 cooperative control activities.

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6. Batterton, Peter G.; Arpasi, Dale J.; and Baumbick, Robert J.: Digital Integrated Control of a Mach 2.5 Mixed-Compression Supersonic Inlet and an Augmented Mixed-Flow Turbofan Engine. NASA TM X-3075, 1974.

TABLE I. - SIGNAL PROCESSING CAPABILITY

	Present complement	Fully expanded
Signal inputs from propulsion system	100	100
Signal outputs to propulsion system	54	54
Analog-to-digital conversion channels	64	64
Period-to-digital conversion channels	10	20
Logical outputs	64	64
Digital-to-analog conversion channels	26	42
External priority interrupt	10	22

TABLE II. - FACILITY SPECIFICATIONS

Digital computer	
Magnetic core memory size, words	16 384
Word length, bits plus parity	16
Memory cycle time, nsec	750
Add time, μsec	1.5
Subtract time, μsec	1.5
Multiply time, μsec	4.5
Divide time, μsec	8.25
Load time, μsec	1.5
Store time, μsec	1.5
Indirect addressing	Infinite
Indexing	Total memory
Priority interrupts	28 Separate levels
Index registers:	
Independent	1
In conjunction with lower accumulator	1
Physical size, cm (in.):	
Width	60.9 (24)
Height	157.4 (62)
Depth	76.2 (30)

Interval timers	
Complement	2
Accuracy, clock pulses	±1
Clock rates, kHz	572, 286, 160, 143, 80, 71.5, 40, 35.75, 20, 10
Counter	16-Bit binary
Output	Priority interrupt to computer

Analog acquisition unit	
Number of multiplexers, digitizers, and sample and holds	2
Overall sample rate (maximum), kHz	40
Resolution of digital data, bits	12 (plus sign)
Output code	Two's complement
Number of channels	64
Input range, V (full scale)	±10
Input impedance, MΩ (shunted by 10 pF)	10
Maximum source resistance, Ω	1000
Conversion time, μsec	38
Input settling time, μsec	9
Sample-and-hold aperture time, nsec	500
Safe input voltages, V	±20 sustained ±100 for less than 100 μsec
Total error with calibration, percent	0.073

TABLE II. - Concluded. FACILITY SPECIFICATIONS

Frequency acquisition unit	
Number of channels	10
Nature of input	Continuously varying or pulsatile
Resolution of digital data, bits	12
Switch selectable clock rates, kHz	20, 80, 100, 400, external
Overall accuracy, bits	±1
Update rate	Once per cycle of input frequency
Maximum input frequency, kHz	1
Input amplitude range	100 mV to 30 V peak to peak

Analog output unit	
Total number of digital-to-analog conversion channels	26
Resolution (10 channels), bits	12 (plus sign)
Resolution (16 channels), bits	11 (plus sign)
Output voltage range, V full scale	±10
Output current (maximum), mA	10
Output impedance, Ω	<1
Accuracy (12 bit), percent of full scale	±0.1
Accuracy (13 bit), percent of full scale	±0.05
Slew rate, V/μsec	1
Settling time for 10-V step to within 0.05 percent of final value, μsec	20

Logical output unit	
Number of electronic switch outputs	32
Number of contact closure outputs	32
Maximum voltage, V	30
Maximum current, mA	100

Priority interrupt processor	
Number of channels	10
Input impedance, kΩ	47
Input voltage range, V	±10
Comparator switching	Trigger on rise or fall
Comparator hysteresis	Adjustable from 35 mV to 650 mV
Comparator output, V	+7
Monostable multivibrator:	
Pulse width, μsec	3
Pulse height, V	+7
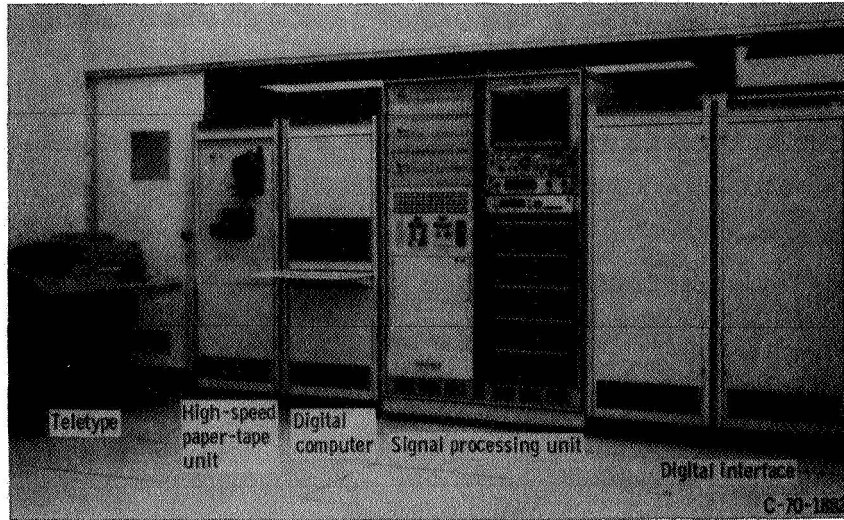
Figure 1. – Digital computer propulsion control facility.
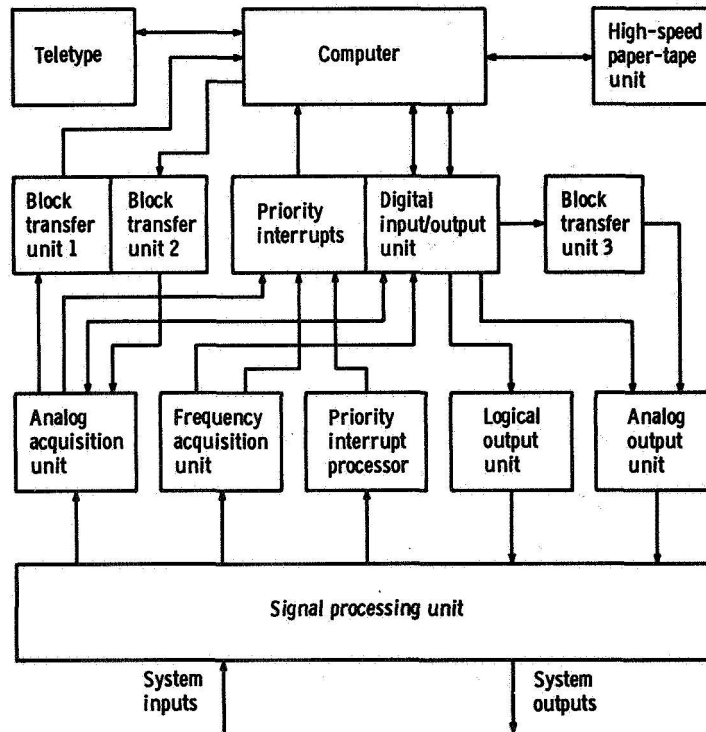
Figure 2. – Block diagram of digital control facility.

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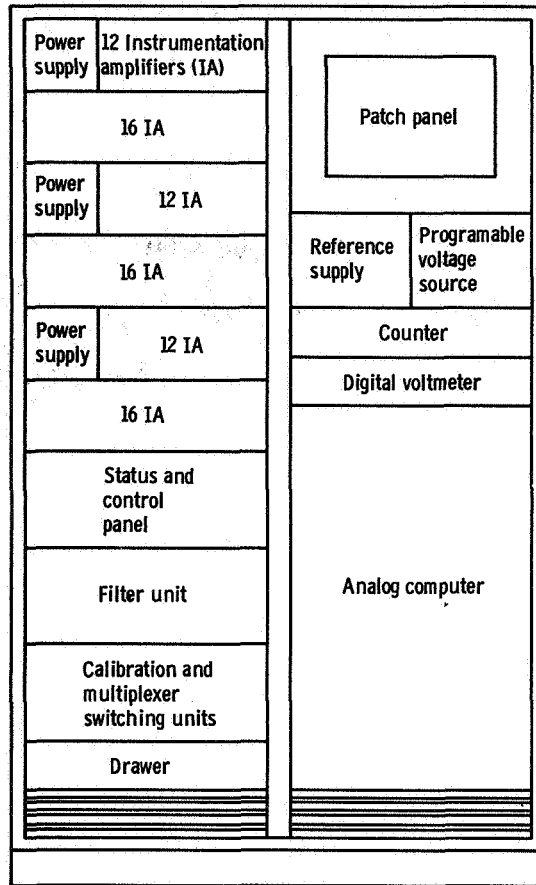
Figure 3. – Cabinet layout of signal processing unit.
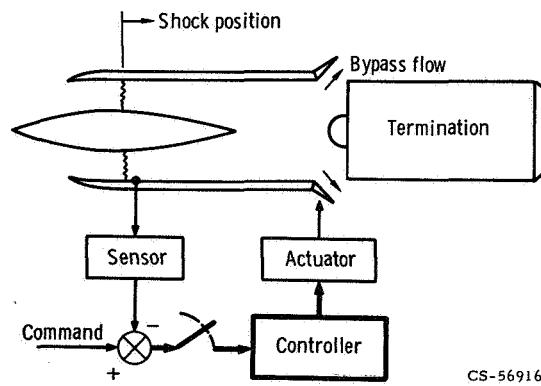
Figure 4. – Block diagram of shock position control system.
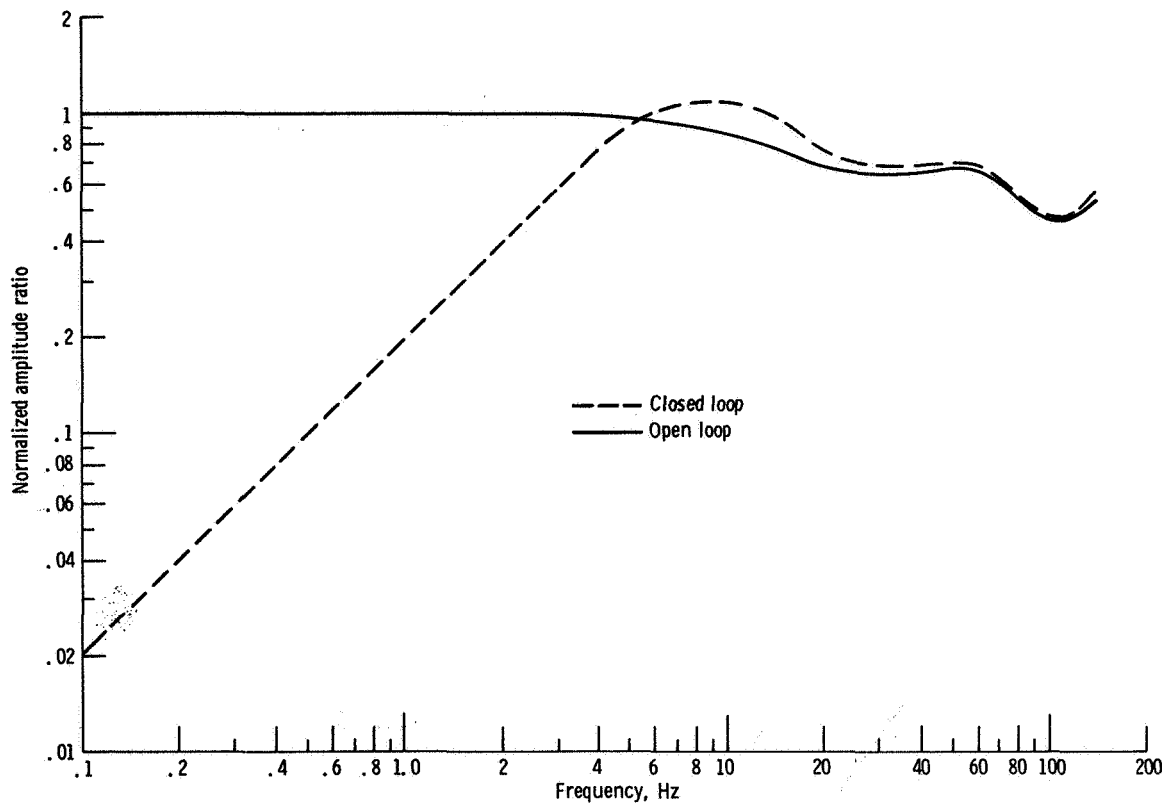
Figure 5. - Comparison of experimental open-loop and closed-loop frequency response of normalized amplitude ratio of shock position to airflow disturbance.
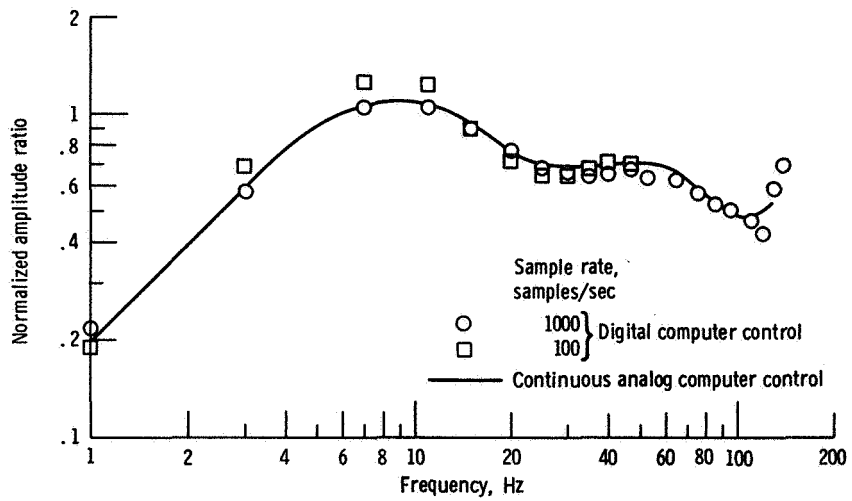
Figure 6. – Comparison of experimental closed-loop frequency response performance using analog computer control and z-transform digital computer control at two different sampling rates.
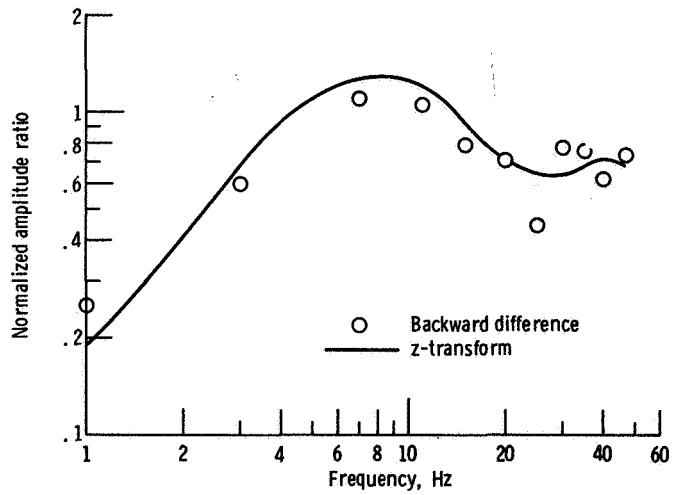
Figure 7. – Comparison of experimental closed-loop frequency response for z-transform and backward difference digital computer algorithms using sample rate of 100 samples per second.

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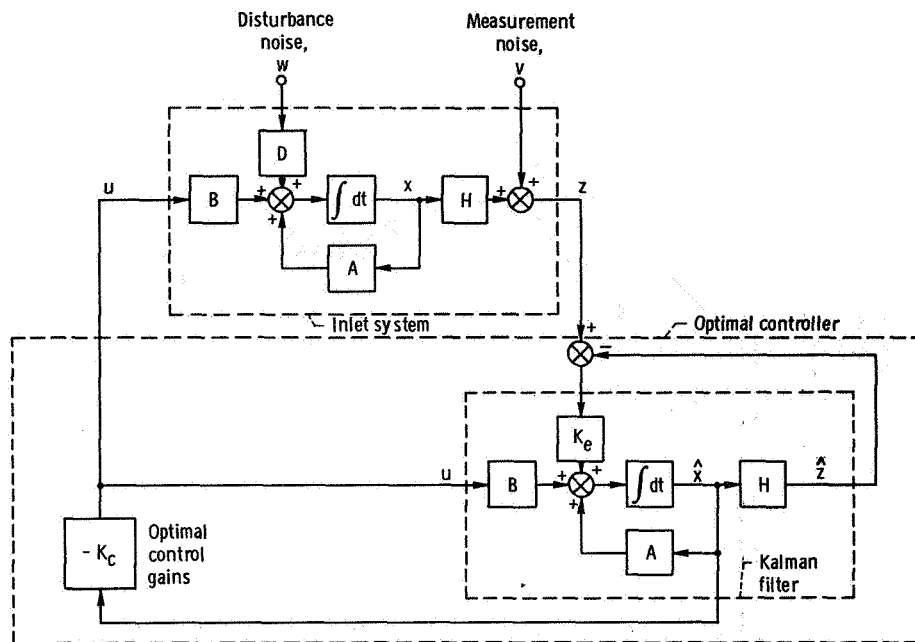
Figure 8. – Block diagram of combined optimal regulator – state estimator. (Symbols used are conventional modern/optimal control notation.)
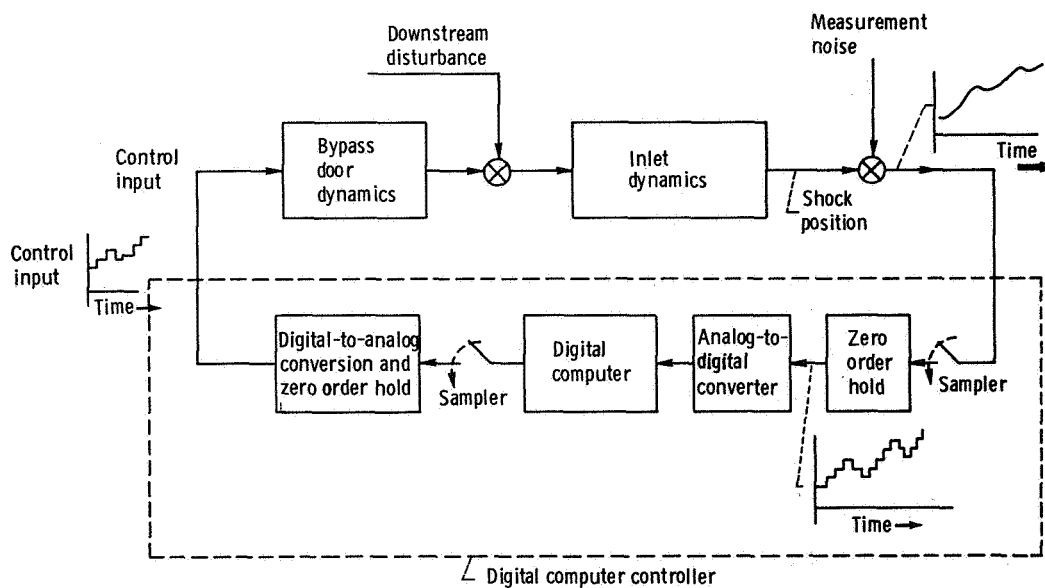
Figure 9. – Block diagram of digital computer inlet control system.
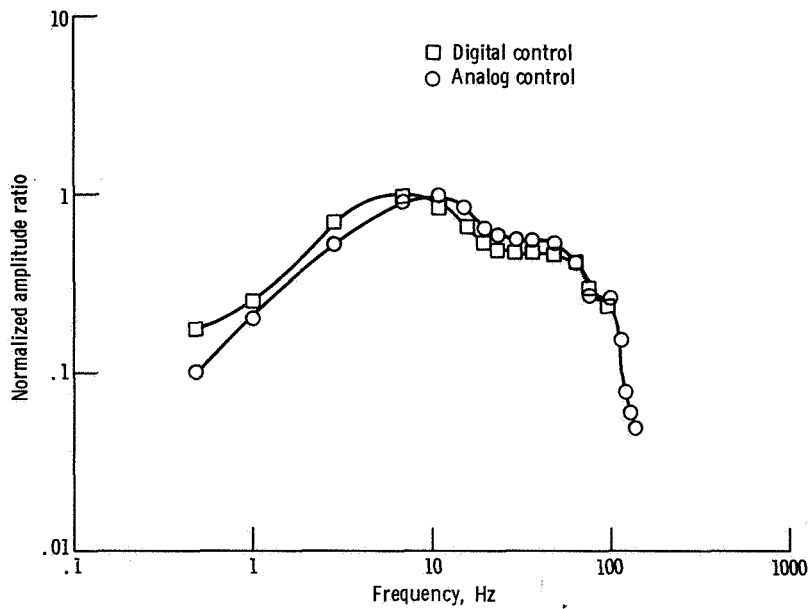
Figure 10. - Comparison of closed-loop experimental frequency responses of shock position to disturbance airflow amplitude ratio using analog and digital computer implementations of optimal regulator - state estimator control law.
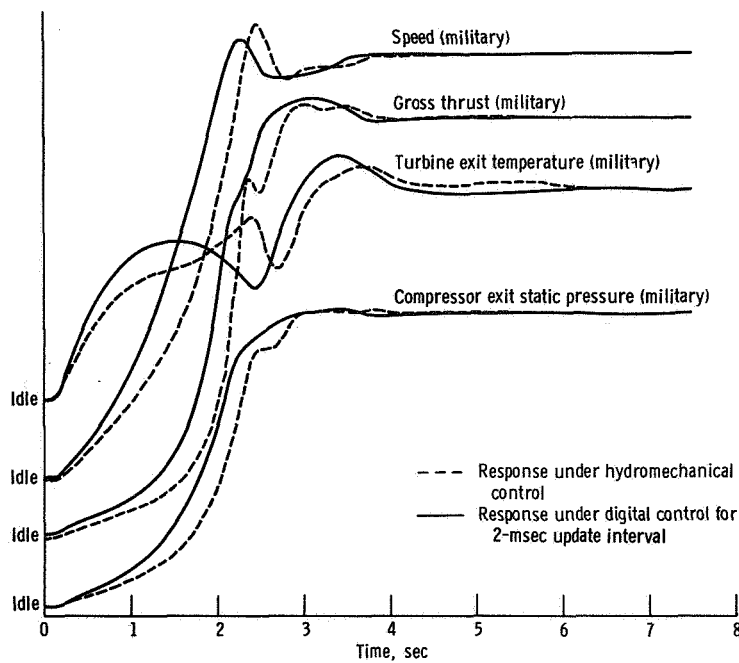
Figure 11. - Comparison of digital and hydromechanical controls for throttle step from idle to military.
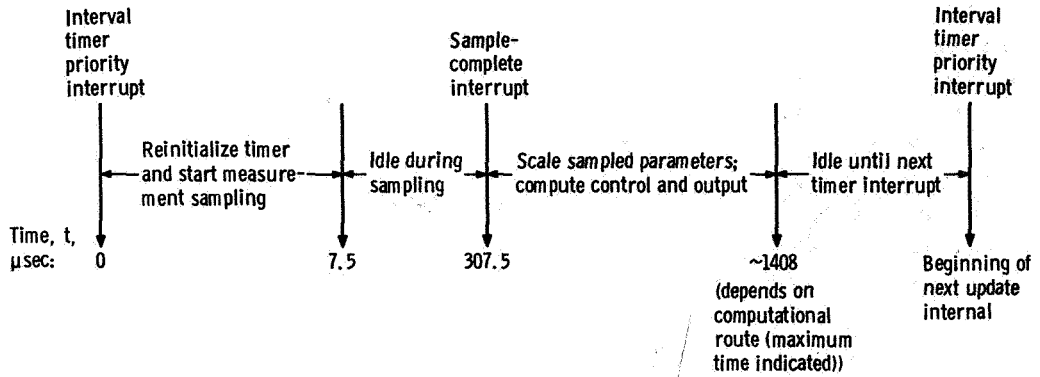
Figure 12. - Timing diagram for typical update interval.
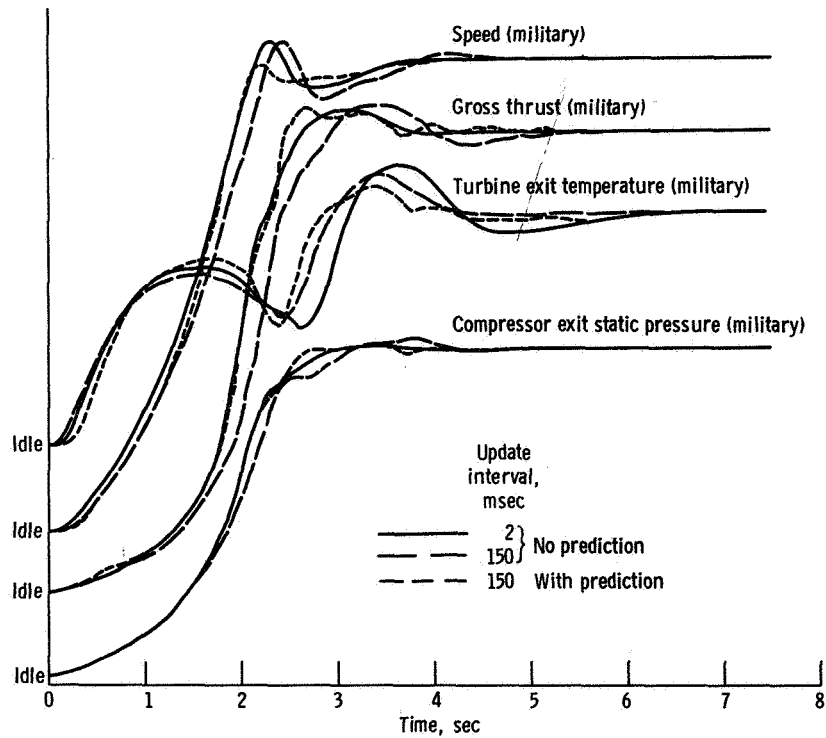
Figure 13. - Effects of prediction on engine response for throttle step from idle to military.

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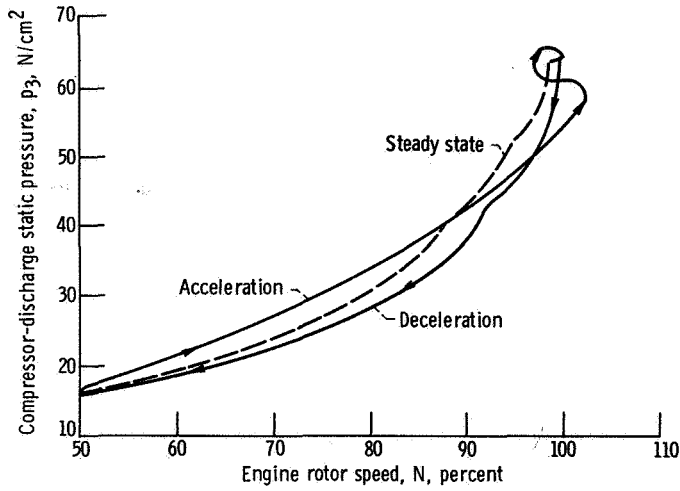
Figure 14. - Compressor-discharge static pressure as function of engine rotor speed for normal digital control.
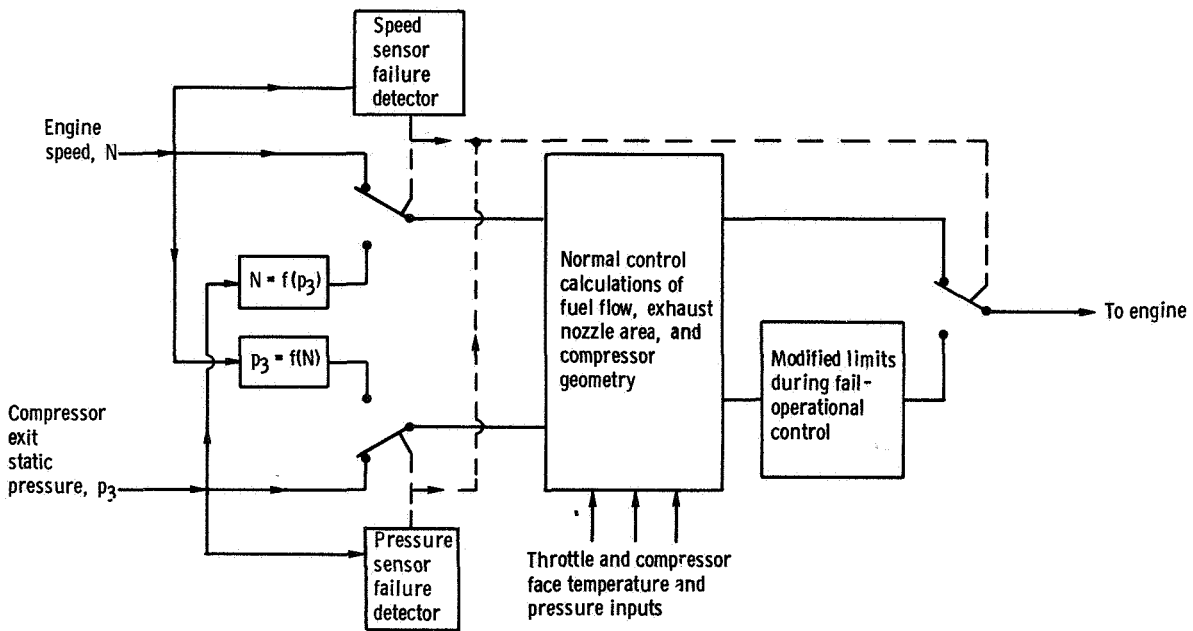
Figure 15. - Block diagram of fail-operational control.
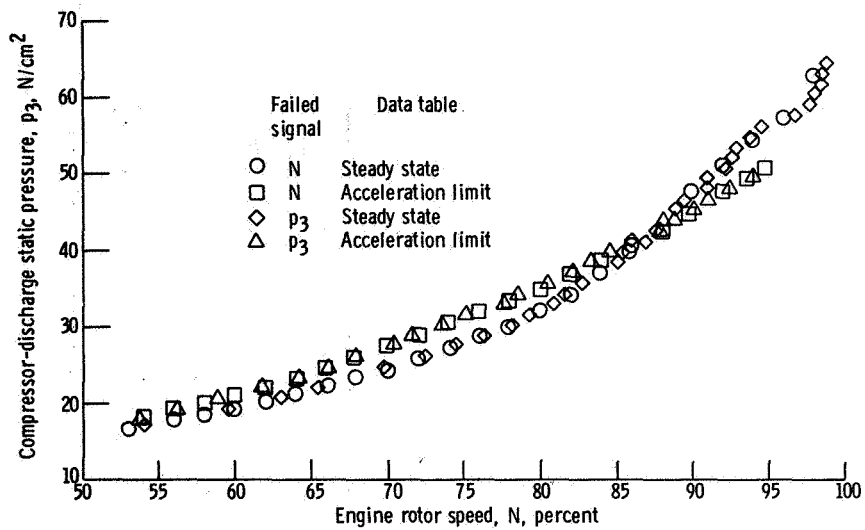
Figure 16. - Plot of tabulated values of compressor-discharge static pressure and engine rotor speed as functions of each other for fail-operational control programs.
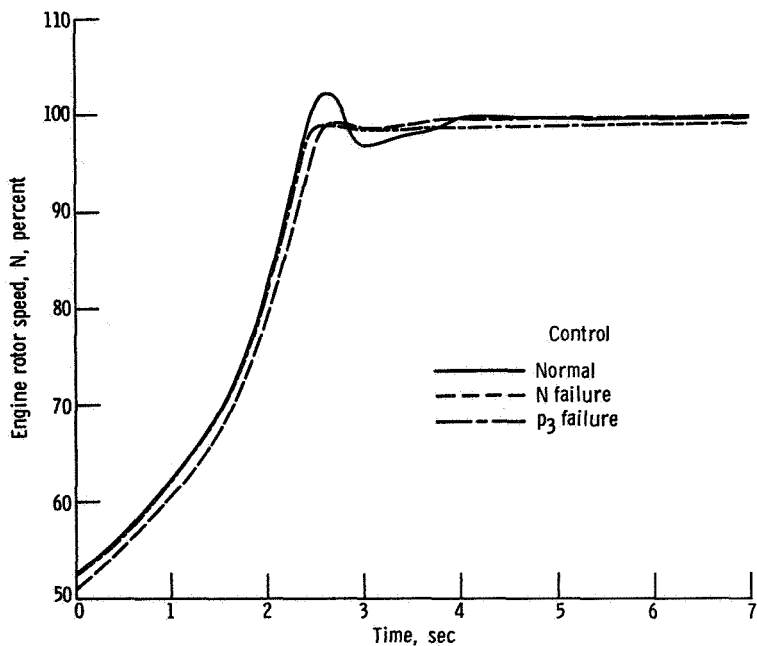
Figure 17. - Step responses of engine rotor speed from idle to 100 percent rotor speed for normal digital control and engine rotor speed and compressor-discharge static pressure p_3 fail-operational controls.
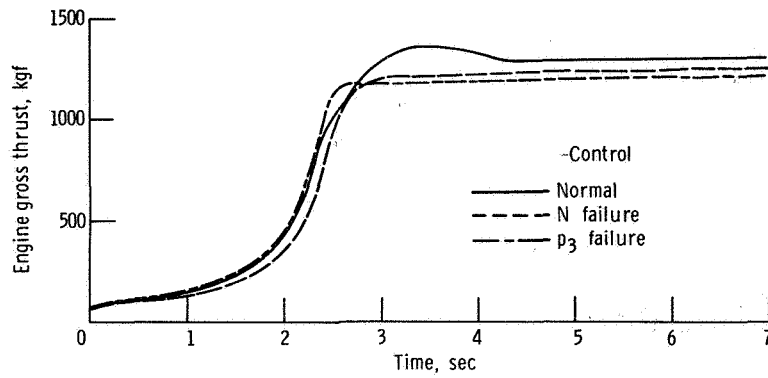
Figure 18. - Step responses of engine gross thrust from idle to 100 percent rotor speed for normal digital control and engine rotor speed N and compressor-discharge static pressure p_3 fail-operation controls.
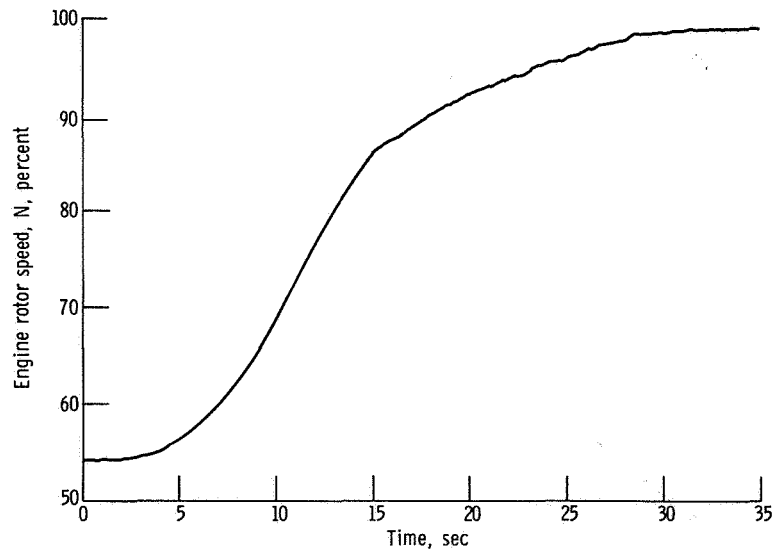
Figure 19. - Step response of engine rotor speed from idle to 100 percent rotor speed for combined engine rotor speed N and compressor-discharge static pressure p_3 fail-operational control.

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PART 2

ADVANCED CONTROL TECHNOLOGY AND ITS POTENTIAL
FOR FUTURE TRANSPORT AIRCRAFT

DRYDEN FLIGHT RESEARCH CENTER
EDWARDS, CALIFORNIA

AUGUST 1976

DESCRIPTION AND TEST RESULTS OF A DIGITAL SUPERSONIC PROPULSION

SYSTEM INTEGRATED CONTROL

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NASA Lewis Research Center

SUMMARY

A digitally implemented integrated inlet/engine control system was developed and tested on a mixed-compression, Mach 2.5, supersonic inlet and augmented turbofan engine. The control matched engine airflow to available inlet airflow so that in steady state, the shock would be at the desired location and the overboard bypass doors would be closed. During engine induced transients, such as augmentor lights and cutoffs, the inlet operating point was momentarily changed to a more supercritical point to minimize unstarts. The digital control also provided automatic inlet restart.

INTRODUCTION

Advanced propulsion systems such as those found in the B-1, F-14, and F-15 are quite complex. As future supersonic transport aircraft are designed, the propulsion systems of those aircraft will require control systems even more complex than those found in current aircraft. Some aspects of these supersonic propulsion system control problems are discussed in references 1 to 4. This increase in complexity has led to an upsurge in interest in digital controls for advanced propulsion systems because of the inherent flexability of the digital computer.

There has been little actual experience with the combination of a turbofan engine and mixed-compression supersonic inlet. This is the first experimental test in the United States to study the interactions of such a system, and to determine its controlability. Several difficulties can arise from the use of such combinations in the area of overall reliability and efficiency, and in providing sufficient stable operating range for the inlet while minimizing the probability of engine stall.

There are several inter-related control problems for this engine and inlet. Any changes in augmentor operation will result in temporary changes in fan airflow. An ex-

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ample of an airflow disturbance for an augmentor light-off is shown in figure 1. This is a result of the fact that, at the high Mach numbers, the fan operating point is generally at a low corrected speed on the fan operating map. Relatively large airflow changes can therefore result from small changes in fan pressure ratio. These airflow changes can cause inlet unstarts if of sufficient magnitude and if the rate of change is outside the control bandwidth of the inlet terminal shock control system. An inlet unstart that was caused by an augmentor light-off transient is shown in figure 2. The unstart causes a rapid dropoff in the fan inlet pressure with a corresponding drop in propulsion system thrust. This occurs while the terminal shock is being expelled from the inlet. The engine compressor is also stalled by this pressure disturbance. Also shown in figure 2 is the indicated turbine inlet temperature. This temperature shows approximately 15 percent increase over the initial value with the inlet unstart and engine stall. This could possibly overtemperature the engine in actual flight, but that did not occur during testing because of low engine inlet temperatures. During the restart of the inlet, high distortion is generated. This may cause a second engine stall and this is shown in the figure. The unstart problem would be less apt to occur if the inlet terminal shock could be positioned further downstream in the inlet throat providing greater margin against an unstart. But this results in poor inlet pressure recovery (poor efficiency) and greater distortion at the engine.

This project was undertaken to determine the nature of a control system such that the aforementioned problems could be avoided or at least minimized and at the same time minimize overboard spillage of inlet capture air. The approach taken was to tie together the inlet, engine, and augmentor control systems in an appropriate manner so as to: first, in steady state, monitor inlet shock position and overboard bypass door command and adjust engine airflow to match available inlet airflow; second, for augmentor induced transients, provide information to the inlet and engine control systems which can be used to prevent inlet unstart and/or engine stall; and third, provide automatic restart should the inlet unstart. Since there was considerable logic involved in this type of control system and since much experience has already been gained in the use of digital computer control of supersonic inlets (ref. 5) and of an augmented engine (ref. 6), it was decided that this control would be implemented on a digital computer. This is the first attempt at simultaneously controlling both a supersonic inlet and engine with the same digital computer. It was also felt that this choice would provide support material for the Air Force-NASA cooperative digital integrated control flight program (ref. 7) now under contract.

N_H high rotor speed, rpm

N_L low rotor speed, rpm

P   total pressure, N/cm^2

p   static pressure, N/cm^2

T   total temperature, ^{o}C

W_f fuel flow, kg/sec

θ (T + 273. 15)/288. 15

Subscripts

eng   main engine

zone1   zone 1

zone2   zone 2

0. 5   inlet cowl lip station

1   inlet geometric throat station

1. 1   inlet throat exit station

2   engine fan inlet station

2. 2   engine low-pressure compressor discharge station

3   engine high-pressure compressor discharge station

4   engine high-pressure turbine inlet station

5   engine low-pressure turbine discharge station

APPARATUS AND PROCEDURE

Testing of the digital integrated control was conducted in the Lewis 10- by 10-Foot Supersonic Wind Tunnel. The propulsion system was composed of a mixed-compression inlet coupled to a dual rotor turbofan engine. Figure 3 shows the system installed in 10- by 10-Foot Supersonic Wind Tunnel. Table I lists the average tunnel free stream conditions. A brief description of the inlet, engine, and computer are provided in the following sections. Complete descriptions of the inlet, engine, and computer are provided in reference 8.

Inlet

The Lewis designed inlet is an axisymmetric, mixed-compression inlet with translating centerbody and 45 percent internal supersonic area contraction. The inlet is designed for Mach 2. 5 operation with a TF30 engine. The inlet has a capture area of 0. 707 square meters and measures 180 centimeters from the cowl lip to the fan face. The inlet is equipped with eight slotted plate bypass doors which are used to position the inlet terminal shock.

Engine

The engine used in this investigation is a Pratt and Whitney TF30-P-3. The TF30-P-3 is an axial, mixed-flow, augmented, twin spool, low bypass ratio turbofan engine with a variable area convergent primary nozzle. The engine includes a three-stage axial-flow fan mounted on the same shaft with a six-stage axial-flow low-pressure compressor. This unit is driven by a three-stage low-pressure turbine. A seven-stage axial-flow compressor driven by a single-stage, air-cooled turbine makes up the high-pressure spool. The compressor is equipped with 7th stage (low-pressure compressor) and 12th stage (high-pressure compressor) bleeds. The 7th stage bleed is operated by aircraft systems and the 12th stage is normally operated automatically by the engine control system. The 12th stage bleed was set closed for this test.

The augmentor consists of a diffuser section, five concentric ring fuel manifolds (zones), three V-gutter ring flame holders, a combustion chamber liner, and a fully modulating flap-type convergent primary nozzle. Variable thrust augmentation is accomplished by adjusting fuel through the fuel manifolds. Augmentor ignition is by means of two "slugs" of fuel injected into the engine gas stream, one upstream of each turbine. This "hot streak" continues aft and ignites the augmentor fuel. The augmentor zones are turned on sequentially, each reaching a predetermined level before proceeding to the next. Note that only the first two zones of the augmentor were actually used for this test program.

The standard TF30-P-3 fuel control systems consist of a hydromechanical main fuel control (MFC) and a hydromechanical combined augmentor and exhaust nozzle control (A/B-ENC). A single power level commands the MFC and A/B-ENC. However, to provide access to the augmentor at the nonstandard wind tunnel conditions, the A/B-ENC was completely removed from the engine and replaced with servocontrolled throttles for fuel flow control, a position servo for the exhaust nozzle, and solenoid valves for generation of the logic signals used by the augmentor ignitor.

Integral with the MFC is a so-called "Weapons Derichment Port" to which for some

engine installations an electrically operated valve is connected to allow derichment of fuel during the firing of aircraft weapons. A servocontrolled throttling valve was attached to this port to allow the bypassing of fuel. By setting the power lever angle (PLA) to a high enough value, the MFC computer would provide acceleration fuel flow. The excess fuel could then be bypassed and engine speed regulation obtained externally of the MFC. The hydromechanical control could then be used for startup as well as emergency procedures during the tunnel operation.

Instrumentation

Sixteen steady-state transducers were used to measure the inlet terminal shock position. These transducers start at a distance of 23 centimeters from the cowl lip and extend to a point 66 centimeters from the cowl lip. The last two transducers were located 5.08 centimeters apart while the others were located 2.54 centimeters apart. The dynamic pressures were measured with strain-gage-type transducers connected to the cowl with short tubes. The frequency response of this pressure measuring system had negligible dynamics in the range covered in these tests (0.1 to 100.0 Hz).

There are four dynamic transducers located 66 centimeters from the cowl lip and positioned 90° apart circumferentially around the cowl. These static pressure signals were electrically averaged and identified as $p_{1.1}$. In addition to these transducers, dynamic transducers were included to measure total and static pressure at the geometric throat (P_1 and p_1, respectively), and static pressure near the cowl lip $p_{0.5}$.

All engine pressures used engine supplied probes; that is, the p_3 signal comes from the pressure signal tube going to the MFC. All pressure signals were sensed by strain-gage-type pressure transducers. The fan inlet temperature T_2 was sensed by a thermocouple, but the high-pressure turbine inlet temperature T_4 is the Pratt and Whitney supplied signal which is based on the temperature rise across the compressors and the low-pressure turbine discharge temperature. The low-pressure rotor speed was sensed by a magnetic pickup and gear located in the "bullet nose". The high-pressure rotor speed was sensed by a magnetic pickup and gear located on the gear box. All fuel flows were measured by turbine flowmeters. The two speeds and the fuel flows were converted to high level analog signals for use by the digital integrated control and recording equipment. The nozzle exit area and the compressor bleed positions were obtained from potentiometers.

Digital Computer

The digital integrated control was implemented on a digital computer, located at the analog computer facility in a building approximately 500 meters from the test facility. It was connected to the test facility via land lines with ground isolation amplifiers at the receiving end of each line. A small desk-top-size 10-volt general purpose analog computer was also used for signal conditioning and biasing of both sensed model parameters and returned control commands. The analog computer was located at the test facility. The digital system consists of four major units.

(1) A digital computer with 16 384 words of memory, a read-restore memory cycle of 750 nanoseconds, and a word length of 16 bits.

(2) A digital interface capable of converting both analog and frequency signals to computer compatible digital words and converting computer generated words to analog and logical outputs.

(3) A signal processing unit which provides signal conditioning and monitoring capability between the digital interface and the propulsion system to be controlled.

(4) Programming peripherals consisting of a high-speed paper-tape reader and punch, and a teletype.
The capabilities of the system are given in table II and a comprehensive description is available in reference 9.

Procedure

The inlet, engine, and control system were tested at zero angle-of-attack. No angle-of-attack data were obtained. At angle-of-attack the inlet control requires more shock position instrumentation than was provided in this inlet. The mass flow delivered to the engine was varied by adjusting the amount of airflow bypassed by the disturbance doors of the inlet. This would allow the observance of the behavior of the system to steady-state, step, and sinusoidal disturbances in airflow. The inlet was unstarted by momentarily reducing the inlet throat bleed until the throat Mach number dropped low enough for the inlet to unstart. Behavior of the control to unstarts could then be determined. For testing of the augmentor control, the primary method of disturbing the control was the PLA. Step changes in the PLA were used. To get in condition for these tests, the engine would be started with the hydromechanical MFC while the inlet was controlled using an electronic analog control.

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CONTROL DESCRIPTION

The goals of the integrated control are summarized as follows. By matching engine airflow to available inlet airflow, inlet pressure recovery is maximized and spillage airflow is minimized, and both these effects usually maximize inlet performance. This is the primary goal of the digital integrated control and is our definition of an airflow match between the engine and inlet. The TF30-P-3 is a turbofan engine, and the bypass ratio of the fan varies, depending on conditions, from about one to two. Augmentor transients such as zone lights and cutoffs disturb the fan airflow directly and these disturbances propagate up into the inlet relatively unimpeded when compared to turbojets. Therefore, the second goal of the control is to provide a more stable operating point while attempting augmentor transients. The last goal for the control is to provide automatic inlet restart should an unstart occur.

A description of the basic inlet and engine control systems is provided in the next section followed by a brief description of the integration of these controls to achieve the aforementioned goals. A more detailed description of the integrated control is provided in reference 8.

Basic Control Functions

There are three basic control functions for this mixed-compression inlet and augmented turbofan propulsion system. These are: (1) inlet terminal shock and restart control, (2) engine rotor speed regulation and fuel flow limiting control, and (3) augmentor and exhaust nozzle control. A brief explanation of each of these control functions follows.

The basic control problem of a mixed-compression supersonic inlet is that of maintaining the terminal shock in the throat to maximize inlet pressure recovery but not allowing the inlet to unstart (allowing the terminal shock to be expelled from the inlet). The usual method of control is to manipulate overboard bypass doors to bypass inlet airflow which in turn positions the terminal shock. By increasing bypass airflow, the shock is pulled downstream in the inlet throat and the reverse occurs if bypass airflow is decreased. Thus a control which senses shock position is used to drive the overboard bypass doors.

The second part of the inlet control is that of starting the inlet. Starting is defined as causing the externally located terminal shock to enter the throat region of the inlet. (The inlet is unstarted when the terminal shock is located forward of the cowl lip.) Starting is accomplished by increasing the ratio of throat area to capture area until the throat goes supersonic, and extending the spike increases ratio of the throat area to capture area for this inlet. Once started, the spike returns to its design position. The started (or unstarted) condition is detected by the presence of supersonic (or subsonic)

airflow at the cowl lip.

For the TF30-P-3, speed regulation is obtained normally by using PLA to schedule a high rotor speed reference in the MFC. The speed reference and actual speed are used in a droop governor to provide a ratio of fuel flow to burner pressure which, when multiplied by burner pressure, determines fuel flow to the engine. Speed regulation is obtained in that manner. The MFC also limits maximum fuel flow during acceleration to avoid turbine inlet overtemperature and/or compressor stall, and limits minimum fuel flow during deceleration to avoid combustor blowout and/or compressor stall. The MFC also provides operating point information (high rotor corrected speed) to the augmentor/exhaust nozzle control, and a signal from the augmentor control indicating that an augmentor blowout has occurred. The augmentor blowout signal causes the MFC to switch from the speed governor to a special fuel flow schedule. This fuel flow schedule reduces fuel flow to the engine to avoid overspeeding the low rotor.

The augmentor control uses PLA to command a level of augmentor fuel flow and to determine which zones should be lit. The zone fuel flow schedules are also ratios of fuel flow to burner pressure schedules because burner pressure is used as a measure of engine core airflow. Thus changes in engine bypass ratio are taken into account to bias those augmentor zones which are in the fan duct airstream. The exhaust nozzle is positioned to drive the error in the MFC determined fan operating point to zero. The fan operating point schedule is a ratio of burner pressure to turbine discharge total pressure p_3/P_5 as a function of high rotor corrected speed. The rate of change in P_5 is used to indicate that the first zone of the augmentor is lit or that an augmentor blowout occurred.

Control Integration

The inlet and engine are defined as being matched when the shock is at the desired location and the bypass doors are closed. Therefore a signal can be generated which could tell the engine to increase speed (and thus airflow) if more airflow is available and conversely if less airflow is available. It is this type of scheme which was developed to satisfy the primary goal for the integrated control. The overall integration loops are shown in figure 4. The nonaugmented engine operation will be discussed first. The airflow match signal is defined as the bypass door command signal, or, if the bypass door command is zero, the shock position error signal. This signal is used to drive a proportional plus integral control which produces a shift in the high rotor speed demand to the engine speed governor. Note, PLA normally generates the base speed demand schedule.

During augmentation, the exhaust nozzle is also available to adjust engine airflow. Therefore, during augmented operation, the airflow match error signal biases the fan operating point schedule. This is done in such a manner as to cause the exhaust nozzle

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to open more than normal if the bypass doors are open or to close more than normal if shock is supercritical. This action was made proportional to allow the integrator in the speed demand shift logic to reset the airflow match signal to zero by adjusting engine speed. This allows the nozzle to return to its normal schedule, which is desirable since significant changes in the fan operating point can lead to engine stall.

The aforementioned scheme will operate successfully except during augmentor transients and inlet unstart-restart. Therefore, additional logic signals were used for these special cases.

Augmentor transient signals are generated by the augmentor/exhaust nozzle control to tell the engine and inlet controls that engine induced airflow transients can be expected. The signals are simply logic signals that indicate whether or not the augmentation level has reached that commanded by PLA. One augmentor transient signal sets the airflow match signal to zero which causes the speed demand control to hold its present value until the transient is over.

Since the response of the bypass door control may not be capable of handling the augmentor induced airflow transient, another augmentor transient signal is used by the inlet control to command the shock to a more supercritical location appropriate to the expected airflow transient. Because the speed demand is held constant, the engine speed will not change and the bypass doors will open to move the shock to the more supercritical location. Having both the shock positioned supercritically and the bypass doors partially open is desirable when large airflow transients are expected from the engine.

The augmentor blowout signal causes the MFC to switch from the speed governor to a special fuel flow schedule. Thus, during this time, the speed demand is placed in the hold mode.

The unstart-restart signal for the engine is the unstart-restart signal used by the inlet control except that, as far as the engine is concerned, the restart is not complete until the spike has returned to its design point. The unstart-restart signal causes the value of the shift in speed demand to be reset to zero. The augmentor/exhaust nozzle control uses the unstart portion of this signal to cause an automatic shutdown of the augmentor. This is based on the assumption that the engine will stall when the inlet unstarts and it is felt that the augmentor should be turned off with engine stalls.

The digital integrated control briefly described here and more completely in reference 8 used no additional sensed inlet or engine variables than would be used for the conventional controls.

RESULTS AND DISCUSSION

The control described in this report was tested with three inlet configurations. The results presented here are for the inlet configured with 10-hertz bandwidth inlet overboard bypass doors. By 10-hertz bandwidth, we mean that the position servo frequency response of the overboard bypass doors exhibited a first-order rolloff at approximately 10 hertz. Two other configurations were tested. These were one with 80-hertz bandwidth bypass doors and one with 10-hertz bandwidth bypass doors and with a controlled variable bleed at the inlet throat. The results of these latter two configurations may be found in reference 8. In discussing the results of the 10-hertz bandwidth bypass doors, the major differences in the results of the other configurations will be mentioned.

The digital control sampled the inlet variables and calculated the inlet control output commands once every 5 milleseconds. The engine variables were sampled and the engine control output commands calculated once every 50 milleseconds. These sample times are representative of the differences in the dynamics of the inlet and engine. Details of how the computer functioned with the different time steps and shared the same multiplexer are described in reference 8.

Inlet terminal shock position could not be dynamically measured directly in the inlet. Therefore, a throat exit static pressure signal was obtained which could be used as a dynamic measure of shock position for feedback to the control. The relation of this pressure $p_{1.1}$ to shock position is shown in figure 5.

Figure 6 shows the action of the control to a square-wave-type disturbance of inlet airflow. The magnitude of this disturbance was 0.85 percent peak-to-peak of the engine total corrected airflow of 68 kilograms per second. At the step closing of the disturbance bypass doors, the control bypass doors step open to correct for the error in shock position $p_{1.1}$. Closing the disturbance bypass doors increases the airflow available to the engine. The engine speeds then increase to allow the control bypass doors to close. With the control gains that were used, this process was underdamped and the speeds would overshoot momentarily pulling the shock to a slightly supercritical position. When the disturbance bypass doors open, the available airflow to the engine is reduced. Thus at the step opening of the disturbance bypass doors, the shock was pulled to a slightly supercritical position. Engine speeds then reduce allowing the shock to return to the desired position. The system is still underdamped, but less than for the step closing of the disturbance bypass door transient. The control thus was able to match engine airflow to available inlet airflow and achieve the result of no overboard bypass airflow in steady state while maintaining the shock at the desired position. This was the first goal of the integrated control.

The second goal of the control was to minimize inlet unstarts during augmentor transients. Figure 7 is an augmentor transient from light-off to maximum zone 2. As the PLA is advanced into augmention, the control commands the shock to a more supercriti-

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cal position (lower $p_{1.1}$) in anticipation of the augmentor light-off disturbance. In this case, $p_{1.1}$ was reduced from 5.50 to 4.97 newtons per square centimeter. This positions the shock 17.5 centimeters downstream of the throat instead of the 3 centimeters during normal operation. The control bypass doors open to achieve this result. The engine speeds were not allowed to reset the control bypass doors during the augmentor transient. Total fuel flow shows the increase in fuel flow as the first augmentor zone starts flowing. The augmentor does not light-off right away since the fuel is filling the manifolds. The large jump in the turbine discharge pressure P_5 indicates that the augmentor has lit-off and the exhaust nozzle is released. The exhaust nozzle slews open to reduce the error in the ratio of p_3 to P_5 and thus maintains the engine at the desired operating condition. As the error in p_3/P_5 is reduced, the augmentor fuel flow is allowed to increase to maximum zone 1. The second step on the total fuel flow trace is the fuel flow for the second zone turning on. After a manifold fill delay, the second zone is allowed to increase to its maximum. After the augmentor has reached the desired level of operation, the shock command is returned to its nominal value and the bypass doors are closed again. Thus by pulling the shock back to a more supercritical position, the additional inlet stability margin could be obtained to avoid an inlet unstart due to an augmentor light-off transient.

Figure 8 is an augmentor transient where the augmentor is turned off from maximum zone 2. Again the shock is positioned to a more supercritical value during the transient. As PLA is reduced out of augmentation, the control reduces the fuel flow first in zone 2 then zone 1. The exhaust nozzle area decreases to maintain the desired ratio of p_3 to P_5. At the minimum fuel flow for zone 1, the fuel flow is cut off abruptly. This is shown in the figure as the drop off of fuel flow. The exhaust nozzle then returns to its nominal area. However, the shock is pulled to a more supercritical position while the nozzle is closing. Once the nozzle has returned, the shock is returned to its nominal position and the control returns the control bypass doors to their closed position. Again the control achieves the desired results of no unstarts during augmentor transients and in steady state the engine airflow is matched to the available inlet airflow.

As mentioned earlier, the control was also tested with 80-hertz bandwidth bypass doors and with 10-hertz bypass doors with a throat bleed control. The response of the 80-hertz bandwidth bypass door control to the square-wave disturbance was essentially the same. However, because of the greater response capability of the door servos, the inlet terminal shock control was better able to handle the augmentor transient airflow disturbances. The net result was that it was not necessary to position the shock to such a supercritical value. For the 80-hertz bandwidth bypass door control, the value of the $p_{1.1}$ command was reduced to only 5.40 newtons per square centimeter instead of 4.97. This positioned the shock 6.5 centimeters downstream of the inlet throat instead of 17.5 centimeters as was required for the 10-hertz bandwidth bypass door case.

The throat bleed for this inlet consisted of a slot just upstream of the geometric throat. This slot dumped into a volume which was bled overboard through four servo controlled butterfly valves. This bleed was used as a "shock trap" by monitoring the inlet throat Mach number and opening the valves if the throat Mach number dropped too low. By including this control with the 10-hertz bandwidth bypass door control, it was also possible to reduce the supercritical value of shock position during the augmentor transient. The value of $p_{1.1}$ command of this control was reduced to only 5.35 newtons per square centimeter. This positioned the shock 8.5 centimeters downstream of the throat. During normal operation, the shock is positioned 3 centimeters downstream of the throat.

The last goal of the integrated control was to provide automatic inlet restart should an inlet unstart occur. Figure 9 shows an inlet unstart with the engine at the maximum zone 2 condition. Immediately following the unstart the augmentor control portion of the integrated control starts to close the exhaust nozzle and to shut off the augmentor fuel flows. The inlet control portion of the integrated control starts extending the spike to increase the ratio of throat area to capture area until the throat goes supersonic again and the inlet restarts. The $p_{1.1}$ command is adjusted to maintain a choked condition in the inlet throat to avoid inlet buzz. Just before the inlet is restarted the $p_{1.1}$ command is reduced considerably. The reason for this is that this inlet generates considerable distortion under restart conditions and this was an attempt to avoid a second engine stall. A second engine stall occurs anyway just after the inlet restarts. This causes a second unstart, but the inlet again restarts without further engine stalls. This characteristic of the second engine stall during restart when unstart occurred during augmented engine operation is not understood at this time. The augmentor is shut down during the inlet restart sequence and does not relight without removing the PLA from augmentation and then returning it to augmentation. The control, however, does bring the inlet and engine back to the match condition after the spike has been returned to its design position.

SUMMARY OF RESULTS

The general problems associated with the mixed-compression inlet and augmented turbofan engine should be similar to those experienced with this particular combination. The results of this test program indicate that the problems of control of an augmented turbofan engine and mixed-compression inlet can be minimized by integrating the engine and inlet control systems. This integration required no additional instrumentation than that normally required for this combination of engine and inlet.

The digital integrated control demonstrated an on-line digital control that provided integration of both augmented turbofan engine and mixed-compression supersonic inlet

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control systems. The control matched engine mass flow to available inlet mass flow. By monitoring inlet terminal shock position and overboard bypass door command, the control adjusted engine speed so that in steady state, the shock would be at the desired location and the overboard bypass doors would be closed. The control thus obtained maximum mass flow recovery as well as maximum pressure recovery consistent with inlet stability. During engine induced transients, such as augmentor lights and cutoff, the inlet operating point was changed to a more supercritical point and thus minimized unstarts. The digital control also provided automatic restart of the inlet should an unstart occur, and provided automatic augmentor operation.

For the system tested here, an improvement in response and damping could be expected with further effort and could also lead to additional sensed parameters. Also, some of the areas not investigated for this control system were the effects of including a turbine inlet temperature limit, and of a mechanical limit on either rotor of the engine. However, these areas are details that could be included in the next effort on applying digital integrated control to the mixed-compression inlet and augmented turbofan engine. In addition, the control tested in this study matched engine airflow to available inlet airflow while maximizing inlet recovery. Other approaches to the integration might be to maximize thrust specific fuel consumption or overall thrust subject to the appropriate restrictions. It is possible that these approaches would result in a different match between the inlet and engine than in the control described in this report.

Lewis Research Center,
    National Aeronautics and Space Administration,
        Cleveland, Ohio, May 3, 1974

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TABLE I. - TEST CONDITIONS

Mach number	2.5
Free stream total pressure, N/cm^2	9.3
Free stream total temperature, K^a	297
Specific heat ratio	1.4
Reynold's number index[b]	.86
Engine total corrected airflow, kg/sec	70.8

[a]Standard day free stream total temperature at
  20 000-meter altitude would be 488 K.
[b]Ratio of Reynold's number at station 2 to
  Reynold's number at sea-level static.

TABLE II. - DIGITAL CONTROL COMPUTER SYSTEM CAPABILITIES

Digital computer	
Magnetic core memory size, words	16 384
Word length, bits plus parity	16
Memory cycle time, nsec	750
Add time, μsec	1.5
Multiply time, μsec	4.5
Divide time, μsec	8.25
Load time, μsec	1.5
Indirect addressing	Infinite
Indexing	Total memory
Priority interrupts	28 Separate levels
Index registers	2
Interval timers	2

Analog acquisition unit	
Overall sample rate (maximum), kHz	20
Resolution of digital data, bits	12 (plus sign)
Output code	Two's complement
Number of channels	64
Input range, V full scale	±10
Conversion time, μsec	38
Total error with calibration, percent	0.073

Analog output unit	
Total number of digital-to-analog conversion channels (DAC)	26
Resolution (13 bit DAC; 10 channels), bits	12 (plus sign)
Accuracy (13 bit DAC), percent of full scale	±0.05
Resolution (12 bit DAC; 16 channels), bits	11 (plus sign)
Accuracy (12 bit DAC), percent of full scale	±0.1
Output voltage range, V full scale	±10
Slew rate, V/μsec	1

Priority interrupt processor	
Number of channels	10
Input voltage range, V	±10
Computer switching	Trigger on rise or fall
Comparator hysteresis, mV	Adjustable from 35 to 650
Comparator output, V	7

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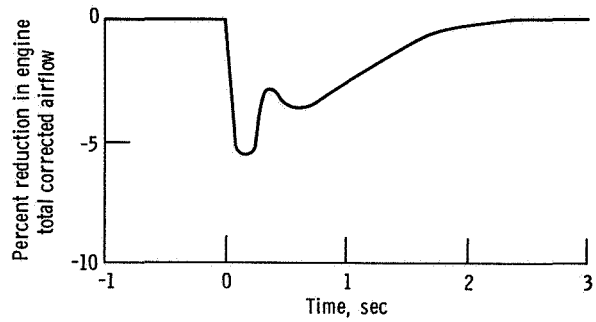
Figure 1. - Typical augmentor light-off airflow distur-
bance as seen at inlet.
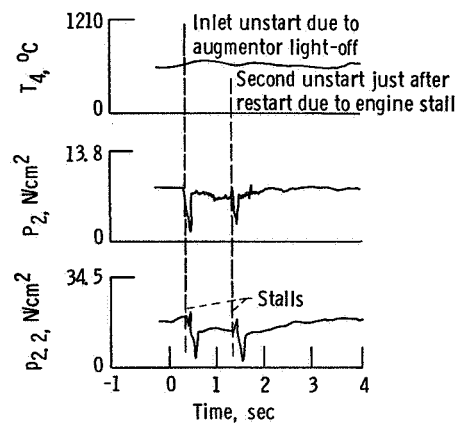
Figure 2. - Augmentor light-off induced
inlet unstart and subsequent inlet re-
start and second engine stall.
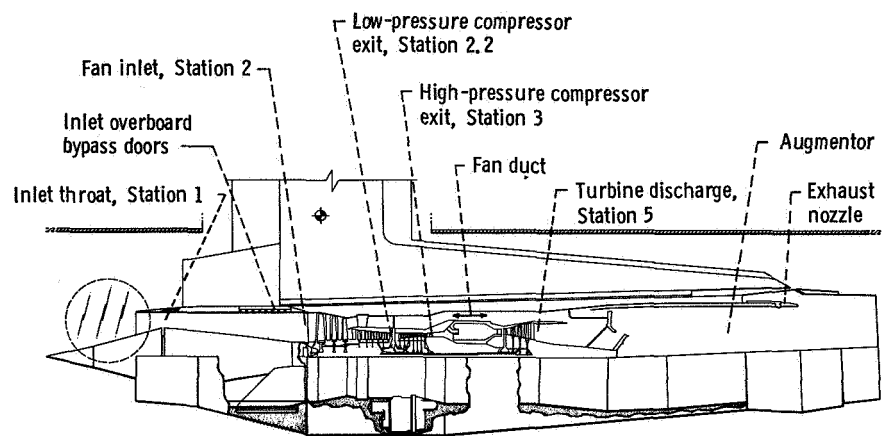
Figure 3. - Cross section of 55-45 axisymmetric mixed-compression inlet and TF30-P-3 turbofan engine.
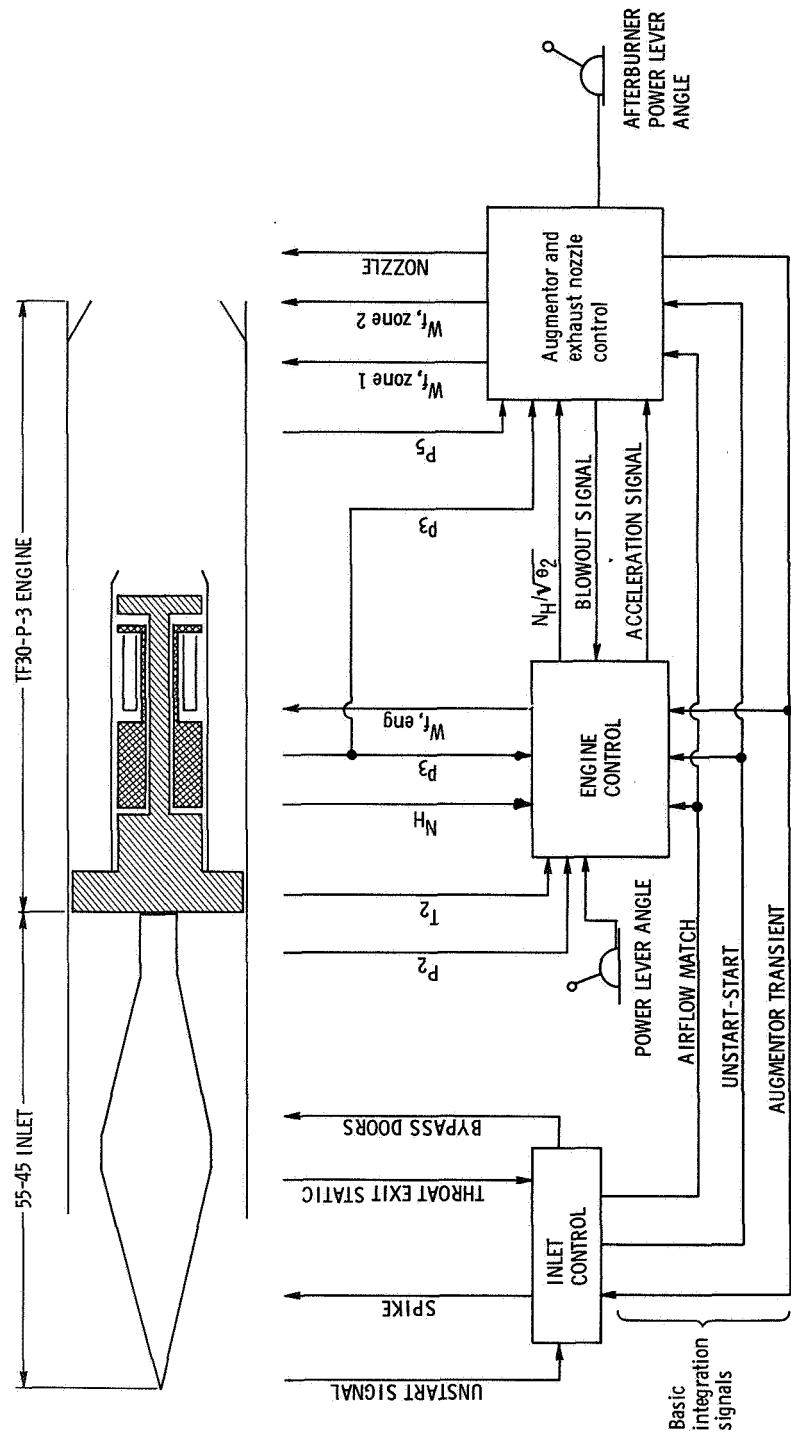
Figure 4 - Overview of digital integrated control.

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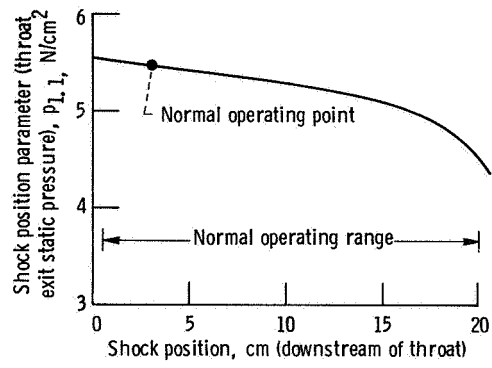
Figure 5. – Relation of throat exit static to shock position.
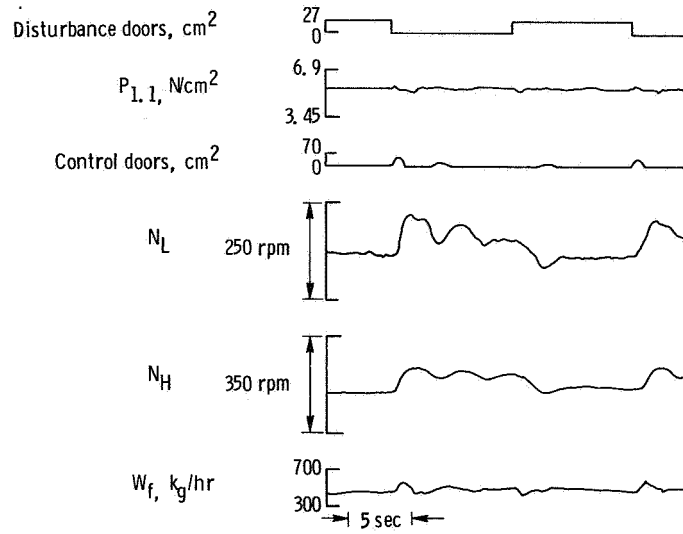
Figure 6. – Square wave airflow disturbance.

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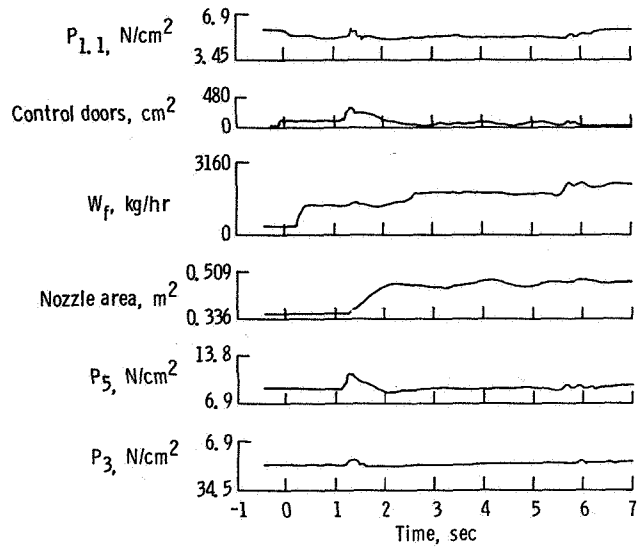
Figure 7. - Augmentor transient, light-off to maximum zone 2.
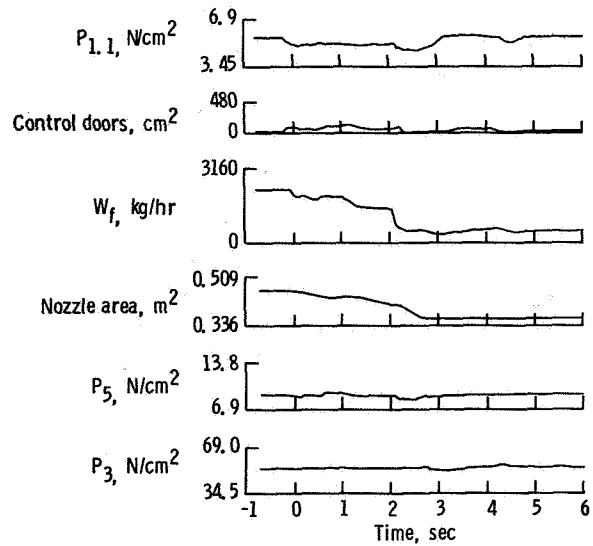
Figure 8. - Augmentor transient, turn-off from maximum zone 2.

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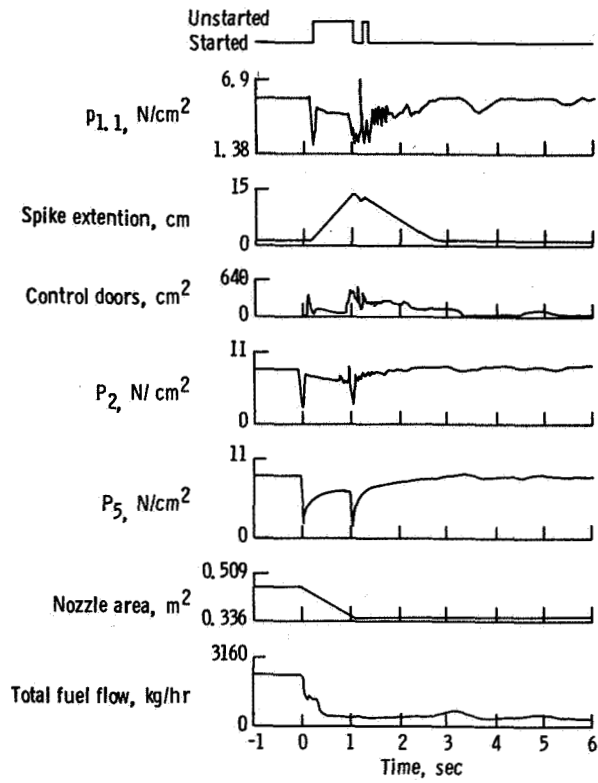
Figure 9. – Inlet unstart and automatic restart with unstart occurring at maximum zone 2 augmentation.

SESSION V

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A FLIGHT INVESTIGATION OF A TERMINAL AREA NAVIGATION

AND GUIDANCE CONCEPT FOR STOL AIRCRAFT

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INTRODUCTION

Studies have shown (e.g., refs. 1 and 2) that short-haul aircraft may provide an effective transportation system that can operate into city centers and suburban facilities. To provide the detailed data base required for the design and development of such a short-haul system, a joint DOT/NASA STOL Operating Systems Experiment Program has been initiated. As a part of this joint program, NASA/Ames has developed an experiments program with the overall objective of providing information that will aid in the choice of terminal area guidance, navigation, and control system concepts for short-haul aircraft, and investigating operational procedures.

In a short-haul transportation system, various levels of avionics systems capability may be needed. Simple, low-cost systems may be adequate for navigation, guidance, and control of aircraft operating in low-density traffic conditions and relatively good weather. More complex and costly automated systems may be economically justifiable for operations in high-density traffic conditions and poor weather. The test data obtained in this program will provide a basis for the selection of system capability to meet operational requirements (e.g., runway requirements, weather minimums, etc.) and will also provide means for estimating the system acceptability and system cost.

A digital avionics system referred to as STOLAND has been purchased and installed (without servos) in the NASA CV-340 twin-engine transport aircraft. Nineteen test flights have been made since October 1973 to obtain preliminary STOLAND performance data in the manual flight director mode using time-controlled guidance.

STOLAND is also installed (with servos) in the powered-lift Augmentor Wing Jet STOL research aircraft (fig. 1) described in reference 3 and a DeHavilland DHC-6 Twin Otter STOL aircraft. Investigations will soon be conducted in these aircraft to obtain performance data on both simple and sophisticated avionics system concepts and the corresponding STOL operational procedures. This report briefly describes the system concept and presents the more significant flight test results obtained in the CV-340 aircraft.

STOLAND is an integrated digital avionics system having a computer of sufficient size, speed, and capability to perform all terminal area navigation, guidance, and control functions, and to automatically control and guide a STOL test vehicle along a curved reference approach flight path. Included in the system are the autopilot modes considered standard for commercial transport aircraft and an autothrottle. This system was built by Sperry Flight Systems to meet stringent performance and environmental requirements. The major components of the system are a Sperry 1819A general-purpose digital computer and a data adapter that interfaces all the navigation aids, displays, controls, and servo actuators (fig. 2). The navigation aids include VHF omnirange (VOR), distance measuring equipment (DME), tactical air navigation (TACAN) receiver, instrument landing system (ILS), microwave modular instrument landing system described in reference 4 (MODILS), inertial navigation system (INS), and radio altimeter.

The system components installed in the cockpit of the aircraft (fig. 3) include the Sperry RD202A horizontal situation indicator (HSI), control wheel, electronic attitude director indicator (EADI), multifunction display (MFD), MFD control panel, mode select panel (MSP), status panel, and data entry panel. During automatic operation, the pilot monitors the system operation through the various cockpit displays. During flight director operation, the pilot uses the same set of displays for guidance information along the reference flight path and to monitor the system. An illustration of the approach flight path flown in the CV-340 is shown in figure 4. It consists of a long inbound leg (waypoints 1-10), a 180° turn to final approach with a 5° glide slope occuring half way around the turn (waypoints 10-12), and a final straight-in approach (waypoints 12-14).

The navigation system used for the approach provides estimates of position and velocity with respect to a runway coordinate system, which has its origin at the glide-slope intercept point (fig. 4). The position and velocity estimate are generated using ground navigation aid information blended in a complementary filter with inertial information obtained from body-mounted accelerometers and attitude sensors, and air data obtained from a barometric altimeter and an airspeed sensor. The ground navigation data are obtained from TACAN except when the aircraft is in MODILS coverage after passing point A (fig. 4). The navigation system also estimates wind velocity utilizing air data. In the event of a momentary loss of ground radio navigation aid information, navigation is accomplished by dead reckoning using air data. Upon regaining radio information, the system automatically switches back to the use of radio data. A detailed description of the navigation system is presented in reference 5.

The guidance system used for the approach is based on a flight path, stored in the airborne computer, which is specified by waypoints (X,Y,Z coordinates) and associated information such as the radius of turn between waypoints and the maximum, minimum, and nominal airspeed between waypoints. The approach guidance is initiated when the aircraft captures the rear extension of the

straight line between waypoints 8 and 9 (see dotted line, fig. 4). At waypoint 8, controlled time of arrival (4D) guidance is initiated. Slightly before waypoint 10, a predictive bank angle command is given, and just before waypoint 11, a constant vertical acceleration maneuver is performed to acquire the 5° flight-path angle. The short straight-in section (waypoints 12-13) is the last segment using the 4D guidance laws given below. The remaining flight path to flare is flown with similar lateral and longitudinal guidance laws except for the system gains, which are relatively low from waypoints 1 to 13, and are high from waypoint 13 to flare to assure precise path tracking.

For lateral tracking the guidance law is:

$$\phi_c = K_1 Y_{err} + K_2 \dot{Y} + \phi_p$$

where

Y_{err} cross track error

$\dot{Y}$ cross track velocity

ϕ_p equals zero, for a straight line track

and

$$\phi_p = \tan^{-1} \frac{V_g^2}{Rg}$$

for a circular track where

V_g ground speed

R radius of turn

g acceleration due to gravity

For vertical tracking the guidance law is:

$$\theta_c = \frac{K_3}{V_g} h_{err} + K_4 \gamma_{err} + \frac{K_5}{V_g} \int h_{err} \, dt$$

where

$\gamma_{err} = \gamma_{nom} - \gamma_I$ (γ = flight-path angle)

h_{err} altitude error

γ_I equals $\frac{\dot{h}_I}{V_g}$, inertial flight-path angle derived from the navigation system

As previously stated, 4D guidance is initiated at waypoint 8 (fig. 4). From this point, the system attempts to arrive at waypoint 13 at a given time.

Control of arrival time at waypoint 13 is based only on speed control, which is provided by controlling the throttle as a function of an airspeed error. In the flight director mode, the airspeed command is displayed on the EADI. The airspeed command V_c is defined as the algebraic sum of a prescribed nominal airspeed (V_{nom}) and an error that is proportional to an aircraft position error (ΔS):

$$V_c = V_{nom} - 0.04 \ \Delta S \ (m/sec)$$

where ΔS is the distance along the track from the estimated aircraft position to a moving target, which represents the desired aircraft position. As the aircraft arrives at waypoint 8, the target and aircraft positions are made to coincide. The computed nominal arrival time at waypoint 13 is based on the time it would take to fly from waypoint 8 provided the aircraft flew the path exactly at the nominal airspeed and there was no wind. To account for winds, the position of the moving target is recomputed every 10 sec based on the latest estimate of wind velocity and direction. This new computed target position assures that the target will arrive at waypoint 13 at the nominal arrival time while moving at the nominal airspeed. If the wind were changing during the approach, the computed positions of the target would have step changes every 10 sec which would result in excessive throttle activity. To limit the throttle activity, the time rate of change in the value of ΔS in the above equation is limited to 6.1 m/sec.

RESULTS AND DISCUSSION

As previously noted, the primary purpose of flight tests in the CV-340 was to validate the operation of the STOLAND system and to obtain a preliminary insight into the navigation and guidance system performance. The data presented are from a set of 20 simulated IFR (hooded) approaches conducted during the latter stages of the tests.

For the CV-340 flights, aircraft position data were provided by a modified NIKE-HERCULES tracking radar. These tracking data were smoothed with a minimum mean-square filter to obtain a best estimate of the actual aircraft position.

The data presented in this report are referenced to a coordinate system whose origin is at the MODILS glide-slope intercept point (GSIP) on runway 35 at Crows Landing NALF (see fig. 4). The XY plane is tangent to the earth at the origin; the X axis is positive in the direction of landing, the Y axis is positive to the right, and the H (altitude) axis is positive up. Representative performance of the guidance and navigation systems along a typical approach is discussed, as well as summary data for all approaches.

Performance for a Typical Approach

The reference flight path and an example of a typical approach are shown in figure 5. The top half of the figure shows the reference path and the

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downrange-crossrange (X vs Y) plot of aircraft position, and the lower part
shows the corresponding altitude-downrange (H vs X) plot. The waypoints are
shown for reference. The sum of the system errors is represented by the
lateral and vertical deviations from the reference path.

As shown in figure 5 the approach was initiated at about 520 m altitude,
about 280 m to the right, and 30 m above the reference path. During the turn
to final approach, the aircraft remained to the right of the path and then
acquired the runway centerline, maintaining that course for the remainder of
the approach. The aircraft remained about 10 to 30 m above the reference path
during the whole approach. The major error prior to MODILS acquisition can be
attributed to the effect of a TACAN DME bias. The errors attributable to the
navigation and the guidance systems are discussed below.

Navigation- Figure 6 presents the lateral (cross track) and vertical
navigation errors for the approach shown in figure 5, and the envelope of
errors experienced in the 20 simulated IFR approaches. The error presented is
the difference between the onboard estimate of the aircraft position and the
tracking radar measured position. The error shown in these traces is the
combined effect of errors due to ground navaid and airborne receiver signal
errors, off-nominal atmosphere effects, small errors in the ground radar track-
ing data, and the basic navigation system errors resulting from software/hard-
ware mechanization. The waypoints are labeled for cross reference with
figure 5.

The envelope of lateral navigation errors at initiation of the approach
at waypoint 8 are as large as 200 m. These errors converge to a maximum less
than 70 m at the initiation of the turn at waypoint 10, where they start to
increase again to values as large as 150 m. Examination of the data indicate
that these navigation errors result from TACAN errors in both range and azi-
muth. A short time after passing waypoint 10, a transition from TACAN to
MODILS navigation is initiated. Navigation errors then converge smoothly to
less than 15 m after transition to MODILS is completed.

The envelope of the time history of the vertical navigation error shows
errors as large as 24 m at initiation of the approach at waypoint 8. The
vertical navigation errors are always positive and are probably a result of a
bias in the baro-altimeter. It should be noted that the baro-altimeter refer-
ence was set prior to each approach based on information radioed from the con-
trol tower, which gives a correct barometric altitude at the runway level only.
After transition to MODILS and the start of the descent at waypoint 11, the
baro-altimeter measurement is slowly blended with and replaced by the
more accurate MODILS data to prevent a step change in estimated altitude at the
initiation of glide-slope tracking. The vertical navigation error converges
to a constant value of approximately 5 m. This bias is unexplained at this
time, although it is speculated that several error sources could be the cause.
For example, a MODILS DME error of about 60 m could result in the 5-m error.
It is clear that more accurate navigation is required for final flare - e.g.,
a radio altimeter or a second, more accurate elevation scanner.

Guidance- Figure 7 presents the lateral and vertical guidance errors for
the approach shown in figure 5 and the envelope of errors experienced in the

20 simulated IFR approaches. The error shown is the difference between the onboard estimate of position and the reference flight path. The waypoints are labeled for cross reference with figure 5. The envelope of time histories of the lateral guidance error shows errors as large as 400 m at the initiation of the approach at waypoint 8; prior to switching to MODILS, these errors converge to smaller values. On switching to MODILS from TACAN, the lateral navigation error decreases while the lateral guidance error increases, reaching a maximum at about waypoint 11. This increase in the lateral guidance error results from a TACAN range bias error that causes the aircraft to fly on the right of the reference path from waypoint 8 to point A (see fig. 5). Upon switching to MODILS, which is a more accurate navigation aid, the navigation estimate indicates that the aircraft is flying to the right of the reference path, thereby generating a lateral guidance error while the navigation error converges to a small value. As a result of the low gain of the guidance system, the aircraft is guided slowly to the reference path. After passing waypoint 11, the lateral navigation and guidance errors converge to small values. As shown in figure 7, the envelope of the lateral guidance error converges to about ±20 m between waypoints 13 and 14 (i.e., 1600 m from touchdown). The envelope of vertical guidance error shows errors as large as 15 m at the initiation of the approach at waypoint 8 and is generally above the desired path. The magnitude of the error represented by the envelope remains approximately constant between waypoints 8 and 10. As shown by the solid line in figure 7, transients occur in the vertical guidance error when the navigation switches from TACAN to MODILS and at approximately waypoint 11 when the descent is initiated. The switching transient decays and the vertical guidance error envelope converges to about ±3 m between waypoints 13 and 14 as a result of the high-gain guidance law and high-gain navigation filters used during the final straight-in approach.

Summary Performance Data;

Errors Prior to Flare (h ≃ 30.5 m)

Navigation- Figure 8 shows the difference between the aircraft position as measured by ground radar and the onboard position estimate as the aircraft passed through a window positioned at a nominal altitude of 30.5 m on a 5° glide slope. (The symbols represent data obtained from flights on two different days.) The data show that the aircraft was to the left of the runway centerline and above the glide slope for the majority of the approaches. For these data, the vertical mean error is 2.4 m above the reference glide slope with a lateral mean error of 1.9 m to the left of centerline. The 2σ errors about the mean are ±2.6 m in altitude and ±4.2 m in the lateral direction.

Guidance- Guidance errors measured at an altitude of 30.5 m are presented in figure 9. The reference in this case is the MODILS 5° glide slope as computed by the navigation equations. If the guidance errors were zero, the data points would be clustered on the estimated glide-slope centerline which is the origin of the graph. For these data, the vertical mean error is 0.8 m below the glide slope with a lateral mean error of 0.8 m to the left of centerline. The 2σ vertical and lateral errors about the mean are ±2.2 m and ±6.8 m, respectively.

Comparison of Flight Data with CTOL Requirements

The test flight data were compared with FAA Category II flight director certification criteria for CTOL aircraft to determine whether the navigation system under investigation might be feasible for a flight director landing on a STOL runway in marginal weather. The FAA criteria are included in figure 9. The FAA criteria from AC 120-29 state that on the localizer,

"From an altitude 300 feet above runway elevation on the approach path to the decision altitude (100 feet), the flight director should cause the airplane to track to within ±25 microamperes (95-percent probability) of the indicated course. The performance should be free of sustained oscillations."

and on the glide slope,

"From 700 feet altitude to the decision altitude (100 feet), the flight director should cause the airplane to track the center of the indicated glide slope to within ±75 microamperes or ±12 feet, whichever is the larger, without sustained oscillations."

Based on a conventional CTOL runway arrangement, these criteria would translate into allowable deviations of about ±3.7 m (12 ft) vertical and ±21 m (69 ft) laterally for a CTOL aircraft at a longitudinal location defined by the 30.5-m (100-ft) altitude point on a 2.7° glide slope.

Figure 9 indicates that the 2σ errors measured in the test flights are within those prescribed for CTOL Category II system landing minima (shaded in fig. 9). Additional testing is needed to define the performance criteria for STOL aircraft certification for Category II weather minima. This comparison of the test flight data with the FAA criteria is not entirely valid, because the landing system, the wind environment, the glide slope, and other parameters were different from those outlined in the FAA advisory circular. Nevertheless, it gives some measure of the system performance.

Speed Control and Longitudinal Guidance

Figure 10 presents the longitudinal guidance error (ΔS), the commanded airspeed, the true airspeed, and the ground speed for the approach shown in figure 5. Also shown are the nominal airspeed specified for the reference path (fig. 5) and the boundaries of the allowable airspeed commands, designated by the unshaded area, which are based on the aircraft performance capabilities. A comparison of the ground speed and true airspeed in figure 10 indicates the strong headwind conditions experienced by the aircraft on the flight path between waypoints 8 and 10. Under such conditions, the aircraft should fly at an airspeed above the nominal to meet the specified arrival time. As shown, the longitudinal error, ΔS, increased linearly and the airspeed command increased above the nominal airspeed for the first 3000 m of track distance. From waypoints 10 to 11, ΔS decreased linearly at its rate limit, as the aircraft caught up with the target and the commanded airspeed approached the

nominal. In this approach a longitudinal error, ΔS, of 76 m, which is equivalent to a 1.3-sec time error, remained to be corrected at waypoint 13.

Time-of-Arrival Errors at Waypoint 13

Figure 11 is a histogram of the time of arrival errors at waypoint 13 for the simulated instrument (hooded) approaches. For these tests, the mean time-of-arrival error is 3.7 sec (late) with 2σ deviation of ±3.4 sec. The mean time-of-arrival error obtained during these tests may result from the TACAN range error which caused the actual longitudinal distance flown to be longer than the reference path. Additional data are required to establish the system performance for all TACAN errors.

It is interesting to note that current manual guidance techniques enable air traffic controllers to deliver CTOL aircraft to the runway within about ±15 sec of the predicted arrival time (ref. 6). This capability corresponds to a single runway acceptance rate of about 40 IFR arrivals per hour using current separation standards. Using the improved capability of the automatic time of arrival guidance system described here it would be possible to increase the runway acceptance rate by about 40 percent (see ref. 6).

CONCLUSIONS

Results are presented for 20 flight director approaches made during an investigation of a STOL approach and landing concept using the NASA CV-340 aircraft. Results of these limited tests led to the following conclusions:

1. Blended radio/inertial navigation using TACAN and a microwave scanning beam landing guidance system (MODILS) permitted a smooth transition from area navigation (TACAN) to precision terminal navigation (MODILS).

2. Guidance system (flight director) performance measured at an altitude of 30.5 m was within that prescribed in FAA AC 120-29 for Category II CTOL operations on a standard runway.

3. Time of arrival at a point about 2 mi from touchdown was about 4 sec ±3 sec (2σ) later than the computed nominal arrival time.

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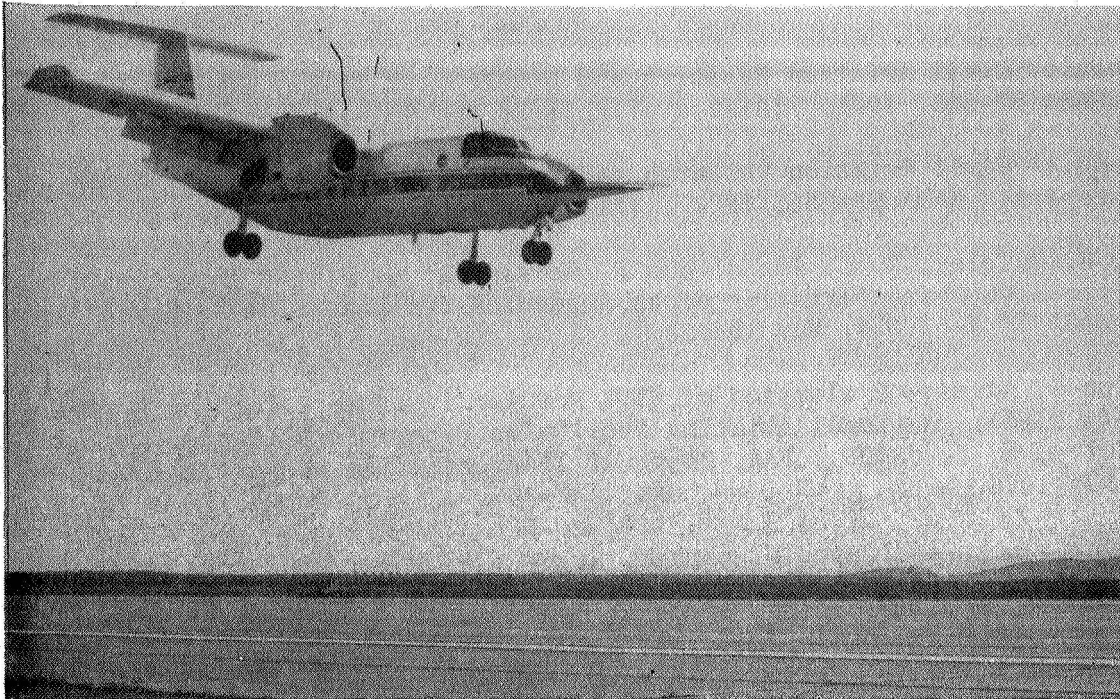
Figure 1.- Augmentor wing jet STOL research aircraft.
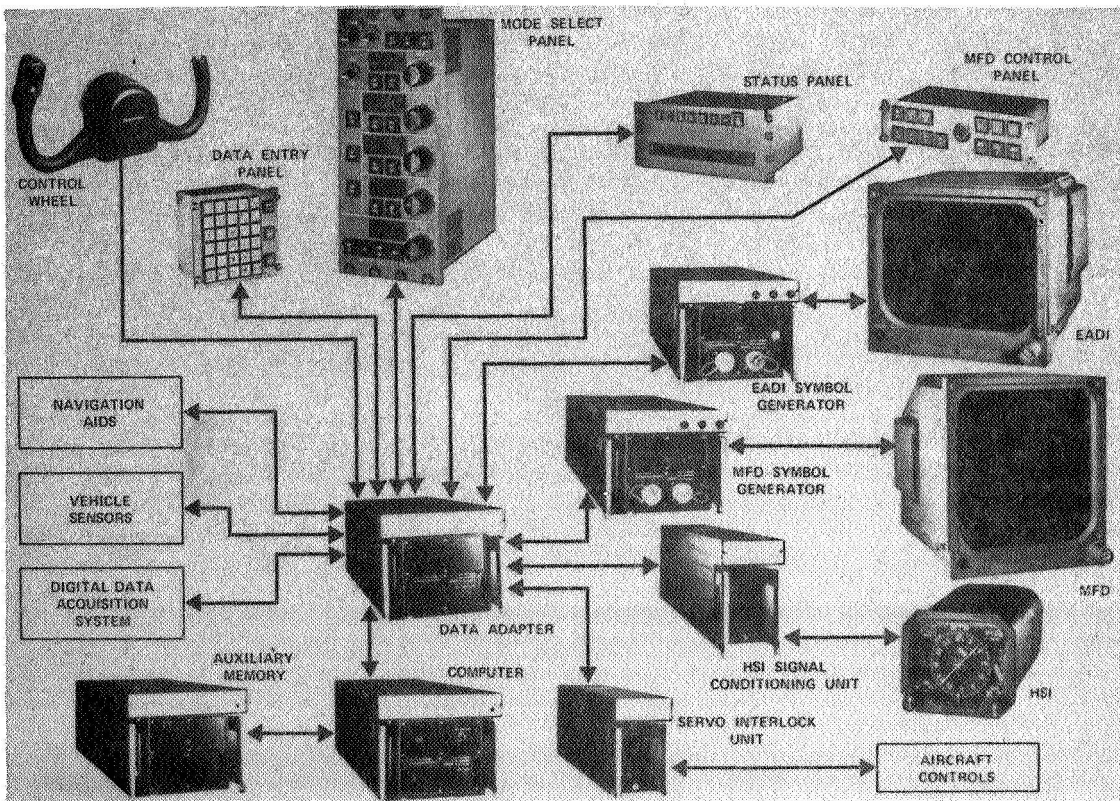
Figure 2.- STOLAND flight-test system.

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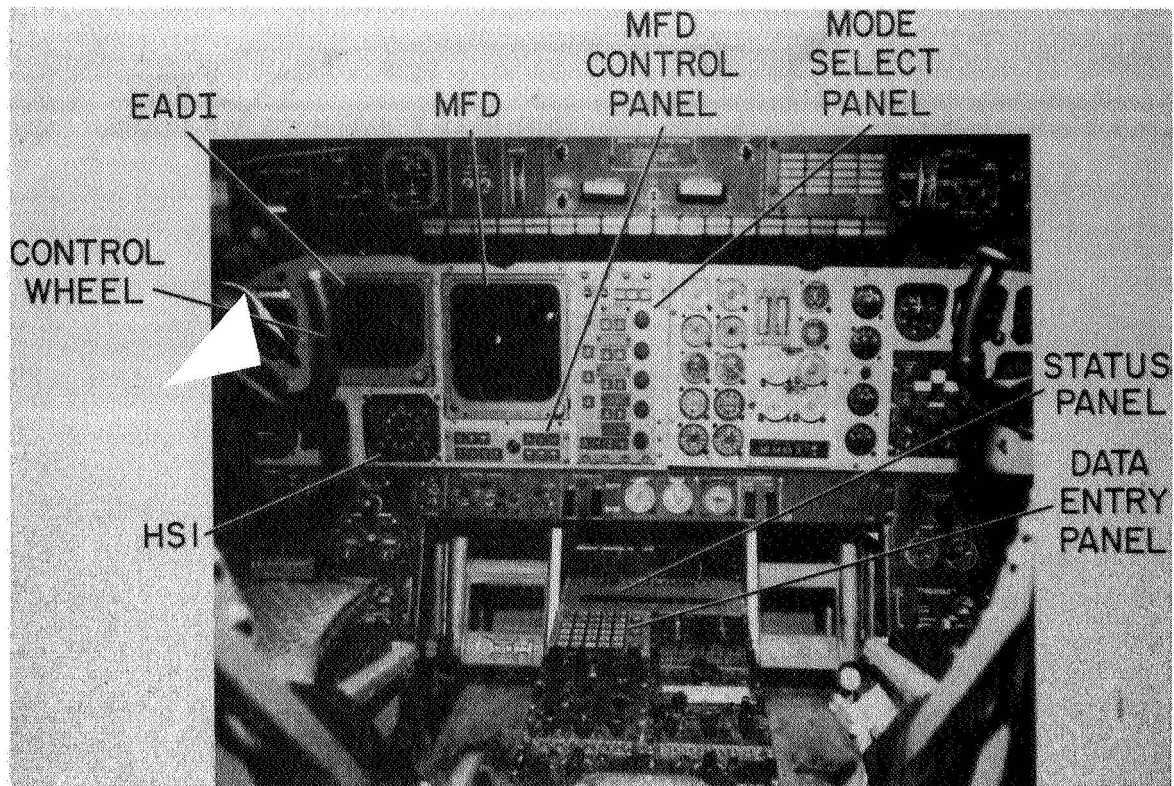
Figure 3.- STOLAND cockpit installation.
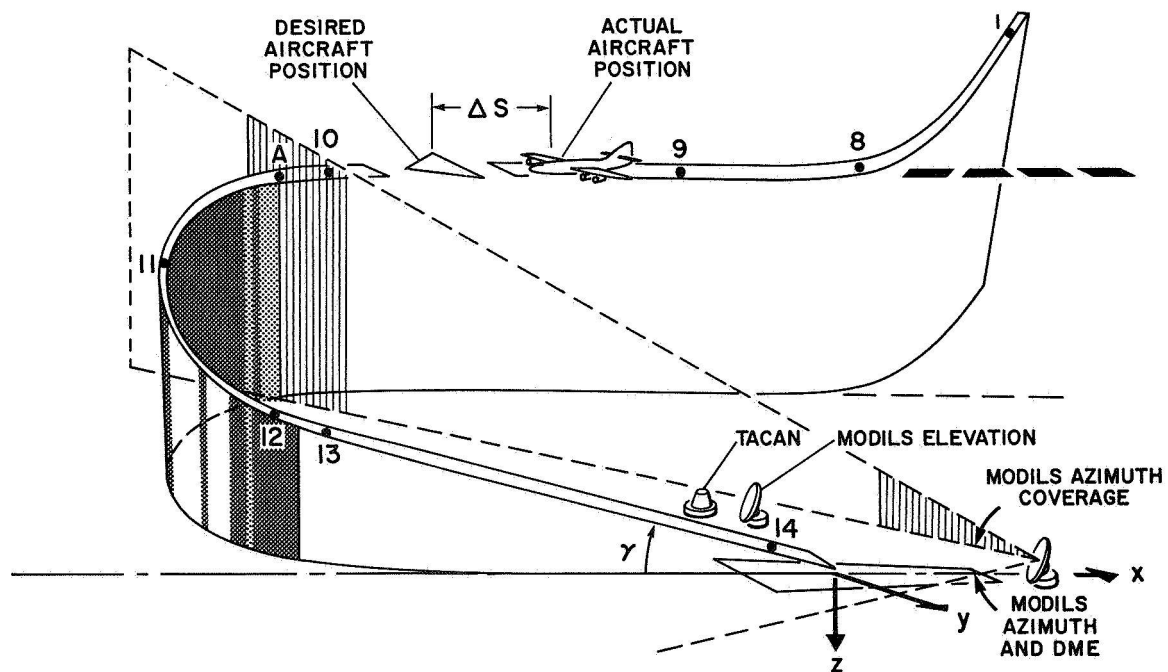
Figure 4.- Approach flight path.
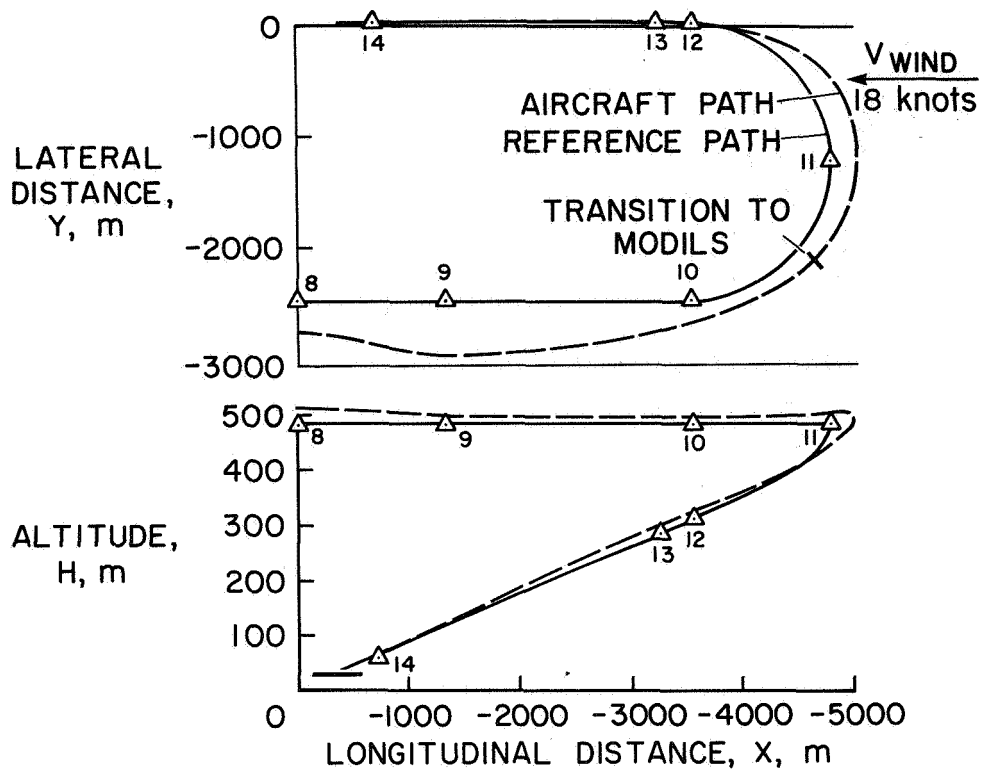
Figure 5.- Typical flight path.
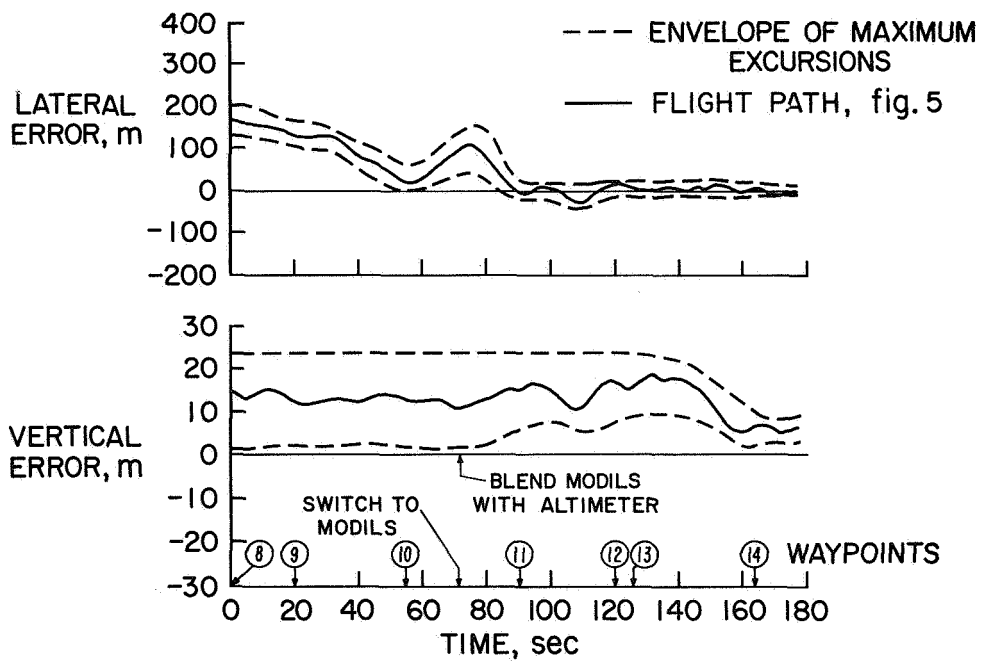
Figure 6.- Navigation errors.
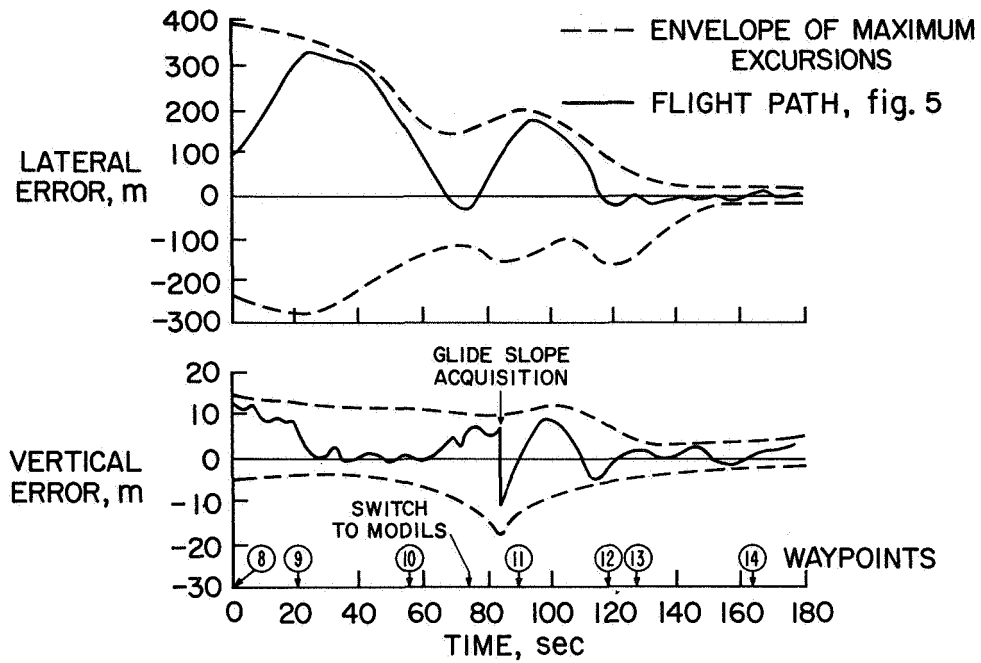
Figure 7.- Guidance errors.
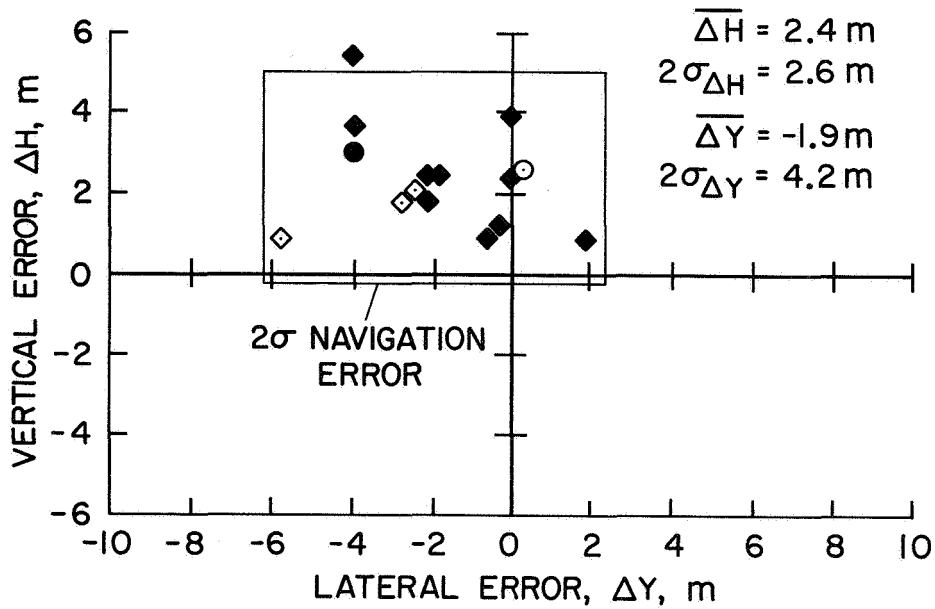
Figure 8.- Navigation errors at 30.5 m.

$$\overline{\Delta H} = -0.8 \text{ m} \qquad \overline{\Delta Y} = -0.8 \text{ m}$$
$$2\sigma_{\Delta H} = 2.2 \text{ m} \qquad 2\sigma_{\Delta Y} = 6.8 \text{ m}$$
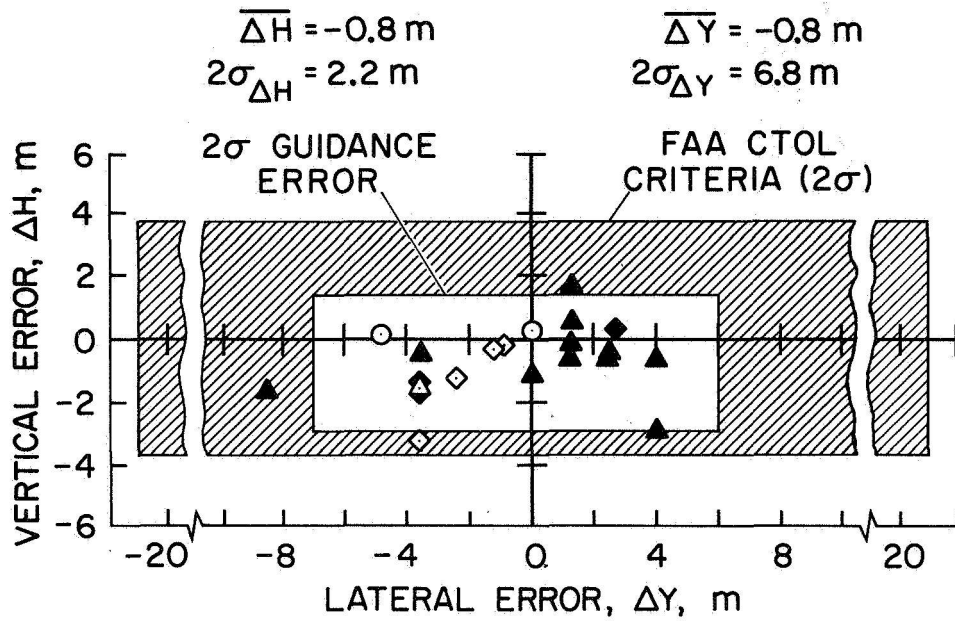
Figure 9.- Guidance errors at 30.5 m.
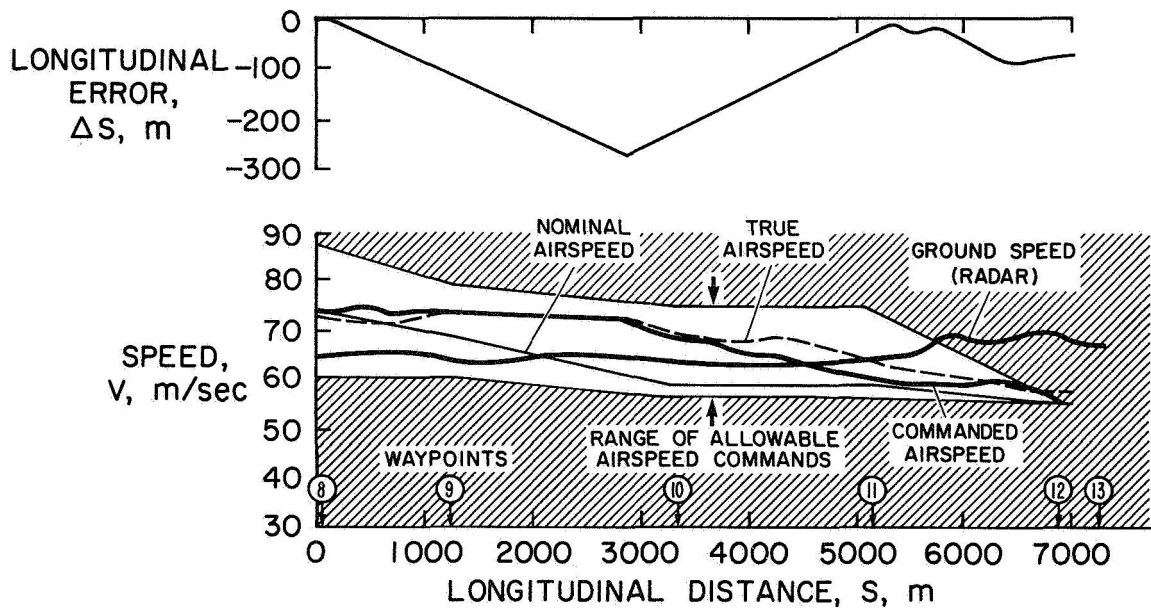
Figure 10.- Longitudinal guidance.
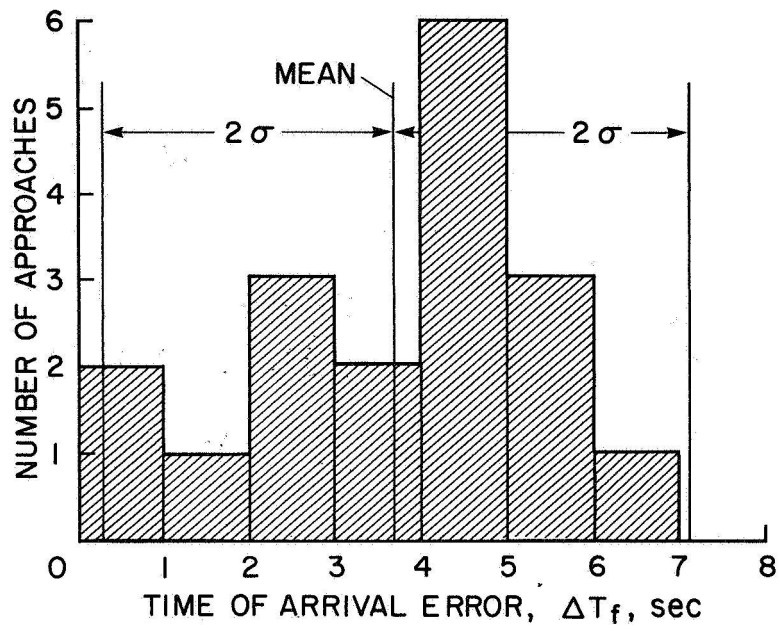
Figure 11.- Time-of-arrival error at waypoint 13.

SOME SYSTEM CONSIDERATIONS IN CONFIGURING

A DIGITAL FLIGHT CONTROL – NAVIGATION SYSTEM

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SUMMARY

A trade study has been conducted with the objective of providing a technical guideline for selection of the most appropriate computer technology for the Automatic Flight Control System of a civil subsonic jet transport.

The trade study considers aspects of using either an analog, incremental type special purpose computer or a general purpose computer to perform critical autopilot computation functions. It also considers aspects of integration of non-critical autopilot and autothrottle modes into the computer performing the critical autoland functions, as compared to the federation of the non-critical modes into either a separate computer or with a R-Nav computer.

The study is accomplished by establishing the relative advantages and/or risks associated with each of the computer configurations.

INTRODUCTION

To justify an investigation of the impact of introducing a new technology into an existing commercial field, two considerations must be ascertained:

1. The motivation behind seeking new technology, and

2. The real advantages to be gained by introducing a particular technology.

Automatic Flight Control systems of civil jet transports have reached the stage of seeking a newer electronic technology. Digital control systems are the candidates.

The purpose of this paper is to describe a method of conducting the background trade studies to define the risks and advantages of a technological change. Although the application of the method is illustrated in terms of civil aircraft, the principles are basic and are applicable in many different areas of industry.

MOTIVATIONS

The analog automatic flight control systems installed on civil jet transports represent significant contribution to the overall cost of development to the airframe manufacturer. In the most recent aircraft, the wide bodied jumbo jets, the automatic flight control systems (AFCS) accounted for development and certification monies ranging from $10,000,000 to $30,000,000 (1969 to 1970 dollars). On the average, the production costs of the wide-bodied jets' AFCS are $300,000.

Cost of ownership has also become substantial, considering that airlines maintenance figures show an annual maintenance cost of 1.5¢/$100 of initial system cost. That amounts to $4500 annually, or $90,000 over the normal life span of the aircraft.

In general, such high costs have been incurred because of increased performance and safety requirements. A particular point is the general requirement for automatic landing systems, resulting in increased redundancy in the sensors, computers, and actuators of the AFCS.

Technological advances in the analog art, in terms of computer architecture and electronic component packaging, have managed to keep costs under reasonable control. For example, considering only the AFCS electronics, a dual-pitch simplex monitored roll configuration of 1966 vintage costs the same as a total duplex pitch-roll system developed in 1969. This is in spite of the fact that the latter system has approximately 40% greater capability due to redundancy and increased operational requirements.

However, the situation doesn't appear to be stable. That is, advancement of analog state-of-the-art isn't sufficient to maintain an adequate margin against further total cost increase for future airplanes. One possible solution is to change the system technology from analog to digital to provide a more competitive condition in meeting yet higher performance and safety requirements.

Substantial investigation and development has been conducted with digital flight control systems (DFCS). However, the accumulated data and conclusions are not directly transferable to civil transports because the greatest majority of the programs have been militarily oriented. The result is that the basic ground rules of development rely on calculated risk levels for safety, performance, and costs which could not be justified for commercial aircraft.

Therefore, any attempt to realistically judge the attendent risks and advantages of developing a commercial DFCS stumbles over the absence of hard trade data. "Absolute" data is available for analog systems because of comprehensive, empirical do's - dont's derived from past experience. Such data are not available for commercially feasible DFCS.

A reasonable comparison - or trade study - methodology can be developed in the absence of "absolute" data by establishing a relative comparison referenced to a known quantity. In the present case, the known quantity is represented by an analog AFCS design in which there is a high level of confidence that it will comply with a significant requirement; the high level of confidence resulting from the absolute data embodied in established design techniques and practical experience.

The reference system can then be arranged in terms of known risk parameters. A comparison of each risk parameter, individually with a counterpart parameter of a DFCS, can be conducted in a relative sense to determine the increment of risk incurred with the DFCS (a negative increment spotlights an advantage). In effect, a sort of chain rule is established which allows evaluation of the newer technology system in known and understood terms of the older technology system.

IDENTIFICATION OF REAL ADVANTAGES

In order to arrange and select appropriate risk parameters, it is necessary to identify the risk points of the analog AFCS. A general survey of latest generation analog systems will result in the following conclusions:

1. Computational Accuracy

    Operational amplifier techniques have reduced computational tolerances to levels between 2 and 5%. However, considering the total AFCS - i.e., sensors, guidance signals, actuators, as well as the computers - further reduction of computational tolerances loses significance in view of the tolerances and inaccuracies of the sensors, guidance signals, etc., which typically range between 8 and 20%.

2. Reliability

    Design and packaging techniques have resulted in analog AFCS computers with mean time between failures of thousands of hours. Manufacturer warranties of 3000-4000 hours are not uncommon. However, with system-wide MTBF's of 200-300 hours, it can be seen that the computers' contribution to system railure rates is relatively insignificant. Therefore, substantial design activity to further increase computer reliability will not pay off proportionately in overall system reliability.

    Another aspect of system relaibility is its availability - a direct function of the system owner's ability to maintain the system. In this respect, analog systems have been shown by experience to be deficient.

    Build-in-test-equipment (BITE) is generally provided in all modern analog equipment. However, each test feature, being itself analog, requires additional circuitry dedicated to testing only. The increased complexity generated by BITE motivates the designer to restrict BITE to within the individual computer. System-wide tests are prohibitive.

    The end result is that fault isolation - to indicate appropriate maintenance activity - within the computer is relatively efficient (about 86% in the 747). But the "system effectiveness", defined as

$$\left[\frac{CONFIRMED\ FAILURES}{TOTAL\ COMPONENT\ REMOVALS} \right] \times 100\%,$$

ranges between 20 and 50%. Thus, more than half the owners maintenance actvities are inappropriate.

3. Redundancy Requirements

Within the scope of commercial jet transports, existing and imminent, redundant systems have relatively little application outside of yaw damping (simple stability augmentation) and automatic landing. More exotic requirements - flight critical modal suppression or control configured vehicles stability systems - are anticipated to be well beyond the next generation of civil aircraft.

Consequently, analog technology has been successfully applied to existing redundancy requirements since 1966.

4. General Cost Considerations

Each new generation of aircraft is accompanied by a redesign of the analog AFCS. Invariably the redesign is necessary to incorporate newer packaging techniques to maintain reliability and reduce costs. In effect, the AFCS is tailor-made.

Peripheral costs are induced by the tailoring. Test equipment, technician training, etc., must be revised each time an airline re-equips.

The general conclusions are that an effective comparison between a digital and analog AFCS must be parameterized to show substantial advantages in terms of system maintainability and costs. Structuring the trade to prove that a digital system is as good as an analog system, or to high-light relatively insignificant advantages will not provide the supporting data necessary to introduce digital technology into commercial AFCS service.

Therefore, selection of risk parameters associated with maintainability on a systems basis, and cost reduction (particularly through reasonable integration of system functions) will provide the most effective trade study.

PRELIMINARY SELECTION OF CANDIDATE SYSTEMS

Systems can be examined under two aspects, viz, 1) organization, 2) level of redundancy. These factors interact to some extent, but generally speaking, system organization is the more fundamental factor. Accordingly, candidate systems are initially selected by consideration of alternate system organizations.

A variety of system organizations are available once the decision to employ digital technology has been made. Potential candidate systems range between the extremes of a central computer that performs all electronic computation, (total integration) to a one-for-one replacement of analog LRU's (Line Replaceable Units) with digital LRU's. The number of potential system candidates must be reduced to make detailed trade studies between alternate systems feasible.

The extremes, or limiting cases, in the type of system organization may be disposed of by general considerations. For example, the one-for-one replacement of analog computing LRU's by digital elements obviously negates the advantage of time shared digital computing elements in addition to proliferating I/O requirements. Plainly, it offers no advantages in the present application. Indeed, to the authors' knowledge, it has never seriously been proposed as a viable digital flight control system and it is mentioned and disposed of here for the sake of logical completeness.

The other limiting case - total integration, wherein a number of disparate computations such as air data, navigation, cruise autopilot, etc., are performed in one computer - has been seriously proposed for a number of applications. From certain aspects this is an attractive candidate. Specifically, such a system organization yields the minimum number of LRU's, minimizes interface complexity and simplifies system test. Nevertheless, this arrangement must also be rejected as inappropriate for the application under study.

The rejection is based on a consideration of the significance of various computations that would be performed in a central computer. Some of these computations are dispatch critical; i.e., the computations must be available if the airplane is to be dispatched. Air data computations are an example of computations that fall in this category. Other computations, such as cruise autopilot modes or autoland are not necessary for dispatch. It is highly desirable from an airline point of view, that a "deferred maintenance" policy be employed to the extent possible. That is, airlines desire to be able to defer maintenance action until such action is convenient from the standpoint of airplane schedule or location. The integration of dispatch critical and nondispatch critical functions in a common computer is not compatible with a deferred maintenance policy. Furthermore, reliability of the dispatch critical computations will suffer from piece part considerations alone. It should be noted that for some applications, such as an RPV, where all computations are required for mission success, total integration might be the logical choice for system organization.

There still remains a large number of potential candidate systems even after the limiting cases have been rejected. The rationale for further reduction to several candidates most promising for detailed trade studies is based on classifying the functions and assessing the redundancy requirements. These are shown in Table 1.

Examination of Table 1 reveals that there are only two functions that are classified as flight critical; Category III Autoland and Yaw Damping. Both of these functions are accordingly assigned a fail-operational redundancy requirement. There is a significant difference in these two computations however, since the yaw damping function is assumed necessary for high altitude and high Mach number flight (normal cruise envelope). Therefore, an operational yaw damper is required for unrestricted dispatch. The redundancy requirement for this function results from the requirement to maintain artificial yaw damping until a speed-altitude reduction can be effected.

In contrast, the autoland function is flight critical only during those times that Category III conditions prevail; in addition, this function is not required for dispatch. The economic penalty for the nonavailability of the yaw damping function is consequently much more severe than the penalty for the nonavailability of Category III autoland.

TABLE 1

FUNCTION	CLASSIFICATION	REDUNDANCY REQUIREMENT
CATEGORY III AUTOLAND	FLIGHT CRITICAL	FAIL OPERATIONAL
AUTOTHROTTLE	NON-CRITICAL	NONE
CRUISE AUTOPILOT MODES	NON-CRITICAL	NONE
YAW DAMPER	FLIGHT CRITICAL AT HIGH MACH & ALTITUDE; REQUIRED FOR UNRESTRICTED DISPATCH	FAIL OPERATIONAL
NAVIGATION	NON-CRITICAL[1]	NONE
FLIGHT DIRECTOR	NON-CRITICAL	NONE

The remaining functions are seen to be classified as non-critical and similar in redundancy requirements. A logical candidate for further study is consequently obtained by structuring the system on the basis of a critical/non-critical division of functions. This results in a system wherein fail safe functions are performed in dual Nav/Flight Control computers and the flight critical autoland is performed in a triplex computer arrangement. In the following discussion this system structure is designated as a "Federated System".

Another candidate system (Integrated System) is obtained by performing all auto-pilot and autothrottle functions, regardless of criticality, in a set of triply redundant computers and navigation functions in separate computers.

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[1] Subsequent to 1980 this classification may change to dispatch critical with a minimum redundancy requirement of fail-op, but without a requirement for graceful degradation of capability after first failure.

Based on the previous discussion, three system configurations are developed (Figures 1 through 3). The analog computer arrangement in Figure 1 provides the "reference" for established technology. It should be noted that this particular arrangement shown is not presently in service. Rather, it is a logical evolution of system arrangement based on current requirements, and represents the level of technical risk acceptable if a change in electronic technology - to digital - were not also under consideration. (To attempt the trade study using systems technology of, say, 1969, would insert a definite bias factor which could unrealistically effect the conclusions.)

Two types of digital computer technology are considered: General Purpose (GP) and Incremental (ICP). The latter shares many of the characteristics of analog machines; accordingly, similar system architecture (Figure 1) is postulated for systems employing these machines. The similar characteristics make it possible to treat the analog and the incremental systems as synonymous except for software development and control.

Application of the general purpose digital computers to the AFCS are illustrated in Figures 2 and 3. These configurations were selected to provide comparative evaluation of significant design considerations while minimizing unnecessary system variables. Figure 2 represents an integrated autopilot system which provides the greatest feasible reduction of equipment and interface complexity. Figure 3 represents a system arrangement which provides greatest possible isolation of flight critical modes to reduce the risks of failure modes compromising system safety requirements.

The selection of these three candidate systems thus provides a means of evaluating contrasting major design factors, that is:

1. Direct evaluation of digital (General Purpose or Incremental) vs. analog technology by consideration of Figure 1 versus Figure 3;

2. Direct evaluation of the impact of substantial integration by consideration of Figure 2 versus Figure 3; and

3. Direct evaluation of maximum feasible benefits of the digital approach by consideration of Figure 1 versus Figure 2.

After selecting the basic candidate systems, major variations within a system configuration may also be considered, as shown by comparing the federated DFCS illustraded in Figures 3 and 4. The effect of including variations will be to provide a band of merit in the eventual study results. Such a band of merit provides a means of further assessing the sensitivity of system risks/advantage to configuration.

TRADE STUDY METHODOLOGY

The identification of key parameters is fundamental in conducting trade studies. Two sets of parameters were identified to evaluate the alternate systems, viz: "System Parameters" and "Trade Parameters".
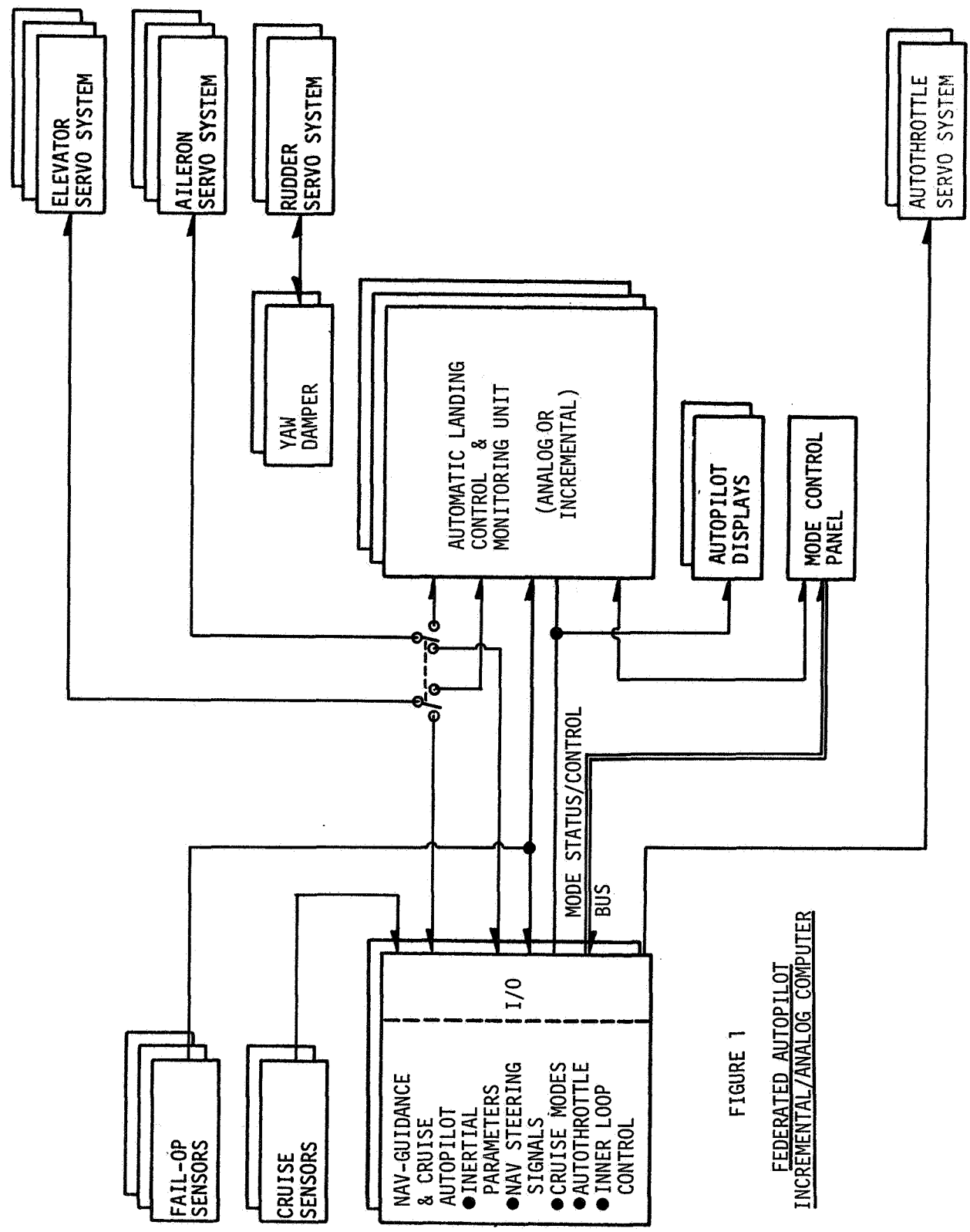
FIGURE 1

FEDERATED AUTOPILOT
INCREMENTAL/ANALOG COMPUTER

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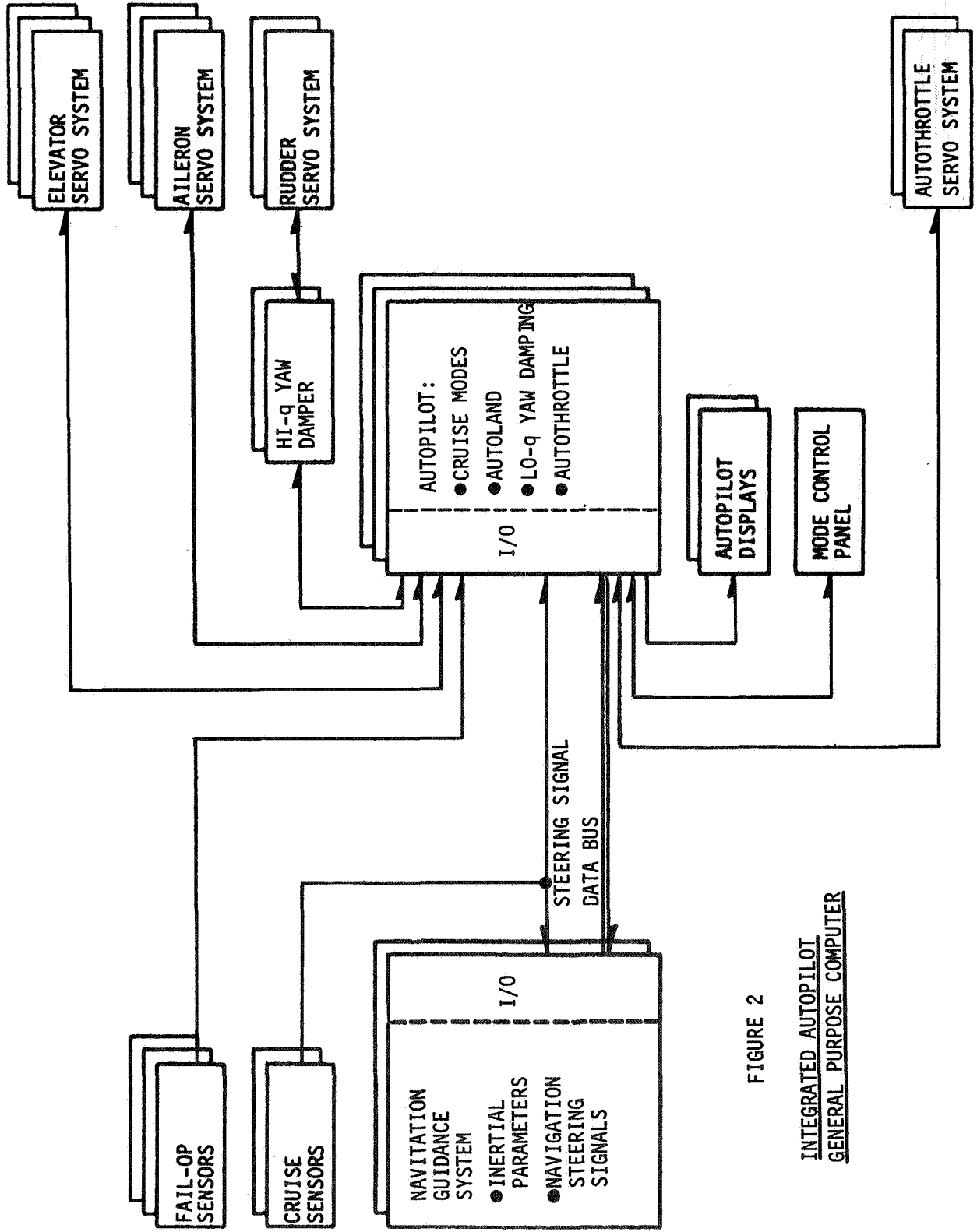
FIGURE 2

INTEGRATED AUTOPILOT
GENERAL PURPOSE COMPUTER
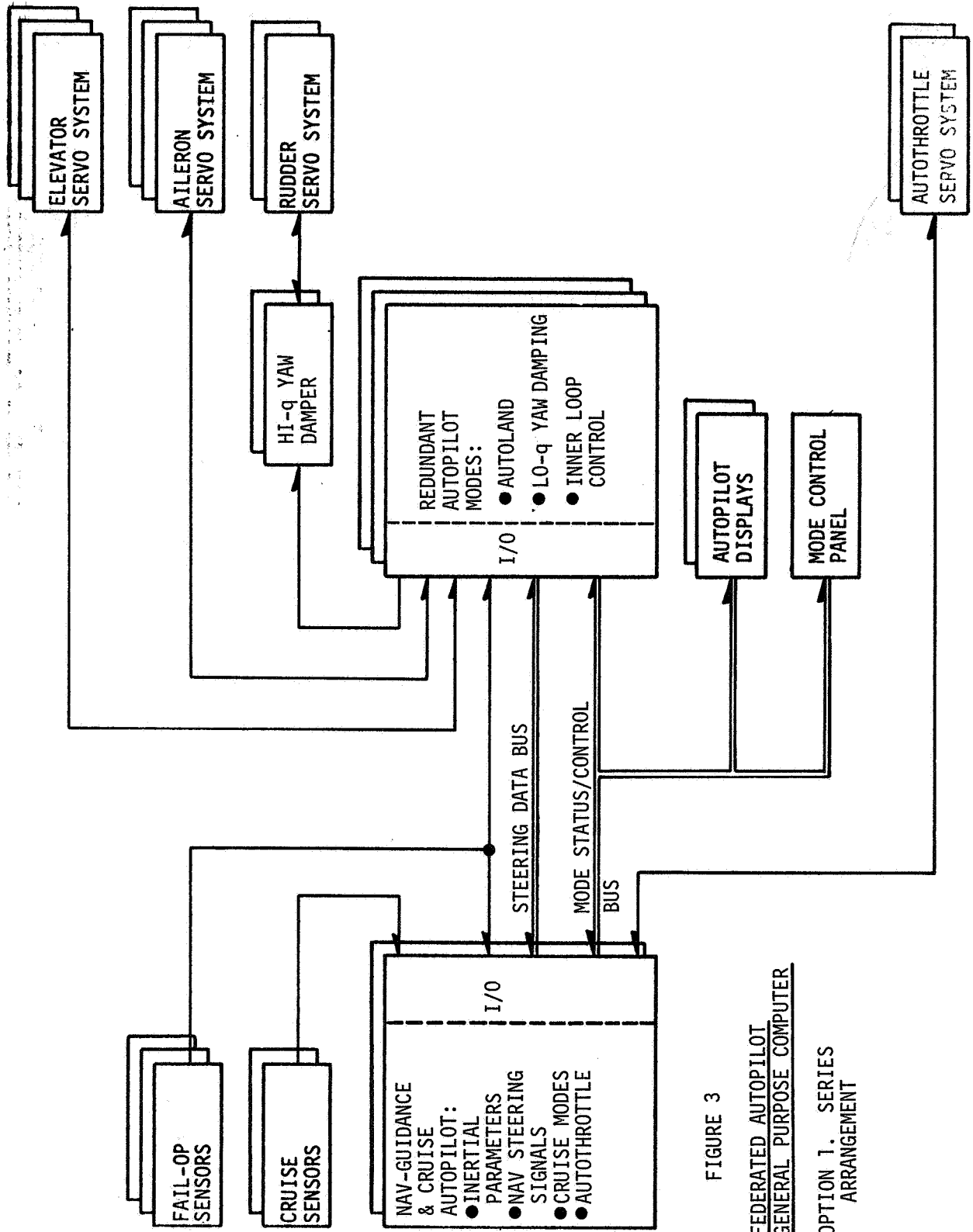
FIGURE 3

FEDERATED AUTOPILOT
GENERAL PURPOSE COMPUTER

OPTION 1. SERIES
ARRANGEMENT

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ELEVATOR SERVO SYSTEM

AILERON SERVO SYSTEM

RUDDER SERVO SYSTEM

AUTOTHROTTLE SERVO SYSTEM

HI-q YAW DAMPER

REDUNDANT AUTOPILOT MODES:
● AUTOLAND
● LO-q YAW DAMPING
● INNER LOOP CONTROL

I/O

AUTOPILOT DISPLAYS

MODE CONTROL PANEL

MODE STATUS/CONTROL BUS

FAIL-OP SENSORS

CRUISE SENSORS

NAV-GUIDANCE & CRUISE AUTOPILOT
● INERTIAL PARAMETERS
● NAV STEERING SIGNALS
● CRUISE MODES
● AUTOTHROTTLE
● INNER LOOP CONTROL

I/O

FIGURE 4

FEDERATED AUTOPILOT
GENERAL PURPOSE COMPUTER

OPTION 2. PARALLEL
ARRANGEMENT

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Trade Parameters were selected to use as a basis of comparison between major features of each system. The major features were designated as System Parameters. Trade Parameters are weighted according to a Relative Advantage/Risk Factor rationale. System Parameters are weighted in accordance with their relative importance to the overall makeup of the system. System Parameters along with their weighting (relative importance factors) are given in Table II.

TABLE II

SYSTEM PARAMETER	RELATIVE IMPORTANCE FACTOR
Software Development, Verification and Control	1.0
I/O Equipment	1.0
System Test	0.7
Sensor Signal Selection and Fault Detection	0.6
Mode Logic and Interlocks	0.5
Interties	0.5
Processor & Memory Sizing	0.2
Control Law Implementation	0.1

Trade Parameters are defined as follows:

Reliability — The impact which the System Parameter under consideration has on the system integrity, operational availability and ability to meet safety requirements (autoland and dispatch critical functions).

Testability — The requirements imposed on system test in terms of hardware/software by the System Parameter being evaluated.

Monitorability — The requirements (in terms of hardware, software and engineering development) to provide failure detection for those elements of the System Parameter being evaluated.

Maintainability — The impact on system fault isolation to the LRU level.

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Growth Capability   The ability of the particular parameter to
                       accommodate growth due to expanded system require-
                       ments, or improvements.

Cost   The impact of the parameter on system cost in terms
                       of hardware requirements and/or engineering develop-
                       ment cost.

Trade Parameter weightings are given in Table III.

TABLE III

RELATIVE ADVANTAGE	RELATIVE RISK	WEIGHTING FACTOR
Definite Advantage		2
Probable Advantage		1
No Advantage	No Risk	0
	Minor Risk	-1
	Moderate Risk	-2
	Severe Risk	-3

It will be noted that the weighting system is balanced at "definite
advantage" vs. "moderate risk". Hence, a severe risk will negatively
influence a definite advantage making it less desirable.

Detailed definition of the descriptive terms of Table III are given in
Table IV.

The manner in which the System Parameter/Trade Parameter weighting factors are
combined is shown schematically in Figure 5. A comparison across the systems under
study, for a given System Parameter is used to select the Advantage/Risk weighting
factor or score. Engineering judgement enters, of course, into selecting the
Advantage/Risk score. However, two factors work to minimize purely subjective
influences. First, a careful choice of System Parameters will isolate the most
significant aspects of the system structure; likewise the choice of Trade Para-
meters displays those features or system characteristics that are regarded as
significant in choosing between competing systems. Thus on this level, tacit
assumptions are either exposed or rendered nugatory. Secondly, the Advantage/Risk
scores are selected only after detailed comparative studies of the System Parameters
under the aspect of the Trade Parameters are made. Again, this procedure works to
minimize the influence of subjective factors. In addition, the procedure isolates
any relatively high risk items in the system configuration that is finally selected

TABLE IV

DESCRIPTIVE TERM	DEFINITION
Definite Advantage	A feature that has, through past experience, been demonstrated to be a definite advantage to the airplane system or design task in terms of: Performance simple design task, desing validation, customer acceptance.
Probable Advantage	Feature may result in a sugnificant improvement; or extentent of improvement is minor; or anticipated benefits are based on extrapolation of similar experience in non-related applications.
No risk	Feature has been successfully applied in past commercial airplanes; or feature has been successfully applied in similar situations without difficulty.
Minor Risk	Has been a problem in past applications, but satisfactory solutions were found; or has not been applied before, but current evidence shows that it can be done without difficulty; or associated problems are defined and solutions are available.
Moderate Risk	Has been a problem in past applications and solutions were difficult to achieve; or depends on a new feature with only limited substantiation in practive (i.e. Lab, etc.): or no specific well defined solutions are currently available although the problem is well defined.
Severe Risk	Has been a failure or near failure in past applications; or depends on a new, unsubstantiated feature; or may require radical system redesign if solution is deemed unacceptable; or problems are not clearly defined.

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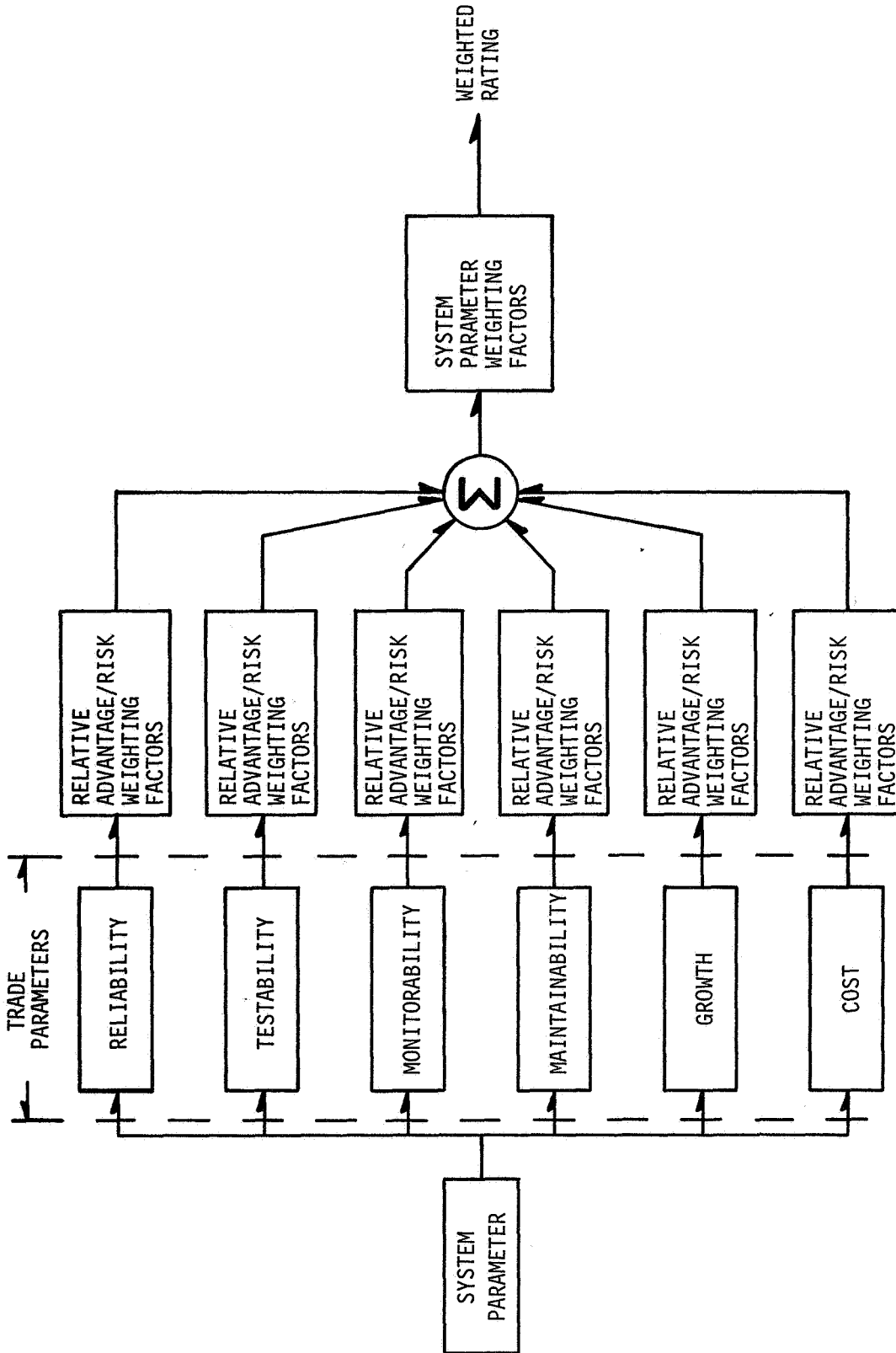
FIGURE 5
TRADE STUDY STRUCTURE

and consequently it serves to focus design efforts on critical items.

A typical example of this procedure taken from a recent trade study is given in the Appendix.

STUDY RESULTS

Application of the above methodology to the systems of Figures 1, 2, 3, and 4 yields the data of Figure 6. Summing the weighted rating of this figure for the various system parameters gives the following overall figure of merit for the systems shown in Figure 7.

The choice of an analog or incremental system is not warranted because of the negative overall ralative rating. The lack of relative advantages for these systems are a function of the nature of the computers. Specifically, they perform only a part of the automatic flight control system tasks, namely control law calculations. The remainder of the tasks - self tests, mode logic, etc. - must be performed by additional, external means.

The figure of merit indicates that the integrated system has the greatest overall potential. However, by reviewing the results for each of the system parameters as displayed in Figure 6, it can be seen that a potentially high level of risk is associated with software development and control. This clearly indicates that a major follow-on effort is necessary to resolve the issue and reduce the risk.
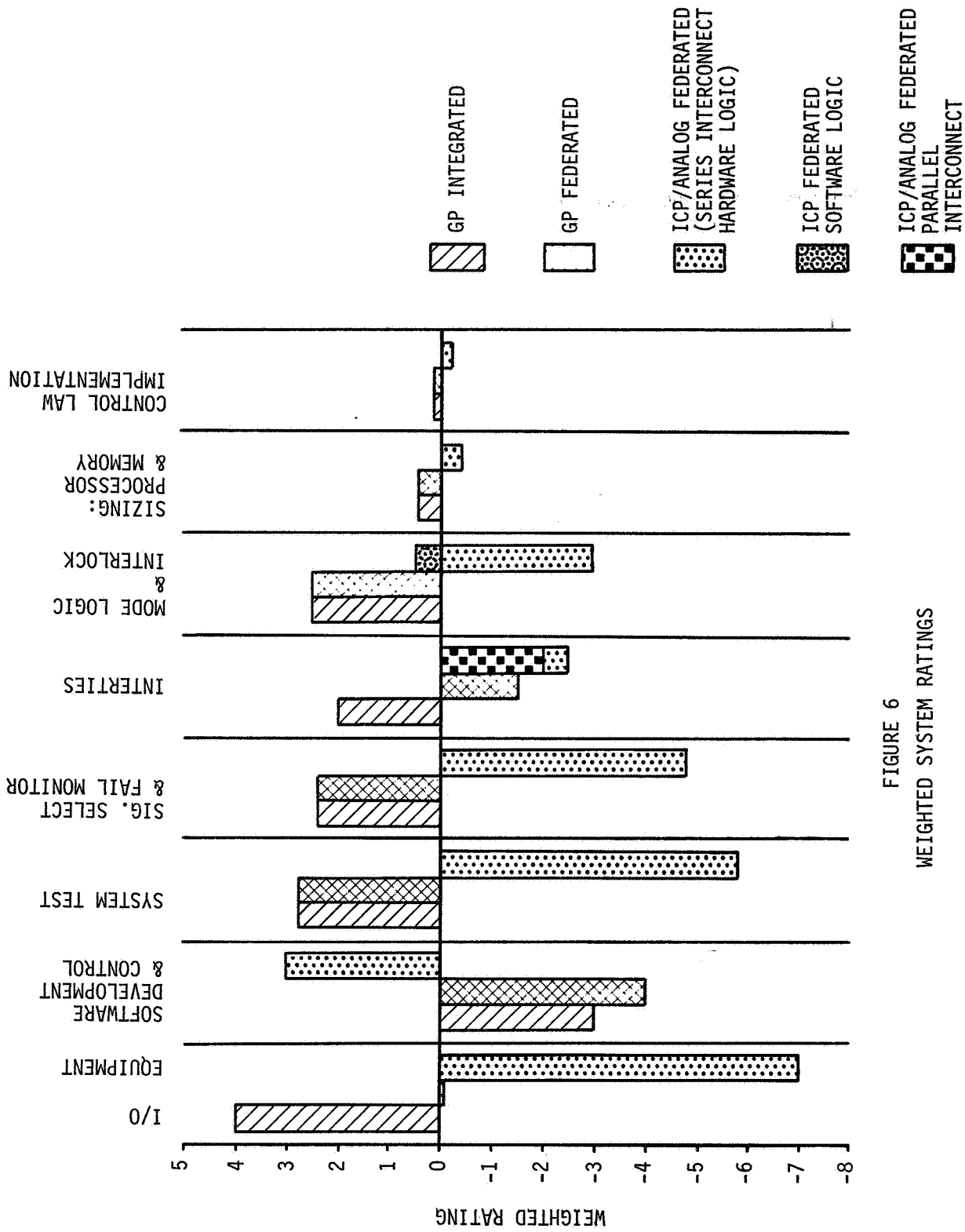
FIGURE 6
WEIGHTED SYSTEM RATINGS

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INCREASING CONFIDENCE

+20

+10 — 11.2 (GP INTEGRATED CONFIGURATION)

0 — 2.7 (GP FEDERATED CONFIGURATION)
NOTE: OPTION 1 OR 2 DO NOT AFFECT
FIGURE OF MERIT

-10

-20 — -16.5 } (OPTION 1) } (RANGE FOR ICP/ANALOG FEDERATED CONFIGURATION)
— -20.5 (OPTION 2)

DECREASING CONFIDENCE

FIGURE 7
RELATIVE MERIT FIGURES
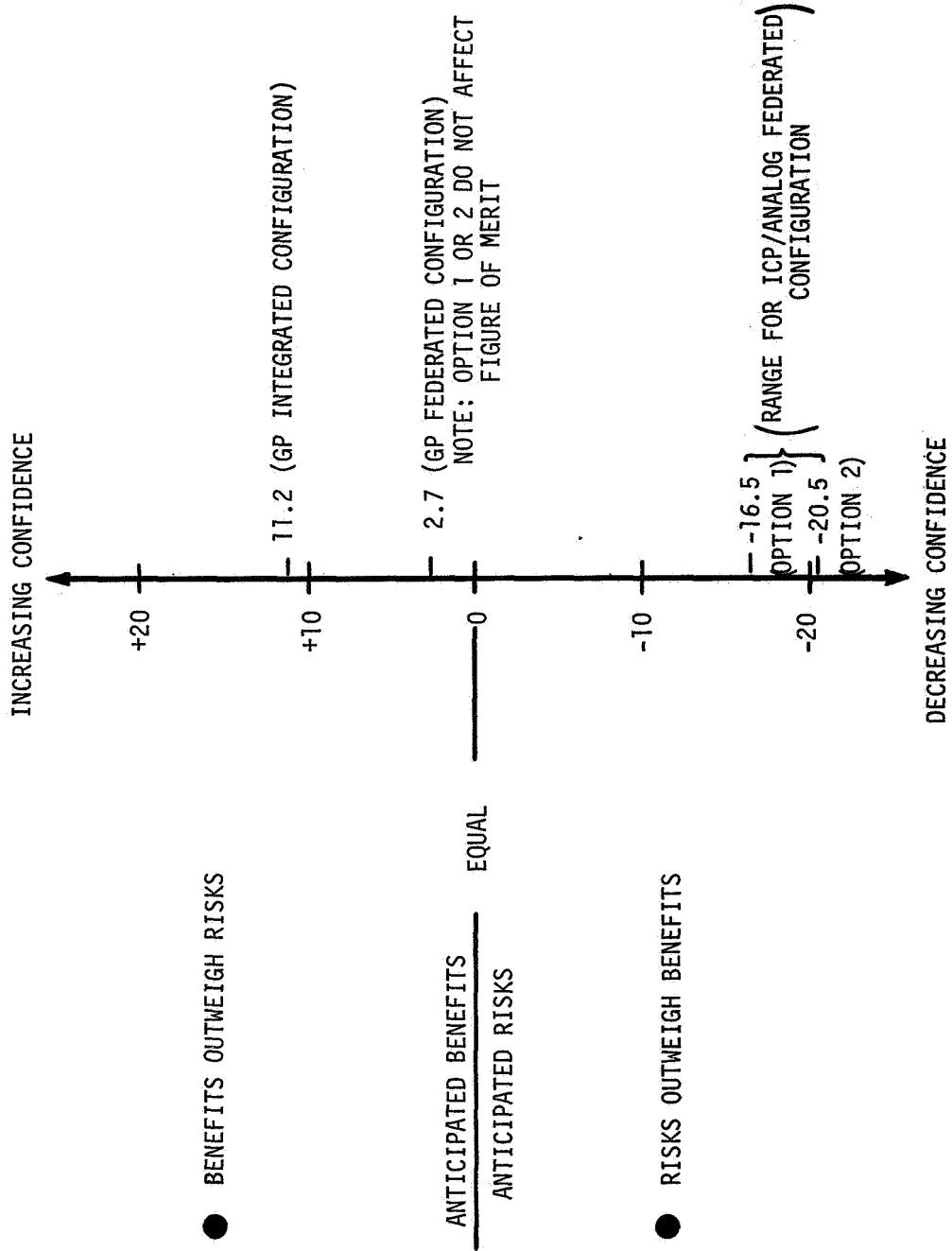
BENEFITS OUTWEIGH RISKS

$\dfrac{\text{ANTICIPATED BENEFITS}}{\text{ANTICIPATED RISKS}}$ EQUAL

RISKS OUTWEIGH BENEFITS

UNCLASSIFIED RISK OR ADVANTAGES

Certain aspects are important in the final selection of a system which are not readily quantifiable, such as vendor support, commonality of equipment, customer choices, ARINC implications, or organizational aspects. The fact that the ICP computer is available from a single source would seem to be a risk with regards to the above consideration since by selecting that architecture one would effectively select the supplier. Customer choices and ARINC implications tend to increase the risk incurred by including the non-critical autopilot and autothrottle functions in the R-NAV computer.

Commonality in the various computers used in the airplane would benefit the customer by reducing his maintenance and possibly inventory costs.

The advances in the digital computer hardware state-of-the-art, through large scale integration and improved semi-conductor devices, reduces cost while increasing computational capacity as well as increasing predicted reliability by reducing the number of interconnections within the computer. However, there is the risk incurred in the early stages of application of new technology.

Failure modes effect and criticality analyses (FMECA) present an area of severe risk for digital systems. The risk is in terms of assessing the effort required to do the FMECA and the probable success achievable. Results of the studies done in the "DOT/SST follow-on" program indicate that any attempt at a FMECA according to the traditional approach may be a gargantuan task even with computer aided evaluations. Similarly, contact with vendors have not revealed any clear methodology for performing a thorough FMECA of digital computers.

Further study is required to assess the FMECA bounds that must be attained to meet certification requirements with a digital autopilot.

The FMECA risk may be alleviated by system design such that the safety is assured by "isolated" simple monitoring devices which are amenable to a thorough FMECA.

With regards to the relative comparison of the ICP and GP computers there is no appreciable difference in the FMECA risk.

CONCLUSIONS

Interpretation of the results of a "relative merit" trade study - such as previously described - can be made only within the frameowrk of level of confidence. One system configuration which rates relatively lower than another cannot be concluded as infeasible. Rather, the confidence of achieving the desired advantages is less than the confidence associated with the higher rates system.

The results of a relative merit trade study, carefully performed, can provide quantified conclusions which clearly indicate the best engineering solution for the system architecture. Also, weakness of the chosen system are identified in such a

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manner as to indicate the degree of urgency for follow-on engineering efforts to reinforce the weak points.

Having made relative comparisons against a known quantity (in this case, the established technology), reasonable predictions can be formed in terms of the actual engineering effort required to introduce the newer technology.

ACKNOWLEDGEMENT

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APPENDIX

The following considerations are typical of the judgement required to assess the risk or advantage increments between candidate systems for a given parameter.

The system parameter discussed in this appendix is typical of the various parameters which must all be considered to complete the study. For the example described in this paper, there were eight major parameters identified.

I. INPUT/OUTPUT STAGE

GENERAL NOTES

1. General Purpose Computer Configurations

All interfaces for incoming and outgoing signals are accomplished within the I/O stage.

Incoming signals are individually conditioned in dedicated signal-conditioning circuits. Two multiplexing units are required, one each for critical and non-critical analog signals. The output of each MUX goes through common time gating circuits and a single A/D converter, then into a parallel-load/serial-output buffer register.

Digital inputs are loaded into their respective buffer registers preparatory to being gated into the computer memory for storage. The A/D converter buffer register, and the digital input buffer registers are gated as serial data into the computer by a common gating circuit.

This arrangement is necessary to allow card-level isolation between critical and non-critical signals. Common circuitry is always downstream of adequate buffering.

The output signals generated by the computer are treated in a similar fashion, i.e., a single D/A conversion followed by individual signal conditioning as required.

Servo amplifiers for elevator and aileron position servos are included as part of the I/O stage.

The high speed yaw dampers are independent analog systems comprising control law calculation, engage and disengage control, and servo loop electronics in a package separate from the AFCS computer. However, low speed (flaps down) yaw damping is augmented by turn coordination and yaw damping control generated within the AFCS computer. Channels A and B provide the upper and lower yaw damper augmentation respectively. Channel C's augmentation may be used for monitoring purposes and as a switchable hot-spare for either, upper or lower yaw damper.

1. General Purpose Computer Configurations (continued)

The I/O stage also includes an interface between the generated auto-throttle commands and the autothrottle (dual) servos. In a similar arrangement to the yaw damper augmentation signals, channel C serves as a monitoring function and switchable hot spare for autothrottle control.

2. Analog Computer Configuration

The description of the analog "I/O" essentially follows that given in Note 1, with the following exceptions:

a. Obvious deletion of MUX requirements.

b. "Brickwall" configuration, i.e., federated configuration does not include interface mixing of critical and non-critical signals. Only critical signals are routed into the analog computer.

c. All yaw damping functions are eliminated from the analog computer as illustrated in Figure 1.

A COMPARISON OF ALTERNATIVE FLIGHT CONTROL COMPUTER CONFIGURATIONS

INTEGRATED GP	FEDERATED GP	FEDERATED ICP/ANALOG
INPUT/OUTPUT	INPUT/OUTPUT	INPUT/OUTPUT

INTEGRATED GP

INPUT/OUTPUT

The integrated I/O comprises the functions described above and is integrally packaged with the computer. (A minor risk in the possibility that computer volume will dictate separating the I/O stage into a separate package requiring some additional effort to build, test, and install the I/O).

As discussed in other sections, mode logic, monitoring, self test capability, and such, are functions resident in the computer itself. Therefore, the integrated configuration represents the least complex I/O of the three configurations under study.

FEDERATED GP

INPUT/OUTPUT

Separating the non-critical from the critical functions has only a moderate impact on the I/O stage associated with the critical computer. The signal input is cut approximately in half materially reducing the dedicated signal conditioning circuits. Output functions are reduced only by elimination of auto-throttle servo interface. The resulting reduction in I/O stage electronic complexity is somewhat offset by the continuing requirement for an integrated self test of the AFCS for maintenance purposes. That is, additional self test program control discretes will be required to interface between the critical system I/O and the non-critical system. The net effect, however, is a general decrease in complexity of the critical I/O stage which is a definite advantage when certification and certification-documentation requirements are considered.

The trade off must also consider the impact of locating non-critical functions and their associated interface requirements in a host machine (such as the RNAV) or in completely separate packaging. In either case, substantial additional equipment must be located where it otherwise wouldn't exist. The result is a moderate risk in terms of cost, reliability, and flexibility (growth) of the overall system but with no particular advantage accruing from the separation. Additionally, it might be expected that the separation will further complicate management and

FEDERATED ICP/ANALOG

INPUT/OUTPUT

ICP

This I/O stage follows the same description as for the general purpose integrated and federated configurations and also includes the following notable increases in hardware complexity:

a) Sensor Signal Select and fault detection circuitry

b) Discrete logic voter and monitor

c) Mode logic

d) Hardware required for System Self Test

These additional functions, assumed to be resident in the software of the general purpose machine, will be performed by dedicated hardware in this configuration. That hardware may take the form of a specially made whole word machine, or a combination of whole word computation and solid state switching mode logic (similar to that employed by contemporary auto-pilot systems). In either event, this represents equipment/circuitry yet to be developed and thus is considered as a moderate risk in terms of reliability, monitorability, testability, and growth capability.

Further, the I/O state is to be separately packaged from the computer resulting in an estimated moderate cost risk.

A COMPARISON OF ALTERNATIVE FLIGHT CONTROL COMPUTER CONFIGURATIONS

INTEGRATED GP	FEDERATED GP	FEDERATED ICP/ANALOG
	conduct of the required self testing leading to an assessment of minor risk.	ANALOG The assessment of the analog interface generally follows that of the ICP except as qualified by General Note 2. Also, the analog interface does not include signal sensor selection. This decrease in complexity is not reflected in terms of significant circuit reduction however. Therefore, the overall assessment remains similar to that of the ICP, i.e., moderate risk in terms of reliability, monitorability, and testability. Growth capability is very difficult to achieve, therefore it represents a severe risk. The interface is inherently integrated with the analog system, thus costs are predictable. Consequently, no risk can be established here. However, no signi-ficant advantage is identified either.

SESSION VI

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DESIGN OF A CONTROL CONFIGURED

TANKER AIRCRAFT

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SUMMARY

A study was conducted to determine the benefits that accrue from using
control configured vehicle (CCV) concepts and the techniques for applying these
concepts to an advanced tanker aircraft design. Reduced static stability (RSS)
and flutter mode control (FMC) were the two primary CCV concepts used in the
design. The CCV tanker was designed to the same mission requirements specified
for a conventional tanker design. A seven degree of freedom mathematical model
of the flexible aircraft was derived and used to synthesize a lateral stability
augmentation system (SAS), a longitudinal control augmentation system (CAS),
and a FMC system. Fatigue life and cost analyses followed the control system
synthesis, after which a comparative evaluation of the CCV and conventional
tankers was made. This comparison indicated that the CCV weight and cost were
lower but that, for this design iteration, the CCV fatigue life was shorter.
Also, the CCV crew station acceleration was lower but the acceleration at the
boom operator station was higher relative to the corresponding conventional
tanker. Comparison of the design processes used in the CCV and conventional
design studies revealed that they were basically the same.

INTRODUCTION

In an Air Force sponsored study, conducted by the Boeing Company, a CCV
tanker was designed to satisfy the same mission requirements specified for an
advanced conventional tanker design. The purpose of this study was to deter-
mine the performance characteristics, control characteristics, methodology
for applying CCV concepts, and the design process resulting from applying CCV
concepts. In addition to RSS and FMC, the two CCV concepts applied, maneuver
load control (MLC), ride control (RC) and gust load alleviation (GLA) were
three other concepts given consideration.

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The objectives of the study were to:

1. Define a CCV configuration

2. Synthesize a control system for the configuration

3. Compare the CCV and conventional tanker designs

To achieve these objectives, the scope of the study was expanded to include:

1. A parametric analysis to determine a nominal size, weight, and geometry for the CCV.

2. Derivation of a point design by refinement of the nominal CCV.

3. The derivation of flexible, rigid body and gust equations of motion for the control system synthesis.

4. Flying quality, fatigue and cost analysis.

The design procedure used in the study is shown in figure 1. First, the CCV was sized in a parametric study, which included two CCV concepts. Next, the CCV point design was defined; and, finally, a control system was synthesized for the point design.

Overall, the methodology used in the CCV and conventional tanker studies were the same. Equations of motion were obtained using finite element methods, and the control synthesis was accomplished using the corresponding transfer functions and root loci. Neither the finite element, transfer function, nor root locus methods are peculiar to CCV design.

CONFIGURATION DEFINITION

Mission and Ancillary Requirements

The specified mission requirements to which the CCV was designed are:

1. Design refuel range

2. Off-load 348,900 pounds of fuel at the design refuel range, Mach .68 and 30,000 ft

3. Personnel and cargo capability

4. Cruise speed: Mach .75

5. Rate of climb with one engine out

6. Takeoff ground roll

7. Landing ground roll

Some of these requirements are depicted pictorially in figure 2. Other quantities used to design and evaluate the CCV are:

1. Gross weight

2. Operating gross weight empty

3. Flying qualities

4. Ride qualities

5. Fatigue life

6. Cost

Parametric Analysis

In a parametric analysis a matrix of CCV configurations was generated; and, although each configuration was smaller than the conventional tanker, each had the same mission capability. Furthermore, each configuration had the same wing loading (W/S), and thrust as the conventional tanker. Because of the application of RSS and MLC however, each configuration had less drag, a smaller operating weight empty (OWE), takeoff gross weight (GW) and wing area than the conventional tanker. From these configurations the smallest CCV tanker was selected for more detailed study and design.

The selected CCV tanker configuration was refined through a detailed design of the aerodynamic, propulsion, and structural subsystems. The result of this refinement, which was constrained by the mission requirements, was the CCV point design developed in the study.

Of the two CCV concepts used in deriving the point design, RSS had the most extensive impact on the CCV external geometry, size and weight. Reduction of the pitching moment requirements accounts for the impact of RSS because these requirements largely determine the size and location of the tail and control surfaces, and the location of the wings and landing gear relative to the center of gravity (C.G.). The influence of RSS is summarized in Tables I and II in which a comparison of various CCV and conventional tanker components is shown. (The starred items in Table I were not determined by RSS.)

In sizing the CCV the only factor attributed to MLC was a 10 percent reduction in wing weight because it was assumed that any higher stresses occuring in the lighter wing structure could be alleviated by an active MLC system. However, for the reason discussed subsequently a MLC system was not synthesized for the CCV, eventhough the assumption of a 10 percent reduction had been applied to the point design.

Point Design Description

The most prominent feature of this design, illustrated in figure 3, is the absence of a horizontal tail. Other design features include a low wing and four engines, of which two are wing and two are fuselage mounted. A tricycle landing gear and a boom operator station located in the aft fuselage section also characterize the configuration. Pitching and rolling moments are obtained from the wing mounted elevons, and the rudder is used to generate yawing moments. Because of RSS, the airframe is statically unstable at some heavy gross weight conditions which includes the takeoff condition.

The CCV point design represents an attempt to maximize the size and weight reductions; and the resulting size reduction may be observed in figure 4, in which the external features of the CCV and conventional tankers are compared.

TABLE I

CCV AND CONVENTIONAL COMPARATIVE SIZE

ITEM	CONVENTIONAL	CCV
WING AREA (FT^2)	10,640	8,984
WING SPAN (FT)	275.	251.4
FUSELAGE LENGTH (FT)	197.	125.
FUSELAGE MAXIMUM DIAMETER (FT)	18.	18.
HORIZONTAL TAIL (FT^2)	2,310	0
VERTICAL TAIL (FT^2)	1,173	571.2
CRITICAL ENGINE MOMENT ARM (IN)	767.	300.
DESIGN WEIGHT (LB)	1,000,000.	835,900.
OWE (LB)	334,100	250,300.

TABLE II

CCV AND CONVENTIONAL WEIGHT COMPARISON

ITEM	CONVENTIONAL WEIGHT (LB)	CCV WEIGHT (LB)	Δ (LB)
WING	120,370	82,650	37,720
HORIZONTAL TAIL	12,200	0	12,200
VERTICAL TAIL	7,070	3,430	3,640
FUSELAGE	52,900	35,660	17,240
SURFACE CONTROLS	9,730	6,590	3,140
HYDRAULICS	5,160	5,080	80
AIR CONDITIONING	2,070	1,190	880
LANDING GEAR	46,790	39,110	7,680
OTHERS	73,500	73,950	-450
WEIGHT EMPTY	329,790	247,660	82,130
DESIGN WEIGHT	1,000,000	835,900	164,100

CCV MODEL

The equations of motion used in the control synthesis included rigid body and elastic structural modes of motion. Finite element techniques, a detailed treatment of which may be found in reference 1 and other sources in the literature, were used to derive the elastic equations of motion. Briefly, the finite element method is a technique in which the structure is modeled by a finite number of nodes (fig. 5) connected by beams or plates which act as structural springs. The structural motion is described by the displacement and rotation of the nodes, at which the forces and moments are assumed to be applied.

For lifting surfaces, an aerodynamic finite element method called the double lattice technique (ref. 2) was used, and the application of this technique entailed dividing each lifting surface into a finite number of trapezoids (fig. 5). A set of coefficients relating the velocity normal to the element and the lift on the element is provided by the method. Also produced by the, method is the dynamic coupling between elements which for example, describe the wing-fin coupling responsible for the flutter mode.

Ordinary differential equations in time with coefficients that are functions of geometry are obtained from the application of the finite element technique. From a procedure for simplifying the equations and a Boeing Company transfer function computer program, the corresponding transfer functions used in the control synthesis were generated. Although the original equations represented as many as seventeen degrees of freedom, only seven degrees of freedom were used in the control synthesis; and these consisted of three rigid and four elastic (structural) modes of motion.

CONTROL SYSTEM SYNTHESIS

In synthesizing the control system no novel design techniques peculiar to control configured vehicles were used.

Transfer functions for the seven degree of freedom model and root loci were used to synthesize a logitudinal CAS, a lateral SAS and a FMC system. The criteria to which these systems were designed are given below.

Design Criteria

The control systems were designed to existing military specifications. The logitudinal CAS and the lateral SAS designs were based on the rigid body flying quality requirements of MIL-F-8785B(ASG), and the FMC system design was based on the $1.15V_D$ criteria of MIL-A-8870, where V_D is the design limit speed. The flying quality requirements are expressed in terms of adequate stability, maneuver response, natural frequency, damping ratios, time constants, and time to double; and the flying qualities are characterized by satisfactory stability, short period, phugoid, dutch roll, spiral and roll subsidence modes of motion.

Lateral SAS Synthesis

The roll axis was unaugmented; and, hence, the lateral SAS consisted of a rigid body yaw damper only. The yaw damper, comprised of a washout circuit and a first order filter, provided the dutch roll damping required by reference 3.

Lateral FMC Synthesis

An antisymmetric flutter analysis revealed the need for a FMC system because the free airframe failed to meet the $1.15V_D$ criterion; that is, the

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second structural mode fluttered at a velocity of 450 KCAS, which is below the $1.15V_D$ of 477 KCAS.

A zero root locus analysis (ref. 4) was conducted to determine the best sensor location and which of the elevons to use for flutter suppression. The philosophy of the zero root locus approach is that the distance between a sensor or surface zero and a structural pole is indicative of the coupling between the sensor or surface and the structural mode. A greater distance implies more coupling or more influence of the surface on the structural mode (fig. 6). The result of this analysis was the selection of an accelerometer location (fig. 3) and the inboard elevon for flutter suppression.

The accelerometer and inboard elevon were incorporated in the design of a FMC system, which consisted of an inner and outer feedback loop. The inner loop was the aforementioned yaw damper and the outer loop was a compensated acceleration loop that controlled the inboard elevon for flutter suppression.

Longitudinal CAS Design

Analysis showed the absence of a longitudinal flutter problem and, hence, only a CAS design was required. Structural coupling was present in the symmetric axis, however, due to the presence of a structural pole in the vicinity of the short period poles at high speeds. Elimination of this coupling (fig. 7) was achieved by using the outboard elevon to which the structural mode was more strongly coupled. Thus, the outboard elevon provides pitch damping and attitude control as well as structural decoupling.

Maneuver Load Control System

Steady state loads were determined for the augmented CCV on the basis of the inflight and taxi load requirements of MIL-A-008861A and MIL-A-008862A. Since the structure was adequate for the loads defined by these specifications, a MLC system was not designed.

Gust Response And Fatigue Analyses

Turbulence response analyses were conducted to determine the fatigue and ride characteristics of the augmented CCV; and since the ride was deemed acceptable and the fatigue life satisfied MIL-A-008866A, neither a GLA nor a RC system was synthesized.

The gust analysis was conducted in the frequency domain, and the atmospheric turbulence was a von Karman power spectral density. The ride quality and the fatigue life were determined, respectively, from the turbulence parameters for acceleration (A) and fatigue damage (N_0).

COMPARATIVE EVALUATION

The aforementioned mission requirements specified for the CCV and conventional tankers are also measures of aircraft performance. Although an absolute numerical comparison of these performance quantities is not generally available, a comparison based on normalized requirements is provided in Table III. A value of unity is assigned to each requirement and the performance quantities are expressed relative to this value. Since both aircraft were designed to have the same range, fuel off-load capability, cruise speed, and cargo-passenger capacity, these quantities have the same value. Values for the rate of climb, takeoff and landing ground rolls differ, however, because the corresponding capabilities were not identical for both tankers. Although both aircraft had a higher rate of climb and a shorter takeoff ground roll than required, the CCV had a faster rate of climb but a longer takeoff ground roll than the conventional tanker. Furthermore, the CCV landing ground roll equaled the slippery runway requirement, whereas that of the conventional tanker was shorter than required.

TABLE III

COMPARISON OF CCV AND CONVENTIONAL TANKER PERFORMANCE

PERFORMANCE QUANTITY	CONVENTIONAL PERFORMANCE	CCV PERFORMANCE	CCV / CONVENTIONAL
RANGE	1	1	1
FUEL OFF-LOAD	1	1	1
PERSONNEL-CARGO CAPACITY	1	1	1
CRUISE SPEED	1	1	1
ENGINE OUT RATE OF CLIMB	2.1	4.2	2
TAKEOFF GROUND	1.2	1.1	.92
LANDING GROUND ROLL (μ = .1)	1.1	1.0	.91

Also compared were the flying quality, flutter, ride, fatigue, and cost characteristics of the two tankers. Except for the roll performance specification which neither aircraft met, the augmented aircraft satisfied all of the flying quality requirements. The roll performance specified by reference 5 is for 30 degrees of bank angle in 2.5 seconds, but the CCV and conventional tankers required 2.7 and 3.75 seconds, respectively, to reach the 30 degrees. Both tankers exceeded the flutter requirements and both had satisfactory ride qualities. However, the CCV crew and boom operator station accelerations were respectively 34 percent lower and 15 percent higher than the corresponding conventional tanker accelerations. From a fatigue life analysis it was learned that the CCV and baseline tankers accumulated 57 and 44 percent, respectively, of their design fatigue lives. Finally, a cost summary comparison, based on the purchase of one hundred tankers, revealed that the CCV will cost 20 percent less (fig. 8).

One of the most important comparisons was between the GW and OWE of the two aircraft. The CCV was 16 and 25 percent lighter in GW and OWE, respectively. The primary importance of these lighter weights is in the potential economic advantages. For example, a 25 percent OWE reduction offers a significant benefit to commercial airlines which may be occasionally confronted with low load factors.

It is important to note that the above comparisons were based on a single CCV design iteration and that additional iterations could alter the performance and other characteristics of the aircraft. Nevertheless, these comparisons provide a valid basis for the conclusions that follow.

CONCLUSIONS

From the results of the CCV and Conventional Tanker studies the following conclusions may be drawn.

1. Significant reductions in GW, OWE and cost are the major benefits resulting from the application of CCV concepts to transport type airplanes.

2. Of all the CCV concepts, RSS has the most extensive impact on the airplane configuration arrangement design and produces the largest reductions in weight and drag.

3. The 16 and 25 percent reductions in GW and OWE are representative of the maximum reductions possible for the specified mission.

4. The application of CCV concepts will not necessarily improve all of the aircraft performance quantities. For example, the 16 percent CCV weight reduction was accompanied by longer takeoff and landing distances, and a reduced fatigue life. However, additional design iterations could shorten the ground roll distances; and the fatigue life could be improved by structural redesign or the inclusion of other CCV concepts such as MLC.

5. The utility of CCV concepts are mission sensitive. For example, analyses determined that GLA and RC systems were unnecessary, but a low level mission which would increase the probability of encountering larger gust intensities could reverse these results.

6. A CCV is an airplane the design of which is based on

    a. The waiver of the free airframe logitudinal static stability requirement.

    b. The use of control systems to perform new tasks such as MLC, FMC, GLA and RC.

7. Although active controls were included in the preliminary design stage, the preliminary design process for the CCV is standard in that, first, the airframe is statically designed after which active control systems are designed.

8. New handling quality criteria are needed because a demarcation between the short period and phugoid modes is lacking at some flight conditions for the RSS airframe.

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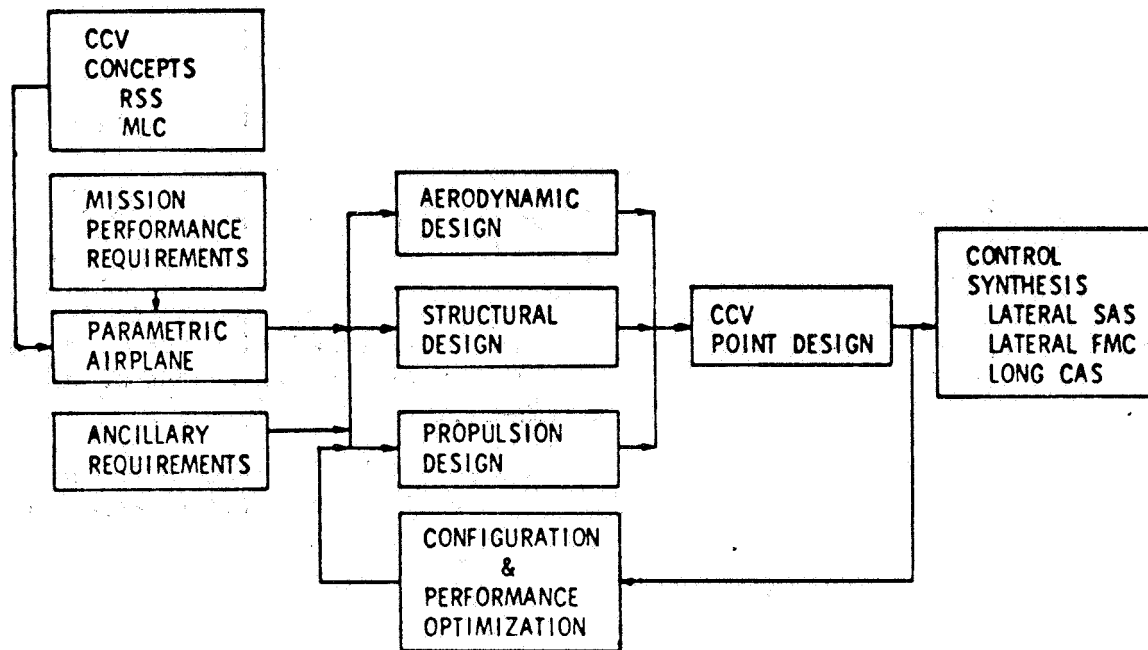
Figure 1.- CCV design process
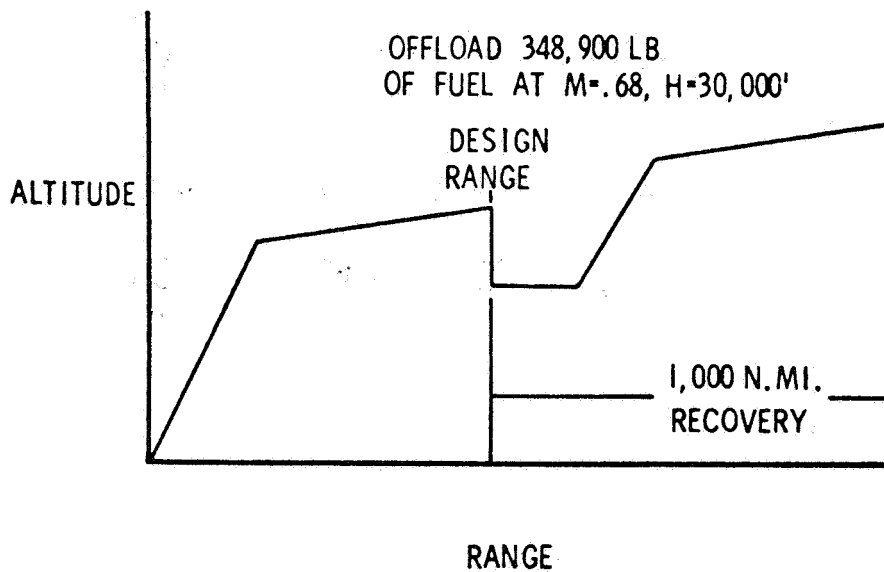
Figure 2.- Mission Profile

FLUTTER MODE
ACCELEROMETER
LOCATION

FLAPERON

ELEVON 1

LEADING EDGE
SLAT
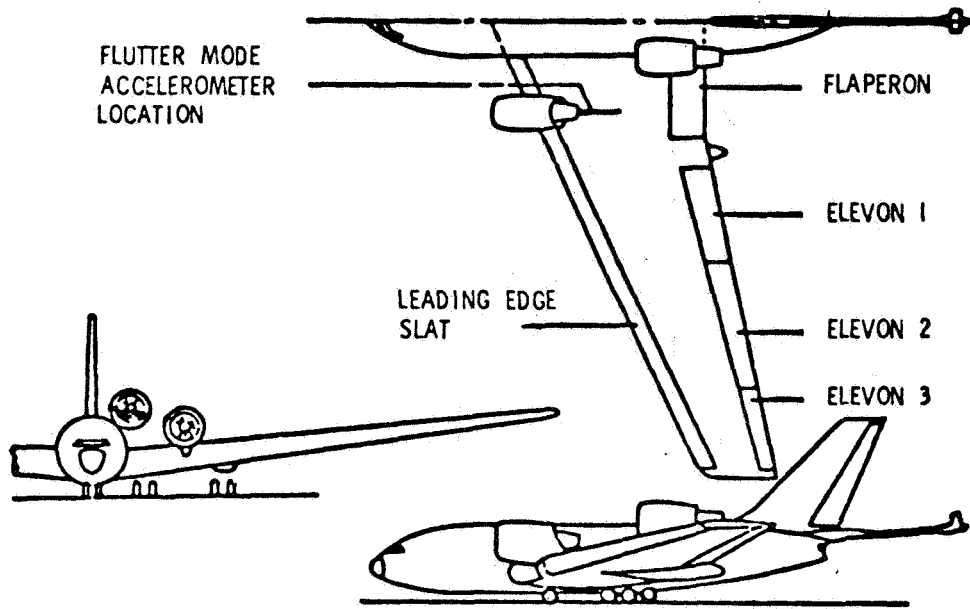
ELEVON 2

ELEVON 3

Figure 3.- CCV point design

BL 300

BL 767

CONVENTIONAL
CCV

50.8

59.3

138.3'

236.8'
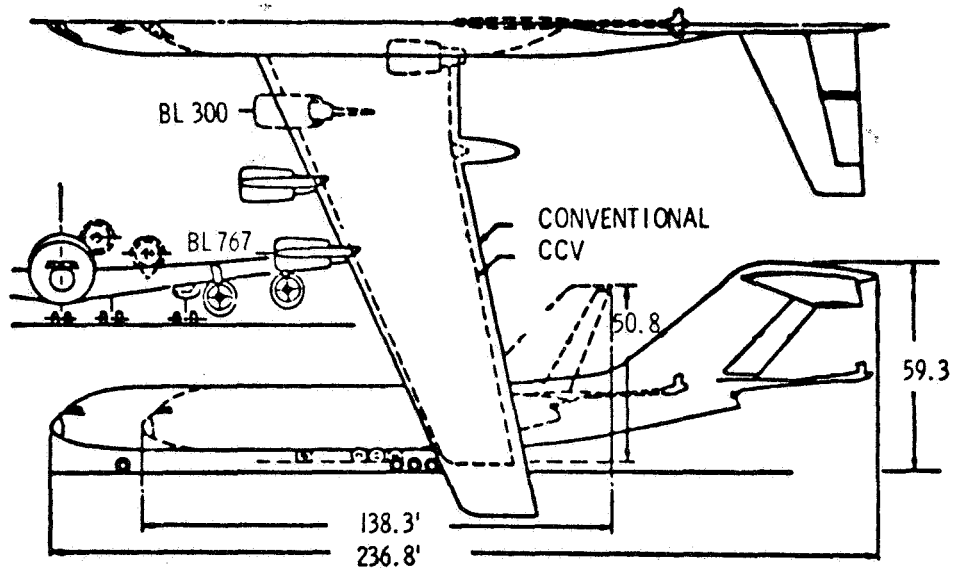
Figure 4.- Size comparison of CCV and
conventional tanker configurations.

STRUCTURAL MODEL

AERODYNAMIC MODEL
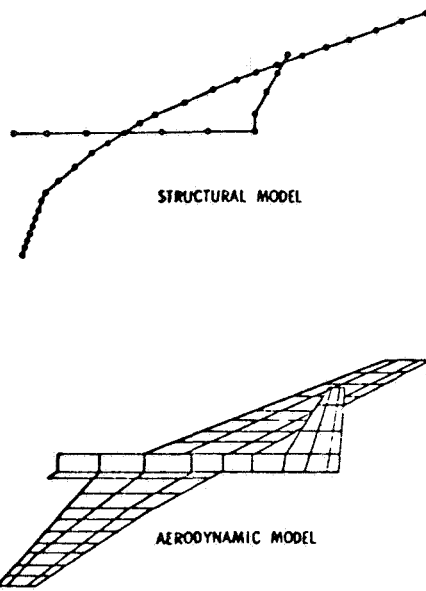
Figure 5.- Finite element models.
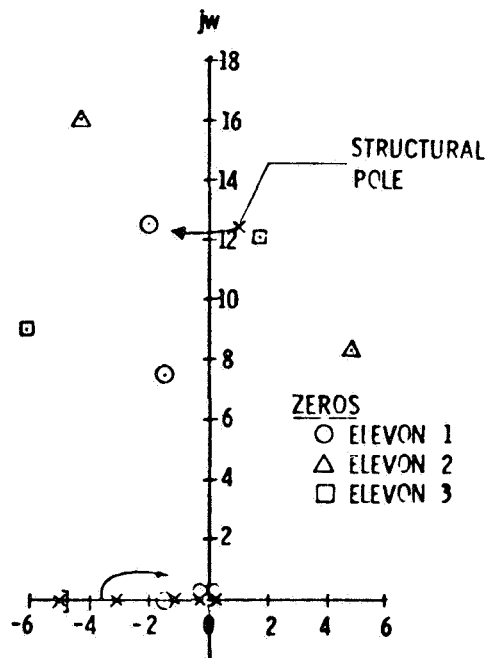
Figure 6.- Flutter control root locus with yaw
damper and alternate elevon zeros.
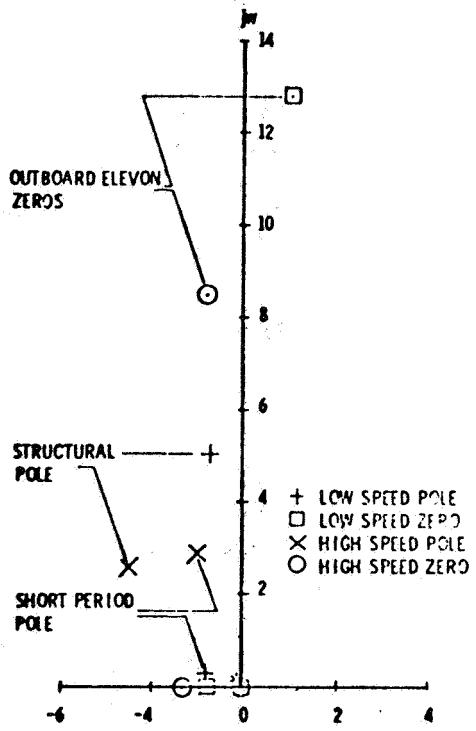
Figure 7.— Effect of speed on short period-
structural coupling.
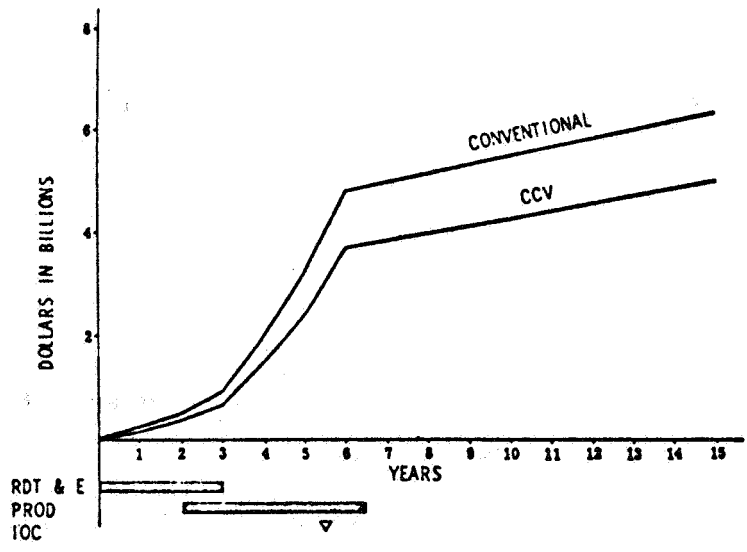
Figure 8.— Life cost comparison.

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STUDY OF AN ACT DEMONSTRATOR

WITH SUBSTANTIAL PERFORMANCE IMPROVEMENTS

USING A REDESIGNED JETSTAR

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and

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SUMMARY

A study has been made of the feasibility of modifying a JetStar airplane into a demonstrator of benefits to be achieved from incorporating active control concepts in the preliminary design of transport type aircraft. Substantial benefits are shown in terms of fuel economy and community noise by virtue of reduction in induced drag through use of a high aspect ratio wing which is made possible by a gust alleviation system. An intermediate configuration was defined which helps to isolate the benefits produced by active controls technology from those due to other configuration variables.

INTRODUCTION

Active controls is a developing technology which could offer substantial payoffs for the air transport industry. Three aspects must be developed before active controls is ready for application. These are: highly reliable fly-by-wire systems, implementation of active control functions, and integration of the active control system into the airframe preliminary design process. The first of these, fly-by-wire, is being adequately addressed in several programs such as the F-8 Digital Fly-By-Wire (DFBW) (refs. 1 and 2). The second aspect, implementation of active control functions, is progressing rapidly in programs such as the B-52-CCV flight tests (ref. 3). Although for single design points, the flight tests have validated the procedures and modeling techniques used in the designs. Active control functions are also being introduced into operational aircraft in order to expand aircraft capabilities. For example, the C-5A Lift Distribution Control System (ref. 4) reduces wing fatigue.

Limited uses of active controls are also finding their way into initial designs to improve performance. For example, a relaxed static stability system is part of the basic YF-16 augmentation system (ref. 5).

The third aspect, integration of active control systems into the preliminary design, has not progressed as rapidly. It is only through the leverage of resizing the airframe that maximum performance benefits are possible. The ATT system studies (ref. 6) included active controls in their integrated preliminary designs, but the designs were never implemented and flight tested, thus verification of the predicted benefits was not possible.

Active controls, then, is clearly emerging as a viable technology for certain airplane applications. Whether or not it will provide realizable benefits for civil transport aircraft is unclear. There is a serious lack of flight verification that promised performance benefits are actually achievable for transports. Recognizing this situation, the NASA is considering various approaches for demonstrating the benefits possible from ACT in a way that would develop confidence within the air transport community. One approach being considered is to redesign, modify, and flight test an existing jet transport to determine the ACT benefits.

This paper presents the results from a feasibility study into the reconfiguration of a Lockheed JetStar, making full use of active controls in the redesign, in order to minimize fuel requirements. The emphasis was on the integration of active controls into the preliminary design in order to maximize performance benefits. In order to more effectively integrate the various aspects of active control, a digital fly-by-wire system was assumed to be available for system implementation.

SYMBOLS

AR   aspect ratio

C   wing chord

$C_{L_{CR}}$ cruise lift coefficient

C_{L_α} lift curve slope, per radian

L/D   lift-to-drag ratio

n_z normal acceleration, g

S   wing area, feet^2

W   weight, pounds

Λ wing sweep angle, degrees

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STUDY FORMULATION

Study Objectives

The feasibility study summarized in this paper had as its primary objectives to determine whether substantial performance benefits could be shown from a synergistic redesign of the JetStar airplane utilizing Active Controls Technology (ACT) concepts, to quantify these benefits, and to direct the configuration development toward the most substantial benefits possible in the reduction of fuel consumption. The utilization of other advanced technologies was encouraged if the interaction would enhance active control system benefits. This latter objective was directed primarily at supercritical wing technology, since it was considered an important aspect in order to make the study results applicable to future transports. An assessment was then to be made of the applicability of these benefits to transport class aircraft in general.

Ground Rules

The most important consideration was to minimize fuel consumption. The Model 1329-6A JetStar was to be used as the baseline aircraft to which modifications would be made. The design was to adhere to the following ground rules:

(1) Maintain current design cruise Mach number (0.82).

(2) Maintain or improve long range cruise speed, ride qualities, handling qualities, range, and payload.

(3) Limit redesign to the wing and empennage. Avoid major redesign to the fuselage and related subsystems.

(4) Assume the availability of a full-time digital fly-by-wire system with a reliability equivalent to that of the basic aircraft structure.

(5) Restrict new technologies considered to those that will be ready for production application by 1980.

REDESIGN STRATEGY

The approach used in the JetStar redesign was to increase the lift-to-drag ratios in all flight regimes by exploiting the use of active controls technology. Increased L/D in takeoff, climb, cruise, approach, and landing produced a direct reduction in fuel required. Summarized in figure 1 are the major elements of the redesign strategy. Beginning with the reference JetStar aircraft, the first step was the application of supercritical wing technology in the redesign of the wing. The higher wing thickness ratios at a given cruise Mach number obtainable for supercritical airfoil sections offer the possibility of achieving adequate mission fuel volume inside

the wing. The redesign then followed two separate paths, one leading to an Intermediate Configuration without ACT and the other leading to an ACT Configuration which made maximum use of an active control system in addition to supercritical wing technology. The reason for defining these two configurations was to isolate the contribution to improved performance due to ACT alone. The rationale utilized in the ACT Configuration evaluation followed the sequence given on the right side of figure 1:

With the supercritical wings, reduce the wing sweep angle and increase the aspect ratio to reduce the induced drag.

Maximize the increase in aspect ratio and minimize any resultant weight penalty by the use of ACT to control acceleration response and wing bending moments.

Resize the empennage by the use of ACT relaxed static stability.

Wing Optimization

Some of the effects of the wing redesign process which were anticipated are illustrated in figure 2 for no ACT and ACT. The general trends in L/D, wing root bending moment, and fuel consumption are shown for variations in aspect ratio. Other parameters such as wing thickness, sweep, and area affect the performance as well, but were expected to have a lesser effect than aspect ratio. It is seen that L/D would be improved at the same aspect ratio as that for the JetStar by the deletion of the external tanks. Increase in aspect ratio should then provide major improvements in wing efficiency. However, as seen in the second graph, an increase in wing root bending moment at the same aspect ratio as the JetStar would accompany the higher lift curve slope of the supercritical wing section. Further increases in aspect ratio would incur substantial increases in bending moment and would be reflected in increased wing weight. An ACT system which reduces bending moment offers the potential for sizable reductions in wing weight, which would be reflected in reduced fuel consumption. This simplified description suggests that optimized wings would have aspect ratios of approximately 7 for no ACT and approximately 9 for ACT. A more detailed examination including all of the wing parameters, the various practical constraints, and the ACT system burden was necessary to see if the initial estimates of aspect ratio and fuel consumption were attainable.

Wing/Fuselage Mating Constraints

If a JetStar were to be used as an ACT demonstrator, several geometrical constraints would be necessary in order to minimize modification costs relative to the fuselage and major subsystems. Major constraints would be the preservation of spar attach points, the main landing gear attachment structure, and stowage provisions for the gear indicated in figure 3 by the heavy lines. An indication of the impact of these constraints on two candidate wing sweeps (0° and 20°) can be seen in the figure. At 20° sweep the gear support structure occupies the area of the inboard flap panel and there is insufficient depth to house the gear. At 0° sweep the gear is

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accommodated, but there is a severe angle in the rear spar. From these two candidate wings, the impact of the wing/fuselage mating constraints is seen to be strongly dependent on the specific wing configuration being considered. The design would require an iterative process in which a candidate wing geometry is selected on the basis of performance considerations. It would then be examined from the standpoint of geometrical constraints. If it did not meet these constraints, a different configuration would be considered.

INTERMEDIATE CONFIGURATION

Selection of the Intermediate Configuration without benefit of ACT was heavily dependent upon matching the ride quality of the basic JetStar. Analysis showed that the worst case for the JetStar was in high speed descent. The criterion used in this study was a ride comfort index which was based on acceleration response and was proportional to wing lift curve slope divided by wing loading. To satisfy ride quality requirements, therefore, the Intermediate Configuration had to have a relatively low aspect ratio wing with moderate sweep to reduce the lift curve slope and gust sensitivity. Trade studies showed that a wing sweep of 30° provided satisfactory lift curve slope and fuel volume capability. The matrix of candidate wing geometries for the Intermediate Configuration is shown in figure 4 plotted against the ride comfort index normalized with respect to the JetStar. The configurations shown have a wing sweep of 30°, aspect ratios from 4 to 6, and wing loading represented by cruise lift coefficients from 0.30 to 0.40. For ride qualities equal to or better than those of the JetStar, the range of possible configurations varies from aspect ratio = 4, $C_{L_{CR}}$ = 0.34 to aspect ratio = 6 where wing loading must be increased to an equivalent $C_{L_{CR}}$ of approximately 0.40.

The selection of the Intermediate Configuration is summarized in figure 5. The carpet plot shows fuel consumption in pounds per nautical mile plotted as a function of aspect ratio and cruise lift coefficient for the required 1850 nautical mile range. The data are provided for a constant wing sweep of 30°. Boundary curves superimposed on the carpet are for fuel volume, ride comfort, and rear spar location. Those boundaries result in a small range of feasible configurations which satisfies all requirements. The selected configuration has an aspect ratio of 5 and will cruise at a lift coefficient of 0.38. The fuel consumption is approximately 8 percent lower than that of the JetStar based on fuel used to accomplish the mission.

A plan view of the Intermediate Configuration is given in figure 6. The wing has an aspect ratio of 5.0, a sweep of 30° at the quarter chord, a wing area of 490 square feet, and a thickness-to-chord ratio of 16 percent at the mean aerodynamic chord. No change in the basic JetStar empennage is required for this configuration. These characteristics compare to those of the basic JetStar which has an aspect ratio of 5.27, a sweep of 30°, a wing area of 542 square feet, and a thickness-to-chord ratio of 11.2 percent.

ACT CONFIGURATION

Results from parametric studies to determine a candidate ACT Configuration are given in figure 7. The carpet plot of fuel consumption for candidate configurations for a sweep angle of 5.5° is based on a match of the cruise segment range requirement of 1850 nautical miles. Wing/fuselage mating constraints were satisfied at this wing sweep. Cruise altitude is assumed to be constant at 40,000 feet. All candidate configurations shown on the carpet plot satisfy the ride comfort criterion. Selection of the ACT Configuration is obtained from the intercept of a line representing adequate fuel volume and a value of minimum fuel consumption which is achieved with an aspect ratio of 9 and a cruise lift coefficient of 0.38. This gives a fuel consumption figure of 5.5 pounds per nautical mile. The start-of-cruise wing loading is 60.4 pounds per square foot for the ACT Configuration compared to 65.5 pounds per square foot for the JetStar. The ride comfort index of the ACT Configuration is 69 percent of the value of the JetStar and the Intermediate Configuration, which is a substantial improvement in acceleration response to turbulence.

Loads Analysis

The limited scope of this feasibility study necessitated restricting the loads investigations wherever possible; accordingly, a single flight condition was selected as being typical of the likely design condition. The condition selected was the cruise speed case (350 knots) at 20,000 feet altitude which represents a suitable datum; the effects of the major increase in wing lift curve slope of the supercritical wing also peak at about this cruise Mach number of 0.78.

The decision was then made to base the gust analyses on the discrete gust case. The short duration of the study did not permit comprehensive spectral density analyses of the several configurations envisaged, and the nonlinearities due to control system limitations (hinge moment, authority, and rate) were likely to be more significant at the larger gust velocities. Hence, the FAR 25 gust of 50 feet per second with a (1 - cosine) profile over a length of 25 chords was selected as the study basis. The overall lift-curve-slope value was 10.2 per radian for the ACT Configuration.

Some results of the loads analyses given in figure 8 show wing root bending moments for both maneuver and gust load conditions. Values for the JetStar airplane are noted by the symbols. The results show that gust loads are more critical for the aspect ratio 9 wing than those due to maneuver conditions. Studies of aircraft response to gusts with various gust alleviation system characteristics resulted in the selection of full-span trailing-edge flaps with a flap actuation rate of 60 degrees per second as the most effective system. The results of dynamic load response at a flap control rate of 60 degrees per second show a reduction in flexible wing root bending moment from 13.2×10^6 in-lb to 9.4×10^6 in-lb, which is of the same magnitude as the rigid wing with no ACT but is over twice the value for the JetStar. The impact of a wing with higher root bending moment than the JetStar is the need for a sizable wing carry through structure. A doubling of the bending moment is near the practical limit for increasing the strength of this carry through structure.

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The results further indicate that maneuver loads are relatively insignificant, and no appreciable benefit would result from the incorporation of a maneuver load control system for this particular configuration. The amount of load alleviation obtainable for the flexible aircraft is much less than that for the rigid aircraft; therefore, the dynamic structural response must be included in any control system analysis. Preliminary flutter analyses, conducted to establish the torsional stiffness required for flutter and divergence prevention, revealed no apparent problems.

The time history of the root bending moment response given in figure 9 shows the substantial reduction of the initial gust load peak as a result of the ACT system. There is little effect on the second (negative) load peak. The gust-induced peak load occurs at about 0.2 second after entering the gust, which is long before any overall pitch response can occur. The basic objective of the active control system, therefore, is to destroy this lift, rather than to change the angle of attack. In an up gust, an upward flap deflection is required together with a proportional downward elevator deflection to counteract the pitching moment. The rapidity of the gust velocity buildup requires the high flap rates discussed previously.

ACT System Burden

The design to this point has assumed the availability of an ACT system; however, the penalty for providing such a system must also be assessed to determine practicality. Ideally, a relationship between system burden and system capability could be established for incorporation into the wing definition process. Unfortunately, this relationship is not easily defined, as illustrated in figure 10. Complexity and weight, indications of the system burden, are shown as a function of control system capability. This relationship is difficult to quantify; thus, only a subjective indication of increasing system penalty with increasing capability is shown.

Lacking a well-defined relationship, an ACT system configuration was assumed which would meet the needs of the design. Figure 11 itemizes the major features for this system and indicates where it might be placed on the penalty versus capability plot, specifically for weight as the penalty and reduction in bending moment as the capability. The gust alleviation portion of the ACT system involves five trailing-edge surfaces and actuators on each half of the wing. These surfaces serve as high-lift devices in addition to the active control function. The surfaces are pivoted at the 75-percent chord. The angular displacement limits for active control are ±20° in all segments. Each surface segment is supported on three hinges, and the segments are operated by dual-tandem hydraulic actuators. For a 29-percent reduction in root bending moment corresponding to an actuator rate limit of 60 degrees per second, the system would weigh approximately 370 pounds and would require approximately 6 gallons per minute hydraulic flow capacity. This burden was judged to be reasonable from practical considerations.

ACT General Arrangement

A plan view of the ACT Configuration selected is shown in figure 12. The characteristics of the ACT wing necessary to satisfy the objectives of this study consist of

an aspect ratio of 9.0, a wing sweep of 5.5°, a wing area of 560 square feet, and a wing thickness-to-chord ratio at the mean aerodynamic chord of 12.7 percent.

The horizontal tail has been reduced in size by 40 percent as compared to the tail of the basic JetStar. Of this reduction, 75 percent is made possible by a smaller tail size requirement for the ACT Configuration wing to achieve the same stability level as that of the basic JetStar. The remaining reduction is made possible by a relaxed static stability system.

Plan views of the Intermediate Configuration and the ACT Configuration are compared in figure 13. Large differences are apparent in aspect ratio, wing sweep, and horizontal tail size. Both aircraft have supercritical wing sections. The Intermediate Configuration has no active control technology applied. As stated earlier in the section on REDESIGN STRATEGY, in order to isolate the benefits attributable to ACT, all comparisons of performance were made between the Intermediate Configuration and the ACT Configuration.

COMPARISONS

Comparison of Weights

A comparison of weight buildup for the JetStar, Intermediate Configuration, and ACT Configuration is presented in figure 14. The major differences occur in the wing weight and mission fuel components. The Intermediate Configuration wing is about 700 pounds heavier than the JetStar wing, primarily because of higher root bending moments resulting from use of the supercritical wing. The wing weight of the ACT Configuration is almost identical to that of the Intermediate Configuration, but it should be noted that for an aspect ratio of 9.0, a considerable penalty would have been incurred without the benefits of active controls in reducing root bending moments. The mission fuel requirement is shown to progressively decrease from the JetStar to the ACT Configuration as a result of the improved lift-to-drag ratios of the Intermediate Configuration and the ACT Configuration. Finally, small changes in systems weight are reflected in the "miscellaneous" block, and the ACT Configuration benefits from a 266-pound reduction in horizontal tail weight because of its smaller size. The takeoff gross weight is 38,378 pounds, 37,821 pounds, and 35,470 pounds for the JetStar, Intermediate Configuration, and ACT Configuration, respectively. Although the takeoff gross weight has been reduced a small amount, this is a side effect of the most important consideration of the study—minimization of fuel consumption.

Comparison of Fuel Usage Benefits

The benefits in fuel consumption were derived from the difference between the Intermediate Configuration and the ACT Configuration. The fuel required to accomplish the mission (less reserves) was used to calculate these benefits. Thus the reduction in fuel consumption of the Intermediate Configuration over that of the

JetStar is 8 percent, and the ACT Configuration reduction over the JetStar is 27 percent. The direct benefit of active control technology, i.e., the ACT Configuration over the Intermediate Configuration, is 20 percent.

Comparison of Fallout Benefits

An analysis of performance characteristics under FAR 36 rules indicated that the use of an active control system would reduce approach noise by 6 EPNdB and takeoff flyover noise, under cutback power, by 8 EPNdB. The benefits in terms of community noise would result directly from the increase in lift-to-drag ratio in the high-lift configuration. The application of active controls technology would improve the ride comfort by 31 percent.

CONCLUSIONS

A study has been made of the feasibility of modifying a JetStar aircraft to demonstrate benefits which may be achieved through active controls. The specific conclusions of the study are:

(1) A 20-percent reduction in fuel consumption was attributable to active controls.

(2) No penalty was incurred in any other performance parameter in order to achieve a fuel consumption benefit.

(3) Additional benefits in the reduction in community noise and improved passenger ride qualities were indicated.

(4) The general relationship between control system burden and capability was not readily attainable. For the specific gust loads alleviation and relaxed static stability system studied, the burden was judged to be reasonable from practical considerations.

PROJECTIONS FOR NEW DESIGNS

The applicability of the results of a feasibility study of this type to transport class aircraft in general is difficult to assess, but it is felt that some generalizations are in order. The results of this study are consistent with those of other similar studies, such as the ATT system studies, in that we can expect benefits in transport aircraft performance from incorporating ACT in the design. It should be noted, however, that the performance increment for a new design transport is uncertain. It would be erroneous to assume that the magnitude of the benefits obtained in this study would be realized in all new transport designs. ATT studies showed the ACT benefits to be highly configuration sensitive. In general, the design strategy employed for a new ACT transport would be essentially the same as that used in

this study, which includes wing optimization to satisfy system requirements of fuel volume, ride quality, and stability and control. There is a need for a more realistic definition of a ride quality criterion, since this is an important design parameter for ACT aircraft.

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REDESIGN STRATEGY

JETSTAR

APPLY SUPERCRITICAL WING TECHNOLOGY

INTERMEDIATE CONFIGURATION

REDUCE WING SWEEP
INCREASE ASPECT RATIO

MAXIMIZE ASPECT RATIO
MINIMIZE WEIGHT BY ACT

RESIZE EMPENNAGE BY ACT
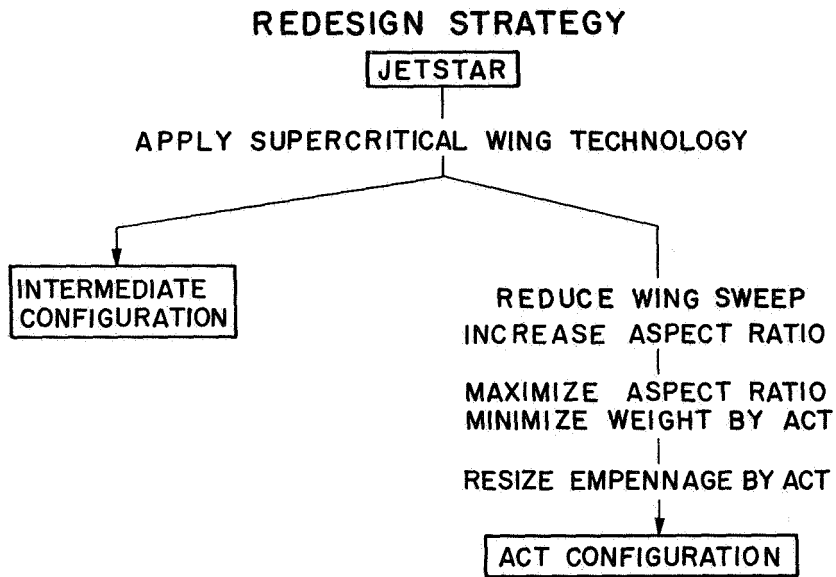
ACT CONFIGURATION

Figure 1

WING OPTIMIZATION

L/D MAX

⊗ BASIC JETSTAR

ROOT BENDING MOMENT IN. LB. x 10^{-6}

NO ACT

ACT

FUEL CONSUMPTION LB/NM

NO ACT

ACT
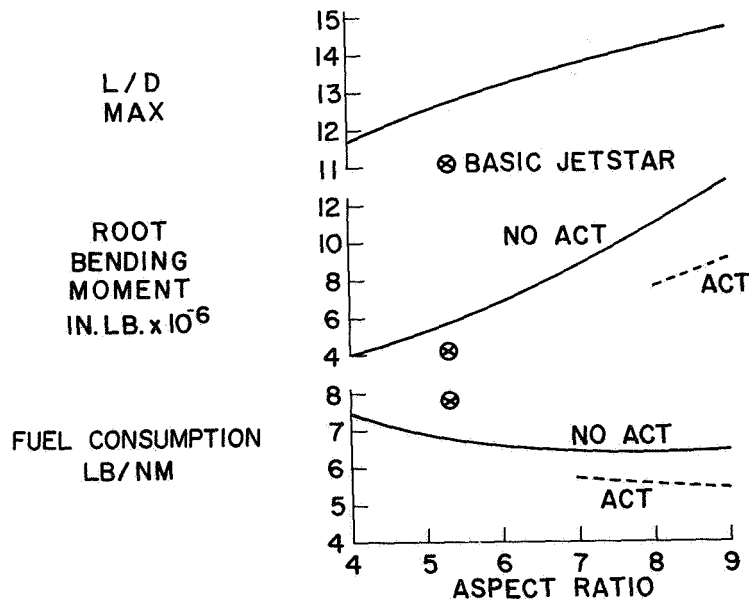
ASPECT RATIO

Figure 2

WING - FUSELAGE GEOMETRY CONSTRAINTS

SWEEP ANGLE = 0°   SWEEP ANGLE = 20°

65%C 75%C   75%C 80%C
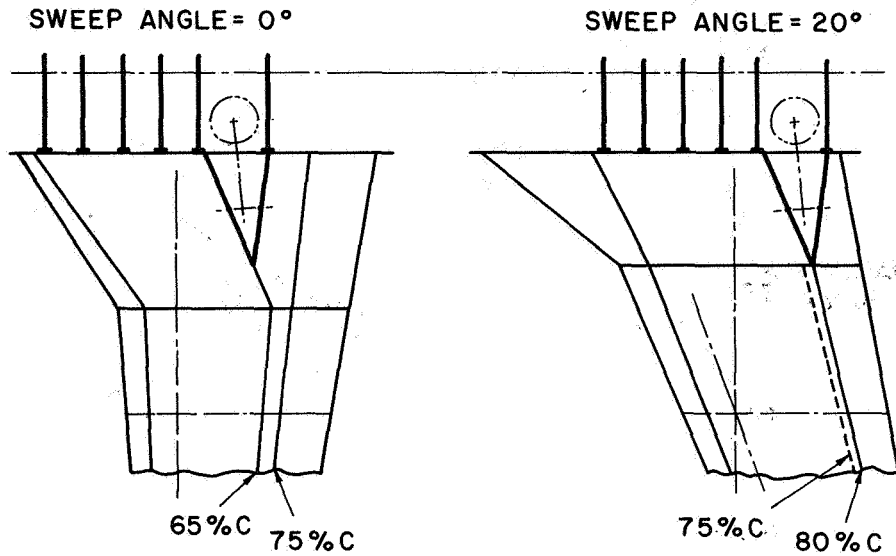
Figure 3

RIDE COMFORT FOR COMBINATIONS OF ASPECT RATIO AND CRUISE LIFT COEFFICIENT

$$\text{RIDE COMFORT RATIO} = \frac{\dfrac{C_{L_\alpha}}{W/S}}{\left(\dfrac{C_{L_\alpha}}{W/S}\right)_{JETSTAR}}$$
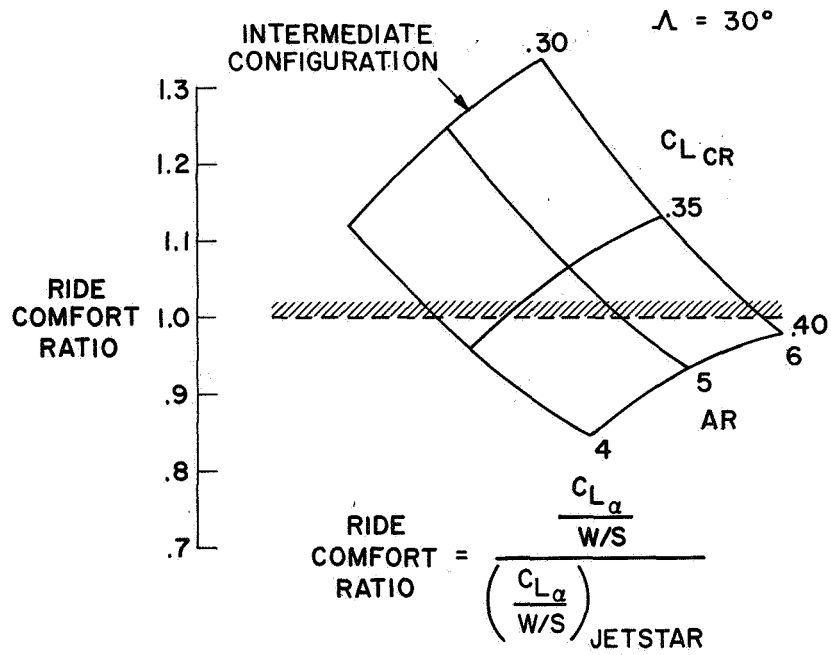
Figure 4

INTERMEDIATE CONFIGURATION SELECTION
$\Lambda = 30°$
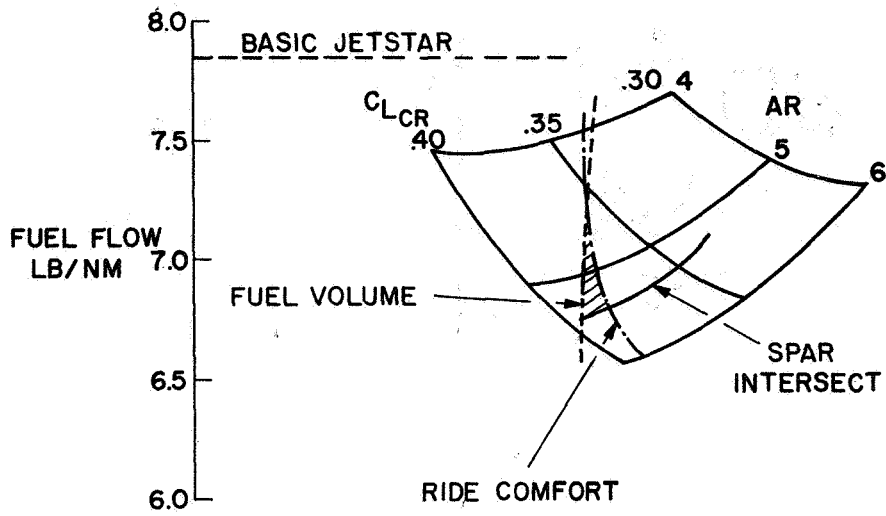
Figure 5

INTERMEDIATE CONFIGURATION

AR = 5.0
Λ = 30°
S = 490 SQ.FT.
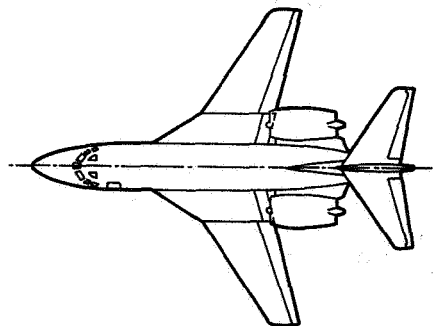
Figure 6

ACT CONFIGURATION SELECTION

$\Lambda = 5.5°$
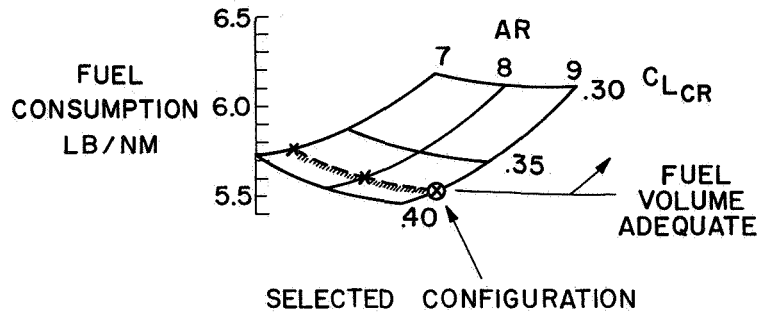
Figure 7

EFFECT OF ACT ON WING BENDING MOMENT
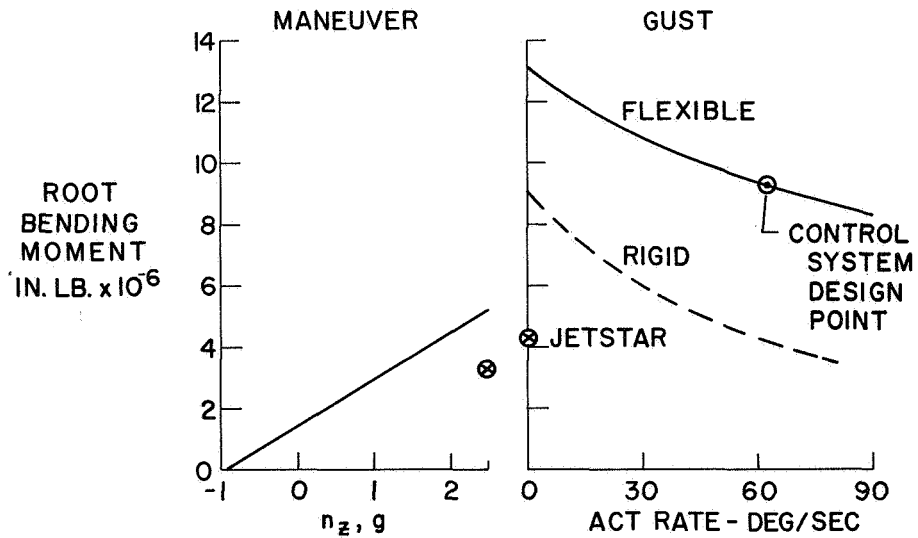
Figure 8

BENDING MOMENT DUE TO DISCRETE GUST
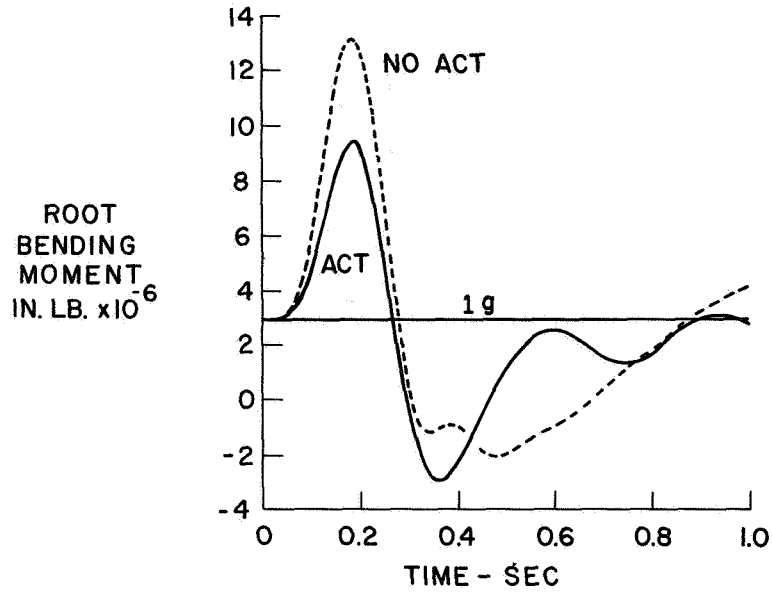
Figure 9

ACT SYSTEM BURDEN
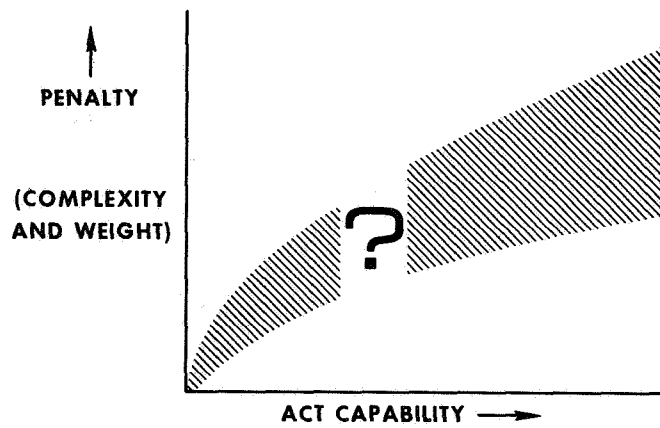
Figure 10

EXAMPLE FOR GUST ALLEVIATION

CANDIDATE SYSTEM

    10 DUAL-TANDEM ACTUATORS

    60 DEG/SEC RATE LIMIT

CAPABILITY

    29 PERCENT LESS ROOT BENDING MOMENT

PENALTY

    370-POUND SYSTEM WEIGHT

Figure 11

ACT CONFIGURATION

AR = 9.0
Λ = 5.5°
S = 560 SQ. FT.
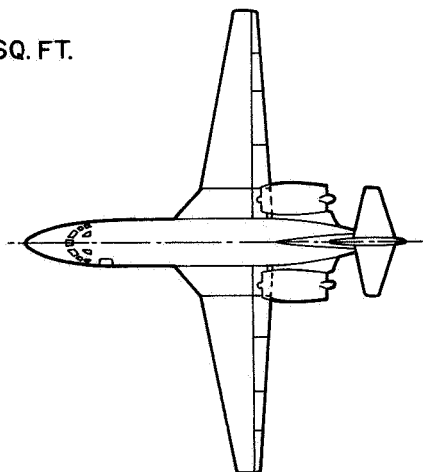
Figure 12

AIRPLANE CONFIGURATIONS

INTERMEDIATE
AR = 5.0
Λ = 30°
S = 490 SQ.FT.

BASELINE ACT
AR = 9.0
Λ = 5.5°
S = 560 SQ.FT.
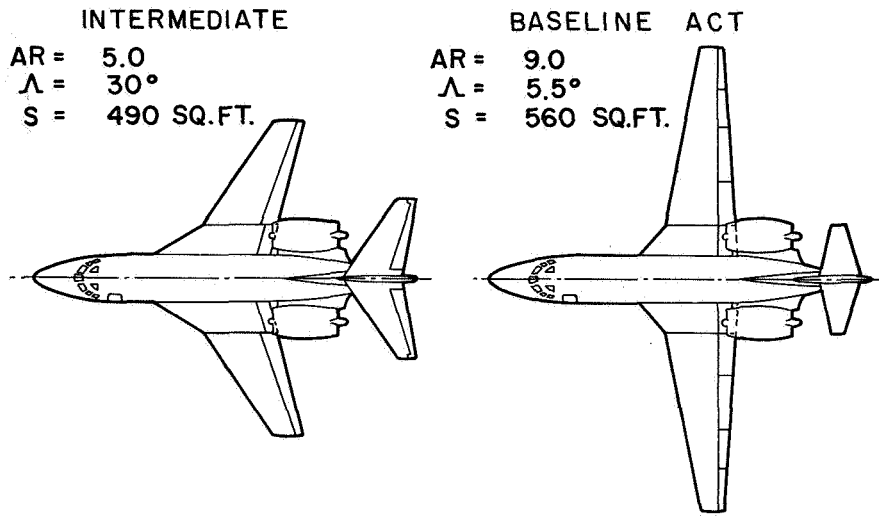
Figure 13

COMPARISON OF WEIGHTS
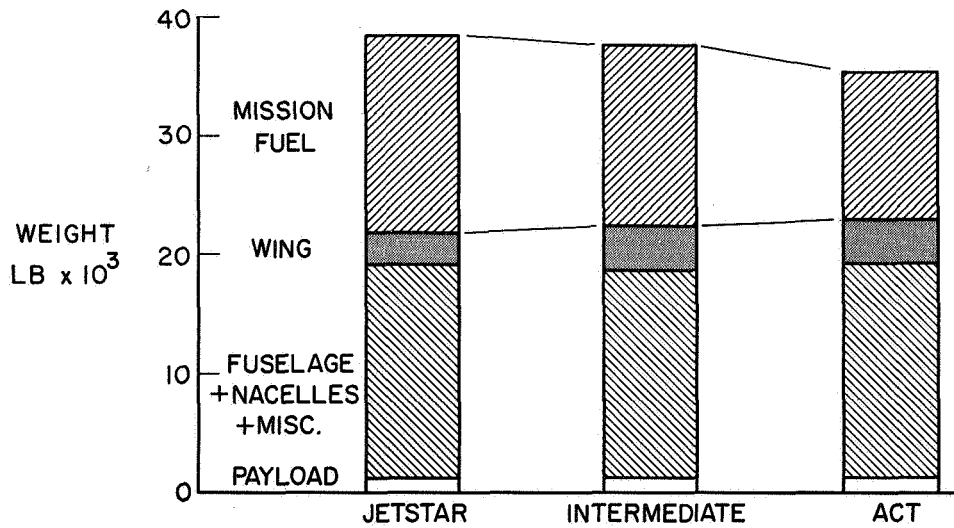
Figure 14

A SUMMARY OF THE APPLICATION OF ACTIVE CONTROLS TECHNOLOGY

IN THE ATT SYSTEM STUDIES

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SUMMARY

The application of active controls technology to subsonic, long-range transport aircraft was investigated in three Advanced Transport Technology system studies. Relaxed stability requirements, maneuver and gust load alleviation, and active flutter suppression were the concepts considered. A different configuration was investigated for each of the three airframe manufacturers, and each had a somewhat different approach to the application of active controls technology. Consequently, the results varied in magnitude between the contractors, but several trends were noted. Relaxed stability requirements resulted in the largest benefits - reduced weight, increased return on investment, and decreased direct operating costs. Maneuver load alleviation, gust load alleviation, and flutter suppression resulted in much smaller benefits. Prior to application of active controls technology, a research and development program directed toward fulfilling data base requirements, establishing effective design techniques and criteria, improving systems maintainability and reliability, and demonstrating technology readiness must be completed.

INTRODUCTION

In mid-1970, NASA initiated an Advanced Transport Technology (ATT) Program directed toward defining and developing advances in technology which would contribute to a superior subsonic long-haul transport aircraft. The Langley Research Center played a lead role in carrying out the airframe technology portion of this program. Systems studies were initiated with three airframe contractors early in the program. These were Boeing, General Dynamics-Convair, and Lockheed-Georgia. Subsequently, assessments from the airline viewpoint were made by United and American Airlines.

The major objectives of the systems studies were to

- Incorporate projected advances in aerodynamics, structures and materials, flight controls (including active control concepts), avionics, propulsion, and auxiliary systems into conceptual configurations

- Identify and quantify the potential benefits and costs of the technology advances

- Define and recommend research activities required to bring the advanced technologies to a state of readiness for commercial application by the end of this decade

The purpose of this paper is to broadly summarize the results and recommendations of the system studies which are pertinent to the application of active controls technology. A brief synopsis of the various approaches and the constraints encountered during the course of the studies is included in order that the benefits might be better understood. The reader is encouraged to obtain the listed references if interested in more details. The August 1972 issue of the Astronautics and Aeronautics (ref. 1) provides an overview of the Advanced Transport Technology Program and the airframe manufacturers' final reports are listed as references 2 through 7 of this paper.

CONCEPTUAL CONFIGURATION STUDIES

Each of the airframe companies studied several configurations having varying cruise Mach numbers, ranges, and payloads. Figures 1, 2, and 3 show a representative high-speed configuration from each contractor. Other configurations having design cruise speeds as low as $M = 0.90$ were studied. The Boeing configuration and the General Dynamics configuration are similar in concept, but differ considerably in a number of details. Both are $M = 0.98$, 196-passenger, 3000-nautical-mile design range, three-engine configurations. The primary differences are engine and horizontal tail locations. Lockheed concentrated on a $M = 0.95$, 400-passenger, 5500-nautical-mile design range, four-engine configuration. These are the configurations which will be discussed for the remainder of the paper.

In arriving at the above configurations, a "baseline" aircraft was defined which incorporated relaxed stability requirements in order that the best cruise performance (lowest trim drag) might be obtained without regard to maintaining inherent stability requirements. These baseline configurations had airframes designed for 100-percent strength and stiffness. The configurations were then examined to determine the applicabiltiy of gust and maneuver load control, ride quality control, and active flutter suppression.

Each of these functions were examined to identify potential benefits, system functional design, cost, and weight. Benefits associated with relaxed stability requirements were identified by "backing-off" to a configuration having conventional stability characteristics.

Relaxed Stability

Boeing's initial longitudinal design philosophy was to select the minimum horizontal stabilizer volume coefficient, V_H, which would provide the required center-of-gravity range as illustrated in figure 4. The balance limits selected in this phase provided that the aft-most center-of-gravity location would be limited to the most-forward maneuver point location encountered in the

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flight envelope. (The maneuver point is defined as that center-of-gravity location at which the stabilizer deflection required for a constant load factor increment becomes zero during a constant speed pull-up.) Ability to trim the aircraft at the landing approach condition with a reasonable tail lift coefficient (C_{L_t} = -0.80) determined the forward center-of-gravity portion.

The early design philosophy resulted in configurations which were unstable in large portions of the flight envelope but stable during cruise, as shown in figure 5. In later phases of the study, it was found that this balance philosophy did not result in the best cruise performance, particularly for a M = 0.98 configuration with two wing-mounted engines. The wing was positioned further forward resulting in a more aft loading envelope which in turn allowed a more aft cruise center-of-gravity position to be maintained. The horizontal stabilizer volume coefficient which provided a compatible aft center-of-gravity limit was found to be larger than the minimum volume coefficient selected earlier, as shown in figure 4.

In sizing the vertical tail, two criteria were considered: a minimum directional stability level (C_{n_β} = 0.002 deg^{-1}) and engine-out control. For the configuration shown in figure 1, the minimum directional stability level was found to be the limiting criterion, based on a two-segment full-span rudder. With the vertical tail sized in this manner, lateral-directional dynamic instability exists over a large portion of the flight envelope, as shown in figure 6. This instability would require a flight-critical augmentation system.

General Dynamics, in investigating relaxed stability requirements, followed a similar, but somewhat different, design philosophy. The configurations investigated were similar in size to Boeing's, with the horizontal tail being a low rather than a high T-tail arrangement and two wing-mounted engines rather than all three aft. Figure 7 illustrates the horizontal tail volume selection for both conventional and relaxed longitudinal stability requirements. In both cases, nose gear unstick and ability to trim in the high lift configuration are considered in establishing the forward center-of-gravity location. For the conventional case, the requirement that the static margin be greater than or equal to zero sets the aft center-of-gravity limit. The criterion selected for the aft center-of-gravity limit in the case of relaxed static stability is the ability to trim the high-speed configuration to a wing-body lift coefficient of 1.0 with a maximum horizontal tail deflection of 15°. This will leave about a 40-percent control power reserve to handle the dynamic aspects of upset disturbances. An operational forward-to-aft center-of-gravity range of 10-percent M.A.C. was maintained. Figure 7 implies that a 25-percent reduction in horizontal tail area may be obtained by employing relaxed longitudinal stability concepts.

Preliminary studies conducted by Genral Dynamics indicate that a further reduction in horizontal tail area (about 20 percent) may be obtained by incorporating a geared trailing-edge control on the all-movable horizontal tail. Balance characteristics of such a configuration are shown in figure 8.

The major impact of this balance concept identified by General Dynamics may be summarized in terms of the changes in structural weight and drag at the trimmed cruise condition. For the Mach 0.98, two-wing and one aft-mounted engine configuration shown in figure 2, the savings are

1. Decreased drag at cruise = 7 counts (0.0007)

2. Structural weight savings due to decreased drag = 690 lb (313 kg)

In backing off to a configuration with conventional inherent stability, General Dynamics determiend only the penalty due to the trim drag increment and did not determine the weight penalty associated with changing the size of the horizontal tail. Thus, the structural weight savings shown are due only to the decreased trim drag and resulting fuel savings.

Implementation of this relaxed stability concept resulted in a configuration which is stable in cruise, but unstable in other portions of the flight profile, requiring an artificial stability system. Figure 9 illustrates a typical flight profile with corresponding values of Mach number and static margin.

Lockheed's ground rules were that their configuration would have a 20-percent M.A.C. center-of-gravity range and a positive static margin of 3-percent M.A.C. Thus, for the configuration shown in figure 3, the forward c.g. limit is constrained by nose wheel lift off, and the aft c.g. limit by the ability of the augmentation system to provide a minimum of 3-percent static stability.

Figure 10 illustrates Lockheed's balance philosophy, assuming an augmentation system with angle-of-attack (α) feedback. Note that as the α gain (K) is increased, the stability line rotates downward. The horizontal tail volume is established by the value of K for which the control system experiences rate or displacement saturation. Using this approach, a reduction in horizontal tail volume of 0.54 was obtained which resulted in a 4.64-percent decrease in ramp weight and a 6.11-percent decrease in required thrust.

The vertical tail sizing philosophy was essentially the same as those of Boeing and General Dynamics.

<center>Load Alleviation and Flutter Suppression</center>

In the application of maneuver load alleviation (MLA), gust load alleviation (GLA), and active flutter suppression (FS), it was found that these functions were not independent and had to be considered at the same time. Each of the contractors included effects of aeroelasticity, multiple load sources, and a number of different flight conditions. Implications of fatigue and ride qualities were also considered in the application of MLA, GLA, and FS.

For MLA, Boeing considered using both inboard and outboard control surfaces to shift the maneuver induced load inboard. Figure 11 shows the potential wing

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box weight savings, considering only strength requirements, in terms of control surface lift and moment capabilities. The wing of the baseline aircraft was shown by analysis to be flutter free up to the required 1.2 V_D. Figure 12 illustrates the impact of removing material by an MLA system on the flutter (stiffness) requirements. The additional material required to prevent flutter is shown as a function of the material removed by the use of MLA. It should be noted that this analysis was based on a configuration with no wing-mounted engines. Configurations with wing-mounted engines would possibly have greater flutter requirements.

A fatigue analysis was then conducted based on the number of ground-air-ground cycles and the percent damage due to gusts. Figure 13 illustrates the additional material required to achieve acceptable (gust-induced) fatigue damage rates as a function of the amount of material removed through the application of MLA. Since the fatigue increment is large relative to the MLA weight reduction, the need for a gust alleviation system to reduce the gust-induced fatigue damage is indicated.

GLA was considered in order to reduce material requirements for fatigue and to improve ride qualities. A center-of-gravity accelerometer feedback driving a wing trailing-edge surface to reduce gust-induced vertical accelerations which operated in conjunction with the pitch control surface to maintain attitude was the control system concept considered. Figure 14 illustrates the results of a two-degree-of-freedom power spectral density gust analysis. Airplane response in terms of root-mean-square center-of-gravity accelerations and the associated flap angles are shown as a function of acceleration feedback gain. A gain of 150 deg/g was selected. Figure 13 shows the amount of material required for fatigue as a function of the material removed when both MLA and GLA are employed.

The application of an active flutter suppression system in conjunction with the MLA and GLA system was also investigated. The control system concept arrived at was an outboard trailing-edge surface responding to a wing-mounted accelerometer signal fed back through a compensation filter. Difficulty was encountered in maintaining stability of both higher and lower frequency airplane modes while controlling the somewhat violent flutter mode at 3.8 Hz. A root locus plot for one of the more promising filter designs is shown in figure 15. Although successful stabilization of the flutter mode was indicated, active flutter control was not included in the final configuration because the added weight due to the control system was approximately equal to the structural weight savings.

General Dynamics considered application of a "wing design load control" to their configuration. This concept was used to reduce wing maneuver loads, as well as gust-induced loads. Implementation concepts which were considered include:

1. Inboard flaperon
2. Outboard spoiler
3. A combination of inboard flaperon and outboard spoiler

Figure 16 illustrates the weight savings obtained through the use of each of the above concepts. Note that the net savings shown are the differences between structural weight reductions and control system weight additions.

Since gust-induced loads were found to be critical on the forward fuselage, further structural weight savings were possible using the inboard flaperon. Reductions in rms gust response all along the fuselage were also found using this concept. Figure 17 illustrates the gust responses at different fuselage locations with and without the active control system. However, since the structural weight savings would not offset the weight associated with the inboard flaperon system and since the unaugmented ride qualities were considered satisfactory, the inboard flaperon was not included in the final results.

General Dynamics also considered the application of active flutter suppression to the configuration incorporating the wing design load control system. Various sensor and control surfaces were considered, in several combinations. Figure 18 shows the degree of damping obtained with several of these control system concepts, as well as the damping for the unaugmented airplane. Two fuel conditions are shown.

Difficulties were encountered in maintaining stability of a higher frequency mode while stabilizing the critical flutter mode. Also, the results shown in figure 18 were based upon feeding back idealized response signals. Figure 19 shows the results of a study on approximating such signals with accelerometers and compensation networks. This work, which was not done on exactly the same configuration as that of figure 18, indicates that when sensors and compensation networks were included, successful stabilization of the flutter mode is not achieved. However, due to a lack of detailed aerodynamic data on supercritical wings with leading-edge (tip) controls, no attempt was made to optimize the proposed flutter suppression system. It was anticipated that successful stabilization could be achieved, but the benefits would be smaller than those predicted assuming ideal feedback signals. The most promising concept appears to be the combination leading- (tip) trailing-edge control system investigated by Nissim (ref. 8).

Fatigue damage calculations were performed to determine the effects of MLA and GLA on the aircraft service life. Three configurations were investigated

1. 100-percent strength without active control system (ACS)
2. 100-percent strength with ACS
3. Reduced strength with ACS

Fatigue damage rates were calculated for two wing stations and two fuselage stations. Figure 20 summarizes the results for the three configurations. Damage rates which caused gust, maneuver, and ground-air-ground cycle are presented and all values are normalized to the 100-percent strength without ACS configuration.

Lockheed investigated the application of MLA, GLA, and active FS to the configuration shown in figure 3. They found the use of MLA and GLA for reducing peak loads to be inappropriate for their configuration. The maximum allowable

wing-bending deflections were limited by ground clearance during rough surface taxi and the maximum dihedral for acceptable stability and control during cruise. Thus, the wing of the large, four-wing-mounted engine configuration was bending-stiffness critical and no benefits were obtained from the application of MLA and GLA.

In investigating possible application of an active fatigue load alleviation system, it was found that for this configuration, the ground-air-ground cycle was the major source of wing fatigue damage. Thus, it was concluded that the benefits of a fatigue load alleviation system in reducing the fatigue damage on their recommended configuration was negligible. 〉

An active flutter suppression system which would be used only for that portion of the flight envelope between V_D and $1.2 V_D$ was considered. This was done in view of the catastrophic nature of most main-surface flutter instabilities and the low probability of making a first-generation flutter-suppression system absolutely reliable. A flutter analysis was conducted and it was found that approximately 575 lb (260 kg) of stiffness material could be removed in lowering the flutter speed from $1.2 V_D$ to V_D. An active flutter suppression system was not synthesized; however, the weight of such a system was estimated and found to be about 320 lb (145 kg). Thus, a maximum net structural weight saving of about 255 lb (115 kg) per aircraft was indicated. In resizing the aircraft, this becomes a 500-lb (227-kg) or a 0.17-percent reduction in operating weight. Lockheed concluded that these benefits would not justify the added cost, complexity, and risk of an active flutter suppression system for their recommended design.

SUMMARY OF RESULTS

The benefits of interest are weight savings and economics for the selected configurations. In some cases, there is a fairly wide spread in the benfits indicated by the various contractors since they investigated different configurations and had different basic ground rules. The high-speed configurations discussed in this paper were found to benefit more from ACT than did the lower speed configurations investigated in the system studies. The fact that the benefits of active controls are dependent on the configurations being studied is well recognized. Research and development recommendations of the contractors do agree quite closely. The recommendations presented herein are general and somewhat broad in scope. For the more detailed, task-level recommendations, the reader is referred to references 3, 5, and 7.

Benefits

Figure 21, from the Boeing study, shows the changes in configuration resulting from the application of active controls. Comparing the conventional technology airplane to the advanced technology airplane with active controls, one can see the differences in horizontal and vertical tail areas. Figure 22

summarizes the weight benefits attributable to the application of active controls as predicted by the contractors. Some caution should be used in examining this figure. Note:

1. Boeing indicates benefits attributable to MLA and GLA and FS in terms of structural weight, not resized aircraft TOGW or OWE.

2. The weight savings shown for RSS by General Dynamics includes no weight savings based on resizing the vertical tail, but is based only on the reduced trim drag.

In addition to weight savings, each of the contractors was able to minimize the trim drag through relaxing stability requirements. This resulted in operational benefits, such as reduced fuel requirements, which will be reflected in Direct Operating Cost (DOC) improvements to be discussed below. None of the contractors included systems specifically for improving the ride qualities, as these were predicted to be adequate. However, reductions in fuselage accelerations of 20 to 40 percent were considered feasible. Fatigue damage rates due to gust and maneuver loads were either improved or at least not increased due to the application of MLA and GLA, as shown in figures 14 and 20. The impact on the ground-air-ground cycle fatigue damage, however, does appear to be detrimental.

Two economic measures were utilized: Return on Investment (ROI) and Direct Operating Costs (DOC). Figure 23 summarizes the percent increase in ROI resulting from the application of active controls. Again, note that the General Dynamics results for relaxed stability include only the effect of reduce trim drag. Boeing used a somewhat different approach in their economics study and did not show the effects of only active controls on ROI or on DOC. Airplane price, which was an input to the ROI calculations, was estimated to be 4.0 to 6.0 percent lower when active controls were used (refs. 4 and 6).

Figure 24 summarizes the findings with respect to DOC, which includes such factors as: maintenance (airframe, engines, avionics, etc.), fuel usage, insurance, and other operating expenses. In general, application of active controls would reduce structural weight which would result in lower maintenance costs for the airframe and engine. However, avionics maintenance costs would increase. Reduced trim drag would result in lower fuel costs. As can be seen, the overall effect of applying active controls was seen to be beneficial in terms of DOC. More comprehensive economic studies have since been completed and are contained in reference 9.

Recommended Research and Development

The recommendations of interest will be summarized under three broad headings:

A. Research and Technology (R&T) Base
B. Integrated Design Concepts
C. Technology Demonstration

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A. Research and Technology Base

   1. Conduct analytical and experimental (wind-tunnel) evaluation of charac-
      teristics of leading-edge and trailing-edge devices designed for opera-
      tion on supercritical wings. Both static and dynamic data are required
      for speeds through the transonic flow regime.

   2. Develop improved aeroelastic methods for flight controls analysis.
   The accuracy of this method should be established by comparisons with
      wind-tunnel and flight test data and the sensitivity of airplane
      balance and flight control design to the accuracy of the method should
      be established.

   3. Develop control laws which are compatible with advanced onboard com-
      puting systems and which maintain effectiveness of the system over the
      entire operating flight envelope.

B. Integrated Design Concepts

   1. Conduct a detailed study of structural design criteria and handling
      qualities requirements for vehicles designed with active control con-
      cepts included.

   2. Carry out a survey of operational flight conditions to point out the
      critical load cases. This survey should cover the effects of angle-
      of-attack and Mach number variations, aeroelasticity, control surface
      deflection and rate limits, and both clean and high lift configurations.

   3. Develop design methods which are more suitable for use in preliminary
      design allowing rapid trade studies between active and passive
      techniques.

   4. A detailed design study should be conducted, integrating the active
      controls early in the design process (control configured vehicle
      concept), optimizing the control systems, and establishing the
      resulting benefits.

C. Technology Demonstration

   1. Design validation under actual flight conditions will provide the
      degree of confidence required prior to incorporation of active control
      concepts into commercially certifiable transport airplanes. This
      flight test program could be accomplished using existing airplanes.

Airline Assessment

     Under contract to NASA, United Air Lines, Inc., conducted an assessment
of the system studies (ref. 10). American Airlines was awarded a similar con-
tract and, although their results are not published as yet, they appear to be
reaching conclusions quite like those of United.

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The primary area of concern to an aircraft operator, is that of systems reliability and maintainability. They recommended that a great deal of effort be put into systems which would allow the operators to detect system degradation and apply preventive or progressive corrective actions prior to complete system failure. The maintenance procedures should be given a great deal of thought by both the manufacturers and the airlines to insure that the maintenance program which evolves will be simple, timely, and responsive to the airline desires. It was felt that the benefits offered by active control concepts would be seriously degraded if necessary to include mechanical backup systems in the aircraft. All, or major portions, of an active control system will be required to be operative prior to flight. Consequently, the level of redundancy must be such that dispatch will be possible with one system inoperative and must sustain a second failure in flight.

Demonstration and service life evaluation in flight of realistic active control systems was considered to be almost essential. Several on-going and complete active control demonstration programs, such as the Air Force CCV program with the B-52, were noted. However, the airlines would like to have years of operational experience rather than hours. It was recommended that gust/maneuver load control and ride quality control systems be retrofitted into several contemporary aircraft in such a manner that current operations are not disrupted. This would allow protracted service life evaluation. It was felt that this approach would not only benefit future aircraft, but could prolong existing aircraft life. Such retrofit systems would be designed such that the aircraft would be able to dispatch with the system failed. Thus, redundancy requirements would be much less critical for these installations.

There was more concern expressed by the airlines about the technology readiness of relaxed stability and flutter suppression systems, primarily because of the flight-critical nature of these functions and apparent remoteness of the solution to the system's reliability problem. They felt the basic study programs should be accelerated utilizing both ground-based and research flight experiments. Contemplated hardware for relaxed stability could be installed in current aircraft, performing other functions, to gain in-service life data. United stated that, from their experience, there is no substitute for the aircraft as a test bed and no laboratory or test cell yet has adequately simulated the aircraft environment.

CONCLUSIONS

Although somewhat different approaches were taken in these system studies, a number of interesting possibilities for applying active controls technology were indicated. However, there is a need for further in-depth studies which would introduce the active control concepts earlier in the design process and in a more integrated manner.

Relaxing the stability requirements offered the greatest benefit and was the only concept included in the initial design process. Flutter suppression

offered the smallest benefit according to the results of two of the contractors and was identified as being the concept most removed from the current state of the art by all of the contractors. Maneuver load alleviation and gust load alleviation were found to yield significantly smaller benefits than relaxed stability requirements; however, the contractors pointed out that these concepts should be introduced at the initiation of the design process in order to maximize the benefits.

Each of the contractors pointed out areas where the benefits could have been greater if more data and/or more time were available for design refinements and system optimization. Problem areas or areas of concern encountered by the contractors in the course of the system studies were reflected in the recommended research and development tasks. Based on the results of these studies, it appears as though active controls technology can provide significant benefits when applied to subsonic, long-range transport aircraft. However, application will require completing a research and development program directed toward fulfilling data base requirements, establishing effective design techniques and criteria, improving systems maintainability and reliability, and demonstrating technology readiness.

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4. Anon.: Study of the Application of Advanced Technologies to Long-Range Transport Aircraft. Vol. I – Technology Applications. General Dynamics, Convair Aerospace Division, NASA CR-112090, May 8, 1972.

5. Anon.: Study of the Application of Advanced Technologies to Long-Range Transport Aircraft. Vol. II – Research and Development Requirements. General Dynamics, Convair Aerospace Division, NASA CR-112091, May 8, 1972.

6. Lange, R. H., et al.: Study of the Application of Advanced Technologies to Long-Range Transport Aircraft. Vol. I – Analysis and Design. Lockheed-Georgia Company, NASA CR-112088, 1972.

7. Lange, R. H., et al.: Study of the Application of Advanced Technologies to Long-Range Transport Aircraft. Vol. II – Research and Development Requirements. Lockheed-Georgia Company, NASA CR-112089, 1972.

8. Nissim, E.: Flutter Suppression Using Active Controls Based on the Concept of Aerodynamic Energy. NASA TN D-6199, March 1971.

9. Anon.: Economic Impact of Applying Advanced Engine and Airframe Technology to Transport Aircraft. NASA CR-132268, August 1973.

10. Anon.: Assessment of the Application of Advanced Technologies to Subsonic CTOL Transport Aircraft. United Airlines, Inc., NASA CR-112242, April 1973.
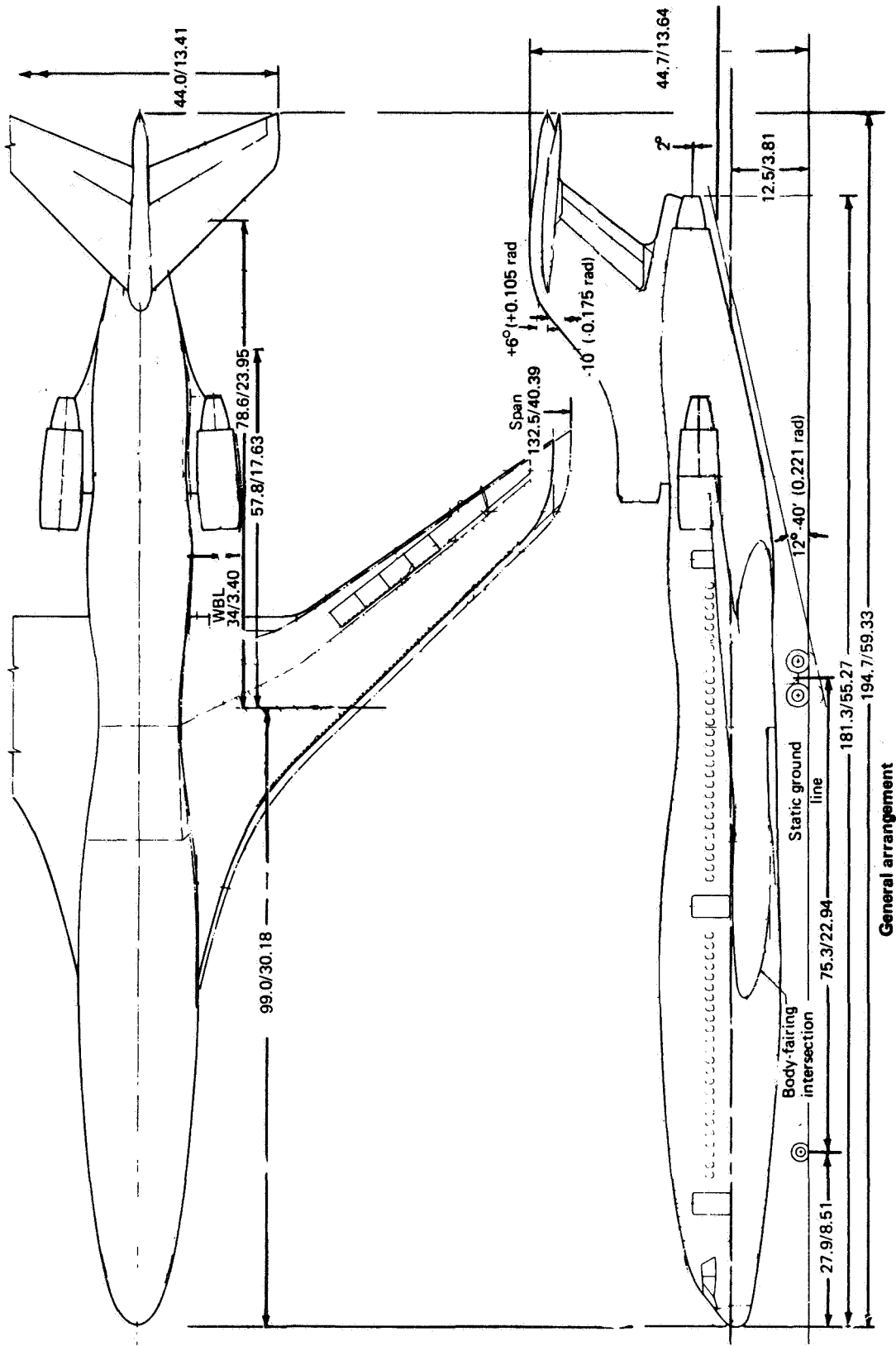
FIGURE 1. BOEING M=0.98, 195 PASSENGER CONFIGURATION
(From ref. 2)

615
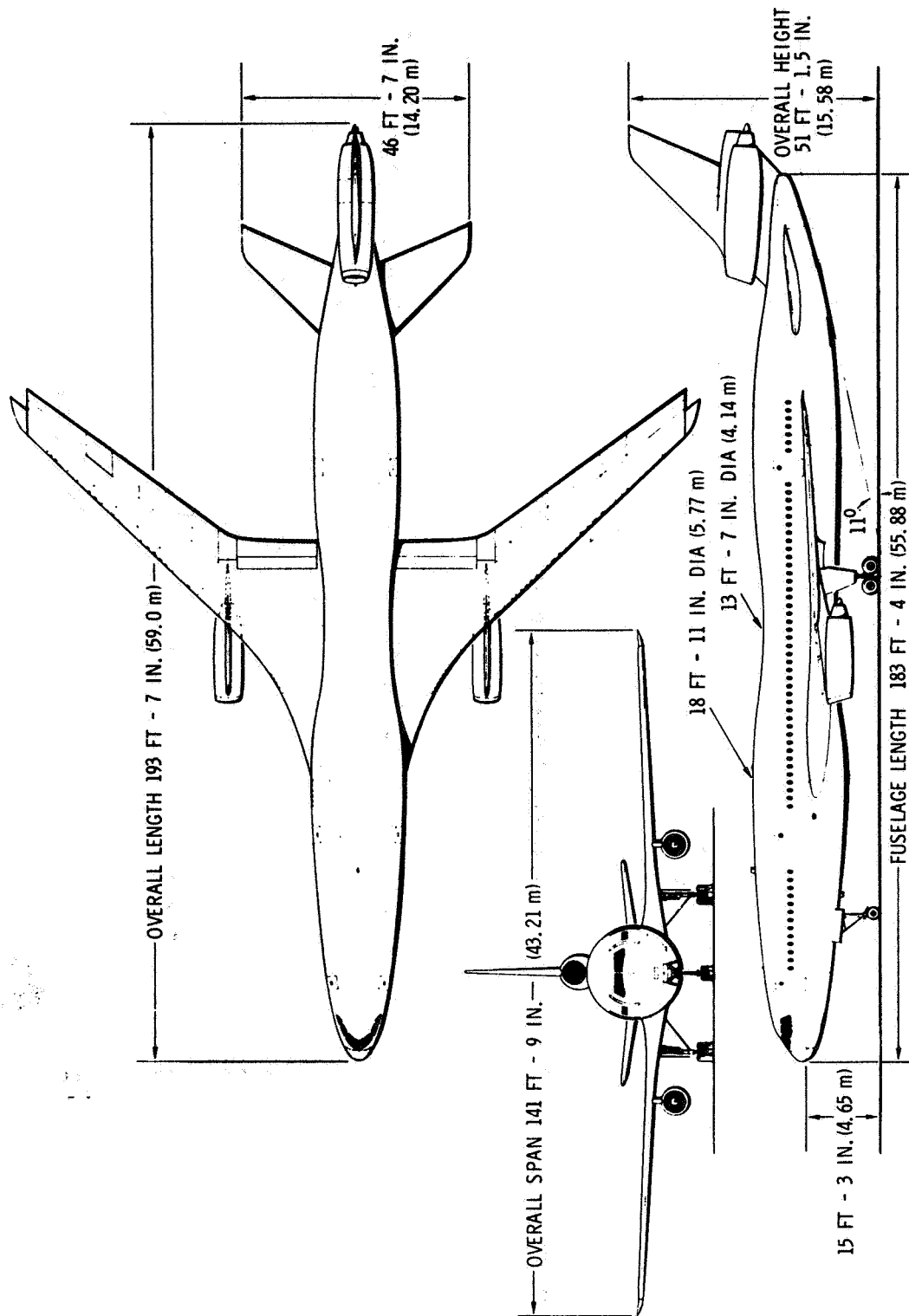
FIGURE 2. GENERAL DYNAMICS M=0.98, 195 PASSENGER CONFIGURATION
(From ref. 4)

46 FT - 7 IN.
(14.20 m)

OVERALL LENGTH 193 FT - 7 IN. (59.0 m)

OVERALL HEIGHT
51 FT - 1.5 IN.
(15.58 m)

18 FT - 11 IN. DIA (5.77 m)

13 FT - 7 IN. DIA (4.14 m)

11°

FUSELAGE LENGTH 183 FT - 4 IN. (55.88 m)

15 FT - 3 IN. (4.65 m)

OVERALL SPAN 141 FT - 9 IN. (43.21 m)

616
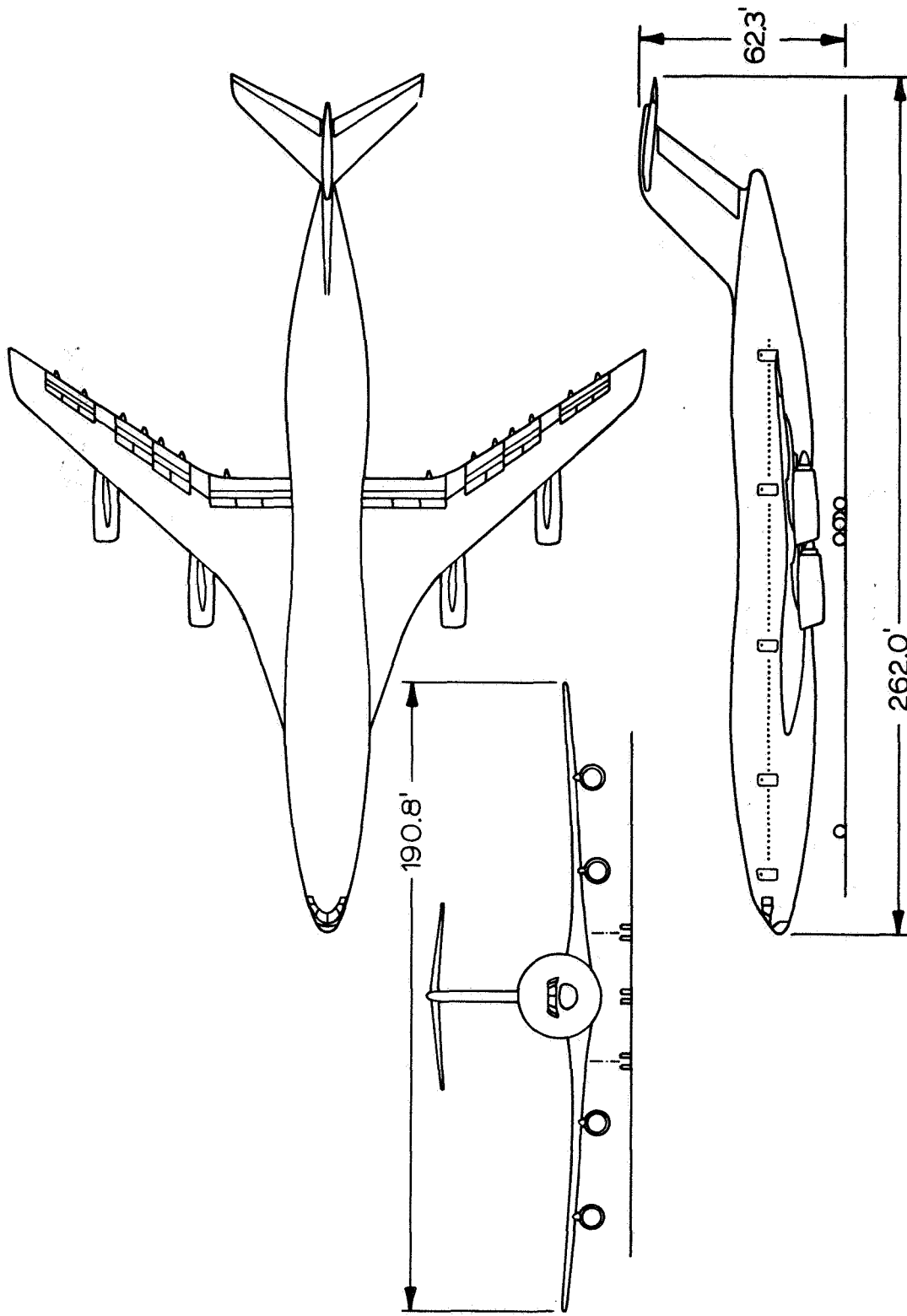
FIGURE 3. LOCKHEED M=0.95, 400 PASSENGER CONFIGURATION
(From ref. 6)

62.3'

262.0'

190.8'
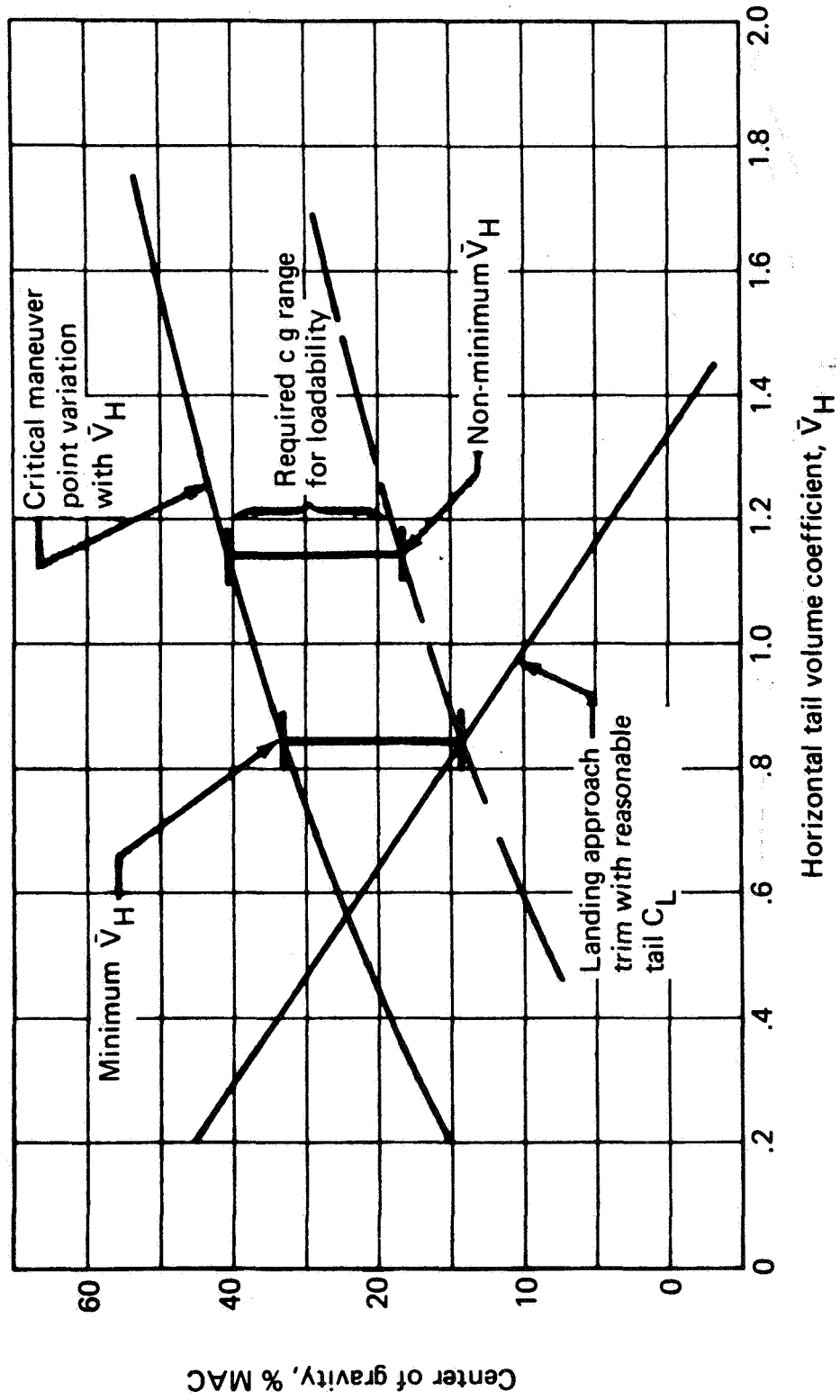
FIGURE 4. BOEING HORIZONTAL TAIL SIZING PHILOSOPHY
(From ref. 2)
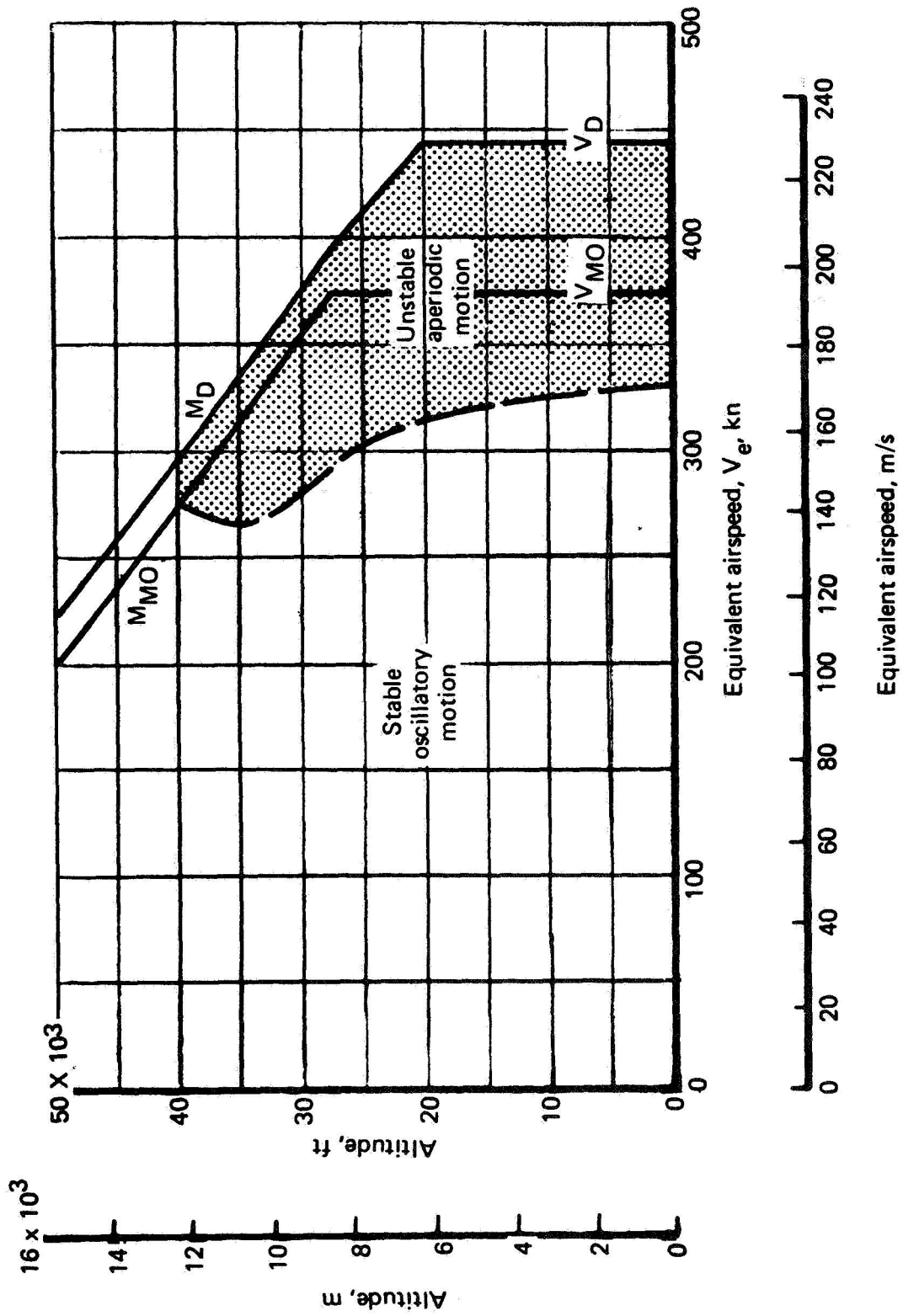
FIGURE 5. UNAUGMENTED LONGITUDINAL DYNAMIC STABILITY
(From ref. 2)

619
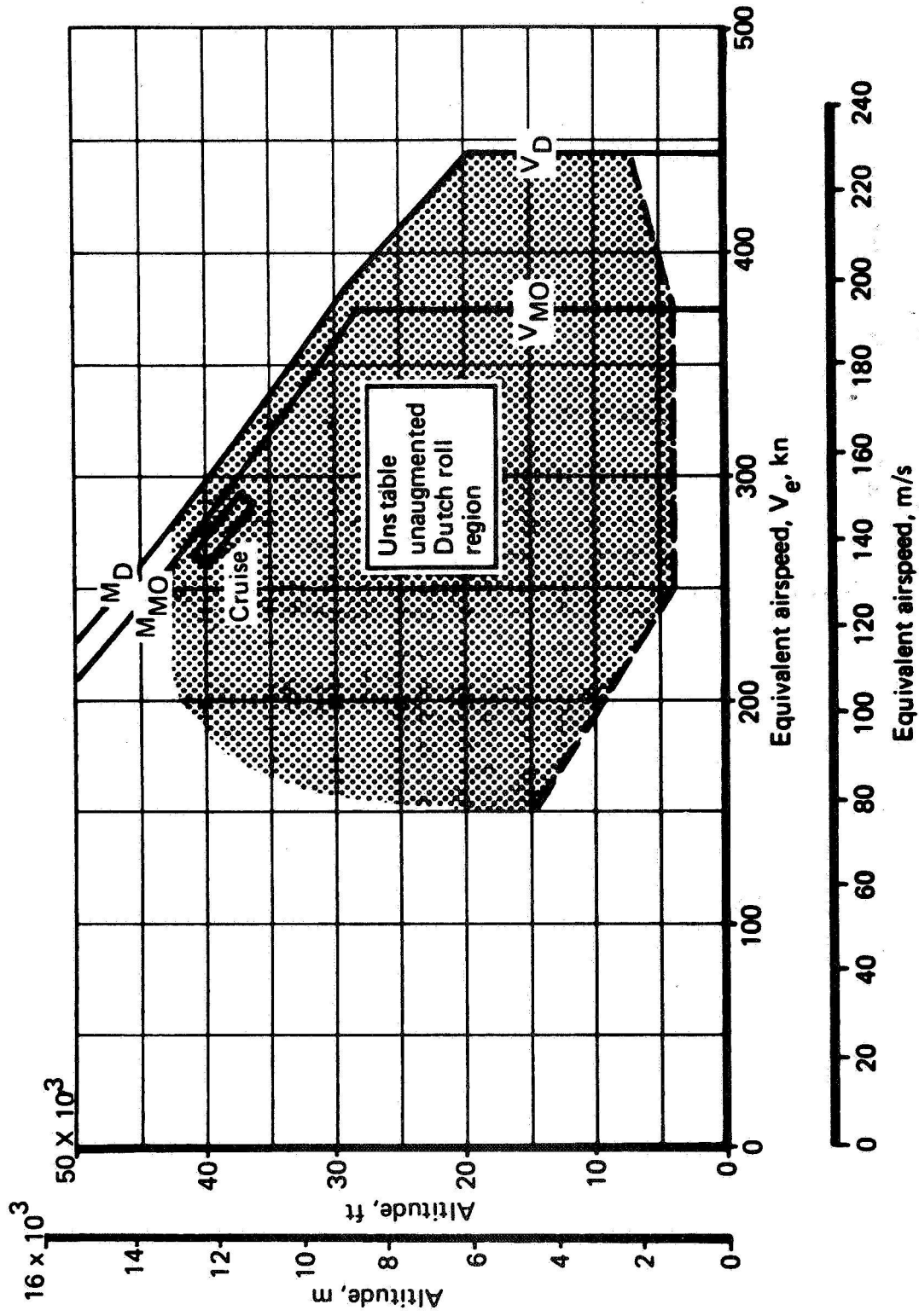
FIGURE 6. LATERAL-DIRECTIONAL DYNAMIC STABILITY, UNAUGMENTED

(From ref. 2)
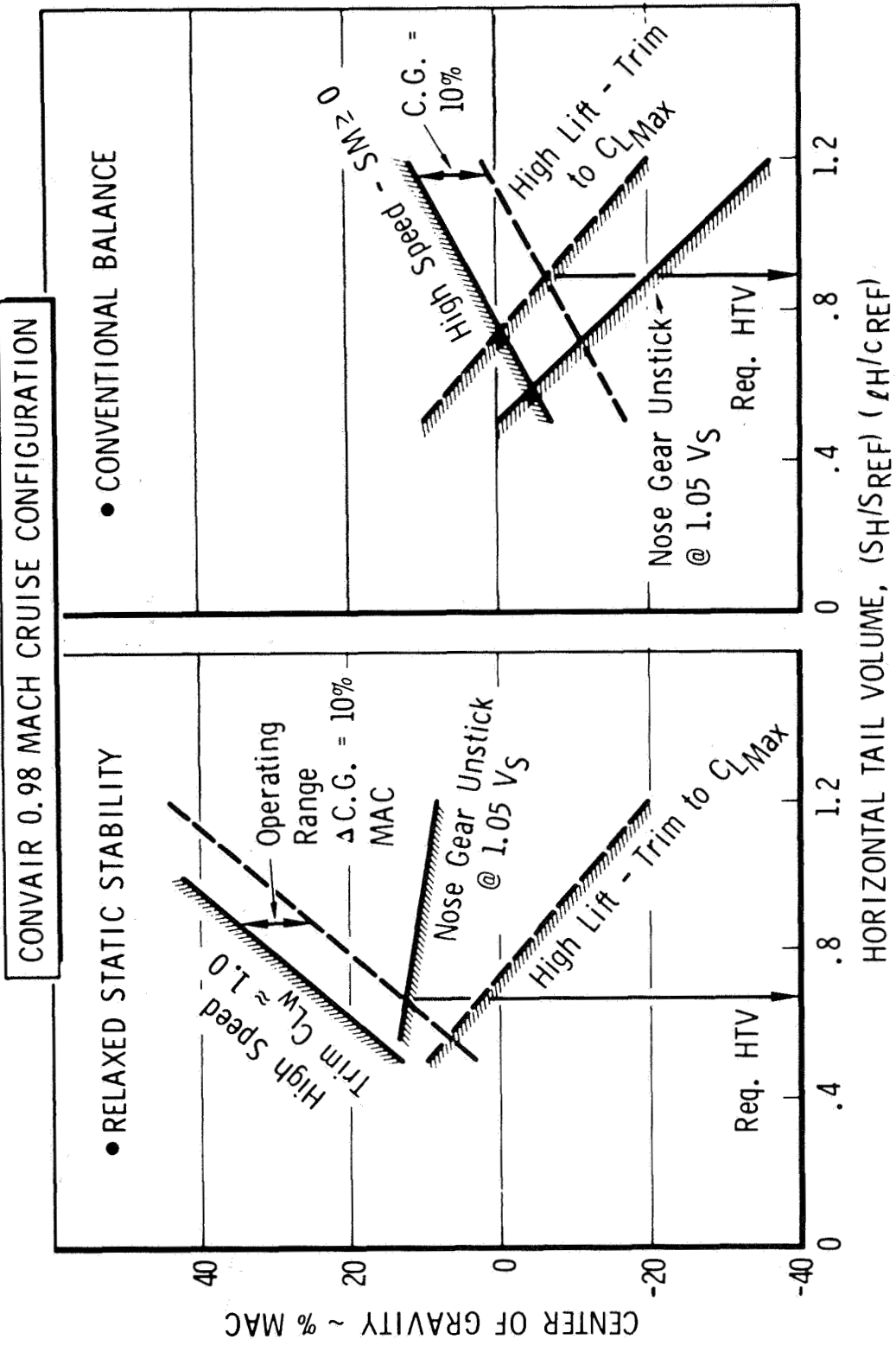
FIGURE 7. GENERAL DYNAMICS HORIZONTAL TAIL SIZING PHILOSOPHY
(From ref. 4)

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FIGURE 8. BALANCE CHARACTERISTICS – ALL MOVABLE HORIZONTAL TAIL PLUS GEARED TRAILING EDGE
(From ref. 4)

622
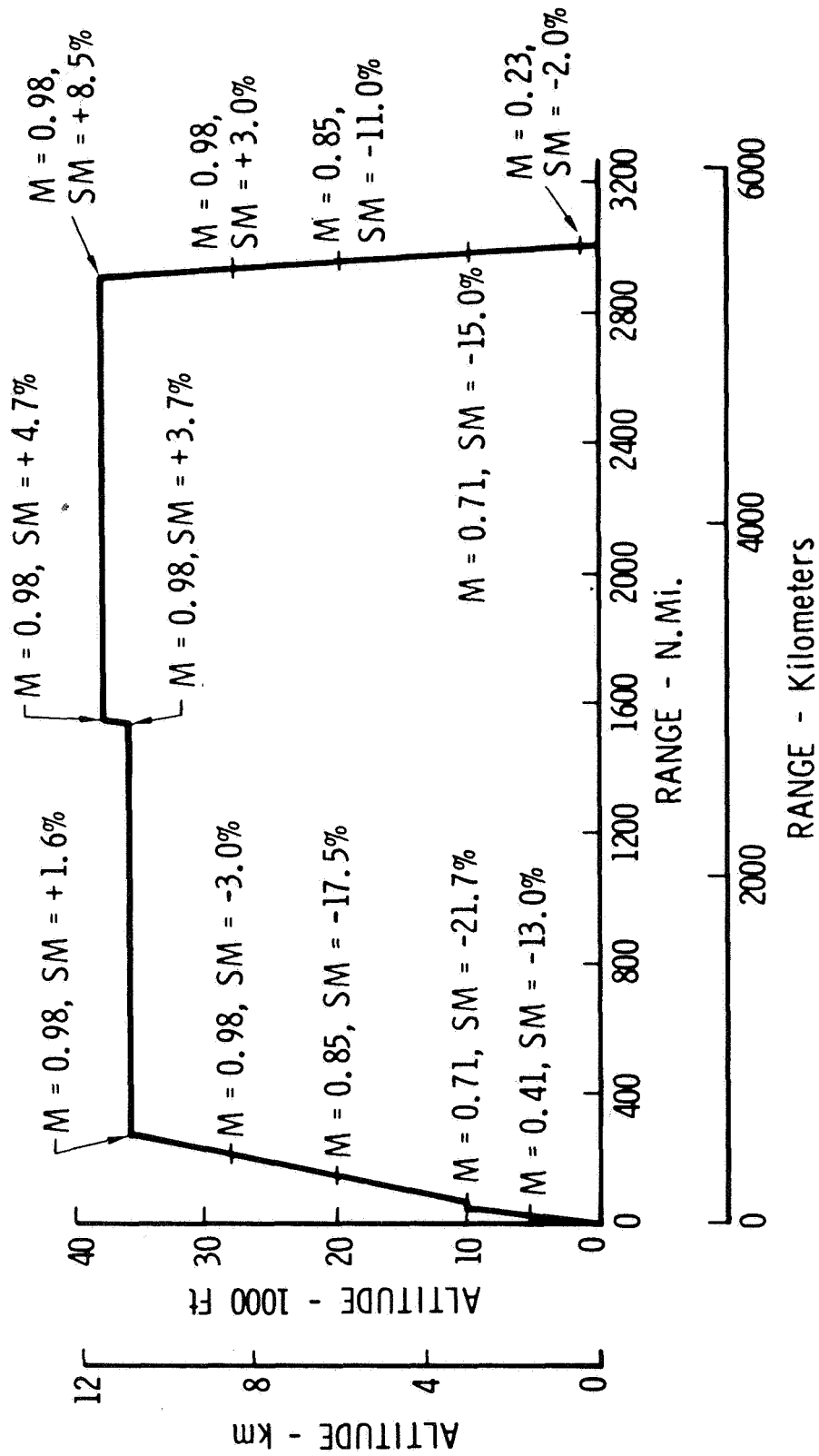
FIGURE 9. VARIATION OF AERODYNAMIC STATIC MARGIN DURING TYPICAL FLIGHT PROFILE

(From ref. 4)

(a) Normal tail

(b) Active control tail
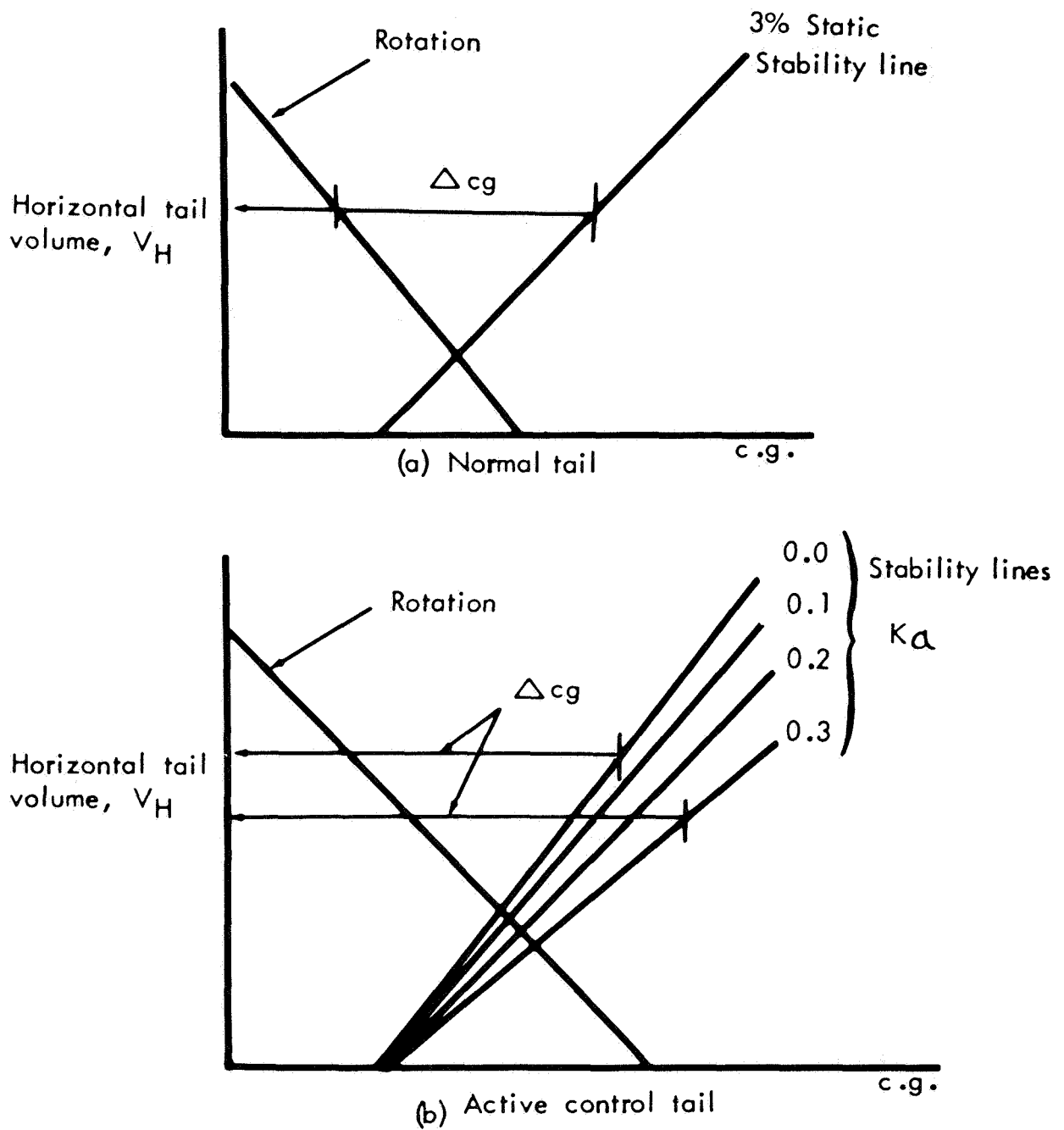
Typical horizontal tail sizing for a given set of flight conditions.

FIGURE 10. LOCKHEED HORIZONTAL TAIL SIZING PHILOSOPHY
(From ref. 6)
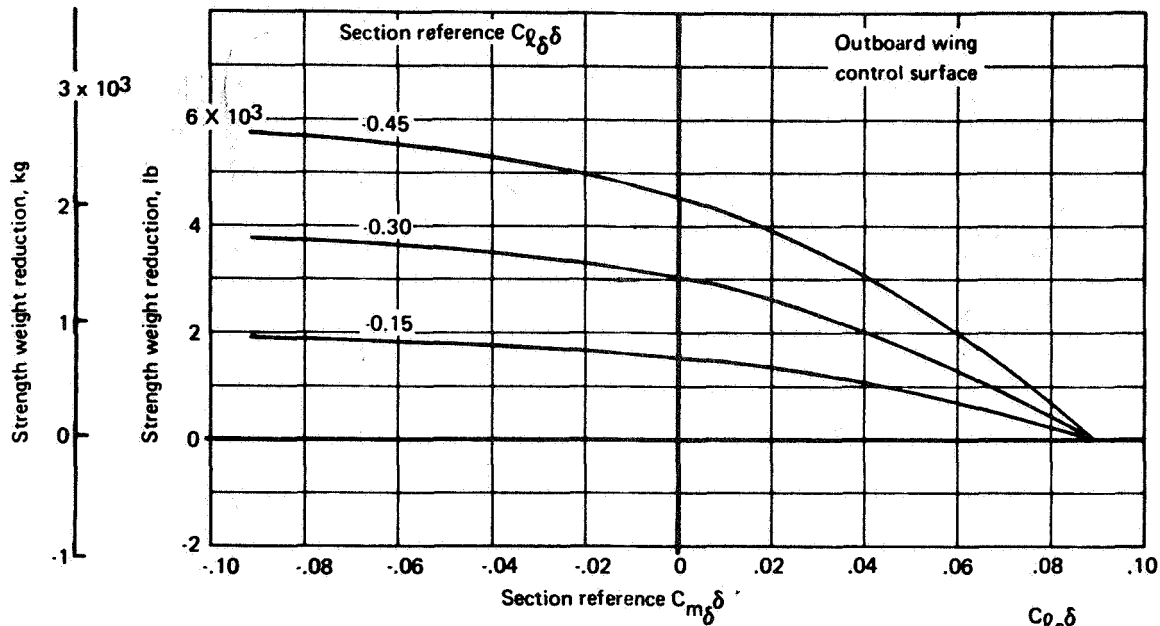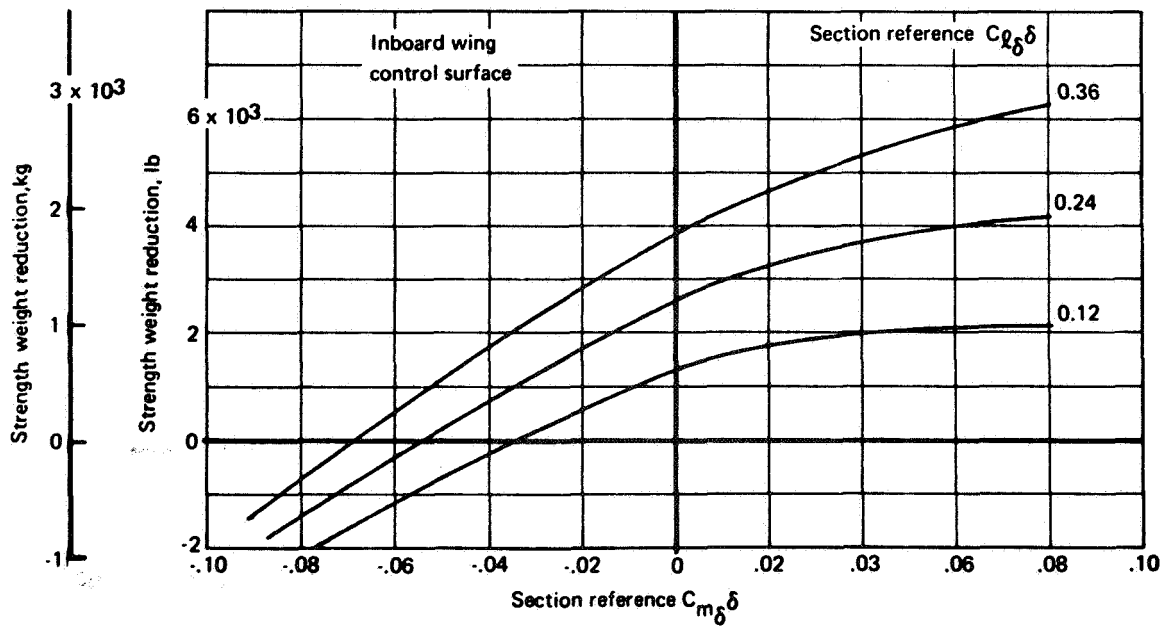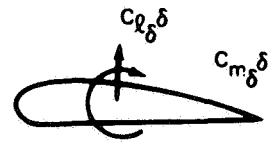
FIGURE 11. POTENTIAL WING BOX MATERIAL SAVINGS BY MLA
(From ref. 2)
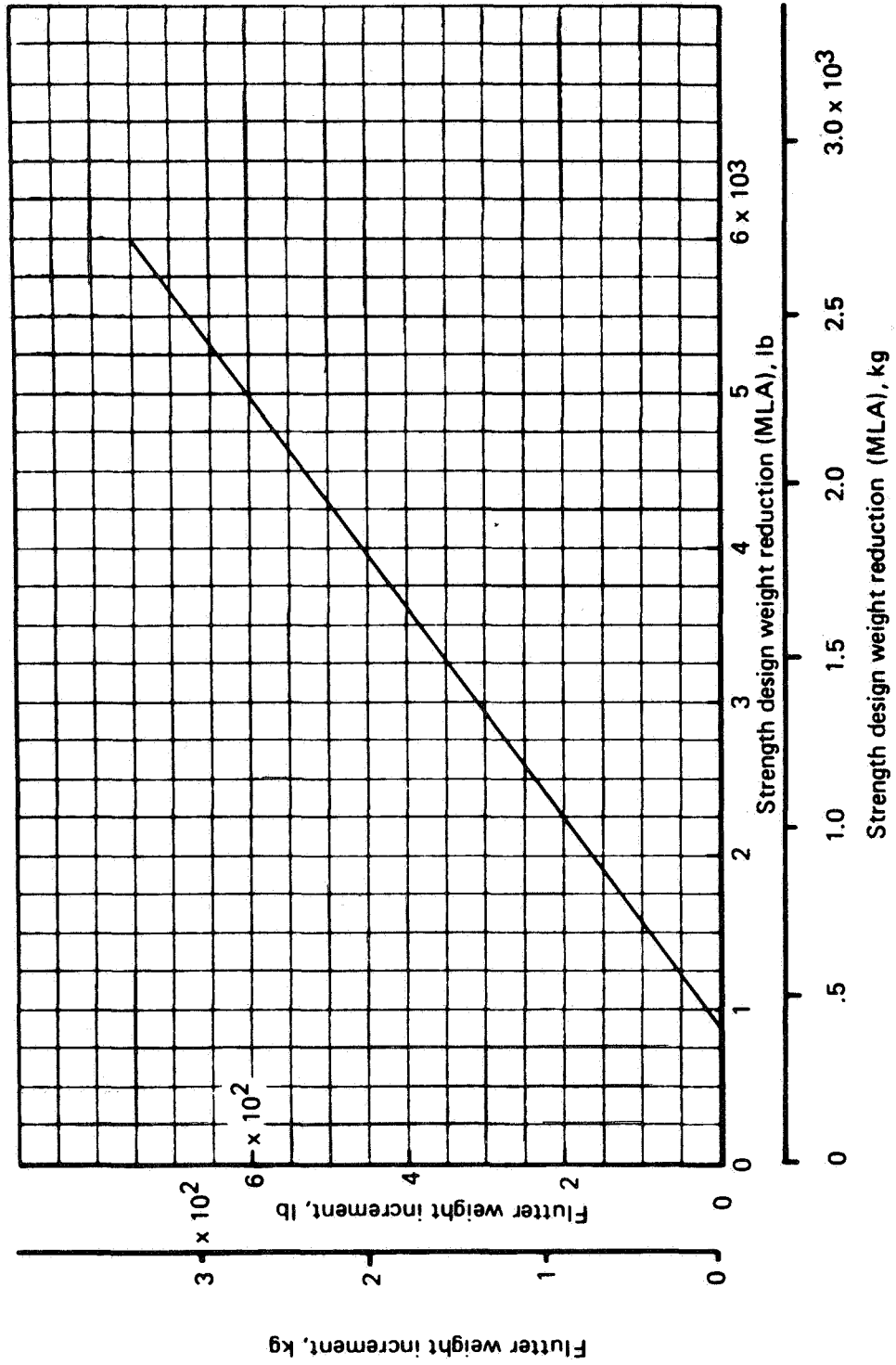
FIGURE 12. FLUTTER REQUIREMENT
(From ref. 2)

626
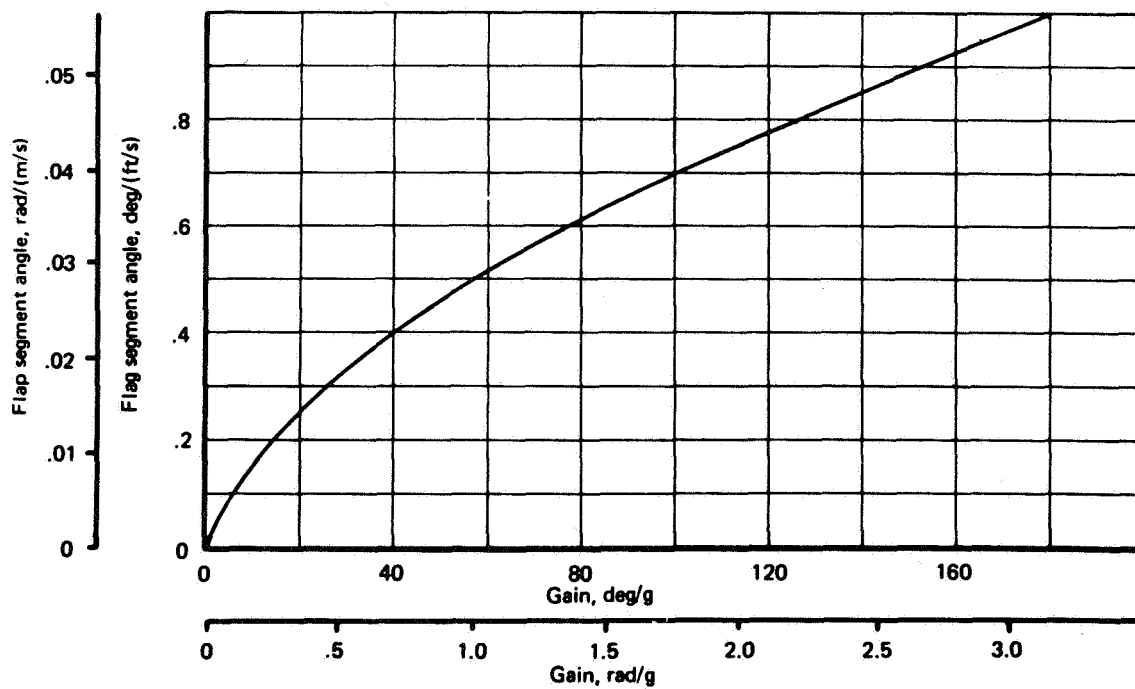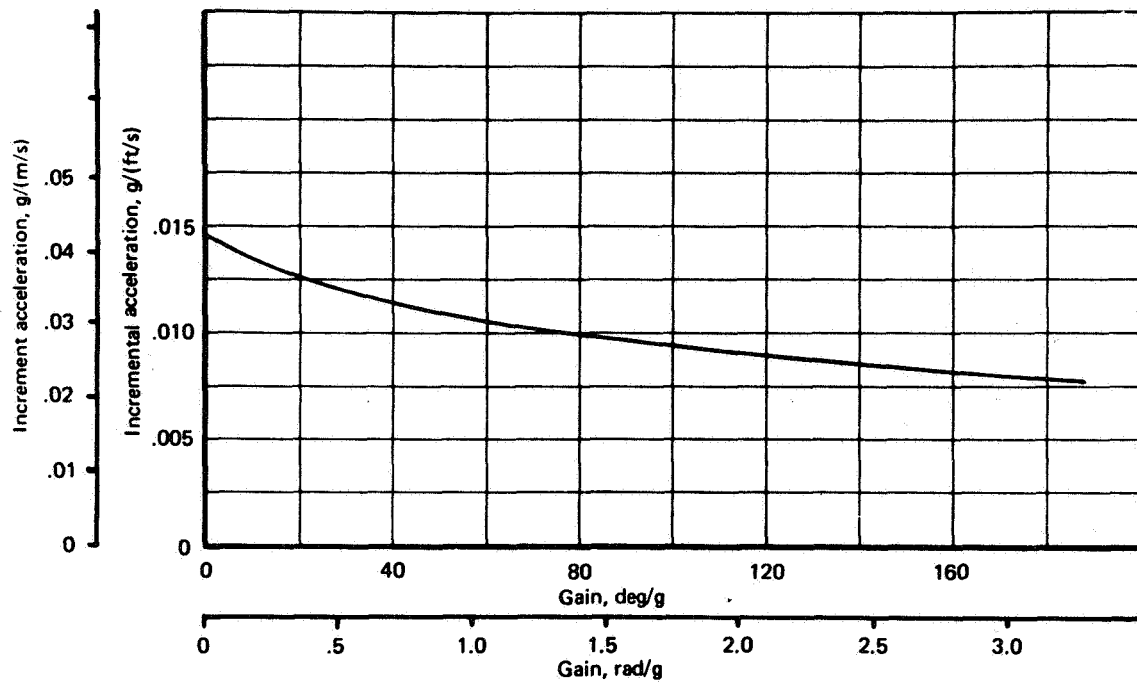
FIGURE 13. AIRPLANE RESPONSE (GLA)
(From ref. 2)
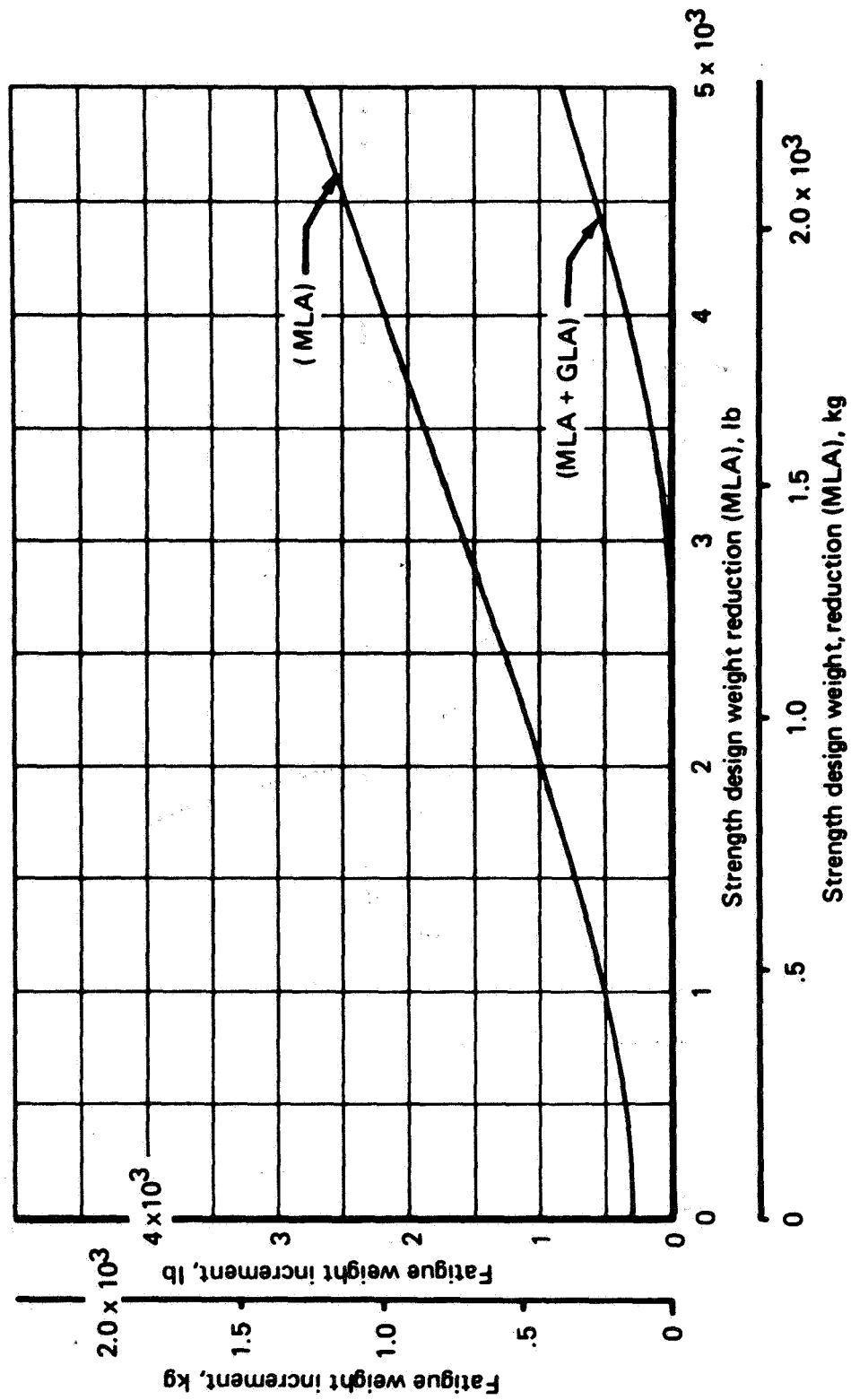
FIGURE 14. GUST LOAD ALLEVIATION POTENTIAL
(From ref. 2)

Flight condition
1.2 V_D
22,700 ft (6919 m)
Full fuel
Symmetrical
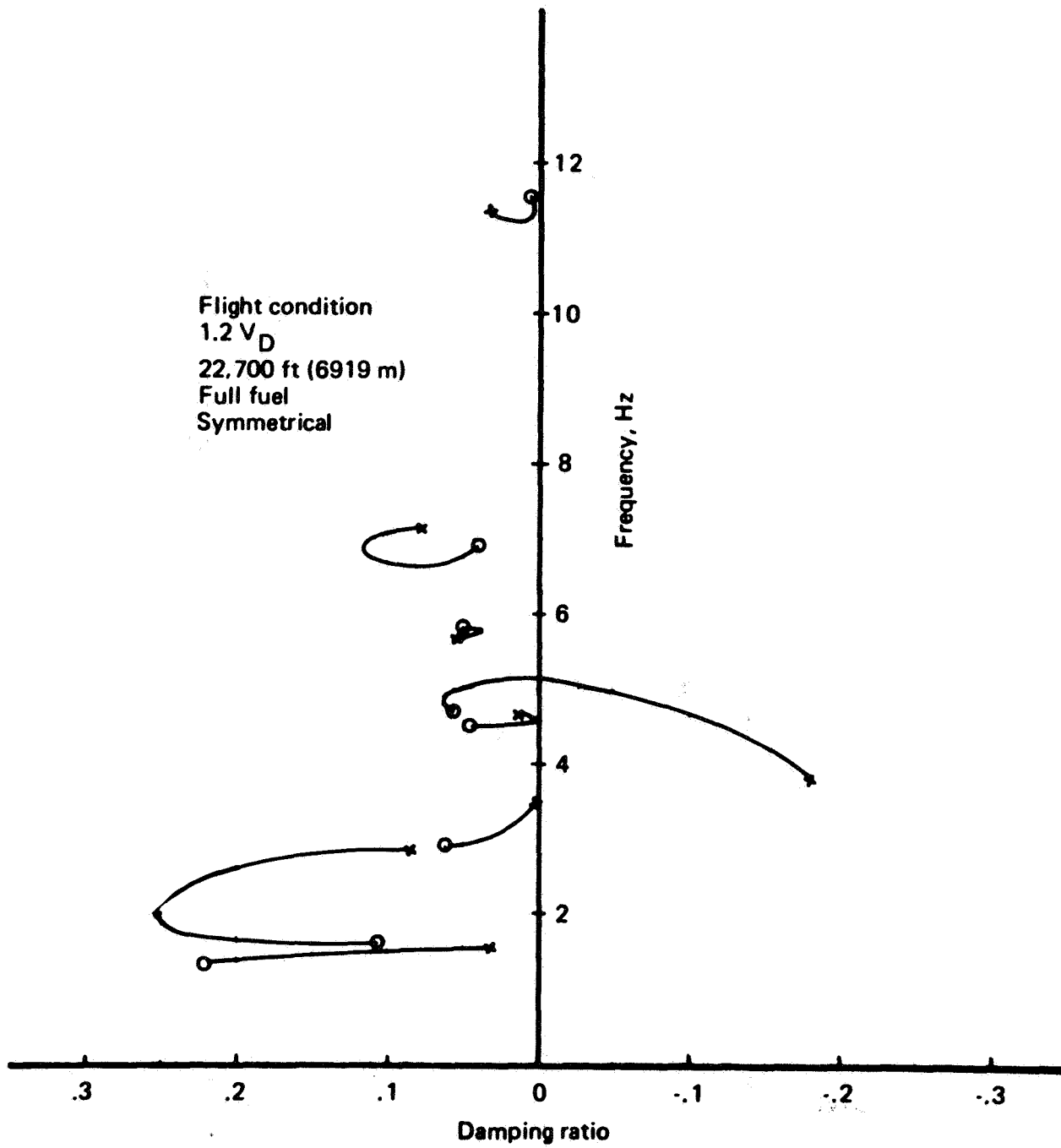
FIGURE 15. FLUTTER MODE CONTROL ROOT LOCUS PLOT
(From ref. 2)
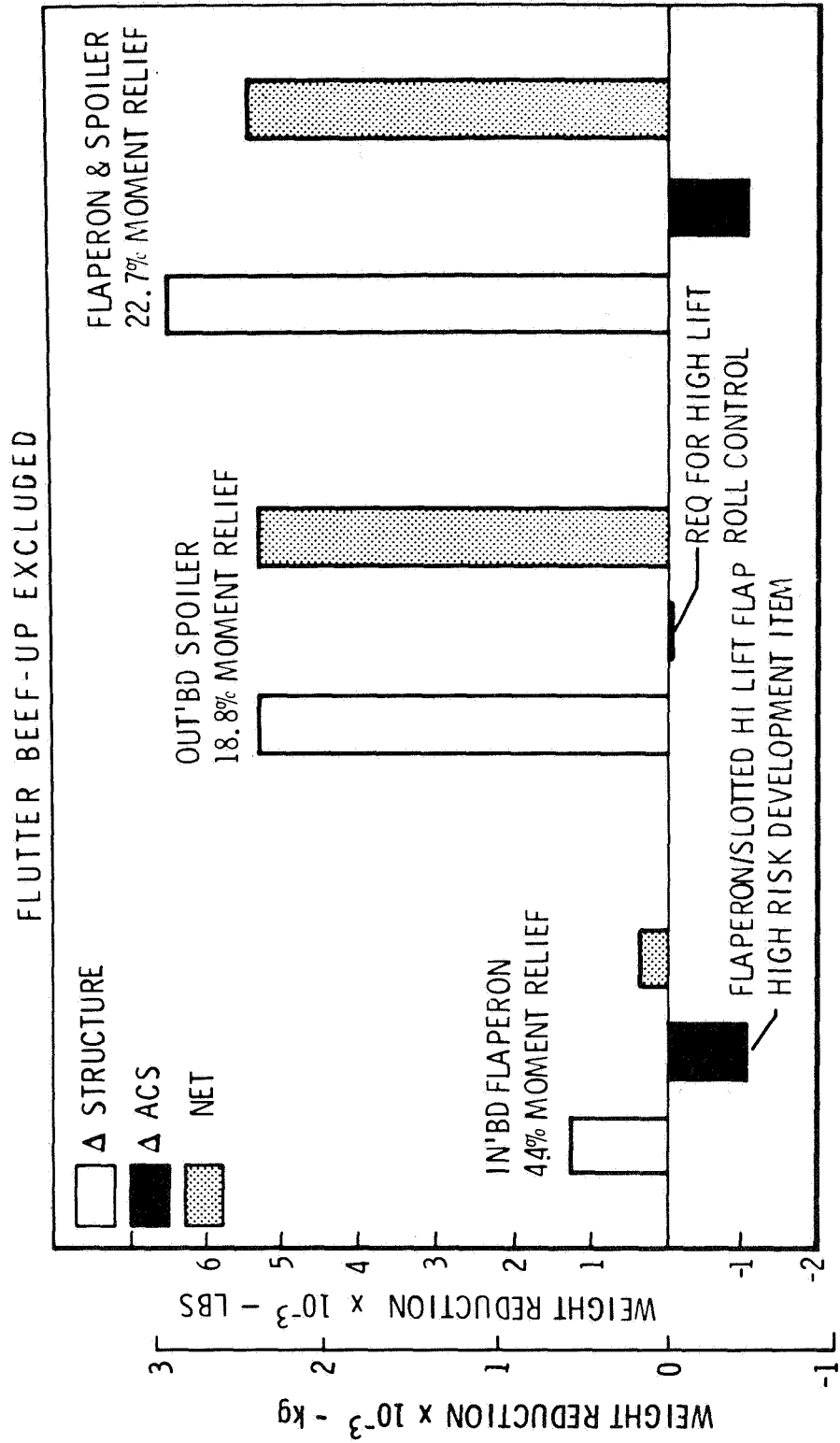
FIGURE 16. WING DESIGN LOAD CONTROL
(From ref. 4)
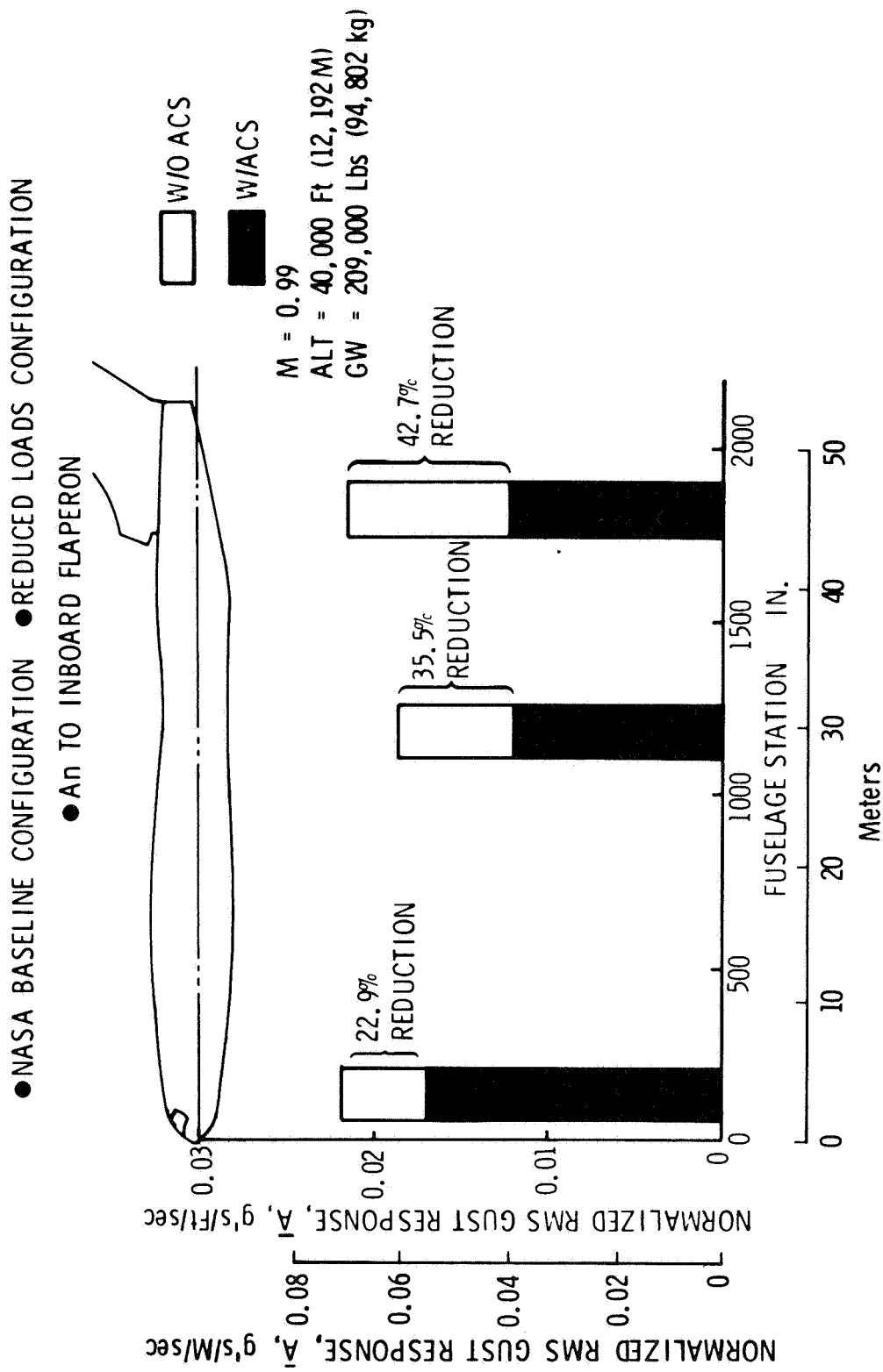
FIGURE 17. RIDE QUALITY BENEFITS OF GUST ALLEVIATION
(From ref. 4)

631
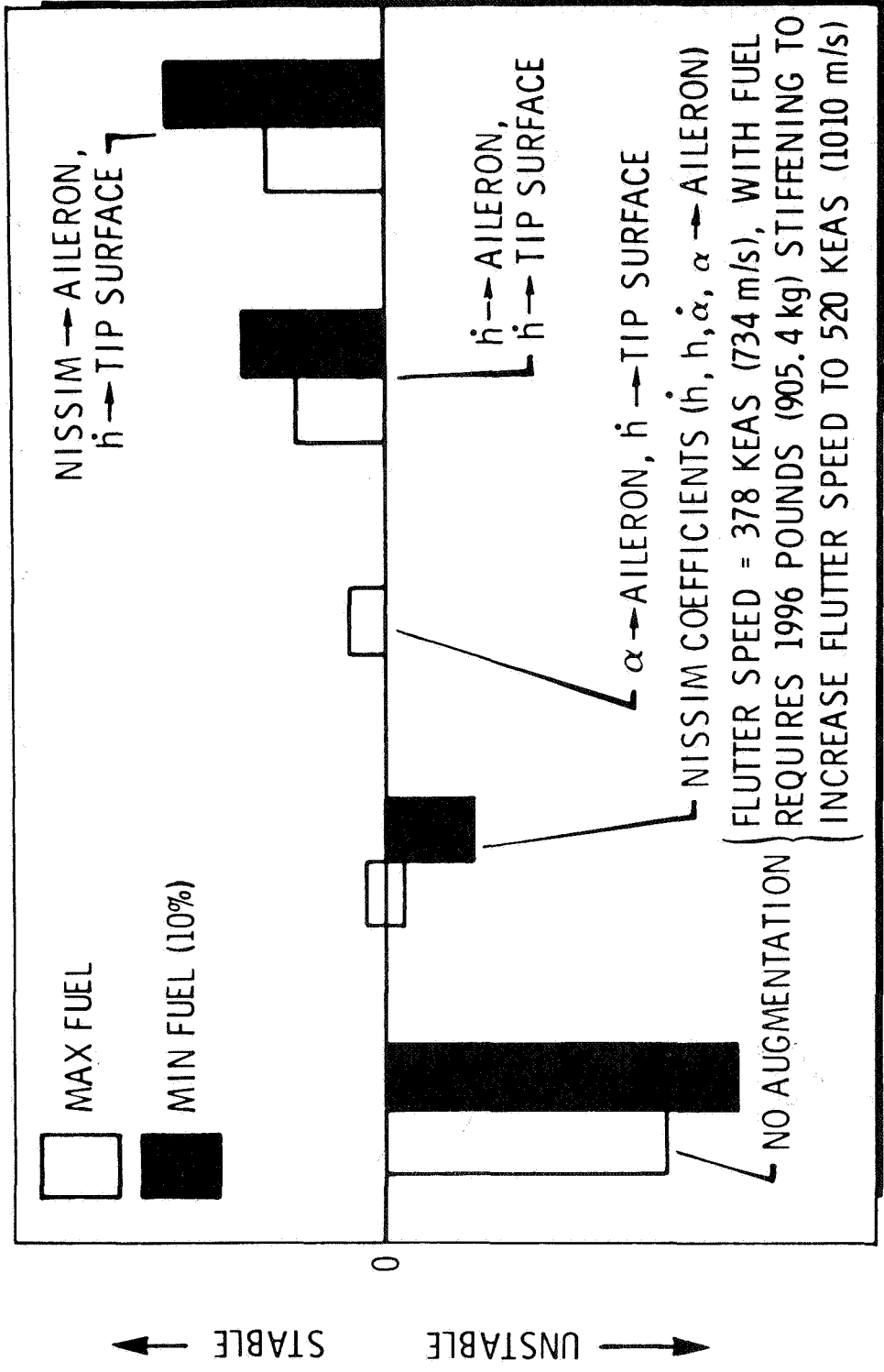
FIGURE 18. FLUTTER SUPPRESSION - REDUCED DESIGN LOADS WING
(From ref. 4)
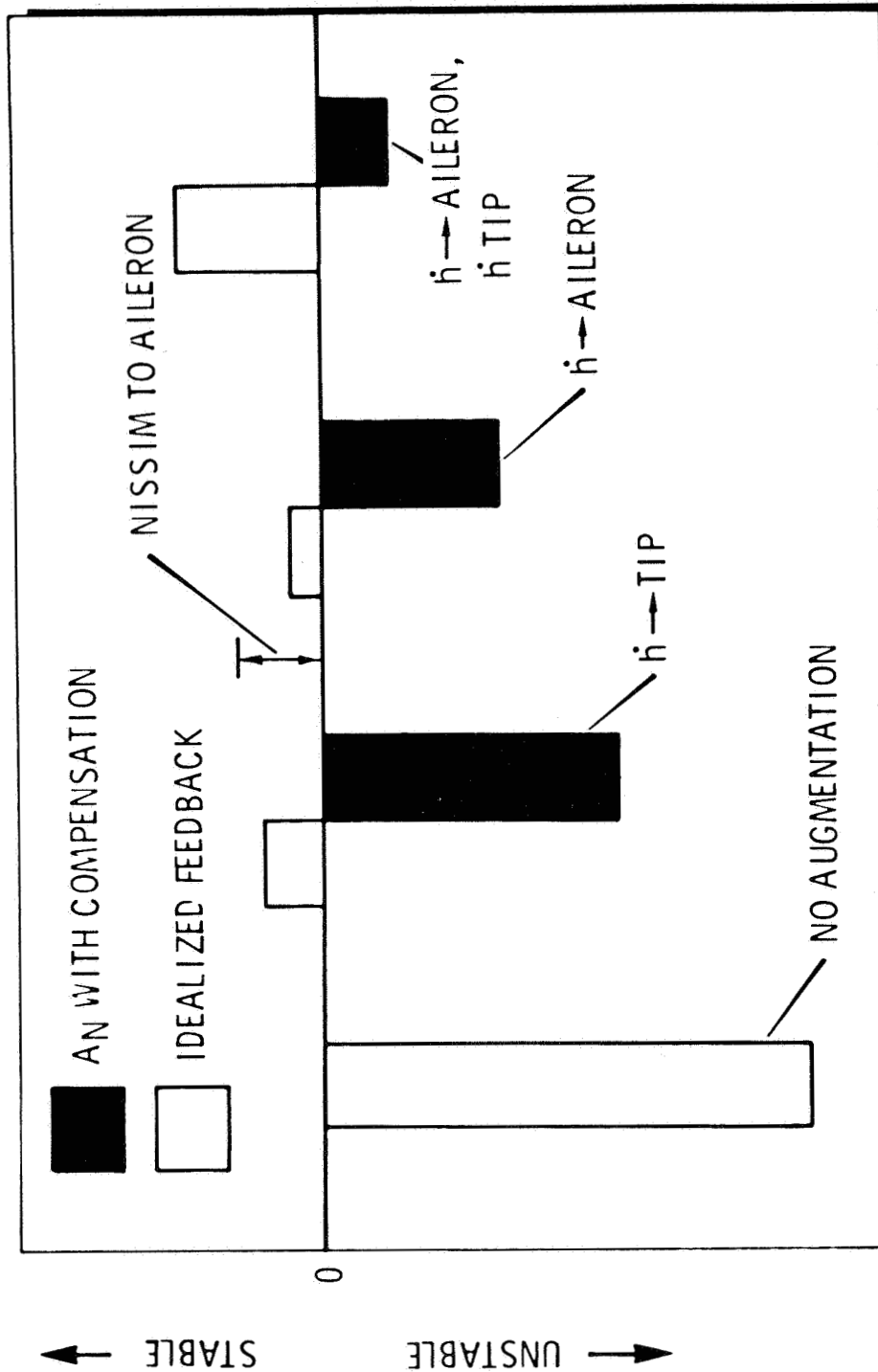
FIGURE 19. FLUTTER SUPPRESSION – CURRENT TECHNOLOGY IMPLEMENTATION
(From ref. 4)

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FIGURE 20. FATIGUE DAMAGE RESULTS
(From ref. 4)

Control Point	Relative* Fatigue Damage		
	Study Configuration		
	100% Strength without ACS	100% Strength with ACS	Reduced Strength with ACS
Wing @ η = 0.187			
Gust	1.00	.39	.65
Maneuver	1.00	.09	.64
G.A.G.	1.00	.65	1.10
Wing @ η = 0.702			
Gust	1.00	.25	.55
Maneuver	1.00	.50	.50
G.A.G.	1.00	.90	1.16
Fuselage Sta 957 (24.2 m)			
Gust	1.00	.37	.49
Maneuver	1.00	.42	.58
G.A.G.	1.00	.82	1.08
Fuselage Sta 1196 (30.4 m)			
Gust	1.00	.29	.42
Maneuver	1.00	.13	.27
G.A.G.	1.00	.75	1.29

* Data normalized on 100% strength without ACS configuration.

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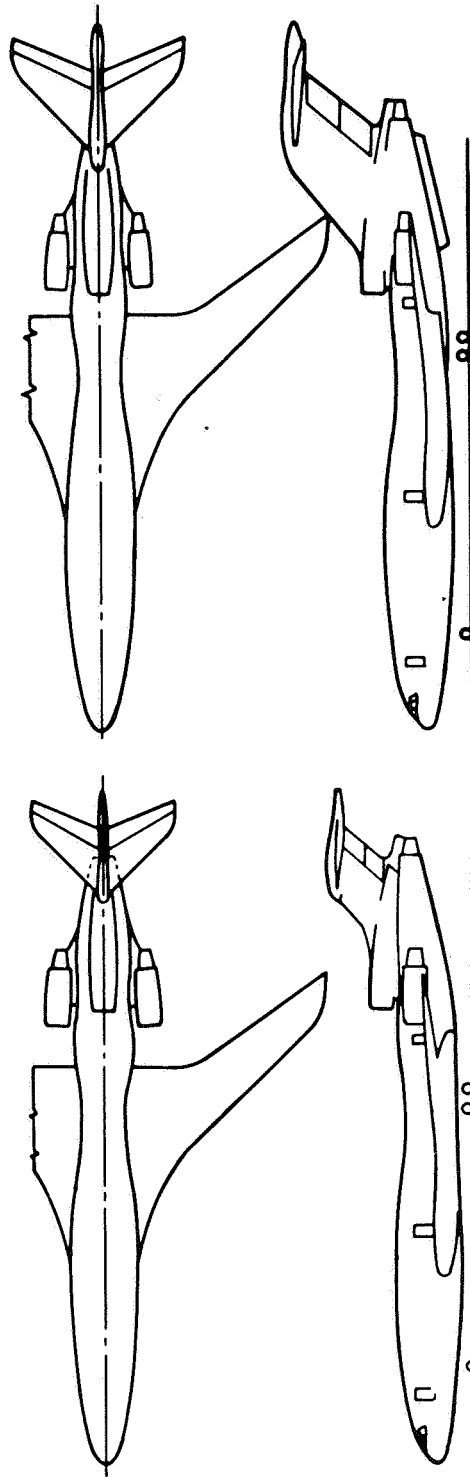
Advanced Technology
-   Flight Critical Stability Augmentation
-   Maneuver Load Alleviation
-   Gust Load Alleviation
-   Optimum Cruise C. G. and
    Reduced C. G. Range

Conventional Technology
-   Statically Stable
-   Fail-Safe Yaw Damper
-   Inherent Acceptable Stall

FIGURE 21. ACTIVE CONTROLS TECHNOLOGY APPLICATION
(From ref. 2)

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CONTRACTOR	RELAXED STABILITY	MLA & GLA	F. S.	TOTAL
BOEING (REF. 2)	11.0% T. O. G. W.	1.4-3.0%	STRUCTURAL WEIGHT	11.0% T. O. G. W.
GENERAL DYNAMICS (REF. 4)	1.9% T. O. G. W. (TRIM DRAG EFFECT)	2.4% T. O. G. W.	2.8% T. O. G. W.	7.1% T. O. G. W.
LOCKHEED (REF. 6)	16% O. W. E.	-0-	-0-	16% O. W. E.

FIGURE 22. PERCENT WEIGHT SAVINGS

CONTRACTOR	RELAXED STABILITY	MLA & GLA	F. S.	TOTAL
BOEING (REF. 2)	N. A.	N. A.	N. A.	N. A.
GENERAL DYNAMICS (REF. 4)	4.0% (TRIM DRAG EFFECT)	4.7%	2.8%	11.5%
LOCKHEED (REF. 6)	8.0%	-0-	-0-	8.0%

FIGURE 23. PERCENT IMPROVEMENT IN RETURN ON INVESTMENT

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CONTRACTOR	RELAXED STABILITY	MLA & GLA	F. S.	TOTAL
BOEING (REF. 2)	N.A.	N.A.	N.A.	N.A.
GENERAL DYNAMICS (REF. 4)	1.5% (TRIM DRAG EFFECT)	1.9%	1.1%	4.5%
LOCKHEED (REF. 6)	8.7%	-0-	-0-	8.7%

FIGURE 24. PERCENT IMPROVEMENT IN DIRECT OPERATING COSTS

A SURVEY OF ACTIVE CONTROLS BENEFITS

TO SUPERSONIC TRANSPORTS

Kermit G. Pratt
NASA Langley Research Center

ABSTRACT

Results are drawn from in-house studies and from several contracted system studies of the impact of advanced technologies on the design of an arrow-wing configuration. Information presented includes estimated benefits, effects of combinations of active control concepts, and constraints. Emphasis is placed on characteristics that are uniquely related to a large airframe featuring a slender body with a fixed wing of low aspect ratio, high sweep, and small thickness ratio.

SUMMARY

The benefits of the application of active controls to supersonic transport airplanes are surveyed. Results are drawn from in-house studies and from several contracted system studies of the impact of advanced technologies on the design of an arrow-wing configuration. The characteristics that are uniquely related to a large flexible airframe featuring a slender body with a fixed wing of low aspect ratio, high sweep, and small thickness ratio are discussed, particularly with regard to the need for the various active controls concepts and to the constraints to benefits. The results indicate that significant benefits can be obtained with a configuration that is inherently longitudinally unstable in subsonic flight and is stabilized by active controls. These benefits may be increased by use of center-of-gravity control and angle-of-attack limiting. Benefits from maneuver and gust load alleviation may be small. In any case, load alleviation most likely will require that flutter suppresion be used as well. Flutter suppression in itself may provide some saving in structural weight. Ride quality control by a mode suppression system may be needed for passenger acceptance. For safety, active lateral control should be considered for limiting the magnitude of the transient motion due to an engine unstart.

INTRODUCTION

In the course of the United States Supersonic Transport (SST) program it was necessary for the designers to utilize active controls to stabilize an inherently unstable vehicle in order to achieve an economically competitive and safe airplane. This concept is frequently referred to as relaxed static stability.

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Subsequently, as a part of research efforts to advance supersonic technology, several studies, both in-house and by contract, were undertaken by the NASA to explore the potential for improvements in SST designs by a more extensive use of ACT. Results from these NASA studies together with some from the U.S. SST program are summarized herein in terms of estimated benefits, effects of combinations of ACT concepts, and constraints. The candidate ACT concepts included relaxed static stability, load alleviation, and mode and flutter suppression. Emphasis is placed on characteristics that are uniquely related to a large airframe, featuring a slender body with a fixed wing of low aspect ratio, high sweep, and small thickness ratio.

The information is organized in the following sequence. Information sources are identified and briefly described. Fixed-wing SST characteristics that are pertinent to ACT are reviewed. Results from the various sources are collected under three main topics that reflect the manner in which the airplane is affected by groups of the various concepts. Relaxed static stability, center-of-gravity control, and angle-of-attack limiting are discussed under the heading of Performance, Airframe Efficiency, and Handling Qualities. Flutter suppression, maneuver load alleviation, and gust load alleviation are considered under the heading of Wing Structural Weight. Gust acceleration alleviation and mode suppression are placed under the heading of Ride Quality.

SYMBOLS

A	gust sensitivity factor, $\sigma_{\Delta n}/\sigma_{w_g}$
AC	aerodynamic center
c.g.	center of gravity
c	chord
c_{av}	average chord
c_ℓ	local lift coefficient
C_L	lift coefficient
C_{L_α}	lift curve slope
$C_{L_{max}}$	maximum lift coefficient
C_{m_0}	zero-lift pitching-moment coefficient
$\left(\dfrac{c}{c_{av}}\dfrac{c_\ell}{C_L}\right)$	spanwise lift distribution coefficient
g	acceleration of gravity

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L/D	lift-to-drag ratio
M	Mach number
MAC	mean aerodynamic chord
Δn	incremental vertical acceleration
q	dynamic pressure
w_g	gust velocity
w_s	running weight of structure
y	distance along span
α	angle of attack
σ	root-mean-square value
ACT	active control technology
FAA	Federal Aviation Administration
GAG	Ground-Air-Ground (cycle)
GLA	gust load alleviation
HSAS	hardened stability augmentation system
MLA	maneuver load alleviation
NASA	National Aeronautics and Space Administration
RSS	relaxed static stability
SAS	stability augmentation system
SCAT	supersonic commercial air transport
SST	supersonic transport
TOGW	take-off gross weight

INFORMATION SOURCES

The survey reported herein is based on information from the sources described below. The numbers designating each source are used for identification in the subsequent sections of this paper.

1. The U.S. SST Program, FAA

2. SST Technology Follow-on Program, FAA

3. Studies of the impact of advanced technologies applied to a conceptual supersonic aircraft configuration, NASA

4. Langley Research Center in-house studies

The subject matter from Sources 1 and 2 deals with the stabilization by active controls of an inherently longitudinally unstable airplane. An illustration of this airplane is shown in figure 1. The airplane structure and control system design was developed in depth. Aeroelastic effects are considered. Material pertinent to active controls is documented in reference 1.

Source 3 consists of three contract design studies related to an arrow-wing configuration, illustrated in figure 2, which was derived from the NASA SCAT-15F concept. The consideration of active controls constituted only a small fraction of the total effort. The active control concepts treated by the individual contractors are listed below.

Contractor	ACT Concepts	Cruise Mach No.
a.	Relaxed static stability	2.7
	Center-of-gravity (c.g.) control (c.g. location measurement)	
	Angle-of-attack limiting	
b.	Relaxed static stability	
	Maneuver load alleviation	2.2
	Gust load alleviation	
	Flutter suppression	
c.	Relaxed static stability	2.7

The results from Source 3 are not published.

Source 4 consists of two studies:

a. A preliminary assessment of active controls benefits to an arrow-wing configuration (fig. 2). The effects of relaxed static stability, maneuver and gust load alleviation, flutter suppression, and ride quality control were considered.

b. Follow-on design development of the arrow-wing configuration. Studies are in progress on alternate methods of balancing the airplane to improve

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performance. Changes in the wing camber and twist of the baseline airplane to improve the cruise lift-drag ratio and the use of relaxed static stability and angle-of-attack limiting for subsonic flight are being investigated.

The results from Source 4 are not published.

SST CHARACTERISTICS

Some of the characteristics, unique to the fixed-wing SST configuration, result in design problems that active controls may resolve. However, some of these characteristics also place constraints on the benefits of active control application. These characteristics are summarized below, together with remarks on their effects. Some of the geometric characteristics are illustrated in figures 1 and 2.

Large Sweepback and Low Aspect Ratio

Advantages
1. Higher cruise lift-drag ratio

2. Lower sonic-boom overpressure

Disadvantages
1. Low maximum lift coefficient ($C_{L_{max}}$) requires low wing loading for reasonable landing speeds

2. Long chord lengths, together with structural requirements, limit the ratio of trailing-edge control surface chord to wing chord to small values. The ratios of control surface areas to wing area are small.

3. Structure contains a large amount of minimum gage material

Small Wing Thickness Ratio

Advantages
1. Lower drag

2. Low stiffness plus sweepback provides some inherent load alleviation by aeroelastic effects

Disadvantages
1. Low stiffness results in reduced control effectiveness from aeroelasticity

2. Low flutter speeds

3. Low natural structural mode frequencies

Long, High-Fineness Ratio Fuselage

Advantages
1. Provides adequate payload volume
2. Lowers sonic-boom pressure

Disadvantages
1. Low natural bending frequencies
2. Aeroelastic effects

Aft-Mounted Engines

Advantages
1. Favorable airframe interference for propulsion efficiency
2. Low noise in passenger compartment

Disadvantages
1. Creates balance problem due to aft-located heavy weight
2. Contributes to lower flutter speeds
3. Space occupied by engines reduces available area for trailing-edge control surfaces

Large Dynamic Pressure

Disadvantages
1. Aggravates adverse aeroelastic effects such as loss of control effectiveness

RESULTS

Performance, Airframe Utilization Efficiency, and Handling Qualities

Relaxed Static Stability

Essentially all recent SST studies (Sources 1, 2, 3a, 3c, and 4b) have advocated the use of a hardened stability augmentation system (HSAS) to provide safe handling qualities for an SST configuration that is inherently statically unstable at subsonic speeds. Hardened means that the reliability must equal that of the airframe structure. Source 3b considers a neutrally stable airplane with a nonhardened SAS. Benefits include either increased range for a given payload (416 km (225 n. mi.) from Sources 1 and 2) or increased payload for a given range. Benefits from Source 3b were expressed in terms of a reduced take-off gross weight (TOGW) of a resized airplane having a fixed

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payload and range. The reduction in TOGW was estimated to be about 18,000 kilo-grams (40,000 lb) for a baseline TOGW of 338,000 kilograms (750,000 lb).

These benefits accrue from an improved lift-drag (L/D) ratio for both cruise and low-speed flight, along with a more efficient utilization of air-frame volume, while retaining safe handling qualities. The need for an HSAS arises from two SST characteristics. One is the shift in aerodynamic center with Mach number as illustrated in figure 3. The other is the aft location of the center of gravity (c.g.) for the operating weight empty condition due to engine locations. These combine to make extremely difficult the longitudinal balancing of the airplane while avoiding or minimizing (1) the need for bal-lasting, (2) unproductive portions of the fuselage that must be kept empty of payload fuel, (3) large tail areas and loads, (4) high trim drag, and (5) unacceptable handling qualities. Some of the considerations of the problem are described in reference 1.

The U.S. SST design features a configuration (fig. 1) that is inherently statically unstable longitudinally (in fact, the c.g. is aft of the maneuver point) at subsonic speeds. An active flight control system was designed to provide good handling qualities for normal operations. This system was backed up by an HSAS designed to provide poor but safe handling qualities with a reliability equal to the airframe structure. In essence, the HSAS is a pitch-rate feedback control that produces an apparent positive maneuver margin. There remained a negative static margin resulting in an unstable phugoid mode; this, however, could be controlled safely by the pilot. This design is documented in reference 1 together with some design guidelines and criteria. The flexi-bility of the airframe was taken into account. A particularly significant problem identified was the difficulty of providing control gains that were high enough for rigid-body mode stabilization without destabilizing the lower fre-quency elastic modes.

Source 3b included a preliminary design of an active control system which consisted of a stability augmentation system (SAS), gust load alleviation, and gust acceleration alleviation (rigid-body mode acceleration) for ride quality improvement. The airplane was considered rigid and the aft-most c.g. location was limited to the neutral point. Thus, the SAS would not need to be hardened as the aircraft could be controlled without it. In this application the benefit stemmed from a reduction in tail volume, hence, decreases in structural weight and in drag. The procedure used in the preliminary design of the system of combined ACT concepts included an optimal method and system practicalization.

In Source 4b, currently under way, the pilosophy is to increase L/D for cruise and lift for landing by arranging an upload on the horizontal tail. For cruise, the airplane is designed to be inherently statically stable. Lift is increased and drag is decreased by means of a small upload on the tail, created by a suitable wing camber and twist ($C_{m_O} > 0$). For landing, the lifting tail load is obtained by designing the airplane to be inherently statically unstable; therefore, an HSAS is required. Thus an advantage is taken of the AC shift with Mach number. This approach is in general agreement with that taken in Source 3c.

In contrast to results of some other relaxed static stability applications, particularly to subsonic transports, this approach will not allow a smaller tail size because the airplane concept features flaps for take off and landing and the associated pitching moments size the tail.

Safe application of the relaxed static stability concept will require the use of an angle-of-attack limiting system or a larger tail surface than required only for stability in order to avoid problems such as lock-in stall or an excessive sink rate.

Center-of-Gravity Control

Even with an HSAS, achievement of a highly efficient SST with good handling qualities is difficult due to the need to allow a substantial tolerance for c.g. location. The benefits of a relaxed static stability HSAS might be greatly enhanced if the c.g. location appropriate to the particular flight speed could be tightly controlled automatically. Source 3a recommends research on defining the requirements for an onboard c.g. measurement system that is a prerequisite to c.g. control.

Angle-of-Attack (Alpha) Limiting System

As previously mentioned with regard to the benefits of relaxed static stability, an angle-of-attack limiter would enhance the benefits of an HSAS. This recommendation is also made in Source 3a which points out the hazard of a lack of warning to the pilot that the airplane is approaching an excessive angle of attack. This may result in a locked-in stall due to exceeding the control authority of the HSAS, or an excessive sink rate. Source 3a suggests the following research: (1) Establish criteria for longitudinal stability and control at the alpha limit; (2) establish any limitations to the applications of an alpha limiter on an SST; (3) synthesize a system for a selected airplane; and (4) validate the system by flight test over the desired flight envelope. An outstanding need is an alpha sensor that is accurate and reliable in an environment featuring a wide range of Mach number, dyynamic pressure, and temperature, and such hostile agents as rain, hail, and bird strikes.

Wing Structural Weight

The potential benefits of maneuver and gust load alleviation, and flutter suppression were explored in Sources 3b and 4a. Both sources recognized that the several concepts must be considered in terms of their aggregate effects and of constraints imposed by structural requirements for other than the controlled quantities. The need for this is discussed with the aid of figure 4 from Source 4a. This chart indicates the structural requirements of the arrow-wing configuration in terms of the individual spanwise distributions of the weight of structure necessary for each of the items listed on the right. These curves are conceptual, not calculated. However, the relative positions of the flutter, maneuver load, and gust load curves are believed to be representative. The extent of the wing area for which some of these structural requirements are dominant for the baseline arrow wing is roughly indicated in figure 5.

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Flutter requirements are likely to be critical for a substantial portion of the wing structure. Thus, a potential benefit from use of a flutter suppression system is indicated. More important, however, is the need for flutter suppression in order to realize any benefits from load alleviation. If flutter is suppressed then the maneuver load becomes critical, and, in turn, if maneuver load alleviation is effective, the gust load may then be critical. If gust load alleviation is effective, the benefits of a combined flutter suppressor, and maneuver load and gust load alleviation system will ultimately be limited by the structural requirements of other loads, such as landing, 6-g crash, and fuel overpressure, and by static stiffness and minimum gage requirements.

If appreciable reduction of the structural material is obtained from load alleviation, the burden on the flutter suppression system is increased over that required to only remove the flutter weight penalty with respect to the unalleviated wing. For baseline structures that are flutter free, effective load alleviation may require flutter suppression.

To summarize, some of the normally noncritical structural requirements may become critical, contingent on the use of active controls. It is also probable that structural requirements not subject to active controls will significantly constrain the benefits from active controls.

Flutter Suppression

To provide adequate flutter speeds by conventional techniques for an arrow-wing airplane, it is estimated in Source 3b that the weight of material added to the strength-designed wing is in the range of 1800 to 2700 kilograms (4000 to 6000 lb). A candidate flutter suppression system was designed that reduced this penalty by about 680 kilograms (1500 lb). This study was a relatively small effort. Presumably, a larger effort might provide a system of greater effectiveness.

Maneuver Load Alleviation

Maneuver load alleviation (MLA) was considered in Sources 3b and 4a. The results, based on calculations for a rigid airplane, varied from 5 to 9 percent reduction in wing root bending moment, depending on the flight condition assumed to be critical. The estimated attendant reductions in structural weight ranged from 450 to 1010 kilograms (1000 to 2200 lb). These figures are probably opti-mistic because the constraints from other structural requirements, discussed earlier, were not imposed.

The effect of these constraints is illustrated conceptually in figure 6. If the requirements for gust and other loads and for minimum gage, etc., exceed that for the alleviated maneuver load, only a fraction of the reduced weight benefit can be realized as indicated by the shaded area in figure 6. The utilization of gust load alleviation would relax, but not eliminate these con-straints. Another constraint, not included in the study, is the effect of aeroelastic deformations on MLA performance. The influence of aeroelasticity on control surface effectiveness is touched upon subsequently in this section

of the paper, but the overall alleviation of loads on the flexible wing is not evaluated.

In assessing benefits of MLA an adverse side effect must be recognized. Effective MLA will increase the mean (one g) stress level over that of the unalleviated wing. This will substantially increase the fatigue damage rate. In view of the predominant effect of the ground-air-ground cycle on fatigue, this may be a significant additional structural requirement. On the other hand, the MLA concept can be used to increase fatigue life if the strength requirements of the unalleviated airplane are retained.

For the sake of generality, it is of interest to examine the properties of the control surfaces for maneuver load alleviation. As can be observed for the arrow wing in figure 7, the total area of the usable control surfaces is a small percentage of the wing area. An increase in control surface area by increasing the span of the control is precluded by the space required by the engines. An increase in control surface chord is restricted by the wing box structure. It is likely that the outboard surfaces 1 and 2, shown crossed out in figure 7, will not be usable due to loss of effectiveness from aeroelastic deformation. The inboard surface 3 between the engines may also suffer a large loss in effectiveness in supersonic flight. However, the load alleviation inherent in flexible sweptback wings at high dynamic pressure reduces the need for active alleviation at supersonic speeds. The need for MLA is likely to be highest at transonic speeds having dynamic pressures that are lower than those for cruise. For the transonic condition, the effect of aeroelasticity on surfaces 3 and 4, shown shaded in figure 7, is not as severe. These surfaces were used in load alleviation calculations in Source 4a.

The influence of these small separate surfaces on the theoretical spanwise aerodynamic load distribution is shown in figure 8 from Source 4a for the arrow wing. These results were obtained using Woodward aerodynamics for a $M = 1.2$, lightweight condition, assuming a rigid structure and maximum control surface deflections of 20°. The reduction in net (aerodynamic and inertia) bending moment can be shown to be about 5 percent at the root and about 9 percent at the mid-semispan station.

Although only effects on bending moment were examined, the additional chordwise loads accompanying the control surface deflections may be significant. Also significant may be the differences between theoretical and actual loads at limit load levels due to nonlinear aerodynamic phenomenon such as flow separation on control surfaces and pressure limiting.

Gust Load Alleviation

Gust load alleviation (GLA) was considered in Sources 3b and 4a. GLA is defined herein as the reduction of the rigid-body-mode gust load responses. The load increments from vibration of structural modes are not accounted for. It is assumed that these would be controlled by a mode suppression system which is mentioned under the subject of ride quality control. The results for a rigid airplane in terms of structural weight reduction, assuming no constraints from other structural requirements, varied from zero to about 225 kilograms (500 lb)

As in the case of maneuver load alleviation, these constraints may reduce the higher value cited. The magnitude of benefits can vary depending on bookkeeping methods. If GLA is needed to realize the benefits of MLA then the somewhat greater benefit of MLA may be attributed to GLA as well.

The reason for the small benefits of GLA to the arrow-wing configuration is that the airpalne is somewhat less sensitive to gusts than subsonic jets. The gust load factor for the arrow-wing airplane was estimated in Source 4a to be about 2.0g in contrast to the 2.5g maneuver limit load factor. At first glance, this seemed surprising in view of the low wing loading (lowest value is approximately 1900 newtons/meter^2 (40 lb/sq ft). However, the low wing loading is compensated for by the characteristically low value of lift curve slope for highly swept, low-aspect-ratio wings.

For reasons given in the discussion of MLA, the available control surfaces for GLA are the two inboard surfaces. It is of interest to note that the sense of the deflection of these inboard surfaces for GLA is opposite to that for MLA. For example, for the alleviation of a positive maneuver load the trailing edges of the controls should deflect downward, whereas for the alleviation of a positive gust load the trailing edges should deflect upward. For outboard control surfaces, were they effective, the sense of the deflection for MLA and GLA would be the same.

Ride Quality

The unpleasant accelerations during flight in turbulence can be regarded as arising from two sources; (1) the response of the airplane rigid-body modes and (2) the vibratory response of the elastic modes. These are illustrated schematically in figure 9. The total vertical accleration response is shown by the sketch at the top and consists of the sum of high frequency structural oscillations and lower frequency rigid-body-mode responses. The use of mode suppression by means of small canards or other auxiliary control surfaces to reduce the structural vibration may be necessary as suggested by the flight experience with the XB-70 airplane and by Source 3b. (Source 3b indicates that need is marginal.) Effective mode suppression would then leave the rigid-body-mode acceleration as indicated by the middle sketches in figure 9. The rigid-body-mode accelerations can be controlled by gust acceleration alleviation. However, for the SST these lower frequency responses are not likely to be objectionable on the basis of gust sensitivity estimates in Sources 3b and 4a.

The gust sensitivity, $\bar{A} = \dfrac{\sigma_{\Delta n}}{\sigma_{w_g}} \approx 0.01$ (ratio of root-mean-square values of acceleration and gust velocity) for the rigid-body modes is well below values for subsonic jet transports. It is just as well, for the effectiveness of the available control surfaces to alter the wing lift for the reduction of low frequency gust accelerations, indicated by the bottom sketch in figure 9, is low.

In general, effective use of maneuver and gust load alleviation (of loads from rigid-body-mode responses) will tend to increase the severity of structural

vibrations over that of the unalleviated wing and, therefore, increase the need for mode suppression. Incidentally, the mode suppression may provide a degree of flutter suppression and vice versa. It could be advantageous to combine the two concepts.

Other

Although they are not found in most lists of active control systems, there are two other concepts that may benefit an SST.

One is the concept of automatically controlling the airplane lateral transient accompanying an engine unstart at supersonic speeds. Conceivably, following an unstart the airplane could be disturbed so rapidly that the pilot could not apply corrective action before the vehicle exceeded design loads or a controllable angle of attack or sidelsip. The second concept is an actively controlled landing strut to reduce the loads and unpleasant motions of the elongated SST during taxi runs. Research on this concept is being conducted at the Langley Research Center.

CONCLUDING REMARKS

The information surveyed indicates that some significant benefits to SST designs may be obtained through active controls. There is considerable agreement that a large transport will require active stabilization of an inherently statically unstable condition at subsonic speeds. The benefit of the relaxed static stability may be increased by use of center-of-gravity control and angle-of-attack limiting. Benefits from maneuver and gust load alleviation may be small for the arrow-wing concept. In any case, load alleviation most likely will require that flutter suppression be used as well. Flutter suppression in itself may provide some saving in structural weight. Ride quality control by a mode suppression system may be needed for passenger acceptance. For safety, active lateral control should be considered for limiting the magnitude of the transient motion due to an engine unstart.

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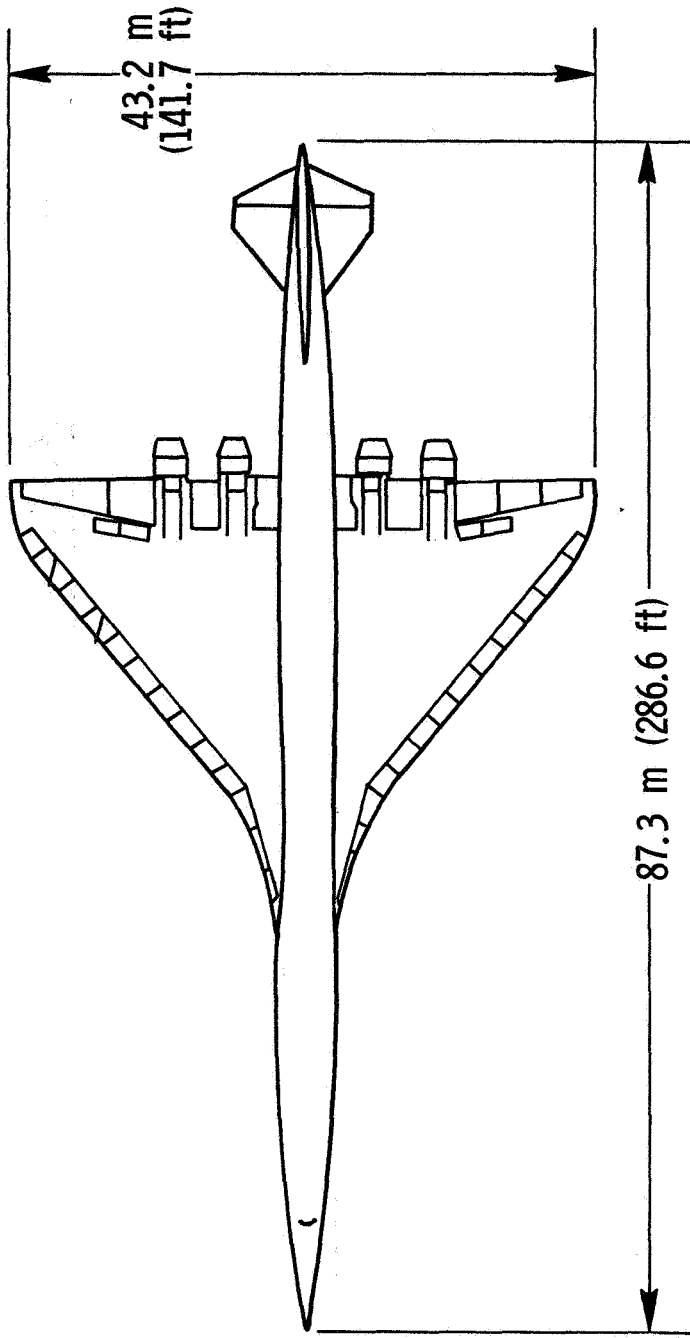
Figure 1.- United States SST design.
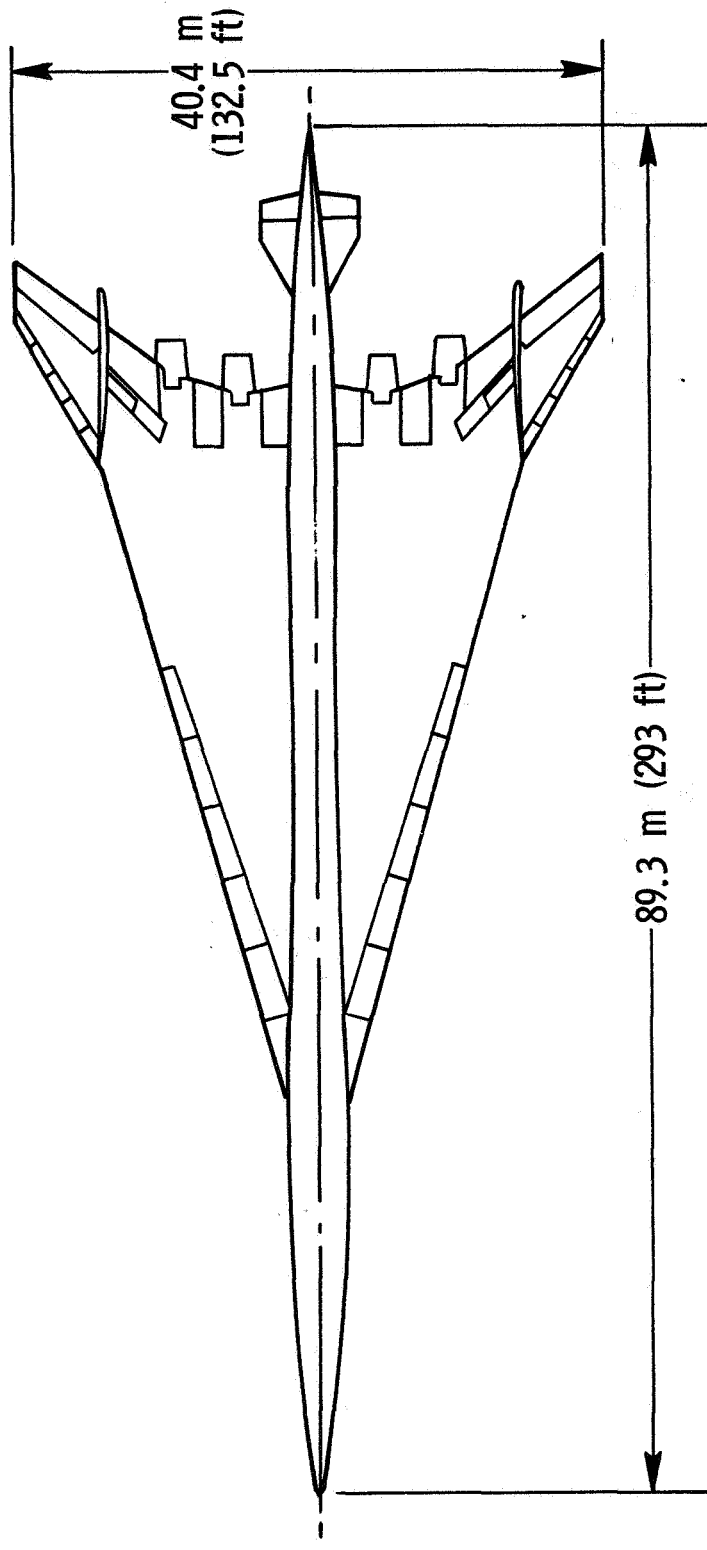
Figure 2.- Arrow-wing design.
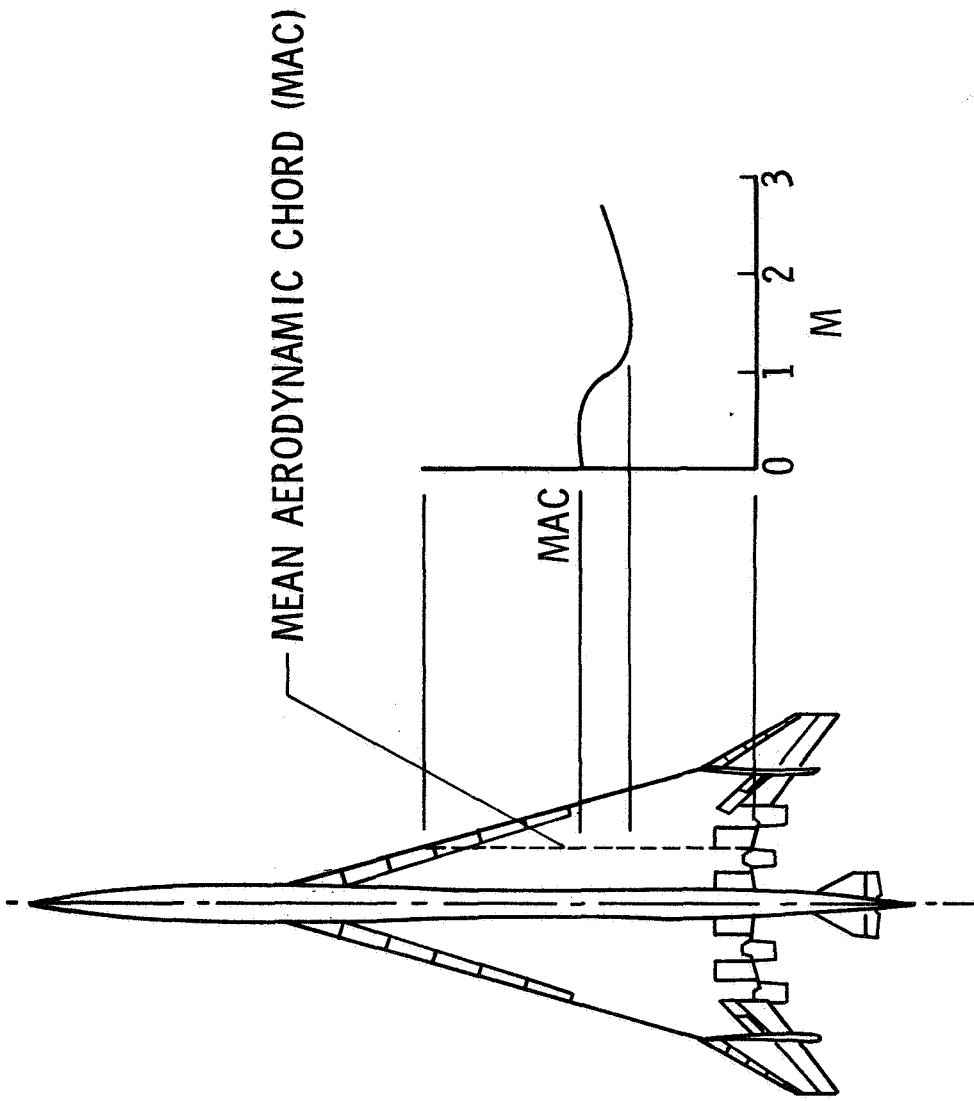
Figure 3.- Variation of airplane aerodynamic center (AC) with Mach number (M).
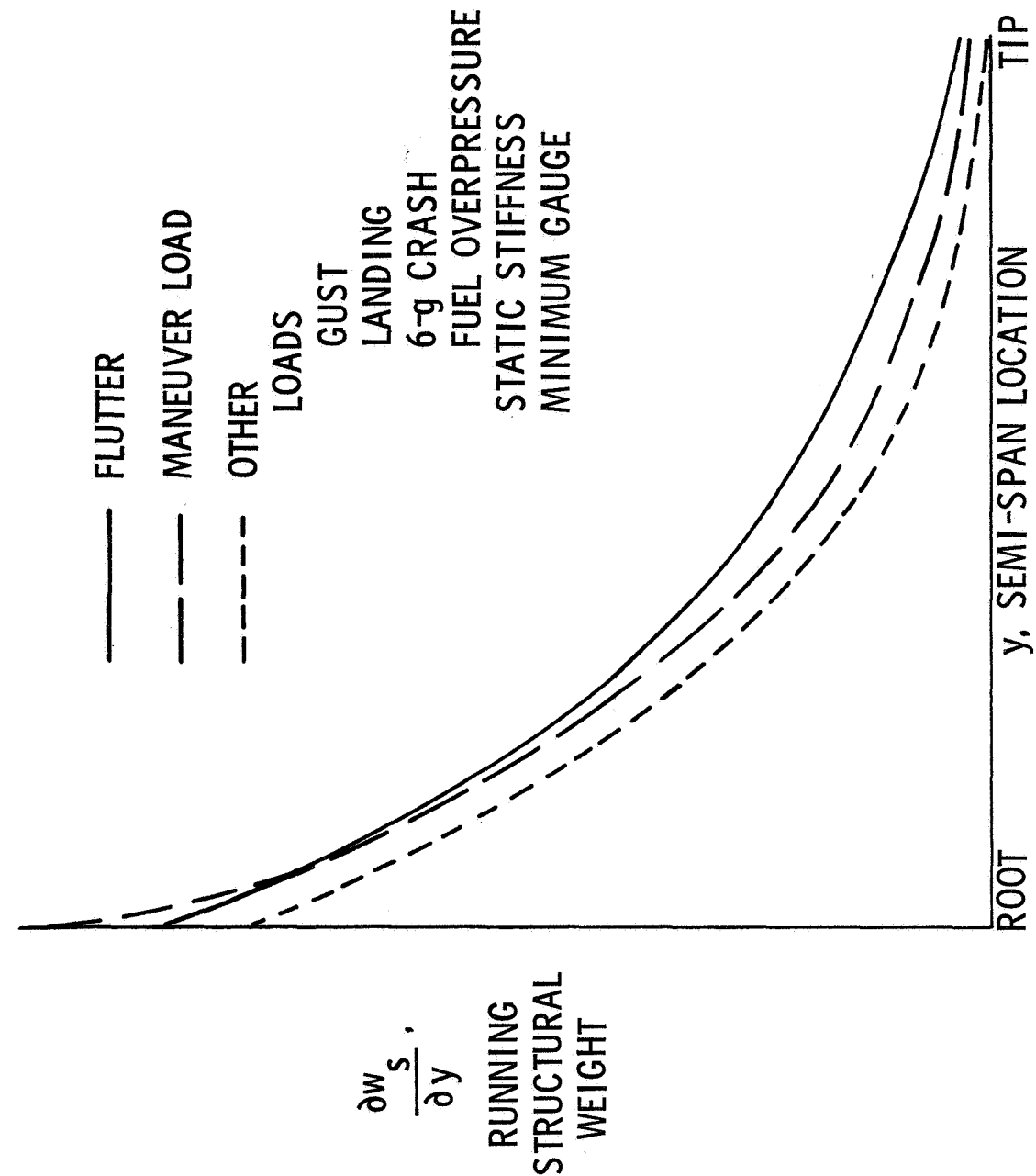
Figure 4.– Wing structural requirements of baseline arrow-wing airplane.

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FLUTTER

MANEUVER LOAD

MINIMUM GAUGE
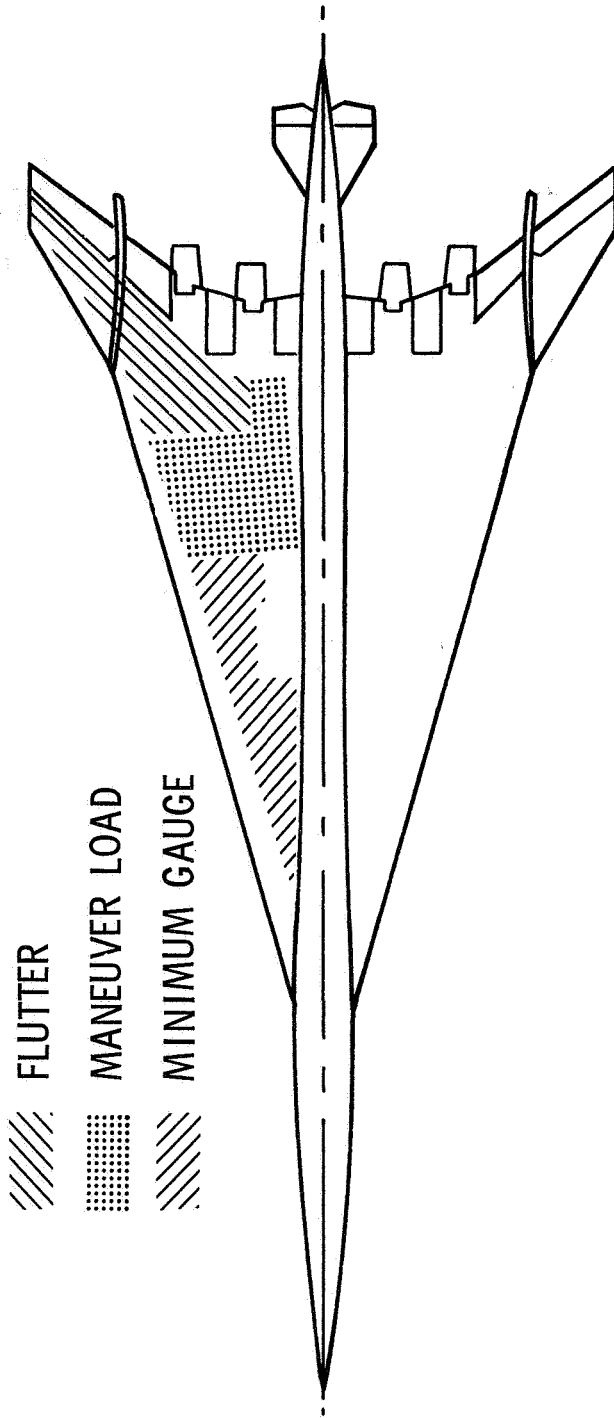
Figure 5.- Critical structural requirements for baseline arrow wing.

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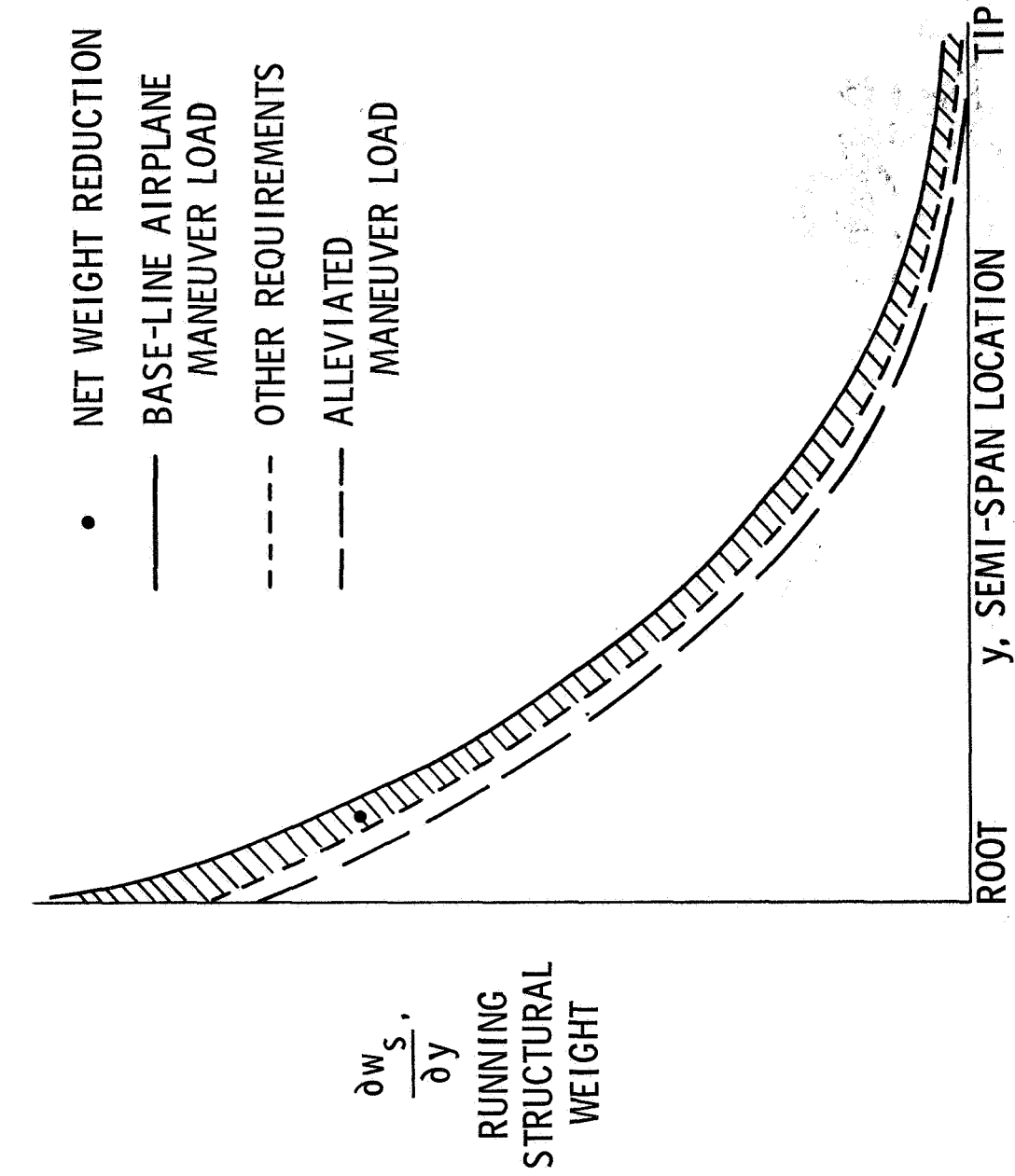
Figure 6.– Structural requirement constraint for maneuver load alleviation.
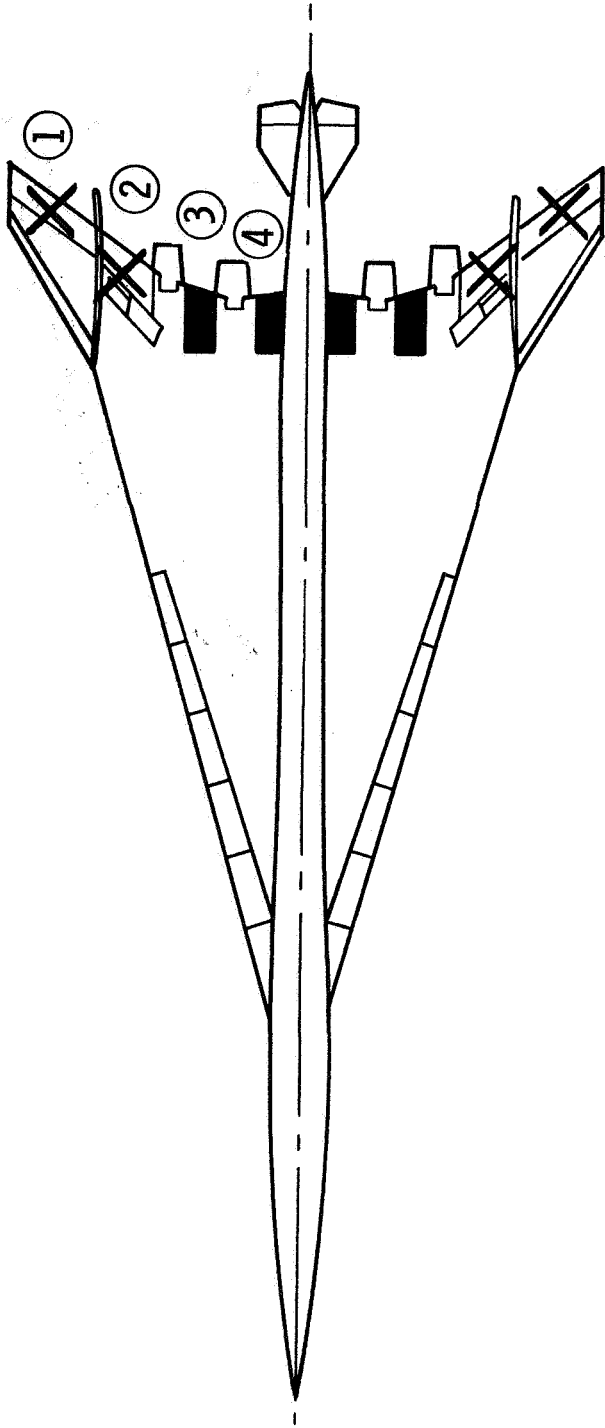
Figure 7.- Control surfaces for maneuver and gust load alleviation.
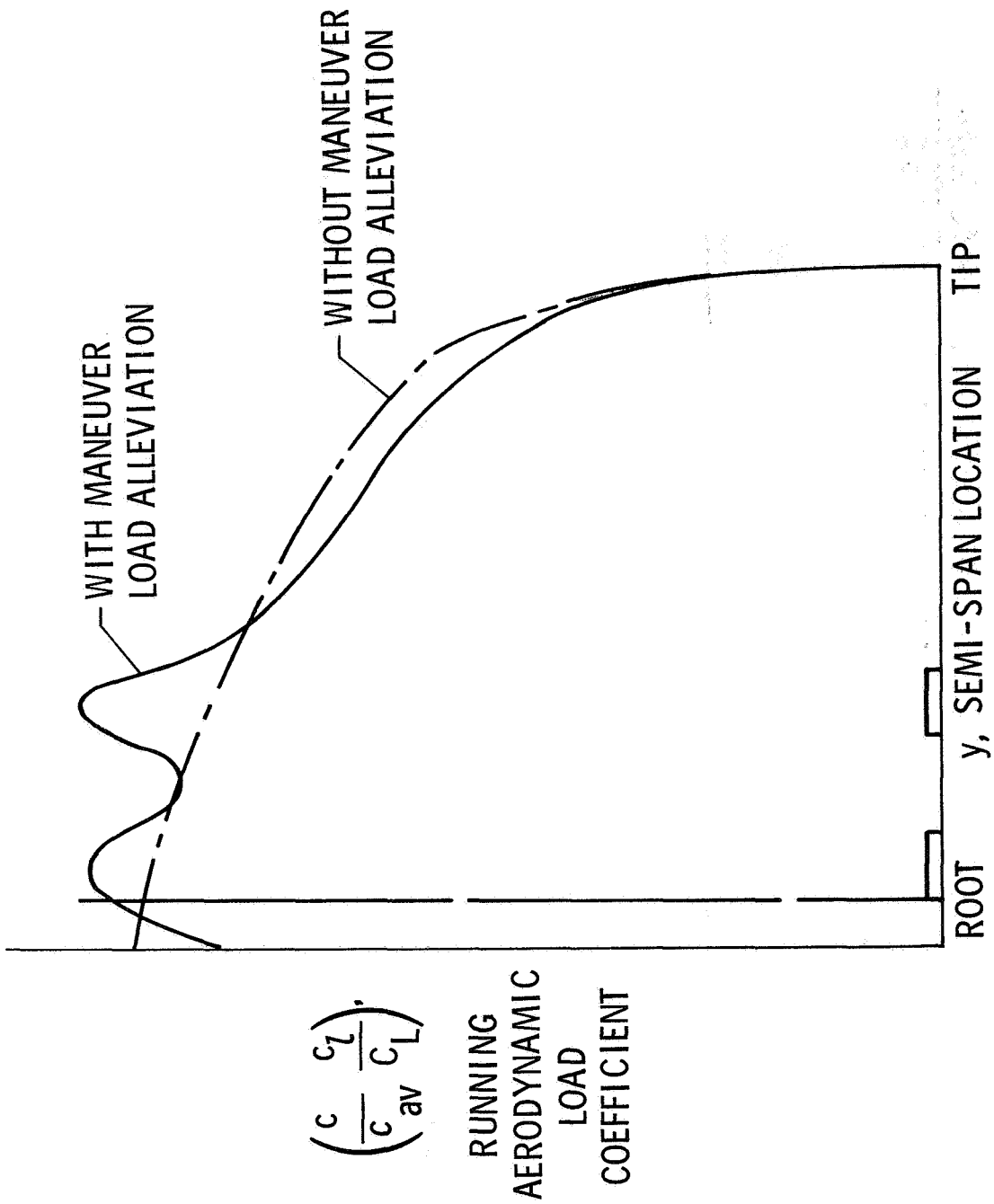
Figure 8.- Spanwise aerodynamic load distributions at limit load, Mach number 1.2, control deflection 20°.

TURBULENCE

WITHOUT ACTIVE CONTROL   VERTICAL ACCELERATION

WITH MODE SUPPRESSION

PLUS GUST ACCELERATION ALLEVIATION
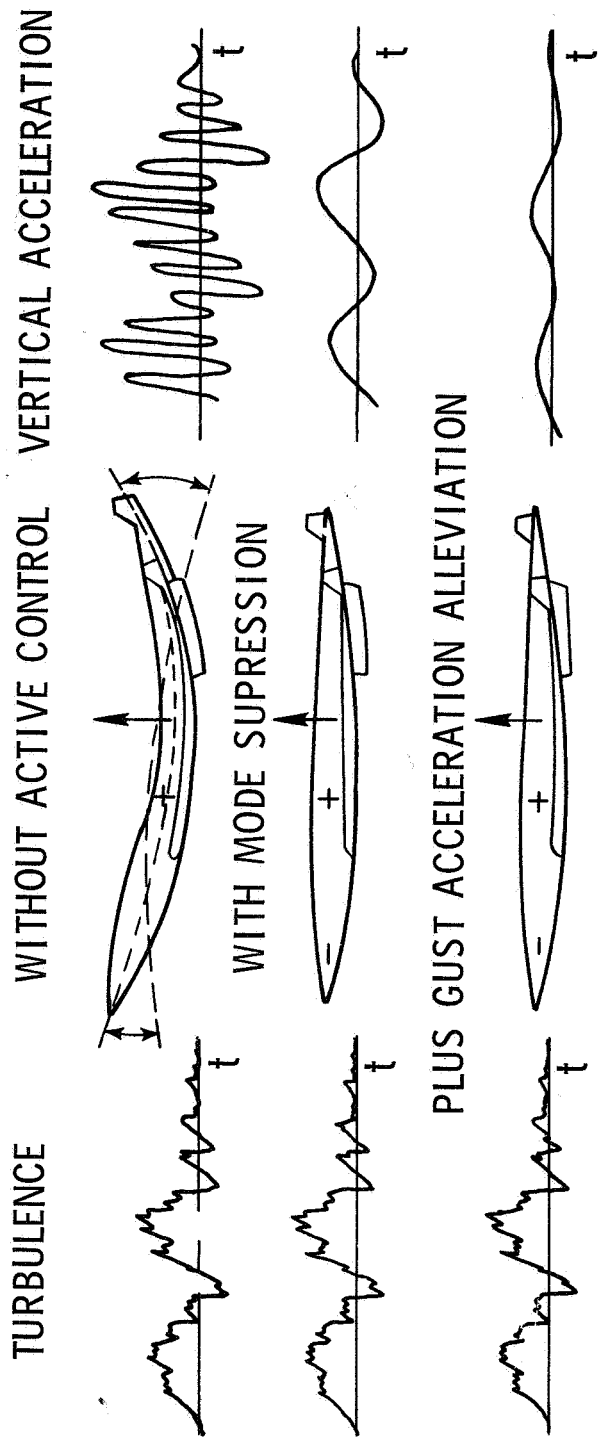
Figure 9.- Ride quality control with mode suppression and gust acceleration alleviation.

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ESTABLISHING CONFIDENCE IN CCV/ACT TECHNOLOGY

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SUMMARY

Despite significant advancements in Controls Configured Vehicles/Active Controls Technology (CCV/ACT) in the past decade, few applications of this promising technology have appeared in recent aircraft designs. This paper briefly summarizes the status of CCV/ACT, describes some of the constraints which are retarding its wider application, and offers some suggestions toward establishing an increased level of confidence in the technology.

INTRODUCTION

Major advancements have been accomplished in flight control technology during the past decade, particularly in the areas of fly-by-wire, active controls and, more recently, digital controls. The next generation of U. S. commercial transports must take advantage of benefits achievable from these advanced techniques to remain competitive in the world market. European aircraft industries have major advanced flight control programs underway and are making significant progress in this field. The United States space program, research aircraft programs and military advanced development programs have brought ACT digital FBW (fly-by-wire) technology to a level where substantial benefits can be realized in the near future. However, the commercial aircraft industry, airline industry, and government civil aviation agencies must be convinced that an aircraft designed around this advanced technology will achieve predicted performance and be safe, reliable, operationally practical and cost effective. Commercial acceptance of any new technology will occur only when sufficient test data are generated to clearly demonstrate that these criteria can be met with reasonable risk on a new airplane design.

Most of the progress to date in this field has been accomplished primarily on four aircraft: the XB-70, B-52, F-4 and F-8. Programs on these aircraft are making significant necessary contributions, but the programs are experimental in nature, conducted to demonstrate concept feasibility under carefully restricted flight conditions in evacuable military aircraft with ejection seats. This paper briefly summarizes the state-of-the-art of CCV/ACT technology and suggests some approaches to the problem of developing a wider level of confidence in that technology.

BACKGROUND

In the past decade, potential benefits of advanced flight control technology have been shown by a large number of theoretical analyses and by several USAF and/or NASA flight demonstration programs. Table I summarizes briefly the results of most of these efforts. References 1 and 2 provide a more complete summary.

Major Air Force experimental flight research programs involving load alleviation and fatigue damage rate reduction by structural mode control techniques were the B-52 LAMS (Load Alleviation and Mode Stabilization) and the XB-70 GASDSAS (Gust Alleviation and Structural Dynamic Stability Augmentation System) programs. Concurrently, an advanced stability augmentation system (SAS) was developed and incorporated on the B-52G and H fleet to reduce fatigue damage rate during low level, high speed flight. The Air Force Control Configured Vehicle (CCV) research program has completed flight demonstration of four ACT concepts at selected flight conditions on a B-52E aircraft: ride control, flutter mode control, maneuver load control, and augmented stability. In addition, the compatibility of a LAMS system with these four concepts was also demonstrated. Goals for each concept were successfully achieved individually and collectively during the program.

Other flight programs have incorporated limited ACT concepts in recently designed military and commercial aircraft. Reduction of lateral gust loads on the L-1011 transport with an advanced yaw damper resulted in a 20 percent reduction of limit design loads. A Gust Response Suppression System has been developed for the 747 to improve passenger ride qualities in the aft section. The system is currently being evaluated by Qantas Airways. A ride control system is being designed for the B-1 strategic bomber, using structural mode control techniques, to improve crew ride qualities during terrain following missions. An Active Lift Distribution Control System (ALDCS) is being designed for the C-5A airplane to reduce wing design limit maneuver and gust loads and wing fatigue damage rate. The General Dynamics prototype lightweight fighter, the YF-16, has a quadruply-redundant analog, FBW control system without mechanical backup. Relaxed inherent stability is integrated into the aircraft design to reduce drag and gross weight.

The first serious commitment to including an ACT concept in a commercial transport occurred during the recent National SST program. The SST was configured with relaxed longitudinal static stability to achieve necessary gains in range-payload from reduced gross weight and drag. Experience gained from development of fail-passive and fail-operational/fail-passive autoland systems at Boeing during the 1960's provided confidence that a suitable flight control system could be developed to meet SST safety and operational requirement

The resulting SST longitudinal command and stability augmentation system providing basic airplane safety was fail-operational squared (fail-operate after second failure), utilizing quadruply redundant sensors, analog electronic channels and actuators. A mechanical reversion back-up mode was retained (a discussion of this system is contained in Reference 3).

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Cancellation of the SST program precluded thorough development and flight test evaluation of the SST flight control system. Advanced technology items which include electronic display and control system components were, however, government funded for further development under the DOT/SST Technology follow-on program (Contract DOT-FA72-WA-2893).

Full realization of advanced control function potential on production aircraft depends on fly-by-wire control systems with a reliability consistent with the function criticality. Two programs, the Air Force F-4 680J Survivable Flight Control System and the NASA F-8C digital fly-by-wire program, are directed toward developing and flight demonstrating FBW systems on fighter aircraft. As a result, Reference 4 states that "with successful completion of the 680J flight test program, analog fly-by-wire control techniques, equipment mechanization and fundamental criteria are now fully validated".

Most advanced FBW flight control systems have used analog implementation techniques. Research is now underway to exploit the advantages of digital control, demonstrated, in part, by the Apollo space program. The recent extremely rapid progress in microcircuitry has made digital control hardware competitive with analog hardware in terms of cost, reliability, size and weight. Further, digital techniques offer significant advantages for advanced control laws, redundancy logic and built-in testing functions. One of the first programs to study digital flight control implementation problems on aircraft is the NASA F-8 program which successfully demonstrated a single channel digital FBW primary flight control system with a triply redundant analog backup system. Other digital control research programs, such as the Digital Avionics Integrated Systems (DAIS), the SST Follow-On Technology, and the planned Tactical Aircraft Digital System (TADS), are contributing to this technology base. Other Air Force programs are investigating the application of multiplexing techniques to flight control systems. Further, research efforts within the U. S. and European flight control system component manufacturers are studying fiber optics for providing signal transmissions immune to electromagnetic interference.

An overall assessment of advanced flight control technology over the past decade indicates that considerable progress has been achieved:

o Performance of CCV functions has been flight demonstrated on a large flexible aircraft

o Digital and analog FBW systems have been flight demonstrated on fighter aircraft

o A prototype lightweight fighter has been designed around CCV analog FBW techniques.

CONSTRAINTS ON THE USE OF CCV/ACT TECHNOLOGY

Despite the large amount of analytical and flight test data available, no CCV/ACT concepts are currently in general use in commercial transport aircraft. Only the simplest form of augmented stability--the yaw damper-- is in widespread use in commercial aircraft today. Primary applications are to improve handling qualities and to increase the comfort level of the crew and passengers. In a few instances, a yaw damper was necessary for certification. Although these are examples of beneficial applications, the systems were generally added after the airplane was designed, sometimes after the first model flew. In most instances, a much greater benefit would have been possible if a full-time directional stability augmentation system had been assumed from the beginning of the design.

There are a number of constraints that have effectively delayed the widespread implementation of these systems. A most fundamental constraint is risk, principally on the part of the airframe manufacturer. As has been pointed out, the maximum potential benefit of these advanced concepts is achieved if they are incorporated into the design at the outset. However, the final assessment of the benefit results from an exhaustive design process that is expensive and time consuming, and for which the correlation with hardware results is not at all certain. The real risk is that a major problem may arise after program commitment of an airplane design predicated on successful system performance.

Figure 1, reproduced from Reference 46, illustrates this concern. At program go-ahead, with only 3% of the eventual total program cost actually spent, management action can influence total program cost by 20% at most. Consequently, a program that relies on advanced systems will either require a significant increase in analysis confidence, or a program structure like the U. S. SST where an engineering prototype precedes production commitment. In other words, one way of eliminating the risk is to have a "proof before use" program plan, which adds to program time and cost.

Another constraint is the cost of these systems, including development, certification, and maintenance cost. Bright spots in the cost picture are the rapidly developing field of digital systems for aircraft applications and the reductions in analog/digital system cost disparity.

A third constraint is the lack of confidence in the analysis tools and the correlation between analytical models and the real world. For example, if a new airplane were to depend on flutter mode control for flutter safety, there would be little margin for error between the analytical model and the hardware. Yet the state of the art of flutter analysis can accomplish this today only by "fine tuning" the analysis with hardware data.

A final constraint is the reluctance on the part of the user, the airlines, to increase maintenance costs. Consequently, there is great reluctance to buy a system, almost independent of its performance benefits, unless there is a proven method of keeping the maintenance burden in hand.

It is generally true that maintenance constitutes about one-quarter of the total direct operating costs for current airplanes. Therefore, complexity such as discussed here should be accompanied by systems designed to hold the line on, or lower, maintenance costs. Digital systems, with improved self-check capability, may provide a solution to this problem.

REMOVING THE CONSTRAINTS: DEVELOPING CONFIDENCE

Commercial realization of the benefits associated with advanced control concepts will occur only when these constraints are removed through comprehensive development and demonstration of necessary methods and components.

Flight Demonstration

Most of the progress to date in this field has been accomplished primarily on four aircraft: the B-52, F-4, F-8 and XB-70. Programs involving the military aircraft are making significant necessary contributions, but the programs are experimental in nature, conducted to demonstrate concept feasibility under carefully restricted flight conditions in evacuable aircraft with ejection seats.

The next logical program should expand this technology base by developing and flight demonstrating an operationally practical advanced digital fly-by-wire control system on a commercial aircraft. The system should be designed to function throughout the flight envelope, from takeoff to landing, under normal and extreme operating conditions. It should include appropriate redundancy management, automated system test, system control, and system status and advisory displays. Extensive flight testing must be conducted to define system performance (compared to analytical predictions), reliability, failure effects, and maintainability requirements under conditions representative of commercial airline operation. Realistic design criteria and design guidelines should be developed, based on results of the program, for critical and noncritical control functions. This program should also be responsive to technology recommendations expressed in the NASA Research and Technology Advisory Council report (Reference 47).

A flight demonstration program formulated to satisfy these objectives and requirements could best achieve credible test results by utilizing a current state-of-the-art, operational, commercial aircraft as the test vehicle. The NASA RSFS airplane is well qualified as a test vehicle for demonstrating certain elements of an advanced control system. This aircraft is currently

being converted into a commercial-type research vehicle under Department of Transportation SST Technology Follow-On Program and NASA Research Support Flight System contracts (References 55 and 56). Arrangement of the new experimental flight control, navigation, and display equipment being installed for the RSFS is shown in the cutaway view of Figure 2. The aircraft features an aft flight deck (AFD) from which a two-man crew may fly the airplane from takeoff to landing with controls electrically coupled to the standard 737 flight control system. Advanced electronic systems include triply redundant digital automatic flight control computers and an advanced digital navigation, guidance, and display system. A data acquisition system provides experimental data for postflight analyses. The complete system (sensor to control surface output), which has an electronic fail-operative capability, is limited to a single thread actuation capability.

The current and planned use of the aircraft is for extensive NASA research programs regarding flight in the terminal area. The experience obtained during these programs in the operation and performance of the fail-operational sensor and computer system will be directly applicable to advanced control system development.

The aircraft, as currently configured, has the capability to provide meaningful performance, reliability, and maintainability test data in a limited-cost flight program. The aircraft could also be modified to provide sufficient system redundancy and capability to more completely model, and thereby provide better data on, the performance benefits, reliability, and maintainability of these systems. This possibility is being explored by Boeing-Wichita under contract to NASA-Langley.

Redundancy Management

The redundancy required for a flight-critical control system depends to a great extent on the mechanization scheme adopted, as well as the failure characteristics of the system under consideration. Similar considerations apply whether the element is a mechanical actuator or an electronic element. DOT-sponsored research (References 53 and 54) examined the elements of the U. S. SST prototype control system and identified problem areas associated with the redundancy of those systems, e.g., channel interactions, failure detection, and failure effects on system performance. A NASA-sponsored study (Reference 27) reviewed ten current actuator redundancy mechanization schemes and identified two concepts that would meet advanced airplane flight control system requirements.

These results are cited as evidence that work is proceeding in this area. But it must be pointed out that since computation and actuation are key elements of any control system, the promise of advanced controls will not be realized until the technology for providing adequate reliability with reasonable system cost is in hand. Appropriate research must be carried out in this area of redundancy management to ensure that the design capability is available when needed.

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Improved Analysis Techniques

Existing methods cannot provide the technological base for the design of airplane configurations that rely on a control system for structural design load reduction, flutter envelope expansion, or stability when balanced for optimum performance. Some of the more fundamental problem areas are: transonic, nonlinear, and unsteady aerodynamics; interaction of structural deformation and control surface deflections with aerodynamic loading; and the dynamics of large flexible structures. Reference 47 points out that these problems "have been painfully evident to aeroelasticians for at least three decades."

Current analytical capability must be expanded to provide adequate treatment of these problem areas. Analytical methods for the rapid incorporation of experimental data into the analyses should be pursued with the objective of successfully treating separated flow regions and nonlinearities due to discontinuities. The methods resulting from this work should be verified through correlation with wind tunnel and flight test data. A few tentative steps are being taken in this direction, but much additional work remains.

In the past, wind-tunnel testing of dynamically scaled airplane models has proven economically desirable to predict airplane dynamic characteristics prior to flight testing. As aircraft become more dependent on stability augmentation systems, wind-tunnel testing of aeroelastic models to prove control concepts will become increasingly more attractive to increase confidence in analyses, as discussed in Reference 48.

In 1967, AFFDL and NASA-Langley jointly initiated a program to demonstrate an active modal suppression system on a one-thirtieth scale B-52E aeroelastic model in the Langley transonic dynamics tunnel. This model includes aileron and elevator actuation systems and provisions for a cable mount system (Reference 49). Model gust responses have been obtained using the airstream oscillator system installed in the tunnel (Reference 50). Boeing-Wichita is assisting NASA in developing a ride smoothing system for the model using 50 Hz bandwidth aileron and elevator actuation systems. Subsequently, canards and flaperons were added for RC and FMC testing, which is now nearing completion. In 1974 a MLC system will be tested.

In addition, wind tunnel tests have been conducted at NASA-Langley on a SST wing model which utilizes a FMC system (References 51 and 52). Wider use of such models will be of great benefit in CCV system synthesis and test.

Results of the recently completed B-52 CCV program indicate that precise mathematical models may not be quite as vital as stated above. Inaccuracies in the math model may be made tolerable by intelligent location of force producers, and use of motion sensors located at several different points in the structure to be controlled (Reference 57).

CONCLUDING REMARKS

Within the past decade a great amount of work has been performed to demonstrate benefits of active controls technology, yet today applications of this technology are few. The best way to develop confidence in these concepts is to flight demonstrate the concepts on a commercial transport under normal and extreme operating conditions. Such a program will clearly demonstrate and establish confidence in CCV/ACT technology.

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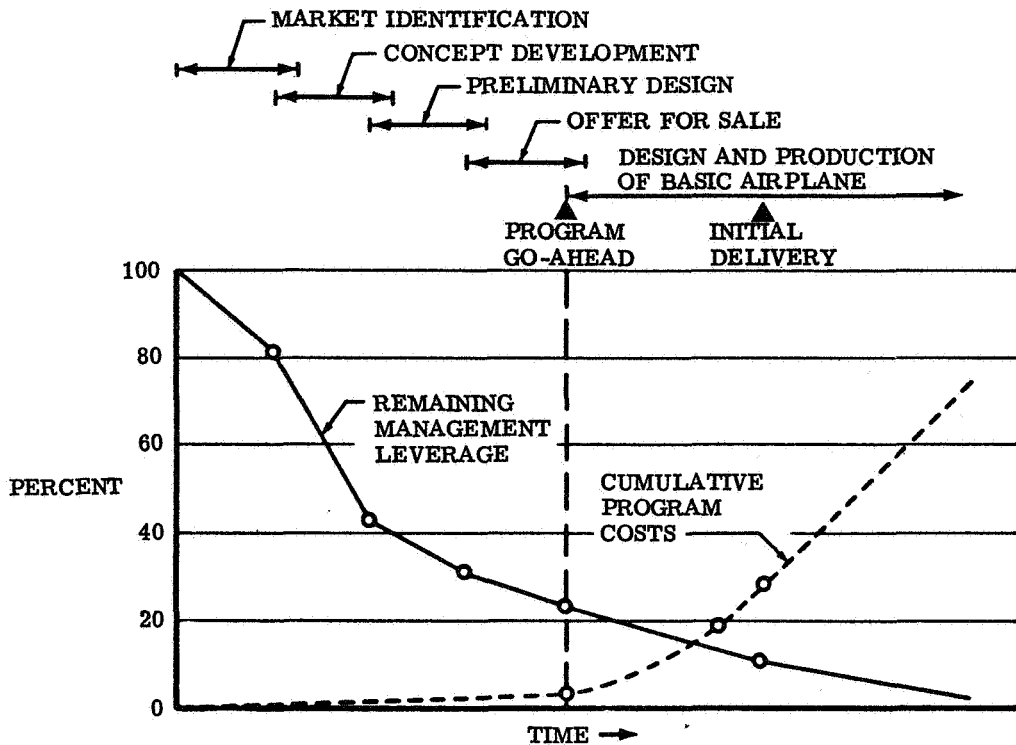
FIGURE 1. COST MANAGEMENT OF A TYPICAL COMMERCIAL PROGRAM
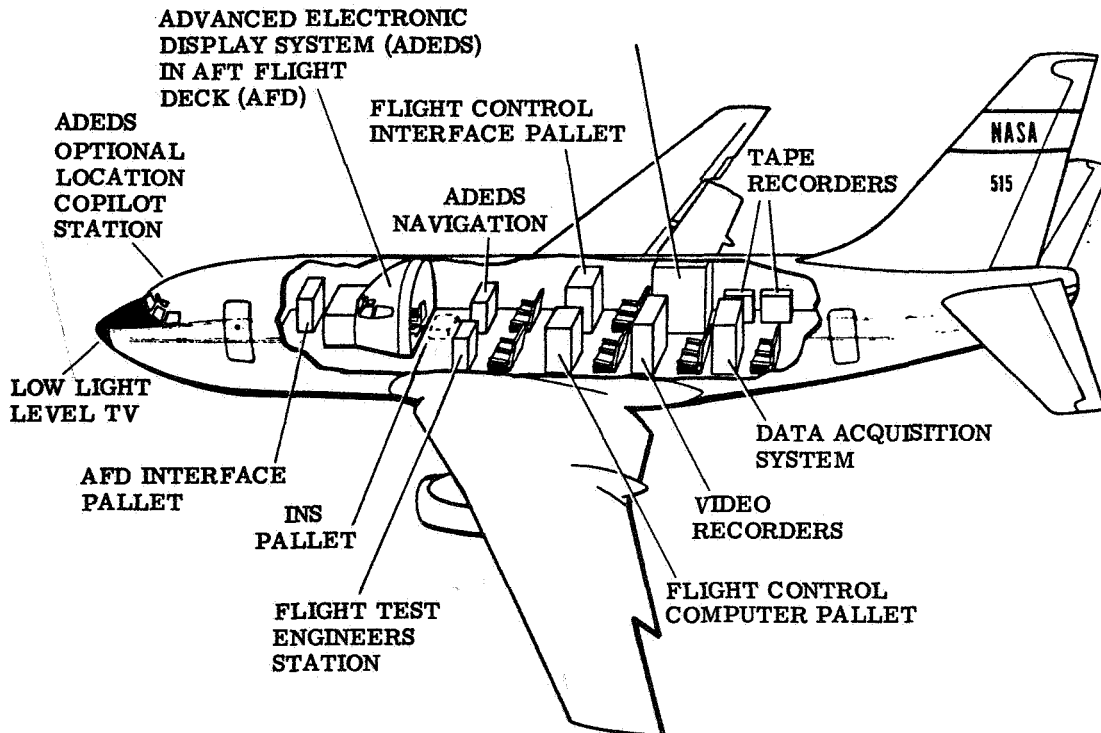
FIGURE 2 - RESEARCH SUPPORT FLIGHT SYSTEM -
INTERNAL ARRANGEMENT

TABLE I
CCV/ACT PERFORMANCE PAY-OFF STUDIES

	CCV/ACT CONCEPT / A/P STUDIED OR TESTED	AUGMENTED STABILITY (AS)	GUST LOAD ALLEVIATION (GLA)	FATIGUE REDUCTION (FR)	MANEUVER LOAD CONTROL (MLC)	RIDE CONTROL (RC)	FLUTTER MODE CONTROL (FMC)
BOMBERS AND TRANSPORTS	B-52	13.7% G.WT. REDUCTION POSSIBLE. SEE REFS. 5-6. ___ CCV PROGRAM (REF. 57) FLIGHT DEMO. WITH NEUTRAL STATIC LONGITUDINAL STABILITY.	SEE REFS. 8-18	SEE REFS. 8-18	5% G.WT. REDUCTION POSSIBLE SEE REFS. 5-6. ___ CCV PROGRAM FLIGHT DEMO. 10% REDUCTION IN WING ROOT BENDING DUE TO MANEUVERS. (REF. 57)	CCV PROGRAM (REF. 57) FLIGHT DEMO. 30% ACCELERATION REDUCTION AT PILOT'S STATION, BOTH VERTICAL & LATERAL.	CCV PROGRAM (REF. 57) FLIGHT DEMO. 30% EXTENSION IN FLUTTER PLACARD.
	XB-70			SEE REFS. 19-25		SEE REFS. 19-25	
	C-5A			SEE REFS. 26-27	LOAD DISTRIBUTION CONTROL SYSTEM (LDCS) FLOWN IN 1969. (REF. 27)		
	LOCKHEED DOUBLE-DELTA SST					50% VERTICAL ACCELERATION REDUCTION. (REF. 28)	
	LIGHT WING-LOADING STOL TRANSPORT		MECHANICAL FLAP LWL DESIGN APPROX. 12% LIGHTER G.WT. THAN EXT'L BLOWN FLAP DESIGNS. (REF. 29)			VERTICAL ACCEL. $\leq .11$ g'S RMS. LATERAL ACCEL. $\leq .055$ g'S RMS. (REF. 30)	
	LARGE JET TRANSPORT (700,000 LB. CLASS)				MLC PROVIDED 10,000 LB PAY-LOAD INCREASE. (REF. 31)	35-50% RED'N. IN GUST-INDUCED LATERAL ACCELERATION IN 747 AFT BODY (REFS 2 & 58)	
	U.S. SST	6,000 LB. WT. SAVINGS. 2.5% CRUISE DRAG REDUCTION (REFS. 32 & 45).				SEE REF. 33	SEE REF. 33
	ADVANCED TECHNOLOGY TRANSPORT	SEE REFS. 34-35	SEE REFS. 34-35		SEE REFS. 34-35		
	B-1					70% REDUCTION IN CREW STATION ACCELERATION. SEE REFS. 36 - 38	
	L-1011		20% REDUCTION IN LATERAL GUST DESIGN LOADS (REF. 27)				
FIGHTERS	TSR2	10% WEIGHT DECREASE, 7% PROFILE DRAG RED'N. (REF. 39)					
	F-4	SEE REFS. 40-42			SEE REFS. 40-42		150 KT. FLIGHT ENVELOPE EXPANSION (REF. 43)
	70,000 LB. FIGHTER (BOEING MODEL 818)	14% G.WT. RED'N POSSIBLE (REF. 6)			7.5% G.WT. RED'N POSSIBLE. (REF. 6)		
	YF-16	APPROX. 10% NEGATIVE STABILITY MARGIN (REF. 44)					

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ACTIVE CONTROL TRANSPORT DESIGN CRITERIA

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and

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INTRODUCTION

The question of design criteria for active control transports is one of the key issues involved in the design. The reason for this is that if one is to realize benefits in the form of increased range, decreased weight, etc., he must be able to apply design criteria which take into consideration the design improvements afforded by active controls. The work presented in this paper draws heavily from the report of an industry panel sponsored by NASA in 1972-73 to study vehicle design considerations for active control applications to subsonic transports. This work is soon to be published in a NASA document, reference 1. Additional background material has been drawn from references 2 through 16, which are not cited individually. In this paper today we will define what is meant by active control and then define those functions which were considered by this panel and should be considered in any detailed study of design criteria. We will also touch briefly on the FAA regulations governing transport aircraft design.

ACTIVE CONTROL TECHNOLOGY

The question of just what kind of an airplane configuration satisfies the definition of an active control aircraft is difficult. Several designations for this type of aircraft have been used (fly by wire, CCV, etc.) but an aircraft utilizing active controls can, in general, be identified as one in which significant inputs (over and above those of the pilot) are transmitted to the control surfaces for the purpose of augmenting vehicle performance. These inputs, derived from various sensors and properly processed, can be utilized to provide reduced trim drag and tail area through stability augmentation, reduce structural fatigue, alleviate maneuvering loads, suppress flutter, and improve ride comfort. If applied in a meaningful manner early in the vehicle design, ACT can have a significant impact on vehicle weight and geometry, thus leading to the designation of a "control configured vehicle" (CCV).

The term "fly by wire" describes a method of system implementation whereby electrical commands are used. This approach is suited to the application of active controls in that it provides an ideal interface between the basic

command system and the sensor and signal processing elements.

One frequently reads in the literature items which would lead one to believe that the active control transport will be a sudden and rather drastic innovation from the long line of transport development over the last 40 years. As a matter of fact, it is not a sudden transition, but a continuing growth in the technology of transport aircraft design. Every modern day aircraft, to some extent, incorporates some of those functions which we rather loosely tie together under the name of active control technology. It became apparent in the early twin engine transports that the pilot had difficulty exerting sufficient stick force to move the control surfaces of the aircraft. The designers rather ingeniously provided the pilot with aerodynamic tabs in order to reduce his workload and make the aircraft easier to control. As aircraft continued to grow, hydraulic-powered control systems were implemented. Although these early systems were designed in a manner which still provided the pilot with a mechanical linkage to the surface in the event of hydraulic failure, the modern day transports (the DC-10, L1011, the 747) now completely depend on the hydraulic system, and the designer (and the pilot) must rely on the reliability of the redundant systems which supply the power for the control surfaces.

Along with this reliance on hydraulic systems, the pilot has also experienced an increase in cockpit workload from the many other systems which must function properly for the economical and safe operation of the large transport aircraft. In return, flying qualities and comfort have improved, reducing pilot effort and fatigue. The pilots are slowly learning to accept the fact that certain critical conditions must be automatically detected and appropriate remedial action taken without pilot activity. In this context then, the incorporation of further active controls on the transport aircraft is not a sudden transition but a steady progression toward a more modern and efficient transport design.

Design criteria and FAA safety regulations have generally responded to design innovations such as active control rather than leading these technical advances. It is important at this time, with active controls of various kinds becoming more and more common, that design criteria and Federal safety regulations lead the effort rather than follow these new designs. The panel concluded that most of the immediately available active control techniques have been well explored theoretically and, in fact, have been and are being demonstrated each day on a wide variety of experimental and military aircraft. This demonstration program is illustrated in Table 1.

The important conclusion to be drawn from this table is that when discussing active control technology, one is dealing with a technology which in some cases is well advanced, including operational experience on transport aircraft. Certainly if one compares this, say for instance, to the introduction of jet engines on aircraft, one would be forced to the conclusion that the relative state of readiness of active controls approaches that of jet engines at the time they were introduced into commercial aircraft. It is also important to note, however, the disparity between the status of various functions. For instance, the yaw damper is well received and in fact may be

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mandatory for safe handling qualities, and has many thousands of transport flight hours behind it. On the other hand, flutter control is by comparison only in its infancy. This leads to the conclusion that we must approach active control technology not as an all-inclusive blanket addition to an aircraft, but in a step by step procedure with each new subsystem being carefully verified on the basis of cost effectiveness, need, and reliability.

The above table does not consider the experience gained in the many missiles and spacecraft, both manned and unmanned, which have flown with complete automatic control and hands-off operation. Every Apollo mission from launch to splashdown is a demonstration of active control technology. The rapidly increasing technology of remotely piloted vehicles is also quickly adding to the storehouse of knowledge on how to take off, land, and navigate in a hands-off, completely automatic mode. Indeed, one must consider that more than 25 years ago the first hands-off flight of an aircraft was demonstrated from takeoff to landing.

ACTIVE CONTROL FUNCTIONS

Relaxed Inherent Stability

Relaxed inherent stability is conventionally defined as a reduction in the stability of the short-period attitude modes of rigid-body aircraft motion. That is, reductions in inherent stability result from the reduction of aerodynamic restoring moment with respect to angle of attack or angle of sideslip or a reduction of aerodynamic damping for the unaugmented (basic) aircraft. In principle, relaxed inherent stability can also refer to reduction in stability for other modes of aircraft motion.

This is a very important departure because the basic stability parameters in both the pitch and yaw axes have established the criteria for a considerable portion of the aircraft design. It is, however, one of the prime areas for the application of active control technology. Desirability of relaxed inherent stability arises from the possibility that with smaller tail volumes significant reductions in total aircraft drag and gross weight can be realized with invariant payload and mission. This is substantiated by the results of industry ATT and AST studies which show that relaxed inherent stability combined with center of gravity control offers the largest payoff for the aircraft in terms of gross weight reduction.

Pitch Stability

Relaxed longitudinal stability is one of the largest areas of potential benefit to be derived from the application of active control technology. We will not, in this paper, go into the details of how one implements active controls for the relaxed stability condition, but we will discuss some of the design criteria involved. First, the basic considerations influencing wing location and horizontal tail surface size and location are affected. The horizontal tail area, for instance, is normally set for a conventional design

to meet stability and control requirements over the desired center of gravity range. Typically, the forward center of gravity limit tail area requirements have been set by trim capability or by control required to develop maximum lift in the landing configuration. The critical condition depends on the type of control system selected, i.e., separate trim and control surfaces or a single surface providing both control and trim. Aft C.G. limit requirements have generally been set by minimum levels of static longitudinal stability. For the active control relaxed stability design, the horizontal tail area may be set by either the landing case or by the pitching moment required for take-off rotation at forward C.G. and by the reduced level of stability or by the pitching acceleration required for control in the presence of gusts and other external disturbances at aft C.G. These points are illustrated in Figure 1. The active controlled aircraft is rebalanced with a farther aft center of gravity range and a smaller horizontal tail.

The deficiencies in inherent stability might be compensated for by augmenting C_{M_α} and C_{M_q}. The degree of instability allowable will be determined not only by increasing stabilization control power requirements but also by the variation of trim drag. As the balancing tail load changes from a down load to an up load, the longitudinal component of the tail lift vector changes from a thrust to a drag, significantly increasing tail drag. Minimum trim drag usually occurs near zero static margin, as illustrated in Figure 2. The exact center of gravity location for minimum trim drag is dependent on the particular configuration and even on the wing aerodynamic design.

As shown in table 1, some experience has been gained with relaxed inherent stability. Many jet transports have augmented static longitudinal stability where the augmentation is a function of airspeed. However, the magnitude of relaxation possible with active control will change the design criteria. Perhaps one of the most disturbing ideas that accompanies this changing criteria is that we have now replaced the easily calculated inherent stability requirement with a possible pitching acceleration requirement based upon the rather uncertain magnitude of airplane response required under varying conditions of flight and levels of atmospheric disturbance.

Flying qualities criteria may also be affected by dependence on augmentation, especially in the pitch axis. These will be discussed later.

Directional Stability

As shown in table 1, this is the area where active control has seen the largest and most widespread application in transport aircraft. We have seen the yaw damper (an augmented directional stability and control system would more completely describe the systems currently flying on large transport aircraft) progress from a system which was a nice passenger comfort add-on feature to a system which must be operating in order for the aircraft to be cleared for flight. Despite this, there is probably much less to be gained by relaxed directional stability than by relaxed longitudinal stability. Currently, vertical tails are sized to provide static directional stability, dynamic lateral-directional stability, and asymmetric thrust control. Minimum control speed criteria are either critical or close to it in sizing the vertical tail

on most transport designs with wing-mounted engines. Selection of the minimum control speed criteria may be somewhat arbitrary, but two things are generally considered:

1) The air minimum control speed must be less than the landing approach speed at all gross weights.

2) Ground and air minimum control speeds may dictate the minimum takeoff runway length and should be set to provide the desired capability.

With relaxed inherent stability and if asymmetric thrust control is not limiting, the tail size may be reduced to the level where stabilization control or airplane control response, as during a crosswind landing decrab maneuver, become limiting. In either case, new and unfamiliar design criteria are required.

Control of Aircraft Center of Gravity and Inertia

This area of active control has also been growing rather rapidly. At least one transport aircraft requires a sequence of wing fuel management in order to maintain the necessary margins against flutter. Maintenance of the C.G. within limits on current transports also dictates certain management sequences. It is therefore not a very great step to add to these procedures some requirements for maintaining an optimum C.G. location and/or inertia distribution for the actively controlled transport. It is this distribution of inertia for the entire aircraft as well as the equivalent C.G. location which acts with the control surface active control system to provide the optimum gains with relaxed inherent stability.

Automatic center-of-gravity control can offer significant design advantages in the following ways.

o Reduction of the design center-of-gravity range at given flight conditions may allow further reduction in the horizontal tail volume coefficient (refer to the indication of "CG range" on figure 1)

o Minimization of total drag with respect to center-of-gravity location during cruising flight, as illustrated in figure 2.

Ride Quality

Ride quality control refers to automatic control system functions which reduce to acceptable levels the accelerations to which passengers and crew are subjected. Factors such as low wing loading, poorly damped dynamic stability, structural flexibility, atmospheric turbulence, and high speed, low altitude flight all contribute to poor ride comfort.

Ride quality problems have tended to be secondary considerations with respect to resolution of structural load and flexibility problems. In fact, it was stated by two members of the panel that ride quality is not a major trade factor in design, because the criteria for ride quality in the commercial environment are:

o Ride must be merely acceptable to passengers

o Ride must be competitive with contemporary commercial aircraft

In addition, the aircraft must be readily controllable in turbulence.

The control techniques for improving ride quality are fairly well established both theoretically and operationally. Many commercial transports have some degree of ride quality control provided by means of conventional control surfaces. The yaw damper systems of modern jet transports improve ride quality even though their fundamental purpose is to improve handling qualities.

Active control for gust load alleviation has demonstrated greatly reduced response to turbulence, thus assuring a greater comfort for passengers. A typical reduction in aircraft response to turbulence obtained during the B52 LAMS and CCV programs is shown in Figure 3. It will be noted that the decrease in response to turbulence is sensitive to the aircraft structural modes and that a uniform reduction at all frequencies is impossible. This led to a good deal of discussion among the panel members as to the criteria for ride quality. While certain maximum limits for ride comfort are relatively easy to establish, the panel decided that detail criteria for ride comfort still need a considerable amount of research in order to establish workable design criteria. In either case it is doubtful that ride quality design criteria will result in weight savings, so the competitive pressure to supply a smoother ride will probably dictate the control system design criteria.

Load Control

Load control refers to the use of passive or automatic control functions for the purpose of regulating the net load and distribution of load applied to the aircraft structure.

There are four main facets of load control. To some extent, all must be considered simultaneously to achieve a well-balanced design although some may receive considerably more emphasis than others. Three facets of load control which are specifically discussed in this subsection are maneuver load control, gust load control, and fatigue damage control. Flutter control might also be included as a fourth facet of load control because flutter is the result of a particular kind of loading. Flutter, however, tends to be disassociated from other types of loading for reasons which will be explained in the flutter control subsection which follows.

The question of load control was perhaps as controversial as the question of relaxed inherent stability, and several important points were raised regarding each type of load control.

Maneuver Loading

Maneuver loading is that portion of forces acting on the airframe which result from maneuvers required to maintain the aircraft on the intended flight path. The distribution of this loading over the airframe can have a powerful effect upon the shear forces and bending moments which must be transmitted at

given points in the structure. The ability to tailor the distribution of maneuver loading over the airframe is maneuver load control. Maneuver load control can have a significant impact upon structural implementation and even upon configuration.

The impact of tailoring maneuver load distribution may be far-reaching. If the maximum reduction in fatigue loading is to be achieved, maneuver load control would be desirable during all maneuvering. When applied to the wing, this usually implies an "unloading" of the outer wing, thus reducing the root bending moment, as illustrated in Figure 4a. A high wing loading transport may possibly be limited in cruise altitude by maneuver requirements such as those specified in the British Civil Airworthiness Requirements. Unloading a portion of the wing would tend to reduce maneuver capability, particularly if wing stalling occurs inboard. Thus, maneuver load control might tend to limit wing loading or dictate a new approach to wing aerodynamic design. This situation may be avoided by utilizing maneuvering flaps to increase lift on the inboard portion of the wing, Figure 4b. Additional aerodynamic and structural design considerations would still be required, along with new modes of control akin to direct lift control.

Gust Loading

Gust loading is that portion of forces acting on the airframe which result from atmospheric disturbances.

Gust-load control is accomplished by the following means:
o Controlling the aircraft in such a way as to produce a net incremental load factor which tends to cancel the net gust-induced load factor. Because of aircraft inertia, this is best accomplished with direct lift control devices.
o Controlling the distribution of the incremental load which tends to cancel the gust-induced load in such a way that their distributions are similar.
o Augmenting damping for modes excited by gusts.

The extent to which gust-load control is effective in performing all three listed functions can have a significant impact upon the structural strength and fatigue requirements.

Experience cited for the panel indicated that the impact of maneuver and gust-load control on reduction of structural requirements tends to be significant only when both maneuver and gust-load control are practiced simultaneously. If only one of these load-control objectives is addressed, then the other source of loading becomes critical before any significant reduction in structural requirements is realized.

Fatigue

Cyclical loading is produced by forces applied to the airframe which result in stress-level oscillations in the structure. Fatigue damage results

from accumulated stress cycles at given stress levels and at critical points in the airframe. Fatigue damage control is a technique for reducing the fatigue damage rate by using active controls to reduce the number of transient cycles at the higher stress levels to which the structure is subjected during operation.

The frequency range of damaging loads extends from once per 100 flights (e.g., from very "firm" landings) to the once per flight of the so-called ground-air-ground (GAG) cycle and to the characteristic frequency of the response to turbulence. The transition between the ground mean loading and the airborne mean loading of the GAG cycle accounts for as much as 80% of fatigue damage on the lower wing skin on some contemporary transport aircraft. Most of the remaining damage accrues from incremental loads in the 1/4- to 1/2-g range.

Since the mean-to-mean fluctuation of the GAG cycle is not amenable to control, active control offers potential reduction of longitudinal loads only for the incremental load fluctuation about the mean level of the GAG cycle. Large potential for load reduction exists for lateral loads because there is no GAG cycle effect.

Much of the panel discussion centered around the application of the classical, rather arbitrary approach of a discrete gust versus the more modern approach of "rational probability analysis" coupled with careful mission analysis. The majority of the panel agreed that we must go even further in developing statistical methods and performing mission analyses in order to realize the benefits to be gained from the application of active controls to load alleviation. The obvious point here is that if careful mission analysis is applied to the calculation of the fatigue life of the aircraft and if the load alleviation control systems are assumed active during the entire life of the aircraft, the weight of the aircraft structure could be reduced for the same fatigue life. Studies confirming this are still in progress and it is difficult at this time to come up with definite criteria. However, the panel agreed that the combination of maneuver load control plus gust load alleviation can result in reductions of load fluctuation.

Other Load Limiting

Other forms of load limiting are also useful. Surface actuator capability not only limits the airplane maneuver envelope but tends to limit the maximum load on the surface itself. Many examples of load limiting are in use today on jet transports. Flap blowback or deflection limiting is in use on several aircraft to limit structural loads. Rudder deflection limiting as a function of flap angle and airspeed is also commonly employed. As other active control modes are used to reduce structural weight and margins, the use of these approaches will have to be considered in concert with the other control modes in a synergistic design procedure.

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Envelope Limiting

Envelope limiting refers to those functions in an active control system that prevent or discourage operation of the aircraft outside its design or operating envelope.

Every transport aircraft currently has some form of envelope limit warning and envelope limiting, although not usually in the ACT sense. Envelope limit warning takes the form of stick shaker systems which warn of an approach to the stall and overspeed warning systems which warn that maximum operating speeds have been exceeded. Envelope limiting is provided by pilot strength limitations, control surface actuator capability, stick pushers, autopilot authority, and autopilot automatic cutoffs (ACO), for example. The limits provided by pilot or actuator strength may or may not be within the structural design envelope of the aircraft. For instance, the pilot does, in some flight regimes, have the capability of exceeding the design limit loads about all axes.

The concept of envelope limiting is now being applied to fighter aircraft to allow use of the full maneuver envelope without danger of a stall-spin departure. For transport aircraft, the incorporation of active control could supplement the present warning and limiting features with an automatic function which prevents the aircraft from entering into a forbidden flight regime. Angle of attack and sideslip limiting could avoid post-stall loads and flight characteristics problems, and reduce vertical tail loads. Overspeed limiting could reduce the required margin between maximum operating and design dive speeds, as shown in Figure 5, reducing design loads and allowing a lighter structure. The possibility of atmospheric-caused upset must be considered in establishment of minimum margins. It would then be necessary to assure that the flight control system will satisfactorily handle this job even in the back-up or degraded operational modes to assure that the aircraft is operated within the criteria established for strength of the structure. The panel felt, however, that G-limiting might not be desirable, as there have been several cases where the ability of an aircraft to exceed the design limit load factor may have avoided a catastrophic accident following upsets at low altitudes.

Flutter Control

Flutter control refers to the use of automatic control functions which alter the apparent structural mass or stiffness, or aerodynamic damping. It was the unanimous opinion of the panel that active flutter control must be considered as part of ACT even if it may not find commercial application in the near future. At present, the nature of the control law for achieving the required augmentation seems extremely sensitive to the unsteady aerodynamic forces and is also sensitive to the mass and stiffness distributions of the airframe. It should also be stressed that the flutter certification of the aircraft and the flutter safety margins will be influenced by the presence of other active control functions. For instance, in the case of relaxed inherent stability, it is necessary to have a relatively wide bandwidth control system to cope with the unstable short period mode roots. This control system will tightly couple with the basic flutter modes of the wing-nacelle-fuselage combinations on a large transport aircraft. This will mean that the safety

margin criteria for flutter will be a function of the control system loop gains and general design. Criteria will also have to be carefully developed to account for backup modes of operation of the flight control system.

DESIGN CONSIDERATIONS AND REGULATIONS

Key elements in bringing ACT to the point of commercial application are:
o Availability of proven design criteria
o Limitations on ACT applications that may be imposed by
   regulations
o Availability of proven design practices to guide the combined
   application of ACT functions.
We are concerned mainly with the first two items in this paper.

Design criteria are derived from many sources. Perhaps the most important are the manufacturer's experience and design philosophy. Studies performed or financed by NASA and DOD provide a large fund of suggested criteria and data which the designer uses in selecting his criteria for application.

For military aircraft, mandatory military specifications are usually applied to obtain what are considered to be good characteristics. In the civil or commercial world competition usually ensures that the aircraft have the best characteristics obtainable, within reason. Safety is therefore the primary purpose of the airworthiness requirements contained in Part 25 of the Federal Aviation Regulations. These requirements must always be kept in mind, as they are the standard by which airworthiness of the aircraft will be judged. Besides the U.S. FAA regulations, the designer must also consider the requirements that may be imposed by other nations on aircraft offered for sale within their territory. Among nations having specific airworthiness requirements are the United Kingdom, France, the Netherlands, Germany, Italy, and Australia.

Existing Federal Airworthiness Regulations (FARs) in Part 25 do not place many significant constraints on the application of ACT. Those constraints which are imposed tend to be of the following kinds:
o Interpretations of the fundamental regulation intent were not made
   in a context which included ACT.
o Practical considerations for demonstrating compliance sometimes
   require arbitrary maneuvers, tests, or environments which have
   no counterparts in normal or degraded modes of operation.
o The view of acceptable safe practice tends to be consistent with
   the current or recent past state of the art but not to the projected
   state of the art.

Existing regulations [FAR 25.21(e)] already recognize that acceptable flight characteristics may depend upon a stability augmentation system or upon other automatic or power-operated systems. This clearly admits ACT systems as well. Revisions to the regulations found necessary for ACT will probably initially take the form of special conditions for certification.

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In the following paragraphs we will discuss some of the important design criteria and regulatory problems affecting the implementation of ACT.

Reliability - Safety

The immediate reaction of most designers when faced with consideration of ACT is to raise the question of reliability and safety—"that thing isn't replacing structure in my airplane until it has demonstrated the same reliability as primary structure".

It is apparent that safety must not be compromised, and that the criteria for catastrophic failure will be basically unchanged. The required level of overall function reliability is achieved in control and vital power systems by increasing redundancy for those functions that do not have the desired reliability. For example, controllability of the wide-body jet transports is dependent on integrity of the hydraulically powered controls. Reliability for safety of flight is provided by multiple hydraulic systems. After some number of failures, it is, of course, advisable to terminate the flight at the nearest suitable airport in order to minimize exposure time in a non-redundant configuration.

One difference, however, is that failures of presently utilized active control functions do not usually result in reductions in structural capability under normal flight conditions, whereas proposed ACT functions will, in effect, replace primary structure. This does not necessarily mean that these functions must be as reliable as the basic structure, however. The strength requirements will be met already considering at least one failure, so that no reduction in necessary capability should occur for the first failure. An assessment of situation severity and a list of means available for reducing risks presented by failures in ACT functions is given in Table 2. There are three principal means of controlling the risk:
   o Control system redundancy
   o Actuation and/or surface authority distribution
   o Reduced operating envelope
The ultimate levels of reliability will be required only for those functions upon which safe termination of the flight depends.

Autoland systems are presently achieving the required reliability, but for only a short exposure period during each flight. Figure 6 shows the required MTBF as a function of the number of systems required to achieve a probability of complete failure of not more than 1×10^{-9} during a three hour flight.

The problems with reliability are likely to occur within the sensing, computing, and display functions which are today largely restricted to flight guidance and control systems (FGCS). Typical MTBF values for these systems are in the order of 300 to 800 hours. Although individual system reliability improvement is still required, Figure 6 shows that the overall reliability goal may be satisfied with a reasonable number of redundant systems. Characteristic systems for this application will include multiple channel command paths in which failures will be annunciated, thus providing the pilot with system

degradation information enabling him to take corrective action prior to total system failure. Ultimately however, improved reliability goals and techniques must be derived and imposed, but must always include a sensible system failure mode and annunciation capability.

An associated problem is the FAA requirement for determining that safety-related systems are functioning prior to dispatch. Difficulties in determining sensor status have prevented taking credit for automatic cut-offs (ACO) in limiting the consequences of autopilot hardover failures, in some cases. This will require design of systems which can be satisfactorily checked on the ground.

Reliability is presently established in a manner whereby elements of the system can be specifically identified in a reliability block diagram and the reliability of each element is available. The reliability of the avionics elements contributing to the flight safety of a control configured vehicle will be significantly more complex. Not only are there many more elements, but the software is an additional facet which must be evaluated. Accomplishing the failure and probability analyses of these complex systems is a major task in itself, and is not within the present state of the art for those ACT functions not yet fully developed. In some cases, failure analyses have been required to prove that certain types of failures were impossible, which in itself may be a nearly impossible task.

Reliability - Economics

The economics we refer to here is that of dispatch reliability, not maintenance costs, although the latter are certainly important.

A typical design goal for dispatch reliability is that, mechanically, the aircraft shall be capable of departure within 15 minutes of the scheduled time 99 percent of the time. This goal is very stringent and is currently being achieved consistently by only one transport aircraft, the DC-9. The design of this aircraft emphasized simplicity and reliability, whereas the design of later aircraft has emphasized performance, with a resulting increased complexity.

This dispatch goal produces a desire to have your cake and eat it, too. The benefits of more complex systems are desired but it is also desirable to allow dispatch with as many things as possible inoperative or missing. It is common to find flight manuals and minimum equipment lists filled with information for covers, doors, and fairings missing or for hydraulic pumps, yaw dampers, Mach trim systems, autopilots, antiskid, and thrust reversers inoperative. In many cases, the benefits to be obtained from, and therefore dependency on, some systems are limited by the criteria for inoperative dispatch.

The goal of 1% delay rate is typically allocated among the various aircraft systems as shown in Figure 7. The pilot controls and FGCS are allotted 0.005% and 0.10%, respectively. The small size of these percentages does allow some increase without having a major impact on delay rate, but the accompanying impact on maintenance and spares availability may be significant.

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Flying Qualities

Design criteria for flight characteristics, or flying qualities of transport aircraft seem to be in good shape, judging by pilot acceptance of the wide-body jet transports. There has been a steady improvement in flying qualities but, at the same time, some increase in the possible number of degraded situations due to increased system complexity and failure modes.

Transport aircraft flying qualities research in the U.S. has received more of the attention it deserves in recent years after previously having to try to adapt fighter-derived criteria.

Since transport aircraft tend to be developed by evolution rather than revolution, their flying qualities and criteria tend to evolve similarly. The FAA regulations concentrate on classical stability characteristics, primarily static, and on steady state control requirements. Control response and aircraft dynamics receive scant mention, although awareness is much higher during actual aircraft evaluation. The need for positive static stability is still debated, but is defended on the grounds of safety, i.e., reduced pilot workload and fatigue plus a tendency to stay put or even recover from a disturbance during periods of inattention.

Automatic and augmented flight control systems have tended to evolve along a line different from that of basic or inherent flight characteristics and control modes. With the advent of fully-augmented active control systems, it is time that the proper modes and parameters be determined.

The primary axis of concern is the pitch axis. In the past, the provision of adequate inherent pitch stability has tended to emphasize long period characteristics: static longitudinal stability, longitudinal maneuvering stability, and speed or flight path stability. When these characteristics are satisfactory, and the configuration is a relatively conventional one, dynamic stability (short period mode) is generally completely satisfactory. The elevator or longitudinal control is, over the long term, an airspeed control and the throttles are primarily a flight path control in straight flight; in a somewhat simplified sense. In actual practice, thrust changes usually produce some trim change also, thus affecting the trimmed airspeed. With the usual nose-up trim change with increased thrust, applying forward throttle will actually result in a slower airspeed but an increased climb angle.

The initial response of the aircraft to rapid control usage is not the same as the final effects on trimmed flight, however. Elevator inputs produce a change in angle of attack, seen by the pilot as an attitude change, which only gradually manifests itself as a change of airspeed. The immediate normal acceleration and the ultimate change in airspeed will cause a change in flight path and, as a result, in altitude unless the throttles are adjusted to maintain the long term path.

Advancing the throttle produces an initial acceleration which is gradually transformed into a change in flight path angle unless restrained by the elevator control. If there is a large effect of thrust on pitching moment, attitude changes will also occur.

Because of these immediate responses, the controls are used in this manner when accurate flight path tracking is required over the short term. In fact, many pilots believe this is the only correct definition of the control modes.

The usual implementation of automatic flight control systems has been based on this short-term control response. Autopilots on propeller-driven and early jet transports typically incorporated attitude and altitude hold modes. Later autopilot designs incorporate vertical speed, airspeed, and Mach hold modes, the latter two more in the line with the long term elevator-as-airspeed-control principle. Later autopilots also include turbulence modes, usually a loose attitude hold with pitch rate damping. This mode evolved from experience when it was determined that attitude control offered the best chance of avoiding upsets when flying in turbulent or stormy weather.

The advent of the autothrottle system, which tries to maintain airspeed with the throttles, dealt a body blow to the elevator-airspeed control proponents. The final blow was administered by the introduction of control wheel steering (CWS), in which the pilot flies the airplane through a rate command, attitude hold mode of control. This system can reduce the pilot's workload because the airplane is essentially always in trim when the controls are released.

What is the effect of these control modes? Since the elevator is inherently a displacement control, mechanizing it as a rate control significantly changes the airplane's characteristics. Conventional maneuvering stability and static stability become meaningless, as the airplane has neutral or no stability in terms of these flight parameters. Singly or in combination, autothrottles and CWS can produce neutral or divergent flight path stability on what would otherwise be a stable aircraft. This is graphically illustrated in Figure 8, which shows airplane response following a pilot-induced upset during landing approach. The basic airplane, Figure 8a, is inherently stable and recovers to the trim attitude and airspeed. With autothrottles engaged, 8b, the attitude and flight path diverge following the upset. Control-wheel-steering, 8c, prevents attitude divergence but also maintains the airplane at the commanded upset attitude as the flight path diverges. To the credit of CWS, it must be said that it is much less susceptible to external disturbances than to pilot-induced upsets.

These CWS systems do not allow compliance with the stability requirements of FAR 25.173 and .175. They have been certificated basically as autopilot control modes under the requirements of FAR 25.1329 and Advisory Circular 25.1329-1A. They are not considered as primary control modes and have therefore not been evaluated against the basic stability requirements. These requirements therefore present a possible problem area in the implementation of active controls, depending on the control modes selected.

Two types of augmentation would be required to match inherent stability characteristics: angle of attack stability and pitch damping. The latter is fairly easily accomplished but the former requires direct measurement or a combination of measurement and computation. Computed angle of attack is within the current state of the art, although accuracy of either computed or measured

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α may be marginal for use at high airspeeds. In any case, considerable work needs to be done to specify the proper flying quality parameters for airworthiness evaluation. Both the industry and the FAA are active in this area and some changes may result from the formal review of the regulations to be held later this year.

The envelope limiting function of ACT may also negate the regulation stalling speed and characteristics requirements. It would seem appropriate in this event to substitute the control limited minimum speed concept.

Structures

The basic impact upon structural design criteria due to the application of active control is in the area of structural loads. In this area it is not only desirable but also feasible to retain a considerable portion of the structural design criteria which have led to the current generation of transport aircraft. For instance the 1-cos gust, as currently applied to aircraft load calculations, is perhaps not conceived on the most rational basis, but it nevertheless serves as a standard, and it is not necessary to modify it just to permit active controls in the design.

Maneuver design criteria, on the other hand, should be reviewed for active controls application. For instance, the basic -1, +2.5g load factor criterion is deeply entrenched in our current transport design philosophy and designs major portions of the structure. Instances are cited where transport aircraft have had to develop this maximum load factor in order to survive an upset. These instances caused the panel to adopt a negative position on g limiting, as mentioned previously. From a design criteria standpoint we need to re-examine the conditions leading to these maximum load factor maneuver requirements and determine whether active controls prevent one from ever getting into this region or perhaps whether active controls can cause even more exaggerated maneuvers. In either case the change in structural weight of the aircraft as a function of this maneuver requirement is considerable.

Another instance where criteria changes are necessary is in the computation of aircraft fatigue life. Here again a considerable portion of structure is designed for fatigue, and as in the maneuver load factor case, the active control system has a considerable influence over aircraft structural response and hence fatigue life. It is not clear that the practical active control system will necessarily reduce the response (and hence the structural weight) of all portions of the aircraft, but it is clear that future criteria must deal directly with the input data required to perform rational probability and mission analysis studies. This conclusion was strongly supported by the panel.

Another instance where new structural design criteria must be developed for the active controlled aircraft is in the area of abrupt maneuver requirements. The loads developed on the structure during the abrupt maneuver will be very dependent on how one chooses to mechanize the control system. For instance the transient loads developed during an abrupt time sequence of elevator deflections may be considerably different from the transient loads

developed during a similar deflection history of a flying tail, although each may produce roughly the same aircraft C.G. acceleration. For the aircraft which depends on a functioning active control system at all times the abrupt maneuver criteria must deal with defining the conditions which cause the abrupt maneuver, rather than defining the control surface time history. The following are examples of these modes

    a) Transient caused by switching from primary to backup systems

    b) Transients caused by control system failure modes such as, "hardover command".

    c) Evasive action for collision avoidance.

Control Systems

The criteria for detail design of conventional control systems are predominantly developed by the manufacturers. These include instructions regarding design to provide safety, ease of maintenance, and to prevent incorrect assembly, for example. The implementation of active controls will necessitate the expansion of these rules to include much more sophisticated applications. In the past (with the possible exception of the yaw damper) transport aircraft have been designed and certified to operate without an operational autopilot. For the actively controlled transport the flight controller becomes a primary design consideration along with structures, aerodynamics, and propulsion systems. It should be noted, however, that a start in this direction is being made with the design of the YC-14 and YC-15 advanced medium STOL transport prototypes.

One area which received considerable attention from the panel is that of establishing a math model of the airframe and deriving design criteria for establishing parameter perturbation analyses on the model. This is an area that has received considerable attention in missile and launch vehicle control system design. Unsteady aerodynamics and structural dynamic parameters were singled out by the panel as being the principal problem areas. It was felt that the accuracy of existing prediction methods was inadequate for optimum ACT system design. This problem is being approached by improving the methods and by exploration of insensitive flight control systems. A related problem is the variation in structural dynamic and aerodynamic parameters due to changes or differences in fuel and payload distribution that may occur during one flight as well as between flights, along with the variation of airspeed, altitude, and Mach number encountered. Again, the insensitive approach may prove to be the best way to handle this variation in parameters.

The active control system will also be much more demanding on control system components which are subject to wear. Because of the higher gains required by the active control system, control system components will have to meet tighter specifications, and remain within these specifications throughout the useful life of the control system. This requires new design criteria for components such as hydraulic valves and actuators whose phase and gain characteristics are affected by wear. It will also require tighter tolerances on control surface hinges in order to prevent low amplitude, fatigue causing, limit cycle oscillations. At the same time, the automatic controllers must handle out-of-tolerance conditions. These conditions can occur due to

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manufacturing tolerances, aging, wear, material failures, off-nominal power supplies, and dynamic characteristics caused by changes in environmental conditions.

As flight control systems become more complex, built-in test equipment (BITE) takes on greater importance as a means for improving safety, operational reliability, and maintenance costs. The design requirements for built-in test equipment must include not only static end to end checks of the control system but dynamic checks as well. The BITE requirements should include the capability for these status and performance checks by continuous on-line tests, inflight pre-engage operational status tests, channel comparison monitoring, and ground maintenance tests. The inflight tests must be capable of detecting failures to the functional system level. The ground checks must isolate failures to the line replaceable unit (LRU) level. The complexity of the systems as compared with the level of capability of average maintenance personnel will require very stringent design requirements to preclude faulty maintenance and provide ease of fault isolation and correction. It is important to note that the background of missile control system experience will do little to help us formulate design criteria associated with many hours of continuous operation.

As one of the special conditions in the transport certification procedure, it is specified that the airplane will operate safely for at least 5 minutes with the primary electrical system inoperative. The current means of complying with this requirement should not be seriously impacted by the incorporation of additional ACT functions. For instance, several aircraft have air-driven electrical generators for emergency use, and the addition of more ACT functions will only add to the electrical load.

FAR 25.671 requires that the aircraft be controllable if all engines fail. Here again the current means for supplying electrical and hydraulic power, in the event of all engines having failed, should be sufficient to satisfy the needs of additional ACT functions.

CONCLUSIONS

It is clear from the information outlined in this paper and from the work of the NASA Panel, that a great deal of work remains to be done in the area of detail design criteria and design practice. It is also apparent that the overall improvement that one can achieve by going to active controls is, with but a few exceptions, not being held back by current regulations and basic design criteria.

The area where the most work needs to be done is in the detail design criteria of the control system itself. The problems center around the derivation of reasonable design criteria for the design of advanced flight controllers. Other problems are the achievement of the reliability goals and production of hardware which can be maintained and manufactured at costs comparable to the rest of the aircraft critical components.

As this work progresses, more ACT functions will be proven to be both reliable and practical, and will be incorporated into the advanced transport designs.

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TABLE 1: ACT FUNCTION APPLICATION EXPERIENCE

ACT FUNCTION	AIRCRAFT	STATE OF READINESS	PAYOFF & TRADE DATA	SYSTEM MECHANIZED	FLIGHT TESTED	OPERATIONAL EXPERIENCE
Relaxed Inherent Stability Augmentation	Military	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
	Experimental	⇒===⟹ (through FLIGHT TESTED)				
Center of Gravity Control	Military	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
	Experimental	⇒===⟹ (through FLIGHT TESTED)				
Ride Quality Control	Military	⇒=================================⟹ (through FLIGHT TESTED)				
Yaw Damper	Military	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
	Commercial Transport	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
Maneuver Load Control	Military	⇒=================================⟹ (through FLIGHT TESTED)				
Gust Load Control	Military	⇒=========================⟹ (through FLIGHT TESTED)				
	Commercial Transport	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
Fatigue Damage Control	Military	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
Flutter Control	Military	⇒=================================⟹ (through FLIGHT TESTED)				
Envelope Limiting	Military	⇒==⟹ (through OPERATIONAL EXPERIENCE)				
	Commercial Transport	⇒==⟹ (through OPERATIONAL EXPERIENCE)				

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TABLE 2:   DEGRADED SITUATION SEVERITY AND MEANS AVAILABLE

FOR MODIFYING RISKS PRESENTED BY FAILURES

FUNCTION		SEVERITY OF SITUATION WITH FUNCTION DEGRADATION	MEANS AVAILABLE FOR MODIFYING RISKS PRESENTED BY FAILURES
Relaxed Inherent Stability Augmentation		Moderate-Very	Redundancy + Authority distribution Reduced operating envelope CG management
Load Control	Maneuver	Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
	Gust	Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
	Fatigue Damage	Negligible	Reduced operating envelope
Flutter Control		Very-Extreme	Redundancy + Authority distribution Reduced operating envelope
Ride Quality Control		Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
Envelope Limiting		Negligible-Moderate	Redundancy Reduced operating envelope
CG Control		Negligible	Reduced operating envelope

FIGURE 1. HORIZONTAL TAIL AREA REQUIREMENTS

FIGURE 2. TRIM DRAG

POWER SPECTRAL DENSITY, $\dfrac{(\text{ACCELERATION})^2}{\text{FREQUENCY}}$

(B-52 FLIGHT TEST - REFERENCE 8)

AUGMENTATION OFF

AUGMENTATION ON

2.2 FPS GUST

1×10^{-3}

1×10^{-4}

1×10^{-5}

1.5   2.0   2.5   3.0   3.5   4.0   4.5   5.0

LATERAL FREQUENCY - Hz
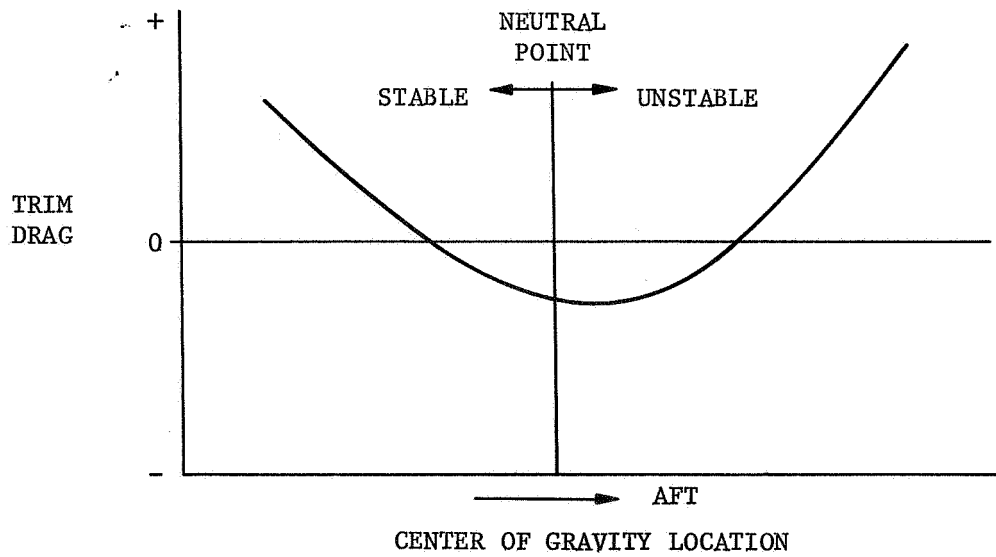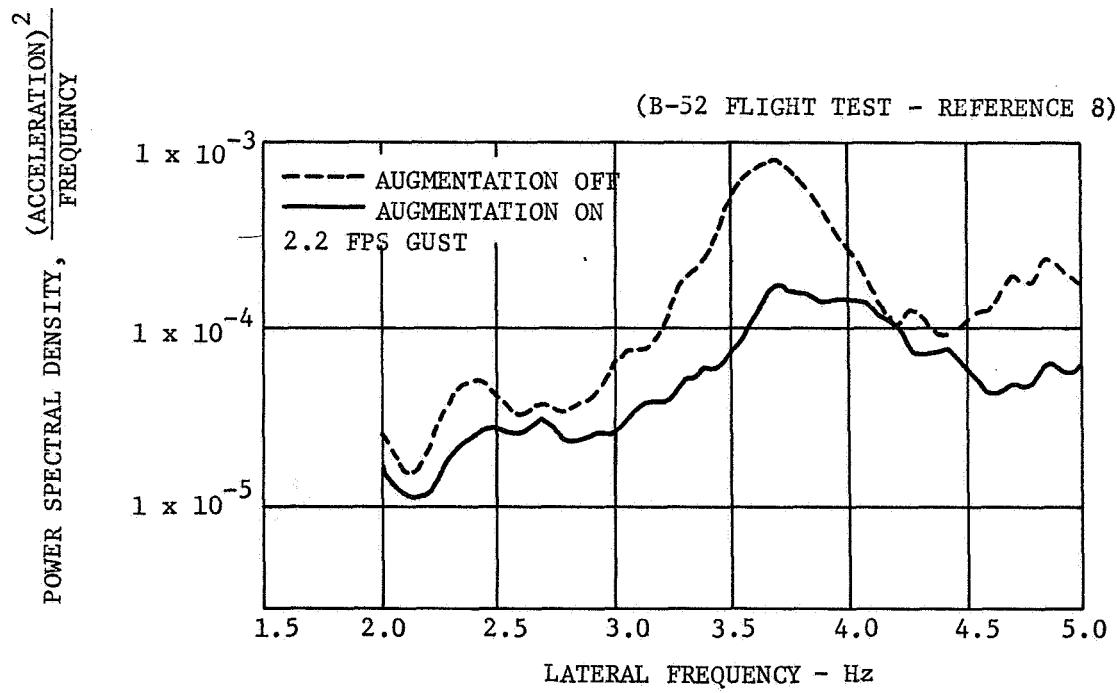
FIGURE 3. RIDE QUALITY IMPROVEMENT
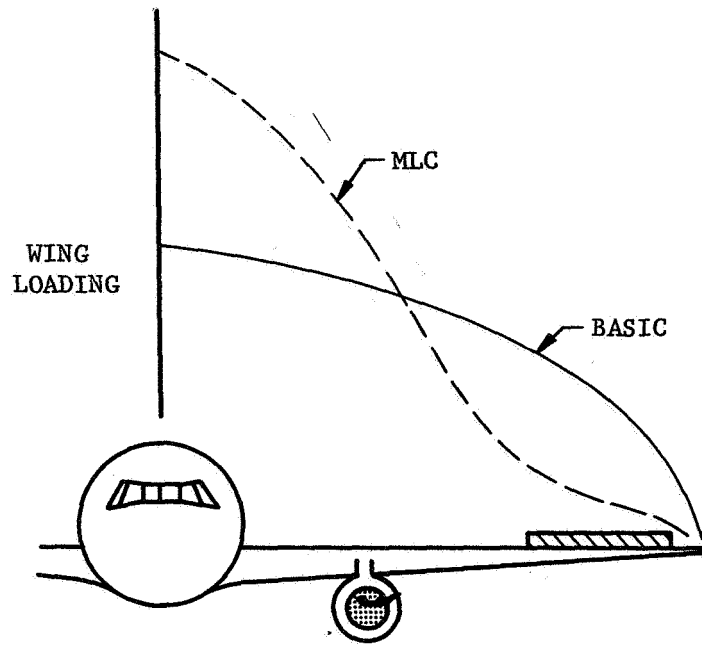
4a. UNLOADING OUTER WING
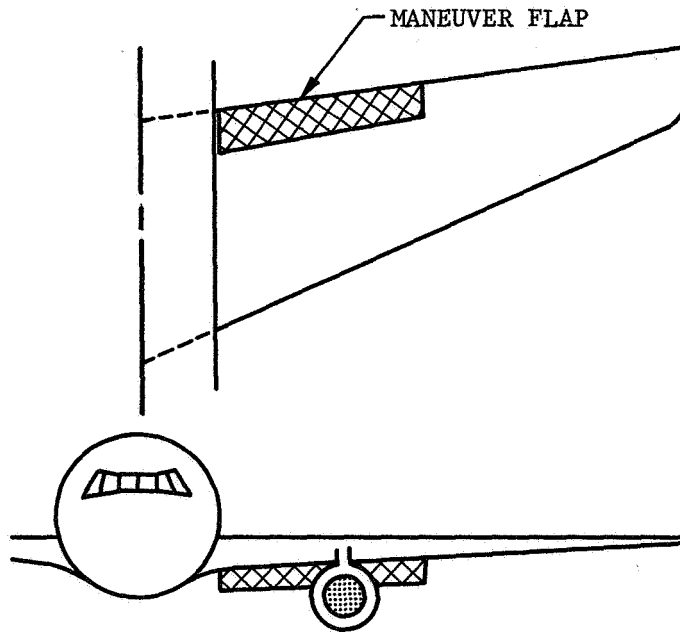
4b. LOADING INNER WING

FIGURE 4. MANEUVER LOAD CONTROL

700
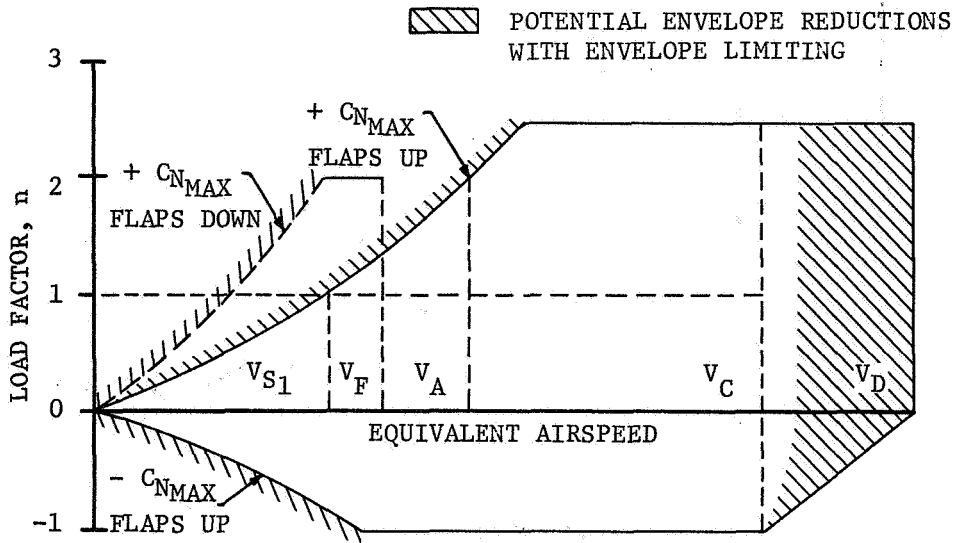
FIGURE 5. DESIGN MANEUVERING ENVELOPE

ASSUMPTIONS
FLT TIME = 3 HOURS
PERFECT DETECTION
PERFECT SWITCHING
ACTIVE STANDBY
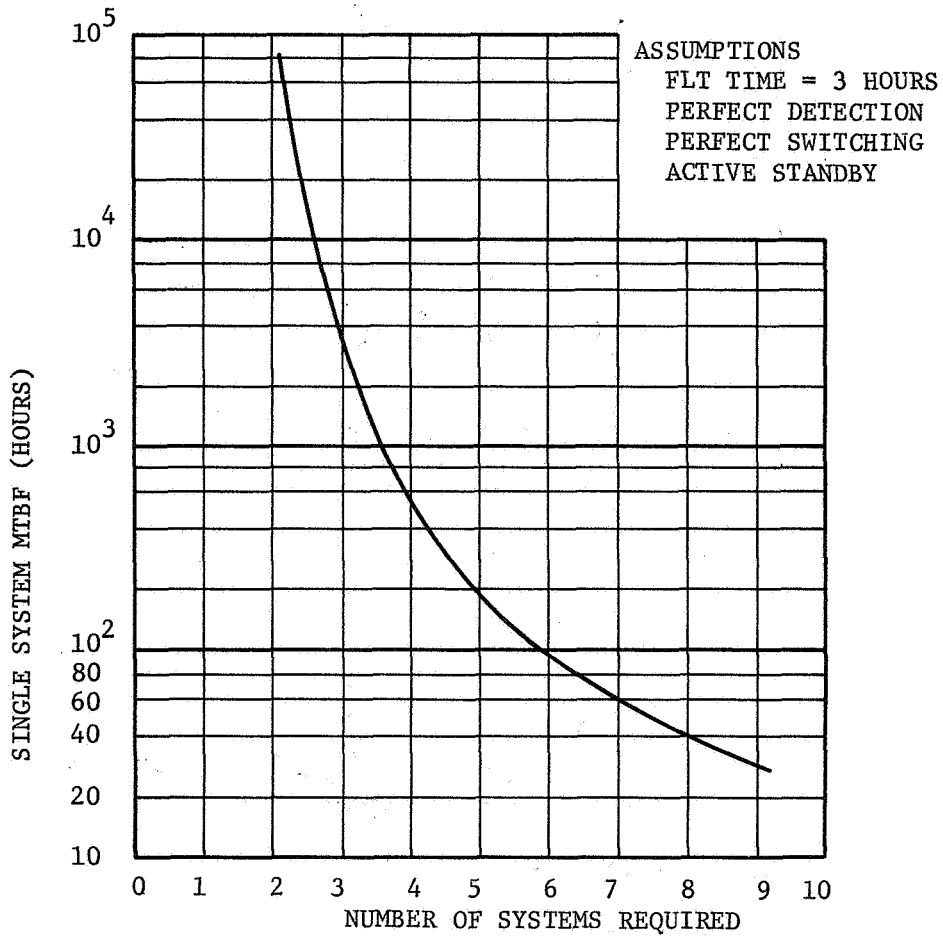
FIGURE 6. SYSTEMS REQUIRED TO PROVIDE PROBABILITY OF 10^{-9} FOR COMPLETE SYSTEM FAILURE

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DISPATCH

99% ON TIME

1% DELAY

DELAY DESCRIPTION
TOTAL 1%

INTERIORS 0.04%

PILOT CONTROLS 0.005%
CARGO SYSTEMS 0.020%

STRUCTURE 0.150%

HYDRO-MECH 0.125%

POWER PLANT 0.340%

ELECTRICAL 0.100%

ENVIRON. 0.100%

FLT GUID 0.100%

COMMUNICATIONS & RATIO 0.020%
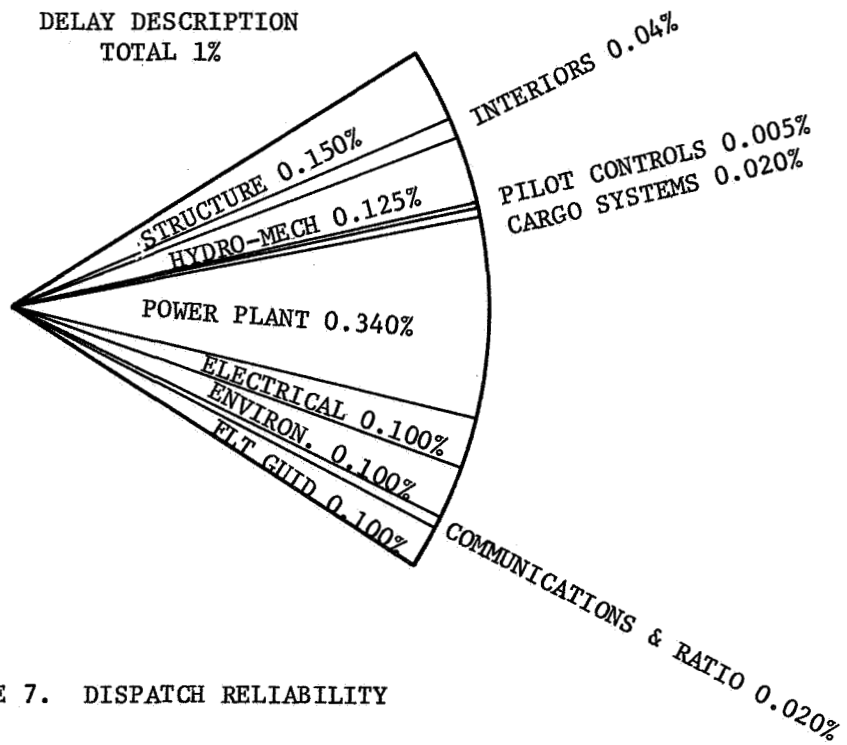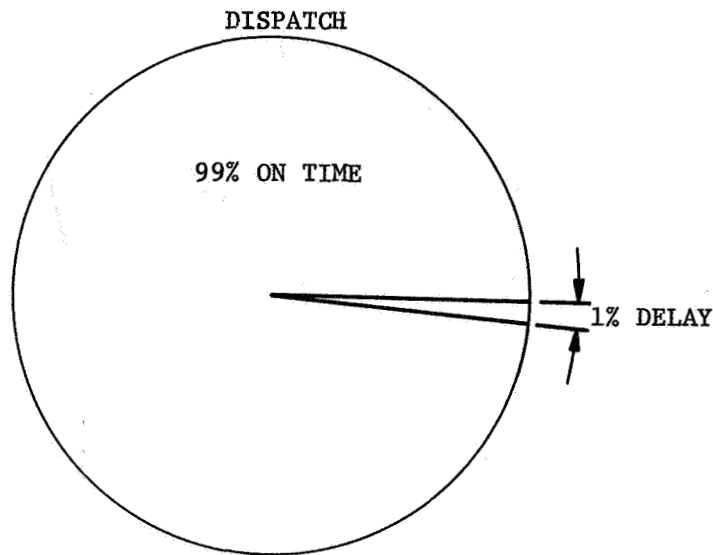
FIGURE 7.   DISPATCH RELIABILITY

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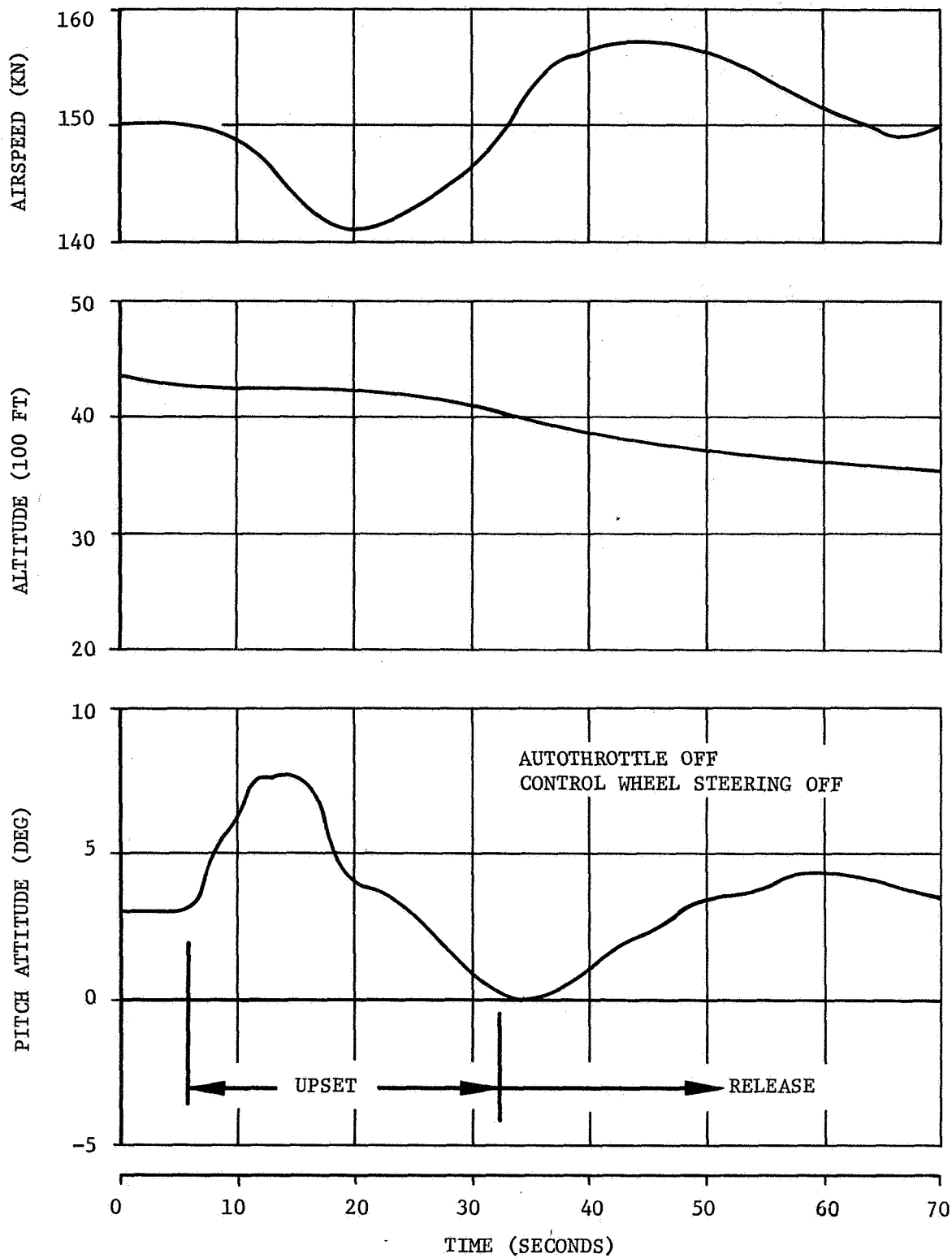
FIGURE 8a. APPROACH LONGITUDINAL STABILITY

703
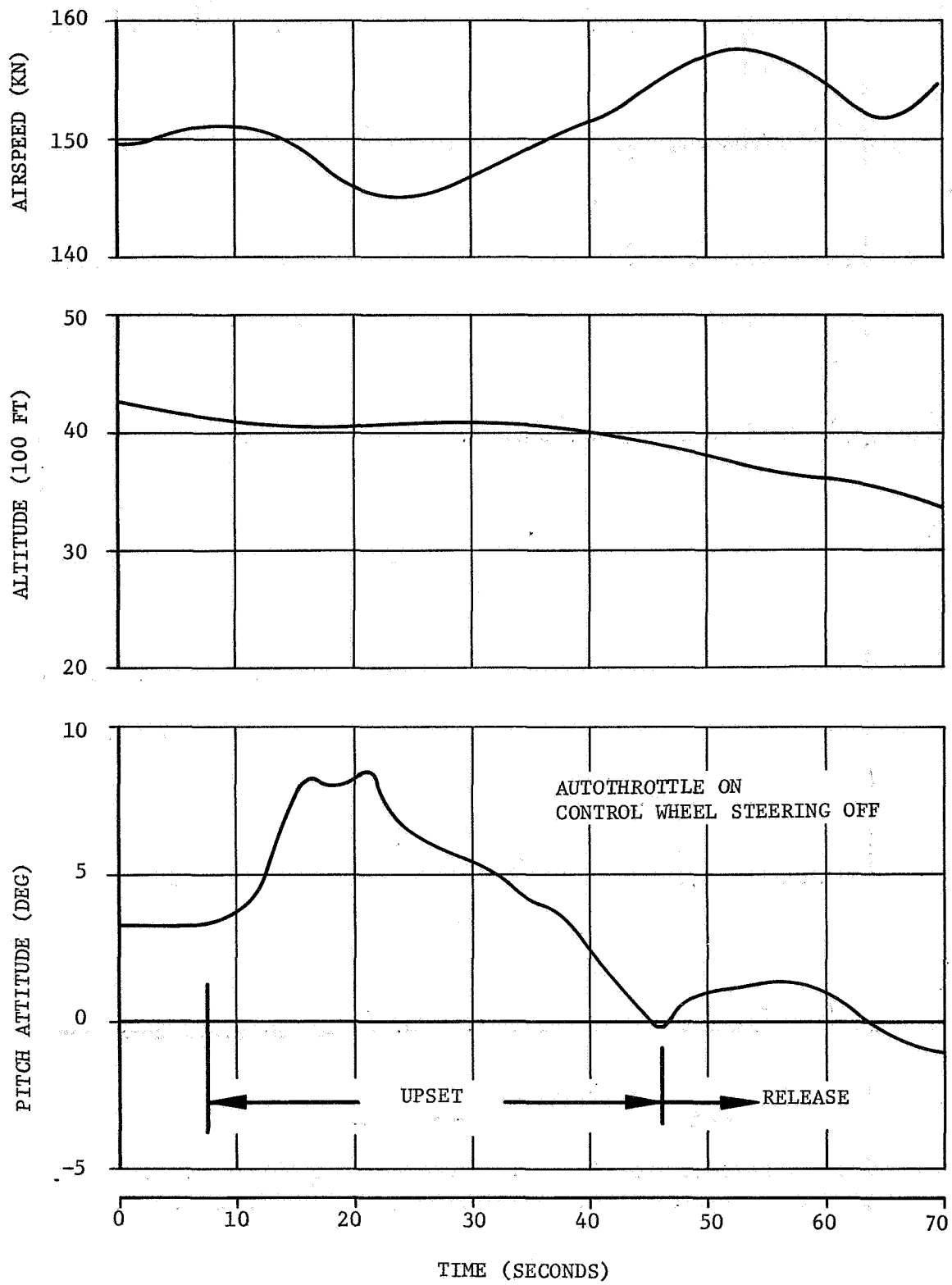
FIGURE 8b. APPROACH LONGITUDINAL STABILITY

704
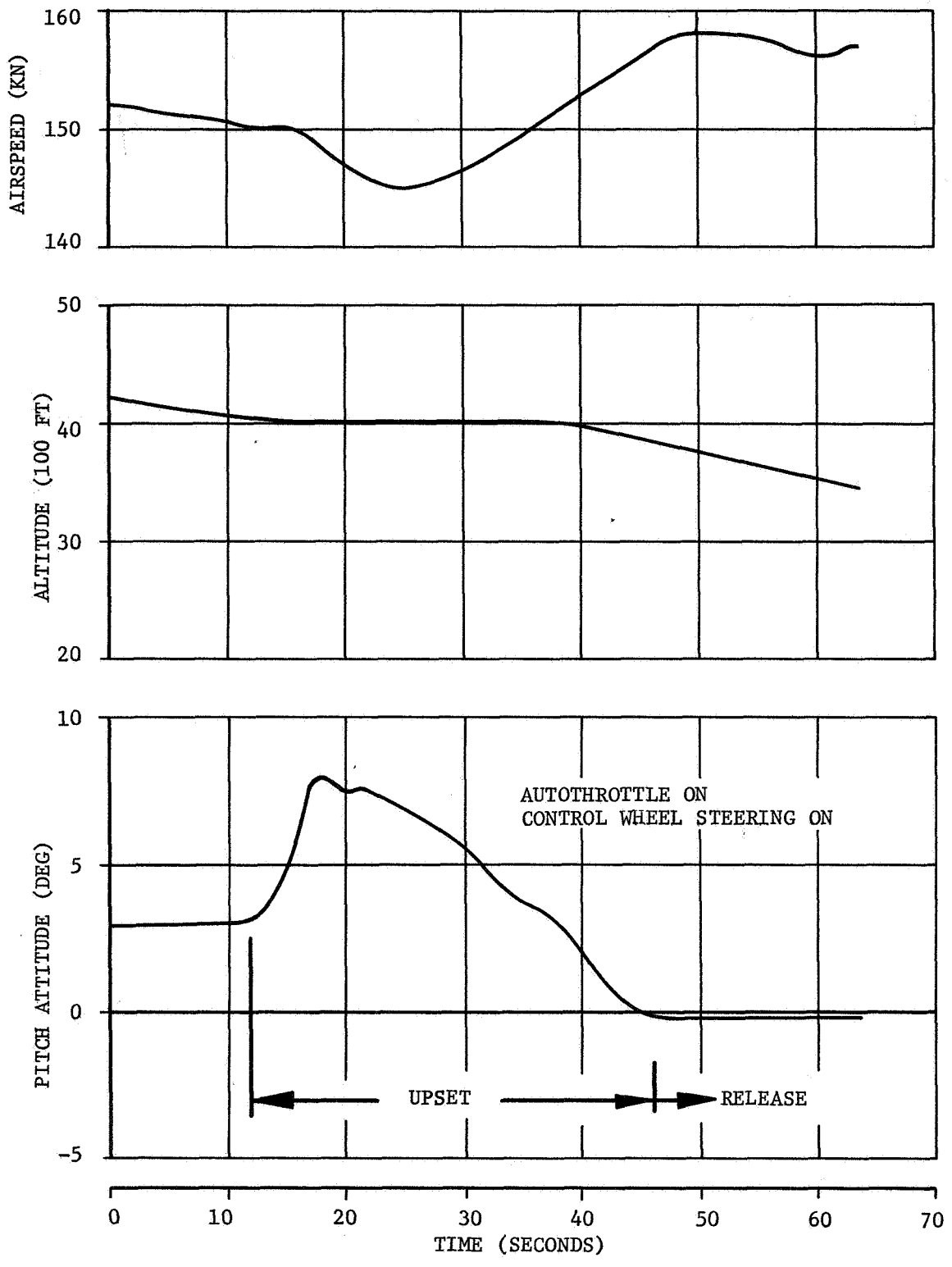
FIGURE 8c. APPROACH LONGITUDINAL STABILITY

ADVANCED CONTROL TECHNOLOGY AND

AIRWORTHINESS FLYING QUALITIES REQUIREMENTS

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INTRODUCTION

Advanced control technology poses a difficult task for the authorities faced with specifying airworthiness flying qualities requirements--and for the manufacturers who must comply with and anticipate these requirements. Requirements for advanced civil transports employing this technology must be carefully framed, such that public safety is ensured and technological advances in civil aviation are not discouraged. It is no secret that excessively complex and overstringent requirements discourage innovation, while clear and flexible requirements (for example, those that give credit for reliability in systems) encourage development and advances in technology.

The specification of flying qualities requirements involves consideration of the complete pilot-airframe-systems loop, the task, and the environment. Figure 1 suggests the complexity of this job; many of these advanced civil configurations tend to be large and flexible and dependent on complex control systems for enhancement of stability, control effectiveness, and control feel characteristics over enlarged flight envelopes, and for numerous automatic control modes. The result is a greatly increased emphasis on failure effects that degrade flying qualities. Key questions being faced include: How good must the flying qualities be in the failure condition? Which failures and combinations must be demonstrated? And how must they be demonstrated?

French and British authorities, in preparing for Concorde SST certification, authored a new form of flying qualities requirements that rely heavily on probabilistic analyses (TSS Part 3, ref. 1). In TSS 3, the required standard of flying qualities varies according to the likelihood of the flight condition occurring, and thus considers the wide range of flight phases, system failure effects, and atmospheric environment. Although it is being applied to Concorde by European authorities and some features of the method have been utilized in U.S. military specifications, the TSS 3 approach has met with mixed reactions among the U.S. civil aviation community because of concerns over the practical implementation of the method.

Since 1969, an ongoing NASA/FAA research program has used the Ames Flight Simulator for Advanced Aircraft (FSAA) in the development of certification criteria for supersonic cruise aircraft. NASA, FAA, industry representatives, and British and French airworthiness authorities are participating in this program. The question of proper accountability of failures has arisen on numerous occasions. These experiences have brought to a focus the

need to review the present treatment of failure cases in the requirements and to examine some of the questions associated with implementation of the TSS 3 type of concept.

This paper, which reports on the findings to date from a continuing study of the subject, comprises the following: a review of the treatment of failure cases in various flying qualities requirements; a description of methods used and relevant lessons learned from recent Autoland certification programs as an example of applied probability procedures; a discussion of uncertainties about the TSS approach; and finally (because these procedures indicate an increasing reliance on simulation methods), a description of three recent experiences with marginal configurations that demonstrate the potential significance of elements sometimes omitted from simulation tests.

CURRENT TREATMENT OF FAILURE CASES IN VARIOUS FLYING QUALITIES REQUIREMENTS

Aircraft flying qualities requirements deal primarily with controllability, stability, and handling characteristics. Civil and military requirements were reviewed for the manner in which failure cases were covered, the amount of flying qualities degradation allowed, the conditions under which failures were to be assessed (for example, introduction of atmospheric effects), and methods for demonstrating compliance. (As used throughout this paper, the term "failure" includes malfunctioning as well as failure to function; degraded system performance below specified tolerances represents a failure to function properly.) Documents reviewed included Federal Aviation Regulations applicable to transport category airplanes (FAR 25, ref. 2), Tentative Airworthiness Standards for Supersonic Transports (TASST, ref. 3), industry recommendations (AIA committee report, ref. 4 and SAE Aerospace Recommended Practice 842B, ref. 5), Franco-British Concorde TSS Standards (TSS, ref. 1), and U.S. military specification (MIL-F-8785B, as described in ref. 6).

Federal Aviation Regulations - FAR 25 and TASST

For orientation, an outline of FAR 25 is shown in figure 2. Flying qualities requirements are contained in "Subpart B - Flight," which is further broken down into topic headings. Although flying qualities are closely interrelated with many performance requirements (many of which involve engine failure conditions), this discussion is primarily concerned with those items indicated by an arrow, and the related paragraphs in "Subpart D - Design and Construction" and "Subpart F - Equipment."

Failure Cases in FAR 25- Philosophy towards treatment of failures has undergone significant change in recent years. For years, about the only multiple failure cases were two-engine-inoperative control requirements and a requirement that the airplane be controllable with all engines inoperative. Only single control system failures were considered. In April 1970,

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Amendment 25-23 incorporated a number of changes into FAR 25 dealing with system failures and introducing the consideration of multiple failures. Stability augmentation systems and automatic systems were dealt with specifically. Some of the new requirements came from the tentative SST requirements and were recognized to be generally applicable and needed because of the increasing dependence on more complex systems of the new generation of subsonic transports. The example shown in figure 3 illustrates the present treatment of control system failures. As indicated, FAR 25.671 requires the capability of continued safe flight and landing after any single control system failure or after any combination of failures not shown to be extremely improbable.

Current FAA interpretation of the terms "probable," "improbable," and "extremely improbable" is shown in the sketch below.

FREQUENCY OF OCCURRENCE PER FLIGHT HOUR
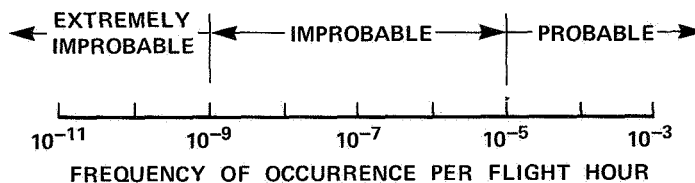
Failure cases in the other aircraft systems must also be analyzed under "Subpart F - Equipment." In addition to requiring the capability of continued safe flight and landing after any failure condition not extremely improbable, FAR 25.1309 requires that the systems and associated components be designed so that

> "the occurrence of any other failure conditions which
> would result in injury to the occupants, or reduce the
> capability of the airplane or the ability of the crew
> to cope with adverse operating conditions is improbable."

In addition, nearly all requirement sets contain catchall paragraphs which are in general terms, but provide evaluation pilots basis for rejection of unsatisfactory situations not covered specifically. An example of this is FAR 25.143 which states

> "(a) The airplane must be safely controllable and
> maneuverable during - (1) takeoff; (2) climb; (3) level
> flight; (4) descent; and (5) landing. (b) It must be
> possible to make a smooth transition from one flight
> condition to any other without exceptional piloting
> skill, alertness, or strength and without danger of
> exceeding the limit-load factor under any probable
> operating conditions (including the sudden failure of
> any engine)."

The phrase "under any probable operating conditions" is certainly subject to interpretation as including failure cases.

Failure Cases in TASST- In TASST (ref. 3), the FAA presented tentative airworthiness standards for study, trial application, and comment during the detail design and prototype phase of supersonic transport development. A number of changes were proposed in this document other than those already discussed, including requirements to cover automatic and manual trim system malfunctions, additional two-engine-inoperative controllability and maneuverability requirements, and extended "flutter, deformation and fail-safe criteria" to consider combinations of failures not shown to be extremely improbable. In the "Stability" section, it was recognized that areas of flight (for example, supersonic cruise) may exist with operational requirements such that the use of reliable automatic flight control systems could be accepted in lieu of the demonstration of classic static stability, provided the loss of automatic flight control would not result in unsafe handling characteristics.

The "Structures" section of TASST is closely related to flying qualities. Paragraph 25.301(e) states that

"For supersonic aircraft, loads must be determined
within the design flight envelope considering the ef-
fects of stability augmentation and automatic flight con-
trol systems, including probable failures and changes in
systems characteristics which can be expected in service.
All malfunctions and failures of these systems must be
considered under FAR 25.671 and FAR 25.1309 within the
normal flight envelope except those shown to be extremely
improbable."

Careful consideration of the complete pilot-aircraft-systems loop and the environment appear very important in satisfying this requirement. Also note that this introduces the assessment of failure effects outside the normal flight envelope (see sketch). The application of flight simulation techniques would appear to be essential for this task.

NORMAL FLIGHT ENVELOPE —
PROBABLE AND IMPROBABLE
FAILURES ($> 10^{-9}$ per hour)
MUST BE CONSIDERED IN
THIS REGION

DESIGN FLIGHT ENVELOPE —
PROBABLE FAILURES
($> 10^{-5}$ per hour) MUST BE
CONSIDERED IN THIS REGION

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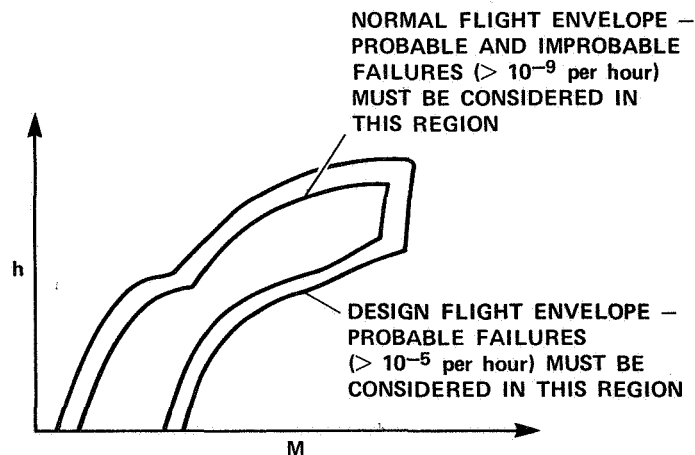
Turbulence and Flexibility Effects- In the introductory discussions to both the "Controllability and Maneuverability" and the "Stability" sections of TASST, it was recognized that the effects of turbulence on the pilot environment should be evaluated. In addition, it was pointed out that the structural flexibility and stability characteristics of supersonic transports will undoubtedly aggravate the pilot environment problem. Flight experience with the XB-70 and F-12 series aircraft lends considerable weight to these statements—as have some piloted simulator experiences with large flexible configurations to be described later. It is very likely that many of the advanced transport designs will exhibit greater flexibility than current subsonic transports and, as will be shown, the effects on handling characteristics in failure-mode operations can be very significant.

Industry Recommendations

AIA Study Group Proposals- In 1970, a special project group representing the Aircraft Industries Association (AIA) published the results (ref. 4) of a study to guide the modernization of the Federal Air Regulations. In reference 4, proposed "modernized" requirements are presented as a set of safety standards generally applicable to all transport aircraft types. These standards describe basic characteristics of the aircraft system that must be achieved to ensure safe operation. In addition, means for showing partial or complete compliance with the individual standards are included. Two fundamental requirements formed the foundation for all the standards proposed:

> "1. The aircraft must respond to commands of the controlling intelligence in a consistent manner and with the precision appropriate to the task.
> 2. Probable subsystem failures must not result in conditions likely to be catastrophic due to human inability to cope with them."

These modernized standards specify three modes of operation (manual, command, automatic), conveying clearly that the controlling intelligence is not always considered to be the human pilot. They state further that if man is the controlling intelligence, he should be considered a subsystem of the total aircraft system. In this way, the standard dealing with operation following failures accounts for failures of human origin in addition to other subsystem failures. This standard states,

> "Operation following probable failure of any subsystem that affects flight safety shall not unduly restrict flight operation after corrective action is taken. The degree of restriction permitted shall be inversely related to the probability of failure."

It then presents requirements related to the ability to take corrective action, either by the crew or by automatic means. The acceptable means of compliance deal more specifically with the failures, and include paragraphs

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that parallel FAR 25.671(c) and 25.672(c) (fig. 3).

SAE Design Criteria- Recommended design criteria for handling qualities of civil transport aircraft (SAE ARP 842B, ref. 5) differ in character from the safety requirements described previously. These criteria represent advisory design information as defined by the SAE and were originally modeled after the format of the military specifications of the early sixties. These criteria appear to have avoided the use of probability terminology and include consideration of single and dual control system failures.

> "... Following the [single] most critical failure in
> the [power or boost] flight control system, the planned
> flight may be completed without a significant degrada-
> tion of flying qualities. ... Following the second
> most critical flight control system failure, it shall
> be possible to complete the flight, after takeoff, to
> a suitable airport from the V_2 transition to enroute
> climb to cruise to a safe landing with the most critical
> engine inoperative at the most critical phase of flight."

They further state that failure of any artificial stability system or powered-actuated trim system should not result in an unsafe flight condition. Some of the quantitative criteria (for example, lateral control) are redefined for the failure cases to accept degraded capability.

The significance of aeroelastic effects is recognized in paragraph 2.1.7 of ARP 842B which states

> "Since it can be expected that aeroelastic effects
> will play an important role in supersonic transport
> design, it should be clear that all requirements for
> flying qualities are applicable to the elastic air-
> frame."

Franco-British TSS 3

General Description and Objectives- A new approach to flying qualities requirements was developed by the French and British airworthiness authorities in preparation for the certification of supersonic transports, Concorde in particular. First published in 1969 as TSS 5 and since changed to TSS 3 (ref. 1), these requirements are currently being applied to Concorde. Their most significant feature is the extensive use of probabilities and systems analysis methods in defining the minimum acceptable flying qualities for a given flight situation, considering the flight phase, aircraft configuration, failure state, and environment. The severity of the requirement is directly related to the probability of occurrence of the flight situation. This concept has since been utilized in the British Provisional Airworthiness Requirements for Civil Powered-Lift Aircraft (ref. 7) and in modified form in the current U.S. military specification MIL-F-8785B.

This approach provides the following significant advantages:

1. a more systematic and complete coverage of all likely flight conditions, whereas past methods have tended to be limited to anticipated critical regions

2. consideration of atmospheric environment effects in a more complete manner

3. a running assessment of the relative risk level throughout the design and development phases for a new aircraft, which provides insight for design modifications

4. a method for defining those cases that can be eliminated from demonstration because of the low probability of occurrence.

The TSS standards are intended to provide the same safety levels for supersonic transports as for subsonic airplanes introduced into service at the same time. These objectives include the following: "For all airworthiness causes the total probability of Catastrophic Effects should be Extremely Remote [$<10^{-7}$ per hour of flight], and the total probability of Hazardous Effects should be remote [$<10^{-5}$] or Extremely Remote." (See table 1 for definition of terms.) Akin to FAR 25, these objectives state that "No single Failure or combination of failures not considered Extremely Improbable shall result in a Catastrophic Effect." They further require that "Remote Failures shall not result in Hazardous Effects" and that "Recurrent Failures shall result only in Minor Effects."

The TSS 3 requirements are categorized into three groups, corresponding to the accident causes attributed to flying qualities: (1) handling - a workload consideration, (2) maneuverability, and (3) involuntary exceedance of airplane limits caused by disturbances due to failures or atmospheric conditions. Various specific criteria are included which, depending on the probability of occurrence of a given "state" (categorized as frequent, occasional, exceptional, and non-exceptional), must be satisfied. There are also a number of requirements, based on judgment and experience, which require demonstration regardless of the estimated probability of occurrence.

Theoretical Application- Figures 4 and 5 illustrate theoretical application of the TSS 3 concept. (Reference 9 points out that practical application requires many simplifying assumptions, although little information on these assumptions has been found in the literature.) First, the various possible flight "tasks" and their associated probabilities of occurrence per flight are defined. As shown in figure 4, a task is defined by four primary elements plus the secondary workload: (1) the flight subphase, for example, localizer capture; (2) state of the atmosphere; (3) state of the aircraft, which includes possible failures; and (4) flight technique. Elements 1 and 4 represent lists prepared by the applicant while elements 2 and 3 represent four-dimensional matrices. The probability P_n of a given task per flight is then calculated from estimates of the probabilities of (1) performing a given subphase per flight, (2) encountering a given atmospheric state during the subphase, (3) having a given aircraft state during the subphase, and (4)

using a given flight technique.

Figure 5 represents the author's interpretation of a method described in TSS 3 for showing compliance with the general handling requirement. Pilot evaluation of a given task identifies a class of difficulty C, which is then converted to the probability P_u that a pilot will not be able to accomplish the workload. The probability of a handling incident during a given subphase per flight is determined by summation over the classes of difficulty of the product of P_u and P_n for that subphase. The total probability of a handling incident per flight is computed by summation over all the subphases that make up a flight. For partial compliance, this total probability must then be less than a safety index, which has been defined as an acceptable risk level.

TSS 3 states that the demanded safety level is to be demonstrated by a limited number of flight tests proposed by the applicant. (Justification for tests omitted is also required.) The majority of these are to be conducted in calm air or low turbulence. Compliance with requirements for flight in turbulence are to be demonstrated by a limited number of flight tests, supported by theoretical studies and simulator tests.

From this brief description, it is clear that numerous questions can be raised regarding the practical application of this approach and that considerable simplifications are needed. In the next section, simplifications are described which have been made in the application of a similar procedure in the U.S. military specification. While all the uncertainties are not laid to rest, discussions in the following sections address many of the expressed concerns, and point the way for continuing work.

U.S. Military Specification (MIL-F-8785B)

Similarity with TSS Concept- MIL-F-8785B (presented with background information in ref. 6) serves dual roles as design requirements and as evaluation criteria. At a 1971 AGARD meeting, a paper (ref. 9) was presented comparing the TSS 3 concept and MIL-F-8785B. It concluded that they are basically the same in intents and goals, although one distinction was made: in addition to assuring that there will be no limitations on flight safety due to deficient flying qualities, MIL-F-8785B demands that mission effectiveness will not be compromised. Similarity of the two criteria is not coincidental; discussion following presentation of the AGARD paper acknowledged the significant contributions made by M. Wanner, representing the Service Technique Aeronautique of France and a strong advocate of the TSS 3 concept, during the preparation of MIL-F-8785B.

A number of simplifying assumptions have been made to permit practical application of MIL-F-8785B, including:

(1) No probability assessment is made for aircraft mass and mass distribution. A probability of 1 is used for all points in the envelope. Thus, probability of state of the aircraft is dependent

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on failure probabilities only.

(2) No attempt is made to estimate the probability of the state of the atmosphere. The required flying qualities are associated with the state of the airplane. (A number of specific flying qualities requirements must be met with specified turbulence conditions, however.)

(3) The probability of being in a given area of the flight envelope has been assumed equal to 1, due to inability to specify this value.

"Levels" of Flying Qualities- Three levels of flying qualities are defined in MIL-F-8785B, as shown in table 2. Cooper-Harper pilot ratings generally associated with the three levels are also shown. Exceptions to these relationships exist, however. For example, level 3 flying qualities for a landing task would correspond to a pilot rating no poorer than 6.5 (requires adequate performance; see fig. 6).

The minimum required flying qualities are defined separately for airplane normal states and airplane failure states. For airplane normal states, level 1 flying qualities are required within the operational flight envelope, and level 2 within the service flight envelope (fig. 7). For airplane failure states, the probability of encountering level 2 flying qualities must be less than 10^{-2} per flight within the operational flight envelope and the probability of encountering level 3 flying qualities must be less than 10^{-4} per flight in the operational flight envelope and less than 10^{-2} in the service flight envelope.

Theoretical Compliance Procedure- Figure 8 illustrates the procedure outlined in MIL-F-8785B for determining theoretical compliance with the failure state requirements. Airplane failure states that have a significant effect on flying qualities are first identified and the corresponding probabilities of encounter per flight are computed, based on the longest flight duration to be encountered during operational missions. The degree of flying qualities degradation associated with each airplane failure state is determined in terms of levels as defined in the specific requirements. The most critical airplane failure states are then determined (assuming the failures are present at whichever point in the flight envelope being considered is most critical in a flying qualities sense), and the total probability of encountering level 2 flying qualities in the operational flight envelope due to equipment failures is computed. Likewise, the probability of encountering level 3 flying qualities in the operational flight envelope is computed. The computed values are then compared with the requirements.

Concept Recommended for Civil Airworthiness Application- Many of the military specifications were originally recommended by Cornell Aeronautical Laboratory, Inc. (now the Calspan Corporation) under contract to the Air Force Flight Dynamics Laboratory. In 1973, Calspan completed a review of the "Flight" subpart of the Yellowbook (Tentative Airworthiness Standards for Powered Lift Transport Category Aircraft) for the FAA. The final report (ref. 11) proposed that the Yellowbook be revised to a new format based on

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many of the ideas used in the military specification and in the British Provisional Airworthiness Requirements for Civil Powered-Lift Aircraft (ref. 7).

General Observations

Based on review of the various requirements, several observations can be made. All elements of the aviation community have acknowledged the need for increased attention to failure effects and have made the transition from single failure to multiple failure philosophy. The prediction of system failure probabilities and their effects has become a significant factor in flight certification of aircraft employing stability augmentation, automatic, and powered control systems. For aircraft employing active controls technology to full advantage, the system failures and effects analyses are even more important. This results in a growing need for close integration of the systems and flying qualities disciplines. Present U.S. certification practice appears to treat the systems and flying qualities evaluations somewhat separately, with the effects of failures often defined by analytic means in the systems studies. While this procedure may have served adequately in the past, the foregoing observations suggest that they will, at the very least, require reexamination for future applications.

General recognition is apparent that atmospheric effects (e.g., turbulence) can influence an airplane's handling characteristics significantly and should be considered, although the method of including this is loosely defined. The high cost of flight testing, the large number of cases to be evaluated, the desire to assess in specified atmospheric conditions, and at marginally safe conditions can be expected to increase the reliance on piloted flight simulators for much of this work.

FAILURE CASE ANALYSES IN AUTOLAND CERTIFICATION

Systems safety analysis procedures used in recent Autoland certification programs represent current examples of the application of probability procedures to the certification of total airframe-systems combinations, including consideration of atmospheric effects. Because of the close relationship with the evaluation concepts previously discussed, the procedures used in the Category IIIA automatic landing programs for the McDonnell Douglas DC-10 and the Lockheed L-1011 (refs. 12-14) were reviewed and relevant findings are noted.

Procedures

The procedure described in ref. 13 appears to be generally representative of the programs for both airplanes. The certification process, which represents the final cycle of studies made in the design and development phases, used progressive simulation and testing, as indicated in figure 9,

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in order to minimize the amount of flight testing required. The first step was the use of high-speed repetitive-operation simulation methods to accomplish the millions of landings required for establishment of the low probability results in a reasonable time period. In the second phase, several thousand simulated landings were made using the actual flight hardware computers. The hydraulic control systems hardware ("iron bird") was then added to the simulation in order to pick up effects of any hardware imperfections. Finally, a minimal number of flight test demonstrations (on the order of a hundred) were made to verify the high end of the performance probability curves. Some of the simulated failure effects were verified by inserting failures into the autopilot during actual approaches. Each of these phases was used to verify the results of the preceding phase.

Environmental conditions for these simulations included turbulence and wind shear, with levels specified in FAA Advisory Circular 20-57A. In the DC-10 program, key performance characteristics of the sensors, analog computer, and mechanical controls were varied between simulation runs within the normally expected ranges using a Monte Carlo sampling routine (ref. 12).

Not evident in the procedure just described is the reliability and safety analysis, a considerable task consisting of an integrated combination of several kinds of analyses and computer simulation techniques. This extensive process is described in detail in ref. 12. Suffice to say that it involved identifying all possible single and multiple faults in the system and their effects, eliminating all single faults that were hazardous, and establishing that no multiple fault in the system having a probability of occurrence greater than 10^{-9} per landing was hazardous.

Relevant Findings

Integrated Programs Necessary- Ordinary numerical reliability analyses were recognized at the outset to be inadequate for fully assessing Autoland system reliability and safety. Because of the basic system complexity, the airborne-ground systems interfaces, and the numerous pilot-aircraft interfaces, integrated programs of laboratory testing, computer analyses, and simulation were found necessary.

Design Guidance Provided- Certification considerations began with the design phase. As the system design evolved, the reliability and safety analyses provided continued assessment of compliance and identified areas requiring design modification. Consideration of failure effects significantly influenced the design of many other aircraft systems, for example, electrical supply.

Multiple Failure Analysis Found Manageable- The multiple failure analysis appeared at first to be an almost impossible task, requiring the combination of all possible failures in all possible sequences and analyzing the result. This task became manageable by first defining what was hazardous and then working backward to find all combinations of faults that could produce the event.

Definition of Atmospheric Disturbances Needed- Atmospheric disturbance effects can become primary design factors. In some flight tests, for example, a condition not anticipated to be critical--a quartering tailwind--was found to be serious. Other flight test experience has indicated that present specified wind shear values may be inadequate. The potential significance of such disturbances makes accurate definition of the atmosphere essential.

Broad-based Engineering Judgment Necessary- A fundamental merit and a hazard of the probability approach are revealed in this quotation from ref. 13:

> "The probability approach to analysis seems, from experience, to have great merit in that the necessity to calculate very low probability numbers forces on the analyst a discipline that makes him study the system in greater detail. The danger in the approach is that the analyst may place too much emphasis on the techniques he has developed, and lose sight of the many assumptions implied in these techniques. In short, there is a danger of placing implicit belief on the accuracy of a calculated number. This danger can be avoided by the use of highly skilled engineers who are capable of understanding system and aircraft operation as well as the detailed working of the circuits to be analyzed."

Concluding Observations

Although the Autoland systems safety analysis procedures described herein appeared extremely cumbersome at the outset, in actual practice they became manageable--while providing significant payoffs in terms of design guidance and improved safety. It should also be noted, however, that while the effort involved in an Autoland certification program is undoubtedly large, the effort appears small when compared to the total effort required in rigorously applying the procedures of TSS 3 to an advanced transport aircraft over its entire flight envelope. The number of cases to be considered for Autoland is limited: the Autoland process is concerned primarily with the final few minutes of flight, and the controlling intelligence can be mathematically modeled more readily than can the human pilot.

UNCERTAINTIES REGARDING THE TSS APPROACH

While the potential advantages of the TSS 3 type of approach have been shown to be very significant (partially verified by the Autoland experience), numerous questions and uncertainties have been raised regarding its practical implementation. These can be grouped under the following headings: (1) reliance on probability methods and reliability prediction, (2) size of the evaluation matrix, (3) use of the pilot rating scale, (4) definition of the

atmospheric environment, and (5) use of simulation methods. In the following discussion, each of these has been addressed in order to provide some insight into these topics, to dispel some fears, and to indicate where further work is needed.

Reliance on Probability Methods and Reliability Prediction- Concern has been expressed with regard to the ability to define some of the required probabilities, such as the flight subphase, pilot technique, and atmospheric environment. Conservative engineering estimates of the first two should be possible with careful study. Definition of the probability of a given atmospheric environment appropriate to a given subphase requires more research (to be discussed later).

In reliability and safety analyses, there is always the danger of "blind faith" in the calculated number. As pointed out in TSS 3 and in refs. 8 and 15, these methods are used as an aid, not as the sole criterion; it is essential that they be combined with good engineering judgment and experience. The practical limitations of a given method must be taken into account and experience with other aircraft in service must be factored into the total assessment. For example, the present safety assessment of redundant systems goes far beyond the failure analysis by considering possible effects of errors by the crew and maintenance personnel, as well as the effects of events outside the aircraft which could affect more than one channel at a time. Redundant systems are checked for common faults to ensure, for example, that both electrical systems are not routed through a common wiring bundle or under galleys and toilets, or that lines from both hydraulic systems are not supported by a common bracket.

A common question asks how probability values of the order of 10^{-7} per flight hour can be estimated with confidence. Reference 16 points out that this is exactly the reason for the philosophy that no single failure can create a catastrophic flight condition. The period of proof-testing required to prove this failure rate would be impractical. However, the individual failure rates of interest in multiple-failure analyses, of the order of 10^{-3}, can usually be estimated with reasonable confidence. It is also intended that critical system failure records be kept on new aircraft entering service over the initial period of operation to verify the reliability estimates.

Evaluation Matrix- The matrix of conditions requiring evaluation under the procedure described in TSS appears awesome. However, considerable simplification appears possible and merits continued study. Also, in practice, the number of failures to be investigated normally turns out to be a manageable number (ref. 8). The fault analysis usually shows a limited number of ways a system can malfunction following a variety of single and multiple faults. Many can be discarded because the result is not serious or the probability of occurrence is clearly satisfactory.

Use of the Pilot Rating Scale- Concern has been expressed over making a pilot rating scale a part of legal regulation, to be used in determining the minimum safety level of an airplane. Questions faced whenever the pilot

rating scale is used become especially significant when it is the minimum
safe boundary being defined. Typical questions are: What pilots are to do
the rating, how many, how to extrapolate from the flight test situation to
the operational one, etc. These questions and others are worthy of careful
study and resolution. Considerable worthwhile discussion on many of these
issues is contained in ref. 10. Many of the questions raised, however, are
not unique to the TSS procedure, but are equally applicable to the present
evaluation process where the subjective opinions of the airworthiness pilots
are key factors in defining the acceptability of a given airplane.

Definition of the Atmospheric Environment- Definition of the significant
elements of the atmospheric environment and associated probabilities is an
area receiving considerable attention in the U.S. and in Europe, and justifi-
ably so. The influences of atmospheric disturbances become especially trouble-
some as flying qualities are degraded and the workload approaches the satura-
tion point.

Many of the turbulence models currently being used in simulation studies
have been tailored to match power spectra measurements. Other concepts are
being studied. For example, recent work in the U.K., stimulated by Autoland
experience, is investigating the use of discrete gust patterns (ref. 17), and
work is continuing in this country under NASA sponsorship at the University of
Washington and elsewhere to develop "non-Gaussian" models. Also, an investi-
gation devoted to verification or improvement of present methods for modeling
aircraft response to turbulence appears worthwhile.

Simulation Methods- The preceding discussions leave little doubt that the
use of simulation methods will play an increasingly key role in the design,
development, and certification of advanced transport aircraft. The applica-
tion of simulators to the certification demonstration process must not be
approached naively, but with appreciation for the limitations of these methods
and for the degree of fidelity (math model, pilot station layout, visual dis-
play, motion, etc.) required for specific tasks. Representation of the appro-
priate workload level, for example, is an important factor in evaluating mini-
mum safe handling qualities.

For a safety assessment as defined in TSS 3, development of this simula-
tion capability early in the design phase, with progressive updating of the
airplane model and the pilot/pilot station interface, appears essential. Ac-
quisition of data for improvement of simulation fidelity must be factored into
layout of early flight tests. Accurate representation of failure annunciators
and warning devices must be incorporated as they are defined, as they are im-
portant elements in the evaluation of a proposed system's acceptability.

RELATED SIMULATION EXPERIENCES

The preceding discussions lead to the conclusion that the final acceptance
of many aircraft failure states may be based largely on simulator evaluations
(and engineering judgment). Three recent simulation experiences have empha-

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sized factors which, with more stable configurations, might have been considered of secondary importance, but became critical components requiring accurate representation in the simulation of marginal configurations. The first emphasizes the significance of turbulence effects, the second indicates the importance of motion cues in critical tasks, and the third demonstrates control limitations that can be imposed by structural mode effects.

Turbulence Effects

In 1972, parallel studies were conducted on two simulators to investigate the SAS-failed approach and landing of delta-wing transports (refs. 18 and 19). Three research test pilots performed ground-based evaluations on the NASA/Ames six-degree-of-freedom Flight Simulator for Advanced Aircraft (FSAA), followed by flight evaluations on the USAF/Calspan Total In-Flight Simulator (TIFS), both shown in figure 10. In a matrix of twenty test configurations, seventeen were unstable longitudinally. The primary task was an ILS approach under IFR conditions, breakout to VFR conditions at 91-m (300-ft) altitude, visual approach and landing. A series of approaches with added tasks included crosswind approach, glide-slope error correction, localizer error correction, and moderate (0.91 m/sec or 3.0 ft/sec rms) turbulence. Of these, the turbulence task proved to be the most critical, although the turbulence intensity used was not uncommon (probability of encountering turbulence of 0.91 m/sec or greater is on the order of 0.1 to 0.3).

Pilot rating data from both investigations is shown versus a divergence parameter, time to double amplitude of angle of attack T_{2_α}, in figure 11. Pilot ratings from the FSAA study are shown by the shaded band, with the scatter primarily attributable to interpilot variation. Values of T_{2_α} of 6 sec and greater were found to be acceptable for the emergency case. As T_{2_α} decreased (divergence rate increased) below this level, pilot ratings show that handling characteristics deteriorated rapidly.

Initial examination of the TIFS pilot rating data showed considerable scatter due to the varying turbulence intensities encountered during the flights. In analyzing the data, Calspan used measurements of the actual turbulence environment to compensate the pilot rating data for each configuration. These results are shown in figure 11 for gust intensities of 0.46 m/sec (1.5 ft/sec) and 0.91 m/sec (3.0 ft/sec). Although the difference between the two levels of turbulence intensity appears small, the differences in subjective evaluation were significant.

Motion Effects

An investigation was conducted at Ames recently to identify the role of cockpit vertical acceleration cues in the landing task (ref. 20). A piloted simulator having very large amplitude vertical motion (24 meters total travel) was utilized in a test series in which the fidelity ("washout") of the vertical acceleration reproduction was deliberately varied over a wide range, representing simulators with varying amounts of available vertical travel. The external

visual scene was provided by a black and white uncollimated TV monitor. The airplane simulation represented a large sweptwing business jet transport. Three levels of static longitudinal stability were simulated, corresponding to 15 percent static margin, neutral, and 5 percent unstable static margin.

The results indicated that vertical motion cues were utilized in the landing task and were particularly important in the simulation of aircraft with marginal longitudinal handling qualities. Figure 12 shows a measure of landing performance, altitude rate at touchdown, plotted against the motion washout filter natural frequency ω_m. The corresponding vertical travel requirements are shown along the top scale. The data indicate that the effect of motion was relatively inconsequential for landing of the stable configuration with good flying qualities, although an oscillatory tendency was observed without motion. However, with the configurations having marginal longitudinal handling qualities, significant degradation was apparent in achievable performance as the motion was constrained (and thereby distorted). At values of ω_m of 1.0 and above, divergent flight path oscillations were common and touchdowns were essentially uncontrolled in many landings.

Structural Mode Effects

In another simulation program in which a very large flexible aircraft was represented complete with structural modes, a very significant degradation in flying qualities resulted from the pilot station motions caused by fuselage bending. Evaluations of the completely unaugmented airplane without motion and body bending resulted in pilot ratings of 5.0 - 5.5 (fig. 6). With motion and body bending, the structural modes were easily excited and the pilots were unable to use the sharp pulse inputs (in pitch) normally used for control of an unstable airplane. This prevented the use of effective control techniques and yielded a pilot rating of 9.

Recommendations

These examples have illustrated the fact that careful attention must be devoted to defining the simulation requirements for a given task. A high degree of sophistication is often required in evaluations of marginal cases if confidence is to be placed in the results. Practical design and evaluation procedures will very likely, by necessity, rely on simplified simulations (very limited motion, no structural mode representation, etc.) for the bulk of the work. It is emphasized that verification testing of critical cases should be planned in simulation facilities which provide a high fidelity of the total task presentation.

CONCLUDING REMARKS

Advanced transport aircraft designs have become increasingly dependent on complex flight control systems in order to improve their flight characteristics.

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In this report, various civil and military flying qualities requirements have been reviewed with regard to their treatment of failure cases and consideration of atmospheric environment effects. There appears to be common acceptance of the philosophy that no single failure should create an unsafe flight condition, nor should any combination of failures that are not extremely improbable. Although consideration of atmospheric environment effects in handling assessments is required, the method for doing this is often ill-defined.

There is an increasing need for an orderly procedure for combining the systems analyses (reliability and fault analyses) with the flying qualities evaluation process, taking the likely atmospheric states into account. Such a procedure can aid in the achievement of design economies and a level of safety equivalent to that of current transports; this is a challenging task since the contribution of system failures to catastrophic effects is at present a very small proportion of the total. Review of the probability-based procedures described in the Anglo-French TSS 3 shows that additional development effort is needed to simplify implementation. Simplified procedures described in the U.S. military specification and lessons learned from recent Autoland programs appear useful for continuing studies devoted to this purpose.

In order to minimize flight testing and to enable evaluation in specified atmospheric conditions and hazardous failure cases, simulation techniques will be used extensively in such procedures. Some recent simulation experiences emphasize that turbulence effects significantly influence the pilot evaluations of marginal configurations, and that evaluation of such conditions sometimes requires more accurate representations of the pilot/aircraft interface (pilot station motions, etc.) than are provided in many current engineering simulations.

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Table 1. Definition of probability terms (ref. 8)

Failures	Effects	
	Type	Definition
Recurrent failures	Minor effects	Can readily be counteracted by crew and may involve: (a) small increase in work load. (b) moderate degradation in performance or handling. (c) slight modifications to the permissible flight envelope.
Remote failures	Major effects	May produce: (a) significant increase in crew work load. (b) significant degradation in performance or handling characteristics. (c) significant modification of the permissible flight envelope. but will not remove the capability to continue a safe flight and landing without demanding more than usual skill on the part of the flight crew.
Extremely remote failures	Hazardous effects	These effects may be more than major providing that the overall risk of catastrophe is extremely improbable, taking into account likely crew action.
Extremely improbable failures	Catastrophic effects	Resulting in fatalities.

Recurrent. (Frequency of occurrence up to about 10^{-5} per hour of flight.) Expected to occur from time to time in the life of an airplane.

Remote. (Of the order of 10^{-5} to 10^{-7} per hour of flight.) May happen a few times during the total operational life of a type of aircraft. For example, a remote failure includes failure of two engines in one flight.

Extremely Remote. (Not expected to occur more often than 10^{-7} per hour of flight.) Unlikely to occur during the total operational life of all aircraft of a type, but nevertheless has to be considered as being possible.

Extremely Improbable. So extremely remote that it can be stated with confidence that it should not occur.

Table 2.- Flying qualities levels from MIL-F-8785B

Level	Description	Corresponding pilot rating (in general)
1	Flying qualities clearly adequate for the mission flight phase	1 - 3.5
2	Flying qualities adequate to accomplish the mission flight phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists	3.5 - 6.5
3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A flight phases can be terminated safely, and Category B and C flight phases can be completed.	6.5 - 9+
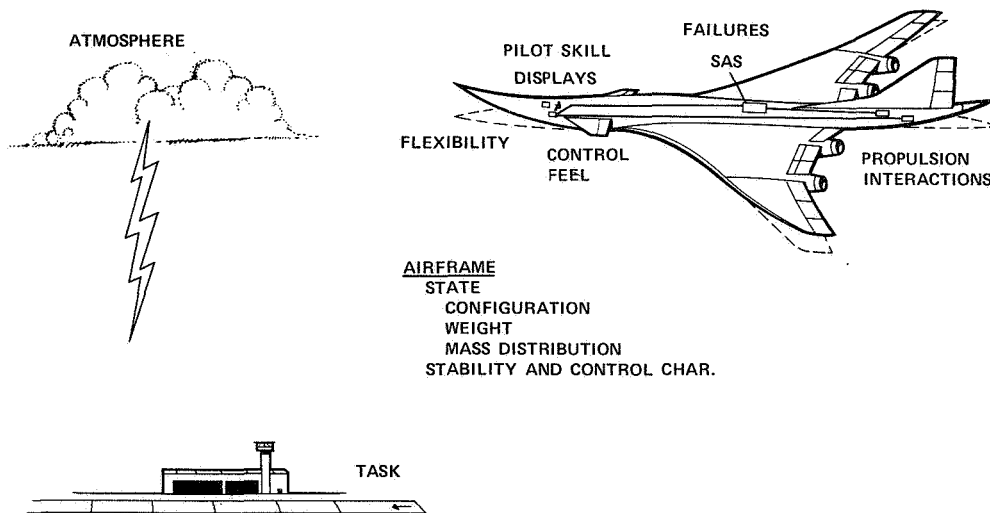
Figure 1.- Factors influencing flying qualities

FAR 25 AIRWORTHINESS STANDARDS: TRANSPORT CATEGORY AIRPLANES (ref. 2)

SUBPARTS

A. GENERAL

B. FLIGHT

    GENERAL

    PERFORMANCE: RECIP. ENGINE POWERED AIRPLANES

    PERFORMANCE: TURBINE ENGINE POWERED AIRPLANES

    CONTROLLABILITY AND MANEUVERABILITY   ◄—

    TRIM   ◄—

    STABILITY   ◄—

    STALLS   ◄—

    GROUND AND WATER HANDLING CHARACTERISTICS   ◄—

    MISCELLANEOUS FLIGHT REQUIREMENTS   ◄—

C. STRUCTURE

D. DESIGN AND CONSTRUCTION   ◄···

E. POWERPLANT

F. EQUIPMENT   ◄···

G. OPERATING LIMITATIONS AND INFORMATION

Figure 2.- FAR 25 outline identifying sections of interest.

SUBPART B   FLIGHT

25.181 DYNAMIC LONGITUDINAL, DIRECTIONAL,
AND LATERAL STABILITY.
    ANY SHORT PERIOD OSCILLATION OCCURRING
BETWEEN STALLING SPEED AND MAXIMUM
ALLOWABLE SPEED APPROPRIATE TO THE
CONFIGURATION . . . MUST BE HEAVILY DAMPED
WITH PRIMARY CONTROLS (1) FREE AND
(2) IN A FIXED POSITION.

SUBPART B   FLIGHT

25.21(e)
    IF COMPLIANCE WITH THE FLIGHT
CHARACTERISTICS REQUIREMENTS IS
DEPENDENT UPON A STABILITY
AUGMENTATION SYSTEM OR UPON ANY
OTHER AUTOMATIC OR POWER-
OPERATED SYSTEM, COMPLIANCE MUST
BE SHOWN WITH PARAGRAPHS 25.671
AND 25.672.

SUBPART D   DESIGN AND CONSTRUCTION

25.671 CONTROL SYSTEMS — GENERAL

(c) THE AIRPLANE MUST BE SHOWN BY ANALYSIS, TESTS, OR
    BOTH TO BE CAPABLE OF CONTINUED SAFE FLIGHT AND
    LANDING AFTER ANY OF THE FOLLOWING FAILURES OR
    JAMMING IN THE FLIGHT CONTROL SYSTEM AND SURFACES
    . . . , WITHIN THE NORMAL FLIGHT ENVELOPE, WITHOUT
    REQUIRING EXCEPTIONAL PILOTING SKILL OR STRENGTH.
    PROBABLE MALFUNCTIONS MUST HAVE ONLY MINOR EFFECTS
    ON CONTROL SYSTEM OPERATION AND MUST BE CAPABLE OF
    BEING READILY COUNTERACTED BY THE PILOT.
    (1) ANY SINGLE FAILURE, EXCLUDING JAMMING.
    (2) ANY COMBINATION OF FAILURES NOT SHOWN TO BE
        EXTREMELY IMPROBABLE, EXCLUDING JAMMING
    (3) ANY JAM IN A CONTROL POSITION NORMALLY
        ENCOUNTERED . . . UNLESS THE JAM IS SHOWN TO
        BE EXTREMELY IMPROBABLE OR CAN BE ALLEVIATED . . .

25.672 STABILITY AUGMENTATION AND AUTOMATIC AND
        POWER-OPERATED SYSTEMS

(b) THE DESIGN . . . MUST PERMIT INITIAL COUNTERACTION
    OF FAILURES OF THE TYPE SPECIFIED IN PARAGRAPH 25.671(c)
    WITHOUT REQUIRING EXCEPTIONAL PILOT SKILL OR STRENGTH,
    BY EITHER DEACTIVATION OF THE SYSTEM, OR A FAILED
    PORTION THEREOF, OR BY OVERRIDING THE FAILURE BY
    MOVEMENT OF THE FLIGHT CONTROLS IN THE NORMAL SENSE.

(c) IT MUST BE SHOWN THAT AFTER ANY SINGLE FAILURE . . .
    (1) THE AIRPLANE IS SAFELY CONTROLLABLE WHEN THE
        FAILURE OR MALFUNCTION OCCURS AT ANY SPEED OR
        ALTITUDE WITHIN THE APPROVED OPERATING LIMITATIONS
        THAT ARE CRITICAL FOR THE TYPE OF FAILURE BEING
        CONSIDERED;
    (2) THE CONTROLLABILITY AND MANEUVERABILITY REQUIRE-
        MENTS OF THIS PART ARE MET WITHIN A PRACTICAL
        OPERATIONAL FLIGHT ENVELOPE . . . DESCRIBED IN THE
        AIRPLANE FLIGHT MANUAL; AND
    (3) THE TRIM, STABILITY, AND STALL CHARACTERISTICS ARE
        NOT IMPAIRED BELOW A LEVEL NEEDED TO PERMIT CONTINUED
        SAFE FLIGHT AND LANDING.

Figure 3.- Example of control system failure treatment in FAR 25.

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Figure 4.- Elements defining the "task" in TSS 3 (refs. 1 & 16).

Figure 5.- One possible method described in TSS 3 (ref. 1) for showing compliance with handling requirement.

HANDLING QUALITIES RATING SCALE

Figure 6.- Cooper-Harper pilot rating scale (ref. 10).

Figure 7.- MIL-F-8785B minimum flying qualities requirements.

Figure 8.- MIL-F-8785B procedure for determining theoretical compliance
with airplane failure state requirements.

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Figure 9.- Progressive simulation and testing in Autoland certification (ref. 13).
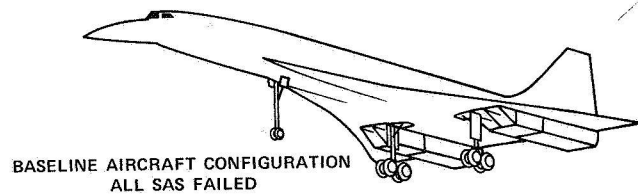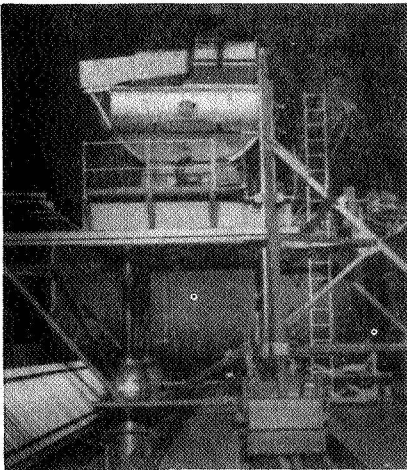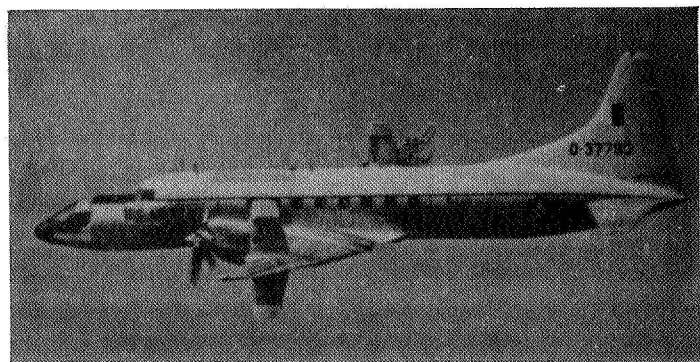
Figure 10.- Simulation study of minimum longitudinal stability for SAS-failed landing.
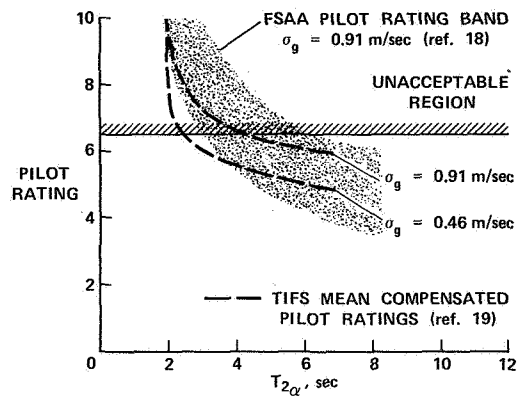
Figure 11.— Pilot rating vs time to double amplitude $T_{2\alpha}$, showing effect of turbulence intensity (σ_g is rms value). Landing task, unaugmented delta-wing transport.
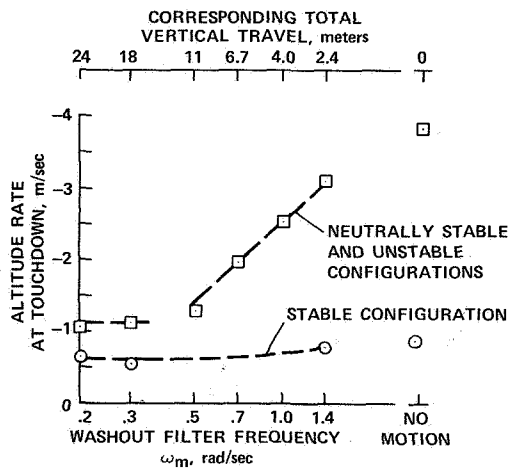
Figure 12.— Vertical motion effects on landing performance (ref. 20).

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HANDLING QUALITIES REQUIREMENTS FOR

CONTROL CONFIGURED VEHICLES

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SUMMARY

The rapid emergence of fly-by-wire and control-configured vehicle concepts challenges us to account adequately for their potential effects on flying qualities. Failure mode probabilities and consequences must be considered. Adequate controllability must be provided for aerodynamically unstable aircraft at extreme flight conditions. Nonclassical overall dynamics of highly augmented aircraft create the need for new approaches to specifying design criteria. New control modes such as direct force require definition of boundaries for usefulness as well as desirability. These considerations are being incorporated in the continuing effort at the AF. Flight Dynamics Laboratory to review and revise the formal military flying qualities requirements. This paper will review the rationale and present current results addressing the above considerations with regard to Military Specification MIL-F-8785B, "Flying Qualities of Piloted Airplanes".

INTRODUCTION

Recently we were asked to clear for flight testing an airplane which, without added ballast, was predicted to be somewhat unstable if the stability augmentation system (SAS) should fail. Considering the expected degree of inaccuracy in aerodynamic and reliability predictions, we recommended putting the center of gravity somewhat forward of the SAS-off maneuver point--where stick force and deflection per g go to zero. Contrary to the MIL-F-8785B requirement, we did not feel compelled to insist on a c.g. location that would assure static speed stability.

"What?", our Laboratory Deputy Director asked. "Here we've put so much of our resources into developing control-configured vehicles to tolerate relaxed static stability, and now you tell me all that refinement isn't necessary--you say a plain unaugmented airplane can fly that way safely. Have we wasted all that time and money?"

Well, there is more to CCV than that in several dimensions, including the degree of allowable bare-airframe instability. But he had made a valid point one that has bothered some of us all along. We know through observation that pilots can control a moderately unstable vehicle in the right circumstances. Haven't helicopters been flying for a long time--and unstable airplanes too! Quoting Amos Root's observations of the Wright brothers' experiments at the

Huffman Prairie in the summer of 1904[1],

>"When I first saw the apparatus it persisted in going up and
>down like the waves of the sea. Sometimes it would dig its
>nose into the dirt, almost in spite of the engineer. After
>repeated experiments it was finally cured of its foolish
>tricks, and was made to go like a steady old horse. This
>work, mind you, was all new. Nobody living could give them
>any advice. It was like exploring a new and unknown domain.
>Shall I tell you how they cured it of bobbing up and down?
>Simply by loading its nose or front steering-apparatus with
>cast iron. In my ignorance I thought the engine was not large
>enough; but when fifty pounds of iron was fastened to its 'nose'
>(as I will persist in calling it), it came down a tolerably
>straight line and carried the burden with ease. There was a
>reason for this that I cannot explain here... Over one hundred
>flights have been made during the past summer. Some of them
>reached perhaps 50 or 60 feet above the ground. On both these
>long trips seventy pounds instead of fifty of cast iron was
>carried on the 'nose'."

Or read Maj. Gen. Benjamin D. Foulois' account[2] of his experience at Ft. Sam
Houston in 1910 as the U.S. Army's airplane pilot:

>"We wanted to develop the airplane into a stable platform
>for air reconnaissance work. Old Number One was the last of
>the Kitty Hawk models, and with its two elevators out in
>front it was about as stable as a bucking bronco. We con-
>tinued experimenting there while the Wright brothers made
>modifications back at Dayton, Ohio. When one of the elevators
>up front was moved around to the back, stability improved
>somewhat but not enough. I later found out that by using just
>one elevator, the rear one, I had a platform that worked very
>well. I could let go of the levers and make notes and
>sketches. It got to be an airplane that could be used for
>real military reconnaissance."

Charles Gibbs-Smith writes[3]

>"So when the Wrights built their first glider in 1900 it
>incorporated two ideas which the brothers were to utilise
>throughout their early work--the intentionally unstable aero-
>plane which could be kept flying satisfactorily only by the
>pilot's skill, and the warping of the wings for control in
>roll. 'We therefore resolved', wrote Wilbur, 'to try a
>fundamentally different principle. We would arrange the
>machine so that it would not tend to right itself.'"

This was truly instability, as we have seen from the preceding accounts. It
was exactly that concept of instability--to a manageable degree--that led
Lilienthal, Chanute and the Wrights to succeed where the "chauffeurs" of highly

stable airplanes could not achieve controlled flight. But the early fliers
had a rather high accident rate which must be attributed in part to the
vehicle's instability. Our tolerance today may be less, even for emergencies,
considering the higher speeds and poor weather to which our flying now is
subject.

Even the Wrights soon recognized the need for improvement. In addition to
ballasting for a forward c.g. and moving the canard surface to the tail in
order to move the neutral stability point aft, they also investigated automa-
tic means[4]. It is interesting to note that their Patent No. 2913 for automa-
tic stabilization preceded Gen. Foulois' rearranging the control surfaces with
the Wrights' help. For pitch, "a pivoted vane acting under the influence of
wind pressure" sensed angle of attack to control a supply of compressed air
which actuated the elevator. A pendulum was specified "for lateral control".
Operation of these devices would not move the pilot's control levers. In 1914
Orville Wright was awarded the Collier Trophy for his work on automatic
stabilization.

<div align="center">BACKGROUND OF CURRENT ACTIVITY</div>

Why, then, have we been less willing in recent times to accept instability,
even for emergencies? A number of reasons, each with some degree of validity,
have led to this conservatism:

> Until recent times the failure rates of stability augmenta-
> tion equipment gave the expectation of frequently experiencing
> the basic-airframe characteristics. Greater redundancy was
> not attractive because of the increased cost and the maintenance
> burden to keep it all operating.

> Little is yet known about the cumulative effects of several
> poor flying qualities together, except that an aircraft that is
> safe with any one "unacceptable" quality can become unflyable
> with some combinations of these characteristics. Further, a
> number of plausible single and multiple failures can degrade
> several handling qualities. Loss of just the pitch axis of
> augmentation, for example, could degrade damping, frequency,
> maneuvering force gradients, friction and backlash. A pilot-
> induced-oscillation could not be stopped by clamping the
> control stick if $d\,\delta_e/d\,n_z$ is unstable.

> Viable designs have generally been possible with basically
> stable airframes--at least for conventional airplanes.

> Little experience has been obtained to define instability
> boundaries suitable for the speeds, tasks, and weather that
> are now commonly encountered in operating aircraft.

From the data collected for MIL-F-8785B[5] the tolerable amount of instability is a function of total damping; the data are insufficient, however, to draw a valid requirement. "After studying the available data, it is obvious that many factors influence the amount of instability which can be handled. Because even a small instability can be quite dangerous under some circumstances, it was decided to require the airplane to be statically stable even for Level 3."

We need to reexamine these conservative requirements in order to provide more guidance on the circumstances and amounts in which instability is safe. We solicit the opinions of those present.

MIL-F-8785B AND CCV'S

In developing MIL-F-8785B we gave much thought to the conditions for allowing degraded flying qualities. We wanted to account as much as possible for real-world problems without overly complicating the requirements. Two causes of degradation were considered. Flying qualities giving less performance or requiring more pilot attention are allowed outside the military-specified Operational Flight Envelopes. This allows some capability for adapting to changes in mission without unduly penalizing a design for having a larger flight envelope than required. After relatively infrequent failures (nominally once per hundred flights) this same level of degradation, Level 2, is allowed in the Operational Flight Envelope, and further degradation is allowed outside those boundaries. Only rarely (once in 10,000 flights) is degradation beyond Level 2 allowed in the Operational Flight Envelope. In any case Level 3 is a relatively safe floor. Degradation beyond Level 3 requires special consideration on a case-by-case basis, thus in principle giving the procuring activity the power of decision. The Special Failure States which are subject to this approval are of several categories. In some cases other specifications or design practices give acceptable assurance: the basic aircraft structure is a common reliability standard. In other cases judgment must be used to establish a point of diminishing returns: two, or three or four hydraulic systems are used to power essential flight controls, for example. There also will be cases in which failure is expected to be extremely remote in probability, but the cost of a change or addition to preclude the failure or limit its effect is small enough to warrant disapproval of a Special Failure State. In still other cases approval may be granted if special design or test requirements are met.

Despite an occasional opinion to the contrary, MIL-F-8785B does apply to CCV's - as far as the specification goes. Although the 8785B treatment of response to atmospheric disturbances is weak in general, clearly the requirements and the Level structure apply to conventional stability and control augmentation. The Special Failure States provide a mechanism "to assure that the flight safety, flying qualities and reliability aspects of dependence on stability augmentation and other forms of system complication will be considered fully". The limitations for CCV application are a lack of requirements

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on direct force control, and the expression of many requirements in terms of classical modal parameters. I would like to evoke discussion of these matters now, at this meeting.

Thrust/speed brake requirements were considered but omitted as beyond the scope of the specification. We are having second thoughts on that now, and will try to arrange with the propulsion people for adequate coverage somehow between the two disciplines. A lack of experience with direct lift or side-force controllers still precludes definitive requirements for those control modes—despite the Japanese' successful use of an automatic maneuvering flap in air combat in 1943; on the outstanding Kawanishi Shiden (George) fighter.[6]

THE FORM OF DYNAMIC REQUIREMENTS

Reference to short-period, dutch-roll, etc. modes is not as much a hindrance to CCV application as one might first suspect. The idea, of course, is to state the requirements in a form we are familiar with, in terms consistent with the aircraft characteristics that form the data base. Conventional stability augmentation modifies the parameters but not the form of the response. Recent flight control system designs, however, show a tendency to introduce additional dynamic modes at frequencies on the order of the aircraft response frequencies, giving rise to overall motions unlike the conventional response. A'Harrah[7], for one, has pointed out the difficulty in associating short-period requirements with a particular pair of poles on a root locus. Nevertheless it is often possible to find an equivalent classical aircraft which matches the response of a more complicated dynamic system reasonably well over a suitable time period or frequency range. Then it should be valid to compare those equivalent parameters with modal requirements. We realize the need for a more generally applicable alternative and hope to do better, at least with longitudinal requirements, in our current revision effort.

Alternative longitudinal requirements are being investigated which should be more generally applicable, but at first these will seem to be of less direct use to the airframe designer. One possibility is Neal and Smith's[8] closed-loop criteria which utilize pilot-vehicle analysis with a specified pilot describing function and parameter adjustment rules. Other possibilities, semi-empirical in origin, involve properties of the open-loop Bode phase angle vs frequency curve. Ideally a requirement should apply to all of:

> The complete airplane attitude response including all pertinent modes (e.g., both phugoid and short period)

> The airplane plus flight control system (i.e., including lags and time delays)

> The various control element forms resulting from current flight control augmentation concepts

The basic inner attitude response features which are
necessary regardless of outer-loop control problems
or auxiliary control (e.g., direct lift)

Variations in pilot control technique (e.g., closed-
loop bandwidth) with control task or flight phase.

(adapted from Ref. 9).

CURRENT ACTIVITY REGARDING LONGITUDINAL REQUIREMENTS

We are also examining "envelope" criteria in the time and frequency
domains--for example Malcom and Tobie's C*[10] and the McDonnell Aircraft re-
finement.[11] C* is a rational parameter to investigate and the envelopes
facilitate design. While the specific criteria which have been developed
may work for the particular configurations investigated, they seem to lack
validity in general application. The refined C* and Ċ* criteria do not seem
to match the time-history ratings of Ref. 8 Vol. II much better than the origi-
nal C* criteria do. However, as reference 12 points out, it is not realistic
to expect any single criterion to encompass all potential faults, especially
for high-order or multi-mode systems.

Since the publication of MIL-F-8785B in 1969, a number of research con-
tracts have been sponsored by the AF Flight Dynamics Lab both to generate
data and to develop new requirements that encompass new technology. Among
the proposed requirements currently being reviewed are the Calspan proposed
longitudinal maneuvering criteria in reference 13. Longitudinal attitude and
normal acceleration control is related to frequency response characteristics,
considering desirable pilot compensation needs. An attractive feature is
elimination of the need to identify short period frequency and damping - a
real advantage for highly augmented airplanes. However, measurement of a
slope and phase angle from the pitch frequency response amplitude versus phase
angle plot is required. This does necessitate knowledge of the aircraft/
flight control system longitudinal frequency response function. The practica-
lity of identification with currently available computer algorithms and flight
test data commonly recorded is being evaluated. Also being investigated is
the practicality of generating an equivalent transfer function which would
allow presentation of requirements in terms of "equivalent" parameters or,
perhaps, required pilot compensation parameters. This concept of incorporating
pilot workload and transfer functions relates requirements more directly to the
designer; but the difficulty of accurately fitting an arbitrary frequency
response curve with a specified transfer function form is significant.

A recent experimental program[14] studied the task dependence of require-
ments such as those described above. Using the AF variable stability T-33,
variations in pilot rating were shown for some high-order configurations
as a function of evaluation task. The configurations most affected by task
variation all exhibited relatively high dominant natural frequencies. In
evaluating the Calspan proposed requirements, Mayhew[15] illustrated the
influence of closed-loop bandwidth on Neal and Smith's flying qualities
parameters. He has also shown the relationship between the proposed

requirements and the current familiar short period criteria. While the Calspan proposal includes some provision for bandwidth variation, further evaluation will determine if additional provision is required.

A different approach to flying qualities criteria, amenable to use in the design phase, is based on the "paper pilot" concept first proposed by Anderson[16]. Reference 12 developed a computerized method of handling qualities analysis based on this idea which showed relatively good correlation for "conventional" airplanes - but less successful for designs representative of CCV technology. However, this result is not conclusive because the empirical nature of the criteria involved require a good data base for validation. Such a basis does not exist for CCV airplanes. Hence, the general approach does warrant further study for future application.

CURRENT ACTIVITY REGARDING LATERAL-DIRECTIONAL REQUIREMENTS

The present lateral-directional dynamic requirements are intended to minimize undesirable yaw due to roll, and dutch-roll excitation. These goals may be satisfied by the basic airplane design or by incorporating augmentation (with proper attention to reliability). Consequently, these requirements are consistent to a high degree with CCV design approaches. Specification of response characteristics such as p_{osc}/p_{av} is consistent with the philosophy being explored for the longitudinal requirements, though modal items are not.

Reference 9 has proposed a new requirement for heading control which is intended to address the problem of adverse yaw more directly. The approach is to evaluate the roll-yaw control coordination required in a turn against a desirable standard for a coordinated turn. Obviously this criterion could be applied to design of a CCV system as well as evaluation of conventional airplanes. General applicability of this criterion (or some variation thereof) to CCV designs incorporating different control modes to achieve heading control remains to be investigated, although reference 9 indicates the criterion is insensitive to airplane class or type. Also, as with the proposed longitudinal requirements, the practicality of measuring or identifying the response characteristics needed remains to be established.

Direct side force control is frequently mentioned in conjunction with CCV and as noted previously is an area where definitive flying qualities data are scarce. Before such data can be generated, a complete understanding of the way pilots employ direct side force in various tasks (Flight Phases) must be developed. For example, they may in some cases employ side force to perform either a flat turn or side slip in tracking. Another application could be to trim out a crosswind effect. Also, the effect of interaction with other controls and with other subsystems such as displays must be explored. Recent efforts at AFFDL have looked at the weapon delivery task[17] and STOL landing[18]. Additional work currently underway will hopefully bring us to the point of developing some new requirements.

Display interaction and cockpit controller characteristics in general require further study before definitive requirements can be developed to encompass some aspects of CCV technology. In some cases, it is simply a matter of generating data. For example, pilot rating and performance data are necessary to develop quantitative requirements on force levels and gradients (including nonlinearities) for sidesticks. Display interaction must be considered when evaluating flying qualities as a function of task and also as a function of control mode. For example, the evaluation of direct side force control for weapon delivery mentioned above considered only fixed gunsights. To complete the evaluation it will be necessary to consider the effect of active gunsights on the pilot's use of direct side foce.

LIMITING FACTORS

In concluding, then, we reiterate that in many respects the current flying qualities requirements are compatible with CCV technology. In some areas, new requirements or expansion of old ones is needed. In these areas, where new requirements are being formulated, we are certainly considering CCV and where necessary attempting to gather new data. The following basic flying qualities considerations, however, might be termed as limitations on the general application of CCV technology.

How much static instability can be tolerated safely? An absolute bound is apparent from "critical task" studies[19], which show that divergence of a simple system is controllable if its time to double amplitude is within certain bounds, depending upon pilot workload. Boeing SST simulations[20] found a criterion of $T_2 \geq 6$ sec to set the safe aft c.g. limit. The critical task has also been used as a side task in pilot-vehicle studies, the magnitude of the controllable unstable time constant being a measure of pilot workload[21]. The amount of divergence, then, which can be handled safely is seen to depend upon the amount of attention a pilot can devote to controlling it. That in turn is a function of the task's inherent difficulty (e.g., landing approach vs cruise) and the level of other flying qualities (e.g., concurrent failures of command augmentation or in another axis of stability augmentation)[9].

Another necessary limit on static instability is the amount of control remaining for recovery. Proposed criteria have ranged from little more than static balance[22] to MIL-F-83300's[23] half the nominal control moment (for forward flight) and specified attitude changes in 1 second (for hover). Sensitivity to gusts is a consideration. Any requirement is bound to be somewhat arbitrary because experience is limited. Here too we solicit opinions and data.

Control surface rate must also be adequate, even in emergency conditions. A 1972 General Dynamics study shows convincing time histories of the wild maneuvers that can result from insufficient surface rate for stability augmentation. In an internal study, Watson, Bennett and Kouri systematically varied the parameters of "a small CCV fighter airplane design", seeking generalized design criteria. That at least is a start toward a specification requirement.

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From considerations leading to the current requirements[5], we have the following discussion relative to instability and Failure States of the airplane.

The Level 3 requirements generally apply in the worst possible Failure States. Except for approved Special Failure States, then, MIL-F-8785's static stability requirement does not permit basic-airframe speed instability (elevator surface fixed). Cases will arise, however, in which the procuring activity is asked to consider allowing basic-airframe instability as a Special Failure State. Even if the reliability of stability augmentation should be judged sufficiently high, or if the degree of instability seems acceptable in itself, a number of aspects of combined airframe-flight control system behavior in normal operation need to be examined before accepting appreciable instability in a Special Failure State.

Obviously, extremes of either stability or instability require more control to balance the airplane throughout an angle-of-attack range. In the stable case, at the control limit the airplane at least has a restoring tendency. But when an airplane has an unstable variation of elevator-surface position with airspeed, the surface position required to maintain off-trim airspeeds is in a direction which reduces the control available to initiate recovery to the trim speed. If the unstable gradient is large enough, the pilot could fly far enough off the trim speed that there would be no elevator control available for recovery. With the elevator against the stops, the airspeed would continue to diverge and the pilot would be powerless to prevent it from doing so. Examples of this behavior can be found in Mach tuck for subsonic airplanes and during wave-offs for some propeller-driven airplanes.

For Airplane Normal States, then, over the entire permissible range of speed and altitude, safety comparable to that of a stable basic airframe would require pilot-control and control-surface authority to balance the airplane at positive and negative ultimate load factors, with some margin of control power remaining, wherever the basic airframe is unstable. (In flight test, of course, limit load factor would not intentionally be exceeded.) For a given configuration, the elevator surface and control positions for balance determine the amount of control authority left for stabilization and control. The relative authority and interactions of command, augmentation and trim controls are important considerations. Authority and rate saturation may be particularly important for dual-purpose controls such as elevons. With aerodynamic instability and higher-order flight control system dynamics, limit cycles also become of increasing concern.

In both Normal and Failure States, the augmentation must maintain appropriate levels of stability in responses to both control and disturbance inputs. For a basically unstable airframe, the sizes of these inputs should be stated specifically, rather than taking a primarily qualitative approach. Some margin above structural design gusts and turbulence might be suitable. The required augmentation authority may exceed the pilot's control authority. Hard-over failures should be made impossible in the flight control system; engine-failure transients conceivably could be critical. Large control inputs of various forms and phasing should be considered. The response to disturbances

during commanded maneuvers must be considered. The effect of flight at off-trim conditions on all these factors must be examined.

Particular attention is needed for the stall and spin recovery requirements. Increased dependence on control systems and artificial stability makes survivability after damage or failure an important consideration for high-angle-of-attack flight.

Stall limiters and departure preventers are already developed as fixes for current fighter airplanes—the F-111 Stall Inhibitor System[24] and the A-7 departure preventer[25], for example. Manufacturers whose aircraft do not need such devices expound on the air combat advantage attainable at extreme angles of attack: rapid deceleration, for example, to change positions with an enemy attacking from the rear. Certainly aerodynamic design for stall/post-stall stability remains an important consideration for CCV design, in order to avoid completely uncontrollable situations. The Air Force Flight Test Center's Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes, MIL-S-83691A, rightfully stresses the need to demonstrate extreme resistance to loss of control. The required testing subjects all aircraft to a degree of "gross" abuse beyond normal maneuvers. Highly maneuverable aircraft are to be even more completely wrung out. Thus limiters, while certainly useful, can supplement but not replace aerodynamic design at high angle of attack.

In determining the adequacy of stall limiters, control authority and rate, one must choose the size of disturbance to be allowed for. Turbulence level is important; both MIL-F-8785B and the proposed MIL-F-9490D flight control system specification give models and intensities for turbulence up to thunderstorm intensities. Single disturbances are likely to be critical. These include gusts, wind shear, wakes of buildings, etc. near the runway and jet wakes. The British revisers of AvP 970 flying qualities requirements are considering, in addition to Gaussian turbulence, pairs of ramp gusts to evoke the worst response. Glyn Jones' development of this approach is proceeding.[26]

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THE F-12 SERIES AIRCRAFT APPROACH TO DESIGN FOR

CONTROL SYSTEM RELIABILITY

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SUMMARY

This paper presents a review of the F-12 series aircraft control system design philosophy as it pertains to functional reliability. The basic control system, i.e., cables, mixer, feel system, trim devices, and hydraulic systems are described and discussed. In addition, the implementation of the redundant stability augmentation system in the F-12 type aircraft is described. Finally, the functional reliability record that has been achieved is presented.

INTRODUCTION

The F-12 series aircraft were designed more than a decade ago, yet they included concepts which have only recently become popular and even acceptable. One of these is the fact that, to a certain extent, the F-12 aircraft are control configured vehicles (CCV). They were designed with the objective of minimizing trim drag to enhance the range capabilities. This, of course, immediately implies either very low or no static stability requiring the full time services of a pitch stability augmentation system (SAS). At high Mach numbers, the Mach effects reduce the directional stability. Since an engine failure or inlet unstart can produce a violent transient, it is rather obvious that the services of a full time yaw stability augmentation system is also important, both from the standpoint of pilot comfort and prevention of structural damage to the aircraft. These factors dictate a full time stability augmentation system in both the pitch and yaw axes and with a functional reliability comparable with that of the basic vehicle itself. This paper presents descriptions of the basic aircraft control system and the redundant stability augmentation systems that permitted us to achieve the necessary functional reliability.

MANUAL CONTROL SYSTEM

The configuration of the F-12 series aircraft is illustrated in Figure 1. The shaded areas show the hydraulically actuated aerodynamic control surfaces. The large inboard and outboard elevons are utilized for pitch and roll control. Pilot control stick motion is separated into pitch and roll commands by the elevon mixer assembly located in the aircraft's tail cone. The outboard elevon is slaved to the inboard elevon through a crossover linkage system which transmits commands across the hot aft nacelle. The crossover linkage contains a preloaded spring cartridge to avoid structural damage should the outboard surface jam. The rudders are really all-movable vertical tails to provide the necessary controllability during engine failure or inlet unstart.

The aircraft contains four hydraulic supply systems; two of these are dedicated to the control system. The two control system hydraulic supplies are designated as System A and System B. The elevon surface actuators are arranged such that alternating cylinders are supplied by Systems A and B Thus, if either hydraulic system is lost, the remaining system will continue to provide power for the actuation of all of the surfaces. The verticals have a similar load sharing arrangement.

Since the A and B valves porting hydraulic oil to the surface actuators are on a common shaft and in close proximity to each other, it is necessary to protect against intersystem leakage in the event of the loss of one of the hydraulic supplies. This is done by providing "scavenger" jet pumps in the return area for both systems. This results in the return of any leakage oil back to the reservoir of the appropriate supply instead of loss into the failed supply.

The roll/pitch elevon mixer is a relatively simple device containing the roll and pitch feel springs and trim actuators. No complexities such as bobweights or q-bellows are employed, and as a result, has proven to be quite reliable. The feel springs for the verticals are located in the stub fin and are incorporated into the yaw trim actuators.

Transmission of pilot stick commands to the elevon control surfaces is achieved by dual cable systems to the mixer and from thence to the summing levers of the inboard servo elevon valves. The rudder pedal motion is also transmitted via a dual cable-pushrod system to the summing levers of the vertical servos. The variations in required cable length due to temperature effects and flexure of the relatively long fuselage is compensated for by the use of tension regulators.

Electrical power is provided by two identical generators, each driven by one of the engines. The generators are synchronized and in normal operation share the load. If either generator fails, an automatic relay system disconnects the failed system transferring the total load to the remaining generator. If both engines quit, causing loss of both generators, a battery/inverter supplies power to the essential bus until the engines are restarted.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The design goal of the automatic flight control system for the F-12 series aircraft was to provide optimum handling qualities in the primary flight regimes of the aircraft. However, another consideration was to provide as simple a system as possible in order to enhance reliability. Since the vehicle was the first supersonic cruise vehicle, and thus would spend the greater portion of its flight time at high Mach number cruise, the handling qualities had to be optimum at these conditions. In addition, it was also imperative to provide good response and controllability in the critical areas of the flight envelope consisting of takeoff, landing and refueling. All other flight conditions were considered transitional where handling qualities could be less than optimum in the interest of simplicity.

The automatic flight control system of the F-12 series aircraft consists of the stability augmentation system (SAS), the autopilot and the Mach trim system. The autopilot is primarily to provide pilot relief modes, and although high reliability for this function is desirable, it is not essential to safety of flight. Thus, the only protective measures taken in the implementation of the autopilot is the provision of duplex fixed authority limits set to prevent excessive transients for hardover failures. The pilot can also disengage the autopilot by depressing a trigger switch on the control stick.

The Mach trim system is also not a safety of flight parameter. Its function is to provide speed stability in the subsonic and low supersonic speed regime during manual flight. Loss of this function, however, requires increased pilot attention and workload in maintaining airspeed.

To protect against runaway trim failures, a trim power switch is located directly ahead of the pilot's left knee for easy access. This is necessary for two reasons; loss of pilot mobility due to the pressure suit and the multiplicity of circuit breakers. This switch cuts power to all trim systems before a runaway trim can cause the requirement of excessive forces to hold the aircraft in trim. Once the runaway condition is stopped, the pilot can locate and pull the proper circuit breakers and then reengage the trim power switch to restore power to the unfailed systems.

STABILITY AUGMENTATION SYSTEM (SAS)

As was stated earlier, the F-12 series aircraft have very low pitch static stability and yaw directional stability at design flight conditions. This requires a greater dependence on stability augmentation during maneuvers and during engine-out transients. This, of course, dictates significant percentages of full manual authority. The comparative authorities are shown in Table I. The magnitude of these authorities is such that pitch or yaw hardover failures could be catastrophic at certain flight conditions. This, combined with the fact that the pitch and yaw SAS functions are essential to safety of flight, dictates they be implemented with a functional reliability comparable to that of the basic aircraft or that of a fly-by-wire system.

SAS REDUNDANCY AND LOGIC

Because of the importance of the yaw and pitch SAS's, they are implemented with triple-redundancy in sensors, electronics and gain scheduling. The roll SAS is not critical, both from the standpoint of handling qualities and transients due to hardover failures. However, the roll SAS is the inner loop for all of the lateral autopilot modes. Thus, to ensure the desired pilot relief and comfort, the roll SAS has a dual mechanization.

The servos for the pitch axis are two dual tandem series servos, each dual servo driving an inboard elevon. The tandem pair are coupled to each other by a stiff spring such that both servos will track even if one is disengaged. If either servo were to jam, the other will still perform its function by distorting the spring. This does mean that if the "downstream" servo of a tandem pair were jammed, the pitch SAS function would only appear on the other elevon resulting in half gain and coupling into roll. However, great care is exercised in providing adequate filtering of the hydraulic fluid and in addition all main metering spool valves are designed to shear any metal chips that might get by. Thus, the probability of jamming is minimal. The yaw axis employs four series servos, whiffle tree summed in pairs, with each pair driving a separate vertical. The roll SAS uses two series servos, one for each inboard elevon.

The gain scheduling is obtained from triple-redundant differential pressure sensors and altitude switches. These are not part of the Central Air Data Computer, and comprise an entirely separate but simple sensing package. Because of the high reliance placed on the pitch SAS to provide static stability, an additional backup pitch damper (BUPD) is mechanized.

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This consists of a separate pitch rate gyro and electronics located in a controlled environment that can be switched into either the A or B servos. This system has a fixed gain and is to be used only below 50,000 feet and at subsonic speeds. To date, there is no record of the BUPD ever having been used. The purpose of the BUPD was only to provide adequate handling qualities for refueling and landing in the event that the basic pitch SAS failed due to overheating of the normal pitch gyros.

Simple block diagrams of the pitch, yaw and roll SAS mechanization are shown in Figures 2, 3 and 4. It is seen that the triplex systems shown for the pitch and yaw axes employ a monitor channel whose only function is to provide a reference for voting. The interceptor version was modified in that all three channels of both the yaw and pitch SAS are active contributing one-third of the total command. This is illustrated in Figure 5 showing the yaw SAS. In that configuration, when the voting logic removes a failed channel, the gain of the remaining two channels is increased by a factor of 1.5. Override provides full control gain from a single channel. On the surface, it would appear that the availability of the additional functional channel would enhance the overall reliability. However, the additional mechanization complexity tends to offset the reliability advantage.

The sensor and electronic circuits of the yaw and pitch SAS utilize triple redundancy in such a manner that a single failure is fail-operational with no change in system performance. This is achieved by a voting scheme which selects the "disagreeing" channel and disengages it as shown in Figures 2 and 3. A second or third failure depending on failure sequence results in total disengagement of that axis. The use of tandem servos in the pitch axis eliminates the need to double the gain in the remaining operational channel in order to maintain full system performance. However, the yaw axis electronic gain in the remaining operational channel is automatically doubled to maintain performance because of the whiffle tree summing mechanization of the series servos.

Only two channels, A and B, are functional; the M channel is used as a reference model. After total disengagement of an axis, if either the A or B channels are still functional, the pilot can exercise a logic override switch and obtain single channel performance.

The servos in both the yaw and pitch channels are essentially quadruple, but with dual hydraulic supplies. The A hydraulic supply powers a right and a left servo that are both being driven by the A electronics. The B supply powers the remaining two servos which are driven by the B electronics. The left and right servos for each hydraulic supply are compared and if they fail to track, that channel is immediately disengaged.

The remaining channel with its associated electronics then properly controls both the left and right surfaces with a gain equivalent to that of the complete system.

The failure monitoring logic is dual redundant, i.e., each comparison is independently duplicated. Since a system failure upstream of the servos produces two disagreements in the voting scheme, a single disagreement does not cause a channel disengage, but turns on the M channel warning light. A single indicated failure of servo logic will cause the related servo channel to disengage and turn on the associated warning light.

The roll SAS is mechanized as a simple dual system with one channel and servo for each side. A cross-monitor is employed in the servo feedback loops that disengages both channels in the event of disagreement. The disengagement is indicated by a failure light between the two channel switches. Disengaging and reengaging both switches recycles the failure logic to verify the failure. The pilot then exercises logic override by switching off both channels and manually engaging one channel at a time to test and select the operational channel. The gain of this channel is automatically doubled.

Certain types of servo position pickoff failures would result in limit cycle oscillations which would not be detected by the servo logic. Therefore, a separate monitor circuit is provided in each servo channel to detect open and short circuits in the pickoff primaries and secondaries.

In order for the pilot to evaluate his situation in the event of failure in the pitch and yaw SAS, a display of lights is presented to him on the Function Select Panel located on the right console as shown in Figure 6. If any of the lights are on, the pilot pushes the illuminated buttons to recycle the logic. Should this fail to reinstate the channels, the pilot can then assess his situation in pitch and yaw as shown in Table 2 on the assumption that the light indication represents the first failure. Subsequent failures use the same lighting sequence and as a result, the particular type failure cannot necessarily be isolated.

One of the major contributors to the maintenance of the F-12 flight control system reliability is the Mission Recording System (MRS). Each essential parameter of the various vehicle subsystems is monitored and properly signal conditioned for use in a magnetic tape recorder. The sampling rate for each parameter is once very three seconds. During the interval between samples, certain of the more significant parameters are monitored by peak-hold circuits which are reset when sampled. In the SAS each active element is monitored. This

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includes all sensors, gain scheduling devices, amplifiers, servos, and logic. This is then made use of in two ways. The first is obviously fault isolation; the second is the evaluation of logic performance during system checkout. For the latter, the pilot exercises the SAS logic prior to each flight and then again as soon as the flight is terminated. This is done by activating the logic checkout switch shown adjacent to the function selector panel in Figure 6. This initiates a preprogramed, built-in test sequence interrogating all SAS logic and AFCS disengage functions. Careful perusal of the resultant data tape then reveals the status of SAS system and the disengage logic. MRS utilization has also shown that it is possible to detect incipient failures. Although it is possible to achieve this through use of special software in the data processing, this has not been done. Thus, to date, this type of examination is performed visually by the data reduction technician.

RELIABILITY EXPERIENCE

The Honeywell Corp. was subcontracted to provide the automatic flight control system and the air data computer (ADC). The design requirements were established by Lockheed. Extremely close coordination and teamwork between Lockheed and Honeywell was maintained in order to meet the design goals for the system. How well these design goals were attained can be illustrated by the experience with the SAS. Honeywell designed and built the nation's first triple-redundant, fail-operational SAS for the F-12 series aircraft in the pitch and yaw axes. In the thousands of operational flight hours since the inception of the program, the pitch and yaw SAS has suffered only two functional failures. One was a maintenance error where incomplete installation of the rate gyro packages exposed the electrical connectors to high Mach ram air temperatures resulting in loss of the pitch axis. The second incident occurred when both the pitch A and B servos failed in the same flight. There were other instances where all three channels were simultaneously disengaged due to power transients during generator failures and subsequent switchover to the remaining generator. However, the channel disengage logic was immediately recycled and the system functioned normally. The one hardware failure during operational usage can be equated to a mean time between failure (MTBF) approaching 150,000 hours vs. a predicted MTBF of 19,000 hours. These numbers are based on total system operating ground and flight hours in an operational environment and exclude Category I and Category II flight testing since initial testing always involves some problem areas and system modification.

CONCLUSIONS

The functional reliability of the F-12 aircraft control systems has met and exceeded all expectations. This has been accomplished even though the aircraft and many of the control system components must operate in the most adverse sustained thermal environment experienced by any aircraft in the world. It must be noted, however, that the system design stressed reliability through simplicity. This resulted in minor compromise of handling qualities during what are considered transitional flight conditions. This would probably not be acceptable for commercial vehicles. Thus, for such applications, more elaborate scheduling and control laws would be required placing additional burdens on functional reliability. Although the F-12 flight control system was not specifically designed as a fly-by-wire system, it has demonstrated all the attributes that are required, and has provided a basis for the development of pilot acceptance of such systems.

Table I - Manual Versus SAS Authority

AXIS	MANUAL AUTHORITY		SAS AUTHORITY	PERCENTAGE MANUAL AUTHORITY	
	$M < 0.5$	$M > 0.5$		$M < 0.5$	$M > 0.5$
PITCH	$-24°, +11°$	$-24°, +11°$	$-2.5°, +6.5°$	10, 59	10, 59
ROLL	$\pm 24°$	$\pm 14°$	$\pm 4°$	17	29
YAW	$\pm 20°$	$\pm 10°$	$\pm 8°$	40	80

Table II - Failure Indications

LIGHTS - (YAW OR PITCH)	FAILURE
A AND M	A ELECTRONICS
B AND M	B ELECTRONICS
M	M ELECTRONICS
A	A SERVO
B	B SERVO
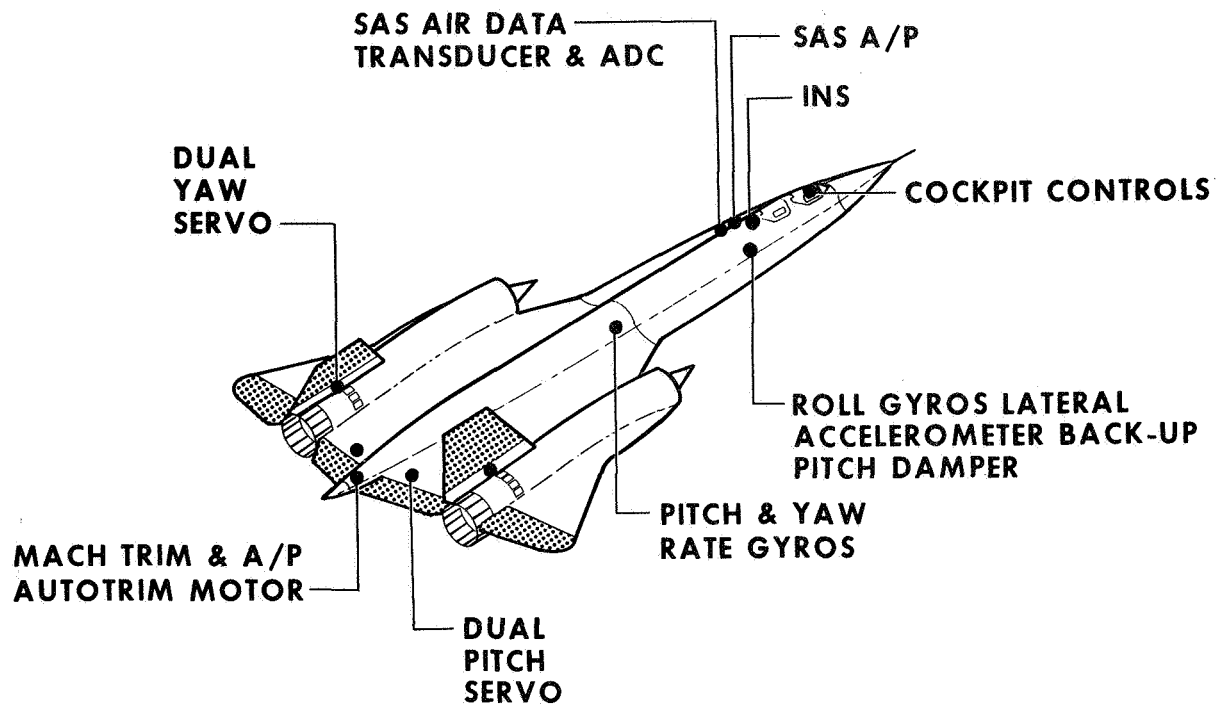
Figure 1 - AFCS Component Locations
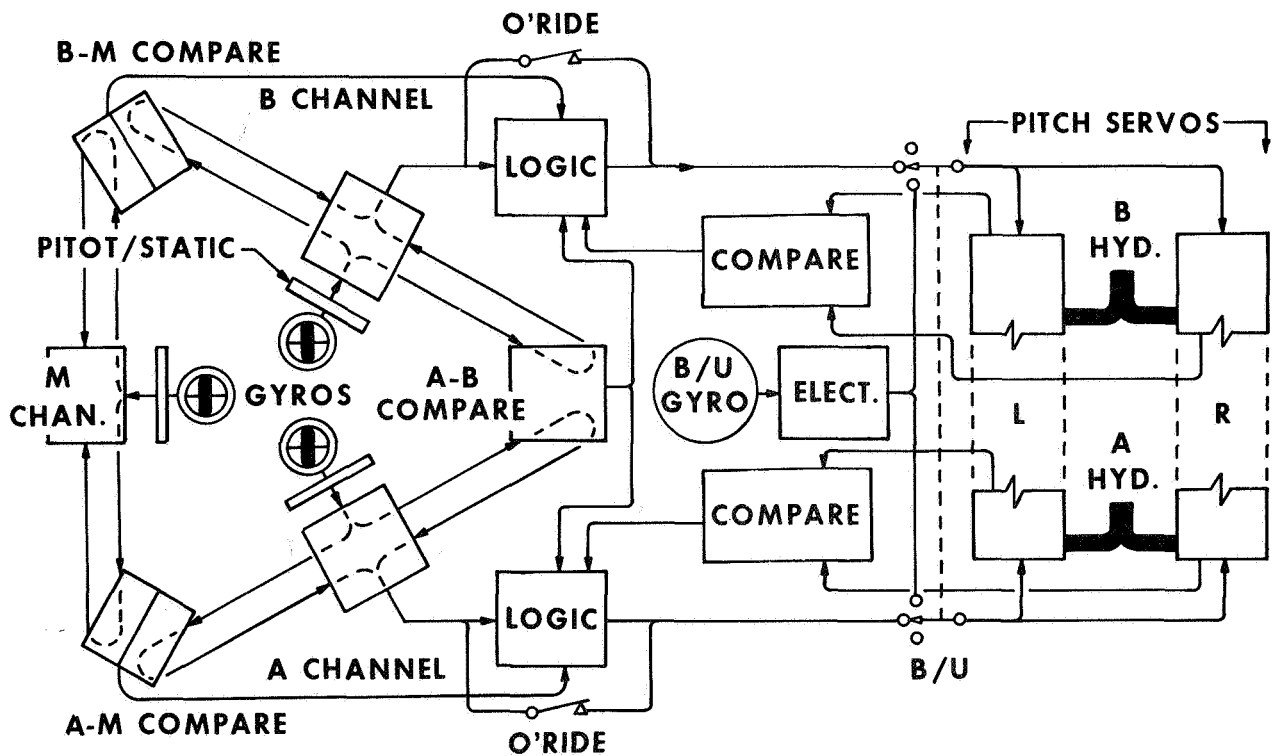
Figure 2 - Flight Controls - Pitch SAS
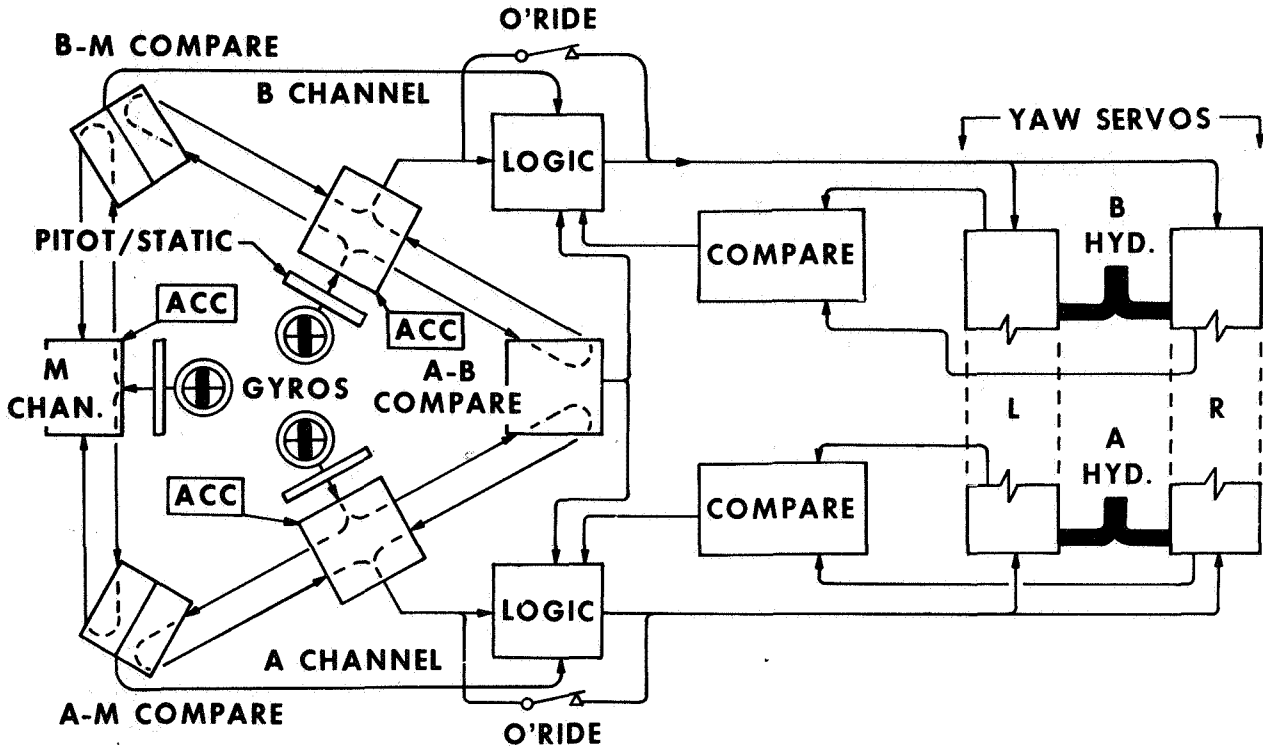
Figure 3 - Flight Controls - Yaw SAS
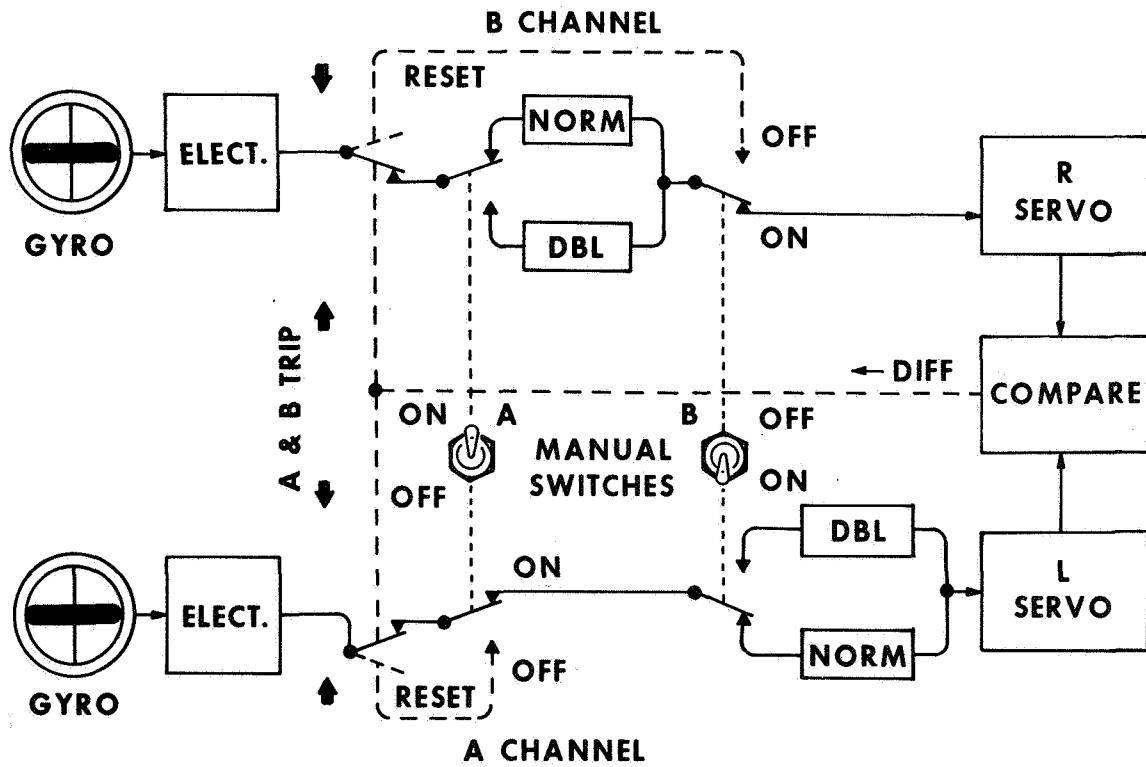
Figure 4 - Flight Controls - Roll SAS

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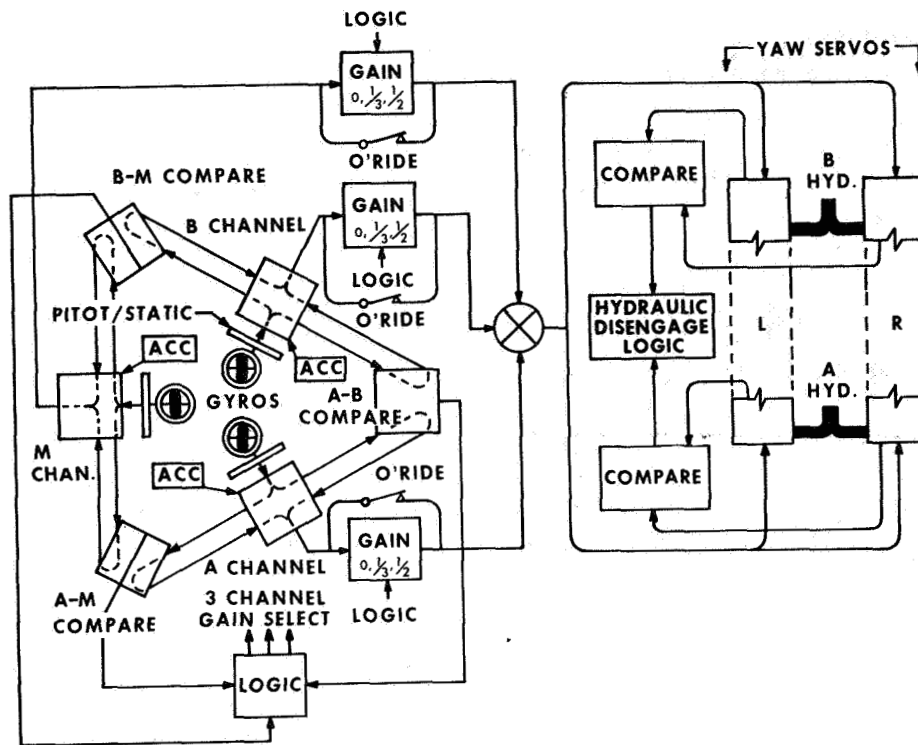
Figure 5 - Flight Controls — Yaw SAS (Interceptor)
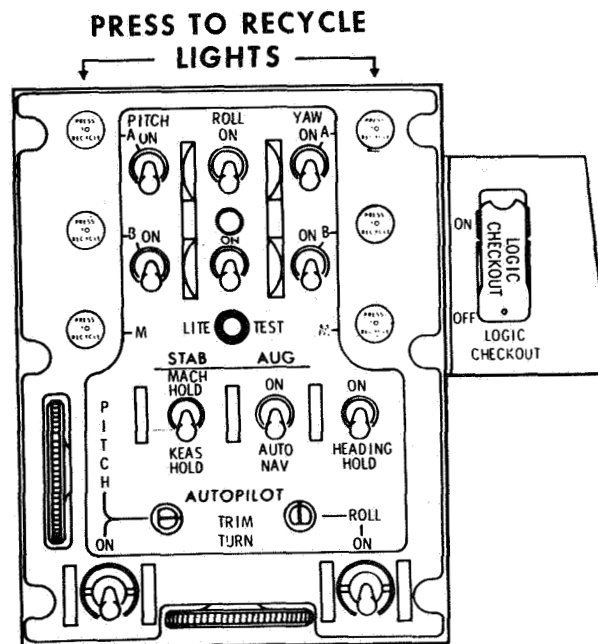
Figure 6 - SAS A/P Functions Selector Panel

B-52 STABILITY AUGMENTATION SYSTEM RELIABILITY

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SUMMARY

The B-52 SAS (Stability Augmentation System) was developed and retrofitted to nearly 300 aircraft. It actively controls B-52 structural bending, provides improved yaw and pitch damping through sensors and electronic control channels, and puts complete reliance on hydraulic control power for rudder and elevators. The system has now experienced over 300,000 flight hours and has exhibited service reliability comparable to the results of the reliability test program. Development experience points out numerous lessons with potential application in the mechanization and development of advanced technology control systems of high reliability.

INTRODUCTION

The B-52 SAS (Stability Augmentation System) was developed and retrofitted on nearly 300 aircraft in order to achieve the following objectives:

a. Minimize fatigue damage due to structure deflection in turbulence.

b. Improve capability of withstanding extremely high velocity gusts.

c. Improve yaw and pitch damping

d. Increase rudder and elevator authority.

e. Improve crew ride.

It was necessary to place unusual emphasis on system reliability, for two principal reasons:

a. On the yaw and pitch axes, replacement of the original mechanical (servo tab) system by a hydraulic actuator system introduces the possibility of total loss of rudder and elevator control in flight due to hydraulic failures.

b. The use of an electronic system with relatively high rudder and elevator authority introduces the possibility of sudden unscheduled displacements or "hardovers" of the control surfaces due to electrical faults, with obvious flight safety implications.

REDUNDANCY MANAGEMENT

Figure 1 is a simplified schematic diagram of the SAS. Yaw damping and elastic mode suppression signals are generated by combining rate gyro outputs with lateral accelerometer outputs, and the gains are scheduled according to airspeed (high gain at low airspeed and vice versa). For the pitch axis, only rate gyro signals are used; the gain is fixed and independent of airspeed. There are two essentially independent hydraulic power supplies, each having a main pump and an emergency pump. The main pumps are electrically powered; the emergency pumps are simply hydraulic transformers (motor-pump packages), driven by separate existing utility hydraulic systems and provided with flow limiters to avoid crippling the utility systems in the event of loss of fluid from a SAS system. The control surface actuators are of tandem type, normally powered by both hydraulic supplies.

The system is basically FO-FS (fail operational on first failure, fail soft on second), with the following exceptions:

a. If two lateral accelerometer channels fail, all three accelerometer channels drop out, while the yaw axis continues to operate on the yaw rate gyro signals only.

b. If two gain scheduling channels fail, all three channels revert to a low gain that is safe at all airspeeds.

These two features provide a substantial decrease in the number of two-failure combinations that can cause yaw axis disengagement or loss of function.

The basic redundancy management concept is relatively straightforward. At various points in the three-channel sensor-electronics subsystem, voters and comparators are used, as shown on Figure 2. For example, the three inputs at the left of the diagram may represent three rate gyro outputs, while the three outputs at the right may represent three channels of an electronic control unit. If any input disagrees with the median signal by more than the preselected error threshold, the comparator trips and latches itself in the tripped mode. In this mode, the comparator swamps the discrepant input so that it will not be selected by any voter as a median signal. In some cases the swamping signal is a hard-over; in other cases, it is a 400 Hz square wave. Also, the comparator shuts off its normal "O.K." signal to the logic circuitry, thus preparing the logic to take proper action in the event of a subsequent second failure. On the yaw axis, the failure of one channel also sends a "channel failed" signal to the pilot, warning him that redundancy has been lost and that yaw damping will be automatically disengaged in the event of a second similar failure. Loss of yaw damping is not a highly critical failure mode, but it poses a slight threat to flight safety by requiring manual damping of Dutch roll, which may be difficult with certain adverse combinations of high gross weight, high altitude, poor visibility, and turbulence. No such warning to the pilot is required for single channel failures in accelerometer, gain scheduling, or pitch axis channels, as these pose no threat to flight safety and require no special crew action.

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FLIGHT SAFETY RELIABILITY

In early discussions, Air Force representatives expressed a clear desire to state the system reliability objective in terms of aircraft loss rate. This required analysis in considerably greater depth than ordinary reliability calculations for a redundant system. It was necessary to:

a. Define each potentially critical failure mode of the system in terms of the effect on control surface motions.

b. Compute the probabilities of occurrence separately for each of these modes during each phase of a standardized mission profile.

c. Compute the probability of aircraft loss for each mode in a variety of flight conditions (altitude, airspeed, and presence of nearby aircraft such as in aerial refueling) with proper allowance for probabilities of various turbulence intensities and visibility conditions.

d. Combine the above to obtain a total predicted B-52 loss rate attributable to SAS failure.

CRITICALITIES

During the prototype program, hundreds of SAS failures were simulated in piloted flight simulators and the resulting aircraft motions were recorded. Five or more different pilots were used for each combination of SAS failure mode and flight condition. After each simulation, the pilot was asked to estimate the percentage of SAC pilots that would have been unable to avoid loss of the aircraft. The results were averaged to arrive at a probability of aircraft loss for each combination. These results were combined with the probabilities of given turbulence conditions, visibility conditions, and autopilot status to yield a criticality matrix suitable for use in the aircraft loss prediction program. Criticality, as used here, is defined as the probability of aircraft loss _if_ the given system failure mode occurs during given flight conditions.

In the past, there has been a widespread tendency to treat criticality as a dichotomy. To label a failure mode as "critical" meant that it would invariably cause loss of the aircraft, and to label it as "non-critical" meant that it would never cause loss of aircraft. In other words, criticality was assigned only two possible values: zero and 100 percent. It is true, of course, that many failure modes have criticalities of zero, and some failure modes, such as gross failure of a primary structure, have criticalities of 100 percent. But in any attempt to make a realistic prediction of the flight safety reliability of a control system, it must be recognized that many of the failure modes will have intermediate criticalities. They may approach 100 percent with unfavorable combinations of flight conditions, and may be essentially zero with favorable combinations of flight conditions.

The probability of occurrence of each potentially critical system failure mode during each phase of the mission was computed using conventional methods, but with certain refinements as subsequently discussed. These probabilities of

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occurrence were compiled into a failure mode occurrence probability matrix. Figure 3 is a simplified diagram showing the principal factors entering into the construction of these two matrices. The two matrices are constructed and combined in a computer program to predict aircraft losses.

In many cases it was found that the criticality of a given system failure mode was not necessarily determined by the mission phase or flight conditions in which the failure occurred, but by subsequent conditions. Many failure modes are relatively noncritical in high altitude cruise, for instance, but leave the system in a degraded state that may have a much greater criticality in subsequent mission phases such as low level penetration or landing. Since high altitude cruise accounts for a large portion of the mission duration, most of the failures will tend to occur during cruise, but many of the resulting aircraft losses will occur during a subsequent mission phase. For other failure modes, the surprise factor is predominant; the probability of aircraft loss is chiefly dependent on the pilot's skill and corrective actions immediately after the failure. These considerations were taken into account in the computerized program.

BITE

The system includes BITE (Built-In Test Equipment) which serves two main purposes:

a. It permits a quick preflight checkout to determine, as far as practicable, that all components in all channels are unfailed before takeoff.

b. It facilitates diagnosis by identifying the failed LRU.

Neither of the above BITE functions is achieved with 100 percent certainty. A careful analysis was made to determine which failure modes of which components could not be detected by BITE or by any feasible preflight check. For each such "hidden" failure mode, suitable ground check intervals were established. Whereever a hidden mode, in combination with other component failure modes, could produce a potentially critical system failure mode, the computation of the probability of system failure mode occurrence was based on the established ground check interval and not merely the time since takeoff. This makes a significant difference in the probability of a given two-failure or three-failure combination, as compared to the conventional method of computing redundant system reliability, which is based on the implicit assumption that all parts are unfailed at takeoff.

SNEAK FAILURE MODES

In addition to this "hidden" failure mode problem, we also encountered several "sneak" failure modes. A sneak failure mode may be roughly defined as one which produces unexpected effects that tend to negate part of the redundancy. Such modes exist chiefly because of inadequate FMEA (Failure Mode and Effect Analysis). For example, the voters used in the prototype design contained two sneak failure modes. In one of them, a single voter fault would produce a

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hardover signal on all three channels simultaneously. In the other, a single voter fault would cause a single hardover originating upstream to be propagated downstream on all three channels. These problems were corrected in the production design.

Another fertile field in which sneak failure modes typically abound is in the area of electronic module power supplies. Naturally, the three-channel redundant configuration of the electronics and sensors employed separate power supply modules to power the electronics on each channel. Here again sneak failure modes were found. For example, one power supply module failure could disable a channel and at the same time prevent the logic circuitry from taking proper action. Such modes were "designed out" wherever they appeared.

FAILURE MODE AND EFFECT ANALYSIS

As might be suspected from the above remarks, the task of analyzing failure modes and their effects was of paramount importance in making a realistic flight safety reliability analysis for the SAS. The FMEA is a traditional task that is usually called for in reliability programs, but the output, in many cases, is of little value in realistic computation of the reliability of a redundant system. Among the typical shortcomings are:

a. Excessive emphasis on what fails rather than how it fails; insufficient recognition of failure modes other than open circuit and short circuit.

b. Inadequate definition of effects on the system; use of catch-all phrases such as "loss or degradation of output"; phrases such as "Loss of +5 VDC power" without any attempt to describe what happens to the system when the +5 VDC power is lost.

c. Endless repetition of the obvious and neglect of the nonobvious.

d. Failure to explain the functioning of the system or assembly and its components so that the FMEA will be meaningful to personnel not highly familiar with the design.

e. Inadequate explanation of redundancies, where applicable; failure to recognize that while two assemblies may be in parallel with respect to the more common or obvious failure modes, they may be effectively in series with respect to less obvious failure modes.

Although formal FMEA reports at the assembly level were generated in the SAS reliability program, there was no attempt to compile a system-level FMEA in the usual format which is not well suited for delineating the effects of redundancies. Instead, the FMEA was effectively combined with the quantitative flight safety reliability analysis as illustrated by Figures 4 and 5. These figures represent two of the system failure modes. The notations f_{49}, f_{70}, etc. represent hourly failure rates of the various subassemblies in the applicable subassembly failure modes. In other words, they represent blocks on a series-parallel block diagram or a fault tree. Each critical system failure mode has a

separate diagram or a separate branch on a fault tree, with blocks representing only those failure modes of subassemblies or components that contribute to the given critical system failure mode. Notations such as h_{71}, g_{67}, etc. are the applicable mode failure rates of subassemblies in an off-line or standby status. W represents the probability of icing conditions that would incapacitate a pitot head with a failed heater. The symbol H refers to the 300-hour periodic check for pitot system leakage, which is the failure mode denoted by f_{81}. The notations T_1 and T_2 refer to time since takeoff; for example, if a mission phase starts 5.52 hours after takeoff and ends 7.52 hours after takeoff, T_1 = 5.52 and T_2 = 7.52. Insofar as potentially critical modes are concerned, the FMEA is thus represented by a collection of critical system, failure mode formulations similar to Figures 4 and 5. We have attempted the task of modifying the usual FMEA format to make it useful in redundant system analysis, but are not satisfied with results to date.

Many component failure modes were simulated in laboratory tests, in order to evaluate failure mode effects that were not clearly predictable.

BLOCK DIAGRAMS AND FAULT TREES

Series-parallel block diagrams and fault trees are sometimes thought of as two different techniques for redundant system reliability analysis, although when properly used they convey identical information. The chief differences between these two approaches, as traditionally used, are:

a. Blocks on the fault tree generally represent events or specific failure modes of components, while blocks on the series-parallel diagram have sometimes been used to represent the total failure rates of components.

b. The fault tree is generally constructed beginning at the top or system level and working down to the detail or functional module level; with the block diagram, there is a tendency to start at the component level and work up to the system level.

In the B-52 SAS analysis, we used two teams, one starting at the top and working down, and the other starting at the bottom and working upward. Comparison of the results provided a useful cross-check and helped to minimize the chance of overlooking critical combinations. As long as the blocks represent specific failure modes of the modules or components, there is no significant difference between the two diagramming techniques, and the choice between them is reduced to a matter of personal preference.

RELIABILITY TESTS

The reliability programs for both the prototype and production contracts included extensive system reliability testing in general accordance with MIL-STD-781. Ordinarily, system reliability tests are conducted primarily for the purpose of MTBF measurement or verification of compliance with MTBF

requirements. For the SAS, the system tests were regarded primarily as opportunities for failure cause analysis in order that corrective actions could be initiated at the earliest possible date. It is almost axiomatic in the industry that the first MTBF test will show an MTBF of about one tenth of the predicted value. (Maybe we were just lucky; our first prototype MTBF test on the SAS indicated an MTBF of about one fourth of the prediction, instead of one tenth.) Most of the failures in the MTBF tests, as well as in the flight test program and operational mockup ("Iron Bird") tests, showed clear causes in a careful failure analysis, and corrective actions were initiated for the subsequent production articles.

MTBF testing under the production contract was divided into four phases:

Phase A consisted of about 1800 hours of operation on an incomplete system - partly with prototype hardware and partly with early production (unqualified) hardware.

Phase B involved 2000 hours of operation on early production hardware.

Phases C and D involved 515 hours each, using fully qualified production hardware.

The purposes of Phases A and B was to determine where reliability improvements were needed, at the earliest practicable date. The purpose of Phases C and D was to demonstrate attainment of the required MTBF.

The reliability test environments, both prototype and production, included cold soaks and operation at ambient temperatures up to 71°C (160°F). Initially, the prototype test included periods of applied vibration at 33 Hz and 2g amplitude. Vibration attempts were finally abandoned for the following reasons:

a. This low frequency was not found to produce any significant effects on equipment failure rates.

b. This type of vibration bears practically no relation to the vibration encountered in jet aircraft.

c. Any significant increase in frequency would require a totally new test setup. The supporting jig was marginal even at 33 Hz.

EFFECTS OF WEAROUT

It is widely assumed that scheduled replacements in service will avoid the occurrence of normal wearout failures. MTBF is consequently often considered as a function of random failure rates only; and since MTBF is customarily demonstrated by tests that typically operate each specimen for 500 hours or less, normal wearout is seldom significant in MTBF demonstrations. As a result, we see so-called MTBF values of 10,000 or even 50,000 hours quoted for mechanical and hydraulic equipment items, based only on their "random" failure rates under the assumption that scheduled replacement will avoid normal wearout problems.

MTBF in service, however, is a distinctly different problem. Scheduled replacements are seldom specified or practiced except where there is a clear-cut safety implication. As a result, the effective MTBF on such equipment is often far less than a pure "random failure" consideration would indicate.

SERVICE EXPERIENCE

For this reason, we kept two sets of books on the SAS MTBF -- one set based on random failure rates only, and the other including estimated normal wearout effects. Table I shows the resulting difference in predicted system MTBF, and also shows the failure experience in service for calendar years 1972 and 1973. The following conclusions may be noted from this table:

a. The hydraulics subsystem shows a distinct rise in failure rates from 1972 to 1973. The 1973 rates agree closely with the prediction that includes wearout effects.

b. The sensor-electronics subsystem shows a decrease in failure rates from 1972 to 1973, in spite of expected wearout effects in the six gyros. This indicates a mixture of two different kinds of apparent infant mortality effects:

   (1) The usual infant mortality experienced in electronic equipment, in spite of burn-in prior to delivery.

   (2) An improvement in the maintenance organizations' familiarity with the equipment, resulting in better repairs and fewer unnecessary replacements.

c. Field experience on the system as a whole agrees closely with the prediction that included estimated effects of normal wearout.

The last two columns at the right of Table I are based on detailed analysis of two field data samples which both indicated that about one third of the reported electronic failures might be attributed to trial-and-error trouble-shooting or other diagnostic errors. This situation is believed to be improving with time and experience gained in the field.

Table II shows the various types of mission reliabilities experienced in service in the 1972-1973 period. There were no corresponding quantitative requirements or predictions.

Table III shows the SAS flight safety reliability requirements and predictions. The predictions were calculated both with and without normal wearout effects. There have been no losses to date attributable to the SAS. There were several early occasions of loss of one hydraulic power supply in service, due to fatigue failures of main pump rigid discharge lines which happened to be in resonance with the pump pulsation frequency. Actually, a similar failure had previously occurred in system reliability testing, but no importance was attached to it, since the test chamber space limitations required the use of

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plumbing configurations somewhat different from those of the aircraft. The lesson learned from this experience is that every effort should be made to use aircraft plumbing configurations in system reliability tests, particularly where there are conceivable resonance or fatigue problems.

The system MTBF tests indicated surprisingly low reliability for certain simple widely used standard or semistandard hydraulic components such as accumulators and pressure switches. Although corrective actions were initiated, the field reliability experience on these components is still disappointing.

CONCLUDING REMARKS

The next few years will see extensive development of electronic-hydraulic flight control systems of fly-by-wire and controls-configured-vehicle types, performing highly essential functions and with extremely high reliability requirements. The B-52 SAS program has provided useful experience for the development of such systems, and has demonstrated the need for close attention to the following considerations:

- Optimization of redundancy management.

- Meaningful Failure Mode/Effects analyses with particular emphasis on effects of redundancy and redundancy management and on early detection of possible sneak failure modes. References 1, 2, and 3 all provide useful guides for failure mode effect analysis.

- Laboratory simulation of failure modes to verify effects and serve as an added guard against sneak failure mode effects.

- Piloted simulator programs to measure pilot reaction to failure modes where applicable, under various visibility and turbulence conditions.

- Adequate consideration of wearout effects in mechanical/hydraulic components.

- Quantification of system failure mode criticalities to permit better allocation of effort and redundancy.

- Adequate BITE to avoid takeoff with possible hidden failure modes.

- Suitable periodic checks for detection of possible hidden failure modes not feasibly detectable by BITE.

- Proper reflection of periodic check interval in reliability predictions, for modes not detected by BITE.

- Adequate BITE fault isolation capability to facilitate proper system repair.

- Definition of reliability requirements for supplier-designed components

in terms of failure mode effects and redundancy management as well as the customary MTBF requirements.

● Establishment of schedule that permits adequate reliability testing to find areas for reliability improvement at earliest possible time before final design freeze.

● Vigorous failure analysis and reliability corrective action program, not only in reliability tests but also in other test areas (qualification, iron bird, flight tests, etc.)

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2. Crown, Peter L.: Design Effective Failure Mode and Effect Analysis. Proceedings, 1969 Annual Symposium on Reliability, Chicago, January 1969.

3. Greene, K., and Cunningham, T. J.: Failure Mode, Effects, and Criticality Analysis. Proceedings, 1968 Annual Symposium on Reliability, Boston, January 1968.

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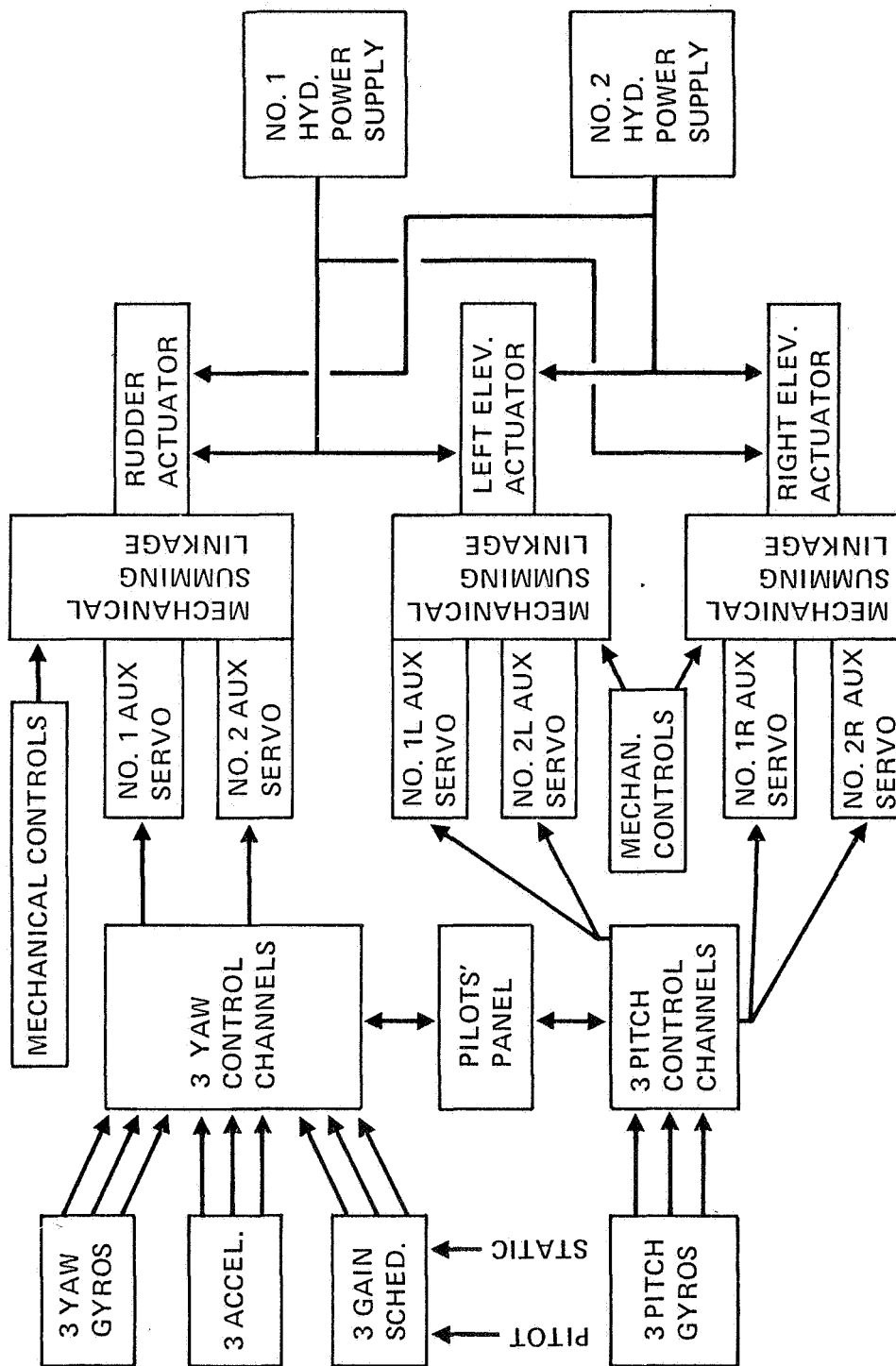
Figure 1. – B-52 SAS Diagram.

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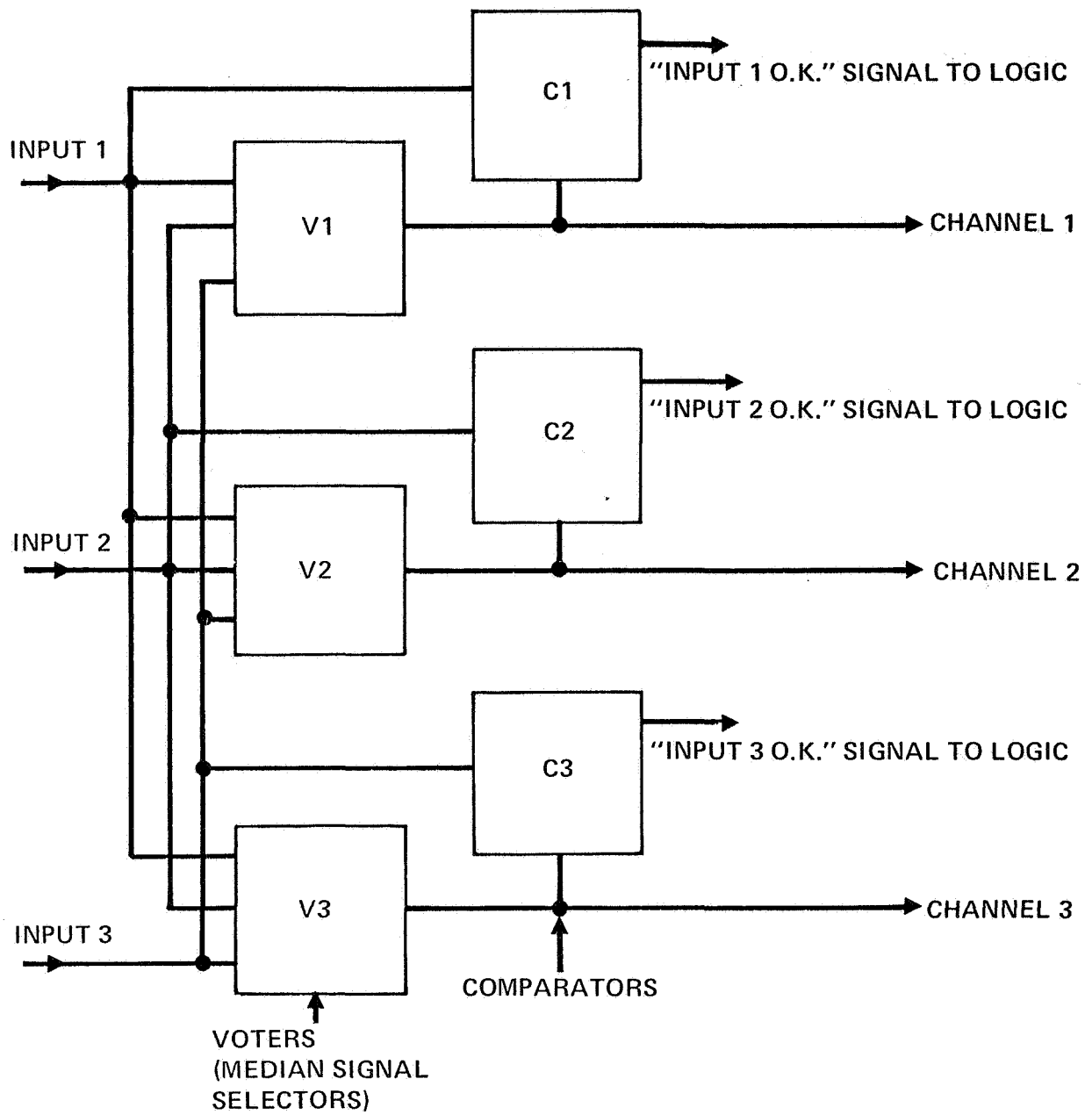
Figure 2. - Typical Voter-Comparator Diagram.
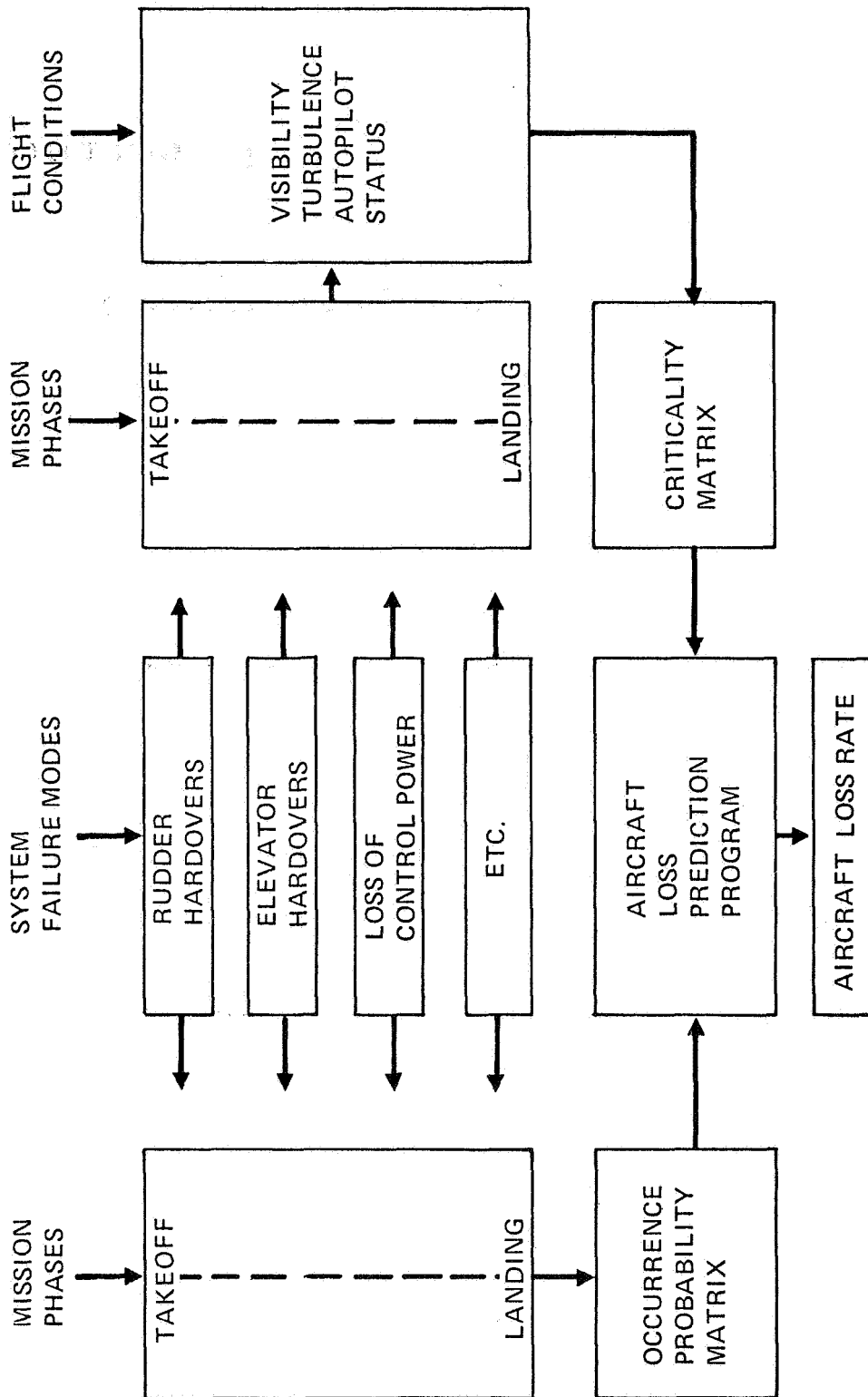
Figure 3. — Matrices and Loss Predictions.

MODE 25. MOMENTARY RUDDER HARDOVER TO SAS AUTHORITY LIMIT OR LESS, CUT OFF BY PROMPT COMPARATOR TRIP PRODUCING LOSS OF YAW DAMPING.

CONDITIONS PRODUCING MODE 25 ARE:

ITEM

PROBABILITY

(1) OPEN LOOPS ON BOTH SERVO CHANNELS (EITHER SEQUENCE)

$(f_{49} + f_{70} + f_{87})(f_{50} + h_{71} + f_{87})$ $(T_2^2 - T_1^2)$

(2) FAILURE OF NO. 1 SERVO CHANNEL (ANY SOURCE OF COMPARATOR TRIP OTHER THAN OPEN LOOP), FOLLOWED BY OPEN LOOP ON NO. 2 SERVO CHANNEL

$(1/2)(f_{28} + f_{40} + f_{45} + f_{47} + 2f_{60} + f_{66} + f_{68} + 2f_{86}$
$+ f_{88} + f_{89})(f_{50} + f_{71} + f_{87})$ $(T_2^2 - T_1^2)$

(3) FAILURE OF NO. 2 SERVO CHANNEL (ANY SOURCE OF COMPARATOR TRIP OTHER THAN OPEN LOOP), FOLLOWED BY OPEN LOOP ON NO. 1 SERVO CHANNEL

$(1/2)(f_{28} + f_{44} + f_{46} + f_{48} + f_{60} + g_{67} + g_{69} + f_{86} + f_{88}$
$+ f_{89})(f_{49} + f_{70} + f_{87})$ $(T_2^2 - T_1^2)$

P_{25} = PROBABILITY OF OCCURENCE OF MODE 25 BETWEEN T_1 AND T_2

= $[\{f_{49} + f_{70} + f_{87}\}\{f_{50} + h_{71} + f_{87} + (1/2)(f_{28} + f_{44} + f_{46} + f_{48} + f_{60}$
$+ g_{67} + g_{69} + f_{86} + f_{88} + f_{89})\} + (1/2)(f_{28} + f_{40} + f_{45} + f_{47} + 2f_{60} + f_{66}$
$+ f_{68} + 2f_{86} + f_{88} + f_{89})(f_{50} + f_{71} + f_{87})][T_2^2 - T_1^2]$

Figure 4

MODE 33: SUSTAINED RUDDER OSCILLATION (FLUTTER) AT AIRSPEED ABOVE 300 KNOTS EAS, DUE TO EXCESSIVE YAW SAS GAIN.

CONDITIONS PRODUCING MODE 33 ARE:

ITEM	PROBABILITY
(1) LOSS OF PITOT PRESSURE:	
(1a) FAILURE OF BOTH PITOT HEATERS, MULTIPLIED BY PROBABILITY OF MODERATE TO SEVERE ICING CONDITIONS	$(f_{80})^2 (W)(T_2^2 - T_1^2)$
(1b) FAILURE OF EITHER PITOT HEATER (MULTIPLIED BY PROBABILITY OF ICING), AND LARGE LEAK IN OPPOSITE PITOT LINE	$2W(f_{80})(f_{81})(300/2)\overset{*}{} T_2 - T_1$
(1c) LARGE LEAKS ON BOTH PITOT LINES	$(f_{81})^2 \overset{*}{(300)}(T_2 - T_1)$
(1d) SINGLE FAILURE IN PITOT MANIFOLD VALVE, PSU MANIFOLD OR INTERCONNECTING HOSE	$f_{82}(T_2 - T_1)$
(2) DEGRADATION OF PSD GAIN IN YAW SERVO POSITION FEEDBACK LOOP	$f_{83}(T_2 - T_1)$
(3) STUCK SOLENOID VALVE ON NO. 1 YAW SERVO, COMBINED WITH ANY TRIP OF NO. 1 SERVO COMPARATOR WHICH LEAVES SENSOR COMMAND APPLIED TO SERVO	$(f_{90})(f_{40} + f_{60} + f_{86} + f_{88} + f_{89})(T_2^2 - T_1^2)$

P_{33} = PROBABILITY OF OCCURRENCE OF MODE 33 BETWEEN T_1 AND T_2

= $(f_{80})^2(W)(T_2^2 - T_1^2) + W(f_{80})(f_{81})\overset{*}{(300)}(T_2 - T_1) + (f_{81})^2 \overset{*}{(300)}(T_2 - T_1) + f_{82}(T_2 - T_1)$

$+ f_{83}(T_2 - T_1) + (f_{90})(f_{40} + f_{60} + f_{86} + f_{88} + f_{89})(T_2^2 - T_1^2)$

*H = 300 HOURS FOR f_{81}

Figure 5

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TABLE I

MTBF COMPARISONS

ITEM	FAILURES PER THOUSAND FLIGHT HOURS					
	PREDICTIONS BASED ON TEST EXPERIENCE		AFM-66-1 SERVICE DATA			
			COUNTING ALL REPORTED ELECTRONIC FAILURES		COUNTING 2/3 OF REPORTED ELECTRONIC FAILURES	
	NO WEAROUT	WITH WEAROUT	1972	1973	1972	1973
SENSOR/ELECTRONICS SUBSYSTEM	5.077	7.459	9.756	8.705	6.504	5.803
HYDRAULICS	2.553	7.271	5.306	7.564	5.306	7.564
MISCELLANEOUS	1.697	1.697	0.601	0.857	0.601	0.857
SYSTEM	9.327	16.427	15.663	17.126	12.411	14.224
MTBF, HOURS	107	61	64	58	81	70
MTBF GOAL	100	—	—	—	—	—

TABLE II

SAS MISSION RELIABILITY COMPARISONS

BASIS: SAC AIR VEHICLE PERFORMANCE REPORTS, 1972 AND 1973

ITEM	RELIABILITY
FLIGHT RELIABILITY:	
PROBABILITY OF NO FLIGHT ABORT DUE TO SAS	99.96%
PROBABILITY OF NO SAS FLIGHT ABORT OR MAJOR DEGRADATION* IN FLIGHT	99.58%
DISPATCH RELIABILITY:	
PROBABILITY OF NO LATE TAKEOFF OR CANCELLATION DUE TO SAS	99.73%
COMBINED RELIABILITY:	
PROBABILITY OF NO SAS FLIGHT ABORT, MAJOR DEGRADATION, LATE TAKEOFF, OR CANCELLATION	99.31%

*INCLUDES LOSS OF PRESSURE FROM ANY OF THE FOUR PUMPS.

TABLE III

SAS FLIGHT SAFETY RELIABILITY

	FLIGHT SAFETY RELIABILITY	AIRCRAFT LOSS RATE DUE TO SAS, PER 10^6 FLIGHTS
GOAL	99.999182%	8.18
PREDICTION (NO WEAROUT)	99.999798%	2.02
PREDICTION (WITH WEAROUT)	99.999508%	4.92
EXPERIENCE TO DATE	NO LOSSES	NO LOSSES

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LOCKHEED L-1011 AVIONIC FLIGHT CONTROL REDUNDANT SYSTEMS

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SUMMARY

Two of the Lockheed L-1011 automatic flight control systems - yaw stability augmentation and automatic landing - are described in terms of their redundancies. The reliability objectives for these systems are discussed and related to in-service experience. In general, the availability of the stability augmentation system is higher than the original design requirement, but is commensurate with early estimates. The in-service experience with automatic landing is not sufficient to provide verification of Category III automatic landing system estimated availability. Component reliability is, however, generally tracking expectation.

INTRODUCTION

The L-1011 TriStar has been in airline operation since April 1972 as one of the current generation of wide-body jets. In service at present, there are about 80 units of the current model which is a short to medium range airplane that cruises typically at $M = .85$, $H_p = 33,000$ feet. Maximum takeoff and landing weights are 430,000 and 360,000 pounds. Figures 1 and 2 show the airplane dimensions and flight control surfaces, respectively.

The Avionic Flight Control System (AFCS) of the L-1011 is highly redundant in comparison to such systems of the previous generation of aircraft. This redundancy to a certain extent is manifest in the so-called "cruise" autopilot portions of the AFCS, but this was more or less a fallout of the need for high redundancy in the Category III Automatic Landing System (ALS). The configuration of the "cruise" portion of the AFCS yaw control channel was also affected by this Category III requirement.

It is intended in the following discussion to provide brief descriptions of the automatic yaw cruise control system and of the automatic landing system, these descriptions to provide the background for judging system redundancy in comparison to other systems familiar to the reader. It is further intended to present in-service derived data describing the reliability of these two systems and to relate this experience to expectation.

AFCS OVERVIEW

The complete AFCS including Category IIIa automatic landing was certifi-
cated at the time of initial airplane FAA certification in April 1972. It
has been subsequently so certified by Canada (MOT), Great Britain (CAA),
Japan (JCAB), and West Germany (LBA). In the total fleet to date, there
have been about 160,000 revenue flight hours accumulated for approximately
80,000 flights.

Briefly the AFCS consists of four subsystems:

Stability Augmentation System (SAS)
Autopilot/Flight Director System (APFDS)
Speed Control System (SCS)
Flight Control Electronic System (FCES)

The components which comprise the AFCS are listed by subsystem in
Table 1. For total systems function, these components interface with other
airplane elements such as sensors-air data, attitude references, radio
navigation and altimetry systems, electrohydraulic and electrical flight
control servos, flight instruments, control panels, etc.

The SAS functions include yaw damping, turn coordination, runway align-
ment during automatic landing and automatic steering during the landing
rollout.

The APFDS provides for automatic control of the airplane from takeoff
to landing. There are the usual modes of

Roll and Pitch Attitude Hold with Control Wheel
    Steering (CWS) and Turbulence Configuration Control
Altitude Select and Hold
Vertical Speed Select and Hold
Airspeed Hold on Pitch
Mach Hold on Pitch
Heading Select and Hold
VOR and Area Navigation
Localizer Capture and Track

In addition, there are the common axis modes of

Approach
Approach/Land (Autoland)
Go-Around
Takeoff

The pitch commands for Go-Around and Takeoff are derived in the SCS with
Takeoff being a flight director mode only.

The SCS autothrottle modes are:

Airspeed Select and Hold
Stall Margin Control

The latter is primarily an approach/land mode which uses angle of attack as the basic reference. And as just mentioned, the SCS also provides for the Go-Around and Takeoff modes using angle of attack as a reference.

The FCES provides a number of functions such as electrical pitch trim, Mach trim, Mach feel, stall warning, altitude alert, primary flight controls monitoring, automatic ground speed brakes and direct lift control. All of these functions operate when either pilot or autopilot is in control.

With these descriptive remarks as background, further discussion is confined to the SAS cruise control system and to the Automatic Landing System. Each of these systems has operational availability/reliability requirements which we shall examine and relate to the reliabilities achieved in service use.

STABILITY AUGMENTATION SYSTEM (SAS)

System Mechanization

Figure 3 depicts the cruise configuration of the SAS. Each of the two yaw computers contains two computation channels that output identical servo commands to an in-line monitored electrohydraulic servo. Four aileron position transducers and three rate gyros service the four computation channels of the total system. The rate gyros provide for Dutch roll damping inputs and the aileron transducers provide for turn coordination.

Figure 4 shows one channel of the SAS cruise computation. It is seen that the gains are scheduled with flap position and the gyro path has the usual low frequency washout filter plus a high frequency cut-off. The aileron input path has a limited washout to remove aileron trim effects and in addition an adjustable dead zone such that turn coordination only comes into play for sufficiently large aileron inputs. The passed signal is subject to gain changing to match the gyro channel and to low pass filtering. The voter output to rudder surface response can be approximated by a two Hz second order servo for small amplitudes. However, the primary control surface servo is severely hinge moment limited in cruise flight.

It is noted that in Figure 4 the output of the computation comprises one input to a voter. The other inputs are derived from the other three computation channels of this dual-dual mechanization. As one would expect, there are two computations and two voters per yaw computer with two voter outputs required to drive one SAS electrohydraulic servo as depicted in

Figure 5. The two voter outputs provide for driving the EHV coils in a push-pull arrangement with two sets of dual monitors acting to shut off the servo loop hydraulics if a fault is detected.

There are also monitors in front of the voters which control the signal configuration of the voter inputs as shown in Figure 6. This figure illustrates the concept whereby the monitors control switching logic that substitutes signal ground or an alternate computation for a faulted channel. Figure 7 shows the voter input crossfeeding for the complete dual-dual system.

In addition to servo and computation monitors, there are rate gyro monitors and electrical power monitors. The latter operate into the servo engage logic while the former monitors operate into the voter switching logic.

Design Objectives and Performance

The function of the cruise mode of the SAS is, of course, to provide improved Dutch roll damping for enhancement of passenger comfort and handling qualities and for reduction of fin loads. This reduction of vertical tail loading, in continuous turbulence, due to the action of the SAS was reflected in the definition of limit design loads.

Early in the development of the L-1011, the effectiveness of the SAS was investigated to determine performance and reliability objectives for the SAS from a loads viewpoint. It appeared that a minimum damping ratio of 0.3 and a timewise availability of 97% were modest design objectives that would yield significant load reductions. It was subsequently found, however, that higher damping ratios could be achieved over most of the climb, cruise and descent flight regimes as seen from the data given in Table 2. Only at low speeds, where effects on fin loads are not critical, are the damping ratios less than 0.3.

It also became evident that a 97% availability requirement was a very conservative estimate of system reliability. On the basis of guaranteed failure rates, the single channel failure rate was calculated to be about 10^{-3} per hour and to preclude the possibility that an airplane might be flown without SAS for a protracted period, it is required that at least one of the two channels be operative for dispatch. Recognizing that for most flights both channels of SAS are operative, even 99.9% timewise availability would appear to be conservative.

A complete discussion of the effect of SAS availability on loads is given in reference (1) from which Figure 8 is taken. This figure illustrates the definition of design loading for vertical tail shear with 0, 97% and 100% SAS availabilities. It is based on a mission analysis criterion

whereby the frequency of exceedance of a load quantity is calculated for operations over specified design flight profiles. The turbulence environment as statistically described for each segment of a profile is applied to the airplane/load transfer function to derive exceedance curves (with or without SAS operating) for each segment. The segment exceedances are summed over the total of all profiles to determine a load vs. frequency-of-exceedance curve for the mission.

It can be seen from Figure 8 that the major reduction $(\frac{H}{F} = 0.70)$ is realized by having at least 97% availability and further reduction comes less readily with 100% availability realizing a ratio of $\frac{G}{F} = 0.65$. These results are for a fully linear system and saturation effects reduce the benefits somewhat. In summary, however, with 97% availability the net reduction in fin loading is better than 25% relative to what it would be if no SAS were available.

It would be very surprising if the in-service reliability indicated a SAS availability of less than 99.9%. The component MTBF values are tracking guarantees as indicated in Table 3. There have apparently been only five complete in-flight losses of SAS and a very few delays as a result of lack of immediate parts replacement. These instances with one exception were associated with dispatch for many consecutive flights with a failed computation channel. The number given above for in-flight losses covers a period in which revenue flight hours were accumulated with an average flight time of two hours. We believe that there have been no other instances of complete loss to date, and that it is conservative to use only that period for which detail records have been evaluated in estimating the total system failure rate. On the basis of actual total in-flight losses during the period evaluated, the SAS availability would be

$$1 - \frac{(5 \text{ losses}) (\frac{1}{2} \text{ average flight time})}{\text{total flight hours}}$$

or about 99.98%. The individual SAS channel in-flight failure rate was also examined and it was found that 60 channel failures were experienced in a 30,000 flight hour (2-hour flights) period. This indicates a SAS channel MTBF of 1000 hours which is commensurate with the data of Table 3.

SAS Conclusions

With respect to the yaw stability augmentation system, the following conclusions can be drawn:

o   97% availability is an extremely conservative value upon which to base design loads.

o   With current technology of design, manufacturing and
    airline maintenance, single channel SAS reliability
    should be adequate to support fin loading design
    criteria as established for the L-1011.

o   L-1011 dual channel SAS provides fin load alleviation
    for all practical purposes equivalent to 100% SAS avail-
    ability.

AUTOMATIC LANDING SYSTEM (ALS)

System Mechanization

The principal elements of the ALS are the APFDS and SAS and their
respective sensors in the configurations established with the Approach/
Land (A/L) mode selected. The system in total definition includes much
more than these units but these have, by far, the most effect on system
reliability and availability. Reliability is used here in the sense of
the system capability to complete a landing. It relates directly to
safety, particularly in low weather minima operations. It was, of course,
the Category III requirement that dictated the extent of redundancy in the
ALS. This redundancy is depicted in some generality in Figure 9 for the
pitch and roll control axes. Each of these axes uses three accelerometers
(normal or lateral) and three attitude inputs. Pitch computations use
only derived pitch rate; roll uses both attitude and roll rate signals.
The Autoland Sensor signals are glideslope error and radio altitude for
pitch and localizer error for roll. Only two each of the Autoland Sensors
are used but each has dual outputs with high integrity self-monitoring.
For example, the probability of the two signals from one G/S receiver being
faulted at a critical time without warning is less than 10^{-9}.

The same theme of APFDS redundancy is carried over into the SAS in
the A/L mode as seen in Figure 10. Here, the exception is that only two
compass systems are utilized which do not have the integrity of an Auto-
land Sensor. The redundancy requirement, however, is not as great for
yaw control as it is for pitch and roll. (In the development program,
automatic landings with no automatic yaw control have been demonstrated
without any significant effect except that the pilot had to control the
rollout.) The compass inputs are actually compared in the SAS computers
and used to define a reference heading error which is memorized. The
compass signals are switched out at 150 feet and integrated rate gyro data
is used from there to touchdown. (The radio altitude signals used to
control this function are omitted from Figure 10.) During this time, a
maneuver is performed whereby the aircraft fuselage is aligned with the
runway and a wing down is held against crosswind.

This use of the compass points out the difference between the safety
and availability aspects. For Category IIIa conditions, the align

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capability is required, at present, and if one compass system fails on the approach above the alert height (100 feet for U.S. Carriers), a missed approach is executed. Safety implications are minimal, but as far as availability goes the day is lost.

As would be expected, the fail-operative pitch, roll and yaw (below 150 feet) mechanizations closely follow that as depicted in Figures 5, 6, and 7 for the cruise yaw control. Four computation channels for each axis are needed for the fail-operative condition and two or three for the fail-passive condition. The latter configuration is acceptable for Category II operations while the former is required down to the alert height for Category IIIa. There are minor differences in each servo control and monitoring mechanizations, but the basic concepts of Figure 5 are applied. For Category III, of course, it requires two servos per axis while one is acceptable for Category II.

Much is left unsaid about other subsystems of the ALS, such as

o   Speed Control System
o   Automatic Pitch Trim
o   Direct Lift Control
o   AFCS Mode Progress and Warning Indicators
o   Flight Instrument Systems
o   Hydraulic Power Sources
o   Electrical Power Sources

In the interest of completeness, however, Table 5 is given to provide a brief summary of the major elements of the total ALS. It is also noted that a more complete description of the AFCS is given in Reference 2.

ALS Objectives and Development Results

There were three L-1011 program objectives with respect to the ALS.

1.   Achieve a Category IIIa certification with a
     system having the potential for Category IIIb.

2.   Develop a maintainable system.

3.   Develop a system which has a reasonably high
     availability.

There is no doubt we held tenaciously to achievement of the first objective and we like to believe we have done the same with the other two. It may not have always been apparent, but we believe we are tracking fairly well even though it is perhaps too early to have all things proven out.

It is a fact that we certified for Category IIIa with the FAA on schedule; but, as you are probably well aware, the manufacturer's certification is only the first of a series. Each operator must verify its capability to use the system to the satisfaction of the same regulatory

agency. One L-1011 operator has accomplished this; others are working at it. In the meantime, we are beginning to look toward achieving a Category IIIb capability.

One of the things an operator must show to achieve an ALS certification is his ability to maintain the system. An indication of this capability is a comparison of failure rates achieved with those used in the Lockheed certification analysis. In effect, MTBF tracking limits are defined. Table 4 shows a list of MTBF lower limits and their currently estimated values. The data given in this table are for the significant contributors to the total disconnect probability (below the alert height). If the MTBF's of all the listed units were at the lower limits, the total disconnect probability would be potentially a factor of two higher, still within acceptable limits. These "lower limits" are not absolute limits in view of the fact that the two factor does not put the disconnect probability to an unacceptable level and further one low MTBF value could be compensated by a high one. To a certain extent the limit is a tracking limit to signal for more detail examination of a potential trouble area. So far, however, things seem to be tracking fairly well.

With respect to ALS availability, there is very little data to display. The one airline operator that has received a Category IIIa certification has shown in his initial data gathering period results to support the certification requirement. The reported results support the reliability prediction but do not allow correlation with the availability estimates of Figure 11. This figure gives a prediction of the Category IIIa ALS availability as an operational day (14 hours) progresses. It is assumed that 10 hours are reserved for maintenance and that the ALS is apparently restored to a fault-free condition by the start of each day. Mature failure rates were used to make the prediction.

The curve of Figure 11 may well represent an upper value on availability for the ALS, but at this time we cannot say. We shall find out, however, as we are now embarking on a program for evaluating availability in cooperation with one overseas operator. And we feel confident that the system will prove out well.

ALS Conclusions

The progress with the L-1011 to date has shown that certifying and supporting the maintenance of a highly redundant automatic landing system can be accomplished in a scheduled manner much like any other flight control system. Further, it is expected that future progress will serve to demonstrate that the redundancy and complexity will not detract from the economic benefits of system utilization.

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1. Hoblit, Frederic M.: Effect of Yaw Damper on Lateral Gust Loads in Design of the L-1011 Transport. AGARD Presentation, the Hague 7-12 October 1973, published in AGARDograph No. 175.

2. Mineck, D. W., Derr, R. E., Lykken, L. O., Hall, J. C.: Avionic Flight Control System for the Lockheed L-1011 TriStar. SAE Presentation for Aerospace Control and Guidance Systems Committee Meeting No. 30, 27-29 September 1972, published by Collins Radio Company, Cedar Rapids, Iowa.

Table 1. - L-1011 Avionic Flight Control System Equipment List

Stability Augmentation System (SAS)

2	Yaw Computers
3	Rate Gyros
2	Aileron Position Sensors (dual)
2	Rudder Position Sensors (dual)

Autopilot/Flight Director System (APFDS)

2	Pitch Computers
2	Roll Computers
2	Pilot's Control Wheels
2	Mode Annunciators
2	Warning Indicators
1	Mode Select Panel (5 modules)
3	Normal Accelerometers
3	Lateral Accelerometers

Speed Control System (SCS)

1	Speed Control Computer
1	Autothrottle Servo
2	Longitudinal Accelerometers

Flight Control Electronic System (FCES)

1	FCES Computer
1	Trim Augmentation Computer
2	Angle of Attack Sensors
2	Stick Shakers
1	Surface Position and Pitch Trim Indicator
10	Surface Position Sensors
2	Control Panels

| | FLIGHT CONDITIONS (MID CG) | | | | | | DUTCH ROLL MODE DAMPING RATIO AND DAMPED NATURAL FREQ | | | |
| | | | | | | | SAS ON | | SAS OFF | |
CONFIGURATION	SPEED KEAS	MACH NO.	ALTITUDE KFT	WEIGHT KLBS	FLAPS DEG	GEAR	ζ	ω_d HZ	ζ	ω_d HZ
Climb	246	.45	10	404	UP	UP	.32	.13	.09	.15
Climb	356	.65	10	308.5	UP	UP	.72	.14	.12	.22
Climb	358	.8	20	400	UP	UP	.56	.15	.07	.21
Cruise	310	.86	33	350	UP	UP	.45	.18	.10	.20
Cruise	260	.86	37.5	300	UP	UP	.43	.15	.07	.18
Cruise (M_{MO})	352	.90	26.5	300	UP	UP	.55	.21	.11	.24
Dive (M_D)	412	.95	21.5	350	UP	UP	.53	.25	.13	.28
Dive (M_D)	258	.95	42	300	UP	UP	.41	.17	.11	.18
Cruise $(1.4\ V_S)$	221	.74	38	300	UP	UP	.22	.14	.05	.15
Cruise	216	.435	15	308.5	UP	UP	.33	.13	.11	.15
Descent	246	.45	10	308.5	UP	UP	.49	.13	.12	.17
Holding	256	.4	1.5	308.5	UP	UP	.50	.13	.13	.16
Holding	160	.292	10	308.5	DOWN	UP	.29	.10	.08	.13
Approach (1.3 V_S)	139	.21	0	308.5	DOWN	DOWN	.26	.10	.09	.12
LANDING (1.3 V_S)	133	.2	0	308.5	DOWN	DOWN	.24	.09	.09	.12
LANDING (1.3 V_S)	141	.213	0	348	DOWN	DOWN	.21	.09	.06	.12
LANDING (DLC ON)	133	.2	0	308.5	DOWN	DOWN	.26	.09	.10	.12
LANDING (1.4 V_S)	143	.262	10	308.5	DOWN	DOWN	.21	.09	.05	.12

Table 2. - L-1011 Dutch Roll Characteristics With and Without Yaw SAS

Table 3. - SAS Reliability Summary

Item	No. Of Units per SAS Channel	No. Of Units Per Aircraft	Latest Point Estimate MTBF per Unit	Latest MTBF Estimate @ 90% Confid.	Mature Unit MTBF
Yaw Computer	1	2	10,900	6,800	4,600
Rate Gyro	2*	3*	10,200	6,300	6,400
Aileron Position Sensor	1	2	--- †	47,100	222,000
Rudder Servo	1	2**	--- †	23,500	24,000
Control Panel	1	2**	--- †	23,500	206,000
Hydraulic Source	1	2	5,500/9,600	3,300/4,800	10,000
Electrical Source	2*	3*	. --- †	23,500	167,000

* One gyro is shared by each SAS channel as is one electrical source.

** Any elements common to both SAS channels are negligible re MTBF estimates.

† There were no failures in reporting period.

Table 4. - Estimated MTBF's vs MTBF Lower Limits

Item	MTBF Lower Limit	Latest MTBF Point Est.	Mature MTBF
Roll Servo	4,000	*	20,000
Pitch Servo	9,750	*	49,000
Roll Computer	1,675	1,800	3,350
Pitch Computer	1,350	3,400	2,700
Yaw Computer	2,300	10,900	4,600
ILS Receiver	1,750	5,400	3,500
Radio Altimeter	1,750	9,100	3,500
Vertical Gyro	1,860	5,200	3,720
Lateral Accelerometer	31,700	**	63,500
Normal Accelerometer	31,700	**	63,500
Warning Indicator	9,600	9,100.	48,000
Aural Warning Unit	30,000	***	150,000
Hydraulic Source	2,000	5,500	10,000

* No reported failures in 54,000 servo flight hours.

** No reported failures in 81,000 accelerometer flight hours.

*** No reported failures in 27,000 flight hours.

Table 5. - Automatic Landing System Major Elements

Item	No. Req.	Remarks
Pitch Computer	2	⎫
Roll Computer	2	⎬ Each computer is dual channel.
Yaw Computer	2	⎭
Roll A/P Servo	2	⎫
Pitch A/P Servo	2	⎬ Each servo is in-line monitored.
Yaw A/P Servo	2	⎭
Aileron Position Sensor	2	Each sensor is dual.
Rudder Position Sensor	2	" " " "
Yaw Rate Gyro	3	Each has limited in-line monitoring.
Mode Annunciator	2	
Warning Indicator	2	
Mode Select Panel	1	
Normal Accelerometer	3	
Lateral Accelerometer	3	
Attitude Gyro	3	Each has limited in-line monitoring
Radio Altimeter	2	⎫ Each has dual outputs with high
ILS Receiver	2	⎭ integrity monitoring.
Speed Control Computer	1	Computer is dual channel
Autothrottle Servo	1	Servo is in-line monitored
Longitudinal Accelerometer	2	
FCES Computer	1	Provides for fail-op/fail-pass DLC
DLC Servo	2	Each is in-line monitored
Trim Augmentation Computer	1	Provides for fail-op/fail-pass auto pitch trim.
Angle of Attack Sensor	2	Each has limited in-line monitoring.
Air Data Computer	2	Each has limited in-line monitoring.
Altimeter	2	
IAS/M Indicator	2	
VSI	2	
ADI	2	
HSI	2	
Radio Altitude Indicator	2	
Compass System	2	
Hydraulic Source	2	
Electric Source	3	

WING AREA
3456 SQ. FT.

FUSELAGE
DIAMETER
235 IN.

55 FT 4 IN.

178 FT. 8 IN'

15 FT 1 IN.

FLOOR HEIGHT

36 FT.

155 FT 4 IN.
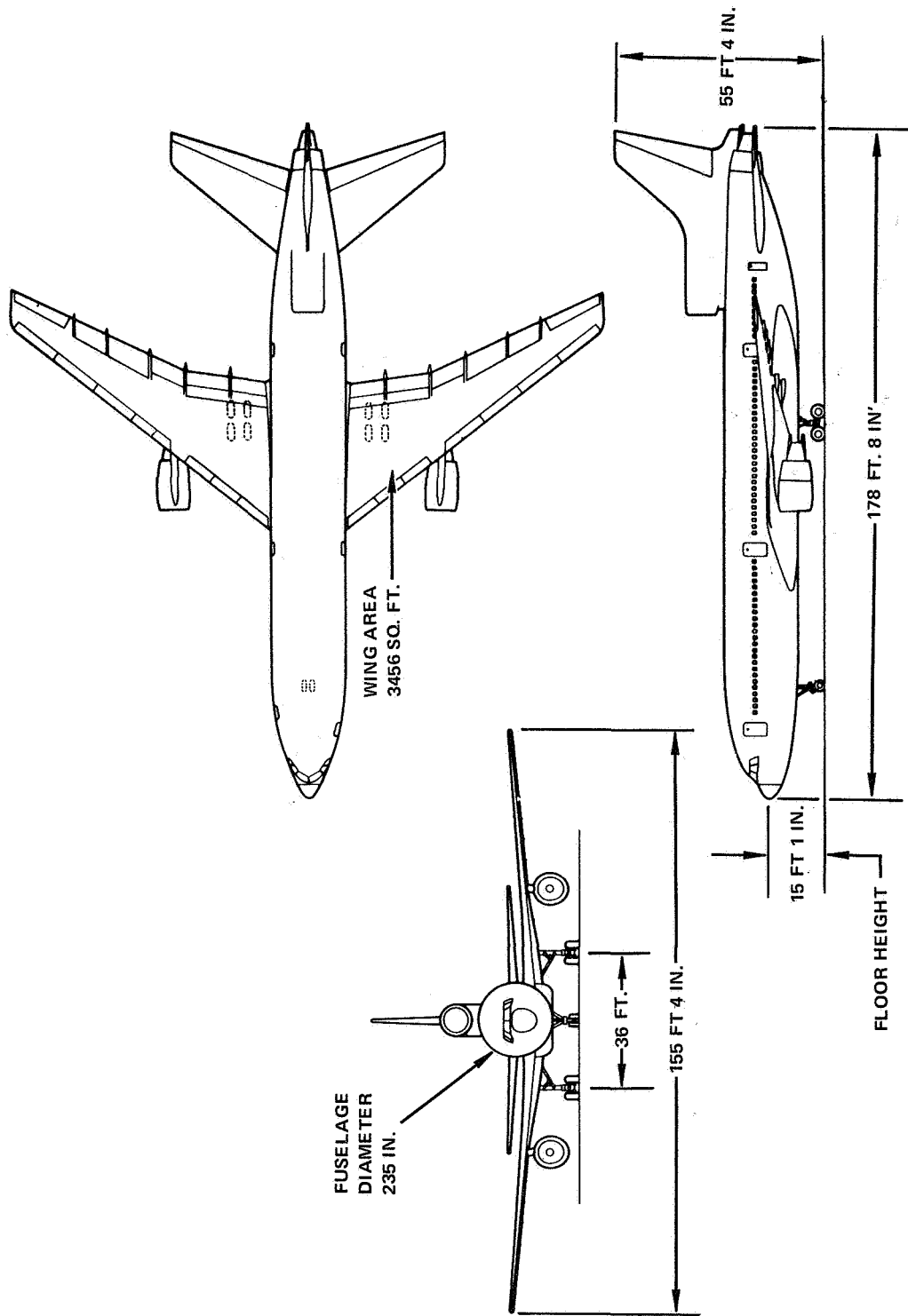
Figure 1   L-1011 General Arrangement
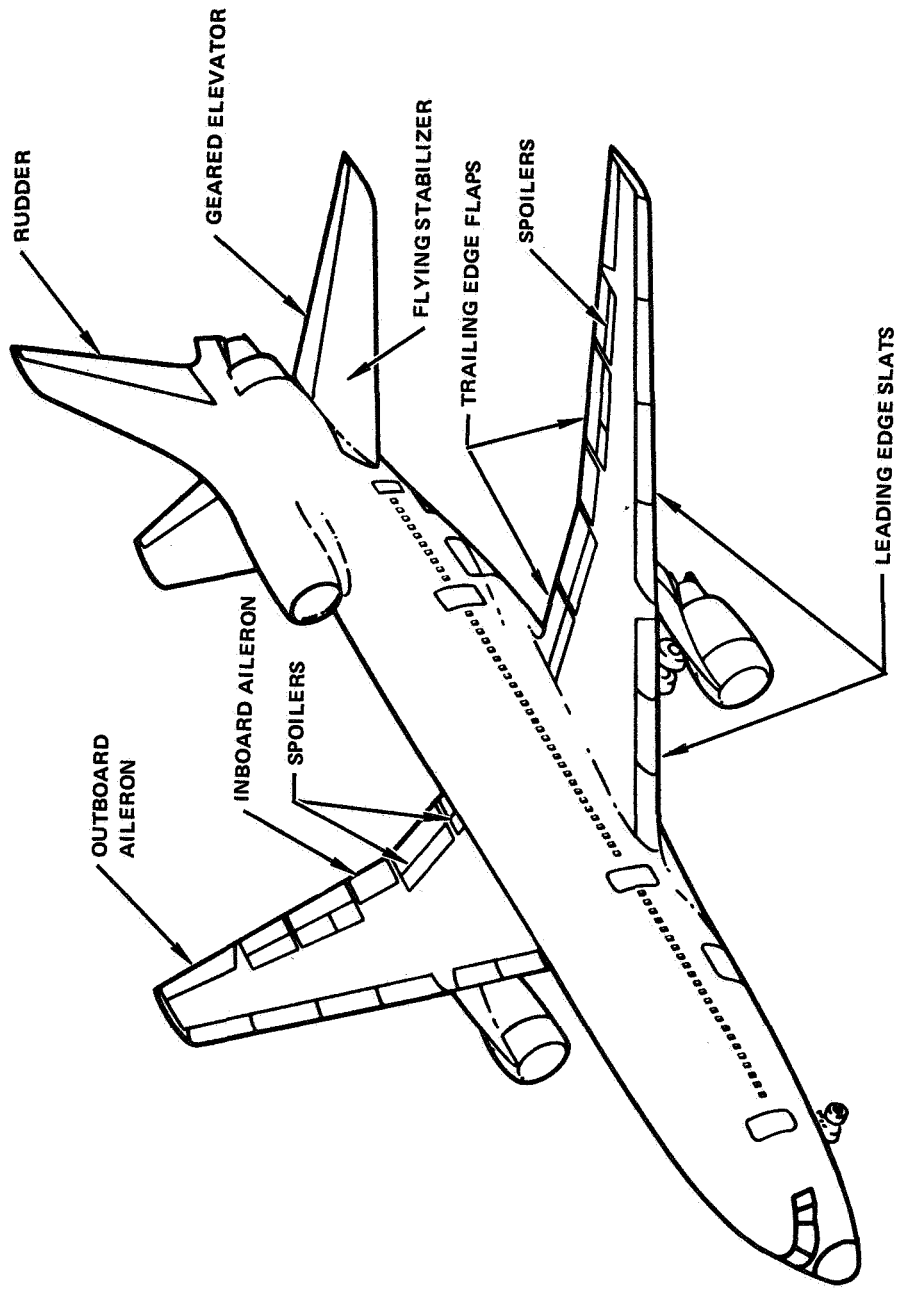
Figure 2 Control Surface Arrangement
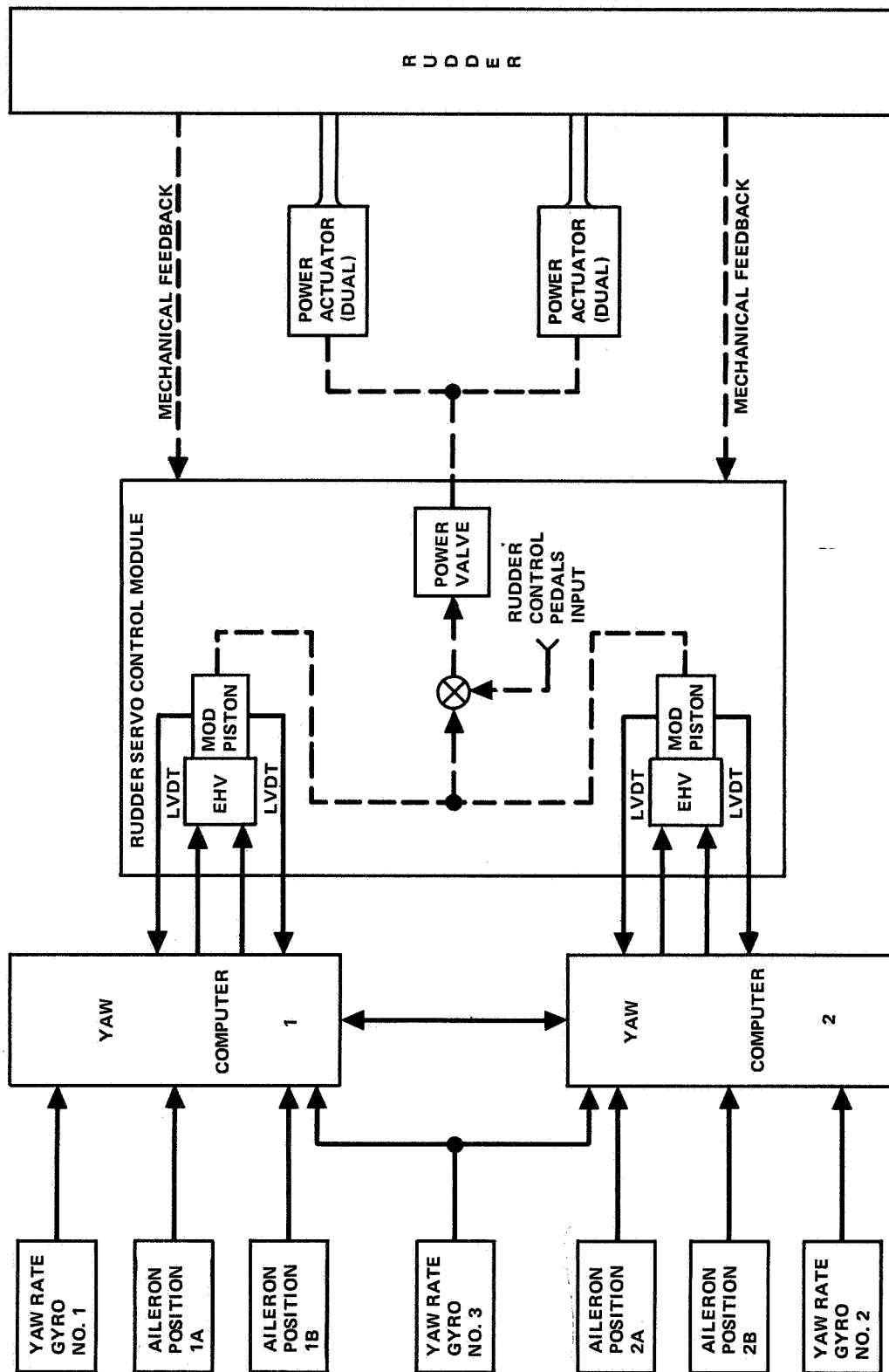
Figure 3 Stability Augmentation System–Cruise Mode

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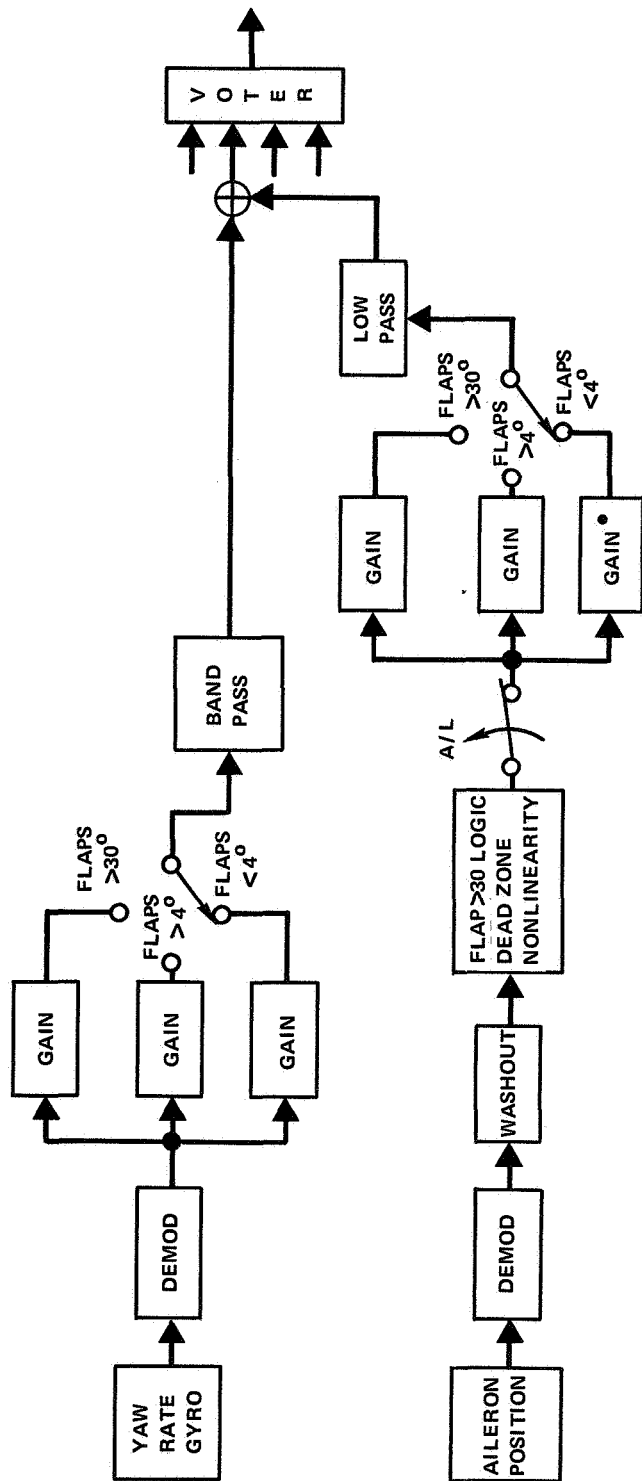
Figure 4 Yaw SAS Cruise Mode Computation
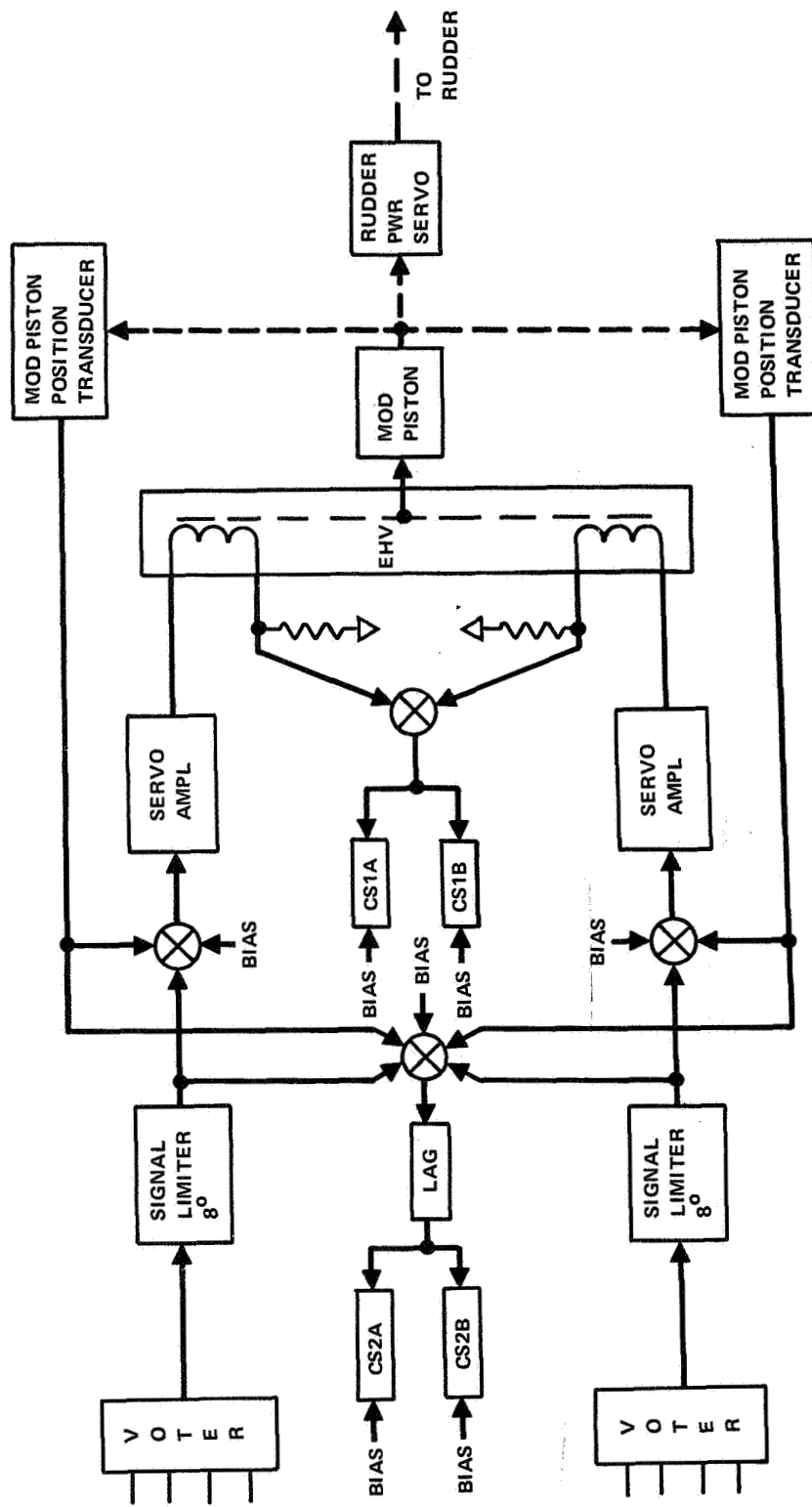
Figure 5 Yaw SAS Servo Loop
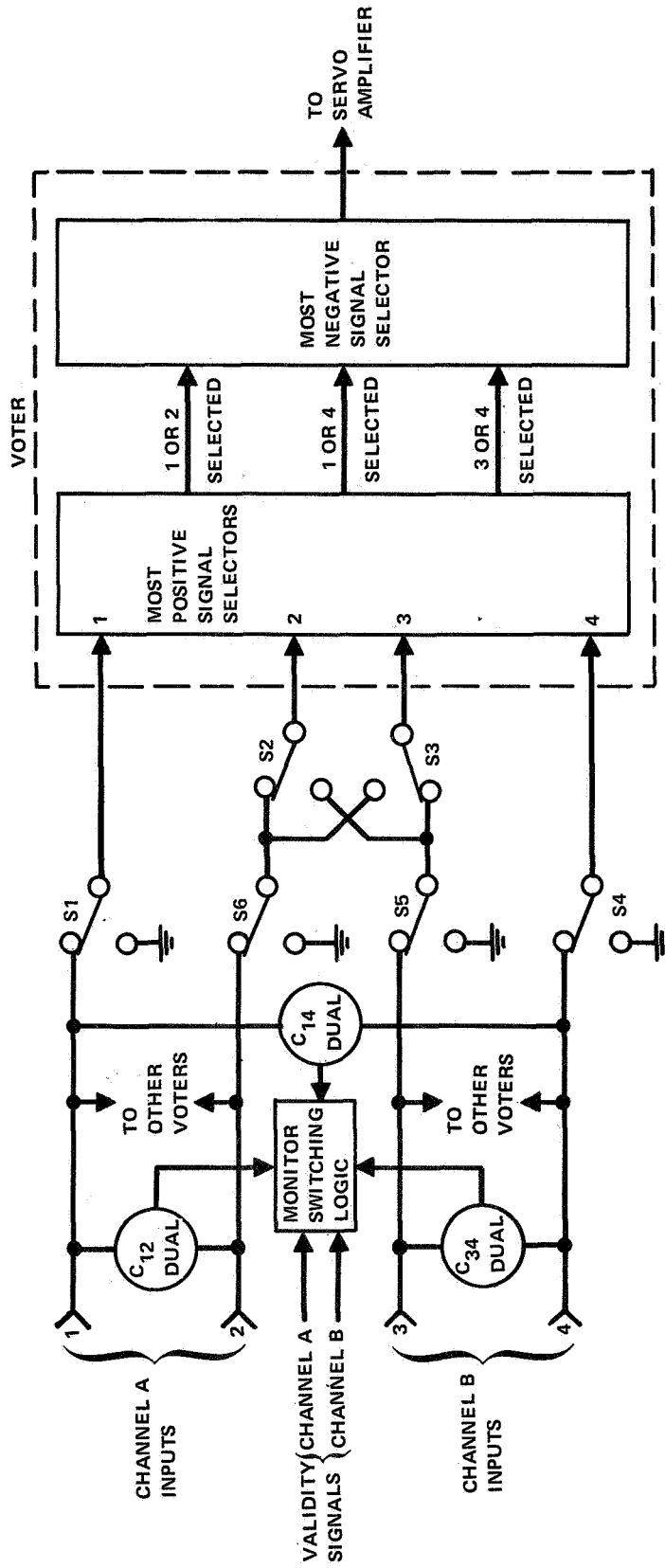
Figure 6 Voter Input Switching

797
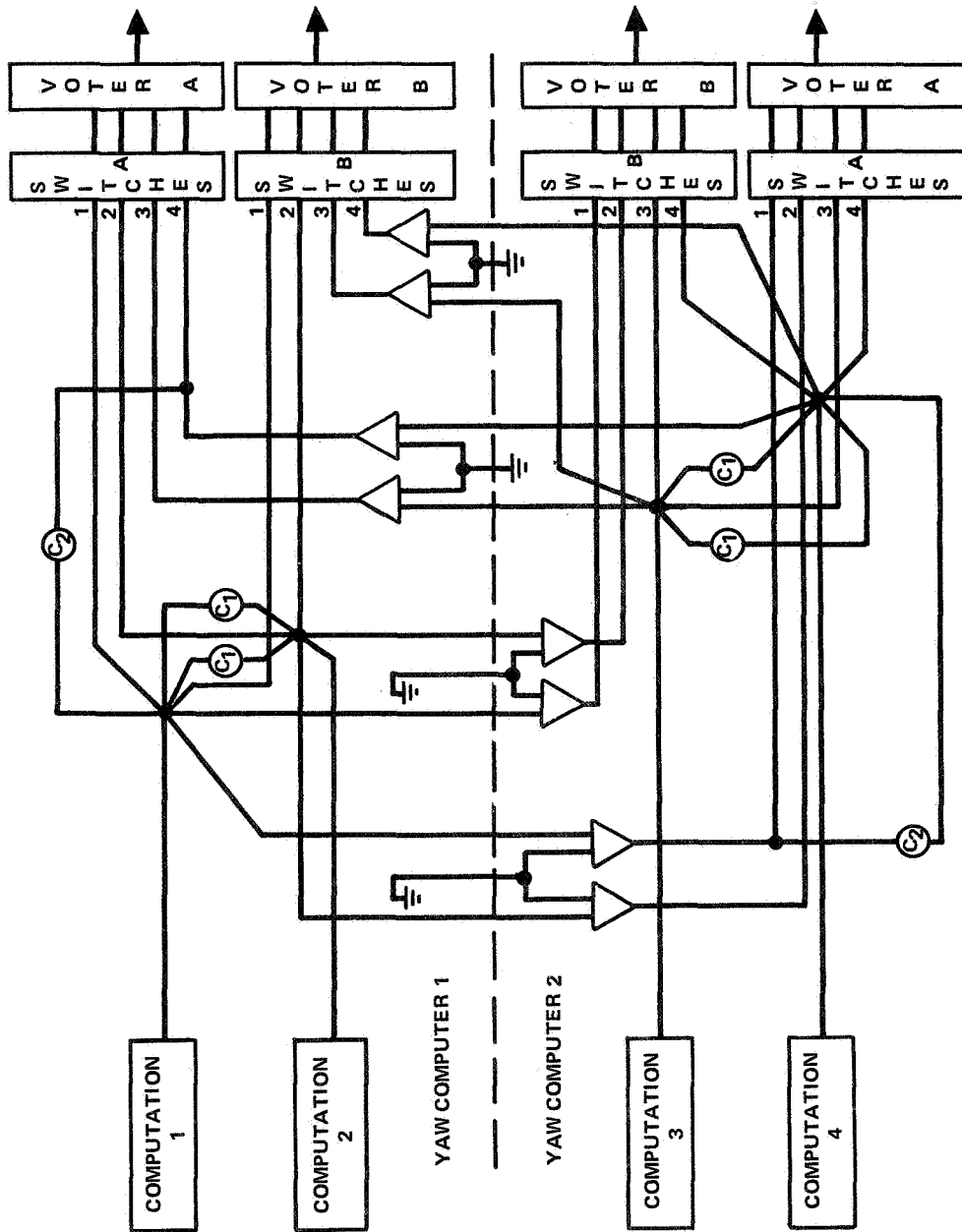
Figure 7 Crossfeeding of Computation Signals

798
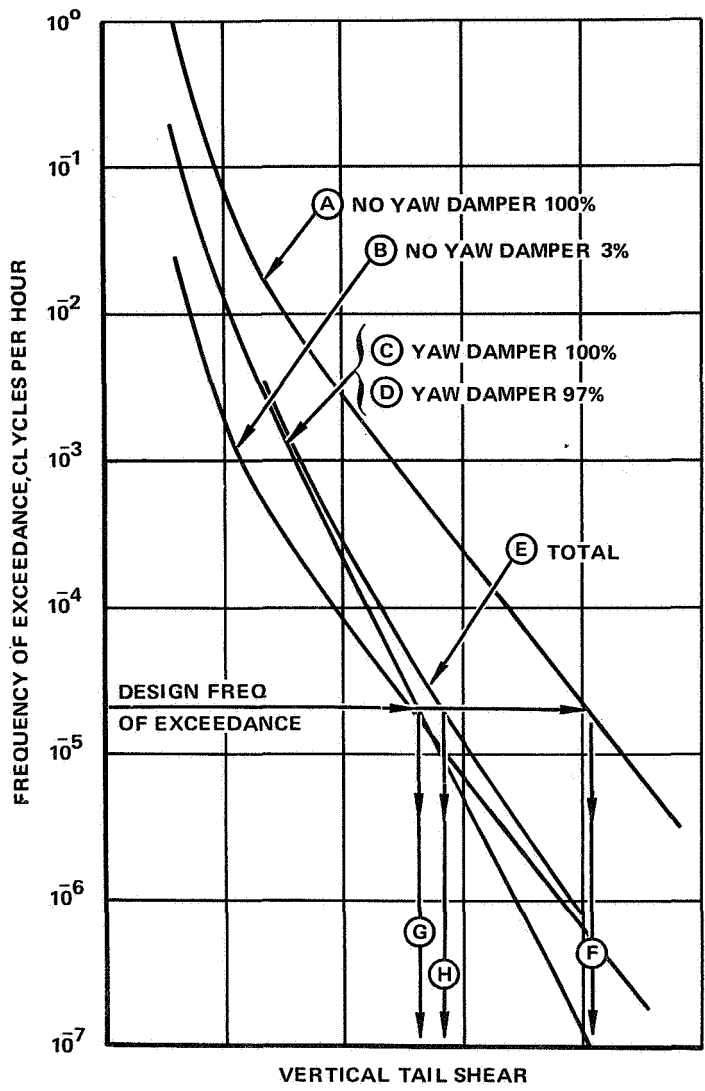
Figure 8 Frequency of Exceedance of Vertical Tail
Shear With and Without Yaw Damper
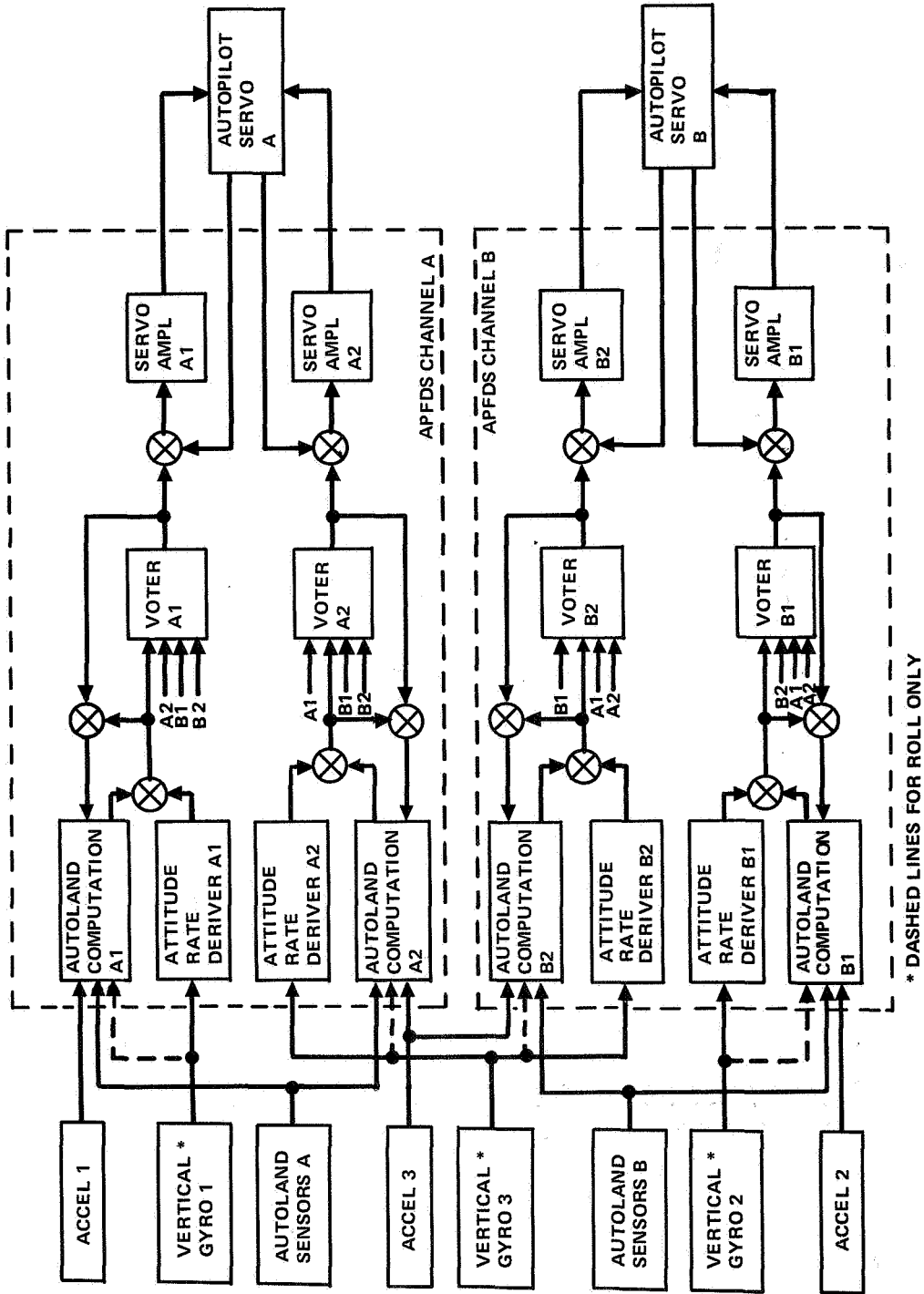
Figure 9 APFDS Approach/Land Mode Configuration

800
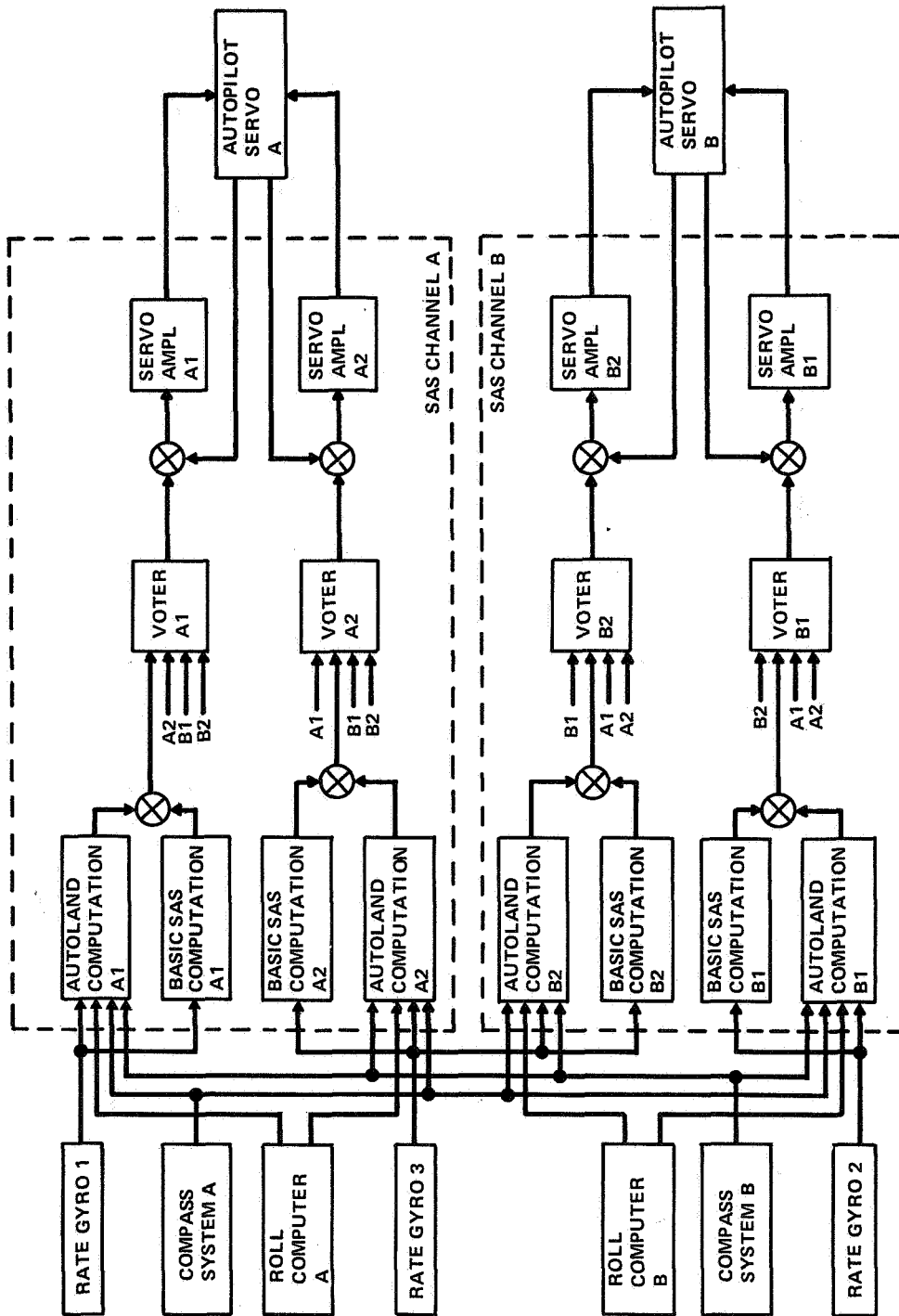
Figure 10 SAS Approach/Land Mode Configuration

801
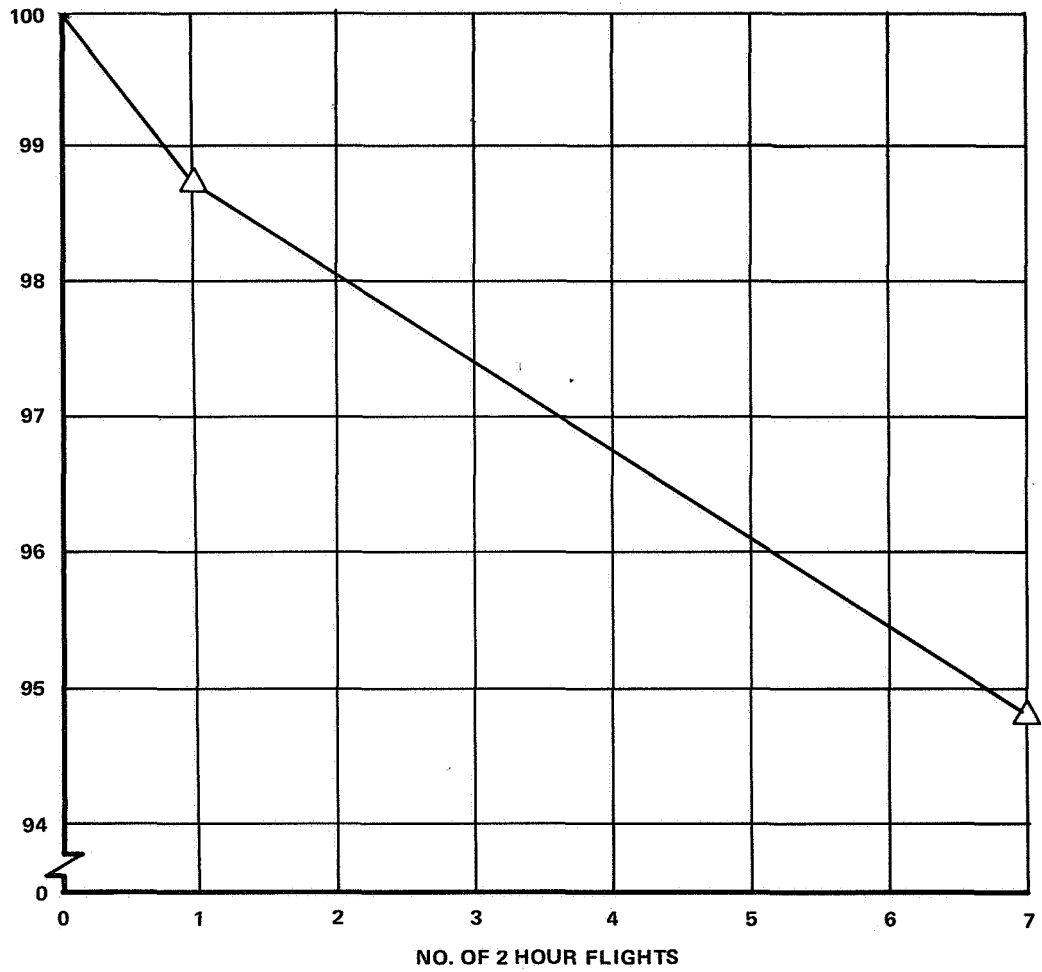
Figure 11 Automatic Landing System Availability - Cat. IIIa

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THE ACT TRANSPORT—

PANACEA FOR THE 80'S OR DESIGNER'S ILLUSION?

Panel discussion

Active control technology is being promoted as a panacea for the transports of the 1980's, reaping performance gains, fuel savings, and increased return on investment. Are these projections realistic or merely designer's illusions?

A panel discussion was held at the symposium which attempted to make an objective and pragmatic assessment of the standing of active control technology. The discussion focused on the standing of active control technology relative to civil air transport applications, the value as opposed to the cost of the projected benefits, the need for research, development, and demonstration, the role of government and industry in developing the technology, the major obstacles to its implementation, and the probable timing of the full utilization of active control technology in commercial transportation.

The panel moderator was Joseph Weil, Director of Research at the NASA Flight Research Center. The panel members were William E. Lamar, Deputy Director, Air Force Flight Dynamics Laboratory; Richard P. Skully, Director, Flight Standards Services, Federal Aviation Administration; Arthur J. K. Carline, Manager, Advanced Transport Technology, Fort Worth Division, General Dynamics Corporation; Clifford F. Newberry, Director of Engineering, Wichita Division, The Boeing Company; Franklin W. Kolk, Vice President, Systems Planning, American Airlines, Incorporated; and Lloyd L. Treece, Vice President, Flight Operations-Control Division, United Air Lines, Incorporated.

The following is an edited transcription of the prepared statements of the panel members and the subsequent open discussion between the panel and the audience. A list of attendees is presented in the appendix.

T. L. K. Smull: Welcome to the ninth session of the symposium, which is a panel discussion on the topic "The ACT Transport—Panacea for the 80's or Designer's Illusion?" The moderator for the panel is Joseph Weil, Director of Research at the NASA Flight Research Center. This session is being tape recorded, and a transcription will appear in the proceedings.

J. Weil: Some of you may feel somewhat perplexed at this point. You may be wondering whether active control technology and control-configured vehicles are

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ready for general application to advanced transport design or whether they are being oversold. Another question is whether events should set their own pace or the government should increase its support of this technology. The last paper yesterday, by Dick Holloway, provided an indication of what might be done to exploit the new concepts.

This morning we are fortunate to have on our panel six distinguished visitors who will give us the benefit of their experience. We have allowed each panelist the option of using 10 minutes to express any general views he might have on the overall topic of discussion. The panel will then focus its discussion on three inter-related questions: what are the potential payoffs of active control technology, what are the biggest obstacles to its implementation, and what new programs are needed to expedite its use in commercial transports. We are anxious to have enough time to discuss these subjects, because we feel they are extremely important. At the end of the panel discussion, we will accept comments and questions from the audience.

At this point I would like to introduce Ken Carline, who will begin the discussion.

COMMENTS BY PANEL MEMBERS

A. J. K. Carline: What is active control technology anyway? In the past, we've always been sure to relate advances in technology to the way they affect the airlines. I'm talking now in the context of this particular symposium, which is related to transport aircraft, although we've also heard some discussion of fighters.

The payoff—what's the payoff? I'm not sure that we really know what the payoff is. We've heard about taking weight out of the wing based on maneuver and gust load alleviation systems, but then we have a problem with fatigue, and we have to put some weight back in because now the wing has a fatigue life of only 5000 hours instead of 30,000 hours or something like that. So I'm not sure that all the money we've spent and all the studies we've done have shown a payoff yet. I think we ought to determine what we are doing with active control technology—it's got to pay off. Nothing I've seen yet proves that there is a payoff. We've seen General Dynamics and Boeing and Lockheed comparisons, and the benefits vary from 1 percent to 12 percent, which, I think, emphasizes the problem. I think we ought to spend some money on some really meaningful studies, something on the order of $1 million instead of $10,000, and get some meaningful answers on the real payoff. We ought to get the airlines in the act as well, not after the fact, the way we usually do. I was rather disconcerted to find that the panel organized to develop design criteria didn't include a member of the airlines. So I think we ought to determine the payoff. I believe there is one, but I'm not sure how much of one it is.

The other thing we ought to look at closely is how we can get the question of reliability sorted out. And we ought to think about how we could certify an airplane. Then we should implement active control functions, in, say, 10 or 12 cargo airplanes and find out what they do for us. Maybe putting an active control system into a cargo airplane will extend its fatigue life from 40,000 hours to 60,000 hours.

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At the same time, I think we ought to have a good demonstrator airplane with active control functions, and not one where we can only alter the wing because we can't move the landing gear or something, as with the JetStar. We mustn't have too many restraints, or the answers won't be meaningful. I suggest that we build an airplane with a digital fly-by-wire system with no mechanical backup. Incidentally, the Concorde has flown 2700 hours, in monsoons and in Alaska, and they have never had a failure which would have embarrassed them if they had not had a mechanical backup system, so the record is pretty good.

Finally, we should take a serious look at flutter suppression, which I think worries people quite a bit.

First, let us determine the benefits of incorporating active control technology systems individually and in combination. We may need different combinations for different purposes. For a short haul we may need one combination, for a medium haul maybe another, because they have different and sometimes conflicting requirements. The studies should include detailed maintenance costs, where this is possible, and equipment redundancy requirements. Then, if the studies show the systems to be worthwhile, and I don't think we've really demonstrated that yet, NASA should sponsor two programs. In one, a technology demonstrator aircraft, for example a business jet with minimum restraints, could be fitted with the most promising active control functions, including a digital fly-by-wire system, which I believe to be the most promising. In the other, a small fleet of cargo airplanes could be modified to incorporate one or more active control functions in order to accumulate a bank of reliability and maintainability information. These programs would prove or disprove the studies we've done. I also think we'd do well to track the record of the Concorde control system. I talked for 2 hours with maintenance people last week, and they gave me a lot of information that showed its record to be good. They hadn't had any really significant failures. And the track record of the equipment was pretty good. I also think we ought to do a lot more research on flutter suppression. I think we are a long way from taking material out of the wing. Perhaps in time, in some future commercial transport, but I don't think it will be the next one.

R. P. Skully: I would like to start by saying that the FAA anticipates the incorporation of active control technology into civil transport aircraft with confidence and a sense of readiness. I'd like to mention some of the things we have done and are doing to prepare for the application of active control technology and control-configured concepts in the transport airplanes presented to the FAA for civil certification and commercial operation. First, the Federal Aviation Regulations have already been amended to accommodate the unprecedented technological advances of the decades just past. For example, a few years ago the captain's instruments were really kept separate, and when integrated systems came into being, we amended the rules to require that their design be such that the loss of display of information essential to safety in flight would be extremely improbable.

In addition, the operating rules have been changed to recognize inertial navigation systems and low weather minimum landing systems. In both cases, accuracy and reliability had to meet stringent criteria before we would approve using the equipment in operational aircraft. Area navigation systems of varying degrees of

sophistication have been accepted into the national airspace system. Digital distance-measuring equipment has been taken in stride, and altitude alerting and many other systems have come into being within the framework of the existing operational and airworthiness rules. The ground proximity warning system has been certificated and is being used today in some transport aircraft.

When a new aircraft is presented for FAA certification, we might find that it has flight characteristics or design features that were not envisioned when the rules were first written. We then apply what we refer to as special conditions to make certain that the current high level of safety is maintained when these new features are incorporated. Recognizing that our regulations do not always reflect the state of the art, we've initiated a new system of periodic airworthiness reviews. The last such review conference was held as I recall in 1960. Over 1000 changes to the regulations were proposed and are now being commented on by all interested parties in industry and government. In December of this year we will have a public meeting in Washington, D. C. in which the spokesmen for the various organizations will have an opportunity to present their views. We plan to have a 2-year cycle to minimize delays in implementing amendments to the regulations. This airworthiness review conference is being scheduled for 8 working days, and we are anticipating many people from outside the United States.

Another thing we are proposing is the introduction of flight simulation as a substitute for a significant portion of the airworthiness certification process. As most of you know, we have already authorized the use of approved flight simulators for certain pilot certification and proficiency requirements. The simulator will be used to plan and practice the certification flight program and to make preliminary evaluations of new aircraft, so that critical flight conditions can be pinpointed. Flight tests will be limited to the validation of these critical conditions. This will provide a way for industry to test its ideas against FAA standards, and, where appropriate, the FAA can develop new standards to cover new aircraft capabilities. This, in turn, will offer industry the potential for creating new markets and perhaps prompt international competition.

The responsibility for developing this proposal into a successful program should be shared by NASA, the FAA, and the aviation industry. The role NASA plays may be to provide advanced simulators and data reduction facilities. Automated data processing for the simulator data is needed, of course. In addition, NASA engineering support could help the industry and the FAA to become more familiar with NASA's facilities and provide a useful exchange of research information. Industry can provide a mathematical model for the vehicle, validated, if possible, by prototype testing. Industry could also be responsible for the bookkeeping and updating of the mathematical model, provide the engineering and pilot support for the programs, and participate with the FAA during the simulator tests. The FAA can provide engineering and pilot participation in the simulator tests of the vehicle's mathematical model and establish the requirements for aircraft certification. Of course, both NASA and the FAA would assume responsibility for the proprietary rights of the industry.

The FAA is ready to pursue, with your support, new areas of technology, including new applications of propulsive lift, advanced structures, synthetic

stability, digital controls, and other new designs. We're looking forward to working with all these groups in the near future.

W. E. Lamar: The question of transitioning technology is, of course, of considerable interest to people at the Air Force Flight Dynamics Laboratory. The only reason the laboratory exists is to develop new technology and see that it is applied. If it isn't applied, people wonder what the laboratory is for. So the application of new technology is of paramount importance to us.

In answer to one of the comments, I believe we have made progress. I don't mean just the laboratory; industry, NASA, and this nation have all helped to develop the basis for this technology. Many aircraft that incorporate active control technology are actually flying, demonstrating a portion of the technology and in some cases a significant portion. The YF-16 airplane is a brand new vehicle which incorporates a fly-by-wire system and relaxed static stability. So we know how to do it, we know that we can make this technology work.

There are several questions, however. First, what is the real need, and what is the payoff? Now here, I think, there is a lot of room for work. Ken Carline brought this up, and I couldn't agree with his remarks more. Analyses must be made in depth to make it clear that there really is a payoff, and that the payoff doesn't vanish when you get to the suboptimization that results when you look at the whole system. You've got to be sure that the payoffs remain. I remember the Boeing experience and the supersonic transport. They considered active flutter suppression using the flight control system. As I remember, estimates of 9000 pounds in weight savings were made because of the flutter suppression system. As the design progressed and they got into the problems of the total system, a lot of the apparent savings vanished. So you've got to make sure that the studies are in enough depth to have a total system viewpoint. You need that confidence.

It's likely that the first application of this technology will be to provide fixes for current aircraft. The C-5 airplane is an excellent example. Studies of the application of load alleviation and mode stabilization to the C-5 aircraft were made long ago. At that time there was very little need for that technology. Now it is being applied, and I think the papers by Lockheed showed the depth of the studies necessary to find out the best way to apply it. Now, if the application of active load distribution technology improves the aircraft's life by a factor of two, the improvement is significant; it's a tremendous payoff, one that essentially saves an airplane, because you fly double the time.

I think that there are a lot of cases in which this technology will be used to fix problems, but if it is going to be applied to new aircraft I think we'll have to have a crisis of some type, or a national need. That means we need the technology in hand, ready to go. The space program got started because of the Russian sputnik, and the intercontinental ballistic missile program got started because of the missile gap. We couldn't get any money for structural development until an F-111 wing fell off, and then we ran into problems with the C-5 airplane. There has to be a crisis of some type. Sometimes it is in a safety area or in a C-5 type of area.

To apply this to the airlines, and I must say to the Air Force too, maintainability must be determined, because overhead and maintenance costs are taking a big portion of the total dollars available. If this keeps up we won't have any money for new systems, so we'll have to do something about insuring maintainability. We've got to make sure the risks and uncertainties are understood, that there are no surprises that appear after we fly a number of months. This has happened with many aircraft, like the T-tailed aircraft, for example. So you've got to know what the costs are, and our ability to ascertain costs in advance is really only in a beginning stage. People don't have much confidence that we can estimate costs properly. We need a thorough study to do that.

Then there's the question of criteria, specifications. Military specification 8785B in principal provides the criteria, but there is a need for a specific meeting on reliability requirements. The specifications are undergoing revision. I think what's required here is a concerted effort to determine acceptance criteria. There are pretty good criteria for engines: they have a 50- or 60-hour preliminary flight rating test (PFRT), and if an engine passes that test, it's considered adequate for a new airplane. Later, there's a model test, which is more thorough, and when it passes that it's ready to go into production. Now, because of some engine problems they are now changing the engine specifications somewhat, and trying to tailor them more to the usage requirements of the airplane. We need to do the same thing in the flight control area. We need to understand just what the technology people must do to prepare the technology for transition. But this means that the users have to get together with the certifiers and the contractors and agree what kind of proof is necessary to make the transition in the technology. Then maybe we can start filling the gaps.

And there are quite a few gaps. For example, we're still not sure about the effects of lightning on fly-by-wire systems. Right now we do not permit our F-4 fly-by-wire airplane to fly in lightning. We do not permit the YF-16 airplane to fly near lightning either. I'm sure that as the program proceeds things will be done to determine the effects of lightning. These are unusual problems, but we've got to solve them, and make sure we are completely ready for operation. We've got to get clear acceptance criteria, and right now they are not clear.

There are many approaches that one can take to application, but certainly a fix-up approach, as on the C-5 airplane, where the technology is applied step by step in nonflight-safety areas, is the first step. For example, when gust load alleviation is applied and it works, you get gust load alleviation and you save some fatigue damage. When it doesn't work, you get a little more fatigue damage on one flight, but next time you fix it. The problem is to make sure that it doesn't screw up some other system and interact in the wrong way from a flight safety standpoint. That's a way to apply the technology safely and get experience. Certainly the Air Force is getting a lot of experience with command augmentation systems. They are basically the same as fly-by-wire systems. They work, and we get a good understanding of their reliability, so we are much more willing to go to full dependence on electronic systems.

From the airline's viewpoint, I would think that putting a system in a nonpassenger cargo airplane might be a good way to acquire experience with the technology.

Maybe the pilots will want ejection seats, which is different from normal airline practice, but it's a way to get lots of time and experience with the technology in the airline environment. You can also apply the technology with systems that have backups. When we first flew the F-4 fly-by-wire flight control system, it did have a mechanical backup. After flying a while we had enough courage to take it out. So keeping the backup in at first may be a way to build up enough confidence to take it out. Later on, you can apply it to completely new designs. Again, there ought to be clear acceptance criteria. I think these are things we need to do.

C. F. Newberry: When the apostle Paul wrote his letter to the church of Corinth, he commented that they compared themselves among themselves and commended themselves. He said that if their spiritual life was as good as they indicated, it should have affected the way they were living. I've spent 2 days at this meeting now, and I think I have somewhat the same feeling. As we compare our technology as experts among experts, we should ask ourselves why we aren't using this technology. Dick Holloway addressed this question a little bit yesterday, and I'd like to reconsider some of his comments and questions.

First, we're faced with a balance between the benefits and the risks of this technology. We want to tip the scales in favor of the benefits. The risks are safety and economics. From an airline's standpoint, the economic risk may be the system's reliability and maintainability. From the manufacturer's standpoint, the risk may be product liability or the cost of retrofitting a fleet if the technology is introduced into an operating fleet prematurely. The other part of the economic risk may be letting the competition get ahead of you. The benefits, of course, include such things as lower cost, better performance, or both.

There is a decided difference between the acceptable risk-to-benefit ratio for the military and the commercial airlines. The military often has the opportunity to test new systems in prototype airplanes or at least to fit the system into an experimental situation and to try it out to evaluate the risk before committing itself to production. This is not generally true in the commercial airlines. The one notable exception to this is the Boeing Model 367-80 (Dash-Eighty), which introduced the -707 fleet. There again, it was a high risk, high payoff situation. Therefore, it important for the risk to be minimized before introducing new technology into commercial aviation and expecting it to be accepted.

We've reviewed various aspects of active control technology in the last couple days, and all of us can draw our own risk curves. There are different levels of risk for different concepts. The noncritical aspects, such as load alleviation, fatigue reduction, and ride control are pretty well accepted. I think the risk of producing these would be low. If there is a problem, the airplane can recover safely after it is switched out of the system. What little reduction we might have in fatigue life during landing would be of no consequence.

The fly-by-wire and stability augmentation systems are a bit more risky. I think the date of application depends on whether the application is military or commercial. I would like to congratulate General Dynamics for applying a fly-by-wire system to the YF-16 airplane. If they're successful, the next military application will be a lot easier. If they're not successful, it's back to the drawing board

for all of us. But I might remind you that on many of the commercial airplanes some of you will be going home on, the system is "fly-by-fluid," and that of course was not too acceptable a few years back. The flutter mode control system is in a more experimental stage, and I think we will have to do much more work in this area, looking at explosive flutter and other aspects, before it will be accepted.

Not only does our technology need to be developed but we need to understand its applications. We have the ability to evaluate the performance benefits of the concepts we can flight test. The benefits of the concepts are configuration sensitive, but we have a reasonably good ability to flight validate them. However, when it comes to the ability to make predictions on the basis of preliminary design techniques, we come up rather short. You've heard discussions of the ability to represent airplane structural modes for paper airplanes or for newly designed and introduced airplanes. Well, I don't share quite all the pessimism, but I do think that we need to do more work in this area.

What I feel is lacking, however, is persuading the designer to take full advantage of these concepts. Ask how many rivets a designer leaves out of an airplane because he has an active control system. Or how much thinner he is willing to make the lower wing skin because maneuver load control is available to him. Our experience to date is that active control technology has been used like Band-Aids. We've been willing to patch up the deficiencies of existing airplanes by using some of these concepts. A history of active control technology applications over the past 10 years includes the B-52 airplane, which had a stability augmentation system that was developed in 1964. The B-52 airplane was designed as a high altitude bomber. In 1958 it was given the role of flying low, and it didn't take it very long to develop a fatigue problem. Now, this stability augmentation system was designed to alleviate part of that fatigue problem, a Band-Aid, if you will. We generally refer to this system as the ECP-1195 system. That system started at the same time as or slightly before the research program called load alleviation and mode stabilization, and there is a 7-year period from the time the program began until load alleviation and mode stabilization was incorporated in a fleet. Now, perhaps finishing the research a little earlier would have reduced the time; however, we saw from Tom Disney's report that it is taking several years to incorporate active controls in the C-5 airplane. And again it is a case of patching up a deficiency, a Band-Aid.

We've also had intensive research for 10 years in the area of active flight controls or control-configured vehicles. On Tuesday Dr. Kurzhals showed bar charts indicating that research in active control technology would take another 8 years, and if so I question the idea that we're on the threshold of a revolution. Instead, we're just continuing an evolutionary process, and maybe that's the way it ought to be. However, if it's true that we require an additional 8 years, I think we ought to change our acronym from CCV for control-configured vehicles to CCC for creation of control careers. Perhaps I'm being a little unfair or impatient in wanting to get on with it, but I believe that NASA has an important role in bringing active control technology into usable shape.

Dick Holloway mentioned yesterday, and I'd like to reiterate, that we need an airline type of airplane to fly with these concepts incorporated in it. It should fly

an airline route, and it should be subject to the same conditions the airlines are subjected to each day. I don't feel that this would be exorbitantly expensive. I think it would be research money well spent. I've heard comments on various ways to bring this about, and I think we ought to consider some of these and investigate this way to spend some of the research money. The other area I'd like to suggest that NASA do some research in was mentioned by others, including Dr. Perkins, and that is preliminary design. So let's create a real design, using some of these concepts. So I say to NASA, you get the money, we've got the ideas.

F. W. Kolk: I think economics is the key to all this. We've got to have a pay-off. We have to have not only a predictable but an achievable payoff. The airline community has been enamored of a number of things that have had a great effect on our airplanes but which in some cases have had a rather indifferent payoff. One example of that is the all-weather landing system. If we think about it, the all-weather landing system has been worthless so far. That isn't to say it won't be worth something some day, but the admission price has been fantastic and the show hasn't started yet, so to speak. We can't afford another debacle like that. So let's figure out what our real payoffs are and be sure that we get them and be sure that we don't spend too much money getting them.

Acceptance is another problem. In some sessions, people calculated the basic system reliability to be somewhere between 10^{-5} and 10^{-6} (failures per flight hour). Keep in mind that the loss of the airplane is at the other end of this probability thing. I think I also heard Dick Sliff say just a few minutes ago that the FAA is thinking in terms of 10^{-10} for this sort of thing. It seems to me that our airworthiness code is somewhere on the order of 10^{-6} to 10^{-7}. It has been a long time since we were on the airworthiness circuit and had to learn probability, but I think those are the correct numbers.

Now maybe we're at 10^{-5}, but maybe from experience the FAA is right and we need 10^{-10}. It's a long way from 10^{-5} to 10^{-10}. This is going to be a probability game. Now, when we have probability, we have several problems. One of them is that we have a bunch of airplane drivers and they're not much interested in probability. They haven't really been schooled in it as a discipline. All they want to know is whether it will happen or won't happen on an absolute basis; they don't want to be dead. I think we could cause them quite a bit of concern with this kind of thing.

Then, of course, there's the business of the accuracy of the predictions. If you're going to have a system in which somebody comes up and puts a chart on the wall and says it has a reliability of 10^{-10}, how does he know? How can he prove it without spending 20 years testing the components to get failure rates? When you have failure rates like this, it implies that you either have a lot of junk in the airplane with some pretty complex interreactions to protect against failures or you've got things that are so reliable that it's not in your ability to create a failure within your lifetime. I think we have a problem.

Then, of course, there is the infant mortality problem—what happens in the first 500 hours or 600 hours after the introduction of a device into airline service when all of a sudden it doesn't work. Maybe the airline people will understand it and maybe they won't, but we've had to live through a few of these clambakes.

Well, it's not all bad. We've been sneaking up on active control technology for quite a while. Somebody said we've been flying by fluid for years, and we have. We've been flying airplanes around with increasing amounts of power boost to the point where they're really totally powered now. The difference between power boost and fully powered controls is simply the feedback ratio to the pilot. When it becomes infinite, the pilot can't do anything anyway if something happens. There are a lot of airplanes flying with manual reversion provisions. But if a pilot ever has to revert to manual control, he has a pretty limited flight envelope. So we've already faced that problem, although we haven't called it fly by wire.

Another problem is psychological. The airline community has been brought up on hard-earned truths that were learned in the DC-3 era or with early DC-4 airplanes or during World War II. In those days something was reliable if it was a bar of iron or a thick steel cable. If you wanted to have something not quite as good as that but with more muscle, you made it hydraulic. That was somewhat less acceptable. If you really wanted to get fancy and stick your neck out, you made it electric, but the last thing you did was make it electronic, because everybody knew that wasn't going to work and it didn't. Now it's 30 years later and it seems that the order of the reliability of those things has reversed. But most people in the airlines haven't found that out yet. We've got an education problem.

Then you've got the syndrome in which here's the technology looking for a mission. I think that several of our panel speakers have touched on that. I don't know what you do with active control technology. I mean I think what you do is disconnect the pilot mechanically and fly through an electrical system, which sounds pretty good, but I don't know what the benefits are. The full benefits will only come out in a totally new vehicle design, and this totally new vehicle design is going to be pretty hard to come by. Commercial aviation won't be able to afford any kind of totally new vehicle design for a few years. So since we're not going to have an immediate chance for a full-scale application, I would say you have to slug it out and find out what you can on an interim basis, Band-Aids if you will, and keep on making improvements.

L. L. Treece: When Joe asked me to appear on this panel, he asked me if I wanted to prepare a 10-minute speech. I said I thought I'd do everybody a favor and not prepare a speech. I do have some thoughts on what I've seen here in the past few days, though. First, I'd like to repeat that those of us from the business end of the airplane are interested primarily in safety of operation and the creation of enough redundancy to insure that. Of course, to prove this we need an adequate test program. In addition, the economic consideration is all important to the industry. Some things have been presented here that look very appealing from an economic standpoint. Much work has been done, and I think we've come a long way already insofar as pilot acceptance is concerned. We've seen a degree of acceptance in the Caravelle, B-727, B-747, and other airplanes of the fluid line with the controls instead of the cable we're used to. And as far as the acceptance of fly by wire with the control wheel steering and so on is concerned, I don't think we'll have any difficulty selling it to the pilots once the safety aspects are proven. I don't know too many airlines that are going to put ejection seats in a $25 million investment to launch three pilots, who don't want to go anyway under those conditions, into the air to test a system. We're going to have to find a better way to do it than that.

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PANEL AND OPEN DISCUSSION

Weil: What I would like to discuss first is the application of active control technology to a medium- to long-haul conventional transport aircraft.

Kolk: I believe that there will be great tangible benefits in incorporating active control technology to these aircraft. The performance of the airplane will improve and therefore fuel consumption will decrease because the center of gravity will be back where the horizontal tail helps rather than hinders. This implies flying some unstable airplanes and also implies putting the landing gear back where the airplane won't go over on its back when it's taking off or sitting on the ground. Certainly this is of paramount importance if we're ever going to have a supersonic transport. So I think I would make that an objective, probably an initial objective. I think the other possible objective is to take weight out of the wing by using a system for load relief. I think this is where a failed system will be a big problem. You can actually fly an unstable airplane, provided that it doesn't get too unstable statically, but it's pretty difficult to fly an airplane without a wing because something electric has failed. I would put that order of priority on it.

Carline: I think without question relaxed static stability and fly-by-wire control systems should be considered for the next airplane, possibly with backup systems. Then, possibly at the same time, I think there's a case for improving the fatigue life of the airplane. We've had a lot of cracks in airplanes that have cost a lot of money to repair. I think the use of active controls to improve an airplane's fatigue life or to reduce the incidence of fatigue damage would be a good objective for the era we're talking about.

Kolk: Yes, but I'd like to point out that the budget is limited. Take an airplane like the B-707. It sort of gets tired at 30,000 hours, so you reskin it for a couple of hundred thousand dollars and get another 30,000 hours out of it. Other airplanes I'm familiar with, like the DC-6 airplane, have gone through this cycle; in fact, I think there are DC-6 airplanes flying around that have been reskinned twice to keep them going. And it's actually a pretty economical way to extend the life of a structure. So unless you can do the job cheaply enough to make it cheaper than just reskinning, you haven't saved anything.

Carline: It's a question of economic payoff. It's a trade, if you like—you've got to weigh one against the other. What about the audience?

J. A. Gorham: A few years ago I had some responsibilities on the L-1011 airplane to do with controls and cockpits and avionics, and I was guilty of persuading all the airlines to try all-weather landing systems. I'm not going to argue about that right now.

Much has been said about the possible benefits of active control technology in terms of saving structural weight and space in the airplanes, center of gravity static margins and so on. I firmly believe that active control technology has a role to play, and I think that with an intelligent program by NASA and industry we'll find out what it is and we'll make the tradeoffs. I think enough has been said

about the need to make tradeoffs with maintainability and reliability and the need for engineering proofs. The idea of the 1985 era worries me a little. Do you mean beginning to design a new transport in 1985 or that it will be flying then? I think a better definition of what we mean by the 1985 era would be useful.

Weil: We had in mind putting it in operation shortly after 1985, not starting a cycle that might go to 1995.

Gorham: I don't believe that. Looking at some of the timetables, I think it might be possible to begin a new airplane somewhere between 1980 and 1985, but not to have it in operation by that time if it's completely control configured and employs active control technology to a reasonable degree.

One thing that hasn't been mentioned is the role active control technology could play in cockpit design. One of the things that I would like to have done with the L-1011 airplane, since a quadruplex automatic landing system has been installed that can be depended upon in landing, is to cut all the cables and take out the control column, which obscures the lower row of instruments. So if we are going to have fly-by-wire systems, let's get rid of the control column and improve the display area in the cockpit. That's obviously one advantage.

Secondly, we have programs at Langley, which I'm concerned with to some extent, on the B-737 terminally configured vehicle in which we're developing all sorts of advanced electronic displays. As most of you who have been involved with commercial or even military transports know, there just isn't any place to put electronic maps except behind the throttles, so let's get rid of the throttle levers too. If we're going to go fly by wire on the primary controls, we can do it on thrust control as well.

In other words, there are advantages up at the front end that to my knowledge haven't been mentioned during this symposium. That's where the pilots are going to see the benefits, and that might just help us persuade them that this thing is worthwhile.

As far as major obstacles are concerned, I think most of them have been discussed already. New commercial transports are begun, not because we plan it or want it, but because of competition. There will be more world competition now, not just competition within the United States, and when we start to race, if I'm anywhere involved, I don't want to try on a new pair of track shoes. I want to know that the shoes are going to take me to the end of the race already.

J. J. Tymczyszyn: I think we're all missing one point, and that's the application of active control technology to vortex wake turbulence alleviation. We all realize that active control technology is a powerful tool, but it hasn't been fully explored yet, and that is one point we should be thinking about in the near future.

Weil: That's a good point, except that some of the small airplanes that intersect these wakes probably won't have it.

W. G. Wells, Jr.: We've been listening, of course, to a discussion of 5 or 6 years of NASA's experience with the merits and demerits of active controls. I'm keenly interested in Mr. Treece's and Mr. Kolk's comments, also Mr. Newberry's, about the great need for reliability, acceptance by the pilots, and economic viability. NASA has worked with the F-8 digital fly-by-wire airplane, but my question is addressed to Mr. Newberry's recommendation that someone ought to fly an airline type of airplane for an extended period of time to acquire this type of information. Now, NASA has not been in the protytype business, and it seems to be buffeted in various directions—told to get out of or into the prototype business. I'd like to get some type of reaction from the panel as to whether this is an appropriate thing for NASA to be involved in in the future. That is, whether it should undertake a proto- type program.

Newberry: Well, in my view, the role that NASA ought to play is one that's helpful. In the early days, NACA developed airfoils when other people didn't have the opportunity or capability to. I think that through the years NASA has become oriented towards basic research. Certainly I'm in favor of basic research, but I think that any government agency ought to assist and not to resist. If indeed it's the greatest help to put this airplane into service, to drive it around and develop so many hours, and as Frank mentioned it takes a long time for confidence to grow, I think it needs to be done in a good environment. Ted Bowling gave a paper on B-52 stability augmentation system reliability, and in it he showed a growth curve. The growth curve had a slope of 0.56. Now, the average growth curve for those is 0.3. The reason for the difference is that a great deal of attention was paid to that particular program. Something was done about every little thing that showed up. We recently installed a forward-looking infrared system in a low light television system on the B-52 airplane and again we had an extensive reliability program. That program too had a growth curve of approximately 0.56. It takes a lot of attention to get reliability. It takes the minute examination of resistors, solder joints, and what have you. There's just no substitute for time, and we need to get started if we're going to have it. So I think that a prototype program is a good project for NASA. It's a project the rest of us can't afford. I don't think the air- lines can afford to set an airplane aside and fly it without passengers just to get time. I don't think a manufacturer can do that. I think this is a role that govern- ment can play and can be helpful in doing so.

Kolk: I'd like to add to that. I think that not only is the role a proper one for NASA—it's also a role that in other areas NASA has already begun to play. I think that NASA's role in the JT8 refan program is similar, and the end result looks like a finished product. They have also worked in other programs like this, and I think it's a good thing for the total community for NASA to do it. Now, in terms of actually choosing an airplane to fit with active controls, you have to look around at what's available. If you want to use an airplane that will eventually have other uses, you'll have some other problems besides the control system. You'll have to have an airplane with a lot of redundancy built into its design. The earlier genera- tion jet transports do not have this redundancy, and you'd wind up with a whole new airplane by the time you built it in. However, the newer airplanes, particu- larly the trijets and the B-747 airplane, do have redundancy built in because they all envisioned all-weather landing systems. So the guts of the airplane can take it and these may be the airplanes to use. I don't know how you're going to make the

transition from getting that kind of hardware together and demonstrating that it'll fly to putting it into passenger service, which is the only way you're going to get the kind of time on it to prove it.

A. B. Barraclough: I'd like to address the question of the obstacles to greater commitment to active control technology and turn it around and ask what can be done to aid its application, specifically by NASA. One thing that hasn't received much attention is the requirement for reliability data. The requirement is to acquire data in such a way that the data can be used at the drawing board level. One of the significant international benefits of the last generation of aircraft was that there was a data base you could go to and find out the reliability of a given component and how the airlines used it. You could go to a maintenance manual and see where it was used in the system and what it looked like. You could rearrange it to use in your own system and come up with some reasonably good probability figures which told you its safety, its unscheduled removal time, its mean time between scheduled removals, and its maintenance man-hour costs—in effect, everything from its cost to its everyday usage. With some useful trade factors, you could then compare all of these different costs, put them on a unit basis and come up with some kind of total trace of cost. You could compare an electrical system with a mechanical system, a pneumatic system, or whatever. You could then go to the chief designer and say that this system was better than this one on a rational basis. He could of course decide one way or the other. But it is a useful tool. One thing that can be done with the electrical systems is to set up a system that allows information to be collected that can be used at the drawing board level. This requires familiarity with the information system, the ability to become familiar with it, which means some kind of publication, and finally dispersal throughout the industry. So the question for industry is what they can do for NASA along these lines.

Weil: I gather that you're suggesting that NASA or some government organization underwrite this type of thing?

Barraclough: No, I wouldn't say NASA specifically, but I think there is a major obstacle, which is that we don't have a data bank with which we can compare things.

Lamar: The Air Force has a system much like the one you're discussing. The system collects data in quite some detail on component removals, the time between removals, and the cause of the problem. The data are analyzed right down to the basic level. Of course, the problem is that that kind of information does not exist for the new systems we're talking about because there is no flight experience with them. It does exist for a lot of command augmentation systems that do have electronic components, however.

Barraclough: I understand. I didn't mean to bypass the Air Force system, but the point is that there is no system that addresses itself to the question of active controls and flying controls by wire.

Weil: We've heard quite a bit about ongoing programs and programs that are planned for 5, 6, or 7 years from now. The space shuttle certainly is one. How much confidence are these programs going to produce compared with what exists

right now? Is there any way to increase their relevance or to change their direction in such a way that they could be made more pertinent to the airlines?

Kolk: The problem is that active control technology is sort of a technology looking for a mission. I would like to have a better idea of exactly what active control technology will accomplish when it is applied to transports. That will provide a road map for making decisions, and until you have one you cannot address the issue intelligently.

Weil: This is a little far afield from the conventional transport area, but the Boeing YC-14 airplane has a digital flight control system which is pseudo fly by wire. It will fly within the next couple of years. If we have a reasonable number of hours on a vehicle of that type, would that provide the type of confidence needed for a long-haul conventional transport?

Kolk: Every little bit helps!

Newberry: Another question is whether ongoing government and industry programs adequately address the obstacles to the use of active control technology. Unless I misunderstood, Mr. Skully said that in 1960 the FAA held a conference to update its regulations. From what Bill Lamar and I have presented, most of the action has taken place since 1960. Now is that the aggressive action the FAA is giving us in regard to these regulations or did I misunderstand?

Skully: Frankly, the FAA has been putting out regulations on a more or less ad hoc basis over the last decade. The FAA is following the Concorde activities. The French and British hope to have it ready to be certificated next spring, and of course that is a fly-by-wire piece of equipment. I had the privilege of riding in it from Boston to Miami and returning, and there were a few things going on that surprised me. The approach mode was made with the autothrottle. A question was raised earlier as to why you have the throttle, and it's a good point. The throttle is there just because it's traditional. The captain was flying the Concorde manually, and he programed his airspeed with the autothrottle. The autothrottle was just providing the thrust necessary to maintain his reference speed.

We're looking at our landing distance requirement again from a certification standpoint. The Concorde doesn't have flaps, and it doesn't have spoilers. We are working with NASA quite actively to try to determine a better way to assess runway slipperiness. All these efforts will help to establish or modify the regulations.

C. L. Seacord: There are two rather new programs that are intended to address the obstacles. One is to determine the measure of acceptability of the advanced systems.

We've had some experience recently with trying to find out what's required for the autoland sensors in terms of reliability for the all-weather landing system. Maybe integrity is the right word these days. It's extremely difficult to find a realistic, usable failure rate probability number. There's talk about changing the probability from 10^{-7} to 10^{-9}. When you look at the reason for doing so there really

isn't one. Neither is there a good way to measure what we have. I think one worthwhile activity for the FAA is the reevaluation and restatement of the integrity requirements and the way in which the requirements are measured.

In addition, there is a series of operations that could be performed to generate the data that the airlines would like to have and undoubtedly need. They don't need to have a prototype airplane or two prototype airplanes flown a few hundred hours a year. They need, as several people have already mentioned, data for on the order of 50,000 hours of flight in a realistic transport environment. The only way these data can be obtained is by installing some of this equipment, representative fly-by-wire equipment, whether it's being used for that or not, on airplanes in scheduled service. Perhaps they operate in a parallel, duplicate way, so you can throw a switch and take it out of the system and the airplane can go on about its business. This is a program that neither the aircraft industry nor an airline is likely to pay for; therefore, I think it is up to a government agency or a combination of DOT, NASA, and the military. I think that even prior to that, though, you need to try to figure out how you're going to run the big program. Because I think that even if someone popped up with $10 million right now and said "Go do it," there would be about 4 years of confusion about what you were going to do and what you would record and how you would analyze what you did record.

So I think you need a program to define the requirements, to determine what is good enough, what's reliable enough, and how to measure it. Then there should be an introductory program, probably involving flight tests of a representative jet airplane, to develop techniques for the large program. The large program would then consist of the government procurement of the systems and their installation and record keeping for them. The systems should be used in regularly scheduled service to produce at least 50,000 hours of data.

G. O. Thompson: It seems to me that programs with clearly defined goals are the ones that make major contributions. I think one reason so much was accomplished in the Saturn-Apollo program was that the goal was so clearly defined. You may recall that in a movie von Braun produced, he stated that that was one of the most important reasons that that program succeeded. It had a clear goal: go to the moon, return, and land safely, by 1970. That goal was accomplished. That goal was kept in front of everyone. It seems to me that one of the biggest problems in active control technology is that neither we nor NASA has a clearly defined goal. I'm somewhat familiar with NASA's plans. I think that one of the biggest contributions we could make would be to motivate NASA's management to establish a clearly defined goal within the framework the panel has discussed and set a time period for that goal.

Weil: As I understand you, you're saying that NASA should bite the bullet and instead of going to the moon establish a goal of perhaps 20 percent to 25 percent improvement in performance or fuel savings and then go after it?

Thompson: I'm saying that NASA needs clearly defined goals for commercial transports comparable to those that were established for space.

Weil: How do you justify that to the Office of Manpower and Budget? I think the answer to that is that we have to run cost-benefit studies, and if we come up with a ratio of benefit to cost of 20 or 25 to 1, and believe it, I think the risk is good.

J. T. Rogers: As a conservative structures guy, I would like to see an effort made to separate the benefits of using control configurations from the benefits to an actual airplane. For example, the load alleviation studies generally have talked about moments, but you'll find when you design a wing that torsion plays a fairly important part and that all the controls we have talked about are large torsion producers. So one of the things I think would contribute a lot would be to separate the items that contribute a large payoff from items that fall in the gray area of "is it or is it not a gain."

Newberry: In this field, as in many fields, we have a great deal of synergism. When we start to introduce one or two things we get additional benefits. One of the things we saw in the C-5 presentation was that it had a restriction similar to one we had on the B-52 airplane, and that is the use of control surfaces that were already there. Those control surfaces were deliberately designed not to stir up structure modes. Now we're constrained to use them to damp structure modes. I think if the designer has some freedom to apply the concepts we're talking about we'll see many more benefits. The fact that there is torsion is obvious if you're going to use only a trailing-edge device. Why not use the trailing edge and the leading edge together and eliminate that sort of thing? You're right, we need to sit aside and look at these benefits as they are, but I think that we ought not be too quick to say that we'll throw out anything under 10 percent. That one thing may be the catalyst that brings other benefits into being, so it becomes beneficial for the total active control airplane.

P. G. Felleman: As far as NASA funding a large program to demonstrate safety or reliability or whatever by implementing active controls in a large fleet of aircraft is concerned, I don't think that is a goal NASA should be involved in. I think NASA should be bringing technology to a state where it is feasible and available. When the cost benefits come along, for example, when there is another 30-percent, 40-percent, or 100-percent increase in fuel costs, the airlines will be quick to look for things that will reduce those costs, and that will make active controls the economically viable thing to do. It happened in the inertial navigation business. Inertial navigation was not developed for the commercial aircraft industry. It was developed for other purposes. When the airlines saw the economic feasibility of using inertial navigation, it became available to them.

Newberry: I don't think it's very progressive to say that because NASA has had a certain role over the years, it ought to keep that role and not step into another area.

J. K. Wimpress: I think I agree with Dick Holloway's comments yesterday, that control-configured vehicles and active control technology are really just a part of the aeronautical engineer's bag of tricks. They aren't going to revolutionize the whole appearance of the airplane. They're just other things that will have to be integrated into the airplane. And I think it's difficult to set goals for that kind of thing.

I think back 20 years when the airlines were dragged into the jet age. At that time they didn't want anything new either. They predicted dire things for the jet engine; they used too much fuel, you couldn't even stand to taxi out with them; nobody knew what their reliability was; they had terrible balance problems; how were they ever going to maintain them. Of course, once jet engines were in service, the airlines found that they set an entirely new standard and that the problems weren't nearly as great as anticipated. The engines used by the first jet transports were military. They were developed for the military and went through the kind of process Bill Lamar discussed for evaluation. If you look at the number of hours on the jet engine at the time it went into commercial service, it was actually quite low compared with the number the airlines began putting on it, and yet the engine served well. In the case of the engines, then, the commercial incentive got to be such that the engine was constantly improved, and engines like the turbofanjet were developed not for the military but for the commercial people. The point is that the airlines were willing to accept an engine entirely new to them on the basis of military experience that was relatively low, yet large enough to be statistically valid. I can see the same path for the fly-by-wire control system. The military will have to take the lead; they'll put it on some of the airplanes they're going to use over an appreciable length of time, and that will develop enough time to be statistically valid and it can then be put into commercial service. In our thinking we should also distinguish between the electronic control and so-called control-configured vehicles. Confidence has to be developed in electronics and electric systems and not in the ability of the control surface to move and create an aerodynamic load that will favor the airplane. The former can certainly be developed in the way that I've described. I think the latter has just developed as part of the preliminary design process.

R. E. Coykendall: I think that we in the airlines are somewhat impressed with what the military has done with some of these systems and the expertise that has been developed. On the other hand, we also feel that the military is somewhat enamored of the airlines' philosophies and practices. That is to say, they are now coming to the airlines, asking us to show them how to maintain vehicles on a long-term basis. This presents an opportunity for a program wherein the military and the airlines pool their information on the maintainability of aircraft and aircraft systems in particular. That could be turned to real advantage in that it would show what the airline maintainability requirement for active control technology really is. Do you agree, Frank?

Kolk: That's basically right. You know, we've got a whole host of gadgets on airplanes that are there for a good reason, and if they go awry, funny things happen. I think one of the most startling pieces of machinery I ever had anything to do with was the stick pusher. We operated a fleet of 30 airplanes for a number of years with stick pushers and I never knew the stick pusher to bomb out on us. It always worked when it was supposed to work and it didn't go off when it wasn't supposed to go off. You can come up with all kinds of examples of things that will have to work full time, with no bail-out route, to take full advantage of active control technology.

So the military people get into active control technology and General Dynamics wants to expand the maneuver envelope for their lightweight fighter, so they make

the tail work for them instead of against them. It was a big payoff in an intensely competitive situation. It's a pretty interesting system, but the point is that at least on the face of it they seem to have made it work and for the first time. This kind of background is going to help. Now I think the airlines should be a little less chary of sharing some of the information that they have. They have so much information in bits and pieces collected over the years that it's a monster of a chore just to get it all in one place. Some of that material might relate to these problems. Some of our experience with electronics may also pertain to some of these things, and I would like to see something set up on a cooperative basis. Certainly we can try. And certainly some of the things we found out about engines are of interest to the military people, because I understand that they have to make them work the first time now or they don't sell them. We have the same problem. All of us are faced with this problem. We've got to minimize risk, and how do we devise a system that minimizes risk? Maybe NASA can serve as a catalyst for this.

I think this meeting is significant, because this is the first time in 20 years that I've seen this many people in a room talking about airplanes. I've been going to meetings for a long time and I want to congratulate everyone for coming and I want to congratulate NASA for inventing some way to get everyone together, which I was afraid was a lost art these days. Just talking like this is going to help. There's something there and we need to use it. It's not a cult. It's a tool, and now it's a question of rolling up our sleeves and getting on with the job. Anything constructive has got to be taken in a constructive way and I think we're all willing to do that.

R. E. Kestek: The problem we seem to be working on is benefits for commercial transport. We pointed out that the safety required to fly your grandmother is of prime importance yet difficult to achieve. How do you do it? You need her on board to pay for the flight unless you have a large amount of money from some other source. In past programs, the airlines relied on the efforts of the military, which I think has some possibilities. Some people have talked about that. Sitting here, an idea occurred to me. There is a commercial airliner in military service that is being serviced by the commercial airlines. That is the T-43 airplane, and I believe it's being serviced by United Airlines. One of our problems is to get the airlines and the military to talk to each other, and here is a vehicle that is identical to an airline vehicle, being flown at high speeds and low altitudes, where fatigue is a problem and ride is a problem. Here is a vehicle with a need for active control technology, and it is being serviced by the airlines, who we are trying to get the information to. It is being flown by the military, so we can install a system in it for a reasonable price. It seems as though that would be a good approach to take to investigate the various aspects of this problem.

Coykendall: To comment on this question, yes, we are under contract to the Air Force to maintain a fleet of T-43 airplanes. Not all of the actual manpower is ours, but the maintenance program is and four of our people are stationed at the Air Force base in Sacramento to supervise the program. I'm not aware of any restrictions on exchanging information in that program.

In this case, the Air Force came to an airline and said that it thought the way the airlines maintained airplanes over the long term had some advantages compared

with the way the Air Force did it and asked the airlines to do it for a while. This presented an opportunity for the Air Force to experience monitoring the results and collecting the information necessary for long-term maintenance. I don't think it's even necessary to have active control technology systems as such installed in those airplanes. What I am referring to is giving the military the opportunity to observe airline objectives and goals in maintenance and maintainability.

C. D. Bardick: If we take the stick out of the cockpit, and I guess we would take the rudders out too, and we take the throttles out of the cockpit and put a couple of little switches in there, I wonder how the pilot is going to feel about looking at the instruments and all the information that is presented to him for the purpose of flying the airplane by hand through the stick, rudder, and throttle. Maybe NASA should look at the interface between the automatic control systems, which are creeping into commercial vehicles in increasing numbers, and the human operator, whose role is changing from being the operator to being more of an assistant manager. Are we, in fact, providing the airline captain with the kind of information he needs to manage these automatic control systems in essentially a nonoperator's role? Maybe NASA should undertake it, because if an airline does it, it's kind of touchy for airline management and the Airline Owners and Pilots Association (AOPA). It's kind of a touchy subject for the Boeing, Lockheed, or Douglas people to get involved with, and it's kind of a touchy subject for the FAA to get involved with, so it seems as if NASA may be the only organization that can touch it without having its fingers burned. Since we have an airline captain on the panel, maybe he would like to address the subject of the flight crew's role in increasingly automatic airplanes.

Treece: I'd like very much to talk about it. First, we accepted the wheel in transports years ago as opposed to the stick, and now we're back to the stick. So I think we're amenable to something new. I think that there is a general movement in the industry to enlarge the role of airline captain to that of manager. You should realize that he's managing a pretty expensive segment of the airlines and that he is a manager. We're encouraging airline captains to manage better, and they have done a much better job. If you look at our fuel costs and the efficiency with which we have operated over the last 2 or 3 years, I think it is self evident that they are challenged by this and that they are doing a better job of managing. We talked at great length with some of the people in the FAA with respect to removing the control column, the throttles, and the rudders, and it opens up a lot of space we badly need for indicators and navigational equipment and that sort of thing. I think there's going to be some sort of resistance among the pilots to removing these traditional things, but it certainly won't take long to convince them if it is in fact a better way. I don't have any objection to it. I think it could be sold very easily once it has been shown that it's a better way.

Somebody made a remark a while ago about buying new equipment. Not too many airlines are beating a path to airplane manufacturers' doors these days looking for new equipment over and beyond what they're already committed for. There's some thought that some of us have too much, so we're not looking for any new problems at the moment. But the airlines will adopt, and not reluctantly, something that is more efficient, safer to operate, or has some other type of advantage. This is no different than in the past. I don't think the airlines are going to

get together and sell the manufacturers on active control technology or control-configured vehicle equipment. The manufacturers are going to have to grab this ball and convince the users that this is a better way to go.

Gorham: I had some comments a while ago, but in view of what's been said I've modified them a little. I was going to say that a new program is essential to establish the benefits of active control technology, and I think we've talked that to death, probably because it's pretty obvious that the tradeoffs have to be pretty well established to know what investigations you have to make. We're investigating active control technology. Fine, but is there anything in the structural area, the cockpit area or any other part of the airplane which the tradeoffs show might bring benefits if changed or modified? Let's not get to a point in 5 years' time where the technology of fly by wire has been thoroughly investigated and is a tool that could be used and when we do the tradeoffs we find some other technology gives a greater payoff. A broad cut of tradeoffs must be established to decide what other areas of technology might relate to the incorporation of active control technology.

Another point I'd like to make is that something happens because there's a need for it. This is getting back to Frank Kolk's point, which I don't take too much umbrage at, but which I will remember for a while, about all-weather automatic landing systems. I well remember the airlines' introducing a system called aircraft integrated data system (AIDS) 7 or 8 years ago. For those of you who don't know what AIDS is, it is a very complex recording system which a certain major airline hoped to install in an airplane. It involved more electronic boxes than were on the airplane at the time, and it was hoped that it would improve the reliability of the lesser avionics that were already being carried. It was kind of irrelevant. I remember standing up just like this in New York, and the speech I made was that I had sat there for 4 days and heard a detailed description of a solution, but that I didn't really know what the problem was. So there are systems that go into airplanes where everybody has been mistaken.

Multiplex entertainer, a complex and difficult system, was introduced, and it fell into a lot of problems on the Boeing B-747, the Douglas DC-10, and the Lockheed L-1011 airplanes. However, when the airlines asked if they could take it off, and we asked if they would accept a 1000-pound weight penalty for taking it off, which is the weight of the wiring, of course they said no. My point is that there was really a big advantage. Some way had to be found to make it work, and we did.

Finally, Mr. Seacord made a point about all-weather automatic landing systems and the need to look at the reliability of the sensors. I take exception to that, and I think the airlines and Mr. Skully should too, because we now have at least three airplanes certificated for all-weather automatic landing systems, and I'm sure the FAA and its British counterpart wouldn't have given that permission if they hadn't been satisfied with the sensors' reliability. His point on 10^{-7} and 10^{-9} is semantic, really. Without going into any details, one involves an individual risk and the other involves a collective risk. It's just a different way to do the bookkeeping.

Finally, I regard the aircraft industry as being all of us, not as separate from NASA, DOT, and the airlines. Even consultants, I think, should be included in the airline industry.

Skully: One of the comments I certainly supported was about establishing clear goals. I think that to attain these goals, and there's more than one, we'll have to make a well coordinated effort. It might be helpful to look at some other programs. One that two of my colleagues and I are very much involved in, or have been, is the two-segment approach program. I am happy to see Mr. Wells from the House staff here, because the FAA has been beaten on the head pretty severely. NASA was funded by Congress to develop the two-segment approach. Frankly, I don't know what went wrong. I don't know why we're in the state that we're in at the moment. American Airlines picked up the project and did a great deal of work on the B-707 airplane—Frank Kolk was the master mind, followed by United Airlines. Lloyd Treece and I have flown United's effort in the B-727 and DC-8 airplanes. We just finished the advance notice for rule making. It went over like a lead brick. The comments were due at the end of June, and I'm almost afraid to read them. The position of the Aerospace Industries Association (AIA) is that they are very much against it. The Airline Pilot's Association (ALPA), the AOPA, the National Business Aircraft Association (NBAA)—any organization you want to name thinks it's just terrible. The point I'm trying to get at here is that we've spent a lot of time, effort, and money, and I don't know if it's going to fly or not. Obviously, the objective is to reduce noise. I might add that I'm a little surprised that I haven't heard any comments during this symposium about what active control technology might do in terms of opening or keeping open some of the critically closed-in airports. If it does, it has a payoff.

Lamar: I believe ongoing programs in the Air Force address the major obstacles to utilizing this type of technology. Of course, Air Force cargo aircraft do have command augmentation systems in them. We are getting a lot of experience with them, and that experience is directly relatable to fly by wire. I think the next step would clearly be the fly-by-wire transport. Once we depend on fly by wire, we can without too much hesitation incorporate the control-configured vehicle concepts that have been shown to provide real payoffs in the design studies. What we're trying to do, of course, is to make options available to the designers. There are gaps in the program, and we're trying to fill them. For example, there is a lot of work under way right now and being planned to insure the satisfactory integration of digital avionics, so that the capabilities of digital processes are exploited in the military subsystems of the aircraft. We are also trying to exploit them for digital flight control. We are moving towards more digital flight control and the use of multimode capabilities. We are working on the displays, the controllers, and the other components that go with it. We are looking at what it takes to get the human operator integrated into it in the most economical fashion.

The Air Force is concerned about overhead and maintenance costs, operational costs. For that reason we are interested in pursuing any lessons learned by the airlines. If there is any way we can work together, I am sure that we will be willing to do so. I think we ought to develop joint programs between NASA, the Navy, and the Air Force to make our dollar go as far as possible to achieve this new technology. The basic program plans are under way, but they are underfunded.

Newberry: I would like to comment on what actions and coordination are needed. I think that this meeting itself is necessary and a first step in bringing industry, the airlines, and the aviation community together. I think that NASA and the Air Force should be complimented for putting together this symposium. I think I speak for many others in saying that it has been an enjoyable symposium, enjoyable in that it has provided an opportunity to meet old friends. All of us tend to become too busy working in our own areas to communicate with others involved in the technology. This symposium has provided an opportunity for the interested and affected parties to discuss this important technology.

Weil: Our time has run out for the panel discussion. I think it was quite productive. We at NASA appreciate the constructive comments on our programs from the airlines and industry, and I'm sure your comments will affect our thinking on future programs. I would like to thank the panel members and the audience for their participation.

SESSION X

SOME EXPERIENCES USING WIND-TUNNEL MODELS

IN ACTIVE CONTROL STUDIES

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SUMMARY

A status report and review of wind-tunnel model experimental techniques
that have been developed to study and validate the use of active control
technology for the minimization of aeroelastic response are presented. Model-
ing techniques, test procedures, and data analysis methods used in three model
studies are described. The studies include flutter mode suppression on a
delta-wing model, flutter mode suppression and ride quality control on a
1/30-size model of the B-52 CCV airplane, and an active lift distribution
control system on a 1/22-size C-5A model.

INTRODUCTION

Dynamic and aeroelastic wind-tunnel models have played an important role
in the development of aircraft and space technology. In many instances model
tests are the most economical means, both in terms of time and cost, of
determining needed data as compared to other methods such as analysis and
flight tests. Models can be used to obtain results at conditions where ana-
lytical results are known to be inaccurate, for instance, transonic speeds.
Ordinarily model results can be obtained in a more timely manner than flight
results, and model tests are more amenable to conducting extensive parametric
studies than are flight tests. Aviation applications of dynamic models have
included such diverse areas as flutter, gust response, and landing loads while
space application includes, among others, launch vehicle buffeting and ground
wind load studies. Some of the many uses of models in aerospace applications
are described in references 1 and 2, for example. Perhaps the extensive use
of models is best illustrated by the fact that a literature search under the
category of dynamic models gives a listing of over 2500 publications. This
continued use of models has resulted in modeling technology reaching a rather
advanced state of development. However, new technology and advanced concepts
are continually being developed which offer new challenges to modeling tech-
nology. Active control technology is one of the latest challenges.

The addition of active controls to models adds a new complexity to model-
ing technology. In particular, control surface and actuation systems must be
miniaturized, usually under severe weight restrictions, and new testing tech-
niques must be developed.

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The NASA Langley Research Center has embarked on a research program to develop experimental techniques so that wind-tunnel models can be employed to study and validate active control systems used to minimize aircraft aeroelastic response. Although the major thrust of this work is experimental, considerable emphasis is also being placed on the development of analytical techniques. This paper presents a status report and review of the experimental work that has been accomplished to date. An earlier status report is presented in reference 3. Some experiences in the testing of three different models in the Langley transonic dynamics tunnel are reported herein. In addition to presenting some basic experimental results from the three studies, such topics as model design and construction, active control system implementation, and wind-tunnel test techniques are discussed. Some comparisons between model experimental results and analysis are made, and in one instance some comparisons between model and flight test results are presented.

The first model program is a flutter suppression study using a delta-wing model. This research model, which is a simplified representation of a contemporary supersonic transport design, was used to develop basic flutter suppression modeling technology and to evaluate the aerodynamic energy flutter suppression concept developed by Nissim in reference 4. In addition to the results presented in reference 3, some later data obtained by using the delta-wing model are presented in reference 5. The second model study used a 1/30-size dynamically scaled aeroelastic model of the B-52 control configured vehicle (CCV). Both flutter mode control (FMC) and ride quality control (RQC) systems were implemented in this model study which was done in cooperation with the Air Force Flight Dynamics Laboratory. Some B-52 model flutter suppression results are given in reference 6. The Boeing Company, Wichita Division, has provided contractual assistance during both the delta-wing and B-52 model studies. The third model study was also done in cooperation with the Air Force and used a 1/22-size C-5A model. Under contract to the Air Force, the Lockheed-Georgia Company designed and built the model, and provided technical support for the wind-tunnel tests. The model was equipped with an active lift distribution control system (ALDCS) which used active controlled ailerons and horizontal tail to redistribute the dynamic wing loading in order to decrease the wing root bending moment. This C-5A model study was performed in conjunction with the development of a proposed lift distribution control system for the full-scale aircraft. The evolution of the proposed aircraft system is described in reference 7.

DELTA WING FLUTTER SUPPRESSION STUDY

General

The delta-wing model study was the first active control flutter suppression study undertaken at the Langley Research Center. In general, this program was initiated to develop the basic technology required for active control modeling studies and, in particular, to demonstrate experimentally that flutter can be suppressed by using active controlled aerodynamic surfaces. The flutter suppression concept chosen for implementation was the aerodynamic energy method

developed by Nissim (ref. 4). Simply stated, this aerodynamic energy concept says that flutter cannot occur if, for all allowable oscillatory motions, positive work is done by the wing on the surrounding airstream. That is, energy is transferred from the wing to the airstream.

A photograph of the delta-wing model mounted in the transonic dynamics tunnel is presented in figure 1, and model geometry is shown in figure 2. The 1.28 aspect ratio model planform was a cropped delta with a leading-edge sweepback angle of 50.5°, a taper ratio of 0.127, and a circular arc airfoil section with a thickness-to-chord ratio of 0.03. Two high-fineness ratio bodies were mounted on the wing lower surface to simulate engine nacelles. The model was cantilever mounted to a rigid mounting block that was bolted to the tunnel sidewall. The mounting block was enclosed in a simulated fuselage fairing which extended ahead of and behind the wing. This mounting arrangement brought the wing root outside of the tunnel wall boundary layer. The model was equipped with leading- and trailing-edge aerodynamic control surfaces. The controls were actuated by an electrohydraulic system which was controlled by a feedback system that was implemented on an analog computer located in the tunnel control room.

Design and Construction Considerations

Design.- Since the delta-wing study was of a research nature, it was not necessary that the model scale any particular full-scale airplane wing. However, for research studies to be as relevant as possible, it is desirable that the models used be representative of current or proposed configurations. Consequently, the delta-wing model design was based on a contemporary supersonic transport configuration. In particular, this model was a simplified 1/17-size version of the Boeing 2707-300 configuration. The design objective was to have a model that had similar flutter characteristics to those of the prototype and would flutter well within the operating boundary of the transonic dynamics tunnel. Other constraints to the model design were that the construction technique was to be as simple as practical and that construction cost was to be kept to a minimum. In developing the delta-wing final design, some preliminary wind-tunnel studies were made by using different size models that differed from one another in stiffness and mass properties. Some results of this study are reported in reference 8. From the results of this study a final design was selected. The flutter boundary of the final design (called configuration C in ref. 8) is very similar to the boundary (for one weight condition) of a rather expensive dynamically scaled replica-type model of the prototype configuration when both sets of data are scaled to airplane values.

Construction.- The construction of the delta-wing model was relatively simple. The basic structure was an aluminum alloy insert which tapered in thickness in the spanwise direction. Portions of the insert were chemically milled to simulate spars and ribs. The insert was covered with balsa wood that was contoured to give the desired airfoil section. The balsa wood was covered with one layer of fiber glass cloth which was doped to the wood. The two engine nacelles were made of steel tubing with balsa wood nose and tail streamlining fairings. The nacelles were ballasted with lead weights to give

the desired mass and inertia properties. Relative to the basic wing structure, the nacelles were rigid. The fuselage fairing was constructed of wood. The basic structure of the leading- and trailing-edge control surfaces was a metal tubing axle with balsa wood bonded to the axle. Two hardwood ribs were incorporated in each surface to provide additional chordwise stiffness. Each control surface was covered with a thin sheet of fiber glass cloth that was doped into place.

Flutter Suppression System

The flutter suppression system implemented on the delta-wing model was based on the aerodynamic energy concept described in reference 4. The implementation of this method used both leading-edge and trailing-edge control surfaces. The deflections of these control surfaces are related to the dynamic motions of the wing through a control law which relates control surfaces rotations to wing displacement and rotation. The resulting matrix equation is shown in figure 3. The elements of the C and G matrices are real numbers whose magnitudes are determined by aerodynamic energy considerations. In theory it is possible to determine values of the C and G matrix elements so that for all allowable wing motions energy is always transferred from the wing to the surrounding airstream and flutter cannot occur. However, in practice it is not necessary that flutter be precluded from occurring at all flight conditions, but only that the flutter speed be increased by some predetermined amount. That is, flutter cannot occur within some specified flight envelope.

Control laws.- Three different control laws were used for the delta-wing model. The three control laws are shown in figure 4. The first two (A and B) used both leading-edge and trailing-edge control surfaces. Control Law C used only a trailing-edge surface. Since the initial part of the study was aimed at demonstrating the basic aerodynamic energy concept, the first control law (Control Law A) used was that given in reference 4. The values used for the C and G matrix elements were the same as those Nissim developed by using two-dimensional unsteady aerodynamic theory. Control Law B was similar to the first except that three-dimensional unsteady aerodynamic theory (doublet lattice method) was used to determine the terms in the C and G matrices. Control Law C was also developed by using three-dimensional aerodynamics, but used only the trailing-edge control surface. In implementing both Control Laws B and C on the model, some difficulties were encountered. In effect, the system was so sensitive that if the model was disturbed in still air the control surfaces would begin to oscillate and drive the model. The exact reason for this problem has not been determined, but it is believed to be due to inertia coupling between the control surface and the wing portion of the model. This difficulty was cured by compromising the analytical values for the coefficients in the G matrix. The original G matrix values are shown in parentheses in figure 4. Analytical study results indicated that the required adjustment in G matrix values had little effect for Control Law C. However, for Control Law B there was a considerable degradation of the expected flutter dynamic pressure increase when the G matrix values were decreased. However, Control Law B still gave better performance in terms of increase in flutter dynamic pressure than that calculated for Control Law A.

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Implementation.- Some of the physical components of the flutter suppression system are shown in the photograph presented in figure 5. A simplified block diagram of the system is presented in figure 6. The wing motion was sensed by two accelerometers that were located in line with the inboard edges of the control surfaces. The accelerometers were located at 30 and 70 percent of the chord. The accelerometer output signals were fed through signal conditioning equipment to an analog computer which was located in the tunnel control room. The aerodynamic energy control law was programed on the analog computer. The integration and differencing operations required to process the acceleration signals were also programed on the computer. (A portion of the analog computer may be seen in the upper left of figure 5.) The analog computer processed the accelerometer signals to determine appropriate actuator command signals. Command signals were passed to hydraulic servovalves which were mounted in the fuselage fairing at the model root. The servovalves controlled the supply of hydraulic fluid to miniature actuators that were mounted in the model at the inboard edge of each control surface. Control surface angular position was determined by using miniature silicon solar cells attached to each actuator shaft. Hydraulic and electric lines were routed to the actuators and sensors in trenches cut into the balsa wood which covered the aluminum insert. The model was also equipped with several resistance wire strain-gage bridges which were used to monitor model response. Although it is not indicated in figure 6, provision was provided for introducing external command signals to the control surfaces. External command signals to the trailing-edge surface were used for performing frequency sweeps and could be introduced with the flutter suppression system either operating (closed-loop) or not operating (open-loop).

Control surface actuators.- Initially it was decided to mechanize the control actuation system with an electromechanical system. The original concept was to mount an electric torque motor external to the model (inboard of the model root) and transmit the torque to the control surface by mechanical shafting. Considerable design effort with accompanying laboratory experimentation was expended in trying to come up with a satisfactory electromechanical system with little success. The major difficulty was associated with the shafting which had to exhibit little wind-up yet be light weight and not contribute any appreciable increase in stiffness to the basic wing. Finally, it was decided to switch to a hydraulic actuation system with the actuators located in the model at the control surfaces. Since no miniature hydraulic actuators existed that were small enough to fit within the model aerodynamic contour and provide the required torque, not to mention the light weight requirement, it was necessary to design and fabricate special actuators. The actuator design and fabrication is described in reference 9. The actuator is essentially a closed compartment that is separated into two chambers by a vane which rotates on a shaft that attaches to the control surface axle. The amount of shaft rotation is determined by the difference in hydraulic pressure between the two chambers. The actuator weighs 56.7 grams (0.125 lb) and is capable of providing a 4.52 N-M (40 in-lb) torque output with a 6.9×10^3 kN/m^2 (1000 lb/in^2) supply pressure over the frequency range from 0 to 25 Hz. A photograph of an assembled actuator attached to the trailing-edge control surface is presented in figure 7. The development of these miniature actuators represents a significant contribution to active control modeling and is not

limited to the delta-wing model application. In fact, similar actuators were used in the C-5A study to be discussed later in this paper. It was also necessary to design and fabricate special control-surface position indicators. Here again the space available was one of the most significant design constraints. The new position sensor (ref. 9) is a rather simple device that uses two silicon solar cells that are mounted on a common base that is attached to the actuator shaft. The solar cells are illuminated by a stationary light source. The intensity of the illumination changes as the shaft rotates, and a voltage is produced which is linearly proportional to the tangent of the shaft rotational angle.

Control surface location.- Since the specific aerodynamic energy control law developed in reference 4 was based on two-dimensional unsteady aerodynamic theory, it was necessary to conduct an analytical study to determine an appropriate location for the control surfaces. This study is described in reference 10, and some of the results are shown here in figure 8. Three possible control surface locations were considered as well as different locations of the model motion accelerometer sensors. In all cases the accelerometers were located at 30 and 70 percent of the local chord. The combination of control surface and sensor locations used for the delta-wing model was the midspan surfaces with the accelerometers alined with the inboard edge of the surfaces. This combination gave the second best increase in flutter dynamic pressure. The most improvement was obtained for outboard control surfaces and outboard sensor locations, but the use of this combination would have been very difficult since the wing was very thin in this region. It should be pointed out that the mathematical model used to generate the data presented in figure 8 was slightly different from the final delta-wing model so the expected flutter dynamic increase for the model would not be expected to be exactly those shown in the figure.

Test Techniques

Wind tunnel.- As was the case for all of the model studies described in this paper, the delta-wing model was tested in the Langley Research Center transonic dynamics tunnel. This facility is specially designed for and almost totally dedicated to the testing of dynamic aeroelastic models. The closed-circuit, single-return tunnel has a 4.88-m (16-foot) rectangular test section with flow expansion slots in all four walls. The tunnel flow conditions are continuously controllable over the Mach number range from about 0.07 to 1.2 at total pressures from near vacuum to slightly above one atmosphere. Either air or freon may be used as the test medium. All results reported herein were obtained by using freon.

Subcritical response.- In active flutter suppression studies it is not only desirable to determine actual flutter data points, but it is also necessary to determine stability information at conditions below the flutter boundary. The desired information is the damping of the critical flutter mode. Two techniques have been used with considerable success for determining subcritical damping levels. Both methods are based on the assumption that the response is that of a single-degree-of-freedom system. The first technique is

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based on the procedure described in reference 11 and is referred to as "randomdec." Unlike most subcritical response procedures, randomdec does not require that the system be excited by special shakers, but depends on flow turbulence to supply the necessary input. The randomdec method is illustrated schematically in figure 9. The system response is assumed to be composed of three components – the responses to a step, to an impulse, and to a stationary random force. The system response to a step force is obtained by an ensemble average of a number of time sweeps, since the response to an impulse and to a random force average to zero. The time averaging was accomplished by using a small special-purpose computer. In the implementation here the different time segments were averaged sequentially. That is, the computer processed all the results for one time sample before beginning to collect the average data for the next sample. The averaging process for each time sample was started when the output signal reached a predetermined level. The model sensor output was passed to a gating circuit. When the preset signal level was reached, the gate was opened and the signal passed to the computer and averaged with values from previous samples. Electronic filters were used to isolate the frequencies of modes of interest. The averaged signal has the appearance of the damped oscillation of a single-degree-of-freedom system. The system damping is obtained from this decaying oscillation. Although the randomdec method has been used quite successfully in many cases to determine subcritical damping level, the method is not free from problems. Two difficulties are worthy of mention here. The first is noise contamination of the signal. At low levels of flow turbulence, the output of the model response sensor is relatively low. However, since the electronic noise level is independent of sensor output, the signal-to-noise level is relatively low and the result is low-quality decay signatures. Fortunately, this difficulty is most severe at conditions removed from the flutter condition. As the flutter condition is approached the system response naturally increases and the signal-to-noise level increases. The second problem is when there are two or more structural frequencies in close proximity to one another. Although signal filtering is useful, it is very difficult to completely filter out the unwanted mode. Although the band-pass filter is set for a very narrow range of frequency, the signal level outside the band is not completely attenuated because of filter roll-off. This results in a beat occurring in the randomdec decay signature and makes determining quantative values of the damping difficult. The decay looks like that of a coupled two-degree-of-freedom system.

The second technique, described in more detail in reference 12, requires the measuring of the forced response of the model. This method is illustrated schematically in figure 10 and will be referred to as the Co-Quad method. The model is excited by a sinusoidal force of varying frequency and the corresponding dynamic response is measured. Special electronic equipment is used to resolve the response into in-phase (called Co for coincident) and out-of-phase (called Quad for quadrature) components relative to the sinusoidal command signal. The damping of the system is obtained for each structural mode from the variation of the coincident component transfer function with frequency. Each resonant condition is treated as if it were that of a single-degree-of-freedom response, and the damping is obtained by using the formula shown in the figure. For active control models the Co-Quad method is easily implemented since an active controlled aerodynamic surface can be used to provide the

sinusoidal force input. For the delta-wing model frequency response data were obtained by oscillating the trailing-edge control surface. Co-Quad response data were obtained in terms of the ratio of accelerometer output $\ddot{h}_1$ to command signal $\delta_{t,c}$ to the trailing-edge control. The difficulties encountered with this method were similar to those described for the randomdec method, namely, noise and closely spaced resonant frequencies. However, in contrast to randomdec, the noise in this case is not primarily instrumentation noise but is the random response of the model which is superimposed on the sinusoidal response. The Co-Quad method requires a longer data gathering period than the randomdec technique. Typically about 30 seconds were required for randomdec while the Co-Quad frequency sweeps of about 4 minutes were used. The Co-Quad method is somewhat dangerous to use at conditions very near the flutter condition since the addition of the sinusoidal force to a model that already has significant response results in extremely large amplitudes as the forcing frequency sweeps through the critical flutter mode.

For the delta-wing model both the randomdec and Co-Quad methods were successfully used. In general, the randomdec method appeared to be the better of the two methods. The randomdec results, as judged by the qualitative appearance of the randomdec decay signature, appear to get better as flutter condition is approached. In contrast the Co-Quad method appeared to give the best results the farther you were away from the flutter condition. Where damping data were obtained by using both methods, the results were within what would be expected to be the experimental scatter band.

Results

Flutter.- Flutter studies of the delta-wing model were conducted at Mach numbers M of 0.6, 0.7, 0.8, and 0.9. Tests were performed both with (closed-loop) and without (open-loop) active controls. For the open-loop studies the control surfaces were kept at 0° deflection by applying hydraulic pressure to the actuators. The pressurized system acted as a stiff spring to keep the rotational frequency of each control surface many times higher than the wing flutter frequency. Once the open-loop flutter boundary of the wing was established, an evaluation of the effects of each of the three control laws on raising the boundary was made. However, studies for Control Laws A and B were restricted to M = 0.9 because of a high-frequency, large-amplitude oscillation of the leading-edge control. This phenomenon occurred around 65 Hz, as compared to the flutter frequency of from 11 to 12.5 Hz. It is believed that oscillatory motion was introduced in some manner by the mechanization of the leading-edge control, and was not a consequence of the control law, since the motion was also observed to a lesser degree with the control loop open.

A comparison of calculated and experimental results showing the effect of each control law on raising the open-loop flutter boundary is presented in figure 11. The results are presented in terms of percent increase in dynamic pressure at M = 0.9. By using Control Law A a 12-percent increase in dynamic pressure was obtained. The observed flutter motions for both open- and closed-loop operations were similar. The calculated increase for Control Law A is in excellent agreement with the experimental values. An earlier analytical

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treatment for this control law was reported in reference 3 and showed a
21-percent increase in the flutter dynamic pressure. The differences between
theory and experiment in reference 3 were attributed in part to the inability
of the aerodynamic theory to adequately predict control surface pressure dis-
tributions. Early in the design of the delta-wing model static hinge-moment
measurements were made to aid in the design of the control actuators. It is
shown in reference 13 that the calculated values of hinge moment are somewhat
higher than those that were measured. For the present analytical investigation
the theoretical unsteady aerodynamic forces for the leading- and trailing-edge
control surfaces were adjusted to take into account the differences between
measured and calculated static hinge moments. The analytical results for
Control Law B indicate a predicted increase of 24 percent. The experimental
results demonstrate a minimum increase of 22 percent. Experimental results for
Control Law B do not represent a closed-loop flutter point since further
increases in dynamic pressure were restricted by the high-frequency oscillation
of the leading-edge control surface mentioned earlier. Of the three control
laws investigated, the largest increase in flutter dynamic pressure was
obtained with Control Law C. A minimum increase in dynamic pressure of 30 per-
cent was obtained with this control law. The model was not tested to the
closed-loop flutter condition since the goal for these tests was set at a
30-percent increase in dynamic pressure assuming that closed-loop flutter was
not encountered. The analytical results indicate a 34-percent increase.

The effectiveness of Control Law C in suppressing the flutter motion is
vividly demonstrated by the time history of the wing bending strain-gage out-
put shown in figure 12. Time is increasing from left to right. The tunnel
dynamic pressure was slowly increased until open-loop flutter occurred (see
left of figure). At this point the flutter suppression system was turned on
as is indicated by the vertical dashed line on the right side of the figure.
Note that when the system is turned on, oscillatory flutter motion is rapidly
damped to a closed-loop no-flutter condition. The degree of confidence in the
control system was such that when open-loop flutter was encountered, the active
control loop was closed to suppress the motion.

In order to evaluate the active control system at other Mach numbers,
Control Law C was both analytically and experimentally studied from M = 0.6
to M = 0.9. The results obtained are presented in figure 13 in terms of the
variation of flutter-speed-index parameter with Mach number. The experimentally
measured open-loop flutter boundary and the closed-loop no-flutter points for
each Mach number are presented. At M = 0.8 a 9.4-percent increase in flutter-
speed-index (20 percent in dynamic pressure) is shown. Unfortunately, at this
point the model was damaged due to saturation of the closed-loop system because
of limited available actuator angles (±9°). Saturation caused the analog com-
puter amplifiers to overload and forced the control surface to go hard against
its stop resulting in open-loop flutter. The model was repaired and tested at
Mach numbers of 0.7 and 0.6. A modest increase in flutter-speed-index of
5.7 percent (12 percent in dynamic pressure) was demonstrated at these two
Mach numbers.

A comparison of calculated and experimental results (Control Law C) is
also presented in figure 13. The calculations for the open-loop system show
reasonable agreement at all Mach numbers.

Subcritical response.- Some subcritical response data obtained by using the Co-Quad technique are presented in figure 14 for a Mach number of 0.90. Both the in-phase and out-of-phase response in terms of the ratio of accelerometer output $\ddot{h}_1$ to trailing-edge command signal $\delta_{t,c}$ are presented. The curves on the left of figure 14 represent the model for open-loop operation at a dynamic pressure approximately 5 percent below the open-loop flutter boundary. The curves to the right in this figure show the model closed-loop response (Control Law C) at the open-loop flutter dynamic pressure. A qualitative measure of the active controls in reducing the forced response of the system is evident by the reductions in peak amplitudes around the flutter frequency of 11 Hz. Also shown in figure 14 are calculated response data. Note that the analysis does predict well the general behavior of the response. For these calculations, the effectiveness of the trailing-edge control was reduced by the ratio of measured-to-calculated static hinge moments.

B-52 MODEL STUDY

General

The planned B-52 model program includes studies in four active control areas. These areas are flutter mode control, ride quality control, maneuver load control, and relaxed static stability. To date a portion of the planned flutter mode control (FMC) and ride quality control (RQC) tests have been completed. The completed flutter suppression and ride quality control wind-tunnel tests are described herein. The flutter mode control portion of the model program is being conducted in cooperation with the Air Force Flight Dynamics Laboratory with contractual support being supplied in all four areas by The Boeing Company, Wichita Division.

The B-52 model program actually began in the late 1960's when a 1/30-size dynamically scaled aeroelastic model of the B-52E aircraft was constructed for use in symmetric gust studies in the transonic dynamics tunnel. Although provision was provided for incorporation of active controlled midspan ailerons and elevator in this original model, only gust response studies without active control were conducted. The results from these tests are not published. With the initiation of the B-52 CCV airplane program, it was decided to convert the B-52E model to a model of the CCV aircraft and expand the model program to include the four active control areas mentioned above. The B-52 CCV airplane program is described in reference 14, and some flight results are presented in reference 15. These CCV model tests offer the unique opportunity of validating wind-tunnel model techniques since flight data would be available for comparison with the model results. Although some modifications to the structural stiffness and mass were required in converting the original model to a CCV model, most of the modifications were associated with the installation of new aerodynamic control surfaces which included outboard ailerons, flaperons, and a pair of fuselage-mounted horizontal canards. A photograph of the complete free-flying model mounted on the two-cable suspension system in the transonic dynamics tunnel is presented in figure 15.

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The objectives of the B-52 model wind-tunnel tests were to demonstrate the effectiveness of the FMC system and the RQC system, and to obtain data for correlation with analysis and airplane flight results. The design of the model systems was based on the corresponding CCV aircraft systems. The FMC system used active controlled flaperons and outboard ailerons. A pair of fuselage-mounted horizontal canard surfaces was used for the RQC system. The locations of the control surfaces are shown in figure 16. The feedback loops for both systems were implemented on an analog computer. The model control surfaces were actuated by using an electromechanical system.

Design and Construction

Scaling.- The B-52 model was a 1/30-size dynamically scaled aeroelastic model of the B-52 CCV airplane. The model weighed about 26 kg (57.75 lb) and had a wing span of 188 cm (6.16 ft). The model was designed to match the dynamic similitude parameters of reduced wavelength, mass ratio, and Froude number. Some of the scaling relationships and corresponding model/airplane flight conditions are presented in figure 17. Since the airplane flight conditions are at relatively low Mach numbers where compressibility effects are small, it was not considered necessary to match the Mach number between the airplane and the model. It is fortunate for the B-52 model study that flight conditions were at relatively low speeds since Mach number and Froude number scaling are difficult to satisfy simultaneously while still matching both reduced wavelength and mass ratio. A discussion of the conflicting requirements of Mach number and Froude number scaling is presented in reference 2.

Construction.- The construction technique used for the B-52 model was one that has been successfully used for a number of years in building aeroelastic models. Some details of the model construction are shown in figure 18. Aluminum alloy spars and beams were used to provide the basic stiffness of the wings and fuselage, respectively. Segmented pods constructed of wood frames covered with thin plastic sheets were attached to the spars and beams to provide the proper aerodynamic contour. The empennage was not elastically scaled. Both the horizontal and vertical tail were relatively stiff, but did have the proper total weight and center-of-gravity location. The engine nacelles were rigid streamlined bodies that had the proper inertia properties. The nacelles were attached to the wing spars by flexible beams which simulated the pylon stiffness. External fuel tanks were attached near the wings tips. The tanks were ballasted to simulate the mass that had to be added to the airplane to produce a flutter condition within the airplane operating boundary.

Test Techniques

Mounting system.- The B-52 model was mounted in the transonic dynamics tunnel by using a modified version of the two-cable suspension system described in reference 16. A portion of the cable support system is shown in figure 15. The model was supported by two cable loops, called flying cables, which were attached to the model at a common point. The cables were routed through low friction pulleys located on the tunnel walls. The forward cable loop was in the vertical plane, and the aft cable loop was in the horizontal plane. The

cables were kept under tension by stretching a soft spring in the rear loop. This mount system provided freedom for the model to translate laterally and vertically and to rotate about the pitch, roll, and yaw axes. In additon to the flying cables, four additional cables were attached to the model to provide emergency restraint (see fig. 15). These snubber cables extended out through the tunnel walls to a shock absorber system and a remotely controlled actuator. These cables were slack during normal test operations. The model was essentially flown in the tunnel test section on the mount system by a pilot located in the tunnel control room. For this model the pilot remotely operated the horizontal stabilizer to provide pitch control. For many models, external roll control is also provided, but this was not done for the B-52 model. Proper roll attitude (wings level) was obtained by manually setting small trim tabs located on each wing by a trial-and-error process. Once a satisfactory tab setting was obtained during the first test run, it was not necessary to change the setting for later runs.

For flutter model testing the primary mount system design requirements are that the model must be stable on the mount system, and that the frequencies of all rigid body modes must be well separated from the frequencies of the structural modes. For longitudinal ride quality control studies, there is the additional requirement that the short-period mode must be simulated as accurately as possible. Since the two-cable system introduces some spring restraints to the model that do not exist in free flight, the short-period mode is affected, and an additional rigid body mode (primarily a vertical translation mode) is added. In designing the B-52 model mount system, particular attention was given to properly simulating the airplane short-period mode and to keeping of the rigid body translational mode frequency as low as possible.

Flutter mode control (FMC).- For the most part the wind-tunnel test techniques used for the B-52 model FMC studies were the same as those used for the delta wing. Both the randomdec and Co-Quad subcritical response techniques were used. Some additional techniques were also used in an effort to determine subcritical damping from transient response data. In one technique the model was disturbed by sinusoidally driving the aileron and then abruptly removing the driving force to give a transient response. In another method a transient response was generated by driving the elevator with a one-cycle sine wave pulse. Neither of these two methods was very satisfactory for determining damping data since the transient response was almost, sometimes totally, obscured by the response of the model to tunnel turbulence. It is interesting to mention that transient response methods were satisfactorily used to determine damping during the B-52 CCV flight tests. Apparently the ratio of turbulence response to control surface input response was higher for the model than for the airplane. The lack of success with transient methods for the B-52 model does not mean that transient response damping determination techniques cannot be developed for model use, but rather means that more development work needs to be done.

Ride quality control (RQC).- Part of the RQC tests were accomplished by using an airstream oscillator system to provide a symmetric sinusoidal gust input to the model. The oscillating vane system consists of a set of biplane vanes installed on each side wall in the entrance cone to the tunnel test section. The vane system is shown in figure 19. The vanes are sinusoidally

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oscillated (either symmetrically or antisymmetrically) through mechanical linkages by a hydraulic motor and flywheel arrangement. A vertical velocity component is induced in the flow in the center portion of the test section by the trailing vortices from the vane tips. The installation and early use of the vane system in the transonic dynamics tunnel is described in reference 17. The gust vane system has been calibrated and some typical results are presented in figure 20 in the form of a contour plot. The data shown are the variation of flow angle of attack with frequency and lateral position across the tunnel. Note that the gust angle decreases rapidly with increasing frequency and that there is some variation in flow angle with lateral position. Model response measurements were made with the RQS system on and off while the airstream oscillator system frequency was varied from 1 to 16 Hz. Also, frequency sweeps were made using external sinusoidal command signals to the model canards. The canard frequency was continuously varied over the frequency range from 4 to 24 Hz. Transfer functions were determined using the Co-Quad technique.

Active Control Systems

The B-52 model used active controlled outboard ailerons and flaperons for the FMC system. A pair of horizontal canards were used for the RQC system. The actuation systems for all of these surfaces were of the electromechanical type as opposed to the electrohydraulic system used on the airplane. The control surfaces were actuated by electric torque motors mounted in the model fuselage. The motors mechanically connected to the control surfaces through a rather complex mechanism of linkages. The complexity of the system can be seen by examining the photograph shown in figure 18. A more detailed description of the actuation system is presented in reference 10. The control laws were implemented and an analog computer located in the tunnel control room. Each control law was wired to a separate removable patch panel.

FMC system.- The design of the FMC system was based on the results of previous experience and analyses of the B-52 airplane. These results indicated that stabilizing aerodynamic forces are produced over the entire flutter oscillation cycle when the incremental lift generated by the control surfaces lags the wing displacement by 90°. Thus, the FMC feedback system was designed to produce the required phase lag between lift and displacement at the flutter frequency. The airplane FMC system is described in reference 18. A simplified block diagram of the model FMC system is presented in figure 21. The FMC system was redundant since there were two independent feedback loops. The first loop used the outboard ailerons as the active aerodynamic surfaces. Accelerometer signals from both the left and right wings were averaged and passed through a shaping filter to generate the aileron feedback command signal which was routed to the single aileron actuator. The flaperon loop was similar to the aileron loop except that each flaperon had its own actuator. In concept the model and B-52 CCV airplane systems are the same, the only difference is in the actuator dynamic characteristics. That is, a comparison of the two transfer functions would show a difference. However, over the frequency range of interest, the two actuators do have similar dynamic characteristics. Note that provision was provided at summing junctions (see upper left of fig. 21) for introducing external command signals to the actuators. The external command signals were used to

drive the control surfaces for model excitation. The command signals can be used when the FMC system is either operating (closed-loop) or not operating (open-loop).

RQC system.- The RQC system was designed to provide about a 30-percent reduction in the RMS vertical acceleration level at the pilot's station. A simplified block diagram of the RQS system is presented in figure 22. Pilot station acceleration signals are fed back through a shaping filter to produce the required canard command signals. In the RQC system it was necessary to add compensation to account for the differences in dynamic characteristics between the model and airplane actuators. The design of the model RQC system is described in reference 19.

B-52 Results

FMC system.- The primary objectives of the model FMC system studies were to establish the open-loop (FMC off) flutter velocity, to demonstrate the effectiveness of the closed-loop system (FMC on), and to obtain data for cor-relation with model analysis and full-scale flight tests.

During the FMC studies the open-loop flutter velocity was determined, and both open- and closed-loop subcritical response measurements were made above and below the open-loop flutter velocity. As described previously, several experimental techniques were used for determining the subcritical response characteristics. In general, the most useful results were from the forced response Co-Quad technique. Representative measurements of the in-phase and out-of-phase components of the wing acceleration $\ddot{z}_{WBL\ 78.3}$ to aileron command displacement $\delta_{a,c}$ as a function of frequency are shown in figure 23. These results are approximately 6 percent in velocity below the measured open-loop flutter point. The curves to the left on this figure are the frequency response of the open-loop system; the curves to the right represent the closed-loop response. The effectiveness of the FMC in reducing the forced response of the system is readily apparent by comparing the resonant response peaks of the open- and closed-loop systems.

The randomdec technique worked best as the flutter speed was closely approached and the damping in the flutter mode became very small. It was especially useful here since it was considered hazardous to apply external excitation. A typical response time history trace for the right wing accel-erometer $\ddot{z}_{WBL\ 47.8}$, and the associated randomdec signature, taken approxi-mately 3 percent in velocity below the flutter point, is shown in figure 24.

A comparison of calculated and measured flutter mode damping versus air-speed for the model is presented in figure 25. The measured values were obtained from the forced response technique while the calculated values were obtained from the characteristic roots of the equations of motion. The compar-ison shows the analysis to be conservative by about 10 percent in predicting the open-loop flutter velocity. This difference may be attributed in part to the fact that the measured structural damping of the model was somewhat higher than the damping used in the flutter analysis. Both experimental and analytical

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results show that the FMC system provides a substantial increase in damping near the open-loop flutter velocity. The measured closed-loop data show the system to be less effective than analytically predicted. This difference is believed to be due to hysterisis in the outboard aileron actuator system combined with a reduced effectiveness of the control surfaces that were not accounted for in the analysis. The maximum velocity tested with the closed-loop system was 48.3 m/sec (158 ft/sec); however, no damping values were measured above 47.2 m/sec (155 ft/sec) (indicated by a dashed line in fig. 25).

A comparison of measured flutter mode damping versus airspeed for the model and full-scale airplane is shown in figure 26 in terms of airplane velocity. The airplane damping values were obtained from transient response records. As indicated in this figure the model open-loop flutter speed is about 7.9 percent higher than the airplane flutter speed. This difference is attributed to minor variations in model mass and stiffness from the required values combined with some cable-mount effects on the rigid body dynamics of the model. The calculated airplane flutter speed was about 8.3 percent below the measured point. Thus a consistency does exist between measured and calculated flutter velocities for both the model and airplane in that the analysis was conservative in both cases by about the same amount. The data in figure 26 show that the model and airplane have the same closed-loop damping trends. In both cases the closed-loop system significantly increases the damping near the open-loop flutter velocity. Although some differences in damping level do exist, it is felt that the correlation between model and airplane is quite reasonable. As indicated in the figure, both the model and airplane were tested above the open-loop flutter velocity.

RQC system.- The objectives of the RQC studies were to demonstrate the effectiveness of a ride control system in reducing the acceleration at the pilot's station and to obtain data for correlation with analysis and full-scale flight. During the RQC studies the open-loop (RQC off) and closed-loop (RQC on) response of the model to external excitation was measured. The first series of tests that were performed involved measuring the response of the model to a sinusoidal gust field generated by the oscillating vanes. The frequencies of the primary modes of interest were at 2, 13, and 17.5 Hz. Sample results obtained from the in-phase and out-of-phase components of the pilot station acceleration $\ddot{Z}_{nose}$ as a function of vane frequency are presented in figure 27. The curves to the left are the open-loop response; the curves to the right, the closed-loop response. Attenuation of the closed-loop response around 2 Hz is apparent. However, the response in the 13-Hz mode is so low that the effect of the ride control system is not obvious. These results for the higher modes are due to the fact that the effectiveness of the oscillating vanes in generating the gust field falls off rapidly at the higher values of frequency (see fig. 20). The canard surfaces were used to generate the excitation for the higher modes. Results at the same test condition obtained from a canard frequency sweep are shown in figure 28. The canard amplitude was 2°. The data are presented for both the open- and closed-loop system in terms of the ratio of pilot station acceleration $\ddot{Z}_{nose}$ to canard command signal $\delta_{c,c}$ as a function of canard frequency. The effect of the RQC system on the higher modes is now evident. It appears that some combination of testing techniques is required to accurately define the system response curves.

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C-5A ALDCS MODEL STUDY

General

In an effort to reduce wing fatigue damage and thereby prolong the service life of the C-5A fleet, the Air Force has contracted with the Lockheed-Georgia Company to develop and flight test a C-5A airplane with an active lift distribution control system (ALDCS). This system is designed to reduce the incremental inboard-wing stresses experienced during gusts and flight maneuvers. The ALDCS uses existing controls on the airplane - ailerons to unload the wing tips and elevators to keep the airplane in trim. Specific design goals for the ALDCS are to reduce the symmetric flight incremental wing root bending moment by at least 30 percent while limiting any increase in torsional moment to less than 5 percent. A detailed description of the airplane ALDCS is presented in reference 20.

A wind-tunnel study of a dynamically scaled aeroelastic model equipped with the proposed ALDCS was undertaken for the following objectives: (1) to determine the ALDCS effectiveness; (2) to investigate the adverse coupling of structural modes due to the ALDCS, particularly wing flutter; and (3) to obtain experimental data for correlation with analysis and to guide flight tests. A photograph of the 1/22-size model used in the study is presented in figure 29. The model program was a joint effort of the Air Force, the Lockheed-Georgia Company, and the Langley Research Center. The model study was performed concurrently with the development of the airplane ALDCS and was completed within a 9-month period prior to the beginning of airplane flight tests. Basically, the model program involved the modification of an existing 1/22-size flutter model to match Froude number scaling, the incorporation of the ALDCS in the model, and wind-tunnel tests in the transonic dynamics tunnel.

Some unique features of the C-5A model study were that it was the first scaled model study of a lift distribution control system, and the model had an onboard hydraulic system. This hydraulic system included pump, fluid cooling system, and servovalves that powered the control surface actuators. The weight of this system (about 16.3 kg or 36 lb) was absorbed as onboard cargo in the fuselage. The C-5A model study required the application of many specially developed wind-tunnel systems and model technologies such as (1) the two-cable suspension system for minimum restraint to permit the model to be essentially free-flying, reference 16; (2) although not used, a special roll control system was available which altered the mount system cable angles in case the aileron deflections needed to keep the model in roll trim because excessive, reference 21; (3) a lift-simulation (cable-pneumatic spring) device that provided the capability of varying the model lift coefficient for given test conditions, reference 22; (4) an oscillating vane system which generated sinusoidal gusts, reference 17; and (5) aileron actuators on the model wing which were basically duplicates of those developed for the delta-wing flutter suppression model previously described.

Model

Scaling.- Although a 1/22-size flutter model of the complete C-5A airplane was available from earlier flutter clearance studies in the transonic dynamics tunnel, the decision was made to rescale and modify the model to match the airplane Froude number. This allowed a closer simulation of the aerodynamic loading and dynamic characteristics of the airplane, thus, a better evaluation of the ALDCS could be made. For a Froude number scaled model, the mass ratio and reduced wavelength are also matched at the selected design test conditions. The model scaling factors were derived so that Froude number was matched for the model at a Mach number of 0.263 and a dynamic pressure 2.394 kN/m^2 (50 lb/ft^2) in the wind tunnel with freon as the test medium. The corresponding airplane flight conditions were a Mach number of 0.58 and a dynamic pressure of 19.87 kN/m^2 (415 lb/ft^2) which corresponded to an altitude of 1524 m (5000 ft). It was assumed that at these relatively low subsonic Mach numbers the compressibility effects were not important and that there would not be any significant differences between the aerodynamic characteristics of the model and the airplane.

Design and construction.- The C-5A ALDCS model was designed to scale two airplane configurations, wing fuel loadings of 0 ·and 33 percent with approximately 113 400 kg (250 000 lb) of cargo for both cases. The 1/22-size model had a wing span of 3.037 m (9.96 ft). The 0-percent fuel configuration weighed 65.1 kg (143.5 lb), and the 33-percent fuel configuration weighed 77.2 kg (170.2 lb). To minimize costs, components of an existing flutter model were used as much as possible in the ALDCS model. A spar and pod type of construction was used. Some of the construction details are shown in figures 30 and 31. The metal spars carried the structural loads and simulated the stiffness characteristics. The balsa wood pods duplicated the aerodynamic shape and were ballasted (with the spars) to simulate the mass characteristics. The ALDCS model required close simulation of the wing properties. Therefore, new wing spars, engine-pylon spars, and fuselage spars were constructed. (The fuselage signficantly affected the wing dynamics.) Some new wing and fuselage pods were also constructed. The existing flutter model empennage was used in the ALDCS model; consequently, the empennage stiffness was not properly scaled. In reworking the empennage to incorporate the horizontal tail active control mechanism, an attempt was made to simulate the required scaled mass properties of the overall empennage; however, the final empennage was considerably under weight. The ailerons were scaled and consisted of a metal spar covered with balsa wood which was faired to give the proper scaled aerodynamic shape. The aileron-wing gap was not sealed on the model, although the gap was kept as small as practical.

With the exception of the empennage, the model simulated the mass and stiffness of the airplane quite well, including the important wing structural mode frequencies and mode shapes. It was concluded that the model adequately represented the airplane for the purposes of the ALDCS study.

Control systems.- The aerodynamic surfaces used for active controls on the C-5A airplane consisted of the ailerons and the elevators. However, for practical model design considerations, the all-movable horizontal tail was used

to provide active pitch control instead of the elevators. An appropriate compensation was made in the control law to account for this difference.

All active control surfaces on the model were actuated by an onboard hydraulic system. The aileron actuators were of the same design as those used for the delta-wing research model described earlier. The aileron also could be remotely controlled (by the model pilot) to permit static roll trim control. The tail actuator controlled the dynamic pitch angle of the complete horizontal stabilizer and was simply a hydraulically actuated piston (see fig. 30). The piston housing was mechanically coupled to an electric motor drive system (also controlled by the pilot) which could move the complete piston unit and thus vary the tail static pitch angle for model pitch trim. The piston drove the tail dynamically and worked against two coil springs which attempted to keep the piston in a centered position. In the event of a hydraulic failure, the centering springs would keep the horizontal stabilizer at its static trim setting, whereas the ailerons would tend to become free-floating and self-alining with the wing countour.

The power for the hydraulic system was provided by an onboard hydraulic pump (see fig. 31). This pump was an aircraft system that was adapted to the model and had an output pressure of 11.3×10^3 kN/m^2 (1600 lb/in^2). For lengthy operation of the hydraulic system, it was necessary to cool the hydraulic fluid, and a water cooling jacket was provided onboard the model. The water was externally pumped to the jacket through flexible plastic hoses which were secured to the instrumentation umbilical cord.

Servovalves were used to control the hydraulic pressure supplied to the control surface actuators. These valves were of the same type as those used for the delta-wing model. The model actuation system was compensated by electronic circuitry to give frequency response characteristics that closely matched the transfer functions of the airplane actuators.

Instrumentation.- Bending and torsional moments on the model were measured by using resistance-wire strain gages mounted at several spanwise stations on the wings and at the roots of the vertical and horizontal tail surfaces. Aileron hinge moments were measured by using strain gages mounted on each aileron pivot arm. Vertical acceleration on each wing near the aileron and at the fuselage center of gravity was measured by using accelerometers. Fuselage angle of attack was measured by using a servoaccelerometer. The angular position of each control surface was measured by using potentiometers. Fuselage center-of-gravity pitch rate was measured by using a pitch rate gyro. Tension in the model support cables was measured by using load cells.

Control Law

A simplified diagram of the active control systems used in the C-5A ALDCS model is presented in figure 32. There were two active control systems operating on the model, the basic aircraft pitch stability autmentation system (pitch SAS) and the ALDCS. Both commanded symmetric actuation of the control surfaces. The pitch SAS employed a feedback from the pitch rate gyro at the

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fuselage center of gravity to actuate the horizontal tail. The ALDCS employed feedbacks from both the pitch rate gyro and the fuselage center-of-gravity accelerometer to actuate the horizontal tail and feedbacks from the wing tip accelerometers to actuate the ailerons. Note that the acceleration signals from the two wings were summed in order to filter out unsymmetrical motions. The capability of supplying external command signals to the control surfaces was included. The gains K_{STAB}, K_{AIL}, and K_{SAS} were scheduled signal gains manually set according to a predetermined Mach number dynamic pressure schedule.

Tests and Procedure

A summary of the C-5A ALDCS model test configuration and test parameters is presented in figure 33. The 33-percent wing fuel configuration was tested first because this was a more realistic flight condition, and, hence, considerably more data were obtained with this configuration. The 0-percent fuel configuration, after a brief ALDCS effectiveness check, was extensively investigated to determine the ALDCS effect on flutter because this wing configuration had the lowest flutter speed (ALDCS off).

Model support and test techniques.- The free-flying mount system used for the C-5A model was essentially the same as that previously described for the B-52 model with the exception that the pulleys were mounted in the model fuselage rather than on the tunnel wall. The model was restrained by using the snubber cables during the flutter tests at Mach numbers above about 0.7 because the model was unstable in a Dutch roll type mode. During the dynamic load tests, the model was not only snubbed but was also tied down by cables attached to the nose and rear of the fuselage.

In the static aerodynamic studies, the simulated lift device shown in figure 34 was employed. Briefly, this device consisted of a single cable attached to the fuselage at the model center of gravity which would exert a down force to the model as needed. This was accomplished by attaching the cable to a piston essentially floating in an air cylinder which was located in the plenum outside of the test section. By varying the air pressure on the top side of the piston, a down load could be transmitted to the model. In operation, as the model angle of attack was varied, the air pressure to the cylinder was adjusted to compensate for the additional model lift and to maintain the model at its normal flying position near the center of the tunnel. By measuring the tension in the cable, the additional lift on the model could be measured.

Aileron and stabilizer frequency sweeps were extensively employed in the dynamics tests. In these sweeps, an external sinusoidal electrical signal was supplied to the control system to actuate symmetrically the control surfaces and generate aerodynamic forces. The model response to these aerodynamic excitation forces was measured. The frequency sweeps ranged from about 0.5 to 20 Hz. Several control surface amplitudes were used. Symmetric aileron and stabilizer step/ramp functions were also employed to excite the model. Part of the ALDCS effectiveness studies involved use of sinusoidal gusts generated by the tunnel oscillating vane system. Both symmetric and antisymmetric gusts were

used in the tests. The gust vane frequency was varied from 0.7 to 16 Hz. In order to reduce the static bending moments, a large portion of the tests were performed with the ailerons set at a nominal angle of +5° trailing-edge up (called uprig angle).

Static aerodynamic measurements.- With the model free-flying and the lift-simulation-device cable attached, static aerodynamic data were measured at dynamic pressures of 1.92 and 2.39 kN/m^2 (40 and 50 lb/ft^2). For each of the three aileron uprig angles of 0°, +10°, and -10°, the model was varied through the angle-of-attack range about the normal flying attitude. The added lift was compensated for by adjusting the down force in the lift cable. Thus, static aerodynamic data, such as model C_{L_α} and C_{L_δ}, and the effect of the aileron on the wing lift distribution could be derived.

Dynamic load measurements.- The purpose of these tests was to determine the dynamic wing and empennage loads produced by oscillating the ailerons and stabilizer. Data were obtained for aileron and stabilizer frequency sweeps at various amplitudes. During these tests the model was restrained by the snubber cables and tie-down cables at the nose and rear of the fuselage. This fuselage restraint was used in an effort to structurally uncouple the wing and empennage approximating a cantilever root condition so that the experimental data could be correlated with analysis where the wing and empennage were treated independently as cantilevered structures.

ALDCS effectiveness tests.- The ALDCS effectiveness tests were made with the model free-flying and a nominal aileron uprig of +5°. The effectiveness of the ALDCS in reducing wing loads was examined for a variety of test variables as shown in figure 33. An earlier version of the control law, identified by analysis as destabilizing a higher structural mode, was also tested.

Flutter tests.- The test procedure was to vary Mach number M and dynamic pressure q along an essentially constant total pressure line with the model ALDCS off. At discrete points along each constant pressure path, the model response to an aileron step was measured. The aileron step was repeated with the ALDCS on. These M-q sweeps were initiated at a low density level in the tunnel, and the sweeps repeated at higher density levels until the envelope was cleared or flutter was obtained. Because of a model Dutch roll type of instability on the cable mount, it was necessary to conduct a portion of the flutter tests with the model snubbed.

Results

The major objectives of the C-5A ALDCS model study were successfully accomplished. The model and the active control systems appeared reasonably representative of the airplane, and the model ALDCS achieved its design goal in reducing wing dynamic bending moment. However, because the model suspension system significantly distorted the rigid body modes, the effect of the ALDCS on these modes was not determined. The ALDCS effect on the model wing flutter characteristics appeared to be negligible, probably because the wing flutter

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mode was antisymmetric, whereas the ALDCS was designed to attenuate symmetric loads only. Some typical results are presented in figures 35 to 38.

The model appeared to simulate reasonably well the overall static aerodynamic characteristics including wing load distribution of the airplane. However, the ailerons were not as effective as those of the airplane. A given model aileron deflection produced less (ranging from about 15 to 35 percent) of a load change than was produced on the airplane for the same deflection. Aileron gains of 1.6 times nominal were therefore included in the test parameters. The dynamic effects on the model loads due to the control surface oscillations compared favorably with analysis.

The ALDCS effectiveness was established best by the aileron frequency sweeps. The aileron sweeps excited higher frequency modes better than either the stabilizer sweeps or the sinusoidal gust sweeps. At the present time, the step/ramp response data have not been reduced sufficiently to determine their quality. The response of the model wing to a typical aileron sweep at the scaled flight condition is shown in figure 35. For these sweeps, the model scaled the 33-percent wing fuel configuration, the aileron amplitude was set for ±5°, the ALDCS was at nominal aileron gain, and the pitch SAS was on. The normalized wing bending moment at the wing root station is shown on the left plot; the normalized wing torsional moment is shown on the right. The frequency of the wing first bending mode was about 4 Hz. Another mode which contributed significantly to the model response had a frequency of about 11 Hz. It can be seen that the major effect of the ALDCS is to reduce the bending moments by about 50 percent at the 4-Hz mode; the torsional moments were also reduced by the ALDCS. The distribution of the bending and torsional moments along the wing span is shown for the same conditions in figures 36 and 37. In figure 36 the bending moments pertain to the 4-Hz mode. The torsional moment in figure 37 is for the 11-Hz mode where the torsional moments are greatest. It can be seen that the load reduction experienced at the root is obtained in nearly the same proportion over the entire span.

In the flutter tests, the 33-percent wing fuel configuration did not flutter within the scaled flight envelope. The 0-percent fuel configuration experienced antisymmetric wing flutter at the two points shown in figure 38. The data in this figure are presented in the form of airplane equivalent airspeed. In each instance, the ALDCS had no effect on the flutter. The model flutter occurred at a frequency of about 13 Hz and appeared to consist of a combined higher wing bending and torsional mode with most of the motion on the outboard portion of the wing. A similar type of wing flutter occurred during earlier flutter model tests.

CONCLUDING REMARKS

In this paper the experiences to date in testing aeroelastic models equipped with active controls in the Langley Research Center transonic dynamics tunnel have been described. Such items as model design, construction, and test techniques have been described in discussing three model experimental programs.

Also, some typical data results have been presented. The three model studies were a simple delta-wing flutter suppression model, a 1/30-size dynamically scaled aeroelastic model of the B-52 CCV, and a 1/22-size dynamically scaled aeroelastic model of the C-5A aircraft. The delta-wing model was used to evaluate the aerodynamic energy concept of flutter suppression. The B-52 model was equipped with flutter mode control and ride quality control systems, while the C-5A model was equipped with a lift distribution control system. In all three studies active controls were successfully implemented on the models. The delta-wing flutter suppression system did provide an increase in flutter dynamic pressure, and the experimental results are in reasonable agreement with analytical trends. Both B-52 model systems provided improved performance, and the FMC experimental results compare favorably with analytical and flight data. The C-5A ALDCS did provide a significant reduction in incremental dynamic bending moment on the wing with no apparent effect on the flutter characteristics.

Experiences with these models have indicated that the addition of active controlled aerodynamic surfaces has, indeed, added complexity to aeroelastic modeling technology. However, no insurmountable obstacles have been encountered in these three studies, and the success to date indicates that much useful information can be obtained from model test results.

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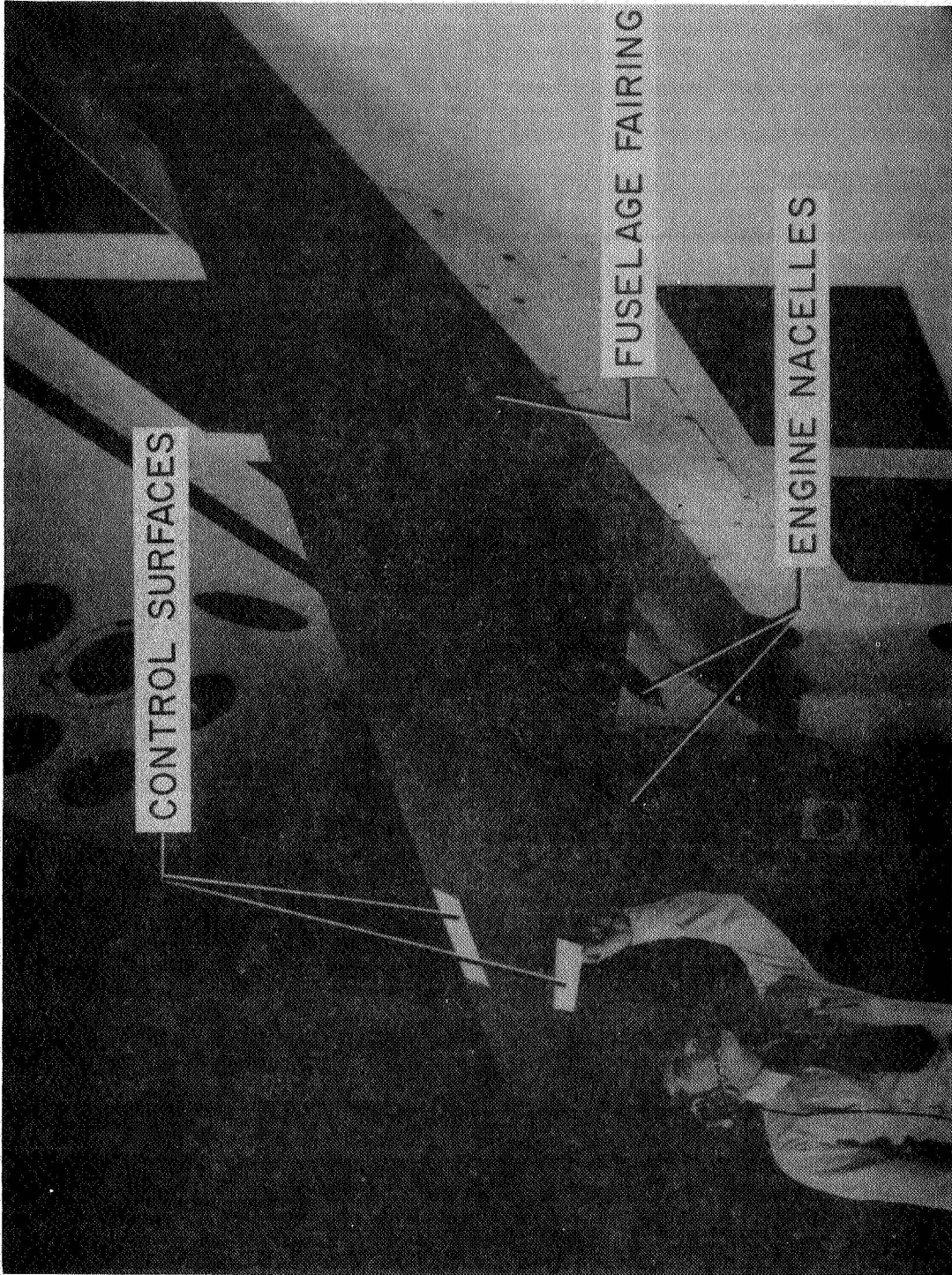
Figure 1.- Delta-wing model mounted in transonic dynamics wind tunnel.

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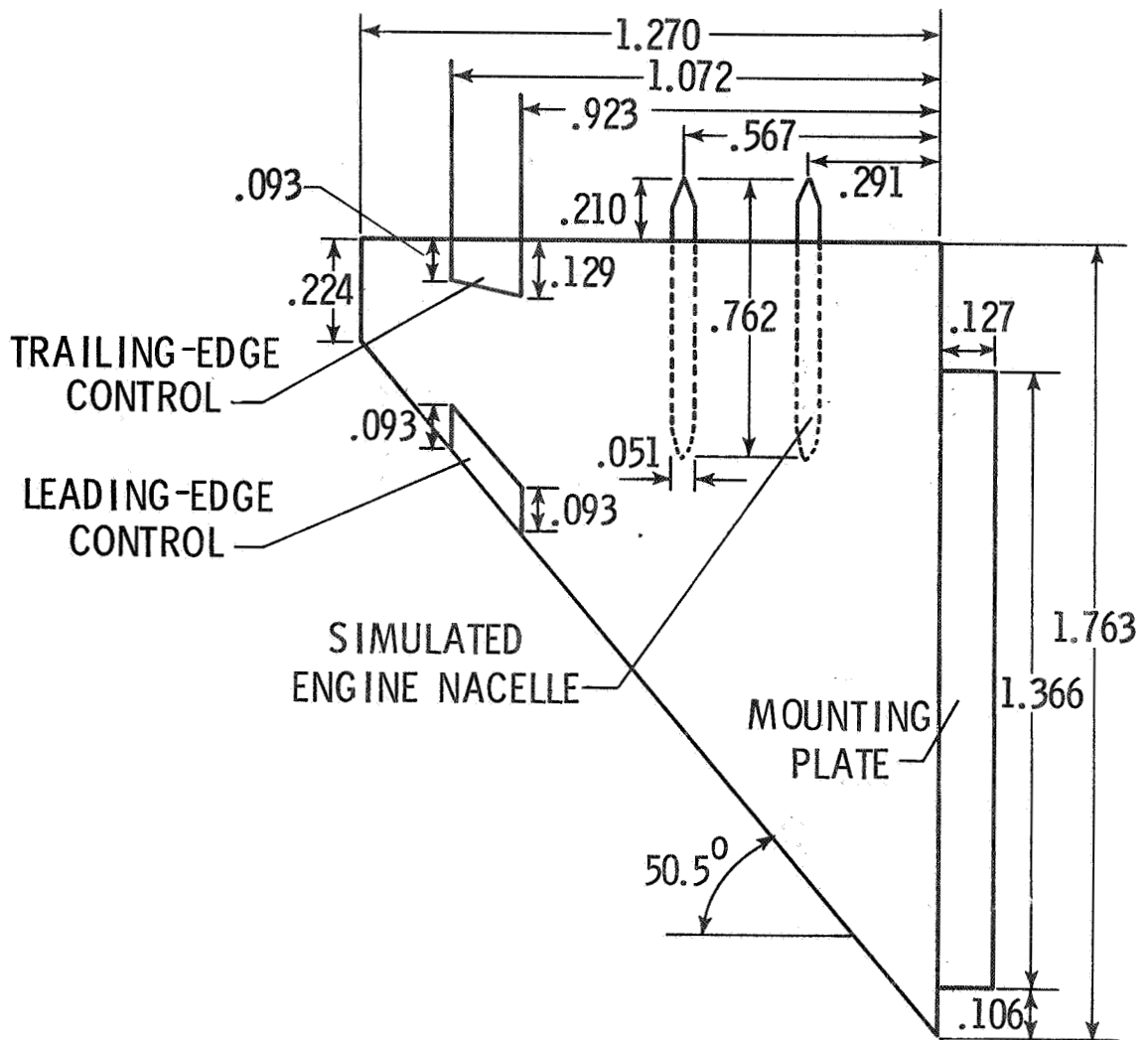
Figure 2.- Sketch of delta-wing model. (All linear dimensions are in meters.)

$$\begin{Bmatrix} \beta \\ \delta \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{12} \\ C_{21} & C_{22} \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix} + i \begin{bmatrix} G_{11} & G_{12} \\ G_{21} & G_{22} \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix}$$
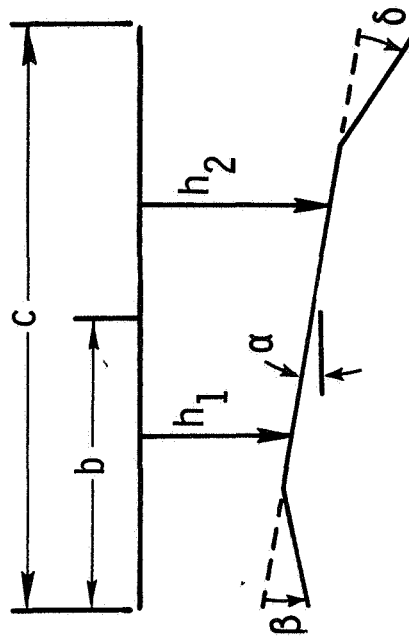
SECTION A-A
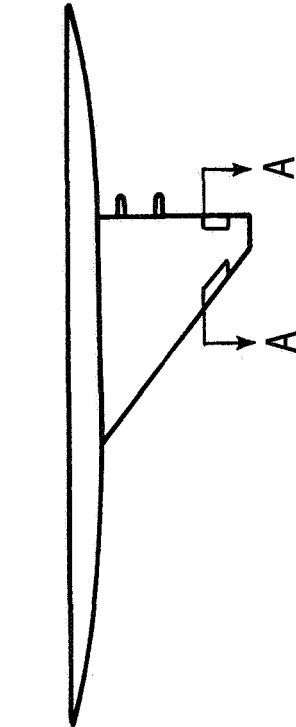
Figure 3.- Basic·control law parameters.

CONTROL LAW A

$$\begin{Bmatrix} \beta \\ \delta \end{Bmatrix} = \begin{bmatrix} 0 & 5.6 \\ 0 & -1.4 \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix} + i \begin{bmatrix} 0 & 1.5 \\ .6 & .2 \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix}$$

CONTROL LAW B

$$\begin{Bmatrix} \beta \\ \delta \end{Bmatrix} = \begin{bmatrix} 0 & 0 \\ 2.7 & -5.3 \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix} + i \begin{bmatrix} 0 & 2.5 (6.0) \\ 1.0 (4.0) & .75 (2.0) \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix}$$

CONTROL LAW C

$$\begin{Bmatrix} \beta \\ \delta \end{Bmatrix} = \begin{bmatrix} 0 & 0 \\ 2.7 & -5.3 \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix} + i \begin{bmatrix} 0 & 0 \\ 2.5 (4.0) & .75 (2.0) \end{bmatrix} \begin{Bmatrix} \dfrac{h_1}{b} \\ \alpha \end{Bmatrix}$$

Figure 4.- Delta-wing model control laws.
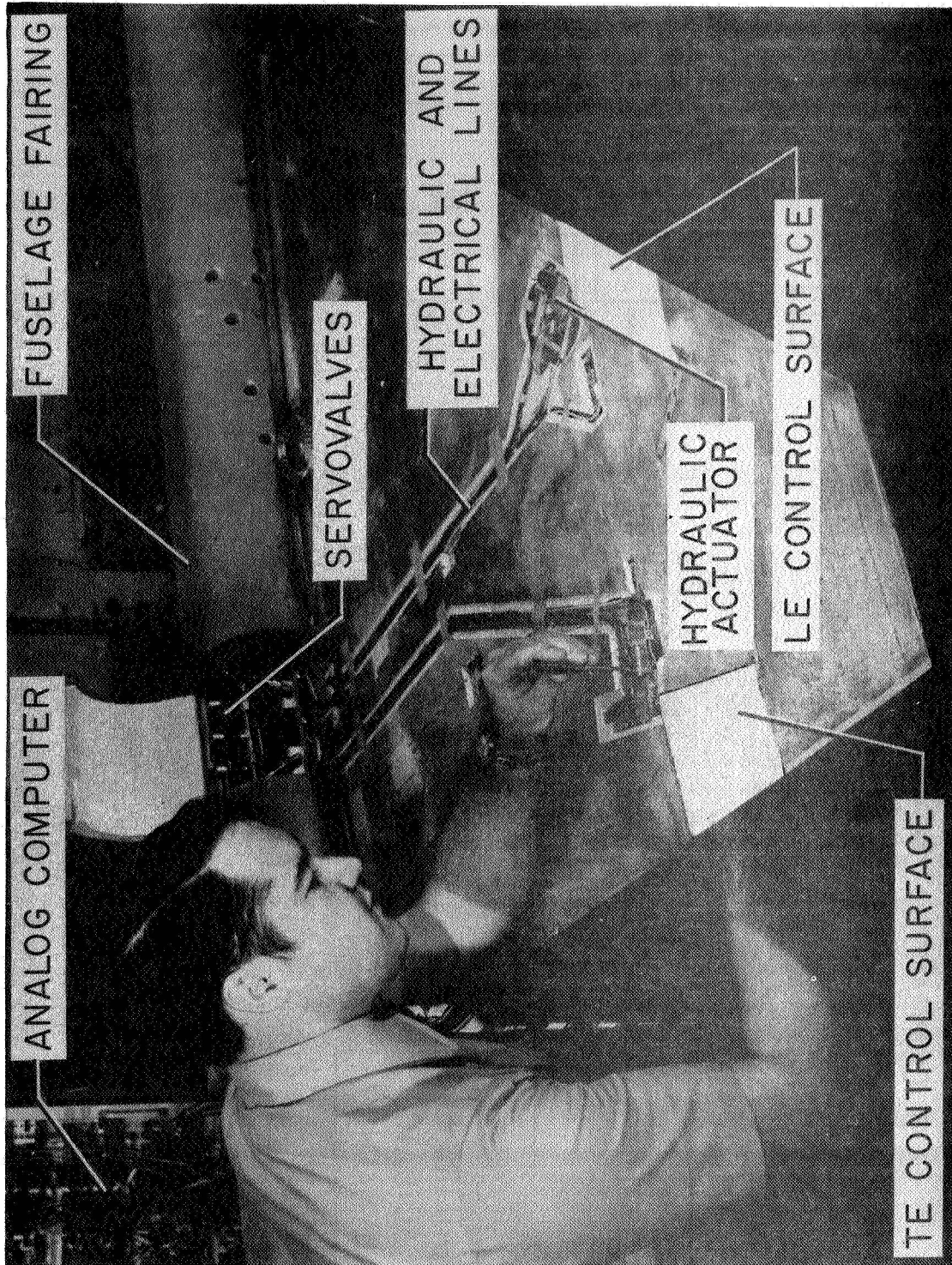
Figure 5.- Components of delta-wing flutter suppression system.

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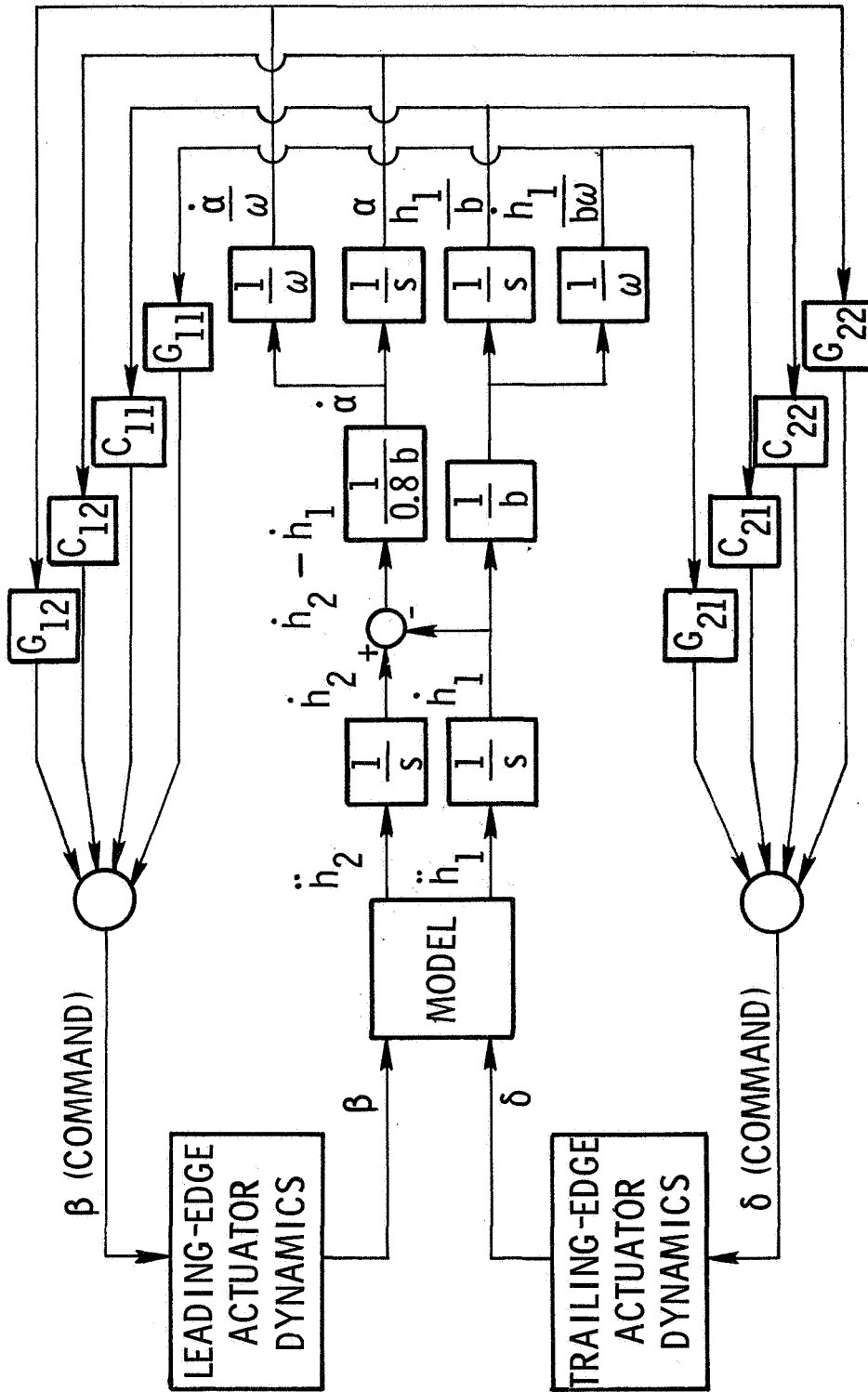
Figure 6.— Simplified block diagram of delta-wing flutter suppression system.
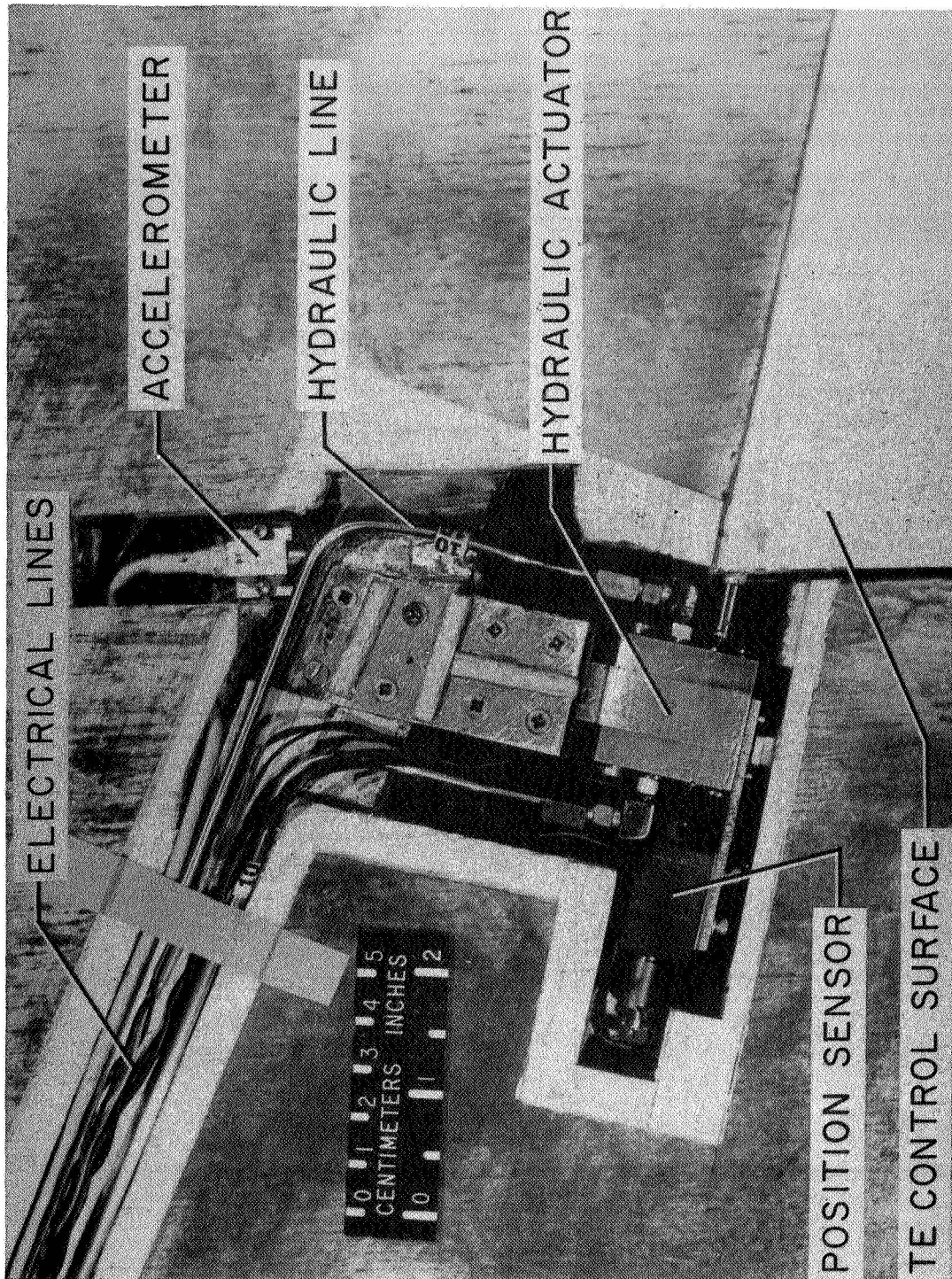
Figure 7.— Delta-wing trailing-edge control actuation system.

PERCENT INCREASE IN
FLUTTER DYNAMIC PRESSURE

SURFACE LOCATION	ACCELEROMETER LOCATION		
	INBOARD EDGE	CENTER LINE	OUTBOARD EDGE
INBOARD STRIP	18	22	21
MIDSPAN STRIP	26	17	8
OUTBOARD STRIP	6	25	36
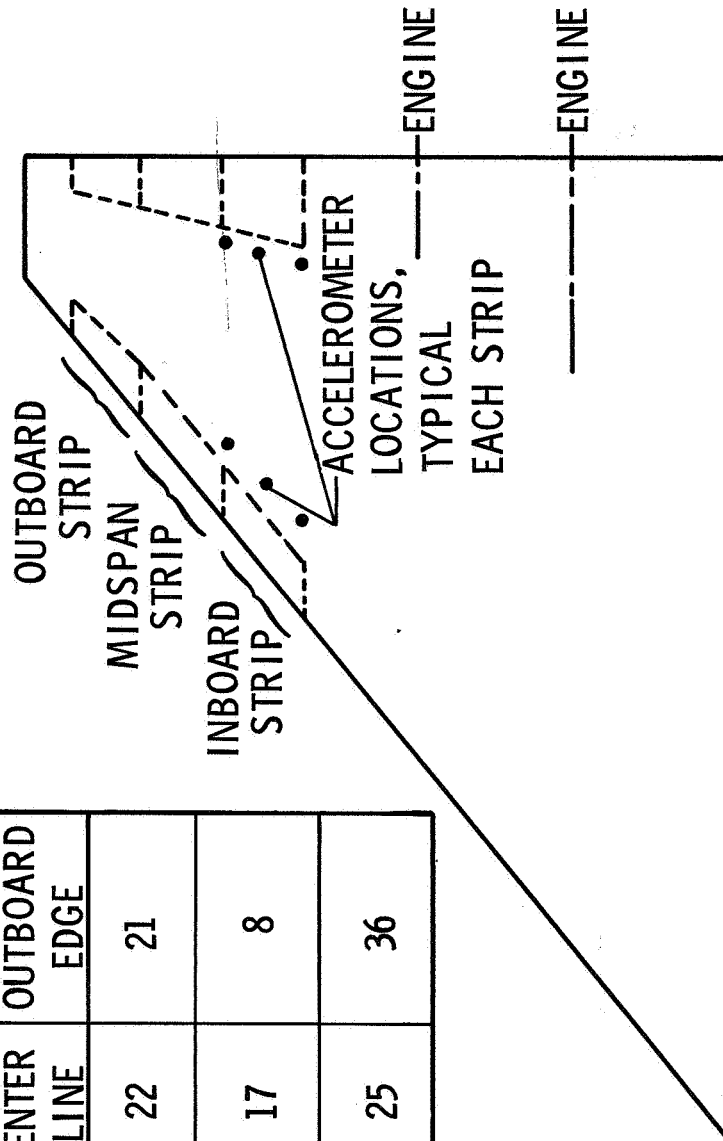
Figure 8.– Effects of control surface and accelerometer locations on flutter dynamic pressure.
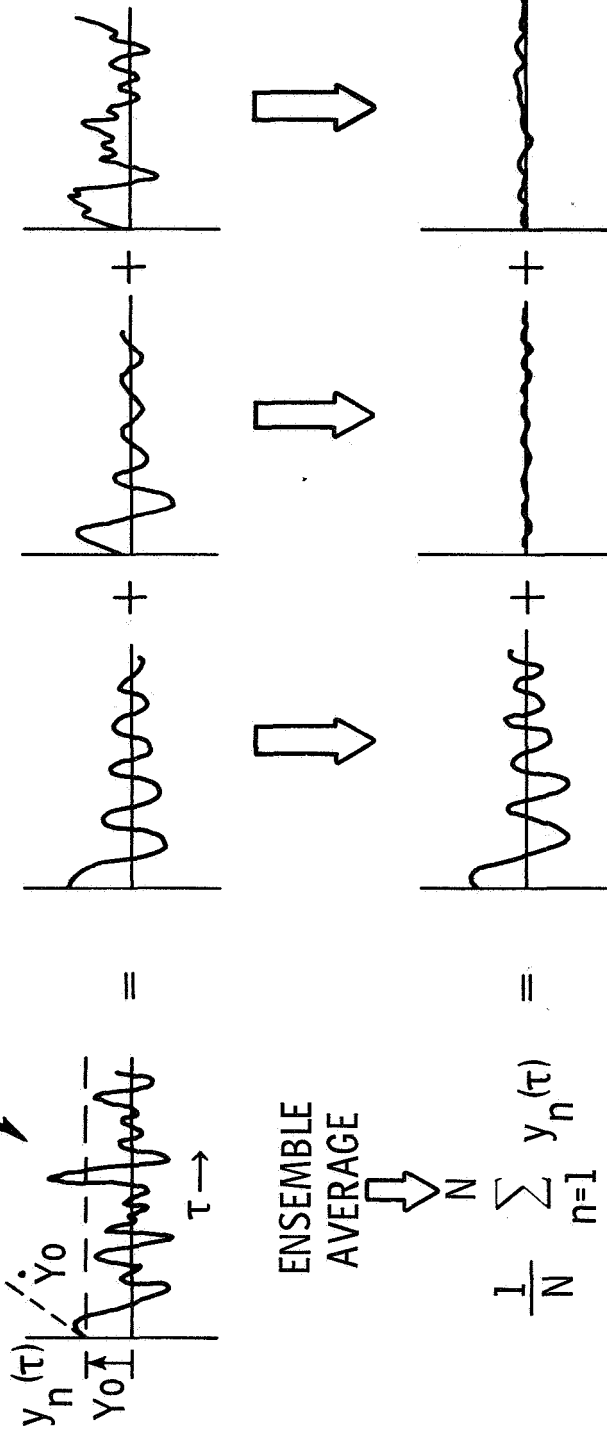
Figure 9.– Basic randomdec concepts.

863

$$\text{DAMPING} = \frac{\left[f_A/f_B\right]^2 - 1}{\left[f_A/f_B\right]^2 + 1}$$
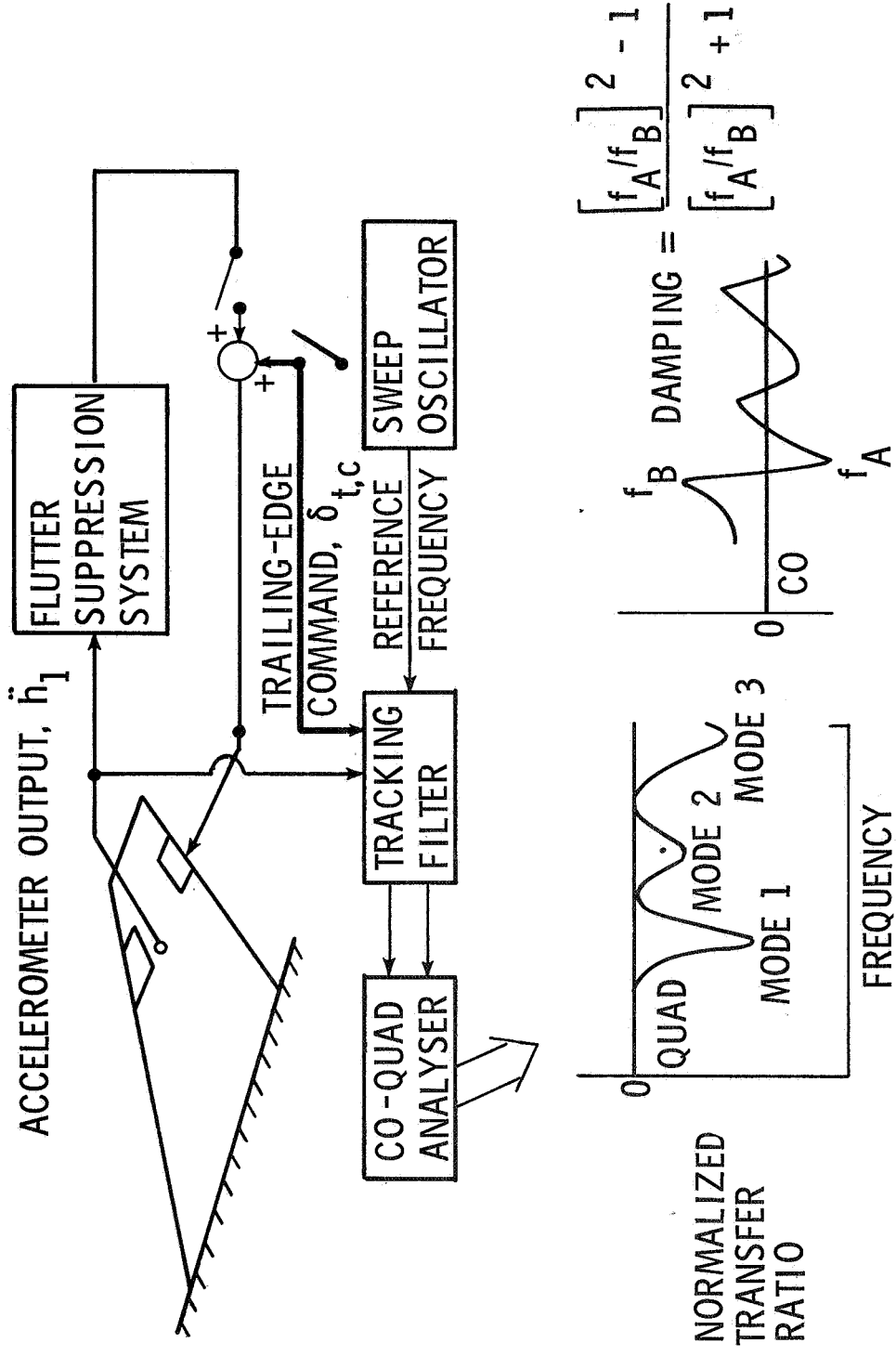
Figure 10.– Co-Quad subcritical response method.
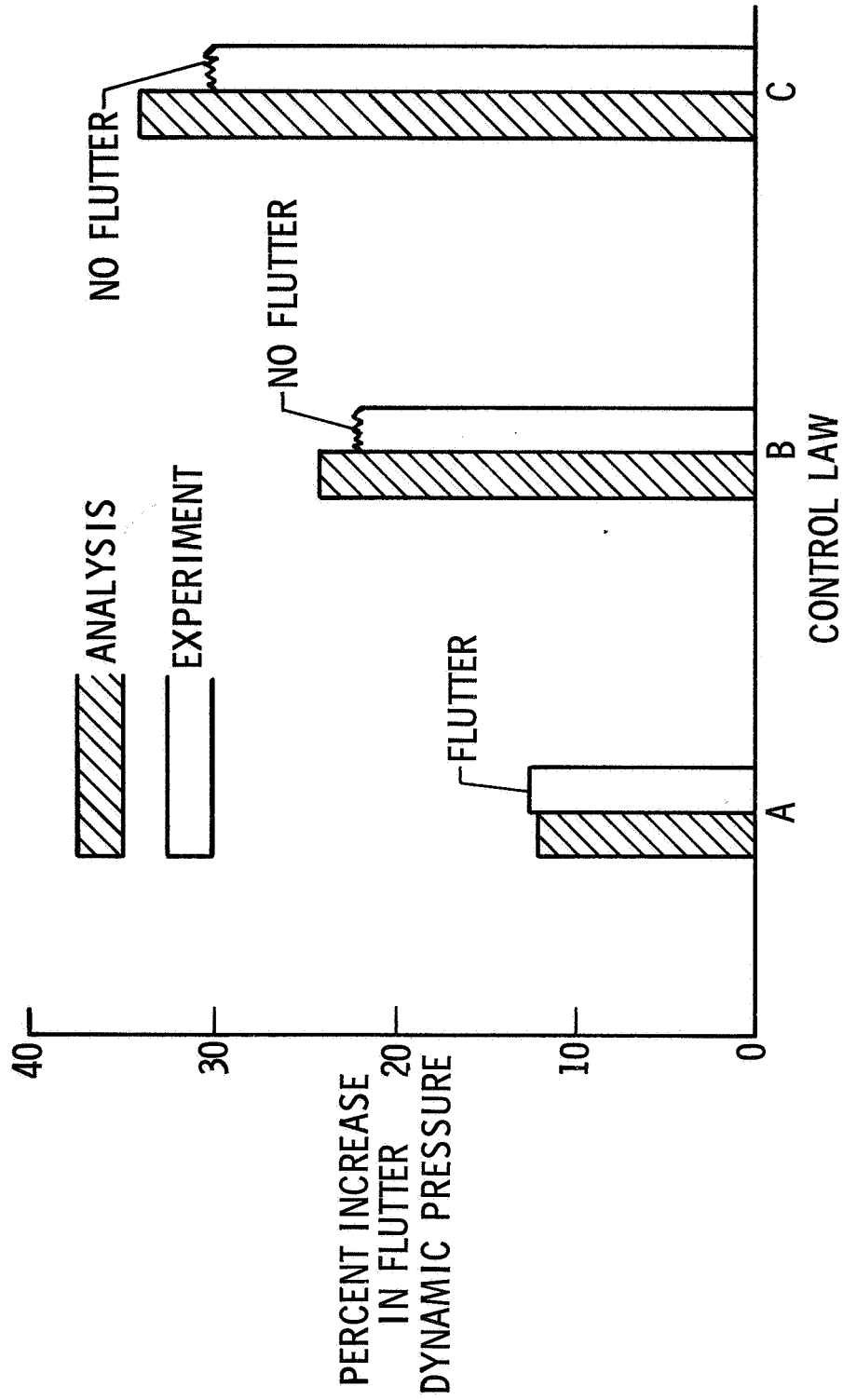
Figure 11.- Effect of different control laws on flutter dynamic pressure at M = 0.9.
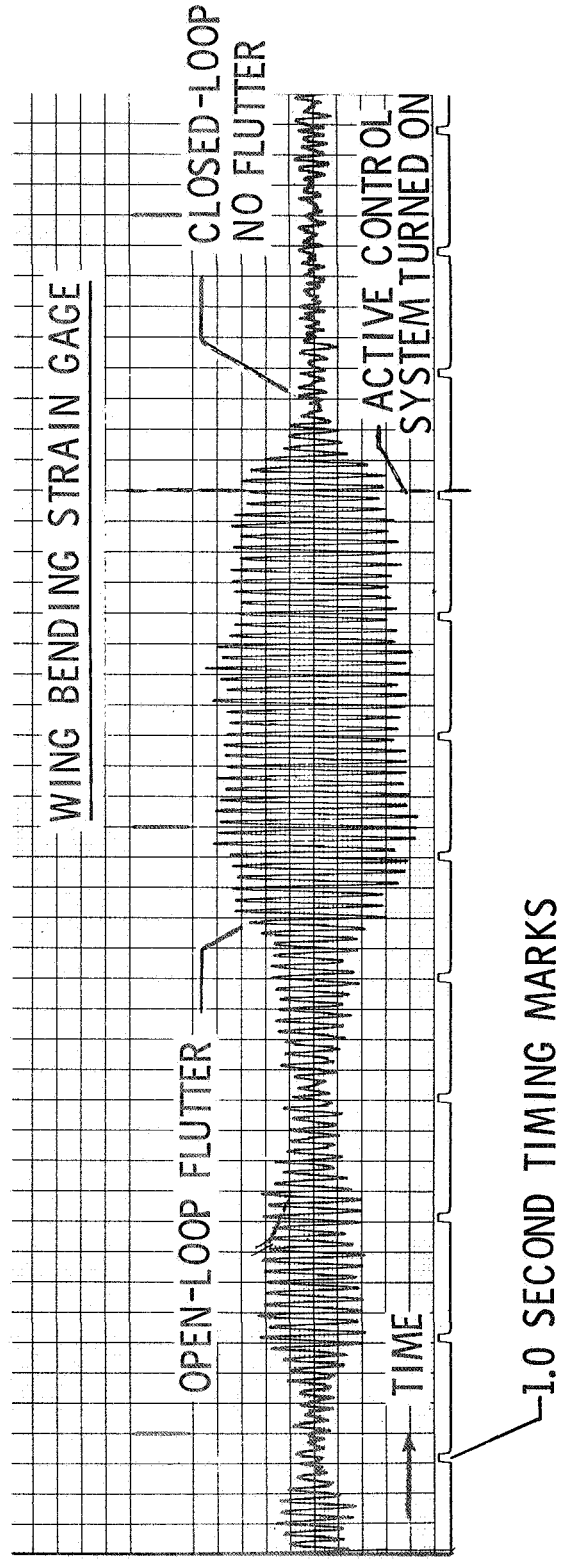
Figure 12.- Time history trace showing effective operation of delta-wing flutter suppression system (Control Law C).

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VELOCITY
INDEX
PARAMETER,
$\dfrac{v}{b\omega_r\sqrt{\mu}}$

EXPERIMENT

OPEN-LOOP ●

CLOSED-LOOP ▲
(NO FLUTTER)

ANALYSIS

OPEN-LOOP --□--

CLOSED-LOOP ◇
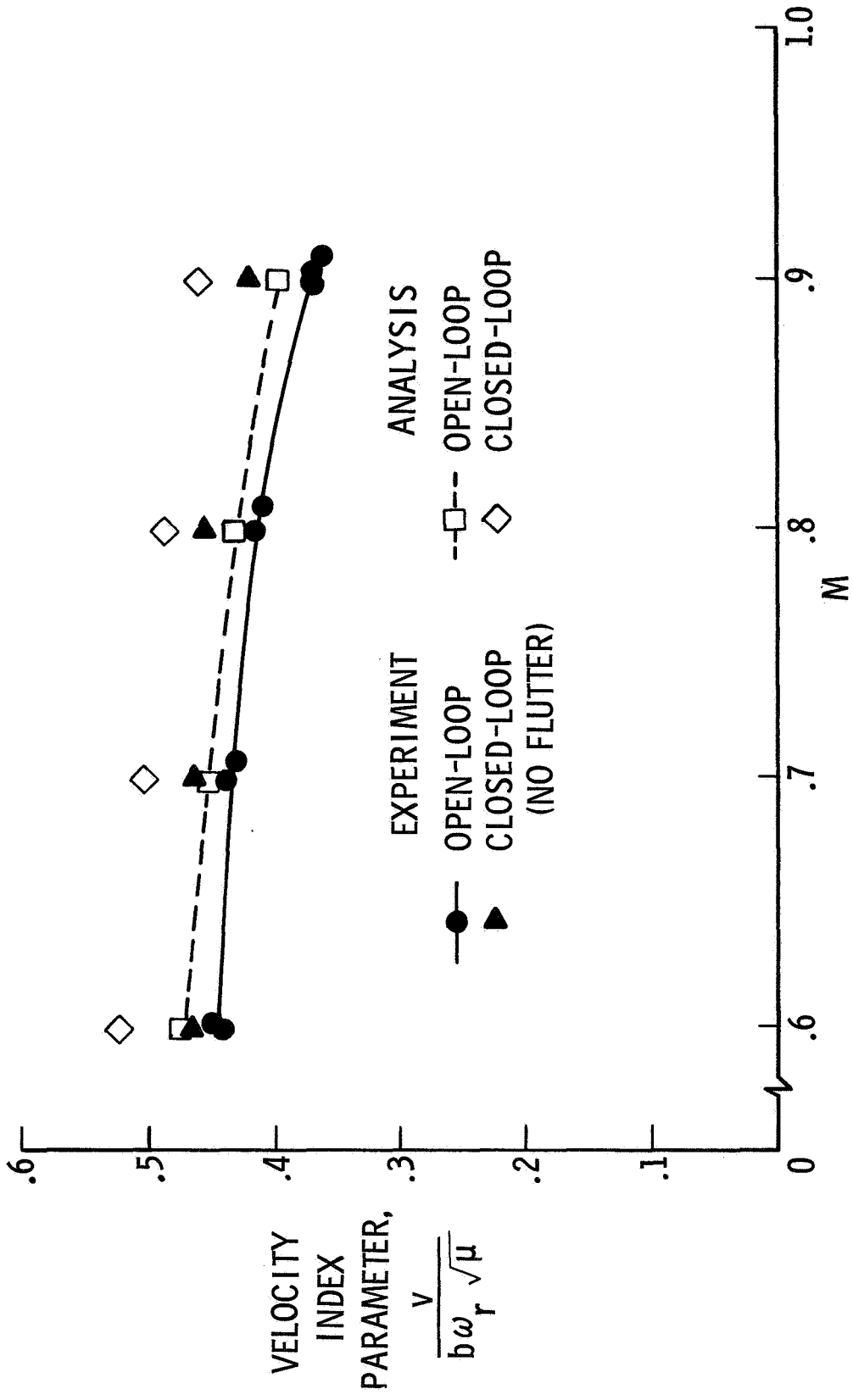
Figure 13.— Measured and calculated variation of flutter velocity index parameter with Mach number.

867
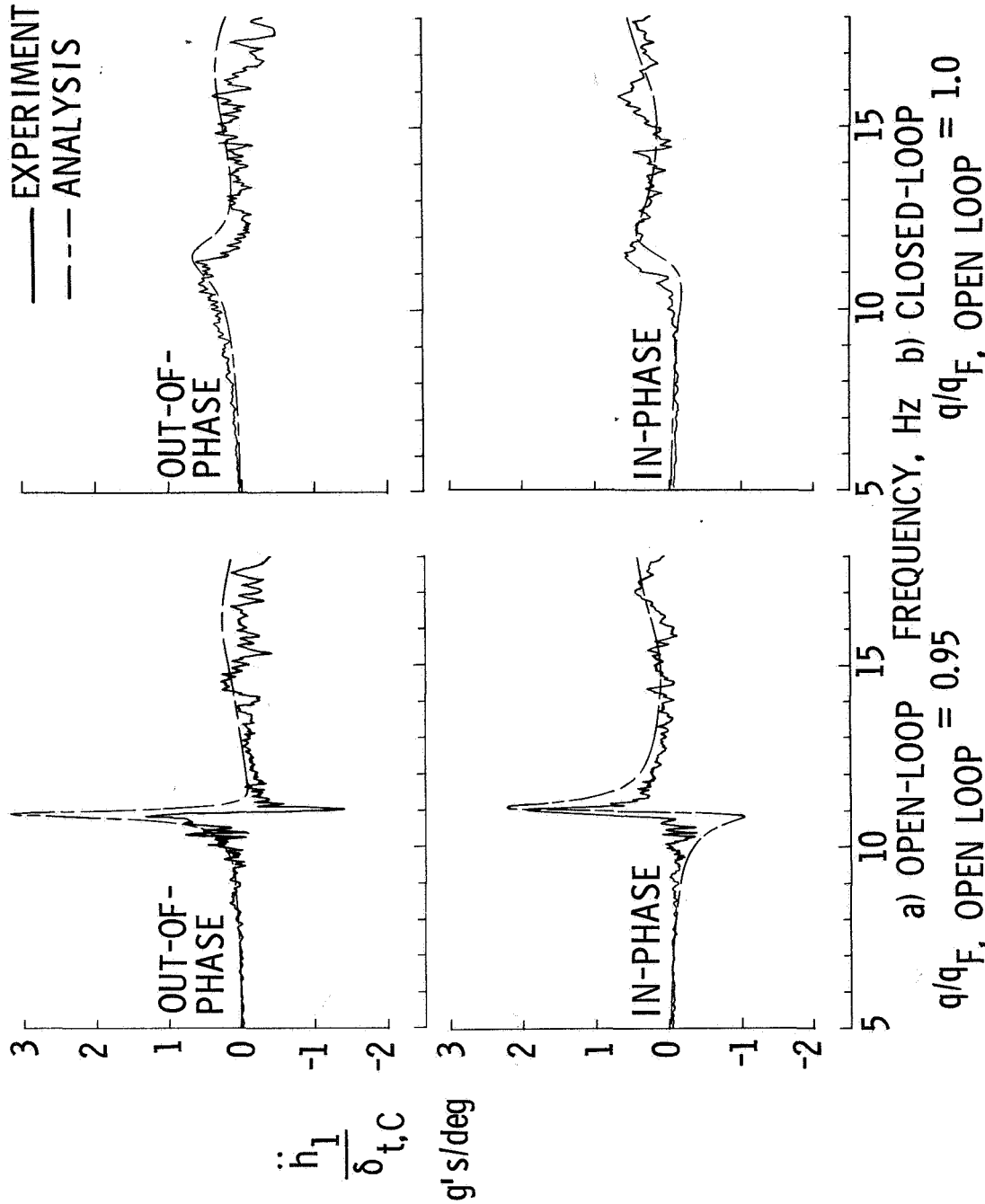
Figure 14.- A comparison of measured and calculated forced response to trailing-edge-control excitation at M = 0.9.
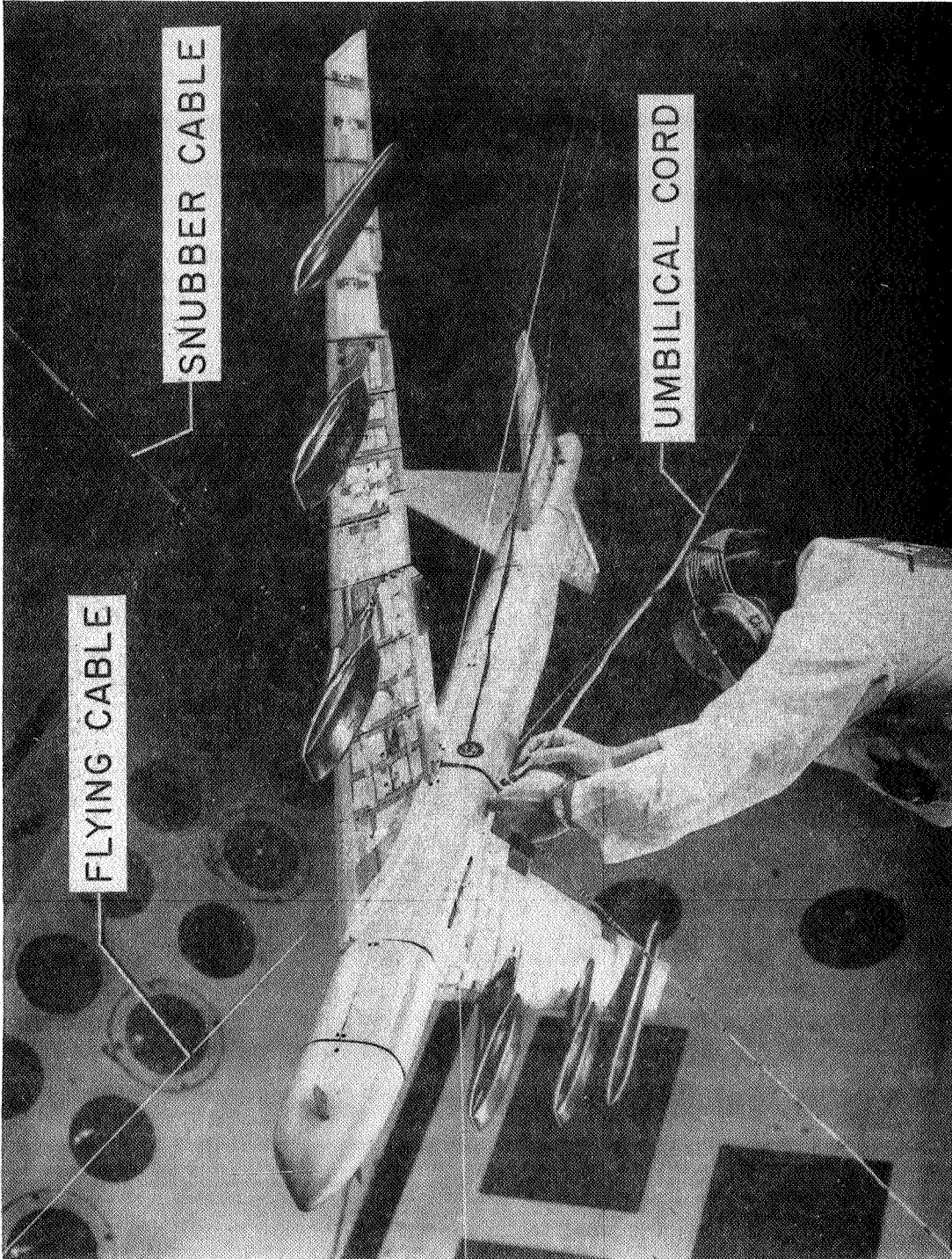
Figure 15. B-52 model mounted in transonic dynamics tunnel.

BALLAST

OUTBOARD FLAPERON SEGMENT

OUTBOARD AILERON

OUTBOARD ACCELEROMETER
$\ddot{Z}_{WBL}$ 78.3cm

INBOARD ACCELEROMETER
$\ddot{Z}_{WBL}$ 47.8cm

HORIZONTAL CANARD

PILOT STATION ACCELEROMETER
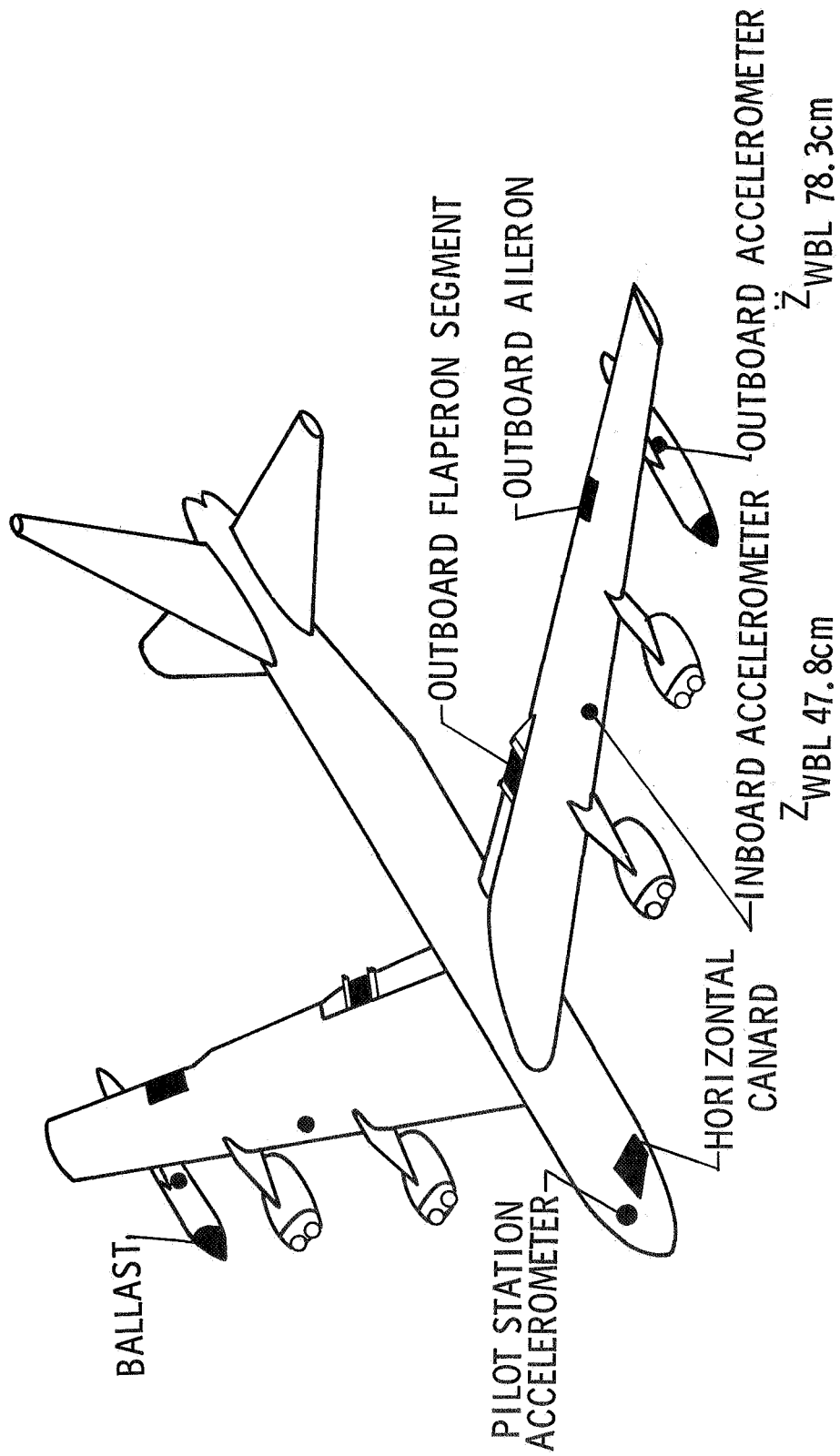
Figure 16.- B-52 model control surface and sensor locations.

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- FROUDE NUMBER, $\dfrac{\text{FLUID ACCELERATION}}{\text{GRAVITY ACCELERATION}}$, $\dfrac{V^2}{Lg}$

- REDUCED WAVELENGTH, $\dfrac{\text{FLUID VELOCITY}}{\text{VIBRATION VELOCITY}}$, $\dfrac{V}{\omega L}$

- MASS RATIO, $\dfrac{\text{MASS OF BODY}}{\text{APPARENT MASS OF FLUID}}$, $\dfrac{m}{\rho_f L^3}$

MODEL / AIRPLANE DESIGN CONDITIONS

	AIRPLANE	MODEL
GROSS WEIGHT	170 097 Kgm	25.67 Kgm
ALTITUDE	6 400 m (FMC) 1 646 m (RQC)	————
DENSITY	.631 Kg/m^3 (FMC) 1.041 Kg/m^3 (RQC)	2.572 Kg/m^3 4.123 Kg/m^3
MEDIUM	AIR	FREON (95%)

Figure 17.- B-52 model scaling parameters.

FUSELAGE SPAR

WING POD

FLAPERON

WING SPAR

TORQUE MOTOR

CONTROL SURFACE LINKAGES
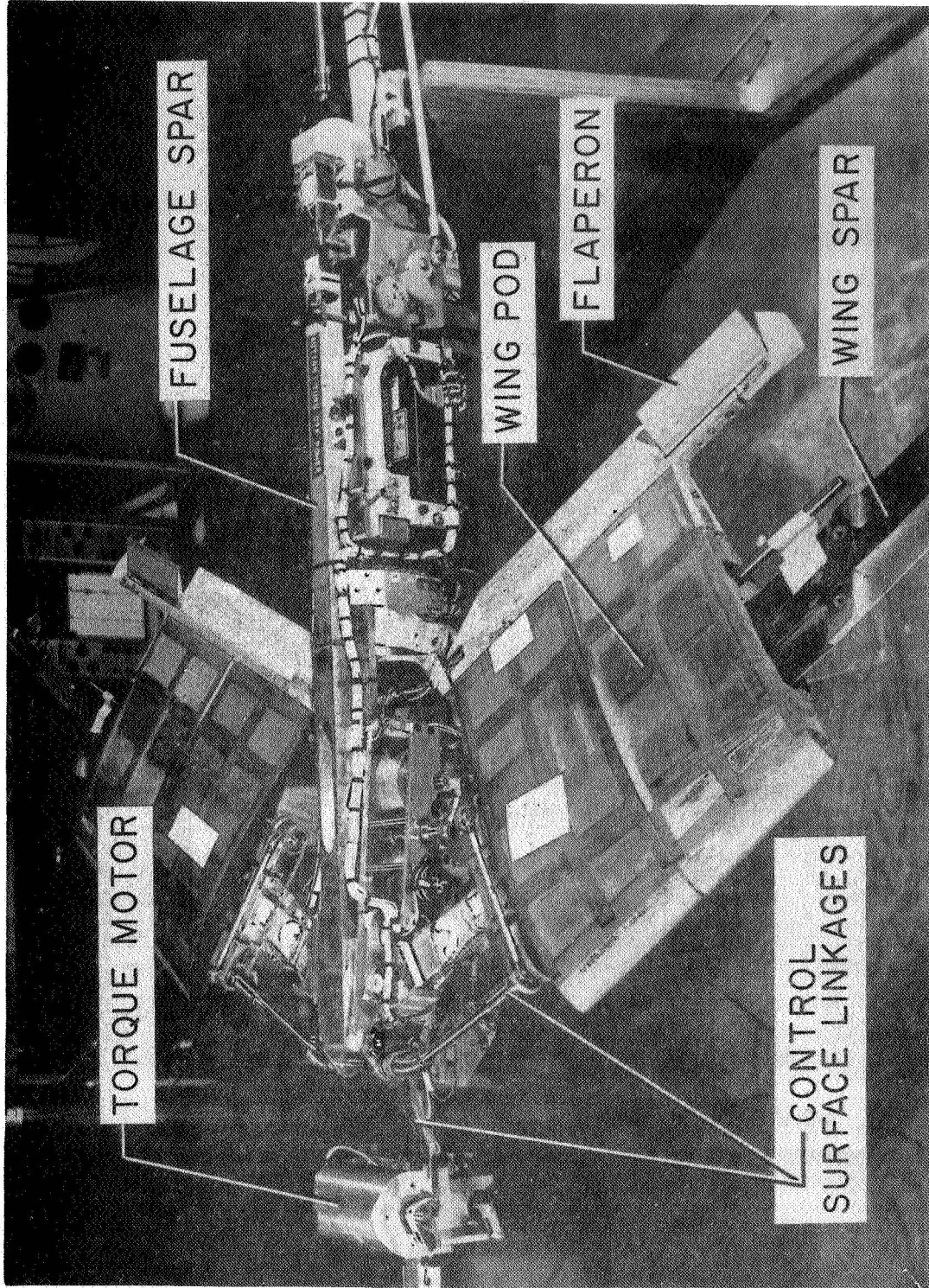
Figure 18.- Exposed view of B-52 model.

872
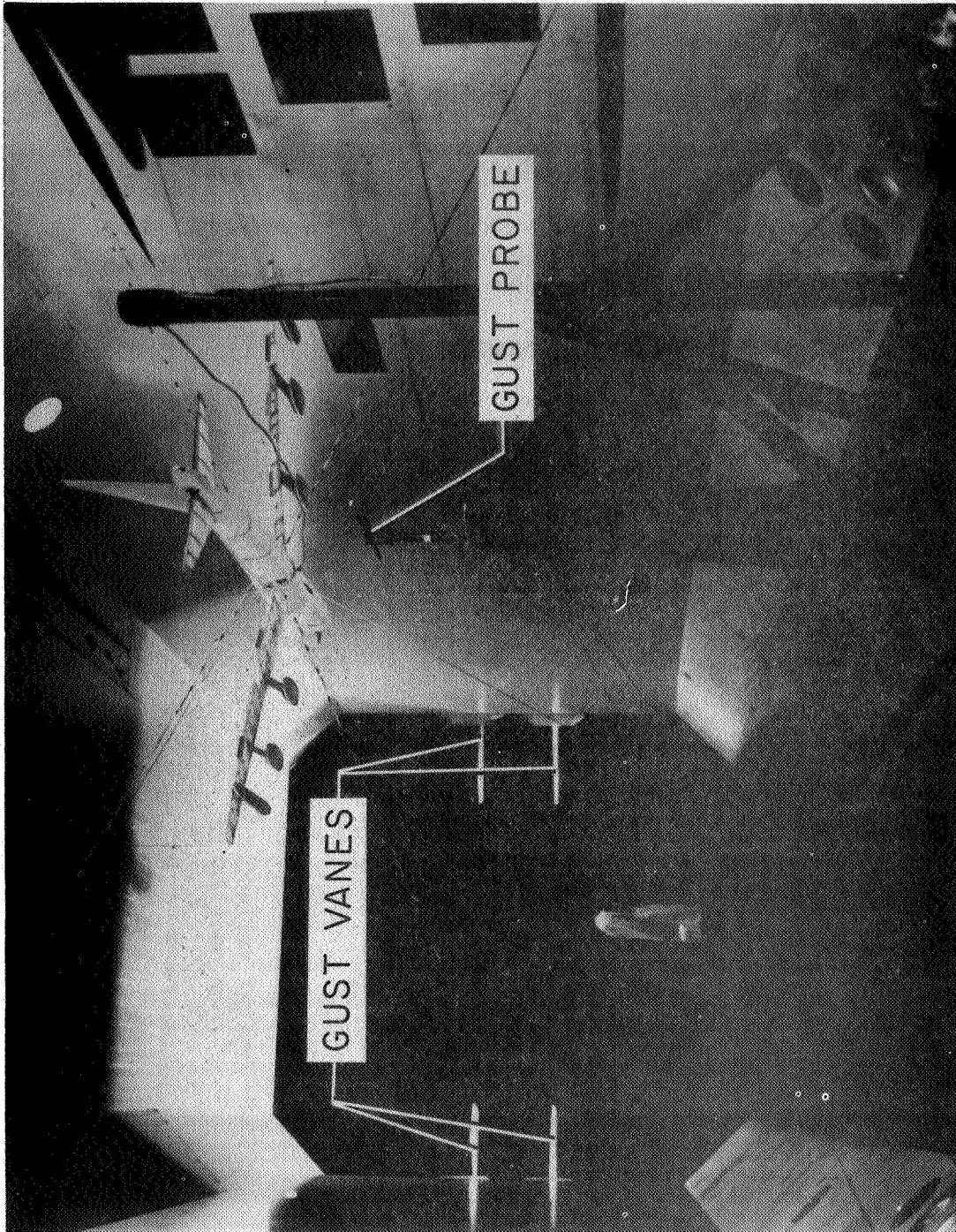
Figure 19.- View of B-52 model showing gust generating vanes.
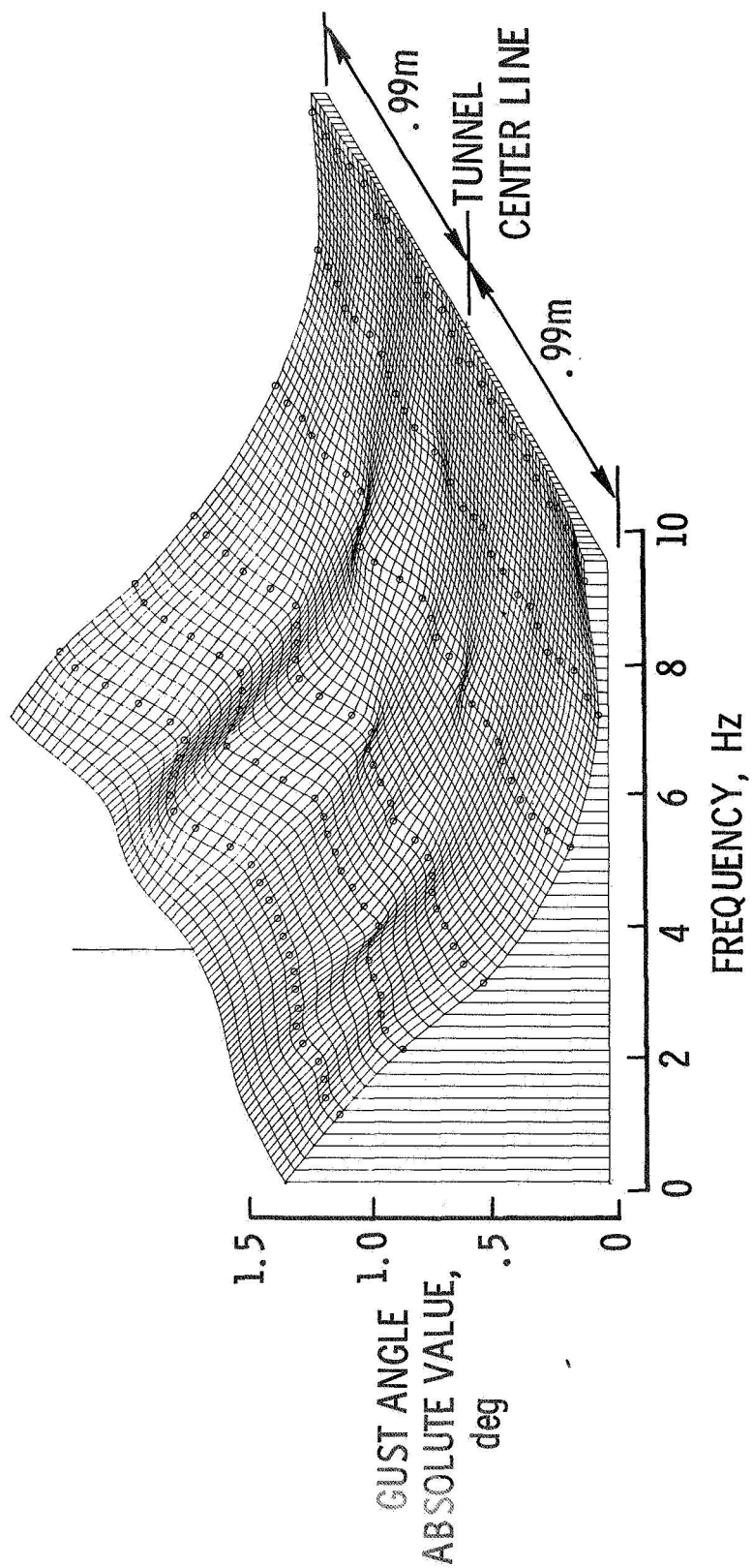
Figure 20.– Typical variation of gust flow angle with frequency and tunnel lateral position for an oscillating vane angle of 6° and a velocity of 35.4 m/sec.

874
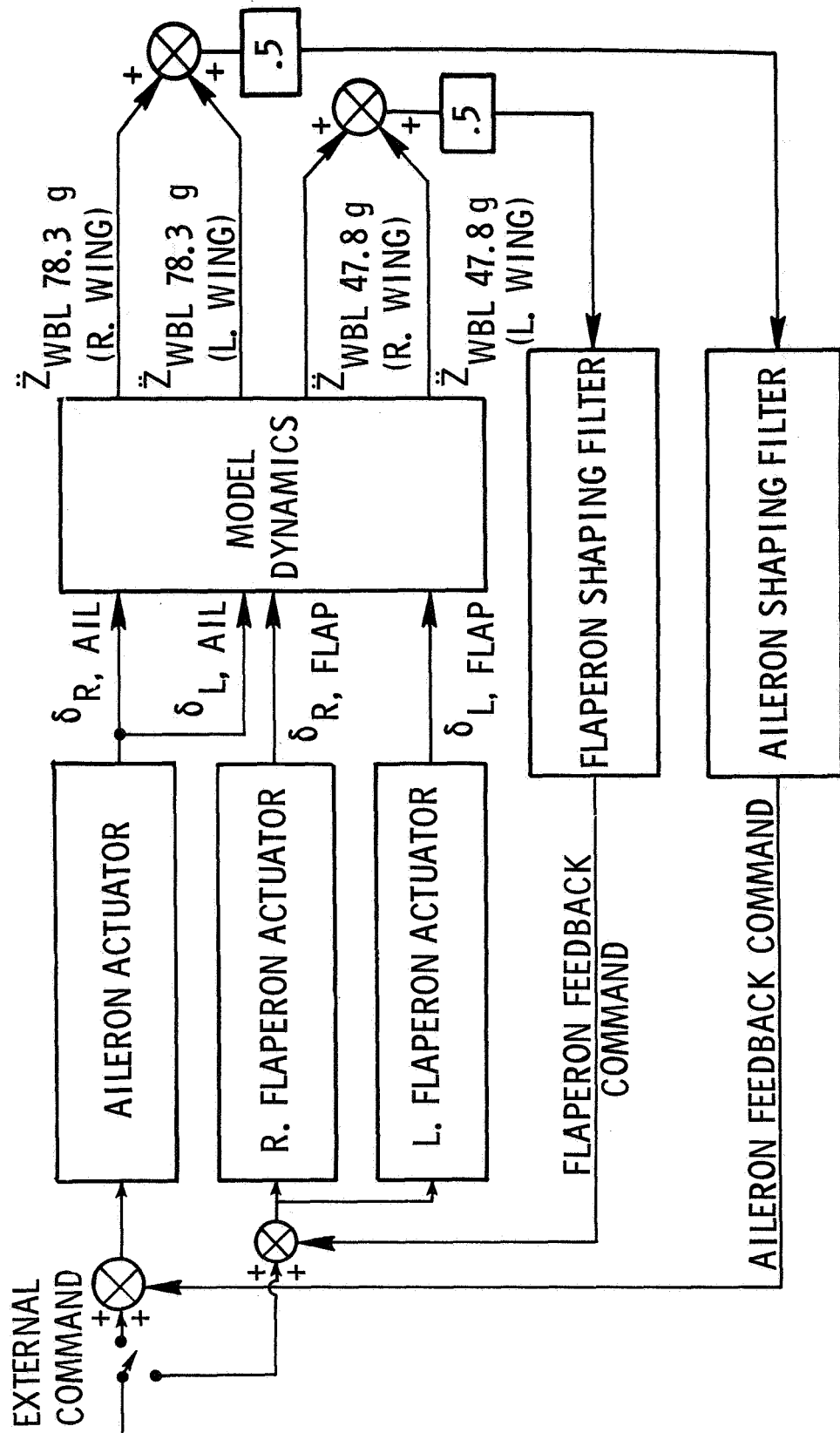
Figure 21.— Simplified block diagram of B-52 model FMC system.
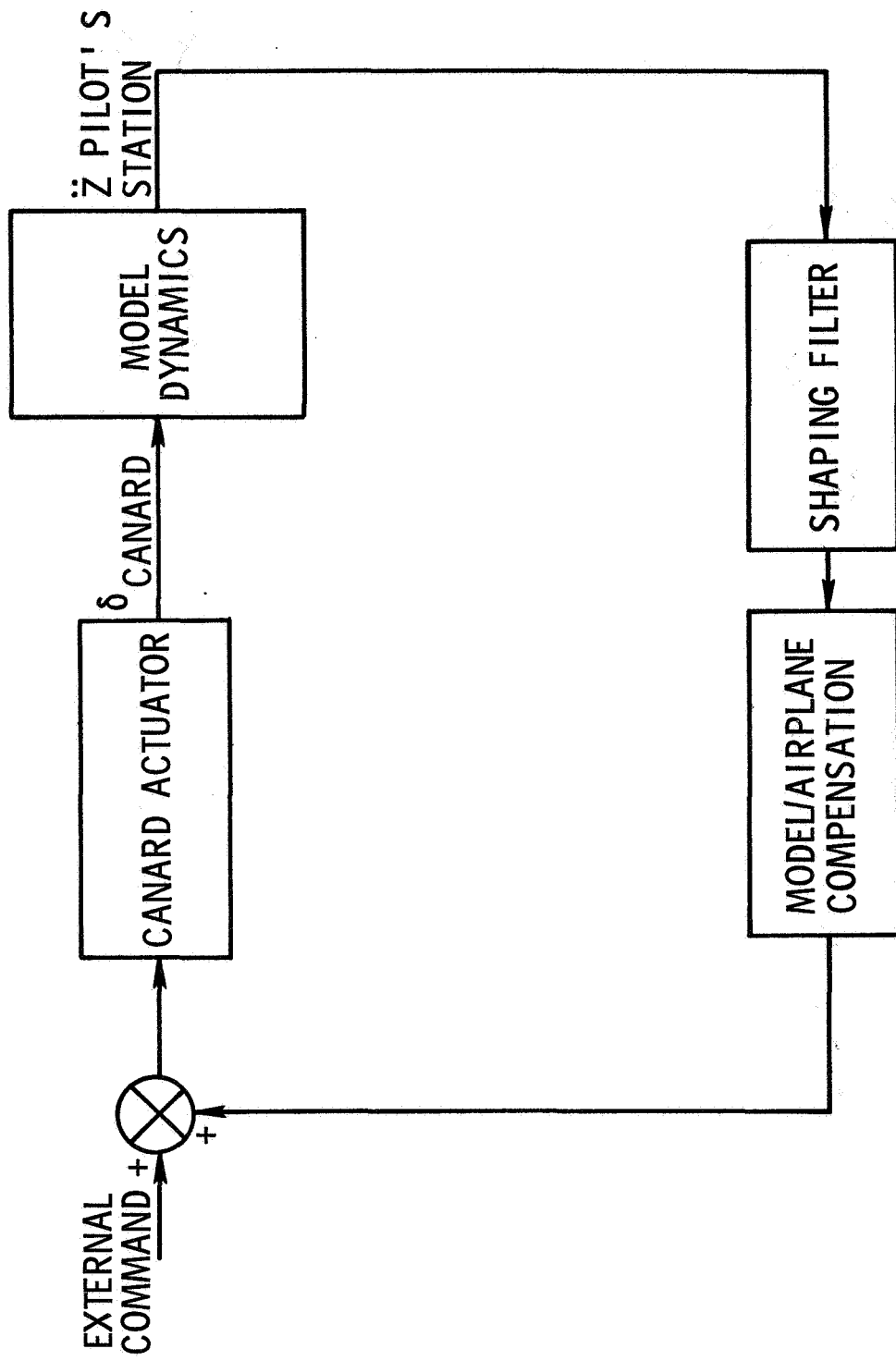
Figure 22.- Simplified block diagram of B-52 model RQC system.

876
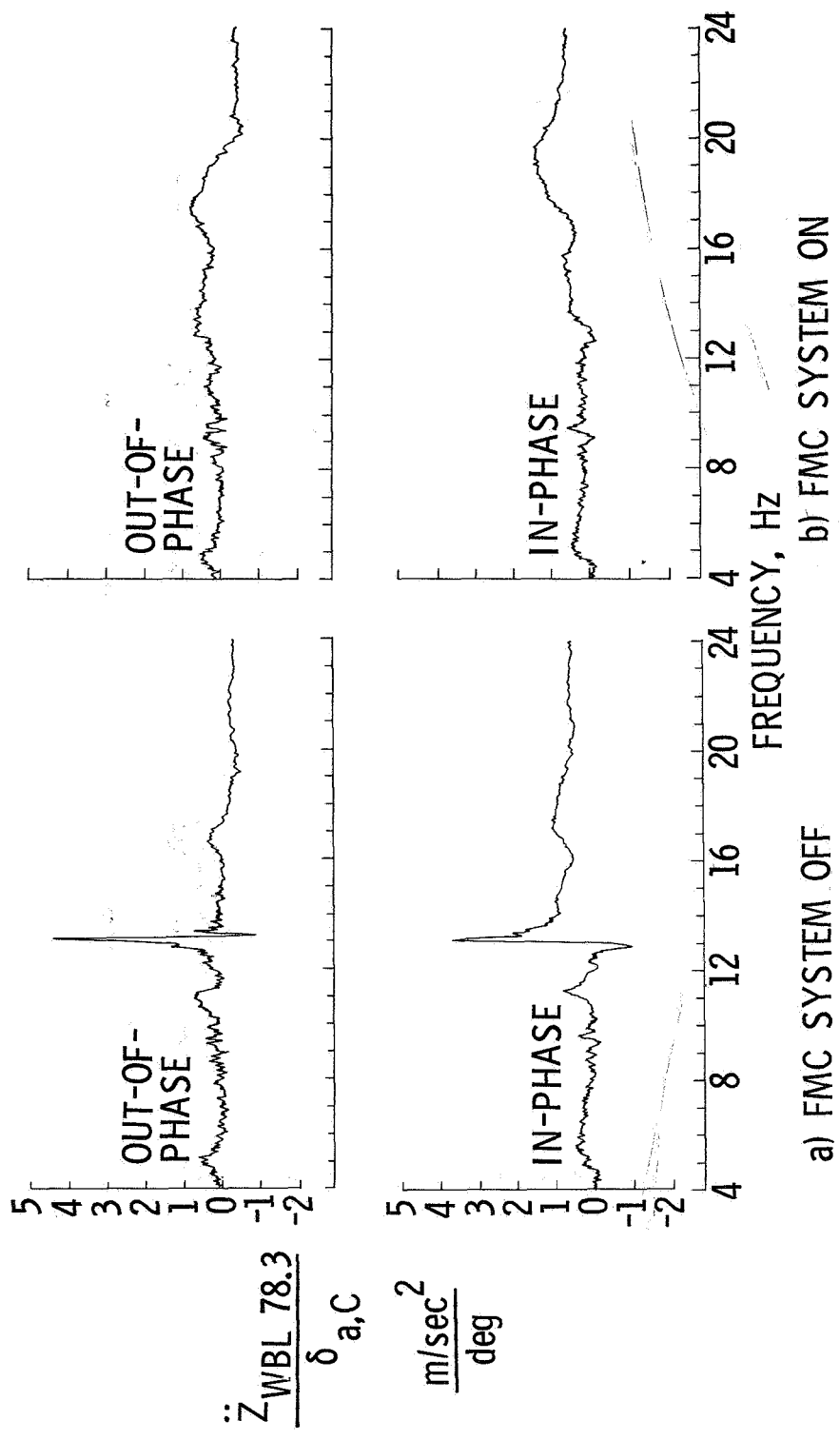
Figure 23.— Measured frequency response of B-52 model to aileron excitation at a velocity 6 percent below open-loop flutter velocity.
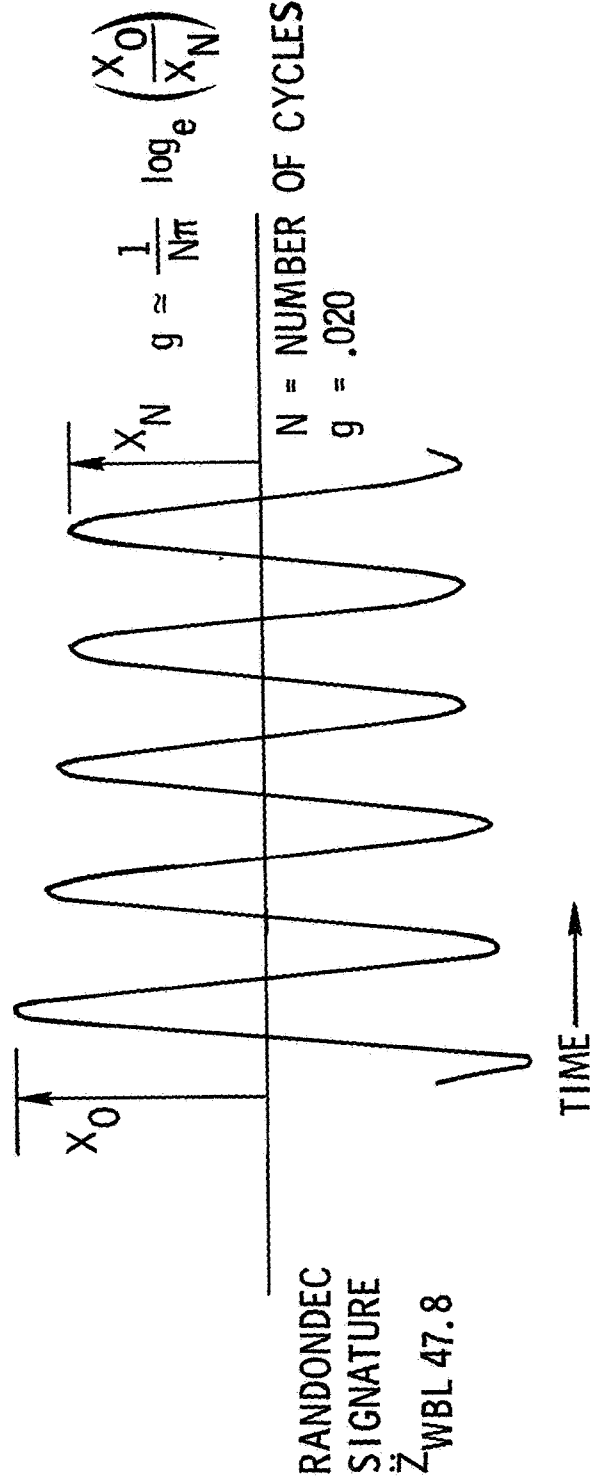
Figure 24.- Wing response as measured with randomdec technique at a velocity of 3 percent below open-loop flutter velocity.
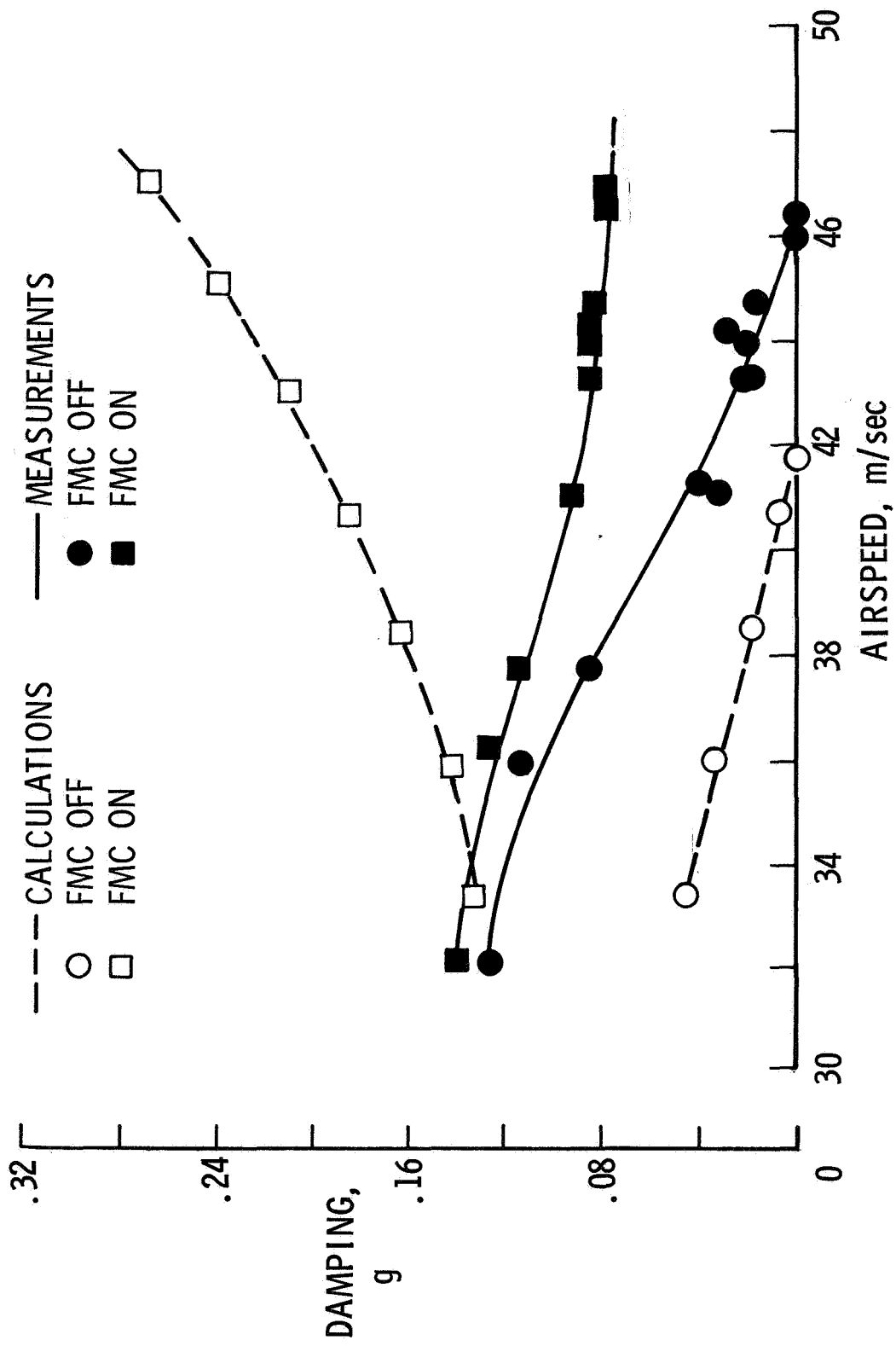
Figure 25.- Comparison of measured and calculated damping for the B-52 model.

879
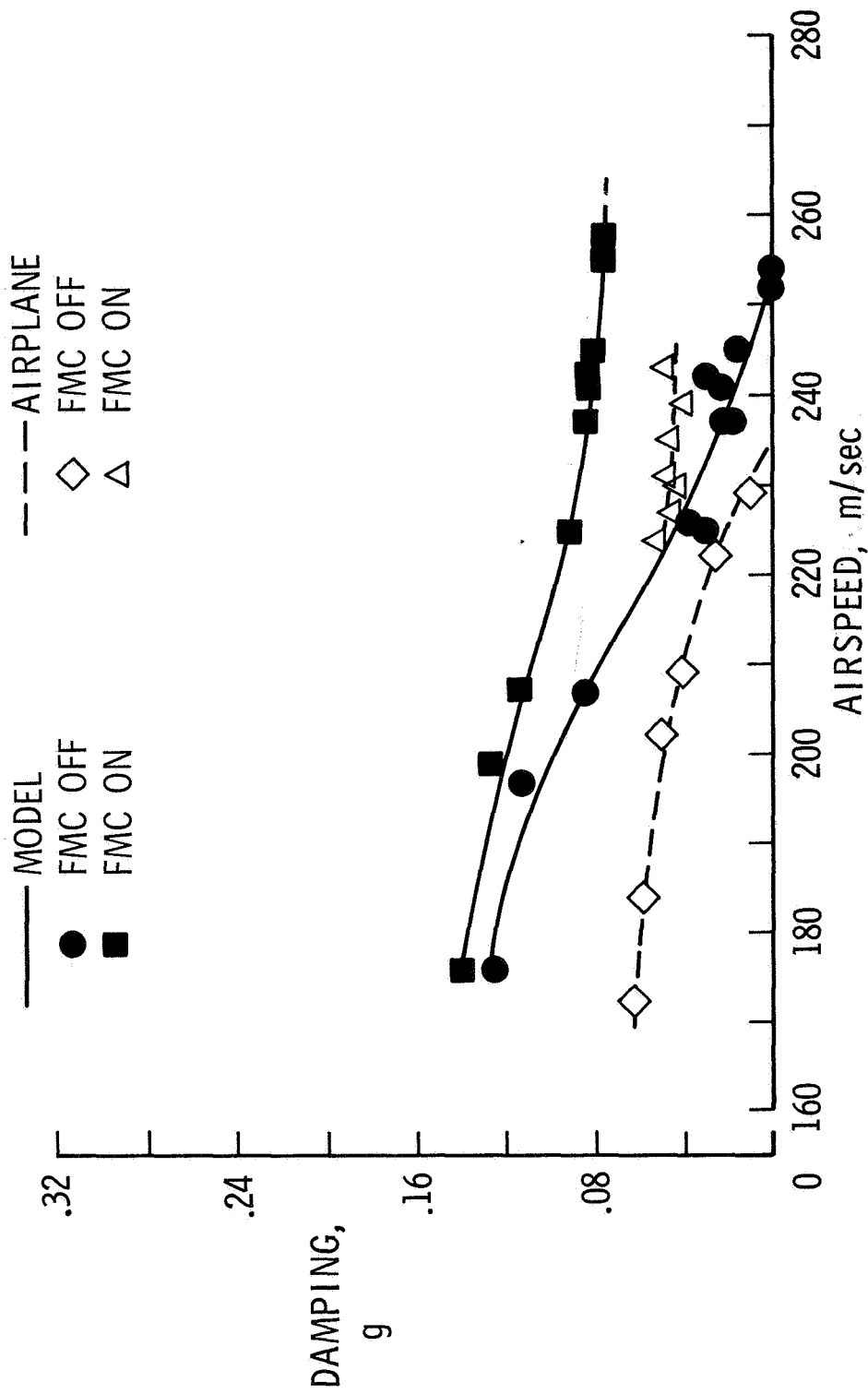
Figure 26.— Comparison of B-52 CCV model and airplane damping characteristics of the flutter mode.
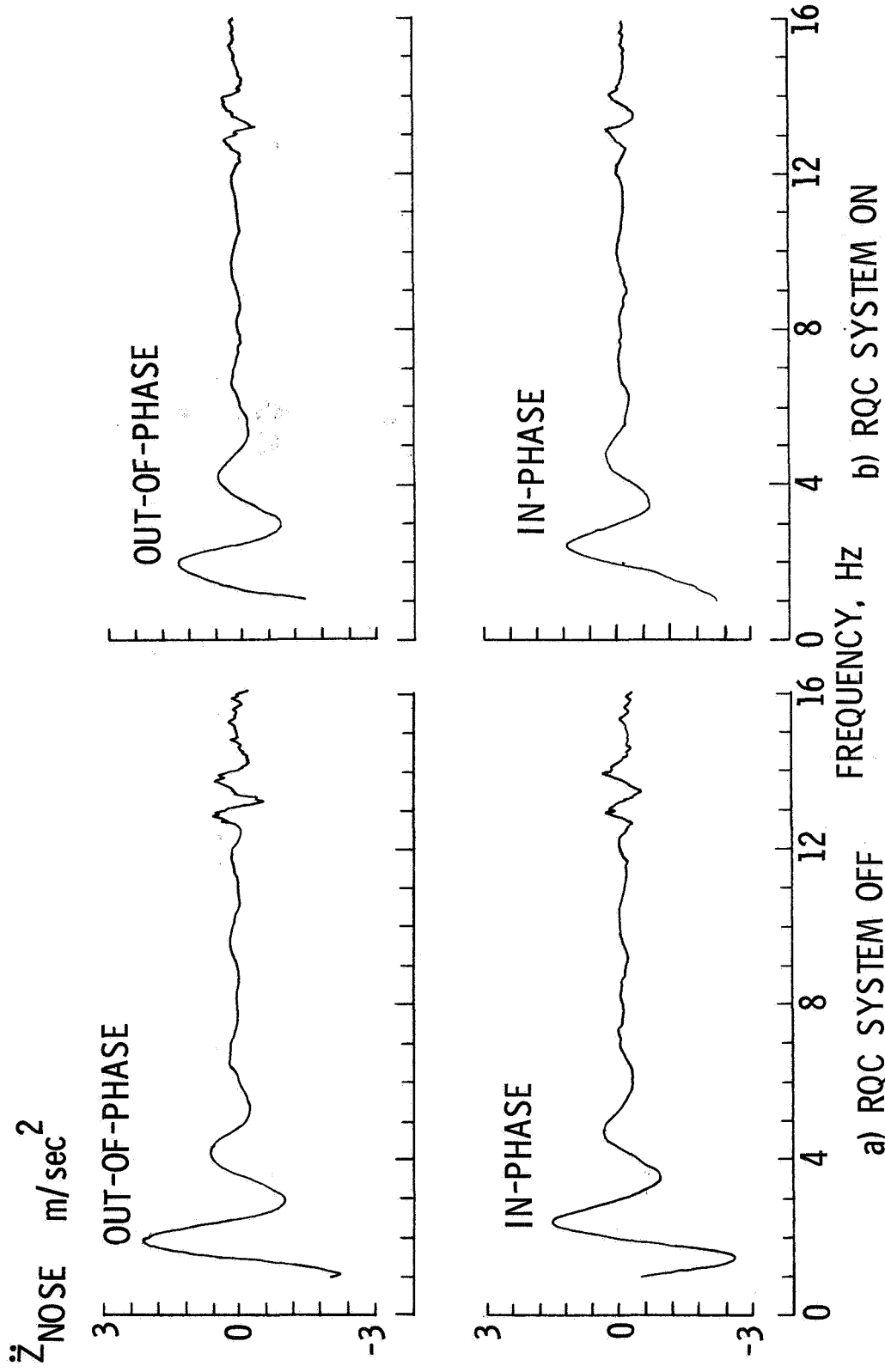
Figure 27.— Measured frequency response of B-52 model to oscillating gust vane excitation for a vane angle of 6° and a velocity of 33.7 m/sec.
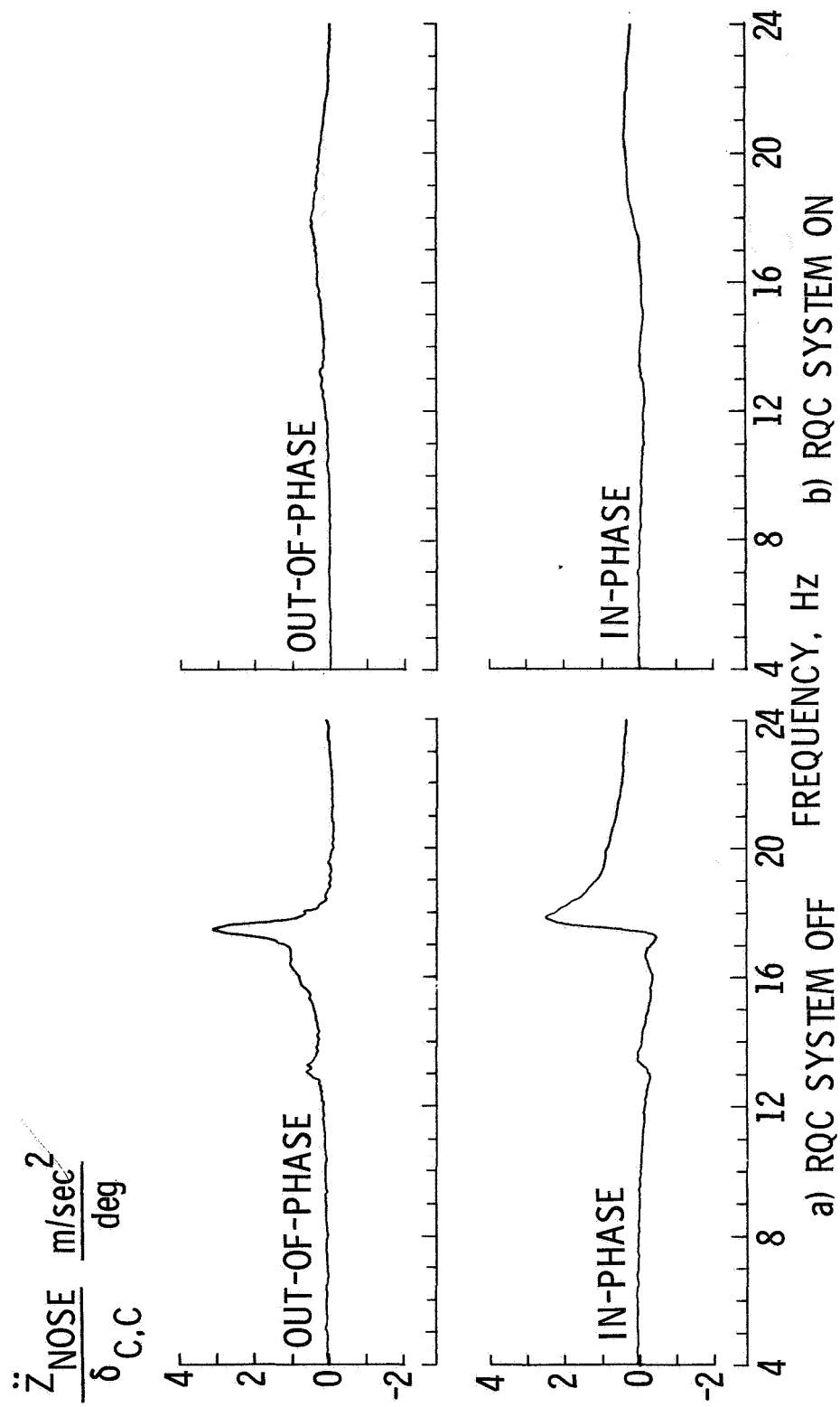
Figure 28.— Measured frequency response of B-52 model to canard excitation at a velocity of 33.7 m/sec.

882
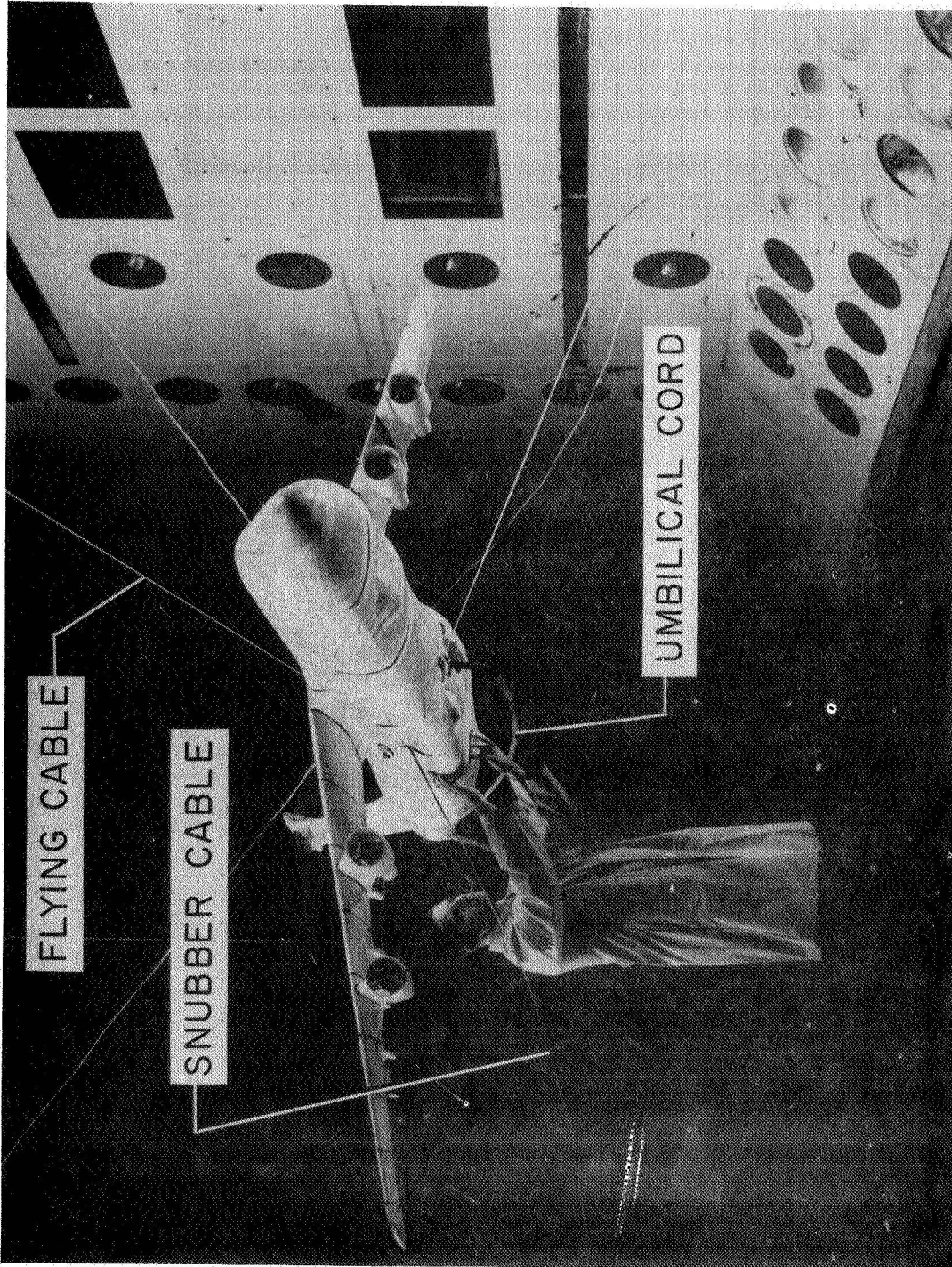
Figure 29.- C-5A model mounted in transonic dynamics tunnel.

Labels on figure: FLYING CABLE, SNUBBER CABLE, UMBILICAL CORD
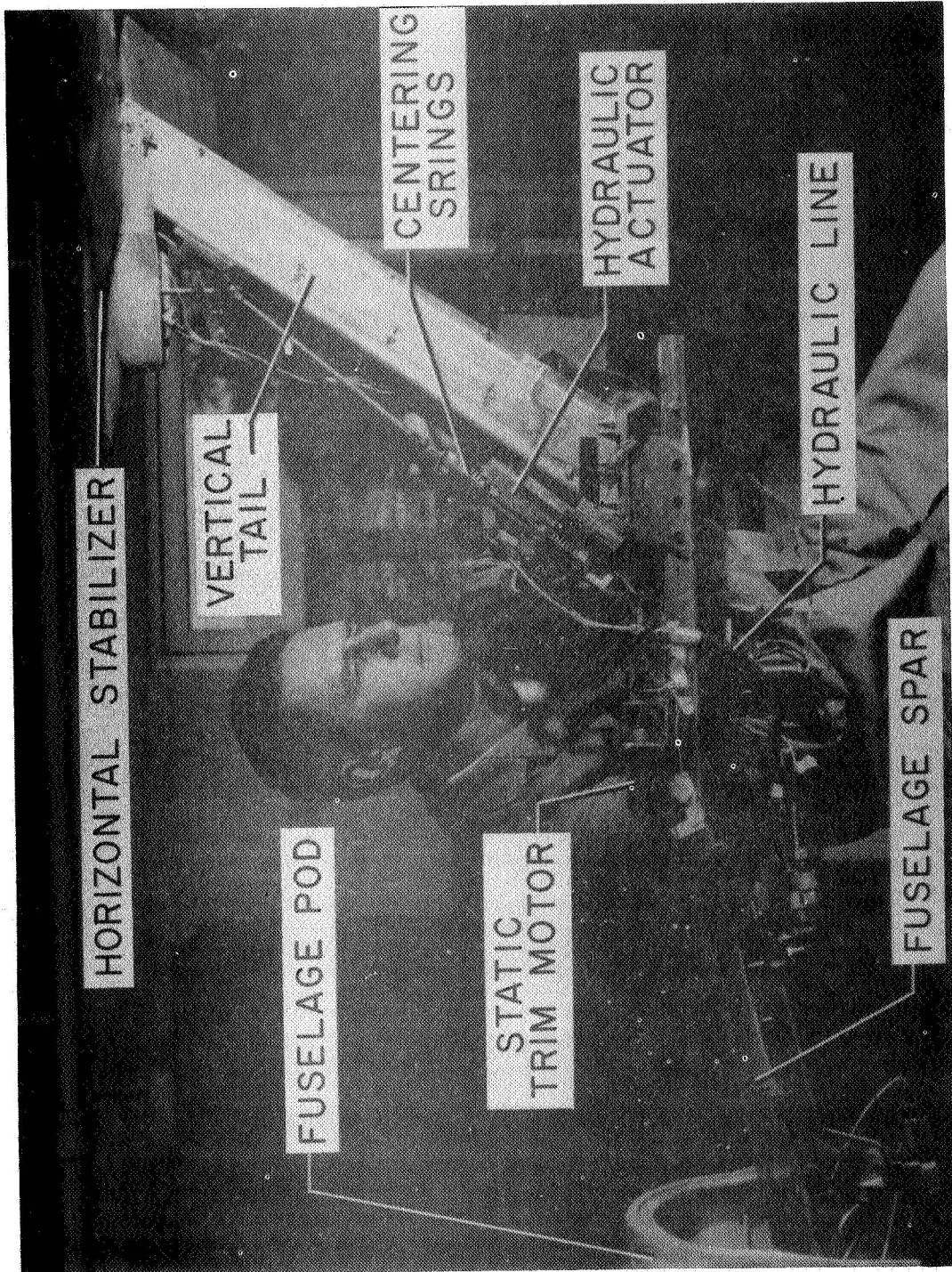
Figure 30.- Exposed view of rear portion of C-5A model fuselage.
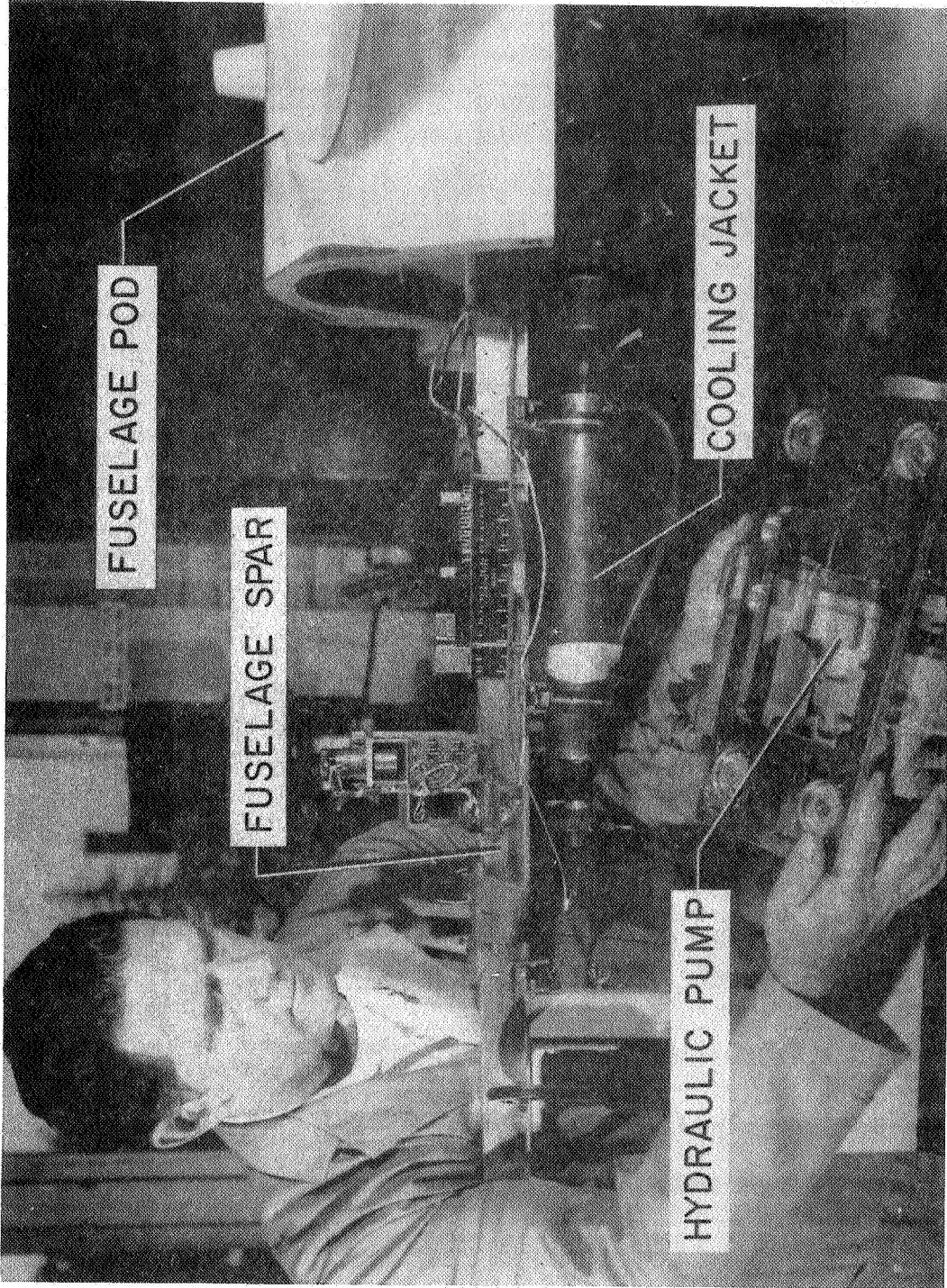
Figure 31.- Exposed view of forward portion of C-5A model fuselage.
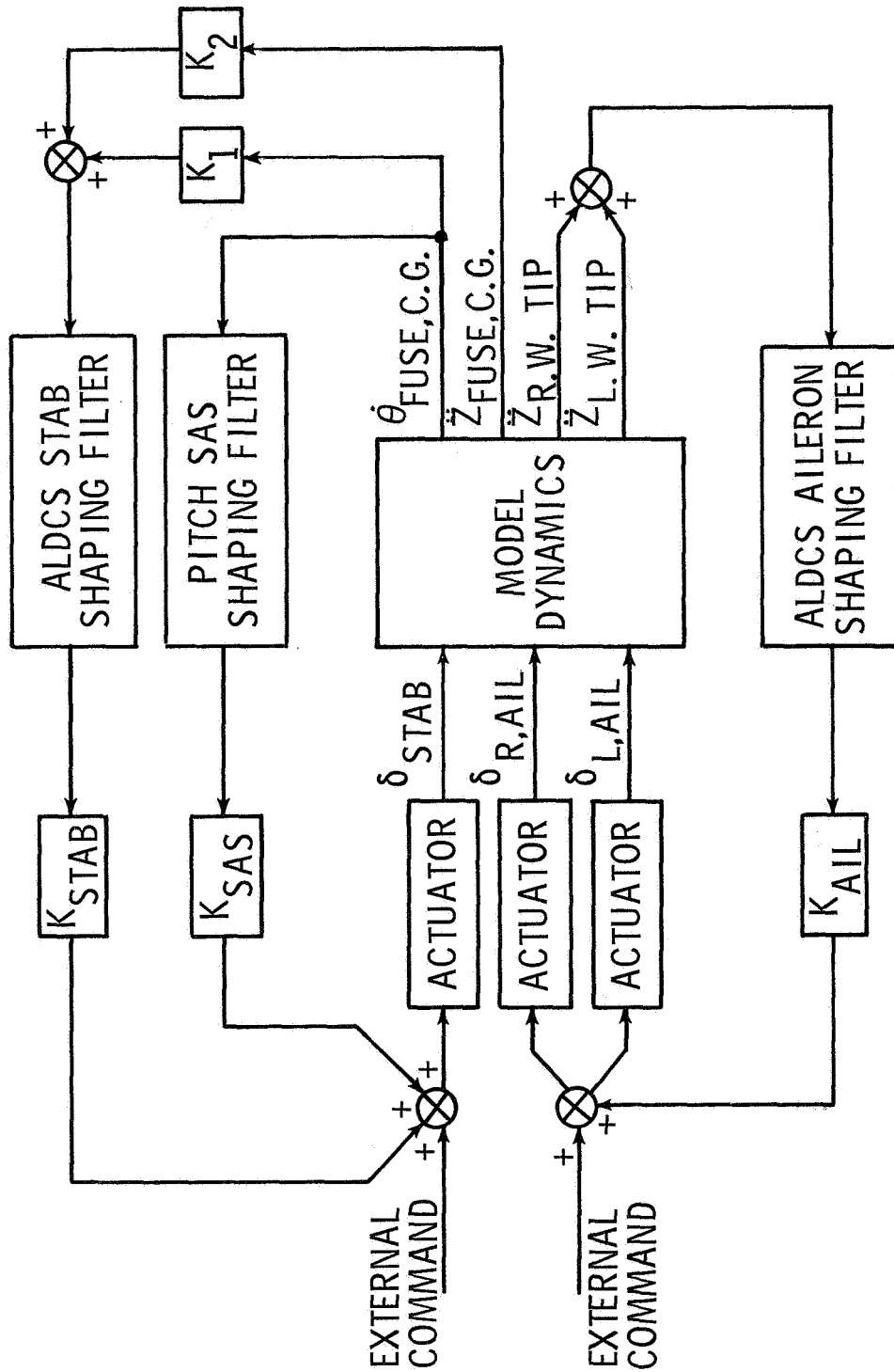
Figure 32.— Simplified block diagram of C-5A active lift distribution control system (ALDCS).

PURPOSE	FIXED TEST CONDITIONS	PRIMARY TEST VARIABLE	OTHER TEST VARIABLES
MODEL TEST CONFIGURATION: 33-PERCENT WING FUEL			
STATIC AER- ODYNAMICS	MODEL FREE-FLYING LIFT DEVICE USED ALDCS OFF	ANGLE OF ATTACK	AILERON UPRIG
DYNAMIC LOADS	MODEL SNUBBED AND RESTRAINED ALDCS OFF AILERON UPRIG + 5°	AILERON AND STABILIZER FREQUENCY SWEEP	AILERON SWEEP AMPLITUDE STABILIZER SWEEP AMPLITUDE
MODEL TEST CONFIGURATIONS: 0-PERCENT AND 33-PERCENT WING FUEL			
ALDCS EF- FECTIVENESS	MODEL FREE-FLYING AILERON UPRIG +5°	ALDCS OFF/ON	AILERON SWEEP AMP. AILERON STEP/RAMPS STAB. SWEEP AMP. STAB. STEP/RAMPS SYM. GUST SWEEP ANTISYM. GUST SWEEP INCREASE ALDCS AILERON GAIN UNSTABLE CONTROL LAW PITCH SAS OFF/ON
ALDCS EF- FECT ON FLUTTER	MODEL SNUBBED MODEL FREE-FLYING AILERON UPRIG + 5°	ALDCS OFF/ON	DYNAMIC PRESSURE MACH NUMBER AILERON STEP AILERON UPRIG +8°

Figure 33.- C-5A ALDCS model test summary.

887
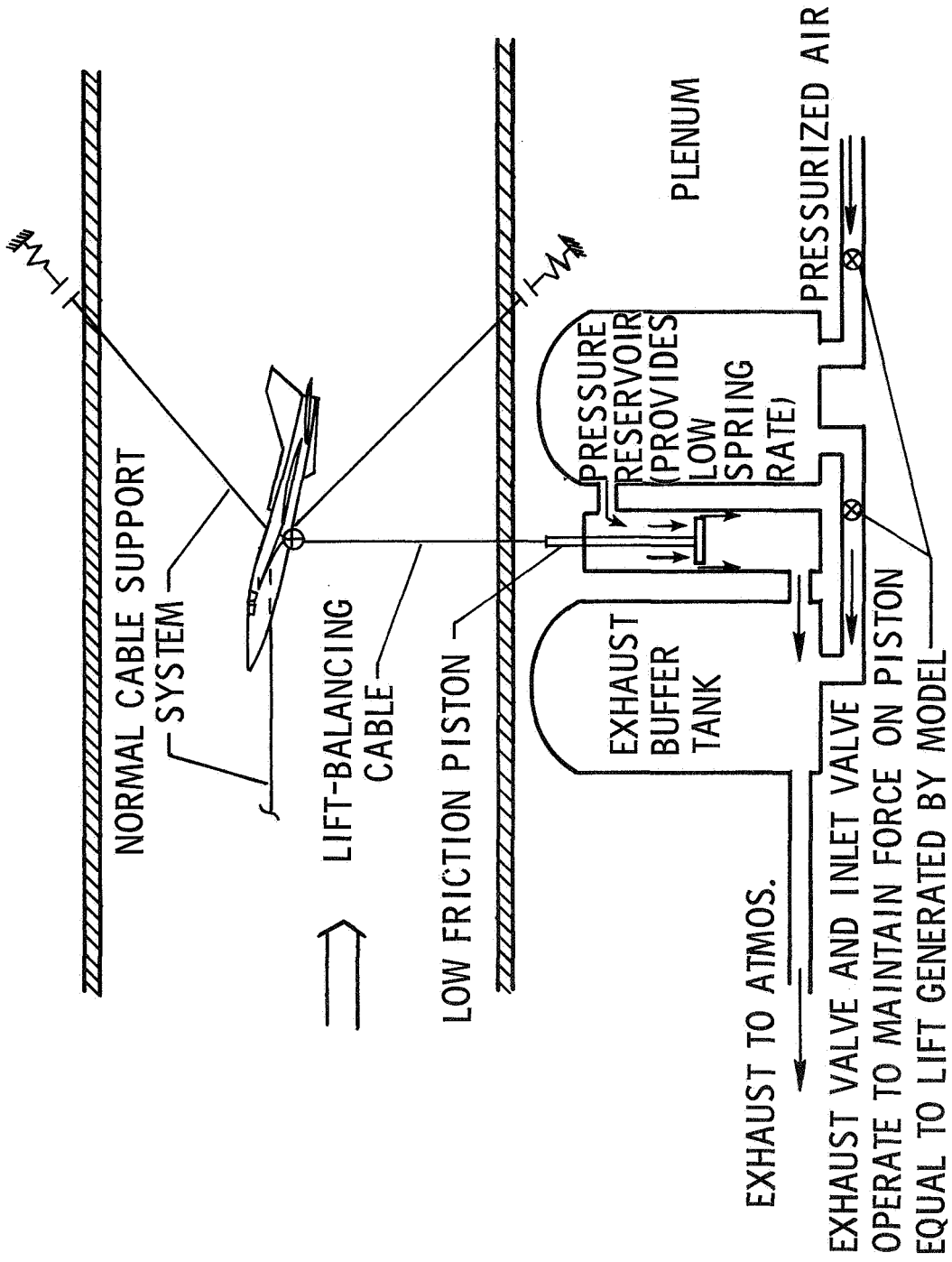
Figure 34.— Schematic diagram of wind-tunnel model lift control device.

NORMAL CABLE SUPPORT SYSTEM

LIFT-BALANCING CABLE

LOW FRICTION PISTON

PRESSURE RESERVOIR (PROVIDES LOW SPRING RATE)

PLENUM

PRESSURIZED AIR

EXHAUST BUFFER TANK

EXHAUST TO ATMOS.

EXHAUST VALVE AND INLET VALVE OPERATE TO MAINTAIN FORCE ON PISTON EQUAL TO LIFT GENERATED BY MODEL

888
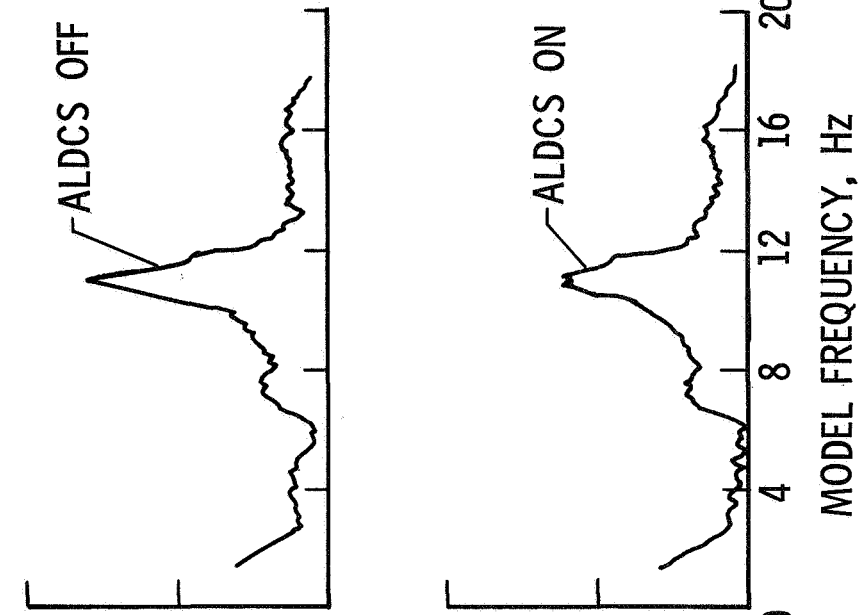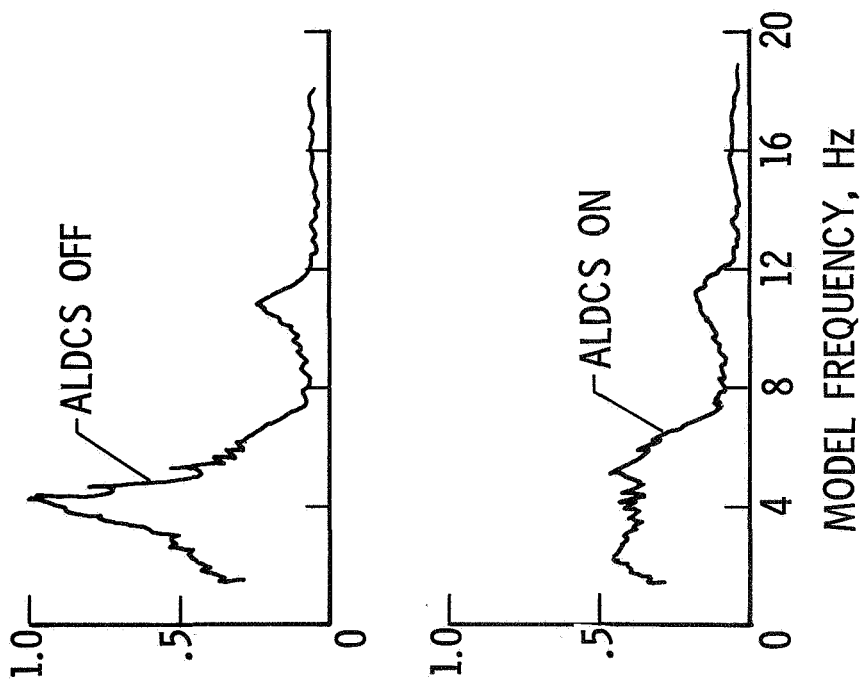
Figure 35.- Variation of C-5A model normalized wing bending and torsional moments with aileron frequency.

889
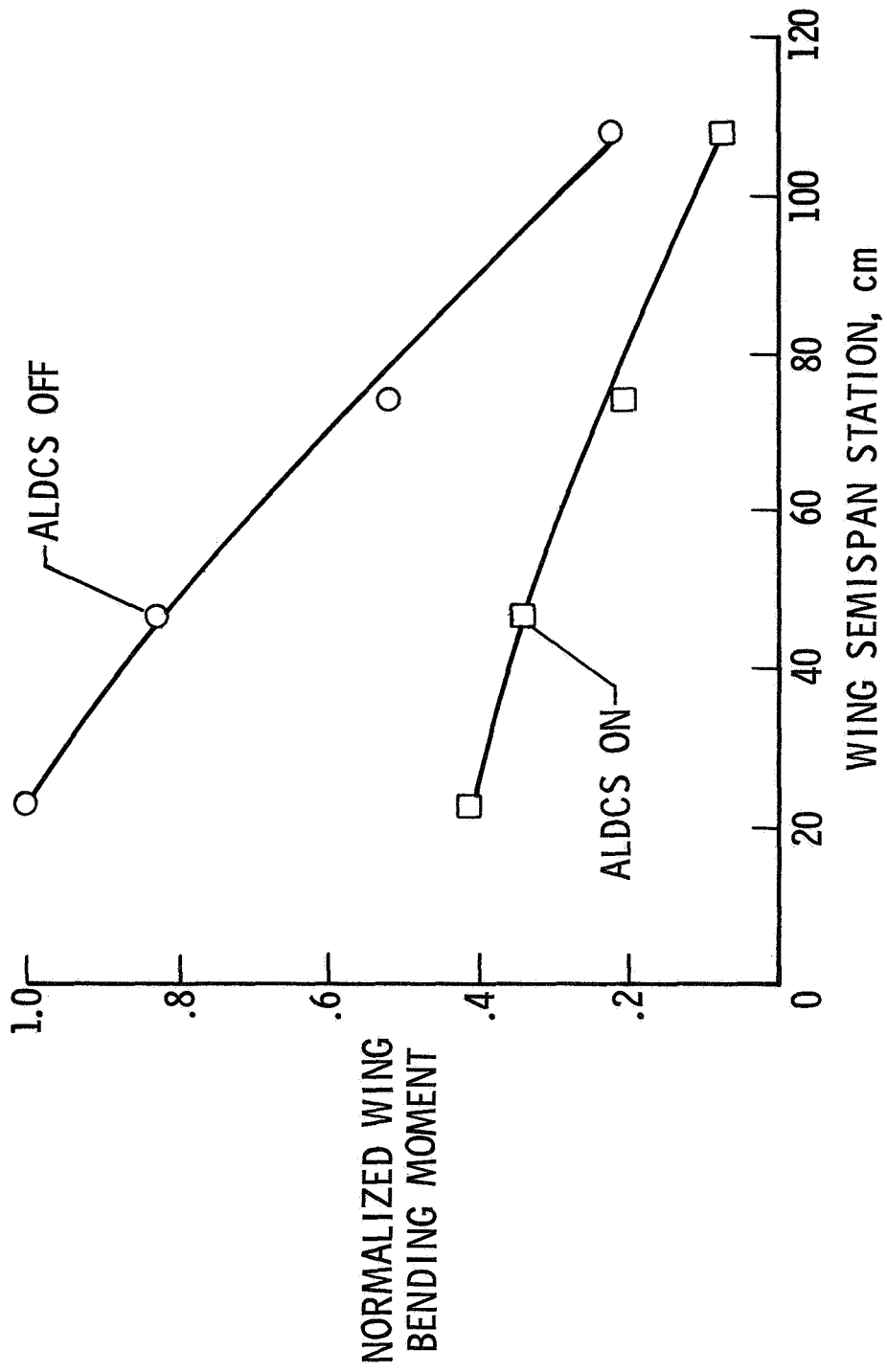
Figure 36.- Spanwise distribution of C-5A model normalized dynamic wing bending moment at a frequency of 4 Hz.
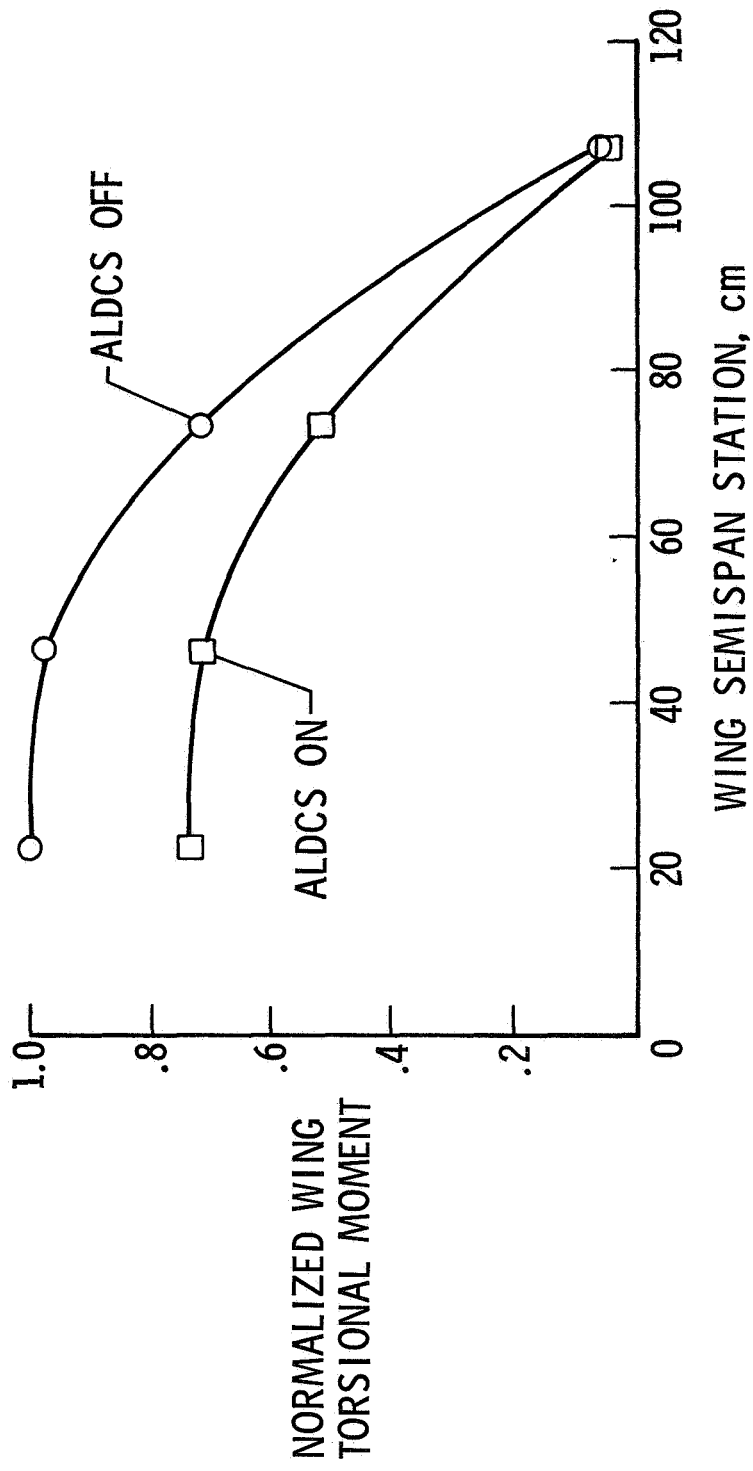
Figure 37.— Spanwise distribution of C-5A model normalized dynamic wing torsional moment at a frequency of 11 Hz.
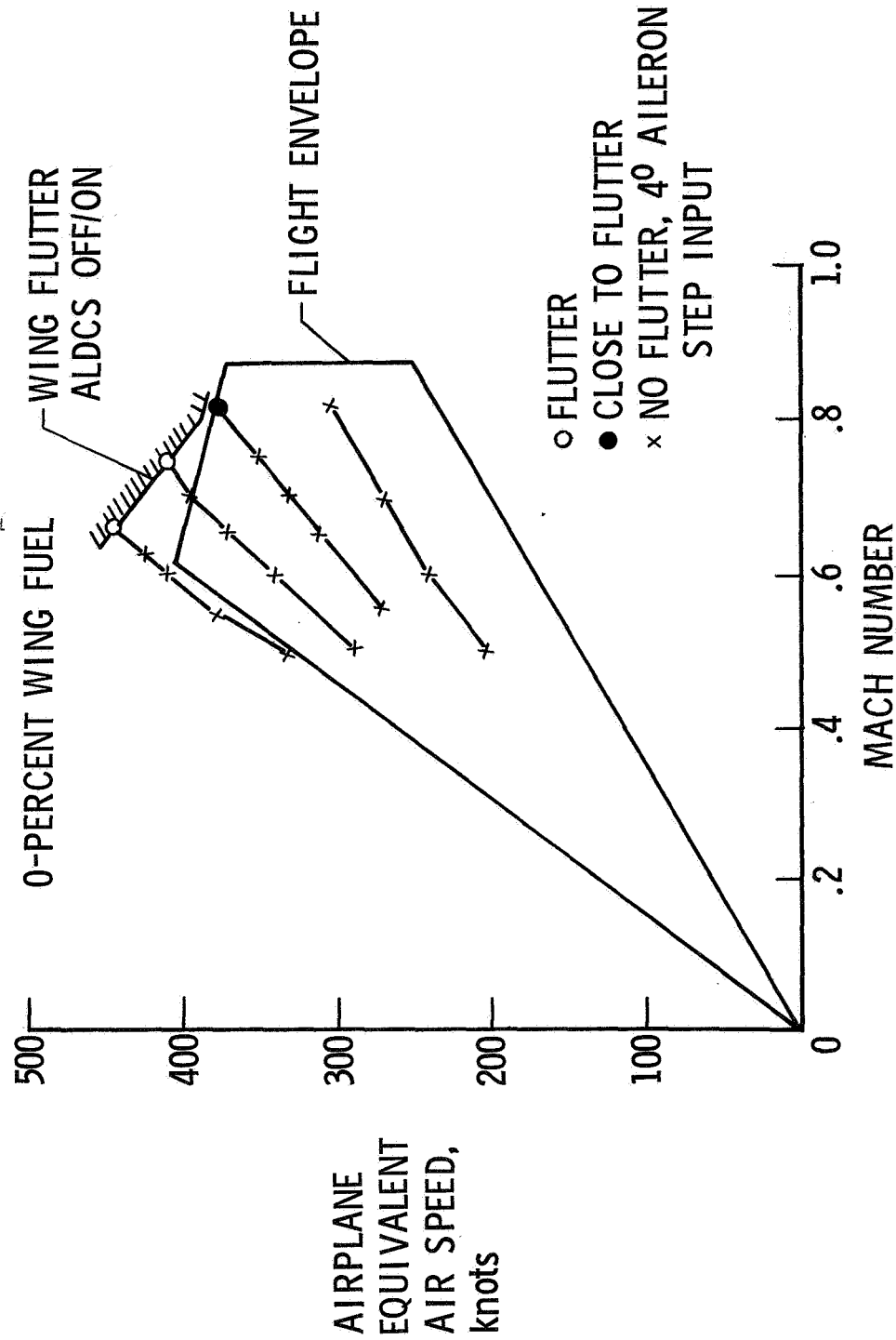
Figure 38.- Effect of ALDCS on C-5A model wing flutter.

A REMOTELY AUGMENTED APPROACH TO FLIGHT TESTING

OF ADVANCED CONTROL TECHNOLOGY

Herman A. Rediess, Eldon E. Kordes, and John W. Edwards
NASA Flight Research Center

ABSTRACT

A new technique for flight testing advanced control system concepts has been developed at the NASA Flight Research Center as an outgrowth of the remotely piloted research vehicle (RPRV) program. The control laws are implemented through up-down data links and a general-purpose ground based digital computer that provides the control law computations. The advantages of this remotely augmented approach over onboard systems are: reduced hardware development time and cost; ease of computer programing and verification; and increased flexibility for changing control laws. This paper describes the techniques, discusses selected flight-test results of a large-scale F-15 RPRV program, and presents plans for applying the technique to future RPRV and manned aircraft programs.

DEVELOPMENT OF METHODS FOR THE ANALYSIS AND

EVALUATION OF CCV AIRCRAFT

Robert C. Schwanz
Air Force Flight Dynamics Laboratory

ABSTRACT

The development of an advanced, computerized method for the analysis and evaluation of the aeroelastic stability and control parameters of controls-fixed and controls-free flight vehicles is presented. Specifically, the contractually developed Level 2.01 FLEXSTAB Computer Program System is described. Technical areas in aerodynamics, dynamics, and control system synthesis are defined in which further research and development are planned to extend the analysis capability of the System for future CCV applications.

INTRODUCTION

The development of computer programs to analyze and evaluate the aero-elastic characteristics of controls-fixed aircraft has intensified in past years. This increase in interest has been due to the design and construction of large transport aircraft, such as the Boeing 747, McDonnell Douglas DC-10, and Lockheed L-1011, and of the high performance aircraft, such as the SST and the B-1. Recent articles by Stauffer, Lewalt, and Hoblit (Reference 1) and Rowan and Burns (Reference 2) describe methods and capabilities currently available within the industry. This paper describes the Level 2.01 FLEXSTAB Computer Program System (Reference 3) that has been developed by the Control Criteria Branch for the aeroelastic stability and control analyses of both controls-fixed and controls-free military aircraft. This development included careful consideration of the applicability of the method to military mission objectives, the variety of potential military users of the method, the man-power and computer costs involved in using the method, and the criteria that govern the application of the method. This development has taken particular care to solve some of the problems that are unique to the analysis and evaluation of Control Configured Vehicle (CCV) aircraft operating at reduced frequencies that are small.

As noted, there are mission and user requirements placed upon any CCV-type aircraft analysis and evaluation method developed for the USAF. The mission objectives of these aircraft impose severe and complex requirements, requiring that the method analyze:

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The subsonic, transonic, supersonic, and hypersonic speed regimes.

The complex, three-dimensional, aerodynamic interference flow fields of transport, bomber, and fighter aircraft.

The structural dynamics of both low and the high structural aspect ratio lifting surfaces of aircraft.

The static and dynamic stability of aircraft with both unusual inertial distributions and large translational and rotational rates of motion.

An examination of the major research and development organizations within the Air Force Flight Dynamics Laboratory (AFFDL) and the Aeronautical Systems Division at Wright-Patterson AFB indicates that there are three types of users who impose additional requirements:

Those concerned with conceptual design and development only. These users desire a fast, inexpensive, and proven method that performs reliable design and evaluation of new and innovative aircraft.

Those concerned with the responsibilities of following and monitoring the development of a new aircraft by the contractors, from the conceptual design to the flight test phases. Again versatility and cost/effectiveness are imperative due to the limited manpower that is available.

Those concerned with developing new and advanced technologies for future applications. These users desire a well documented and versatile method that can be easily modified to prove their ideas before major computer program development is initiated.

As noted, each of these users has a unique problem that the method must address. Fortunately, these user requirements for high speed, low cost, and versatility are compatible and can be met using the large digital computer and aerodynamic and structural finite element theory.

The development of an analysis method to meet these requirements was begun in 1971 by the Control Criteria Branch of AFFDL. A 1973-1974 target date was set for the completion of the analysis method to ensure support for the development of the Ride Control System of the B-1 aircraft and the Reduced Static Stability systems proposed for several Lightweight Fighter aircraft configurations. At that time, descriptions of the existing analysis methods within the industry were meager and only a limited amount of financial resources were available for the planned contractual work.

An obvious technique for acquiring a CCV analysis and evaluation method would have been to purchase the most accurate and versatile, contractor-developed, design method available. Discussions with the aerospace contractors indicated that their existing design methods in flight controls, aerodynamics, structural analysis, and dynamics could suffice in the short term. However,

due to the loose and often undocumented federation of computer programs used by each contractor, the AFFDL Control Criteria Branch would have had to purchase the "expert" who developed each program to implement this technique. Thus, it was impossible to make a direct purchase to satisfy the identified needs.

The search was then diverted to the newly developing controls-fixed aircraft analysis and evaluation methods that contained the necessary mathematical sophistication. The prime candidate in the 1971-1972 time period was the Level 1.01 FLEXSTAB Computer Program System being developed by the Boeing Company, Seattle, Washington, under contract to the NASA Ames Research Center (Reference 4). This program could not meet the USAF requirements in the transonic and hypersonic speed regimes, had a limited unsteady aerodynamics computational capability, had no turbulence analysis capability, and had a restricted mathematical representation of high aspect ratio structures via elastic axes and lumped masses. However, the programs were well documented, were somewhat modularized, and with some modifications could meet the majority of the requirements of the USAF. Of considerable benefit was the fact that NASA had already spent approximately $500,000 developing the aerodynamic and structural modules of FLEXSTAB. Thus, with limited expenditures by the Control Criteria Branch, the modules for flight control systems analysis could be added to meet both the time schedule and budget objectives.

The development plan to construct the control system analysis modules was formulated and coordinated with NASA Ames Research Center (ARC). It was decided that NASA/ARC would concentrate their immediate resources in further checking the accuracy of the aerodynamics and structures modules of Level 1.01 FLEXSTAB. Meanwhile, the Control Criteria Branch would implement Level 1.01 at Wright-Patterson Air Force Base (WPAFB) to measure the accuracy and efficiency of the programs using current military bomber, transport, and fighter aircraft as test cases. In addition, the AFFDL would begin a contractual effort to develop the control system analysis modules. The combination of these aerodynamic, structural, and control system analysis modules would then form the basis for Level 2.01 FLEXSTAB to be used by NASA and USAF in the stability and control analysis of conventional and CCV-type aircraft.

The Level 2.01 FLEXSTAB Computer Program System is in the final check-out phase. The contracted work is scheduled for completion in November, 1974. The Control Criteria Branch has been evaluating a pre-release version of Level 2.01 since February, 1974 with the primary applications being made to the C-5A, B-52E, B-1, and F-111 TACT aircraft. An oral technical presentation is scheduled for October, 1974 at the AFFDL. Several papers will be presented by the contractor in coming technical meetings to more fully describe Level 2.01 and to illustrate its application to the B-52E and other aircraft.

DESCRIPTION OF LEVEL 2.01 FLEXSTAB

The development of the control system analysis modules and the inter-facing of them to the Level 1.01 aerodynamic and strutures modules has been done under an AFFDL research and development contract with the Boeing Company, Seattle, Washington. As a first step in the contract, the principal investigators established firm guidelines to meet the Statement of Work from the Control Criteria Branch. Their initial report (Reference 5) on the contract summarized these guidelines:

No predetermination of "important motion parameters" per previous short cuts in analysis. The equations of motion should not be "tailored" for conventional aircraft, e.g., the method would not neglect the forward speed degree of freedom of the "body-fixed" axis system as currently practiced by most structural dynamacists.

The initial conditions of motion to be as general as possible. Only the initial conditions of linear and angular accelerations were eliminated because of theoretical aerodynamic problems.

Reasonable restriction of the number of feedback and pilot inputs and of the order of the control system filters.

Hinge moment aerodynamic effects represented as an option.

Compatibility with the Level 1.01 FLEXSTAB program to take advantage of NASA sponsored improvements. In fact, the NASA coding requirements for the Level 1.01 FLEXSTAB were specified by the Control Criteria Branch.

No higher user skill level than the capability to create simple Fortran statements as taught to Freshmen engineering students.

Independence from the specialized math models of actuators, of control surface/actuator coupling, and of turbulence Power Spectral Density shapes.

Capability for the analysis of both the open and closed loop system responses due to deterministic gusts and due to random turbulence.

Input data formats that minimize the preparatory work required by the user.

Minimization of the complexity of modules within each program to permit the user to understand the basic calculations by the program.

A program structure that provides for a maximum number of accurately determined complex number roots.

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Computer overlay structure that minimize the computer cycle time in production runs.

Above all else, it was established that program risks must be identified and reported before the initiation of any detailed engineering and coding. These high risk areas were to be avoided until a program existed that could numerically demonstrate the analysis and the evaluation problems in these areas and, at that point, USAF or NASA funding on specific high risk/high payoff areas could be initiated. The test case specified to check Level 2.01 FLEXSTAB is the B-52 LAMS (Reference 6) at Flight Condition 1: $M = 0.569$, Altitude = 4,000 ft, Weight = 350,000 lbs., and Center of Mass = $0.298\bar{c}$.

Physical Structure of Level 2.01 FLEXSTAB

As mentioned, the Level 1.01 FLEXSTAB programs were intended for the stability and control analysis of controls-fixed aircraft. In order to improve the efficiency and suitability of the Level 2.01 Systems for both controls-fixed and controls-free aircraft analysis and in order to meet the stated guidelines, the 16 computer programs of Level 1.01 (Figure 1) were overlayed and restructured. This meant modification of all four sections of FLEXSTAB: Airplane Definition, Airplane Stability Evaluation, Graphical Display, and Auxiliary Program sections. The net result in Level 2.01 FLEXSTAB is 13 computer programs (Figure 2) that are interconnected by cards and magnetic tapes.

A more detailed description of the physical structure of the Level 2.01 FLEXSTAB System can be accomplished by contrasting Level 2.01 programs to those of Level 1.01. Then, current government and industrial users of Level 1.01 can more easily visualize Level 2.01 FLEXSTAB. To further facilitate this contrast, Level 2.01 programs have been segmented into the same four analysis sections as before and individual program acronyms of Level 1.01 are maintained wherever possible. Since these program acronyms have been in active use for more than three years, it is hoped most readers will have some familiarity with the terminology of Level 2.01 FLEXSTAB.

Specifically then in the Airplane Definition section of Level 2.01 FLEXSTAB, the Geometry Definition (GD) and the associated CALCOMP program (GDPLOT) of Level 1.01 have been combined in Level 2.01 to facilitate the conversion of the scaled configuration drawings of the aircraft into the spatial mathematical descriptions required by the subsequent downstream programs of Level 2.01. The Aerodynamic Influence Coefficient (AIC) program of the Level 2.01 System is structured to include both the steady and low-frequency unsteady aerodynamic programs of Level 1.01. The Internal Structural Influence Coefficient (ISIC) and Normal Modes (NM) programs and the External Structural Influence Coefficient (ESIC) program of Level 1.01 have been modified to provide structural data to mathematically represent 15 types of flight control sensors. In the airplane Stability Evaluation Section of Level 2.01, the Stability Derivative and Static Stability (SD+SS) and the Characteristic Equation Rooting (CER) programs of Level 1.01 have

been combined to facilitate the controls-fixed aircraft analysis. In the position previously occupied by the CER program of Level 1.01, a new Linear Systems Analysis (LSA) program for controls-free aircraft analysis is included in Level 2.01. The Level 1.01 FLEXSTAB Time Histories (TH) program is modified so that Level 2.01 may analyze the response of controls-fixed and controls-free flexible aircraft that are perturbed by deterministic gusts or control surface disturbances. The remaining Graphical Display programs of Level 2.01, i.e., Elastic Axis Plot (EAPLOT), Normal Modes Plot (NMPLOT), and Pressure Distribution Plot (PDPLOT), and the Auxiliary programs of Level 2.01, i.e., Corrected Aerodynamic Influence Coefficient (CAIC), and Structural Loads (SLOADS) are only slightly altered from their Level 1.01 FLEXSTAB form.

The net cost/effectiveness of these changes from the Level 1.01 to Level 2.01 FLEXSTAB, as well as the contracted and the in-house modifications to the CDC computers at WPAFB, is shown in Figure 3. Initially, the Level 1.01 FLEXSTAB, as implemented at WPAFB in February 1973, required 17 workdays, 72 manhours, and 18,000 computer resource seconds to perform a single design point analysis of a high aspect ratio aircraft using a moderate sized math model that consisted of 200 aerodynamic influence coefficients (AIC's). A computer resource second is defined to be the total of the Central Processor seconds added to one-half the Input/Output seconds (CP + 1/2 IO).

Presently, the Level 2.01 FLEXSTAB analyses require substantially less workdays, manhours, and computer seconds at WPAFB. As an example, the Control Criteria Branch performed a conceptual design analysis (Reference 7) of a Spanloader aircraft, Figure 4, inspired by the Lockheed Spanloader presented in Reference 8. This in-house analysis required approximately 3 workdays, 5.7 manhours, and 10,000 computer resource seconds. In fact, a substantially greater number of Level 2.01 FLEXSTAB analyses were performed, at approximately 1/2 the computer costs of the Level 1.01 FLEXSTAB analyses. An examination of the differences in the workdays indicates that 10 days were removed due to the purchase of a second CDC 6600 computer for WPAFB. The remaining 8 days were reduced to 3 via the streamlining of the program and the specialization of the System to the computer software of the CDC computer system at WPAFB. The reduction in the expended manhours can be attributed to several things. Approximately 20 hours of the reduction can be attributed to experience gained in using the System over the past two years. The major manpower savings is accomplished in the operation of the ISIC program due to the creation of an interface program that simplifies the input of elastic axis and lumped mass locations, thus eliminating user-generated errors. Manpower savings were also accomplished in the GD and SD+SS programs by rewriting the input data formats to highlight redundant data inputs and to redefine the input data.

Analysis and Evaluation Capability of Level 2.01 FLEXSTAB

The basic analysis and evaluation capability of the Level 2.01 System is substantial, being best illustrated by detailed descriptions of what each major program in the System will do. In general, the Level 2.01 System

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estimates the static and dynamic stability and control parameters of controls-fixed and controls-free flexible aircraft over a Mach number range of 0.0 to approximately 3.5. The System is applicable to complex, three-dimensional aircraft configurations, e.g., vertical tails located on the outboard portions of the wing, T-tails, nacelles suspended from struts, and close-coupled canards and wings. The steady aerodynamic theory of the System is an advanced version of the method developed by Woodward, Tinoco, and Larsen (Reference 9), in which constant pressure vortex panels, constant strength source panels, line doublets, and line sources represent the linear potential flow aero-dynamics about the flight vehicle. In addition, the aerodynamic program of the System contains a recently-developed (Reference 10), low-frequency unsteady aerodynamic approximation that extends the basic steady aerodynamic method to the calculation of unsteady aerodynamic derivatives, e.g., $\dot{\alpha}$, $\dot{q}$, δ_S, $\dot{\beta}$, $\dot{r}$ and $\dot{p}$ derivatives, as well as the "generalized modal aerodynamics." The low-frequency approximation is unique in that it has the same general, three-dimensional capability of the steady aerodynamics method at both subsonic and super-sonic speeds. This feature eliminates the redefinition of the influence coefficient geometry with Mach number and also the "diaphragm region" of the existing Mach Box supersonic method.

There are two structural influence coefficient methods within the System. One method is based upon the elastic axis/lumped mass approximation usually employed on high aspect ratio aircraft. In this method (ISIC and NM programs of Figure 2), the structure of the aircraft is replaced by a connected con-figuration of beams with specified bending and torsional stiffness properties. This math model is used to calculate the stiffness and the flexibility matrices, the aircraft inertial characteristics, and the invacuum vibration eigenvalues and eigenvectors of the free-free structure. The structural and inertial matrix coefficients are transformed into a mathematical format that is compatible with the steady and unsteady aerodynamic influence coefficients. Once the aerodynamic and structural influence coefficients exist in a compat-ible numerical format, the subsequent programs in the System can calculate the stability and control parameters of the rigid and flexible aircraft.

The second structural influence coefficient method in the System (ESIC program of Figure 2) contains numerical routines that accept influence coef-ficients from an "external" finite elements programs, such as NASTRAN, and then converts them to a form compatible with the aerodynamic influence coef-ficients. Thus, ESIC provides the System user with an accurate structural influence coefficient representation of the aircraft that is most useful in the advanced design cycle of aircraft development.

As noted before, the aerodynamic and structural influence coefficients are summed for the static and dynamic aeroelastic stability and control calculations by the other programs in the System. These System programs (SD+SS, LSA, and TH programs of Figure 2) calculate the stability and control derivatives, the static and dynamic stability, the aerodynamic loads on the maneuvering aircraft, and the deformed shape of the flexible aircraft. The SD+SS program of the System (Reference 11) allows for initial conditions of non-accelerating dynamic motions consisting of constant-magnitude, angular pitch, roll, and yaw rates and linear translational rates, singularly or in

combination. The engine gyroscopic effects are included at the user's option. Once the initial conditions of motion are defined in SD+SS and the trim or specified shape of the aircraft defined, the analyses of symmetric, asymmetric, or coupled perturbation dynamic motions is possible using (1) a characteristic equation rooting method, if the perturbation equations are linear ordinary differential equations, or (2) a Runge-Kutta integration method of the TH program, if the differential equations are nonlinear or the system is excited by deterministic gust and control disturbances.

The Stability Derivative and Static Stability (SD+SS) program also contains numerous options that connect the Level 2.01 FLEXSTAB to the existing experimental and semi-empirical methods of analysis. As an example, the stability and control data measured during wind tunnel tests of rigid force and hinge moment models of aircraft may be incorporated as tables of data, e.g., $C_L(\alpha, \delta_S)$ and $C_n(\alpha, \beta)$, or as derivatives at the trim point, e.g., C_{L_α} and C_{n_β}. If the wind tunnel measurements are unavailable, a semi-empirical method such as Datcom (Reference 12) may be used instead.

The Linear Systems Analysis (LSA) program of Level 2.01 FLEXSTAB is of particular interest to analysts of CCV-type aircraft. As implied previously, the computer mechanization of the routine engineering computations involved in CCV analyses improves the analysis cycle time and analysis accuracy, because the computer does not make mistakes due to fatigue or boredom. The specific CCV-type calculations that have been mechanized in LSA (Reference 13) are:

The construction of an LSA precompiler, that accepts user specified control system transfer functions as a ratio of high order polynominals in the Laplacian variable, and then internally rearranges these elements into a standard matrix format for Root Locus, Bode, Nyquist, and Power Spectral Density analyses.

The construction of accelerometer, rate gyro, angular position, air speed, and inertial velocity sensor equations as an integral part of the overall LSA calculations.

The construction of a Pade´ polynomial approximation of the gust and the turbulence penetration exponential, per user specified tolerances.

A built-in Von Karman Power Spectral Density model to represent the turbulence excitation of the flexible, controls-fixed and controls-free aircraft. The user may input his own turbulence model if he so desires.

Options that permit a user to "delete" or "reduce" selected invacuum modes via the MODAL TRUNCATION or the RESIDUAL FLEXIBILITY formulations that are discussed in the next section of this paper.

An option to punch on cards, in a standard format, the matrix equations of motion of the aircraft and the sensors for the user

to input to flight simulators and other interfacing stability and control computer programs.

The input and output data of the programs are graphically presented by five CALCOMP programs that are interconnected to the main analysis programs of the System by overlay, cards, or magnetic tape (TAPE 99 in Figure 2). These CALCOMP programs present the geometric orientation of the aerodynamic and structural elements of the math model of the aircraft, the invacuum eigenvectors, the aerodynamic load distributions, and the time history responses.

Two programs in the Level 2.01 FLEXSTAB are intended primarily for design. One, Corrected Aerodynamic Influence Coefficient (CAIC program of Figure 2) is used to correct the aerodynamic influence coefficients for non-linear effects via correction factor matrices. The second, Structural Loads (SLOADS program in Figure 2), calculates the aerodynamic and inertial component loads along the elastic axes, if the beam structural math model is employed.

ESTABLISHMENT OF CRITERIA FOR THE APPLICATION OF LEVEL 2.01 FLEXSTAB

The Level 2.01 FLEXSTAB contains numerous user options that permit varied types of aeroelastic stability and control analyses. A unifying concept in the System is, that no matter which of the options are selected, one of the major results is the creation of the equations of motion using the aerodynamic, structural, and inertial matrices. These equations of motion, and the attendant sensor and loads equations, consist of three inter-related formulations, QUASI STATIC, RESIDUAL FLEXIBILITY, and MODAL TRUNCATION, that describe the dynamics of the controls-fixed and controls-free aircraft.

The industry surveys mentioned at the beginning of this paper indicated that all of the formulations were used to a degree, but that the QUASI STATIC and MODAL TRUNCATION formulations were the most common. As examples:

The XB-70 GASDSAS, B-52E LAMS, B-52E CCV, C-5A ALDCS, F-4 Survivable Flight Controls, F-4 CCV, F-111, and F-15 aircraft projects applied the QUASI STATIC and MODAL TRUNCATION formulations.

The initial AFFDL sponsored studies of a CCV-type bomber, transport, and fighter aircraft applied the QUASI STATIC and a combination of the MODAL TRUNCATION and RESIDUAL STIFFNESS formulations.

The SST design studies applied the QUASI STATIC, MODAL TRUNCATION, and RESIDUAL STIFFNESS formulations. The B-1 design studies applied the QUASI STATIC, MODAL TRUNCATION and RESIDUAL STIFFNESS formulations.

The industry and government have developed criteria for the selection of the QUASI STATIC formulation. However, there are few criteria to guide the

selection of the other formulations of the equations of motion. The criteria are necessary because they:

Force the flight control analysis to be consistent with the flutter and structural loads analyses of CCV-type aircraft.

Provide a qualification for the associated handling quality and ride quality criteria studies.

Provide a rational to the USAF and the other government agencies to be used in the evaluation of competing CCV-type aircraft.

Identify the configuration development problems created by the application of each formulation.

Place upper limits on the complexity to be tolerated in CCV-type control systems that are designed using each of the approximate formulations.

Determine the risks associated with the relaxation of criteria.

The AFFDL Control Criteria Branch initiated a study in 1971-1972 to supply these criteria as part of the Development plan for Level 2.01 FLEXSTAB. As a first step, the six linear formulations of the equations of motion were identified and mathematically related using the notation of the FLEXSTAB documentation:

EXACT - The motion of the structure is determined by eigenvalue (root) and eigenvector (mode shape) solutions of the equations of motion for the elastic aircraft. The mode shape coordinates contain complex numbers. The accuracy of the solution is limited by the existing computerized routines that calculate the complex number eigenvalues and eigenvectors.

MODAL SUBSTITUTION - The motions of the structure are assumed to be related to the orthogonal, invacuum eigenvectors (mode shapes). The eigenvectors contain only real numbers.

RESIDUAL STIFFNESS - The mode shapes representing the elastic motion in the MODAL SUBSTITUTION formulation are separated into "retained" and "deleted" modes. The deleted modes are represented in the dynamic stability analysis as quasi static aeroelastic corrections, using a correction factor related to the deleted modes and the stiffness matrix of the free-free structure.

RESIDUAL FLEXIBILITY - Similar to the RESIDUAL STIFFNESS formulation, except the quasi static aeroelastic correction is related to the retained modes and the flexibility matrix of the free-free structure.

MODAL TRUNCATION - The deleted modes of the RESIDUAL STIFFNESS and

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RESIDUAL FLEXIBILITY formulations are not represented by any
correction factor. This is the most common dynamic aeroelastic
formulation reported in the literature.

QUASI STATIC - The motions of the structure are assumed to be
in-phase with the rigid body motions. The method is used
primarily for the conceptual and preliminary design of handling
quality and reduced static stability control systems for elastic
aircraft with a wide frequency separation between the axis
system motions and the structural deformations.

A contrast of the computational difficulties and the unique features
of each of the formulations is found in Tables 1 and 2. As shown, the EXACT
and MODAL SUBSTITUTION formulations consist of a large number of equations
that must be solved simultaneously and, in most cases, their number precludes
their use in the design of flight control systems. The RESIDUAL STIFFNESS
and RESIDUAL FLEXIBILITY formulations provide equivalent numerical results,
despite the differences in matrix formulation.

During the analytical studies to mathematically relate the various
formulations, it became apparent that a general criteria for the selection
of each formulation could be stated in terms of the major assumptions that
are required to derive each formulation (Reference 14). These major assump-
tions are presented in Figure 5. An examination of Figure 5 reemphasizes
that it is relatively easy to decide when the QUASI STATIC formulation is
appropriate. However, the decision on the appropriateness of the RESIDUAL
formulations or the MODAL TRUNCATION formulation is considerably more
difficult. The difficulty arises due to the necessity to numerically evalu-
ate the significance of the "structural spring forces," A8, and the "aero-
dynamic forces of structural deformation," A9 upon the performance of the
flight control system.

Presently, most of the aeroelastic stability and control design methods
in use in the industry do not possess the capability to evaluate these terms
for their numerical significance to the dynamics of the flexible aircraft.
In contrast, the Level 2.01 FLEXSTAB is specifically engineered and coded
to provide the USAF with the capability to consider both formulations, and
thus, to evaluate the numerical significance of A8 and A9 when applied to
the design of any proposed aircraft. This new capability in Level 2.01
should provide additional information concerning the interaction of modern
flight control systems with the structural dynamics of aircraft.

The aeroelastic stability and control parameters, to be calculated
with the Level 2.01 FLEXSTAB during the check-out using the aircraft presented
in Table 3, will provide more specific numerical criteria for the selection
of either the RESIDUAL FLEXIBILITY or MODAL TRUNCATION formulations of
Level 2.01 FLEXSTAB. The bomber/transport aircraft category is receiving
first attention due to their significant aeroelasticity at all flight condi-
tions. Once the numerical criteria are generated for this category, the
fighter category of aircraft will then be considered. Here, the emphasis
will be placed upon the unique fighter aircraft maneuvers that are comprised

of large rates of rotation and high load factors.

FUTURE DEVELOPMENT OF LEVEL 2.01 FLEXSTAB

The Level 2.01 FLEXSTAB is nearing the completion of the first cycle of funding action that was intended to provide AFFDL and the USAF with the capability to perform basic analysis and evaluation of conventional and CCV-type aircraft. As mentioned previously, the contractor and the AFFDL Control Criteria Branch decided early in the program that the high risk technical areas should be identified prior to beginning the extensive engineering or programming work in these high risk areas of analysis. There are two areas of high risk that have been identified for AFFDL in-house and contractual studies in FY75-76:

> The application of the low-frequency unsteady aerodynamics to the calculation of turbulence and gust induced aerodynamic forces.

> The identification of a suitable test case to verify the engineering and the coding of Level 2.01 FLEXSTAB.

The test application of the low frequency unsteady aerodynamic method to the calculation of unsteady aerodynamic stability and control derivatives such as $C_{L_{\hat{\alpha}}}$, $C_{m_{\hat{\alpha}}}$, $C_{n_{\beta}}$, $C_{\ell_{\dot{p}}}$, and $C_{m_{\dot{\delta}_S}}$ and the low-frequency, generalized aerodynamic forces has been "successful" to date. By successful is meant that "reasonable" correlation has been achieved on most test cases. The doublet lattice and the unsteady aerodynamic strip theory methods provide partial checks at subsonic speeds. At supersonic speeds there are no comparable theoretical methods that can represent the complex flow field around three dimensional aircraft configurations. Ideally, the Level 2.01 estimates should be compared to experimental data, as well as existing analytical data. However, the comparison to the experimental data will require the development of the parameter estimation method for flexible aircraft, to be discussed in the latter paragraphs of this section of the paper.

The low frequency unsteady aerodynamics have proven to be marginally acceptable to unacceptable for the calculation of atmospheric gust and turbulence induced aerodynamic forces. The problem in the Level 2.01 turbulence analyses is that the calculation of the Power Spectral Density of a parameter such as vertical acceleration due to vertical gusts, a_z/w_g, requires the integration of the square of the frequency domain representation of the a_z/w_g transfer function, multiplied by the turbulence Power Spectral Density. This integration over all frequencies does not converge due to the neglect of the higher order unsteady aerodynamic effects by the low frequency aerodynamics method. The contractor, Air Force Office of Scientific Research, and the Control Criteria Branch have studied the numerical problem in detail and identified the contributions of the individual terms of the transfer functions. There are four possible solutions:

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Incorporation of the Kussner-Wagner functions per conventional design practices.

Addition of the doublet lattice aerodynamic methods for turbulence analysis at subsonic speeds.

Retention of the next higher order frequency terms in the asymptotic expansion of the unsteady aerodynamic potential flow equations.

Expansion of the unsteady potential flow equations for "large frequencies" and then "matching" of the low and high frequency solutions for intermediate frequencies.

The first option is the obvious short term solution for Level 2.01, since only a small increase in the computer costs is involved in using the System. The incorporation of the doublet lattice method into the System is attractive, since it has become an accepted design method. Unfortunately, this solution may require extensive modification of the System, and thus eliminate some of the unique CCV analysis options currently available, e.g., the inclusion of the forward speed degree of freedom in the dynamics, the multiple equation of motion formulations discussed in the preceeding section of this paper, and the very general initial conditions of motion. NASA has contracted to study this problem in detail. In addition, the Air Force Office of Scientific Research has funded fundamental studies related to the unsteady aerodynamics methods applied to stability and control analyses. The third and fourth solutions are theoretically interesting, but unproven mathematically. Regardless, the incorporation of the latter 3 solutions is a relatively long term process requiring several extensive program check cases. These check case data are presently being collected during studies at AFFDL using the existing Level 2.01 programs and will be available for the future unsteady aerodynamic improvements to the Level 2.01 FLEXSTAB. The incorporation of the Kussner-Wagner functions and the correlation of Level 2.01 FLEXSTAB analytical estimates to the C-5A or the B-52E flight test data has been planned for FY75.

As mentioned, a new computer program requires extensive verification of the engineering equations and of the program coding. The development of the 13 programs of Level 2.01 FLEXSTAB compounded the verification problems, in that the check data on aircraft technology integration and the check data on the correlation of existing design methods to flight test data is practically non-existent. Ample amounts of wind tunnel test data on rigid aircraft models are available, along with comparisons to the other analytical methods. These data verify only the steady aerodynamic methods. Some static and dynamic structural data from ground vibration tests are available, but test conditions and parameters are not entirely suited to computer program check-outs; often these ground tests do not have a comparable flight test counterpart. The static-elastic aircraft models, cantilevered from stings or struts during wind tunnel tests, provide excellent checks of basic static aeroelastic calculations, but again little data is presently available. The cable-mounted flutter models provide some verification of dynamic

aeroelastic calculations, although cable friction and umbilical cord drag add incalcuable factors.

The Level 2.01 contractual test case consisting of the B-52E LAMS at Flight Condition Number 1 has provided mixed results. This is because the LAMS data were not intended for check cases for new computer programs and, thus, they were not qualified and correlated in any great detail to the results of the LAMS design methods. For example, the generalized structural damping added to each structural mode was not documented and has been assumed to be $\zeta = 0.03$ in the Level 2.01 check case. Additionally, typographical errors, such as sign errors in the summation of the LAMS feedback loops exist inadvertently in the formal AFFDL documentation. Numerous additional questions arise in the correlations between the results of the Level 2.01 FLEXSTAB and the flight test that cannot be answered because the basic LAMS calculations were not preserved.

To date, the AFFDL Control Criteria Branch has been unable to find a suitable operational aircraft that can check the Level 2.01 program to the degree desired. As such, the Control Criteria Branch has decided to select the best data from the aircraft and aircraft model wind tunnel tests that are presented in Table 3. Each aircraft or model checks an area of major calculation within the System. To whatever extent possible, the experimental data and the Level 2.01 analytical estimates will be compared to the estimates of the contemporary, parochial analytical methods. The contrast of the Level 2.01 calculations to the calculations of parochial design methods of the aerospace industry is particularly important, since it qualifies the inaccuracies of FLEXSTAB, while providing a historical background to measure the progress of research and development. The major contracted test cases are with the B-52E and the F-111 TACT aircraft and flexible model; the remainder are in-house check cases. The manpower and computer cost of the in-house effort, approximately 60,000 dollars, is considerably less than the costs of a single wind tunnel test of either a rigid or a flexible model of an aircraft.

It should be noted that the data collected during these check case studies of Level 2.01 are extremely valuable. The data provide a mathematical representation of the current USAF vehicles for AFFDL support to the System Program Office (SPO) and for the development of future analytical methods by AFFDL.

In addition to funding the unsteady aerodynamics improvement and the additional B-52E and F-111 TACT test cases, AFFDL Control Criteria Branch has decided to begin studies in four areas directly or indirectly related to Level 2.01:

> Modification of the System to allow the analysis of sting- and strut-mounted flexible models that are tested in wind tunnels.

> Creation of a structural loads analysis module that interfaces with the System and that provides a numerical measure of the

effectiveness of the CCV control system. This necessitates the study of the sensor equations that are appropriate to each of the math models of the dynamics of the aircraft.

Creation of an optimal control synthesis module that may be interfaced with the System.

Creation of a parameter estimation method for flexible aircraft to provide experimental check data for Level 2.01 from flight tests of aircraft.

In FY75-76, the Control Criteria Branch will study the difficulties involved in adding the capability for the analysis of static aeroelastic models that are tested in wind tunnels. Conceptually, this modification to Level 2.01 requires relatively minor changes to the System: the elimination of the "inertia relief" and the "free-free flexibility matrix" calculations in ISIC/NM, ESIC, and SD+SS programs (Figure 2) that are required for aircraft, but not for static aeroelastic models. The F-111 TACT flexible model serves as a check case for this modification, as well as an element in the overall System verification discussed previously. Pressure data will be incorporated into the analyses of the F-111'TACT flexible model and aircraft to further facilitate the numerical checks of the coding.

The FY75 studies will also investigate and define the form of the loads analysis equations that reflect the improvements or the degradations to the structural loads due to the operation of the CCV-type control systems. This study will include an investigation into the elastic correction factors on the accelerometer equations of motion found in the QUASI STATIC and the RESIDUAL FLEXIBILITY math models of flexible aircraft dynamics.

The incorporation of the optimal control synthesis methods as a feature of the Auxiliary Programs of the Level 2.01 FLEXSTAB is currently being studied in-house by the Control Criteria Branch (Reference 15). The aerodynamic program (AIC program of Figure 2) is particularly attractive in this study, in that it formulates the equations of motion in the time, Laplacian, and frequency domain of mathematical analysis. This uniqueness of Level 2.01 FLEXSTAB means that most if not all of the useful optimal control synthesis methods can be interfaced with Level 2.01 FLEXSTAB. This work by the AFFDL Control Criteria Branch is closely coordinated with the Active Controls Aircraft Office at NASA/ARC to ensure that funding duplication is avoided and that independent work by the AFFDL and NASA is available to all. The FY75-76 study in optimal control will use the C-5A as the test case.

The application of parameter estimation methods to flexible aircraft is receiving substantial attention from the Control Criteria Branch (Reference 16). There are several motivating factors that force the development of this type of Auxiliary Program for the Level 2.01 FLEXSTAB . First, the comparison of the steady and unsteady aerodynamic parameters, experimentally determined by flight tests, to the analytically calculated values, is essential to ultimately verify the accuracy of Level 2.01 FLEXSTAB, or any

other aeroelastic analysis program. Without these types of verification to qualify the precision of the analytical estimates of the aerodynamic parameters of importance to CCV design, the innovative use of CCV concepts may be penalized by design risk factors that are assumed to be too large. Since the existing parameter estimation methods treat the aircraft as a "rigid" vehicle, a new method must be developed.

Second, the practical necessity of removing all excessive structural weight, whether through conventional design practices or through active control systems, has resulted in vehicles that are more aeroelastic than previous vehicles with similar operational missions. To a degree, all flight vehicles, including fighter aircraft, are aeroelastic. The degree of aeroelasticity depends upon the particular flight condition (Mach number, dynamic pressure, and mass distribution) at which measurements or observations are made. In order to minimize the technical risks involved in the design of these type of high performance vehicles, a prototype or a pre-production vehicle is often constructed prior to committing a large amount of resources to a production vehicle. The SST aircraft are obvious examples. The intent of the prototype vehicle is to demonstrate that the design meets all the mission objectives. This demonstration entails flight tests of the prototype to verify the math models employed in the design and to isolate any configuration problems that would be objectionable in the production vehicle. Again, since the existing parameter estimation methods treat the flight vehicle as a "rigid" structure, they eliminate the possibility of explicitly identifying important aeroelastic parameters that affect the response of the aircraft. This reason further necessitates the development of a parameter estimation method for flexible aircraft.

The effort in parameter estimation consists of both in-house and contractual work planned through FY78. As a first step, the Control Criteria Branch is developing an in-house program that is based upon the maximum likelihood method. The test data for this program will consist of B-52E CCV flight test data that will be selected to minimize the anticipated numerical problems discussed in Reference 16. The B-52E CCV analytical start-up data for the in-house method has been generated using the Level 2.01 aerodynamic and structural finite element representation presented in Figure 6. A contracted effort will compare the B-52E CCV aircraft analytical model to flight data estimated by a method being developed by a contractor. Both groups of these parameter estimates will be correlated to flight test data measured during the tests of the dynamic response of the B-52E to step, ramp, and sinusoidal motions of the control surfaces. As part of the in-house effort, the ISIC and NM programs of Level 2.01 FLEXSTAB will be evaluated relative to the methods applied to the B-52E CCV aircraft by the contractor. The purpose of the evaluation is to identify any inaccuracy that could be introduced in the in-house developed parameter estimation method due to the theoretically calculated values of the generalized mass and stiffness and of the invacuum mode shapes.

The next phase of the effort will involve investigating the high risk/ high payoff areas of parameter estimation of flexible aircraft and further developing a production computer program. The final phase of the effort

will include an extension of the linearized methods to nonlinear analyses.

CONCLUDING REMARKS

The Level 2.01 FLEXSTAB Computer Program System has the potential to meet most of the immediate needs of the AFFDL and the USAF for an analysis and evaluation tool of conventional and CCV aircraft. Its capability for varied aerodynamic and structural finite element representations of the con- trols-fixed and the controls-free aircraft provides versatility and allows cost/effective analyses at the conceptual, preliminary, and advanced design levels of aircraft development. The modular independence of the thirteen programs that comprise FLEXSTAB facilitate improvements to the aerodynamic, structural, dynamic, and flight control program elements. In fact, several USAF and NASA efforts are presently underway or planned to enhance FLEXSTAB to create an increase in capability for the Level 3.01 FLEXSTAB System.

Of particular importance in Level 2.01 FLEXSTAB is the availability of the QUASI STATIC, the MODAL TRUNCATION, and the RESIDUAL FLEXIBILITY formulations of the dynamics of aircraft. These multiple formulations pro- vide a further capability for cost/effective analysis of the dynamics of both the controls-fixed and controls-free flexible aircraft. The numerous options in the Level 2.01 programs, for the inclusion of experimental data to improve accuracy, and for the interface of output data to flight simulators, optimal control synthesis methods, and parameter estimation methods, should make the Level 2.01 programs a key element in the development of the flight control systems of future military aircraft.

The AFFDL Control Criteria Branch and other organizations are currently applying Level 2.01 FLEXSTAB to the analysis and evaluation of all categories of military aircraft. This accrued experience is available to other inter- ested organizations through a liaison officer in the Control Criteria Branch. This officer is responsible for any request to the AFFDL for a copy of the Level 2.01 FLEXSTAB programs, for monitoring the successes and failures of other Level 2.01 users, and for answering user questions. Presently several contractors and government agencies are taking advantage of the service.

Most promising is the decision by the Mechanics Department of the Air Force Institute of Technology (AFIT) to use the Level 2.01 FLEXSTAB programs and documentation as an illustrative tool to teach the intricacies of combining technologies during the design of modern aircraft. These USAF students provide valuable, constructive criticisms to the AFFDL Control Criteria Branch. Their related thesis work should provide new and often innovative ideas that could be incorporated into future Levels of FLEXSTAB. Finally, their background in finite element programs, such as FLEXSTAB, prepares them for the tasks of following the contractors of the USAF during the conceptual, preliminary, advanced design, and flight test phases of new aircraft development, or during the contractual modification of operational aircraft.

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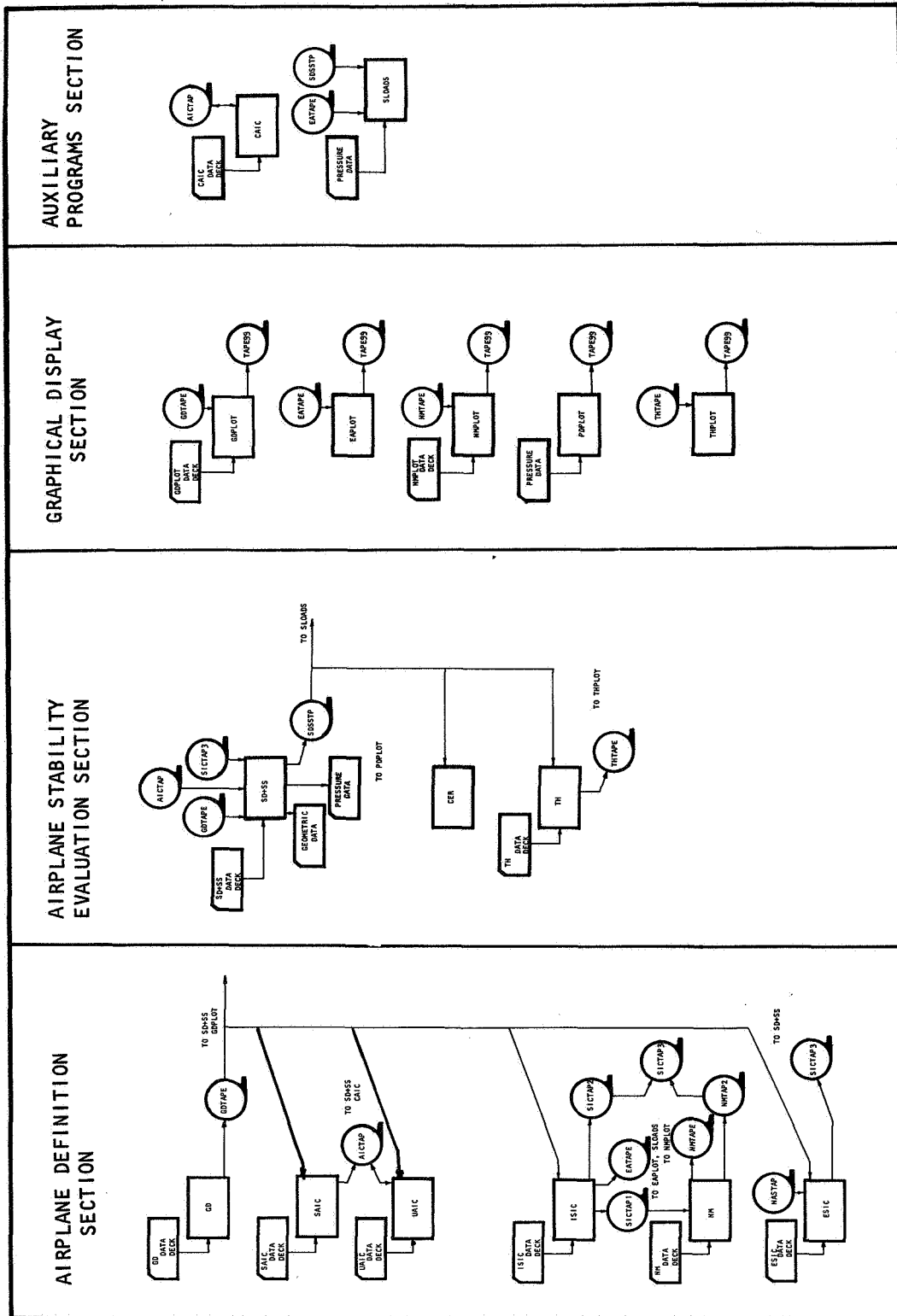
Figure 1. Level 1.01 FLEXSTAB Computer Program System
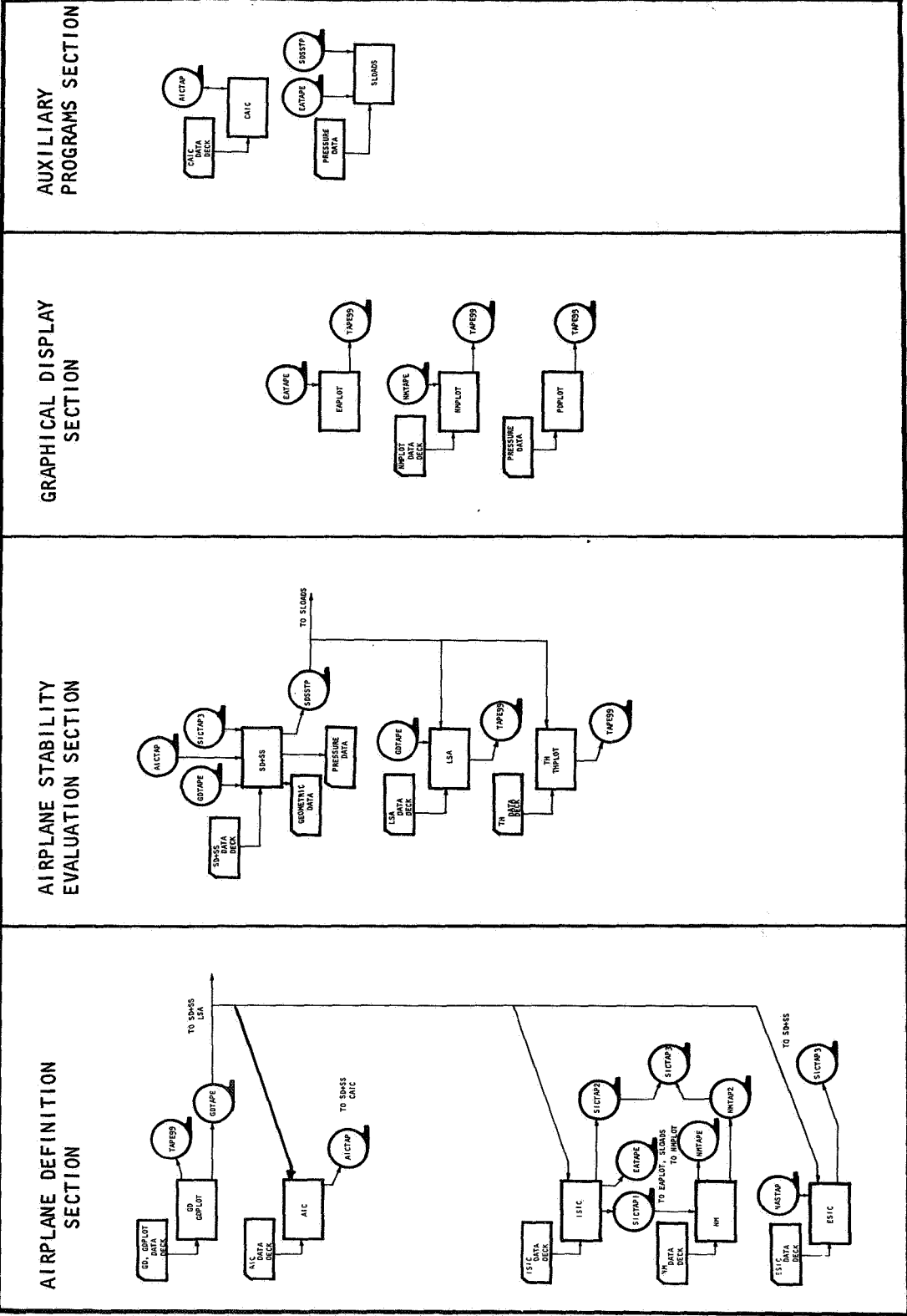
Figure 2. Level 2.01 FLEXSTAB Computer Program System

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200×200 AIC ANALYSIS
ONE ENGINEER/ONE AIRCRAFT CONFIGURATION
ONE ANALYSIS POINT (M, q, Wt, STATICS, DYNAMICS)

(QS) – QUASI STATIC
(MT) – MODAL TRUNCATION
(RF) – RESIDUAL FLEXIBILITY

LEVEL 1.01 FLEXSTAB (1973)

No Computer Priority
2 Day Turnaround

Program	CP + 10/2(sec)
GD	30
SAIC	4,000
UAIC	2,000
ISIC	8,000
SD+SS(QS)	4,000
CER	100
	18,130

ISIC

SAIC

UAIC

GD

ISIC   SD+SS

CER

Manhours

Workdays

LEVEL 2.01 FLEXSTAB (1974)

No Computer Priority
1 Day Turnaround

Program	CP+10/2(sec)
GD	19
AIC	2,655
ISIC	937
SD+SS(QS)	1,321
	4,932
NM	308
SD+SS(MT)	1,952
SD+SS(RF)	2,950
	10,142

GD
AIC
ISIC
NM
SD+SS

SD+SS

Manhours

Workdays

Figure 3. Costs for Aeroelastic Stability and Control Analysis at WPAFB

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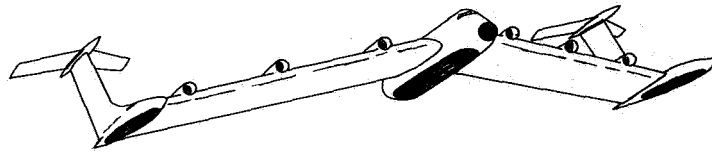
a. Lockheed Spanloader Aircraft
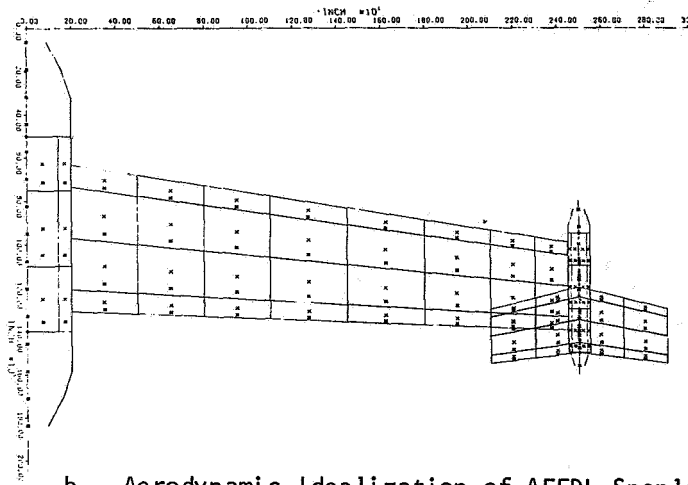
b. Aerodynamic Idealization of AFFDL Spanloader
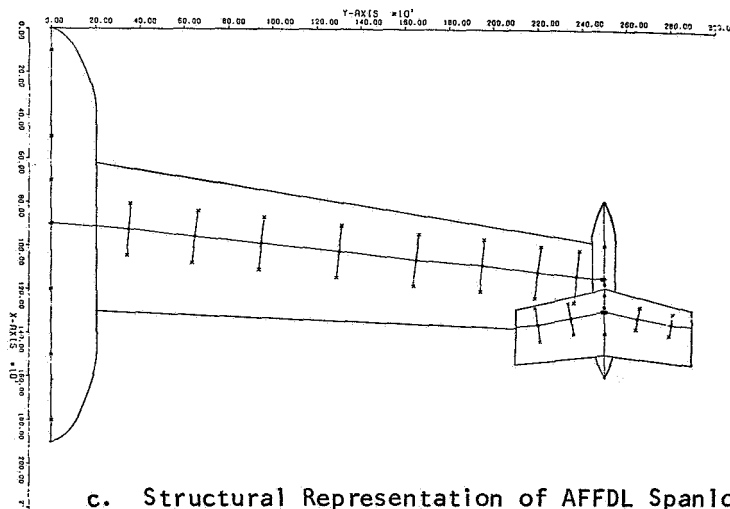
c. Structural Representation of AFFDL Spanloader

Figure 4. Application of Level 2.01 FLEXSTAB to Conceptual Design
Analysis of a Spanloader Aircraft

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Table 1. Computational Difficulties Associated with the
Linear Formulations of the Equations of Motion

| METHOD | ECONOMICS | | EIGENVALUE ROUTINE | | STRUCTURAL DATA REQUIRED |
	PROGRAM SIZE	PROGRAM RUN TIME	REAL NUMBER	COMPLEX NUMBER	
EXACT	LARGE	?	-	LARGE	[K]
MODAL SUBSTITUTION	LARGE	?	LARGE	LARGE	[K]
RESIDUAL STIFFNESS	MEDIUM	LONG	LARGE	MEDIUM	[K]
RESIDUAL FLEXIBILITY	MEDIUM	LONG	MEDIUM	MEDIUM	[K] AND [C]
MODAL TRUNCATION	MEDIUM	MEDIUM	MEDIUM	MEDIUM	[K]
QUASI STATIC	SMALL	SHORT	-	SMALL	[K]

Table 2. Unique Features of the Formulations Describing Lightly
Damped Aircraft Dynamics

METHOD	ADVANTAGES	DISADVANTAGES
MODAL SUBSTITUTION	PERMITS COMPLETE ANALYSIS OF LIGHTLY DAMPED AIRCRAFT MOST ACCURATE	LIMITED TO LINEAR OR PIECEWISE LINEAR SYSTEMS SLOWEST, MOST COSTLY
RESIDUAL STIFFNESS	REDUCED: NUMBER OF UNKNOWNS ANALYSIS CYCLE TIME COMPUTING COSTS	NEGLECTS DYNAMICS OF DELETED MODES REQUIRES ALL INVACUUM MODES
RESIDUAL FLEXIBILITY	SAME AS RESIDUAL STIFFNESS REQUIRES ONLY INVACUUM MODES OF INTEREST	REQUIRES FREE-FREE FLEXIBILITY MATRIX
MODAL TRUNCATION	SAME AS RESIDUAL FLEXIBILITY COMMON ANALYSIS TECHNIQUE	NEGLECTS STATICS AND DYNAMICS OF DELETED MODES
QUASI STATIC	FASTEST, LEAST COSTLY	NEGLECTS ALL DYNAMICS OF STRUCTURAL MOTION LEAST ACCURATE

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DATA
● AERODYNAMIC
● STRUCTURAL
● INERTIAL
● GEOMETRIC

INERTIAL

INERTIAL
OR
NON-INERTIAL
COORDINATES

NON-INERTIAL
BODY FIXED AXIS

LINEAR

LINEAR
OR
NON-LINEAR
SYSTEM

NON-LINEAR

EXACT
FORMULATION

STATIC

STATIC
OR DYNAMIC
AEROELASTIC
ANALYSIS

DYNAMIC

A1

A2

QUASI STATIC

EXACT
FORMULATION

NO

IS
SYSTEM
DAMPING
SIGNIFICANT

YES

A3

A4

EXACT
FORMULATION

MODAL
SUBSTITUTION

A5

A6

A7

RESIDUAL
STIFFNESS

RESIDUAL
FLEXIBILITY

A8

A9

MODAL
TRUNCATION

MODAL
TRUNCATION

ASSUMPTIONS

A1 The aerodynamic forces proportional
    to structural velocity and acceler-
    ation are zero.
A2 The structural inertial and damping
    forces are zero.
A3 Structural damping forces are neg-
    ligible.
A4 Aerodynamic forces due to elastic
    deformation are negligible.
A5 The aerodynamic forces due to
    deleted modal velocity and acceler-
    ation are zero.
A6 The structural damping and inertial
    forces due to deleted modal deforma-
    tion are zero.
A7 The structural damping of retained
    modes on deleted modes is zero.
A8 The structural spring forces of
    deleted modes are zero.
A9 The aerodynamic forces due to
    deleted modes are zero.

Figure 5. General Criteria for the Selection of the Formulation
of the Equations of Motion in Level 2.01 FLEXSTAB

Table 3. Initial Level 2.01 FLEXSTAB Check Case Aircraft, AFFDL In-house and Contract

TEST CASE	THEORETICAL											SEMI-EMPIRICAL				DETERMINISTIC GUST/CONTROL		RANDOM TURBULENCE	
	GD	AIC	ISIC	NM	ESIC	SD+SS(R)	SD+SS(QS)/SD+SS(MT)/SD+SS(RF)	LSA(R)	LSA(QS)/LSA(MT)/LSA(RF)	TH(R)	TH(QS)/TH(MT)/TH(RF)	CAIC	SD+SS(QS)/SD+SS(MT)/SD+SS(RF)	LSA(QS)/LSA(MT)/LSA(RF)	TH(QS)/TH(MT)/TH(RF)	TH(R)	TH(QS)/TH(MT)/TH(RF)	LSA(R)	LSA(QS)/LSA(MT)/LSA(RF)
Check Original NASA/ARC																			
707	•••	•	•	•	••	••	••						•		•				
YF-12 (M<1.)	••	•			••	••	••			•	•		•		•			LSA(R)	LSA(QS)/LSA(MT)/LSA(RF)
YF-12 (M>1.)	•																		
Bomber/Transport																			
B-52E LAMS	•••	••••	••••	••••	••••	••••	••••	•••	•••	•	•	••	•••	•••	• •	•••	•••	•••	•••
B-52E CCV	•••	••••	•••	••	•••	••••	••••	•••	••••	•	•	••	•••	••	• •				
C-5A ALDCS	••	•••	•••		•••	••••	••••	••••	••••									••••	••••
CCV KC-X	••	•	•					•••	•••										
Strategic Bomber (M<1.)																			
Strategic Bomber (M>1.)																			
CMW Strategic Bomber (M<1.)																			
CMW Strategic Bomber (M>1.)																			
Fighter Aircraft																			
F-111 TACT (M<1.)	•••	•••			••••	••••	••••	••	••	• •	••	••••	•••••	••	• •	••	••	••	••
F-111 TACT (M>1.)	•••	•••						••	••										
TACT FLEX MODEL (M<1.)	•																		
TACT FLEX MODEL (M>1.)																			
F-15 CMW	••	••	••	•	••	•••	••	••	••	• •	••	•	••	••					
YF-16 CCV																			

(R) RIGID ANALYSIS
(QS) QUASI STATIC ANALYSIS
(MT) MODAL TRUNCATION ANALYSIS
(RF) RESIDUAL FLEXIBILITY ANALYSIS

a. The B-52E CCV Aircraft

b. Aerodynamic Idealization of B-52E CCV

c. Structural Idealization of B-52E CCV

Figure 6. Application of Level 2.01 FLEXSTAB to the B-52E CCV Aircraft

METHODOLOGY FOR DESIGN OF ACTIVE CONTROLS

FOR V/STOL AIRCRAFT

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ABSTRACT

An effort is underway at the Ames Research Center to develop techniques for the design of integrated, fully automatic flight control systems for powered lift STOL and VTOL aircraft. The paper describes the structure of the control system which has been developed to deal with the strong non-linearities inherent in this class of aircraft; to admit automatic coupling with the advanced ATC requiring accurate execution of complex trajectories; and to admit a variety of active control tasks. The specific case being considered is the Augmentor Wing Research Aircraft.

INTRODUCTION

NASA through its STOL and VTOL research programs is investing substantial resources in developing powered lift technology. In all cases, the wide range of lift coefficient required to cover all flight conditions between cruise and landing is achieved by in-flight modification of aircraft configuration. These modifications result in drastic changes in control characteristics of the aircraft, and, particularly in the high-lift transition and landing configurations, the aircraft response to control inputs is very nonlinear. Moreover, the presence of powered and direct lift generators increases the total number of controls available to the pilot who must continually make decisions on control techniques. Finally, the coming short-haul transportation system will be required to satisfy stringent environmental constraints which will necessitate accurate execution of complex trajectories. Accurate, unaided manual tracking of complex trajectories by manipulating a large set of inter-acting controls of an aircraft whose control characteristics are non-linear and rapidly changing represents an unacceptably high pilot work load. Active control technology has the potential to provide a means for reducing the pilot work load to an acceptable level by integrating control functions in such a way as to generate desirable handling qualities without reduction in the per-formance of the aircraft as an element of the advanced civil air transportation system. The advantages of active control technology are potentially even more substantial in military applications of STOL and VTOL aircraft. Both the Advanced Military STOL and the Sea Control Fighter VTOL must utilize the maneuvering capacity of the basic aircraft to the fullest. The tracking of complex penetration trajectories must be sufficiently accurate for proper execution of mission, and the pilot work load associated with flying must not adversely affect his ability to perform other tasks. Again, the maneuverability, accuracy, and level of pilot work load can be improved by means of active

control technology. At the present time, however, the practical problems of applying the technology to powered lift aircraft are not well understood. In order to provide the required data base, an applications-oriented program has been initiated at the Ames Research Center. The objectives of this program are to generate design guide lines and to provide flight test confirmation required for incorporation of active control technology into this class of aircraft. The present paper describes the progress made in one segment of this program, namely, the development of a methodology for the design of automatic trajectory control systems for powered lift aircraft.

THE AUGMENTOR WING RESEARCH AIRCRAFT

The specific case being used in the development and tests of the design methodology is the Augmentor Wing Research Aircraft. The aircraft is a de Havilland C-8A "Buffalo" modified according to the general arrangement shown in figure 1. The aircraft is powered by two turbofan engines. The relatively cold flow from the front fans is ducted through the wing and fuselage to the augmented jet flap, blown ailerons, and fuselage boundary layer control systems. The hot gas flows through two pairs of nozzles which can be rotated in flight to provide vectoring of the hot thrust through a $98°$ range. The hot and cold thrusts are nonlinear functions of the throttle setting. The nozzle servos move the nozzles in unison in response to a single nozzle angle command. The system is quite fast, being limited to 90 ($°$/sec.). The throttle-to-thrust control system is relatively slow with a bandwidth of approximately 1 (rad./sec.).

The cold flow has a pronounced effect on the lift and drag polars of the aircraft. For example, figure 2 shows the wing-body polars for two flap settings. The independent variables in the plots are the aircraft angle of attack, α, and the cold thrust coefficient $C_J = T_c/QS_W$, where the cold thrust T_c is a nonlinear function of throttle, and density and temperature of the air; Q is the dynamic pressure, and S_W is the wing area. Of particular significance for the design of flight path control systems is the large variation in the basic aerodynamic characteristics of the aircraft.

Certainly, there is a large change between the cruise configuration (flap = $4.5°$) and the landing configuration (flap = $65°$). But present indications are that the nonlinearity is significant even over a much smaller region. For example, figure 3 shows the total lift and drag coefficients, including the effects of the hot thrust for the case of constant flap, throttle and speed which corresponds to a typical landing configuration with angle of attack and nozzle angle ν in the active control mode. Point A_1 in the figure represents equilibrium flight along the $-7.5°$ glide slope. Point A_2 represents level flight. Also shown are the derivatives of the total force coefficient at these two points. As the aircraft is maneuvered from point A_1 to point A_2 the changes in these derivatives may adversely affect closed loop dynamics. But of greater concern is that if the maneuver is performed by means of a feed-forward command based on the linear model at point A_1, then the aircraft will be out of trim at A_2 by $\Delta C_L /C_L = 4.7\%$. Because of bandwidth limitations imposed on the altitude control loop ($\omega_n \leq 0.5$ rad./sec.) by unsteady aerodynamics, the error in trim results in an altitude error $\Delta h > 6$ ft. Similarly, transition from A_2 to A_1 will end up at A_{21}; the corresponding error $\Delta h > 16$ ft.

Of course this hangoff error can be removed by means of an integrator, but the removal will be too slow for many maneuvers. Consequently, the transition between A_1 and A_2 must be considered to be nonlinear.

The design problem is further complicated by the presence of redundant controls. Thus, the two-dimensional total force coefficient $C = (C_D, C_L)^T$ is a function, say $C(F,T,\alpha,\nu)$, of four variables, namely flap, throttle, angle of attack, and nozzle angle. For example, figure 4 shows the plot of $C(F,T,\alpha,\nu) = C_0$, where C_0 corresponds to steady flight along $-7.5°$ glide slope. It may be noted that the plot is rather nonlinear. The problem is to be able to generate online optimum trim values of the controls (F,T,α,ν) for any admissible trim values of (C_D, C_L).

DESIGN APPROACH

The approach is motivated by the following line of reasoning. Let equation (1) be the system state equation.

$$\dot{x} = f(x,u) \tag{1}$$

The control u is restricted to a set U which may depend on the state x. A trajectory $(x_0(t),\ t \in T)$ is flyable if for all $t \in T$, there is a control $u_0(t)$ such that

$$\dot{x}_0(t) = f(x_0(t), u_0(t)) \tag{2}$$

The trim problem is to find a control u_0 satisfying (2), given that the trim trajectory is flyable. The solution will be an inverse of (1), namely a function (g,F), which we call the trimmap, such that for all $(\dot{x},x) \in F$,

$$f(x, g(\dot{x},x)) = \dot{x} \tag{3}$$

The corresponding trim control is given by

$$u_0 = g(\dot{x}_0, x_0) \tag{4}$$

Usually, trim refers to cases with constant u_0. Here u_0 may vary with time. Note that when the controls are redundant, state equation (1) alone is not enough to define the trimmap (g,F), and additional conditions must be introduced to resolve the redundancy.

The trim problem may be difficult to solve; but, evidently, its solution to required accuracy is the essential first step in the design of automatic flight path control systems. The next step usually taken is to design a control system based on perturbation models. Thus, given a flyable nominal trajectory $(\dot{x}_0, x_0) \in F$ trimmed by u_0 according to equation (4), the linear model (5) is obtained for the perturbations $\delta x = x - x_0$ and $\delta u = u - u_0$.

$$\delta\dot{x} = f_{x_0} \delta x + f_{u_0} \delta u \tag{5}$$

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Then, the application of the methods of linear control theory yields the perturbation control law (6).

$$\delta u = K \delta x \tag{6}$$

Since the coefficients in (5) depend on the nominal trajectory, the process must be repeated for sufficiently large number of nominal trajectories $(\dot{x}_0, x_0) \in F$ until F is adequately covered. The result is a scheduled gain matrix $K(\dot{x}_0, x_0)$, and the complete control law is

$$u = g(\dot{x}_0, x_0) + K(\dot{x}_0, x_0)(x - x_0) \tag{7}$$

The major drawback of this approach is that when the state equation (1) is highly nonlinear, the procedure for choosing the proper set of nominal trajectories to cover the flight envelope F is, at present, rather unclear. For this reason we at Ames have decided to investigate a different approach.

Consider the trim equation (4). Suppose that initially $x = x_0$. Then in the absence of modeling errors the control u_0 will maintain $x = x_0$. The tracking will be perfect even if at some point the acceleration of the nominal trajectory is perturbed from $\dot{x}_0$ to $\dot{x}_0 + \delta\dot{x}_0$, provided that $(\dot{x}_0 + \delta\dot{x}_0, x) \in F$. The corresponding control is

$$u = g(\dot{x}_0 + \delta\dot{x}_0, x) \tag{8}$$

Now, suppose that initially $x - x_0 = \delta x \neq 0$, but that the error can be removed by means of a flyable trajectory. Then there is a perturbed nominal acceleration $\dot{x}_0 + \delta\dot{x}_0$ which will take x into x_0 by means of the control law (8). That is, the feedback for the control of process uncertainties can be closed through the trimmap as in equation (8), rather than after the trimmap as in equation (7). Such control by means of continual adjustments in commanded acceleration forms the basis of the Ames approach. The emphasis is shifted from perturbation models on F to flyable perturbations in commanded acceleration. The next section describes the resulting structure of the control system.

FULL FLIGHT ENVELOPE AUTOPILOT

The proposed structure of the autopilot is shown in figure 5. The plant represents the basic aircraft together with attitude and throttle servosystems, and sensors. Everything to the left is the autopilot. It consists of four blocks - trimmap, wind filter, compensator, and command generator - which carry out the following functions.

Trimmap computes the active control u_c to generate acceleration with inertial coordinates $\dot{V}_{si}$. For the case shown, the active controls are the commanded attitude and nozzle angle; while the redundant controls are the throttle and flap. Any other partition of the controls is treated similarly. The total commanded aerodynamic force F_{sc} is transformed into estimated stability coordinates F_{vc} from which commanded roll ϕ_v, angle of attack α_c, sideslip angle β_c,

and nozzle angle ν_c are computed online using the nonlinear inverse function g. The commanded attitude direction cosine matrix is given by

$$A_{cs} = E_2(\alpha_c)E_3^t(\beta_c)E_1(\phi_v)E_2(\gamma_v)E_3(\psi_v) \qquad (9)$$

The attitude control system (servo) may operate directly on A_{cs}. In case Euler angles are required, they are given by $A_{cs} = E_1(\phi_c)E_2(\theta_c)E_3(\psi_c)$. In any case, commanded attitude and nozzle are defined.

Wind filter computes smoothed inertial coordinates v_g^* of aircraft velocity relative to the airmass from body mounted air velocity sensors, and inertial velocity and attitude of the aircraft. The relative velocity is needed in the trimmap to locate stability axes and to convert forces into coefficients. Note that only inertial coordinates of wind are filtered. The aircraft velocity is unaffected. Hence, in the absence of sensor errors and wind, $v_g^* = V_g$.

Trimmap, wind filter, and attitude and throttle control systems form an acceleration controller. The input is $\dot{V}_{si}$; the output is the actual acceleration $\dot{V}_g$ of the aircraft. Moreover,

$$\dot{V}_s = \dot{V}_{si} + e \qquad (10)$$

where the error e depends on the inaccuracies of inversion g and wind estimates, the presence of unsteady aerodynamics in f such as alpha dot effects, and on the attitude and throttle servo dynamics. It is the purpose of the compensator to generate corrective accelerations $\dot{V}_{sm}$ to compensate for the error e of the acceleration controller. Inertial coordinates of position, velocity, and acceleration are transformed into approximately longitudinal, lateral, and normal errors by means of the direction cosine matrix A_{vc} computed from the commanded inertial velocity V_{sc}; the errors are weighted by constant gain matrices K_1, K_2, and K_3 commensurate with the acceleration capacities of the aircraft in these directions, and the result is filtered to insure compatibility with attitude and throttle servo dynamics. The corrective acceleration is transformed back into inertial space and added to the command $\dot{V}_{sc}$ to give the input $\dot{V}_{si}$. In this way, the feedback is closed around the process uncertainties e so that the representation

$$\dot{V}_s = \dot{V}_{sc} \qquad (11)$$

is sufficiently accurate provided that $\dot{V}_{sc}$ is admissible, namely $(\dot{V}_{sc}, V_s)$ is flyable and the bandwidth of $\dot{V}_{sc}$ is suitable restricted.

The last major block of the autopilot is the command generator. Its purpose is to porvide only admissible commands to the acceleration controller. One of the subblocks defines the autopilot mode. For the case shown in the diagram, 27 modes are available. Every mode defines whether position, velocity or acceleration is to be tracked in each of the three axes. Thus, mode (0,0,0) requests three axis acceleration tracking; mode (1,1,1) requests three axis velocity tracking; etc. As an example of the use of modes suppose that the autopilot is in mode (2,2,2) tracking position of a 4-D trajectory commanded by the air traffic control (ATC) as the aircraft penetrates a heavy, localized turbulence. The mode may have to be changed to, say, (1,1,1) or possibly even (0,0,0). On exiting the turbulence, the mode may be returned

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back to (2,2,2). The command generator must generate an admissible trajectory for bringing the aircraft back on the trajectory commanded by the ATC.

The ATC trajectory may be transmitted continuously to the aircraft, or more likely, it may be generated onboard from a given set of trajectory parameters. The latter may be transmitted by the ATC or selected by the pilot. In any case, if the commanded trajectory $(R_{sc}^*, V_{sc}^*, \dot{V}_{sc}^*)$ is discontinuous in any of the variables (e.g. "step down altitude by 500 feet" or "change glide path from $-7.5°$ to $-2°$".), the command generator must generate the required flare maneuver.

Such flare maneuvers are generated by means of transition dynamics. This subblock consists of a stable state equation, initial conditions, and an output map. At the time of the initiation of the transition dynamics, $(R_{sc}, V_{sc}, \dot{V}_{sc}) = (R_s, V_s, \dot{V}_s)$. At the end of the transient, $(R_{sc}, V_{sc}, \dot{V}_{sc}) = (R_{sc}^*, V_{sc}^*, \dot{V}_{sc}^*)$. The dynamics of the transient are made compatible with the acceleration controller by a proper selection of the state equation. To ensure continuity in position, velocity, and acceleration, the state equation must be at least three-dimensional (and three-axis). In the diagram, a linear state equation is shown. Nonlinear transition dynamics are currently being designed to permit arbitrarily large initial deviations from the ATC command.

The feasibility of the autopilot has been tested by application to the unmodified C8A and the Augmentor Wing Research Aircraft for which detailed simulations are available at Ames. Present indications are that the proposed structure is feasible, although final evaluation must await flight tests which are scheduled in 1976.

The present paper presented an overview of the proposed design methodology. Several reports, currently in preparation and soon to appear, discuss the methodology in greater detail.

CONCLUSION

The proposed design approach has several advantages, among which are the following.

(1)   The approach is applicable to a large class of aircraft with nonlinear dynamics.

(2)   The approach is nearly algorithmic.

(3)   The approach is invariant for a wide spectrum of tracking accuracy requirements.

(4)   There is an effective trade-off between tracking accuracy requirements and computer requirements and a priori knowledge of system dynamics.

Present indications are that the proposed design methodology is feasible, but definite evaluation must await flight tests.

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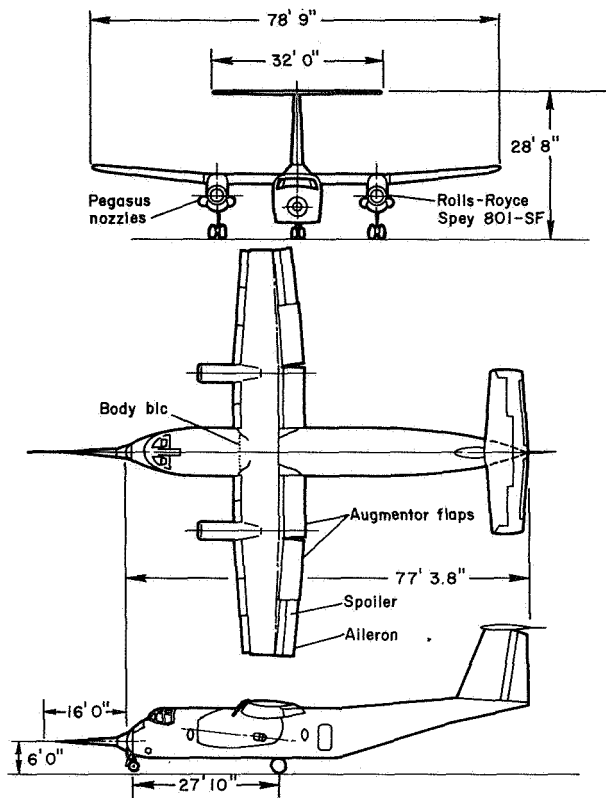
Figure 1.   Modified C-8A, General Arrangement
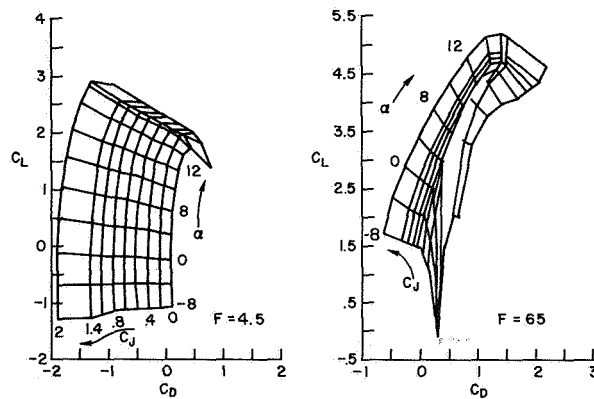
Figure 2.   Typical Wing-Body Polars of the Augmentor Wing
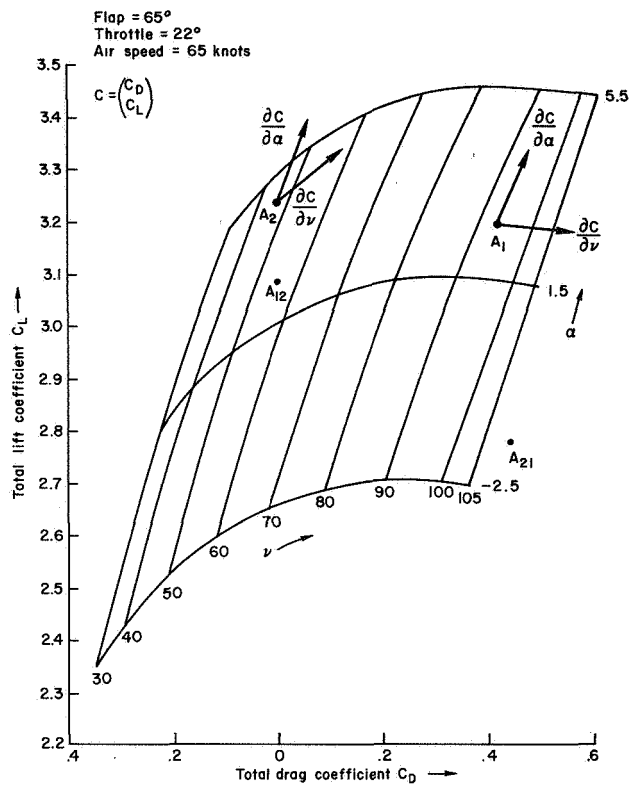
Figure 3. Total Force Coefficient
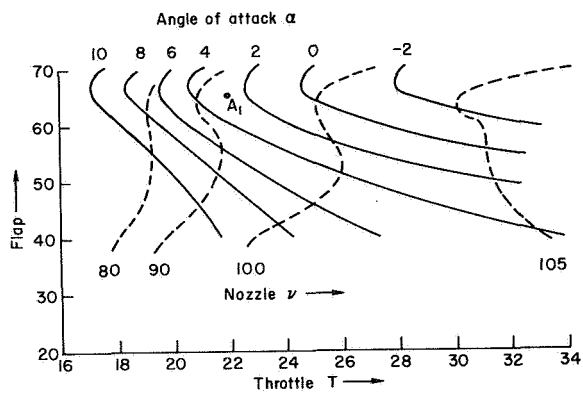
Figure 4. Controls for One Value of Total Force Coefficient

Figure 5. Full Flight Envelope Autopilot

ACTIVE CONTROL SYSTEM TRENDS

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ABSTRACT

The Active Control Concepts which achieve the benefit of improved mission performance and lower cost generate the system trends. The system trends are towards improved dynamic performance, more integration and digital fly-by-wire mechanization. These system trends yield new analytical issues and implementation requirements:

- Higher bandwidth, more dynamic coupling, stochastic and deterministic inputs.

- Limited control power.

- Multiple control loops, more interaction, multiple and conflicting criteria.

- Reliability (safety-of-flight requirements) and low cost.

New tools and approaches have been or are being developed to address the new analytical and implementation issues:

- Quadratic Optimal Control
- Multiloop Frequency Response
- Digital System Analysis

- Large Scale Integration
- Microprocessor technology
- Digital Architecture
- Software technology

INTRODUCTION

Active control system trends are towards improved performance, more integration and digital fly-by-wire mechanization. Most active control concepts are well known and will be briefly noted. The benefits are better mission performance and lower cost. The active control concepts and benefits determine the system trends.

The new analytical issues result from the active control system trends. An active control system has a first order effect on aircraft performance. More aircraft and controller configurations must be analyzed. The control

design problem is larger and more complex. The designer must consider higher bandwidths, more dynamic coupling, stochastic and deterministic disturbances and commands and limited control power. Simultaneous implementation of some active control concepts yields design problems with multiple sensors, multiple responses and multiple control loops with multiple and conflicting criteria.

New tools are needed to address these analytical issues. Better aircraft mathematical models are required. Other tools are classical and modern controller synthesis and analysis approaches (quadratic optimal control, multivariable frequency response and digital systems analysis).

New implementation needs also result from the active control system trends. The designer is confronted with safety-of-flight control system requirements. Control system cost becomes a crucial factor in achieving control configured vehicle cost benefit.

New approaches to meet the implementation requirements are digital mechanization, advances in architecture to use microprocessors and to achieve fault tolerance, large scale integration and software technology.

These topics will be discussed in subsequent sections:

- Active control concepts and benefits

- Control systems trends

- Analytical issues

- Analytical tools

- Implementation needs

- Implementation technology trends.

ACTIVE CONTROL CONCEPTS AND BENEFITS

Active control concepts are well known and have been studied either on paper or by flight test on several airplanes:

- Relaxed static stability (e.g. C-5A, F-8, F-16, JA-37)

- Ride Smoothing (XB-70, B-52, B-1, C-5A, YF-12, JA-37)

- Flight envelope limiting (F-101, F-104, F-8)

- Maneuver load relief (C-5A, B-52)

- Gust load relief (C-5A, B-52, YF-12)

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- Structure mode damping (B-52, C-5A, YF-12)

- Flutter mode damping (B-52, YF-12)

- Flight path and attitude coupling (B-52 direct lift control, C-5A, Advanced Fighter Technology Integration Program, C-130 gunship).

Other concepts which are related to the propulsion system can be included in a list of active control concepts.

- Propulsion integration (TF-30 and Joint Technology Demonstrator Program)

- Flight Propulsion Coupling (YF-12 and Flight Propulsion Control Coupling Program)

A rather extensive data base is evolving for these concepts [1 - 8].

The benefit of active control is improved mission performance (payoff) and reduced cost. The measure of performance is dependent on the particular aircraft's mission. It could include payload and range in a transport type aircraft and flight envelope and maneuverability in a fighter aircraft. The cost is total system life cost in dollars. The goal is to maximize the ratio payoff to cost.

ACTIVE CONTROL SYSTEM TRENDS

Active control system trends fall into three cateogires:

- Performance

- Integration

- Mechanization

Improving system performance increases the difficulty of the design problem and can affect the implementation cost. The designer must consider wider bandwidths and more dynamic coupling like in structural and flutter mode suppression and ride quality control. He must design control systems for both stochastic and deterministic commands and disturbances like maneuver and gust load control. When we push control configured vehicle concepts to the limit the designer is generally faced with limited control power. This can generate a requirement for flight envelope limiting or be a design constraint during flight envelope limiting.

Integration is required to implement more than one CCV concept along with conventional autopilots and control and stability augmentation systems. It is also a result of trying to improve performance by implementing favourable coupling. Control system or mode integration presents the designer with multiple sensors, multiple responses and multiple control loops with more inter-action between variables. It can also present the designer with a multiple and

conflicting criteria. Examples of the integration trend are the B-52 CCV program, the YF-12 Cooperative Autopilot Propulsion Control System program and the TF-30 Integrated Propulsion Control System program.

The system mechanization trends are towards safety-of-flight requirements and fly-by-wire mechanization. Safety-of-flight requirements come from the performance requirements of some CCV concepts like relaxed static stability. The SR.-71 is a specific example. Safety-of-flight will be a requirement if future aircraft are designed to rely on stress relief or mode stabilization for structure integrity. Fly-by-wire mechanizations reduce cost and improve performance. Fly-by-wire has been demonstrated on the F-4, F-8 and C-141.

The mechanization trends are towards digital implementation. This is caused by the cost projections of digital hardware. Production digital hardware is here today on the JA-37 digital flight control. It is coming soon on the Space Shuttle digital flight control and main engine control, and the Integrated Propulsion Control System Program. Cost is the primary factor in the digital versus analog tradeoff.

ACTIVE CONTROL ANALYTICAL ISSUES

The analytical issues resulting from the systems trends and active control concepts pose a more difficult design and specification problem for the buyer and supplier. Active control system directly affect aircraft performance. The control design engineer is also confronted with:

- higher bandwidths

- coupled dynamics

- stochastic and deterministic commands and disturbances

- limited control power

- multiple variables

- multiple and conflicting criteria

- increased interaction

- digital specifications

- extensive system tradeoffs.

The origin of most of these issues was discussed in the previous section.

Two additional issues are digital specifications and extensive system tradeoffs. When digital mechanization is a candidate it is necessary to select and specify digital variables:

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- sample rate and multiple sample rates

- wordlength and multiple wordlength

- computational delay and multiple computational delay.

This is a new analysis and design problem. Extensive tradeoffs between active control and aircraft concepts and configurations are necessary to optimize the benefit. Both aircraft and controller design and analysis speed and cost are the issue.

ANALYTICAL METHODS

In order to achieve the potential benefits of active control technology we usually require accurate and relatively complete descriptions of the aircraft dynamics. More design iterations of these sophisticated configurations are usually required.

NASA and the Air Force are funding significant efforts to develop computer programs to generate the aerodynamic and structural models from the standpoint of the control system designers needs and also to develop the programs necessary to rapidly synthesize and analyze active control configurations.

In addition, technology can be developed which will make the control system relatively insensitive to modeling deficiencies and errors. This is important for both reliability and performance reasons.

Modern control technology modulation to the well developed classical techniques presently exist to aid the designer. These are in the areas of Optimal Control, Digital Control and Multiple Input/Output Control.

OPTIMAL CONTROL. Quadratic optimal control synthesis techniques makes it relatively easy to handle:

- multiple control inputs

- multiple sensor outputs

- multiple responses

- multiple criteria

- stochastic and deterministic commands and disturbances

- digital and analog mechanizations

- constant or time varying dynamics

- short and long mission segments.

These features were recognized back in the early sixties. Since then over 30 man-years of development have gone into making the quadratic optimal methodology a practical design tool at Honeywell [9-12], probably many times that effort have been carried on throughout the United States. These developments of the quadratic design methodology have been directed towards:

- lowering the design cost

- improving control system performance

- constraining the designs to simpler hardware

- automated modeling

- consideration of the data and model uncertainties.

Today the quadratic design methodology for controller synthesis is a design and analysis software package.   It can be used to:

- configure control systems

- compute control laws

- evaluate control system performance

- perform extremely rapid tradeoffs between competing configurations and control laws.

The ultimate benefits of this design tool are:

- mission oriented performance

- lower design cost

- better performance or cheaper hardware

- controller designs for complex systems.

MULTIVARIABLE FREQUENCY DOMAIN.   It is necessary to take a new look at the stability criteria when systems or CCV concepts are integrated and multiple control loop designs are implemented.   Vector frequency response or multiple variable frequency response is a generalization of classical gain and phase stability margins.   It is valid and meaningful for multiple loop systems [13-16].   This concept can be used to write specifications for integrated or coupled systems.   It can also be used in the design process to achieve the specification or improve the design.

DIGITAL CONTROL.   Digital implementation confronts the designer with a new problem - specifying digital variables.   In the past, the analyst has designed control laws to meet performance specifications, the systems engineer has put together control modes and switching and the circuit engineer has designed the hardware.   Interaction between these three functions was minimal.   In a digital implementation the analysts control law performance is dependent on hardware variables (wordlength and computer speed) and system or software variables (sample rate and computational delay).

Designing digital control laws can be accomplished several ways:

- digitize analog design

- direct digital design (classical or optimal)

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No matter what technique is used the issue remaining is what values of the digital variables are acceptable. The digital variables can be specified by digital analysis software. This software must compute performance and stability measures as a function of the digital system variables, i.e., sample rate, wordlength, computational delay or multiple values of these variables. The performance and stability measures include such things as pole and zero locations, gain and phase margins, rms and discrete responses and frequency response. Rapid analysis by this software can yield precise specifications [17].

Digital control also yields new capability like nonlinear control, adaptive control, long memory and tight tolerances. These new capabilities to date are largely unexploited.

IMPLEMENTATION NEEDS

The implementation needs that result from the active control system trends are increased system reliability, lower cost, and size and weight improvements:

- Increased system reliability - The safety of flight requirements of active controllers demand improvements in system reliability. This can be achieved through a combination of improved component reliability and through extensive and effective redundancy to achieve fault tolerance.

- Reduced cost - In order to realize the predicted improvements in performance it is important that implementation costs do not increase. Since the computational function required for active controllers are increased, the implementation cost per function must decrease.

- Size and weight improvements - Again the performance gains predicted through the use of active controllers can be maximized if the implementation size and weight of the controller can be reduced.

IMPLEMENTATION TECHNOLOGY TRENDS

There are a number of current developments and trends in digital system implementation technology that will contribute to satisfying the implementation needs discussed in the previous section. These trends and their anticipated impact are discussed below. This section deals exclusively with digital implementation technology since that is where the most significant gains can be expected.

LARGE SCALE INTEGRATED CIRCUITS. The availability of large scale integrated (LSI) circuits, in which hundreds of logic functions are implemented on a single chip, is allowing implementation of digital systems with improvements in both cost and reliability.

The major impact will be from standard (off the shelf) LSI which will be available in increasingly complex functional building blocks. In 1978, all of these will satisfy commercial specs, 25 percent will satisfy extended thermal specs, and 5 percent will satisfy mil specs. Some of these building blocks are:

- Memory modules - RAM's (Random Access Memories), ROM's (Read Only Memories), and EAROM's (Electrically Alterable ROM's) will be available at lower cost as the number of bits per chip increases. This not only impacts cost directly but it also allows more modular architecture to be used since small memories are economical.

- Programmable Logic Arrays - These are complex functional building blocks that are programmable to allow performance of a number of different logic functions.

- Microprocessors - LSI is making it possible to implement microcomputers at extremely low cost. More on microprocessor trends and their impact are discussed later in this section.

Custom LSI will also be available. In those cases where the function cannot be conveniently implemented with standard LSI, custom LSI can be justified for surprising low volumes. In reference 18 it is shown that for a volume of less than 100 systems, (the sample system consisted of 500 gates of random logic) it is cost effective now to use custom LSI as compared to standard Small Scale Integration/Medium Scale Integration (SSI/MSI) implementation. At a volume of 1000 systems, the cost advantage (including both recurring and non-recurring costs) is nearly an order of magnitude.

LSI, either standard or custom, or a combination of the two will provide major cost advantages now and increasingly so in the future.

The use of LSI implementation also provides advantages in terms of reliability, primarily because of a reduction both in the number of ICs and in the number of interconnects in the system. Reference 18 shows that in a typical digital subsystem of 500 gates of random logic, an order of magnitude reliability improvement can be realized with LSI implementation as compared to SSI/MSI implementation.

LSI implementation will also provide significant improvements in size and weight compared to the standard SSI/MSI implementation. The sample system of Reference 18 shows more than order of magnitude improvement.

MICROPROCESSOR TECHNOLOGY. The LSI technology has spawned a significant new technology called microprocessors. The microprocessor is defined as a standard programmable LSI which consists of a parallel arithmetic unit, a control unit, and a general purpose parallel data bus for memory and external device communications. This chip (or chip set) can be combined with LSI memory chips to realize a general purpose microcomputer for extremely low cost.

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These microprocessors (commercial spec) are now being manufactured in high volume by semiconductor vendors. It is projected that by 1978, 200K instruction per second microprocessor chips (or chip sets) can be purchased for from $15 to $25 each in volumes of 100 or more. Thus 8 bit and 16 bit microcomputers can be implemented for as little as $200 to $400. While extended spec and mil spec microprocessors will undoubtedly cost more, they will be available.

These microcomputers, having the programmability of conventional general purpose computers, will be used in two ways:

- to perform computational functions

- to replace hard-wired logic

In both applications a significant fact is that there is no longer a driving force to use the device efficiently. System level cost trade-offs tend to lead to dedicated use of these devices for certain functions even though this may result in, for example, the device being kept busy only 20 percent of the time. This leads to important new trade-offs in the area of system architecture, which is discussed next.

ARCHITECTURE. By system architecture we mean the overall organization or configuration of the building blocks of the system. The significant trend here that will help satisfy the implementation needs of advanced flight control systems is a trend towards more distributed systems. From a digital computer point of view, this means networks of minicomputers or microcomputers rather than the uni-processor architecture. The low cost of the computer modules will lead to dedication of a computer module to a specific function rather than time-sharing or multi-programming to allow a computer module to handle several functions. This is done primarily to reduce software costs, particularly the executive program.

Extensive research and advanced development activity is going on now on distributed computer systems with emphasis on bussing techniques and executive techniques [19]. The distributed computer approach has the following potential payoffs:

- Cost - Since there is only one computer building block in the system.

- Expandability - Since the bussing and executive will allow a variable number of computer modules to be present.

- Fault tolerance - Techniques are needed to provide backup if a module performing a critical function should fail. Much more work is required in this area but since all computer modules are identical, it holds promise of being able to satisfy fault tolerance requirements without high levels of redundancy. The use of the small dedicated building block allows redundancy to be applied to varying degrees throughout the system depending on the criticality of the function being

performed.   The interconnection mechanism is a critical resource in this system so special purpose hardware for fault tolerance may be required for it.

SOFTWARE.   A trend towards using a library of software modules which can be tailored and linked to fit the software requirements of a specific system will have important impacts on both system cost and reliability.   This is a significant change from current avionics software practice in which ad hoc techniques are used on a system by system basis, producing software that is both expensive and unique.   Being able to select and tailor already validated modules to satisfy a new requirement also contributes to reliability because validation and verification of the software will tend to be more complete.

A trend towards the use of higher order languages is an important companion of the library of modules trend in order for the library to be transferable from one computer to another.

SUMMARY.   The implementation needs of higher reliability and reduced costs appear to be achievable due to the following implementation technology trends:

- Large Scale Integrated Circuits can provide today order of magnitude advantages in cost, reliability, and size and weight compared to standard SSI implementations.

- Microprocessors are rapidly becoming available at extremely low costs. It is projected that 200K instruction per second microprocessor chip sets satisfying commercial specs will be available at $15 to $25 each.

- Distributed computer architecture consisting of a variable number of identical computer modules interconnected by busses are being developed. These architectures have potential advantages in hardware costs and in satisfying fault tolerance requirements.

- Software trends towards re-use of software through use of a library of modules will pay off in terms of both cost and reliability.

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THE IMPLEMENTATION OF FAIL-OPERATIVE FUNCTIONS IN

INTEGRATED DIGITAL AVIONICS SYSTEMS

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SUMMARY

System architectures which incorporate fail-operative flight guidance
functions within a total integrated avionics complex are described. It is
shown that the mixture of flight critical and non-flight critical functions
within a common computer complex is an efficient solution to the integration
of navigation, guidance, flight control, display and flight management. Inter-
facing subsystems retain autonomous capability to avoid vulnerability to total
avionics system shutdown as a result of only a few failures.

INTRODUCTION

The advent of the airborne digital computer in an attractive practical
configuration (from the standpoint of cost, size and power) has set the stage
for the emergence of a variety of new avionics system architectures. Despite
the continuing growth in requirements for navigation, guidance, control and
data management functions, the industry is faced with relentless pressures to
hold system costs to pre-1970 levels. We require increased system sophistica-
tion, but cannot afford increased cost or increased complexity and its con-
comitant reliability penalty. The solutions appear in new avionics architures
that feature a high level of integration and consolidation of functions. In-
deed, the trivial answer to any cost trade-off study of competing avionics
architectures is the totally integrated system where a single central computer
(of sufficient speed) performs all required functions so that the cost of
functional growth is measured only by the cost of the memory increment. This
solution does not acknowledge the complicating factors of flight critical fail-
operative requirements and the related problems of fault isolation and redun-
dnacy management.

The usual approach to defining a system architecture that must provide
some fail-operative functions is to separate subsystems into fail-operative
and non-fail-operative categories. In this paper it is shown that this type
of separation does not result in the most efficient mechanization of the de-
sired function. An alternative integrated system architecture that starts
with the requirements for the fail-operative autoland and stabilization and
control functions is described. It soon becomes apparent that the majority of
information interfaces needed for these fail-operative functions are also used

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for the other guidance, navigation, display and data management requirements. The system architecture and safety techniques used to mechanize the fail-operative requirements can be made completely compatible with the generally accepted methods of implementing the non-flight critical functions.

Expanding from the fail-operative flight guidance system, additional interfaces are added to achieve the remaining navigation, control and display functions. These additional functions are treated differently in terms of interface hardware and software mechanizations because the rather elaborate monitoring and fault isolation routines for fail-operative performance are not required.

The vulnerability of such integrated systems to the total loss of avionics functions with only two failures, such as the loss of two central computers, must be avoided. Consequently, the system architecture must make provision for continued although degraded operation through the retention of autonomous capability in the various interfacing subsystems. These back-up provisions generally appear as residual hardware functions in contrast to the software functions which are provided by the primary or central integrated mode of operation.

This paper presents a brief rationale for the selection of a totally integrated avionics architecture over two other competing candidates. The organization of the totally integrated system and the techniques for achieving fail-operative performance for flight critical modes are described. The vulnerability to total system shutdown is analyzed, and methods of protecting against that vulnerability are suggested. In general, the practical feasibility of such a totally integrated avionics system appears to be limited only by questions regarding the manageability of the system software.

SYMBOLS AND ABBREVIATIONS

θ	Pitch Attitude	Q	Dynamic Pressure
Φ	Roll	P_s	Static Pressure
ψ	Heading	P_T	Total Pressure $(P_T - P_s) = Q_c$
$F\theta$	Column Force	$P_F(t)$	Probability of failure in time duration t
$F\Phi$	Wheel Force		
A_x, A_y, A_z	Linear body axis accelerations in x, y, z direction	T_T	Total Temperature
		T_s	Static Air Temperature
h	Altitude	V_c	Calibrated Airspeed
M	Mach number	INS	Inertial Navigation System

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ILS	Instrument Landing System	MLS	Microwave Landing System
CWS	Control Wheel Steering	MFD	Multi-Function Display
DME	Distance Measuring Equipment		

RATIONALE FOR CANDIDATE SYSTEM ARCHITECTURE SELECTION

Three generic candiate avionics system architectures illustrate the requirements, considerations, and controversies surrounding the selection of an integrated avionics approach for future transport aircraft. These three candidates are:

1) The Federated System — a combination of new computers for each required class of functions. This is a direct extension of today's technology, but the argument is made that computers are becoming sufficiently inexpensive that we can afford the separate computers of the federated concept. This argument does not address the problem of intercomputer communication and interface complexity.

2) The Integrated System with Separate, Fail-Operative Flight Control Computers — a major acknowledgment of the need for integration but, nevertheless, it continues to duplicate the majority of sensor interfaces in order to separate the fail-operative guidance functions.

3) The Integrated System with Self-Contained, Fail-Operative Flight Control Functions — this system involves a minimum of interface duplication.

Trade-off analyses of these three configurations can be performed to prove any desired conclusion merely by applying the desired arbitrary weighting to one or more criteria of interest. Therefore, rather than perform a quantitative trade-off we will illustrate how a single parameter, "the interface complexity," varies with each of the candidate architectures. It is contended that interface complexity is the single most significant factor that influences cost, complexity and reliability of digital systems. When the computation and logic are performed in software, the largest hardware function is the acquisition and distribution of the data required by the computer. If we minimize the scope and complexity of that function, we create the simplest, least expensive and most reliable system. With this viewpoint in mind, we can compare the three candidates with reference to Figures 1, 2 and 3 which illustrate some of the typical interactive elements of the system requirements.

Figure 1, the federated combination of computers, is an extension of the 1970 state of the art where integration exists primarily to the extent of sharing sensor sources through relatively standardized interface mechanizations.

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The navigation computer, in this concept, is responsible only for area navigation, receiving navigation sensor and Inertial Navigation System (INS) inputs. The flight control computers retain their traditional autopilot and flight-director modes, including autoland; hence the triplex redundancy for the fail-operative requirement. Note that in all candidate systems, a separate flight control electronics function is shown in order to emphasize the fact that a considerable amount of electronics are required in addition to control law and logic computation. This electronics is associated with servo actuator drives, engage and shutdown controls, power conditioning for transducer excitations, and some signal conditioning. Dual, independent air data computers feed the navigation computers, the flight control computers, and dual EPR/autothrottle computers. Redundant navigation receivers representing the ILS function feed both the flight control (autoland) computers as well as the navigation computers.

This candidate is rejected because it represents the extrapolation of the traditional and presumably unsatisfactory approach to avionics. The problem of unwieldly interconnections and equipment growth is not adequately handled by this configuration. More interfaces are generated, and the number of black boxes grows, as we can readily see in Figure 1.

The second candidate (Figure 2) makes a reasonable attempt at integrating functions and minimizing black boxes and interfaces by using the navigation computer as the new integrating element. That computer complex incorporates all navigation, including air data computation and thrust management/autothrottle computations. It also includes flight path guidance computations other than those associated with autoland. The weakness of this approach is the use of three additional computers and their associated interfaces for the basic autopilot plus autoland guidance functions. The input interfaces required for the flight control computers are: VHF navigation receivers (ILS), air data (h, Q, $\dot{h}$, V_T), attitude and heading, radio altitude, accelerometers (A_z and A_y), and a considerable amount of mode selection logic. All of this information, with the possible exception of radio altitude, is also required in the navigation computer. Moreover, if provision is made for growth to MLS, then the MLS localizer, glide slope and DME will be required interfaces for both the flight control and the navigation computers. What then is the reason for also moving this information to a separate set of flight control computers? It can only be the edict that flight control functions are flight critical, as implied by the fail-operative requirements, while the other functions are not. Hence, if one assumes that fail-operative capability is achieved with a minimum of triplex redundancy, Candidate 2 is a natural conclusion.

The simplest interfacing of sensors is achieved with the third candidate (Figure 3). This system mechanizes the fail-operative autoland functions with two computers. These computers are shown interfacing with a triplex actuator control mechanization, although that interface could readily be quadruplex. Since the autoland architecture does not differ from the system architecture requirements of the non-flight critical navigation functions, those navigation functions are incorporated in the same computer complex. Triplex navigation functions are interfaced with both computers, as in the other candidates, but

only one set of interfaces is required. This interface reduction is representative of the significant minimization of electronics and wiring when this level of functional integration is implemented.

Candidate 3 is based on technology advances made in recent years where techniques have been developed that permit 100-percent fail-operative performance with dual digital computers. We define 100-percent fail-operative as follows: If the probability that the best contemporary triplex or quadruplex fail-operative system will respond properly to all failure situations is P_1, and the probability that the dual digital system will respond properly is P_2, then

$$P_2/P_1 \geqslant 1.0$$

In effect, this definition acknowledges that all fail-operative systems have loop-holes in such matters as multiple simultaneous failures, but the recommended dual system is at least as good as the best contemporary system in regard to fail-operative integrity.

If the fail-operative functions are mechanized in dual computers and will meet every stringent safety ground rule for Cat. III certification, why not use the same computers (using non-fail-operative techniques) for the other functions? When we follow this approach, the resultant configuration yields a major reduction in interface complexity and a significant reduction in the number of required black boxes.

SYSTEM ARCHITECTURE, REDUNDANCY AND SUMMARY OF FUNCTIONS

The recommended system organization is illustrated in Figure 4. The dual computational redundancy is represented by the pair of data adapters and computers. The autoland and stabilization and control autopilot functions that must be fail-operative are contained within the elements shown on this block diagram. Moving from left to right on the diagram, this is achieved through the use of appropriate redundancy in the required sensors, special hardware techniques within the data adapter, special software monitoring and data handling routines within the computer, and the necessary redundancy to interface the flight control electronics with the aircraft's electro-hydraulic actuation system. The number of flight control electronic units is shown as n where n may be three channels or four. Whether the control electronics is triplex or quadruplex depends upon the specific aircraft application and its servo actuator/control surface philosophy. All other non-fail-operative sensing and computational functions are performed without these special fail-operative techniques, although very thorough monitoring and fault isolation software routines are included for non-fail-operative as well as for the fail-operative functions.

A data adapter, a computer, and a flight data storage unit (mass storage) make up one computer complex. The data adapter is the computer's hardware

interface with the physical world. It isolates the computer from all problems of electronic mechanization so that the computer's only contribution to the system is contained within its software. The data adapter serves as a communications terminal for all data transfers, and as a data conditioning and data conversion center for its computer.

Each computer contains a program for performing all flight control, guidance, navigation, automatic flight planning, air data computation, engine EPR (thrust rating) computation, autothrottle controls and associated display functions. In regard to displays, CRT instruments are recommended for the ADI and HSI. The HSI function is implemented from a Multi-Function Display (MFD) which provides a moving map presentation (or, on pilot selection, a fixed map, moving aircraft display). The computer provides all the electronic map data processing; it receives continuous updates of data from the flight data storage unit, an air-bearing disk memory that provides mass storage of air navigation route logistic data. The computer also contains programs that allow it to perform an automatic central integrated test function that enhances the maintenance management of a major part of the aircraft's avionics equipment. It also presents checklist information on the MFD and includes interactive interfaces with the flight crew through pedestal mounted Control and Display Units (CDUs). These CDUs are normally used for automated flight plan selection and modification; however, their keyboard controls and associated alphanumeric readout (in conjunction with the large data display capability of the MFD), allow a convenient man-computer interface for checklist activity.

As shown in Figure 4, switching controls, activated automatically or by the crew, allow transferring of displays and sensor sources from left side to right side, and vice-versa.

SENSOR SUMMARY

The sensor requirements are covered as general categories in Figure 4. A list of the sensor complement and a discussion of redundancy requirements follows. _In the category of stabilization and control, sensors are:_

- CWS Force Sensors ($F\theta$, $F\Phi$)
- Yaw Rate* (r)
- Pitch and roll Attitude* (θ, Φ)
- Heading ** (ψ)

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*It is recommended that pitch and roll rates be obtained as software-derived rates from the attitude data.

**Heading data free of gimbal errors is desirable because this information is used for coordinate transformations during turning maneuvers in those configurations which are not provided with INS. If ψ is obtained from a conventional 2-degree-of-freedom directional gyro, then a gimbal error correction algorithm is incorporated in the system software.

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- Linear Acceleration Triad (A_x, A_y, A_z)

- Flap Position

- Surface Position

The Air Data Sensors are:

- Static Pressure (P_s)

- Total Pressure (P_T)

- Total Temperature (T_T)

(Note that angle of attack (α) may be computed from inertial and baro data.)

An inertial navigator is shown, although for the configurations that do not include an INS, provision is made for inertial smoothing of radio navigation data, using strapdown accelerometers, plus attitude and heading references. When the INS is provided, its velocity-north and velocity-east information is used as the basis of the smoothing algorithm, and the short-term strapdown inertial computations are not needed. The radio NAVAIDS are:

- VOR

- DME

- ILS

although provision is included in the data adapter for interfacing with the future MLS system and hyperbolic radio navigation systems such as OMEGA.

The radio altimeter is required only for the autoland and instrument approach functions. Engine EPR is needed for the autothrottle EPR mode, and throttle servo rate is needed because the throttle servo loop is closed through computer software.

Redundancy of sensors where fail-operative capability is required is approached by using the three techniques illustrated in Figure 5. The first (Figure 5a) feeds each sensor into each of the dual computing channels. A voting, middle-value selection or averaging algorithm is mechanized in the computer software to ensure that both channel 1 and channel 2 use the same estimate of the sensed parameter. Intercomputer communications, via buffered serial data links, inform each computation channel of the estimated value, (Â, B̂, Ĉ), and whether a sensor discrepancy or anomaly has been detected. The technique of Figure 5a is the most efficient from the standpoint of sensor equipment minimization, least efficient from the standpoint of interface complexity (and wiring), and somewhat more complex in regard to software complexity when compared to the other candidate sensor configurations.

The second technique (Figure 5b) uses quadruplex sensors arranged in pairs. As in the first case, software voting and averaging are used to isolate faults and equalize the estimates in both channels. The third arrangement (Figure 5c) uses internally monitored sensors that generate their own valids to indicate that the data is usable. The serial data exchanges allow channel equalization. When this method is used, appropriate interfacing techniques are employed to avoid the situation where the valid is received, but the data is lost through an open connector pin.

There are many factors which enter into the selection of configuration 5a, 5b, or 5c for a specific sensor. Some of the considerations are logistic. For example, two sets of dual sensors (5b) may be easier to maintain than three individual sensors (5a). Other factors involve safety guidelines and allowable probability that a failure may be undetected. For example, configuration 5c assumes a self-monitored sensor. Modern radio altimeters fall into this category, but it may be argued that the built in sensor monitoring is not 100 percent effective and a finite probability may exist for an undetected radio altimeter failure in the final phases of an autoland approach. We may respond to a stringent safety guideline regarding radio altimeters by adding a third sensor and using the configuration (5a) approach. However, it can be shown that the validity determination for a given sensor may be augmented within the system's monitoring software where state estimations from other types of sensors may be used to verify a given sensor signal. Thus, for example, a radio altimeter signal may be analyzed with regard to its validity by means of comparisons with baro-inertial estimates of the aircraft's vertical velocity. Hence the 5c sensor configuration may be justified over the 5a configuration.

MONITORING CONCEPT FOR DUAL-FAIL-OPERATIVE FLIGHT GUIDANCE FUNCTIONS

Summary

The two halves of the total, fail-operative Digital Flight Guidance System are designated as channel 1 and channel 2 (Figure 6). Channel 1 has a dual internal structure with the two parts designated as channels A and B. Channel 2's subchannels are also designated as A and B. Both channel 1 and channel 2 are autonomous of each other, and each is capable of operating as a fully monitored fail-passive system. Each channel is designed to detect any discrepancy from normal operation and activate safe shut-down controls if the discrepancy is deemed to constitute a system failure.

There are several different monitoring techniques used to achieve 100-percent failure detection in each computer channel. Unlike analog systems, however, we cannot identify a unique set of malfunctions with each type of monitor. There are very large overlaps in the fault detection routines. Four different monitoring algorithms, for example, may detect one failure. In some cases this overlap is exploited to permit partial shutdowns, and in other cases

only a total channel shutdown is permited. The following is a summary of the types of fault detection techniques that are employed:

- Processing of sensor valid discretes

- Sensor data validity and reasonableness checking algorithms

- Sensor data comparison monitoring — variable thresholds dependent upon aircraft state, signal amplitude and signal duration

- Redundant computations internal to the computer using separate computer memory banks and comparison checks of results

- End around I/O checking — all outputs are fed back to the computer via the input conversion sections and verified against the specified output

- Test words continuously checked for all intrasystem communications

- Model and comparison monitoring of servo actuator responses

- Software executive continuously verifies that the required sequence of software tasks is accomplished each 50 millisecond iteration period

- External (to computer), dual hardware monitors examine the computer's output for a required dynamic signal pattern — any computer failure that will prevent the execution of the specified program will cause the pattern to cease.

In addition to the monitoring algorithms, all input signal data are processed so that all redundant control law computations are performed with identical values for all variables. Hence all control output commands must be identical. The servo actuator commands are therefore identical so that servo system monitoring criteria are dependent only upon servo system tolerance. Some cross-channel (between channel 1 and 2) computation equalization is needed, but the amplitude constraint on the amount of equalization is a small percent of the control authority. Cross-channel equalization is needed to correct for small offsets caused by an occasional 50-millisecond time skew between data used in channel 1 and channel 2.

Computer Executive and Hardware Monitor

Descriptions of the input signal screening, monitoring and equalization algorithms are beyond the scope of this paper. The necessary system concepts can be appreciated as extrapolations and improvements over techniques used in contemporary analog systems. However, some additional comment is needed to elaborate on the concept of a 100 percent, self-monitored computer. A computer system verification function is used to generate a prescribed output signal pattern at the end of each iteration cycle only if a checklist of required

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computation routines has been completely satisfied. The instructions for checking off this list are therefore interwoven throughout the entire program so that if any of the required routines is not properly completed, or if a processor function is faulty, the verification signal pattern will not be properly generated. This verification signal is D/A converted and transmitted to the hardware monitor in the Data Adapter where it is compared with a correct signal pattern. A difference in these signals will cause the computer complex to shut down safely (without servo command transients). Since the verification signal is dynamic and must contain correct timing information to be valid, a failure in the verification signal path to the hardware monitor (such as an open or a hardover) will be detected, as well as timing errors in the computer. The computer system verification function serves principally to detect massive computer failures, and does not allow shutdown of partial computation functions as is possible with the software monitoring functions. Nevertheless, there is a very intimate relationship between the software and hardware monitoring functions. This is shown in a simplified representation in Figure 7. In this figure the concept of an executive program which generates a task list as a function of the status logic is illustrated. With the completion of each of its specified tasks, the program acknowledges that it is ready for the next task by setting a task-completion bit. When the real-time interrupt that controls the program iteration rate occurs, a check is made to determine whether all required tasks were completed. If they were not, the computer software recognizes a computation failure and jumps to a failure response routine. It simultaneously neglects to generate the correct output pattern. In this case both the software and hardware monitors will detect a failure, but the hardware monitor will require a few cycles of incorrect output before it will respond. For simplicity, an output pattern in the form of a 10 Hz square wave is illustrated by Figure 7. In practice, more complex, multilevel patterns have been used.

Failures of the digital computer's logic circuitry associated with the execution of specific instructions will result in the condition just described. The airborne program incorporates techniques which deliberately exercise the instruction repertoire so that failures in repertoire logic will cause the program sequence to get lost — that is, the program is forced to a wrong address. The result is a program hang-up or loop where it never reaches completion of the specified tasks. The program will recognize the real-time interrupt, and the machine may be capable of executing shutdown instructions. However, a more fundamental computer failure, such as loss of clock or memory read-write circuitry, will leave the computer in a state where it cannot execute any instructions. In that case, the hardware monitor will detect a fixed state on output D rather than the required dynamic pattern on output D of the figure. It will thereby initiate a system shutdown by commanding a computer power-down and interruption of power to D/A output commands. As mentioned earlier, some dual computation paths are also used within the computer primarily to detect failures associated with single-bit malfunctions in storage of data words.

BACKUP CONCEPTS AND RELIABILITY IMPLICATIONS

Summary of Display/Control Functions

A complete description of the cockpit displays and controls and their interfaces with the redundant computer complex is beyond the scope of this paper. However, it is essential that the software-controlled functions be identified so that we can devise an appropriate back-up strategy for the remote possibility of a total computer shutdown.

Referring to the highly schematic cockpit layout shown in Figure 8, consider normal system operation with computer complex No. 1 driving the left set of displays, and computer complex No. 2 driving the right set of displays. The computer/display interconnection may be switched, either automatically in response to failure detections, or manually by pilot selection. The primary flight displays are:

Multifunction Display

The MFDs primary use is to serve as an HSI incorporating a moving-map display. In this configuration, it provides the HSI pictorial representation of the flight situation with regard to course, course deviation, distance to destination and heading. The reference path is drawn as a solid line connecting waypoints. Projecting from the aircraft symbol is a trend vector depicting the aircraft's predicted location up to a software selectable time into the future. Behind the aircraft is a sequence of dots representing the previous position history. Waypoints, airports, airways, landmarks, VORTAC, VOR, VOR/DME stations are normally displayed on the map. The heading tape is at the top, with a digital readout of aircraft heading. Scale factor selection is provided on the MFD control panel located to the right of the MFD. Scales of 1, 5, 20 and 80 nautical miles-per-inch are provided, but these values are obviously completely under software control. When the landing area is reached, if the scale factor is reduced to 1.0 nautical mile-per-inch, then a runway symbol appears, and a useful presentation in the MLS era when accurate terminal DME and wide-angle azimuth to the landing area is available. The MLS accuracy would permit the use of the fine scale map so that navigation accuracy is consistent with map resolution.

On the left side of the MFD display area, various parameters associated with flight plan progress and 4-D guidance (arrival time) status are presented as alphanumeric readouts.

The map is also displayable in the north-up mode (moving aircraft fixed-map display) upon selection at the MFD control panel. Slewing controls move the map up-down and left-right, with the aircraft symbol remaining fixed at its true location on the map. Mode selection at the MFD control panel permits pilot editing of the map content. Other mode-select buttons delete the map and allow the display to list pages of data, such as that associated with route planning or preflight checklists.

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Electronic Attitude-Director Indicator

This display presents the basic horizon presentation via instrument interfaces that are completely autonomous of the computer system (not under software control). Also independent of software is a digital radio altitude readout in the upper right window. Indicated airspeed appears in a window at the upper left of the screen, and the system software provides a choice of which parameter one can display in the window at the upper center of the screen. Experimental work has been done where this window was used to display distance to touchdown (during final approach) in nearest .1 nautical mile, or vertical speed in feet-per-minute.

Other information displayed and retractable (figuratively) under software control is listed:

- ILS or Flight Path Window

  Raw data deviation from the ILS flight path or computed position error from area navigation flight paths.

- Flight Path Angle Symbol

- Flight Path Acceleration

- Flight Director Command Bars

- Fast-Slow Indication

- Perspective Runway Symbol (This presentation is used when accurate DME information to the landing site is available, as in MLS systems.)

On the right bezel of the EADI is a set of approach progress annunciators. Modes that are armed illuminate amber, and when engaged they illuminate green.

Radio Altitude, Altitude, Vertical Speed, Airspeed/Mach

These indicators are clustered around the ADI in the conventional manner.

Autopilot Flight Director System Mode Annunciator

The mode annunciator is an electronic display containing four alphanumeric readouts that present the autothrottle mode, vertical guidance mode, lateral guidance mode, and autoland mode. These readouts flash if the mode is being captured, and illuminate steady when the mode is in a "track" phase.

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Instrument AFCS/Warning Display

The instrument /AFCS warning display panel provides for annunciation of subsystem failures. A unit is located in the primary viewing area on each side of the instrument panel.

Dual Digital DME and Radio Magnetic Indicator

To the left of the MFD is a basic RMI indicator that has direct interface with the radio receivers and the heading reference systems in order to display bearing to VOR or ADF stations. It also provides dual digital DME readouts through direct digital interfaces with the DME receivers.

ATS/EPR Control Display Panel

This panel, located at the bottom of the center instrument panel, serves as the thrust-rating readout and thrust-mode selector. It also provides the means of engaging the dual autothrottle servos. By selecting either the take-off, maximum continuous, climb, cruise or go-around mode, the computed EPR limit for those modes is displayed in conjunction with the total air temperature. This instrument may also be used to display total and static air temperature and true airspeed.

Mode Select Panel

The Mode Select Panel (MSP) located in the glare-shielded region provides the following control and display capability:

- Dual VHF Nav Receiver frequency readouts (for display of an automatically tuned station) or manual tuning override capability -- located on left and right side of MSP.

- Speed Control mode select and reference readout (airspeed and Mach via pitch or autothrottle control).

- Vertical Guidance mode select and reference readouts. These include flight path angle and/or vertical speed and altitude pre-select displays and controls.

- Autopilot and Flight Director Engage Switches, including flight critical engage switches, turbulence mode control and engage controls for autoland, take-off and go-around.

- Lateral Guidance mode select and reference read-outs. These include heading and course set controls and display redundant navigation sources, plus means for selecting various navigation guidance modes and displays.

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Dual Control/Display Units (CDUs)

Dual Control/Display Units (CDUs) are shown on the left and right side of the pedestal. These CDUs are normally used for automatic flight plan selection and modification. However, their general purpose keyboard controls and associated alphanumeric readout (in conjunction with the large data display capability of the MFD), allows a convenient man-computer interface for checklist activity.

Backup Concepts

The integrated system has many of the same reliability hazards as contemporary systems. If all attitude references fail in flight, many of the system functions and modes are disabled. If all of the NAV receivers fail, a different set of functions and modes are disabled. The superior fault isolation and failure assessment capability of the integrated system allows automatic reconfiguring of the navigation and guidance functions into alternate or degraded modes. The crew can also participate in the reconfiguring of the system data flow and displays through control of instrument switching. The fewer black boxes and the improved failure detection, isolation and annunciation capability results in a significant improvement of overall avionics reliability and utility. There is, however, one potential weakness that disturbs the critics of avionics integration. They cite the possibility of losing all avionics functions as a consequence of losing one or two system elements. This criticism must be addressed, and the recommended approach must be justified in terms of system operational capability in all failure situations as well as with quantitative reliability analyses that show overall MTBF improvement.

First it must be emphatically stressed that most failures, including multiple failures in redundant channels, do not wipe out the system. Three questions must be answered. They are:

- What failures can wipe out the system?

- What is the probability of such an occurrence?

- What are the backup provisions in the event of such a failure occurrence?

The answer to the first question is that the loss of both computer complexes (Computer and Data Adapter) will disable the entire system. The projected MTBFs of the computer and data adapter are 5000 hours each. Considering that only one half of single data adapter failures are totally disabling, the probability of total system loss in a 3-hour flight, $P_T(t) = P_F(3)$ is

$$P_F \text{ (3 hours)} = .81 \times 10^{-6}$$

Making allowances for combinations of other multiple failures which would contribute to a total system disability, it can be stated that the probability

of total system shutdown in a 3-hour flight is about 10^{-6}. Suppose we are being overly optimistic on the projected MTBF and we only achieve one-half the MTBF values specified. Then the $P_F(3)$ rises to 3.24×10^{-6}, or, making provision for other disabling failures, the probability of total system shutdown in a 3-hour flight is about 4×10^{-6} (or four shutdowns per million flights).

The response to the third question shows that the backup provisions are sufficient to allow continued instrument flight (although not to a Cat. II level). The following is a summary of these backup provisions:

- Both EADIs present horizon displays independent of the computers, and the attitude references are manually selectable from alternate sources.

- Both DDRMIs present ADF or VOR bearing (selectable) and aircraft heading from selectable data sources. The VOR radials are selected through the Mode Select Panel course-select knobs which contain course-reference synchros.

- Provision can be made for a direct interface between the heading references and the NAV receivers and the MFD so that a course line pointing to the azimuth scale would represent the desired flight path (localizer or VOR radial). The aircraft symbol would be displaced from the course line by the course-deviation signal. Thus the MFD reverts to a residual HSI through the use of direct, hard-wired interfaces to the required sensors.

- Manual tuning of NAV receivers is independent of the computer system. DME data to two stations is coupled directly from the DME receivers to the DME readouts on the DDRMI instruments.

- Both EADIs present radio altitude independent of the computer system. Also, the radio altimeter display is independent of the computer system.

- Raw data ILS (localizer and glide slope deviation) is presented on the EADIs' ILS window symbol. Course deviation from VOR radials can also be presented on this display if a course resolver is incorporated in the course-set controller on the MSP.

- Pneumatic altimeters, airspeed indicators and vertical speed indicators may be located on the center instrument panel. A self-contained horizon instrument may also be located on this panel. Another means of providing backup air data would be the use of a low cost, mini-air data computer having only three outputs: altitude, altitude rate, and airspeed. These three outputs can be encoded to provide the word stream needed to drive all air data instruments, following the selection of the backup air data by an appropriate instrument switching arrangement. The backup air data would also provide the required encoding for the aircraft's altitude-reporting function.

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• A backup, redundant, hardware yaw damper (with somewhat degraded capability) is included in the flight control electronics. That yaw damper function is independent of the computer system.

This leads to a final observation regarding logistical problems, and a very significant departure from contemporary practice. It would appear that the consolidation of several flight-critical functions within an integrated system would necessitate the requirement that two computer complexes be designated as reliability "dispatch items" by an operating airline. The provisioning of spares on a short-haul route structure would be resisted by airline maintenance policies. Perhaps the minimization of the total number of black boxes would permit the carrying of the spares aboard the aircraft. With advanced fault isolation and maintenance-management techniques inherent in a sophisticated digital system, it might even be possible to consider in-flight repairs using the on-board spares.

SOFTWARE SUMMARY AND CONCLUDING COMMENTS

The system design is organized into a software module grouping with a master executive program that integrates these various modular routines and performs such tasks as timing, system reconfiguring, backup algorithm selection, and part of the monitoring functions. A list of software modules, the estimated time per iteration in an advanced Sperry computer, typical iteration rate requirements and memory storage estimates are given in Table I. The advanced Sperry computer (designated RMM-1) was designed for application in the post-1975 era, and has some extremely high speed and architecture innovations. Add/subtract times range from 350 to 700 nanoseconds and multiply times, including memory access ranges from 1.15 microseconds to 4.2 microseconds (for a floating point multiply). That computer would be provided with a 32K plated wire NDRO memory for this application, but Table I shows that the memory budget is only 17,800 words (not including the integrated test and pre-flight checklist which would be contained in the mass memory [disk] and transferred to the computer resident memory when required). The mass storage requirement is estimated as 8×10^{-6} bits for worldwide logistic data, or 1×10^{-6} bits for regional data only. The disk capability is 10×10^{-6} bits.

A perusal of Table I shows that the advanced computer would be working at less than 10 percent of its available time to complete the entire computation task. An estimate of the computer load using a more contemporary 1974 machine (Sperry 1819B) indicates that the entire task could be done in 70 percent of that machine's available time with memory (main store) consumption of about 26K words. Thus there do not appear to be any serious questions regarding whether the state of the art in avionics can meet the requirements of this type of system. One nagging question persists. Is the software manageable? That is, can such a software system that encompasses so broad a scope of functions, technical disciplines and organizational responsibilities be developed, verified and configuration-controlled in a typical transport aircraft development environment? Fortunately for the author, that question is easily dodged.

The answer is no, if traditional approaches and relationships between participating parties (airframe manufacturers, avionics equipment manufacturers and airlines) are maintained. However, even those digital system pioneers who have survived to regret slogans such as "there are no problems because it's all in the software," will optimistically answer yes if the development environment and responsibilities can be properly disciplined. There is pessimism, however, that industry can achieve that organization and discipline in the near future.
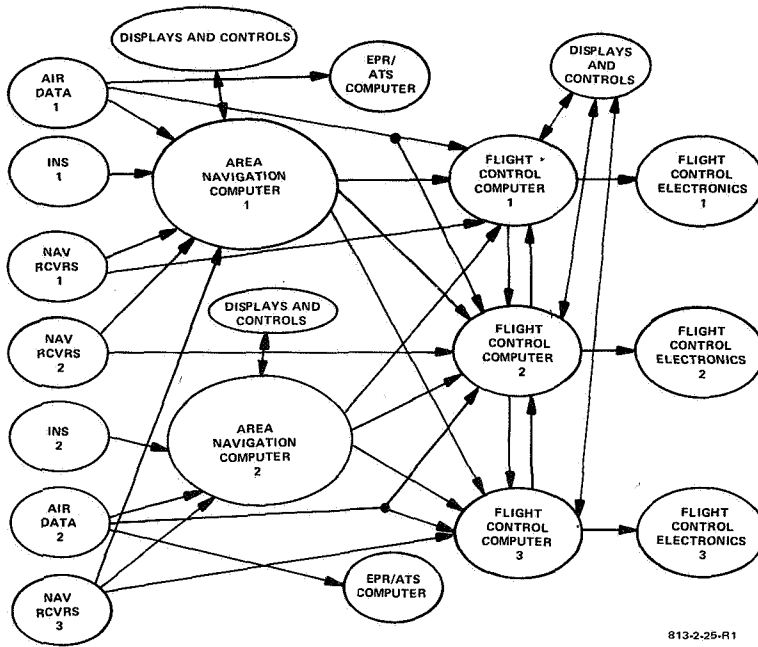
813-2-25-R1

Figure 1
Candidate 1, Federated Computer System
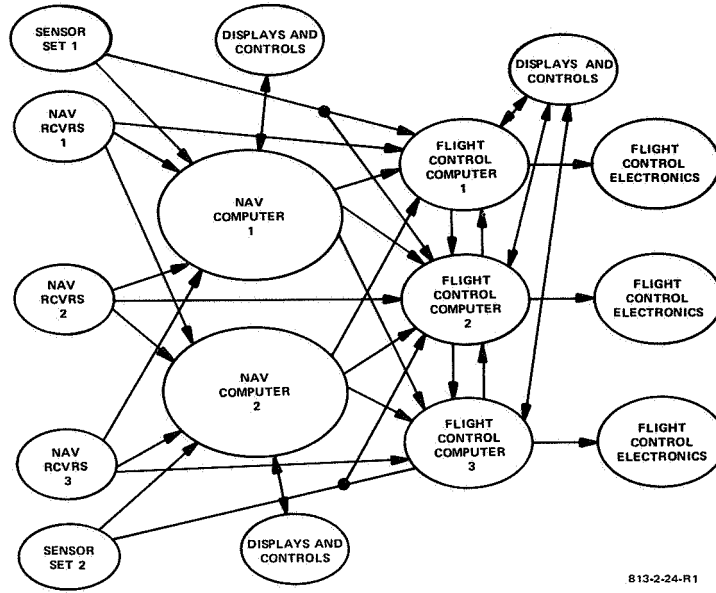
Figure 2
Candidate 2, Dual Navigation Computerization Separate
Fail-Operative Flight Control Computer
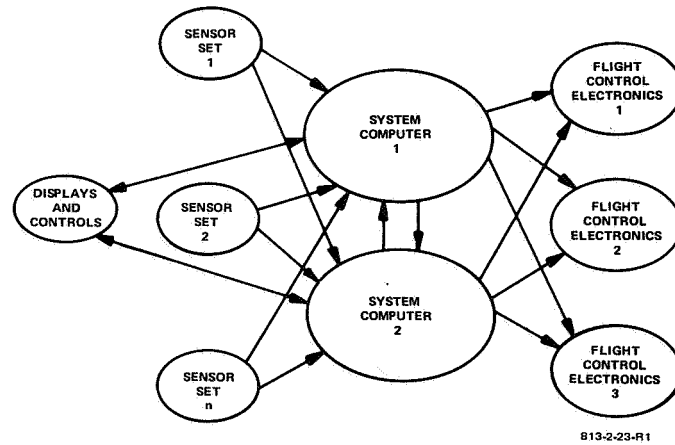
Figure 3
Candidate 3, Integrated Dual Fail-Operative System
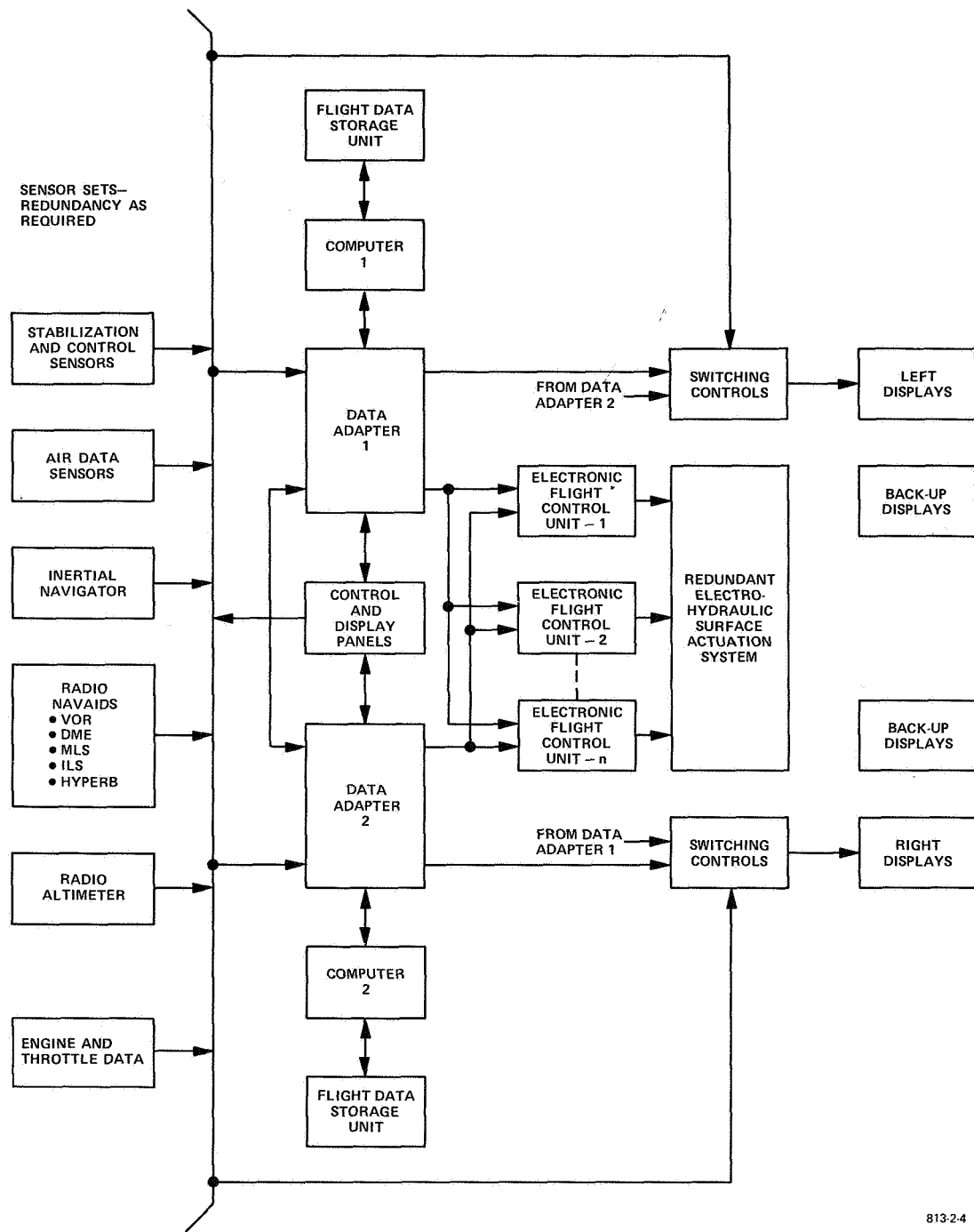
Figure 4
Redundancy Architecture

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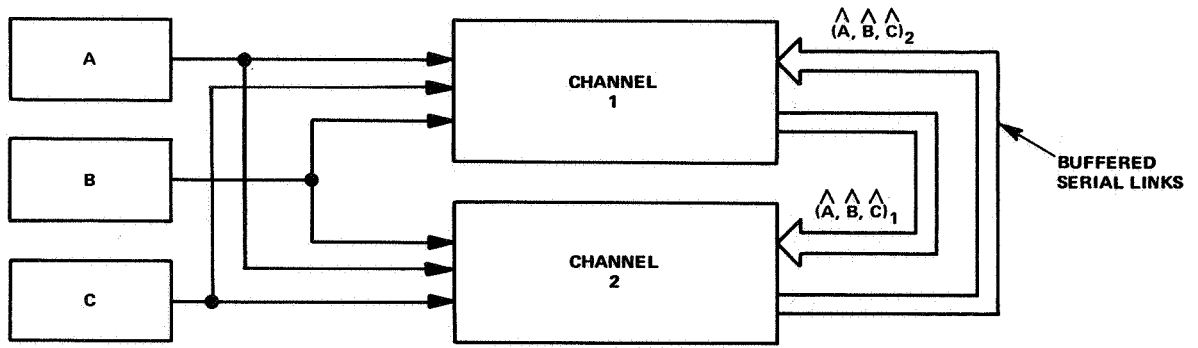
(a) TRIPLEX — SOFTWARE VOTING
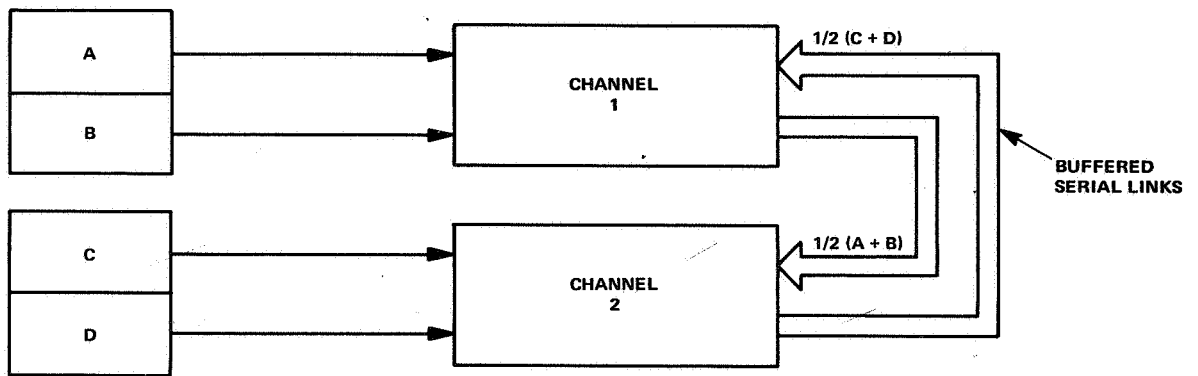
(b) QUADRUPLEX — SOFTWARE VOTING
AND AVERAGING
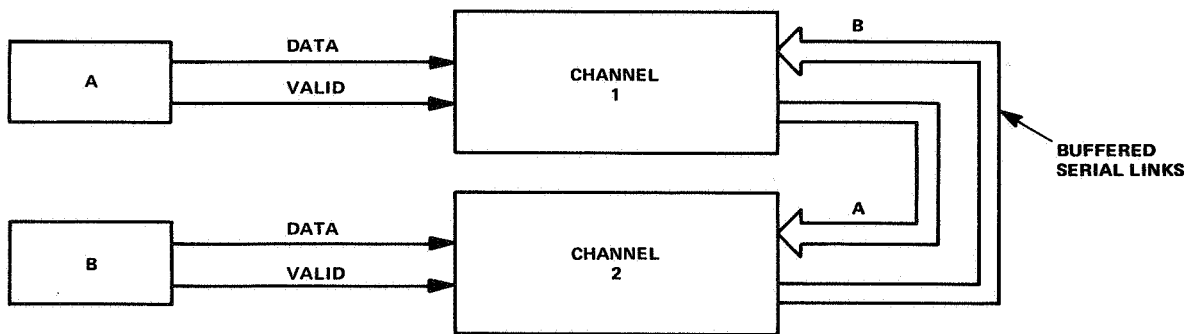
(c) DUAL IN-LINE MONITORING
WITH SOFTWARE AVERAGING

813-2-6

Figure 5
Redundancy Schemes for Sensors
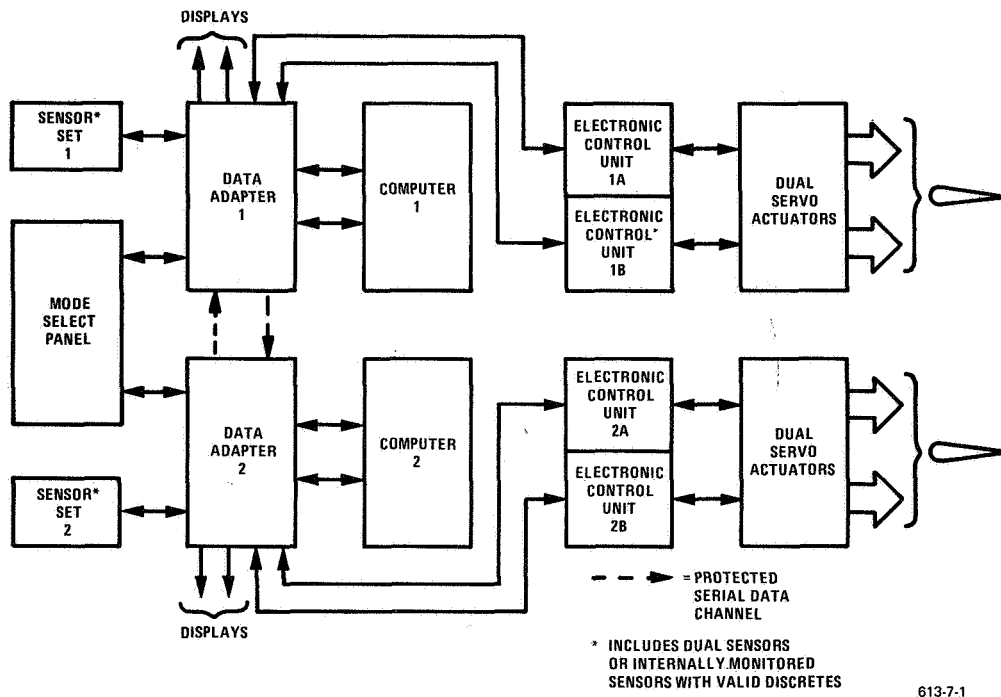
Figure 6
Dual Fail-Operative System Architecture
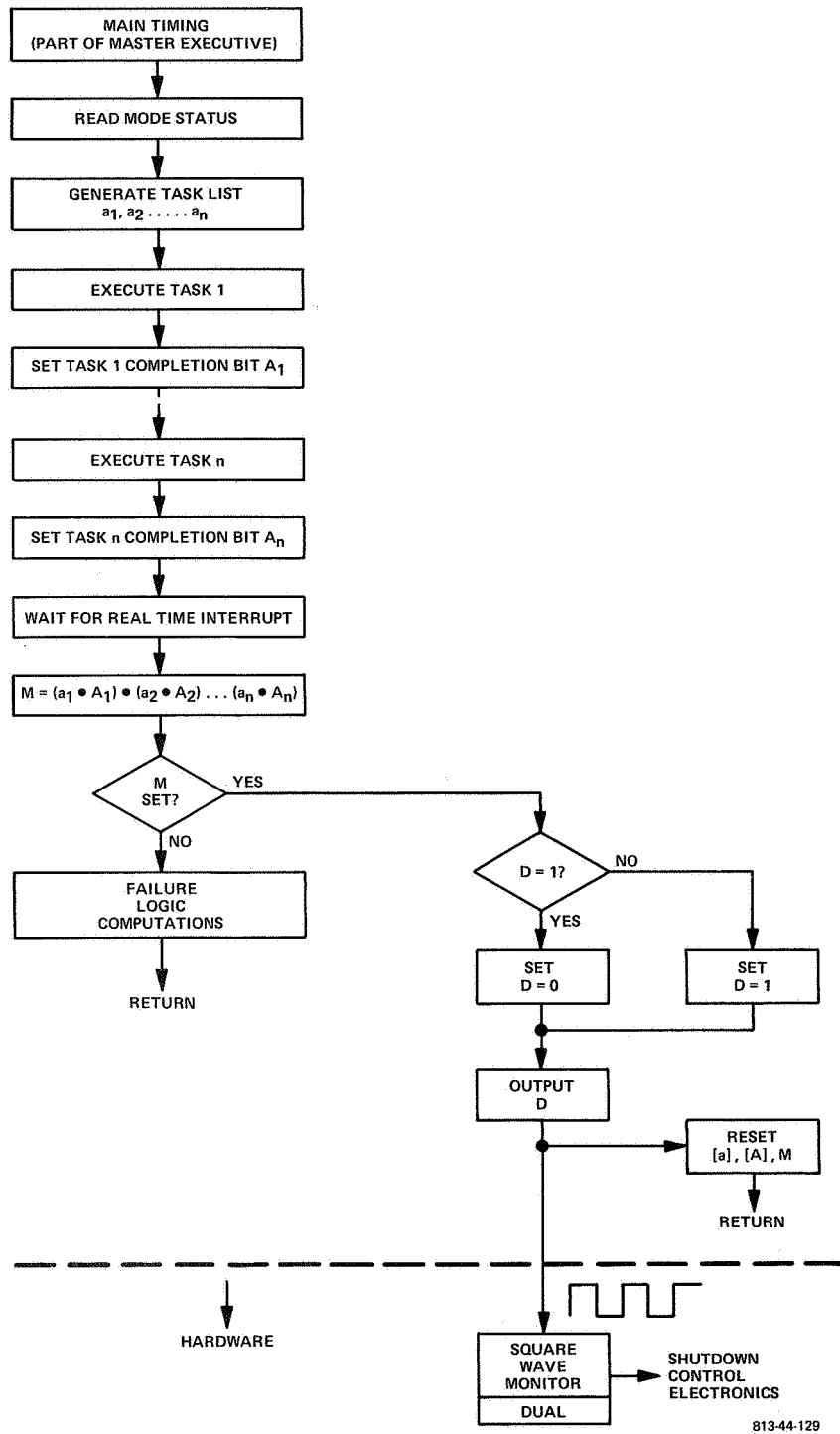
Figure 7
Relationship Between Software Executive Monitor
and Hardware Monitor

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MODE ANNUNCIATOR　　　MODE ANNUNCIATOR

AFCS AND INST. WARNING　　　MODE SELECT PANEL　　　AFCS AND INST. WARNING

SURFACE DISPLAY　A/S MACH　EADI　RAD ALT　ALTIM

ENGINE INSTRUMENT DISPLAYS

A/S MACH　EADI　RAD ALT　ALTIM　SURFACE DISPLAY

FLAP DISPLAY　CLOCK　VERT SPEED

CLOCK　VERT SPEED　FLAP DISPLAY

DDRMI　MFD

DDRMI　MFD

MFD CONTROL PANEL　CDU 1　THRUST RATING AND ATS ENG　CDU 2　MFD CONTROL PANEL
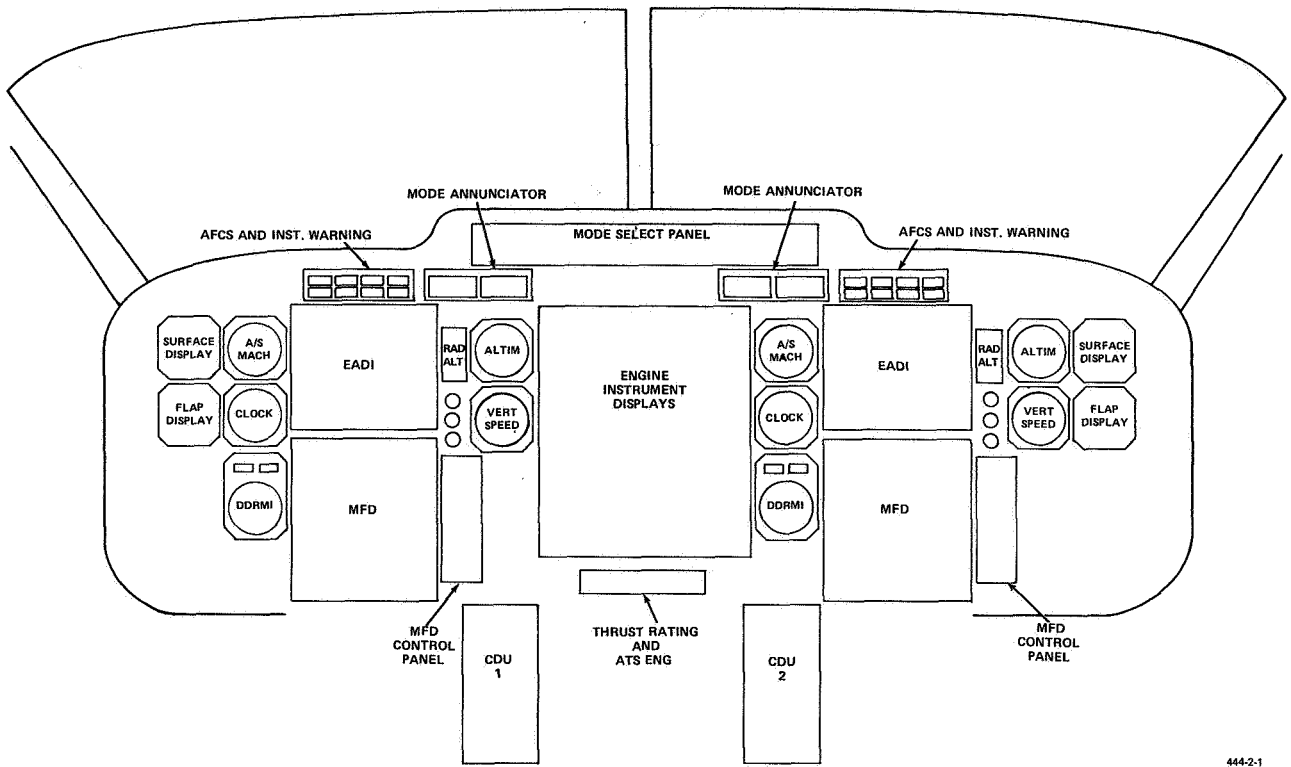
444-2-1

Figure 8
Schematic of Cockpit Display and Control Layout

TABLE I

COMPUTER REQUIREMENTS SUMMARY
(BASED ON RMM-1 COMPUTER)

Function	Typical Time Per Iteration (μ sec)	Required Iteration Rate (per sec)	Memory Storage Requirement (words)
Master Executive	100	1 to 20	1,000
Autopilot/Flight Director Guidance and Stabilization ● Attitude Stabilization ● CWS ● Vertical Guidance ● Lateral Guidance ● Autoland ● Interlocks and Mode Logic ● Panel Communication ● Basic Monitoring	2000	20	5,700
Special Fail-Operative Routines	50 to 700	20	2,000
Navigation ● ρ, θ Nav from Navaids ● Remote Tuning ● State Estimation (filtering) ● Flight Planning (Waypoint Data Processing, Updating, CDU Communication	400	1 to 20	4,000
Air Data Computation h, $\dot{h}$, V_c, V_T, M, T_T, T_s, Q_c, P_s	175	20	800

TABLE I (cont)

COMPUTER REQUIREMENTS SUMMARY
(BASED ON RMM-1 COMPUTER)

Function	Typical Time Per Iteration (μ sec)	Required Iteration Rate (per sec)	Memory Storage Requirement (words)
Autothrottle/Speed Command and Stall Warning (includes α computation)	200	10 to 20	400
EPR/Thrust Rating Computation	125	1 to 5	900
MFD • Communications and Formatting • Map Processing	2,000	1 to 20	3,000
Integrated Test and Preflight Checklist	–	–	4,000 (Resident in mass storage)
Air Navigation Logistic Data • Worldwide • Regional Only	– –	– –	500,000 62,500 (Resident in mass storage)

A FORWARD VIEW ON RELIABLE COMPUTERS FOR FLIGHT CONTROL*

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SUMMARY

We examine the requirements for fault-tolerant computers for flight
control of commercial aircraft and conclude that the reliability require-
ments far exceed those typically quoted for space missions. Examination
of circuit technology and alternative computer architectures indicates
that the desired reliability can be achieved with several different computer
structures, though there are obvious advantages to those that are more
economic, more reliable, and, very importantly, more certifiable as to fault
tolerance. Progress in this field is expected to bring about better computer
systems that are more rigorously designed and analyzed even though computa-
tional requirements are expected to increase significantly.

INTRODUCTION

Current NASA developments in aircraft and aviation systems design require
a great increase in on-board computing. Most of the advanced aircraft
designs--e.g., configuration-controlled vehicles, and certain STOL modes--
require extremely reliable computations. NASA must therefore be assured
that it will be possible to build computing systems having the high capacity
and extreme reliability that its current advanced aircraft designs will
require.

The reliability requirements far exceed those typically quoted for space
missions (95% success after five years). This implies that the probability
of error of spaceborne computers is designed to be on the order of 10^{-6}/hr
for long missions while the acceptable figure for advanced avionic systems
for the commercial environment is on the order of 10^{-9}/hr for short missions.
The commercial environment also has different certification requirements,
not only because of the high public demand for safety, but because the users
are more diversified. Thus the hardware and software components of a computer
for commercial avionics must not only satisfy the reliability criteria of
computer designers, but the reliability must be convincingly demonstrated to
aircraft system designers and users. It is well understood that computers
of the needed power will require a large number of components, and that
this number is so large ($>10^4$) and the assured reliability is so low ($>10^{-6}$

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failures/hr) that some form of built-in fault tolerance is essential. Unfortunately, the simpler forms of fault tolerance (e.g., error correcting codes and triple modular redundancy) are inadequate for computers of the required size.

Realization of this inadequacy has given rise to several research and development efforts in the design of automatically reconfigurable computers. Some examples of computers carried to a fairly detailed design level are STAR (JPL) [ref. 1], EXAM (NASA-ERC) [ref. 2], ARMMS (NASA-Marshall) [ref. 3], and MSC (SAMSO) [ref. 4]. Other recent designs, at a less-detailed level, include SIFT (NASA-Langley) [refs. 5 and 6], an unnamed computer, hereafter called HS (MIT C. S. Draper Laboratory) [refs. 7 through 9]. There has also been considerable research in techniques for designing redundant logic networks and memories, for testing arbitrary logic networks, and for modelling redundant systems. For a discussion of these topics, see reference 10.

These design and technique studies comprise a well-rounded, but relatively unproven art. They do not yet comprise a base of technological practice sufficient for the design of computer systems whose reliability can be specified with a high degree of assurance. This is a consequence of the basic fact that (1) faults and errors can occur in extremely varied ways, and (2) the fault-tolerant behavior of an automatically reconfigurable computer can be extremely complex.

Subsequent sections of this paper examine the computational and reliability requirements, the technology constraints, and estimates of the likelihood of achieving the goals.

COMPUTATIONAL REQUIREMENTS

In this section we consider the computational and reliability requirements of a representative aircraft computer system. The example we choose is that of a commercial transonic four-engine aircraft. We assume that advanced control systems will be required for such functions as flutter control and attitude control. We further assume that an advanced blind landing system would be used.

The requirements are reported in detail in reference 6, and are summarized in table 1. The most critical phase of the flight from a computational standpoint is during an instrument landing. Those applications involved in that phase are indicated with an "#". Small tasks that are not required during that phase do not influence the design of the computer system and therefore have not been estimated to the same accuracy as the more important tasks.

The column headings of table 1 are defined as follows:

Task----------------The name given to the application program.

Criticality Class----1. Immediate safety-of-flight impact.
                     2. Eventual safety-of-flight impact.
                     3. Significant change-of-mission impact.
                     4. Operational impact.
                     5. Economic impact.

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Table 1

COMPUTING REQUIREMENTS FOR EACH COMPUTATIONAL FUNCTION

Task	Criticality Class	Iteration Rate(s)/ Sec	Equivalent MIPS	Memory Required		Missed Iterations
				Inst.	Data	
Attitude control	1	5,20	0.023	1845	230	2-3
Flutter control	1	250	0.069	70	22	2-3
Load control	3,5#	240	0.014	45	15	2-3
Autoland, horiz.	#	20				
Autoland, vert.	1#	160	0.055	750	275	2-3
Autoland, throttle	#	33				
Autopilot	4	5	?	150	100	4-5
Elec. att. control	1#	30	0.077	790	520	?
Supervisor	4	?	?	75	15	?
Inertial	2#	1,25	0.034	2100	150	0-4
VOR/DME	4	5	0.004	250	50	4-5
DME, OMEGA	4	5	?·	400	105	4-5
Air data	4	?	?	110	25	4-5
Kalman filter	4	1/5	0.001	250	65	2-3
Flight data	4	5	0.028	450	100	2-3
Airspeed, altitude	4#	8,16	0.009	360	70	2-3
Graphic display	4#	1,8	0.032	890	5360	2-3
Text display	4	10	0.019	640	8700	4-5
Collision avoidance	4#	1/3,670	0.021	550	650	1-2
Data comm, A/C	#	Non-iterative	0.006	210	400	?
Data comm, ground	4#	≤ 4	0.001	450	112	?
AIDS	5#	1/4 to 4	0.002	650	650	4-5
Inst. monit.	4#	5	0.014	800	100	2-3
Syst. monit.	1-4#	1/2	0.001	900	50	2-3
Life support	1-4#	<1/2	0.001	900	50	3-4
Engine control	1-2#	33	0.119	1300	200	1-2

Tasks to be run during blind landing, the most critical flight mode, are marked "#".
Tasks marked "?" exert a negligible load for the parameter in question.
The column headings are defined in the text.

Iteration Rates/Sec--The number of times per second that the calculation must be carried out. When two figures are quoted, they represent two calculations within the same functional task.

Equivalent MIPS------The Millions of Instructions Per Second to carry out the calculations.

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<u>Memory Required</u>------The number of words of memory required for instructions and data.

<u>Missed Iterations</u>----The maximum number of consecutive iterations that can be missed before the application is jeopardized.

In interpreting the table and discussing its implications on computer architecture, we consider reliability, roll-back delay, main memory requirements, processor speed, processing variations within a mission, and data rates.

Reliability

We assume that the probability of <u>not</u> successfully carrying out the most critical computation should be less than 10^{-8} per mission. These computations, corresponding to criticality classes 1 and 2, could cause an aircraft crash if not carried out or if carried out with gross errors. With this assumed computation reliability, for a fleet of 1000 aircraft flying four daily missions, each of five hours without repair between flights within a day, about one crash due to a computer failure would occur in 100 years. For the other criticality classes, the assumed reliability is not as stringent--a typical failure probability is 10^{-4}--since the failure to carry out these less critical computations results in only a mission change or an economic loss. In a system design, it would be beneficial to so allocate redundancy that each task is carried out with the indicated reliability.

Roll-back

An important parameter of a fault-tolerant computer is the maximum time interval that the computer can be in a roll-back/reconfiguration mode in responding to a failure. During this interval some processing of certain computations may cease, and newly appearing data might be lost. The missed iterations column of table 1 indicates the number of iterations that can be ignored in a given computation without adversely affecting the aircraft. In the worst case (collision avoidance) the system must be "down" for no more than 1.5 msec. Several other critical computations--flutter control, load control, autoland--require reconfiguration times nearly as short. For these computations, it might be necessary to reload programs, which indicates that the computer might be required to be totally engaged in reconfiguration following a failure. Fortunately, the computations with large amounts of data, e.g., display, can tolerate a downtime of approximately 0.5 sec., thus allowing ample time for the possible reloading of data, interleaved with the more critical computations.

Memory Requirements

The application programs for the critical phase require approximately 20K words. This figure is a low estimate for two reasons:

● The difficulty of estimating accurately

● The need for memory space for the executive routines.

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Hence we assume a total memory requirement of 24K words. Note that this is a nonredundant requirement; the demand for fault tolerance will increase this figure. For architecture relying totally on triplication, this storage requirement must be tripled to 72K. For architectures utilizing only single-byte correction (ref. 10) in memory (plus possibly a few extra bytes for double-byte detection and sparing), the figure is about one-third in excess of 24K or about 32K.

<div align="center">Processor Speed</div>

For the critical phase, the application tasks require 0.386 MIPS (millions of instructions per second). Once again we must regard this figure as being low in part due to inaccuracies, but mostly due to the "wasted" CPU power in multiprogramming and the processing of executive routines. For these reasons we assume a processor load of 0.5 MIPS. An important attribute of the computations is their relative independence. That is, the sharing of functions and data among the computations does not substantially reduce the overall memory or processor requirements. Each computation requires access to the state of the aircraft, but most other data can be considered to be local. Hence it is quite simple to impose a multiprocessor discipline on the computations, with almost an arbitrary number of processors.

Under certain allocation of tasks to processors it is not necessary to do any task interruption within a processor. That is, a task can be allowed to run through completion before initiating another task. Five processors each of 0.1 MIPS would enable such an allocation. However, near the end of the useful life of the computer, say if just one or two unfailed processors remain, it is possible that a high-rate task (flutter control) might be allocated to the same processor as a low-rate but long task (graphic display). If such a joint allocation is unavoidable, then interruption of the longer task will be essential.

<div align="center">Processing Variations Within a Mission</div>

All applications marked with "#" are required during an instrument landing. This represents about 60 percent of the total CPU requirement and about 50 percent of the memory requirement. Hence some graceful degradation is possible as, during the mission, tasks will be naturally deallocated as they are no longer needed as part of the flight. Hence, when a task is no longer needed, its memory area can be allocated to another task, or, a failure in a memory module is automatically handled by a memory module with a reduced requirement. However, we note that the degradation with respect to memory is not uniform, assuming that all programs and constants are retained in main memory.* For example, in mid-flight, although not all tasks are being processed, all programs must be stored reliably in the main memory. Hence the

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* The issue of back-up memory in an aircraft environment is yet to be completely resolved. Rugged discs can be obtained but their cost per bit is not significantly less than that for LSI main memories.

graceful degradation with regard to main memory is not exploitable until the last minutes of the flight, and hence is of questionable utility to the architecture.

Data Rates

An important measure of computer power required is the load on the bus structure for transfer of instructions and data. Given a computing load of 0.5 MIPS, we assume that an instruction will, on average, require 24 bits.* Different instructions require varying amounts of data including the following cases:

- 0 bits for register-to-register operations

- 8 bits for byte operations, e.g., text display

- 16 bits for integer operations

- 32 bits for floating point operations.

Based on an estimate that the average data required is 16 bits, the total flow between memory and CPU is 20 Mbits/sec. In some architectures (e.g., the JPL STAR), the bus would have to be capable of maintaining this rate. In the case of the Hopkins scheme, a significant reduction would be achieved by the use of the local CACHE on the processors. An additional reduction is achieved by providing a multi-bus structure or allowing multiple ports into main memory. In the SIFT system, most of the bus load would be in individual modules, with only an estimated one percent between modules.

TECHNOLOGICAL ADVANCES

The most important future development in technology is expected to be the continued improvements in LSI. The cost of LSI circuits will continue to drop throughout the 1970s, and will result in processor and memory costs that are low enough so that extensive redundancy of units is practical from a cost viewpoint. This redundancy can be either by replication or by coding, the latter being more applicable to memories. It is expected that the cost of a computer system to carry out all computation within an aircraft will be comparable with the present cost of existing single-function avionic units (e.g., inertial navigation).

A second advantage in the use of LSI is the small size of such units, making it possible to achieve far more efficient shielding from both electric and magnetic fields, thereby reducing the probability of noise and crosstalk. It is expected that fault modes of this type (which are manifested as data-dependent transient faults) will be insignificant within the central units. However, such faults may still exist in connections to external sensors and actuators.

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*In a 16-bit computer this implies equal number of single- and double-length instructions.

With the use of LSI most of the connections at the device and gate level take place within the semiconductor device, or chip, rather than on a board or through a connector as in the use of discrete circuits. The number of soldered and wrapped joints is estimated to be at least an order of magnitude less than that associated with, say, integrated circuits, thus there would be consequent reduction of faults in the connection system.

LSI circuits, though relatively cheap in high-volume production, have a high development cost. This implies that an efficient design would contain as small a number of different chip types as possible. This affects architectural decisions at two levels. At the unit level (memory, bus, arithmetic unit, control, etc.), there will be strong advantage in using replication of identical units rather than units designed specifically for particular functions. At the logic level, the high development cost of custom built units makes it more attractive to transfer arbitrary logic to a form of memory as in the use of microprogramming.

Replacement and maintenance strategies in a reconfigurable computer are also influenced by LSI. The large number of gates per chip, together with the tendency for a chip fault to affect many gates, implies that groups of registers on the same chip should be replaced, rather than replace small units such as registers.

The choice of LSI technologies is between the lower-speed, lower-cost MOS and the higher-speed and higher-cost bipolar technologies. The total computing power required among the elements of the several candidate architectures is such that MOS will be sufficiently fast for memories, buses and arithmetic units. In addition, the use of a multiprocessor organization permits the attainment of high computation capacity with slower processors. The higher speed of bipolar circuits may still be necessary in the control sections where the microprogram cycle time will typically be an order of magnitude faster than the instruction cycle time. Recent advances in technology have tended to bring the two types closer in both speed and cost.

We note that the choice between different LSI technologies, discussed above, was on the basis of speed and cost. The lower-cost alternative of MOS is possible because of the higher density within the chip, thereby enabling the use of fewer chips. This will have the desirable effect of increasing the inherent reliability due to the reduction in number of chips. LSI memory systems appear to be potentially more reliable than core or plated wire, because of the reduced numbers of discrete semiconductor devices and inter-connections. The use of batteries is deemed to be a fully adequate assurance of non-volatility.

The MTBF for LSI circuits is estimated to be between 10^6 and 10^7 hours. The requirement to achieve a MTBF of 10^9 hours for the whole system can be shown to be achievable by several architectures.

The use of optical coupling between units can provide great protection against damage propagation through several units. The architecture must therefore be more concerned with fault propagation through erroneous data

than by adverse electrical phenomena. The added cost for such protection is substantial, though not prohibitive, so careful design to achieve fault-isolation is required.

DESIGN CONSIDERATIONS FOR FAULT-TOLERANT COMPUTER ARCHITECTURES

In the preceding sections we have discussed the requirements for fault-tolerant aircraft computers, and the impact of new technology on their design. We now consider some representative computer architectures from the viewpoint of cost and reliability.

Many possible computer structures exist to satisfy the requirements and it is not our intent here to survey all existing or possible designs, but rather to look at a small number of designs in order to compare the use of different fault-tolerance techniques. We choose three designs—multichannel, SIFT, and SIFT with coding in memory.

In the multichannel design, a number of identical computers are used with all computers operating identically on the tasks to be performed. The computers are operated in a lock-step mode with all data movement being checked by voters that are connected to the buses. A typical number of channels would be three, four or five, higher numbers being unnecessary and tending to complicate the design of the voters.

In the SIFT design, a number of computers are also used but they do not operate in lock-step mode, and they do not all operate on the same tasks. Error-detection is achieved by comparison of results of calculations carried out in several computers, this comparison being by program, not by a hardware voter. An important characteristic of the design is that the buses connecting computers are constrained so that each computer cannot write into the memory of the other computers. This greatly improves fault isolation between computers. Reconfiguration is also carried out by software in a system executive that is itself replicated to assure adequate reliability.

In the third design to be considered, the processors operate as in the SIFT design, but coding is applied to protect against faults in memory.

We now consider each of the above designs. In all cases we assume a chip failure probability of 10^{-6}/hr. We use the notation that P[event] = probability of the event occurring per hour.

We distinguish between the most critical (MC) tasks where error[*] probabilities should be below 10^{-9}/hr and the least critical (LC) tasks where errors should be below 10^{-4}/hr. We also distinguish those tasks required for automatic

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[*]In this analysis, we do not distinguish between erroneous outputs to actuators and null outputs. A more comprehensive analysis would need to make this distinction.

'blind' landing and other tasks. The landing phase is the most demanding in terms of computing load. We summarize in table 2 a representative set of requirements, where M is memory requirements in thousands of words and P is processor requirements in MIPS.

Table 2

COMPUTATION AND MEMORY REQUIREMENTS

	Landing	Other
Most Critical	P = 0.29 M = 8.8	P = 0.09 M = 2.2
Least Critical	P = 0.9 M = 6.8	P = 0.05 M = 5.5

We assume that words contain, on the average, 24 information bits. We further assume that a memory chip contains 4K bits, and that it requires 30 chips/MIPS to realize the CPU.

Case 1: Multichannel

We assume 10% extra memory and processor requirement to handle the multiprogramming and other executive requirements (interrupt handling, etc.). The multichannel concept requires enough memory in each channel to hold all tasks (23.2K + 10% ≈ 26K), and the CPU must handle the heaviest task load (0.38 + 10% ≈ 0.42 MIPS). Therefore for each channel we have

$$26K \text{ words} = 156 \text{ chips}$$
$$0.42 \text{ MIPS} \approx 13 \text{ chips}$$
$$\Big\} \approx 170 \text{ chips}$$

Assume that the chips in the voter (sufficiently replicated for reliability) are negligible and consider the probability of error for three-, four- and five-channel configurations. The results are displayed in table 3.

Case 2: SIFT With Fault Tolerance Achieved by Uniform Replication

For this case, the strategy is to triplicate all tasks, and when faults occur to reduce the LC tasks to duplicate, then single processors, finally removing them entirely in the event that resources are drastically reduced. We assume 20% overhead for executive plus voting routines.[*]

The memory and processor requirements are as in table 4. The reliability results are displayed in tables 5 and 6, for a SIFT system decomposed into four and ten modules, respectively.

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[*]This estimate (of 20%) is not critical in determining the component count, the cost or the reliability of the design.

Table 3

RELIABILITY ESTIMATES FOR MULTICHANNEL SYSTEM

3 Channel

    Total chips   = 540

    P[1 fault]   = 0.51 X 10^{-3},....voting masks error, discard faulty
                                      channel

    P[2 faults]   = 0.17 X 10^{-6},....system failure

4 Channel

    Total chips   = 680

    P[1 fault]   = 0.68 X 10^{-3},....voter removes faulty channel

    P[2 faults]   = 0.34 X 10^{-6},....voter masks second fault, discard faulty
                                      channel

    P[3 faults]   = 1.2 X 10^{-10},....system failure

5 Channel

    Total chips   = 850

    P[1 fault]   = 0.85 X 10^{-3},....voter removes faulty channel

    P[2 faults]   = 0.58 X 10^{-6},....voter removes faulty channel

    P[3 faults]   = 0.3 X 10^{-9},....voter masks fault, discard faulty
                                      channel

    P[4 faults]   = 1   X 10^{-13},...system failure

Table 4

SIFT PROCESSOR AND MEMORY REQUIREMENTS

	Landing	Other
Most Critical	P = 0.35 M = 10.4	P = 0.11 M = 2.6
Least Critical	P = 0.11 M = 8.2	P = 0.06 M = 6.6

Total memory requirement = 27.8 ≈ 28K

Maximum CPU requirement = 0.46 MIPS

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Table 5
RELIABILITY ESTIMATES FOR A 4-MODULE SIFT

Each memory	$= (28 \times 3)/4 = 21K = 126$ chips $\Big\}$ 136 chips
Each CPU	$= (0.46 \times 3)/4 = 0.35 \approx 10$ chips
Total chips	$= 544$

We denote MC during landing as MC/L etc.

$P[1 \text{ fault}] = 0.54 \times 10^{-3}$, $M = 63K$, $P = 1.05$
During Landing: Remove LC, MC survive
During Other: MC survive, all LC to SIMPLEX, future removal LC/L

$P[2 \text{ faults}] = 0.22 \times 10^{-6}$, $M = 42K$, $P = 0.70$
During Landing: MC/L only survive in DUPLEX
During Other: All MC to DUPLEX in memory, all LC to SIMPLEX, future removal of LC/L

$P[3 \text{ faults}] = 0.6 \times 10^{-10}$, System failure

Table 6
RELIABILITY ESTIMATES FOR A 10-MODULE SIFT

Each memory	$= (28 \times 3)/10 = 8.4K = 51$ chips $\Big\}$ 55 chips
Each CPU	$= (0.46 \times 3)/10 = 0.14 \approx 4$ chips
Total chips	$= 550$ chips

$P[1 \text{ fault}] = 0.55 \times 10^{-3}$, $M = 75.6K$, $P = 1.26$
During Landing: Fault masked, LC to DUPLEX
During Other: Fault masked, LC/O to DUPLEX, Future LC/L to DUPLEX

$P[2 \text{ faults}] = 0.27 \times 10^{-6}$, $M = 67.2K$, $P = 1.12$
During Landing: MC fault masked, LC failed
During Other: Fault masked, Future LC/L fail

$P[3 \text{ faults}] = 0.19 \times 10^{-9}$, $M = 48.8K$, $P = 0.98$
During Landing: MC fault masked, MC/L to DUPLEX
During Other: Fault masked, Future LC/L fail

$P[4 \text{ faults}] = 0.73 \times 10^{-13}$, $M = 40.4K$, $P = .84$
During Landing: Possibility of system failure
During Other: Possibility of LC failure, future MC/L in DUPLEX

Case 3: SIFT with Coding in Memory

The majority of chips for SIFT in Case 2 are used in the memory. We can add protection by using an error detecting/correcting code. The analysis displayed in table 7 is for a single-error-correcting, double-error-detecting code with an assumption of 25% increase in memory cost. A module failure requires failure of one chip in the CPU or two chips in the memory. Low criticality tasks are run in SIMPLEX mode.

Table 7

RELIABILITY ESTIMATES FOR FOUR- AND SIX-MODULE SIFT WITH CODING IN MEMORY

4 Module

Memory per module $= (13 \times 2 + 15)/4 + 25\% \approx 13K = 78$ chips $\Big\}$ 84 chips

CPU per module $= (0.35 \times 2 + 0.11)/4 \approx 0.2 = 6$ chips

Total chips $= 332$

$P[\text{CPU fault}] = 0.6 \times 10^{-5}/\text{per module}$

$P[\text{single memory fault}] = 0.8 \times 10^{-4}$

$P[\text{double memory fault}] = 0.6 \times 10^{-8}$

$P[\text{LC task failure}] = 0.6 \times 10^{-5}$

$P[\text{reconfiguration}] = 0.3 \times 10^{-3}$

$P[\text{second module fail}] = 0.8 \times 10^{-7}$

$P[\text{MC task fail}] = 1.3 \times 10^{-11}$

6 Module

Total chips $= 348$

$P[\text{LC fail}] = 0.4 \times 10^{-5}$

$P[\text{MC fail}] = 0.2 \times 10^{-11}$

The above analysis is portrayed in figure 1 which shows the relationship between number of chips required and the probability of failure of the most critical tasks.

We conclude that all these architectures are capable of achieving the required reliability given sufficient replication. Using triplication, both of these architectures are capable of achieving a reliability of failure in the region of 10^{-6} to $10^{-7}/\text{hr}$. Where reliability requirements are more stringent, as in the case for commercial aircraft, the multichannel approach can only achieve sufficient reliability at a cost significantly higher than that achievable by the SIFT architecture. In both cases, the use of coding in memory can
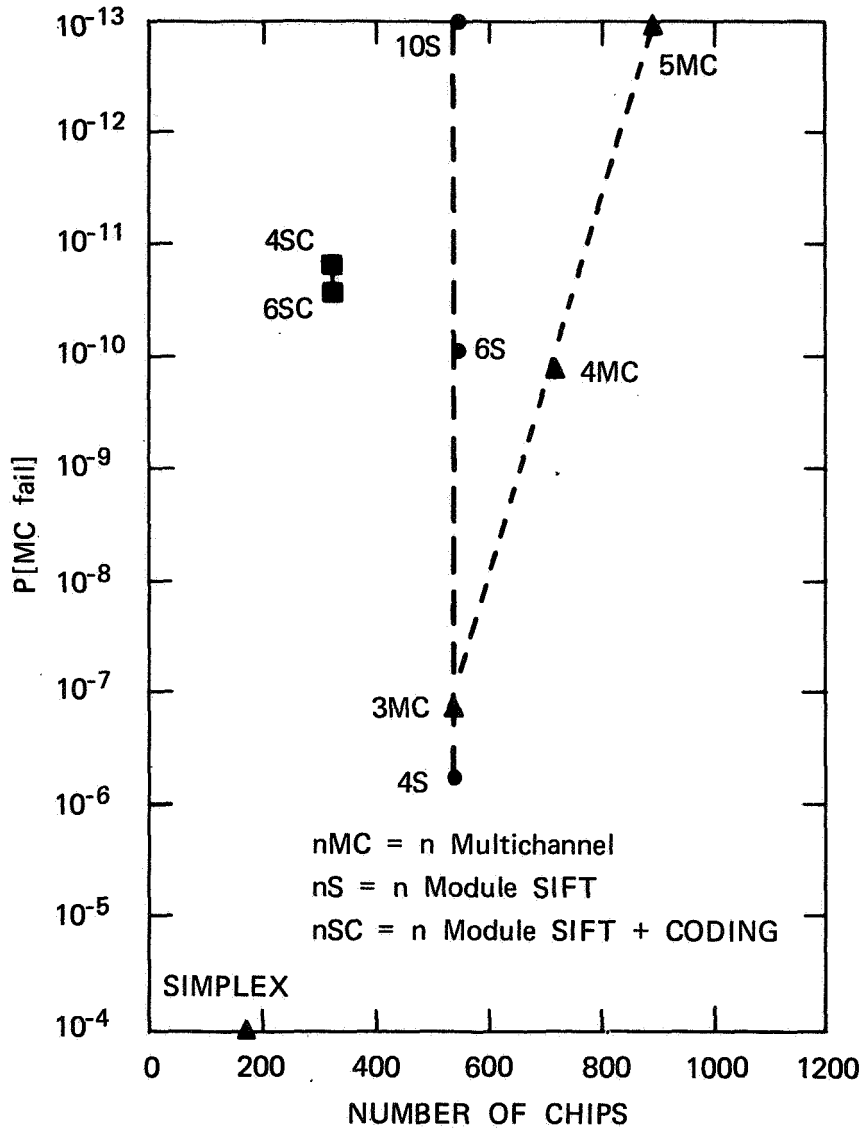
FIGURE 1 PROBABILITY OF FAILURE OF MOST
CRITICAL FUNCTIONS P[MC fail] AGAINST
NUMBER OF CHIPS

have a significant impact on reliability and cost by handling single-error correction and double-error detection in memory in a very economic manner.

FACTORS INFLUENCING FUTURE COMPUTER ARCHITECTURES

We have discussed in the preceding sections the problems of designing fault-tolerant computers for advanced avionics requirements. We now examine the forces that will influence such computer designs in the future, particularly the period 1980-85. We see three types of influences: changes in requirements, advances in technology, and maturity in this specialized design field.

In looking at requirements we expect to see an increase in the computing load. To a large extent, this will be due to the trend towards aircraft designs that requires substantial real-time control systems for critical functions. Obvious examples are in flutter and attitude control. In addition, the operational modes of commercial aircraft will change. We would expect to see more extensive use of automatic blind-landing systems, collision-avoidance systems, and automatic or semi-automatic route-control systems. In summary, we see a greater requirement due both to more advanced aircraft designs and to a wider range of operational modes.

The most significant development of technology in the late 1970s is expected to be the wide application of large-scale integrated (LSI) technology. This will cause several effects. First, we observe that low-cost production of LSI circuits relies upon large-volume production and therefore there will be a strong incentive to use standard circuits whenever possible. This will greatly influence the type of acceptable computer architectures. Design concepts, such as discussed in the preceding section, are the types that will be favored compared with designs relying upon specialized logic to carry out the various functions associated with fault tolerance.

The second effect will be that the demonstrable inherent reliability of a circuit will be available only on large-production-volume devices. This effect will be another force towards the use of standardized circuits whenever possible. A third effect of LSI development will be the availability of back-up storage units based upon electronic (i.e., non-mechanical) technology. Such developments as bubble or charge-coupled memories potentially can be used to hold data either for later use, or for re-entry into main memory after a memory fault.

The third significant force that will influence future avionics computers stems from the increasing maturity in this field. Most designs in the past were arbitrary designs developed in vacuo, i.e., each design effort did not rely upon results of other efforts. There was little that could be taken from one effort to assist another. This is now changing so that the community of fault-tolerant computer designers can borrow from the results of others. Notable examples of this expanding technology base are error correcting/detecting codes, reliable switches, and reliable clocks. We still see deficiencies in the technology base, but expect that with continued research, they will disappear. The most notable present deficiencies are in the field of reliability modeling and in the area of certification. Reliability modeling as an art at present tends only to be able to analyze very idealized systems and must make very

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simplifying assumptions (e.g., that faults are independent and permanent). We expect that reliability modeling techniques will be developed to the point where more realistic reliability analyses can be carried out. In considering any fault-tolerant architecture, one is faced with the problem of certification of the procedures used for achieving reliability. These procedures may be implemented in either hardware or software, but whichever implementation is used there is a need to prove that the desired reliability characteristics are achieved. The present progress in the field of program proving gives us grounds to believe that formal proofs of fault-tolerant behavior will be possible.

To summarize, we see a strong trend towards the use of LSI circuitry with its attendant reduction in the number of devices, thus greatly improving the intrinsic reliability of computers. In addition, we expect advances in the theory and practice of designing, analyzing and certifying fault-tolerant computers for aircraft control applications.

We see the greatest need for improvement in techniques as:

(a) Structures for logic, systems, and software that provide both high levels of fault tolerance and ease of analysis, without the penalty of gross inefficiency or too inflexible a structure.

(b) Economical and accurate methods for verifying the correctness of system hardware and software with respect to fault tolerance and proper servicing of application programs.

However, there appears to be no fundamental reason why very reliable computers cannot be built within reasonable economic constraints. We would envisage such computers to use more than one technique to achieve adequate reliability. The main techniques would be replication, coding and reconfiguration.

CONCLUSIONS

In some new aircraft types under development there is a need for computational resources to handle very critical functions, indeed, the safety of the aircraft will be dependent on the correct functioning of the computer. In addition, the combination of high reliability and substantial computational load needed for future aircraft makes the use of simple redundant computer configurations impractical.

The present reliability art, together with continually improving technology, promises substantial improvements within the next five years for those aircraft applications with only modest computational loads. However, to meet all the larger set of computational requirements that have been suggested, at the necessary reliability levels, advances in the art of fault tolerant computer design will be required.

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THE EFFECTS OF LIGHTNING ON DIGITAL

FLIGHT CONTROL SYSTEMS

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SUMMARY

Present practices in lightning protection of aircraft
deal primarily with the DIRECT EFFECTS of lightning, such as
structural damage and ignition of fuel vapors. There is
increasing evidence of troublesome electromagnetic effects,
however, in aircraft employing solid-state microelectronics in
critical navigation, instrumentation and control functions.
The potential impact of these INDIRECT EFFECTS on critical
systems such as Digital Fly-by-Wire (DFBW) flight controls has
been studied by several recent research programs, including an
experimental study of lightning-induced voltages in the NASA
F8 DFBW airplane. The results indicate a need for positive
steps to be taken during the design of future fly-by-wire sys-
tems to minimize the possibility of hazardous effects from
lightning.

INTRODUCTION

Present practices in lightning protection of aircraft
deal predominantly with what may be called the DIRECT EFFECTS
of lightning, including burning, blasting and physical defor-
mation of skins and structural elements. Existing lightning

protection specifications, such as MIL-B-5087B, (Bonding, Electrical, and Lightning Protection, for Aerospace Systems) concentrate on electrical bonding and its function in minimizing these effects. Other criteria such as FAA Advisory Circular No. AC 25-3A, provide guidance for protection against lightning ignition of flammable fuel-air mixtures. Concern with these effects has been necessary since safety of flight in a lightning environment has heretofore primarily depended upon protection against fuel ignition and structural damage that can be produced by lightning. There is increasing evidence of troublesome electromagnetic effects due to lightning, however, as a result of transient surge voltages induced in aircraft electrical wiring. These voltages have caused both permanent damage and temporary malfunction of equipment.

Earlier vacuum tube electronics were inherently less vulnerable to lightning-induced voltage surges; however, the newer generations of modern, solid state microcircuitry are increasingly more vulnerable to upset or damage from such effects. Because these are electromagnetically induced effects, they are often referred to as the INDIRECT EFFECTS of lightning. Recently, these effects have been receiving additional attention since the flight safety of modern aircraft is increasingly dependent on reliable operation of critical electronic systems. At present there are no standards or specifications applicable to the INDIRECT EFFECTS of lightning.

With the advent of fly-by-wire systems, particularly those with digital computer and control electronics, the indirect effects of lightning very clearly have the potential of presenting a hazard to safety of flight. This hazard may be particularly acute for digital systems. While most practical digital fly-by-wire systems would include multiple redundant control circuits it is possible to conceive of a situation in which the high level electromagnetic interference produced by lightning could interfere with all channels of a fly-by-wire system at once, raising the possibility that there may in fact be no real redundance with respect to lightning effects.

The NASA Flight Research Center has developed and is presently demonstrating a digital fly-by-wire (DFBW) flight control system in an F8 aircraft. Recognizing the possibility of this hazard, a program was implemented with General Electric to evaluate the possible electromagnetic effects of lightning on this flight control system and obtain data for use in minimizing these effects in future generations of fly-by-wire aircraft. The F8 DFBW system was not designed to withstand lightning strike effects. Therefore, the opportunity existed to experimentally determine the severity of effects in this unprotected system, thus providing test data upon which to base design guidelines for protection of future systems.

SYMBOLS

A/C	Aircraft
AGC	Apollo Guidance Computer (DFCS computer)
BCS	Backup Control System
DFCS	Digital Flight Control System
DFBW	Digital Fly by Wire
IR	Structural ohmic resistive voltages
i_L	Lightning current

TEST AND MEASUREMENT TECHNIQUE

A recently developed simulated lightning test and measurement system known as the TRANSIENT ANALYSIS technique offers a means of investigating the electromagnetic effects of lightning without hazard to the aircraft being tested. This technique, the development of which was sponsored by the Aerospace Safety Research and Data Institute of NASA-Lewis Research Center (Ref. 1), consists of injecting current surges into an aircraft, of the same waveshape as those produced by lightning but of greatly reduced amplitude. The responses of the aircraft's electrical circuits to these current surges can be measured and then extrapolated to correspond with full lightning stroke amplitudes to determine if they present a hazard to the equipment under test. During the development of this technique, tests were made to show that the response of an aircraft electrical system was linear with lightning current amplitude and that this extrapolation was valid. The transient analysis technique was utilized in the study of the NASA F8 DFBW aircraft in this program. A photograph of the aircraft and test setup is shown in Figure 1.

The test circuit is shown on Figure 2(a). The airframe is connected to ground at the point nearest the terminals of the circuit being measured via a 36 inch wide, 3 mil aluminum foil. This was attached to the instrument table and the hangar ground about 20 feet away. Use of the aluminum foil provides a very low impedance between the airframe and instrument table. The instrument cable was placed along this foil so that no air gap existed between it and the foil. As shown on Figure 2(b), the lightning current circuit is grounded once and only via this airframe ground foil. Consequently, no simulated lightning

current could flow off of the airframe along this path or the instrument cable shield and get back to the transient analyzer.

Most of the tests were made with a unidirectional simulated lightning stroke current rising to its crest in 2.75 microseconds and decaying to half value after 60 microseconds. This waveform is representative of natural lightning stroke waveforms and is similar to the waveform specified for indirect effects testing of the Space Shuttle. Its crest amplitude was set at 300 amperes to minimize the possibility of interference or damage to any of the electronic systems or components aboard the aircraft. Natural lightning strokes exceed 200,000 amperes about 1% of the time and average about 30,000 amperes in amplitude. Therefore, voltages induced by this waveform must be extrapolated upward by a factor of 100 to correspond with an average lightning stroke or 670 to correspond with a severe 200 kA stroke. The test current waveform is shown on Figure 2a.

It will be noted that damped oscillations appear on the test current wavefront. These are believed to be the result of traveling wave reflections in the transmission line formed by the aircraft and return conductor beneath it. Measurements were made of the current entering as well as leaving the aircraft, verifying that the superimposed oscillations flowed through the aircraft along with the fundamental current waveform. The extent to which oscillations or "jagged edges" occur in natural lightning current wavefronts is not well known, although available oscillographic data (Ref. 2) does show evidence of such occurrences in some strokes.

Induced voltages were measured by a Tektronix Type 545 oscilloscope with a Tektronix Type G differential pre-amplifier. The differential measurement system previously developed for this technique and described in Ref. 1 was utilized. One channel of the measurement circuit was normally connected to the circuit conductor being measured, and the other channel was connected to the DFCS ground, airframe ground or circuit low side, as required for the measurement being made. The pre-amplifier subtracted the signal on the second channel from that on the first so that common-mode errors induced in the instrument cable would not appear in the measurement.

Measurements were made with the DFBW system powered with batteries and operating in the primary mode. Access to most circuits was made with break-out boxes at important interfaces in order to maintain circuit continuity, although some measurements were made at opened interfaces to obtain measurements of open-circuit voltages at cable ends.

DESCRIPTION OF DFBW SYSTEM

The F8 digital fly-by-wire flight control system components are shown in Figure 3. A single digital primary channel and triple redundant electrical analog backup channels replaced the F8 mechanical control system. The primary and backup channels all provide three-axis control of the airplane. The digital channel consists of a lunar guidance computer, inertial measurement unit, coupling data unit, and astronaut display and keyboard, all taken from the Apollo guidance and navigation system. A mode and power panel permits the pilot to request the lunar guidance computer to make mode and gain changes. The three-channel backup control system consists only of surface position command electronics. Specially designed electro-hydraulic secondary actuators interface the primary and backup electronic commands with the conventional F8 control surface power actuators.

Figure 4 shows the general arrangement of the flight control system hardware in the F8 airplane. Five secondary actuators were required, one for the rudder and one each for the two horizontal stabilizers and the two ailerons. The Apollo lunar guidance computer is the heart of the primary control system and performs all flight control computations.

The DFBW system is described in more detail in Reference 3.

TEST RESULTS

Measurements were made at a variety of primary and backup system interfaces. Of greatest interest were the induced voltages appearing at the wiring interfaces with the primary DFCS system, which includes the Apollo lunar guidance computer (AGC). Figures 5, 6 and 7 show some of the measurements. For all of these measurements the simulated lightning current entered the nose and exited from the tail of the aircraft. Figure 5 shows measurements made at the J25 interface on circuits coming from the mode and power control panel and stick, BCS and yaw trim transducers in the cockpit area. These appear as damped oscillations at a fundamental frequency of about 1 megahertz. Most of the voltage has subsided after about 6 microseconds has elapsed. Each voltage shown on Figure 5 is a damped oscillation at a fundamental frequency of about 1 megahertz since all conductors follow the same bundle to the cockpit. The waveforms have slight variations which are probably due to differences in load impedances at each end.

Figure 6 shows voltages induced in the pitch, roll and yaw control sensor circuits coming to the DFCS computer, but the measurements were made at plug P4 with this plug disconnected from the DFCS system. These, therefore, are open circuit voltages and are not necessarily the same as the voltages which might appear at the closed interface, since DFCS input impedances would affect the voltages impressed across them. The characteristic frequencies of the open-circuit voltages measured at pins D-E (osc. 528), G-H (osc. 525), W-X (osc. 523) and Y-Z (osc. 526) have a fundamental frequency of about 1.7 megahertz with lower amplitude oscillations of several higher frequencies superimposed. These are induced in circuits coming from the DFCS stick transducer in the cockpit. The fundamental frequency of voltages measured at pins A-B (osc. 524) and U-V (osc. 527) in circuits coming from the rudder pedal transducer in the tail area is also 1.7 megahertz but without as much of the superimposed higher frequency component. Neither fundamental frequency is the same as that measured at the closed J25 interface in circuits also coming from the cockpit area.

The closed circuit J2 interface measurements shown on Figure 7 are of the same 1 megahertz fundamental as those measured at the J25 interface of Figure 5, except that the polarity is reversed.

DISCUSSION OF RESULTS

Induced Voltages

Study of the induced voltages measured in this system indicates that they are primarily of aperture magnetic flux origin due to the absence of long-duration unidirectional components induced by diffusion magnetic flux appearing inside the airframe when lightning current has diffused to the inside of its skin. Indications of structural IR voltage components are also absent, as expected, since the system is single-point grounded and has no direct reference to the airframe at locations remote from the DFCS pallet where these measurements were made. The single-point ground to the airframe is at the DFCS pallet.

The most prevalent frequency of oscillation of induced voltages measured at the DFCS interface is about 1 megahertz. This is not the same frequency as the oscillations superimposed on the simulated lightning current wavefront, which is 2.6 megahertz. If fact, there is no similarity between this frequency and that of induced voltages measured anywhere in the DFCS system. Fourier transformations were made to determine the frequency spectral distribution of the actual lightning test waveform as compared with an idealized smooth-front waveform.

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Spectral peaks above the smooth-front waveform distribution occur in the test waveform distribution at 2.5, 5 and 8 megahertz, but not at the 1 megahertz frequency of the induced voltages measured at the DFCS interfaces.

The induced voltages reach their maximum during the first several microseconds of lightning current flow, which is when the lightning current and corresponding aperture flux are changing most rapidly. Continued oscillations appearing for several more microseconds are most likely the result of subsequent traveling waves in the circuit being measured. If this is so, the frequency of these voltages is primarily a function of the distributed circuit inductance and capacitances.

The variation in fundamental frequencies and presence of more than one frequency component in a single voltage is probably due to variations in circuit routing and interconnections with other circuits in the system.

The ranges of voltage amplitudes measured at the DFCS interfaces, when scaled to a 200,000 ampere (fast) lightning waveform, are presented in Table I.

Table I - Range of Induced Voltage Amplitudes
(Scaled to i_L = 200 kA)

INTERFACE	INDUCED VOLTAGE AMPLITUDE (0 - Peak Volts)	
	MIN.	MAX.
Stick Trim and MPC Inputs to DFCS(J25)	233	900
Stick Transducer Inputs to DFCS(P4)	40	87
DFCS Control Outputs(J2)	233	400
BCS Control Inputs(J12)	222	422
Mode and Power Control(J15)	833	1132
Mode and Power Control(J14)	213	732
Power Dist. Bay (+28VDC BUS)	160	200
DFCS Ground to A/C Ground	-	666

Voltages measured at other locations in the DFBW system, such as at the secondary actuators and BCS electronics, were of generally similar magnitudes.

Impact on DFCS System

The expected impact of the induced voltages measured in the DFCS system on system operation was analyzed by DELCO Electronics, manufacturer of the DFCS. Comparison of individual component vulnerability data, when available, with induced voltage levels at single circuit interfaces was utilized to determine vulnerability of system components and effect on circuit operation. In other cases, best engineering judgment was used. An example of such an assessment is the attitude (yaw, pitch or roll) gain logic power circuits (pins A-W) from the MPC panel to the DFCS pallet. The schematic diagram of one of these circuits is shown on Figure 8. Induced voltages at the J25 interface are shown on Figure 5 (i.e. osc. 505). The voltages at the J25 interface (DFCS) ranged from 566 to 865 volts, and at the J15 interface (MPC), 1065 to 1132 volts. At the MPC, the induced voltage exceeds the 1000 volt (at sea level) dielectric breakdown rating of the switch. Arc-over may therefore occur either to case and mounting or between contacts, with a possibility of switch failure.

This circuit (+28 VDC) provides a request to the computer to change attitude control loop gain. If the wiper arm of the switch burns open, the computer will notice no gain requests and under this condition is programmed to assume attitude gain position 1. The DFCS control will survive at this gain position. If the switch would short such that two gain-position requests exist, the computer is programmed to assume the lower gain of the two requests. The DFCS control will survive.

At the AGC, the induced voltage exceeds the 500 volt dielectric breakdown rating of the 20K resistor, R2. Arc-over of R2 may then expose capacitor C1 to damaging overvoltage, causing it to short circuit. If it remains shorted during the entire lightning flash, no further damage should occur. If C1 opens between successive strokes of a multiple stroke flash, arc-over(s) of the 1.5K resistor R4 on successive strokes may permanently destroy transistor Q1. If C1 is short circuited, the AGC gain change circuit will be inhibited. Selection of this gain position after the lightning flash will cause the computer to select attitude gain position 1. The DFCS control will survive at that gain position. The same applies if transistor Q1 fails.

As another example, the DFCS digital control output circuits are considered. The schematic diagram of these circuits is shown on Figure 9. Induced voltages measured at the J2 interface are shown on Figure 7 and range from 233 to 400 volts. Those measured at the P12 end ranged between 222 and 422 volts. At the DFCS, capacitor C2 has a 15V rating. Therefore it would

break down as a short circuit. The capacitor could then fail as an open circuit. In either case the remaining circuit components would probably survive the lightning stroke. These are dual circuits which provide attitude commands which are utilized as control surface inputs. The dual outputs are compared to each other for failure detection purposes. Since capacitor C2 can be failed as an open or short circuit, several combinations were considered. If C2 is shorted as a dual output, no failure detection would occur. The pilot would discover that a problem existed only by noting the lack of aircraft response to control stick position. If one of the dual command outputs contained C2 open circuited and the other short circuited, any off-neutral control stick position would trigger the failure detection circuit which would remove that attitude axis control from DFCS to the BCS. In the case of C2 open-circuited as a dual output, DFCS aircraft attitude control could be maintained.

Other individual circuits were assessed in the same manner. Failure in some circuits is likely to degrade DFCS performance, but in others, the consequences appear minimal. It is evident, from Figures 5, 6 and 7, that lightning-induced voltages appear simultaneously in all DFCS circuits.

They also appeared in the 3 BCS channels. Thus, the consequences of simultaneous failures in many circuits must be fully assessed before the total impact on system operation can be determined. This has not been accomplished for this system. In general, however, it was found that many DFCS components are vulnerable to the induced voltages expected from a 200,000-ampere lightning stroke. The most vulnerable components are capacitors, transistors, and relay arc-suppression diodes. The least vulnerable components that may be damaged are switches, relays, forward loop diodes, and inductors.

It should be remembered that the DFCS equipment is an adaption of existing Apollo Lunar Module equipment that was not designed to survive lightning-induced voltages, and also, that a 200,000 ampere stroke is likely to occur only about 1% of the time. The average amplitude is about 30,000 amperes. Under this condition, component vulnerability is reduced.

CONCLUSIONS

This program represents the first experimental investigation of lightning-induced effects on a fly-by-wire system, digital or analog. The results of this study are therefore significant, both for this particular aircraft and for future generations of aircraft and other aerospace vehicles such as the Space Shuttle, which will employ digital fly-by-wire flight control systems. Particular conclusions from this work are as follows:

1. Equipment bays in a typical metallic airframe are poorly shielded and permit substantial voltages to be induced in unshielded electrical cabling inside.

2. Lightning-induced voltages in a typical aircraft cabling system pose a serious hazard to modern electronics, and positive steps must be taken to minimize the impact of these voltages on system operation.

3. Induced voltages of similar magnitudes will appear simultaneously in all channels of a redundant system.

4. A single-point ground does not eliminate lightning-induced voltages. It reduces the amount of diffusion-flux induced and structural IR voltage but permits significant aperture-flux induced voltages.

5. Cable shielding, surge suppression, grounding and interface modifications offer means of protection, but successful design will require a coordinated sharing of responsibility among those who design the interconnecting cabling and those who design the electronics. A set of Transient Control Levels for system cabling and Transient Design Levels for electronics, separated by a margin of safety, should be established as design criteria. Data from this and other experimental programs should be utilized to help establish these criteria.

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1. Walko, L.C., "A Test Technique for Measuring Lightning-Induced Voltages on Aircraft Electrical Circuits", NASA CR-2348, General Electric Corporate Research and Development Report No. SRD-72-065, January, 1973.

2. Berger, K., "Novel Observations on Lightning Discharges: Results of Research on Mount San Salvatore", Journal of the Franklin Institute, Vol. 283, pp. 478-525, June, 1967.

3. Deets, D.A. and Szalai, K.J., "Design and Flight Experience With a Digital Fly-By-Wire Control System in an F-8 Airplane", AGARD paper.
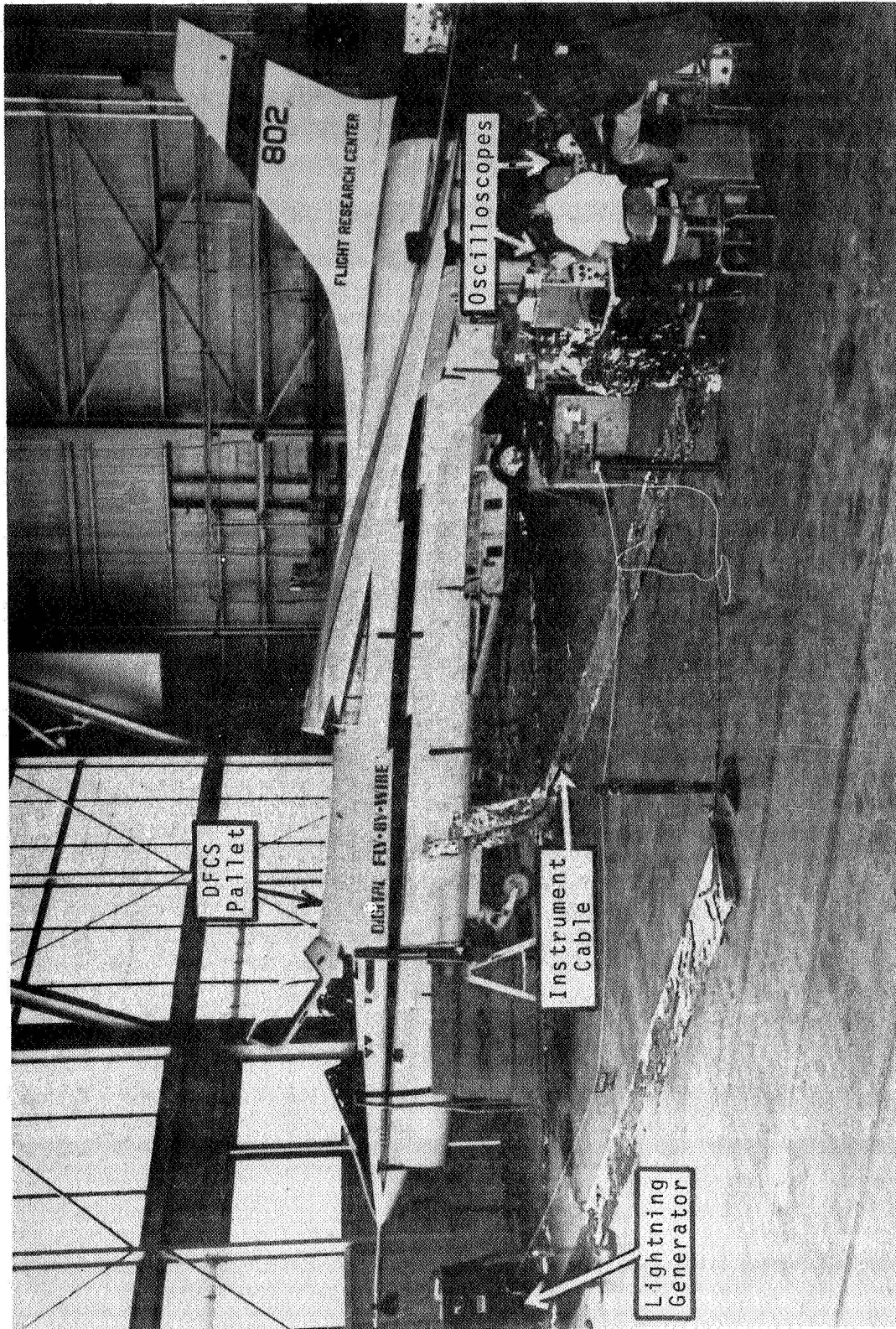
FIGURE 1 – TEST SETUP FOR MEASUREMENT OF LIGHTNING-INDUCED VOLTAGES IN NASA F8 DFBW AIRPLANE.

1000

Simulated Lightning Current
300 Amperes Peak

83 A/Div.

5 μs/Div.
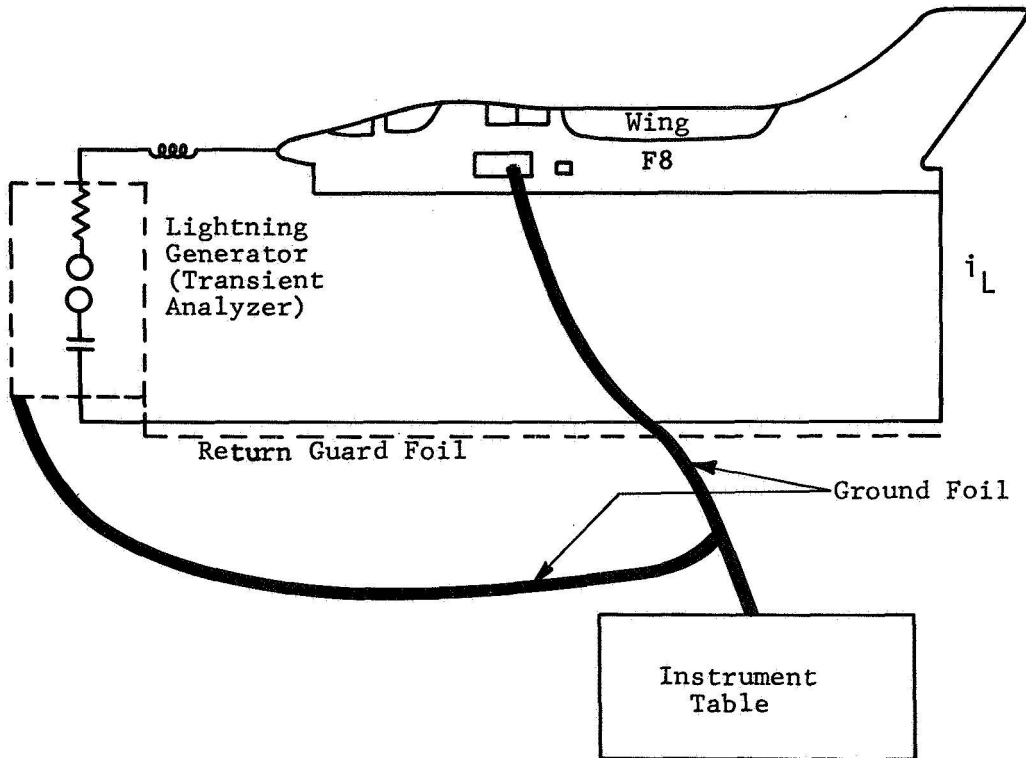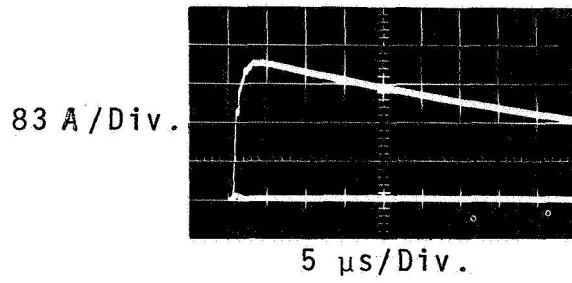
a) Actual Test Circuit
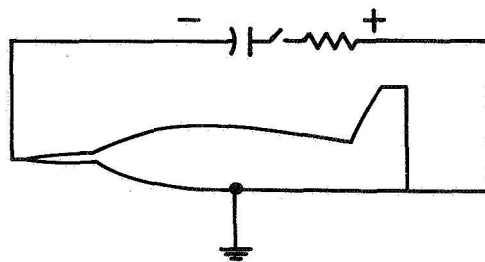
b) Equivalent Test Circuit

FIGURE 2 - SIMULATED LIGHTNING WAVEFORM AND TEST CIRCUIT

1001
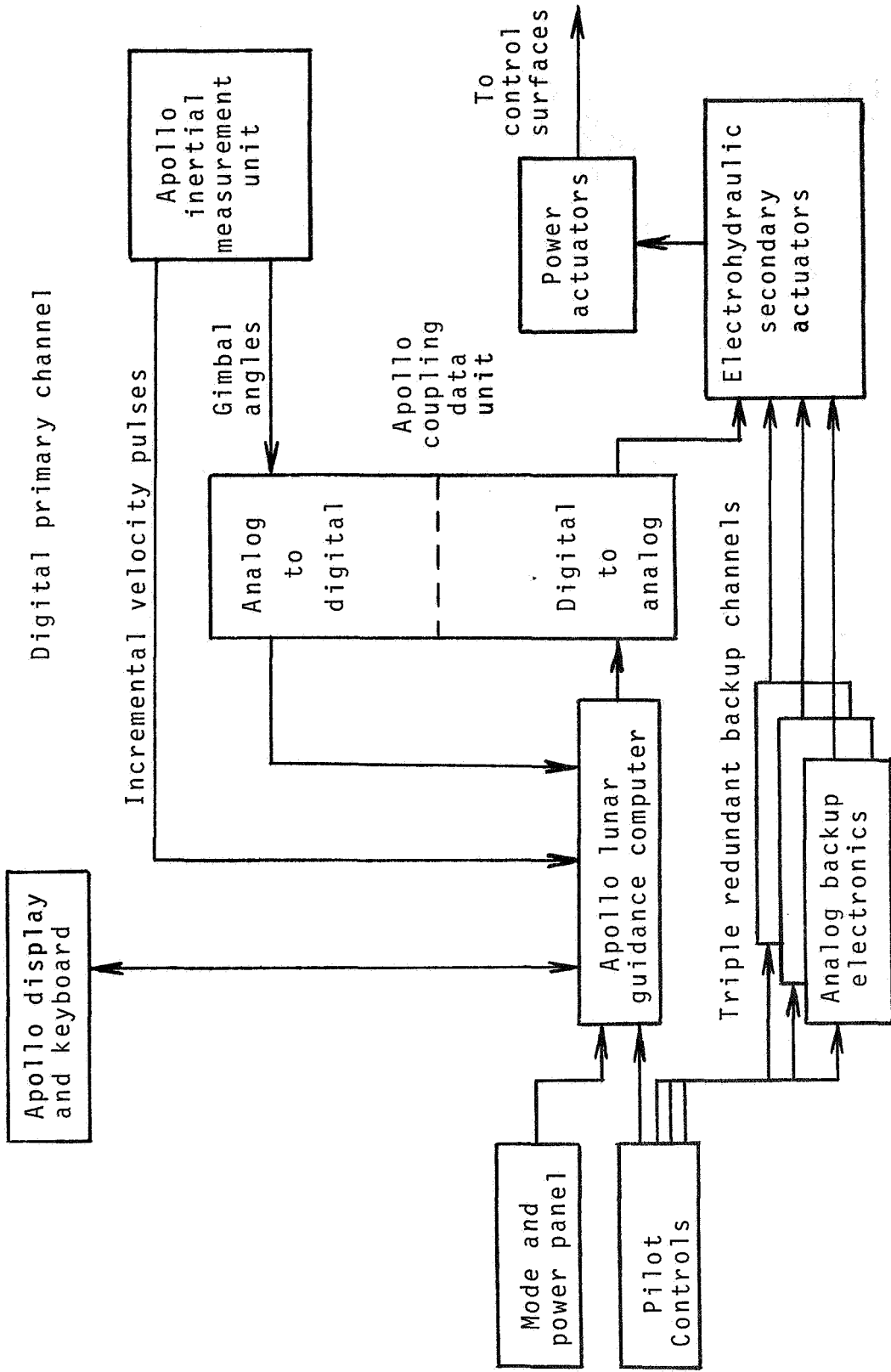
FIGURE 3 - F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM HARDWARE ELEMENTS.

1002

Secondary and
Power Actuators
(Typical)

Primary Digital
System (DFCS Pallet)

Display and Keyboard

Primary and Backup
Control System Electronics

Stick

Mode and
Power Control Panel
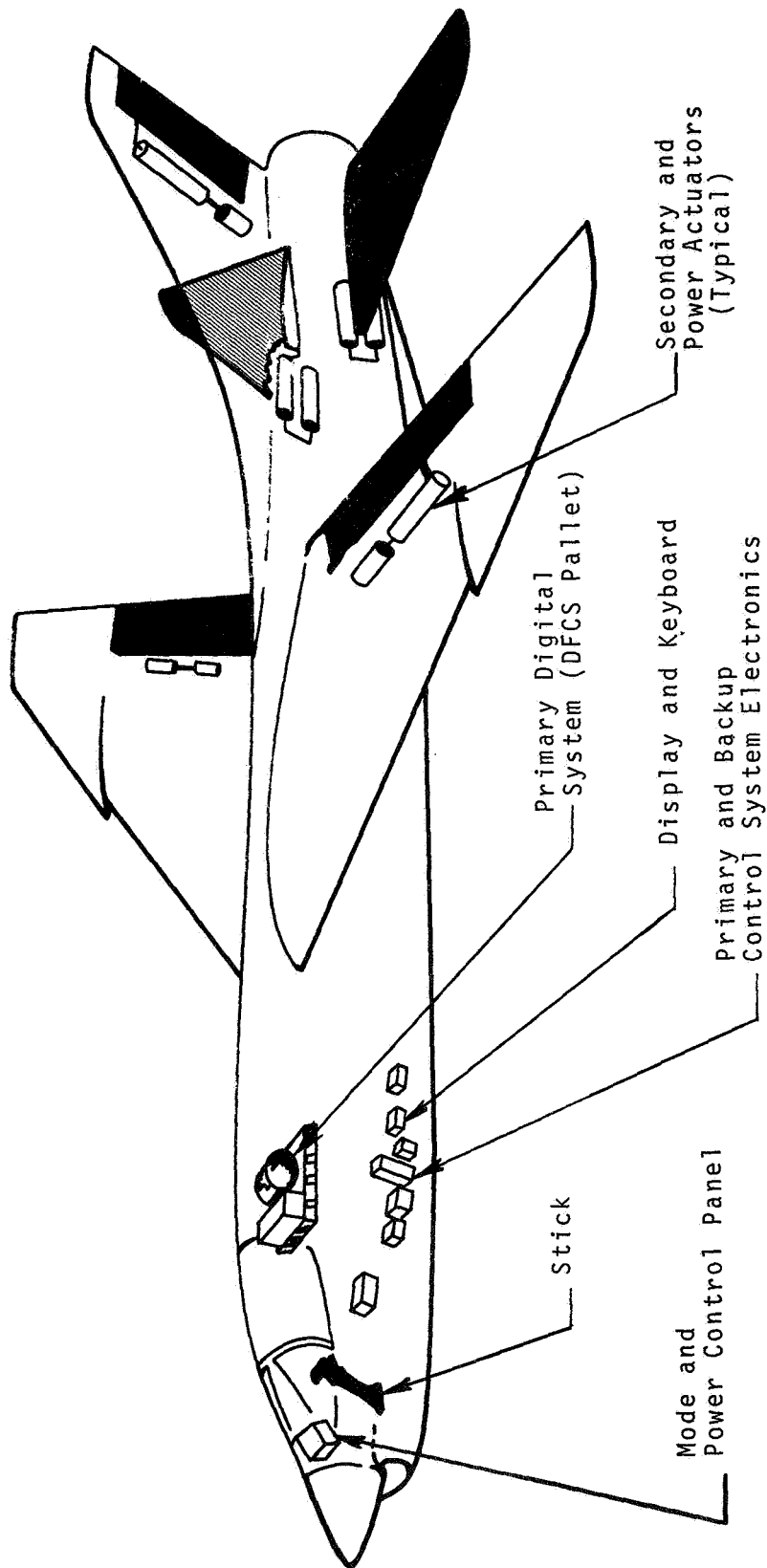
FIGURE 4 - LOCATION OF FLY-BY-WIRE CONTROL SYSTEM HARDWARE IN F8 AIRPLANE.

1003

MODE AND POWER CONTROL PANEL

	J15	P15
YAW GAIN LOGIC PWR	A	
CHASSIS GROUND	B	
YAW GAIN 4 PWR	F	
PITCH GAIN LOGIC PWR	J	
ROLL GAIN LOGIC PWR	L	
PITCH GAIN 4 PWR	T	
ROLL GAIN 4 PWR	V	

STICK TRIM   TB110

PITCH TRIM DOWN	P	A
PITCH TRIM UP	M	D
ROLL TRIM RWD	K	H
ROLL TRIM LWD	N	J
EXCITATION	L	C

BCS TRIM   P113

NOSE DOWN	A
YAW LEFT	B
28 VDC	C
NOSE UP	D
LEFT ROLL	E

YAW TRIM   P114

28 VDC EXCIT	A
YAW LEFT	B
YAW RIGHT	C
CHANNEL #1	

COCKPIT AREA

P25

J25	
A	YAW GAIN LOGIC POWER
D	PITCH TRIM 2 (BCS) DN
E	PITCH TRIM 2 (BCS) UP
F	YAW GAIN 4
J	LOGIC POWER - PITCH GAIN
L	LOGIC POWER - ROLL GAIN
T	PITCH GAIN 4
V	ROLL GAIN 4
W	CHASSIS GROUND
B̄	PITCH TRIM 1 (DFCS) UP
C̄	PITCH TRIM 1 (DFCS) DN
V̄	YAW TRIM (DFCS) LFT
W̄	YAW TRIM (DFCS) RT

DFCS PALLET

(OSCILLOGRAMS RETOUCHED FOR CLARITY.)

ALL OSCILLOGRAMS 0.5 VOLTS/DIV, 1µSEC/DIV.

PINS A-W	PINS D-W	PINS E-W	PINS F-W
No.505	No.509	No.508	No.504

PINS J-W	PINS L-W	PINS T-W	PINS V-W
No.501	No.503	No.500	No.502

PINS B̄-W	PINS C̄-W	PINS V̄-W	PINS W̄-W
No.506	No.507	No.510	No.511

FIGURE 5 - INDUCED VOLTAGES ON MPC AND STICK TRIM
INPUTS TO DFCS AT J25 INTERFACE.

1004
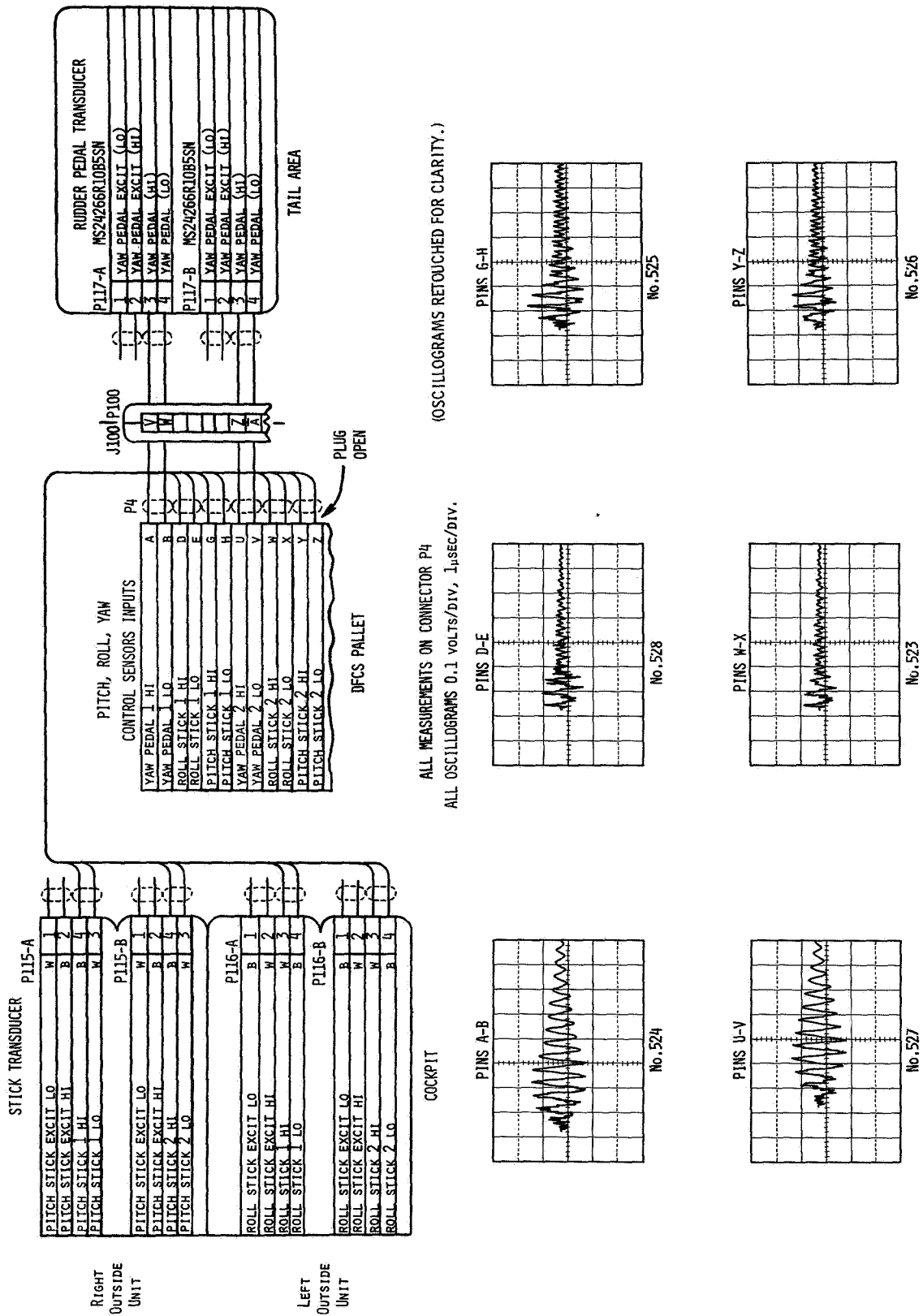
FIGURE 6 - INDUCED VOLTAGES ON CONTROL SENSOR INPUTS TO DFCS AT P4(OPEN).

1005

DFCS Digital Control
Digital-to-Analog
Converter

Primary Control Electronics #1

J2 P2		P12			
A	PITCH SERVO 1 HI	18	PITCH	ACTIVE SERVO CMMD INPUT	(HI)
B	PITCH SERVO 1 LO	19	"	" " " "	(LO)
D	ROLL SERVO 1 HI	43	ROLL	ACTIVE SERVO CMMD INPUT	(HI)
E	ROLL SERVO 1 LO	44	"	" " " "	(LO)
G	YAW SERVO 2 HI	67	YAW	MONITOR SERVO CMMD INPUT	(HI)
H	YAW SERVO 2 LO	68	"	" " " "	(LO)
X	YAW SERVO 1 HI	65	YAW	ACTIVE SERVO CMMD INPUT	(HI)
Y	YAW SERVO 1 LO	66	"	" " " "	(LO)
Z	ROLL SERVO 2 HI	45	ROLL	MONITOR SERVO CMMD INPUT	(HI)
Ā	ROLL SERVO 2 LO	46	"	" " " "	(LO)
B̄	PITCH SERVO 2 HI	20	PITCH	MONITOR SERVO CMMD INPUT	(HI)
C̄	PITCH SERVO 2 LO	21	"	" " " "	(LO)

DFCS PALLET

LEFT GUN BAY

(OSCILLOGRAMS RETOUCHED FOR CLARITY.)

ALL MEASUREMENTS: 0.5 VOLTS/DIV, 1 μSEC/DIV.

PINS A-B

No. 514

PINS X-Y

No. 515

PINS D-E

No. 516

PINS Z-Ā

No. 517

PINS G-H

No. 518

PINS B̄-C̄
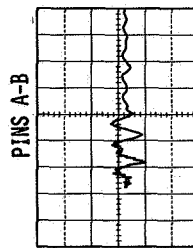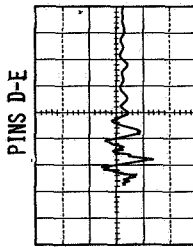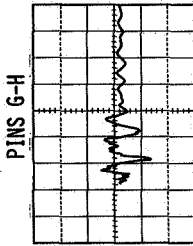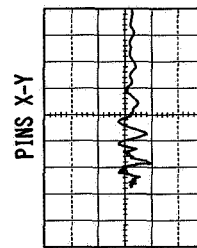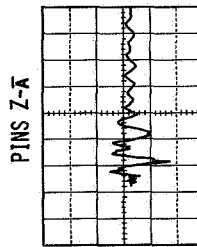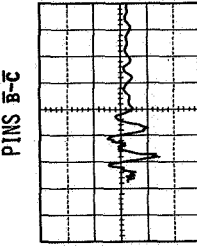
No. 519

FIGURE 7 - INDUCED VOLTAGES ON DFCS DIGITAL CONTROL CIRCUITS AT J2 INTERFACE.
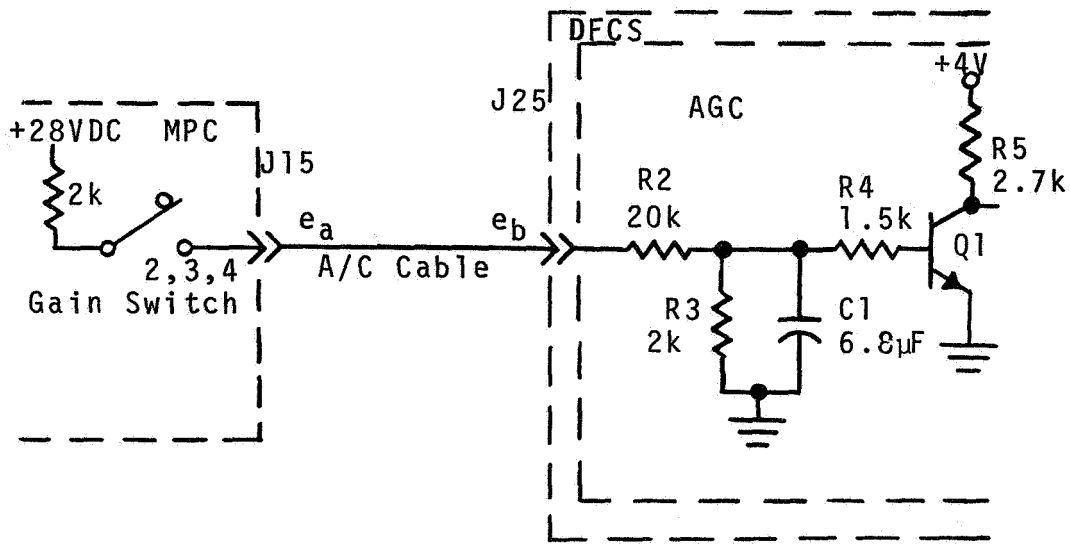
J15 Interface	J25 Interface
e_a average = 1119 $V_{(o-p)}$	e_b average = 677 $V_{(o-p)}$
e_a range = 1065 to 1132 V	e_b range = 566 to 865 V
f = 1.0 MHz	f = 1.0 MHZ
No.of measurements = 5 (J15: Yaw & Roll Gain 4; Not Shown in Figures)	No.of measurements = 3 (J25: Yaw, Pitch & Roll Gain 4 - Fig.6, osc. 505, 500,502)

FIGURE 8 - ATTITUDE GAIN SWITCH POSITION 2, 3, and 4
SIGNAL CIRCUIT FOR DFCS AND MPC INTERFACE.
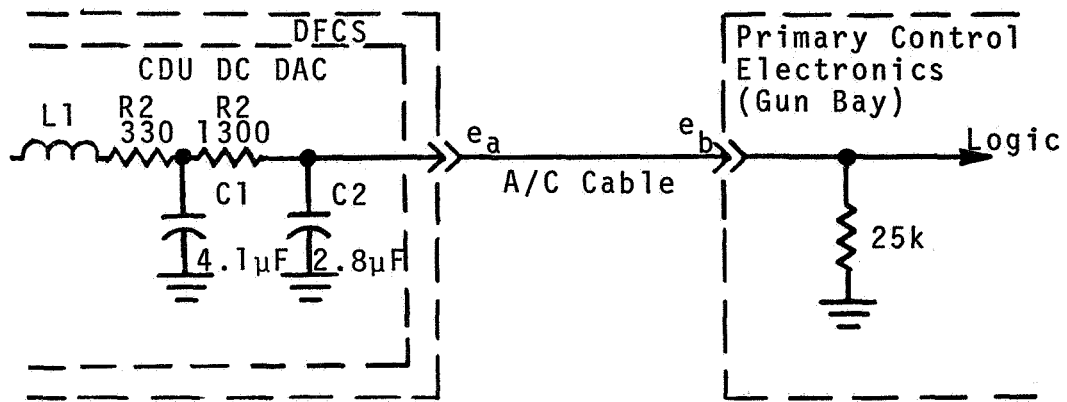
e_a average = 316 $V_{(o-p)}$ e_b average = 355 $V_{(o-p)}$

e_a range = 233 to 400 $V_{(o-p)}$ e_b range = 222 to 422$V_{(o-p)}$

f = 1.0 MHz   1 = 1.0 MHz

No.of measurements = 6   No. of measurements = 3
(J2: Yaw, Pitch and Roll   (Yaw, Pitch and Roll
DACS' 1 and 2, Figure 7)   primary commands)

FIGURE 9 - DFCS DIGITAL CONTROL DIGITAL-TO ANALOG
           CONVERTER OUTPUT SIGNAL CIRCUIT TO
           PRIMARY CONTROL ELECTRONICS.

REDUNDANCY OF HYDRAULIC FLIGHT CONTROL ACTUATORS

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Boeing Commercial Airplane Company

SUMMARY

The constraint of requiring airplanes to have inherent aerodynamic stability can be removed by using active control systems. The resulting airplane requires control system reliability approaching that of the basic airframe. Redundant control actuators can be used to achieve the required reliability, but create mechanization and operational problems. Of numerous candidate systems, two different approaches to solving the problems associated with redundant actuators appear the most likely to be used in advanced airplane control systems.

INTRODUCTION

Future civil aircraft will have to take advantage of all possible gains in aerodynamic efficiency and weight reduction to be economically viable. It has been shown in previous studies by Boeing and others that gains in aerodynamic efficiency and reduction in airplane weight can be achieved by placing the center of gravity aft of the longitudinal maneuver point. The resulting unstable airplane must be augmented through the flight control system to provide acceptable handling qualities. If the stability of the airplane is critical, such that loss of the augmentation would result in loss of the airplane, the control system reliability must approach that of the basic airframe. To meet this level of reliability, special consideration must be given to the control system design. Such considerations include design simplification, derating of components, elimination of electrical connectors, and physical isolation of electrical wiring and hydraulic power. Even then redundancy is usually required to obtain satisfactory reliability from the complex hydraulic actuators and electronic control systems used in airplane flight controls.

Use of redundancy to achieve reliability has always been an accepted engineering design technique. However, the advantages of redundancy are not easily realized in control systems because of signal channel interaction, failure effects, performance degradation after failures, null shift with channel switching and failure detection problems. If force voted multiple hydraulic actuators are used to drive a single load, actuator load sharing also becomes a concern. Methods of insuring proper load sharing can reduce load reaction stiffness, cause poor resolution, and may lead to dynamic instability if not properly designed and built. Monitoring used to effect the orderly shutdown of failed elements may cause inadvertent shutdown of good elements. All of these problem areas with respect to redundant control systems and actuators require careful consideration in control system design and mechanization.

REDUNDANCY REQUIREMENTS

Redundancy requirements for flight control actuation systems can be divided into two areas, the requirement for flutter free control surfaces and the maintenance of critical control surface operation.

The need to minimize airplane weight reduces the permissible use of control surface mass balance as a means of preventing control surface flutter. If mass balance is not used, the surface must be restrained by the surface control system. The Federal Aviation Regulations, Volume III, Part 25, paragraph 25.629, "Flutter, deformation, and fail-safe criteria," requires that an airplane be free from flutter after any single failure in the flight control system, plus any other "reasonably probable" single failure or malfunction affecting flutter. Hydraulic system failures are classified as "reasonably probable" by the FAA. Therefore, when airplane design dictates that control surfaces be restrained by the surface power actuators to avoid the mass balance weight penalty, these requirements dictate a need for at least two surface power actuators and three hydraulic systems for each surface.

Independent of considerations for suppression of surface flutter, surface power actuator redundancy is also influenced by the need to maintain control of the airplane flight path. The Federal Aviation Regulations, Volume III, Part 25, paragraph 25.671, requires, in part, that the airplane must be capable of safe flight and landing after any single failure, excluding jamming, in combination with any probable hydraulic or electrical system failure.

One form of redundancy to assure continuance of control function would be to use multiple aerodynamic surface segments, independently controlled, in each airplane axis. If actuator redundancy were not required for prevention of flutter, each surface could be controlled by a single actuator. Degraded, but safe, operation would be possible if one or more surface segments became inoperable. This feature is used in some current airplanes. However, if the airplane design is such that a limited number of flight control surfaces are available or if all control surfaces in an axis are needed for flight path control, each surface must remain controllable after certain dual control system failures.

Advanced supersonic airplanes will probably be limited in use of control surface redundancy, particularly in the longitudinal axis, because of the need to attain maximum aerodynamic efficiency. The need for minimum weight in an advanced supersonic transport airplane will also limit the consideration of mass balance for flutter prevention. These two factors are sufficient to set the minimum redundancy level for surface power actuators and show the need for redundancy in flight control actuation systems.

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ACTUATOR REDUNDANCY MECHANIZATION

There are two distinct categories of mechanization applied to redundant actuator channels used in aircraft control systems. One type is the parallel active configuration, and the other type is the active/standby configuration. The principle differences between the two types are as follows:

a. Since the parallel active technique implies that the control channels are working together at some point in the control system, the failure of one of the control channels can cause an output performance change. For an active/standby system, the control elements operate independently and failures of the active control element causes transfer to a correctly operating standby channel with no performance degradation.

b. With a parallel active system all of the control channels are working at the same time and the failure of one channel is compensated for by the remaining correctly operating channels (to varying degrees). It is not necessary to rapidly switch the failed channel off. With an active/standby mechanization, rapid transfer between control elements is essential (with the actual required transfer time being determined by the particular application).

There are three options available in mechanization of parallel active systems. The control channels can be brought together and the actuator outputs summed in the following ways:

a. Force voting
b. Velocity summing
c. Position summing

Force voting is the most common technique used in mechanizing parallel active systems. By force voting several actuators on a common output, an output representing the mid value of all input commands can be achieved. Many examples of this type of system exist. The Boeing 747 pitch and roll autopilot actuators (autoland option), and the GE 680J F-4 roll and yaw secondary actuators are typical. One problem with this type of system that does not exist with other types is the force fight that can occur between actuator channels when channels differ in input command or actuator characteristics.

Velocity summing is an alternate parallel active mechanization which does not incur the force fight problems of the force voted systems. Probably the best example of this method is the electromechanical secondary actuator developed by LTV for the 680J F-4 pitch axis. This mechanization uses servo motors summed through differential gear boxes. Net output velocity is the sum of the individual motor velocities and the force output is the sum of the individual force outputs of the servo motors.

Position summing systems have no actuator force fight. However, since the individual actuators are summed by differential linkage, a channel failure or actuator shutdown will reduce total output stroke capability. Each individual actuator must have a larger stroke than the minimum allowable output

stroke to accommodate channel failures. This characteristic restricts the application of the position summing technique to systems that require only small output displacement. It has been used in dual systems for series actuation. Examples are the Boeing 737 dual yaw damper and the dual channel series actuators on the Grumman F-14. Mechanization becomes difficult when more than two actuators are summed because of linkage complexity.

ACTUATOR REDUNDANCY IMPLEMENTATION FACTORS

There are several factors that must be considered when redundant actuators are used. The most significant are those that affect normal operation, operation after failures, and cause interface problems. These are outlined below.

Failure Insensitivity

Failure insensitivity is the ability of a redundant control system to experience failures and automatically continue operation with an acceptable transient. If the system performs a critical function, operation must be maintained in the presence of one or more failures; i.e., be fail operational. However, a fail operational system does not insure minimum control system transients. The criticality of transients has an impact on the detail design of the system. All four methods of redundancy mechanization can be fail operational. However, the number of channels required and failure characterisitcs vary as discussed below:

a. Fail-operational capability can be achieved in parallel active systems by majority voting or averaging three or more active actuators. With three active channels operation continues after the first failure. With four channels operation continues after two failures. In voting systems the first failed channel must be disconnected before the second channel fails for the system to remain operational. In the force voted systems the failed channel is automatically overpowered by the remaining channels and the magnitude of the failure transient can be insignificant. Displacement and velocity summing provide an averaged output but have inherent failure transients and steady state null offset after failure. The magnitude of the transients is dependent upon the system closed loop response.

b. With active/standby systems a failure detection device must assess that the active channel has failed, automatically disconnect it, and switch to a good channel. The failure transient is dependent upon the failure detection level, the switching time and the tracking of the standby channel.

Failure Detection

Detection and indication of failures during operation must be provided so that failed channels or actuators can be disengaged to preserve the integrity of the system. The failure detection system must be designed to detect all types of failures; hardover, passive, and oscillatory and slowovers or ramps which could produce an unsafe situation.

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The ability of the failure detection system to sort out legitimate failures from apparent failures which might occur due to adverse tolerances has an equivalence in reliability. If the failure detection system trips a channel off inadvertently due to an apparent failure, the equivalent mean-time-between-failure (MTBF) for the system may be significantly affected.

Failures in parallel active systems may be sensed by in-line monitoring of actuator characteristics or by cross channel monitoring between active actuators. A method of reducing the number of redundant actuators is to add a model of a working channel and use it for cross channel monitoring. While this extends the system fail operational capability with one less working channel, its effectiveness depends on how well the model matches the actual hardware. In certain applications, where actuators are large and where weight is critical, the model approach may provide a way to minimize the overall weight.

In active/standby systems each channel must be individually monitored for failure detection. Each control channel is usually duplicated or modeled to provide the comparison required to detect failure of the active channel.

Load Sharing

Load sharing is a measure of the ability of multiple actuators with identical inputs to work together in positioning a common output. Load sharing is a problem peculiar to force voted actuators since, obviously, there is no force fighting in an active/standby system when only one system controls at a time, or in position summed and rate summed systems where forces of individual actuators are additive.

Ideally, it is desirable that the load be divided equally among redundant actuators to eliminate any force fighting. However, tracking errors arise due to tolerance buildup in each actuator servo loop and actuator installation that tend to make each actuator seek a unique position even though the input commands are identical. With the actuators tied to a common output all position commands cannot be satisfied and force fight occurs between actuators.

To minimize the force fighting in force voted actuator configurations and assure acceptable sharing of the load, four methods are commonly used:

a. Accurate tolerance control of the feedback loop of the actuator. A mechanical actuator can be mechanized with good tolerance control because of the manufacturing accuracies that are possible and the unchanging nature of the mechanical linkages.

An electrically controlled actuator has command path elements such as summing amplifiers, demodulators, and feedback transducers which can change characteristics with time, temperature, and power. It is generally accepted that the tolerances associated with an electronically controlled actuator are significantly greater than for a mechanically controlled actuator.

b. Compliance between channels. In some applications the structural compliance between actuators allows sufficient individual actuator position difference to reduce force fight through the normal position feedback loop.

c. Low force gain actuators. Low pressure gain servovalves can be used to reduce the force fight resulting from expected valve command differences to an acceptable level. In some applications a feedback path consisting of deflections of the actuators' reaction structure has been sufficient to provide the actuator force gain reduction, and reduced force fight. Another way to reduce actuator force gain is to use actuator pressure as a feedback command. However, there is a limit to the amount of compliance that can be tolerated without reducing the overall actuator stiffness below the minimum allowable level. Reducing actuator force gain (stiffness) has been used successfully where the inputs are reasonably matched, such as a set of surface power actuators signalled by a common mechanical command, or in secondary actuators where the output load is small.

d. Equalization to average load. For cases where the actuators are required to operate into large aerodynamic loads and have uncontrolled input mismatch, any pressure feedback system requires modification to be useful. The individual actuator load must be compared to the average load. Computation of the average load and the individual difference from average requires cross channel comparison. This method does not degrade actuator stiffness but adds complexity and introduces the possibility of cross channel failures.

Input Mismatch

Differences in commands (input mismatch) due to tolerances in an electrical control system, from sensor to actuator, can be quite high, as much as a quarter of full scale command, unless some design precautions are taken to prevent such buildup. It should be noted that differences in commands generated by actuator loop tolerances are an order of magnitude less than those generated by computational elements in the upstream portions of the system. The various methods of redundant actuator mechanization deal with the input mismatch problem as itemized below.

a. Force Voting Systems. In force voted systems the output is the mid value of all input commands. The force fight that occurs due to input command mismatch can be reduced by the same methods used to insure load sharing. In some applications the only possible means of controlling command differences may be the use of electronic signal conditioning to reduce the input mismatch.

b. Velocity Summing Systems. Velocity summed actuators allow the individual channels to cancel command differences by differentially summing rates.

c. Position Summing Systems. Position summed actuators give a single output which is the average of the input commands.

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d. Active/Standby Systems. Usually the active actuator is commanded by a single electronic channel and mismatch is of no concern during operation. Mismatches between the commands of the active and the standby channel are of concern, however, and must be minimized to avoid large surface transients upon switching from active to standby actuators.

SECONDARY ACTUATORS

Surface actuator input signals can be either electrical or mechanical. A dual load path mechanical signal to three power actuators can satisfy all reliability requirements. However, the control signals for critical stability augmentation or fly-by-wire systems will be electrical.

The power associated with the electronic signals for fly-by-wire command, autopilot, and stability augmentation systems must be kept at low levels as a matter of good design. These low level signals are required to command surface actuators that operate at high power levels. To transform the electrical commands to surface displacements controlled by large hydraulic power actuators requires several stages of amplification. Review of current redundant flight control actuation systems shows an almost universal use of small electrically signaled hydraulic actuators as one of the stages of amplification. These small actuators are termed secondary actuators.

It is advantageous to treat the command path and computation and power actuation errors independently by inserting a synchronizing stage between the two functions. The synchronizing stage provides a single valued command and may be an electronic voter or a mechanical output of a secondary actuator arrangement. Some of the advantages of synchronizing are:

a. When surface power actuators are isolated from the upstream command differences, the task of providing adequate power actuator load sharing becomes easier, permitting a simpler and more reliable mechanization.

b. When secondary actuators are used to provide the synchronizing stage they do not eliminate the problems of redundant actuators but the magnitude of the problems are less severe because the secondary actuators operate at significantly lower force levels than the surface power actuators.

SYSTEM SELECTION

Four types of actuator redundancy have been discussed. It has also been shown that prevailing control system designs use secondary actuators as a stage of signal amplification and as a means of command path synchronization. Surface power actuators are usually force voted mechanical input actuators. The system differences are in the redundancy mechanization of the secondary actuators. Active/standby and force voted systems predominate by a large margin with force voted systems the most common.

Although the use of velocity summing solves the problem of force fight there are disadvantages which make this type of system a questionable candidate for future use in critical flight control applications on civil aircraft. The complex gearing could make it difficult to prove that jam-type failures would be extremely remote, as required by FAA regulations. Also, for the same output force the electromechanical actuator is larger and heavier than an equivalent electrohydraulic actuator. One advantage would be the availability of four independent actuator signals in an airplane with only three hydraulic signals. Another advantage for military aircraft is the reduced vulnerability to loss of hydraulic systems.

Position summed systems are difficult to mechanize for more than two redundant channels because of the complex linkage required. In addition the loss of rate and travel capability after failure and the inherent output position transient that occurs with failure are also disadvantages.

The active/standby and the force voted systems have advantages and disadvantages that must be considered in conjunction with the specific airplane and control system application. The most significant differences between the two types of systems are:

Normal Performance

The single channel operation of the active/standby system can give optimum performance. In the force voted system residual actuator force fight can affect output resolution and reduce actuator stiffness.

Failure Transients

Force voted systems can be mechanized to give very small failure transients. The active/standby system must trade failure detection levels and nuisance trips against the allowable failure transient.

Performance After Failure

The active standby systems preserve normal performance in the failure sequence from the active channel to the standby channel and on to the second standby channel. The force voted system may suffer a performance degradation as it fails down. This degradation can be exhibited as reduced resolution capability and force output.

Failure Detection

The active/standby concept requires immediate failure detection to be safe following failures. The force voted concept does not require immediate detection of a failure to be safe. Failure detection is only required to enable a failed channel to be shut down before another failure occurs.

Switching

Each standby channel must be continually monitored to assure that it is

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capable of control if the active channel fails. Further, somewhere in the system a device like a switch or blocking valve is required to operate without prior knowledge of its condition to provide a successful transfer to a standby channel. Force voted systems are comprised of only active channels continually monitoring each other and require no immediate switching to be safe.

CONCLUDING REMARKS

Advanced technology airplanes will require redundant flight control actuators to achieve reliability because operational stability augmentation system will be essential for safe flight and acceptable airplane handling qualities.

Surface restraint to meet the fail safe requirements for flutter prevention and minimum safe controllability requirements will dictate the minimum redundancy levels for control surface power actuators. Airplanes with redundant flight control surfaces may have dual surface power actuators if a third hydraulic system is provided. Control surfaces that are critical for control functions will require at least three actuators per surface in order to meet FAA requirements and provide an adequate level of safety.

Reliability requirements for control systems that amplify autopilot, stability augmentation, and pilot commands and provide inputs to the control surface power actuators are determined by the need to remain operational in spite of control channel malfunctions. Actuation systems with fault corrective capability that will meet the system reliability requirements and satisfy FAA regulations require at least four active channels or three monitored channels. Surface power actuators could be mechanized with this level of redundancy but it has been found to be more efficient to utilize small secondary actuators to provide a reliable single valued mechanical input to three surface power actuators of reduced complexity.

Based on a review and examination of current redundant actuation systems, two concepts were found to be representative of secondary actuator mechanization which meet advanced civil airplane flight control system requirements. The two actuator configurations are a four channel force voted system and a three channel active/standby system. Both of these systems should be considered since they reflect different design philosophies.

Redundant control systems have operating and failure characteristics that are affected by overall control system and airplane design. Redundant actuators should be studied in conjunction with pilot and airplane to understand pilot reaction and airplane response to variations in control system characteristics and failures.

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ACTIVE CONTROL TECHNOLOGY AND THE USE OF

MULTIPLE CONTROL SURFACES

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SUMMARY

Needed criteria for active control technology applications in commercial transports are lacking. Criteria for redundancy requirements, believed to be consistent with certification philosophy, are postulated to afford a discussion of the relative value of multiple control surfaces. The control power and frequency bandpass requirements of various active control technology applications are shown to be such that multiple control surfaces offer advantages in minimizing the hydraulic or auxiliary power for the control surface actuators.

INTRODUCTION

There is a dearth of criteria to aid in the design of flight control systems for commercial transport aircraft which include active control technology (ACT) applications. Such criteria are necessary, however, to permit an orderly design development without fear of costly redesign, as might result from special conditions imposed after the aircraft design was committed to take advantage of ACT. The Federal Air Regulation for transport aircraft, amendment 25-23, sets forth a number of failure tolerance requirements for flight control systems. Paragraph 25.671(C) states, "the airplane must be shown by analysis, test, or both, to be capable of continued safe flight and landing after any of the following failures or jamming in the flight control system and surfaces... 1) Any single failure, excluding jamming... 2) Any combination of failures not shown to be extremely improbable, excluding jamming... 3) Any jam...unless the jam is shown to be extremely improbable, or can be alleviated." Paragraph 25.672 says, "If the functioning of stability augmentation or other automatic... system is necessary to show compliance with the flight characteristics requirements of this Part, such systems must comply with...the following: a) A warning...must be provided for any failure...which could result in an unsafe condition if the pilot were not aware of the failure... b) The design...must permit initial counteraction of failures...by either deactivation of the system...or by overriding the failure by movement of the flight controls in the normal sense... c) It must be shown that after any single failure...the aircraft is safely controllable...at any speed or altitude within the approved operating limitations..."

These regulations, while not known to be written with active control technology applications in mind, may well cover the subject. Certainly, ACT applications will not have less demanding requirements. Considerations of operational, maintenance and cost aspects of potential system redundancy approaches,

necessary to meet these failure tolerance requirements, leads to the conclusion that split surfaces offer unique advantages in mechanizing many ACT applications.

POSTULATED CRITERIA

In the absence of specific regulatory requirements, failure criteria which are believed to be consistent with certification philosophy are postulated and presented in table 1. For each ACT application function the failure requirements for the flight control system, under the heading of Redundancy, are given for several different aircraft designs graded according to the consequence of loss of the ACT function. The failure requirements for the several ACT functions are considered minimum in each case, and are based on the assumption that only that ACT function is involved. In reality, it is difficult to visualize an aircraft designed to utilize only one ACT function; where more than one function is involved it is obvious that the more stringent redundancy requirement would prevail. It should be noted that a failure warning is given to the pilot at each failure level to meet the FAR requirements. It is assumed, in at least some cases, that the operating envelope would be restricted to some defined level following each indicated failure.

An aircraft employing a pure fly-by-wire control system (which is not considered to be an ACT application per se) requires extremely high reliability in the entire flight control system. Such aircraft will likely have no less than two fail-operate redundancy and, as such, might profitably employ ACT applications with only relatively slight increases in the control system complexity. Once the commitment is made to inalterably depend upon the functioning of the sensors, computers, actuators and control surfaces, it makes little difference to safety as to how uncontrollable or structurally sound the aircraft is without the control functions. (In such cases, restricting the operating envelope may be moot.) However, it is in such cases that the full benefits of ACT, in terms of reduced direct operating cost and increased return on investment, will be realized.

CONSEQUENCES OF MULTIPLICITY

The multiplicity of flight control components, channels and power sources to achieve the operational reliability required does not come without its price. The price is in terms of equipment, but it is also in terms of pre-flight tests to establish that there are no latent failures and in maintenance action required by actual failures or false alarms. An STI report, "TFX Handling Quality and Flight Control System Study" (AD 447909L) published in August 1963, is recommended as an excellent reference which "facilitates tradeoffs between potential competing mechanizations" of redundancy in automatic flight control systems. Included in this paper is a matrix of practical redundant mechanizations versus major operational and maintenance qualities. From the data given it is evident, assuming a control surface pulse can be tolerated as the result of switching after a failure is detected, that an active/standby

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TABLE 1

ACT Application	Aircraft design such that loss of function results in:	Redundancy
Relaxed Stability	Undesirable handling qualities	Fail Soft
	Unacceptable handling qualities	Fail op/Fail Soft
	Uncontrollable	Fail op/Fail op
Maneuver Load Control or Gust Load Alleviation	Dynamic and static loads less than limit loads	Fail Soft
	Dynamic and static loads less than ultimate loads	Fail op/Fail Soft
	Dynamic and static loads greater than ultimate loads	Fail op/Fail op
Flutter Mode Control	Flutter predicted above V_D/M_D	Fail Soft
	Flutter predicted above V_C/M_C	Fail op/Fail Soft
	Flutter predicted below V_C/M_C	Fail op/Fail op
Fatigue Life Improvement	Fatigue damage rate increase	Fail Soft or Fail op/Fail Soft
Ride Quality Control	Undesirable ride qualities	Fail Soft
	Unacceptable ride qualities	Fail op/Fail Soft

approach offers significant reductions in the probability of complete failure as compared to triple channel configurations. Even more impressive, the mean time between maintenance actions due to actual failures and false alarms is reduced by an order of magnitude (assuming the same mean time between failures for like components in each case).

The use of split surfaces with active/standby actuator redundancy for each offers an additional feature, namely that uninterrupted operation is assured after any single failure. After any second failure, uninterrupted operation is also assured but with a one in three chance (or less) that reduced performance (authority) will result. The eventuality of a possible lower authority after a second failure may be accommodated by selecting the original authorities above actual requirements, adjusting system parameters after the original or second fault or possibly by operational restrictions after the original or second fault.

MULTIPLE CONTROL SURFACES

The use of multiple control surfaces for individual axes of an aircraft has a long history. Trim controls that are separate from the primary maneuvering controls, for example, is a concept that has been used for many generations; more recently, split controls such as upper and lower rudders and inboard and outboard elevators are not uncommon. There are a variety of reasons why multiple control surfaces have been used including advantage from consideration of auxiliary power demands, operational safety, manufacturing costs (particularly on large aircraft), and flutter characteristics, in addition to accommodating the flight control failure tolerance requirements. When used for failure tolerance reasons, the multiple control surfaces in any axis must be sized such that the total authority exceeds the minimum requirement by some margin. Otherwise, the whole philosophy is fallacious, being analogous to a multi-engine aircraft in which the loss of any one engine results in an inability to continue to fly. This raises the question of what is the minimum authority required. A quantitative answer is strongly dependent upon the aircraft configuration and which, if any, ACT applications are involved. Some general-trend type observations can be made, however, which bear on the use of split surfaces for ACT.

Consider the auxiliary power required for a flight control surface servo or actuator. If it is assumed that a constant pressure hydraulic power source is used, the peak power supplied is simply proportional to the flow demand.

$$\text{Power supplied} = K\,P_S\,Q$$

where K is a constant, P_S is the supply pressure and Q is the flow rate. For a given stroke actuator, the area of the actuator is proportional to the maximum hinge moment (assuming the acceleration forces are small). Flow is the product of actuator area times rate, or proportional to actuator area times surface rate. Thus

$$\text{Power supplied} = K_1\,\dot{\delta}_{max}\,H\,M_{max}$$

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where K_1 is a new constant, $\overset{\bullet}{\delta}_{max}$ is the maximum surface rate and HM_{max} is the maximum hinge moment. If ω is approximately the maximum frequency (in radians per second) at which the control surface in question must respond with some maximum deflection δ_{max}, the power supplied expression may be written as

$$K_1 \ \omega \ \delta_{max} \ HM_{max}$$

(Please note that HM_{max} and δ_{max} typically do not occur at the same flight condition.)

Qualitative values for the factors in this expression are presented in table 2 and have a number of implications. As a reference point, the "bandpass" frequency factor (ω) is shown as "low" and the maximum "control power" factor ($\delta_{max}HM_{max}$) is shown as "high" for the basic maneuvering of the aircraft. The corresponding hydraulic power (P) is the power required assuming no stability or control augmentation. Designing the aircraft with relaxed stability and restoring this stability with augmentation results in a higher required bandpass, although the authority required for the augmentation function may be somewhat lower than that required for the basic maneuverability. The peak hydraulic power for the augmentation function alone is higher than that for the actuation system to drive the baseline control surfaces. In many current aircraft, the control surfaces for these functions are one and the same, with the result that the peak hydraulic power is greater than the reference "P." In view of the power requirements (and compatability of bandpass requirements), it is not uncommon that relaxed stability, fatigue life improvement, and maneuver load control form a set of ACT functions which are considered together.

When ride quality control and gust load alleviation are considered, particularly where the higher frequency gusts are significant, the bandpass and peak hydraulic power requirements are considerably increased. These two ACT functions make another logical set. The bandpass requirements for flutter mode controls, except for isolated or specifically contrived cases, are relatively very high with the result that this ACT application stands alone. It is evident that gust load alleviation and ride quality control, and flutter mode control applications, are prime candidates for dedicated control surfaces. If a single control surface, with its high control power, were used for basic maneuvering and for an ACT function with a high or very high bandpass requirement, the peak hydraulic power demands for the actuators would be extremely high. As indicated by the control powers needed for the ACT functions in table 2, the use of smaller or partial control surface authorities can meet these needs, minimize the hydraulic power required, and minimize the effects of surface pulses or even a hard over, as might result in a failure or multiple failure condition.

One possible flight control design approach suggested by multiple control surface considerations would utilize a set of three-axis multiple control surfaces, with the possible addition of direct lift and side force controls, for maneuvering the aircraft either by the pilot or autopilot. Stability augmentation, maneuver load control and fatigue life improvement functions can conveniently use part of these multiple control surfaces. The servo actuators for those surfaces which are used for stability augmentation or fatigue life

TABLE 2

	Bandpass	Control Power	Auxiliary Power
Basic Maneuvering	Low	High	P
ACT Function			
Relaxed Stability	Medium	Medium	$>$P
Maneuver Load Control	Low	Medium	$<$P
Gust Load Alleviation	High	Medium	$\gg$P
Flutter Mode Control	Very High	Low	$\gg$P
Fatigue Life Improvement	Medium	Medium	$>$P
Ride Quality Control	High	Medium	$\gg$P

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improvement require a higher bandpass than the maneuvering control surfaces actuators. But by using only a portion of the maneuvering controls for these purposes, the hydraulic power demands are significantly reduced compared to a non-multiple surface design.

If the ACT functions of ride quality control and gust load alleviation are added, they might also use portions of the basic maneuvering control surfaces but separate, "dedicated," surfaces located more optimumly would likely be desirable from a system weight and power demand standpoint. The desired location and required high-frequency response of control surfaces providing flutter mode control will, in all likelihood, necessitate separate dedicated surfaces for this ACT function. In any case, the possible use of any "dedicated" control surfaces as ultimate backups to the basic maneuvering controls is an attractive possibility.

The "fullness of time" for ACT applications has arrived. Improved aircraft efficiency in meaningful measures can be achieved and the use of multiple control surfaces can contribute significantly to this achievement without compromising safety or creating a "hanger queen."

APPENDIX

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CONFEREES

ABEL, I.	NASA Langley Research Center
ABERCROMBIE, J. M.	McDonnell Aircraft Co.
ACKERMAN, J. S.	Northrop Corp., Aircraft Div.
ADAMS, A. L.	Hydraulic Research & Manufacturing Co.
A'HARRAH, R. C.	Naval Air Systems Command
AIKEN, W. S., Jr.	NASA Headquarters
ALEXANDER, D.	FAA
ALFORD, W. J., Jr.	NASA Langley Research Center
ALLEN, R.	FAA
AMES, L. E., Jr.	McDonnell Aircraft Co.
ANDERSON, C. A.	General Dynamics Corp., Fort Worth Div.
ANDES, J. P.	Calspan Corp.
ANDREWS, W. S.	Rockwell International, Columbus Aircraft
ANDRISANI, D.	Calspan Corp.
ARNOLD, J. E.	Air Force Flight Test Center
ARNOLD, J. I.	The Boeing Co., Wichita Div.
ASHKENAS, I. L.	Systems Technology, Inc.
ASSEO, S. J.	Calspan Corp.
ASTON, P.	Honeywell Inc.
AYERS, J. L.	NASA Flight Research Center
BAILEY, A. L., Jr.	Honeywell Inc.
BAIRNSFATHER, R. R.	The Charles Stark Draper Laboratory, Inc.
BAKER, A. N.	Lockheed-California Co.
BARBERO, A. T.	Grumman Aerospace Corp.
BARDICK, C. D.	Litton Industries
BARNES, K. H.	Honeywell Inc.
BARRACLOUGH, A. B.	Douglas Aircraft Co.
BATTERTON, P. G.	NASA Lewis Research Center
BAUER, C. A.	NASA Flight Research Center
BAYATI, J. E.	Rockwell International Corp.
BEAVIN, R. C.	Air Force Flight Dynamics Laboratory
BECK, D. M.	
BEELER, D. E.	NASA Flight Research Center
BELL, R. W.	Naval Postgraduate School
BERG, R. A.	Douglas Aircraft Co.
BERNHARD, F. L.	E-Systems, Inc., Montek Div.
BERRY, D. T.	NASA Flight Research Center
BERTHE, C. J.	
BIGHAM, J. P.	NASA Lyndon B. Johnson Space Center
BILLIG, L. O.	The Boeing Aerospace Co.

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BIRD, D. K.   Air Force Flight Dynamics Laboratory
BISCHOFF, D. E.   Naval Air Test Center
BITTENBINDER, D. M.   Air Force Flight Test Center
BLACK, J. M.   NASA Flight Research Center
BLACK, R. E.   Douglas Aircraft Co.
BLACKALLER, D.   FAA
BOHLING, R. F.   Jet Propulsion Laboratory, California
                               Institute of Technology
BOLD, C. E.   General Electric Co.
BOLTON, B. A.   Boeing Commercial Airplane Co.
BOONE, J. H.   Boeing Commercial Airplane Co.
BORGESON, R. D.
BORLAND, C. J.   Rockwell International Corp.
BOWLING, T. C.   The Boeing Co., Wichita Div.
BRANCH, P. R.   Air Force Flight Dynamics Laboratory
BRECHT, M. W.   United Air Lines
BRETOI, R. N.   NASA Ames Research Center
BROKASKI, R. W.   Delco Electronics Div., Santa Barbara
                               Operations
BROWN, R. B.   The Boeing Co.
BROWN, S. R.   NASA Flight Research Center
BRUCE, T. F.   General Electric Co.
BRYSON, A. E., Jr.   Stanford University
BUCKLEY, R. W.   Sundstrand Aviation
BUELL, D. R.   FAA
BUHL, M. L., Jr.   NASA Lyndon B. Johnson Space Center
BURCHAM, F. W., Jr.   NASA Flight Research Center
BURTON, C. L.   LTV Aerospace Corp., Vought Systems Div.
BUSCHER, R. G.   Lear Siegler, Inc., Astronics Div.
BYFORD, F.   Hydraulic Research & Manufacturing Co.
CAMPBELL, D. H.   Lockheed-California Co.
CAPASSO, V. N.   NASA Flight Research Center
CARLINE, A. J. K.   General Dynamics Corp., Fort Worth Div.
CAVAGE, R. L.   Rockwell International, Los Angeles Aircraft Div.
CAZIER, F. W., Jr.   NASA Flight Research Center
CHALK, C. R.   Calspan Corp.
CHANDLER, R. L.   The Boeing Aerospace Co.
CHENEY, D.   FAA
CHENOWETH, C. C.   The Boeing Co.
CHEW, F. E.   LTV Aerospace Corp., Vought Systems Div.
CHOO, D. K.   Northrop Corp., Aircraft Div.
CLIFF, E. M.   Virginia Polytechnic Institute and State
                               University
COBB, J. W.   Collins Radio Co.
COHEN, G. C.   Boeing Commercial Airplane Co.
COMAN, R.   Grumman Aerospace Corp.
CONNER, D. W.   NASA Langley Research Center
COOK, J. L.   Douglas Aircraft Co.
COOPER, A. S.   USAF, Norton Air Force Base
COTTON, L. S.   Sikorsky Aircraft
COYKENDALL, R. E.   United Air Lines

CRAFT, J. B.	NASA Flight Research Center
CRANE, R. C.	Air Force Flight Test Center
CRAWFORD, R. D.	Honeywell Inc.
CRAWFORD, R. E.	Summa Corp.
CREER, B. Y.	NASA Ames Research Center
CROTHER, C. A.	Rockwell International, B-1 Div.
CURTIS, A. R.	Lockheed-California Co.
DAMMAN, L. M.	Air Force Flight Test Center
DASHCUND, D. E.	Princeton University
DAVENPORT, F. J.	Boeing Commercial Airplane Co.
DAVEY, J. A.	Continental Airlines
DAVIES, J. B.	Sperry Flight Systems Div., Sperry Rand Corp.
DAVIS, L. I.	Lockheed Aircraft Corp.
DAVIS, W. J.	Lockheed-California Co.
DeANGELIS, V. M.	NASA Flight Research Center
DeCARLI, H. E.	Lockheed-California Co.
DEETS, D. A.	NASA Flight Research Center
DENNIS, W. V.	Northrop Corp.
DeTALLY, W. A.	Douglas Aircraft Co.
DILCHER, H. S.	Lockheed-Georgia Co.
DISNEY, T. E.	Lockheed-Georgia Co.
DIVES, W. E.	NASA Flight Research Center
DIVOLA, J. M.	Douglas Aircraft Co.
DOGGETT, R. V., Jr.	NASA Langley Research Center
DOLBIN, B.	Lear Siegler, Inc., Astronics Div.
DOOLITTLE, J. H.	Air Force Flight Test Center
DOWNING, J. W.	Douglas Aircraft Co.
DRAZIN, L. A.	Hydraulic Research & Manufacturing Co.
DUFFY, R. T.	NASA Wallops Flight Center
DUKES, T. A.	Princeton University
DUSTO, A. R.	Boeing Commercial Airplane Co.
ECKHOLDT, D. C.	USAF, Aeronautical Systems Div.
EDINGER, L. D.	Honeywell Inc.
EDWARDS, J. W.	NASA Flight Research Center
ELKIN, H. W.	Rockwell International, Los Angeles Aircraft Div.
ELLIS, D. R.	Princeton University
ELSANKER, W. K.	
EMERSON, L. D., Jr.	Pratt & Whitney Aircraft
ENEVOLDSON, E. K.	NASA Flight Research Center
ERICKSON, J. A.	Honeywell Inc.
FAIRCHILD, J. E.	University of Texas at Arlington
FEDZIUK, H. A.	NASA Langley Research Center
FELLEMAN, P. G.	The Charles Stark Draper Laboratory, Inc.
FENG, T. Y.	Celesco Industries Inc.
FINCH, R. E.	Calspan Corp.
FLAIG, J. W.	Naval Air Systems Command
FLANAGAN, R. D.	IBM Federal Systems Div.
FLANDERS, T. A.	Hamilton Standard, Div. United Aircraft
FLYNN, G. R.	The Boeing Co.
FLYNN, J. J.	Lear Siegler, Inc., Astronics Div.
FOLEY, W. D.	General Electric Co.

FORNELL, G. E.   USAF, Headquarters
FOSS, H. C., Jr.   Douglas Aircraft Co.
FOSS, R. L.   Lockheed Aircraft Corp.
FOSTER, J. L.   Collins Radio Co.
FOURNIER, L. E.   General Electric Co.
FRANKLIN, E. T.   Honeywell Inc.
FREDERICKSEN, A. D.   NASA Flight Research Center
FRISBEE, L. E.   Lockheed Aircraft Corp.
FUNK, R. A.
GADBOIS, R. G.   Lear Siegler, Inc., Astronics Div.
GALLAGHER, J. T.   Northrop Corp.
GARCIA, J. O.   Douglas Aircraft Co.
GARCIA, S. X.   Hydraulic Research & Manufacturing Co.
GARDNER, W. G. B.   Pratt & Whitney Aircraft
GARVIN, J. B.   Burroughs Corp.
GATLIN, D. H.   NASA Flight Research Center
GAYLOR, R.   Sperry Flight Systems Div., Sperry Rand Corp.
GEE, S. W.   NASA Flight Research Center
GEORGE, F. L.   Air Force Flight Dynamics Laboratory
GERKE, P. D.   NASA Lyndon B. Johnson Space Center
GILYARD, G. B.   NASA Flight Research Center
GLENN, J. E.   Northrop Corp., Aircraft Div.
GOECKE, S. A.   NASA Flight Research Center
GOGIA, J. K.   Lear Siegler, Inc., Power Equipment Div.
GOLDBERG, J.   Stanford Research Institute
GOLDSTEIN, S. M.   Sundstrand Aviation
GOODERUM, H. G., Jr.   USAF, Norton Air Force Base
GORHAM, J. A.   Gorham Associates, Aviation Consultants
GREENBERG, R. I.   Jet Propulsion Laboratory, California
                                     Institute of Technology
GREENLEY, B. R.   Lear Siegler, Inc., Astronics Div.
GRIMESEY, R. P.   IBM Federal Systems Div.
GROSSER, W.   Fairchild Republic Co.
GROWER, R. G.   Douglas Aircraft Co.
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HALBERT, J. T.   Douglas Aircraft Co.
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HAMILTON, W. T.   The Boeing Aerospace Co.
HAMMELL, A. K.   Honeywell Inc.
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HARGROVE, W. J.   Lockheed-Georgia Co.
HARRIS, L. A.   NASA Headquarters
HARRIS, R. B.   Douglas Aircraft Co.
HARSCHBURGER, H. E.   McDonnell Douglas Corp.
HARSH, R. C.   Kavlico Electronics, Inc.
HART, J. E.   Lockheed-Georgia Co.
HAUSER, F. D.   Martin Marietta Aerospace, Denver Div.
HAZAN, S. I.   Calspan Corp.

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HEIMBOLD, R. L.	Lockheed-California Co.
HELFMAN, S.	Fairchild Republic Co.
HEMMERLY, R. A.	Naval Ship Research and Development Center
HERMANN, P. J.	Iowa State University
HEYMAN, A. C.	Grumman Aerospace Corp.
HILDEBRAND, R. R.	Air Force Flight Test Center
HILL, D. K.	Lockheed-California Co.
HILL, D. P.	The Garrett Corp.
HILL, E. C.	General Electric Co.
HOBLIT, F. M.	Lockheed-California Co.
HOEY, R. G.	Air Force Flight Test Center
HOH, R. H.	Systems Technology, Inc.
HOLLANDER, G. L.	Hollander Associates
HOLLEMAN, E. C.	NASA Flight Research Center
HOLLOWAY, B. K.	NASA Headquarters
HOLLOWAY, R. B.	The Boeing Co., Wichita Div.
HOLMBERG, G. M.	Gates Learjet Corp.
HOOD, R. V.	NASA Headquarters
HOWLAND, D. C.	Cadillac Controls
HUBER, R. E.	Teledyne Ryan Aeronautical
HULLETT, W. E.	University of California at Los Angeles
HUMPHREY, B. G.	Lear Siegler, Inc., Astronics Div.
HUNTER, J. E.	Air Force Flight Dynamics Laboratory
HUPPERT, L. D.	The Garrett Corp.
HUS, R. D.	Hydraulic Research & Manufacturing Co.
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HWANG, C.	Northrop Corp.
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IRELAND, L. E.	Lockheed Aircraft Corp.
IRWIN, H.	FAA
JACKSON, R. B.	NASA Flight Research Center
JACKSON, R. P.	Northrop Corp.
JACOBSON, I. D.	University of Virginia
JACOBUS, F. B.	Collins Radio Co.
JAMES, H. A.	Teledyne Ryan Aeronautical
JARVIS, C. R.	NASA Flight Research Center
JENKS, G. E.	University of Kansas
JOHANNES, R. P.	Air Force Flight Dynamics Laboratory
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JOHNSON, R. G.	NASA Flight Research Center
JOHNSON, R. J.	U.S. Army Aviation Systems Command
JOHNSON, S. H.	Lehigh University
JOHNSTON, D. E.	Systems Technology, Inc.
JOHNSTONE, R. D.	Air Force Flight Test Center
JONES, G. L.	Air Force Flight Test Center
JONES, K. E.	LTV Aerospace Corp., Vought Systems Div.
KACHEL, G. J.	Lockheed-California Co.
KADLEC, P. W.	Continental Airlines
KALVISTE, J.	Northrop Corp.

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KAPLAN, M.	Consulting Aerospace Engineers
KAPLAN, S. J.	Pratt & Whitney Aircraft
KAUFFMAN, R. L.	Douglas Aircraft Co.
KAUFMAN, L. A.	Teledyne Systems Co.
KENNEDY, T. W., Jr.	
KESLER, D. F.	Northrop Corp., Aircraft Div.
KESTEK, R. E.	The Boeing Co.
KEY, D. L.	U.S. Army, NASA Ames Research Center
KEY, L. W.	The Boeing Co., Wichita Div.
KIDDER, R. C.	
KILP, S. G.	Air Force Flight Test Center
KIRSTEN, P. W.	Air Force Flight Test Center
KLAPPROTT, R. L.	Gates Learjet Corp.
KLOTZ, E.	Lear Siegler, Inc., Astronics Div.
KNIAT, J.	Pratt & Whitney Aircraft
KNUUSI, O. A.	Lockheed-California Co.
KOEPCKE, W. W.	Northrop Corp., Aircraft Div.
KOLK, F. W.	American Airlines, Inc.
KORDES, E. E.	NASA Flight Research Center
KOROCH, R. W.	Douglas Aircraft Co.
KOZIOL, D. E.	Rockwell International Corp.
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KREIMENDAHL, B. W.	Consulting Aerospace Engineers
KRIER, G. E.	NASA Flight Research Center
KUEHNE, H. W.	Douglas Aircraft Co.
KUHLBERG, J. F.	Pratt & Whitney Aircraft
KULEWICZ, H.	FAA
KURZHALS, P. R.	NASA Headquarters
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KYLE, R. G.	NASA Langley Research Center
LAMAR, W. E.	Air Force Flight Dynamics Laboratory
LANE, R. E.	Hughes Helicopters
LANG, G. E.	Bendix Research Laboratories
LANGE, R. H.	Lockheed-Georgia Co.
LAPINS, M.	University of Virginia
LARSEN, H. N.	Boeing Commercial Airplane Co.
LARSON, R. R.	NASA Flight Research Center
LASLEY, J. M.	Northrop Corp., Aircraft Div.
LAZARUS, A.	Aerospace Corp.
LEE, A. H.	The Boeing Co.
LEONARD, J. B.	Grumman Aerospace Corp.
LEWIN, N. N.	Grumman Aerospace Corp.
LEWIS, R. B. II	U.S. Army Aviation Engineering Flight Activity
LEWOLT, J. G.	Lockheed-California Co.
LEY, A. C.	Rockwell International, Los Angeles Aircraft Div.
LIECHTY, H. B.	Rockwell International, Los Angeles Aircraft Div.
LIPPIS, R. L.	FAA
LOCK, W. P., Jr.	NASA Flight Research Center
LOCKENOUR, J. L.	Air Force Flight Dynamics Laboratory
LOFTIN, L. K.	NASA Langley Research Center

LOOME, R. G.	Sperry Flight Systems Div., Sperry Rand Corp.
LOSCHKE, R. C.	Lockheed Aircraft Corp.
LUCERO, F. N.	Air Force Flight Test Center
LUNDQUIST, L. E., Jr.	Bertea Corp.
LUTZE, F. H.	Virginia Polytechnic Institute and State University
LYDDANE, G. H.	FAA
LYON, W. J.	Philco Houston Operations
MABLI, R. A.	Lear Siegler, Inc., Astronics Div.
MACKAL, C. H.	
MADAYAG, A. F.	FAA
MADDEN, P. A.	The Charles Stark Draper Laboratory, Inc.
MAGGOS, G. P.	Naval Air Systems Command
MAINE, R. E.	NASA Flight Research Center
MALLOY, W. A.	Delco Electronics Div., Santa Barbara Operations
MANINGER, H. E.	Northrop Corp., Aircraft Div.
MARTIN, R. E.	General Dynamics Corp., Convair Div.
MARTIN, W. A.	Northrop Corp.
MARX, M. F.	General Electric Co.
MASTERS, G. W., Jr.	Aerospace Corp.
McCALLON, L. K.	Air Force Flight Test Center
McCORMACK, W. H.	AiResearch Manufacturing Co., Div. of The Garrett Corp.
McDONNELL, J. D.	Douglas Aircraft Co.
McELREATH, K. W.	Air Force Flight Dynamics Laboratory
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McRUER, D. T.	Systems Technology, Inc.
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MELLEN, D. L.	Control Data Corp.
MENDEZ, J. M.	Aerospace Corp.
MERKEL, E. W.	Cessna Aircraft Co., Wallace Div.
MERRIMAN, R.	Northrop Corp.
MEYER, G.	NASA Ames Research Center
MILLER, W. B.	USAF, Aeronautical Systems Div.
MINOTT, G. M.	Rockwell International Corp.
MOLLOY, R. E.	Bertea Corp.
MONSON, G. D.	Delco Electronics Div., Santa Barbara Operations
MONTGOMERY, T. D.	NASA Flight Research Center
MOORE, K. A.	Continental Airlines
MORGAN, D. E.	McDonnell Aircraft Co.
MORI, A. S.	Rockwell International, B-1 Div.
MORRIS, E.	FAA
MULLER, A. F.	USAF, Norton Air Force Base
MURRAY, R. C.	United Air Lines
MUSE, T. C.	Department of Defense, DDR&E
NAGATA, J. I.	U.S. Army Aviation Engineering Flight Activity

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NAGY, C. J.   Air Force Flight Test Center
NARDI, L. U.   Rockwell International, B-1 Div.
NAUMANN, E. A.   The Bendix Corp., Flight Systems Div.
NEAL, G. L.   Collins Radio Co.
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NEWBERRY, C. F.   The Boeing Co., Wichita Div.
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ONSTOTT, E. D.   Northrop Corp., Aircraft Div.
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PETERSEN, W. R.   NASA Flight Research Center
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RADER, J. E.   Air Force Flight Test Center
RADOVCICH, N. A.   Lockheed-California Co.
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REESING, H. A.   Douglas Aircraft Co.
REUKAUF, P. J.   NASA Flight Research Center
RHODEN, J. A.   AiResearch Manufacturing Co. of Arizona, Div. of The Garrett Corp.

RICCOBONO, J. A.   Douglas Aircraft Co.
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RUTTLE, J. W., Jr.	Martin Marietta Aerospace, Denver Div.
RYDER, D. R.	Boeing Commercial Airplane Co.
RYNASKI, E. G.	Calspan Corp.
SALAS, R. L.	FAA
SAMUELSON, R. M.	Lockheed-California Co.
SARPHIE, T. E.	Sundstrand Aviation
SAWOROTHOW, I.	Northrop Corp.
SCHATZ, E. J.	Fairchild Republic Co.
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SCHENK, F. L.	Lockheed-California Co.
SCHERER, L. R.	NASA Flight Research Center
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SCHMIDT, R. D.	Honeywell, G&AP Div.
SCHULDT, T. J.	Collins Radio Co.
SCHUMACHER, J.	McDonnell Aircraft Co.
SCHWANZ, R. C.	Air Force Flight Dynamics Laboratory
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SCOTT, D. R.	NASA Flight Research Center
SEACORD, C. L.	Honeywell, G&AP Div.
SECKEL, E.	Princeton University
SEEGER, G. R.	The Bendix Corp., Executive Offices
SEEMANN, R. J.	Northrop Corp.
SEITZ, C. R.	Bertea Corp.
SEITZ, W. R.	Bendix Research Laboratories
SESSA, A.	Univac
SHAFFER, J. R.	Douglas Aircraft Co.
SHAFFER, L.	Teledyne Ryan Aeronautical
SHAH, N. M.	Northrop Corp., Aircraft Div.
SHANKS, R. B.	USAF, Norton Air Force Base
SHARMAN, D. T.	FAA
SHARP, P. S.	Air Force Flight Test Center
SHEVELL, R. S.	Stanford University
SHIELDS, M. E.	LTV Aerospace Corp., Vought Systems Div.
SHUCK, L. M.	Sperry Flight Systems Div., Sperry Rand Corp.
SIM, A. G.	NASA Flight Research Center
SIM, G.	Lockheed-California Co.
SIMPSON, B. D.	Hydraulic Research & Manufacturing Co.
SITTERLE, G. J.	NASA Flight Research Center
SKABO, P. S.	Air Force Flight Test Center
SKINNER, G. L.	U.S. Army Aviation Engineering Flight Activity
SKINNER, M. R.	LTV Aerospace Corp., Vought Systems Div.
SKULLY, R. P.	FAA
SMART, J. A.	Eastern Airlines
SMELT, R.	Lockheed Aircraft Corp.
SMITH, D. W.	NASA Ames Research Center
SMITH, H. J.	NASA Flight Research Center
SMITH, J. W.	NASA Flight Research Center
SMITH, M. W.	Continental Airlines
SMITH, P.	LTV Aerospace Corp.
SMITH, S. B.	Air Force Flight Test Center
SMULL, T. L. K.	NASA Flight Research Center

SMYTH, R. K.	Milco International, Inc.
SNYDER, C. T.	NASA Ames Research Center
SODERLUND, G. M.	Lear Siegler, Inc., Astronics Div.
STALONY-DOBRZANSKI, J.	Northrop Corp., Aircraft Div.
STARR, G. A.	LTV Aerospace Corp., Vought Systems Div.
STAUFFER, W. A.	Lockheed-California Co.
STEERS, L. L.	NASA Flight Research Center
STEERS, S. L.	NASA Flight Research Center
STEPHENS, J. W.	Eustic Directorate, U.S. Army
STERBENTZ, W. H.	Lockheed Missiles and Space Co., Inc.
STEVENS, C. E.	Bertea Corp.
STICKLE, J. W.	NASA Langley Research Center
ST. LAWRENCE, D.	Lear Siegler, Inc., Astronics Div.
SWANSON, D. B.	Boeing Commercial Airplane Co.
SWARTZ, C.	House Aeronautics Committee Staff
SZALAI, K. J.	NASA Flight Research Center
TEPLITZ, J.	FAA
TERRY, J. L., Jr.	Eustic Directorate, U.S. Army
TERVO, W. K.	Pratt & Whitney Aircraft
THAYER, W. J.	Moog Inc.
THOMAS, D. M.	The Boeing Aerospace Co.
THOMAS, W. S.	FAA
THOMPSON, G. O.	The Boeing Co., Wichita Div.
THRONDSEN, E. O.	Lockheed-California Co.
TIEDEMANN, T. A.	NASA Flight Research Center
TREACY, J. J.	
TREECE, L. L.	United Air Lines
TRIBUNO, R. F.	Collins Radio Co.
TUCKER, R. C.	Air Force Flight Test Center
TUMILOWICZ, R. A.	Grumman Aerospace Corp.
TYMCZYSZYN, J. J.	FAA
UMBARGER, I. J.	Hydraulic Research & Manufacturing Co.
VAN LEYNSEELE, F.	Calspan Corp.
VAN'T RIET, R.	Douglas Aircraft Co.
VASLEY, G.	FAA
VENANCIO, A. J.	Sperry Flight Systems Div., Sperry Rand Corp.
VETTER, L. J.	King Radio Corp.
VISCONTI, L.	Grumman Aerospace Corp.
WADDOUPS, R.	Hydraulic Research & Manufacturing Co.
WALKER, H. J.	NASA Flight Research Center
WALKER, S. A.	USAF, Aeronautical Systems Div.
WALTER, H. L.	Beech Aircraft Corp.
WANGER, R. P.	General Electric Co.
WATERMAN, H. E.	FAA
WEBEL, L.	Sperry Rand Corp.
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WILSON, F. M.	Douglas Aircraft Co.
WILSON, R. J.	NASA Flight Research Center
WIMPRESS, J. K.	The Boeing Aerospace Co.
WINGER, D. J.	State of California
WINN, A. L.	U.S. Army Aviation Engineering Flight Activity
WINTER, W. R.	NASA Flight Research Center
WITT, D. O.	Hydraulic Research & Manufacturing Co.
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WOLF, L. A.	McDonnell Douglas Corp.
WOOD, N. E.	AiResearch Manufacturing Co., Div. of The Garrett Corp.
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WRIGHT, W. S.	Pacific Scientific Co.
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ZVARA, J.	Aerospace Systems, Inc.