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SURVIVABLE FLIGHT CONTROL SYSTEM
INTERIM REPORT NO. 1
STUDIES, ANALYSES AND APPROACH

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FOREWORD

This report was prepared by McDonnell Aircraft Company, St. Louis, Missouri, 63166, under Air Force Contract F33615-69-C-1827, PZ05, "Development and Flight Test Demonstration of a Survivable Flight Control System." This contracted effort comprises a major portion of development under the Air Force Systems Command Program No. 680J, "Survivable Flight Control System (SFCS)." The work was administered under the direction of the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, by Major Robert C. Lorenzetti, Technical Manager.

The report covers work performed between July 1969 and May 1971.

Principal contributors to this report are the MCAIR SFCS project personnel of many disciplines; the studies, analyses, and reporting were accomplished by them under direction of David S. Hooker, Project Engineer; Robert L. Kisslinger, Senior Project Dynamics Engineer; George R. Smith, Project Electronics Engineer; and M. Sheppard Smyth, Assistant Project Engineer - Design.

The authors wish to acknowledge the contributions of the MCAIR SFCS project personnel for the content reported herein. Acknowledgement also is given to project personnel of the following major subcontractors for contributions to information as follows:

- Sperry, Phoenix, Arizona -- Survivable Flight Control Electronic Set
- General Electric, Binghamton, New York -- Secondary Actuator
- Lear Siegler, Santa Monica, California -- Side Stick Controller
- LTV-Electrosystems, Arlington, Texas -- Survivable Stabilator Actuator Package

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ABSTRACT

The Survivable Flight Control System (SFCS) Program is an advanced development program of which the principal objective is the development and flight test demonstration of an SFCS utilizing Fly-By-Wire and Integrated Actuator Package techniques. The studies and analyses conducted to date have sufficiently defined the system requirements to provide a definition of an approach to the implementation of the SFCS. The results of these studies and the definition of the approach are presented herein. The details of the Control Criteria, Control Law Development, and Hydraulic Power and Actuation Studies are presented in report supplements 1, 2, and 3, respectively.

The SFCS Program is based on the principle of dispersed redundant control elements providing improved control performance and a very stable weapons delivery platform. The SFCS equipment includes:

- A quadruplex, three-axis, two-fail operational Survivable Flight Control Electronics Set which provides the computations for fly-by-wire control. Preflight built in test, in-flight monitoring of SFCS equipment, an adaptive gain and stall warning function, and provisions for in-flight failure simulation are included.

- Quadruplex force-summing electrohydraulic secondary actuators to convert the fly-by-wire flight control signals into the physical motion required to command the existing power actuators of the F-4 test aircraft.

- A quadruplex, two-axis sidestick controller in each cockpit. The front cockpit will also provide fly-by-wire center stick to allow direct comparison.

- A duplex power-by-wire actuator termed the Survivable Stabilator Actuator Package (SSAP). The SSAP will be capable of full-time operation throughout the F-4 flight envelope while receiving only electrical power. The SSAP incorporates a quadruplex velocity-summing electromechanical secondary actuator, and is controlled solely by the fly-by-wire system.

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LIST OF ABBREVIATIONS AND SYMBOLS

ABBREVIATIONS:

AA - Aileron Actuator

AC - Alternating Current

A/C - Air-conditioning

AOA - Angle of Attack Transmitters

AFCS - Automatic Flight Control System

AGSW - Adaptive Gain and Stall Warning

APU - Auxiliary Power Unit

ARI - Aileron Rudder Interconnect

ARPS - Aerospace Research Pilots School

ATP - Acceptance Test Procedure or Aft Cockpit Trim Control Panel

Avg - Average

B+ - Supply Voltage

BIT - Built-In-Test

bit - Binary Digit

B/P - Bypass Valve

CLTR - Caution Light Test Relay

COMM - Communication

CST - Control Stick Transducer

cu. in. - Cubic Inch

CVU or CV - Computer and Voter Unit followed by a -1, -2, -3, -4 indicates red, blue, black, and yellow channel respectively

DATP - Design Approval Test Procedure

dB - deciBels - a measure of gain = 20 $\log_{10}$ (Amplitude Ratio)

DC - Direct Current

DC1 - SFCS DC Power Supply Number One

DC2 - SFCS DC Power Supply Number Two

DC3 - SFCS DC Power Supply Number Three

DC4 - SFCS DC Power Supply Number Four

Deg - Degree

DFG - Discrete Function Generator

DSA - Directional Secondary Actuator

DSSV - Dual Spoiler Servo Valve

EBU - Electrical Back-Up

EDHP - Electrically Driven Hydraulic Pump

EH - Electro-Hydraulic

ELEC - Electronics

EM - Electro-Mechanical

EMI - Electromagnetic Interference

EMR - Emergency

ENG - Engine

EROS - Eliminate Range Zero System (Collision Avoidance)

Ess - Essential

FBW - Fly-By-Wire

FMEA - Failure Mode and Effects Analysis

FREQ - Frequency

F.S. or FS - Fuselage Station

Ft - Feet

FTP - Forward Cockpit Trim Control Panel

FWD - Forward

GD - Gear Down (landing gear extended)

GEN - Generator

GPM - Gallons per minute

G&S - Adaptive Gain and Stall Warning Computer Unit

HDEG - Hydraulic Driven Electrical Generator

HP - Horsepower

HV - Hydraulic Solenoid Valve

HYD, Hy - Hydraulic

Hz - Hertz (cycles per second)

IAP - Integrated Actuator Package

IFF - Identification, Friend or Foe

IFM - In-Flight Monitor

In - Inch

INB'D - Inboard

IND - Indicator

INOP - Inoperative

IR - Infrared

K - Multiplier, 1000 times (Kilo)

KVA - Kilovolt-ampere

LA - Lateral Accelerometer Sensor Unit

Lb - Pound

LC - Line Contactor

L.E. - Leading Edge

LH - Left Hand

LMP - Left SSAP Motor Pump Unit

LRU - Line Replaceable Unit

LSA - Left Lateral Secondary Actuator

LVDT - Linear Variable Differential Transformer

M - Mach, Speed of Sound, Unit of

MBU - Mechanical Back-Up

MCAIR - McDonnell Aircraft Company

MCP, MCDP - Master Control and Display Panel

MCV - Main Control Valve

MED - medium

MGTF - Mobile Ground Test Facility

MHz - MegaHertz (million cycles per second)

Mic - Microphone

MIM - Mechanical Isolation Mechanism

MIMA - Mechanical Isolation Mechanism Actuator

MIM D - Directional Mechanical Isolation Mechanism

MIM P - Pitch Mechanical Isolation Mechanism

MMH/FH - Maintenance Manhour per Flight Hour

MP - Motor Pump (SSAP)

MP1 - Motor Pump One (SSAP)

MP2 - Motor Pump Two (SSAP)

MTBF - Mean Time Between Failure

MTBMA - Mean Time Between Maintenance Actions

MTP - Maintenance Test Panel

NA - Normal Accelerometer Sensor Unit

NAV - Navigation

n/cm^2(1-MeV Si) - Neutrons per square centimeter one million electron volts-silicon equivalent

NiCd - Nickel Cadmium

NSS - Neutral Speed Stability (longitudinal SFCS function)

OUT'B - Outboard

oz - Ounce

PA - Power Approach

PBW - Power-By-Wire

PC - Power Control Hydraulic System

PC-1 - Power Control Hydraulic System One

PC-2 - Power Control Hydraulic System Two

PCP - Pitch Computer Channel

PG - Pitch Gyros

PL - Place

PRG - Pitch Rate Sensor Unit

PSA - Pitch Servo Amplifier Channel or Pitch Secondary Actuator

PSECA - Pitch Secondary Actuator Channel

$PSECA_s$ - Pitch Secondary Actuator Single Failure

psi - Pounds per square inch

PV - Pitch Voter Channel

Q - Hydraulic flow rate (gallons per minute)

R - Resistor, Hydraulic Return Line

Rad - Radian

rad - Unit of Absorbed Radiation Dose

rad(Si) - Silicon Equivalent Dose (rads)

RA_s - Rudder Actuator Single Failure

R&D - Research and Development

RD_s - Rudder Damper Single Failure

REF - Reference

RH - Right Hand

RMP - Right SSAP Motor Pump

rms - Root-Mean-Square

rpm - Revolutions per minute

RPT - Pedal Transducer Unit

RRG - Roll Rate Sensor Unit

RSA - Right Secondary Actuator

SA - Secondary Actuator

SAS - Stability Augmentation System

SCDP - Secondary Control and Display Panel

Sec - Second

SFCES - Survivable Flight Control Electronic Set

SFCS - Survivable Flight Control System

SFCS-Mech Sw - SFCS-Mech Select Switch

SIF - Selective Identification Feature

SIP - State Interpretation Program

SL - Sea Level

SLR - Side Looking Radar

SPDT - Single Pole Double Throw

SPT - Stabilator Position Transducer

SRB - Self Retaining Bolt

SSA - Aft Side Stick Controller

SSAP - Survivable Stabilator Actuator Package

SSAPmtb - SSAP Motor-Tachometer Brake

$SSAP_s$ - Survivable Stabilator Actuator Package Single Failure

SSC - Side Stick Controller

SSD - Signal Selection Device

SSF - Forward Side Stick Controller

STA_s - Stabilator Actuator Single Failure

SVns - Switching Valve - No Switch

SW - Switch, electrical; Switching Valve, Stall warning

Tach - Tachometer

TAS - True Air Speed

TOL - Take Off and Land (longitudinal SFCS function)

TOLsw - Take Off - Land Switch

T/R - Transformer Rectifier

U - Utility Hydraulic System

UHF - Ultra High Frequency

UPS - Utility Pressure Switch

USAF - United States Air Force

Util - Utility

V - velocity, voltage

VAC - Volts Alternating Current

VDC - Volts Direct Current

VR/SP - Voltage Regulator and Supervisory Panel

YRG - Yaw Rate Sensor Unit

$YSECA_S$ - Yaw Secondary Actuator Single Failure

3M - Naval Maintenance Material Management System

4PDT - Four Pole Double Throw

4th - Fourth Hydraulic System

28 VDC - Channel 28 VDC Power

SYMBOLS:

A	Amplitude
A	Tie point on parameter identifier - volts
a	gain margin (dB)
a	Variable parameter of time delay comparator
a,b	Internal signal of parameter identifier - volt
a_{y_p}	Lateral Acceleration measured at pilot's seat
B	U + HV + MIM A + UPS
C	Controlled or metered hydraulic connection point
C (Phase IIB)	SPT + PCP + NA + PG
C (Phase IIC)	PCP + NA + PG
C*	Pitch axis time history response parameter
$C_a(s)$	Laplace transform of total airframe response at a specified location
C^*_N	Normalized C*
$C_v(s)$	Laplace transform of rigid aircraft response
D	28 VDC + TOL_{sw}
D*	Yaw axis time history response parameter
D^*_1	Directional criteria - deg.
DAC	Dual 115 VAC
E	Engine
e_i	Signal selection device input voltages from the four channels, i = 1, 2, 3 and 4
e'_i	Equalized signal selection device inputs, i = 1, 2, 3 and 4
e_o	Signal selection device output voltage
e_{offset}	Offset voltage
e_{null}	Null or zero offset voltage
f_T	Transistor high cutoff frequency

g	Acceleration due to gravity - ft/sec^2
H	Static Trip Level
K	Amplifier gain
k	Ratio of "commanded roll performance" to "applicable roll performance requirement" - dimensionless
$K_a(s)$	Laplace transform of rigid control surface rotation
K_{cs}	Center stick longitudinal force transducer prefilter gain volts/volt
K_E	Equalization feedback loop gain volts/volt
K_F	Longitudinal forward loop variable gain - volts/volt
K_{NZ}	Normal accelerometer output gain - volts/volt
K_q	Pitch rate gyro output gain - volts/volt
KR	Yaw rate gyro output gain - volts/volt
Link P	Pitch Linkage
Link R	Lateral Linkage (One Wing)
Link RD	Rudder Linkage
LL	Link R + AA + DSSV
LR	RAs + RDs + Link RD + YSECAs
LRM	RAs + RDs + Link RD + MIM D
M	MP + MP 115 VAC
M_{δ_c}	Computed pitching angular acceleration due to stabilator deflection - 1/sec^2
M_{δ_i}	Value of M_δ computed by Parameter Identifier for the red, blue, black and yellow channels respectively, i = 1, 2, 3, and 4, -1/sec^2
M_δ, M_{δ_s}	Pitching angular acceleration due to stabilator deflection - 1/sec^2
n	Number of channels operating
N_Z	Normal acceleration
$N_Z(s)$	Laplace transformation of normal acceleration

P	Pressure hydraulic connection
P_G	Angular velocity sensed by roll rate gyro - rad/sec
P_i	Signal path i = 1, 2
Q	Probability of failure
q	Pitch rate, the angular velocity of airplane about Y axis - rad/sec
$\bar{q}$	Dynamic pressure - lb/ft^2
R_a	Laplace transform of rigid control surface deflection
$R_v(s)$	Laplace transform of vehicle control command
S_i	Switching valves hydraulic, i = 1, 2, 3
SSs	SSAPs + Link P
ST	STAs + Link P + PSECAs
STM	STAs + Link P + MIM P
S	Laplace operator
t_i	Time of occurrence of a discrete event, i = 1, 2, 3...
V (Phase IIB)	PV + PSA + PSECA
V (Phase IIC)	PV + PSA + $SSAP_{MTB}$
V_{SAT}	Saturation voltage
α	Ratio of gain in path P_1 to the gain in path P_2
β	Sideslip angle
γ	Phase angle
Δ_e	Magnitude of level shift
ΔP	Differential pressure
δ	Control surface deflection - rad
δ_s	Stabilator deflection - rad
η_4	Normal coordinate for first fuselage torsion mode - dimensionless

η_5	Normal coordinate for unsymmetric wing bending mode - dimensionless
η_6	Normal coordinate for first lateral bending mode - dimensionless
θ	Pitch angle - rad
λ	Failure rate
λA	Axis failure rate
λc	Channel failure rate
τ	Time constant or time delay
μa	Microcircuit amplifier
ω	Frequency - rad/sec
ω_c	Frequency at which gain margin is computed
ω_ϕ	Frequency at which phase margin is computed (Gain 0 db, crossover frequency) - rad/sec
3ϕ	Three phase
$(\dot{\ })$	First derivative of a variable with respect to time
$(\ddot{\ })$	Second derivative of a variable with respect to time
$^\circ$	Degree
$^\circ F$	Degrees Fahrenheit
∞	Infinity

SECTION I

INTRODUCTION AND SUMMARY

The Survivable Flight Control System (SFCS) Program is a flight control advanced development program being performed primarily by MCAIR under contract to the Air Force Flight Dynamics Laboratory. The principal objective of this program is the development and flight test demonstration on an F-4 aircraft of a Survivable Flight Control System utilizing fly-by-wire and power-by-wire techniques.

Recent combat experience has shown that relatively minor damage, in the form of small arms fire can result in aircraft loss due to loss of control. This is brought about by either hits in the hydraulic distribution system which drain the fluid, or hits which sever or jam the non-redundant mechanical flight control linkages. The power-by-wire concept of integrating electric motor driven hydraulic pumps with the surface actuator reduces system vulnerability through elimination of dependence on long exposed runs of hydraulic plumbing. The fly-by-wire concept of redundant and physically dispersed electrical control channels improves survivability by eliminating the single-failure points of the conventional mechanical control linkages.

The SFCS Program is being performed in two phases. Phase I, which included flight test evaluation of a Simplex integrated actuator package, has been completed and is documented in Reference 1. The Phase II program and objectives are illustrated by Figure 1, and include the development and flight test evaluation of a flight control system employing fly-by-wire and power-by-wire concepts.

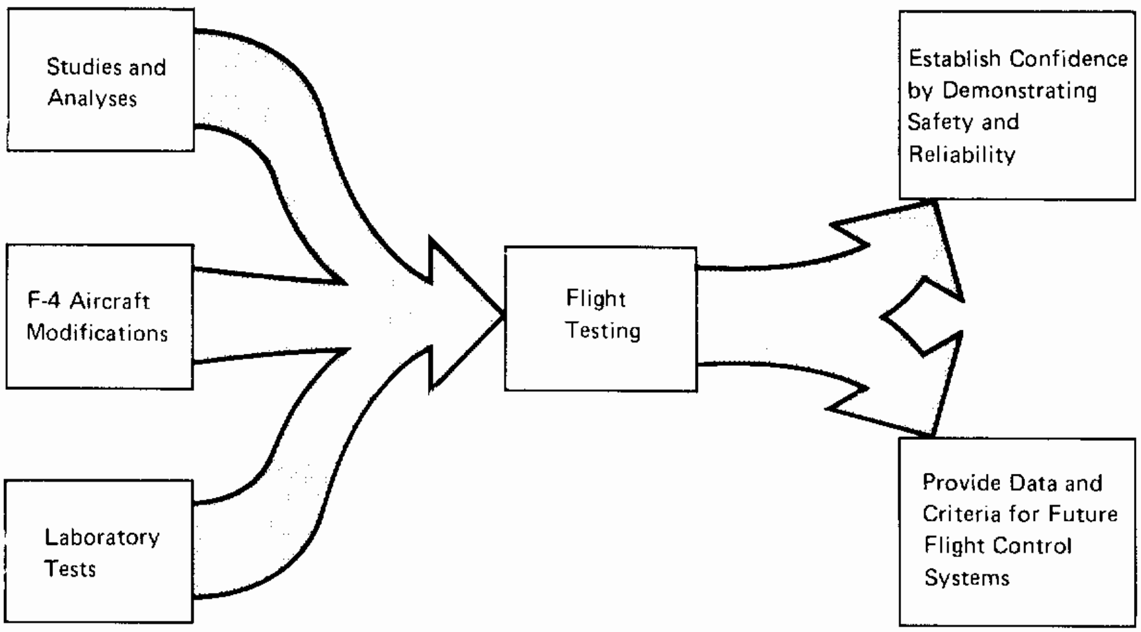


FIGURE 1

PHASE II - PROGRAM AND OBJECTIVES

F-4 WITH SURVIVABLE FLIGHT CONTROL SYSTEM

Fly-by-wire (FBW) is a primary flight control system which uses an electrical signaling path to provide the desired aircraft response to pilot commands, without a mechanical connection between the cockpit controller and the control surface actuator. It can incorporate aircraft motion sensors such that aircraft motion, rather than control surface position, is the controlled variable.

To be accepted by the aerospace industry as more than a research tool, the reliability of the FBW system must meet or exceed the reliability of the mechanical system it is replacing while showing advantages in other areas. The benefits foreseen for an FBW system include:

- Enhanced survivability
- Superior aiming, tracking, and weapon delivery
- Reduced pilot workload
- Flight control design and installation savings
- Decreased cost of ownership
- More airframe design freedom

Power-by-wire (PBW) is the transmission of power from the aircraft engines to the flight control surface actuators by electrical rather than hydraulic means. Hydraulic power is generated by electric motor driven hydraulic pump(s) integral to the actuators. Power-by-wire equipment has been called "integrated actuator packages" in this country, and simply "packaged actuators" in England.

The improved control performance of a fly-by-wire system and the get-home-and-land capability provided by an actuator with an emergency-only electric motor driven pump could be combined to provide a measurable improvement in flight control survivability. An F-4 Simplex Actuator Package with this emergency-only PBW capability was successfully flight tested in Phase I of the SFCS Program, with results reported in Reference 1. However, a truly survivable flight control system requires use of power-by-wire integrated actuator packages which are capable of full-time operation independent of the aircraft central hydraulic systems and their exposed plumbing. The Survivable Stabilator Actuator Package (SSAP) to be flight tested in Phase IIC of the SFCS Program will be a duplex PBW actuator capable of full-time operation throughout the F-4 flight envelope. The SSAP will be controlled by the fly-by-wire system installed and flight tested in Phases IIA and IIB of the program.

The location of the fly-by-wire system components, the SSAP, and the other SFCS equipment in the F-4 test aircraft is shown in Figure 2.

The results of the SFCS studies and analyses to date, and the definition of the SFCS approach are presented in this report. The details of the Control Criteria, the Control Law Development, and Hydraulic Power and Actuation Studies are presented in report supplements 1, 2, and 3, respectively.

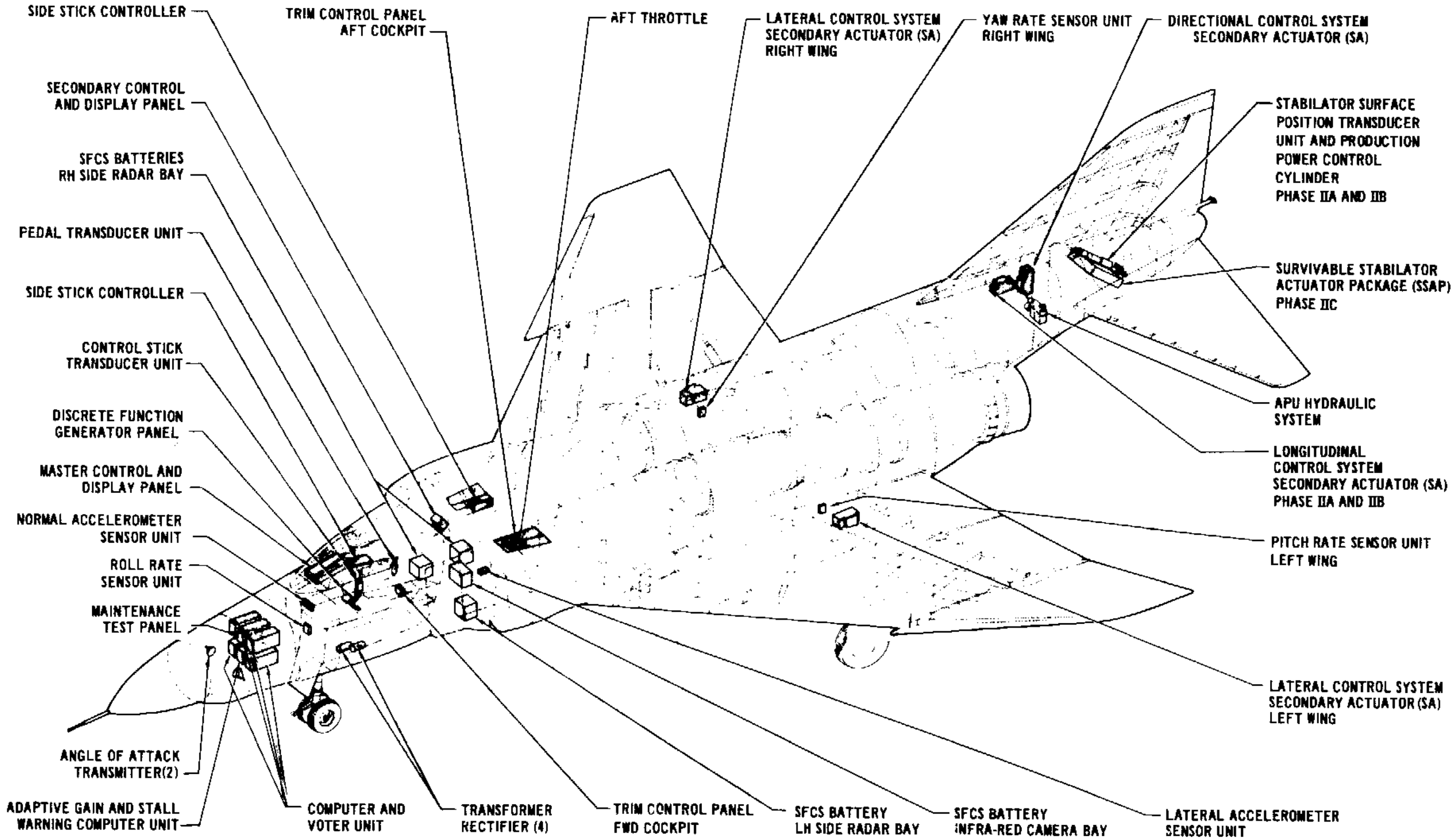


FIGURE 2
SFCS EQUIPMENT LOCATION

SECTION II

DEFINITION OF APPROACH

1. GENERAL

The approach taken by MCAIR to accomplish Phase II of the SFCS Program is defined in this section.

The Phase II effort provides for the development, laboratory test, simulation, installation and flight evaluation of a SFCS. This effort includes the development of the quadruplex fly-by-wire electronics and the redundant integrated actuator package. In this SFCS program, these are often termed Survivable Flight Control Electronic Set (SFCES) and Survivable Stabilator Actuator Package (SSAP), respectively.

2. THE STEP APPROACH

One part of the approach has been and is to separate the Phase II effort into the following four steps:

Phase IIA - Fly-by-wire with mechanical back-up
Phase IIB - Fly-by-wire without mechanical back-up
Phase IIC - Fly-by-wire with Survivable Stabilator Actuator Package in pitch axis
Phase IID - Demonstration flights

Figure 3 illustrates the development approach in progressing from the present F-4 mechanical control system to the Survivable Flight Control System.

Phase IIA flight demonstrates a fly-by-wire flight control system with the mechanical system disengaged, but available for emergency use in the pitch and yaw axes. The mechanical isolation mechanisms function as variable gain devices with the mechanical system having zero gain when fly-by-wire is engaged, and vice versa. In all three phases, secondary actuators convert the four-channel FBW electrical commands into the physical motion required to operate the existing F-4 surface actuators. Phase IIB completely removes the mechanical back-up to demonstrate a true fly-by-wire system.

The basic requirements for the fly-by-wire system, as established by the Air Force Statement of Work, include the following:

- o Provide three-axis flight control through the entire performance envelope of the F-4 aircraft.
- o Blended pitch rate and acceleration or a similar longitudinal control maneuver demand mechanization.
- o Dispersed quadruplex electronics and sensors for a two-fail/operate capability. (Minimum performance degradation after two similar failures.

- Provide superior control performance for precise weapons delivery and tracking.

- An electrical back-up (EBU) mode which allows the pilot to command surface position rather than aircraft motion. This back-up mode consists of a subset of the complete normal-mode fly-by-wire equipment.

- Redundant self-adaptive gain changing to maintain maximum performance capability in all flight conditions.

- Sidestick controllers, with pitch axis vernier capability, in both pilots' stations.

- Cockpit failure monitoring and reset capability.

- Built-in test (BIT) equipment for go/no-go preflight check and fault isolation to a line replaceable unit.

- A design reliability analysis with a system design goal failure rate of $\lambda = 2.3 \times 10^{-7}$ failures per flight hour.

- Interface capability with outer control loops such as autopilot, automatic landing systems, automatic terrain following, and direct lift control systems.

- A selectable neutral speed stability capability.

As illustrated in Figure 3, Phase IIC consists of installing a Duplex Survivable Stabilator Actuator Package (SSAP) in the longitudinal axis. The Phase I Simplex Actuator Package with built-in pump and motor provides emergency get-home-and-land capability upon loss of both aircraft central hydraulic systems. However, the SSAP will be a full-time-operating package capable of controlling the aircraft within the same flight envelope and performance capabilities as the present system. Central hydraulics will be plumbed to the SSAP for experimental flight safety, but will only serve as a back-up source. The SSAP will be controlled by the fly-by-wire control system already installed in the aircraft. Phase IIC includes study and analysis of the reliability, maintainability, and survivability of power-by-wire actuators, plus computer simulations and "iron bird" control system mockup tests of the SSAP.

Approximately 60 data flights will be conducted to optimize, evaluate, and demonstrate the SFCS system performance and handling qualities. Data from the flight tests will be processed and correlated with ground checkout and simulation data continuously throughout the program. The data flights will include mission modes applicable to the F-4 test aircraft, such as weapon delivery, aerial combat, high-speed low-altitude dash, etc.

In Phase IID, additional demonstration flights will be provided for pilots from USAF, Army, Navy, and other interested agencies. These demonstration flights may be interspersed among the data flights of the other three phases.

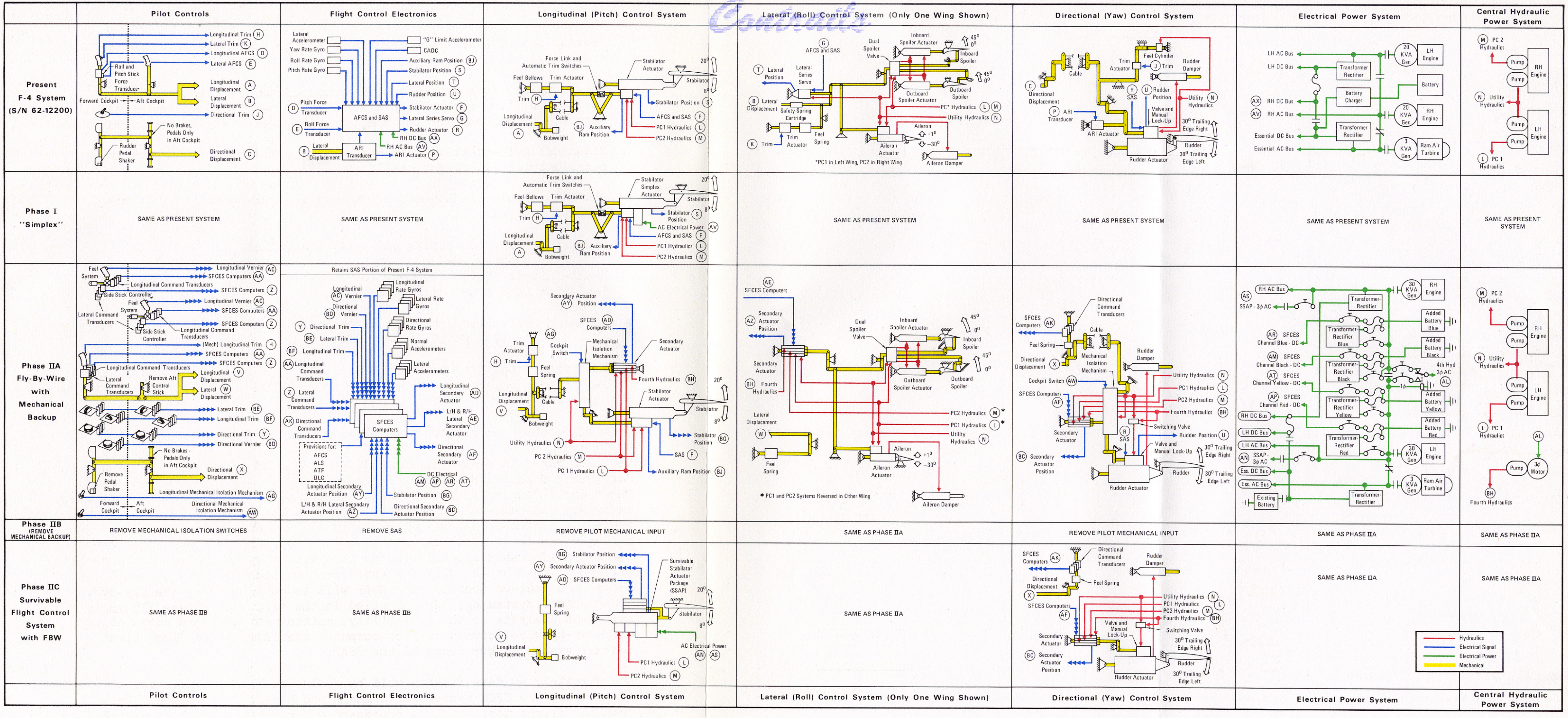


FIGURE 3
SCHEMATIC OF SFCS DEVELOPMENT APPROACH

3. FUNCTIONAL ACTIVITY APPROACH

A second part of the SFCS approach is to separate and organize the total job into the following major activities:

- o Studies and Analyses,
- o Simulation,
- o Design and Modification of the F-4,
- o Laboratory Tests,
- o Flight Testing, and
- o Reports and Data.

Rather extensive studies and analyses have been conducted to help define the approach, the modifications to the aircraft, and the performance and characteristics of major equipments such as the SFCES and SSAP. Many of these studies have been completed and some are continuing. In addition, the approach includes additional studies based on observed characteristics of the hardware and upon results of laboratory and flight tests to define desirable characteristics of an SFCS for future aircraft.

The simulation activity is part of the approach. Simulations have been used both with and without the pilot in the loop to help find desirable and/or undesirable design and performance parameters and to assist in establishing desirable characteristics. The approach includes additional simulations with and without actual hardware to evaluate system component compatibility and to provide correlation with flight test results.

Part of the approach is to proceed, concurrent with studies and simulations, with the design of the modifications to the test aircraft subsystems and structure, and design and establishment of major equipment requirements, based on experience and judgement. The approach includes modifying designs and requirements as indicated to be desirable by results of the studies and simulations.

A comprehensive laboratory testing program is planned to achieve a high level of confidence in the SFCS. Dynamic performance and safety will be stressed as well as installation compatibility. The laboratory tests will include design approval tests and reliability tests of equipment by subcontractors. In addition, MCAIR will perform component evaluation tests as components are received. Also, compatibility testing will be performed using both the control system mockup (Iron Bird) and the six-degree-of-freedom fixed base flight simulator. Furthermore, the system as installed on the test airplane will undergo ground vibration tests, EMI tests, and closed-loop performance tests.

Flight tests of an F-4 test aircraft with the SFCS installed is a major activity included as part of the approach. The objectives of the F-4 flight tests are:

- Evaluate and demonstrate a SFCS utilizing fly-by-wire (FBW) and integrated actuator package (IAP) techniques,
- Obtain data to establish the design criteria, survivability, reliability, maintainability, safety, performance, and testing requirements for application in future installations, and
- Establish confidence in the SFCS technology and broaden the experience level.

4. SUBCONTRACT APPROACH

A third part of the approach has been and is to subcontract the development of some major equipment items to suppliers in order to gain the benefits of their expertise and imagination. The major suppliers and equipment are as follows:

- Sperry, Phoenix, Arizona -- Survivable Flight Control Electronic Set
- General Electric, Binghamton, New York -- Secondary Actuator
- Lear Siegler, Santa Monica, California -- Side Stick Controller
- LTV-Electrosystems, Arlington, Texas -- Survivable Stabilator Actuator Package

The major items of procured equipment are discussed in Section IV.

5. INFORMATION APPROACH

A fourth part of the approach has been and is to keep the Air Force informed by periodic formal status reports and by many more informal telephone discussions and face-to-face meetings. The periodic status reports describe the cost, schedule, and technical situations. Information provided to the Air Force also includes laboratory and flight test plans, interim reports such as this one, and a final technical report which will describe in formal report form results of all the tasks accomplished in completing the contract.

6. SFCS IMPLEMENTATION

a. General

The SFCS implementation is summarized on the next few pages and described more completely in Sections IV and V.

The Survivable Flight Control System (SFCS) is a three axis fly-by-wire primary flight control system combined with an integrated actuator package in the pitch axis. In the NORMAL mode, the pilot commands aircraft motion. In the electrical back-up (EBU) mode, surface position is the controlled variable. The final configuration of the SFCS will provide true fly-by-wire control of all primary flight control surfaces, with no mechanical back-up system in any axis.

b. Redundancy

Quadruplex (four channel) redundancy is used in all on-line system components, including power supplies, to obtain improved system reliability and increased mission completion probability. The system is designed to sustain two similar failures per axis without significant degradation in performance. In addition, an electrical back-up mode is provided which allows direct pilot control of surface position for use in the remote event of three or more similar failures in the computational electronics. Improved survivability obtained through the use of physical isolation or dispersion of components is utilized, insofar as practical for the test aircraft, to demonstrate the principle of survivability through dispersed redundancy. Future studies performed for this program will address the integration of SFCS concepts into future weapon systems with emphasis towards channel separation, component separation, component design arrangements, component placement within the weapon system, and other features which should further enhance weapon system survivability.

c. Mechanical Back-Up

A mechanical back-up mode is provided in the pitch and yaw axes for the initial portion of the flight test program as an additional safety measure. A roll axis back-up mode is not required for the flight test program since landings can be accomplished utilizing the rudder alone for roll control in the event of a complete electronics failure. The mechanical back-up capability is implemented through the use of Mechanical Isolation Mechanisms (MIM) installed in the pitch and yaw mechanical control systems. The MIM functions to select either 100% fly-by-wire or 100% mechanical control. Limited authority fly-by-wire is not used. During the Phase IIB modification, the mechanical back-up modes will be eliminated by removing the MIMs and associated linkages.

d. Built-In-Test

An extensive ground built-in-test (BIT) capability is included in the SFCS. The system automatically tests the SFCS and subsequently indicates a GO or NO GO condition to the pilot and ground crew. Most detected failures will be automatically isolated to an LRU. LRU failure indications will be displayed on a maintenance test panel. The ground BIT will require approximately two minutes, and will be positively locked out during flight.

e. In-Flight Monitoring

In-Flight Monitoring (IFM) is employed to automatically detect and disengage failed channels or secondary actuator elements. Re-set switches with integral status indicator lights are installed on the main instrument panel to provide continuous system status information. Momentary or inadvertent disconnects can be re-set using these switches.

f. Flexibility

Considerable flexibility is provided in the system design to enable side-by-side comparisons of various control modes, methods of applying pilot inputs, and airplane response characteristics. Both center stick and side stick controllers will be provided in the front cockpit for side-by-side evaluation. In addition, numerous parameters are ground adjustable by substitution of discrete electronic components, specifically designed for ease of modification, or simple mechanical adjustments.

g. Instrumentation

A flight test instrumentation system has been designed to enable monitoring of important SFCS and aircraft parameters during flight testing. Those instrumentation sensors which must be located internal to the various LRUs are included in the LRU designs and will be qualification tested with the units. Special connectors are included on the electronic set computer and voter unit LRUs to enable interfacing with flight test instrumentation.

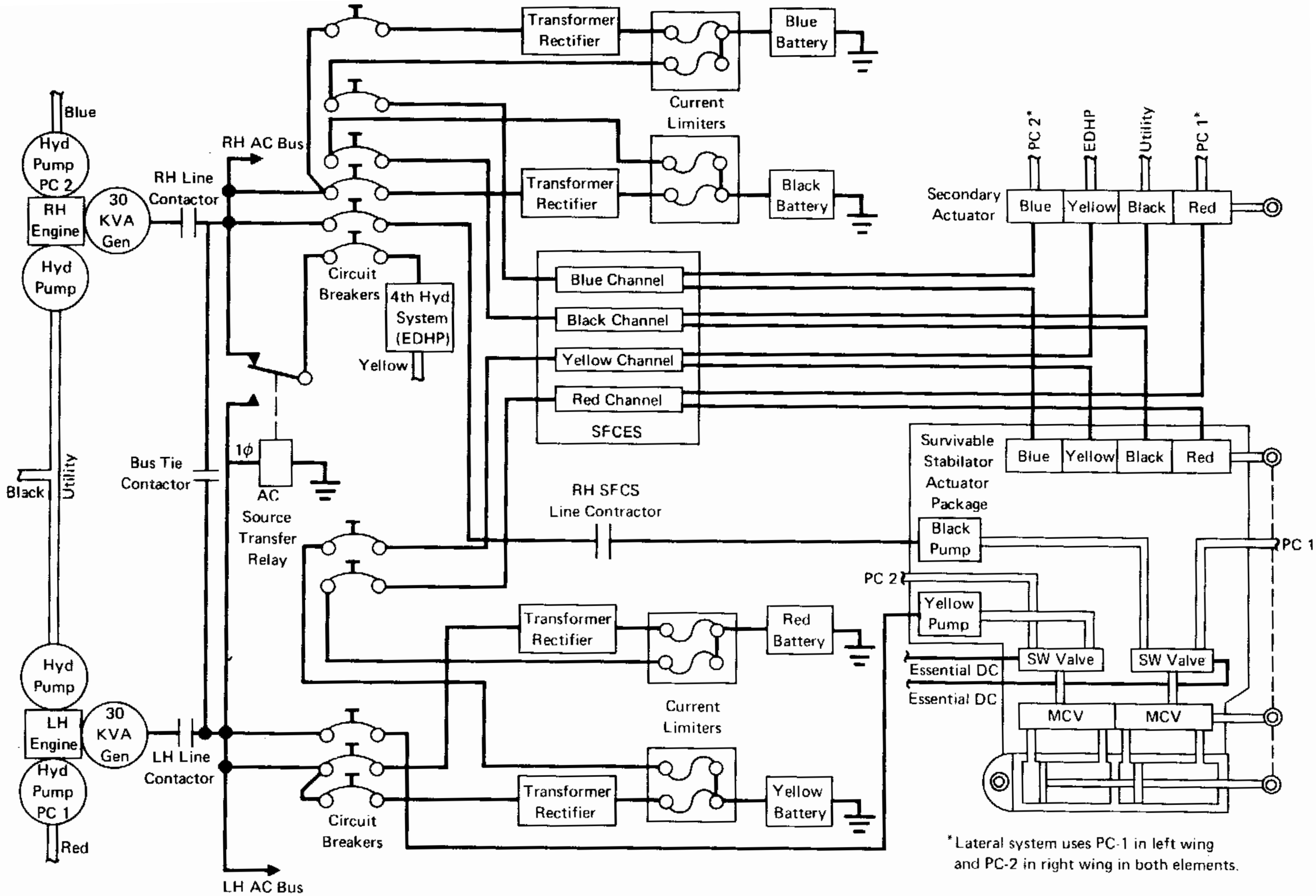
h. Color Coding

A color coding system is utilized throughout all SFCS LRUs, interconnect wiring, connectors and hydraulic plumbing fittings to help prevent installation and maintenance errors. The four channels of each axis are identified by the colors red, blue, black and yellow. Each wire bundle or hydraulic line which must be connected to an SFCS LRU is identified by one of these colors. LRUs which contain elements of more than one channel have color code identification adjacent to the necessary electrical connectors or hydraulic line fittings.

A simplified SFCS power system schematic is presented in Figure 4. This diagram illustrates the color coding by channel of the SFCS and shows the relationship between electrical and hydraulic power sources. The electrical and hydraulic power sources to be used in each channel were selected to minimize the effect of power source failures.

i. Electrical Power

The primary sources of electrical power in the test airplane are two engine-driven AC generators. These generators power the left and right hand AC busses in a split bus configuration. In order to obtain quadruplex power sources for the SFCS equipment, two transformer rectifiers are connected to each of the two electrical busses. In the event of power failure on one of the busses, the remaining bus is automatically switched into the failed bus. Each of the transformer rectifiers is connected in parallel with an aircraft battery and connected to one and only one SFCS channel. The batteries have sufficient capacity to power the SFCS for approximately one hour. Use of the batteries assures continued operation of all four SFCS channels in the event of total AC power failure.



*Lateral system uses PC-1 in left wing and PC-2 in right wing in both elements.

FIGURE 4
COLOR CODED SIMPLIFIED SFCS POWER SYSTEM SCHEMATIC

j. Hydraulic Power Systems

Three hydraulic pressure sources are normally available in the F-4 airplane. These pressure sources and the assigned color code are: Power Control Hydraulic System No. 1 (PC-1) - Red; Power Control Hydraulic System No. 2 (PC-2) - Blue; and Utility Hydraulic System - Black. A fourth hydraulic system is required to maintain quadruplex redundancy for the SFCS in the test airplane. An auxiliary power unit containing an electric motor driven hydraulic pump is utilized to provide the fourth hydraulic system which is color coded yellow. Excitation for the auxiliary power unit is normally supplied from the left hand AC bus with automatic switchover to the right hand AC bus in the event of electrical power failure.

k. Secondary Actuators

Secondary actuators are initially used in all axes of control to convert electrical command signals to mechanical position commands for application to the surface actuators. The separate secondary actuators were used, in preference to integrated secondary and surface actuator units, to permit utilization of existing F-4 surface actuators. Separate quadruplex electrohydraulic secondary actuators are employed. The four elements of the secondary actuators are physically isolated from each other, powered from separate hydraulic sources, and commanded through separate electrical channels of the SFCES. The outputs of the four elements of a secondary actuator are physically connected to provide the single mechanical input required for the surface actuator. The single mechanical connection between each secondary actuator and its respective surface actuator is designed with sufficient strength and integrity to maintain overall SFCS reliability.

l. Survivable Stabilator Actuator Package (SSAP)

The SSAP will be installed in the aircraft replacing the pitch secondary actuator and production surface actuator for Phase IIC of the flight test program. This SSAP includes two integral hydraulic systems and an integrated electromechanical secondary actuator. The integral hydraulic systems are planned to provide complete control of the stabilator surface without utilizing the aircraft central hydraulic systems. However, for purposes of this test program, two central hydraulic systems will be used to provide emergency back-up to the integrated hydraulic systems.

The electromechanical secondary actuator mounted on the SSAP will receive commands through separate electrical channels of the SFCES. The electronics set has been configured to control both electro-hydraulic and electromechanical secondary actuators for the appropriate parts of the flight test program.

m. Primary Control Axes

A brief functional description of each primary axis of control is presented in the following paragraphs to indicate the development approach used in the SFCS design. Simplified functional block diagrams of the longitudinal and lateral-directional axes are presented in Figures 5 and 6 respectively.

(1) Longitudinal Axis

A high gain control loop with both pitch rate and normal acceleration feedback is used in this axis. Both pilot selectable and adaptive gain control are provided. A structural filter is utilized to reduce the loop gain at airframe structural resonance frequencies and help eliminate the effects of structural bending. Both Neutral Speed Stability (NSS) and Take-off and Landing (TOL) functions are included to eliminate the requirement to trim in the clean configuration and to provide a positive speed stability characteristic for take-off and landing. With the NSS function selected, no steady state pilot applied stick force or trim input is required in order to compensate for the change in stabilator position required to trim the airplane due to changes in airspeed and/or altitude. Selection of the TOL mode will result in a requirement to manually trim the aircraft as airspeed and/or altitude are varied. Airspeed changes at subsonic flight conditions will result in positive speed stability; i.e., push forces will be required as airspeed is increased and pull forces will be required as airspeed is reduced. Stall warning is provided through increased stick force and audio tones as the stall is approached. Performance characteristics of the pitch axis are presented in Section III, Paragraph 13a.

(2) Lateral Axis

A fixed gain roll rate feedback loop is utilized to provide a highly responsive roll rate command loop. Pilot inputs are shaped to provide desirable roll rate time constants at all flight conditions. A structural filter is included to attenuate loop gain at structural resonance frequencies. An input from the stall warning circuit is utilized to remove roll rate feedback at high angles-of-attack and thus prevent normal aircraft wing rock from causing spin-inducing lateral control deflections. A shaped lateral stick force output is provided to the yaw axis for use in turn coordination.

(3) Directional Axis

Yaw rate and lateral acceleration feedback signals are used in this axis to improve Dutch Roll damping and frequency characteristics. Both pilot selectable and adaptive gain control are provided. Pilot rudder pedal inputs are applied through a shaping filter to further improve response characteristics. Shaped lateral stick force inputs are applied through variable gain

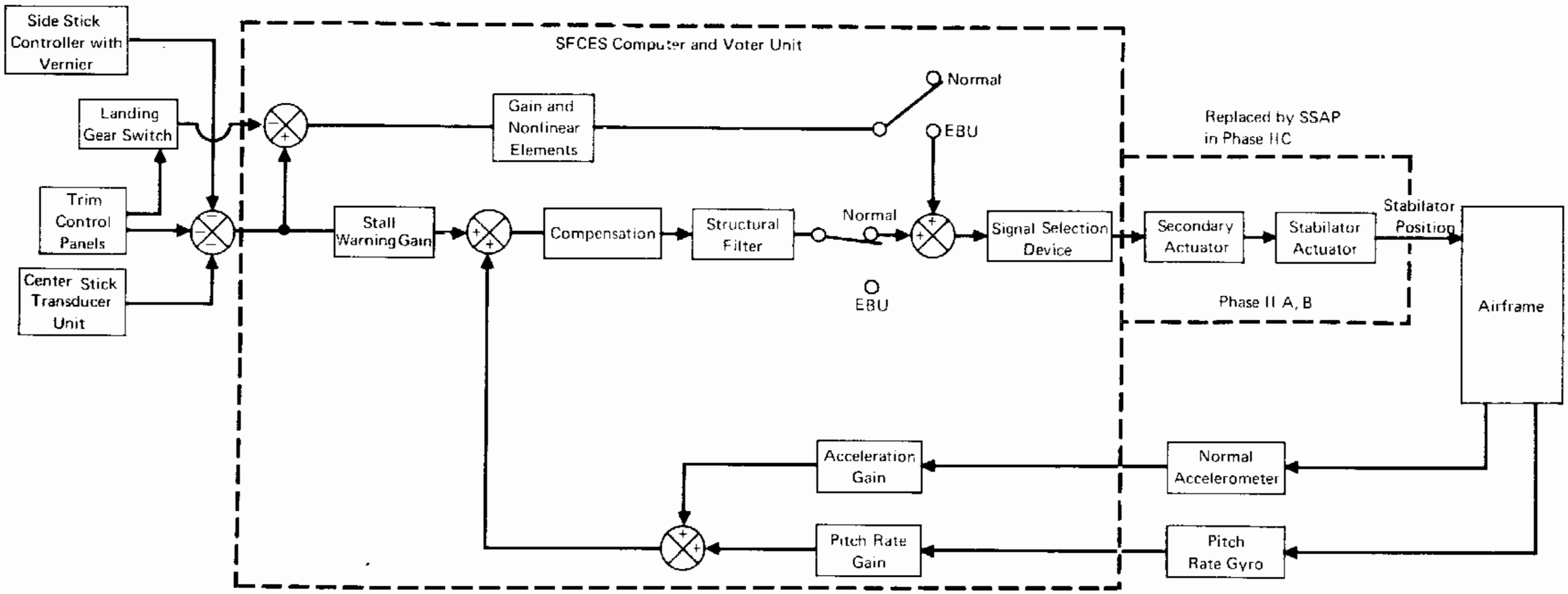


FIGURE 5
SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM
SINGLE CHANNEL OF LONGITUDINAL AXIS

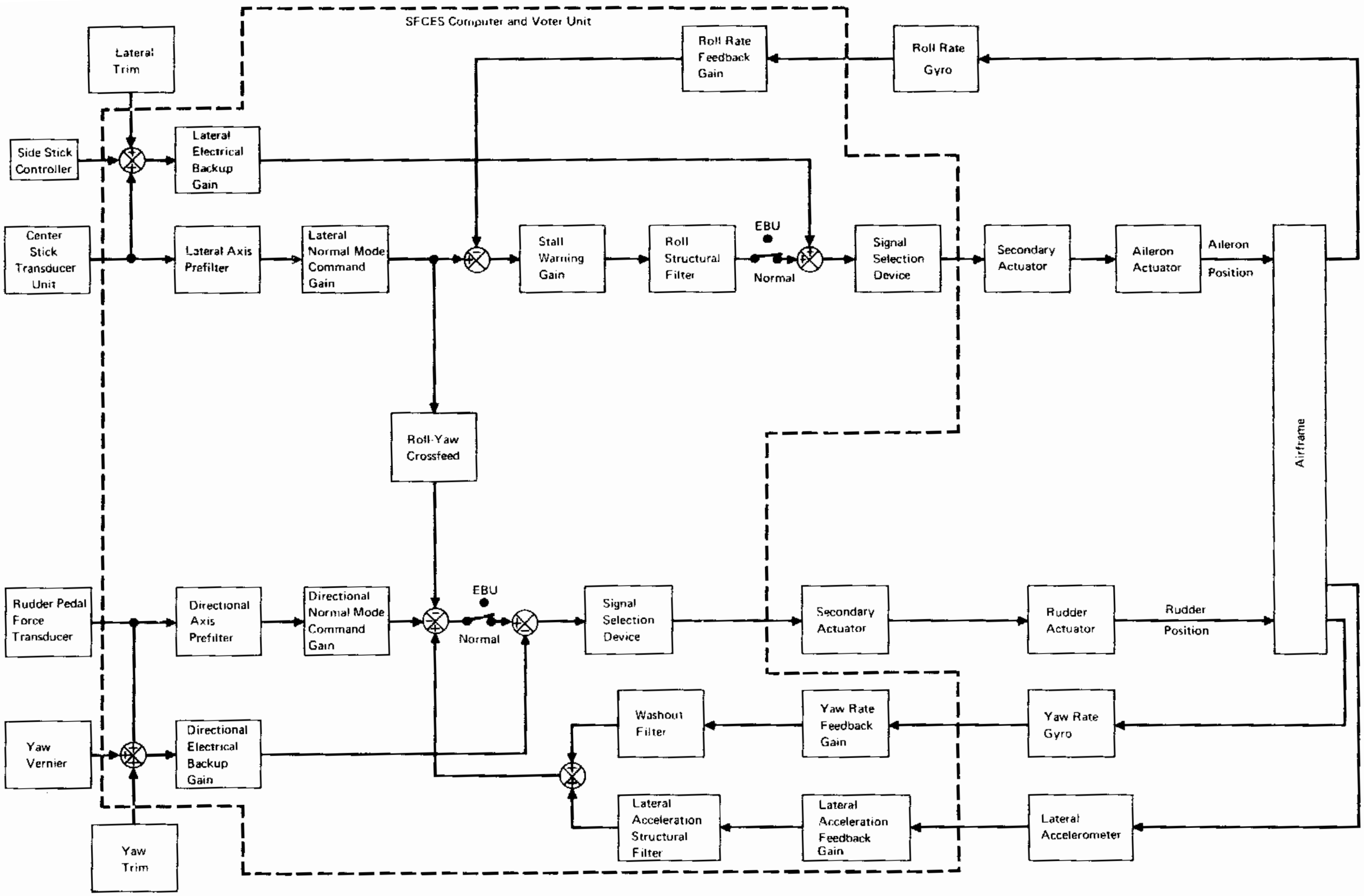


FIGURE 6
SIMPLIFIED FUNCTIONAL BLOCK DIAGRAM
SINGLE CHANNEL OF LATERAL AND DIRECTIONAL AXES

networks to improve turn coordination. The roll to yaw crossfeed circuit is utilized at all flight conditions except high q where the adaptive gain is set to zero, since crossfeed is not required. Performance characteristics of the roll and yaw axes are presented in Section III, Paragraph 13b.

SECTION III

STUDIES AND ANALYSES

1. SINGLE POINT FAILURE ANALYSES

a. Introduction and Summary

A Single Point Failure Analysis is being conducted by MCAIR on the Survivable Flight Control System (SFCS). The purpose of this analysis is to identify those single points of failure where a malfunction might cause loss of flight control of the SFCS test aircraft. The analysis is planned to consist of an end-to-end study of the total flight control installation including power supplies, failure monitoring circuits, aircraft plumbing and wiring and all components of the flight control system.

This analysis has just been started, and to date is limited to available information on current configurations of the Side Stick Controller (SSC), Secondary Actuator (SA), Survivable Stabilator Actuator Package (SSAP) and Survivable Flight Control Electronic Set (SFCES).

The following paragraphs contain a brief description of the function of each of the major procured equipments which comprise the SFCS. A detailed description of procured equipment is contained in Section IV.

Failures identified to date, which might result in loss of flight control are delineated herein.

b. Side Stick Controller (SSC)

The Side Stick Controller (SSC) provides the pilot with a means of transmitting pitch and roll command signals to the SFCES. An SSC is installed in the front and rear cockpits. Fore and aft motion of the stick will introduce four identical signals, derived from the four pitch LVDTs, into the four pitch channels of the SFCES. Lateral motion of the stick will produce four identical signals, derived from the four roll LVDTs, for introduction into the four roll channels of the SFCES. A trigger switch, which is used to activate four single pole double throw microswitches, is mounted in the stick grip to provide a means for disengaging the normal mode of the SFCES and engaging the electrical back-up mode. A pitch vernier thumb wheel, spring loaded to center, is also mounted in the stick grip and provides four outputs, each derived from a separate LVDT, for use in commanding small changes in blended pitch rate and normal acceleration.

Major parts which comprise the SSC are the pitch suspension, roll suspension, pitch vernier, trigger, neutral lock, arm rest adjustment, terminal boards, and connectors. A block diagram depicting these major parts is contained in Figure 7.

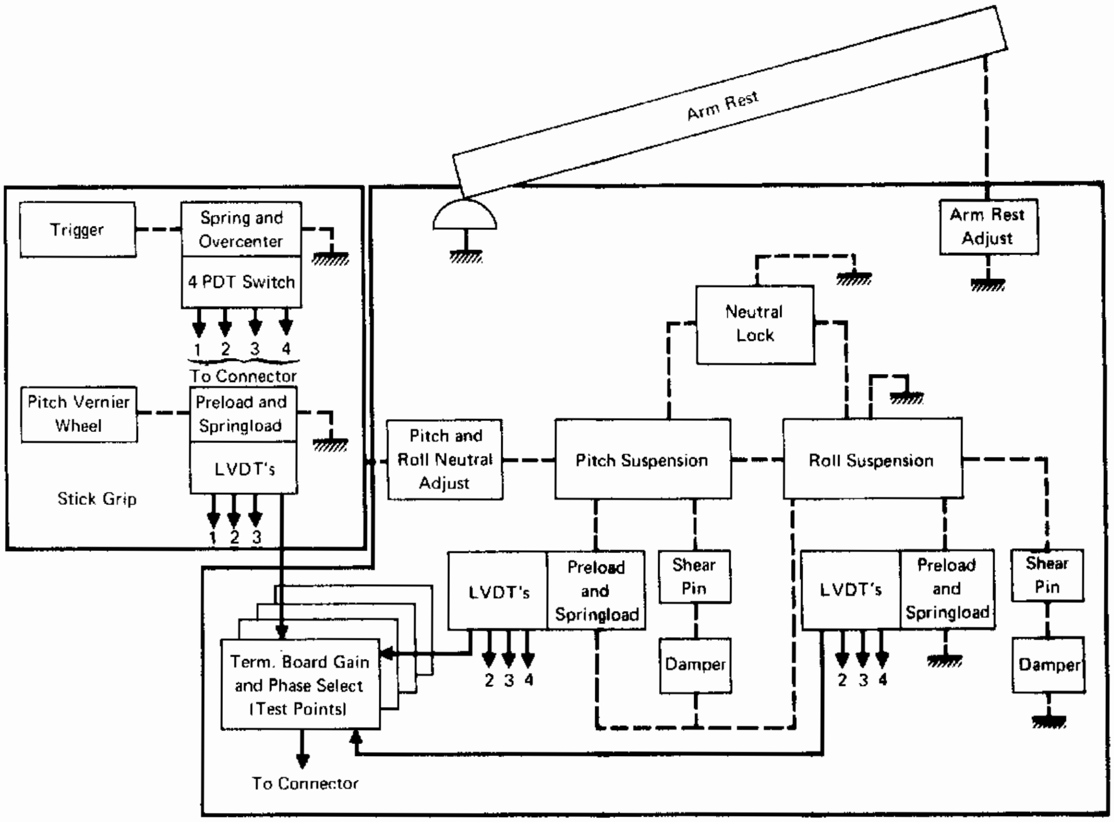


FIGURE 7
SIDE STICK CONTROLLER FUNCTIONAL BLOCK DIAGRAM

Failures which might result in a complete loss of one or both axes of the SSC function (Single point failure) are tabulated in Table I.

c. Secondary Actuator (SA)

The secondary actuator (SA) is a quadruplex, force summing, electro-hydraulic mechanism. It is a self-contained unit consisting of four independent elements coupled to a common output. Each element accepts individual electrical signals and is connected to an individual hydraulic supply. The output is a mechanical motion which positions the slide valve of an existing F-4 power actuator. Each element contains a position transducer for electrical feedback to the SFCES.

The lateral and directional secondary actuators are designed to return to neutral and hold after three failures which are similar. The term, failures which are similar, as used herein, includes all failures within elements which prevent those elements from operating within required limits. The longitudinal secondary actuator is designed to hold in its failed position after three failures which are similar. Each actuator element incorporates a differential pressure sensor that produces a signal which is used to detect a faulty element and provide a signal to affect cutoff by the shutoff

TABLE I
SIDE STICK CONTROLLER SINGLE POINT FAILURES (1)

No.	Part	Failure	Effects	Suggested Action(s)
1.	Pitch Suspension	Jammed	No stick motion possible. No output if jammed at stick neutral position. Fixed output if jammed at other than neutral position.	Maintain control of aircraft through use of trim controls and center stick.
2.	Pitch Preload and Spring Load	Jammed	Same as 1 above	Same as 1 above
		Broken Spring	Stick will not center automatically	Engage neutral lock to center control and null output. Same as 1 above.
3.	Pitch Damper	Locked Up	Same as 1 above	Application of force to stick grip will break shear pin and disconnect damper from system.
		No Damping	Change in "feel system" characteristics	Engage neutral lock. Use center stick.
4.	Roll Suspension	Jammed	Same as 1 above	Same as 1 above
5.	Roll Pre-load and Spring Load	Jammed	Same as 1 above	Same as 1 above
		Broken Spring	Same as 2 above	Same as 2 above
6.	Roll Damper	Locked Up	Same as 1 above	Same as 3 above
		No Damping	Same as 3 above	Same as 3 above
7.	Pitch Vernier Pre-load and Spring Load	Jammed	No vernier motion possible. If jammed in neutral no output will be available from vernier. If jammed away from neutral position system will receive continuous output.	Use trim control to wipe out undesired input.
		Broken Spring	Vernier will not return to neutral	Very limited control authority available through vernier. Effect easily wiped out through use of trim.
8.	Trigger Switch	Jammed	Impossible to engage electrical backup mode via SSC	Use paddle switch on center stick or switches on Master Control and Display Panels to engage backup mode.
9.	Neutral Lock	Jammed While Locked	Side stick will not be usable.	Use center stick
		Jammed While Unlocked	Not possible to lock stick at neutral.	Avoid bumping

(1) Failures which might result in complete loss of one or both axes of the SSC function.

valve. The front frame assembly contains a spring operated centering mechanism or a spring operated brake mechanism, as applicable. A mechanical schematic of the SA is presented in Figure 8.

Each actuator element contains a servovalve, position feedback transducer (LVDT), solenoid shutoff valve, and actuator cylinder. The piston in each actuator element is connected to a common rocker arm assembly by means of a quill shaft.

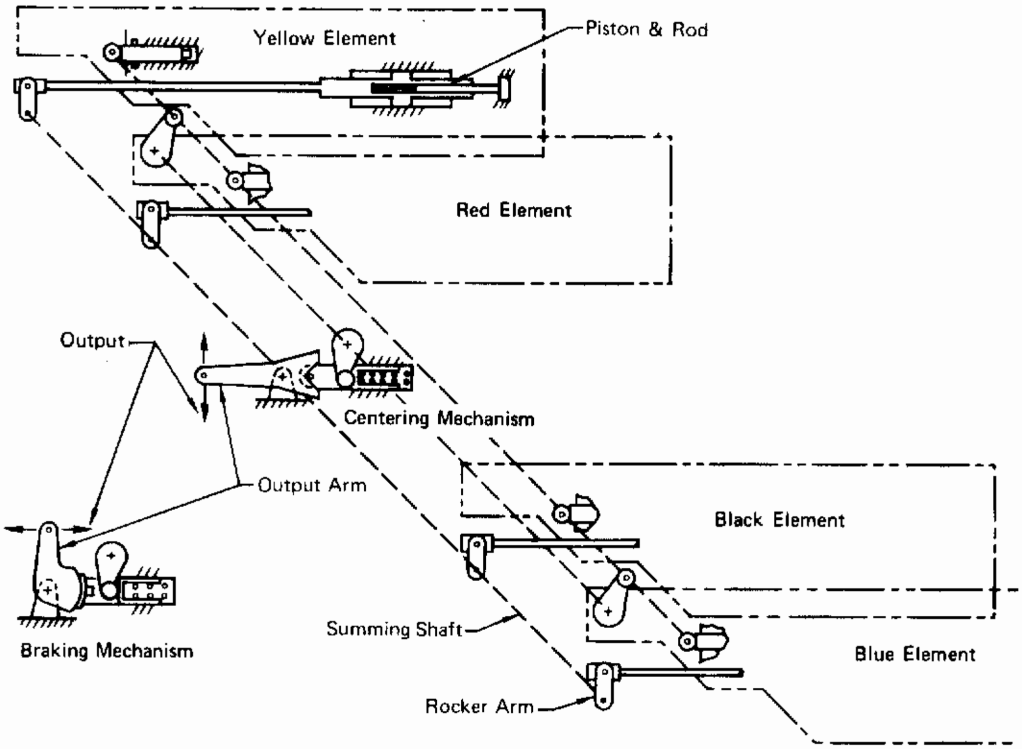


FIGURE 8
MECHANICAL SCHEMATIC, SECONDARY ACTUATOR

TABLE II
SECONDARY ACTUATOR SINGLE POINT FAILURES[1]

No.	Part	Failure	Effect(s)	Suggested Action(s)
1.	Piston or Piston Rod	Seizure or Jam	Total actuator motion stopped	Phase 2A- Revert to mechanical backup in Pitch and/or Yaw Axis) Phase 2B and 2C - None
2.	Rocker Arm Assembly	Seizure	Same as 1 above	Same as 1 above
3.	Summing Shaft	Broken	Loss of output	Same as 1 above
4.	Output Arm	Broken	Loss of output	Same as 1 above
5.	Centering or Braking Mechanism	Jammed	Same as 1 above	If detected during preflight, repair or replace LRU. Engaged in flight only after 3 failures. If jammed open, actuator will not center or hold. If centering mechanism is jammed while deactivated, EMERGENCY ROLL/YAW will not be usable. If brake jams while deactivated EMERGENCY PITCH will be usable as will any other mode. One operating actuator element can overcome braking friction.

[1] Failures which might result in complete loss of the secondary actuator function.

The servovalve consists of a D.C. torque motor and a jet pipe. The solenoid shutoff valve contains a spring and the solenoid winding. The differential pressure transducer contains a spring, spools and sleeve, and an LVDT.

Those failures, identified to date, which might result in a complete loss of the Secondary Actuator function are tabulated in Table II.

d. Survivable Stabilator Actuator Package (SSAP)

The Survivable Stabilator Actuator Package (SSAP) comprises four LRUs. They are (1) a surface actuator which includes a dual tandem cylinder assembly and a hydraulic valve housing assembly, (2) a four-channel electromechanical secondary actuator, and (3) two electric motor driven hydraulic power supplies.

A functional block diagram of the SSAP is presented in Figure 9.

Major components which comprise a motor pump LRU are the electric motor, pump, reservoir, heat exchanger, relief/bleed valve, solenoid valve and filter. The surface actuator LRU major components are the main control valve, piston rod assembly and barrel.

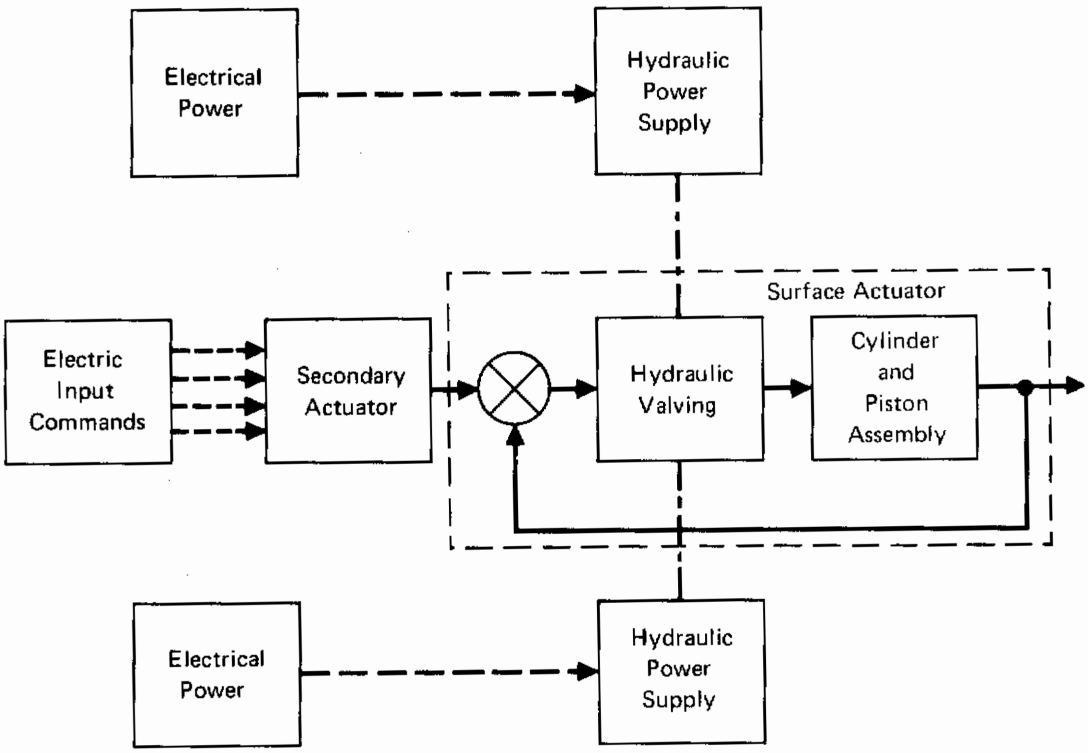


FIGURE 9
SURVIVABLE STABILATOR ACTUATOR PACKAGE FUNCTIONAL BLOCK DIAGRAM

Secondary Actuator components are electric motors, brakes, tachometers, differential gear assemblies, ballscrews and output yoke assemblies. The mechanical schematic for the secondary actuator LRU is presented in Figure 10. A more complete description of the Secondary Actuator LRU is provided in Section IV.

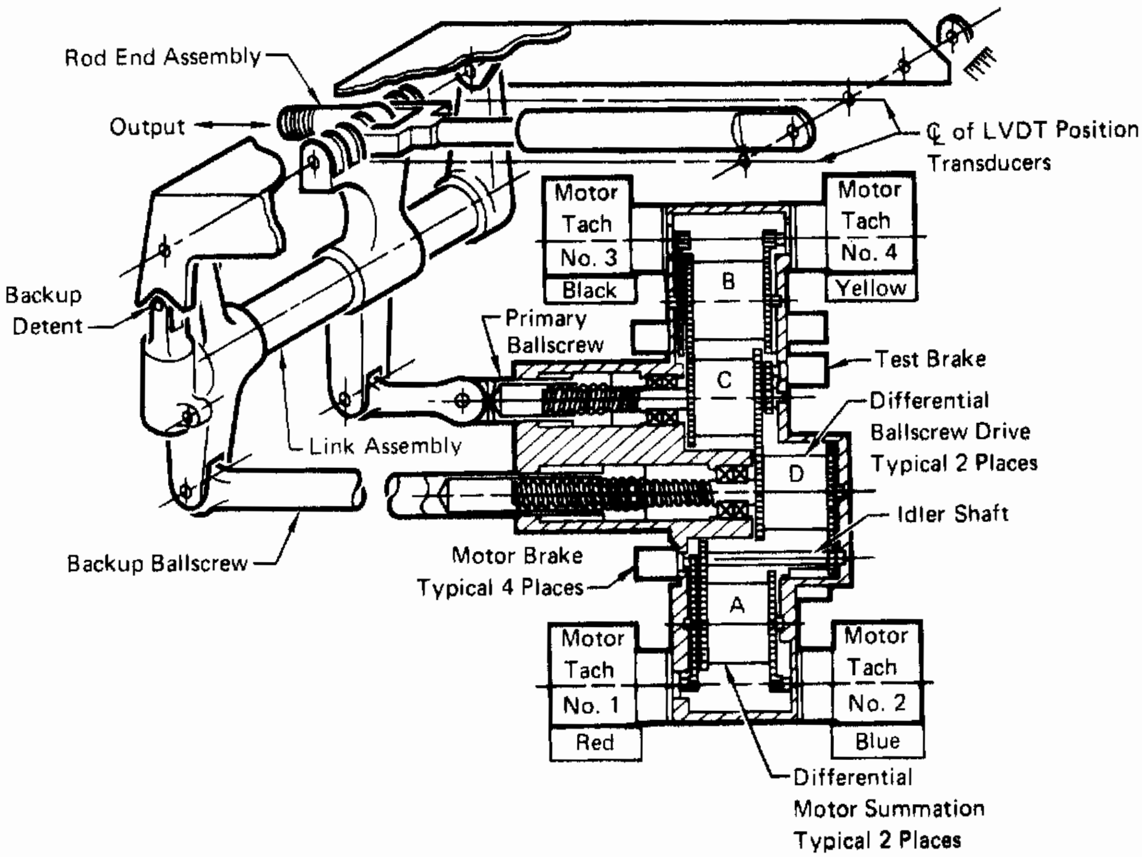


FIGURE 10

ELECTROMECHANICAL SCHEMATIC SECONDARY ACTUATOR

Those failures, identified to date, which might result in a complete loss of the SSAP function are tabulated in Table III.

TABLE III
SURVIVABLE STABILATOR ACTUATOR PACKAGE SINGLE POINT FAILURES (1)

No.	Part	Failure	Effect(s)	Suggested Action(s)
	Secondary Actuator LRU			
1	Motor Tachometer	Sheared pinion gear Sheared motor shaft	Loss of stabilator control	None - Catastrophic
2	Position Transducer LVDT	Jammed	Same as 1 above	None - However, SA has output force capability in excess of 400 pounds to overcome jam
3	Idler Shaft	Sheared teeth	Same as 1 above	None - Catastrophic
4	Ballscrew	Primary ballscrew broken	Same as 1 above	None - Catastrophic
		Primary ballscrew jammed	Loss of primary output	Automatic reversion to secondary ballscrew
5	Link Assembly	Jammed	Same as 1 above	None - Catastrophic
		Broken	Same as 1 above	None - Catastrophic
6	Rod End Assembly	Broken	Same as 1 above	None - Catastrophic
		Jammed	Same as 1 above	None - Catastrophic
	Surface Actuator LRU			
7	Main Control Valve	Jammed	No control of actuator output	None - Catastrophic
8	Actuator Attachment Lug	Broken	No control of surface	None - Catastrophic
9	Output Rod or Piston	Jammed	No actuator output	None - Catastrophic

(1) Failures which might result in complete loss of SSAP function.

e. Survivable Flight Control Electronic Set (SFCES)

The Survivable Flight Control Electronic Set (SFCES) is a quadruplex electronic device designed to provide the pilot with the primary means for controlling the aircraft. The SFCES will accept pilot commands as electrical signals, sum these signals with signals derived from position, rate, and acceleration sensors, perform the necessary signal conditioning, monitoring and signal selection functions and convert the resulting signals into the proper values for controlling the electrohydraulic and electromechanical actuators. The SFCES provides adaptive gain changing and stall warning as off-line computational functions and is able to determine its own operational status through an off-line Built-In-Test capability. An electrical back-up mode, which allows direct pilot control of surface position, is provided in each axis to permit bypassing of computational electronics and rate and acceleration sensors.

The SFCES is comprised of 25 LRUs. A description of each LRU type and its intended function is provided in Section IV of this report.

Because the SFCES was packaged to be retrofitted into an existing aircraft, which limited the degree of dispersion of redundant components that might be accomplished in an aircraft specifically designed for a system of this type, certain redundant components of the set were placed in the same LRU. Those LRUs, not including LRUs which involve off-line computational functions, which contain four redundant components are:

- pitch rate sensor unit
- roll rate sensor unit
- yaw rate sensor unit
- normal accelerometer sensor unit
- lateral accelerometer sensor unit
- control stick transducer unit
- pedal transducer unit
- stabilator surface position transducer unit

The electrical and electronic redundancy of the SFCES is designed to preclude the possibility of an electrical or electronic failure from propagating and resulting in the total loss of any of the above units. No single point failures have been identified to date which will cause complete loss of the SFCES. Those failures which will cause loss of one mode of operation in one or more axes are delineated in Table IV.

Incorporation of the EBU mode into the SFCES will prevent failures of Rate and Acceleration Sensor Units from resulting in a catastrophic situation. The SSC may be used to back up the Control Stick

TABLE IV

SFCES FAILURES RESULTING IN LOSS OF A MODE[1]

No.	Part	Failure	Effect(s)	Suggested Action(s)
1.	Pitch Rate Sensor Unit	Mount	Loss of pitch rate feedback signals	Revert to electrical backup EBU mode
2.	Roll Rate Sensor Unit	Same as 1 above	Loss of roll rate feedback signals	Same as 1 above
3.	Yaw Rate Sensor Unit	Same as 1 above	Loss of yaw rate feedback signals	Same as 1 above
4.	Normal Accelerometer Unit	Sames as 1 above	Loss of normal acceleration feedback signals	Same as 1 above
5.	Lateral Accelerometer Unit	Same as 1 above	Loss of lateral acceleration feedback signals	Same as 1 above
6.	Control Stick Transducer Unit	Broken Grip Loss of Feel System	Pilot has no means of putting in pitch or roll commands Same as above	Use sidestick controller Use sidestick controller
7.	Pedal Transducer Unit	Attachment Broken	Rudder will not respond to pilot inputs	Aileron to rudder crossfeed will provide turn coordination
8.	Stabilator Surface Position Transducer Unit	Attachment Broken Jammed	Change in actuator dynamics None	None required. Revert to EBU mode if desired Surface actuator has sufficient force capability to overcome jam

(1) Failures which might result in partial loss of an SFCES mode.

Transducer Unit and the aircraft may be controlled without benefit of the Rudder Pedal Transducer Unit. Loss of the Stabilator Surface Position Transducer Unit will not cause a catastrophic situation.

f. Other Equipment

To accomplish the installation of the SFCS in the test aircraft, numerous modifications to the existing electrical and hydraulic systems, equipment installations, and support structure are planned to be made. These modifications are being reviewed; and the SFCS single point failure analysis will be updated and future results will be incorporated in subsequent reports.

g. Conclusions

From the single point failure analyses conducted to date on current configurations of the procured equipment it may be concluded that:

(1) Although there are failures of the SSC that could result in the inability of the pilot to control the aircraft through continued use of the SSC, the center stick may be utilized to circumvent the failure and maintain control of the aircraft.

(2) During Phase IIA, failures which preclude continued use of the SAs are circumvented by reversion to mechanical back-up in the pitch and yaw axes. The use of two SAs in the roll axis provides an octamerus roll control system and provides dual redundancy in those areas where a single failure would result in loss of output of one secondary actuator. During Phase IIB and IIC there are single failures which will preclude continued use of the SAs in the pitch and yaw axes. Only those failures which occur in the pitch axis may be catastrophic failures.

(3) The SSAP represents the worst case, at least from the standpoint of single point failures which could be catastrophic. Disregarding failures of the main control valve, attachment points and output rod or piston, which are present in the production actuator as well as the SSAP, the electromechanical secondary actuator appears to be the LRU which will require the most attention to design detail.

(4) The SFCES contains four independent channels for all on-line computational functions. However, it was not advantageous or even possible in the case of such items as the stick force sensor unit, pedal transducer unit and stabilator position transducer unit, to separate and disperse all associated elements. For these elements and the others noted in Paragraph (e), the probability of the failures noted appears to be highly unlikely. Besides, the effects of all of these failures probably can be circumvented by using the appropriate back-up modes.

2. SYSTEM SAFETY ANALYSES

a. General

Engineering effort has been directed towards identifying hazards which could be introduced by the modification of the aircraft to incorporate the SFCS and instrumentation equipment. A Preliminary Hazard Analysis and an Electrical Subsystem Analysis were conducted.

b. Preliminary Hazard Analysis

An analysis was conducted on the Survivable Flight Control Electronic Set (SFCES), Secondary Actuator, Side Stick Controller, and Survivable Stabilator Actuator Package (SSAP) to identify potential Class III (critical) and Class IV (catastrophic) hazards. The hazards identified to date for the equipment with associated hazard classifications are itemized as follows:

Hazard	Classification
(1) Survivable Flight Control Electronic Set	
Rate or Acceleration Sensor Unit	III
Undetected BIT Failure	III
Failure of the IFM	III
Lack of Channel Isolation	III
(2) Secondary Actuator	
Pitch Actuator Summing Shaft Jammed	IV
Piston or Piston Rod Jammed	IV
Output Arm Broken	IV
Summing Shaft Broken	IV
(3) Side Stick Controller	
Jammed Spring Cartridge (Pitch or Roll)	III
(4) Survivable Stabilator Actuator Package	
Sheared Motor Shaft or Pinion Gear	IV
Sheared Teeth on Idler Shaft	IV
Primary Ballscrew Broken	IV
Link Assembly Jammed or Broken	IV
Rod End Assembly Jammed or Broken	IV
Main Control Valve Jammed	IV
Actuator Attachment Lugs Broken	IV
Output Rod or Piston Jammed	IV

The above potential hazards have been studied for possible elimination. It was not feasible or possible to eliminate the hazards listed above. Therefore, the system has been designed to include a high margin of safety to reduce the probability that these hazards will be encountered.

c. Electrical Subsystem Analysis

An analysis was conducted on the split bus electrical power system as designed for the test aircraft. This split bus system consists of two 30 KVA generators, each of which is connected to a separate load bus. Either generator is capable of assuming the total load with the exception of one SSAP motor pump unit, and temporarily operating in an overload condition. However, actuation of the circuit protective devices must occur in the proper sequence before the bus tie contactor will close and permit a single generator to assume this load. When this occurs, AC power will be provided to both the left and right 115 VAC buses by the remaining generator. Power from the 115 VAC buses is supplied to each SFCS channel transformer-rectifier (T/R) which in turn supplies 28 VDC channel power to the SFCES and maintains a full charge on the back-up battery. The loss of the AC generators or a T/R output will result in a shift to battery power for continued SFCES operation.

The electrical design for the power distribution that is to be installed in the test aircraft is the safest design yet for F-4 aircraft. No Class III or Class IV hazards have been found in the SFCS test aircraft electrical system.

3. RELIABILITY ANALYSES

Reliability is a prime consideration in the design of the SFCS. The design goal is a failure rate not exceeding 2.3 x 10^{-7} failures in one hour. The reliability effort to achieve this goal consists of system and subsystem analyses, allocation of reliability requirements, and configuration and component reliability studies including Failure Modes and Effects Analyses (FMEA).

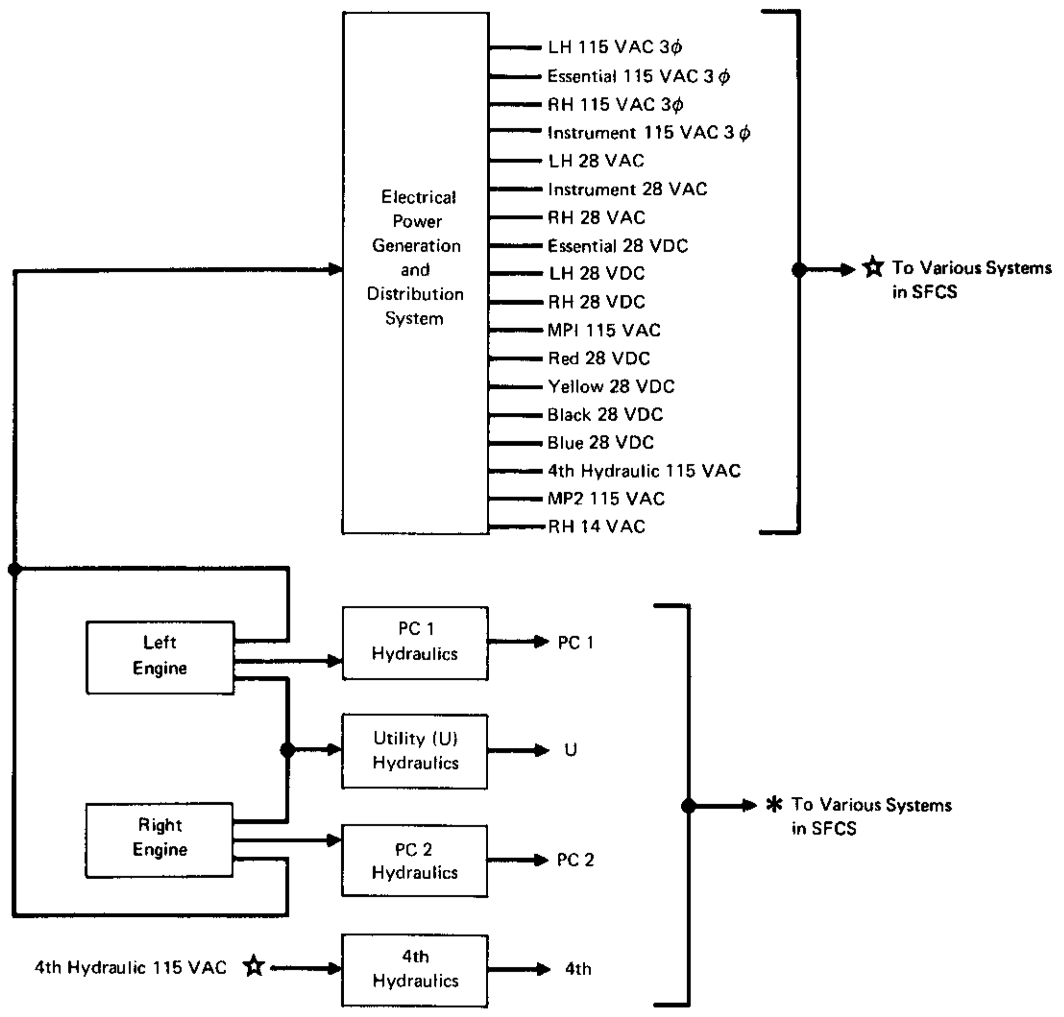
a. System and Subsystem Analyses

The SFCS has been periodically analyzed as significant changes in configuration or failure rates have occurred. A summary of an analysis of the current configuration is presented below using failure rates based primarily on MCAIR F-4 field data and reliability estimates and FMEA's from major Suppliers. The analysis was conducted using the reliability diagrams depicting current SFCS configuration for Phases IIA, IIB and IIC, as shown in Figures 11 and 12 and the State Interpretation Program (SIP) as presented in Reference 2. SIP state diagrams were generated for Phases IIA, IIB and IIC as shown in Figures 13, 14, and 15, respectively. A maximum of three transitions (arrows) to reach the final state was considered, since four or more transitions have a negligible effect on the results. The final state in each diagram depicts an SFCS failure defined as:

- o Either loss of longitudinal control, or
- o Loss of directional and lateral (spoilers on one wing and aileron on the other wing) control.

Component and system failure rates and designations used in the state diagrams are listed in Tables V and VI in alphanumeric order by designation. The probability of loss of control for Phase IIA, IIB, and IIC as predicted by the SIP results are given in Table VII as Q IIA, Q IIB, and Q IIC. The Qs were calculated for flight durations up to 2.5 hours in 0.1 hour increments. The Phase IIA and Phase IIB Qs for a one hour flight (underlined in Table VII) are 10.20 x 10^{-7} and 10.66 x 10^{-7} respectively, while the Phase IIC Q is 2.70 x 10^{-7}.

The difference in the failure rates is a direct reflection of the single point failures and the associated failure rates used. For Phases IIA and IIB, an estimated failure rate of 10.0 x 10^{-7} was used for the stabilator surface actuator, which is consistent with that used in Reference 1. This is the predominant failure mode for Phases IIA and IIB. For Phase IIC, the SSAP catastrophic failure rate, estimated by the Supplier to be 2.6 x 10^{-7}, becomes predominant. Subtracting these failure rates from the respective one hour Qs gives the failure rate of the remainder of the SFCS: 0.20 x 10^{-7} for Phase IIA, 0.66 x 10^{-7} for Phase IIB, and 0.10 x 10^{-7} for Phase IIC. In Phase IIA these remaining failures are contributed mainly by the MIM and linkages; in Phase IIB by the Secondary Actuator and linkages, and in Phase IIC by the SSAP fittings and attaching aircraft parts.



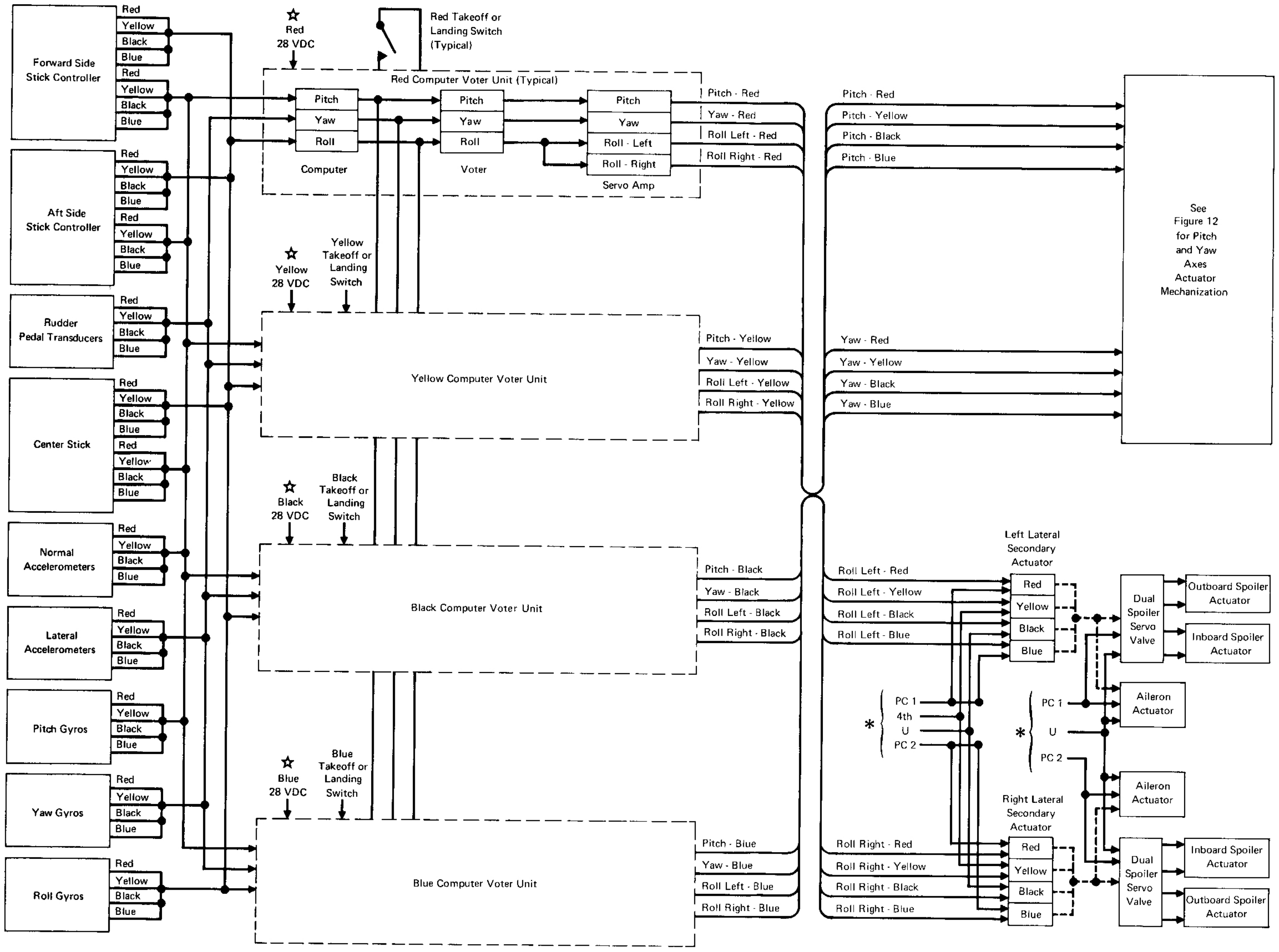


FIGURE 11
PHASE II RELIABILITY DIAGRAM

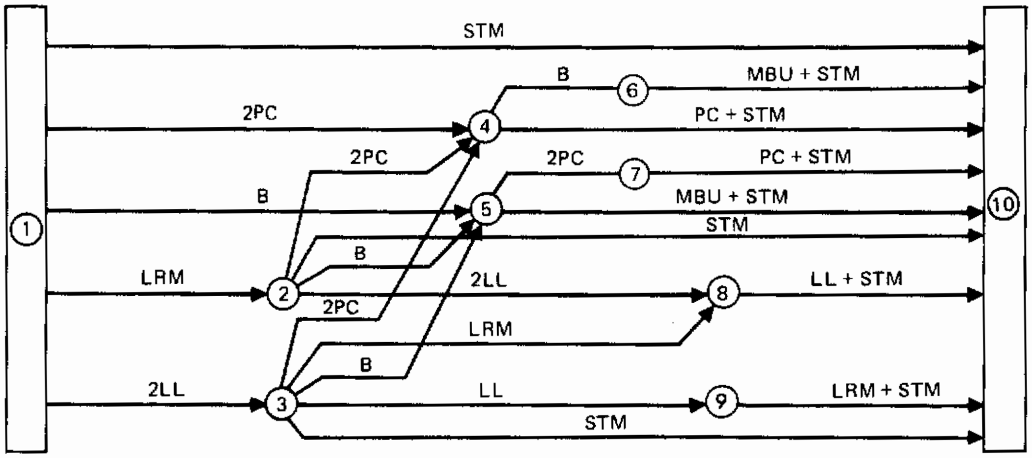
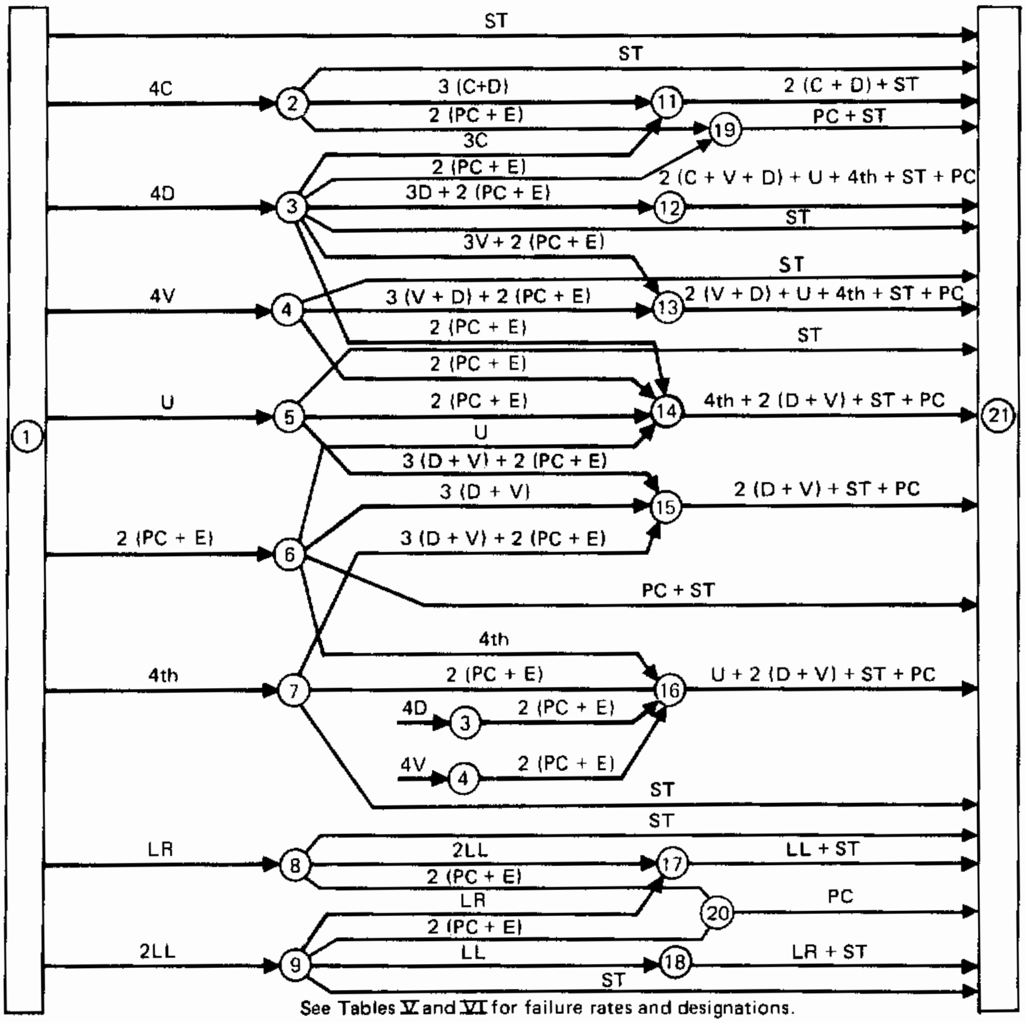
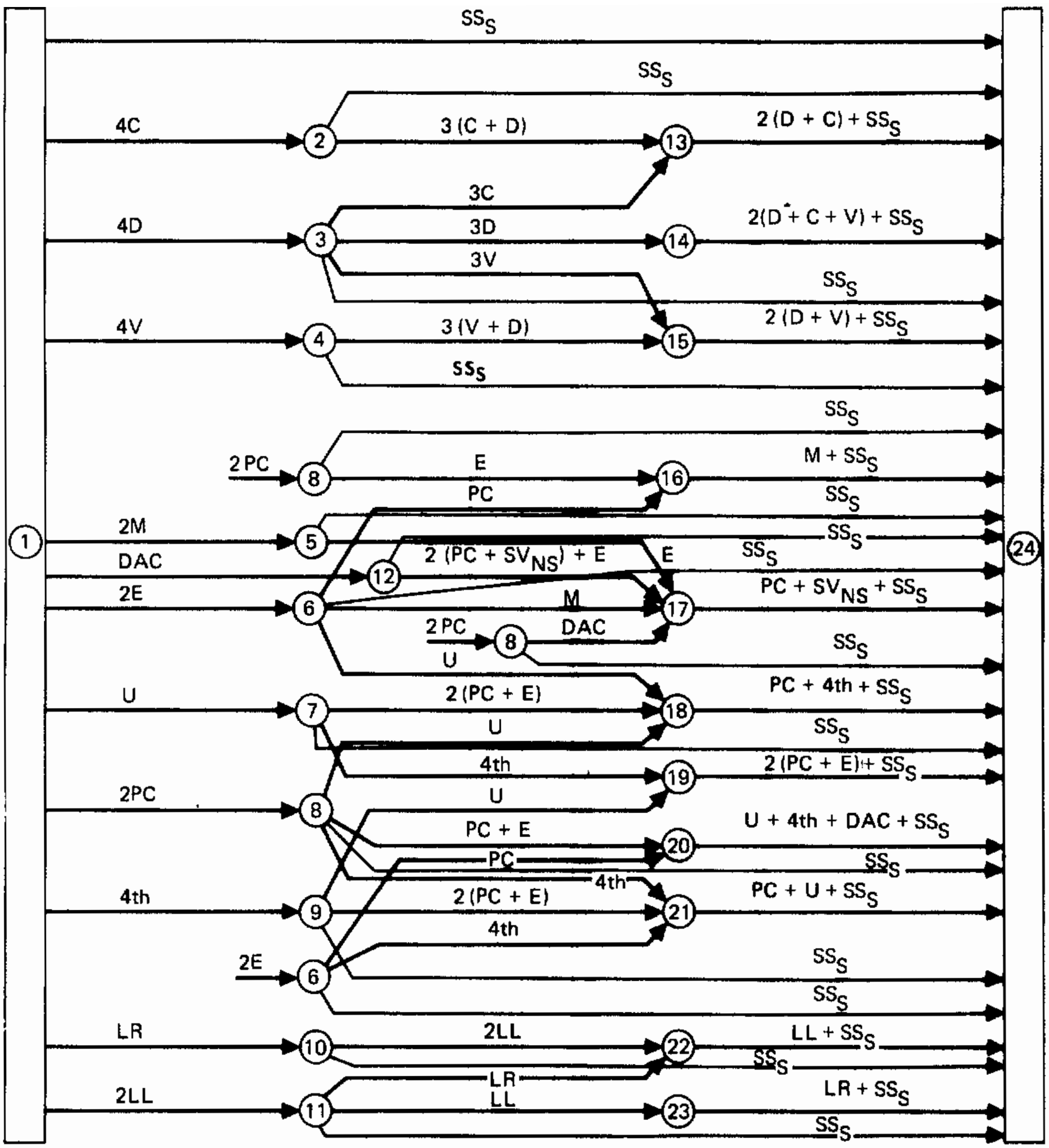


FIGURE 13
PHASE IIA STATE DIAGRAM



See Tables V and VI for failure rates and designations.

FIGURE 14
PHASE II B STATE DIAGRAM



See Tables V and VI for failure rates and designations

FIGURE 15
PHASE IIC STATE DIAGRAM

TABLE V
COMPONENT AND SYSTEM FAILURE RATES

Component or System	Designation	Failure Rate (Per 10^6 Hours)
Aileron Actuator	AA	0.1
Dual Spoiler Servo Valve	DSSV	0.1
Left and Right Hand 115 VAC (Dual Failure)	Dual 115 VAC	14.0
Engine	ENG	25.0
Essential 28 VDC Power	Ess 28 VDC	(0.707×10^{-12})
Hydraulic Solenoid Valve	HV	2.0
Pitch Linkage	Link P	0.01
Lateral Linkage (One Wing)	Link R	0.01
Rudder Linkage	Link RD	0.01
Mechanical Backup (Mechanical Controls)	MBU	0.03
Mechanical Isolation Mechanism Actuator	MIMA	1.0
Directional Mechanical Isolation Mechanism	MIM D	0.01
Pitch Mechanical Isolation Mechanism	MIM P	0.01
Motor Pump (SSAP)	MP	386.0
Motor Pump 115 VAC Electrical Power	MP 115 VAC	1,764.0
Normal Accelerometer	NA	8.1
Power Control Hydraulic System	PC	30.0
Pitch Computer Channel	PCC	25.0
Pitch Gyro	PG	3.8
Pitch Servo Amplifier Channel	PSA	1.0
Pitch Secondary Actuator Channel	PSECA	158.0
Pitch Secondary Actuator Single Failure	$PSECA_s$	0.055
Pitch Voter Channel	PV	1.8
Rudder Actuator Single Failure	RAs	3.3
Rudder Damper Single Failure	RDs	0.1
SFCS-Mech Select Switch	SFCS-Mech Sw	0.1
Stabilator Position Transducer	SPT	8.3
SSAP Motor-Tachometer Brake	SSAPmtb	12.0
Survivable Stabilator Actuator Package Single Failure	SSAPs	0.26
Stabilator Actuator Single Failure	STAs	1.0
Switching Valve - No Switch	SVns	0.1
Takeoff - Land Switch	TOLsw	5.0
Utility Hydraulic System	U	143.0
Utility Pressure Switch	UPS	10.0
Yaw Secondary Actuator Single Failure	YSECAs	0.055
Fourth Hydraulic System	4th	120.0
Channel 28 VDC Power	28 VDC	3.2

The one hour Qs which include all axes for Phases IIA (10.20×10^{-7}), IIB (10.66×10^{-7}), and IIC (2.70×10^{-7}) are less than the one hour Q of 11.45×10^{-7} for a standard F-4 aircraft longitudinal axis as presented in Reference 1. It is therefore anticipated that the SFCS reliability will exceed that of the standard F-4 flight control system in all Phases of the program.

It is anticipated that upon completion of the SSAP design the Phsse IIC catastrophic failure rate may be adjusted downward and that the SFCS reliability goal of 2.3×10^{-7} will be realized.

TABLE VI
COMBINED COMPONENT AND SYSTEM FAILURE RATES

Combined Components or Systems	Combined Designation	Failure Rate (Per 10^6 Hours)
U + HV + MIM A + UPS	B	156.1
SPT + PCC + NA + PG	C (Phase IIB)	45.2
PCP + NA + PG	C (Phase IIC)	36.9
28 VDC + TOL_{sw}	D	8.2
Dual 115 VAC	DAC	14.0
ENG	E	25.0
Link R + AA + DSSV	LL	0.21
RAs + RDs + Link RD + YSECAs	LR	3.465
RAs + RDs + Link RD + MIM D	LRM	3.42
MP + MP 115 VAC	M	2150.0
SSAPs	SSs	0.26
STAs + Link P + PSECAs	ST	1.065
STAs + Link P + MIM P	STM	1.02
PV + PSA + PSECA	V (Phase IIB)	160.8
PV + PSA + $SSAP_{MTB}$	V (Phase IIC)	14.8

b. Allocation of Requirements

The reliability goal for the SFCS is a failure rate not to exceed 2.3×10^{-7} per hour. To achieve this goal, the individual failure rates of the SFCS equipment cannot collectively exceed 2.3×10^{-7}.

The SFCS comprises an electronic set, actuators, linkages and power sources. The results of a preliminary study indicated that the electronics should be allocated a requirement of 1.0×10^{-7} failures per hour and that the remainder of the goal, 1.3×10^{-7} should be shared by the remainder of the SFCS.

(1) SFCES

The SFCES allocation of 1.0×10^{-7} failures per hour was subdivided by allocating 3.3×10^{-8} failures per hour to the electronics in each aircraft axis. Under the premise that an axis is lost after three of the four redundant electronic channels have failed, the failure rate of each channel is given by:

$$\lambda_C = \sqrt[3]{\lambda_A / 4}$$

where: λ_C = channel failure rate

λ_A = axis failure rate

TABLE VII
PROBABILITY OF SFCS FAILURE vs FLIGHT TIME

Flight Time (hrs)	Probability of Failure (1)		
	Q IIA	Q IIB	Q IIC
0.0	0.0	0.0	0.0
0.10	0.1020090D-06 (2)	0.1065166D-06 (2)	0.2700000D-07 (2)
0.20	0.2040361D-06	0.2130666D-06	0.5400004D-07
0.30	0.3060812D-06	0.3196505D-06	0.8100013D-07
0.40	0.4081443D-06	0.4262687D-06	0.1080003D-06
0.50	0.5102254D-06	0.5329217D-06	0.1350006D-06
0.60	0.6123246D-06	0.6396099D-06	0.1620011D-06
0.70	0.7144419D-06	0.7463337D-06	0.1890017D-C6
0.80	0.8165771D-06	0.8530937D-06	0.2160025D-06
0.90	0.9187304D-06	0.9598903D-06	0.2430036D-06
1.00	0.1020902D-05	0.1066724D-05	0.2700049D-06
1.10	0.1123091D-05	0.1173595D-05	0.2970065D-06
1.20	0.1225298D-05	0.1280504D-05	0.3240085D-06
1.30	0.1327524D-05	0.1387451D-05	0.3510108D-06
1.40	0.1429767D-05	0.1494437D-05	0.3780134D-06
1.50	0.1532029D-05	0.1601462D-05	0.4050165D-06
1.60	0.1634308D-05	0.1708527D-05	0.4320201D-06
1.70	0.1736606D-C5	0.1815632D-05	0.4590241D-06
1.80	0.1838921D-05	0.1922778D-05	0.4860286D-06
1.90	0.1941255D-05	0.2029964D-05	0.5130336D-06
2.00	0.2043607D-05	0.2137192D-05	0.5400392D-06
2.10	0.2145976D-05	0.2244462D-05	0.5670454D-06
2.20	0.2248364D-05	0.2351775D-05	0.5940522D-06
2.30	0.2350770D-05	0.2459130D-05	0.6210596D-06
2.40	0.2453193D-05	0.2566528D-05	0.6480677D-06
2.50	0.2555635D-05	0.2673970D-05	0.6750766D-06

Notes: (1) Sufficient to cause loss of longitudinal control or loss of directional and lateral control. (Electrical backup not considered)

(2) D-06 equals x 10^{-6}, D-05 equals x 10^{-5}, etc.

Since the axis failure rate is allocated to be 3.3×10^{-8} failures per hour, the channel failure rate should not exceed 2.02×10^{-3} failures per hour.

It was felt that the best way to specify the reliability requirement was as a series failure rate or Mean Time Between Failures (MTBF). The SFCES has twelve channels, four per axis, which results in a total SFCES failure rate of 24.24×10^{-3} failures per hour to achieve the goal of 2.02×10^{-3} failures per hour per channel. An appropriate safety factor was then applied to the overall failure rate, resulting in an SFCES failure rate of approximately 5.0×10^{-3} or a minimum acceptable MTBF of 200 hours. Application of the 2:1 discrimination ratio of Test Plan IV of MIL-STD-781 resulted in a specified requirement of 400 hours for the SFCES MTBF.

(2) Actuators

The state diagram analysis discussed above contemplated two catastrophic failure situations: (1) loss of longitudinal control or (2) loss of both lateral and directional control. The directional control system was included in the above analysis because an experienced F-4 pilot can, under normal flight test conditions, land the aircraft using the directional control system to compensate for a passively failed lateral control system. Since an allocation is made for establishment of goals at lower equipment levels, it was decided that a pessimistic or worst case approach should be taken. Under this assumption, any possible redundancy contribution from the directional control system was discounted for allocation purposes. This leaves loss of either longitudinal or lateral control a catastrophic event. Therefore, starting again with the SFCS reliability design goal of 2.3×10^{-7} failures per hour, 1.7×10^{-7} and 0.6×10^{-7} failures per hour were allocated to the pitch and roll axes respectively. The roll axis was allocated the lower failure rate because loss of control in both wings is necessary for roll axis loss.

(a) SSAP

The pitch axis allocation of 1.7×10^{-7} has been sub-allocated as follows:

Pitch Axis = SSAP Multiple Failures + SSAP Catastrophic Failures + Four Channel Power Sources + Control Linkage + Four Channel Electronics

SSAP catastrophic failures were specified as a maximum of 5.0×10^{-8} failures per hour. F-4 field data indicates a longitudinal control linkage failure rate of 5.0×10^{-8}. Substituting these rates into the above formula leaves a total failure rate of 0.7×10^{-7} to be allocated among the remaining portions of the SSAP, channel electronics, and channel power sources.

Using the previously established equation for channel failure rate results in a failure of 2.6×10^{-3} failures per hour per pitch channel. In a worst case situation where there is no voting between channels, each pitch axis channel must be sub-allocated as follows:

Pitch channel = SSAP Channel + DC Supply + SFCES Channel + Utility Hydraulic Supply

At the time of the SSAP allocation it was not known if the secondary actuator LRU of the SSAP was going to be Electro-Mechanical (EM) or Electro-Hydraulic (EH). Therefore, as a worst case it was assumed that four hydraulic supplies would be necessary (one for each secondary actuator channel of the SSAP) and the Utility supply was used since it has the highest failure rate of the hydraulic supplies.

Substituting estimated failure rates into this equation resulted in an SSAP channel failure rate as follows:

$$2.6 \times 10^{-3} = \text{SSAP channel} + 0.01 \times 10^{-6} + 1.25 \times 10^{-3} + 170 \times 10^{-6}$$

Solving:

SSAP Channel = 1.17×10^{-3} failures per hour

The total SSAP failure rate was then allocated to be $4(1.17 \times 10^{-3}) = 4.68 \times 10^{-3}$ failures per hour for all four channels. Rounding to 4.0×10^{-3} failures per hour, the SSAP minimum allowable MTBF was specified as 250 hours.

(b) Secondary Actuator

The secondary actuator minimum acceptable MTBF of 1000 hours was allocated by considering the relative complexity between it and the SSAP and knowing that an MTBF at least as good as the SSAP was desirable. A catastrophic failure rate of 5.0×10^{-8} was also specified.

c. Configuration and Component Reliability Studies

Analysis of possible component failure modes and their effect on the overall system is necessary to arrive at a configuration which will have the least susceptibility to multiple-channel failures and still provide an acceptable level of performance. The single point failure analysis discusses those instances where, in MCAIR's judgment, it was either not possible or not practical to provide multi-channel redundancy.

In most cases, however, such constraints do not apply and some manipulation of components and approaches to arrive at the most reliable configuration is possible. The Suppliers are furnishing Failure Mode and Effects Analyses (FMEAs) for their equipment to assist in the effort. Typical studies involving both Supplier and MCAIR furnished components are discussed below.

In considering the hydraulic line material to be used, it was anticipated that introduction of the SSAP would increase the severity of the heat and vibration environment in the stabilator actuator area. If the motor-pumps installed on the SSAP produce vibration, the plumbing will have to survive this environment. Data on the presence of this vibration will, hopefully, be generated during future testing. Since new plumbing was to be added and most of the existing plumbing in the area had to be reworked to provide clearance for the SSAP, the question arose as to whether it would be better to stay with production type aluminum or go to some other material. Titanium was considered briefly but was dropped because of the development lead time and cost involved. Although stainless steel has at various times been judged to be too heavy for overall production use, it had become common practice to use it in field repairs, especially on "problem" lines which gave trouble more than once on the same aircraft. The anticipation of increased problems with aluminum in the stabilator area led to the conclusion that it would be worthwhile in this case to accept the weight penalty and replace aluminum hydraulic lines reworked during the SFCS program with stainless.

The connector and wire bundle philosophy being followed for SFCS provides another example where consideration of the reliability factors involved had a strong influence on configuration and design. It was recognized early in the program that it would be absolutely essential that a fault (opens, shorts between pins or to ground etc.) in a single connector not be permitted to disable the entire SFCS. As a result, the SSC, the GE secondary actuator, and the SSAP all have separate cylindrical connectors for each of the four channels. In the SFCES, the channels are completely separated with a computer for each channel. Rack and panel connectors were used on the computers to accommodate the large number of interconnecting voting status wires, and particular attention is being paid to buffering of this wiring to prevent cross-coupling failure modes. The control stick was one element of the electronics set where space restrictions precluded as much channel isolation as might be desirable; given the space available, it was necessary to run all four channels through a single connector.

Aircraft wiring runs between connectors and LRUs presented a more difficult reliability problem than might be apparent at first glance. Problems in wiring that could affect more than one channel would be just as detrimental to aircraft survival as problems at the connectors.

It was decided to isolate the channel wiring as much as the use of an existing aircraft would allow. While shorts, especially between channels, are a major concern in the control circuits, opens could be more critical in certain power circuits. A number of the most important power circuits are being run in triplicate to head off problems from this failure mode. Since the intent of the triple wire run is to protect against open circuits, and there are similar sets of three wires to each CVU, the three wires to each CVU are bundled together, rather than dispersed.

d. Conclusions

As a result of reliability analyses conducted to date it appears that attainment of the SFCS reliability goal of 2.3×10^{-7} failures per hour is a possibility. It is anticipated that the SFCS reliability will exceed that of the standard F-4 aircraft flight control system in all phases of the program.

4. SURVIVABILITY

a. General

Survivability analyses have been conducted to date for two purposes, namely:

- o A survivability assessment for the Survivable Stabilator Actuator Package (SSAP) degree of redundancy trade study, and
- o Review and evaluation of proposed, and subsequently procured, hardware required for implementation of the SFCS.

b. SSAP Redundancy Trade Study

A limited trade study was conducted to determine the degree of redundancy to be specified for the SSAP. In all, nineteen configurations were considered. From a survivability standpoint, each configuration was evaluated for its ability to continue to provide longitudinal control after sustaining up to three projectile hits on components comprising each of the configurations. The survivability ratings of these configurations were then considered, along with ratings obtained from other disciplines for other parameters affecting redundancy, in making the final configuration selection. Results of the SSAP trade study survivability analysis are presented in Supplement 3.

c. Evaluation of Proposed Hardware

Supplier proposals for the subcontracted components required for implementation of the SFCS were evaluated. Since the SFCS program is an advanced development program and the components are to be installed in an existing F-4 aircraft, component sizing and location are restrained by available space. Because of the imposed space and sizing restrictions, proposed hardware for each similar major component tend to be of nearly the same size, although the design techniques employed to accomplish the desired flight control system functions differed between proposing prospective subcontractors.

In reviewing the survivability attributes of proposals for each major component, consideration was given to the following factors:

- o Redundant channel separation and isolation
- o Compactness of design
- o Effects on total system installation
- o Provisions for structural integrity
- o Reduction of exposed external linkages

o Reduction of exposed external hydraulic plumbing

o Jam release features

o Inherent self-shielding

With one component, namely the secondary actuator, there was significant variance in design concept. Both electromechanical and electrohydraulic designs were proposed. Viewed as an individual component, the electromechanical secondary actuator would appear to be more fragile, i.e., there is a higher probability that a single projectile hit could cause sufficient damage to eliminate the required mechanical ground and thus disable all four channels. The electrohydraulic secondary actuators are more rugged in this respect. From a systems installation standpoint, the electromechanical secondary actuator eliminates the need for the additional hydraulic system plumbing (and its associated vulnerabilities) required to provide the four channel redundancy for the electrohydraulic secondary actuators.

The results of all survivability considerations were then included, along with important factors from other disciplines, in determination of subcontractor selection for each major component.

In the Survivable Flight Control Electronic Set, vulnerability criteria have been compromised to solve the redundant sensor tracking and alignment problems. The rate and acceleration sensors are packaged as quadruplex units for each axis. Also, in the rate sensor package some mechanical separation is sacrificed to achieve an overall package size that can be fitted into the test aircraft. In any event, the presented areas of these sensor packages are quite small and the aircraft can be safely flown in the electrical backup (EBU) mode without use of the sensors. Therefore, the compromises which were made are undoubtedly of minor consequence.

In all other respects, physical and electrical separation of redundant elements and components is being accomplished to the extent practicable for a development system installation in an existing test aircraft. This conforms to the intent of enhanced survivability through dispersed redundancy. For a new aircraft, designed from scratch so to speak, even more consideration could be given to channel separation, component separation, component design arrangements, component placement within the aircraft, etc., all of which should further enhance aircraft survivability.

5. MAINTAINABILITY

The primary elements of the SFCS Maintainability studies are Maintainability Analyses, Incorporation of Maintainability Requirements in Vendor Specifications, and Design Review. The following is a summary of the results of these tasks as accomplished to date. More detailed discussion of the Analyses are given in Appendix I.

a. Analyses

(1) Ground Test of the SFCS DC Power Supplies

Each channel of the SFCS derives its electrical power from an independent DC power supply consisting of a transformer rectifier (T/R) and a Nickel Cadmium (NiCd) battery in parallel. Since the proper functioning of these supplies is essential to the safe operation of the SFCS, it was concluded that an automated check would be implemented in the Survivable Flight Control Electronics Set (SFCES) Built-In-Test (BIT) circuitry. The BIT check consists of a simultaneous test of voltage received by the SFCES and current flow through the battery. The voltage and current levels tested define a properly functioning T/R and a charged battery. This test will be backed up by the implementation of scheduled inspection and servicing of the batteries. In-flight monitoring of the SFCS DC power supplies is not considered necessary. If a particular channel power supply were to be reduced below the required operating level, signal transmission degradation in that channel will be detected by the In-Flight-Monitor (IFM) and that channel will be disengaged.

(2) Probability of a False GO or Potentially Hazardous Condition Existing at the Completion of Ground BIT Check

The ability of the ground BIT to correctly identify the operating condition of the SFCS will help determine the confidence the flight crew has that a successful and hazard free mission may be flown. Therefore, an analysis of whether a GO displayed at the end of the ground BIT check really means that the system is GO, considering GO and NO-GO probabilities and the resulting BIT system displays, was undertaken. The analysis considered the condition of the SFCS functional equipment, the condition of the subsystem, and the display possibilities following missions of varying durations. The results indicate that the probability of indicating a false GO, which is a potentially hazardous condition, varied from 0.000368 to 0.001776 for 1 to 5 hour missions respectively. For a mission of 1.3 hours the probability is 0.000478, or one false GO in 2720 flight hours. Since the calculations were based on worst case conditions, as described in Appendix I, it may be assumed that only a small portion of the failures resulting in a false GO would result in a hazardous condition. Therefore, the probability of such a condition going undetected by ground BIT and resulting in an unsafe flight condition would be considerably less than the figures calculated in the study.

(3) SFCS Support Equipment

The support equipment for the SFCS F-4 modified and added subsystems was identified and listed to aid in identifying the overall SFCS support requirements. A listing of the special and standard test and servicing equipment required to provide maintenance support of the modified systems of the test aircraft may be found in Appendix I.

(4) SFCS Maintenance Manhours Per Flight Hour

MMH/FH figures were estimated for an SFCS F-4 and compared to a conventional flight control configured RF-4C aircraft. Base aircraft for the comparison was an RF-4C less its reconnaissance mission sensors and equipment. The RF-4C MMH/FH figures were further modified to reflect the split bus AC power supply to be used on the test aircraft. The SFCS F-4 was assumed to be configured with an SFCES, SSC, electrohydraulic lateral and directional SA's and an SSAP. The result of modifying the RF-4C to the SFCS configuration was estimated to increase the overall MMH/FH by approximately 0.188. The MMH/FH calculations are presented in Appendix I.

(5) BIT Check of the SSAP Blower Motors

A study was made to determine if a requirement existed for a check of the SSAP motor-pump heat exchanger and Secondary Actuator blower motor operation by the SFCS IFM or BIT circuitry. The study concluded that although incorporation of IFM and/or ground BIT checks of the SSAP blowers was desirable, the benefits did not warrant the additional complexity of the system. This recommendation was based on the availability of flight test instrumentation measurands which would reveal deficient blower operation; lack of confidence in such checks due to the complexity of the circuitry involved; redundancy of the SSAP, and the low probability of multiple failures of the blower motors.

b. Incorporate Maintainability Requirements in Vendor Specifications

The overall goal of the SFCS Maintainability Program is to minimize SFCS maintenance and support requirements to that consistent with the R&D nature of the program. In line with this goal, the procurement specifications and supporting documentation were written to include equipment features, data requirements and maintainability emphasis in areas where the maximum program benefits would accrue without unduly increasing program costs. Four areas where specific equipment features were specified for incorporation in the Supplier's equipment to enhance maintainability are as follows:

(1) SFCS Ground BIT

The ground BIT routine implemented in the SFCES design will provide a major assist in preflight verification and maintenance troubleshooting of the SFCS.

(2) SFCES Maintenance Test Panel (MTP)

The MTP, an LRU of the SFCES, contains the ground BIT circuitry and maintenance displays associated with ground test of the SFCS. The display panel portion of the MTP, as presently configured contains an annunciator panel which lists the LRU's and subsystems which are planned to be fault isolated by the SFCS BIT. The MTP display panel is shown in Figure 60.

(3) Survivable Stabilator Actuator Package (SSAP) Line Replaceable Units (LRU's)

Due to the volume and weight of the SSAP and the restricted access available in the aft fuselage of the test aircraft, it is necessary that the SSAP be constructed in 4 LRU's. These LRU's, as presently configured, can be assembled into the aircraft one at a time, as shown in Figure 66.

(4) Mobile Ground Test Facility (MGTF)

The MGTF was originally envisioned as an analog computer checkout facility to be used to perform closed loop testing of the installed SFCS prior to the first flight and prior to all other first flights following system modifications. Subsequently, the facility was expanded to include limited shop capabilities. As presently configured, the MGTF includes an analog computer, 8 track recorder, SFCES test bench and provisions for use of the SSC suitcase tester, SFCES LRU testers and supporting standard test equipment as required for shop repair of the SFCES and SSC LRU's, and closed loop testing of the aircraft installed SFCS or the SFCES when installed on the system bench in the MGTF.

c. Design Review

Several areas of the test aircraft SFCS installation design have been influenced by the maintenance requirements of the SFCS added equipment. Some of the more significant modifications and provisions added specifically to aid in equipment maintenance and servicing are as follows:

(1) Channel Color Coding

To aid in the identification of the various channels of the SFCS, a system of color coding was adopted. Red, blue, yellow or black will be used with orange to identify the four SFCS channels. SFCS hydraulic lines and wire bundles will be marked with mylar polyester tape. Epoxy paint or lacquer will be used to identify items such as electrical connectors and hydraulic ports on system LRU's or mounting racks by placing the appropriate color combination in close proximity to the connector or port. Rack mounted LRU's, which are interchangeable between channels, such as the SFCES computer and voter units, will not be marked with channel colors. In some cases, the electrical connectors are keyed individually for the four channels in order to insure the proper relationships between channel electrical and hydraulic supplies.

(2) SFCS NiCd Battery Mounting Arrangement

The SFCS NiCd batteries, located on the lower side of the aircraft, will require frequent servicing. Therefore, the battery mounting brackets are arranged to allow them to be swung down for easier battery removal and installation.

(3) APU Servicing Panel

The fourth hydraulic power supply auxiliary power unit (APU) will require preflight inspection for reservoir fill level. Additionally, it is estimated that APU servicing will be required at 10 operating hour intervals. In order to ease these inspection and servicing requirements, an APU servicing panel was added to the aircraft. The panel has a transparent insert which permits viewing of the APU reservoir indicator directly. The APU fill port is located directly behind this panel and is reached by loosening two airlock fasteners and opening the hinged door.

(4) Lateral SA Access Panels

Access panels were added to the aircraft to aid in the routing of hydraulic lines and electrical wiring in the area of the lateral SA's. Additionally these panels may be used to gain direct access to the SA electrical connectors, if necessary, for trouble shooting or interface with the MGTF.

(5) Use of Self Retained Bolts (SRB's) in the SFCS Control Linkages

The MS 27576 SRB provides a fail safe installation as regards nut loss due to human error or mechanical failure. For this reason, the MS 27576 SRB and its associated nut, the MS 21244, have been selected for use in the SFCS. All bellcranks and/or rod end locations that have SRB's installed will be marked with blue paint as a means of identification.

(6) Removable Frame for the Installation of SSAP LRU's

The weight and bulk of the SSAP LRU's requires modification of the frame structure bisecting the access area behind doors 63 and 65. This frame member will be strengthened and modified to allow temporary removal during installation and removal of the SSAP LRU's.

6. AERODYNAMIC STUDIES

The aerodynamic characteristics of the test aircraft were provided for utilization in both analytical studies and six-degree-of-freedom man-in-the-loop investigations of the in-flight handling qualities. These data included the results of recent stall-near stall flight tests and high angle of attack Langley Wind Tunnel Tests. The low and moderate angle-of attack data are contained in Reference 11. The high angle of attack data used for simulations were taken from Reference 16.

7. THERMODYNAMICS

a. General

The SFCS is the primary flight control system. Failures due to overtemperature conditions, if allowed to encompass the redundant SFCS channels, can result in loss of the aircraft. Whether encompassing all channels or not, nuisance failures due to overtemperatures or other adverse temperature conditions could impede to varying degrees the flight testing of the SFCS.

The elemental functions performed by the SFCS in control of the aircraft, viz., actuation and control of actuation, require many upstream stages of power application which produce heat. Ultimately, waste heat - produced in the SFCS or its surroundings - accounts for all of the expended power except for part of that expended by the SSAP. The SSAP provides a directly useful actuation output during half-cycles having opposing aerodynamic loads. However, tending to thermally offset this output, is the return power input to the actuator during half-cycles when the aerodynamic load aids actuator motion. If identical actuation rates and loads are repeated in consecutive half-cycles, the returning input power is virtually equal to the output power. Therefore, at thermal equilibrium, nearly all power delivered to the SFCS is removed as heat. Consumed electrical power in the form of waste heat cannot be removed by wire, its transmission media. This contrasts with the use of the transmission media for hydraulic power. Virtually all the hydraulic power consumed by an actuator remote from the hydraulic power source can be removed by the returning hydraulic flow. For both the electrical and hydraulic cases, heat is produced by power losses incurred in power generation, conversion, storage, transmission, control and final use. The equilibrium temperature of the various equipments at each of these stages of power application is determined by the impedance offered to heat transfer and the capacity and temperature of the heat sink employed for the particular equipment.

The most important factors in the thermal energy balance for the SFCS are presented in Figure 16.

b. Thermodynamic Analyses

The principal thermodynamic analyses performed for this program are summarized below and described more completely in Appendix II.

- Determining the heat expected to be generated.
- Establishing the methods and capability to dissipate heat.
- Establishing appropriate limits for compartment air and structural temperatures.
- Ascertaining equipment vulnerability to cooling system failures for SFCS equipment and aircraft equipment.

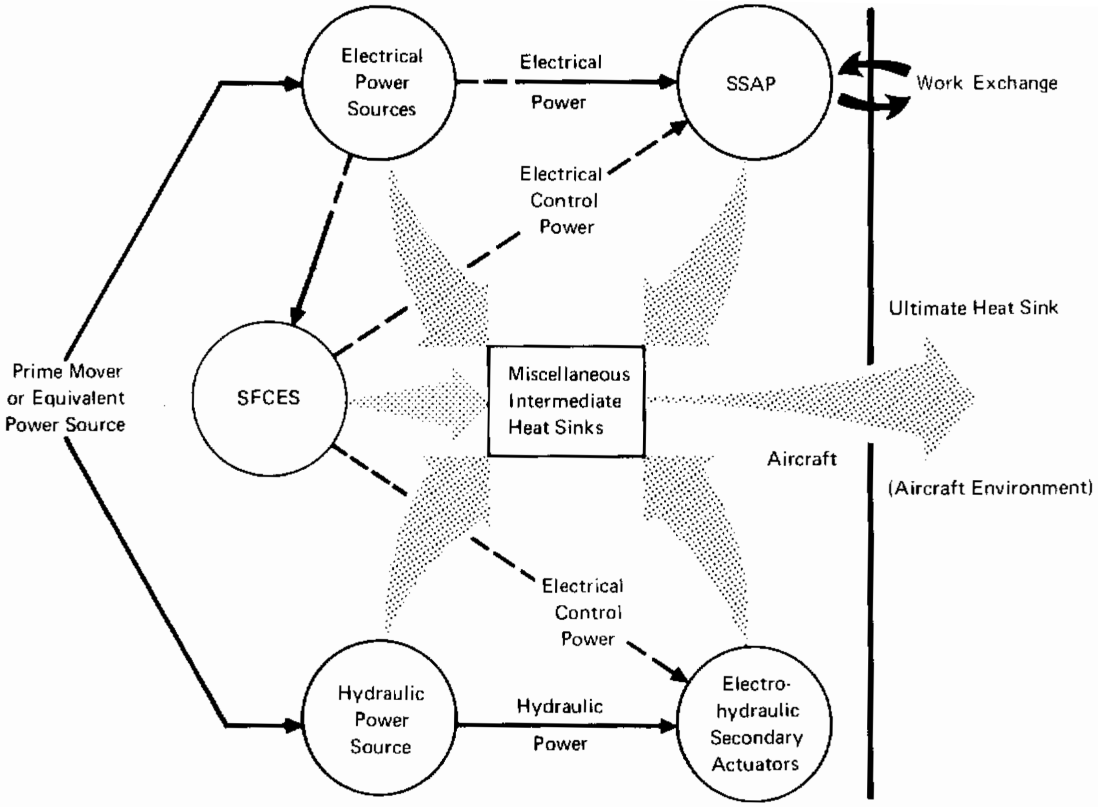


FIGURE 16
ENERGY BALANCE FOR SFCS

o Precluding component temperature problems.

o Determining the proper distribution of ventilating air.

o Assessing and determining methods to minimize, to the extent practicable, the aircraft penalty ensuing from thermal design requirements.

These analyses were performed to assess the thermal impact of the heat dissipation and environment of the special electrical and hydraulic power sources and using equipment of the SFCS. The power sources specifically considered include:

o the transformer rectifiers (T/R's) and batteries

o the fourth hydraulic system

and the using equipment considered includes:

o the Survivable Flight Control Electronics Set (SFCES)

o the Survivable Stabilator Actuator Package (SSAP)

o the Secondary Actuator (SA)

c. Conclusions

The thermal analyses have been used to define a thermal design and testing approach for the SFCS equipment. The approach described in Appendix II provides for the maintenance of acceptable temperatures in the SFCS equipment. Potential thermal problems have been identified and preventive measures implemented in accordance with available data and analyses. The predicted adequacy of the equipment thermal design will be verified empirically in formal quality assurance and other tests. These include simulation of the equipment thermal environment that might result from an aircraft refrigeration package failure. SSAP tests will include simulated failures of the self-contained cooling fans.

8. ELECTRICAL SYSTEM DESIGN STUDIES

Three separate studies were performed as a means of selecting the most reliable electrical system for the SFCS test aircraft. The studies include investigations of:

- o Electrical Power Generation and Distribution, and Load Analysis
- o Wiring Techniques
- o Wiring Dispersion

These electrical system design studies are summarized here and described more completely in Appendix III.

a. Electrical Power Generation and Distribution, and Load Analysis

A design study of the electrical power system including the power sources, distribution, and utilization aspects was accomplished. Design modifications and additions to the test aircraft electrical power distribution system were investigated; variations of the basic split bus system were also studied. This study provides the rationale leading to the test aircraft electrical power distribution system design. In addition, an analysis was performed of the electrical loads expected to be imposed by the existing aircraft electrical equipment plus additional SFCS equipment.

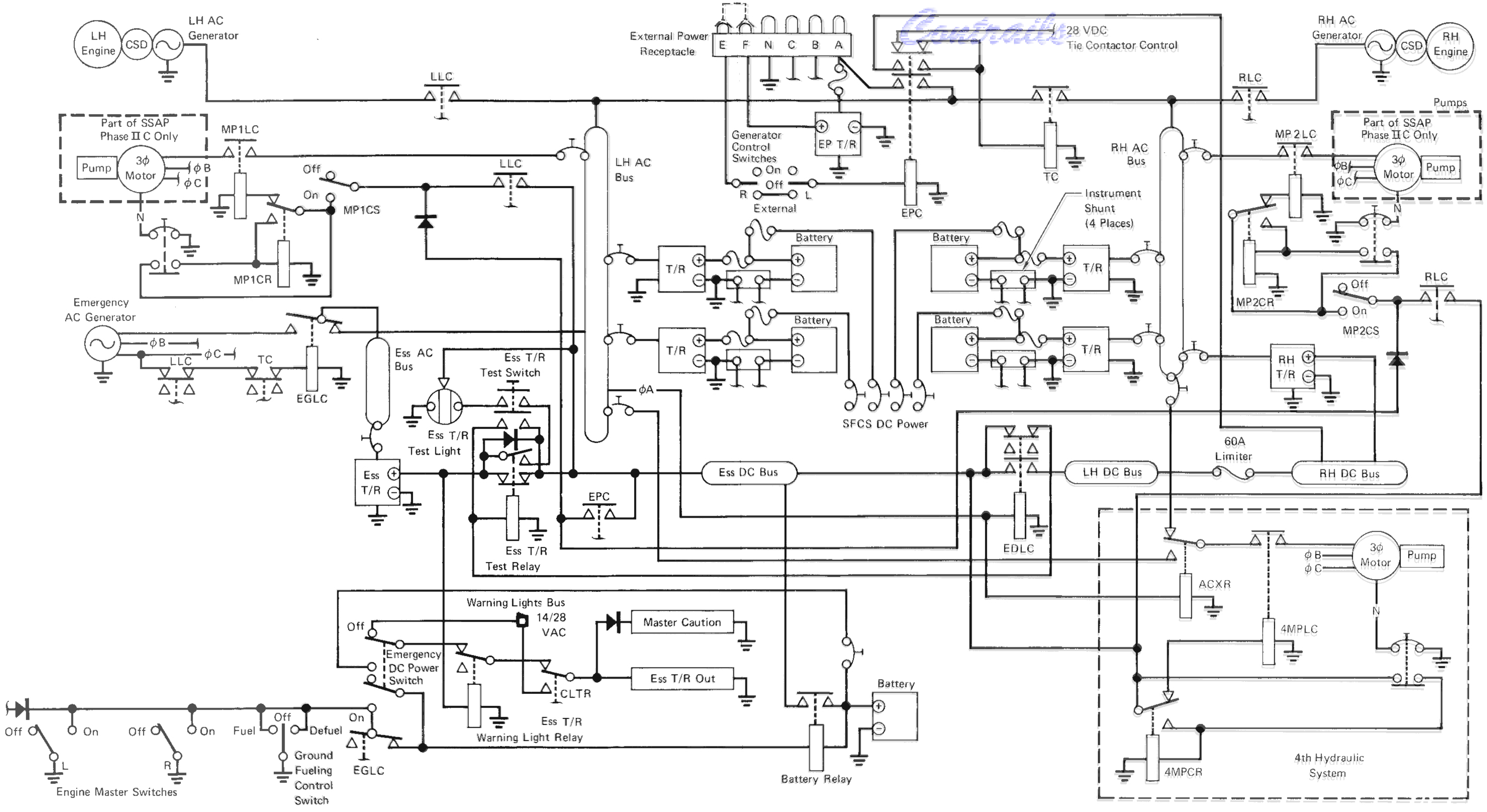
(1) Two basic types of aircraft electric generating and distribution systems are in general use, parallel bus systems and split bus systems. Variations of either basic system are numerous and generally consist of different bus-switching schemes. Bus-switching may be fully automatic, manually controlled, or a combination of automatic and manual control.

A parallel bus system consists of two or more AC generators connected to a common load bus system with suitable synchronization and switch gear and circuit protective devices.

A split bus system consists of two or more AC generators, each connected to a separate load bus. Switch gear and circuit protective devices are used to permit feeding any load bus from one of the operating generators if the generator that normally feeds that bus becomes inoperative.

(2) The decision to use a split bus electric generating and distribution system for the SFCS test aircraft is based primarily on consideration of safety and reliability.

The split bus system planned to be used in the SFCS test aircraft is represented by Figure 17 and consists of two 30 KVA oil cooled AC generators, each connected to a separate isolated main load bus. The two SSAP motor driven hydraulic pumps for the Phase IIC program are to be individually powered from these sources; one



ACXR – AC Source Transfer Relay
CLTR – Caution Light Test Relay
CSD – Constant Speed Drive
EDLC – Essential DC Line Contactor
EGLC – Emergency Generator Line Contactor
EPC – External Power Contactor
EP – External Power
Ess – Essential
LLC – Left Generator Line Contactor
MP1CR – Motor Pump 1 Two-Phase Cutout Relay
MP1CS – Motor Pump 1 Control Switch
MP1LC – Motor Pump 1 Line Contactor
MP2CR – Motor Pump 2 Two-Phase Cutout Relay
MP2CS – Motor Pump 2 Control Switch
MP2LC – Motor Pump 2 Line Contactor
RLC – Right Generator Line Contactor
SSAP – Survivable Stabilator Actuator Package
TC – Bus Tie Contactor
T/R – Transformer Rectifier
4 MPLC – 4th Hyd. System Motor Pump Line Contactor
4 MPCR – 4th Hyd. System Two-Phase Cutout Relay
φ - Phase
N – Neutral

FIGURE 17
SFCS SPLIT BUS ELECTRICAL SYSTEM – PHASE II

from the generator driven by the left engine and the other from the generator driven by the right engine. The motor driven pumps are not switched from one generator to the other during single-generator operation because one generator cannot supply enough power for both motor pump units and the remainder of the aircraft loads. If one of these AC sources fails, an aircraft central hydraulic power system is to be switched into the portion of the SSAP normally powered by the inoperative motor pump.

The design for the SFCS test aircraft electrical system also provides four redundant sources of DC power for the SFCS flight control circuits, each source consisting of a Transformer-Rectifier (T/R) shunted with a battery. Each of the four added batteries is kept charged by an individual T/R, which normally supplies the SFCS DC power. The added T/R's are 100 ampere, nominal 28 VDC, fan-cooled units conforming to MS17976-2. The added batteries are Nickel-Cadmium, 22 ampere-hour units conforming to MS24497-5. If the T/R's become inoperative, the batteries are expected to supply usable DC power for about one hour.

(3) An electrical load analysis has been made to determine the adequacy of the two 30 KVA AC generators and to confirm that the loads are properly apportioned to the various AC and DC buses. Proper load distribution is that which results in nearly equal loading of the generators. The results of this load analysis are presented in Table VIII.

TABLE VIII
ELECTRICAL LOAD ANALYSIS SUMMARY

Operating Conditions	Load(1) or Anchor	Start(1) and Warm-up	Taxi(1)	Takeoff(1) and Climb	Cruise(2)	Cruise(2) Combat	Landing(1)	Emer-(1) gency
LH Generator - (30 KVA Rating)		23.57	24.68	24.74	24.44	24.41	24.68	
RH Generator - (30 KVA Rating)		19.62	19.98	21.54	21.20	21.18	21.43	
Emer. Generator - (3.0 KVA Rating)								2.24
Total	*6.05	43.19	44.66	46.28	45.64	45.59	46.11	2.24
Single-Generator Operation (1-hr Rating = 38 KVA)				35.98	35.33	35.27	35.80	

All entries in kilovolt-amperes (KVA)
* From ground power

(1) 15 minute average
(2) 30 minute average

b. Wiring Techniques

Components and installation techniques of various wiring systems have been considered. In order to provide an electrical installation meeting the stated program objectives, the SFCS will be wired with MCAIR developed COMPACT wire bundles. Figure 18 shows a typical COMPACT wire bundle. The COMPACT wiring system uses high temperature Teflon or Kapton insulated wire assembled into bundles which are then overbraided with a Dacron jacket. This creates a flexible multi-conductor jacketed bundle which provides maximum mechanical protection for the SFCS wiring throughout the test aircraft.

COMPACT wiring has been analyzed to determine if any specialized components or procedures are required for the SFCS test aircraft installation. Special consideration was given to the following:

- o Channeling to avoid maintenance damage
- o Fire Protection
- o Overheat Protection
- o Contraction and Expansion of Components

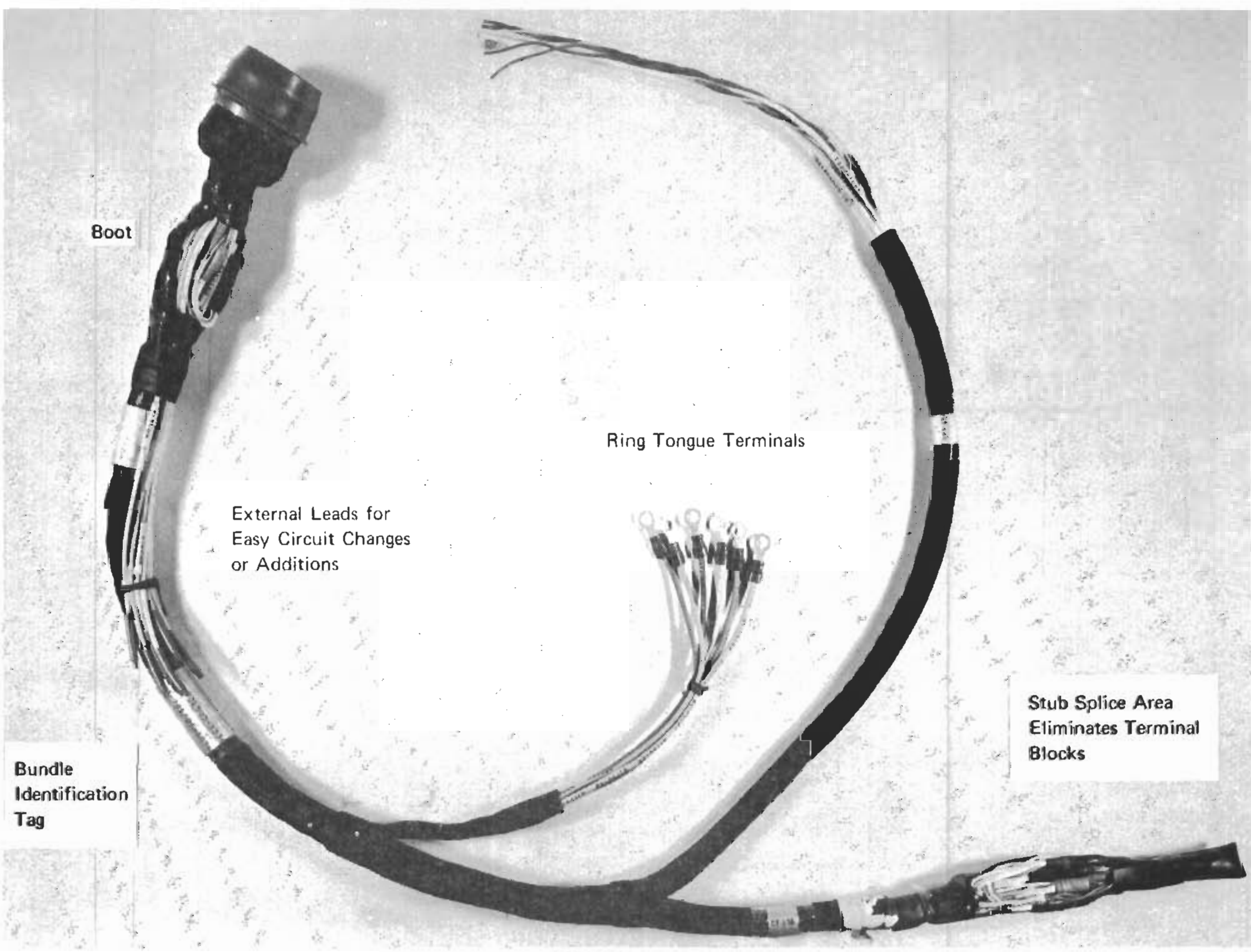


FIGURE 18
TYPICAL COMPACT WIRE BUNDLE

(1) Channeling to Avoid Maintenance Damage

The braid on the COMPACT wire bundles provides good mechanical protection. However, metallic or fiber glass protective channel or conduit will be installed over the bundles in specific areas where the wire bundles are exposed to potential damage.

(2) Fire Protection

The SFCS electrical system is to be protected against overloads, shorts, and mechanical breakdowns which may cause an aircraft fire by the following:

- o Wires of adequate size will be selected to carry the electrical loads.
- o The circuit breakers will be selected with an ampere rating below the current carrying rating of the associated wiring.
- o All relays and switches will be sealed types to contain the arc from "off" and "on" operation.
- o The separation of switches, terminal strips, relays, and circuit breakers will be adequate to prevent flashover between terminals.
- o Adequate clearances will be maintained from all oxygen and hydraulic lines.

(3) Overheat Protection

The standard Dacron covered COMPACT wire bundle has a temperature rating of 300°F, which is sufficient for most applications. When the anticipated ambient temperature exceeds the limits of the Dacron braid, the wire bundle will be covered with Teflon tubing or Nomex high temperature nylon braid to provide a bundle rated to 392°F.

(4) Contraction and Expansion of Components

The wiring will be installed to provide sufficient slack in the bundles to prevent contraction or expansion from causing a strain on the wire terminations.

The service record of the COMPACT wire bundles used on the F-4 has been outstanding with long Mean Time Between Maintenance Action (MTBMA) and low Maintenance Man Hours per Flight Hour (MMH/FH). Data from the Naval Maintenance Material Management System (3M), from the USAF Maintenance Management System (66-1), and from Air Transport Association System (ARA-100), comparing the F-4 COMPACT wiring versus open laced wiring substantiates this record. Table IX summarizes the above data.

TABLE IX
COMPACT WIRE SERVICE RECORD

Type of Aircraft	Type of Wiring	MTBMA ①	MMH/FH ②	Flight Hours ③
F/RF-4B/J	MCAIR Compact	813.0	0.00830	208,952
DC-8	Open, Laced Wiring	342.8	0.00643	155,995
737	Open, Laced Wiring	314.9	0.00520	26,448
720	Open, Laced Wiring	278.1	0.00769	49,772
727	Open, Laced Wiring	258.9	0.00520	163,895
C-123	Open, Laced Wiring	97.2	0.02396	45,862
C-141	Open, Laced Wiring	67.4	0.04367	259,747
F-106A/B	Open, Laced Wiring	21.1	0.19937	56,425
F-105B/D	Open, Laced Wiring	13.0	0.15903	107,336

① Mean time between maintenance actions (Hours)

② Maintenance manhours per flight hour

③ Flight hours on which MTBMA and MMH/FH are based

c. Wiring Dispersion

This study defines the dispersion criteria for the SFCS test aircraft electrical wiring, identifies the areas in the test aircraft where special attention is required by existing structural and routing path limitations, and defines the physical and electrical protective measures used to provide safe and reliable operation of the SFCS.

(1) Dispersion

Dispersion of the electrical distribution system for the SFCS test aircraft is expected to be accomplished by isolating the four channels from each other, electrically and physically, throughout the aircraft.

It is intended that failure of any single component in the aircraft electrical distribution system will not result in the loss of more than one channel of the SFCS. Physical isolation of SFCS wiring, channel-to-channel, and from the rest of the aircraft's wiring, is to be accomplished by the use of COMPACT wiring techniques where each channel will be isolated from each other channel and from all present aircraft wiring. In general, the SFCS electrical installation is to consist of four separate routing paths throughout the aircraft. However, ideal dispersion of wiring could be incorporated into the entire aircraft only if the concept had been in the aircraft's initial design.

Since the F-4 airframe was not initially designed to include quadruplex wire routing paths, design criteria have been established that will provide maximum dispersion of the SFCS wiring consistent with the physical limitation of the SFCS test aircraft.

(2) Installation and Routing

The general installation and routing paths for the SFCS wiring are shown in Figure 19. It is anticipated that all wiring between LRUs will be end-to-end hard wired with no breaks or connectors. Twisting, shielding, and separation of wiring will minimize electromagnetic interference within the SFCS and with existing aircraft systems. Strict separation of SFCS wiring from exterior lighting wiring, pitot heater wiring, and antenna feed lines will minimize possible damage to the SFCS from lightning strike.

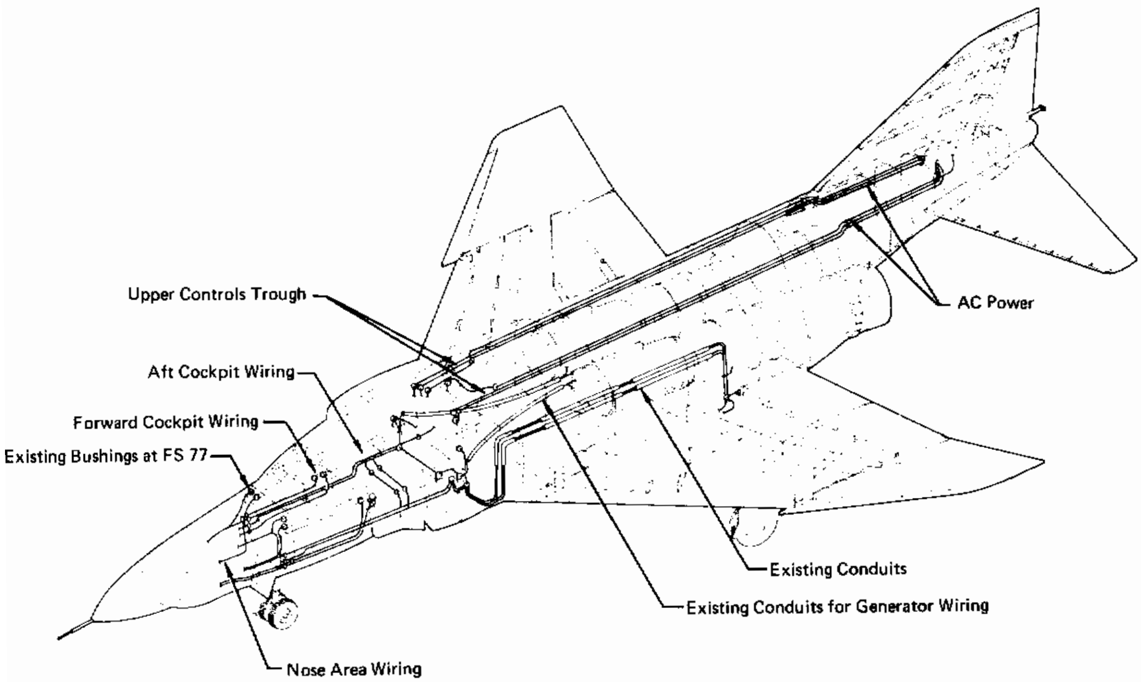


FIGURE 19
SFCS ELECTRICAL WIRE ROUTING

9. HYDRAULIC POWER SUPPLY

a. Three rather extensive hydraulic studies have been conducted. These studies are described in Supplement 3 and summarized below.

b. Hydraulic power requirements for Phase II of the SFCS program have been established. These requirements indicate that four hydraulic systems are required to provide the necessary redundancy for the SFCS. The four hydraulic power systems will be obtained from the three existing airplane systems, PC-1, PC-2, and Utility hydraulic systems, plus the addition of a fourth system which will be powered by an electric motor driven pump. The fourth hydraulic system, which is powered by the production F-4 APU, operates at 1600 psi, while the other three hydraulic systems operate at 3000 psi.

Each hydraulic system supplies power to one of the four elements of each secondary actuator, thus assuring that a single hydraulic system failure will cause the loss of only one channel of each of the quadruplex secondary actuators. Four secondary actuators are used in Phases IIA and IIB to control the surface actuator positions for the left hand and right hand lateral control system, the directional control system, and the pitch control system.

An electrohydraulic, force summing secondary actuator concept has been selected for all phases of the SFCS program for the lateral and directional axes and for Phases IIA and IIB for the longitudinal axis. In Phase IIC an integral velocity summing, electromechanical secondary actuator is provided as part of the SSAP.

c. As a result of the hydraulic power studies reported in Supplement 3, the following conclusions may be drawn:

(1) The steady state hydraulic power drain required by the secondary actuators has negligible effect on the PC-1, PC-2, and Utility hydraulic systems, due to the relatively low power requirements in relation to the potential of these systems.

(2) The fourth hydraulic system as planned is capable of supplying one channel of the secondary actuators and in addition serving as a back-up power source for the rudder surface actuator.

(3) A duplex package powered by two integral hydraulic systems and backed up by two aircraft systems provides the best approach for the SSAP and will be used in Phase IIC of the SFCS program.

(4) MIL-H-83282 (MLO 68-5), a high temperature, less flammable, synthetic hydrocarbon hydraulic fluid is recommended for the SSAP, and its back-up hydraulic systems, PC-1 and PC-2.

d. All other components in the SFCS hydraulic systems are either standard F-4 components or units with long term prior service usage on other jet aircraft. Some of these units have been modified to meet SFCS requirements.

10. ACTUATOR DYNAMIC ANALYSES

a. General

Dynamic analyses were conducted on SFCS actuators to evaluate stability, frequency response, failure transients, and nuisance disconnect characteristics. The capability of the actuators to comply with design requirements was investigated along with studies of actuator monitoring and electrohydraulic versus electromechanical actuator concepts. The results of the analyses reflect the current information on the SFCS actuators and are based on information from References 3, 4, and 5. The parameters utilized represent the best compromise which could be effected to satisfy dynamic requirements and are basically theoretical in nature. Experimental data were used in some instances to define operating characteristics. The results of failure transient and nuisance disconnect analyses as discussed in AFFDL-TR-71-20 Supplement 3 are incomplete due to the lack of some necessary information at the time the analyses were conducted. The analyses results discussed herein were obtained by means of simulation.

The actuator dynamic analyses are presented in Supplement 3 and are summarized below.

b. Secondary Actuator

The secondary actuator is a quadruplex, force summing, electrohydraulic mechanism. It is a self-contained unit consisting of four independent elements coupled to a common output. A schematic of one element is shown in Figure 20.

(1) Stability

When approximated as a second order system, the secondary actuator has a damping ratio near unity, which indicates a very stable system. This is consistent with past experience which has shown that small electrohydraulic servoactuators with relatively insignificant dynamic loading usually have a large stability margin. Should the need arise, the open loop gain can be increased appreciably without causing stability problems.

(2) Frequency Response

The frequency response characteristics represent the overall performance capability of the secondary actuator with nominal parameters. Small signal response usually reveals the influence of nonlinearities such as freeplay, backlash, deadspace, and friction while large signal response identifies the influence of velocity and acceleration saturation. Small signal response, with either four or three elements operating, meets requirements with the exception of phase lag at high frequencies. Dynamic seal friction does add phase lag to small signal response characteristics and this effect was included in the evaluation. The large signal frequency response also shows additional phase lag

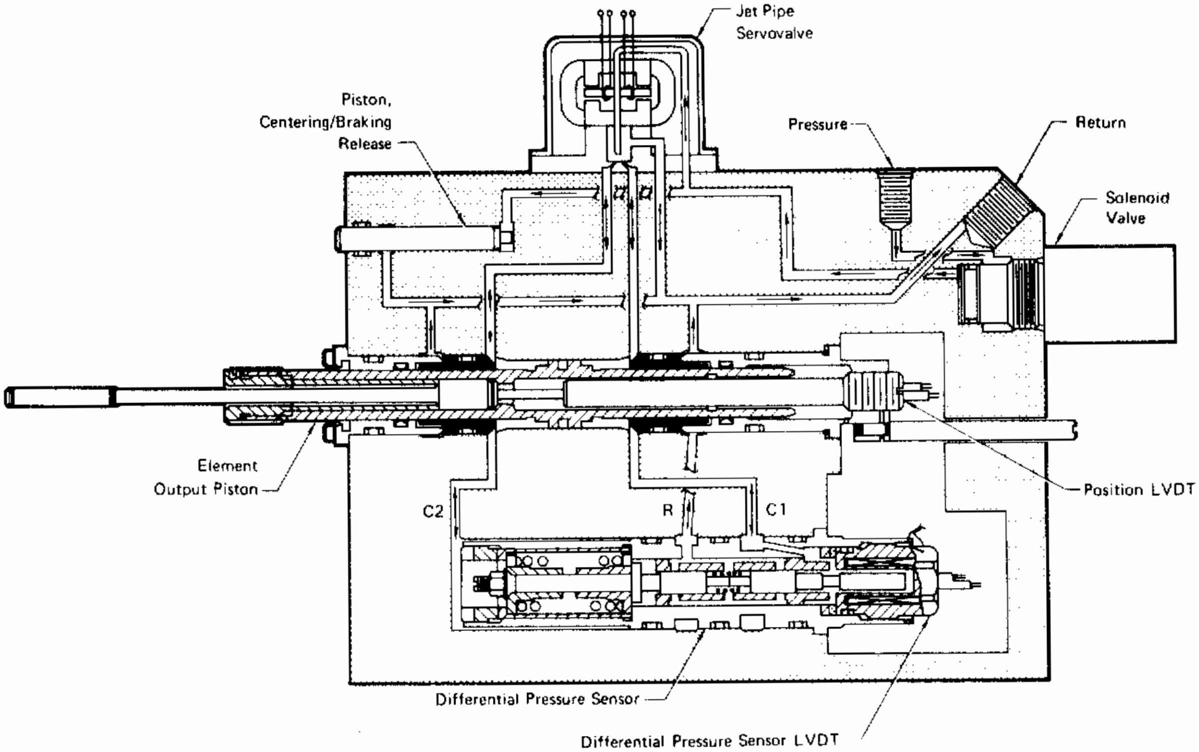


FIGURE 20
HYDRAULIC SCHEMATIC, SINGLE ACTUATOR ELEMENT

at higher frequencies due to velocity saturation in the electro-hydraulic servovalve.

(3) Failure Transients

The analysis of failure transients is incomplete in that the influence of control linkage and surface actuator dynamic characteristics has not been considered due to a lack of information at the time of the analysis. A preliminary linear analysis, which does not include these effects, indicates that the output displacement on first, second and third failures will be acceptable.

(4) Output Velocity and Centering Time

By sizing the single stage jet pipe servovalves properly, the recovery flow with a worst case differential pressure of 1000 psi across the servovalve is sufficient to achieve the required slew rate. With the supply pressure off, the pressure drop through a receiver control orifice at the required slew rate is approximately 30 psi. Thus the loss of one element does not have a significant effect on slew rate.

Centering for lateral and directional secondary actuators is accomplished by a pumped back spring which provides a centering force when all supply pressures are cut off. The spring force needed to meet centering time requirements need only exceed the resisting forces of friction and the receiver control orifice pressure drop by a few pounds. No bypass valves are required since bypassing is inherent in the single stage jet pipe concept.

(5) Compatibility with 1600 psi Fourth Hydraulic System

Due to the combination of line losses and hydraulic pump soft cutoff characteristics, the pressure available at the pitch and yaw actuators in the fourth hydraulic (yellow) channel will be approximately 1400 psi while the pressure at the lateral actuator will be approximately 1350 psi. Since pressure recovery is approximately 75-80 percent, the differential pressure that can be developed across the piston may be as low as 1020 psi. By setting the force limit valve at 1000 psi and the tripout threshold at 930 psi, compatibility with the 1600 psi system is established.

(6) In-Flight Monitor (IFM)

The IFM circuit will utilize a cross element comparison concept in which all differential pressure signals are demodulated and compared in six cross element comparators in the SFCES. The output of the comparators are connected to collection logic for failure detection and shutdown of a failed secondary actuator element. A center tapped LVDT output permits the use of a carrier sensor monitoring scheme which detects shorts, opens and loss of power in the differential pressure sensor.

(7) Nuisance Disconnect

A preliminary statistical analysis assuming a normal distribution of element tolerances indicated that three sigma differential pressures were approximately half of the tripout threshold of 930 psi. This margin appears to be sufficient to assure operation free of nuisance disconnects despite the lact of analytical-experimental correlation for this concept. However, not all of the tolerances in the associated electronics were available when the analysis was conducted. Therefore, the analysis must be updated as the definition of equipment tolerances progresses.

c. Survivable Stabilator Actuator Package (SSAP)

The SSAP is an integrated actuator package utilizing both power-by-wire and fly-by-wire concepts, and is designed to replace the secondary actuator and F-4 stabilator actuator combination for Phase IIC of the SFCS program. The SSAP consists of a quadruplex velocity summing electromechanical secondary actuator and a surface actuator with dual tandem pistons which are powered by two integral motor pump units; it is described more completely in Section IV. The SSAP characteristics are discussed below.

(1) Stability

A stability analysis was conducted for the secondary actuator, surface actuator, and the combined secondary and surface actuators with electrical feedback from the surface actuator. Based on the current information on servomotors, and the associated electronic driver, the electromechanical secondary actuator with tachometer feedback is stable. The surface actuator, which utilizes mechanical feedback, was also found to be stable on the basis of information presently available on the high temperature characteristics of MIL-H-83282 fluid. SSAP stability, i.e., the stability of the loop formed by closing surface actuator electrical feedback around both secondary and surface actuators, can be provided by proper selection of the electrical feedback gain.

(2) Response

Frequency response characteristics for the SSAP were determined for both small and large signals with the effects of soft cutoff pump characteristics, nonlinear valve gain, flow limiting, force limiting, control linkage freeplay and other nonlinearities considered. Small signal response was found to have phase lag in excess of required limits for frequencies above 0.3 Hz due primarily to the effect of control linkage freeplay. Large signal response was found to have phase lag in excess of required limits for frequencies above 5 Hz due to force limiting in the secondary actuator.

(3) Failure Transients

As was the case for the electrohydraulic secondary actuator failure transient analysis, the influence of control linkage and surface actuator dynamic characteristics has not been considered due to a lack of information at the time of the analysis. A preliminary analysis indicates that the output displacement on first, second and third failures will be acceptable. However, a difference in the maximum rpm between servomotors can add to the output displacement on third failures. The effect of the difference in maximum rpm can be reduced by utilizing a short monitor delay time and through selectivity of servomotors to minimize rpm differences.

(4) Nuisance Disconnect

An abbreviated statistical analysis indicates that the probability of nuisance disconnect for the electromechanical secondary actuator is approximately the same as for the electrohydraulic secondary actuator. At the time of the analysis, however, the definition of tolerances for the associated electronics was incomplete. Analysis results will be reviewed as additional information becomes available.

(5) Output Velocity

Both secondary and surface actuators meet velocity requirements under no load conditions. Since the secondary actuator is a velocity summing device, the output velocity decreases by 25 percent for each element lost. However, the no-load velocity with four elements operating is great enough so that the actuator meets the velocity requirements with two elements operating.

An additional output velocity requirement exists when the surface actuator is operating against an output load approximately half of maximum. This requirement assures an adequate recovery rate when the aircraft is in low static stability flight conditions. Calculations indicate that pump pressure-flow characteristics are adequate to meet this requirement.

(6) IFM

In-flight monitoring circuitry for the SSAP secondary actuator is functionally equivalent to that for the electrohydraulic secondary actuator. The error signals to six comparators are obtained from four tachometer windings rather than differential pressure sensors. A special coil is also used to couple excitation voltage to the output at zero rpm to allow the use of a carrier sensor monitoring scheme. A differential speed of 7500 rpm has been tentatively selected as the tripout level.

(7) Voltage Variations

Battery voltage variations from 28 to 20 volts, when charging or not charging, respectively, will not affect the secondary actuator output velocity. The voltage to the brake solenoids will be regulated to 19 volts, however, in order to prevent overheating. By limiting the maximum pulse width to the fixed and control phases of the servomotors as a function of supply voltage, the maximum power and torque are reasonably independent of supply voltage. This assumes that tolerances in the pulse width limiting circuitry are negligible.

Despite the pulse width limiting, transient velocity differences between motors will exist because the effective time constant of the servomotors is a function of the supply voltage. Analyses conducted to date do not indicate that these velocity differences will cause a disengagement problem. Additional velocity differences occur under load since servomotor speed-torque characteristics are also a function of supply voltage.

d. Conclusions

Based upon information available for use in this analysis, the secondary actuator and SSAP will provide stability and control characteristics compatible with the intended SFCS control laws. These conclusions will be checked when hardware characteristics have been explicitly defined.

11. AEROELASTICITY AND VIBRATION

a. General

The installation of a new flight control system and stabilator actuator package introduced three aspects which required consideration - the dynamic aeroelastic stability characteristics of the affected control surfaces, airframe aeroelastic transfer function characteristics, and design and test requirements for vibration, shock, and acoustic noise. Each of these aspects is discussed below.

b. Flutter

(1) Vertical Fin

Based on a review of the vertical fin and rudder flutter analysis as reported in Reference 1, it is concluded that adequate aeroelastic stability margins exist throughout the F-4 flight envelope regardless of the rudder rotational restraint provided by the secondary actuator in the mechanical reversion mode. This is due to the fact that the rudder is statically and dynamically mass balanced to preclude all flutter coupling with other aircraft modes for any rudder restraint stiffness. Retention of the standard production rudder hydraulic dampers precludes transonic buzz.

(2) Horizontal Stabilator

The installation of the SSAP changes the stabilator elastic restraint and the effective stabilator pitch inertia provided by the actuator mass and can, therefore, significantly affect stabilator aeroelastic stability characteristics. The vibration and aeroelastic stability trend studies presented in Appendix IV considered both of these parameters and the results of these studies will be utilized in conjunction with ground vibration test results to establish flight clearances based on aeroelastic stability considerations for the F-4 with slotted leading edge stabilator.

c. Flexible Aircraft Stability Derivatives

(1) The Structural Feedback Problem

Motion of the aircraft is sensed by linear accelerometers and rate gyros located in the fuselage as part of the SFCS. Structural feedback coupling would exist if these devices could detect motions due to structural modes below the control system cutoff frequencies and cause the generation of undesired feedback signals, thus introducing the possibility of a control system instability. It is desirable, therefore, to minimize these signals without severely compromising the control system response characteristics. This requires the consideration of aeroelastic transfer functions in the control system analysis and design.

The block diagram of Figure 21 schematically illustrates the structural feedback elements in relation to the aircraft longitudinal flight control system. The overall aircraft transfer function is given by the Laplace transform $C_V(S)/R_V(S)$, where $R_V(S)$ and $C_V(S)$ represent vehicle control commands and rigid aircraft response, respectively. The dashed area isolates the elements included in the aeroelastic transfer function denoted by the Laplace transform $C_a(S)/R_a(S)$, where $R_a(S)$ and $C_a(S)$ are rigid control surface rotation and total airframe response at a specified location, respectively. It should be noted that the aeroelastic transfer function is open loop and independent of actuator frequency response characteristics.

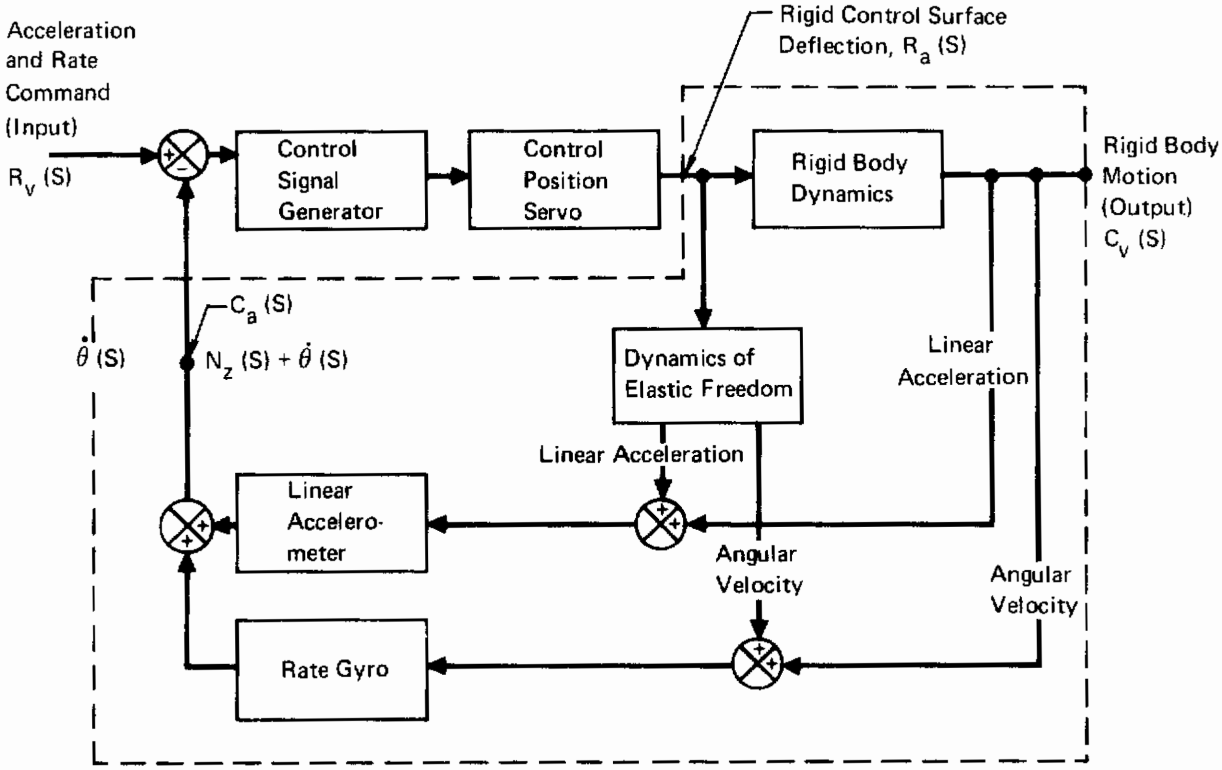


FIGURE 21
BLOCK DIAGRAM OF AIRCRAFT CONTROL SYSTEM WITH AEROELASTIC FEEDBACK

(2) Longitudinal Stability Derivatives

Appendix V presents the derivation of the stability derivatives used in the longitudinal equations of motion. Aeroelastic transfer functions are presented only where appropriate for expository purposes. A discussion of the results and the applicability of the selected approach is presented in Appendix V.

(3) Lateral-Directional Stability Derivatives

Appendix VI presents the derivation of the stability derivatives used in the lateral-directional equations of motion. Particular attention is devoted to defining aileron, spoiler, and rudder control inputs. A discussion of the results and applicability of the selected approach is presented in the Appendix VI.

d. Dynamic Design and Test Requirements

Vibration, shock, and acoustic design and test requirements per Reference 7 have been incorporated in the SFCS Phase II procurement specifications. The objective of these requirements is to provide satisfactory performance and service life for each component.

e. Conclusions

Based on the results of the aeroelastic and vibration studies the following conclusions can be drawn:

- o Adequate rudder stability margins appear to exist at all flight conditions.
- o Ground vibration tests will be required to determine flight clearances based on aeroelastic stability of the stabilator.
- o Longitudinal and lateral-directional stability derivatives have been determined for use in the SFCS design.
- o Design and test requirements have been defined and compliance with these requirements will be monitored during component tests.

12. CONTROL PERFORMANCE CRITERIA

a. General

Past programs for development of longitudinal and lateral-directional handling qualities have been directed toward establishing limiting values of traditional performance parameters (frequency, damping, time constants, etc.) which pilots feel are consistent with desired levels of precision and control during maneuvering flight. The work performed to date, which has been used to update applicable military specifications has been directed mainly toward the specification of handling qualities for aircraft which did not include the use of aircraft motion feedbacks in the primary flight control mode. With the introduction of highly augmented flight control systems and fly-by-wire systems such as the SFCS, increased concern over the adequacy of existing specifications and performance criteria has been expressed. As a result, a control performance investigation has been conducted in an attempt to define short period performance criteria requirements for the SFCS. Performance criteria which are expressed in the time domain and functionally combine the high and low speed transient characteristics desired by the pilot were investigated and results of the associated effort are presented in this report. Applicability to future FBW designs was one of the objectives of the study effort and it was determined that if the formulated criteria are not explicitly dependent on traditional airframe parameters, their use could be applicable to advanced designs. Multi-loop systems of this type will cause significant masking of the basic airframe characteristics and further divorce the fighter aircraft transient response characteristics desired by the pilot for specific inputs from conventional control surface usage. Since candidate criteria developed during this investigation are an expression of fighter pilots' desired handling quality requirements, and are not dependent on airframe characteristics or flight control system mechanization, they are applicable to future SFCS designs. The specific goals and objectives of the investigation were to:

(1) Define SFCS Handling Quality Requirements By Investigating:

(a) to what degree C* handling qualities criteria are compatible with the required mission loop closures

(b) how higher order and nonlinear characteristics affect application of C* criteria

(c) if lateral-directional handling and flying qualities can be incorporated into a new criterion

(d) if control laws should be based upon mission modes or tasks rather than the traditional short period handling qualities and control techniques

(e) if interaxis coupling is desirable and if so to what degree.

(2) Establish a basis of minimum performance requirements for pilot oriented closed loop stability and control.

(3) Establish performance requirements for three axis flight path control for gunnery and bombing aiming accuracies by:

(a) analytically defining, formulating and studying the parameters which significantly affect tracking stability and weapon delivery precision.

(b) evaluating compatibility of C* criterion and lateral-directional criteria with mission tasks.

b. Summary

Analytic studies and man-in-the-loop simulations were performed in order to accomplish the outlined objectives. A generic conclusion pertaining to handling qualities criteria which became apparent during these studies was that a boundary on the time history of flight path rate was necessary but not sufficient to provide good handling qualities. In all axes, a boundary on the rate of change of path rate was required to eliminate higher order effects which could be accommodated by a criterion such as C*.

Pitch axis configurations tests helped establish a revised shape for the C* criterion envelope as shown in Figure 22 and the C* rate of change requirement was generated to be more effective against undesirable response characteristics. Modification of the C* criterion as shown in the figure included elimination of the initial time delay characteristics and the reduction of the lower boundary in the interval between 0.5 seconds to 1.0 seconds.

Acceptability of roll time constant variations tested during the fixed base simulation, appeared to be marginal for values greater than one and nearly optimum for values at 0.50 seconds. Simulation and testing enabled establishment of a roll axis time history criterion, based upon roll rate and roll acceleration, as shown in Figure 23 for lateral step force inputs.

Directional response data and pilot comments obtained during simulated rolling maneuvers indicate that sideslip and lateral acceleration are important directional motion cues which can be combined into a common expression, called D* to serve as an equivalent directional criterion to the longitudinal C* criterion. D* and $\dot{D}^*$ envelopes of acceptability, as shown in Figure 24, were established during analysis and simulation, and are recommended for use in evaluating directional response characteristics for lateral step force inputs.

The D^*_T parameter is a combination of sideslip and lateral acceleration at the pilot seat as expressed by the following equation:

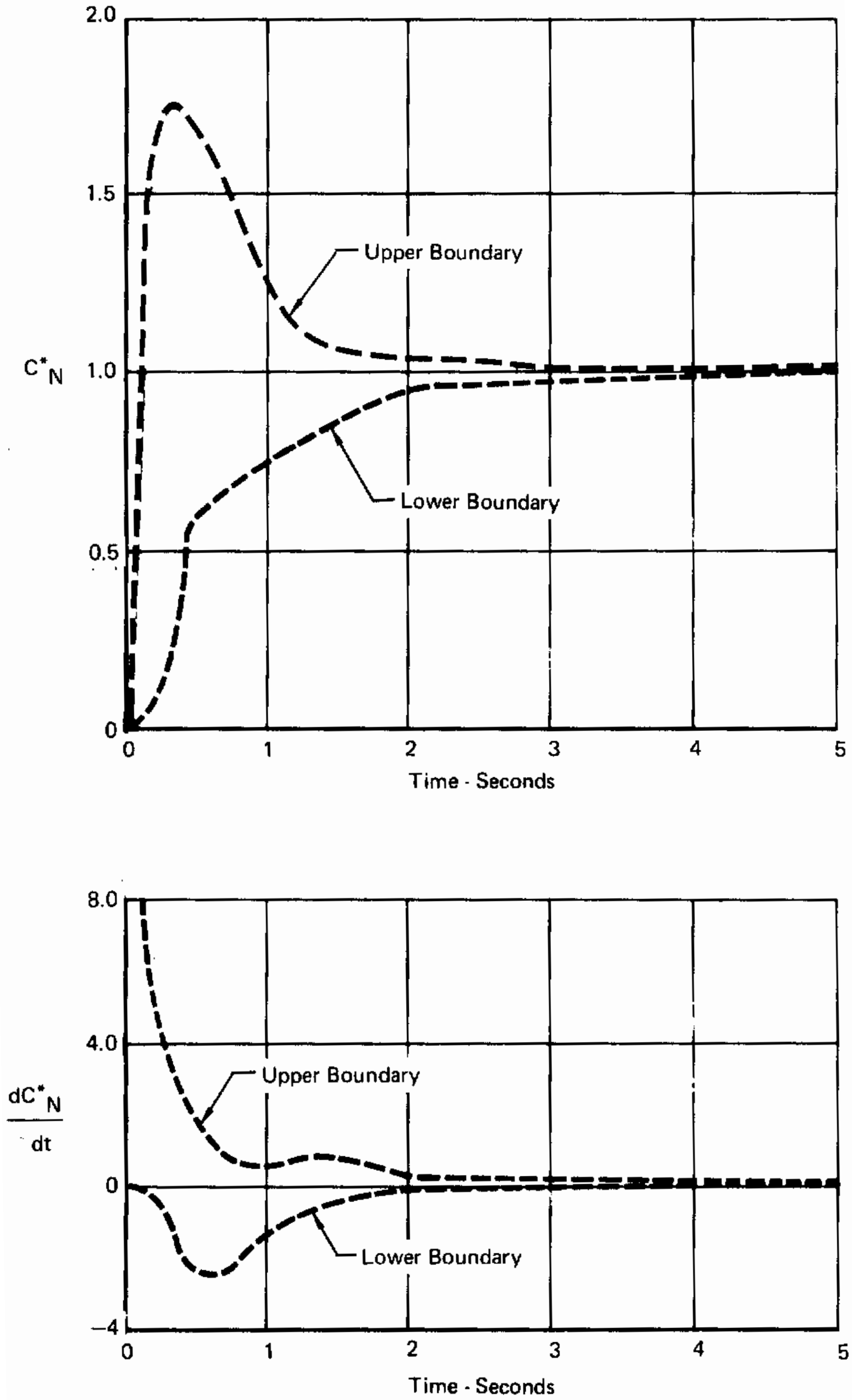
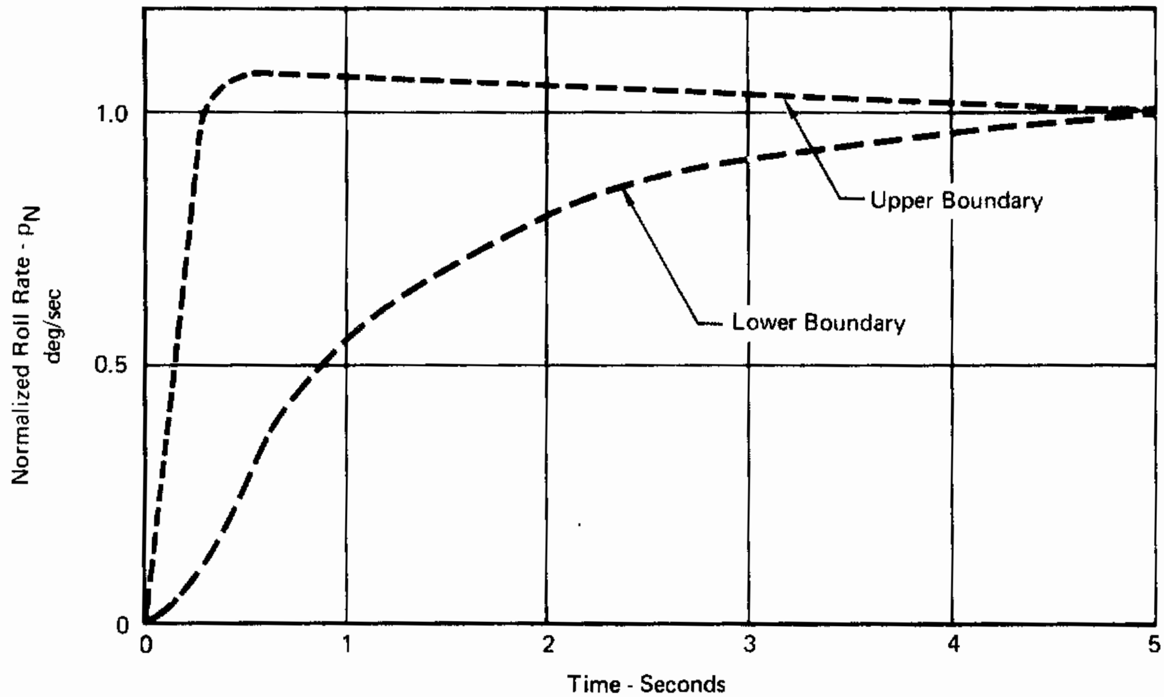


FIGURE 22
SFCS PITCH AXIS TIME HISTORY CRITERION



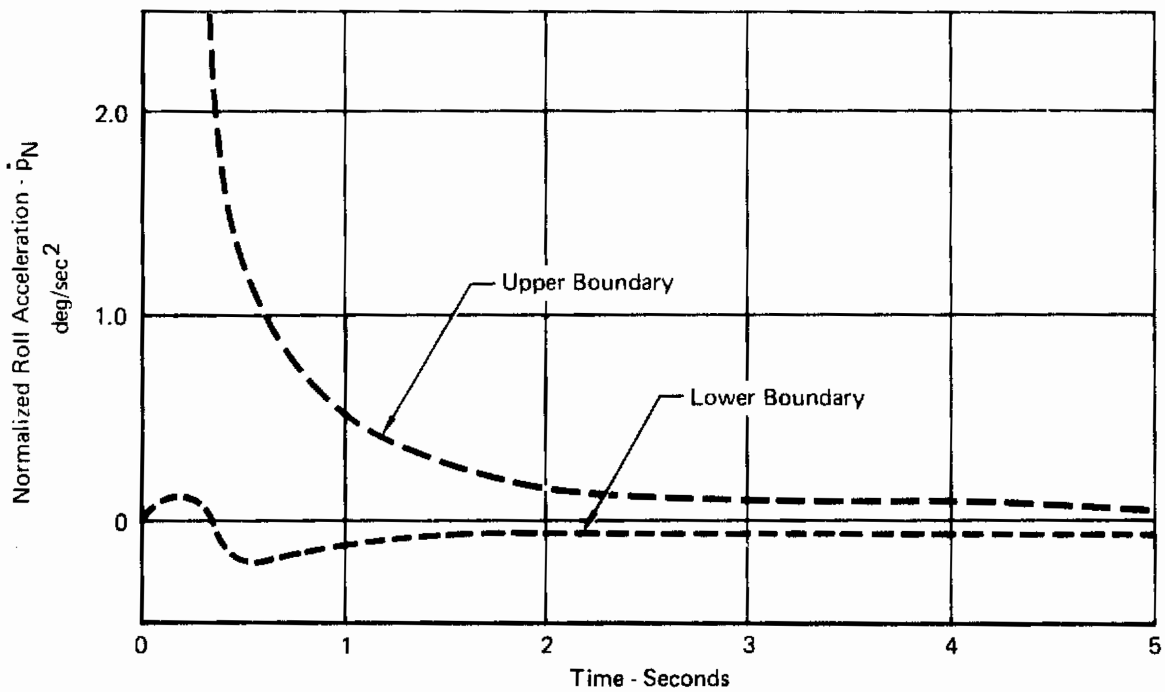


FIGURE 23
SFCS ROLL AXIS TIME HISTORY CRITERION

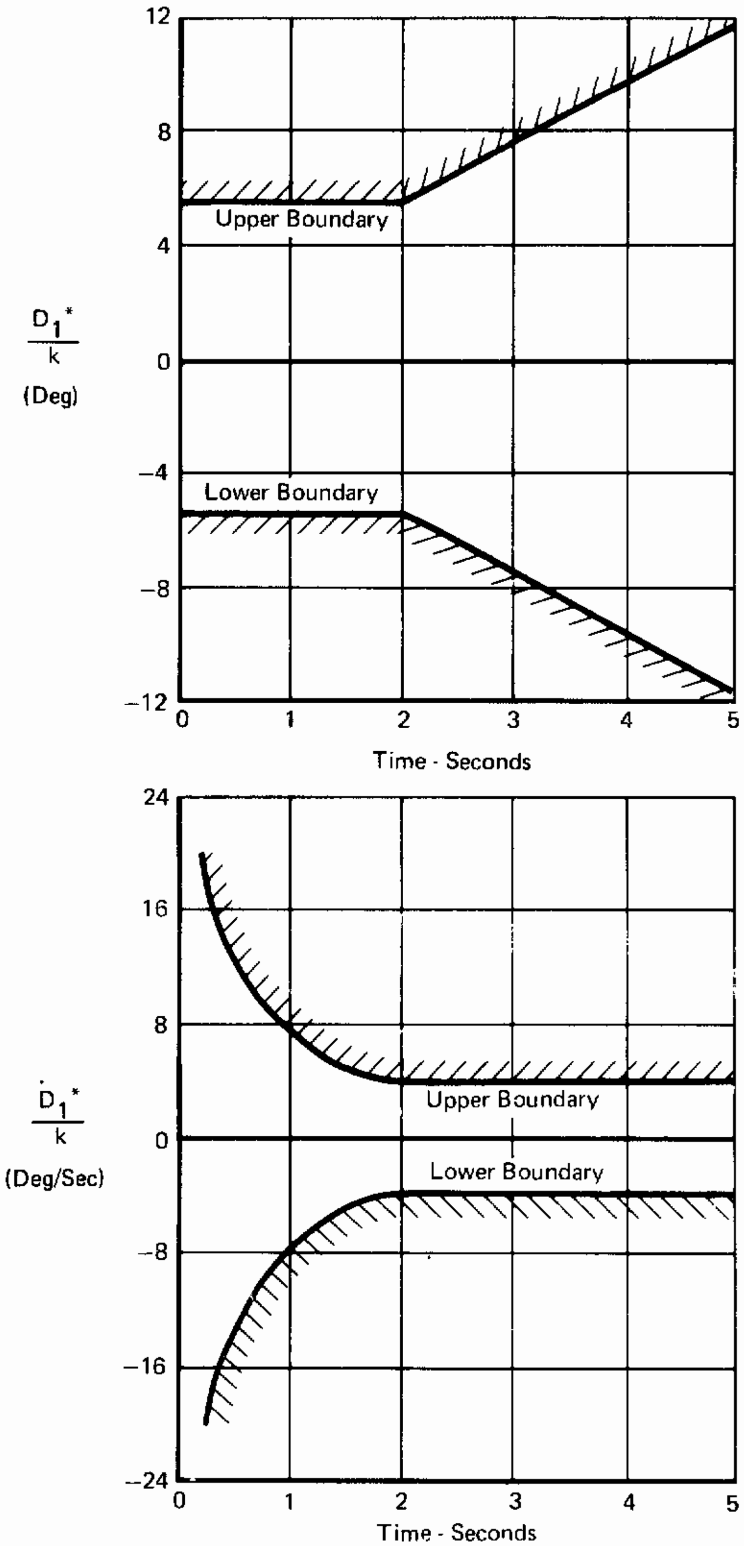


FIGURE 24
SFCS DIRECTIONAL TIME HISTORY CRITERIA

$D_1^* = \beta - .513 \, A_{y_p}$ (deg)

where: β is sideslip angle measured in degrees, and
A_{y_p} is lateral acceleration measured at the pilot seat in feet/second squared.

The development and discussion of the D_1^* criteria are presented in TR-71-20, Supplement 1.

c. Conclusions

The results of the control performance analyses and simulation studies support the following conclusions and recommendations reached during the investigations:

(1) The C* concept is applicable for use in the F-4 aircraft SFCS design. Its applicability is based on favorable results obtained from analysis and simulation studies, and on its strong correlation with advanced flight control system designs used successfully in the industry for closed loop augmentation prior to C* concept inception.

(2) From hybrid man-in-the-loop simulation studies and pilot comments, it is concluded that the C* handling qualities are compatible with the F-4 aircraft mission modes investigated and the mission task oriented basing of control laws, and the attendant mode switching, would not be necessary except possibly for refueling, landing, or other modes not investigated in this study.

(3) Pilots indicated that optimization of aircraft feel system is desirable, but aircraft dynamic response variations normally encountered in more conventional designs are adequately masked by the FBW system.

(4) The adverse effect on C* applicability of time history response abnormalities including high order and nonlinear characteristics can be reduced with modification of lower C* boundary and additional use of C* rate of change envelope of acceptability.

(5) The D* concept, in which lateral acceleration and sideslip angle were combined and used as a lateral-directional counterpart to the C* criterion, was formulated. Preliminary D* and $\dot{D}^*$ criterion boundaries were established for roll command step inputs.

(6) A roll axis time history envelope of acceptability was formulated for lateral step command inputs.

(7) Complete interaxis decoupling, as investigated during the hybrid simulation, did not show significant performance improvement over designs in which maximum ARI effectiveness was realized. Pilot comments indicate preference for zero roll-to-yaw coupling at

low-normal angles of attack. Non-zero yaw-to-roll is desired and used during low roll effectiveness flight conditions for achieving maximum roll rate. Pilots also desire some yaw-to-roll coupling for high angle of attack rolling maneuvers.

(8) Tracking stability and weapon delivery equations were derived, and flight path control performance requirements were investigated. It was found that aircraft gun angle and flight control system dynamics significantly affect pilot-oriented closed loop stability and control. Performance requirements established during the study included the three axes criteria presented above.

d. Recommendations

Flight test verification of the candidate criteria described in this report is necessary in order to establish their validity and encourage their usage in future flight control system design efforts.

e. Additional Information

The results summarized above were generated during an extensive study and man-in-the-loop simulation program. Information on the methods of analysis, assumptions, conditions studied, simulation mechanization, and documentation of the results is presented in Supplement 1.

13. CONTROL LAWS

a. General

The extent and results of the SFCS control law development studies, analyses, and simulations are presented in Supplement 2. A summary of the results and conclusions is presented in this subsection.

b. SFCS Modes and Functions

(1) The SFCS has four operating modes which are separately selectable for each axis of control. These modes are defined as follows:

(a) Normal Mode - This mode is selected from the Master Control and Display Panel (MCDP), shown in Figure 25, by depressing the desired control mode selector switch until a "NORMAL" indication is obtained on the switch indicator. In this mode of operation, aircraft motion is the variable commanded by pilot inputs; i.e., aircraft rates and accelerations are used as feedback signals. This mode is "Fly-By-Wire" (FBW) by definition. Two forms of control are provided in the Normal mode, as follows:

(1) Neutral Speed Stability (NSS) - The NSS is selected by placing the Flight Mode switch on the forward trim panel, shown in Figure 26, to the "NORMAL" position. After the landing gear are retracted this will be

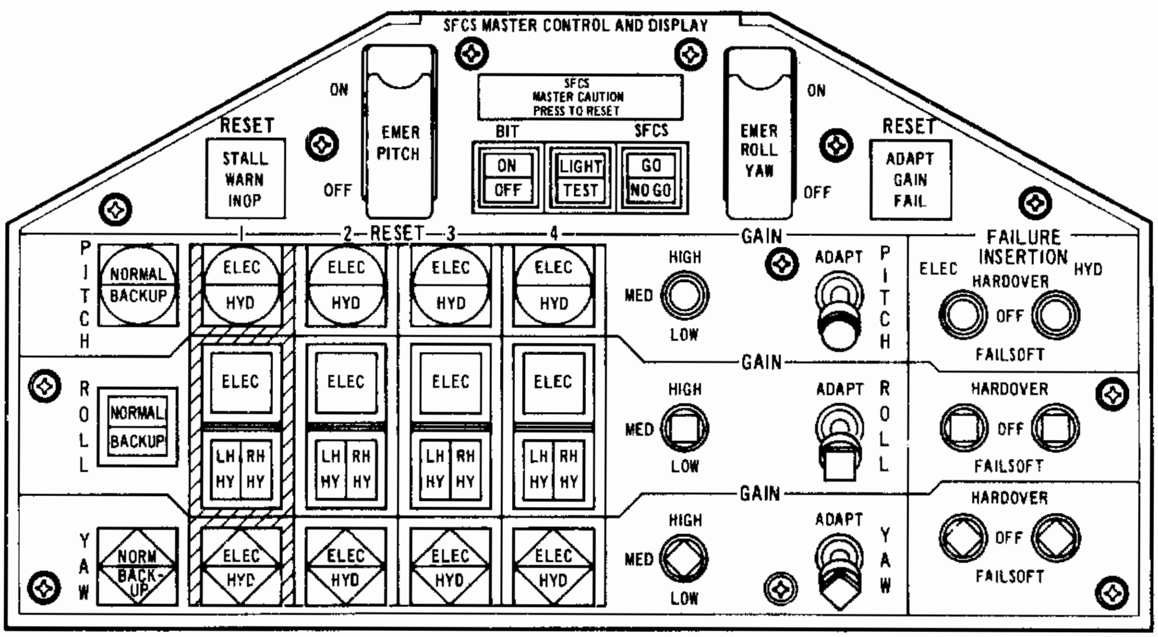


FIGURE 25
MASTER CONTROL AND DISPLAY PANEL

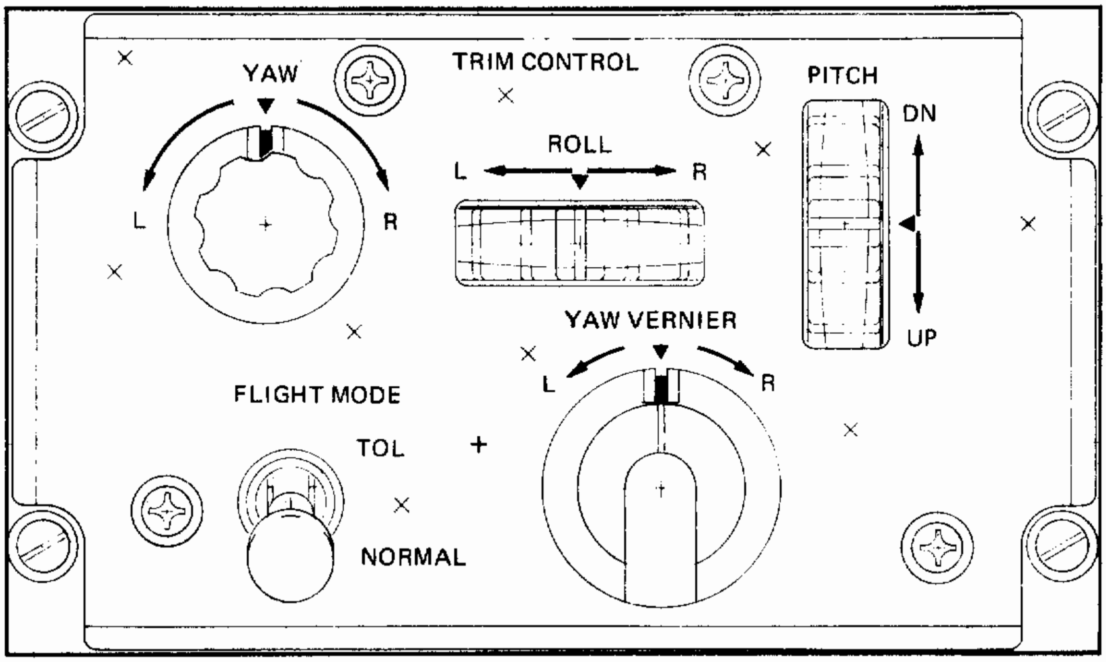


FIGURE 26
TRIM CONTROL PANEL (FORE) LAYOUT

operable. With the NSS selected, no steady state pilot applied stick force or trim input is required in order to compensate for the change in stabilator position required to trim the airplane due to changes in airspeed and/or altitude. An interconnect with the nose gear position switches eliminates the NSS function when landing gear are extended.

(2) Take Off and Land (TOL) - The TOL is selected by placing the Flight Mode switch in the "TOL" position. Selection of the TOL will result in a requirement to manually trim the aircraft as airspeed and/or altitude are varied. Airspeed changes at subsonic flight conditions will result in a positive speed stability characteristic; i.e., push forces will be required as airspeed is increased and pull forces will be required as airspeed is reduced.

(b) Electrical Back-Up (EBU) Mode - The EBU mode can be selected for use in any axis from the MCDP by depressing the desired control mode selector switch until a "BACKUP" indication is obtained on the switch indicator. The EBU mode can be engaged in all axes simultaneously by depressing either the side stick controller trigger switch or the emergency disconnect switch on the center stick controller.

Aircraft motion feedbacks, forward loop compensation, structural filters, adaptive gain changing, stabilator position feedback, and stall warning circuitry are disabled in the EBU mode. However, the EBU is still a quadruplex system and the signal selection devices continue to function.

(c) Mechanical Back-Up (MBU) Mode - The MBU mode is available for use in Phase IIA only. The MBU mode is not available in the roll axis. This mode can be selected by actuating the Pitch and/or Yaw MIM control switches, shown in Figure 27, to the "MECH BACKUP" position or by depressing the emergency disconnect switch on the center stick controller. The pilot commands surface position through mechanical linkages in the MBU mode. Rate and lateral acceleration feedbacks are applied through the stability augmentation system and production series servos to augment airplane damping in the pitch and yaw axes. The SFCS Lateral Control Switch, shown in Figure 27, is provided to permit removal of the SFCS lateral control function. This switch is used to remove power from the engage solenoids on all elements of the lateral control secondary actuators thereby causing them to be automatically centered.

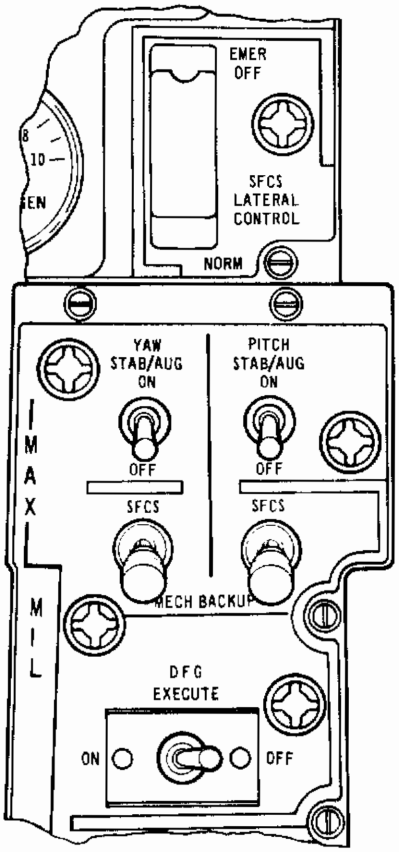


FIGURE 27
PHASE IIA PANEL CONFIGURATION - LEFT CONSOLE

(d) Demand On Mode - The demand on mode can be selected only from the MCDP. Covered switches labeled "EMER PITCH" and "EMER ROLL YAW" are used for this purpose. Selection of the demand on mode disables the comparators used to disengage secondary actuator elements and engages all secondary actuator elements. In all other respects the demand on mode is identical to the EBU mode.

(2) A number of capabilities are included in the SFCS which are classified as "functions". These "functions" are separately selectable and can be engaged as desired. The functions of the SFCS are as follows:

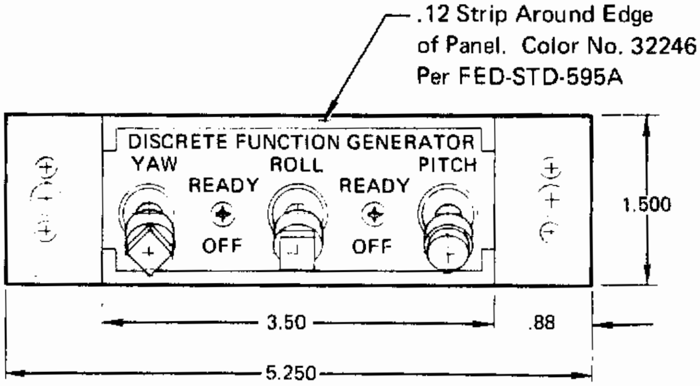
(a) Adaptive Gain Function - The adaptive gain function can be selected in the pitch and yaw axes from the MCDP. The fixed gain control switches must be placed in the "LOW" position and the adaptive gain switches placed in the "ADAPT" position to engage this function. System gains are automatically varied as a function of aircraft response characteristics to obtain the best achievable handling qualities. The roll to yaw crossfeed gains are also varied to obtain turn coordination with this function selected.

(b) Selectable Fixed Gain Function - Any one of three fixed gain values can be selected using the fixed gain switches on the MCDP. Gains in the pitch and yaw axis are independently selectable. These switches can be used to manually select the proper gains for a given flight condition or to investigate the effect of using off-nominal gain values.

(c) Stall Warning Function - The stall warning function is selected by depressing the "STALL WARN INOP" switch on the MCDP after takeoff. This function provides stick force cues to the pilot when the region of impending accelerated stall is entered. In addition, roll rate feedback is eliminated in the high angle of attack region to prevent spin inducing aileron deflections resulting from wing rock.

(d) Failure Insertion Function - Switches are included on the MCDP to enable the insertion of null or hardover failures. Both electronic and actuator failures can be simulated. The failure insertion switches affect only the yellow channel of the SFCS.

(e) Discrete Function Generator - This function provides for the application of step input commands to each axis of control to obtain responses for comparison with simulation data. The axis to which the step input is to be applied is selected on the Discrete Function Generator panel shown in Figure 28, located on the pedestal panel. Application of the step command is accomplished using the Discrete Function Generator execute switch located on the left console.



Note:
1. Front Panel Layout for
MCAIR Part No. 53-044001-33

FIGURE 28
DISCRETE FUNCTION GENERATOR PANEL

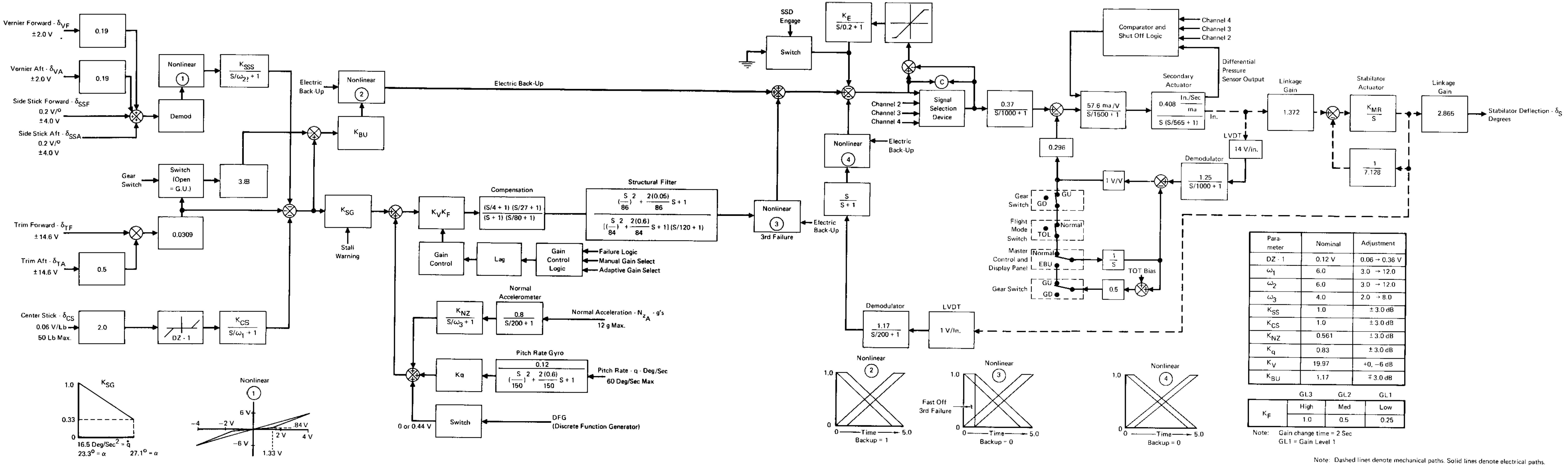
c. Longitudinal Control Law Development

The functional block diagram presented in Figure 29 depicts the SFCS longitudinal control system designed for the Phase IIA and B portion of the flight test program. A summary of the control law features as presently conceived is presented below.

(1) Normal Mode

The longitudinal control system is a fly-by-wire system in which aircraft motion is the controlled parameter. This is accomplished by utilizing normal acceleration and pitch rate feedbacks which are subtracted from the center stick force transducer commands and the side stick position transducer commands to obtain an error signal. The error signal is used as a position command for the stabilator actuator.

The Normal mode of operation provides neutral speed stability (NSS) for all nonterminal flight phases. Since the speed stability is neutral, no steady state pilot applied stick force or trim input is required in order to compensate for the change in stabilator position required to trim the airplane due to changes in airspeed and/or altitude. An interconnect with the nose gear



Parameter	Nominal	Adjustment
DZ · 1	0.12 V	0.06 → 0.36 V
ω_1	6.0	3.0 → 12.0
ω_2	6.0	3.0 → 12.0
ω_3	4.0	2.0 → 8.0
K_{SS}	1.0	± 3.0 dB
K_{CS}	1.0	± 3.0 dB
K_{NZ}	0.561	± 3.0 dB
K_q	0.83	± 3.0 dB
K_V	19.97	+0, −6 dB
K_{BU}	1.17	∓ 3.0 dB

	GL3	GL2	GL1
	High	Med	Low
K_F	1.0	0.5	0.25

Note: Gain change time = 2 Sec
GL1 = Gain Level 1

Note: Dashed lines denote mechanical paths. Solid lines denote electrical paths.

FIGURE 29
LONGITUDINAL SFCS FUNCTIONAL BLOCK DIAGRAM

position switches removes the function which provides the neutral speed stability when the landing gear are lowered. The pilot may also override the neutral speed stability function using the mode select switch on the trim control panel.

The control system interface with BIT is not denoted in Figure 29. The take-off trim is automatically established on the output of the integrator at the completion of BIT.

The neutral speed stability characteristic is obtained by generating an integration in the forward path of the control system. The integration maintains zero steady-state error between force command and the blended pitch rate and normal acceleration feedback. The airplane is kept in trim since any uncommanded pitch rate and acceleration is automatically reduced to zero by the action of the integrator. For nonterminal flight conditions, only occasional trim inputs initiated by the pilot are required to offset any electrical biases which may be present. Since the integration function is removed when the landing gear are extended, nominal trimming action by the pilot will be required during terminal flight phases.

The secondary actuator is used to provide the integration function. This is accomplished by incorporating washout circuits in the secondary actuator ram position and main actuator ram position feedback signals, which transform the position feedbacks into velocity feedbacks for low frequency inputs ($\omega < 1$ rad/sec).

The washout network for the secondary actuator position feedback is mechanized immediately after the demodulator as shown in Figure 29. For the Normal mode with gear up, the transfer function for the feedback circuitry is:

$$\frac{\text{Output}}{\text{Input}} = \frac{1}{1 + 1/S} = \frac{S}{S + 1}$$

The demodulated stabilator actuator position feedback is passed through an identical washout of $S/(S + 1)$. Since the washout circuits reduce both electrical feedbacks to zero for steady-state secondary actuator and stabilator actuator position, any electrical input to the actuator loop is unopposed and is integrated by the secondary actuator. The feedback paths can attain non-zero steady-state values only in the presence of constant secondary actuator and stabilator actuator rates. Thus the steady-state response to a constant input to the actuator loop is a constant stabilator rate. This method of implementing the NSS function was selected since the inherent integration characteristic of the secondary actuator is utilized and the need for an electronic integrator in the forward loop is avoided.

The input to the actuator loop must be zero in order to achieve a steady-state stabilator deflection in response to a steady-state pilot applied input. For center stick input. the steady-state control law derived from Figure 29 is:

$$-(\text{Pilot force})(.06)(2)(K_{c_s}) + N_Z(.8)K_{N_Z} + q(.12)K_q = 0$$

Substituting the values for K_{c_s}, K_{N_Z} and K_q and using the steady-state equation for pitch rate, $q = (1845/V)N_Z$ gives the following expression:

$$(.45 + \frac{184.5}{V})N_Z - .12(\text{Pilot Force}) = 0$$

In order for the above identity to hold, N_Z must be zero in the absence of pilot applied force. When an out-of-trim stabilator condition occurs, uncommanded N_Z is sensed, and the forward loop integration acts to change the stabilator position, thereby re-establishing the above equality.

(2) Electric Back-Up(EBU) Mode

The control system includes an electric back-up mode which consists of a simple forward path connecting the longitudinal force inputs prefilters to the signal selection device (SSD). To minimize transients, selection of electric back-up mode by the pilot causes the direct electric path to be faded in (Nonlinear 2) as the normal SFCS forward path is faded open (Nonlinear 3). The actuator main ram electrical feedback path is also simultaneously faded open (Nonlinear 4). Aircraft motion feedbacks, forward loop compensation, structural filters, adaptive gain changing, stabilator position feedback, and stall warning circuitry are disabled in the EBU mode. However, the EBU is still a quadruplex system and the signal selection devices continue to function.

(3) Adaptive Gain Changing

Pilot selection of adaptive gain changing or fixed gain operation is available. The adaptive gain changer provides 3 gain states in the longitudinal axis which change the forward loop gain, K_F, over a range of 4 to 1 as the aircraft stabilator effectiveness parameter, M_δ, varies due to flight condition and aircraft configuration changes.

The adaptive gain computer provides "hysteresis loops" at the three gain change boundaries, and all gain changes are faded in over a two-second period. After selecting the fixed gain position for a given axis on the Master Control and Display Panel (See Figure 25), the pilot may manually select any of the three gain states. The system is configured to provide stable operation over the entire flight envelope for the low gain value.

By selecting fixed gain operation, the pilot may manually select any of the adaptive gains.

(4) Stall Warning System

The normal F-4 audio tone generator provides warning of impending one g and accelerated stalls. An additional stall warning system is provided with the SFCS to warn of impending accelerated stalls.

A functional block diagram of the stall warning system is presented in Figure 30. The stall warning system provides stick force cues to the pilot when the region of impending accelerated stall is entered. The warning becomes increasingly more pronounced with progression into the stall region. This is accomplished by decreasing the electrical gradient through which the stick force transducer and the Side Stick Controller (SSC) outputs are passed prior to the point at which they are summed with the sensor feedback signals. Thus, the pilot applied maneuvering force for either center stick or SSC must be increased in order to maintain a given maneuver if the maneuver causes entry into the stall region. The warning signal is initiated at 23.3° angle of attack and linearly increases to a maximum at 27.1° angle of attack, at which point the stick force electrical gradient, K_{CS}, (Figure 29) is correspondingly decreased to one-third its nominal value. Cancelled pitch rate is used to generate a stall anticipatory signal. This signal, which is proportional to pitch angular acceleration, activates the stall warning system at 16.5 deg/sec^2 and provides full warning output at 24 deg/sec^2 for zero angle of attack. The angle of attack signals and cancelled pitch rate signals are summed through appropriate gains so that any combination of angle of attack and positive pitch acceleration which satisfies the equation, $\alpha + 1.41\ddot{\theta} \geq 23.3$, will activate the stall warning system. In order that negative pitch accelerations will not cause deactivations of the stall warning when $\alpha \geq 23.3$, only positive pitch accelerations are used. Appropriate filtering is applied to the summed acceleration and angle of attack signals to prevent momentary variations in signal caused by air turbulence from actuating the stall warning system.

(5) Structural Filter

A second order notch and first order lag filter for structural mode attenuation are included in the longitudinal SFCS forward loop. The open loop peak gain amplitudes corresponding to each of the three structural modes were used to obtain the minimum attenuation at each structural mode. These values are shown in Table X. The attenuation is at least 6 dB for all flight conditions investigated. The structural mode attenuation will be greater for fixed low gain operation by 6 to 12 dB.

Switching Logic $\overline{SW} = C_1 + C_2 + C_3 + SMRD\,\dot{\theta}_1 + SMRD\,\dot{\theta}_2$

$V_1 = 1.66V$
$V_2 = 2.42V$
$V_0 = 4.56V$
BIAS = $30^o\alpha$

Note: 0 to 100% Modulation Occurs for Range of 0 to V_0.

Variable	Nominal	Range
K_1	0.5	0.25 - - 1.0
K_2	15000	TBA
K_3	6.0	3.0 - - - 12.0
K_4	TBA	
K_5	TBA	
K_6	TBA	
K_7	0.843	TBA

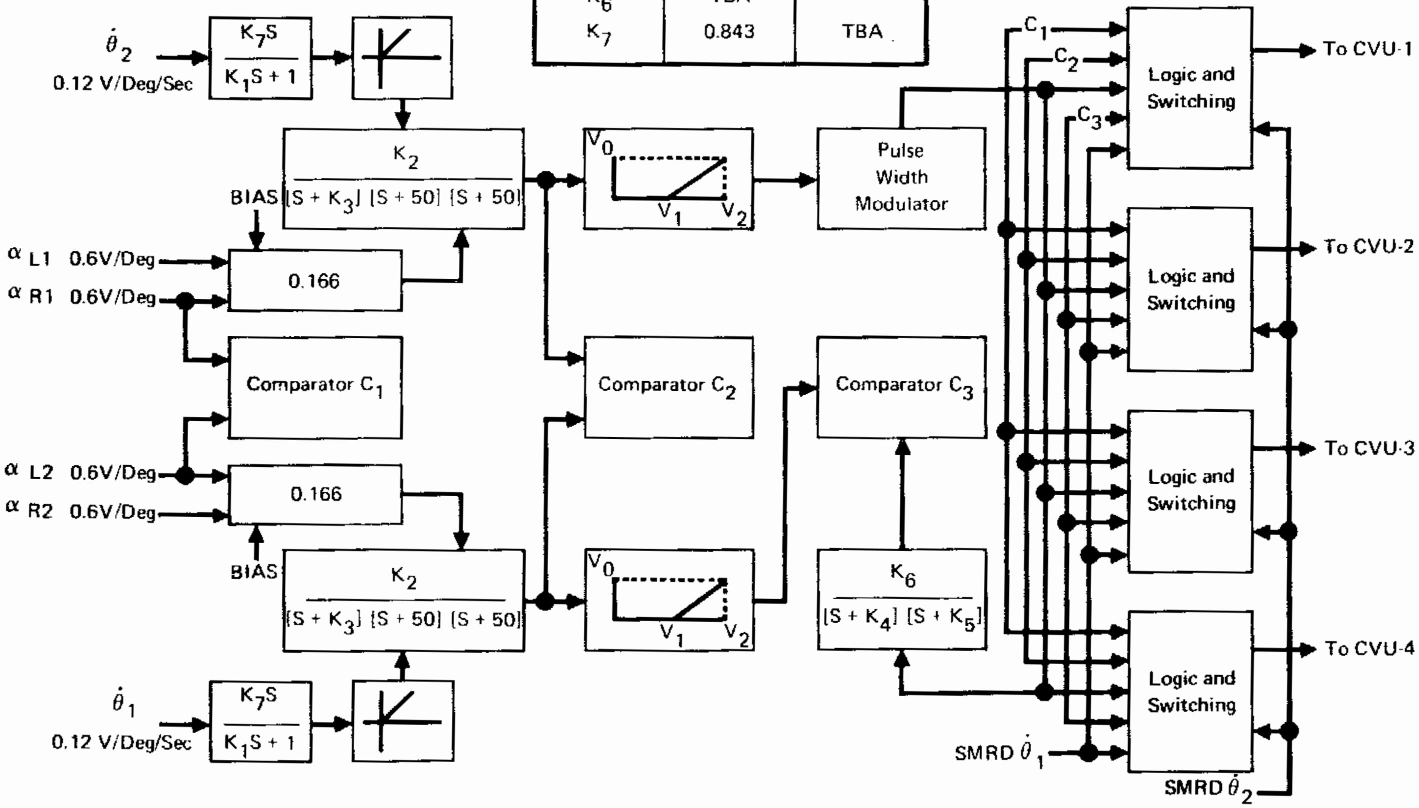


FIGURE 30
STALL WARNING FUNCTIONAL BLOCK DIAGRAM

TABLE X
STRUCTURAL MODE ATTENUATION
(Adaptive Gains)

Mach	Altitude (Ft)	Weight (Lb)	Function (NSS TOL)	K_F	Stabilator Bending Freq (Rad/Sec)	Stabilator Bending Attenuation (dB)	First Vertical Bending Freq (Rad/Sec)	First Vertical Bending Attenuation (dB)	Stabilator Rotation Freq (Rad/Sec)	Stabilator Rotation Attenuation (dB)
0.5	5,000	38,732	NSS	1.0	71	22	88	11	139	−11
0.5	5,000	43,720	NSS	1.0	71	−22	88	−11	139	−10
0.5	25,000	38,732	NSS	1.0	68	−9	87	−10	142	−7
0.5	25,000	43,720	NSS	1.0	68	−9	87	−10	142	−8
0.9	15,000	38,732	NSS	0.5	76	17	90	28	139	−26
0.9	15,000	43,720	NSS	0.5	76	17	90	−28	139	27
0.9	35,000	38,732	NSS	1.0	70	−20	87	−12	139	−10
0.9	35,000	43,720	NSS	1.0	70	20	87	−12	139	10
0.9	45,000	38,732	NSS	1.0	69	10	88	−11	140	−8
0.9	45,000	43,720	NSS	1.0	69	−10	88	−11	140	−8
1.2	5,000	38,732	NSS	0.25	81	16	98	6	160	−15
1.2	5,000	43,720	NSS	0.25	81	14	98	−6	160	16
1.5	15,000	38,732	NSS	0.25	80	16	96	11	160	−18
1.5	15,000	43,720	NSS	0.25	80	16	96	−11	160	−19
1.5	35,000	38,732	NSS	0.5	74	−20	90	22	149	−30
1.5	35,000	43,720	NSS	0.5	75	−21	90	−22	149	29
1.5	45,000	38,732	NSS	0.5	73	−24	88	−18	145	19
1.5	45,000	43,720	NSS	0.5	73	24	88	18	145	−18
1.8	55,000	38,732	NSS	1.0	70	−18	88	−11	144	16
1.8	55,000	43,720	NSS	1.0	70	18	88	−11	144	−16
2.15	36,000	38,732	NSS	0.25	76	−27	90	−20	148	−25
2.15	36,000	43,720	NSS	0.5	76	21	90	20	148	−24

TABLE XI
STABILITY MARGINS (PHASE IIA, B)
(Adaptive Gain)

Mach	Altitude (Ft)	Weight (Lb)	Function (NSS TOL)	K_F	Phase Margin (Deg)	Phase Margin Freq	Gain Margin (dB)	Gain Margin Freq
0.206	SL	32,500	TOL	1.0	47.7	2.3	28.1	36.3
0.214	SL	43,720	TOL	1.0	45.8	2.1	29.5	36.2
0.318	SL	32,500	TOL	1.0	54.2	4.0	20.8	36.4
0.318	SL	43,720	TOL	1.0	49.7	3.3	23.1	36.3
0.5	5,000	38,732	NSS	1.0	57.0	7.1	14.9	36.1
0.5	5,000	43,720	NSS	1.0	55.1	6.5	15.4	36.0
0.5	25,000	38,732	NSS	1.0	45.5	4.0	21.7	35.8
0.5	25,000	43,720	NSS	1.0	41.8	3.6	22.3	35.8
0.84	SL	38,732	NSS	0.25	64.1	6.1	17.4	36.4
0.84	SL	43,720	NSS	0.25	53.5	5.2	18.1	36.3
0.9	15,000	38,732	NSS	0.5	60.0	7.7	14.5	36.3
0.9	15,000	43,720	NSS	0.5	56.4	6.8	15.1	36.2
0.9	35,000	38,732	NSS	1.0	53.9	6.8	15.2	35.9
0.9	35,000	43,720	NSS	1.0	52.0	6.2	15.9	35.9
0.9	45,000	38,732	NSS	1.0	47.7	4.9	19.2	35.8
0.9	45,000	43,720	NSS	1.0	45.0	4.4	19.8	35.8
1.1	SL	38,732	NSS	0.25	75.5	9.7	13.2	37.0
1.1	SL	43,720	NSS	0.25	71.3	8.4	13.9	36.9
1.2	5,000	38,732	NSS	0.25	74.2	10.4	13.3	36.9
1.2	5,000	43,720	NSS	0.25	71.8	9.1	13.9	36.8
1.5	15,000	38,732	NSS	0.25	71.6	9.4	15.5	35.7
1.5	15,000	43,720	NSS	0.25	67.1	8.3	16.2	35.7
1.5	35,000	38,732	NSS	0.5	60.0	9.5	14.3	36.1
1.5	35,000	43,720	NSS	0.5	59.1	8.6	15.0	36.0
1.5	45,000	38,732	NSS	0.5	57.8	7.1	18.1	35.9
1.5	45,000	43,720	NSS	0.5	56.1	6.5	18.7	35.9
1.8	55,000	38,732	NSS	1.0	53.7	8.0	15.2	35.8
1.8	55,000	43,720	NSS	1.0	52.8	7.4	14.9	35.7
2.15	36,000	38,732	NSS	0.25	64.6	8.5	19.5	35.9
2.15	36,000	43,720	NSS	0.5	55.9	9.8	14.2	35.8

(6) Stability and Response Characteristics

Stability margins for the aircraft short period and control system oscillatory modes were determined by the Bode frequency response method. Open loop frequency response plots were obtained for fifteen flight conditions and two aircraft weights. Table XI shows the gain and phase margins exhibited by the system at each flight condition for adaptive gain operation. Adequate stability margins are maintained at all flight conditions.

Time history responses for the aircraft with the SFCS installed are presented in Supplement 2. All flight conditions compare favorably with the evaluation criteria.

d. Lateral-Directional Control Law Development

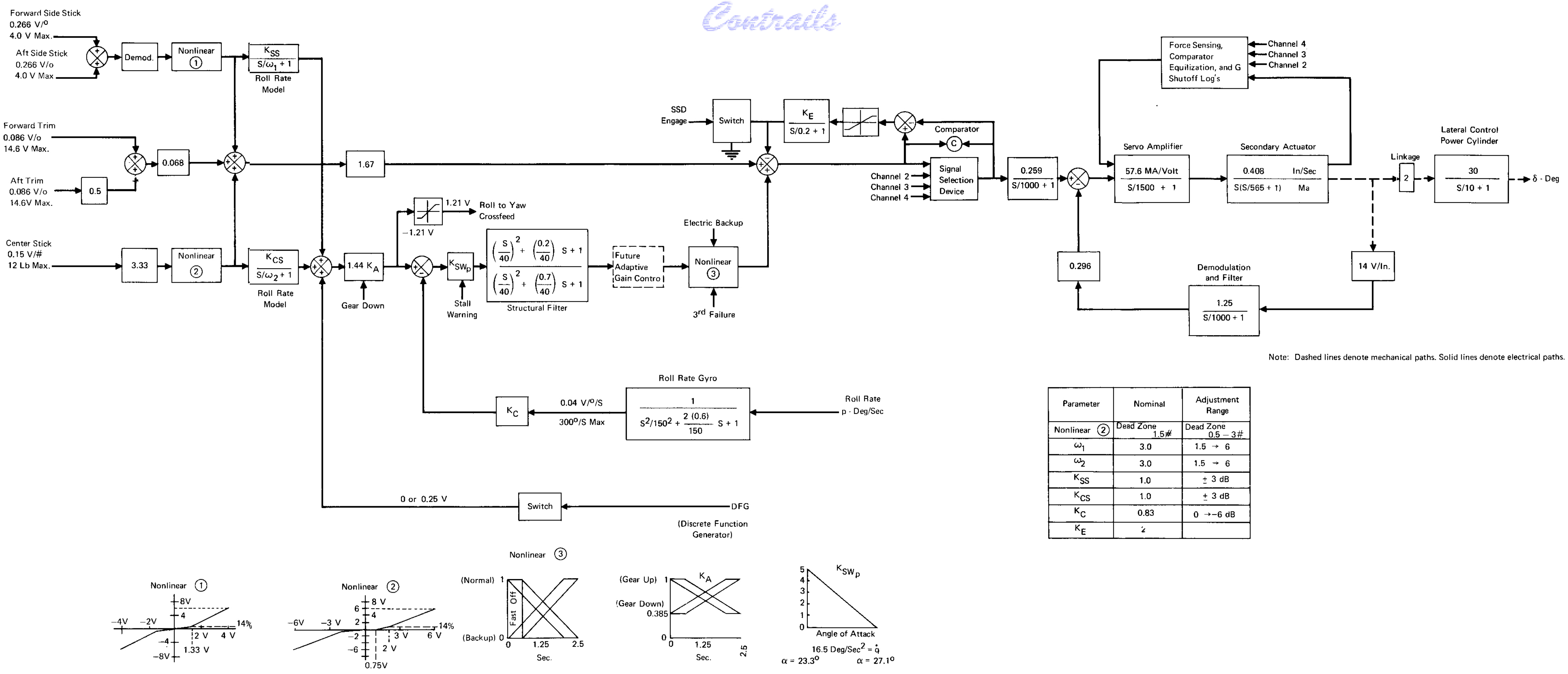
The lateral and directional axes functional block diagrams are presented in Figures 31 and 32 respectively. Each axis has two major modes of operation: a Normal mode in which aircraft motion is controlled through pilot commands and closed loop feedbacks, and an Electrical Back-Up mode in which aircraft motion is an open loop response to pilot commanded control surface deflection. The directional axis has an additional Mechanical Back-Up mode in Phase IIA only.

(1) Lateral Axis

The lateral control system was designed to comply with the criteria in MIL-F-8785B (ASG), and provide a nearly constant roll rate time constant and roll rate to stick force sensitivity throughout the flight envelope. In addition the system was evaluated in terms of the criteria of Figures 23 and 24. A fixed gain in the roll rate feedback loop suffices to comply with all applicable roll criteria. However, provisions have been made for a future adaptive gain function in order to enhance system flexibility.

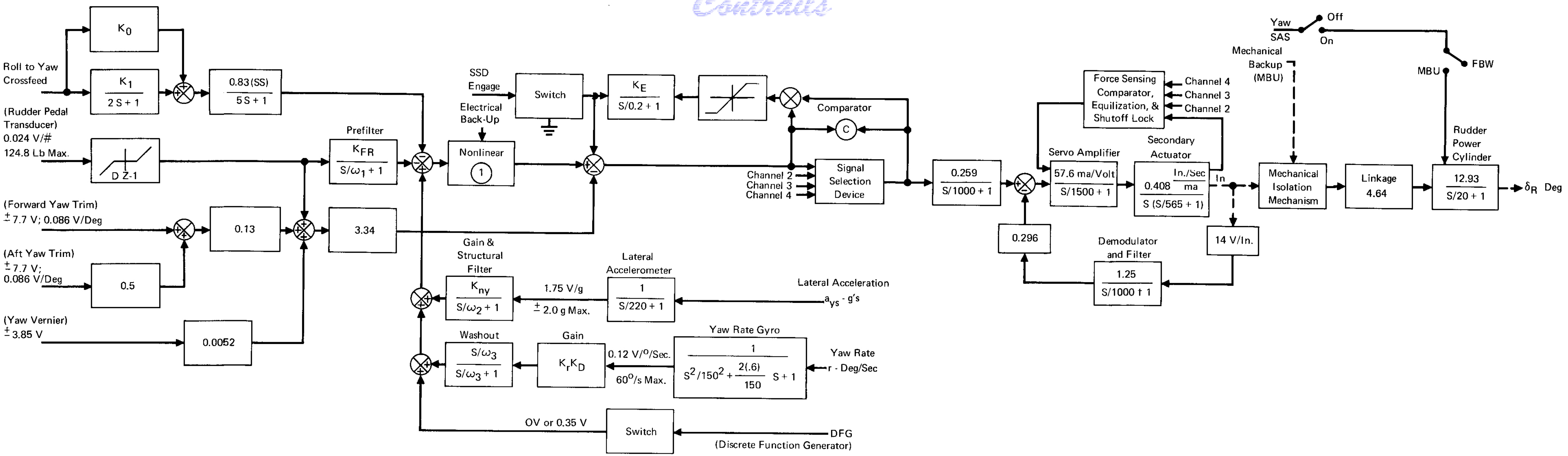
Employing the loop gain of 0.5 radians of aileron per radian per second roll rate, the SFCS roll rate time constant in the Normal mode varies between 0.35 and 0.7 seconds with the more typical values being 0.5 seconds. The roll rate to stick force response in the Normal mode is shaped by the three radian per second prefilter and the relatively high roll rate feedback gain. Although the response does not follow the 0.33 second prefilter time constant exactly, the improvement over the unaugmented F-4 is substantial.

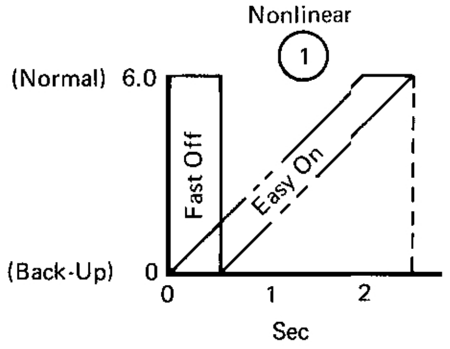
The SFCS roll rate to stick force gradient is also substantially improved over that of the F-4 with SAS. At 15,000 feet altitude the variation in roll rate to stick force with Mach number is about 3 to 1 for the F-4 with SAS. The F-4 with SFCS has about 1.35 to 1 variation under the same conditions. A dual



Parameter	Nominal	Adjustment Range
Nonlinear 2	Dead Zone 1.5#	Dead Zone 0.5 – 3#
ω_1	3.0	1.5 → 6
ω_2	3.0	1.5 → 6
K_{SS}	1.0	± 3 dB
K_{CS}	1.0	± 3 dB
K_C	0.83	0 → −6 dB
K_E	2	

FIGURE 31
LATERAL AXIS FUNCTIONAL BLOCK DIAGRAM





Level / Gain	Gear Up: Adaptive $M_\delta < 6$	$6 < M_\delta < 10$	$10 < M_\delta < 30$	$M_\delta > 30$	Manual Switch Position: High	Med	Low	Gear Down
K_0	0.325	0.15	0.0	0.0	0.325	0.0	0.0	0.35
K_1	0.0	0.05	0.2	0.0	0.0	0.2	0.0	0.2
K_r	0.83	1.67	1.67	3.33	0.83	1.67	3.33	0.83

Note: Gain Change Time = 2 Sec.

Parameter	Nominal	Adjust Range
K_{FR}	0.278	
ω_1	3.0	1.5 – 6.0
ω_2	40.0	
ω_3	0.5	0.25 – 1.0
K_{ny}	0.815	± 3.0 dB
K_D	0.42	± 3.0 dB
K_E	20.0	
DZ-1	5.0 #	2.4 – 5.5 #

FIGURE 32
DIRECTIONAL AXIS FUNCTIONAL BLOCK DIAGRAM

gradient is included in the SFCS roll rate command to reduce sensitivity for small roll maneuvers for precise tracking. This non-linearity maintains good control characteristics by providing for a lower roll rate to stick force sensitivity around neutral and allowing for high roll rate commands without excessive stick force. To provide for additional lower sensitivity at approach, the roll rate command gain is reduced upon actuation of the gear down switch, as shown in Figure 31.

The EBU mode command path remains a part of the Normal mode with the gains selected to provide for maximum control surface deflection capability at maximum stick force.

The use of ailerons at or near stall on a high performance aircraft can precipitate spin. To avoid an aileron induced spin, a signal from the stall warning computer reduces effective roll rate feedback to zero as a function of pitch rate and angle of attack at high angles of attack. The aileron to stick force gain is also reduced to that present in the unaugmented airplane or EBU mode. This design feature was evaluated on the six-degree-of-freedom man-in-the-loop simulation. The results showed that the chances of inducing a spin are lower when the above design feature is incorporated into the lateral axis. In addition, once a spin has occurred, recovery is more readily accomplished as shown in Supplement 2.

(2) Directional Axis

The major requirements for the directional axis are to provide Dutch Roll mode damping and to keep the sideslip excursions in rolling maneuvers sufficiently small so as to provide good tracking performance.

To achieve good damping and minimize sideslip excursions, a combination of yaw rate and lateral acceleration feedback is employed in the directional axis. The high lateral loop gain alone prevents the roll to sideslip coupling in the Dutch Roll mode from exceeding maximums allowable. Yaw rate to rudder feedback augments the airframe Dutch Roll damping. A washout network in this feedback loop prevents opposing rudder deflection during steady-state turn maneuvers. Lateral acceleration feedback aids in reducing sideslip and provides turn coordination especially at high q flight conditions. Turn coordination at low and mid q flight conditions is augmented by the roll to yaw crossfeed network.

A fixed gain is provided in the lateral acceleration feedback loop and a three value variable gain is included in the yaw rate feedback loop. This variable gain is both pilot selectable and controllable by the adaptive gain changer as a function of the longitudinal M_δ parameter. The variable gain is required to minimize the effects of yaw rate feedback on adverse yaw at low M_δ flight conditions, while providing sufficient Dutch Roll damping at high M_δ flight conditions. The gear down switch adjusts the yaw rate gain for the landing configuration.

As in the lateral axis, the directional axis EBU mode is utilized during Normal mode operation with overall gains selected to provide full rudder authority at maximum rudder pedal force. The Normal mode command gain is set to offset the feedback signals at low q where full rudder deflection may be desired. A three radian filter is included in this command path to reduce the initial command gradient and prevent over sensitivity of the rudder pedals due to the added gain.

(3) Roll to Yaw Crossfeed

The directional axis yaw rate and lateral acceleration feedbacks provide satisfactory Dutch Roll mode damping, but do not provide for sufficient improvement in turn coordination. Severe sideslip excursions resulting from rolling maneuvers can exist at several flight conditions. Increasing lateral acceleration feedback is not feasible due to its degrading effect on Dutch Roll mode damping, the higher rms g environment experienced by the pilot during random gusts, and structural mode feedback. Therefore, a crossfeed from the roll rate command signal to the directional axis was selected for the SFCS.

An analysis of the F-4's roll induced sideslip shows that sideslip varies with flight condition, from highly adverse at high angles of attack to proverse at low angles of attack. A fixed network crossfeed therefore would not be sufficient. The SFCS roll to yaw crossfeed is relatively uncomplicated and adequate to provide a significant reduction in sideslip. The network incorporates two variable gains which are changed automatically as a function of M_δ by the adaptive gain changer, or manually by the directional axis manual gain select switch. Four sets of gains, varying from high gains at low M_δ to zero gain at high M_δ, are provided through the adaptive gain changer. A fifth set of gains is used with gear down. A limiter is required to avoid excessive crossfeed commands at low roll power flight conditions.

(4) Structural Modes

Three structural modes have been identified as factors to be considered in the design of the lateral-directional control system. An analysis of these modes show that the lateral loop is primarily sensitive to the fuselage first torsion mode. The low frequency and the high sensitivity of this mode to aileron deflection, coupled with the high SFCS lateral loop gain, required that a 10 dB, 40 radian notch filter be provided in the lateral loop. The directional axis is primarily sensitive to the fuselage first lateral bending mode through the lateral accelerometer pickup. Sufficient attenuation of this structural mode in the SFCS is provided by the low lateral acceleration loop gain and a 40 radian lag filter in the lateral acceleration feedback loop. No filter for structural modes is required in the yaw rate loop.

Table XII shows that adequate phase margins, gain margins, and structural mode attenuation have been achieved using the lateral-directional SFCS. The minimum lateral axis stability margins are 61 degrees of phase and 15 dB of gain. The minimum directional axis stability margins are 58 degrees of phase and 13 dB of gain. Gains at all structural frequencies are -10 dB or less in the feedback loops.

TABLE XII
LATERAL-DIRECTIONAL GAIN AND PHASE MARGINS

Flight Condition Mach/Alt	Yaw Rate Gain (K_i)	Lateral Loop**							Directional Loop***						
		Phase Margin		Gain Margin		Gain (dB) at Flexible Modes			Phase Margin		Gain Margin		Gains (dB) at Flexible Modes		
		γ (Deg)	ω_ϕ (Rad/Sec)	a (dB)	ω_c (Rad/Sec)	η_4	η_5	η_6	γ (Deg)	ω_ϕ (Rad/Sec)	a (dB)	ω_c (Rad/Sec)	η_4	η_5	η_6
0.5/5K	1.5*	80.8	4.38	19.3	19.4	24.5	-36.5	-34.1	90.1	4.5	26.0	49.5	-22.0	-42.3	17.3
0.5/25K	0.75*	96.5	2.61	26.6	19.5	-32.0	-45.0	-39.0	116.2	1.85	37.5	54.5	-30.4	-53.0	-22.3
	1.5	98.0	2.55	25.6	18.4	-32.0	-45.0	-39.0	101.3	2.52	32.3	50.0	-28.3	-48.0	-22.9
	3.0	98.8	2.51	25.6	18.5	-32.0	-45.0	-39.0	89.1	4.0	26.7	45.3	-24.9	-37.0	-24.2
0.84/SL	0.75	63.3	6.9	15.4	20.5	-14.1	-27.0	-17.0	80.4	6.6	22.9	57.5	-15.4	-22.5	-13.4
	1.5*	61.1	7.3	15.4	20.5	-14.1	25.4	17.1	77.2	10.8	17.8	52.2	-13.4	-21.7	14.0
	3.0	63.2	7.2	15.1	20.0	-14.2	25.4	17.2	53.7	17.8	12.4	47.0	10.3	-19.9	15.0
0.9/15K	1.5*	66.3	6.03	16.4	19.6	-17.6	28.0	20.2	82.3	7.9	20.1	52.5	-15.0	-26.2	-15.0
0.9/35K	1.5*	75.4	4.13	19.6	18.6	-25.7	-34.8	-26.8	89.7	4.25	27.2	51.5	-22.9	-41.8	-19.7
0.9/45K	0.75	82.5	2.96	23.1	18.0	-30.0	-38.0	-30.5	107.3	2.12	38.0	54.5	-30.5	-48.6	-22.2
1.2/5K	0.75*	85.5	4.57	21.7	20.0	-11.4	-22.2	-9.1	102.4	6.2	37.0	53.4	-27.3	-24.7	-21.1
	1.5	87.9	4.42	21.4	19.6	-11.4	-22.2	9.1	82.7	6.9	30.8	49.2	-25.8	-25.0	21.6
	3.0*	89.7	4.3	21.4	19.4	-11.4	22.3	9.3	73.1	8.13	24.8	45.0	22.8	-28.4	22.7
1.5/15K	3.0*	100.5	3.25	22.8	19.3	-14.8	-24.8	-11.7	80.1	6.55	30.2	42.0	-30.0	-29.7	-22.1
1.5/35K	1.5*	82.7	3.55	21.0	19.0	-21.2	-30.0	-16.2	82.7	4.6	32.7	44.0	-31.1	-34.5	20.3
1.5/45K	1.5*	87.3	3.0	22.2	18.4	-25.4	-32.0	-19.8	89.3	3.9	33.9	43.0	-32.8	-39.0	-21.3
1.8/55K	1.5*	95.0	2.0	24.9	18.0	-27.7	-33.0	-21.0	88.7	3.0	34.7	33.7	-38.5	-35.6	-20.7
2.15/36K	1.5*	99.6	2.48	24.0	18.3	-16.3	-26.0	-11.4	58.3	4.39	29.4	26.5	-43.9	-27.1	-17.9

η_4 First fuselage torsional mode - 40 Rad/Sec
η_5 Unsymmetric wing bending mode - 68 Rad/Sec
η_6 First lateral bending mode - 80 to 104 Rad/Sec

* Adaptive gain value
** Based on a frequency response of the open lateral loop with the directional loop closed
*** Based on a frequency response of the open directional loop with the lateral loop closed

e. Conclusions

As a result of the studies reported herein, the following conclusions can be drawn:

(1) The SFCS will provide adequate stability and good performance characteristics in the Normal mode.

(2) The electrical back-up mode will provide for safe return and landing in the event of total failure of the Normal mode.

(3) The longitudinal SFCS, when operating in the Normal mode, provides airframe responses which compare favorably with the C* and $\dot{C}$* criteria throughout the F-4E flight envelope.

(4) The lateral SFCS, operating in the Normal mode, provides improved airframe roll response throughout the F-4E flight envelope.

(5) The directional SFCS, operating in the Normal mode, provides Dutch Roll damping which meets the MIL-F-8785B (ASG) requirements throughout the F-4E flight envelope.

(6) The Roll to Yaw Crossfeed provides improved turn coordination and meets the MIL-F-8785B (ASG) requirements except for some very low $\bar{q}$ and very high Mach number flight conditions.

(7) The stall warning mechanization provides good indication of the approach to stall.

(8) Adequate structural mode stability margins for stable operation are provided by the three axis SFCS through the correct placement of feedback sensors and utilization of structural filters.

14. SWITCHING TRANSIENTS

a. Introduction and Summary

Switching transients within the SFCS can result from any one of the following switching functions:

Mode Switching
Gain Switching
Channel Failures

If these transients were permitted to progress uninhibited through the system to produce undesirable surface deflections, objectionable aircraft motions would result. Each of the transient producing functions, the techniques employed to alleviate the transient, and the methods of evaluation are discussed in the ensuing paragraphs.

Switching transients which may be incurred within the SFCS are alleviated by a combination of linearly variable gain functions, easy-on, easy-off circuits and operational procedures.

b. Mode Switching Transients

The SFCS must operate in each of the four following modes:

Mechanical Back-Up
Electrical Back-Up
Demand-On
Normal
   Take-Off and Land (TOL)
   Neutral Speed Stability (NSS)

Mode switching transients can occur any time the operating mode of the SFCS is changed. However, mode switching or changing from TOL to NSS, or vice versa, of the normal mode, with the exceptions noted and discussed in Supplement 2, can occur only upon command of the pilot. Therefore, to simplify the mode switching mechanizations and retain a high level of reliability in these mechanizations, it was decided that the pilot would be required to maintain control of the aircraft, including the smoothing of any transients, during mode switching. Easy-on, easy-off circuits, as described in Supplement 2, have been implemented in the SFCES to delay any immediate effect of a transient introduced by switching between modes, thereby providing the pilot with additional time in which to apply the corrective action required to control the aircraft.

Pilot evaluations of the mode switching transients and of the easy-on, easy-off implementations were made during the course of the selected control system simulations. The results of these evaluations are contained in Supplement 2.

c. Gain Switching Transients

Gain switching transients will occur any time manual or adaptive gain switching is employed while an error exists in the forward loop of the SFCES at the input to the gain control function. In the Normal (NSS) mode of operation, during steady state flight, the error in the forward loop will be very nearly zero and the resultant transient will be correspondingly small. In the Normal (TOL) mode however, an error may exist in the forward loop during steady state flight and a transient may be expected when the gain is changed.

To alleviate this transient, a gain switching time of 2 seconds is planned in conjunction with the gain control to linearly vary the gain between the original and newly selected value, thereby converting a possible step input to the surface actuator to a ramp type input. The gain switching time of two seconds was implemented and evaluated during the selected control system evaluation simulation and found to be an acceptable solution to the reduction of this form of switching transient. See Supplement 2 for the results of this evaluation.

d. Channel Failures

Transient effects of sensor failures and failures within the computational electronics are directly related to Signal Selection Device (SSD) and in-flight monitoring (IFM) circuitry characteristics. The discussions which follow describe the SSD and IFM characteristics and the effect of upstream failures on the output of the SSD. These discussions cover general theoretical operation only. Details of circuit design will be presented in the next Interim Report.

(1) Signal Selection Device

In each channel of each axis a signal selection device is used to provide a common input to the secondary actuators and to monitor the operation of the computational circuitry.

Figure 33 contains a block diagram of the SSD. The SSD is of the operational amplifier type using a voltage summing technique. The high gain amplifiers and common feedback point cause saturation which, in turn, causes the signal selection phenomenon to occur. There is, of course, no mid-value of the four signals produced when this technique is employed. Instead, with all four channels operating, the SSD selects and passes the least magnitude mid-value. The principle by which the SSD selects the least magnitude mid-value can best be described by inductive reasoning. Refer to Figure 34 and let

e_1 = +3 volts

e_2 = +2 volts

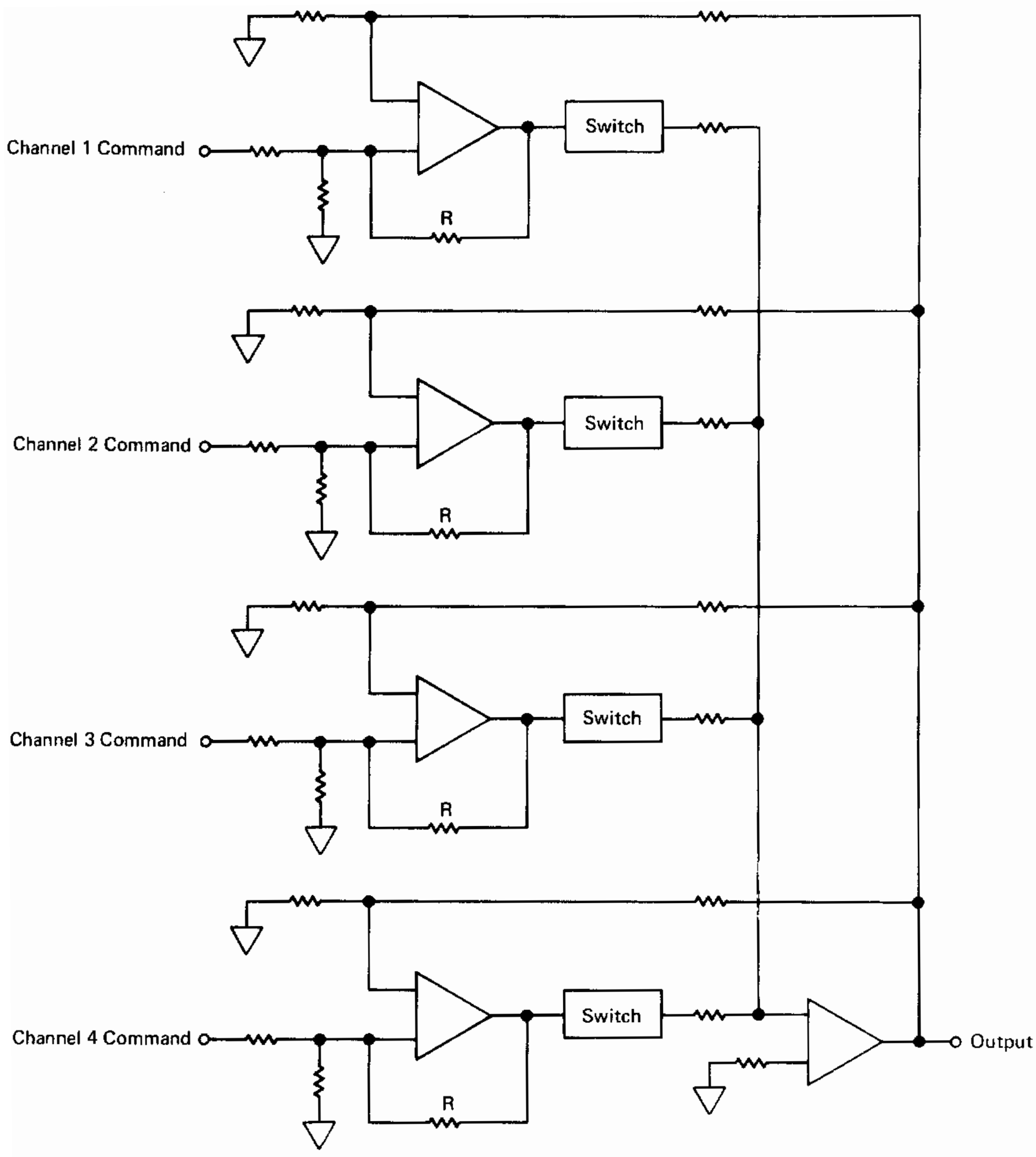


FIGURE 33
SIGNAL SELECTION DEVICE
Four Channel Operational Amplifier Type

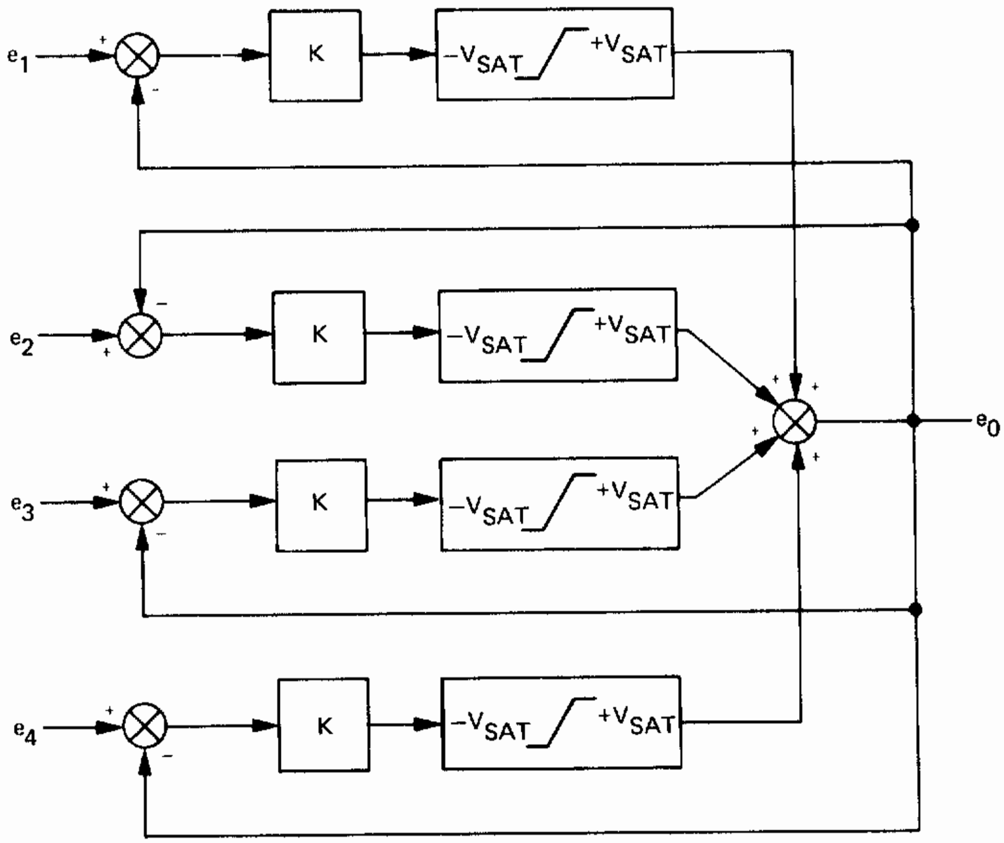


FIGURE 34
SSD BLOCK DIAGRAM

e_3 = +1 volts

e_4 = 0 volts

Assume that the output e_o is equal to one of the inputs, in this instance +3 volts, and the amplifier gains K equal to ∞. For this case the output of three of the four amplifiers will be $+V_{SAT}$ and the sum $+3\ V_{SAT}$. The remaining amplifier will not be capable of providing an output sufficient to cancel this $+3\ V_{SAT}$ voltage therefore this condition cannot exist. Next assume the output to be +1 volt. When this output is summed with each input as indicated in Figure 34 the result is that two amplifiers provide an output of $-V_{SAT}$ and one amplifier provides an output of $+V_{SAT}$. The remaining amplifier, in order to provide an SSD output of +1 volt, must then provide an output of $(V_{SAT} -1)$ volts. The above verifies that the least magnitude mid-value satisfies the condition.

The operating characteristics of the SSD when subjected to the accompanying conditions are presented in Table XIII.

TABLE XIII

OPERATING CHARACTERISTICS OF OPERATIONAL AMPLIFIER TYPE SIGNAL SELECTION DEVICE (SSD)

No.	Condition	Characteristics
1	All input signals are identical	The output will follow the inputs with unity gain and an offset less than the offset of the worst amplifier in the set.
2	All inputs track but with fixed differentials.	The output will follow the least magnitude mid-value with the same offset as in condition (1). There will also be a dead zone around zero equal to the difference between the two mid-values unless the resistor's R, in Figure 27, are included. These resistors prevent saturation for small null tolerances, producing an averaging effect in this region. The influence of this averaging region is controlled by the size of the resistors and has minimum effect on the large signal voting characteristics.
3	One input hardover	Same as condition (2) with the hardover acting as one input (the hardover is not passed through the SSD).
4	A failed channel has been switched out, leaving a three-channel SSD.	The defective channel will be switched out by removing the power to the SSD amplifier. The circuit then operates as a three-channel SSD, and the median signal will be selected.
5	Two channels have failed and have been switched out.	The output will follow the least magnitude of the two remaining inputs or the average around zero.
6	Maximum available output swing	The output swing is determined by the saturation level of the output amplifier and can easily reach ± 10 volts.

The SSD is subject to latent failures that will not show up until a channel fails. A failure in any one of the three high gain amplifiers associated with the signals received from the other three channels will not be detected by the IFM. A second failure, in the computational circuitry of one of the remaining two channels, may result in a monitor race which could result in the good channel being cut off. BIT is designed to detect latent failures.

All failures of the SSD, except for the final summing amplifier, are latent in a four-channel SSD. The SSD inputs are well isolated from SSD failures by the input resistors. The value of these resistors will be chosen so that the least degradation in performance is experienced as a result of latent failures. Failures of the final summing amplifier will be detected by the comparator and the channel output will be shut off. The secondary actuator monitor will also detect these failures and switch off the actuator element in that channel.

(2) IFM Circuitry

The IFM circuitry used in conjunction with the four channel SSD's in the SFCES is of the cross-SSD type.

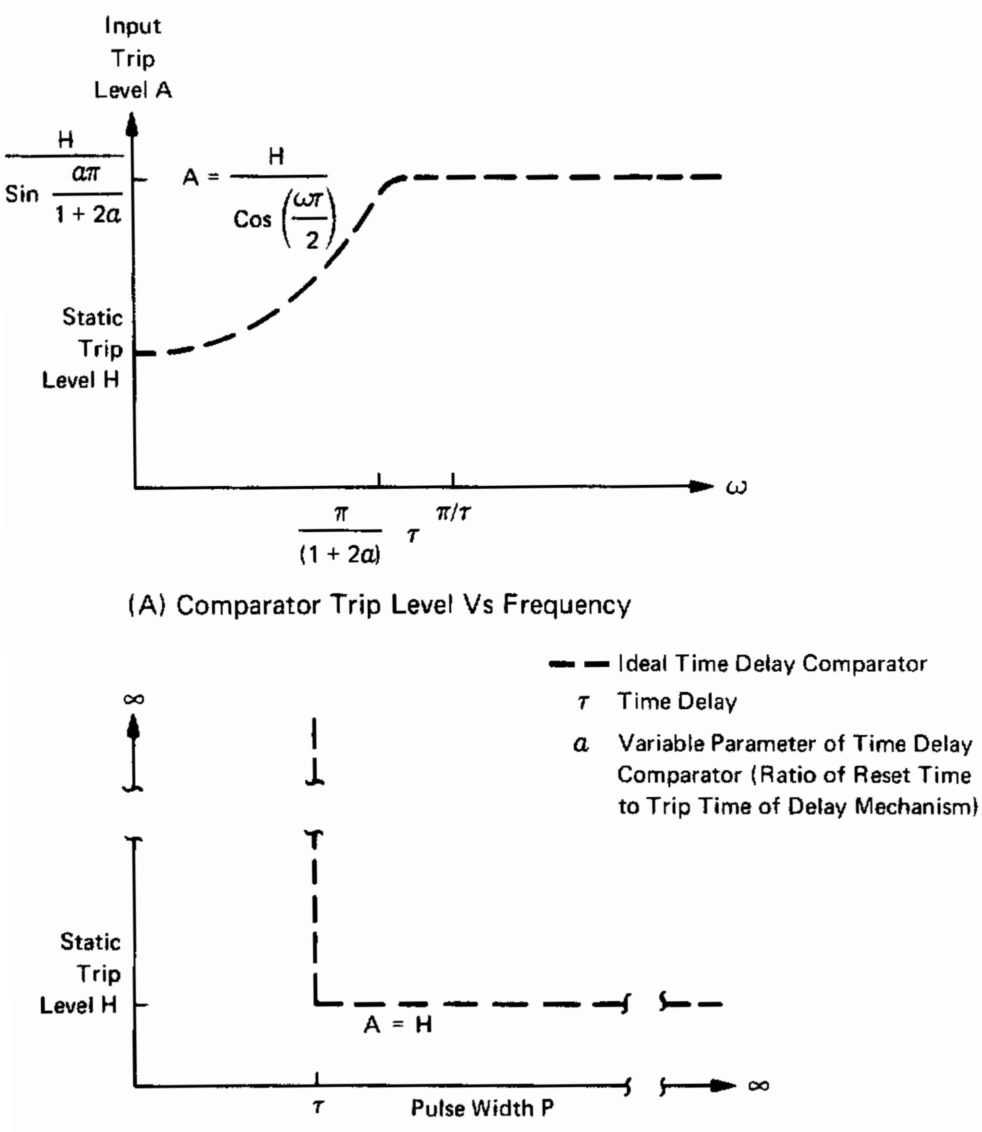
The comparator configuration is of the time-delay type which waits a specific interval of time, presently set at approximately 200 milliseconds, after a failure has occurred before declaring a fault. As demonstrated in Figure 35 (A), the comparator trip level of the time-delay comparator has a flattened characteristic above a frequency determined by the parameter a. Thus, by varying this design parameter, the time-delay comparator trip level can be made nearly insensitive to input frequency or nearly infinite at a specified cutoff frequency. This design flexibility is a definite advantage for the time-delay comparator.

The sensitivity of the time-delay comparator is shown in Figure 35 (B). This figure illustrates the capability of the time-delay comparator to reject high-amplitude, short time duration pulses.

(3) Transient Performance

(a) The transient performance of a four-channel SSD is directly dependent upon the nature of the upstream failure producing the transient and the tolerances of the input signals prior to the failure. Figure 36 illustrates the transient performance of a four channel SSD in the presence of first, second and third failures. The effects of limited averaging in the SSD are small for large input tolerances such as those illustrated in this figure and therefore neglected. A complete description of averaging and its influence on transient levels are discussed further on in this section.

At time t_1 in Figure 36, the voter represented by Figure 34 is subjected to a negative hardover failure in channel 3, which is corrected at time t_2. The initial transient occurs at t_1 and can have a maximum amplitude equal to the maximum channel offset level. The duration of this transient, from t_1 to t_2, depends on the comparator time delay and can be made quite small. The failure correction transient occurs at time t_2 and is accompanied by a level shift in the SSD output. This level shift is the result of establishing a new voted output and, again, has a maximum value equal to the maximum offset between channels. The level shift transient, unlike the initial transient, is independent of comparator characteristics. Therefore, aircraft transients corresponding to this level shift are functions of the tolerance level, SFCES closed loop gains, and pilot response. Second failures, such as that occur-



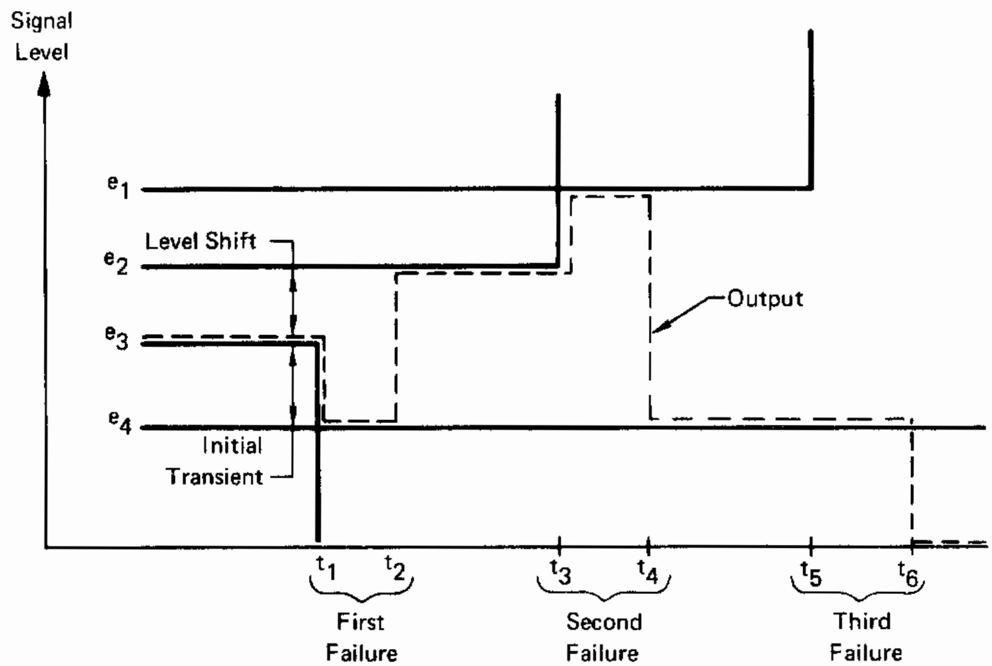
(A) Comparator Trip Level Vs Frequency

(B) Comparator Pulse Trip Level vs Pulse Width

FIGURE 35
TIME-DELAY COMPARATOR CHARACTERISTICS

ring at time t_3, have similar characteristics to first failures. Worst case initial and failure correction transients depend directly upon signal offset levels occurring at the SSD inputs. Equalization as a means of controlling these offset levels is discussed further on in this section.

Third failure transient characteristics can differ greatly from those associated with first and second failures. For failures away from zero, such as illustrated in Figure 36, the failed channel does not influence the SSD output immediately. A failure correction transient, however,



Notes:

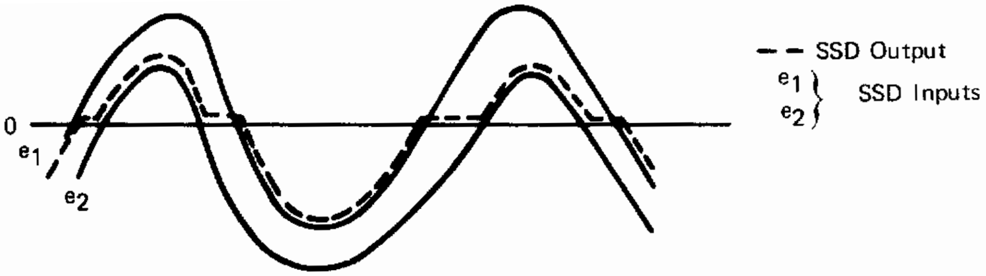
- Transient Magnitude Depends Upon Offset Amplitudes and Distribution
- Worst Case Initial Transient is Equivalent to Maximum Input Offset for First Two Failures, Signal Level for Third Failure.
- Worst Case Failure Correction Transient is Equivalent to Maximum Offset for First Two Failures, Signal Level for Third Failure.

FIGURE 36

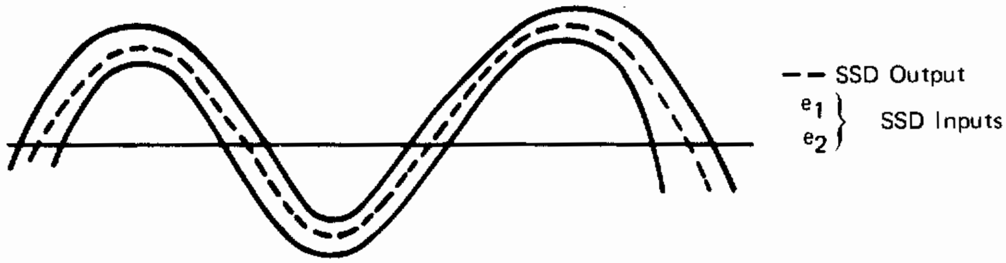
SSD INPUT FAILURE TRANSIENT CHARACTERISTICS

equal to the signal amplitude of e_4 is experienced. Had the failure been in the opposite direction, an initial transient equal to the amplitude of e_4 would have been experienced (the output going to zero) with no associated failure correction transient. This SSD configuration which rejects failures of either polarity away from zero results in the best possible implementation, since zero represents trim in the normal operating mode. Circuits which select the most positive or most negative of the two channels could appreciably increase transient levels.

(b) The SSD shown in Figure 34 uses a common feedback point and amplifier saturation characteristics to produce the signal selection phenomenon. If the gains K in this illustration are set to infinity, the SSD output corresponds to the least magnitude mid-value or zero for four channel operation, the mid-value input for three channel operation, and the least magnitude input or zero for two channel operation. With infinite gains, however, a deadband is incurred at zero when an even number of channels is operating. This deadband effect, shown in Figure 37(a), can be



(a) Two Channel SSD Output for $K = \infty$



(b) Two Channel SSD Output for $K = \dfrac{V_{SAT}}{e_{NULL}}$

FIGURE 37
SSD DEADBAND CHARACTERISTICS

eliminated by reducing the amplifier gain K until null offsets (e_{null}) no longer cause amplifier saturation; i.e., a small averaging region is created around null. This averaging effect will occur only so long as the amplifiers are not driven to saturation. The amplifier gain required to produce the desired averaging effect can be determined as follows:

Let

$$e_1 = e_2 = +e_{null} \tag{1}$$

and

$$e_3 = e_4 = -e_{null} \tag{2}$$

then

$$K = \frac{V_{SAT}}{e_{null}} \tag{3}$$

The effect of this gain reduction in correcting the dead-band problems is shown in Figure 37(b).

The influence of reducing amplifier gains on SSD transient characteristics can be evaluated by setting all inputs in Figure 34 equal except for one input which varies so that its corresponding amplifier output changes from $-V_{SAT}$ to $+V_{SAT}$. Using Mason's rule as applied to non-linear functions, the maximum transient output obtainable in the averaging region can then be derived as a pulse of amplitude

$$\Delta e_{o_{max}} = \frac{2\ V_{sat}}{1 + (n - 1)\ K} \qquad (4)$$

and a Δe_o level shift after the removal of the failed input of

$$\Delta e_o \text{ level shift} = \frac{V_{sat}}{1 + (n - 1)\ K} \qquad (5)$$

where n is the number of active channels. This transient is shown in Figure 38 (a) where t_1 is the time of failure and t_2 is the time of failure correction. Combining equations (3) and (4)

$$\text{Initial transient} = \Delta e_{o_{max}} = \frac{2K\ e_{null}}{1 + (n - 1)\ K} \qquad (6)$$

or

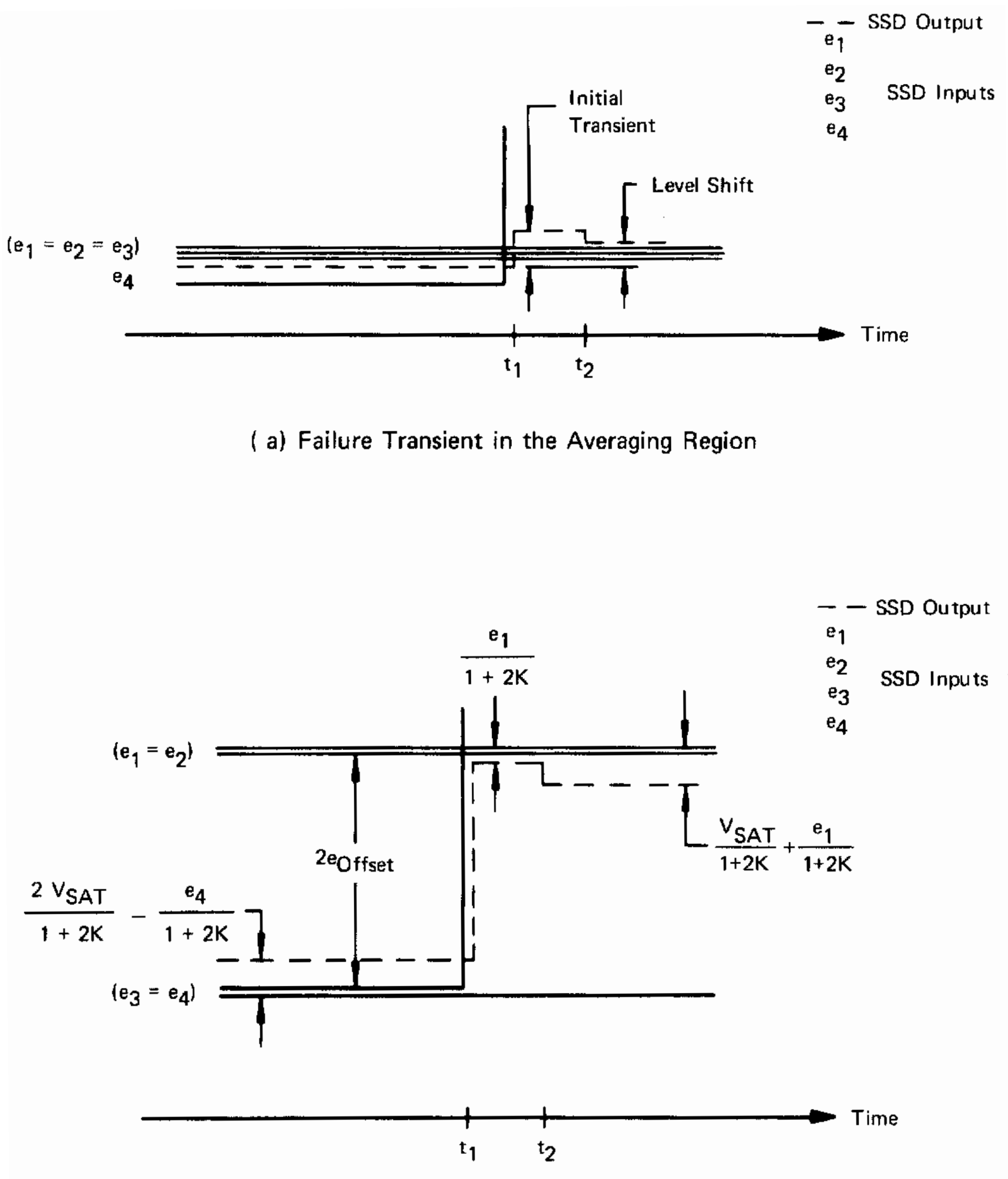
$$\text{Initial transient} \approx \frac{2\ e_{null}}{n - 1} \qquad (7)$$

combining equations (3) and (5)

$$\text{Level Shift} \approx \frac{e_{null}}{n - 1} \qquad (8)$$

Equation (7) indicates that reducing the amplifier gains to eliminate deadband allows the SSD to pass transients directly proportional to null offset levels and inversely proportional to the number of good channels remaining when the SSD is operating in the averaging region. Equation (10) is representative only for failures away from zero. Hardover failures from nominal toward zero will result in both amplifiers saturating and a net zero output. Correcting third failures always results in total SSD shutdown to zero creating level shift equal to the nominal SSD output before failure occurrence.

The SSD with infinite gain K (no averaging region) also exhibits failure transients proportional to channel mismatch if the input failure is of the appropriate polarity or involves the selected input. These results are indicative of SSD operation outside the averaging region where total



(a) Failure Transient in the Averaging Region

(b) Worst Case Failure Transient

FIGURE 38
SSD FAILURE TRANSIENTS

tolerances are much greater than null tolerances and amplifier saturation occurs. In the worst case, this transient can be approximated by a step level shift.

$$\Delta e_{o_{K=\infty}} = 2\, e_{offset} \approx \Delta e_{o_{K \neq \infty}} \qquad (9)$$

where $\pm e_{offset}$ is total channel voltage tolerance including e_{null} in equation (7). The exact transient including the effects of finite gain is shown in Figure 38 (b). As shown, equation (9) is a good approximation for reasonably high values of K. Note that in this extreme case, reducing amplifier gains actually reduces the size of the failure transient.

For expected tolerance levels, equations (7), (8) and (9) indicate that the failure transient limits of 0.5g in pitch, 0.25g in yaw, and 10.0 deg/sec in roll will be exceeded. Equalization techniques, however, may be applied to reduce the magnitude of input tolerances effectively and to allow realization of specified airplane transient limits.

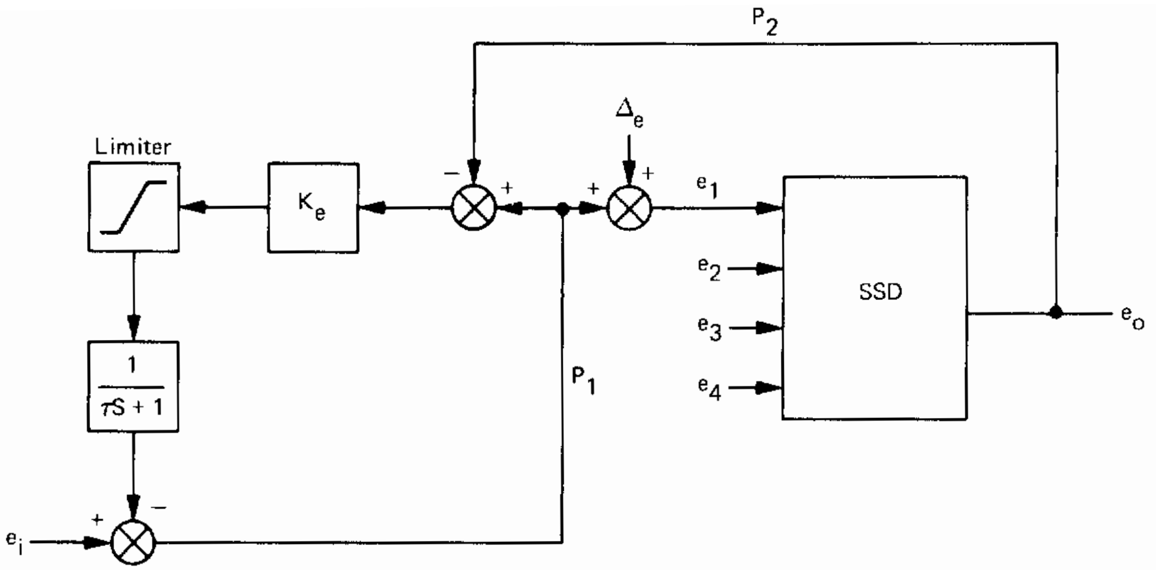
The equalization technique and some practical circuit considerations are shown in Figure 39. Statistically, the equalization loop reduces input offsets appearing at the SSD inputs by a factor of $\frac{1}{1 + K_e}$. Equations (7) and (8) can therefore be written as

$$\text{Initial transient} = \frac{2\, e_{null}}{(n-1)(1+K_e)} \qquad (10)$$

$$\text{Level shift} = \frac{e_{null}}{(n-1)(1+K_e)} \qquad (11)$$

(Note that equations (10) and (11) do not hold in the general case for n = 2 (third failure).)

This initial transient, the level shift at fault correction, and the effect of equalization time constant are shown in Figure 40. The initial transient is of short duration and will appear as an impulse function to the airplane. After correcting the fault, the SSD output shifts levels. If a different input channel is selected as the SSD output, a slow transition towards the unequalized value of this newly selected channel occurs. The speed of this transition is controlled by the equalization time constant λ and has a maximum amplitude determined by the unequalized SSD input tolerances. If the equalization time constant is set at a sufficiently high value (MCAIR studies indicate approximately 5 seconds), pilot corrective action can be effective in reducing airplane transient levels to acceptable levels. For purposes of transient evaluation, the transient waveform in Figure 38 (a) may be used.



Practical Considerations

Equalization Limit Necessary to Allow Slowover Detection.

Circuit is Unstable if $K_e > \frac{1}{1-a}$ Where K_e is the Static Value of $K_{e(s)}$ and a is the Ratio of the Gain in Path P_1 to the Gain in Path P_2.

Offsets in the Signal Selection Device Appear Multiplied at the Circuit Output.
i.e., $e_o = (1 + K_e)\ \Delta e$ Where Δe = SSD Offset.

FIGURE 39
EQUALIZATION OF THE SSD

The initial transient and level shift are described by equations (10) and (11) subject to the following conditions and limitations:

(1) The airplane is in non-maneuvering flight so that the equalization loop has reached a static condition.

(2) The equalization gain is sufficiently high to provide operation within the averaging region.

(3) The SSD is operating around a nominal null condition.

(4) For n = 2 (third failure), equation (10) is true only for failures away from zero and equation (11) is not valid.

Condition (1) will be satisfied at nearly all times, since the equalization bandwidth will be quite high (although

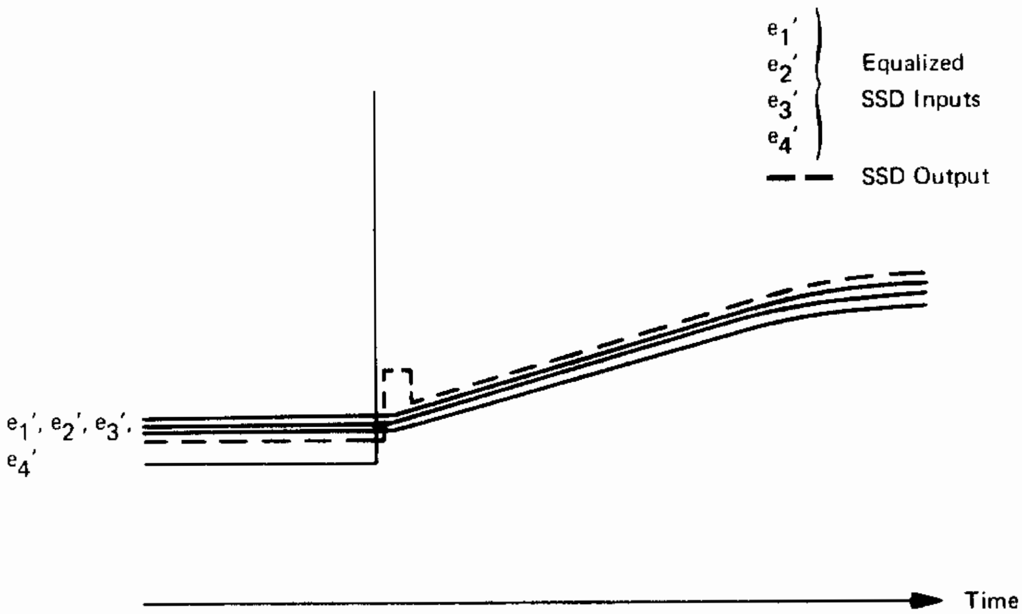


FIGURE 40
EQUALIZED SSD FAILURE TRANSIENT

limited in maximum slew rate by the equalization limiter). Condition (2) must be satisfied to eliminate deadband problems and condition (3) is true for the pitch axis in the Normal (NSS) mode of operation and the yaw rate and roll axis in straight and level flight.

For non-maneuvering operation in the Normal (TOL) mode in the pitch axis and for steady turning flight in the yaw and roll axes, tolerances may exceed null values, and thus result in operation outside the averaging region. Equation (9) then applies. Statistically, equation (9) may be rewritten to account for the effects of equalization as

$$\Delta_{e_o} = \text{Step transient level} \sim \frac{2\, e_{offset}}{1 + K_e} \qquad (12)$$

where e_{offset} is the signal mismatch occurring at the time of failure. The transient waveform which results can be approximated by Figure 38 (neglecting as before the possible long term transition to a new selected output) for the first two failures. Again, equation (12) does not hold for third failures toward zero from a nominal output. For this case, the maximum step transient level is equal to the nominal SSD output at the time of failure.

For maneuvering flight, such that the equalization loop does not have time to reach a static value and the SSD operates outside the averaging region, the transient waveform can also be described by Figure 38 (again neglecting the long term transition to a new level). In this case, the transient amplitude is equal to the signal offset existing at the time of failure. This offset will consist only of gain tolerances associated with relatively small perturbations about an equalized level.

e. Conclusions

- o Switching transients can occur within the SFCS whenever gains, functions, or modes are changed or failures occur within the system.
- o To help prevent and/or alleviate the transients which can result when gains are changed, gain controls which vary linearly over a 2 second period have been implemented in the SFCES.
- o Transients which occur when changing functions and mode switching transients are expected to be suppressed by incorporation of easy-on, easy-off circuits within the SFCES and operational procedures to be discharged by the pilot.
- o Failure transients are suppressed by the proper mechanization of the signal selection device (SSD) and the inflight monitoring circuitry.

15. ELECTROMAGNETIC INTERFERENCE (EMI)

A study of the predicted electromagnetic characteristics of the SFCS equipment and the compatibility to be expected between the SFCS and the other test aircraft systems was conducted. Using the results of this study as a guide, electromagnetic compatibility between the SFCS and F-4 aircraft 62-12200 is planned to be implemented in the following manner:

a. Existing electrical and electronic equipment and installations have established electromagnetic compatibility. It is therefore not necessary to perform any analysis on testing relative to the electromagnetic interaction of these systems. The necessary analysis and testing is limited to the compatibility of the existing installations and the SFCS and vice versa.

b. The SFCES, SSC, SA and SSAP are being designed in accordance with the EMI requirements of MIL-STD-461A (Notice 2) except as noted below. The SFCS equipment listed above will be tested for compliance with applicable portions of MIL-STD-462 by the Suppliers prior to acceptance by MCAIR.

   The requirements deleted from MIL-STD-461 and -462 and additional EMI requirements added to the equipment procurement specifications are as follows:

   - The low frequency conducted tests covered by test methods CE01, CE02, and CE03 were deleted because analysis indicated they were not required.
   - The inverse filter tests covered by test method CE05 were deleted because these tests are alternates for test methods CE03 and CE04 which are required.
   - The test method RS04 radiated susceptibility tests were deleted because the same frequency range is covered in test method RS03.
   - The one volt per meter radiation level of test method RS03 was increased to ten volts per meter to make the test more nearly represent the predicted actual aircraft environment.
   - Test method RS02 was modified to add additional radiated susceptibility tests. The added tests have been developed by MCAIR to test for susceptibility to various stray transient spikes generated in the F-4 aircraft but not adequately covered by the standard RS02 test method.

c. The Suppliers of the SFCES, SSC, SA and SSAP are required to provide MCAIR with a recommendation for aircraft interconnect wiring provisions. The equipment Suppliers also are required to provide external circuit parameters for all wires which terminate at their equipment. External circuit parameters include the voltage and current on the wires, the source and load impedances, frequency and wave form signals, etc. The proposed interconnect wiring and external circuit parameters are studied and analyzed to determine probable coupled voltage levels from one circuit to another. The result of this analysis is used as the basis for specifying twisting, shielding, bonding and grounding requirements which are reflected on the interconnect schematic drawing used in designing aircraft wire bundles.

d. After the SFCS equipment has been installed in F-4 aircraft 62-12200, electromagnetic compatibility, between the SFCS and the existing F-4 electrical and electronic equipment and installations, will be verified by performance of an EMI Ground Test.

The EMI Ground Test is a non-instrumented functional evaluation conducted on the flight ramp with the aircraft located as far as practicable from buildings and reflecting surfaces such as fences, parked automobiles and other aircraft.

The test is divided generally into two parts. In the first part the effect of existing aircraft systems and equipment on the SFCS will be observed by operating the equipment and monitoring the SFCES display panels and aircraft control surfaces. In the second part of the test the SFCS will be operated and existing aircraft systems will be monitored for observable or audible effects.

16. LIGHTNING PROTECTION

Lightning represents a possible hazard to the SFCS in the form of conducted or induced electrical transients in the SFCS wiring. The frequency of reported lightning strikes on F-4 aircraft, considering all flight operations for a five year period, has been approximately 1.8 strikes per 100,000 flight hours. The frequency of lightning incidents on flight test aircraft is much lower. During the flight test period of the SFCS, it is not anticipated that any flights will be made during weather which is known to present a lightning hazard.

Even though it is planned and hoped that the test aircraft not be subjected to lightning strikes, some lightning protection measures have been implemented and these are summarized as follows:

- Non-conducting skin surfaces which are not required for the test aircraft have been replaced with metal panels.
- A maximum resistance value of 0.0025 ohms across mechanical joints in the aircraft skin or structure has been established as the criterion.
- SFCES wiring is being routed at least six inches away from openings in the aircraft skin wherever practical.

Lightning and surge protective devices have been considered for installation on external lights and probes, but there are no current plans to install any of these devices on the test aircraft, for the reasons detailed below.

The lightning protection devices considered for use on the test aircraft included hermetically sealed spark gaps. These devices essentially consist of an open circuit in the form of a precision gap between a power wire and aircraft structure. The theory of operation is that when a voltage substantially above the operating voltage appears on the wire the gap breaks down and conducts the generated current to the aircraft structure. These precision spark gap devices have apparently been applied successfully in a relatively benign environment.

For maximum effectiveness, lightning protective devices should be installed as close as possible to the aircraft skin opening. In the case of the SFCS aircraft this would require installation of lightning spark gap devices in the wing tips and vertical stabilizer. These locations exhibit severe vibration environments. To date MCAIR has not located a spark gap device designed for use in an aircraft vibration environment. Development and testing of such devices for use in the SFCS test aircraft would involve a considerable expenditure of time and money. The use of unqualified devices on the test aircraft could conceivably produce problems not associated with the objectives of the test program. Although a lightning strike is a recognized potential hazard, the flight test will be run such as to minimize the possibility of encountering lightning, an approach which is considered appropriate for the state of the art. It is therefore recommended that no lightning protection devices be installed on the test aircraft.

17. NUCLEAR HARDENING

Hardening of the SFCS against the effects of nuclear weapons is a design consideration. When prudently implemented, it will provide an extra margin of system survivability in a hostile environment. The objectives of the SFCS nuclear hardening program are to:

- o Reduce nuclear vulnerability at no (or minimum) cost,
- o Analyze completed design for "soft points", and
- o Make recommendations for additional hardening (elimination of "soft points") for future designs.

Operational limitations greatly reduce the types of environments and levels of intensity to which the SFCS may be exposed. These limitations are:

- o The system will be installed in an aircraft operating in the sensible atmosphere. Hence, X-rays will be attenuated by the air and will not present a real threat.
- o SFCS components are not part of the aircraft mold line. Hence, they would not be exposed directly to nuclear weapon-produced thermal, gust and overpressure threats.
- o The aircraft is manned. Hence, crew vulnerability to nuclear radiation sets a final, relatively low threshold. Hardening of any part of the system beyond this threshold serves no useful purpose.

Nuclear vulnerability of any system is a complex function of nuclear weapons products (neutrons, gamma rays, X-rays, thermal pulse, gust, and overpressure) and their interaction with system components. Practical considerations for the SFCS:

- o reduce the six damage-causing weapons products to two: Neutrons, and gamma rays, and
- o suggest a maximum practical hardness level which is compatible with crew survivability (50% mission-completion corresponds to about 2,500 rad of whole body radiation).

The total dose is deposited in man by a "typical" environmental mix of 10^{12} (neutrons per square centimeter one million electron volts-silicon equivalent) n/cm^2(1-MeV Si) and 10^8 rad (Si)/sec of prompt gamma rays, and represents a practical limit for crew survival. The associated environmental levels are also upper boundaries for useful hardening and thus establish the SFCS hardening goal.

Radiation up to this level threatens only active electronic components. Hydraulics, mechanical linkages, and passive electronic elements are not significantly affected. As a result, only one of the four SFCS packages, the Survivable Flight Control Electronic Set (SFCES), is potentially vulnerable. Reduction of the vulnerability of this unit will raise nuclear hardness of the SFCS as a whole to meet the goal. It is expected that the design goal levels can easily be met with carefully selected piece parts and simple hardening techniques.

Selection of parts for the SFCES has been completed and reflects a compromise between the desire to harden and available means to do the job:

- Silicon Controlled Rectifiers, prone to degradation and susceptible to latch-up, are not used.

- Operational Amplifiers were selected "off-the-shelf". Test data indicate that their performance in the goal environment is adequate.

- Bipolar Transistors have a high cutoff frequency, f_T > MHz. However, some of the power devices do not meet this requirement. Projected delays in the delivery resulted in the decision not to use their radiation-hard counterparts in the SFCES design. All "soft" parts could be replaced in future designs.

Circuit design to nuclear hardening goal levels is based on inherent hardness (piece parts and components), insensitivity (tolerance to transient pulses), and overdesign (tolerance to gain degradation). Analog circuits exhibit both temporary (transient) and permanent changes when subjected to a nuclear environment. Radiation pulses are typically four decades above the upper cutoff frequencies of the computation electronics,thereby precluding transient effects from this source. Computation circuits use high gain operational amplifiers. Small degradations in gain of these amplifiers is expected to have little effect on the circuit. Discrete transistor circuits are designed to tolerate gain degradation without malfunctions.

Present design of the power supply regulators minimize radiation susceptibility.

Latching circuits are only used in the SFCES monitors. These monitors have built-in time delays (at least 100 milliseconds) which will filter transient radiation effects.

Digital logic is used only in the SFCES ground Built-In-Test circuits. Although vulnerable, these circuits are disabled and cause no problem during flight.

Summary - Nuclear hardening of the SFCS to a level appropriate for a manned aircraft is considered feasible with existing parts and technology.

18. IN-FLIGHT PARAMETER AND GAIN VARIATIONS

a. General

Considerable analytical studies and simulations were conducted to examine variations of gain and other control law parameters, and these investigations are discussed in Supplement 2.

The following paragraphs give the present plans with respect to:

- o Pilot selectable variable parameters,
- o Automatic gain changer, and
- o Ground adjustable parameters.

b. Pilot-Selectable Variable Parameters

Parameters such as accelerometer lags, acceleration to angular rate ratios, pre-filter time constants, and compensation networks could have been made variable in flight. However, the use of many in-flight variable parameters would degrade system accuracy and reliability, increase circuit complexity, and complicate flight test, while contributing little to the SFCS goals. Therefore, only the forward loop gains will be pilot selectable in flight.

Aircraft responses obtained using each of the pilot selectable fixed gains at a number of flight conditions are documented in Supplement 2 of this report. Typical time histories illustrating the effect of loop gain on compliance with the time history criteria are presented in Figure 41. It should be noted that considerable variation in the responses can be obtained by changing the pilot selectable fixed gains, and that at each flight condition there is a pilot selectable gain value which produces responses that comply with the most stringent criterion envelopes. Thus, it can be seen that providing for pilot selection of fixed gain values alone allows evaluation of widely varying response characteristics at each flight condition. Additional figures illustrating the effects of pilot selectable gains are given in Supplement 2.

c. Automatic Gain Changer

(1) Original Parameter Identifier

The original gain changer design utilized an airframe pitch rate and stabilator position sensing network (Parameter Identifier) which was intended to identify the stabilator effectiveness parameter M_δ. The choice of the gain states and the justification for these levels is discussed in Supplement 2 of this report.

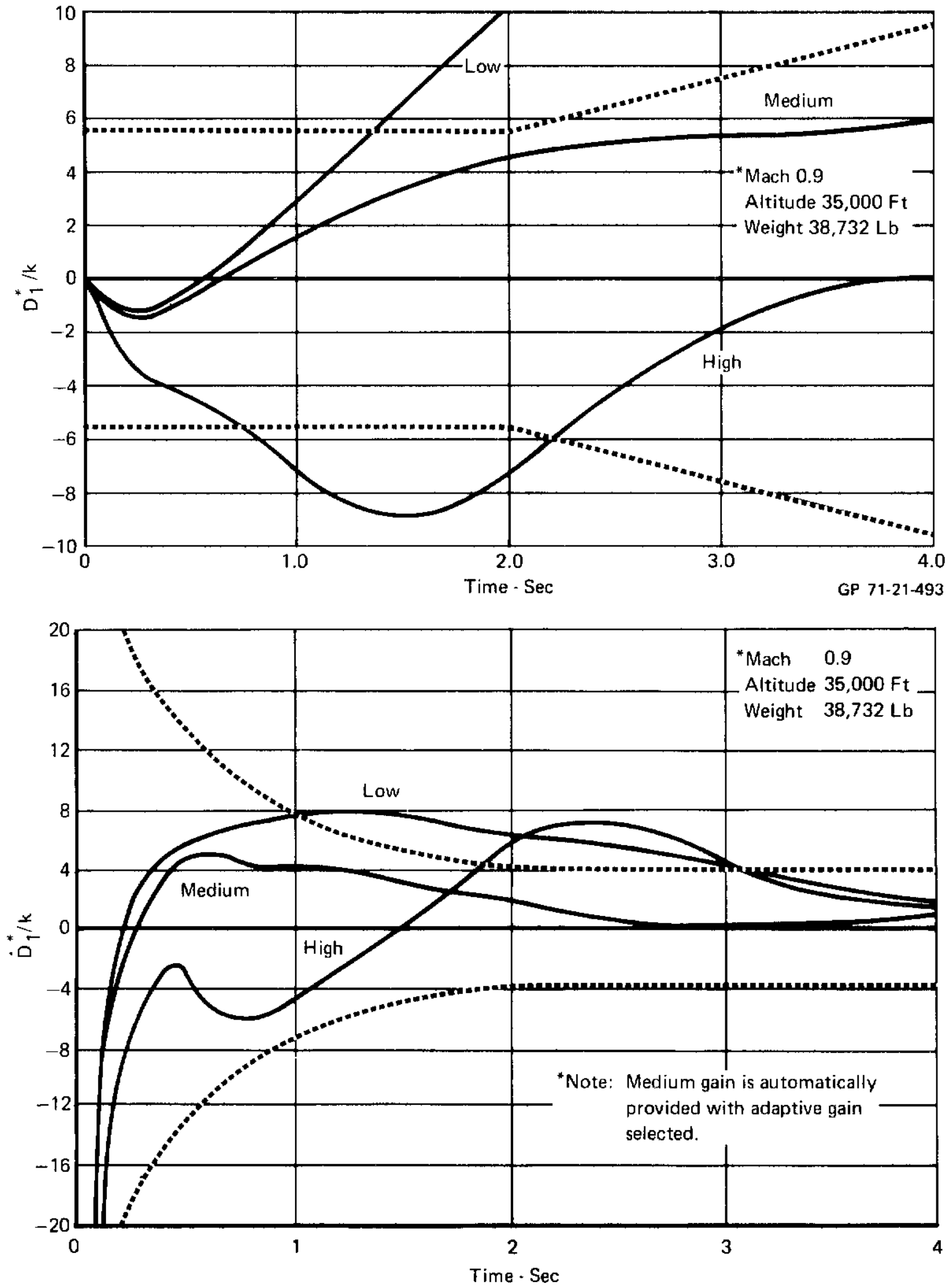


FIGURE 41
DIRECTIONAL RESPONSE VARIATIONS WITH SELECTABLE GAINS

The Parameter Identifier was designed to solve the considerably simplified pitching acceleration equation:

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

This equation is usable only if low frequency inputs below 2 Hz are eliminated from the computation. Since the $\dot{\theta}$ and δ_s inputs must be filtered to eliminate the low frequencies, errors between $\ddot{\theta}$ and $M_\delta \delta$ will be present for only a short time period for a single input. If practical gains are used in the M_{δ_c} computation loop, a number of inputs will be required to cause a significant variation in M_{δ_c}.

During the manned simulation it was found that the parameter identifier did not correctly identify the stabilator effectiveness parameter at low q flight conditions. This was a direct result of a lack of sufficient high frequency pilot inputs to cause computation. It was also found that the general operation of the parameter identifier was unsatisfactory when only directional automatic gain changing was selected due to inaccuracy in computing M_δ. Other simulation programs, which included air turbulence effects, demonstrated that the parameter identifier was quite susceptible to gusts. The parameter identifier was unable to discern the difference between pilot inputs and gust disturbances. Following considerable analyses of possible modifications, it was concluded that a new concept was needed for the SFCS automatic gain changer.

(2) Current Adaptive Gain Changer

A new concept for the automatic gain changer was designed for the SFCS. It employs a four Hertz interrogation signal with a 10% duty cycle to overcome the problems of the original parameter identifier. The revised computational configuration of the adaptive gain changer is shown in the block diagram presented in Figure 42. This automatic gain changer utilizes dual pitch rate sensor inputs and provides gain changing output discretes through selected logic and switch functions to each of the axes.

The input to the adaptive gain computer, $\dot{\theta}$, is shaped by a bandpass compensation network to minimize the effect of all but the 4 cycle per second component of the signal. The 4 cycle per second M_{δ_s} interrogator has a dual function. It allows the identification process to occur in the absence of normal command inputs, and it provides a means of long term signal averaging during those times when signal to noise ratios are very small. To eliminate the possibly degrading effects of continuous excitation from the M_{δ_s} interrogator, control logic is employed to allow a duty cycle of only 10 percent when the M_{δ_s} error signal is below a predetermined level. When this level has not been exceeded, the M_{δ_s} interrogator generates a

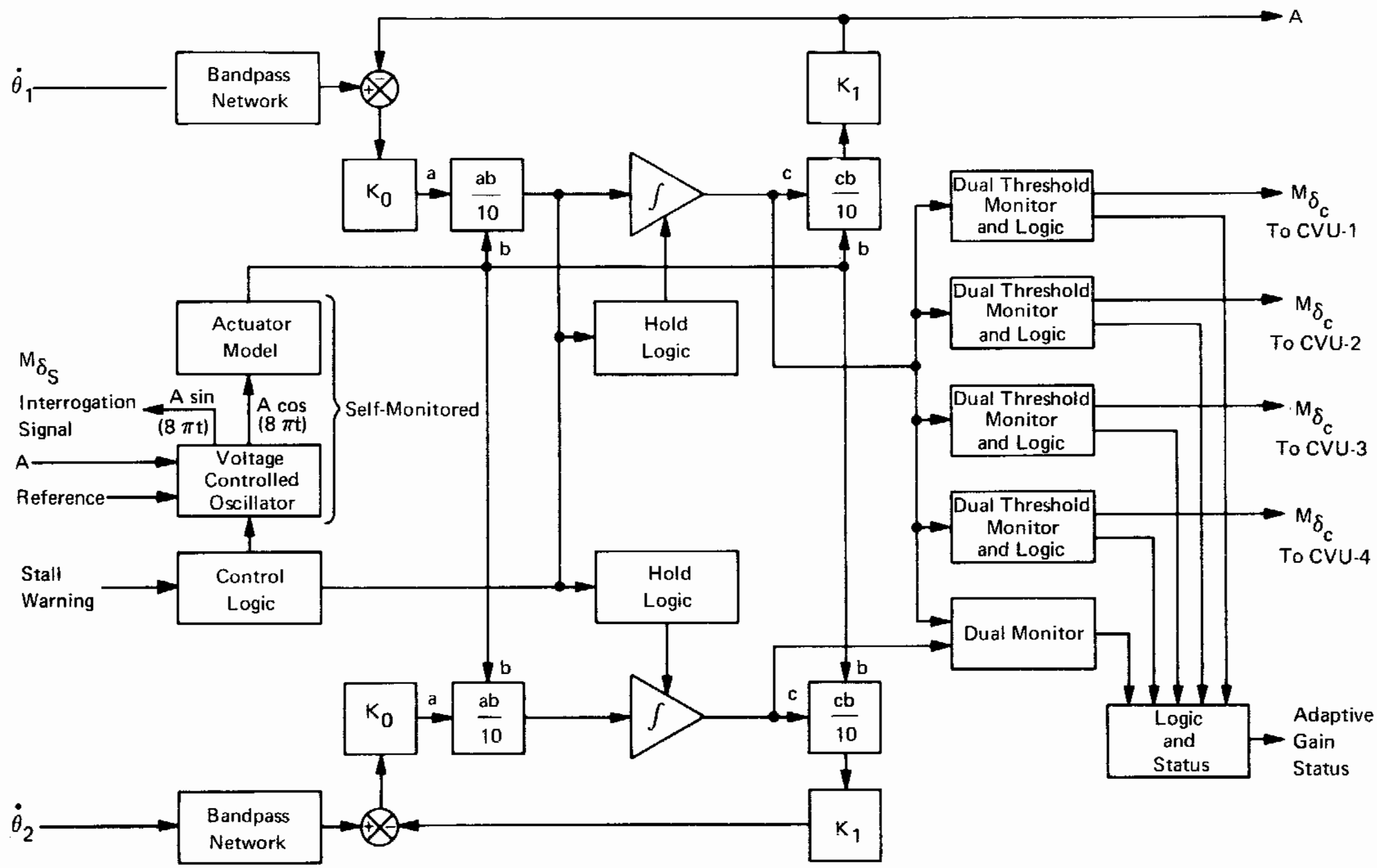


FIGURE 42

ADAPTIVE GAIN CHANGER FUNCTIONAL BLOCK DIAGRAM

one and a half second burst of limited amplitude four cycle per second sinusoidal inputs and then remains idle for approximately 15 seconds. If the threshold is exceeded, however, the M_{δ_S} interrogator remains ON, until the error signal is below the predetermined level. A voltage controlled oscillator providing both sine and cosine terms to produce a 90 degree phase difference is employed to generate the 4 Hz M_{δ_S} interrogation signal. The interrogation signal is fed to the pitch servo amplifiers to produce an aircraft pitch rate $\dot{\theta}$. The interrogation signal with the phase shifted by 90 degrees is also applied to a model of the pitch actuator which computes the value of stabilator deflection, δ. The 90 degree phase shift between the two interrogation signals causes the stabilator model output to be proportional to computed pitch acceleration. The stabilator position generated from the modeled actuator is multiplied by the computed parameter, M_{δ_C} and the product compared against the measured $\ddot{\theta}$ signal. The resulting error signal is employed to modify the M_{δ_C} parameter by the steepest descent approach until the error signal is driven within a predetermined threshold value.

To eliminate continuous slewing of the gain controller, the M_δ parameter is delivered to the individual channels through hysteresis and threshold networks. The threshold networks limit the gain changing range to four distinct levels, and the hysteresis tends to prevent cycling between levels. (Only three of the four gain levels are used in the pitch axis.)

Under normal circumstances the use of a clean δ signal provides a good correlation function and allows rapid, accurate computation of M_δ. Thus a rather wide variation in M_{δ_S} can be accommodated within the 1.5 second computation time even in the presence of severe gusts or pilot input commands. In the event that M_{δ_S} is changing very rapidly over a large range, such as in a powered dive, the error signal will hold the control logic in until the error is below the predetermined level. An activity monitor detects the output of the actuator model and shuts off the computer if a predetermined operating time of approximately 30 seconds is exceeded.

The M_{δ_C} computation is conducted in a dual circuit, and the outputs of the two integrators are compared to provide in-flight monitoring and fault detection. Should a fail be detected, the adaptive gain changer is disengaged and the low manual gain is engaged. This is accomplished by interrupting a solenoid holding circuit to the ADAPTIVE/NORMAL solenoid held switches, thus causing reversion to the NORMAL position. The ADAPTIVE function may only be engaged by first selecting the LOW gain setting. Therefore, the LOW gain is automatically selected on detection of a fault in the adaptive gain changer. The electronics are packaged in the Adaptive Gain and Stall Warning Computer both of whose functions are dual rather than quadruplex. See Supplement 2 for details of the stall warning computation.

Table XIV provides the flow and horsepower demands upon the stabilator actuator for both the production power cylinder and the SSAP for the limits of stabilator travel which can be generated. It can be seen that the average flow and horsepower requirements using the planned duty cycle will be rather small. Further evaluation will be performed with the new gain changer design utilizing both manned and unmanned simulation studies.

TABLE XIV

SURFACE ACTUATOR POWER REQUIREMENTS RESULTING FROM M_δ INTERROGATION SIGNAL

F-4 Stabilator Actuator						
δ^0 (P-P)	Q (Peak)	Q (Avg/Cycle)	Q(Avg/Period)	HP (Peak)	HP (Avg/Cycle)	HP (Avg/Period)
0.15^0	1 GPM	0.64 GPM	0.057 GPM	1.75	1.05	0.10
0.25^0	1.66 GPM	1.055 GPM	0.096 GPM	2.9	1.85	0.17
SSAP Surface Actuator						
0.15^0	0.97 GPM	0.62 GPM	0.05 GPM	0.92	0.59	0.05
0.25^0	1.61 GPM	1.03 GPM	0.095 GPM	1.40	0.92	0.09

d. Ground Adjustable Parameters

Considerable flexibility is being incorporated into the SFCES to enable variations in loop gains and time constants by changing discrete components. This approach was employed rather than in-flight variation for all system parameters except the selectable fixed gains deemed desirable to enhance aircraft handling qualities.

The rationale for this choice, other than the fact that it will result in minimizing rather than maximizing the amount of required flight test time, is the resulting improvements in safety, reliability and packaging. System safety aspects are improved because any change in system characteristics can be checked on the mobile ground test facility prior to use in flight. The mobile ground test facility, which provides a complete closed loop test capability utilizing analog computers to generate aircraft dynamics, is discussed in Section IV. In addition, the pilot is prevented from misapplication of parameter selection by the best possible technique, the absence of the choice.

If flight testing indicates the need for different loop gains or time constants, necessary modifications can be incorporated. However, the SFCS has been designed to obtain handling qualities compatible with the best handling qualities criteria available at this time and numerous or frequent changes to the loop gains and time constants should not be required.

19. SIDE STICK CONTROLLER

The Side Stick Controller (SSC) development study established design and performance criteria which were incorporated into the SSC requirements for the SFCS. Use was made of existing side-stick technology in a concerted effort to obtain the simplest design to meet the requirements of the SFCS program. Thorough evaluations of current SSC designs applicable to the SFCS were made and, based on these evaluations, an in-house mock-up was constructed and installed in an F-4 cockpit mock-up for further study by engineering and pilot personnel. Particular attention was paid to the following items and features considered relevant to program requirements:

- Location of SSC in the cockpit.
- Grip configuration.
- Arm rest configuration.
- Control motion geometry.
- Neutral position adjustment range.
- Control travel limits.
- Artificial feel characteristics.

Appendix VIII provides a detailed account of the SSC development study.

a. Evaluation of Current Designs

Technical reports, pilot reports, and other available data were reviewed. Three current SSC programs were reviewed in depth. AFFDL Technical reports and MCAIR pilot evaluations of the USAF Fly-By-Wire B-47 aircraft and the Cornell University B-26 provided valuable engineering and human factors data. However, the mechanical configurations of these controllers was not adaptable to the SFCS installation. The Aerospace Research Pilots School (ARPS) F-104D SSC design data and pilot evaluations proved to be the most useful in providing a starting point in establishment of the SFCS SSC design criteria. The cockpit location and general configuration of this unit was compatible with anticipated SFCS needs. Control motion geometry and feel forces appeared to be acceptable to most pilots, and the F-104D performance envelope is similar enough to the F-4 to consider the pilot evaluations pertinent.

b. MCAIR SSC Mock-Up

MCAIR designed and fabricated a SSC mock-up to be used in evaluating the F-4 cockpit interface, establishing SSC feel forces and ranges, and arm rest, grip, vernier and trigger configurations. The result of evaluations using this mock-up helped provide the design criteria for the SSC.

c. SSC Development Mock-Up

Lear Siegler, Inc., the SSC Supplier, provided a development mock-up representative of his proposed design and upgraded this mock-up to maintain a representative configuration for engineering evaluation as the design effort progressed. This effort has helped to establish feel forces, damping rates, breakout forces, grip shape, vernier and trigger forces, arm rest contour, grip neutral lock control, and grip neutral adjust controls. Table XV contains the values presently planned for some of the above parameters, and Figure 43 illustrates the most recent SSC development mock-up.

TABLE XV
ARTIFICIAL FEEL FORCES AND GRIP GEOMETRY

	Pitch Axis	Roll Axis	Pitch Vernier	Trigger Switch
Breakout Force	1.75 lb	1.75 lb	3.5 oz	15 lb
Total Force at Full Travel	9.45 lb	3.55 lb	7.5 oz	N/A
Damping (In. Lb/Rad/Sec)	12 △1	2 △1	N/A	N/A
Travel	20° Fwd and Aft	15° Inb'd and Outb'd	$\pm 90^{\circ}$	N/A
Grip Neutral Adjust	$10^{\circ}30'$ Fwd $7^{\circ}30'$ Aft	5° Inb'd and Outb'd	N/A	N/A
Grip Neutral Adjust Knob Location	Outb'd of Grip	Inb'd of Grip	N/A	N/A

△1 At Ambient Room Temperature

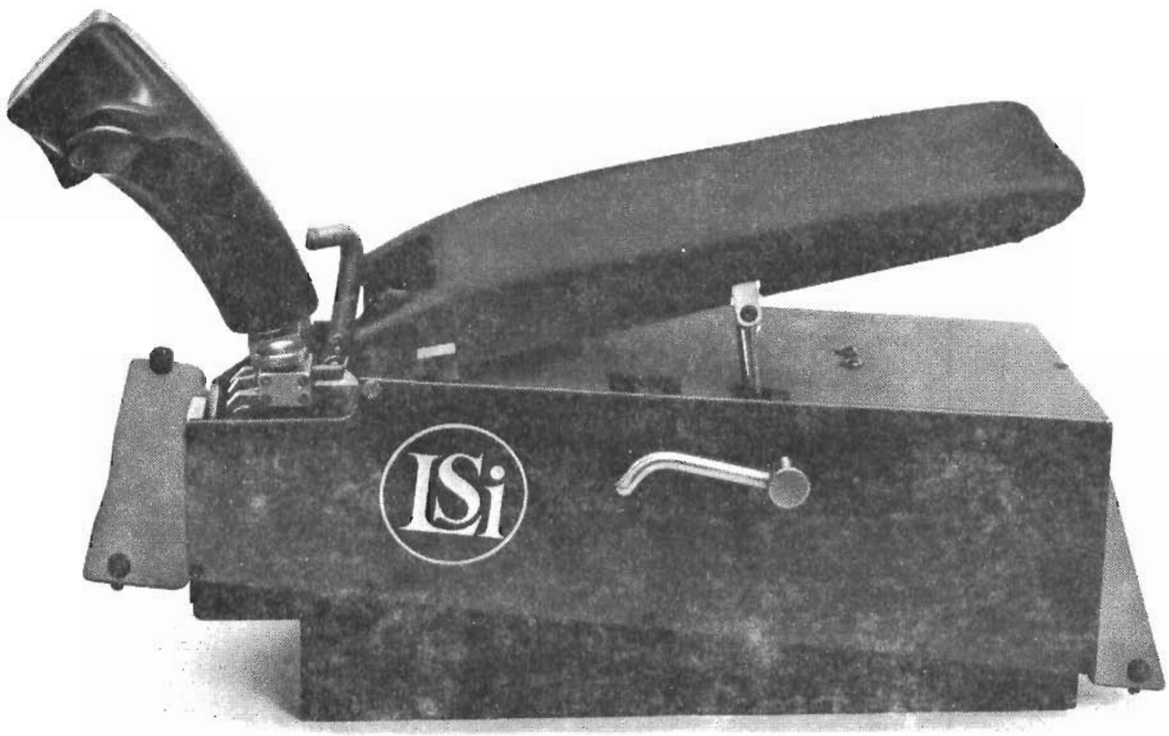


FIGURE 43
SSC DEVELOPMENT MOCKUP

20. TRIM SYSTEM

a. System Description

In the standard F-4 with a mechanical control system, the trim actuators in the pitch, roll, and yaw axes are used to reduce the stick and pedal forces to zero at the aircraft attitude and acceleration desired by the pilot. In this system the stick and pedal positions directly reflect the surface positions. The SFCS FBW trim system operates differently since there is no mechanical link between the stick or pedals and the control surfaces. Bias voltages are inserted by the trim wheels to provide control surface deflections for the flight path desired by the pilot in conjunction with zero stick and pedal forces. The trim voltages in the roll and yaw axes command proportional control surface deflections with no change of rudder pedal or stick position. With the electrical back-up mode selected, the pitch trim control commands proportional control surface deflection as do the roll and yaw trim controls without changing the stick or pedal position. With the Normal mode engaged, the SFCS reacts to a pitch trim input similarly to a force signal from the stick transducer and thus commands the SFCS to generate a normal acceleration and pitch rate proportional to the trim input.

Trade studies were conducted to determine the proper implementation for the SFCS trim systems. Mechanical trim is to be employed in the MBU mode and electrical trim is to be used for both the EBU and Normal Modes of FBW. Considering the results of these studies, the following implementations are planned.

(1) Mechanical Trim

A trim actuator, as discussed in Section V, is to be used in the longitudinal feel system to enable the pilot to trim out the stick forces in the MBU mode. Mechanical trim will be controlled by the beep type trim switch located at the top of the center control stick. The actuator changes the position of the feel spring cartridge relative to aircraft structure, which causes the stick deflection to vary as desired for zero stick force.

Roll axis trim is not required since a mechanical back-up mode is not used in the roll axis.

Yaw axis trim could have been used with the MBU mode but was not considered necessary. Requirements for significant steady state rudder deflections exist only with asymmetric configurations and during de-crab maneuvers for landing, and the pilot can hold the force necessary at such times.

(2) Electrical Trim

Pitch, roll and yaw electrical trim controls will be located on the SFCS trim panel mounted in the left consoles of both cockpits. The factors contributing to the decision were:

- o The top of the stick grip would have to be enlarged in order to provide room for two sets of quadruple switches.
- o Location of the trim controls on the left console allows identical trim mechanizations for the front and aft cockpits.
- o Pilot trimming motions are identical whether aircraft control is being affected from the side stick or the center stick.

Proportional trim is planned to be used, rather than beep trim for the following reasons:

- o Use of proportional trim wheels eliminates the need for quadruplex integrators in the SFCS electronics with their inherent tracking problems.
- o Proportional trim wheels provide control of trim rate.

Several methods of inserting the electrical trim inputs into the control loops were evaluated. An early implementation had the trim signals applied to the secondary actuator feedback loop. However, it was found that this mechanization resulted in significant transients when adaptive gain changes occurred during turning flight. The trim inputs were then moved to a point upstream of the adaptively changed gain. In each axis of control, the electrical trim inputs sum with the filtered pilot command input signal.

The electrical trim authorities originally selected for FBW were:

- o Pitch Normal Mode: The amount of trim necessary to sustain a 60 degree bank angle turn at 360 Knots TAS and sea level;
- o Pitch EBU Mode: Three degrees leading edge up to fourteen degrees leading edge down stabilator position;
- o Roll Normal and EBU Modes: 16.5% of full lateral surface authority; and
- o Yaw Normal and EBU Modes: 33% of full directional surface authority.

b. Trim Modifications

As a result of simulations, longitudinal trim modifications were devised for each mode of the SFCS. Pilots inadvertently used the mechanical trim switch located at the center stick when in the FBW modes. This action led to a mistrimmed MBU condition. To solve this problem, the center stick trim button was disabled during FBW operation. The longitudinal EBU trim authority was found excessive. At some flight conditions, maximum g could be commanded with partial motion of the thumbwheel trim. The pilots felt that a reduction to $\pm 1^\circ$ stabilator for clean and $\pm 5^\circ$ for PA was necessary. In addition, the pilots felt that the capability to trim into a 60° banked level turn at 360 Knots at sea level was not necessary. This lead to a reduction of the Normal mode longitudinal trim by one half.

The roll trim authority was reduced by one half for all modes since the pilots felt 33 1/3% surface authority was too sensitive for small roll rate corrections. No problem was found with the directional trim during the simulation.

21. ELECTRICAL BACK-UP CONTROL MODE

a. General

Implementation of the Electrical Back-Up (EBU) control mode for each of the three control axes has been investigated. The two basic approaches presented in Figure 44 were considered. These operate as follows:

System 1 - EBU control mode normally disengaged. When the EBU control mode is commanded ON, signals through the computation paths are faded out and the control inputs faded into the input of the signal selection device (SSD).

System 2 - EBU control mode normally engaged. When the EBU control mode is commanded ON, signals through the computation paths are faded out leaving only the control inputs as inputs to the SSD.

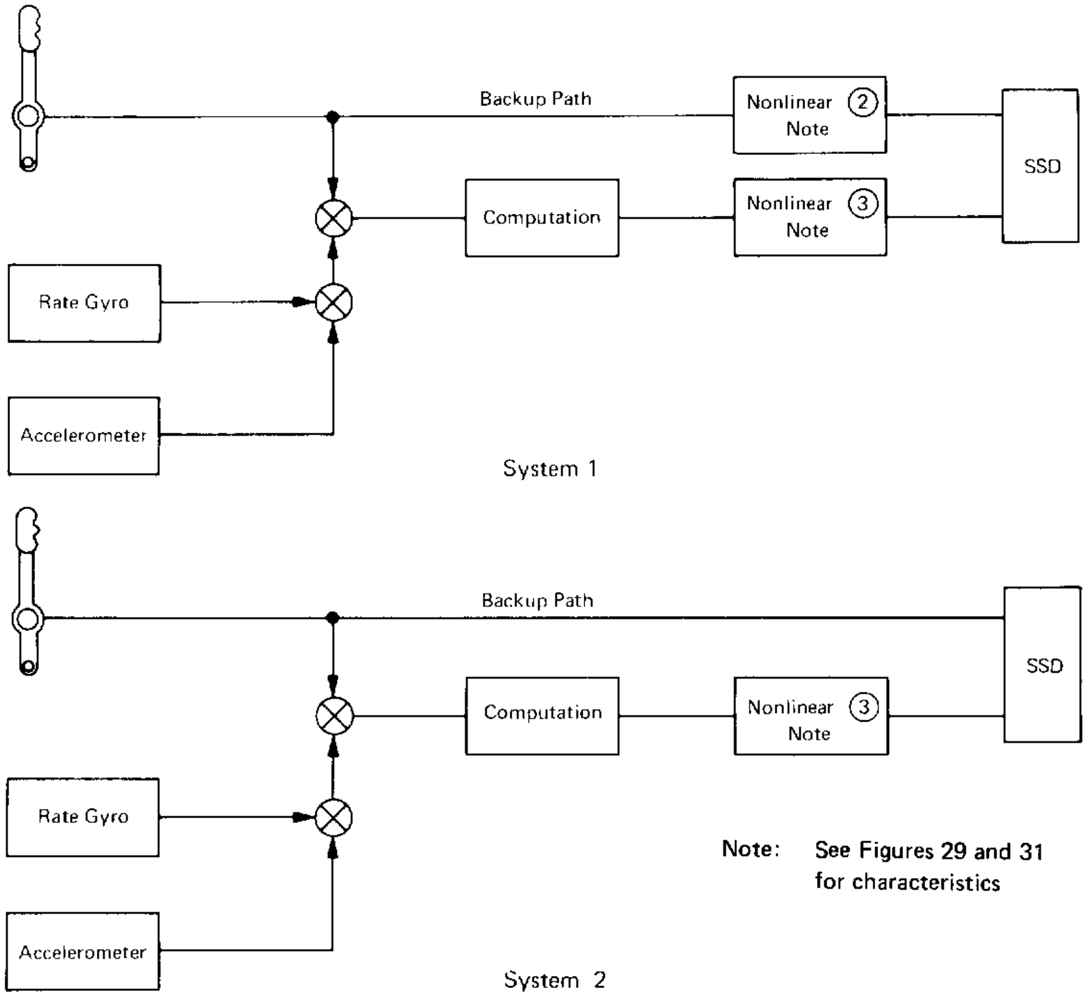


FIGURE 44

ELECTRICAL BACKUP CONTROL CANDIDATES

Handling qualities studies have shown that for the roll and yaw axes, System 2 is not only acceptable but desirable. In the pitch axis, however, handling qualities are degraded if the EBU control mode is active full time. Therefore, System 1 is implemented in the pitch axis and System 2 is implemented in the roll and yaw axes. Both systems employ the SSD during EBU control mode operation to reduce the probability of nuisance disengagements, as discussed below.

b. Discussion

Tradeoffs with respect to maintainability, reliability, power requirements and cost are essentially nonexistent for System 1 and System 2. The only significant difference between these systems is in the addition of a switching circuit required to turn on the EBU control mode in System 1.

A comparison of the failure rates for the two mechanizations would provide an indication of system maintainability and reliability. A cursory study indicated that these would differ by less than one failure per million hours in a system with a series failure rate of approximately 2500 failures per million hours. The difference in power requirements for each mechanization would be measured in milliwatts in a system requiring hundreds of watts per channel. The choice of system implementation, therefore, was dictated by system performance.

As discussed in Supplement 2, the pitch, roll and yaw axes performance analyses resulted in control system implementations which would meet the handling qualities criteria. These performance analyses indicated that the preferred implementation for the EBU control mode was System 1 for the pitch axis, and System 2 for the roll and yaw axes.

The EBU control mode signals and the Normal mode signals are processed by the same SSD in both system mechanizations. This technique was chosen when the signal accuracies (monitor thresholds) required to meet the transient performance requirements following a failure were weighed against the possibility of actuator nuisance disengagements during EBU control mode operation. Each side stick controller channel was required to track within one percent, and the total channel input tracking accuracy at the servo amplifier was required to track within one percent. It was obvious that total servo command tracking tolerances would exceed 1 percent if reasonable electronic tolerances were considered. The obvious solutions to this dilemma were, a signal selection device, higher actuator monitoring levels, or reduced actuator bandwidths.

The possible solutions were further reduced by the fact that the actuator monitoring levels for both the electrohydraulic and electromechanical actuators were already set at near the maximum pressure and rpm for the respective units. To increase to the maximum would have resulted in only a token improvement.

A reduction in actuator bandwidth may have resulted in increased nonlinearities and associated stability problems for a system with the broad closed loop bandwidth exhibited in Supplement 2. System bandwidth must be maintained if desired performance benefits are to be realized.

The employment of the SSD, already existing within the SFCES, offered the best solution to the tolerance dilemma for EBU control mode operation. Its only drawback is the slight decrease in EBU control mode reliability because of the addition of the SSD.

c. The pilot may engage the EBU mode by:

(1) depressing the paddle switch on the center stick.(affects all control axes simultaneously)

(2) depressing the trigger switch on the SSC. (affects all control axes simultaneously)

(3) depressing the desired switch(es) on the Master Control and Display Panel. (one switch for each axis of control)

The pilot may reengage the NORMAL by depressing the desired switch(es) on the Master Control and Display Panel.

d. Conclusions

As a result of the studies summarized above, the EBU mode is presently planned to incorporate the following features:

- The roll and yaw axes will have a direct electrical signal path functioning at all times, and the FBW signals will be faded out upon EBU engagement.

- In the pitch axis the FBW signals will be faded out and the EBU signals will be faded in when EBU is engaged.

- In all axes, the EBU signals will be applied to the secondary actuator through a SSD, to reduce the probability of nuisance disengagements without requiring unrealistically tight tolerances on the electronics components.

Further discussion of the SFCS performance while using EBU mode is presented in Supplement 2.

22. MECHANICAL BACK-UP

a. General

In Phase IIA, the SFCS will have fly-by-wire (FBW) controls, electrical back-up (EBU) control, and mechanical back-up (MBU) control available, not simultaneously, in the pitch and yaw axes. The MBU in the pitch and yaw axes is intended to provide back-up control in the event that problems occur in the FBW control system. It also permits the part time use of manual mode while the integrity of the FBW control system is established in early development flights. MBU will not be utilized in the roll axis since the airplane can be brought home and landed without the roll axis should FBW control not be available.

The presently planned mechanical portions of the MBU and FBW control systems are shown schematically in Figure 45 for the pitch axis. The yaw axis concept is analogous. The combined MBU and FBW control systems differ from the production F-4 control system in that the feel-trim system has been simplified and moved to the forward fuselage and a secondary actuator and Mechanical Isolation Mechanism (MIM) have been added. The MIM provides transition between FBW and MBU control modes.

When confidence in the safe performance and reliability of the FBW control system is established, the MBU will be removed. The MIM and control linkages between the center stick and MIM will be removed. The feel system will remain in the forward fuselage. The control

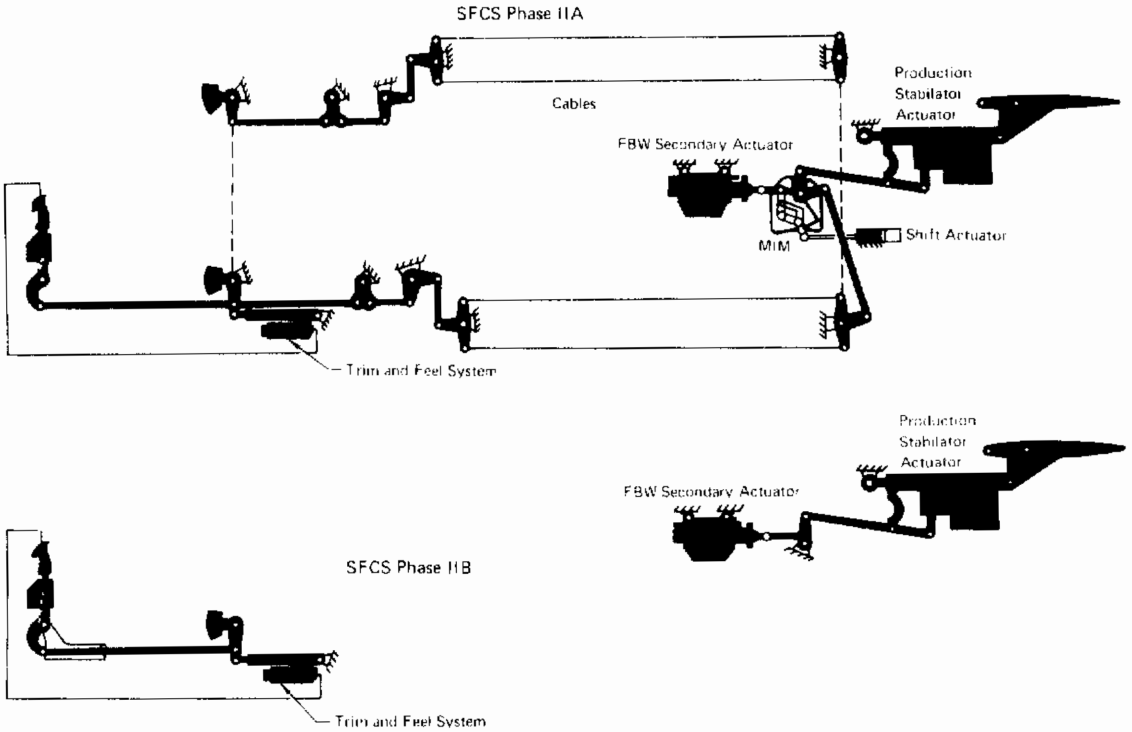


FIGURE 45

MECHANICAL LINKAGE SCHEMATIC - PITCH AXIS

system would be as shown schematically in Figure 45. Subsequent flights, in Phases IIB and IIC of the SFCS program, will utilize FBW control exclusively without MBU.

b. Description of Operation

Operation of the MBU mode is dependent on the functional characteristics of the control linkages, feel system, MIM, and MIM shift actuator. A brief description of the intended implementation of the different portions of the control system is contained in the following paragraphs.

(1) Mechanical Linkages

The mechanical linkages between the center stick and MIM in the pitch axis and between the rudder pedals and MIM in the yaw axis consist basically of the mechanical control systems of the production F-4. Some minor changes to the linkages will be required in the vicinity of the MIMs.

(2) Feel System

In the yaw axis the feel system consists of preloaded springs. The springs provide a breakout force and a force gradient as a function of pedal deflection. Unlike the production F-4, only a single force gradient is provided. MBU mode trim is not included in the yaw axis.

In the pitch axis the feel system consists of preloaded springs, an electrical trim actuator, an eddy-current damper, and an emergency spring cartridge. The springs provide a breakout force and a force gradient as a function of center stick motion. Unlike the production F-4, the feel system force gradient versus deflection is not modified as a function of dynamic pressure. The damper provides stick force proportional to stick rate, and helps prevent stick free oscillations of the center stick. The emergency spring cartridge allows center stick motion in the event of a jammed damper. The trim actuator is used by the pilot to adjust the zero force position of the center stick during MBU operation only.

(3) Mechanical Isolation Mechanism (MIM)

The MIM is the device which provides transition between MBU and FBW modes in the yaw and pitch axes. The MIM has the following characteristics.

- The MIM is a mechanical gain changing mechanism made up of a torque tube, bellcranks and links.
- The MIM design permits independent operation by the output of the secondary actuator of the FBW input bellcrank without influencing the motion of the surface actuator while in MBU mode. This allows the secondary actuator to operate and be

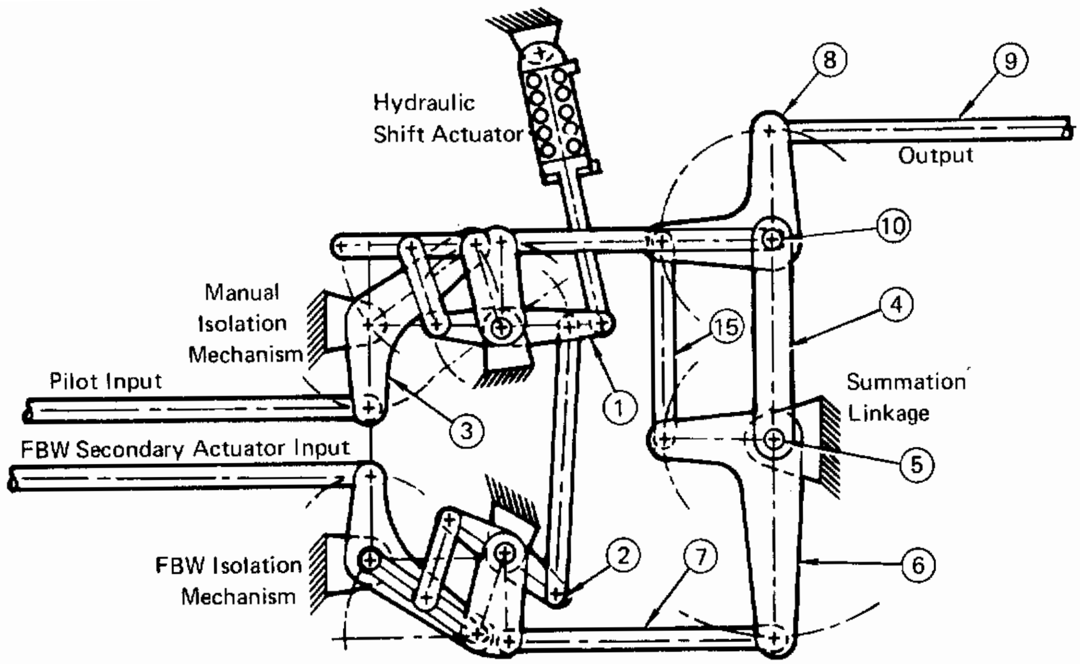
monitored while in the MBU mode.

- The MIM design completely decouples the mechanical controls during FBW operation so that the flying qualities of the FBW system are not degraded.

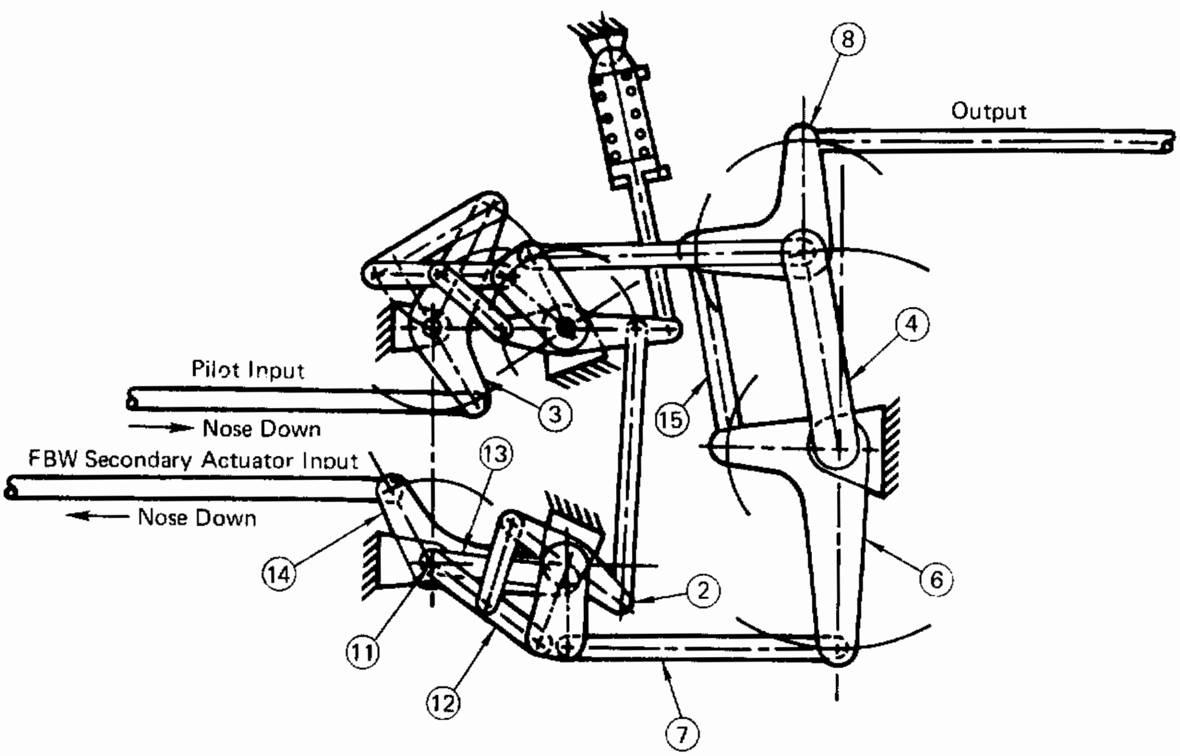
- The MIM is composed of two identical parallelograms, as shown schematically in Figure 46, connected to a shift actuator so that as one parallelogram is phased in the other is phased out. In Figure 46, the shift bellcranks (1) and (2) have been moved to the manual mode of operation. In this position the manual input bellcrank (3), which is shown at mid-travel, will move bellcrank (4) about pivot point (5) while bellcrank (6) is held fixed in place by link (7). When bellcrank (6) is held while bellcrank (4) moves, bellcrank (8) will assume a position as shown in Figure 46. The linear motion supplied to the output control rod (9) is identical for all practical purposes to motion that would occur if pivot point (10) in Figure 46 is held and bellcrank (6) is moved through the same arc as bellcrank (4). The lower part of Figure 46 shows the MIM still in MBU mode except the pilot has selected full nose down position of the center stick. In this position the secondary actuator (SA) also goes to aircraft nose down position and the linkage arrangement at the FBW bellcrank shows that motion of this bellcrank does not produce motion of link (7) and bellcrank (6) is held motionless. The reason for this is that the shift bellcrank (2) has moved pivot point (11) of link (12) and (13) over the pivot point of the FBW input bellcrank (14) and motion of bellcrank (14) is powerless to move point (11). Bellcranks (4), (6) and (8) plus link (15) comprise the summation linkage of the MIM.

- In the FBW mode the MIM shift actuator shifts the two identical link parallelograms so that secondary actuator motion drives the MIM output while pilot input motion is shifted out. Operation of the MIM from secondary actuator motion to MIM output is essentially the same as that described for pilot input motion to MIM output when in the MBU mode.

- The mechanical gains of the MIM can be represented by two dependent mechanical multipliers, A and B, as shown in Figure 47. The outputs of the two multipliers are summed and applied to the output of the MIM which drives the main control valve (MCV) on the surface actuator. The inputs to the MIM are a secondary actuator position input to multiplier A, the MBU position input to multiplier B, and the gain control linkage input to multipliers A and B. As a function of the gain control linkage position, the gains of multiplier A and B vary linearly from zero to one or one to zero in such a manner that the sum of the gains of multipliers A and B is always equal to one. This also is illustrated in Figure 47. The input linkage which controls these gains is driven by the MIM shift actuator which is controlled by a switch in the cockpit. Transition time from 100 percent FBW to 100 percent MBU

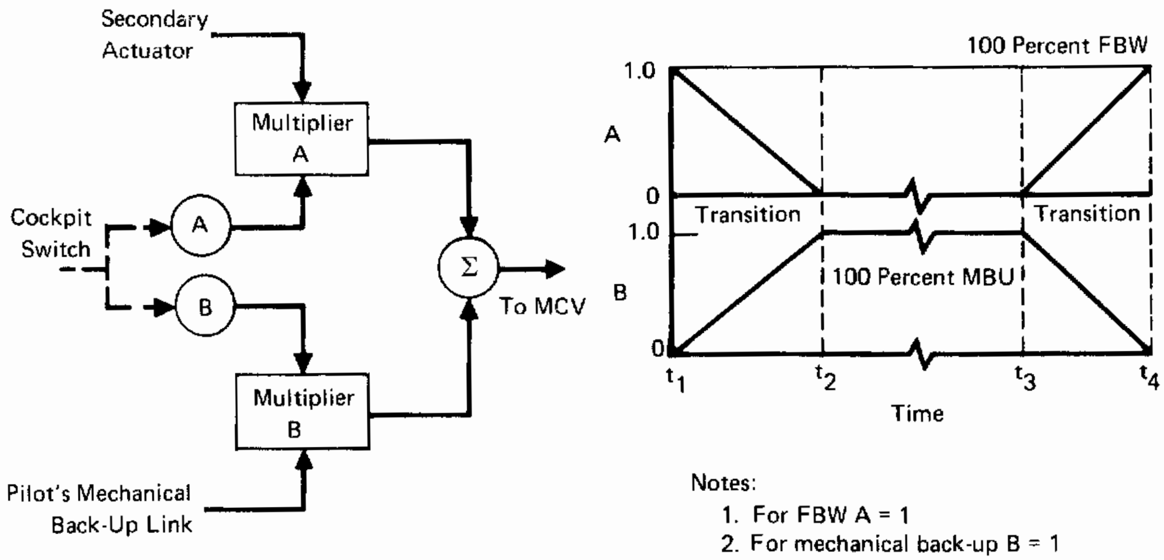


Full Manual Shift Mode - Controls at Mid Travel



Full Manual Shift Mode - Controls at 30° Nose Down

FIGURE 46
MECHANICAL ISOLATION MECHANISM (MIM) SCHEMATIC



(a) MIM Schematic Diagram

Notes:

1. For FBW A = 1
2. For mechanical back-up B = 1
3. During transition, see curve B above
4. A + B always equal to 1

FIGURE 47
MECHANICAL ISOLATION MECHANISM OPERATIONAL DIAGRAM

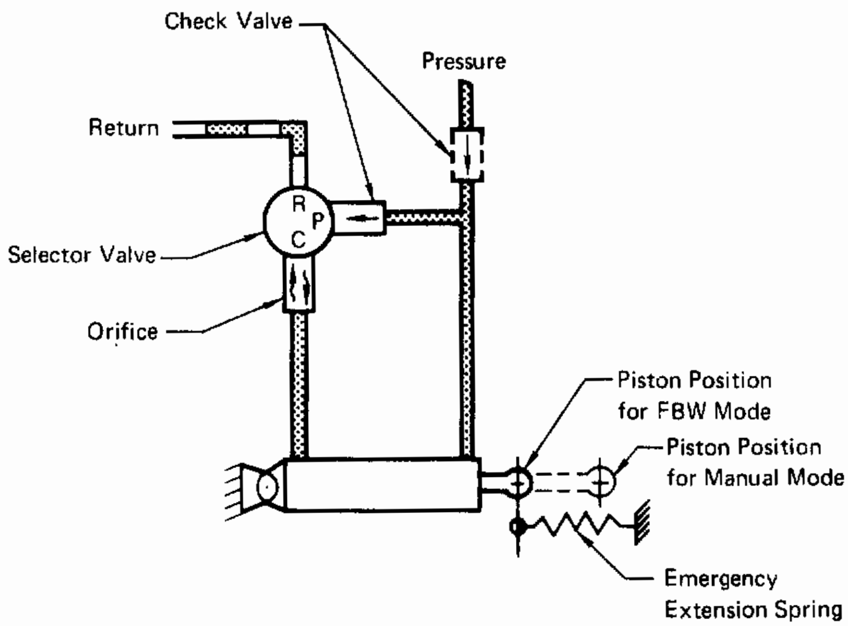


FIGURE 48
MIM SHIFT ACTUATOR HYDRAULIC SCHEMATIC

control or vice versa is approximately 2.5 seconds.

(4) MIM Shift Actuator

The hydraulic schematic of the MIM shift actuator is shown in Figure 48. A single orifice filtered in both directions controls the rate of both extension and retraction of the actuator. Dual extension springs, which are provided external to the cylinder to return the hydraulic actuator to the manual mode, provide the force equivalent of 595 psi hydraulic pressure for the longitudinal MIM and 664 psi for the directional MIM.

c. Safety Considerations

Operational characteristics as they relate to safety are relatively straightforward for the feel system and mechanical linkages between the center stick and MIM. An emergency spring cartrigde allows the center stick to displace should the eddy-current damper jam. The probability of jam type failures in the mechanical linkages is slight. Specific characteristics of the MIM and MIM shift actuator associated with safety are more numerous, however, and are discussed in ensuing paragraphs.

(1) MIM

The presence of the MIM in the control system gives rise to potential loading and transient problems, both of which occur only during transition between modes. High loads can exist on the secondary actuator due to the mechanical gain across the MIM during transition from FBW to MBU. Transients might occur during transition due to the lack of synchronization between MBU linkage and the secondary actuator.

(a) Secondary Actuator Loading

In the yaw axis, conditions can exist in which the MIM backdrives the secondary actuator. Depending on the conditions, the backdriving may be accomplished by either the shift actuator or by air loads when the rudder surface actuator is stalled by air loads. This cannot happen in the pitch axis because the production F-4 pitch surface actuator cannot be stalled by air loads.

One condition under which backdriving occurs involves a stalled right rudder, a continued right rudder effort by the pilot, and a shift from FBW mode to MBU mode. Under this condition, the MIM output cannot move because the MCV is stalled. The mechanical control linkage, neglecting cable stretch, will not back off because of the pilot effort. The result is that the secondary actuator is backdriven with relatively high loading. Backdriving of the secondary actuator would also occur if hydraulic pressure were lost when a right rudder stall is accompanied by a right rudder pilot effort.

The secondary actuator design is consistent with the loads and backdriving rates which would occur under the above conditions. Thus, no problems are anticipated even in the unlikely event that one of the above conditions were to occur.

(b) Transients

When operating under either 100 percent MBU or 100 percent FBW control, the secondary actuator position and the MBU linkage position are unrelated. Therefore, transfer from one mode to the other could cause transients. The severity of these transients can be controlled by controlling the rate at which the transfer is performed.

The rudder is normally at or near zero when in either MBU or FBW modes. Therefore, transients due to transfer from FBW to MBU and vice versa are considered acceptable, and a transition time of 2.5 seconds has been established.

In the pitch axis, the SFCS control laws provide flight path control, and the stabilator is maintained at a position as required to provide zero error command into the servo amplifier, which may or may not correspond to stick position. When using the MBU mode, the stabilator deflection is directly proportional to center stick position, and the pilot uses the trim actuator to establish the zero force position of the stick. Therefore, transition between the SFCS and MBU modes can result in large stick force changes in the absence of pilot action. In addition, while on MBU, the SFCS system cannot be allowed to operate in the Normal (NSS) mode since this could result in a hardover position of the secondary actuator. To prevent this, while on MBU, a switch on the MIM signals the SFCS to revert to the TOL mode. The SFCS mode may be selected by placing the SFCS-MBU switches on the left console in the SFCS position. Figure 27 shows the location of these switches. For further explanation see Paragraph 13.

Various schemes for minimizing pitch axis transients during transition were investigated. These schemes all involved added electronic and/or mechanical complexity in order to slave the MBU position to the secondary actuator position or vice versa. The following specific solutions were evaluated and discarded:

- During FBW control, the pitch trim actuator could be used to slave the position MBU linkage to the position of the secondary actuator. This technique would require the addition of electronics and would result in control column movement during FBW operation. Control column movement would interfere with or at least degrade hand dling quality evaluations.

- During MBU control, the secondary actuator position might be synchronized to the surface actuator position.

- During transition the secondary actuator could be slaved to the surface actuator position or the SFCS can be operated in the TOL function or MBU modes.

The severity of the transient with the pilot in the loop was investigated on the simulator by MCAIR. This investigation showed that the pilot could control and maintain transients to a satisfactory level quite easily without the need for any synchronization, provided the transition time was in the order of 2.5 seconds or greater. It was found necessary, however, to disable the mechanical trim switch on the center stick when in FBW mode. This is required to eliminate inadvertent pilot inputs to the mechanical trim switch while in FBW mode and the transients which would be generated upon transfer to MBU. The necessity for disabling the MBU trim is evident in Figure 49 where the transients for MBU to FBW transition are shown with and without inadvertent mistrim.

Since transition between FBW and MBU is an R&D peculiar problem, the development costs and complexity of synchronization schemes could not be justified.

(2) MIM Shift Actuator

The MIM shift actuator is a double acting single-ram hydraulic cylinder. Pressure from the Utility hydraulic system is utilized by the MIM shift actuators in both the yaw and pitch axes. To help obtain operation consistent with the safety and reliability requirements of primary control systems, the following functional characteristics have been incorporated in the MIM shift actuator.

- The MIM shift actuator is spring loaded to return to the MBU mode in the event of an hydraulic pressure failure or any electrical failure which results in loss of electrical power to the MIM solenoid.

- During flight, hydraulic pressure is maintained to hold the shift actuator in either the MBU or FBW mode, as selected.

- Transition time from MBU to FBW, or vice versa, is approximately 2.5 seconds.

- In the event of an hydraulic failure, the transition time from FBW mode to MBU mode is approximately 3.5 seconds.

- The FBW mode cannot be energized unless either the PC-1 or PC-2 hydraulic system is pressurized. This avoids jamming the MIM output when power is not available to operate the stabilator during ground operation. This feature will be deleted when the MIM is deactivated for Phase IIB.

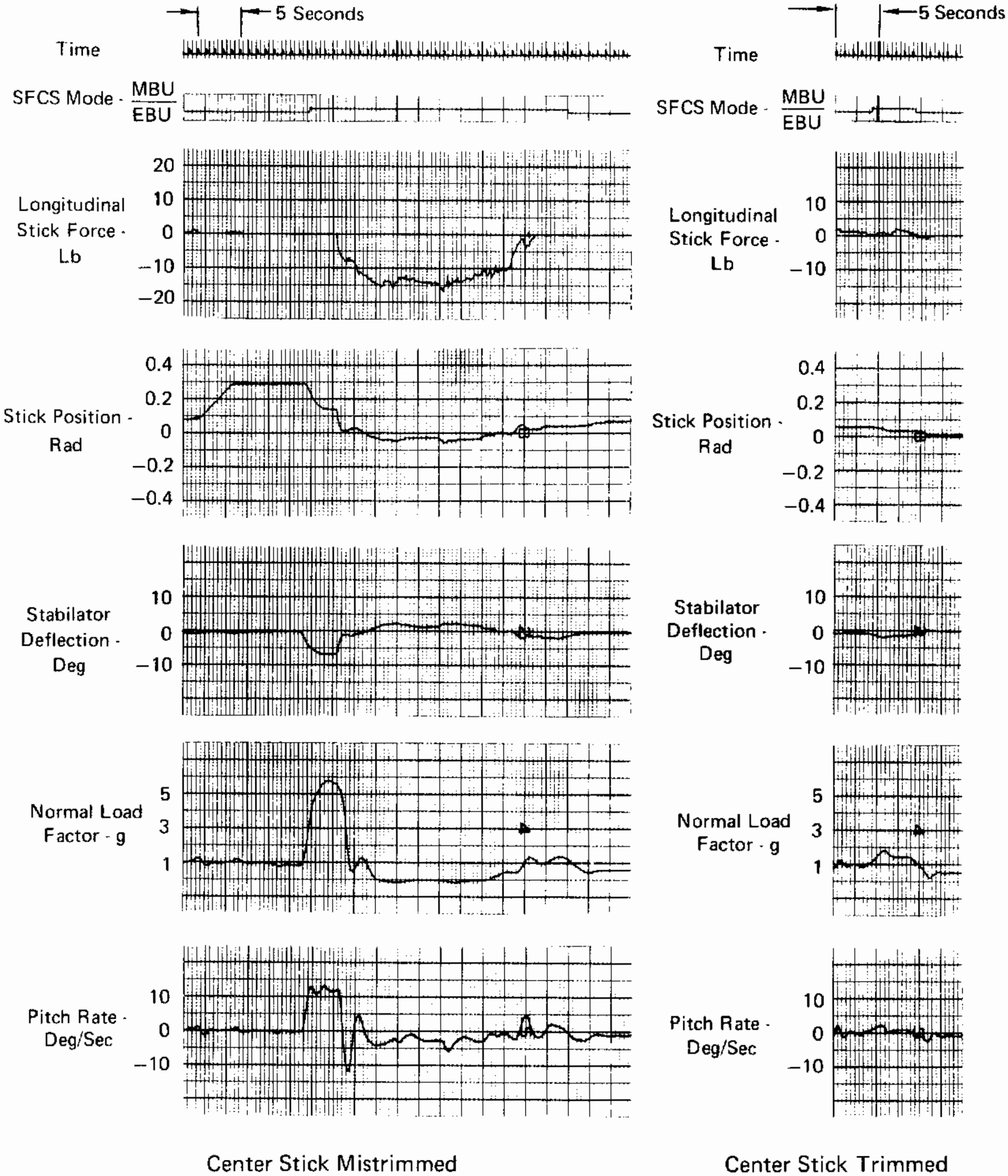


FIGURE 49

TRANSITION FROM SFCS ELECTRICAL BACK-UP MODE TO MECHANICAL BACK-UP MODE

23. INTEGRATED ACTUATOR PACKAGE STUDY

A study was performed to determine the degree of redundancy required for the Survivable Stabilator Actuator Package (SSAP). In this SSAP study, redundancy requirements and the effect of their interface with other subsystems of the flight control system were given primary consideration. The study included the following types of redundancy:

- Full-time duplex with no back-up
- Full-time duplex with central hydraulic back-up
- Full-time triplex with no back-up
- Full-time triplex with central hydraulic back-up
- Full-time quadruplex with no back-up
- Additional redundancy concepts

The terms duplex, triplex, and quadruplex are based on the number of motor pumps in a given configuration.

The extent and results of the study are summarized in the following paragraphs, and a more complete discussion of the study can be found in Supplement 3.

a. Background

In the initial stages of the study, both servopump and soft cutoff pump concepts were considered. The servopump concept was of special interest because of superior thermal characteristics. However, the capability of the servopump concept to meet static and dynamic stiffness requirements comparable to those of the production F-4 stabilator power control cylinder had not been demonstrated experimentally at the time of the study. It was, therefore, concluded that the technical risk associated with the servopump concept was too high for the SFCS program.

Since redundant configurations with as many as four motor pumps were considered in the study, it was necessary to provide four sources of electrical power. For purposes of this study, AC power sources included the left hand bus, right hand bus, an hydraulic driven electric generator (HDEG), and a fourth source normally connected to the left hand bus which switched to the right hand bus when power on the left hand bus was lost. With the expection of the fourth AC source, motor pump loads were not interconnected between left and right hand busses.

b. Discussion

The configurations selected for the trade study are those shown in Figure 50. These configurations are based on the soft cutoff pump concept which utilizes a master control valve. As shown in Figure 50 concepts with from two to four pistons and from two to four motor pumps were considered. The use of back-up hydraulic systems operating through hydraulic switching valves was also evaluated as were flow sharing concepts where the flow demand for one piston is supplied from two hydraulic systems.

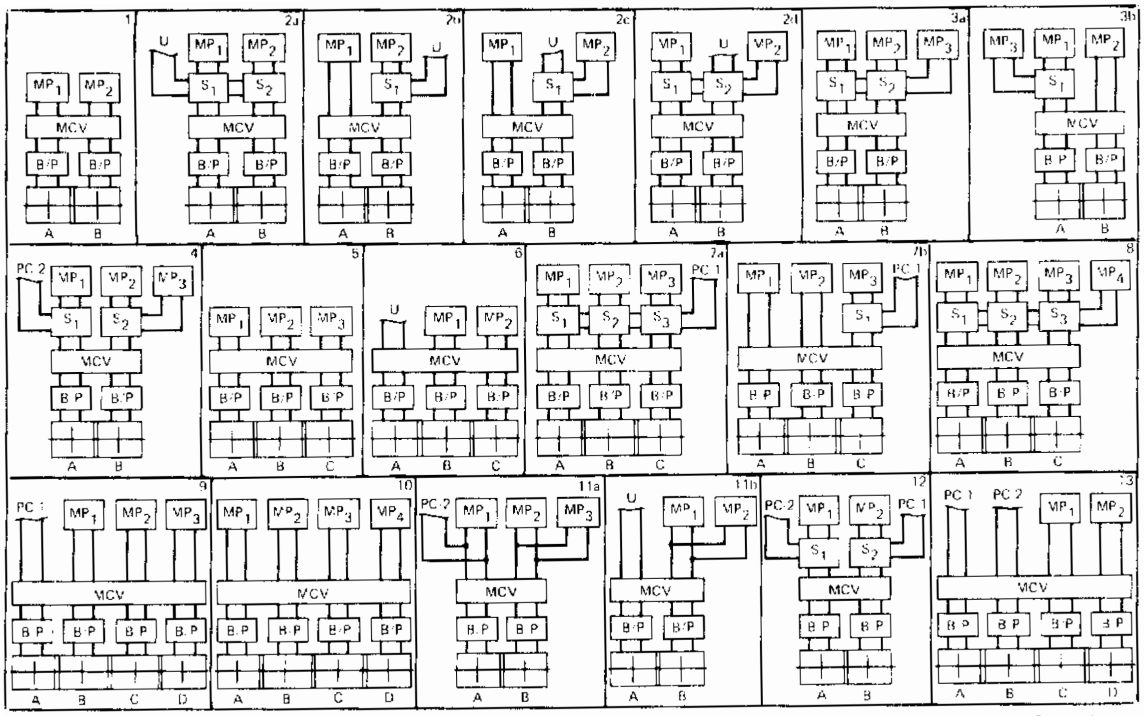
In evaluating the configurations, thirteen technical and functional characteristics were considered as indicated in the list shown in Table XVI. Of the nineteen configurations evaluated, fourteen were rated unacceptable on the basis of one or more requirements, primarily performance, reliability, or compatibility with the aircraft electrical system.

Ordinarily an unacceptable performance rating was due to the inability of a configuration to meet dynamic stiffness requirements after two failures. Similarly, many configurations could not meet reliability requirements after two failures and were accordingly rated unacceptable. The configurations which were not compatible with the aircraft electrical system were rated unacceptable when steady state or starting loads were excessive.

The ratings for the five configurations found acceptable were nearly equal as shown in Table XVI. It was, therefore, necessary to impose the additional considerations of cost and technical risk in order to allow a firm choice. Configurations 4 and 11a require a third motor pump and a HDEG and would, therefore, cost more than configuration 12. Configurations 6 and 13 would incur a greater technical risk than configuration 12 because of the imposing task of developing a triple or quadruple valve and actuator.

c. Conclusions

Configuration 12 was considered to represent the lowest cost and the least technical risk of the acceptable configurations. The use of two central hydraulic systems as back-up in configuration 12 gives the increased safety necessary for successful completion of the SFCS program but in no way compromises the future use of IAPs where central hydraulic system(s) back-up may or may not be desired, depending on the overall aircraft layout, mission, and design philosophy. Therefore, configuration 12 was chosen as the basis for the current SSAP design.



A,B,C,D	Main Actuator Pistons	MP_1	Motor Pump - L/H AC Bus	PC 1	Central Hydraulic System (Primary 1)
MCV	Master Control Valve	MP_2	Motor Pump - R/H AC Bus	PC 2	Central Hydraulic System (Primary 2)
B P	Bypass Valve	MP_3	Motor Pump - Hydraulic (Utility) Driven Electric Generator	U	Central Hydraulic System (Utility)
S	Switching Valve Hydraulic	MP_4	Motor Pump - L/H and R/H AC Buses		

FIGURE 50
SSAP CONFIGURATION SCHEMATICS

TABLE XVI
COMPARISON MATRIX OF SSAP DESIGN APPROACHES

	Wt Factor	Configuration 1	2a	2b	2c	2d	3a	3b	4	5	6	7a	7b	8	9	10	11a	11b	12	13
1) Performance	10	D	D	D	D	D	D	D	A	A	A	D	D	D	D	D	A	D	A	A
2) Compatibility with Aircraft Electrical System	10	D	A	D	A	A	D	D	A	D	A	D	D	D	D	D	A	A	A	A
3) Reliability	10	D	A	A	A	A	D	D	A	D	A	A	A	D	A	D	A	A	A	A
4) Envelope	10								A		B						A		A	C
5) Thermal Considerations	5	C	B	C	B	B	B	C	C	C	B	B	C	B	B	C	A	B	B	B
6) Survivability	5	C	C	C	C	C	C	B	A	B	B	A	A	A	A	A	A	B	B	A
7) Safety	3	C	C	C	C	C	C	C	C	C	C	C	C	C	C	C	C	C	A	A
8) Compatibility with Two-Fail Operate Criteria	3	C	C	C	C	C	C	C	A	A	A	A	A	A	A	A	A	C	A	A
9) Compatibility with SFCS Program Schedule	3	A	A	A	A	A	B	B	B	B	B	B	B	C	B	C	B	A	A	A
10) Compatibility with Central Hydraulic Systems	2	A	C	C	C	C	A	A	C	A	B	C	C	A	B	A	C	B	C	B
11) Maintainability	2	B	B	B	B	B	B	B	B	C	B	C	C	C	C	C	C	C	B	C
12) Weight	1	B	B	B	B	B	C	C	C	C	C	C	C	C	C	C	B	A	B	C
13) Steady State Power Requirements	1	D	B	C	B	B	B	C	B	B	B	B	C	B	B	B	B	B	B	B
Total	65	D	D	D	D	D	D	D	168	D	159	D	D	D	D	D	176	D	177	161

Ratings:
- A Excellent, meets all requirements (3 points)
- B Satisfactory, involves only minor compromise (2 points)
- C Acceptable, involves significant compromises (1 point)
- D Unacceptable
- Not rated

24. ACTUATOR MONITORING STUDY

A study was conducted to determine what Secondary Actuator and SSAP parameters should be monitored in order to provide malfunction indications in the cockpit. Study results indicated, as shown in Table XVII that four signals from each secondary actuator and 12 signals from the SSAP should be displayed in the cockpit.

TABLE XVII
MONITORED PARAMETERS OF SFCS ACTUATORS

SFCS Actuator	No. of Signals	Signals Monitored	Receiver	Displayed	Function
Pitch Axis Secondary Actuator	4	ΔP Sensor*	SFCES	a. Master Control Panel	Indicate failure of a secondary actuator element, electrical source, or hydraulic source
LH Roll Axis Secondary Actuator	4	ΔP Sensor	SFCES	Master Control Panel	Indicate failure of a secondary actuator element, electrical source, or hydraulic source
RH Roll Axis Secondary Actuator	4	ΔP Sensor	SFCES	Master Control Panel	Indicate failure of a secondary actuator element, electrical source, or hydraulic source
Yaw Axis Secondary Actuator	4	ΔP Sensor	SFCES	Master Control Panel	Indicate failure of a secondary actuator element, electrical source, or hydraulic source
SSAP (Secondary Actuator)	4	Servo Motor* Tachometer	SFCES	Master Control Panel	Indicate failure of an element in the SSAP secondary actuator or its electrical source
SSAP	2	Pump Output Pressure Low	SFCES	b. Stabilator Motor Panel	Indicate degradation in output pressure of an SSAP motor pump unit
SSAP	2	Switching Valve Position	Stabilator Motor Panel	Stabilator Motor Panel	Indicate position of switching valve
SSAP	2	Reservoir Switch Fluid Level Low	Stabilator Motor Panel	Stabilator Motor Panel	Indicate low fluid level in reservoir
SSAP	2	Pump over Temperature Switch	Stabilator Motor Panel	Stabilator Motor Panel	Indicate pump over temperature in SSAP motor pump

*In Phase II-C Tachometer Signals will replace ΔP Signals in the Pitch Axis

a. See Figure 61

b. See Figure 82

Four differential pressure signals are monitored in each secondary actuator in order to indicate failure in an element or associated energy source. Passive SA failures are detected in the following manner: The servo is run hardover and the outputs of the ΔP sensors are checked. The servo is run into the stops in both directions. When a normal servo element is stalled or runs hardover into the stops a differential pressure is developed and is sensed by the ΔP sensor. A servo element which has a passive failure cannot develop a differential pressure and there will be no ΔP sensor output. The BIT circuitry will detect the absence of the ΔP sensor output and will indicate a failure. In the SSAP secondary actuator, four tachometer signals will be monitored for the purpose cited above and will replace four differential pressure signals in the pitch axis. The hydraulic source (PC or MP), switching valve position, low reservoir fluid level, pump overtemperature, and low pump output pressure will be monitored on each motor pump.

25. BUILT-IN-TEST (BIT)

a. General

The basic decision to utilize built-in-test (BIT) rather than specially designed ground test equipment to determine the operational readiness of the SFCS was made during the conceptual stage of the SFCS program. The BIT provisions are being designed concurrent with SFCS design, and BIT is expected to be verified prior to installation of SFCS equipment on the Iron Bird and the test aircraft. BIT is being designed to be utilized only when the test aircraft is on the ground and is sometimes referred to as "Ground BIT". For safety reasons, the SFCS is being designed so that BIT cannot be operated when the aircraft is airborne. The BIT goals, concept, features and operation are summarized in the following paragraphs.

b. Design Goals

The BIT design goals include the following:

- Capability of detecting failures of the SFCS with a probability of .98.
- Probability of an erroneous GO indication of less than .005.
- Capability of isolating failures to an LRU with a probability of 0.95.

c. BIT Concept

The concept for BIT in the SFCS includes designing so that BIT can be initiated only when it is safe to do so, and that BIT can be stopped when an unsafe condition is detected. The prerequisites for initiating BIT are:

- The aircraft is not airborne.
- The aircraft is not taxiing or moving.
- It is safe to automatically move control surfaces.

To provide a convenient and safe method of performing BIT, the aircraft wiring is being designed to include BIT consent circuitry as shown in Figure 51. The microphone adapter includes a jumper to energize the BIT interlock relay and a switch hereinafter called the "BIT consent switch". During preflight the microphone is connected to allow the crew chief to communicate with the pilot. When the pilot requests the automatic BIT sequence to be initiated, the crew chief will determine that it is safe to do so and will depress the BIT consent switch, which starts the automatic BIT sequence.

The automatic BIT switches and indicators are planned to be implemented as follows:

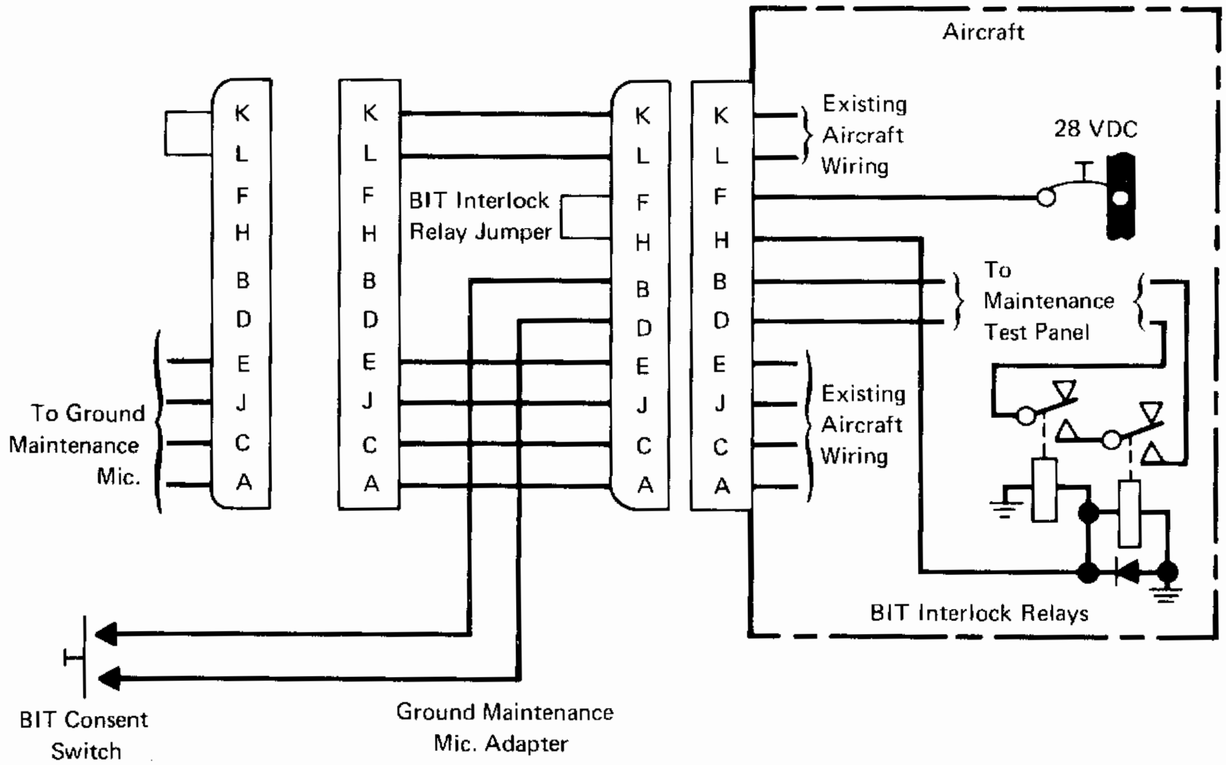


FIGURE 51
GROUND BIT INTERLOCK AND BIT CONSENT CIRCUITRY

(1) The BIT may be initiated from the BIT consent switch only.

(2) The OFF portion of the BIT ON/OFF indicator switches will illuminate when the BIT interlock relay is energized and the BIT test sequence is not in progress. (See MCDP, Fig. 61, and MTP, Fig. 60.)

(3) The ON portion of the BIT ON/OFF indicator switches will illuminate when the BIT sequence is in progress.

(4) The GO lamp will illuminate at the successful completion of the BIT test sequence.

(5) The GO lamps on MCDP, SCDP and MTP can be extinguished by removing the microphone adapter.

(6) The BIT sequence can be interrupted by the following:

- o Releasing the BIT consent switch,
- o Momentarily depressing the BIT ON/OFF indicator switch on the Master Control and Display Panel or the Maintenance Test Panel or,
- o Deenergizing the BIT interlock relay.

(7) The BIT test number is displayed on the MTP to further aid fault isolation.

(8) The NO GO lamp will illuminate if a failure is detected or the BIT test sequence is interrupted for any reason before successful completion.

(9) After the NO GO lamp has been illuminated it will illuminate any time the aircraft electrical power is applied until the BIT test sequence is reinitiated.

d. BIT Features

Some of the salient BIT features planned to be incorporated in the SFCS are presented below:

(1) The test program memory in the SFCES, which is the basic BIT control element, has a storage capacity of 192, 48 bit words. Each 48 bit word is divided into the following functions: a monitor reset code, test input code, operation code, model discretes, failure sorting code and test time address.

(2) BIT provides flight line testing of the SFCS without the use of external test equipment.

(3) BIT isolates failures to a line replaceable unit (LRU) to the degree described in Paragraph III-25f with a probability specified as 0.95.

(4) BIT is designed such that its power is disconnected in flight to prevent inadvertent in-flight operation.

(5) BIT provides end-to-end testing through actual operating modes.

(6) BIT performs dynamic tests wherever practical.

(7) BIT is self-checking with respect to failure detection, by virtue of the fact that the first step of the BIT sequence is a self check.

(8) BIT NO GO indication is provided by latching indicators which can only be reset by restarting BIT.

e. BIT Operation

The complete BIT has not been designed in detail at this time; however, the expected BIT operation is illustrated by a description of the rate gyro test which is presented below.

When the BIT is started by depressing and holding the BIT consent switch, all the logic is reset and the Clock Control and Synchronizing Logic is started. Figure 52 illustrates the planned BIT implementation. As shown, the output of the Clock Control and Synchronizing Logic is fed to the Test Timing Counter and Logic. There are

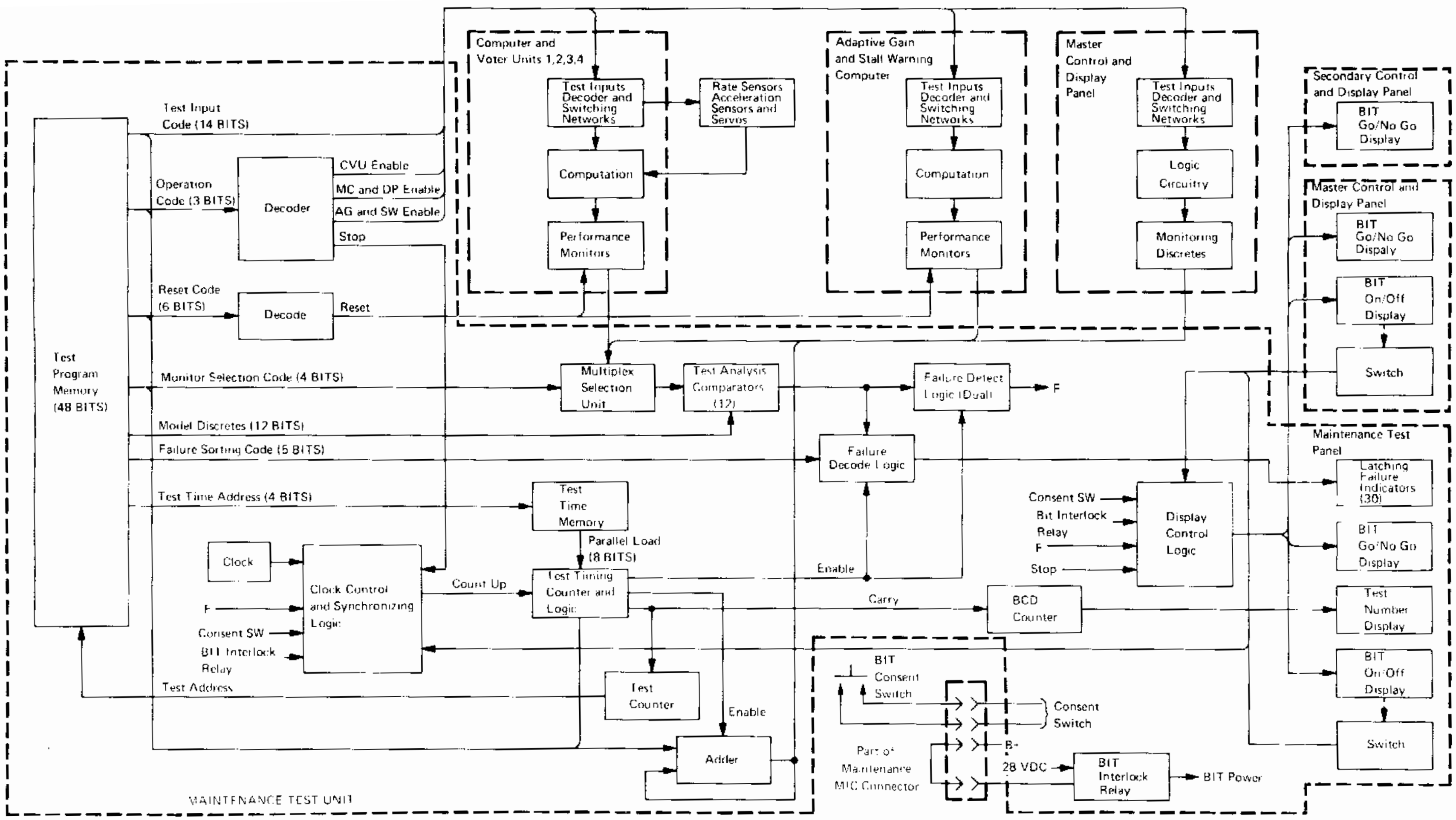


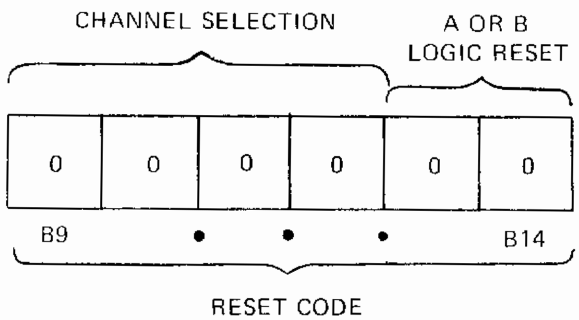
FIGURE 52
BIT IMPLEMENTATION

three outputs from the Test Timing Counter and Logic. They are two Enable outputs, and one Carry output.

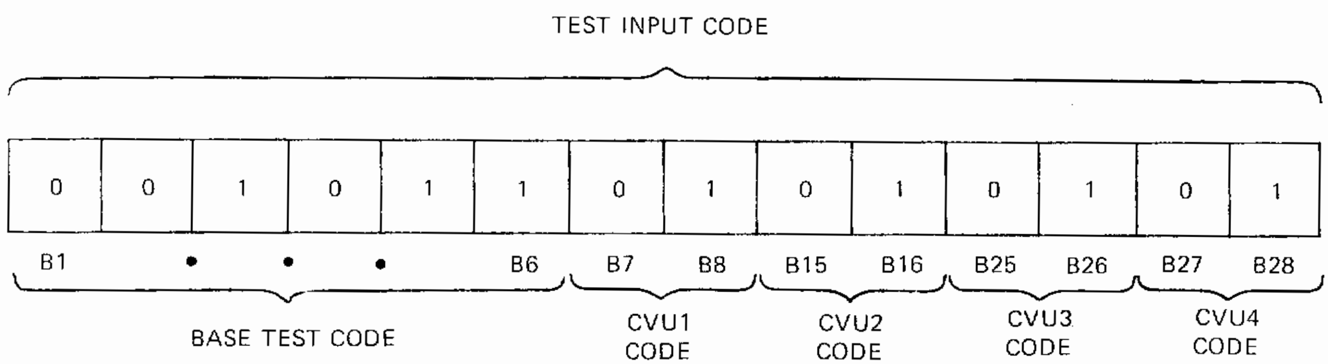
The Enable logic is supplied to the Failure Decode Logic, Failure Detect Logic and the Test Counter. The Output of the Test Counter causes the Test Program Memory to generate seven outputs. These outputs are:

- o Reset Code (6 bits)
- o Test Input Code (14 bits)
- o Operation Code (3 bits)
- o Test Time Address (4 bits)
- o Model Discretes (12 bits)
- o Failure Sorting Code (5 bits)

To illustrate operation of the BIT computer and its interface with various components in the system, consider a typical test involving rate gyro validation. The reset code provides a means of resetting latching monitors within the computer and voter units and also the adaptive gain and stall warning computer. For this particular test, no reset is required and these bits are programmed as follows:



Active inputs necessary to perform a testing function are defined in the test input code. This code contains 14 bits organized as shown below.



The Test Input Code is decoded within the computer(s) and appropriate switches change state, applying stimuli where necessary. In this case voltages are applied to the rate gyro torquer in all four channels simultaneously.

The base test code is outputted to units defined by the operation code. In the example, bits 17, 18 and 19 are programmed as shown to enable the computer and voter unit test input decoders.

OPERATION CODE

B17	B18	B19
0	1	0

Individual CVU codes modify this basic code to allow non-identical inputs among the four channels. For the case shown, all inputs are identical and the test input decoders located in each CVU apply similar stimuli to each of the rate gyros (pitch, roll and yaw). Different codes are available which excite the servos, accelerometers, and CVU internal computational elements.

Three performance monitors, one each in pitch, roll and yaw in each CVU evaluate the rate gyro test and transmit discretes back to the BIT computer. The parameters monitored for each rate gyro are the SMRD and relative amplitude. In this case, a successful test should result in all "1's" (high outputs) from these 12 particular performance monitors. Many other discretes are available for interrogation by the BIT computer but these particular 12 are defined by the monitor selection code. This code, formed by 4 bits as shown below, is fed to the multiplex selection unit which selects the appropriate 12 input discretes and applies them to one input of the test analysis comparators.

MONITOR SELECTION CODE

B41	•	•	• B44
0	0	1	1

The other 12 inputs to the test analysis comparators are obtained from the model discrete portion of the memory word. Since all performance monitors for this sample test should be "1", the model discretes are programmed as follows:

MODEL DISCRETES

1	1	1	1	1	1	1	1	1	1	1	1
B29				•	•	•					B40

Test duration time is stored in the test time memory. Sixteen eight-bit test time locations are available, selectable by the test time address bits. Test times may be programmed in any multiple of 16 milliseconds. Maximum duration of any particular test time is approximately 4 seconds. For the sample test, the 4 test time address bits are programmed as shown below.

TEST TIME ADDRESS

1	0	1	1
B45	• • •		B48

When the selected test time is concluded, as determined by the test timing counter and logic, the dual failure detect logic and failure decode logic is enabled. If a failure has occurred, the failure detect logic forces the clock to stop, thus interrupting the test sequence. A failure sort code, programmed in bits 20 through 24 as illustrated, essentially distributes the test analysis comparators to the appropriate failure indicators by use of the failure decoding logic.

FAILURE SORT CODE

0	0	1	1	0
B20		• • •		B24

The output from the failure decode logic causes the appropriate latching failure indicator to latch, thus indicating the failed LRU.

If the test is successful, the test timing logic advances the test number display and the test counter one increment. This process continues until the testing is complete. At this time, a cumulative adder, which has been keeping a running sum of critical control bits, is interrogated. If a failure in the test program memory has occurred, this running sum will not be correct and a BIT failure is indicated. In the normal case, however, the sum will check and the test will advance one step. The operation code on this last step demands the test sequence to stop and a GO is indicated on all three display panels.

f. Degrees to Which BIT Checks SFCS Components and/or Subsystems

The SFCS components and/or subsystems will be checked to varying degrees. Some components and/or subsystems are not planned to be checked by BIT. These are:

- o Mechanical Isolation Mechanism
- o Mechanical Back-Up

It is planned that these items will be checked by the pilot using a manual test procedure.

The second group of components and/or subsystems are nominally checked by BIT in that active failures will be detected but passive failures will not. Items in this category are:

- o Control Stick Transducer
- o Side Stick Controllers
- o Trim Panels
- o Rudder Pedal Transducer
- o Angle of Attack Probes

It is planned that passive failures in the above items will be detected by the pilot and/or ground crew using a manual test procedure.

The third group of components and/or subsystems is partially checked for passive failures and partially checked for active failures. These items are:

- o Master Control and Display Panel
- o Secondary Control and Display Panel
- o Maintenance Test Panel

It is planned that passive failures in the above items will be detected by the pilot and/or ground crew using a manual test procedure.

g. Additional Information

Additional information on the planned BIT implementation may be found in Reference 5.

26. ELECTRICAL BACK-UP SYSTEM POWER

The SFCS is planned to include an Electrical Back-up (EBU) mode of operation. This mode of operation will provide a direct electrical path to the surface actuators, bypassing the normal FBW sensor inputs and computational circuits, to be used when the normal FBW system ceases to function satisfactorily due to sensor or computer malfunctions.

An early EBU concept called for use of separate force transducers and a separate non-redundant DC electrical power source for mechanization. The use of separate force transducers and an independent power source presents two problems: (1) the interface between the output of the EBU transducers and the servoamplifiers and secondary actuator loops due to mixing of electrical power supplies, and (2) an unrealistic installation if an attempt were made to locate the EBU transducers within the already crowded control stick. Also, the reliability of an independent non-redundant power supply probably would be less than the reliability of the SFCS DC quad-redundant supplies.

In view of these factors, it was decided to implement the EBU system using the same transducers and quad-redundant SFCS power supplies as are used for the Normal SFCS mode.

27. IN-FLIGHT FAILURE SIMULATION

a. General

In-flight failure simulation can add significantly to the value of the SFCS flight test results. Provisions to accomplish this will therefore be included in the SFCS aircraft. MCAIR recognizes that the failure modes inherently added by the equipment required to perform the in-flight failure simulation may degrade the reliability of the SFCS. Therefore, the failure insertion mechanization is designed to be as simple, safe and efficient as practicable. For the reasons stated above, it is planned that the in-flight simulation of failures will be provided using the following philosophy:

- o SFCES, Secondary Actuator, SSAP, and electrical and hydraulic power supply failures will be simulated.
- o Two types of SFCES and actuator failures will be simulated - the most difficult to detect, null failures; and the most dangerous, hardovers.
- o Only those failures will be simulated which are not expected to cause more than one similar failure. Up to six dissimilar failures can be simulated simultaneously in flight.
- o Only one channel of each axis will have provisions for failure simulation.
- o All failure insertion switches will be covered to help prevent inadvertent activation.

b. Implementation

Six failure insertion switches are planned to be provided on the SFCES master control and display panel (Fig. 61) for the purpose of in-flight or ground simulation of failures. Each switch can provide a hardover or null failure by applying either B+ or ground at the failure insertion point. For safety reasons, failure insertion is limited to one channel of pitch, roll, and yaw in the No. 4 (yellow channel) computer and voter units. The failure insertion circuitry exists in all CVUs for commonality, but aircraft wiring allows insertion only into the yellow channel. In each axis, failures may either be inserted into the electronics in front of the voter, simulating sensor or electronics failures, or behind the voter, simulating actuator or hydraulic failures.

The means for inserting these failures are shown in Figure 53. This particular failure insertion method employs the natural current-limiting capabilities of the uA741 amplifier to reduce interface requirements and increase reliability. Since the uA741 is capable of permanently sustaining a short-circuit current (approximately 15 milliamperes), the failure switches can be wired in parallel with

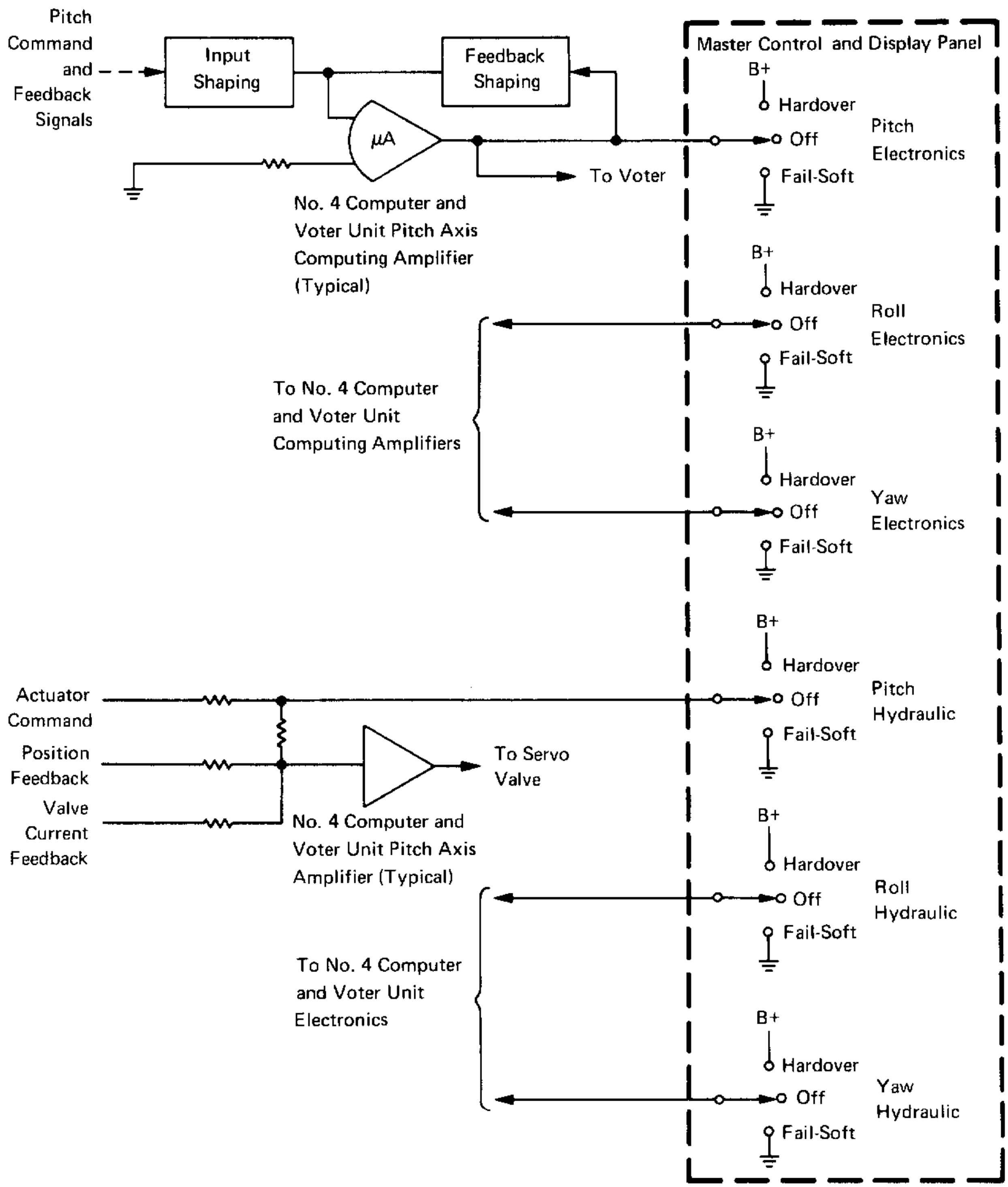


FIGURE 53
FAILURE INSERTION DIAGRAM

the normal operating circuits. Series wiring of the switch would increase the interface wiring and require that control signals be routed through the switch, thus reducing reliability.

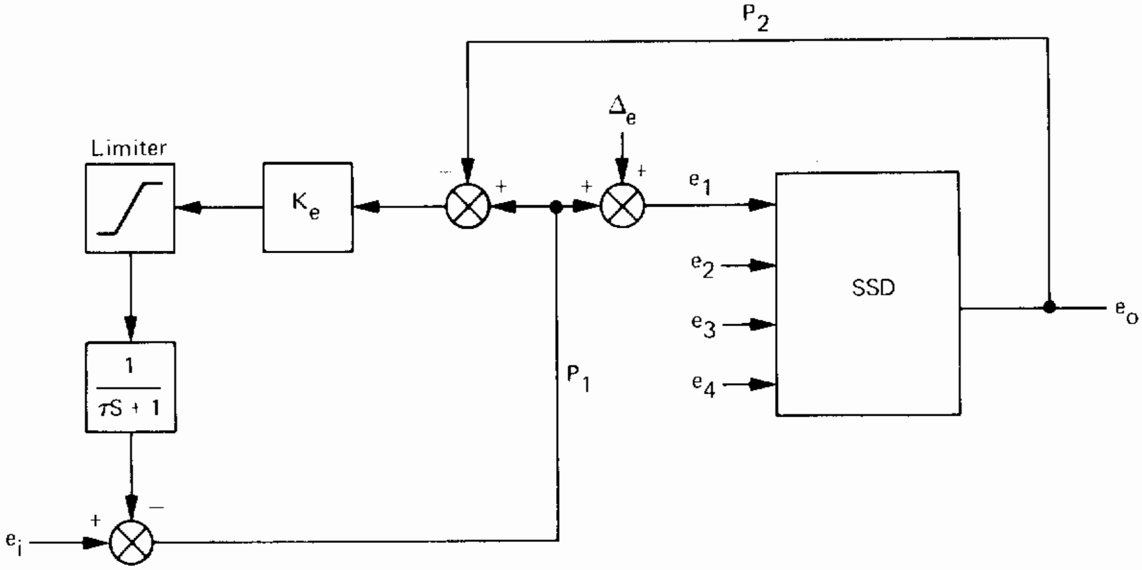
Hydraulic failures are simulated by applying B+ or signal return to the amplifier command input. Applying B+ at the amplifier command input will result in the servovalve producing maximum pressure output. This type of failure will result in failure detection within 500 milliseconds - even without system activity. Output motion resulting from a hardover is proportional to the loop gain, the level at which pressure bypass of the failed channel will occur, and the number of remaining good channels. When hydraulic failures are simulated by placing the input of the amplifier at signal return, failure characteristics can only be described as a function of input signals. In each case however, the input command must result in the failed channel producing an equivalent error signal of a magnitude and duration to exceed the monitor threshold in order for a failure to be detected.

c. Simulator Evaluation

Characteristics of the aircraft transients which can result from the insertion of simulated failures were evaluated during the Selected Control System Evaluation Simulation. Worst case transients were evaluated to determine aircraft controllability following failure insertion. The transient characteristics resulting from simulated electronic failures are a function of the equalization circuit used with the Signal Selection Device (SSD). These characteristics are discussed in the following paragraphs.

Figure 54 shows the equalization technique used with the SSD to reduce the effects of real or simulated failures. Application of this technique results in the improved SSD input tracking shown in Figure 55. In this illustration the effect on tracking accuracies and improvement in transient levels are defined by the waveforms and corresponding equations. The fact that failure transient may be controlled by equalization gain and time constants is indicated. Depending on equalization loop gain, the step values occurring at t_2 and t_3 can be made negligible. For example, a 4 percent channel tracking at the time of failure and an equalization gain of 20 will produce a maximum pitch axis transient contribution at t_2 equal to 0.17 g (assuming a high "q" flight condition and τ much greater than the comparator trip time).

The failure correction transient, which is initiated at time t_3, has an upper limit equal to the channel tracking error; however, this transient occurs at a rate corresponding to the equalization time constant τ, which is approximately 5 seconds. From the pilot's viewpoint, a 4 percent transition of this type would require a 4 percent change in stick force (at high "q") over the transition time interval. A visual indication of the disturbance source is indicated by a warning light on the control panel, minimizing the possibility of pilot confusion. Equalization time constants of 5 seconds or greater should permit pilot corrective action in smoothing the failure correction transient.



Practical Considerations

Equalization Limit Necessary to Allow Slowover Detection.

Circuit is Unstable if $K_e > \frac{1}{1-a}$ Where K_e is the Static Value of $K_{e(s)}$ and a is the Ratio of the Gain in Path P_1 to the Gain in Path P_2.

Offsets in the Signal Selection Device Appear Multiplied at the Circuit Output.
i.e., $e_o = (1 + K_e)\ \Delta e$ Where Δe = SSD Offset.

FIGURE 54
EQUALIZATION OF THE SSD

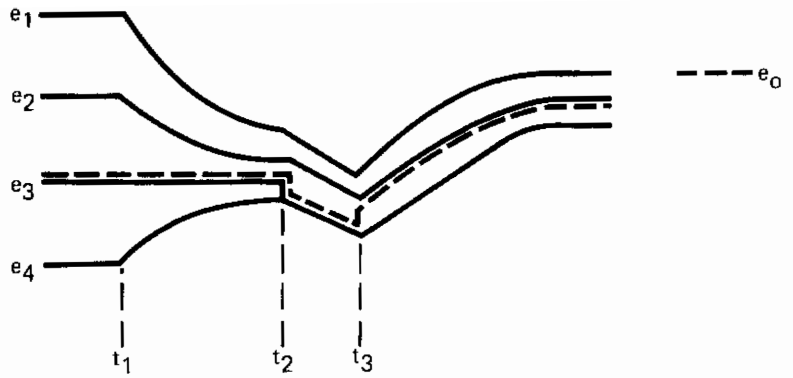
To simulate worst case transients which could result from failure insertion, the following assumptions are made:

o The yellow channel is the channel selected by the SSD.

o Channel tracking is such that the output of two channels are at the extreme high end of the tolerance band and the other two channelsoutputs are at the extreme low end of the tolerance band.

Under these worst case conditions, the output transient will be equal to the maximum tracking error of approximately one volt and will be inserted on a 5 second time constant. If the yellow channel is not the channel selected by the SSD or the tracking is not as assumed above, smaller transients would occur.

The worst case transient input was simulated by inserting a step through a first order lag with a 5 second time constant to the secondary actuator input point. Normal pilot response smoothed this input. The pilots found no difficulty in correcting for the small effect caused by these failures. The failure signals were first individually applied to the SFCS with the pilots taking no corrective



- Input errors reduced by a factor of $\frac{1}{1 + K_e}$ when equalization loop is closed at t_1.

- Initial transient consists of a step $\frac{e_3 - e_4}{1 + K_e}$ at time t_2 and an exponential decay toward e_4 of time constant τ.

- Correction transient at time t_3 consists of a step $\frac{e_2 - e_4}{1 + K_e}$ and an exponential rise of time constant τ towards e_2.

FIGURE 55
EQUALIZED VOTER INPUT FAILURE TRANSIENT CHARACTERISTICS

action while observing the resultant transients. Figures 56 through 58 present the resulting transients with no pilot inputs for the three axes failures at .9M - 15,000 ft. It appears from the characteristics of these time histories that a pilot probably could easily correct for these failure induced transients. No deterioration in system performance was observed following failure insertion.

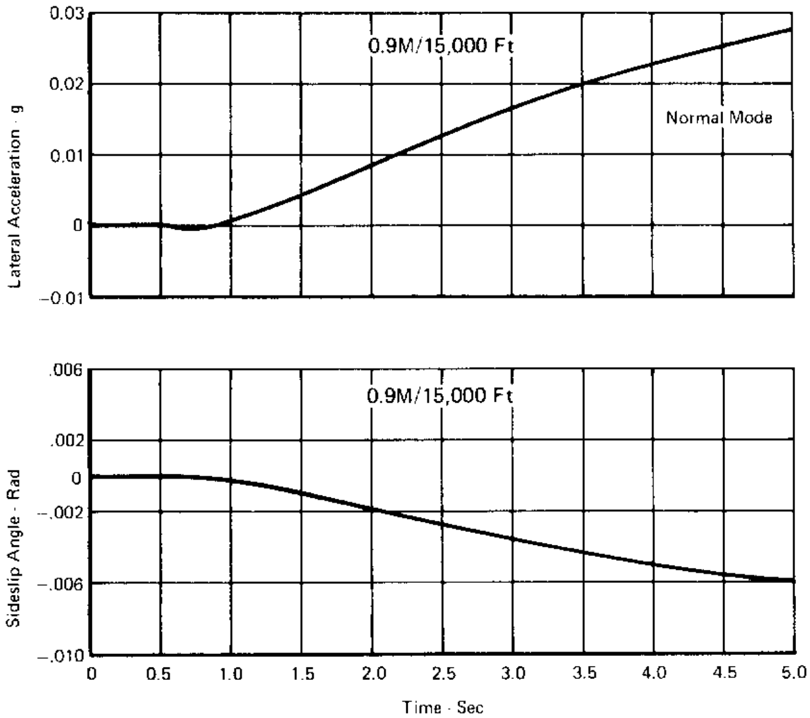


FIGURE 56
YAW AXIS SINGLE CHANNEL FAILURE

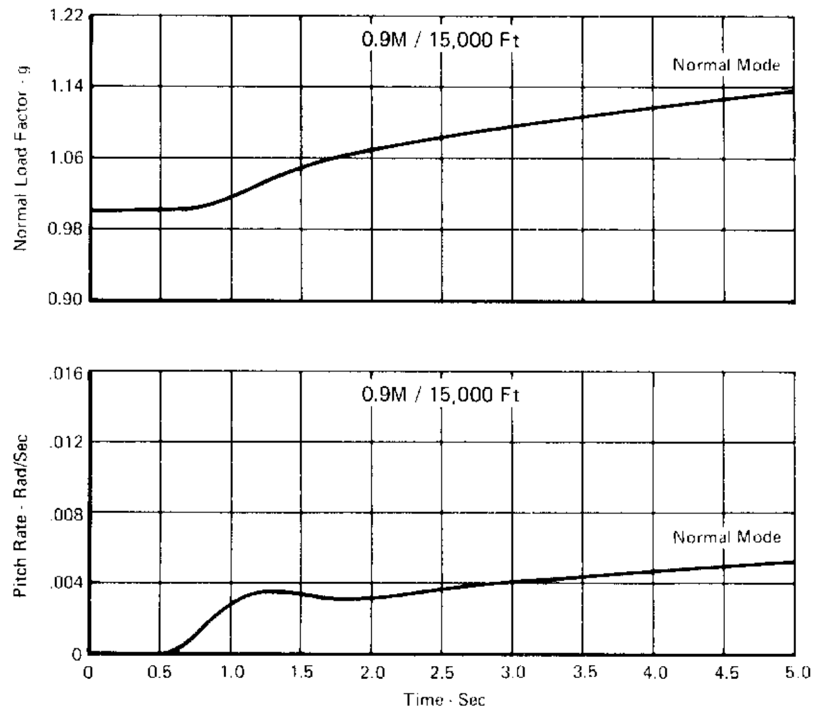


FIGURE 57
PITCH AXIS SINGLE CHANNEL FAILURE

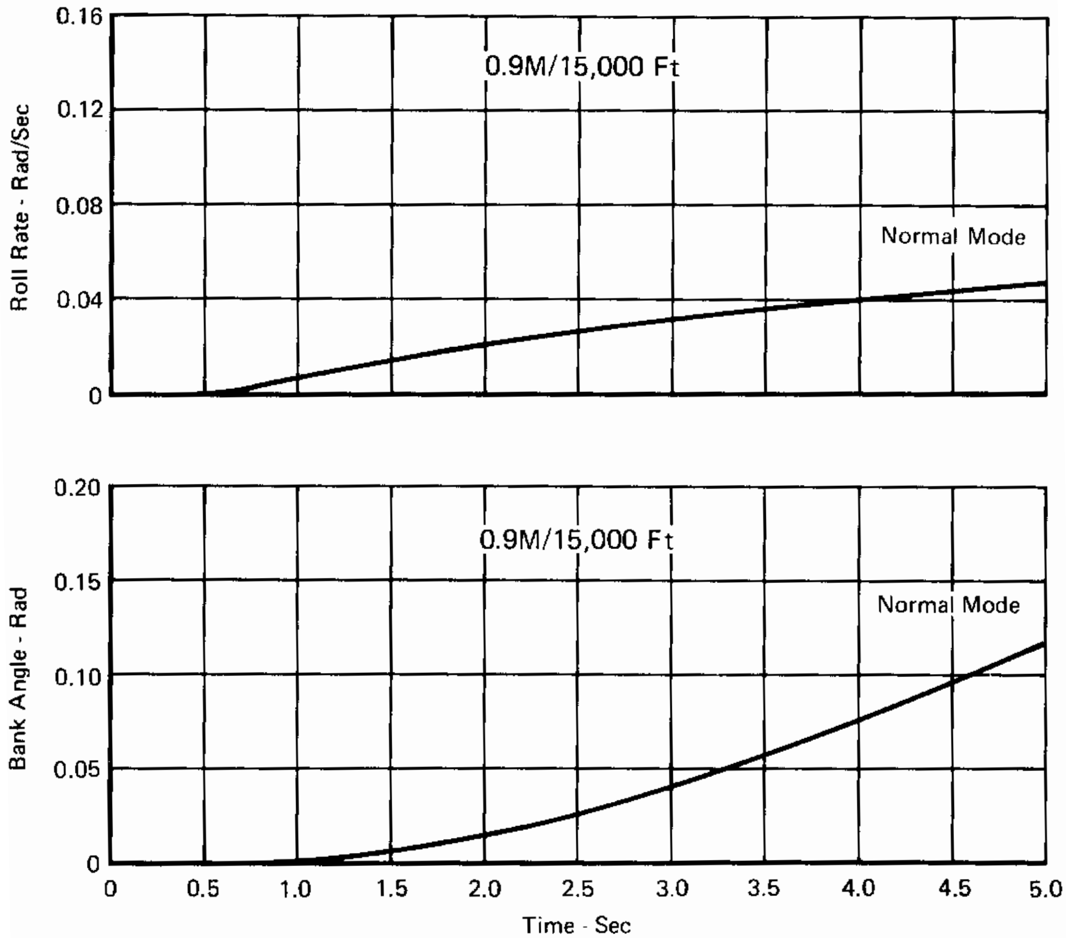


FIGURE 58
ROLL AXIS SINGLE CHANNEL FAILURE

SECTION IV

DESCRIPTION OF MAJOR PROCURED EQUIPMENT

1. GENERAL

This section provides a brief description of major procured equipment. The components, functions and known characteristics of the Survivable Flight Control Electronic Set (SFCES), Secondary Actuators (SA), Side Stick Controller (SSC), Survivable Stabilator Actuator Package (SSAP) and the Mobile Ground Test Facility (MGTF) are delineated.

2. SURVIVABLE FLIGHT CONTROL ELECTRONIC SET

The Survivable Flight Control Electronic Set (SFCES) includes the necessary electronic components for the three-axis, quadruplex, two-fail operational SFCS; it provides:

- Aircraft motion sensors
- Pilot command transducers
- Control and monitoring panels
- Stall Warning functions
- Adaptive Gain
- Analog computers to combine the above and provide quadruplex electrical (FBW) control signals
- Signal selection devices
- In-flight monitoring
- Built-In-Test including fault isolation to the LRU
- Buffered SFCS performance parameters for instrumentation outputs

One ship's set of equipment consists of the units and quantities in the five categories listed in Table XVIII.

Table XIX gives the expected physical characteristics of the SFCES LRUs. The LRUs are to be located in the aircraft as shown in Figure 59.

Additional SFCES information is given in Reference 5.

TABLE XVIII
SFCES EQUIPMENT CATEGORIES

Units	Number Per Shipset	Number of Peculiar LRU's
Computers	6	3
Sensors	5	5
Transducers	3	3
Panels	5	5
Equipment Racks	6	3
Total	25	19

TABLE XIX
SFCES EQUIPMENT PHYSICAL CHARACTERISTICS

Unit	Approximate Weight (lb)	Approximate Volume (cu. in.)
Computers		
Computer and Voter Unit*	29	1050
Adaptive Gain and Stall Warning Computer	18	850
Maintenance Test Panel (BIT Computer)	21	850
Sensors		
Pitch Rate Sensor Unit	4	80
Roll Rate Sensor Unit	4	80
Yaw Rate Sensor Unit	4	80
Normal Accelerometer Sensor Unit	2	70
Lateral Accelerometer Sensor Unit	2	70
Transducers		
Control Stick Transducer Unit	6	40
Pedal Transducer Unit	3	30
Stabilator Surface Position Transducer Unit	4	60
Panels		
Master Control and Display Panel	30	1850
Secondary Control and Display Panel	5	120
Trim Control Panel (Forward)	3	80
Trim Control Panel (Aft)	2	55
Discrete Function Generator Panel	1	30
Equipment Racks		
Tray, Computer and Voter Unit*	3	
Tray Maintenance Test Panel	2	
Tray Adaptive Gain and Stall Warning	2	
Total Per Ship Set	241	8545

*4 Per Ship Set

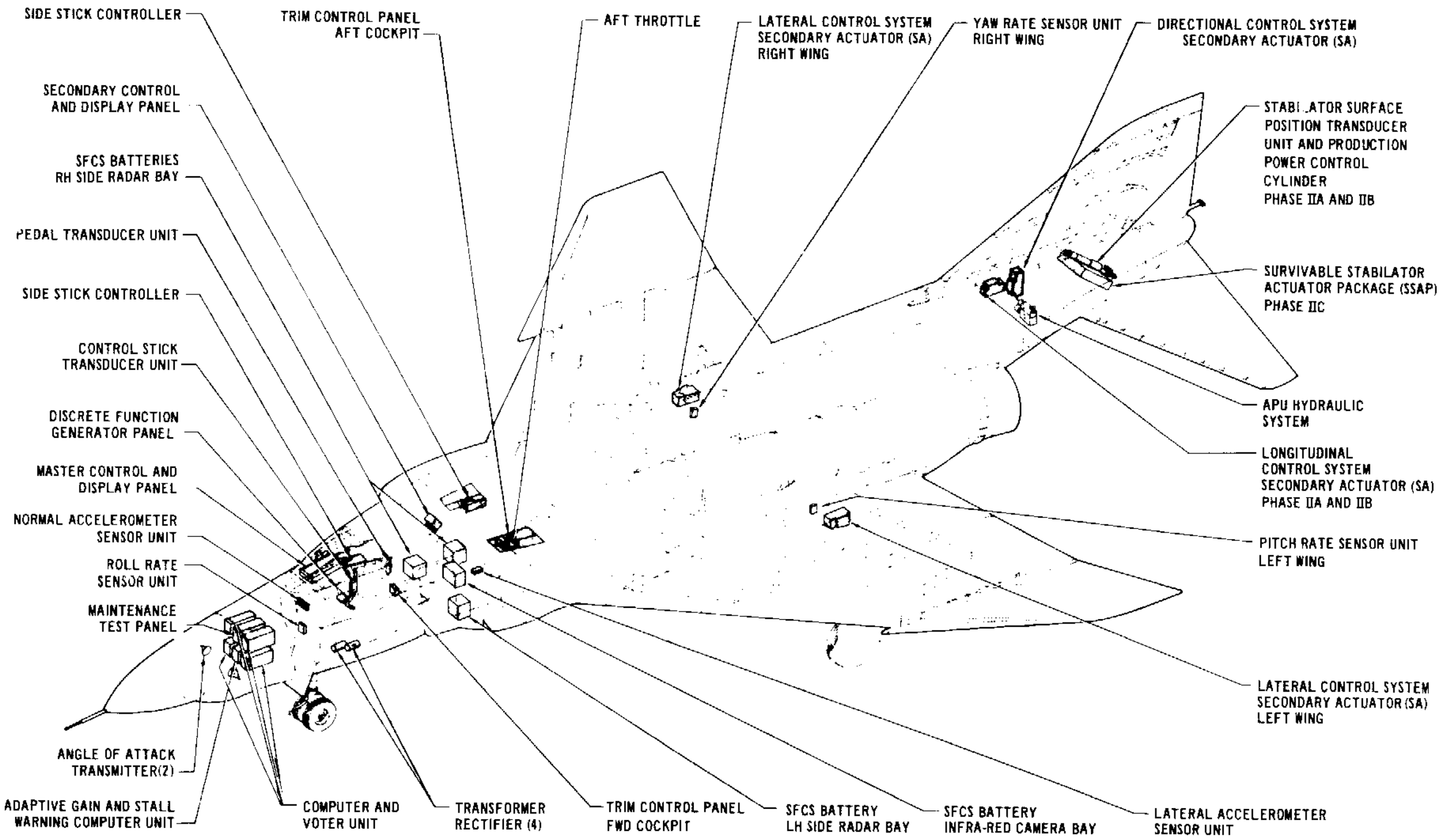


FIGURE 59
SFCS EQUIPMENT LOCATION

a. Computers

(1) Computer and Voter Unit (CVU)

The SFCES uses four tray mounted Computer and Voter Units. These units provide four isolated control paths for the SFCS electronics.

Each Computer and Voter Unit contains:

(a) Analog computational circuitry to implement the control laws for the pitch, roll and yaw axes.

(b) Electronic signal selection devices to select the least magnitude median of the four signals to control the servo amplifiers.

(c) Servo amplifiers to interface with various electrohydraulic and electromechanical actuators.

(d) Power supplies and converters for excitation of one entire channel of the SFCS.

(e) Provisions to interface with the analog computer in the Mobile Ground Test Facility (MGTF).

(f) Electronic buffers to feed the necessary SFCS parameters to the flight test instrumentation.

(g) Monitoring circuitry for ground Built-In-Test (BIT) and In-Flight Monitoring (IFM).

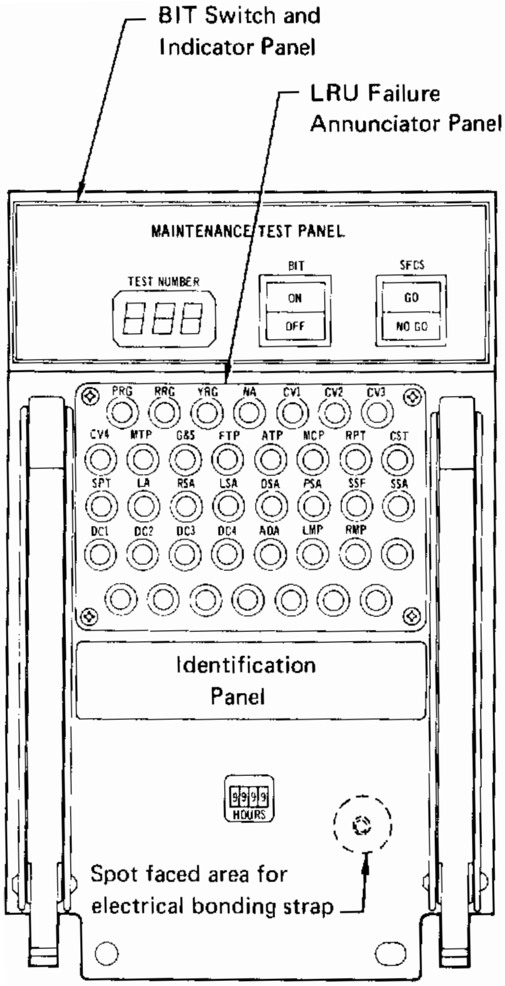
Each unit has 9 metal cards and 4 printed circuit cards. The printed circuit cards contain embedded hybrid, micro-electronic subassemblies and discrete components. Conformal coated cordwood modules are used on the metal cards.

(2) Adaptive Gain and Stall Warning Computer

The Adaptive Gain and Stall Warning Computer Unit contains the necessary computational circuitry to provide the adaptive gain and stall warning functions. These functions are independent, dual redundant, and fail safe. Refer to the Stall Warning block diagram in Section III-13 "Control Laws" and Adaptive Gain block diagram in Section III-18 "In-Flight Parameter and Gain Variations."

(3) Maintenance Test Panel

The Maintenance Test Panel, the front panel of which is shown in Figure 60, performs digital computer functions associated with ground BIT and provides a display of BIT results and status. Read only memories, which are large scale integrated circuits plus medium scale integrated circuits mounted on printed circuit boards, comprise the major portion of the electronics of this computer.



AOA - Angle of Attack Transmitters
ATP - Aft Cockpit Trim Control Panel
CST - Control Stick Transducer
CV1 - Computer and Voter Unit Number One
CV2 - Computer and Voter Unit Number Two
CV3 - Computer and Voter Unit Number Three
CV4 - Computer and Voter Unit Number Four
DC1 - SFCS DC Power Supply Number One
DC2 - SFCS DC Power Supply Number Two
DC3 - SFCS DC Power Supply Number Three
DC4 - SFCS DC Power Supply Number Four
DSA - Directional Secondary Actuator
FTP - Forward Cockpit Trim Panel
G&S - Adaptive Gain and Stall Warning Computer Unit
LA - Lateral Accelerometer Sensor Unit
LMP - Left SSAP Motor Pump Unit
LSA - Left Lateral Secondary Actuator
MCP - Master Control and Display Panel
MTP - Maintenance Test Panel
NA - Normal Accelerometer Sensor Unit
PRG - Pitch Rate Sensor Unit
PSA - Pitch Secondary Actuator
RMP - Right SSAP Motor Pump
RPT - Pedal Transducer Unit
RRG - Roll Rate Sensor Unit
RSA - Right Secondary Actuator
SPT - Stabilator Surface Position Transducer Unit
SSA - Aft Side Stick Controller
SSF - Forward Side Stick Controller
YRG - Yaw Rate Sensor Unit

Indicator or Switch	Display	Significance
BIT ON/OFF Switch	OFF Illuminated Yellow	BIT Power Applied but BIT not in Progress
	ON Illuminated Yellow	BIT in Progress
SFCS GO/NO GO Indicator	GO Illuminated Green	BIT sequence complete and system ready for operation.
	NO GO Illuminated Red	BIT sequence interrupted by a failure or personnel action.
LRU Failure Annunciators	Black	Applicable LRU status GO
	Yellow	Applicable LRU status NO GO

FIGURE 60
SFCES MAINTENANCE TEST PANEL

b. Sensor Units

(1) Rate Sensor Units

Three rate sensor units are provided as part of the SFCES. These sensors detect angular rate in the pitch, roll and yaw axes.

Each rate sensor supplies four independent rate signals to the Computer and Voter Units from four separate rate gyros. The pitch rate sensor unit also supplies signals to the Adaptive Gain and Stall Warning Computer. The rate gyros are torsion bar spring restrained gyros. The four gyros in each sensor unit package are mounted in separate compartments in an aluminum block machined to provide physical isolation.

Each of the rate gyros has a built-in torquer and a spin motor rotation detector to test the sensor during Built-In-Test and to provide a motor speed signal for In-Flight Monitoring.

(2) Accelerometer Sensor Units

Two accelerometer sensor units are provided as part of the SFCES. One unit is used to sense lateral acceleration and the other is used to sense normal acceleration.

Each sensor unit contains four independent accelerometers. Each accelerometer employs a force balance system for positioning a mass to generate a null output from a differential transformer. The balancing force provides an output signal proportional to the sensed acceleration. Each accelerometer has a built-in torquer which is used during BIT to test the accelerometers.

(3) Control Stick Transducer Unit

The Control Stick Transducer Unit converts pilot force inputs into electrical command inputs for the Computer and Voter Units. The Control Stick Transducer has four pitch and four roll outputs from eight semiconductor strain gages. These strain gages are mounted on grounded cantilevered beams to provide an electrical output proportional to the deflection of the flexure.

The unit contains a balance weight on the flexure to help avoid unwanted outputs from the stick transducer when the Side Stick Controller is being used to control the aircraft.

An Emergency Disengage Paddle Switch is included on the Transducer Unit. The paddle switch consists of four microswitches which are used to switch the system:

o from Normal to electrical back-up in Phases IIA, IIB and IIC and

- o from SFCS to mechanical back-up in the pitch and yaw axes through additional circuitry in Phase IIA. The SFCES continues to function in EBU but its outputs are not applied to the control surfaces.

(4) Pedal Transducer Unit

The Pedal Transducer Unit converts pilot force inputs into electrical directional commands for the Computer and Voter Units.

The housing of the Pedal Transducer Unit is a machined spring which is instrumented with four semiconductor strain gages. The strain gages provide an output signal when the applied force is greater than the breakout force. The output signal is proportional to the applied force for forces above the breakout force.

(5) Stabilator Surface Position Transducer Unit

The Stabilator Surface Position Transducer Unit consists of a housing containing four Linear Variable Differential Transformers (LVDTs). This unit is attached to the production F-4 stabilator actuator in Phases IIA and IIB and provides electrical signals proportional to stabilator surface position to the SFCES.

c. Panels

(1) Master Control and Display Panel

The Master Control and Display Panel is illustrated by Figure 61, and provides SFCES pilot control functions and SFCES status information. The panel contains momentary action switches which include indicators. These switches are used to switch between the NORMAL and BACK-UP modes and to reset the failure monitor circuitry. The Nomenclature, indication and function of these switches is summarized in Table XX.

The panel contains solenoid held toggle switches for selection of the adaptive gain function, lever locked toggle switches for failure insertion into the yellow channel and covered toggle switches for selection of the EMERGENCY PITCH and EMERGENCY ROLL-YAW functions. The indicators and switches are shape coded as follows: pitch axis-round, roll axis-square, and yaw axis-diamond. A BIT ON-OFF indicator and switch is included to indicate presence of BIT power and enable the pilot to stop the BIT function upon command. Sufficient electronic circuitry is included for BIT, light drivers and flasher logic.

(2) Secondary Control and Display Panel

The Secondary Control and Display Panel is located in the aft cockpit and repeats the Master Control and Display Panel indications.

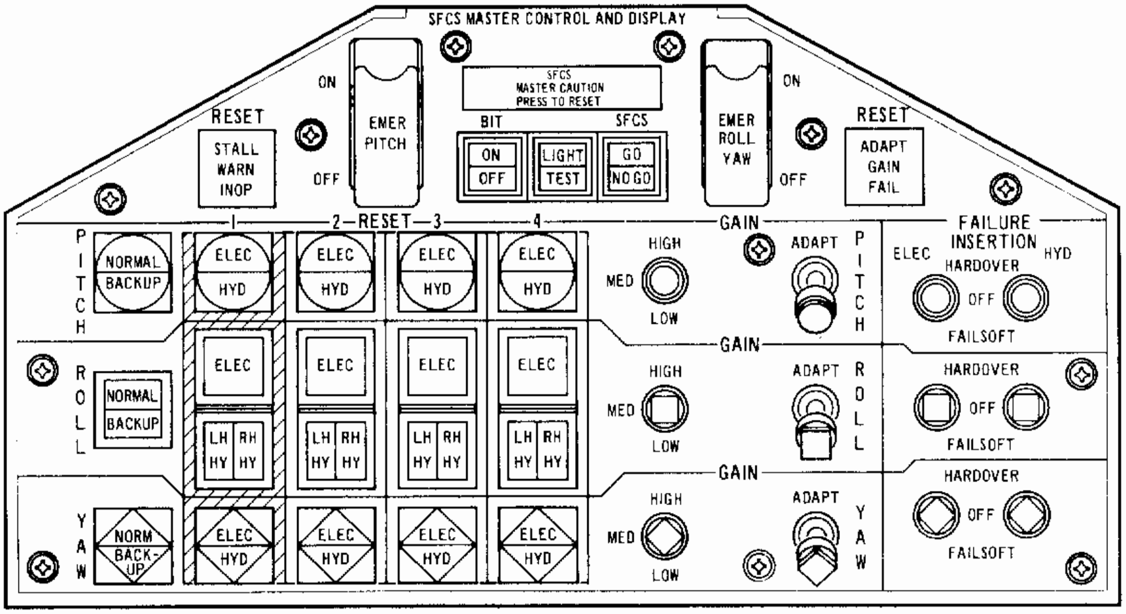


FIGURE 61
MASTER CONTROL AND DISPLAY PANEL

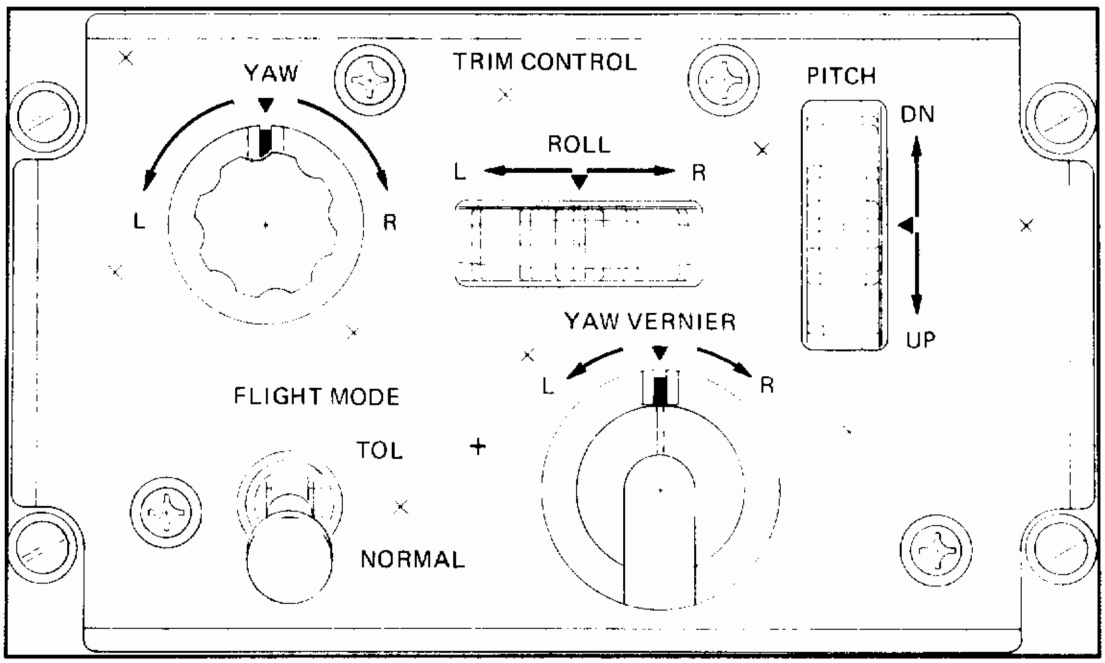


FIGURE 62
TRIM CONTROL PANEL (FORE) LAYOUT

TABLE XX
INDICATIONS AND SWITCH FUNCTIONS OF
MASTER CONTROL AND DISPLAY
PANEL SWITCH INDICATORS

Switch Indicator Nomenclature	Indication	Switch Functions*
PITCH NORMAL/BACKUP	Indicates Mode Status	Switches Modes
ROLL NORMAL/BACKUP	Indicates Mode Status	Switches Modes
YAW NORMAL/BACKUP	Indicates Mode Status	Switches Modes
PITCH ELEC. HYD. (4 PLACES)	Indicates Electronic Set or Actuator Failures	Resets Failure Monitor Circuitry
ROLL ELEC/LH HYD (4 PL) ROLL ELEC/RH HYD (4 PL)	Indicates Electronic Set or Actuator Failures	Resets Failure Monitor Circuitry
YAW ELEC. HYD (4 PLACES)	Indicates Electronic Set or Actuator Failures	Resets Failure Monitor Circuitry
BIT ON-OFF	Indicates BIT Power is applied when Off is illuminated. Indicates BIT Sequence is in progress.	Stops BIT Sequence
LIGHT TEST	All Lights On	Turns On All the Master Control and Display Panel Lamps
SFCS GO/NO GO	Indicates the Results of the BIT Sequence	None
STALL WARN INOP.	Indicates Stall Warn System Status	Resets Failure Monitor Circuitry
ADAPT GAIN FAIL	Indicates Adaptive Gain System Status	Resets Failure Monitor Circuitry
SFCS MASTER CAUTION	Flashes Indicating a Failure	Press to Reset Turns Off Master Caution Light
EMER PITCH	Up Position Demands On	Bypasses Pitch EH or EM SA Monitors
EMER ROLL YAW	Up Position Demands On	Bypasses Lateral and Directional SA

*All switches are momentary action except EMER PITCH and ROLL YAW switches

(3) Trim Control Panel (Forward Cockpit)

The forward cockpit Trim Control Panel is shown by Figure 62, and includes four quadruplex potentiometer assemblies for:

- o Pitch Trim,
- o Roll Trim,
- o Yaw Trim, and
- o Yaw Vernier

A four pole double throw lever-locked NORMAL or TOL switch is included so that the pilot can override the Neutral Speed Stability (NSS) function. The panel contains integrally lighted wheels for Pitch and Roll trim control. A knob is provided for Yaw trim control. A knob which is spring loaded to the center position is provided for Yaw Vernier control. The yaw verner is provided for making small directional corrections.

(4) Trim Control Panel (Aft Cockpit)

The aft cockpit Trim Control Panel has three quadruplex potentiometer assemblies for:

- o Pitch Trim,
- o Roll Trim, and
- o Yaw Trim.

Trim authority of the aft cockpit Trim Control Panel is 50% of that for the forward Trim Control Panels so that the pilot can override any trim signals from the Aft Trim Control Panel.

(5) Discrete Function Generator

The Discrete Function Generator has three shape coded, lever locked, two position, bat handled switches. This panel provides a means of inserting discrete step inputs simultaneously into all channels of the pitch, roll or yaw axes to evaluate the effect of known discrete inputs.

d. Equipment Trays

Six equipment trays are provided per ship's set. There are four for the Computer and Voter Units, one for the Maintenance Test Panel and one for the Adaptive Gain and Stall Warning Computer. The Computer and Voter Unit tray is wider than the Adaptive Gain and Stall Warning Computer tray and the Maintenance Test Panel tray. The trays for the Adaptive Gain and Stall Warning Computer and Maintenance Test Panel are identical except for intended mechanical variations to prevent mis-installation. Provisions for an electrical bonding strap to the computer chassis are included on each tray.

3. SECONDARY ACTUATOR

a. The secondary actuator configuration is a quadruplex, force summing, hydraulically powered actuator which controls surface position in response to electrical signals from the SFCES, and provides electrical information to the SFCES for cross channel monitoring and comparison. The secondary actuator is comprised of four individual elements, which are small actuators, whose force outputs are summed through a rotary linkage, as shown in Figure 63. Each individual element is part of one of the SFCS channels. Figure 64 shows a cross section of a typical secondary actuator element. Each element is driven by a single stage jet pipe servovalve. Each element has a LVDT to provide position feedback to its channel of the SFCES. The working pressure or differential pressure across each element's piston head is monitored by a differential pressure sensor which provides electrical information to the SFCES for cross channel monitoring and comparison.

b. The differential pressure across the element's piston head is converted into an electric signal which is transmitted to the SFCES. When an element is in error, it will fight the other elements and its differential pressure will increase relative to the others. When the differential pressure exceeds a predetermined level, the SFCES logic will indicate that the element has failed and initiates a shut down by de-energizing the element's solenoid operated shutoff valve.

c. The preceding steps are repeated when a similar failure occurs in another element. However, when the third element fails, the SFCES shuts down both elements. The element with the third failure fights the remaining element and the differential pressures in both increases towards the failure voting level. It is not possible to determine which of the remaining two elements is good, thus necessitating the shut down of both elements in the event of the third failure.

d. When a lateral or directional secondary actuator has been totally shut down, a spring driven system centers the output. In the case of a longitudinal secondary actuator a brake holds the actuator in its last position. The centering and braking system is designed to be totally disengaged during normal operation of the secondary actuator to avoid affecting actuator performance.

A more complete description of the secondary actuator is presented in Reference 3. A dynamic analysis of the secondary actuator is presented in Supplement 3.

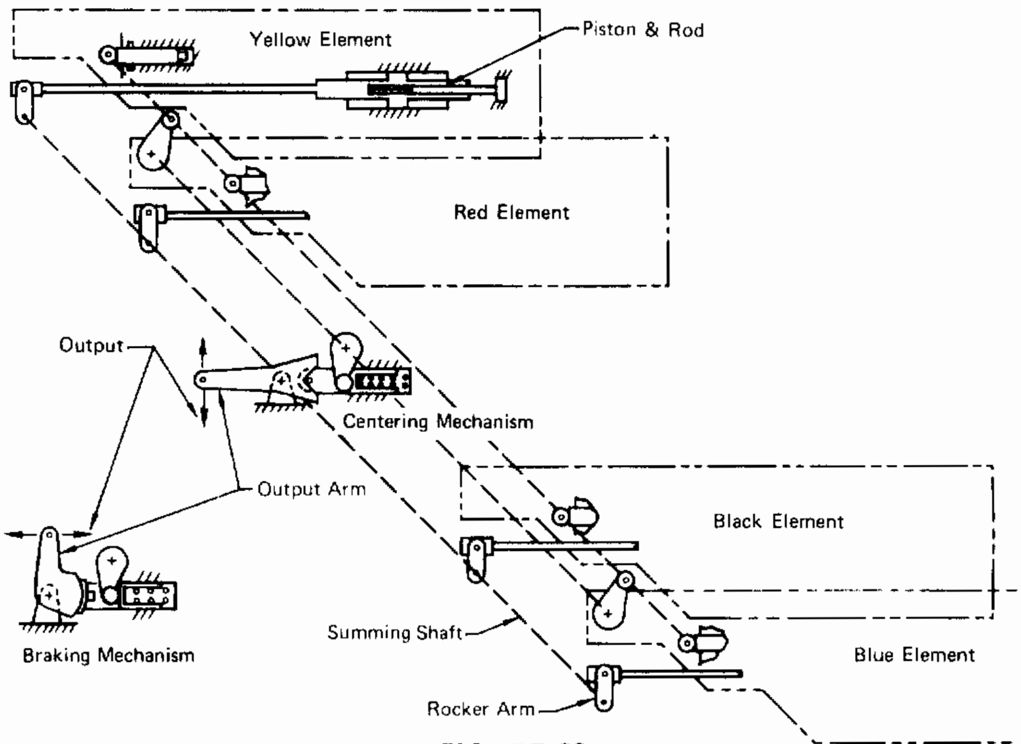


FIGURE 63
MECHANICAL SCHEMATIC , SECONDAIRY ACTUATOR

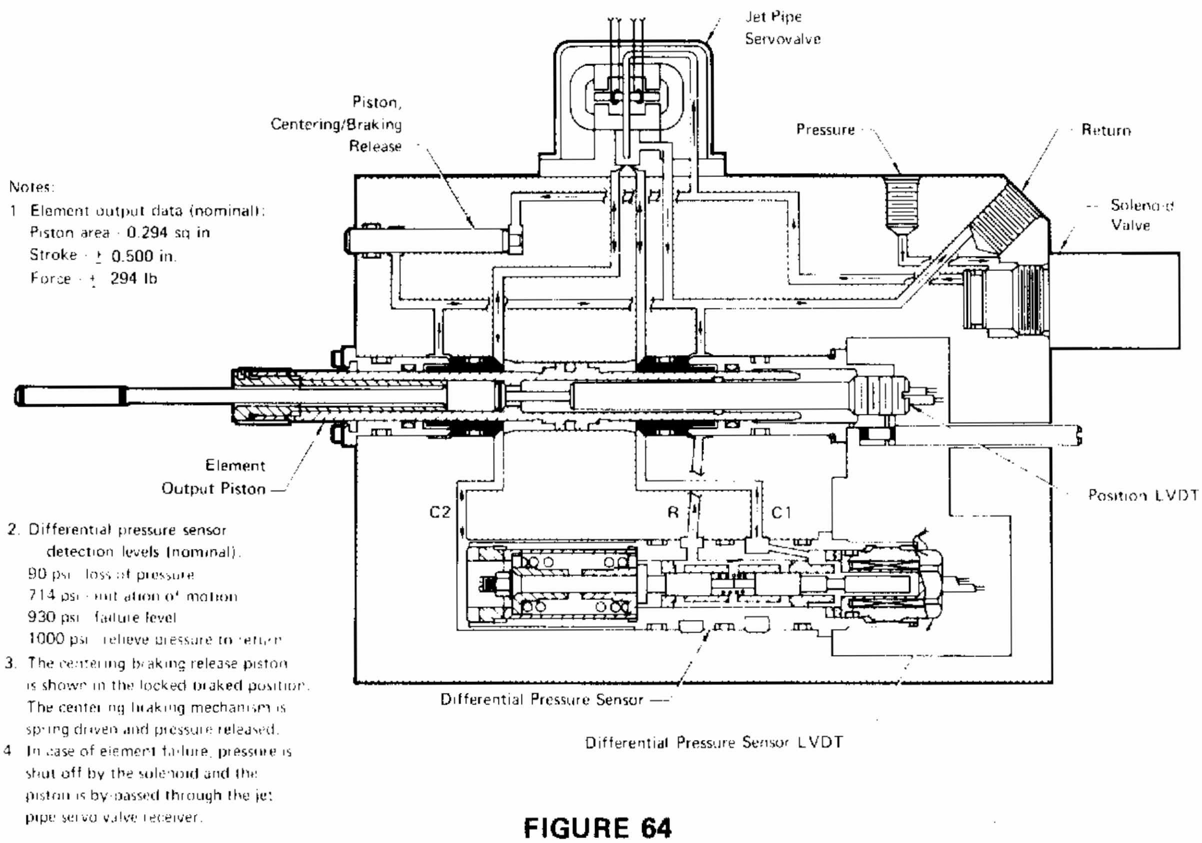


FIGURE 64
HYDRAULIC SCHEMATIC, SINGLE ACTUATOR ELEMENT

4. SIDE STICK CONTROLLER

The Side Stick Controller (SSC) to be used for the SFCS flight test program is designed to provide:

- o Pitch and roll output signals.
- o An adjustable position stick grip with neutral position lock.
- o Pitch Vernier control output signals.
- o Trigger Switch to switch from Normal to Electrical Back-Up mode.
- o Artificial feel forces.
- o Adjustable position arm rest.
- o Suitable mounting provision for forward and aft cockpit installation.

Figure 65 illustrates the layout of the SSC design. Table XV provides a summary of the artificial feel forces and grip geometry.

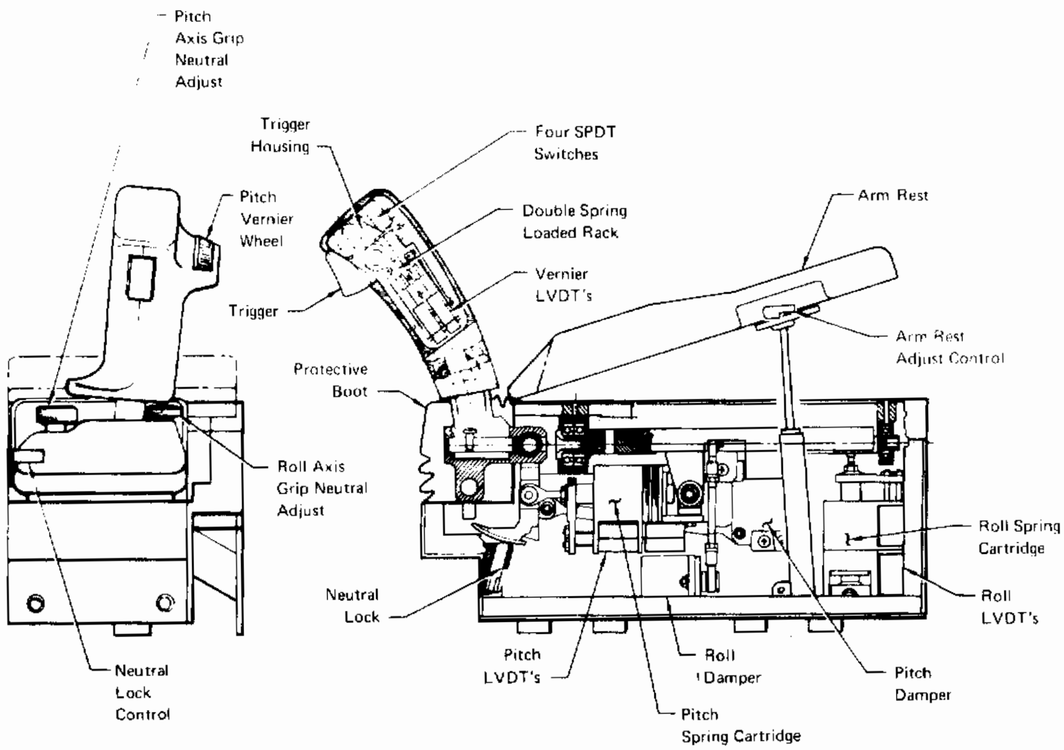


FIGURE 65
SFCS SIDE STICK CONTROLLER

a. Grip

The grip shape was contoured to provide comfortable and convenient access to the pitch vernier command thumbwheel and the trigger switch. The vernier wheel is for small maneuvering commands to allow the pilot to make small corrections when extreme accuracy is required, such as in weapon delivery and/or in-flight refueling. The vernier thumbwheel operates a double rack which is spring loaded to return to center. This mechanism moves the four LVDT cores for the vernier command signal. The trigger switch provides an output signal that is to be used to change the SFCES mode of operation from the normal mode to the electrical back-up mode. The trigger mechanism actuates four Single Pole Double Throw (SPDT) switches. Since inadvertent operation of this switch must be avoided, the action requires a positive 15 pound breakout force. This force is provided by a spring loaded ball detent. Controls are located at the base of the grip to allow the pilot to adjust the neutral position of the grip to the most comfortable position.

A neutral lock is provided to secure the grip in the neutral position and prevent inadvertent stick displacement when the controller is not in use.

b. Artificial Feel

Artificial feel forces, including breakout, spring gradient, and damping are provided in the Side Stick Controller. The breakout and gradient functions are provided by independent spring cartridges in the roll and pitch axes. The force levels mechanized were selected after evaluation of other side stick controllers, a development mock-up, and a man-in-the-loop simulation. The selected levels are 1.75 pounds breakout and 9.45 pounds hardover for pitch and 1.75 pounds breakout and 3.55 pounds hardover for roll. These forces may be varied within the following ranges by replacing springs and adjusting the initial spring load.

Axis	Breakout	Hardover
Pitch	1 - 7 pounds	6 - 20 pounds
Roll	1 - 6 pounds	3 - 10 pounds

The quadruplex LVDTs that provide the pitch output signal are mounted integrally with the pitch spring cartridge. The quadruplex LVDTs for the roll signal are integral with the roll spring cartridge.

Viscous devices are used for damping. The damping is normally set for one overshoot through neutral when the grip is placed hardover and then released. The damping range encompasses the range from zero damping to over damped.

c. Arm Rest

An adjustable position arm rest is provided to allow for variations in pilot physical size. The arm rest is padded and contoured to provide comfortable support to the pilot's arm and wrist. The arm rest position may be adjusted by rotating the arm rest adjust control and moving the rest to the desired position then allowing the control to return to the locked position.

d. Provision for Installation

The SSC unit is designed to be mounted in either the forward or aft cockpit. This may be accomplished by interchanging a plate on one side of the unit. Electrical connection to the SSC is by four identical cylindrical connectors on the bottom of the unit.

Additional information on the SSC is presented in Appendix VII , and by Reference 8.

5. SURVIVABLE STABILATOR ACTUATOR PACKAGE

The Survivable Stabilator Actuator Package (SSAP) is an electrically controlled and powered, dual tandem hydraulic surface actuator for positioning the F-4 stabilator. The package contains, as LRUs:

- a secondary actuator,
- a dual tandem surface actuator, and
- two integrated motor pump hydraulic power supplies.

The package is normally powered by the integrated hydraulic supplies, but has the capability of operating by using the aircraft central hydraulic supplies in a back-up mode. The integrated pump output pressure is routed to the dual tandem main control valve which meters the flow to the surface actuator as a function of the valve opening. The mechanical output motion from the secondary actuator is applied to the surface actuator input linkage. This motion is summed with the surface actuator mechanical feedback, and positions the main control valve as a function of the difference. The secondary actuator positions its output in response to electrical signals from the SFCES. The SSAP arrangement is shown by Figure 66, and the LRUs are described in the following paragraphs.

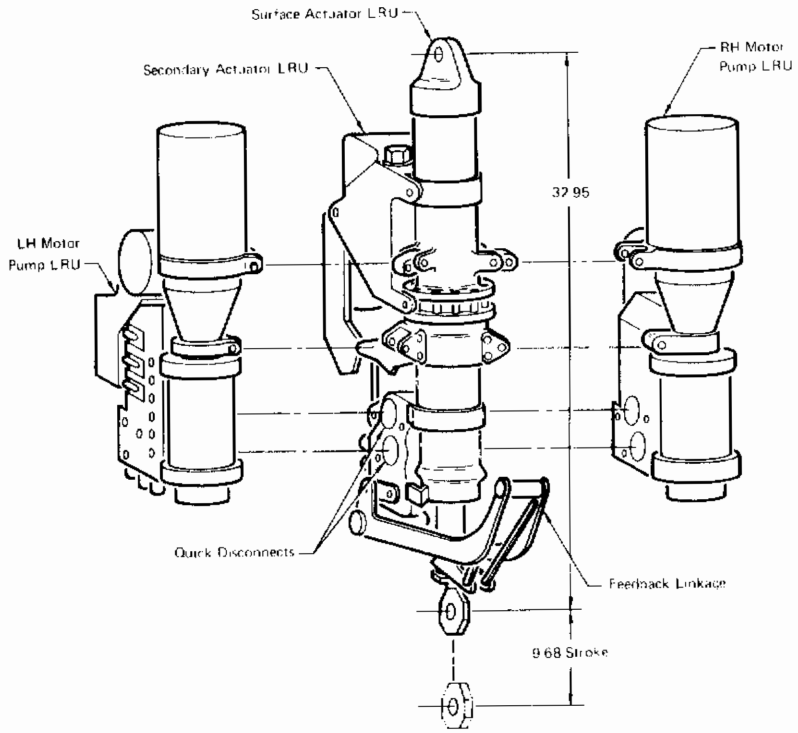


FIGURE 66
SSAP LRU ARRANGEMENT

a. Secondary Actuator LRU

The secondary actuator is a quadruplex electromechanical actuator. It utilizes four AC electric motors and associated gearing to convert up to four electrical signals from the SFCES to a single mechanical output which positions the main control valve of the surface actuator LRU. The rotary motion of the four AC electric motors is converted to linear motion at the single mechanical output by means of the differential gears, ballscrews, and linkage shown in Figure 67. Position transducers (LVDTs) are located at the single output point to provide position feedback signals to the servo amplifiers in the SFCES.

At the input to the servo amplifier within each CVU of the SFCES, the command signal from the signal selection device is summed with the corresponding secondary actuator position feedback signal. The resulting error signal is used to drive the pertinent AC motor in the secondary actuator. The velocity (speed and direction of rotation) of each motor is proportional to the error signal being received from its corresponding SFCES servo amplifier. The AC motor rotations are summed through the differential gears, ballscrews, and linkage to provide a linear velocity and/or position of the single mechanical output. The mechanical output continues to move until the sum of the position feedback signals cancel out the sum of the command signals and the mechanical output comes to rest. A schematic of the electromechanical secondary actuator is shown in Figure 67.

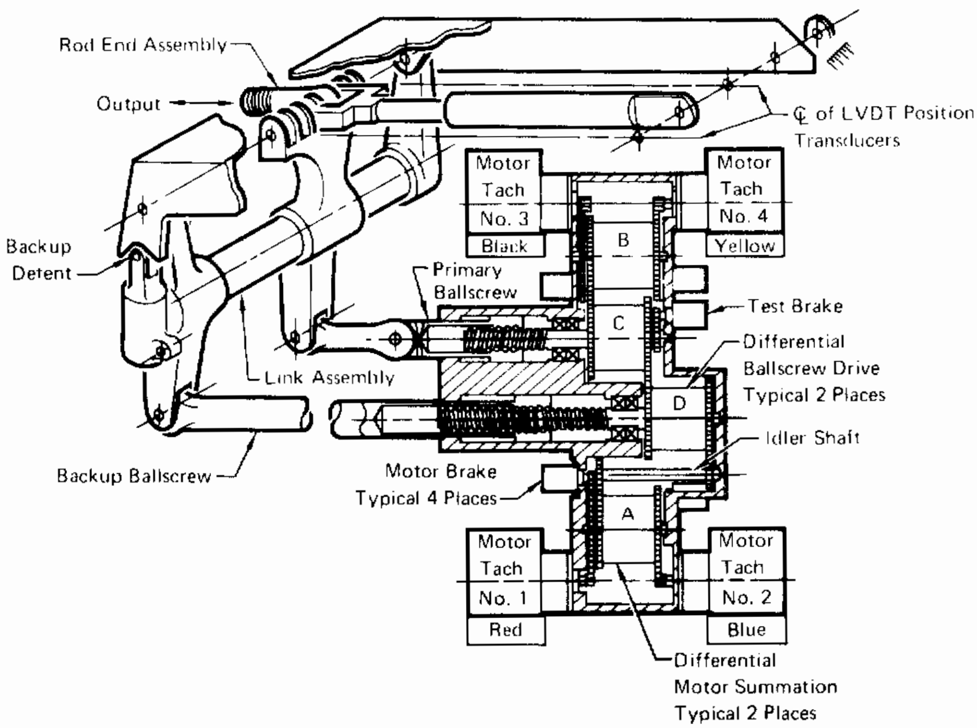


FIGURE 67

ELECTROMECHANICAL SCHEMATIC LTV-E SECONDARY ACTUATOR

As shown in Figure 67, the output velocities from Number 1 and 2 motors sum into the A differential; the output velocities from Number 3 and 4 motors sum into the B differential. A and B sum into differentials C and D. The primary and back-up ballscrews convert direction and velocity of rotation into linear velocity and/or position. During normal operation, when the output is transmitted via the primary ballscrew, the output from differential A is transmitted via the idler gear and differential D directly to the input of differential C. The detent on the back-up output ballscrew causes the output of differential D to remain nearly stationary. The output of differential C is splined to the primary ballscrew, which provides the output motion. Differential D provides an alternate output path in the event of a jam in the primary ballscrew. The pre-load of the detent, which is approximately seven percent of the capability of the actuator, holds the back-up ballscrew output near its center position. Therefore the output motion is normally through the primary ballscrew, since it has the lower impedance. The back-up screw has twice the stroke capability of the primary output to allow full output displacement authority even if the primary output jams in a hardover condition. A switch on the back-up ballscrew provides a signal which indicates when the back-up ballscrew has moved out of detent. This signal is used by BIT only to indicate failure or satisfactory operation of the back-up ballscrew.

In the event of a failure in one channel, the tachometer output of the failed channel will not agree with the other three. The in-flight monitoring circuitry will detect the failure and supply a signal to the SFCES logic. The SFCES will, thereafter, provide a null command to the motor in that channel and apply a brake to that motor. After two failures, the monitor simply compares the velocities of the remaining two motors. In the event that a disagreement occurs, a null command is supplied to both motors and both brakes are applied. A dynamic analysis of the secondary actuator is presented in Supplement 3.

Full output force and travel are available with any one or more of the four channels operating. Thus the change in output displacement for a given command signal is the same with one or more of the four channels operating. However, each time a channel is lost, the maximum output velocity is reduced by 25 percent of that achieveable with four channels operating. Thus the linear velocity of the secondary actuator output is proportional to the sum of the velocities of the individual motors. This is known as the velocity sum concept.

b. Motor Pump LRUs

The two Motor Pump units are self-contained hydraulic systems which supply hydraulic power to the surface actuator. They are functionally independent of other LRUs of the SSAP. The components which comprise each Motor Pump LRU are motor, pump, reservoir, switching valve, lock out valve, heat exchanger and blower, various switches, check valves, transducers, sensors and filters. That portion of the SSAP contained within each motor pump LRU is indicated on Figure 68. The

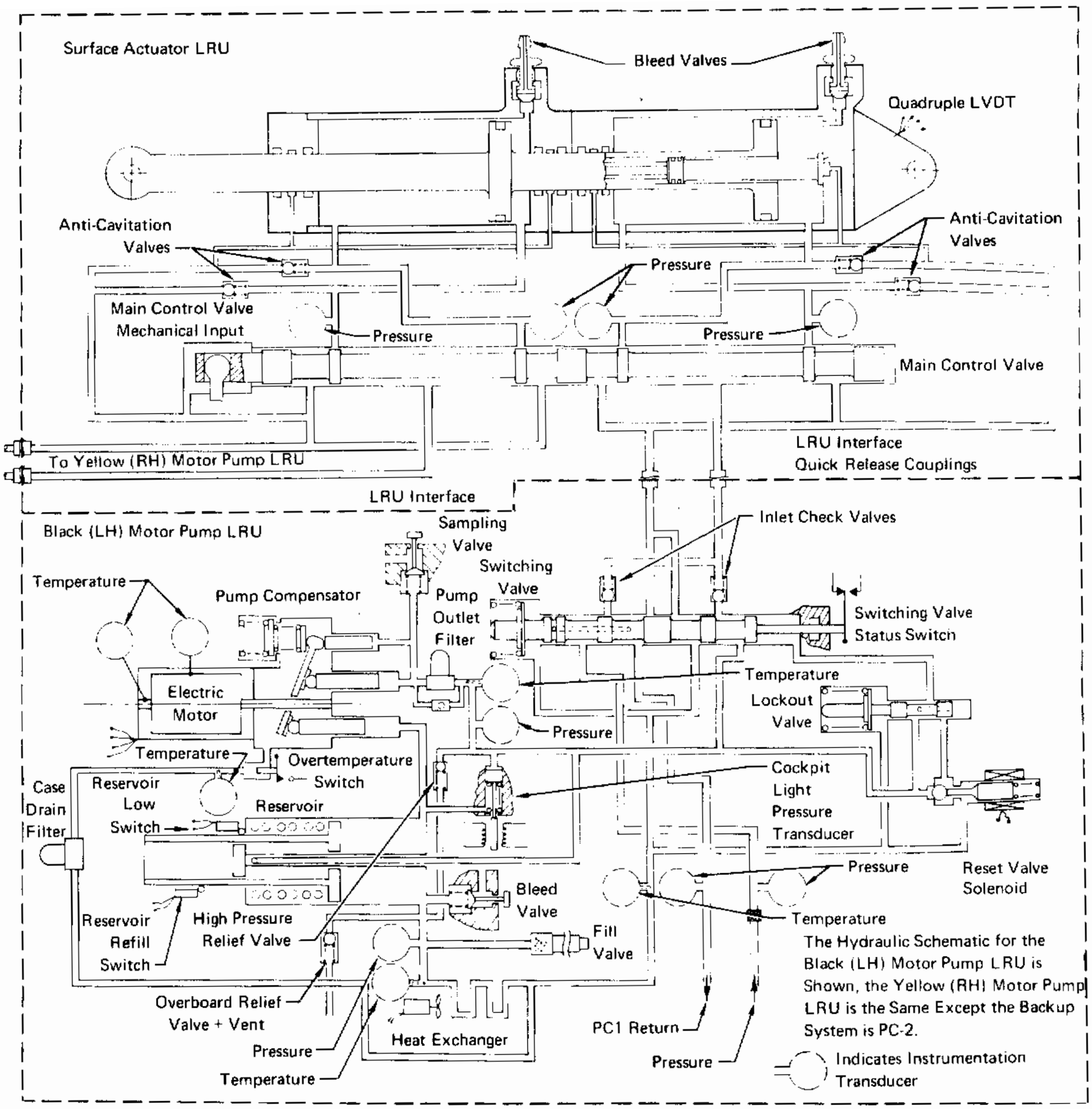


FIGURE 68
SCHEMATIC OF MOTOR PUMP & SURFACE ACTUATOR LRUs OF SSAP

function and description of the components that make up the Motor Pump LRU are discussed below.

Each motor pump LRU contains a 7.5 HP, 3 phase, 115-200 V, 400 Hz, electric motor operating at approximately 7,700 RPM. The in-line piston pump has a nominal displacement of 0.044 cubic inches per revolution, with an output flow-pressure characteristic as shown on Figure 69. The insert in this figure shows how the compensator piston force is balanced against the dual springs, A and B.

The reservoir is a spring loaded, bootstrap type with a displacement of 49 cubic inches at the refill level and 76 cubic inches when full. The full, refill and low condition of the reservoir is indicated visually by the reservoir piston and electrically by microswitches at the refill and low volume points.

The pump outlet filter, which filters to 25 micron absolute and 10 micron nominal, has a cleanable element and a ΔP indicator. The case drain filter has a rating of 15 micron absolute, and does not have a ΔP indicator. No bypass is provided for either filter.

A system relief valve is provided to limit system pressures in the event the pump compensator fails. An overboard relief valve is provided to protect the return system from overpressures caused by overfilling the reservoir.

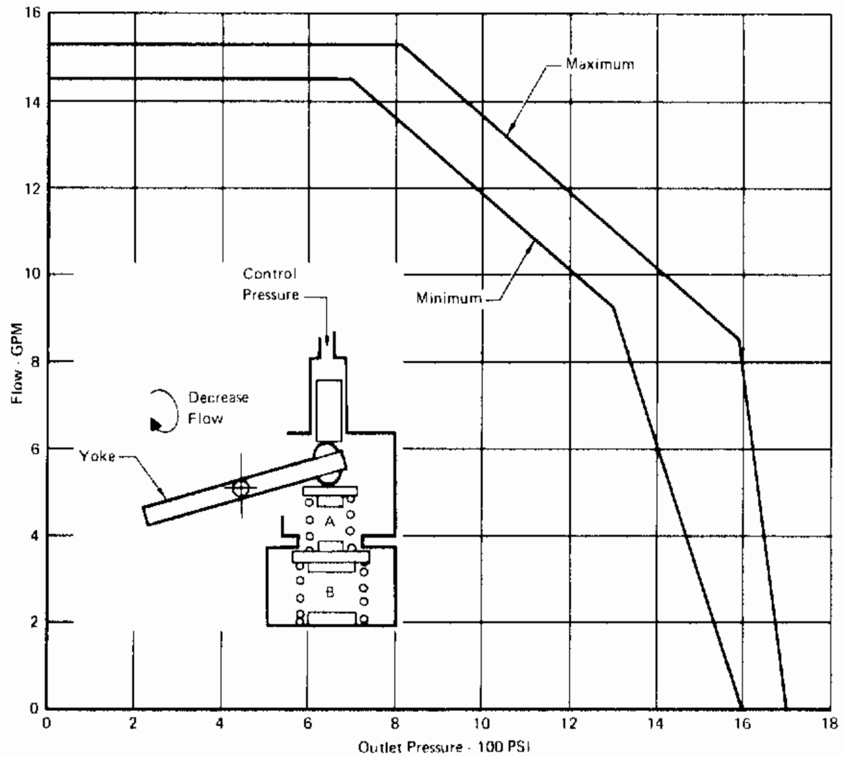


FIGURE 69
PUMP PRESSURE - FLOW CHARACTERISTICS

A pump output pressure transducer provides pressure information to the SFCES for processing. Low pressure is indicated on the SSAP hydraulic system status panel, along with low reservoir level and overtemperature warnings (see Figure 84).

A switching valve is provided to control the selection between normal and back-up hydraulic systems for each half of the main control valve. Three components are utilized to provide the switching function. They are: the solenoid reset valve, the lockout valve, and the switching valve. Figure 68 shows the solenoid reset valve in the deenergized position, the lockout valve in the normal position and the switch valve in the normal (motor pump) position.

Under start-up conditions, the switching valve, the lockout valve and the solenoid reset valve are spring offset into the position allowing the aircraft hydraulic system to supply the main control valve and illuminating the "SWITCH VALVE RESET" push button on the SSAP hydraulic system status panel.

With the motor pump running and operating normally, depressing the "SWITCH VALVE RESET" push button will energize the solenoid reset valve, cause the lockout valve and the switching valve to cycle, and switch off the light behind the push button. If the lockout valve has engaged correctly, the light will not come on again until the motor pump is switched off or fails. Pressure sensor outputs are used by integrator circuits within the SFCES to detect degraded motor pump performance. When the motor pump output fails to achieve the required pressure-time criteria the lockout valve cycles to the start up position, releasing the pressure at the end of the switching valve, causing it to cycle back to the start up position.

The fill valve is a quick release coupling with an integral filter. A filter which has a nominal rating of 40 micron protects the motor pump from contamination in the filling fluid. A bleed valve facilitates bleeding of the motor pump LRU. A sampling valve allows fluid samples to be taken for monitoring of fluid characteristics.

Quick release couplings connect the motor pump LRUs to the surface actuator and prevent fluid leakage when the LRUs are removed.

The heat exchangers have a core which is basically a rectangular flat plate design, and a ducted ventilating fan which draws air across the core and expels it outside the package.

The "OVER TEMP" light on the SSAP hydraulic systems status panel indicates when the case drain fluid temperature exceeds 462°F $\pm$ 12°F.

Temperature transducers which are used for flight test instrumentation are incorporated into the motor pump LRU at the pump outlet, case drain, switching valve return to the heat exchangers, heat exchanger outlet, motor bearing and motor stator.

Pressure transducers which are used for flight test instrumentation are incorporated into the Motor Pump LRU at the pump outlet, reservoir, aircraft supply pressure and aircraft supply return.

c. Surface Actuator LRU

The surface actuator moves the stabilator surface in response to metered hydraulic fluid from the main control valve. The dual tandem main control valve is positioned by a mechanical linkage system which moves in response to the position error between the secondary actuator output and the mechanical feedback from the surface actuator piston rod. The electromechanical secondary actuator is physically attached to the surface actuator and controls its position; however, it does not form any part of the hydraulic circuit. Quadruplex LVDT position transducers are mechanically connected to the surface actuator piston rod and provide stabilator position feedback information to the SFCES.

Figure 68 depicts the hydraulic arrangement of the surface actuator.

The dual tandem main control valve controls the flow of fluid to the surface actuator. The mechanical input to the valve is via a bell-crank through a rotary input shaft and seal arrangement.

Anti-cavitation valves allow flow around the piston to reduce the probability of cavitation in a surface actuator chamber.

Additional detail information on the SSAP is presented in Reference 4 .

6. MOBILE GROUND TEST FACILITY

In addition to the SFCES, Sperry is building a Mobile Ground Test Facility (MGTF) to provide closed loop testing to verify required performance of the complete SFCS installed in the aircraft. Closed loop testing will be accomplished prior to the first flight and prior to each flight where a modification or change to the system has been made. An analog computer will be provided to perform the closed loop testing. The testing will be accomplished by programming the aircraft longitudinal and lateral-directional equations of motion on the analog computer and utilizing actual aircraft surface positions as inputs to these equations. The acceleration and rate solutions to the equations of motion will be applied to the SFCES as though they were obtained from the system accelerometers and rate gyros. Inputs to the closed-loop system will be obtained by applying forces to the pilot controls in the normal manner or by applying electrical inputs directly to the SFCES. The analog computer will provide a high degree of flexibility for changing the programmed mechanization of the simulated aircraft equations of motion and motion sensor dynamics.

A recorder will be provided in the MGTF to record test data from the closed loop system tests.

The facility will be large enough to accommodate the SFCES and Side Stick Controller (SSC) test equipment, test benches, and three men to operate the equipment. The proposed interior layout of the MGTF is shown in Figure 70.

The analog computer, recorders and associated equipment required to accomplish the closed loop testing will be installed in the MGTF, which will have all essential lighting, heating and air-conditioning. A source of 277/480 VAC, 3 phase, 4 wire, 60 Hertz power will be required for the operations in St. Louis or at a remote facility such as Edwards Air Force Base. The facility will have power conversion equipment which can supply the electrical power in the form required for operation of the SFCES and SSC during bench test. All interconnecting cables between the aircraft and the facility necessary for the closed loop testing of the installed SFCS are included as part of the facility. A communication system between the facility and the area surrounding the aircraft is provided. The trailer will have sufficient mobility to be routinely towed to the side of the parked aircraft by an aircraft tug and, when necessary, towed cross-country by an over-the-road tractor unit.

This facility will also serve as a maintenance station where minor electronic modifications, calibrations, LRU test, troubleshooting, and repairs can be performed in close proximity to the aircraft.

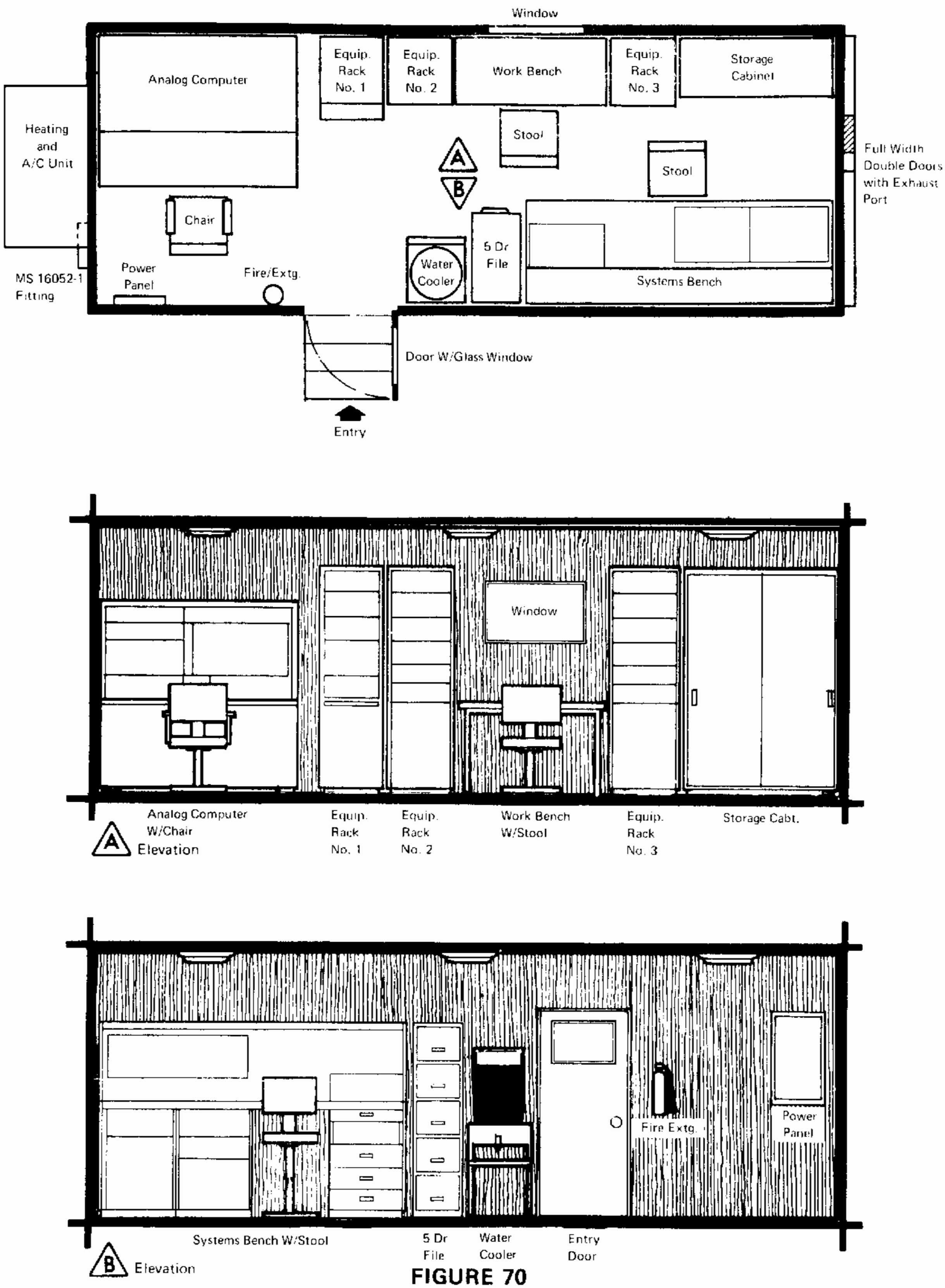


FIGURE 70

INTERIOR LAYOUT OF MOBILE GROUND TEST FACILITY

SECTION V

SYSTEM MODIFICATIONS AND INSTALLATIONS

1. GENERAL

In addition to installing the major procured equipment (except the MGTF), described in Section IV, considerable modifications to the existing test aircraft systems, structure and cockpit are planned, and installation of additional equipment is expected to be made to help accomplish the objectives and requirements of the SFCS program. These modifications and installations are described in this section.

The test aircraft to be used in Phase II of the SFCS program is F-4 S/N 62-12200. The aircraft was originally produced as an F-4B Phantom II and later modified to a YRF-4C. The aircraft was subsequently modified to a YF-4E for a prototype gun installation. This modification included the installation of a gun support structure in the forward fuselage and local strength improvements. The moldlines, compartments, doors, etc., are equivalent to the RF-4 aircraft. The general arrangement is shown by Figure 71.

Subsequently the aircraft was modified for Simplex flight testing, Phase I of the SFCS program. The Simplex flight test program required the incorporation of certain structural and equipment changes to provide a suitable configuration. Reference 1 provides a description of the changes which were accomplished for Phase I of the SFCS Program. The Simplex actuator package has subsequently been removed.

For Phase II of the SFCS program additional structural and equipment changes will be required. New hydraulic and electrical power sources and distribution systems must be designed and installed in the test aircraft to provide the redundancy required by the two-fail-operate concept and to improve the reliability. The studies and analyses presented in this report provide the rationale for the modification and installations described in this section.

Descriptions are provided in the following sub-sections of:

- Flight Control Systems
- Hydraulic System
- Electrical System
- Cockpit
- Other Equipment Installations

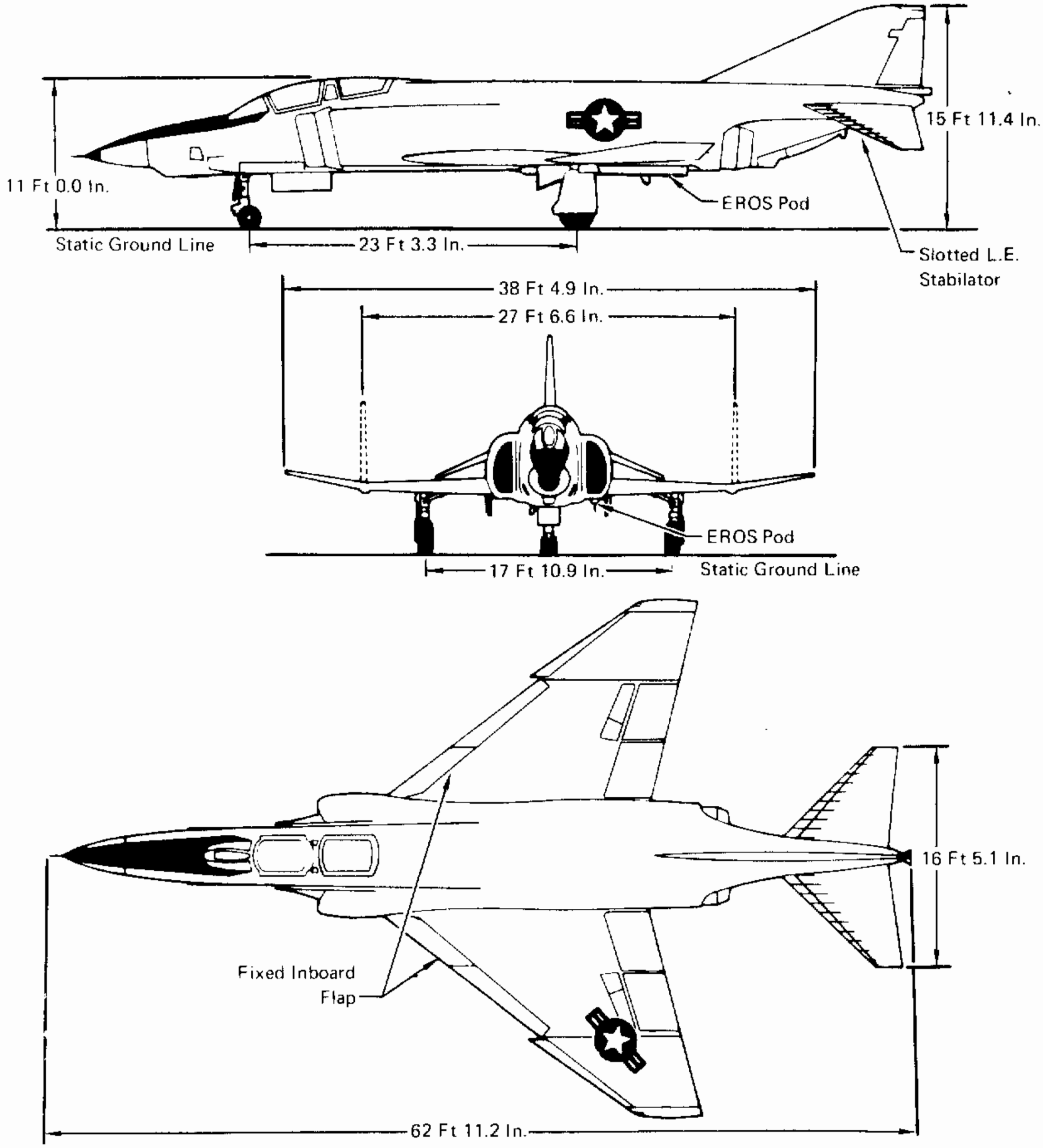


FIGURE 71

GENERAL ARRANGEMENT, F-4 AIRCRAFT, AIR FORCE S/N 62-12200

2. FLIGHT CONTROL SYSTEMS

During Phase IIA of the SFCS program, the longitudinal and directional axes of the SFCS will be backed up with a mechanical control system. The lateral SFCS will be installed in Phase IIA and is designed to remain unchanged throughout the remainder of the program. For Phase IIB, after sufficient confidence in the SFCS has been acquired, both mechanical back-up systems will be deactivated. In Phase IIC the Survivable Stabilator Actuator Package will be combined with the SFCES in the longitudinal axis to produce the final configuration of the Survivable Flight Control System. A description of the flight control system for each phase of the SFCS program is provided below.

a. Phase IIA

For Phase IIA of the SFCS program, each control system is envisioned to be as proposed in the following paragraphs:

(1) Longitudinal Control System

The mechanical longitudinal control system will remain basically the same as the production F-4. The stabilator control surface actuator will be modified by the addition of the stabilator surface position transducer. A Mechanical Isolation Mechanism (MIM) will be installed to provide the means for selecting the SFCS or mechanical back-up mode of operation. The secondary actuator, which accepts electrical signals from the SFCES, will be connected to one input of the MIM. The control cables from the pilot's stick will be connected to the other input of the MIM. The present F-4 feel-trim system will be removed and a simplified feel system will be installed in the aft cockpit. The feel system, which is in parallel with the basic control system, will consist of a feel spring cartridge, safety spring cartridge, trim actuator and eddy current damper. Figure 72 illustrates the longitudinal control system design for Phase IIA.

(2) Lateral Control System

The F-4 has demonstrated the ability to return to base and land safely after the complete loss of the lateral control system. Consequently, the lateral control system will be modified to the SFCS configuration in the initial stage of Phase II and no mechanical back-up system will be installed. The lateral control system will be designed so that it can be de-activated and the control surfaces permitted to return to a neutral position in the event of multiple failures. Figure 27 shows the location of the switch on the left console. The mechanical linkage from the cockpit to the wing will be deleted and a secondary actuator will be introduced into the system near the wing root at the last point common to both the spoiler and aileron actuators. The production F-4 lateral feel-trim system will be removed and a new feel spring cartridge will be added as closely coupled to the control stick as practical. No mechanical trim system will be provided. Figure 73 illustrates the lateral control system

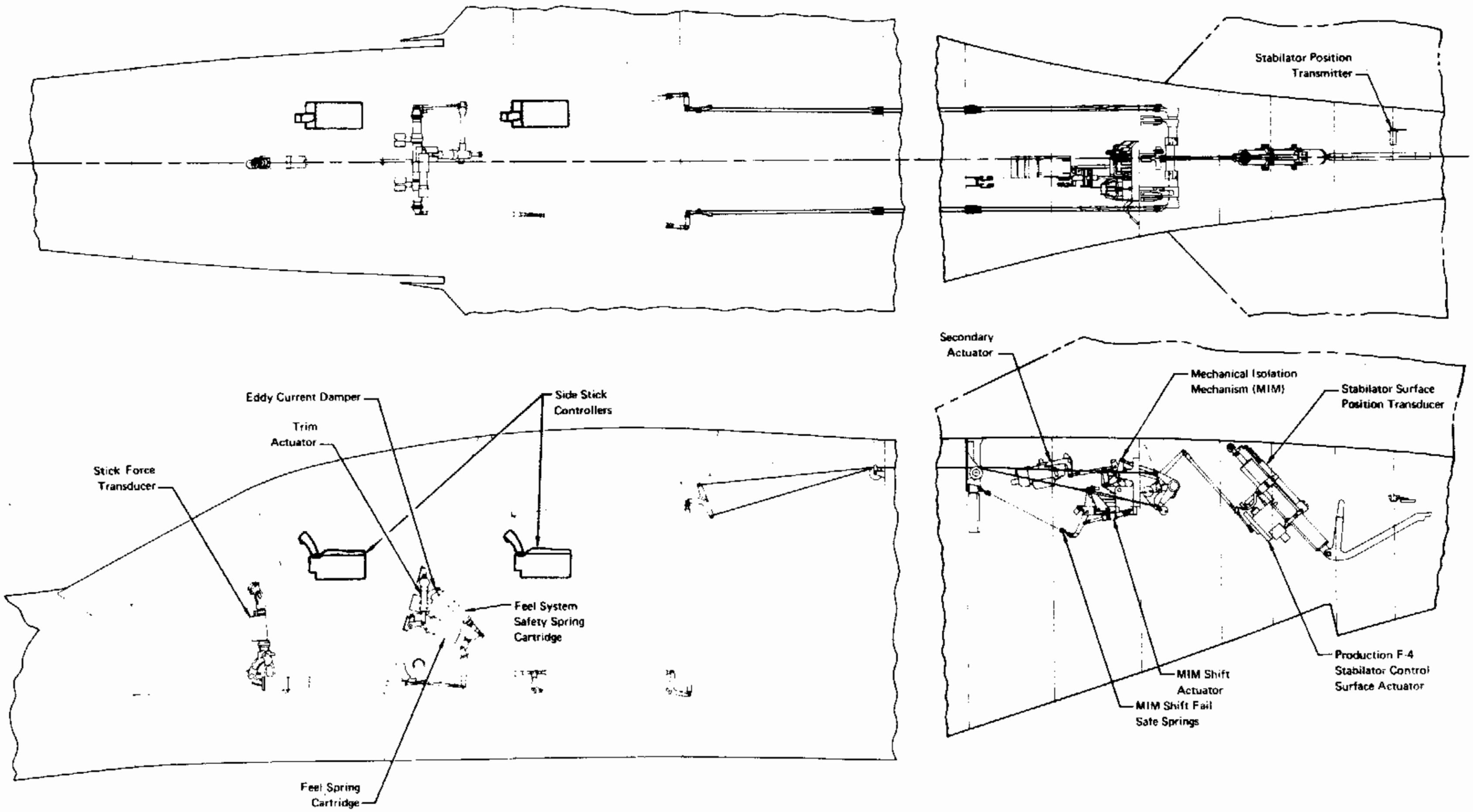


FIGURE 72

PHASE IIA LONGITUDINAL CONTROL SYSTEM

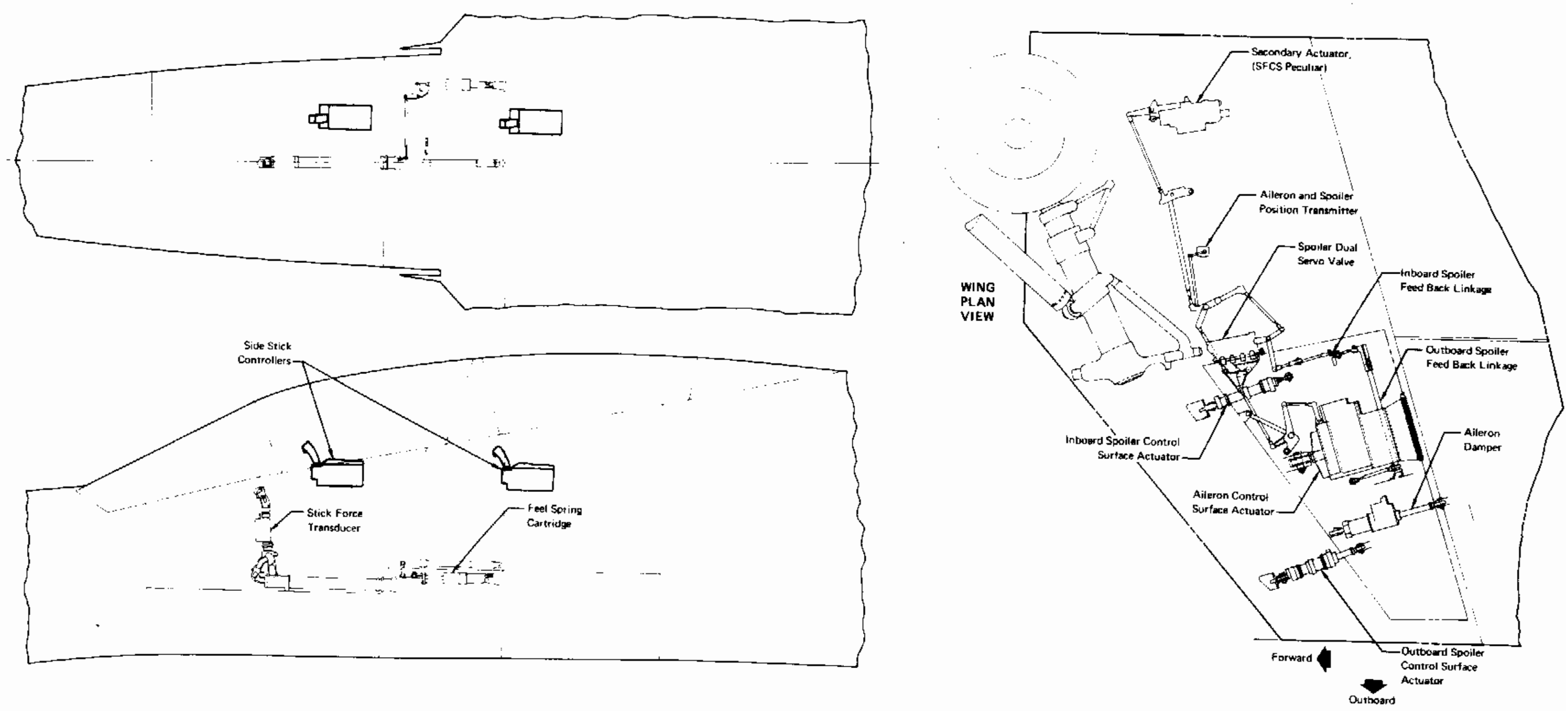


FIGURE 73

PHASE IIA, IIB AND IIC LATERAL CONTROL SYSTEM design for Phase II.

(3) Directional Control System

The directional control system will use the production F-4 control surface actuator. A MIM, which is similar to that installed in the longitudinal system, will be added. A secondary actuator, which accepts electrical signals from the SFCES and provides input motion to the input bellcrank of the MIM, will be installed. The production F-4 directional feel-trim system, located in the aft fuselage, will be replaced with a feel spring cartridge connected to the rudder pedal. The rudder pedals will be retained in both cockpits. The test airplane does not have braking capability in the aft cockpit and none will be added. The stall warning pedal shaker will be removed from the front cockpit rudder pedals because it would introduce spurious signals into the SFCES. Figure 74 illustrates the directional control system design for Phase IIA.

(4) Control Stick and Throttle

The control stick in the forward cockpit will remain the same as the production F-4 except that the production AFCS stick force transducer will be removed and replaced with a quadruplex control stick force transducer. The transducer will sense the force inputs by the pilot to both the lateral and longitudinal axes and provide the input signals to the SFCES. The longitudinal trim switch, used to trim the mechanical control system, will be the same as the production F-4. The function of the longitudinal trim switch will be deleted for Normal and electrical back-up mode operation. The function of the lateral trim switch will be deleted. The aft center stick will be removed. Side Stick Controllers (SSCs) will be installed in both cockpits for fly-by-wire control. The problems of inadvertent inputs to an aft cockpit center stick containing a force transducer make the SSC a more desirable aft cockpit controller for this test program, since it includes a neutral lock. Throttles will be added to the aft cockpit.

b. Phase IIB

For Phase IIB of the SFCS program, the MIM will be de-activated in both the longitudinal and directional systems and the SFCS secondary actuators will be connected directly to the control surface actuators. All mechanical linkage including bellcranks, control rods, and control cables, between the MIM and the pilot controls in the cockpit are planned to be removed. Figure 75 illustrates the longitudinal control system design and Figure 76 illustrates the directional control system design for Phase IIB. The lateral control system for Phase IIB is the same as that used in Phase IIA.

c. Phase IIC

For Phase IIC of the SFCS program, only the longitudinal control

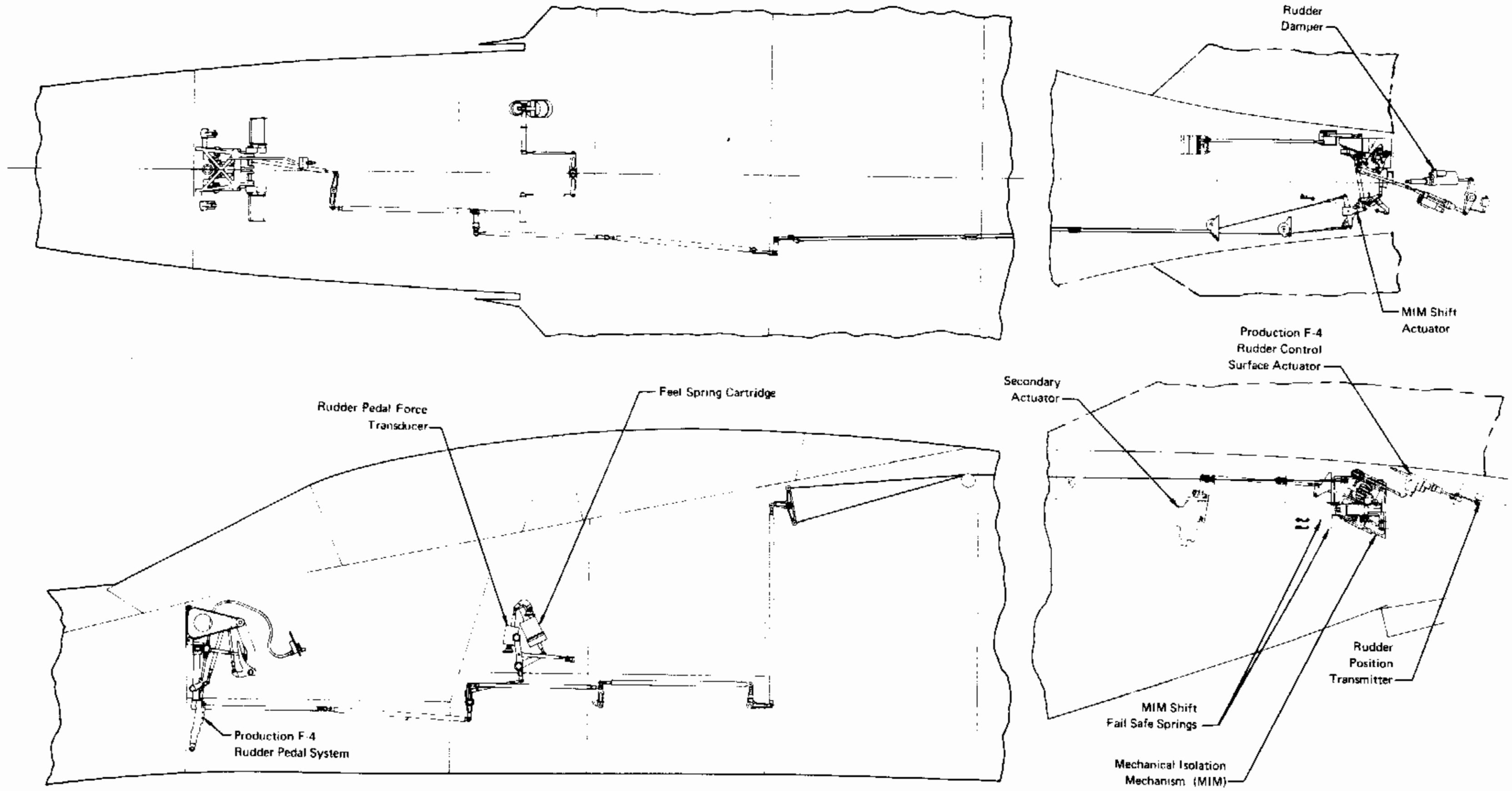


FIGURE 74
PHASE IIA DIRECTIONAL CONTROL SYSTEM

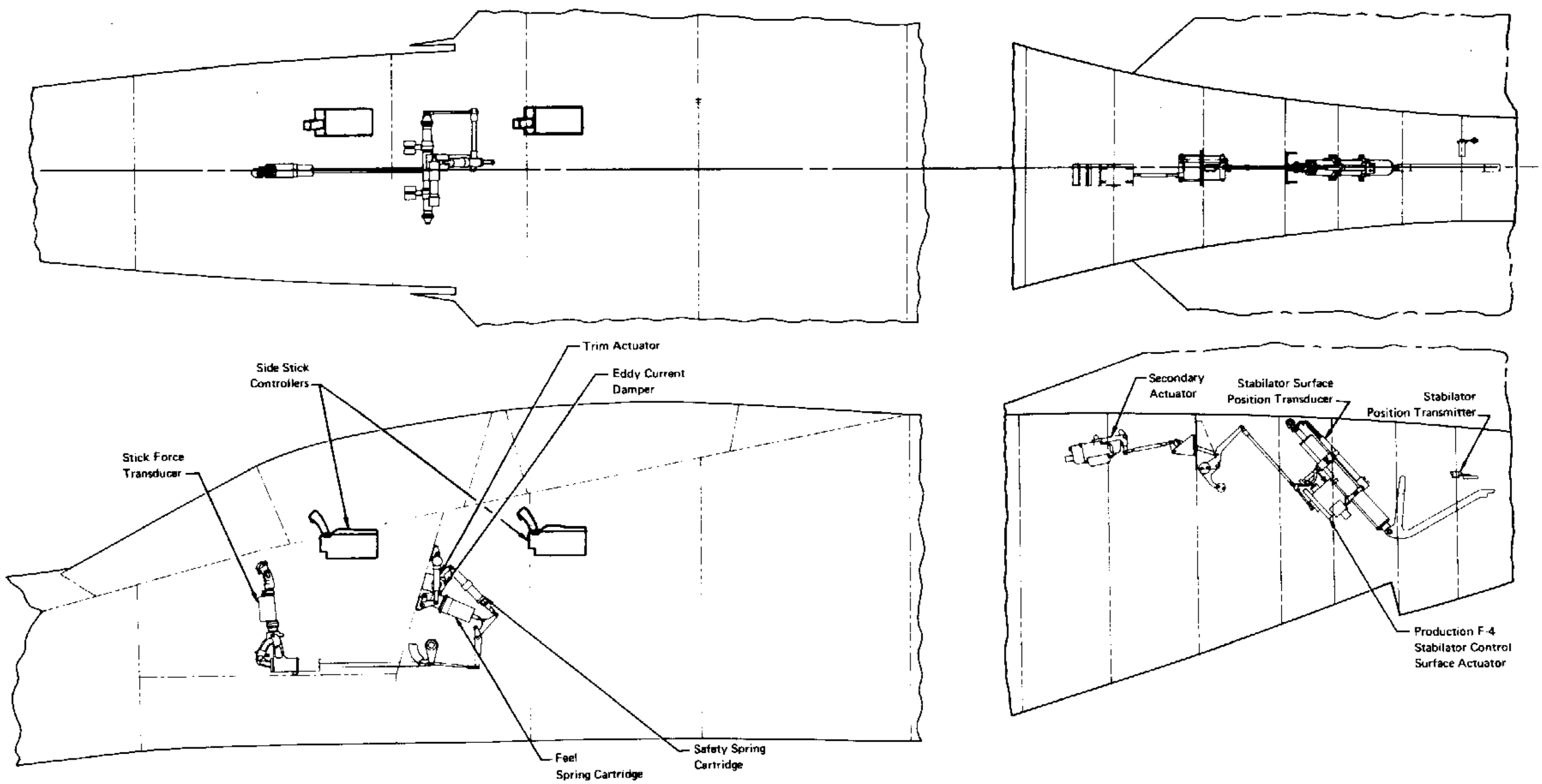


FIGURE 75

PHASE IIB LONGITUDINAL CONTROL SYSTEM

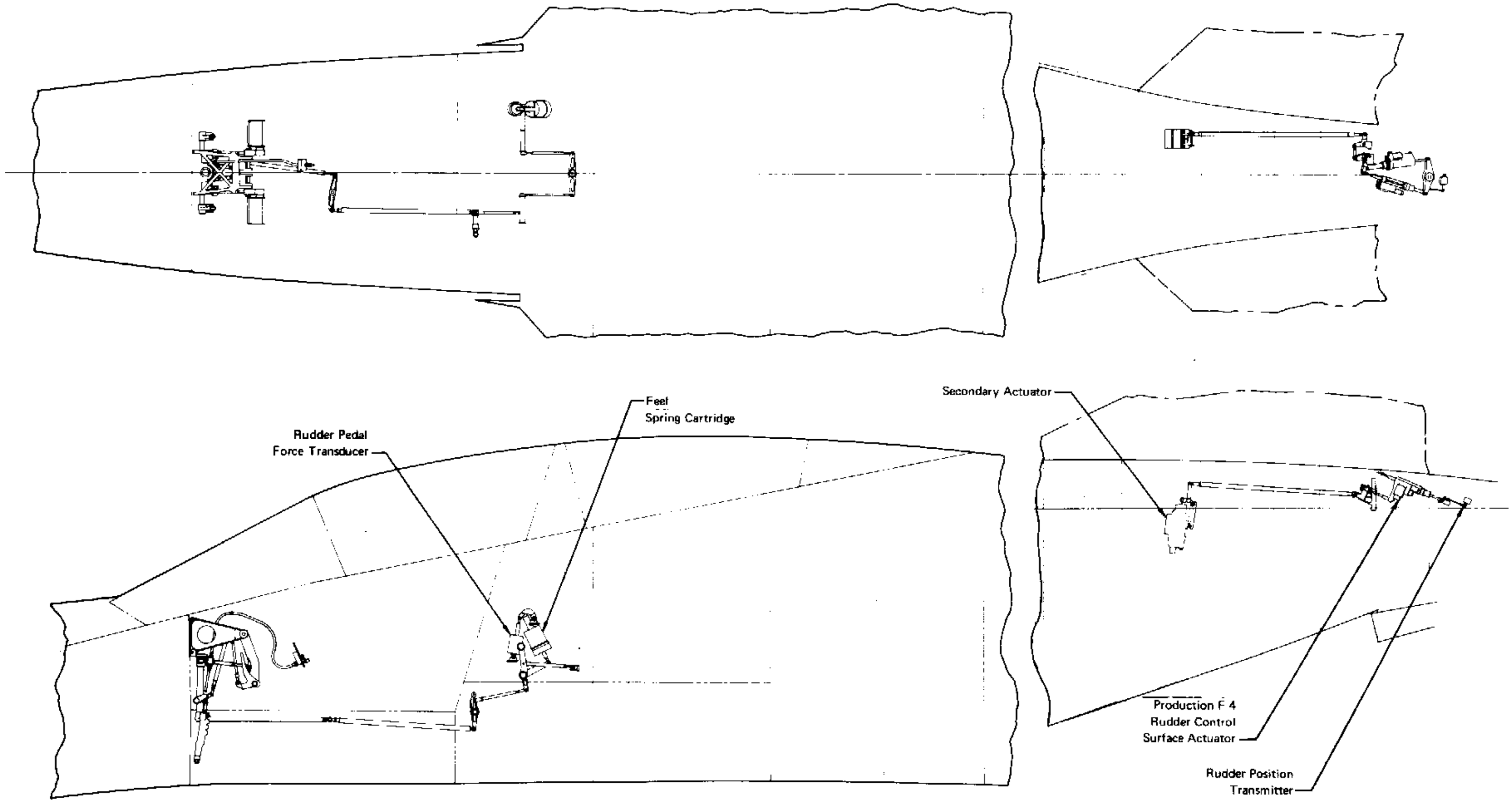


FIGURE 76

PHASE IIB AND IIC DIRECTIONAL CONTROL SYSTEM

system will be modified. The longitudinal secondary actuator and its mechanical linkage to the control surface actuator will be removed. The production F-4 surface actuator also will be removed and replaced with the Survivable Stabilator Actuator Package (SSAP). Figure 77 illustrates the longitudinal control system design for Phase IIC. Both the directional and lateral control systems are designed to remain the same as in Phase IIB.

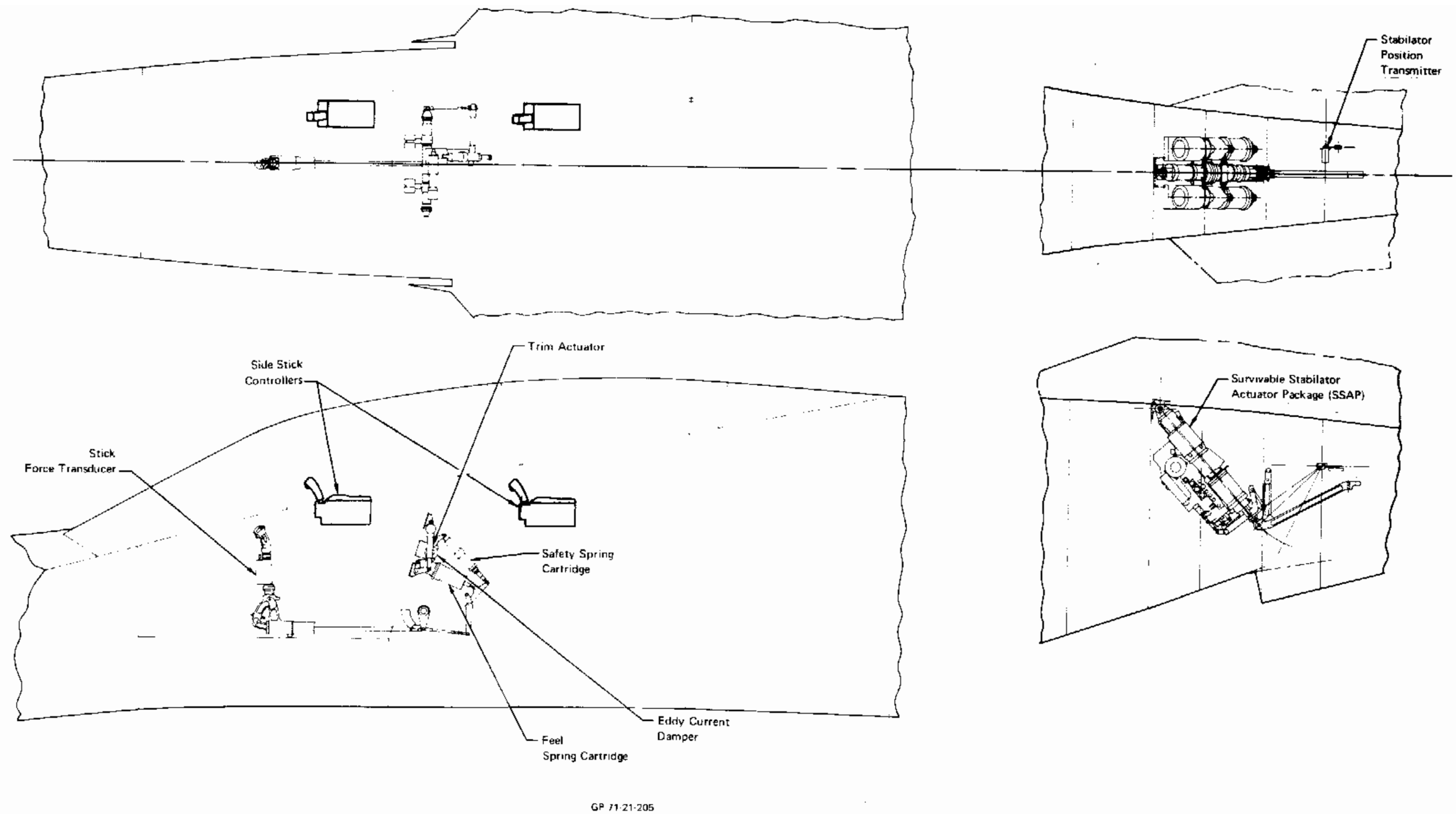


FIGURE 77

PHASE IIC LONGITUDINAL CONTROL SYSTEM

3. HYDRAULIC SYSTEM

The hydraulic components for all phases of the SFCS program will be designed to be compatible with the various systems and components with which they are required to interface. The hydraulic systems installations will be conventional. New and modified lines will be constructed of stainless steel tubing. High temperature fluid is not required in the hydraulic systems until the installation of the SSAP; however, it is planned to install MIL-H-83282 (MLO 68-5) fluid in the PC-1 and PC-2 hydraulic systems before the Phase IIA modification layup in order to gain operational experience with this fluid. A fourth hydraulic system, powered by an electric motor pump, and the aircraft Utility hydraulic system will continue to use MIL-H-5606 fluid since these systems will not be exposed to the higher operating temperatures of the SSAP. Existing hydraulic subsystems, not associated with the primary flight control systems, will not be modified during the SFCS program.

a. Phase IIA

The presently planned Phase IIA configuration of the SFCS hydraulic system is represented schematically in Figure 78. The major changes from the F-4 systems are as follows:

(1) A fourth hydraulic system, powered by an electrically driven motor pump unit located in the aft fuselage, will be added.

(2) The individual elements of the four SFCS secondary actuators are powered by the three central hydraulic systems and the added fourth hydraulic system.

(3) MIM actuators are to be powered by the Utility system.

b. Phase IIB

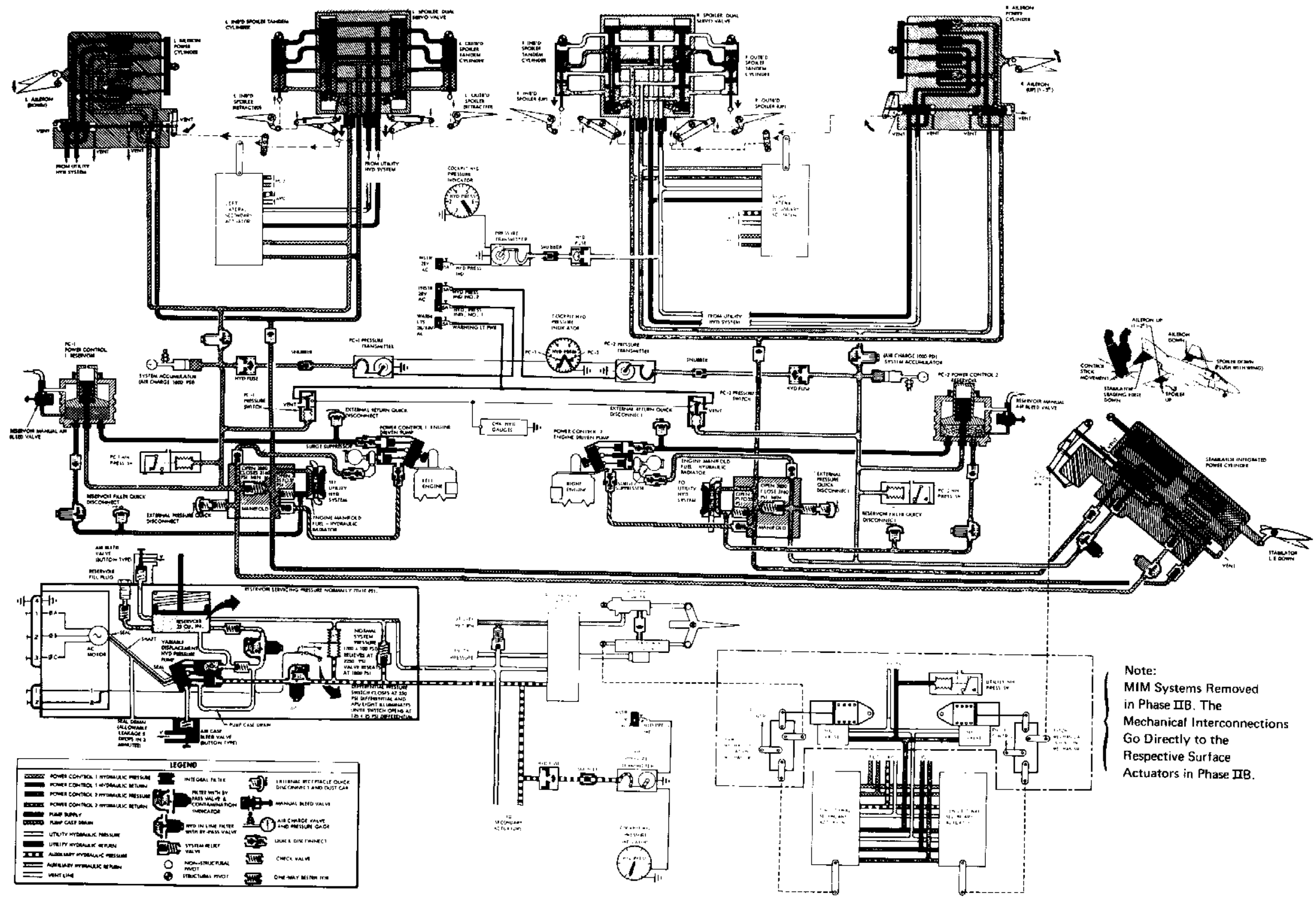
The planned configuration of the hydraulic system design for Phase IIB is represented schematically in Figure 78. This configuration is identical to Phase IIA except the circuitry and components required for actuation of the Mechanical Isolation Mechanism are deleted.

c. Phase IIC

The planned configuration of the hydraulic system design for Phase IIC is represented schematically in Figure 79. The major differences between this system and the Phase IIB system are as follows:

(1) The stabilator secondary actuator and its plumbing will be deleted.

(2) The production stabilator actuator will be replaced by the SSAP, and the PC-1 and PC-2 hydraulic systems will be connected into the SSAP.



Note:
MIM Systems Removed in Phase IIB. The Mechanical Interconnections Go Directly to the Respective Surface Actuators in Phase IIB.

FIGURE 78
SFCS FLIGHT CONTROL HYDRAULIC SYSTEM PHASE IIA AND IIB

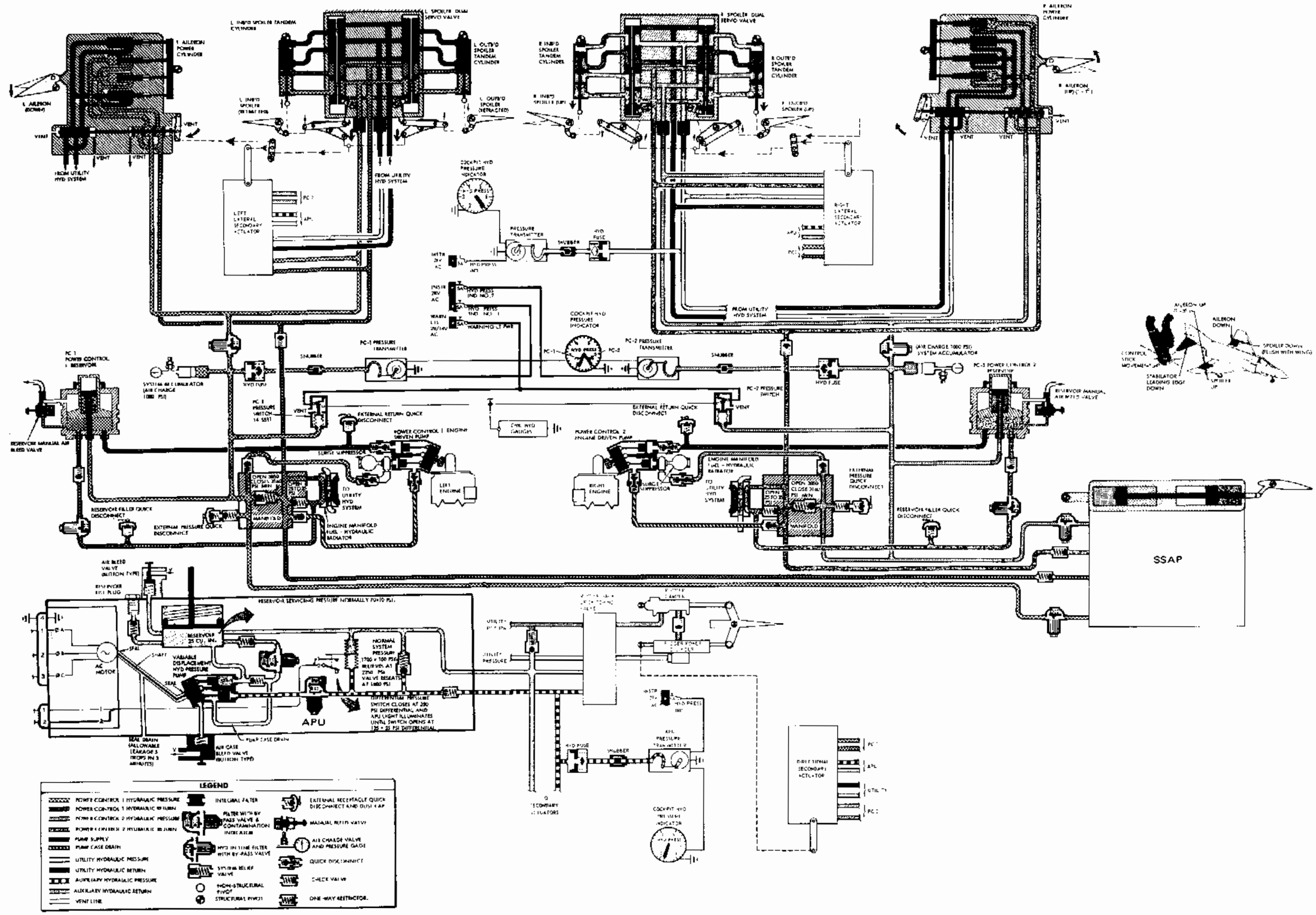


FIGURE 79

SFCS FLIGHT CONTROL HYDRAULIC SYSTEM PHASE II C

4. ELECTRICAL SYSTEM

The planned design of the SFCS electrical system installation is based on criteria established by the Electrical System design studies presented in Section III of this report. The design of the electrical power generation, distribution, and control system and the wiring installation is expected to adhere to production F-4 design and installation practices; however, the design for this test aircraft is planned to be modified as described below to, in general, provide a safer and more reliable system capable of meeting SFCS program objectives.

a. Electrical Power Generation, Distribution, and Control

The electrical power system installation, as currently envisioned, is illustrated in Figure 80.

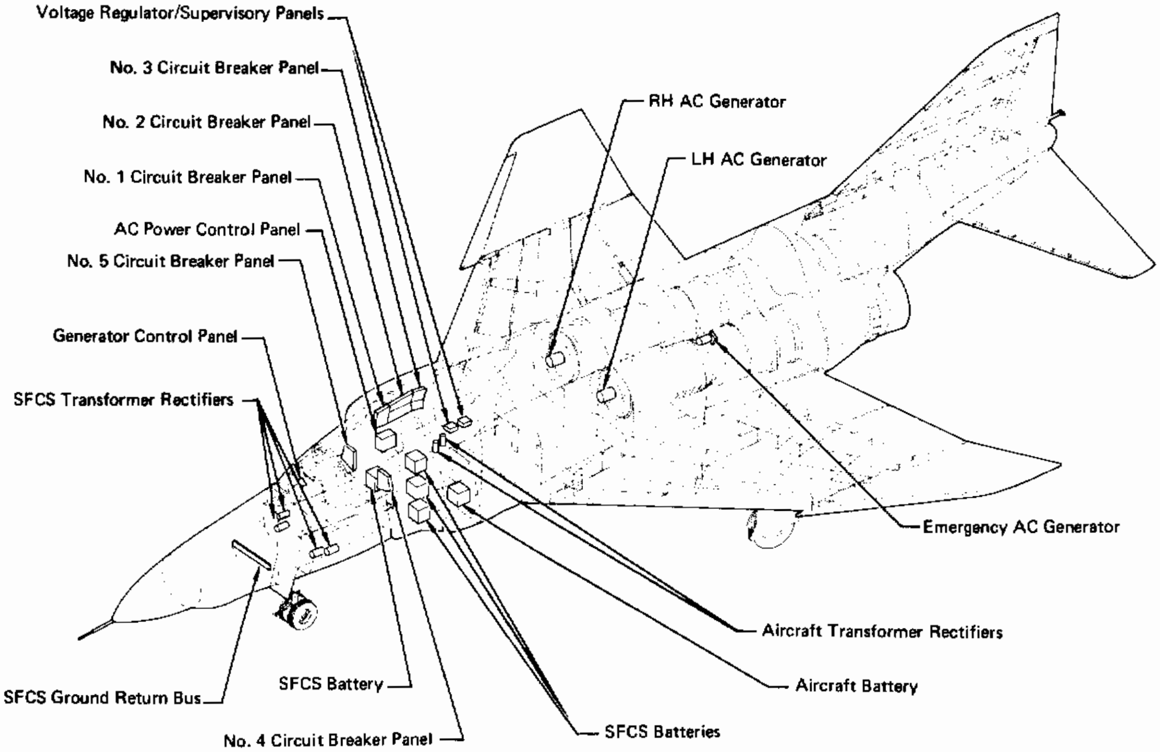


FIGURE 80

SFCS ELECTRICAL POWER SYSTEM INSTALLATION

(1) Primary Electrical Power

The aircraft primary electrical power is 115/200 volt, 3-phase, 400 Hz. This power is normally provided by two 30 KVA oil cooled AC generators, one in each engine air inlet dome. Each generator is controlled and regulated by a separate Voltage Regulator and Supervisory Panel (VR/SP). The VR/SPs, in conjunction with manually operated switches located on the Generator Control Panel, control and monitor the generating system. These units also control the bus switching and generator contactors in the AC Power Control Panel.

(2) Emergency Electrical Power

The aircraft's emergency electrical power is also 115/200 volt, 3-phase, 400 Hz. This power is furnished by a single 3 KVA ram air turbine driven generator capable of supplying the essential AC and DC bus loads. The emergency generator's ram air turbine is normally retracted within the aircraft, and can be extended into the airstream by the pilot, following a double generator failure or failure of the LH generator and bus-tie circuitry.

(3) DC Power System

The aircraft's DC power system consists of two 100 ampere, nominal 28 VDC Transformer/Rectifiers (T/Rs) located in the aft cockpit and a single 11 ampere-hour nickel-cadmium battery located below the cockpit floor in the left Side Looking Radar (SLR) equipment bay.

(4) SFCS DC Power System

The SFCS DC power system consists of four separate, redundant power supplies. Each supply consists of a 100 ampere 28 VDC T/R shunted by a 22 ampere-hour nickel-cadmium battery. Two T/Rs are located in each SLR antenna bay. Three of the SFCS batteries are located in the SLR equipment bays. The fourth SFCS battery is located in the infrared (IR) bay. The output of each SFCS DC supply terminates in Circuit Breaker Panel No. 5 located on the right-hand side of the forward cockpit. The SFCS ground return bus, located in the nose of the aircraft, provides a single tie point for all SFCS DC and signal ground returns. The ground return bus is hardwired to the negative terminals of the SFCS batteries and T/Rs.

(5) Power Distribution System

The power distribution system consists of main load buses and pilot accessible auxiliary load buses. The main load buses are located in Circuit Breaker Panels Nos. 1, 2 and 3 in the aft cockpit. The pilot-accessible auxiliary load buses are located in Circuit Breaker Panel No. 4 which will be added to the left-hand side of the forward cockpit.

It is intended that failure of any single component in the aircraft electrical distribution system will not result in the loss of more than one channel of the SFCS. Physical isolation of SFCS wiring, channel-to-channel, and from the rest of the aircraft's wiring, is to be accomplished by the use of COMPACT wiring techniques where each channel will be isolated from all other channels and from all present aircraft wiring. In general, the SFCS electrical installation is to consist of four separate routing paths throughout the aircraft. However, ideal dispersion of wiring could be incorporated into the entire aircraft only if the concept was considered in the aircraft's initial design. Since the F-4 airframe was not initially designed to include quadruplex wire routing paths, design criteria have been established that will assure maximum dispersion of the SFCS wiring consistent with the physical limitations of the SFCS test aircraft.

b. Wiring Installation

(1) Test Aircraft Rewiring

The test aircraft was previously equipped with numerous equipments which are not now installed and will not be required for the SFCS program. Much of the interconnect wiring for this equipment remains in the test aircraft. The remaining equipment components, not required for this program, will be removed. All unused wiring is planned to be removed or modified to provide a cleaner, more reliable and safer initial configuration of the aircraft from which the installation of the SFCS may proceed in an orderly and efficient manner.

(2) SFCS Wiring

The general installation and routing paths for the SFCS wiring are shown in Figure 81. The installation will be in accordance with MCAIR Process Specifications for Marking, Installation, Identification, Fabrication Modification, and others as applicable which have been previously approved by the Air Force for use on the F-4 aircraft. Twisting, shielding, and separation of wiring are planned to minimize electromagnetic interference (EMI) within the SFCS and with existing aircraft systems. Strict separation of SFCS wiring and exterior lighting wiring, pitot heater wiring, and antenna feed lines are expected to minimize possible damage to the SFCS from lightning strike. All signal returns are to be hardwired; random grounding of signal returns will not be used.

The electrical wiring is expected to be routed from the SFCES Units located in the nose to the forward cockpit through bushings in the forward bulkhead. SFCS bundles will also be routed through the forward fuselage to controls, displays and equipment in the aft cockpit. The SFCS bundles from the aft cockpit to the aft fuselage will be routed via the upper controls troughs above and outboard of the fuel cells to the SFCS components in the aft fuselage.

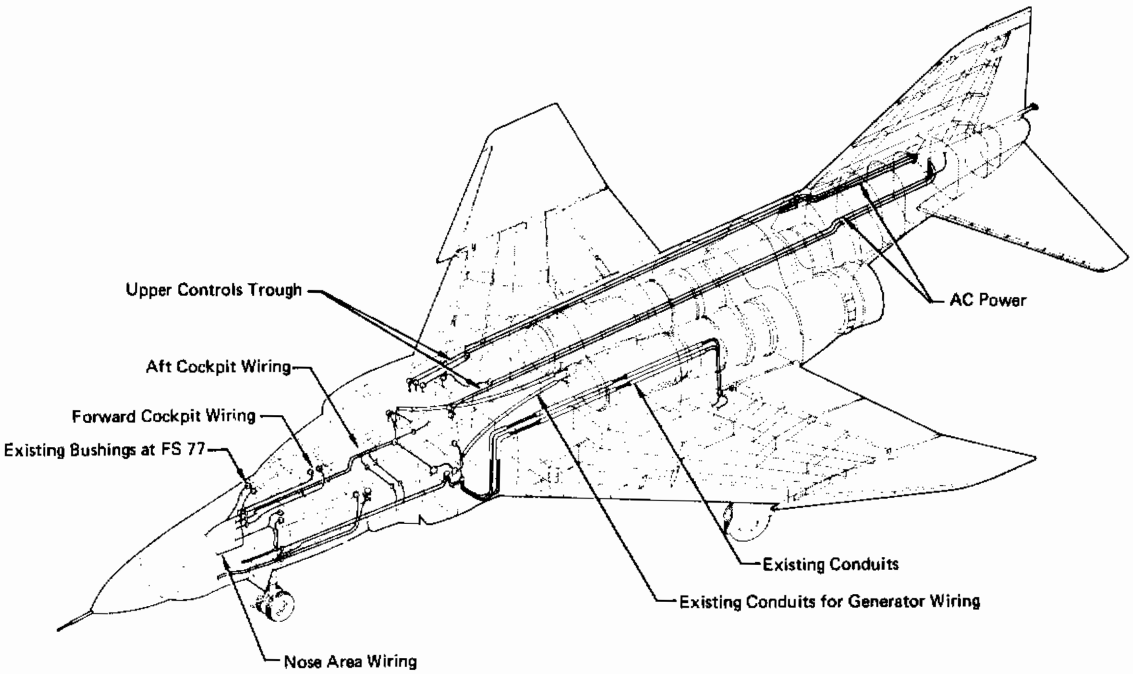


FIGURE 81

SFCS ELECTRICAL WIRE ROUTING

Other SFCS bundles will be routed from the nose through both SLR antenna and equipment bays in separate paths into the lower fuselage. The bundles will then be routed through two conduits on each side of the aircraft and exit into the wing area to the lateral control system secondary actuators and the SFCES rate sensors.

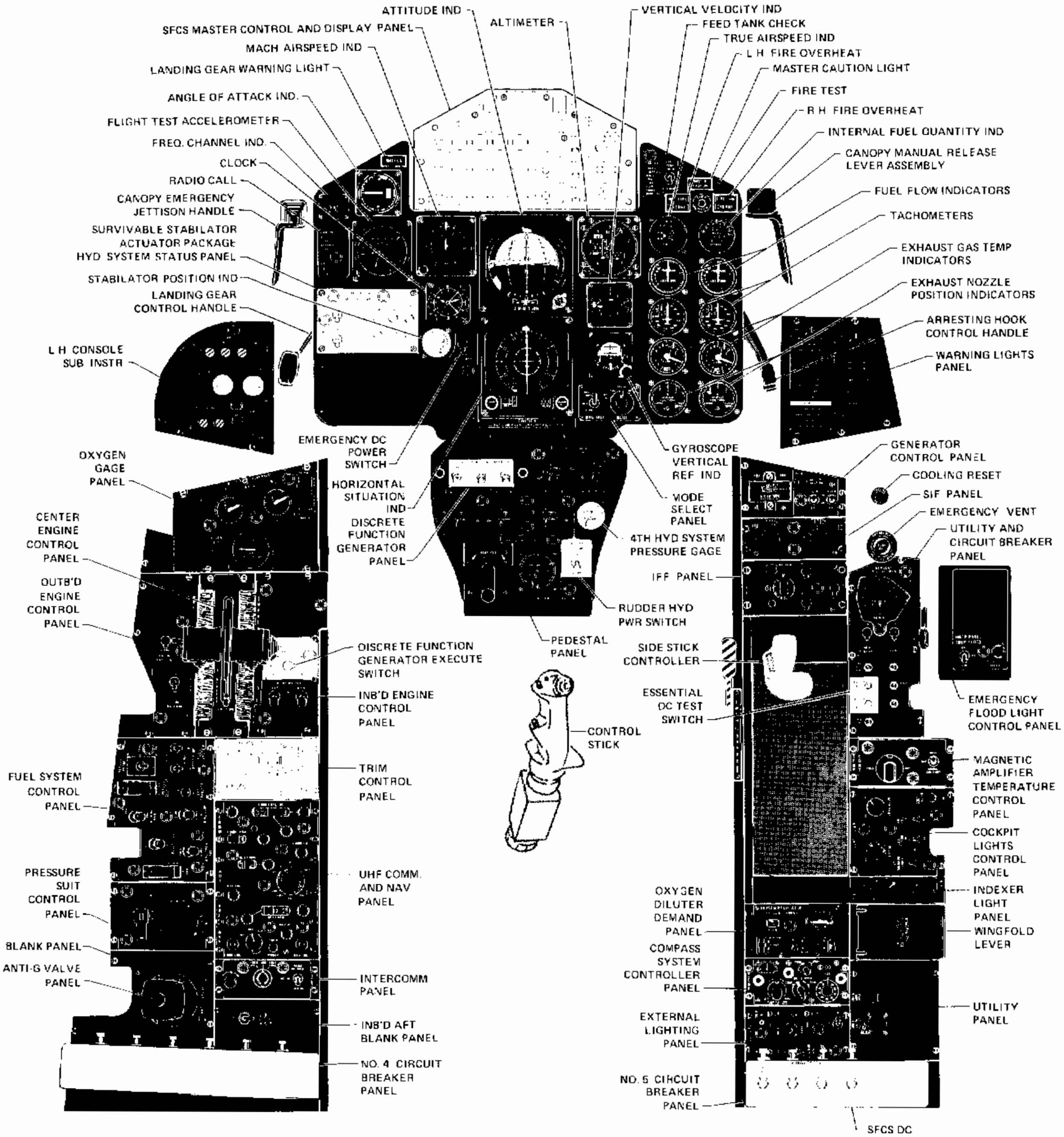
5. COCKPIT

The SFCS cockpit configuration as currently envisioned will retaun the required F-4 primary flight controls and displays. The equipment to be added will be the SFCES controls and displays, the SFCS hydraulic system controls and displays, the control surface position indicators and the electrical power and distribution controls. Surface position indicators are provided since the stick position in a FBW system does not give an indication of surface position due to the motion command system and the revised trim mechanization. The location of equipment in the forward and aft cockpit is designed to provide the pilot with ease of operation, maximum controllability, and maximum visibility of all SFCS controls.

The presently planned forward cockpit configuration, shown in Figure 82, is expected to provide suitable equipment location for the SFCES control panels, display indicators, and the Side Stick Controller. Figure 27 shows the location of controls peculiar to the Phase IIA configuration.

The presently planned aft cockpit configuration, shown in Figure 83, is expected to provide a suitable equipment location for the SFCES controls and displays, and aft cockpit flight control capability.

The equipment being added to the forward and aft cockpits is listed in Tables XXI and XXII.



CODE: EQUIPMENT WITH WHITE BACKGROUND – ADDED FOR SFCS
EQUIPMENT WITH BLACK BACKGROUND – EXISTING

FIGURE 82

FORWARD COCKPIT SFCS PHASE IIC

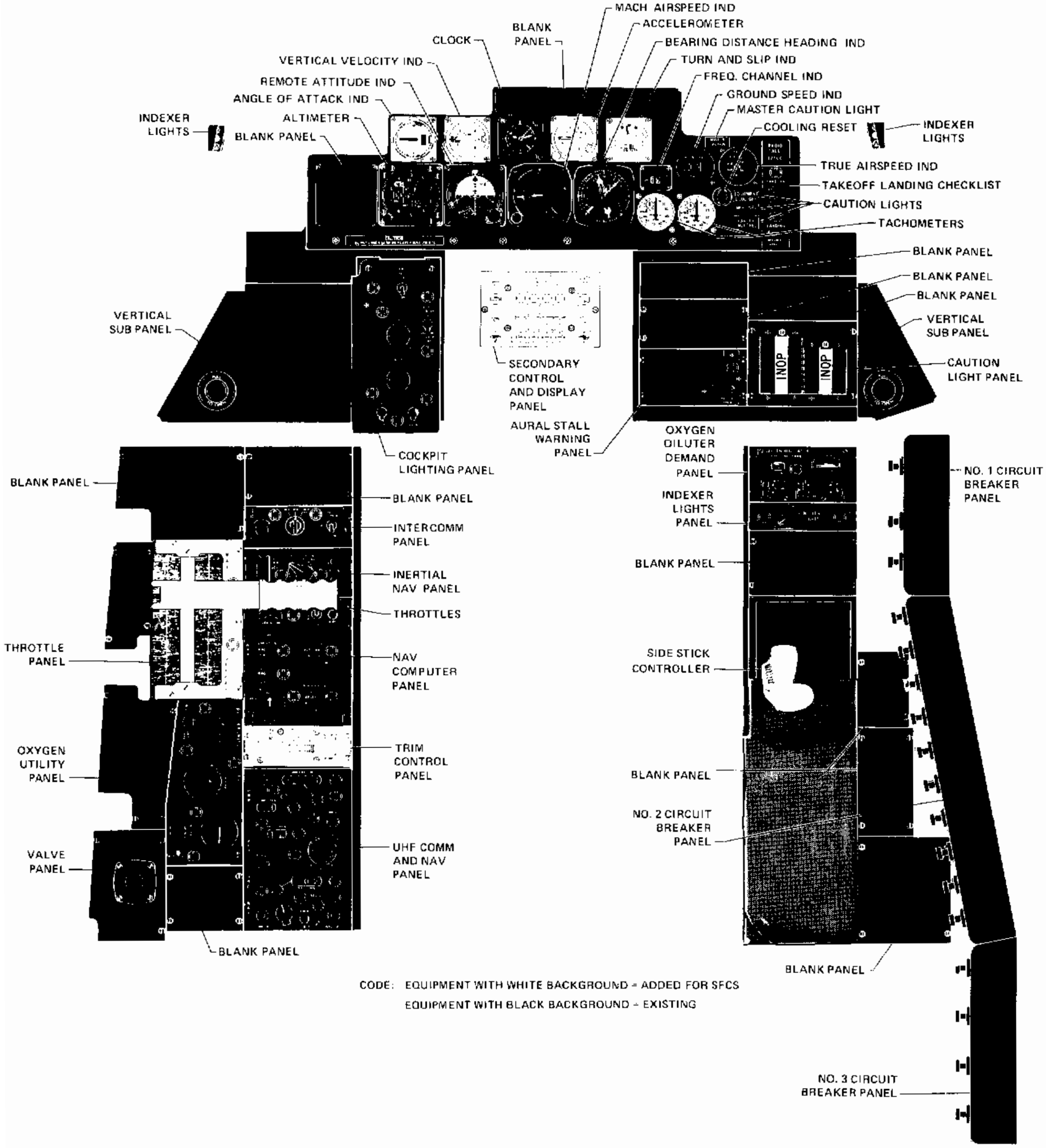


FIGURE 83

AFT COCKPIT SFCS PHASE IIC

TABLE XXI
SFCS FWD COCKPIT ADDED EQUIPMENT

Component Title	Location	Phase Effectivity A	Phase Effectivity B	Phase Effectivity C	Operated by LH	Operated by RH	Function
Master Control and Display Panel	Top of Main Instrument Panel						Equipment monitoring providing failure detection, pilot advisory and mode control
Survivable Stabilator Actuator Package Hydraulic System Status Panel	LH Side Main Instrument Panel						Controls stabilator hydraulic motor pumps 1 & 2, provides failure detection of motor pump system
Discrete Function Generator Panel	LH Side Pedestal Panel						Provides capability of inserting step inputs to the pitch, roll or yaw axis in conjunction with the execute switch
4th Hyd. System Pressure Gage	RH Side Pedestal Panel						Monitors Pressure of 4th hydraulic system
Rudder Hydraulic Power Switch	RH Side Pedestal Panel						Controls backup hydraulic power to rudder
Discrete Function Generator Execute Switch	LH Console Inboard of Throttles						Initiates step input as selected by discrete function generator panel
Yaw Stability Augmentation Switch	LH Console Inboard of Throttles						Engages yaw stability augmentation mode
Pitch Stability Augmentation Switch	LH Console Inboard of Throttles						Engages pitch stability augmentation mode
Yaw Select Switch SFCS or Mech Backup	LH Console Inboard of Throttles						Selects SFCS or mechanical backup system
Pitch Select Switch SFCS or Mech Backup	LH Console Inboard of Throttles						Selects SFCS or mechanical backup system
Trim Control Panel	LH Console Aft of Throttles						Means of applying trim inputs to flight control system (includes NSS override switch)
No. 4 Circuit Breaker Panel	Aft End of LH Console						Circuit breakers for SFCS
Side Stick Controller	RH Console						Means of applying pilot inputs to flight control system
No. 5 Circuit Breaker Panel	Aft End of RH Console						Circuit breakers for SFCES
SFCS DC Power Switches	Top of No. 5 Circuit Breaker Panel RH Console						Controls battery power to SFCES

TABLE XXII
SFCS AFT COCKPIT ADDED EQUIPMENT

Component Title	Location	Phase Effectivity A	Phase Effectivity B	Phase Effectivity C	Operated by LH	Operated by RH	Function
Secondary Control and Display Panel	Underneath Main Instrument Panel						Equipment monitoring providing failure detection pilot advisory and light test
Trim Control Panel	LH Console, Aft of Throttles						Means of applying trim inputs to flight control system
Side Stick Controller	RH Console						Means of applying pilot inputs to flight control system
Dual Control Instruments	Main Instrument Panel						Provides aft cockpit crewmember with basic instruments for flight control
Throttles	LH Console						Provides engine control for aft cockpit crewmember

6. OTHER EQUIPMENT INSTALLATIONS

The installation of the SFCS requires the removal of some existing equipment, the relocation of other existing equipment, and the installation of the new SFCS equipment.

The existing electronic equipment in the test aircraft is installed in the upper equipment bay behind the aft cockpit and in the under console areas of the aft cockpit. The SLR and IR bays are to be used for the SFCS battery complement. The second Angle of Attack Transmitter, added to provide dual angle of attack inputs to the SFCES, is located on the opposite side of the aircraft nose in the same relative position as the existing unit.

Figure 84 illustrates the currently envisioned configuration of the SFCS equipment installation.

Evaluation of possible equipment locations in the SFCS test aircraft has resulted in location of the SFCES Computer and Voter Units, the Adaptive Gain and Stall Warning Computer, and the Maintenance Test Panel in the nose area. The planned installation of these six units provides for:

- o Access for inspection, test, and maintenance by way of existing doors in the SFCS test aircraft.
- o Minimum relocation of existing aircraft equipment and electrical wiring.
- o An efficient structural installation.

The present aircraft structure in this area is relatively clean and sound and there are a number of doors that originally provided access to the cameras and associated equipment. The front panels of these six units are accessible for inspection and test by way of the forward camera door. The wiring and connectors to the rear of these units are accessible by way of two large doors, one on each side of the compartment. These doors also provide access to the Normal Accelerometer and the Roll Rate Sensor, mounted on the aft bulkhead of the compartment.

The Master Control and Display Panel, Secondary Control and Display Panel, Control Stick Force Transducer, Pedal Force Transducer, Trim Control Panels, and the Discrete Function Generator Panel are located in the forward and/or aft cockpits.

The Pitch Rate Sensor is located in the left wing root and the Yaw Rate Sensor is located in the right wing root in the same respective locations as the present F-4 rate gyros.

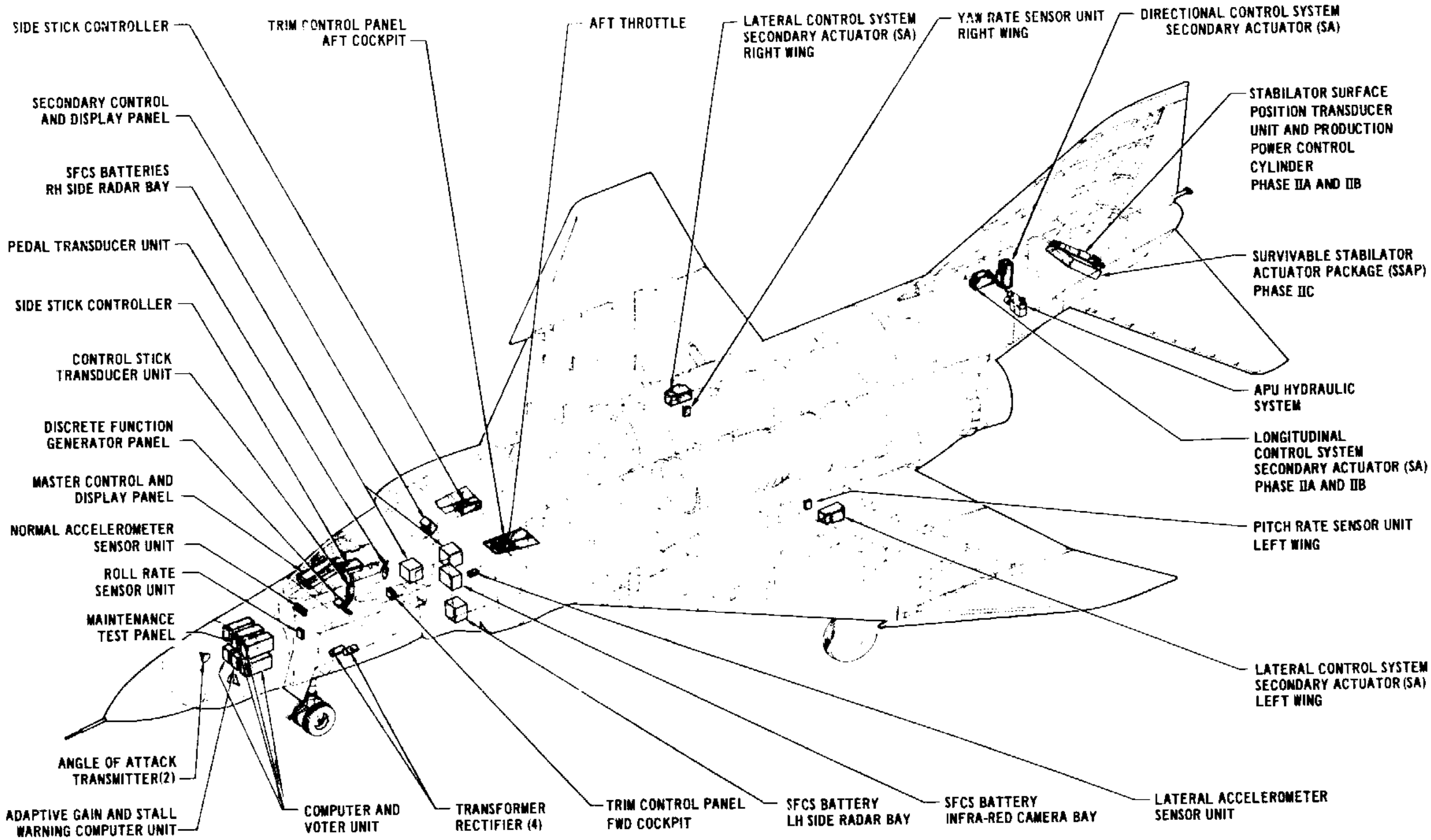


FIGURE 84
SFCS EQUIPMENT LOCATION

SECTION VI

FUTURE EFFORTS

1. GENERAL

The work to be accomplished in the future consists of additional design and fabrication efforts by MCAIR, the modification of the airplane to the SFCS test aircraft configuration, the preparation and/or approval of various engineering data, completion of design and fabrication of major procurement items, the accomplishment of a large number of ground and flight tests, the redesign of test aircraft systems and installations and major equipment whenever such is found unsatisfactory by MCAIR and/or the major subcontractors, and the preparation of two significant technical reports. The reports to be provided are discussed in subsequent paragraphs. In addition, the following paragraphs briefly describe the scope of the ground tests and flight tests planned to be accomplished.

2. TESTS

a. Acceptance Tests

The Acceptance Test is performed on each assembled unit to demonstrate the pertinent physical, mechanical, electrical, hydraulic, static, and dynamic characteristics of the unit. Acceptance Test Procedures (ATPs) which will contain a complete description of the tests to be performed are being prepared by the Suppliers.

b. Design Approval Tests

The Design Approval Test is performed on one or more sets of units and subjects the equipment to the specified environments for a time estimated to be sufficient to determine the ability of the equipment to perform within specified limits while subjected to the predicted installed environmental conditions. Design Approval Test Procedures (DATPs) are being prepared by the Suppliers. The procedures describe the step-by-step test sequence and methods of tests to be performed to demonstrate the functional compliance of the unit with the design and test requirements of the procurement specification.

c. Reliability Tests

Reliability tests are performed using one or more sets of units and are intended to demonstrate the specified Mean-Time-Between-Failure (MTBF) when the equipment is operated in any mode or combination of modes while subjected to any combination of the loads and environmental conditions specified for the Design Approval Tests discussed above. The Reliability Test Procedures are being prepared by the Suppliers.

d. Component Evaluation Tests

Component evaluation tests will be conducted in MCAIR laboratories with procured hardware to assess functional operation using electrical and/or hydraulic power supplies which are representative of those planned for the SFCS test aircraft.

e. Compatibility Tests

The Compatibility tests will be performed by MCAIR using the control system mock-up (Iron Bird) and the six-degree-of-freedom fixed base flight simulator. Components of the SFCS will be interconnected in the same functional relationship as in the actual airplane and all practicable functions and modes of operation of the system will be evaluated for compliance with performance criteria and safe operation.

f. Installed Equipment Tests

All units of the SFCS will be installed in the test aircraft and completely checked out. These tests will include, as applicable, ground vibration tests, EMI tests, and closed-loop performance tests. The closed-loop performance tests will be conducted utilizing the Mobile Ground Test Facility (MGTF).

g. Flight Tests

Flight tests and other work pertinent to these tests will be performed on test aircraft, YF-4E, AF S/N 62-12200 (MCAIR No. 266), for Phase II of the Survivable Flight Control System (SFCS). The objectives of these tests will be to:

- Develop, evaluate, and demonstrate a SFCS utilizing the Fly-By-Wire (FBW) and Integrated Actuator Package (IAP) techniques.

- Obtain data which can be utilized to establish the design criteria, survivability, reliability, maintainability, safety, performance, and testing requirements for application in future weapon systems.

- Establish confidence in the SFCS technology and broaden the experience level.

Parts of the Phase II test program will be performed at the Contractor's facility at St. Louis, Missouri, and parts at Edwards Air Force Base California. Approximately 60 contractor-data flights are expected to be required to complete the testing, planned as:

- Phase IIA - Development and evaluation of: (1) the FBW system for all three axes of control with a mechanical back-up system retained for the pitch and directional control systems; and (2) evaluation of a side stick controller.

- Phase IIB - A continuation of the Phase IIA testing with the mechanical back-up systems removed.

- Phase IIC - (1) Development, evaluation and demonstration of the SSAP installed in the pitch control system, and (2) evaluation of the complete SFCS.

- Phase IID - Demonstration flights of the SFCS for pilots of the Air Force, Navy, NASA, and other interested agencies.

3. REPORTS

A second Interim Report, summarizing the ground and flight tests of Phases IIA and IIB is planned to be published following the completion of Phase IIB flight testing.

A Final Report, summarizing the Phase IIC efforts and results and presenting an overview of the SFCS program results is planned to be published following the completion of Phase IIC flight testing.

APPENDIX I

MAINTAINABILITY ANALYSES

1. INTRODUCTION AND SUMMARY

This appendix contains details of the Maintainability Analyses conducted to help define the approach for the Phase II SFCS program.

The Maintainability Analyses were used to:

- o Define an automatic ground BIT check of the SFCS DC Power Supplies.
- o Establish the probability of a false GO or potentially hazardous condition existing at the completion of the ground BIT check.
- o List the support equipment required to meet the test and servicing requirements of the SFCS.
- o Predict the amounts of maintenance time which may be expected to be expended in servicing of the fourth hydraulic power supply and the SFCS NiCd batteries.
- o Investigate whether a requirement existed to incorporate testing of the various SSAP blowers in the IFM or automated ground BIT.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. MAINTAINABILITY ANALYSES

a. Ground Test of the SFCS DC Power Supplies

The proper functioning of the SFCS DC Power supplies is critical to the SFCS both from the standpoint of normal operation, where they provide conversion of the aircraft primary AC power to DC energy, and emergency operation, where the integral DC power supply NiCd batteries provide the energy to maintain system operation in the event of T/R or primary AC power loss. Study of the DC supplies identified two things that should be determined regarding DC power supply performance. One, that the system was operating properly, and two, that the NiCd batteries were at a sufficient state of charge to sustain operation for a period of time to allow the aircraft to return to base without electronic system failure. As the electronic BIT design evolved and study of the operating characteristics of NiCd batteries progressed, it became apparent that an electronic BIT test could be implemented which would readily identify the operating condition of the SFCS DC power supplies but not necessarily provide a positive identification of the NiCd battery state of charge. This was due to a phenomenon known as apparent loss of capacity which occurs when NiCd batteries are charged from a constant potential source such as is encountered in most aircraft DC power supplies. This loss of capacity is caused by shallow discharging and recharging by means of constant potential. With time, a battery installed in an aircraft and floating on the aircraft buss will experience a loss of capacity. For a more complete discussion of the loss of capacity phenomenon the reader is directed to Reference 15.

The use of a constant current charging system can eliminate this phenomenon but at the cost of increased complexity and isolation of the batteries from the DC bus. Since the SFCES design, in part, was to take advantage of the inherent filtering of the DC power provided by the batteries, it was decided to proceed with the constant potential DC power supply design and to avoid the temporary loss of capacity phenomenon through the implementation of scheduled servicing and inspection procedures. Following is a description of the automated BIT check and scheduled inspection and servicing planned to be performed on the SFCS DC power supplies:

(1) Built-In-Test

Using suitable sensors, the output voltage of the DC power supply T/R is verified during the ground BIT check for a potential greater than 26.0 volts. Simultaneously, current flow through the battery is measured to determine if it lies in the range of 0.5 amps to 5.0 amps. Successful passage of the two checks above provides an indication of proper DC power supply operation as follows:

- Voltage: The primary purpose of the voltage check is to verify that the current levels detected are being supplied at a voltage higher than that of the battery. Without this check, the current levels detected could be within tolerance and with the proper polarity for a charging battery, but the battery depleted due to a partially failed T/R. The voltage selected is a function of the aircraft primary AC supplies, T/R capacity and T/R load.

- Current: The current levels, when detected at the voltage specified, will indicate continuity through the battery loop and indicate the apparent state of charge of the battery, i.e., less than 5 amps implying a charged battery.

(2) Scheduled Servicing and Inspection

The following inspection and servicing tasks were selected to maintain the SFCS NiCd batteries as close as possible to the peak operating condition and maximum state of charge.

- Post flight inspection in accordance with Reference 15.

- Capacity tests and recharging of the SFCS NiCd batteries on a weekly basis initially and subsequently extending this time in one week increments as confidence in the charging system is built up. Batteries for these tests will be removed as soon after the last flight of the day as possible. If temporary capacity loss of the battery is detected, the maximum capacity test frequency will be determined by the point where battery capacity drops consistently to 80% of its full capacity. The capacity test will include steps a, b, c, e, f, and g of Paragraph 3-5 of Reference 15. A complete battery servicing sequence as described by Paragraph 3-3 of Reference 15 will be accomplished at any time battery capacity falls below 70%, current leakage is excessive, visual inspection reveals deficiencies, 60 days have elapsed since the last complete servicing or T/R failure occurs.

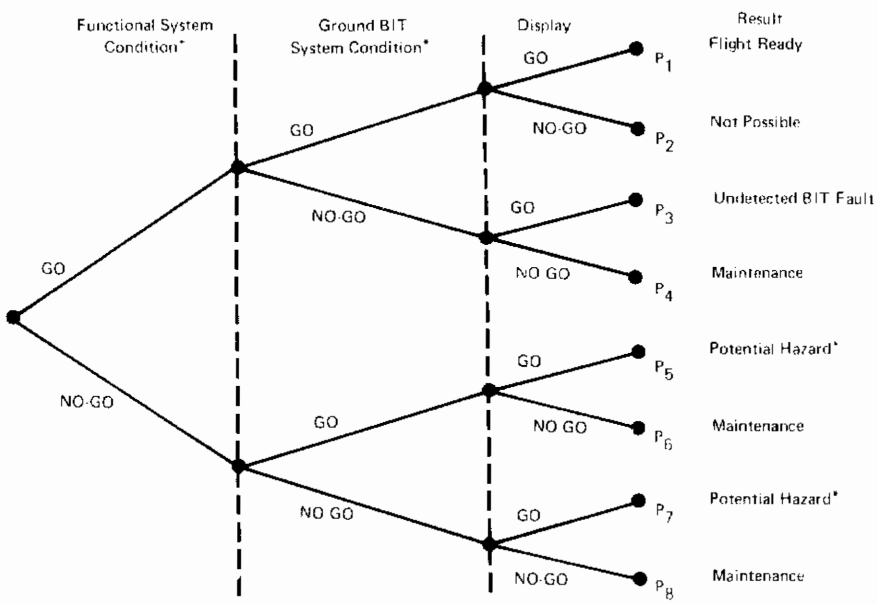
b. Probability of a False GO or Potentially Hazardous Condition Existing at the Completion of Ground BIT Check

A ground BIT check will be performed prior to and following each flight and following any maintenance of the system. Four displays are possible at the completion of a ground BIT check with GO, NO-GO indicators of the type planned for the SFCS. These are: a GO illuminated, a NO-GO illuminated, both GO and NO-GO indicators illuminated, or both GO and NO-GO indicators extinguished. Of these, the occurrence of both indicators either extinguished or illuminated at the completion of the gorund BIT check is self-evident proof of faulty system operation and a requirement for maintenance. For this reason these two conditions do not constitute a hazard and were dropped from further study. Of the remaining two possible displays, the one, NO-GO illuminated will always be the cause of maintenance and the other, GO illuminated, a source of a potentially hazardous situation if it is not displaying a real condition to the flight or ground crew. A false GO would result any time an undetectable fault was present in the system. Faults can be undetectable either because of the inherent limitations of BIT or due to BIT circuitry failures. The probabilities of occurrence of these displays were analyzed and are summarized below.

(1) Figure 85 is the conditional probability tree for either a GO or NO-GO display at the end of the ground BIT check.

Utilizing this tree, the various display probabilities were calculated by multiplying the GO/NO-GO probabilities of the functional system, the ground BIT system and the displays. The results of this procedure are presented in Table XXIII. As shown in Table XXIII, the probability of a false GO or potentially hazardous condition existing at the completion of the ground BIT check varies from 0.000368 to 0.001776 for mission lengths of 1.0 to 5.0 hours respectively. For a mission of 1.3 hours this probability is 0.000478. This equates to one false GO in 2090 ground BIT checks of the SFCS or once in 2720 flight hours.

(2) It should be noted that the false GO calculation is analogus to a "series" MTBF defined as "failure to perform its prescribed function of any part, channel, LRU, axis or set shall be considered a failure regardless of whether such failure could or would cause channel, LRU or system failure". Therefore, based on this worst case type consideration and the configuration of the SFCS, it may be assumed that only a small portion of the failures resulting in a false GO would result in a hazardous



Notes: 1. Potential hazard is based on the series MTBF definition: Failure of any part, channel, LRU, axis or set is considered a failure (NO-GO) regardless of whether such failure could cause channel, axis, LRU or system failure.

2. Refer to Table XXIII for probabilities associated with P_1 through P_8.

FIGURE 85

CONDITIONAL PROBABILITY TREE SFCS GO, NO-GO INDICATIONS

TABLE XXIII

GO, NO-GO INDICATION PROBABILITIES FOR VARIOUS MISSION DURATIONS

Indication and Notation		Description	Mission Durations					
			1 Hour	1.3 Hours	2 Hours	3 Hours	4 Hours	5 Hours
P_1	GO	Flight Ready	0.981546	0.976087	0.963465	0.945717	0.928296	0.911195
P_2	NO GO	Not Possible	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
P_3	GO	Undetected BIT Fault	0.000016	0.000016	0.000016	0.000016	0.000015	0.000015
P_4	NO GO	Maintenance	0.000016	0.000016	0.000016	0.000016	0.000015	0.000015
P_5	GO	Potential Hazard	0.000368	0.000478	0.000730	0.001085	0.001433	0.001775
P_6	NO GO	Maintenance	0.018052	0.023401	0.035772	0.053165	0.070238	0.086996
P_7	GO	Potential Hazard	0.000000	0.000000	0.000001	0.000001	0.000001	0.000001
P_8	NO GO	Maintenance	0.000000	0.000000	0.000001	0.000001	0.000001	0.000001

condition, i.e. hazard category Class III or IV, and the probability of such a condition going undetected by ground BIT and resulting in an unsafe flight condition would be considerably less than the 0.000478 calculated herein.

c. SFCS Support Equipment

As a part of the overall Maintainability analyses, the test and servicing requirements for the SFCS systems were identified. The listing includes both standard and special test and servicing equipment including those items of production F-4 support equipment modified to fulfill the SFCS F-4 requirements. The listing of the SFCS Test and Servicing Equipment may be found in Table XXIV.

d. SFCS Maintenance Manhours Per Flight Hour (MMH/FH)

Estimated MMH/FH figures for the SFCS F-4 were calculated for systems, subsystems and components added or modified by the addition of the SFCS to the basic RF-4 aircraft. It is recognized that these figures will probably not be experienced by the test aircraft due to the R&D nature of the SFCS program. Their purpose is to act as a guide in identifying broad areas where maintenance will be expended rather than as a prediction of the actual resources expected in support of the test aircraft. The basic aircraft selected as the starting point for the calculations is an RF-4C less its reconnaissance mission sensors and equipment. In addition, the base figures were modified to reflect the change from a parallel bus to a split bus AC power supply configuration. MMH/FH figures for SFCS equipment were generated by one of two methods. For equipment new to the F-4 MMH/FH figures were calculated by estimating the mean maintenance manhours to repair for an average repair task and dividing by the estimated mean time between maintenance action for the equipment under study. For equipment the same as or similar to currently existing F-4 equipment, MMH/FH figures were extracted directly from AFM 66-1 Maintenance Data. Figure 86 gives a summary of the overall impact of the change from the F-4 conventional flight control system to the SFCS F-4 flight control system consisting of the SFCES, SSC, roll and yaw electrohydraulic SAs and SSAP. A system and subsystem summary is provided in Table XXV. RF-4C MMH/FH figures are from the AFM 66-1 ON-OFF Equipment Maintenance Summary for January through December 1969 with a base of 116,538 flight hours.

Largest overall changes occur in the Autopilot, Hydraulic Power and Electrical Power subsystems. The Autopilot change amounts to a 0.499 MMH/FH reduction due to complete removal of the Autopilot system. The Autopilot ARI and SAS functions are now accomplished by the SFCES which is included in the Flight Control System figures. The remainder of the Autopilot functions are not duplicated in the SFCS aircraft. The addition of the fourth hydraulic power supply increases the Hydraulic power subsystem by approximately 0.311 MMH/FH. The majority of this increase is a result of the estimated subsystem servicing requirements which contribute 0.240 MMH/FH. The other large contributor to the SFCS F-4 MMH/FH requirements is the SFCS DC power supply which contributes 0.271 MMH/FH to the aircraft total with 97% of this being

TABLE XXIV

SFCS TEST AND SERVICING EQUIPMENT

Class	St	Standard
	Sp	Special (F 4)
	X	Special (SFCS)
Level	FL	Flight Line
	S	Shop
	V	Mobile Ground Test Facility
	M	Factory

Equipment Nomenclature	Designation	Class	Level of Use	Usage	Remarks
Adapter Cable	53 044122 (MCAIR)	X	FL	Interface the AN/PSM 20B AC Electrical System Test Set with the SFCS aircraft modified AC Test Receptacles	
External Electrical Power Source	TBD	St	FL	External electrical power source during system checkout	200/115 VAC, 400 Hz, 3φ, 41 26 KVA minimum
Hydraulic System Filler Unit	2120 (Mod.) (Monen)	St (Mod)	FL	Used to service the PC 1, PC 2 and SSAP reservoirs with MIL H 83282	Modified for use with MIL H 83282
SSAP MP Hoisting Adapters	TBD	X	FL	Used to lift SSAP MP LRU s into the aircraft	Required due to MP weight and limited aircraft access
Headset Adapter Cable	53-044118 (MCAIR)	X	FL	Adapts standard ground to aircraft Intercom Headset to specialized SFCS use	Interconnects F 4 Ground Intercom Plug with Headset Plug. Contains breakout of BIT Consent Switch and BIT Interlock Switch on cable
Hydraulic Test Stand	82130 (Mod.) (Sprague)	St (Mod)	FL	Provides a source of hydraulic power for operation, maintenance and servicing of aircraft PC 1 and PC 2 Hydraulic Systems	a. Modified for MIL H 83282 use b. 20 gpm, 3000 psi
Multimeter	AN/PSM 6	St	FL/S/V	Used to check voltage and resistance values	
Push Pull Spring Gages	TBD	St	FL/S/V	Used to measure breakout feel and friction forces	Scales TBD
Torque Wrenches	TBD	St	FL/S	Used for final assembly of components	Values TBD
DC Power Supply	LB 723 FM OV (Lambda)	St	V	Provides DC power for MGTF requirements	
Analog/Hybrid Computer	AD 5 (Applied Dynamics)	X	S/V	Used to perform system simulations	May be used with the aircraft installed system or with the SFCLS installed on the SFCES system test bench.
Eight Channel Recorder	7729A (Hewlett Packard)	St	S/V	Records results of Analog Computer closed loop simulations	Can be used with SSC, SA and SSAP
Digital Voltmeter	TBD	St	S/V	Standard Test Equipment to support SFCS	
Oscilloscope	TBD	St	S/V	Standard Test Equipment to support SFCS	
Function Generator	203A (Hewlett Packard)	St	S/V	Standard Test Equipment to support SFCES System Test Bench	Alternate Hewlett Packard 205A
Tilt Table	NST 300 (Nikken)	St	S/V	Standard Test Equipment to support SFCES System Test Bench	Or equivalent
Rate Table	722 (Inland)	St	S/V	Standard Test Equipment to support SFCES System Test BEnch	Limited performance testing
Force Scale	DPP 10 (Chattilon)	St	S/V	Used to calibrate and test the Emergency Disengage Switch in the Control Stick Transducer	Or equivalent
DC Micro Volt Ammeter	Model 4072 (Dynamics)	St	V	Standard Test Equipment to support SFCS van mounted factory test fixtures	
SFCS System Bench	68054427001 (Sperry)	X	S/V	Provides complete test capability for the SFCES LRU's	Interfaces with Analog Computer and Standard Test Equipment

TABEL XXIV

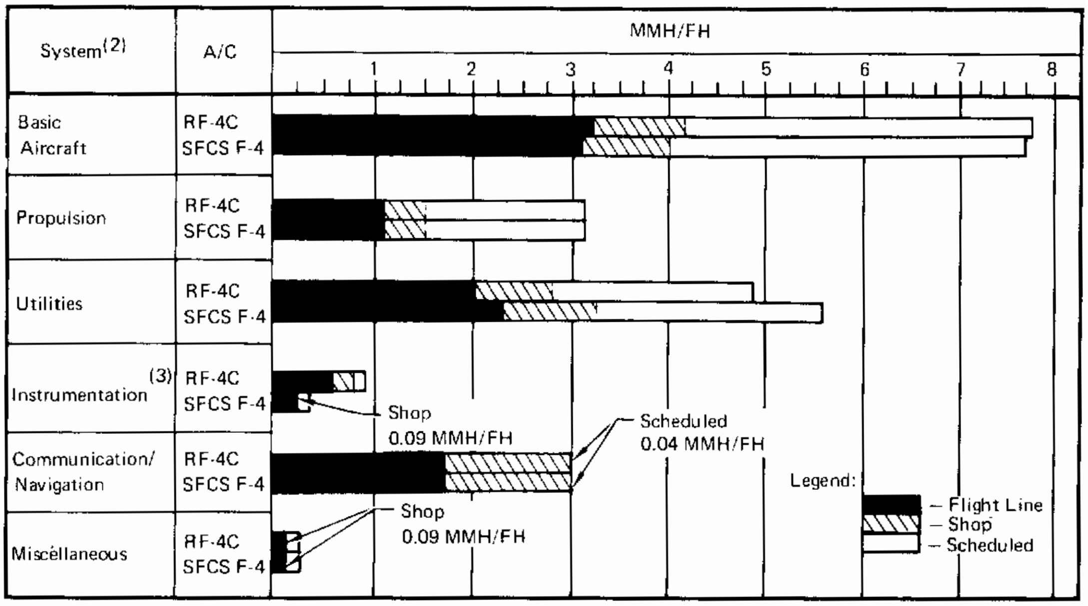
SFCS TEST AND SERVICING EQUIPMENT (CONTINUED)

Equipment Nomenclature	Designation	Class	Level of Use	Usage	Remarks
Computer and Voter Analyzer	T 321949 (Sperry)	X	M/V	Tests SFCES CV's to ATP requirements when used with Standard Test Equipment.	Factory Test Fixture
Adaptive Gain and Stall Warning Computer Analyzer	T 321950 (Sperry)	X	M/V	Tests SFCES G&S to ATP requirements when used with Standard Test Equipment.	Factory Test Fixture
Maintenance Test Panel Analyzer	T-321955 (Sperry)	X	M/V	Tests SFCES MTP to ATP requirements when used with Standard Test Equipment	Factory Test Fixture
Trim Control and DFG Test Fixture	T-321954 (Sperry)	X	M/V	Tests SFCES FTP, ATP and Discrete Function Generator to ATP requirments when used with Standard Test Equipment.	Factory Test Fixture
Control Panel Test Fixture	T-321953 (Sperry)	X	M/V	Tests SFCES MCP and SCP to ATP requirements when used with Standard Size Equipment.	Factory Test Fixture
Sensor Unit Adapter Plate	T-321945 (Sperry)	X	M/V	Provides means of mounting Rate Sensors to the Rate Table	Factory Test Fixture
Stabilator Surface Position Transducer Test Fixture	T-321942 (Sperry)	X	M/V	Provides the excitation and switching required to test the Stabilator Surface Position Transducer when not interfaced with SFCES bench.	Factory Test Fixture
Rate Sensor Unit Test Fixture	T-321946 (Sperry)	X	M/V	Provides the excitation and switching required to test the Rate Sensor LRU's	Factory Test Fixture
Accelerometer Unit Mounting Fixture	T-321945 (Sperry)	X	M/V	Provides the means for mounting and indexing information for aligning the sensitive axis of individual accelerometers in the NA and LA LRU's to the Rate Table.	Factory Test Fixture
Pedal Transducer Unit Load Fixture	T 321957 (Sperry)	X	S/V	Provides means for complete test troubleshooting and repair of the Pedal Transducer when used with the SFCES System Bench.	
Control Stick Transducer Load Fixture	T-321976 (Sperry)	X	S/V	Provides means for complete test, troubleshooting and repair of the Control Stick Transducer when used with the SFCES System Bench.	
Stabilator Surface Position Transducer Position Test Fixture	T 321956 (Sperry)	X	S/V	Used to test the Stabilator Surface Position Transducer when connected to the SFCS Test Bench.	Phase IIA and B Only
Computer Card Extender	68054424017 (Sperry)	X	S/V	Extends CVU, MTP, and G&S Cards for piece part troubleshooting, and repair.	With Computers Installed in Systems Bench
Computer Card Extender	68054424018 (Sperry)	X	S/V	Extends CVU, MTP, and G&S Cards for piece part troubleshooting and repair	
Accelerometer Test Fixture	T 321946 (Sperry)	X	M/V	Provides the excitation and switching required to test the NA and LA LRU's.	Factory Test Fixture
Control Stick Transducer Test Fixture	T-321968 (Sperry)	X	M/V	Provides the excitation and switching required to test the Control Stick Transducer when not interfaced with SFCES Bench	Factory Test Fixture
Pedal Transducer Test Fixture	T-321971 (Sperry)	X	M/V	Provides the excitation and switching required to test the Pedal Transducer when not interfaced with SFCES Bench.	Factory Test Fixture
Rate Table	813 (Inland)	SI	S	Standard Test Equipment used to calibrate and test Rate Sensor and Accelerometer LRU's.	Or Equivalent

TABEL XXIV

SFCS TEST AND SERVICING EQUIPMENT (CONTINUED)

Equipment	Designation	Class	Level of Use	Usage	Remarks
Side Stick Controller Test Set	430544-0 (LSI)	X	S/V	Provides means for complete test, troubleshooting and repair of the SSC.	
Phase Sensitive Voltmeter	PAV-4 (Getsch)	St	S/V	Test SSC	
SSAP Rigging Jig	TBD	X	S	Used to rig the overall length of the SSAP to the SFCS aircraft dimensions.	On aircraft rigging inadvisable due to Actuator LRU weight
SSAP Holding Fixture	401-66965 (LTV-E)	X	S	Used to support the SSAP during shop maintenance.	
Hydraulic Power Supply	TBD	St	S	Required to operate the SSAP in the External Power Mode	0-3000 psi at 30 gpm 0-5630 psi at 1 gpm for proof testing.
Electric Power Supply	TBD	St	S	Required to provide electrical power to the SSAP motor pumps.	115 VAC, 400 Hz, 3 ph, 30 KVA
Voltmeter	403B (Hewlett Packard)	St	S	Used to measure voltage applied to SSAP Secondary Actuator and motor pumps.	
SSAP Test Console	401-66960 (LTV-E)	X	S	Provides closed loop and fault isolation testing of the SSAP.	Used with Standard Test Equipment
Hydraulic Test Stand	TBD	St	S	Provides shop hydraulic power for test of SA and SSAP.	Requirements: a) 0-3000 psi, 5 gpm (SA) b) 0-3000 psi, 30 gpm (SSAP) c) 0-5630 psi, 1 gpm for proof testing
Secondary Actuator Holding Fixture	TBD	X	S	Supports Secondary Actuator (SA) during shop testing. Provides loads and capability of measuring actuator position.	
Secondary Actuator Servo & Logic Test Set	TBD	X	S	Includes servo amplifier, failure logic and manual switches required to conduct failure mode test, and control frequency response tests.	
Transfer Function Analyzer	DA410L-H (Weston)	St	S	Used to measure frequency response and phase shift.	
Battery Charger - Test Set	A/E 24U-10	St	S	Used for servicing MS24497-5 NiCd SFCS batteries.	Alternate: NiCd Battery Test RAC777NC, NiCd Battery Charger RAC505-50



Aircraft Totals:

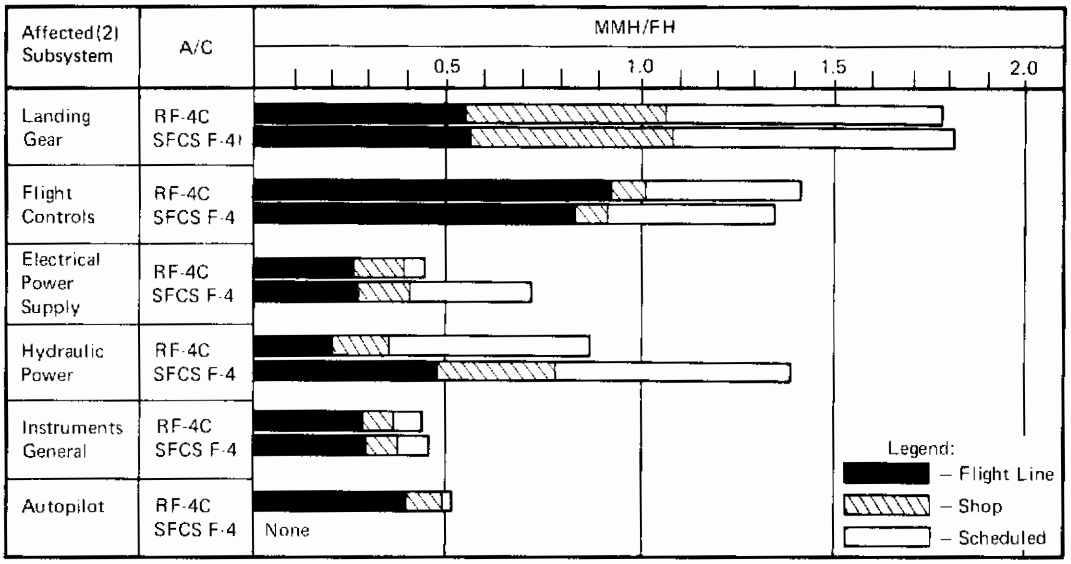
	Flight Line	Shop	Scheduled	Total
RF-4C	8.76549	3.63240	7.60891	20.00680
SFCS F-4	8.58759	3.68056	7.92620	20.19435

Notes:

(1) Less Mission Equipment

(2) See Table XXV for System, Subsystem Listing

(3) Does not include Flight Test instrumentation.



Affected Subsystem Summary

Changes From RF-4C:

	Flight Line	Shop	Schedule	Overall
Landing Gear	+0.01450	+0.00001	No Change	+0.01451
Flight Controls	–0.12398	+0.00248	+0.02475	–0.09775
Elect. Pwr. Sup.	+0.01369	+0.00378	+0.27140	+0.28887
Hydraulic Power	+0.26747	+0.17151	+0.02781	+0.46679
Instrument General	+0.01047	+0.00353	No Change	+0.01400
Autopilot	–0.35905	–0.13315	–0.00667	–0.49887
Overall	–0.17790	+0.04816	+0.31729	+0.18755

FIGURE 86

MAINTENANCE MANHOUR PER FLIGHT HOUR

RF-4C(1) Versus SFCS F-4

TABLE XXV

AIRCRAFT SYSTEM/SUBSYSTEM WORK UNIT CODE SUMMARY

System	WUC(1)	Subsystem	Modifications for SFCS
Basic Aircraft	11	Airframe	
	12	Cockpit and Fuselage Compartments	
	13	Landing Gear	Added Micro Switches
	14	Flight Controls	Includes SFCES, SA's, SSC, SSAP
Propulsion	23	J-79 Turbojet Engine	
Utilities	41	Airconditioning and Pressurization	
	42	Electrical Power Supply	Includes SFCS DC Power Supply
	44	Lighting System	
	451	Hydraulic Power	Fourth Hydraulic Power Supply Added
	452	Pneumatic Power	
	453	Ram Air Turbine	
	46	Fuel	
	47	Oxygen	
	49	Miscellaneous Utilities	
Instrumentation	51	Instruments General	Second Angle of Attack Transmitter Added
	52	Autopilot	SAS and ARI functions included in SFCS,
	55	Malfunction Analysis and Recording	Autopilot functions not included in SFCS.
Communication/ Navigation	71	Radio Navigation	
	72	Radar Navigation	
Miscellaneous	91	Emergency Equipment	
	93	Drag Chute Equipment	
	96	Personnel Equipment	
	97	Aircraft Explosives	

(1) From TO IF-4(R)C-06, RF-4C Work Unit Code (WUC) Manual

for scheduled inspection and servicing of the NiCd batteries.

e. IFM and Ground BIT Check of SSAP Blower Motors

The SSAP contains 6 blower motors, one for each SA element and one for each Motor Pump (MP) LRU heat exchanger. These blowers can be critical to the operation of their respective elements if high temperatures are encountered due to high duty cycle or operation at the extremes of the aircraft flight envelope. Since check of these blowers could be incorporated into the SFCS IFM or Ground BIT design an investigation of whether such checks would warrant incorporation in the SFCS BIT circuitry was made.

(1) The investigation of the incorporation of detectors to sense failure of the SSAP blower motors was divided into the use areas of IFM and ground BIT with the IFM area sub-divided into MP or SA LRU application. Discussion is as follows:

(a) IFM

- o Motor Pump Heat Exchanger Blowers

  The failure of a MP heat exchanger blower will not always result in an overtemperature of the MP LRU and therefore does not constitute a reason to shut down the affected MP. Conversely, an overtemperature condition in a MP LRU will always be cause to shut down the affected MP and as such constitutes a more critical parameter. Hydraulic overtemperature information will be provided to the pilot, therefore IFM of the MP LRU heat exchanger blowers is not recommended.

- o Secondary Actuators

  The failure of the SA element blowers will not always cause failure of the affected SA element. Additionally, manual shutdown of individual SA elements, although possible by opening SA element circuit breakers, probably will not be accomplished except in extreme emergency conditions where three SFCS failures in the longitudinal axis have occurred and the demand on capability of the SFCES is used. Therefore, no IFM checks of the SSAP SA blower are planned.

(b) Ground BIT

- o The addition of SSAP blower checks to the SFCS ground BIT routine provides benefits in increased system confidence and reduced maintenance effort. Although these positive attributes accrue to the ground BIT check of the SSAP blowers, the system complexity added for such checks and the lack of confidence that they will always provide a positive indication of improper operation of the blowers make the addition of circuitry to perform them questionable. Additionally, other factors combine to further dilute the necessity for SSAP blower checks. The first of these is attributable to the redundancy of the SSAP. The possibility of a combination of blower failures, flight conditions leading to high duty cycle of the SSAP, and other multiple SSAP element failures occurring on the same flight is extremely remote. The second negating factor is the availability of flight test instrumentation measurands of SA tach motor temperature, heat exchanger outlet temperature and pump output temperature which may be checked during and after flight to identify any cooling deficiencies generated by loss of these blowers.

(2) In summary then, it appeared that incorporation of IFM or ground BIT checks of the SSAP blowers was desirable, but investigation indicated that the benefits were not worth the cost. Therefore, BIT check of the SSAP blowers is not to be incorporated in the

SFCS design and checks of the operating condition of the SSAP blowers will be accomplished through scheduled inspections of SSAP operation and checks of the flight test temperature measurands mentioned in Paragraph (1)(b) above. Flight test measurands will be checked after each flight. Scheduled inspections will be conducted at a maximum interval of 15 SSAP operating hours.

3. CONCLUSIONS

As a result of the Maintainability Analyses the following conclusions were reached:

- Ground BIT test of the SFCS DC power supplies is feasible and, when supported by scheduled maintenance, is expected to provide a high degree of confidence in the condition of the DC power supplies at the completion of the ground BIT test.

- The probability of a hazardous false GO indication at the completion of the Ground BIT check is extremely low. At the completion of the ground BIT check following a 5 hour flight, the probability is calculated to be less than 0.001776.

- Routine flight line maintenance will be accomplished with a minimum of SFCS peculiar support equipment. No special electronic test equipment is expected to be required for routine maintenance.

- Significant amounts of routine maintenance may be expected to be expended in servicing the fourth hydraulic power supply and the SFCS DC power supply NiCd batteries for the flight test program.

- Ground BIT and IFM checks of the SSAP heat exchanger and SA blower motors, although desirable, are not worth the cost and will not be included in the SFCS equipment.

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX I

ABBREVIATIONS:

amp - ampere

Elect - Electrical

Mod - Modified

P - Probability

ph - Phase

Pwr - Power

Sup - Supply

TBA - To Be Assigned

TBD - To Be Determined

TO - Technical Order

V - Mobile Ground Test Facility (Code)

VAC - Volts Alternating Current

WUC - Work Unit Code

SYMBOLS:

FL	Flight Line
M	Factory
S	Shop
Sp	Special (F-4)
St	Standard
X	Special (SFCS)
ϕ	Phase

APPENDIX II

THERMODYNAMICS

1. INTRODUCTION AND SUMMARY

   a. General

   The SFCS thermal analysis covers the several stages of SFCS power application from the power source equipment to the power using equipment, wherever special equipment for the SFCS program is added to the basic aircraft.

   The electrical and hydraulic power sources of interest are the transformer rectifiers (T/Rs) and batteries, and the fourth hydraulic system.

   The power using equipment of interest are the Survivable Flight Control Electronic Set (SFCES), Secondary Actuators (Sas), and the Survivable Stabilator Actuator Package (SSAP).

   A list of specialized abbreviations and symbols is found at the end of this appendix.

   b. Thermal Goals and Tests

   The desired end result of the thermal analyses and design is maintenance of the proper operating temperature without excessive cost, space, or weight penalties. The major thermal design and testing goals are as follows:

   (1) Modification of the test aircraft to provide acceptable heat sinks for all SFCS equipment.

   (2) Design of the SFCES and the SSAP to:

   - o Minimize heat generation,
   - o Possess the needed heat transfer capability to the heat sinks and
   - o Endure the operating temperatures that will be experienced.

   (3) Implementation of thermal tests properly defined to:

   - o Evaluate SFCS equipment capability to withstand its thermal environment,
   - o Obtain empirical thermal data through ground testing, and
   - o Assess SFCS capabilities in the actual F-4 environment through flight tests.

c. The principal thermodynamics analyses to be performed relate to:

(1) Determining the heat expected to be generated.

(2) Establishing the methods and capability to dissipate heat.

(3) Establishing appropriate limits for compartment air and structural temperatures.

(4) Ascertaining SFCS and aircraft equipment vulnerability to cooling system failures.

(5) Precluding component temperature problems.

(6) Distributing the ventilating air.

(7) Assessing and minimizing, to the extent practicable, the aircraft penalty ensuing from thermal design requirements.

2. THERMAL DESIGN APPROACH

The SFCS is a primary flight control system. Failures due to overtemperature conditions, if allowed to encompass the redundant SFCS channels, can be envisioned to result in loss of the aircraft. Nuisance failures due to overtemperature or other adverse temperature conditions could impede the flight testing of the SFCS.

The thermal studies and analyses have been employed to define a thermal design and testing approach for the SFCS equipment. The approach, which is identified herein, is such that acceptable temperature can be maintained in the SFCS equipment. Potential thermal problems have been identified and preventive measures implemented in accordance with available data and analyses. The predicted adequacy of the equipment thermal design will be verified empirically in formal quality assurance and other tests. The tests include simulated failures of the SSAP self-contained cooling fans and the aircraft refrigeration package. Supplementing the more formal acquisition of thermal data by vendors during MCAIR specified temperature altitude tests are the results of tests that are for vendor subcomponent design approval or development purposes or that are routine checks of procured hardware, such as acceptance tests and iron bird tests. It is envisioned that thermal analysis parameters will be updated as early as is feasible from these data.

3. THERMAL ANALYSES

a. SFCS Transformer-Rectifiers and Batteries

A thermal analysis has been made of the SFCS transformer-rectifiers (T/Rs) and batteries for Phase II and the resulting thermal environment is discussed herein. The T/Rs are used to supply power to the SFCS and charge the back-up batteries.

Two T/Rs are located in the left SLR antenna bay and two in the right. These compartments are ventilated by discharge airflow from the nose

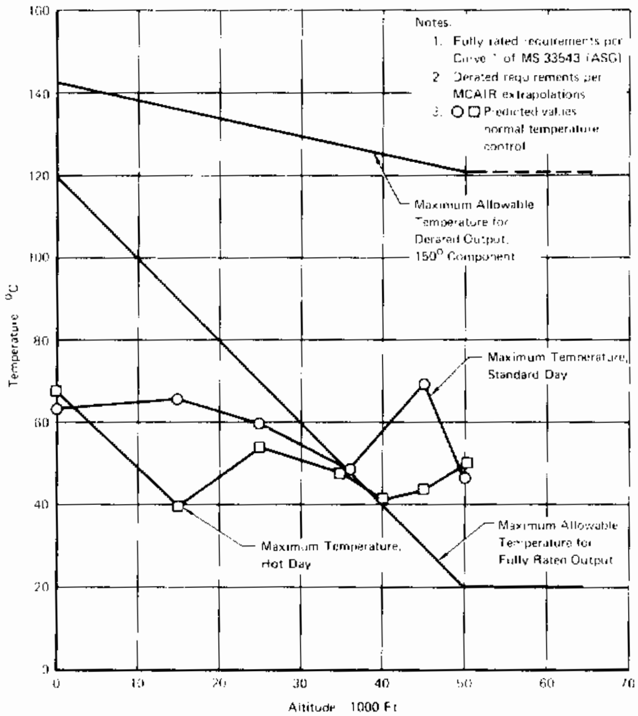


FIGURE 87
TEMPERATURE - ALTITUDE
ENVIRONMENT FOR SFCS TRANSFORMER RECTIFIERS

compartment. The T/R has a self-contained blower. Figure 87 presents the temperature-altitude environment for the SFCS T/Rs for several high speed flight conditions along with the specified and estimated allowables for a fully loaded and derated T/R, respectively. At several flight conditions above 35,000 feet, the air temperatures predicted for these bays, assuming steady state conditions, exceed the maximum temperatures specified by curve 1 of MS 33543 (ASG) for a fully loaded T/R. However, the air temperatures at these flight conditions are well below MCAIR estimates of the maximum ambient temperatures allowable for a T/R derated to 25 amps. The actual T/R load is estimated to be about 20 amps. At this derating, and as long as the aircraft equipment refrigeration package is functioning normally, a maximum silicon diode rectifier temperature of below 100°C is expected during SFCS flights. Further, it appears that at the predicted ambient temperatures, the T/R loading could increase to 65 amps each without the silicon diodes exceeding 150°C, which is a reasonable nominal maximum temperature.

The SFCS batteries experience essentially the same environment as the nearby T/R units. Because of the large mass of each battery, its temperature is not expected to change significantly from initial soak temperatures in the course of a flight. While extremely low initial soak temperatures could cause battery output to be marginal, the ground level atmospheric temperatures in the locale of Edwards Air Force Base and Saint Louis are moderate.

b. Fourth Hydraulics System

A thermal analysis of the SFCS fourth hydraulics system was performed. The extent and results of the analysis are reported in Supplement 3. The fourth hydraulic system has marginal heat transfer capacity. The large thermal time constant that the fourth hydraulics system has when it is supplying flow to its secondary actuator load will permit ground and flight testing during initial program stages without precipitously developing overtemperatures. The analysis indicates that it is feasible to limit fourth hydraulic system temperatures to acceptable levels.

c. SFCES

(1) Summary

The SFCES should provide undegraded performance over the full operating temperature range of the F-4 aircraft. Although all of the equipment subassemblies and discrete components are stressed for the full temperature extremes, other effects must be considered. These effects may be grouped into two areas of interest.

The first area of interest involves eliminating or precluding the possibility of excessive SFCES performance variance because of temperature effects. Early in the program it was thought that these temperature effects could arise from excursions in compartment temperature or from the development of temperature nonuniformities in the compartment locally, and in particular from the development of temperature inequalities between identical components of different SFCS channels. Limited test data to date have not shown this to be a problem. The second area of interest involves assessing and either correcting or accommodating vulnerability of the electronics set to failures of the aircraft refrigeration package. An approach aimed at avoiding and resolving problems in these areas of interest has been established.

The thermal design approach is based on the following SFCES capabilities:

(a) SFCES capability to withstand the environmental temperature expected over the full flight envelope during normal operation of the aircraft refrigeration and temperature control systems. The capability of the F-4 refrigeration packages to provide the predicted cooling capacity is based on previous analyses and flight testing. The environmental requirements imposed upon the SFCES are more severe than that expected when use is made of air from the refrigeration packages.

(b) Acceptable SFCES performance with large temperature excursions and/or differentials between SFCES components. The use of devices with low temperature coefficients and setting the monitor trip levels to reflect system variance over the full temperature range are the primary methods of compensating for component temperature differentials.

(c) Absence of adverse effects on performance, safety, and reliability in the event of an overtemperature condition. On the basis of the 5 minute aircraft limitations in the maximum speed region, the applicable SFCES components were analyzed considering operation in an overtemperature environment for 4.5 minutes with satisfactory results. It was therefore concluded that the overtemperature condition will not have an excessively adverse effect on the factors mentioned.

The above capabilities are, at this point, based on empirical and analytical data, which will be verified or revised in the course of the SFCS program.

The heat dissipation of the SFCES equipment, arranged in accordance with aircraft compartment locations, is presented in Table XXVI. The associated temperature-altitude requirements, that by procurement specification were imposed upon the equipment manufacturer, are presented in Figures 88 through 91. Where indicated, the approximate temperatures expected are presented also. The nose compartment, the former infrared compartment, and the cockpit are the only SFCES equipment locations to be supplied conditioned air from the equipment or cockpit refrigeration packages. No airflow is delivered to the SFCES equipment itself; as is the case in uncooled compartments, adequate heat transfer is expected to be achieved by means not dependent upon air delivery: natural convection, thermal radiation and thermal conduction.

For the nose compartment, Figure 91 is applicable. During normal operation, airflow is to be delivered to the nose compartment from the equipment refrigeration package at rates nominally equal to that for the RF-4C. The airflow is to be distributed in a manner that will provide a uniform compartment temperature to help eliminate potentially undesirable thermal gradients within the SFCES. The nose compartment outlet air temperature is to be controlled to a nominal 35°C as was the case for the RF-4C camera compartment. The nominal outlet air temperature cannot always be maintained, but the temperature of the compartment will be controlled to limits well within the equipment specification requirements. The range of this control is depicted in Figure 91 as the crosshatched region. The temperatures may momentarily go outside the normal control range while the aircraft is being brought to a more benign flight condition in the eventuality of a temperature control or a refrigeration package failure as is provided for in the design requirements and as is discussed in a later paragraph of this subsection.

The environment in the cockpit is subject to the comfort level selected by the pilot, when attainable, but does not exceed those requirements set forth in Figure 90.

The requirements for the wing root compartments are presented in Figure 88. The wing root compartments are not subjected to substantial equipment heat sources and for all practical purposes

TABLE XXVI
HEAT DISSIPATION OF SFCES EQUIPMENT

Compartment	Unit Dash No.	Nomenclature	No. Reqd.	Heat Dissipation Each (Watts) Normal/Other	Subtotal Heat Dissipation (Watts) Normal/Other
Nose					
	–3	Computer Voter Unit	4	44.0	176.0
	*	Secondary Actuator Power Unit	4	16.0/36.0 (1)	64.0/144.0 (1)
	–5	Adaptive Gain and Stall Warning Unit	1	7.5	7.5
	–9	Roll Rate Sensor Unit	1	20.0	
	–13	Normal Accelerometer Sensor Unit	1	2.0	2.0
	–29	Maintenance Test Panel	1	15.0	15.0
					264.5/344.5
Infrared Compartment	–15	Lateral Accelerometer Sensor Unit (3)	1	2.0	2.0
Wing Roots					
	–11	Yaw Rate Sensor Unit	1	20.0	
	–7	Pitch Rate Sensor Unit	1	20.0	
Cockpit					
	–17	Control Stick Transducer Unit	1	4.0	
	–19	Pedal Transducer Unit	1	2.0	
	–21	Master Control and Display Panel	1	2.0/82.0 (2)	
	–23	Secondary Control and Display Panel	1	2.0/93.0 (2)	
	–25	Trim Control Panel (Forward Cockpit)	1	1.4	
	–27	Trim Control Panel (Aft Cockpit)	1	1.8	
	–33	Discrete Function Generator Panel	1	1.0	
Aft Fuselage					
	–31	Stabilator Surface Position Transducer	1	3.0	

Notes:

(1) Heat dissipation for stalled SSAP secondary actuator.

(2) Heat dissipation for BIT.

(3) Same thermal environment requirements as nose compartment.

*Combined with computer voter unit. Separate dash no. deleted.

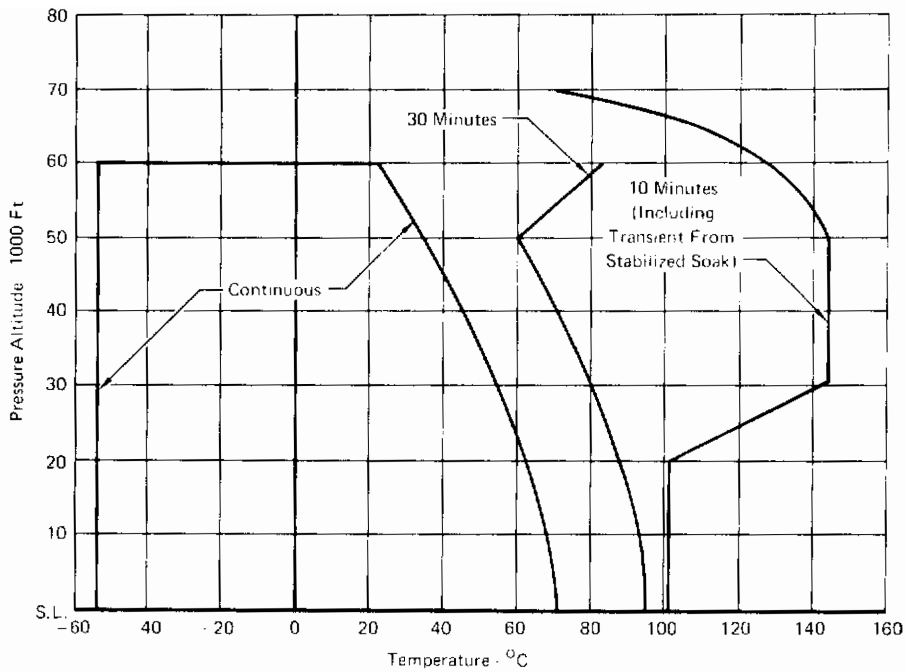


FIGURE 88

TEMPERATURE-ALTITUDE REQUIREMENTS - WING ROOTS

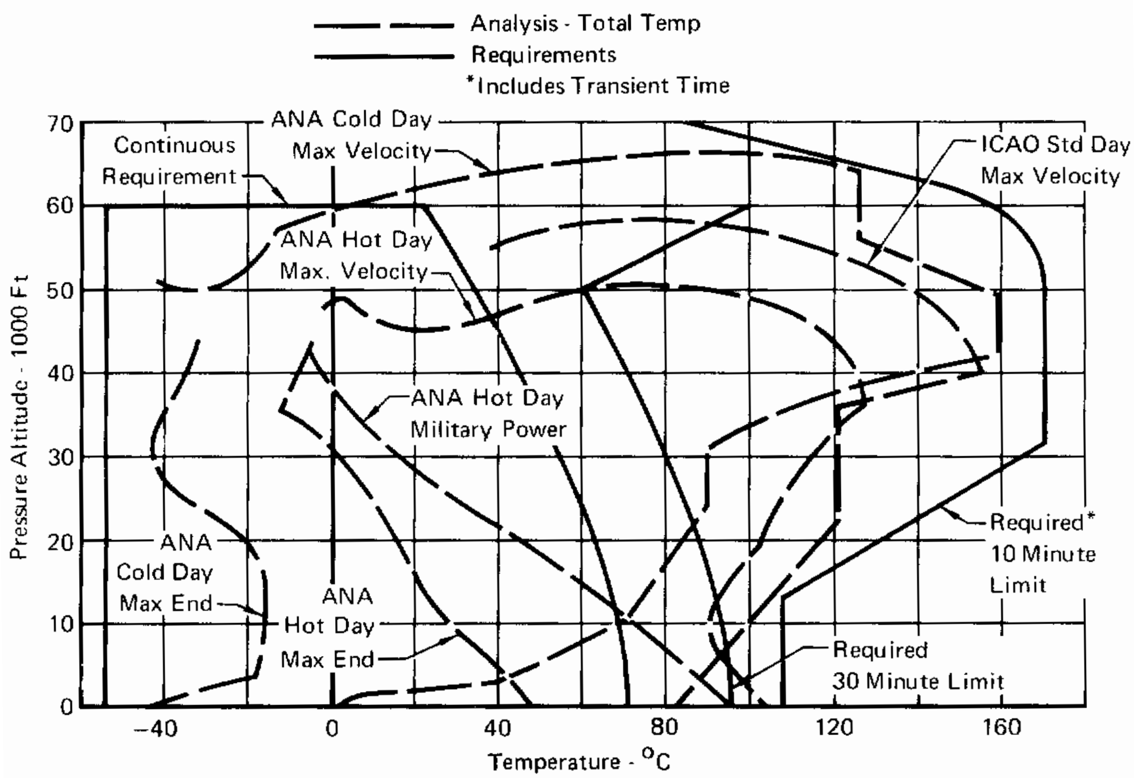


FIGURE 89

TEMPERATURE-ALTITUDE REQUIREMENTS - AFT FUSELAGE

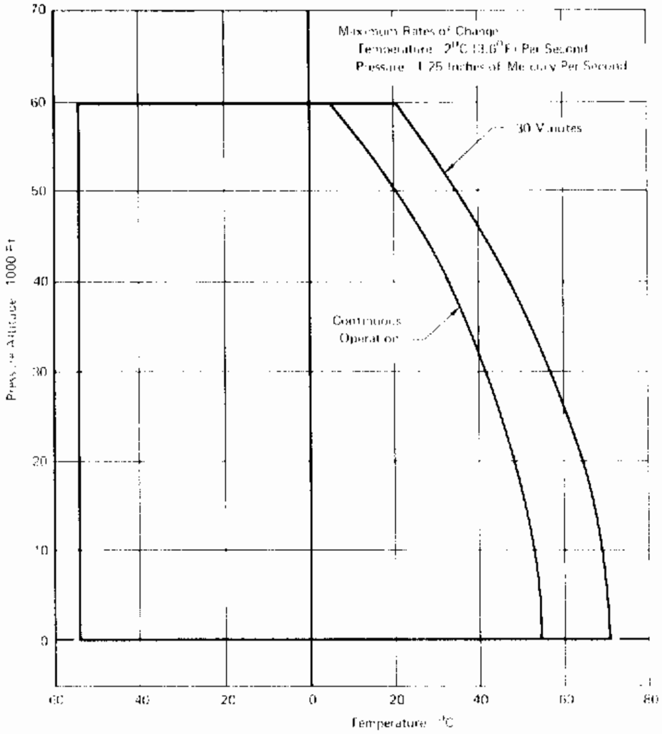


FIGURE 90
TEMPERATURE-ALTITUDE REQUIREMENTS - COCKPIT

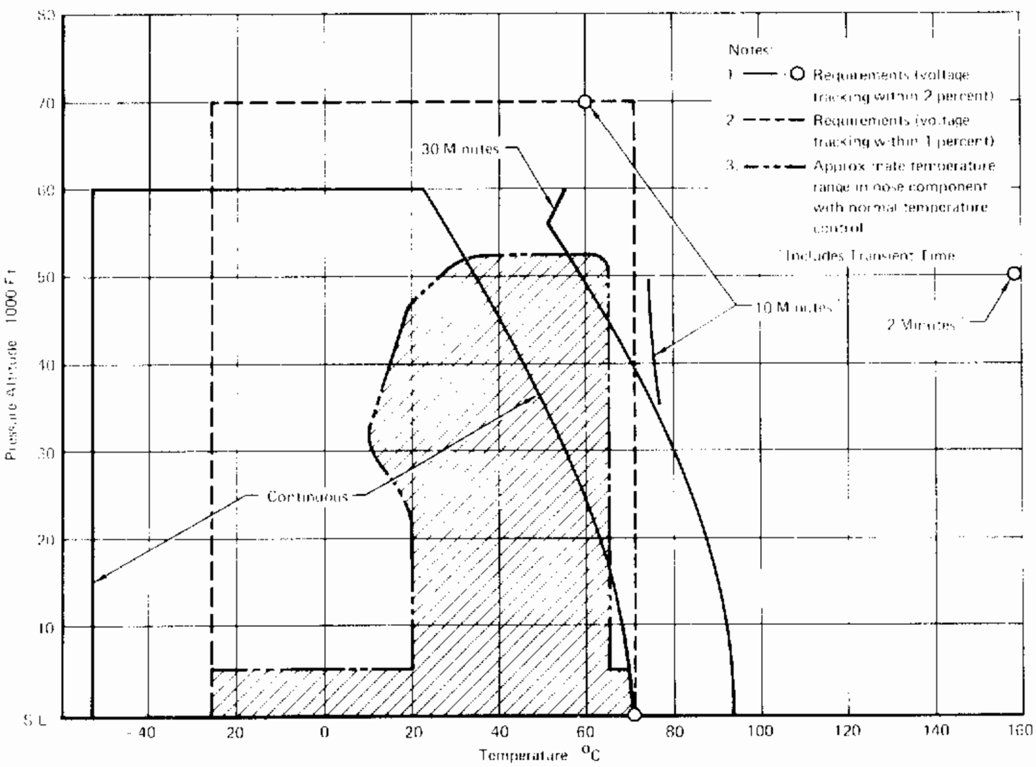


FIGURE 91
TEMPERATURE-ALTITUDE ENVIRONMENT - NOSE COMPARTMENT

cannot reach temperatures outside the adiabatic wall temperature range. During most flight conditions the wing root compartment temperatures lie well within the extremes of Figure 88.

For SFCES equipment located in the aft fuselage, Figure 89 is applicable. As indicated, the free stream total temperature at various flight conditions falls within the equipment specification requirements. For the "short lived" conditions (10 and 30 minutes), aircraft limitations do not allow the requirements to be exceeded. Because the thermal lag and the ram air flow provided offset the effect of heat dissipation within the aft fuselage in Phase IIA and IIB, the maximum analytical total temperatures are not expected to be exceeded.

In summary, the temperature-altitude environments that are provided for the SFCES equipment are satisfactorily confined within the equipment specification requirements. On the other hand, some of the options available to achieve closer control of cooled compartment temperatures were not implemented. By initially avoiding extraordinary temperature control provisions it should be easier to conclude, if no temperature effect problems are encountered in flight testing, that adequate SFCES performance is attainable with "ordinary" or typical temperature control provisions.

(2) Performance With Large Temperature Excursions and Large Temperature Gradients

When the different components of the SFCES are started and operated in different ambient temperatures, the gains and drift characteristics of amplifiers and sensors are affected. These effects are random in nature, so a positive or negative gain or offset variation of any magnitude up to the bounds of the temperature coefficient can be expected from each sensor and amplifier. For this reason, open loop temperature compensation is not a realistic solution to the problem. In the SFCES, two techniques are being employed to control temperature effects. The first is the use of low temperature coefficient resistors, capacitors, and sensors; the second is the setting of monitor trip levels to reflect system variance over the full temperature range for nuisance disconnect prevention.

An 1800 Hz power supply is contained in the CVU for each channel and delivers excitation power to each sensor in that channel. A 1 percent tracking between channels is to be met over a -26°C to +71°C temperature range and a 2 percent tracking between -55°C and -26°C. Because of the random nature of the temperature effects, the most desirable method for meeting a tracking requirement is to design for minimum temperature effects. As of this date, four 1800 Hz reference oscillators have been constructed, calibrated, and tested over the temperature range. These tests have demonstrated random (as expected) variations of less than ± 0.5 percent over the full operating range, and thus indicate that the requirements will be met at the system level.

The randomness of temperature effects negates the dependence on temperature tracking between boxes as a means of assuring electrical parameter tracking. It is not felt that tolerances between two channels with one at +25°C and the other at +71°C will be greater than when both are at +71°C.

The input voltage levels have a minimal effect upon the 1800 Hz magnitude while within the specified limits of 20.0 to 28.3 volts DC. The input power is preregulated so as to minimize the effects of changes and to minimize normal power dissipations within the LRU. The 1800 Hz tracking is therefore expected to be held over the full input power range.

(3) Vulnerability of SFCES to Aircraft Refrigeration Package Failures

A general capability of the quadruple redundant SFCES is that it retains full performance capability after a single failure. However, when installed in the aircraft this capability is of little avail if a single environmental control system (ECS) failure completely or partially disables the SFCES. Ideally, to completely preserve the redundancy concept, either the SFCES must be designed to need no external cooling provisions, or the cooling provisions themselves must feature adequate redundancy. Neither of these two design approaches was feasible for the SFCS program. To determine the proper design approach, the following ones were considered:

- Design the equipment for the severe environment resulting with no cooling air. This may involve the employment of self-contained (or individual) active, passive or expendable cooling provisions which have cost, weight, size, maintainability or schedule disadvantages. For equipment having low heat dissipation, a high inherent heat storage capacity in its own housing and tray and the transient protection offered by its compartment could combine to allow its survival for high speed, high temperature transients.

- Locate SFCES equipment in the cockpit. This would allow full SFCES capability if the equipment refrigeration package failed. If the cockpit package failed, flight speed would be limited anyway by the ability of the crew to tolerate high temperatures. The equipment could readily be designed to crew tolerance temperatures. The lack of available space in the cockpits is the major disadvantage.

- Design the equipment for the basic requirement of normal operation in cooled compartments. Also, design the equipment to, as a minimum, survive the overtemperature period after failure before corrective action becomes effective. As the program progresses, ascertain whether the equipment can actually withstand an extended overtemperature condition.

A combination of the first and last approaches above was selected. The resulting thermal requirements and the approximate environmental control capabilities were previously discussed. When cooling air for the nose compartment is lost, the aircraft flight speed will be reduced or limited in accordance with a predetermined flight placard. A tentative placard which specifies the flight envelope in which SFCES compartment temperatures without cooling air are acceptable is shown in Figure 92. The electronics are required to tolerate the brief overtemperature condition that could prevail if, at the maximum speed condition at 50,000 feet, the equipment refrigeration package failed and hot ram air or its equivalent in temperature is directed to the equipment compartment.

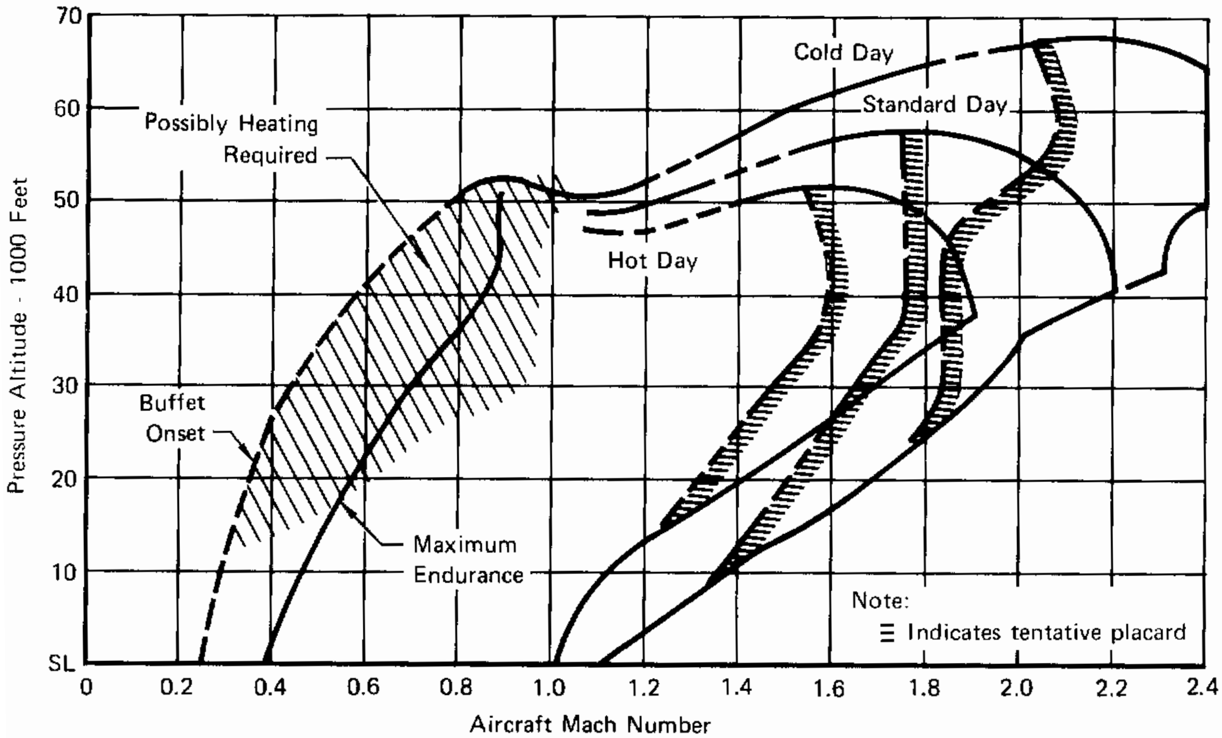


FIGURE 92

SFCS FLIGHT ENVELOPE WITHOUT COOLING AIR

To simulate the period in which overtemperature detection and pilot corrective reaction occur, a one minute period of temperature buildup from 75°C to 159°C is specified. To simulate the period of aircraft deceleration and heat retention, a one minute period in which the temperature declines to 75°C is specified. This overtemperature condition is imposed upon the applicable SFCES equipment as a design requirement and a temperature altitude test requirement. The condition is graphically presented in Figure 93. Also presented in the figure are the simplified temperature-time profiles assumed by the vendor in analyzing both the overtemperature requirement and the overtemperature extension. The capability of the SFCS equipment to survive an extended overtemperature condition indicates that the SFCS could survive a

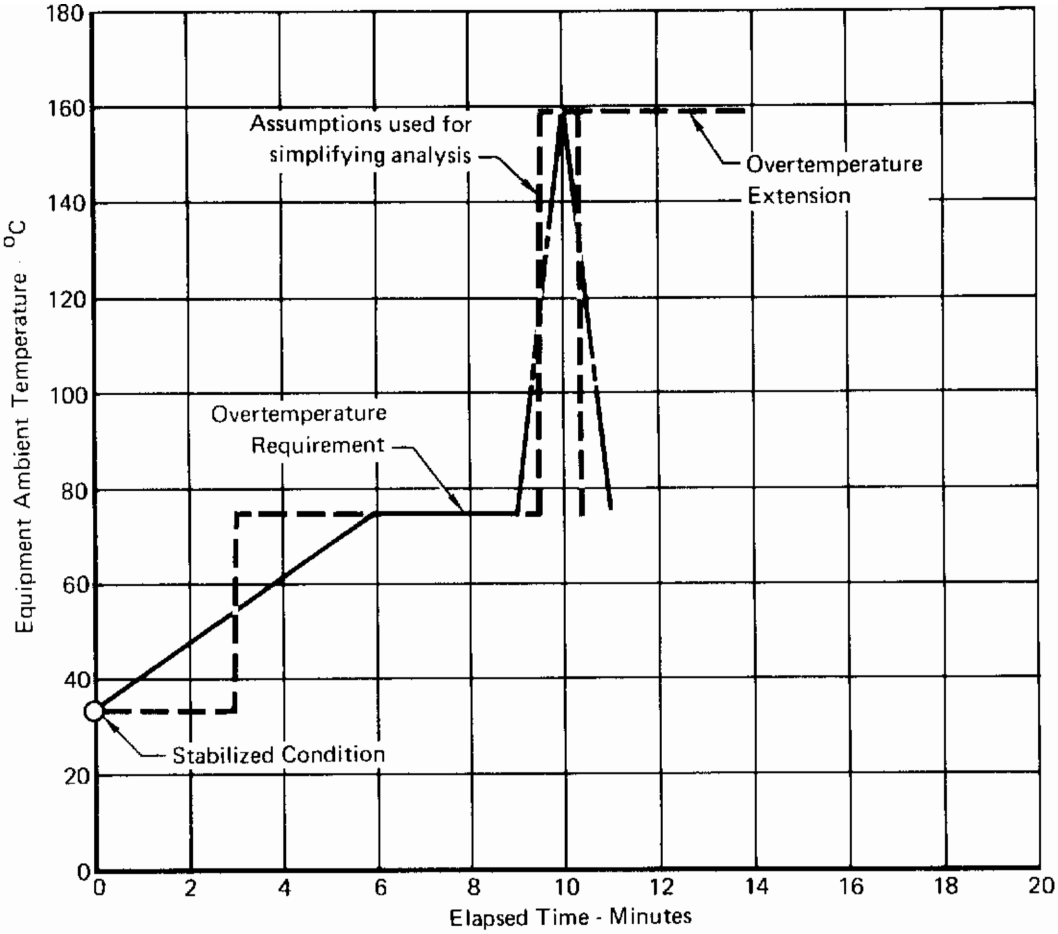


FIGURE 93
OVERTEMPERATURE CONDITIONS

refrigeration package failure at the maximum speed condition even if the pilot failed to immediately decelerate the aircraft. The limitations of non-SFCS equipment, of course, preclude attempting to utilize this capability in flight tests.

A transient analysis was performed to determine the SFCES temperatures ensuing from an extended overtemperature condition. Applicable units include the three electronic computers, the normal and lateral accelerometers, and the roll rate sensor package.

(a) Computers

Table XXVII shows the results of the transient thermal analysis for extension of the overtemperature condition to 4.5 minutes. Component temperatures are within the reliable operating range.

The start temperature listed in Table XXVII is the temperature to which the unit had risen just prior to the initiation of the temperature change from 75°C to 159°C.

TABLE XXVII
SFCES COMPUTER TEMPERATURES
DURING EXTENDED OVER TEMPERATURE CONDITION

Component	Computer Temperature, °C					
	Computer and Voter Unit Temperature		Gain and Stall Computer Temperature		Maintenance Test Panel Temperature ⚠1	
	Start	End	Start	End	Start	End
Encapsulated Module						
Component Surface	67.9	81.6	62.5	75.5		
Semiconductor Junction	69.9	83.6	64.5	77.5		
Power Supply Assembly						
Component Surface	79.2	107.4	72.4	102.8	70.1	101.2
Semiconductor Junction	82.2	110.4	75.4	105.8	73.1	104.2
Printed Wiring Card Assembly						
Semiconductor Junction					69.9	83.6

⚠1 These temperatures are for inflight MTP operation and are for information only. The MTP is automatically switched off in flight.

The extension of the overtemperature condition from 1.0 to 4.5 minutes causes an increase in junction temperatures of approximately 10°C. The SFCES uses silicone semiconductor devices which are operational at junction temperatures in excess of 150°C. The additional 10°C does not result in reliability degradation assuming that the overtemperature transient is not a frequent occurrence.

The overtemperature condition does not significantly accelerate failure modes or wearout conditions because the long term average temperature of the unit is not significantly altered.

The design criteria are such that the channel monitoring levels are not reached at temperature extremes. Rare drift magnitudes and tolerance buildups can be postulated to make a given channel marginal at the upper temperature extreme. The possibility that the extended overtemperature condition would cause the disengagement of a channel is considered remote. However, in the event that a channel should trip, it could be reset after the demise of the temperature transient. The overtemperature condition, therefore, should have a negligible effect on SFCES performance.

(b) Accelerometer Units and Roll Rate Sensor Units

The temperature of the accelerometer and the roll rate sensor units before and after exposure to the overtemperature extension was predicted and is shown in Table XXVIII.

TABLE XXVIII

SFCES SENSOR TEMPERATURES

DURING EXTENDED OVERTEMPERATURE CONDITION

Sensor and Component	Temperature	
	Start	End
Normal Accelerometer		
Surface	52.4°C	95.3°C
Lateral Accelerometer		
Surface	52.4°C	95.3°C
Roll Rate Gyro		
Roll Gyro Surface	67.7°C	103.3°C
Module Component Surface	67.8°C	114.9°C
Module Component Semi-conductor Junction	69.8°C	116.9°C

The analysis shows that the maximum temperature reached by the accelerometer surface after exposure to an extension of the overtemperature condition to 4.5 minutes was 95°C. This temperature does not significantly affect the operation, safety, or reliability of the unit.

The analysis shows that the maximum temperatures reached by the roll rate sensor unit after exposure to an extension of the overtemperature condition to 4.5 minutes are within the reliable operating range and thus do not affect the operation, safety, or reliability of the unit.

d. Survivable Stabilator Actuator Package (SSAP)

(1) Summary

The principal thermal consideration of the SSAP is that it is an integrated actuator package (IAP) that includes two motor pump units whose total power requirement is converted to waste heat virtually all inside the package itself. The package dissipates part of this heat by radiation and convection from the surfaces of the package hydraulic components. An ambient cooled oil-to-air heat exchanger is provided for each motor pump Line Replaceable Unit (LRU) to dissipate the remaining heat generated within the hydraulic components. Each heat exchanger employs a blower at its air outlet to pull ambient air through the heat exchanger core. To achieve the desired heat transfer rate while still maintaining an acceptable heat exchanger size, ram air is ducted to the immediate vicinity of the heat exchanger inlet.

While some of the already heated compartment air is entrained in the ram air stream between the duct and the heat exchanger face, the heat exchanger inlet temperature is significantly below the prevailing compartment temperature.

The electric motor driving each hydraulic pump is cooled by a self-contained blower which forces compartment air across the finned housing of the motor. The electric motor is essentially thermally isolated from the hydraulic components in that it dissipates its heat directly to the compartment. The heat transfer from the motor is primarily by convection, although thermal radiation to the compartment walls also occurs.

The SSAP employs an electromechanical secondary actuator which, like the pump motor, dissipates its heat directly to the compartment. Fans are contained in the secondary actuator to cool the electric motors.

The heat sink employed is ram air from a scoop located in the vertical fin leading edge. It is planned to replace the F-4B type scoop that was originally in the test aircraft with an F4J/K type scoop which has a higher flow capacity. The ram air scoop is attached to ducting which distributes the ram air flow to the blast panel cooling passages, to the parabrake cooling passages, and to the aft fuselage compartment in which the SSAP and fourth hydraulic system are located. The aft fuselage and parabrake ventilating air are discharged through a port located in the parabrake tail cone.

(2) Thermal Considerations

Thermal considerations are a primary concern in IAP design. Potential operating temperatures for an IAP design without the proper heat transfer provisions are unacceptably high. The package temperatures ensuing from operation of such thermally unimproved designs are presented in Supplement 3. A preliminary thermal model of the SSAP design that was eventually selected indicated that, without special heat transfer provisions, a fluid temperature in the neighborhood of 585°F could occur. Further, when this preliminary thermal model is revised to reflect the higher heat generation values recently estimated by the SSAP manufacturer, there are indications that a fluid temperature in the neighborhood of 700°F could occur. Many factors discussed in detail in Supplement 3 contributed to this high temperature. In brief, conventional small volume closed loop hydraulic systems without special heat transfer provisions have inadequate ability to store the heat generated therein or to dissipate it to the normally available ambient air at an acceptable fluid temperature. A general comparison of the thermal characteristics of central hydraulics systems with integrated actuator packages is presented in Table XXIX for the test aircraft. The fourth hydraulics system is of limited heat transfer capacity like the IAP but, unlike the SSAP, is able to operate at 275°F or

TABLE XXIX

HYDRAULIC SOURCE THERMAL CHARACTERISTICS

Thermal Characteristic	Central	IAP	Fourth Hydraulic System
Heat Dissipation Area	Long lines to remote actuators add beneficial heat transfer area.	Small Surface Area	Small surface area. Relatively long lines to remote secondary actuators and the secondary actuators themselves add beneficial heat transfer area.
Heat Storage Capacity	Large Fluid Volume	Small Fluid Volume	Small fluid volume. However, four (or three) secondary actuators, the tubing and the APU itself contribute large heat storage capacity and provide a large thermal time constant.
Duty Cycle	Moderate Percentage of Capacity on Average	Potentially High Percentage of Capacity	Constant at about 17.5% of full flow capacity.
Special Heat Transfer Provisions	Hydraulics to Fuel Heat Exchanger	Ambient Cooled Heat Exchangers	Heat transfer by conduction through secondary actuator housings to central hydraulics systems supplements ambient cooling.
Flight Profile & Maneuvering	Full Capability	Full Capability	Marginal capability to provide fourth hydraulic source for four secondary actuators at low fluid temperatures.
Operating Temperature	Normal $< 85^{o}F$ (Estimated) Maximum $175^{o}F$ (Estimated)	$450^{o}F$ Design Temperature	$\leqslant 275^{o}F$ under normal conditions.

below. The fourth hydraulic system is a special case, however in that it dissipates a major portion of its heat to the central hydraulics system. Additionally, the secondary actuators served by the fourth hydraulic system contribute a great deal of heat storage capacity at a remote location. The SSAP is specified not to depend upon the central hydraulics systems or any other vulnerable heat sink.

(3) Thermal Design Plan

A program of thermal analysis and testing has been devised for the SSAP. For referral purposes, it is designated as SSAP "Thermal Design Plan". The Thermal Design Plan outlines the SSAP thermal design approach and helps assure that thermal considerations receive the proper design emphasis. The plan is intended to help dovetail testing with analysis in the most useful manner promoting the earliest updating of analytical data with test data.

Special data and adjunct data of opportunity from bench tests, vendor motor pump tests, MCAIR iron bird tests, and the temperature altitude tests should provide part by part confirmation of or updating of thermal parameters.

(4) Protective Ventilation

It is conceivable that the SSAP may be able to operate within its own temperature limits but cause overheating of surrounding equipment, such as the fourth hydraulic system. Overheating occurs when the compartment rises to unacceptable temperatures due to the heat it receives from the SSAP. Protective ventilation is supplied in the form of airflow through the compartment to reduce the probability of encountering an excessive compartment temperature rise.

A successful completion of the temperature altitude tests will be indicative that SSAP temperature problems should not appear in the actual flight testing.

The scoop and ram air ducting provided in the test aircraft is believed to offer adequate flexibility in airflow rate and scheme of delivery or airflow distribution to accommodate any reasonably expected deviation in thermal design parameters from predicted values. The temperature altitude tests offer an opportunity to empirically discover any important thermal parameter deviations and to evaluate the functional effects of a thermal design of the lowest reasonable aircraft penalty.

(5) SSAP Transient Temperature Performance

The SSAP design is based on the environmental design requirements presented in Table XXX and the duty cycle and loading requirements presented in Table XXXI. Ram airflow rates, ram air temperatures and operating times are shown for flight conditions throughout the flight envelope in Figure 94.

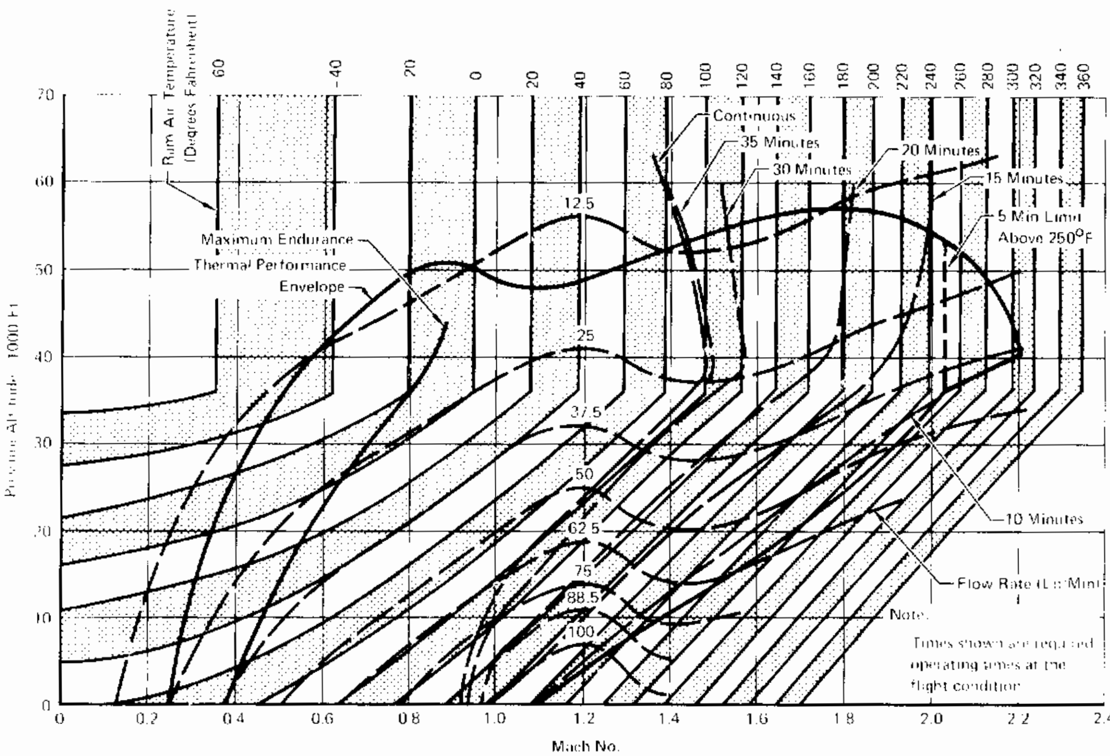
Preliminary predictions of transient temperature performance as a function of package temperatures, compartment temperatures, and heat exchanger heat flows for the SSAP are shown in Figures 95 through 97. The package temperature is the temperature of the return fluid from the switching valves and MCV. Since these data were developed, however, the anticipated heat exchanger arrangement has been revised: the heat exchanger capacity is lower, the heat exchanger blower is at the heat exchanger outlet, and ram air is to be ducted to the vicinity of the heat exchanger inlet. As discussed in paragraph (6), the revised analysis reflecting these changes is presently in progress. The package temperatures with the revised analysis are expected to be lower than those presented in these figures. These figures show most flight conditions to result in a package temperature below 400°F. The package is designed to operate at 450°F. Flight Condition 12 package temperatures are shown to exceed the upper design limit by about 15°F. The analysis for Flight Condition 12 can be rationalized as conservative since a step rise in compartment temperature was assumed as

TABLE XXX
TEMPERATURE ALTITUDE ENVIRONMENTAL CONDITIONS

	Cold Start-Up	Low Speed Cold Soak	Ground Check-out and Taxi	Low Speed Hot Soak	Sea Level Cruise	Sea Level Max Speed	High Alt Cruise	High Alt Soak	High Alt Max Speed	Max Alt High Speed	High Load Condition	High Alt, High Speed, Long Duration	Landing Approach	Touch and Go
Flight Condition Ref. No.	1	2	3	4	5	6	7	8	9	10	11	12	13	14
Mach No.	0	0.4	0	0.39	0.71	1.025	0.94	1.46	2.4	2.15	2.0	2.04	0.21	0.18
Altitude (1000 Ft)	SL	SL	SL	SL	SL	SL	45	50	50	70	35	53	SL	SL
Design Day	$-40^{o}F$	Cold	$125^{o}F$	Hot	Hot	Hot	Hot	ICAO Std	Cold Sta	Cold	ICAO Std	ICAO Std	$125^{o}F$	$125^{o}F$
Duration of Flight Condition (Min.)	–	Cont	Cont	Cont	Cont	20	Cont	Cont	4.5	1	13.5	14	Cont	5
Time to Reach Flight Condition (Min.)	–	3	–	–	–	2.5	–	–	8	1	3.3	15	–	0.5
Previous Flight Condition Ref. No.	–	1	–	–	–	3	–	–	8	8 Plus 3.5 Min at 9	8	3, 7 or 8	–	13
Air Inlet Temp Tr (^{o}F)	–	51	128	120	160	222	35	95	318	185	250	250	135	129
Air Flow Rate W ram (Lb/Min.)	–	–	–	37.5	65	100	15.6	13	30	9.7	40	17.5	17	10
Air Temp. Rise (Tc-Tr)/Kw (^{o}F/Kw)	–	2.37	23.7	6.32	3.66	2.37	15.2	18.2	7.9	24.4	5.94	13.52	13.93	23.7
Wall Temp. Taw (^{o}F)	–	–52	–	118	155	206	26.6	78	266	151	218	220	134	128
Nominal Available Compartment Air Velocity for Heat Transfer Enhancement (Ft/Sec)	2.5	15	2.5	9.0	–	41	32	32	75	59	70	–	4.0	2.2

TABLE XXXI
FLIGHT CONDITION LOADS AND DUTY CYCLES

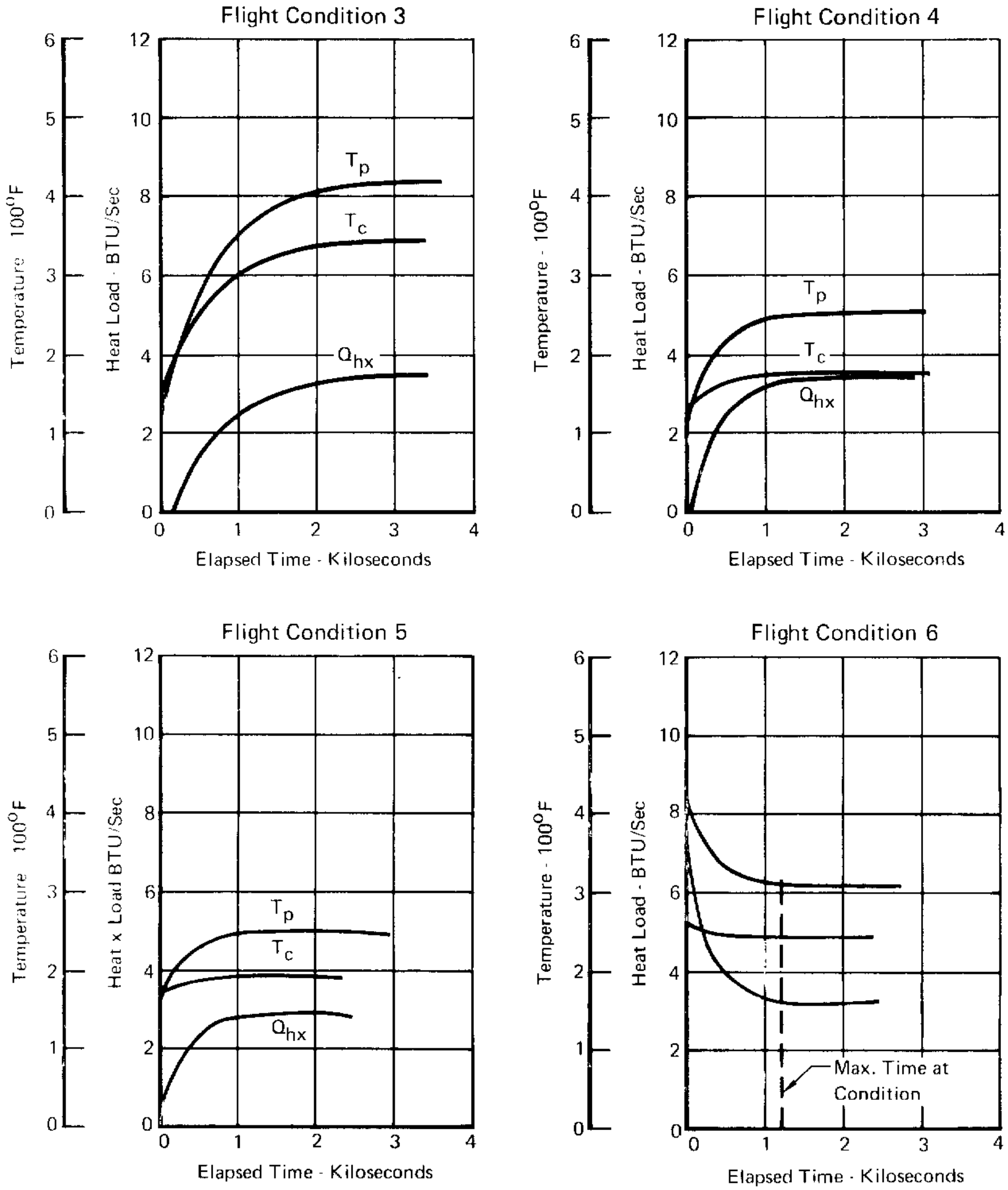
	Cold Start-Up	Low Speed Cold Soak	Ground Checkout and Taxi	Low Speed Hot Soak	Sea Level Cruise	Sea Level Max Speed	High Alt Cruise	High Alt Soak	High Alt Max Speed	Max Alt High Speed	High Load Condition	High Alt, High Speed, Long Duration	Dog Fight and Landing Approach	Touch and Go
Flight Condition Reference No.	1	2	3	4	5	6	7	8	9	10	11	12	13	14
Mach No.	0	0.4	0	0.39	0.71	1.025	0.94	1.46	2.4	2.15	2.0	2.04	0.21	0.18
Altitude (1000 Ft)	SL	SL	SL	SL	SL	SL	45	50	50	70	35	53	SL	SL
Actuator Trim Load (1000 Lb)	0	2.0	0	2.0	4.8	7.5	2.0	5.0	8.3	3.2	8.0	6.0	1.0	1.0
	Small Amplitude Duty Cycle													
Actuator Amplitude Peak to Peak (In.)	0	0	2.0	2.0	0.7	0.6	2.0	2.0	2.0	2.0	1.0	2.0	2.0	2.0
Frequency (Cycles per Sec)	0	0	0.5	0.5	1	1.5	0.5	0.5	0.5	0.5	1.0	0.5	0.5	0.5
Duty Cycle Period (Min)	0	0	Cont	Cont	Cont	Cont	Cont	1	1	1	1	1	Cont	Cont
Time Between Periods (Min)	0	0	0	0	0	0	0	4	4	4	4	4	0	0
	Maneuver Duty Cycle (Intermittent)													
Actuator Amplitude Peak to Peak (In.)	8.0	0	4.0	4.0	2.0	1.2	4.0	4.0	3.4	4.0	1.8	4.0	4.0	4.0
Frequency (Cycles per Sec)	0.25	0	0.5	0.5	1.0	1.5	0.5	0.5	0.5	0.5	1.0	0.5	0.5	0.5
Loading Period (No. per Cycles)	1	0	2	2	1	2	2	2	2	2	2	2	2	2
Time Between Periods (Sec)	45	0	30	60	60	60	60	60	90	90	90	90	20	20



Total Temperature and Flow Rate ICAO Standard Day

FIGURE 94

RAM AIRFLOW FOR SSAP VENTILATION



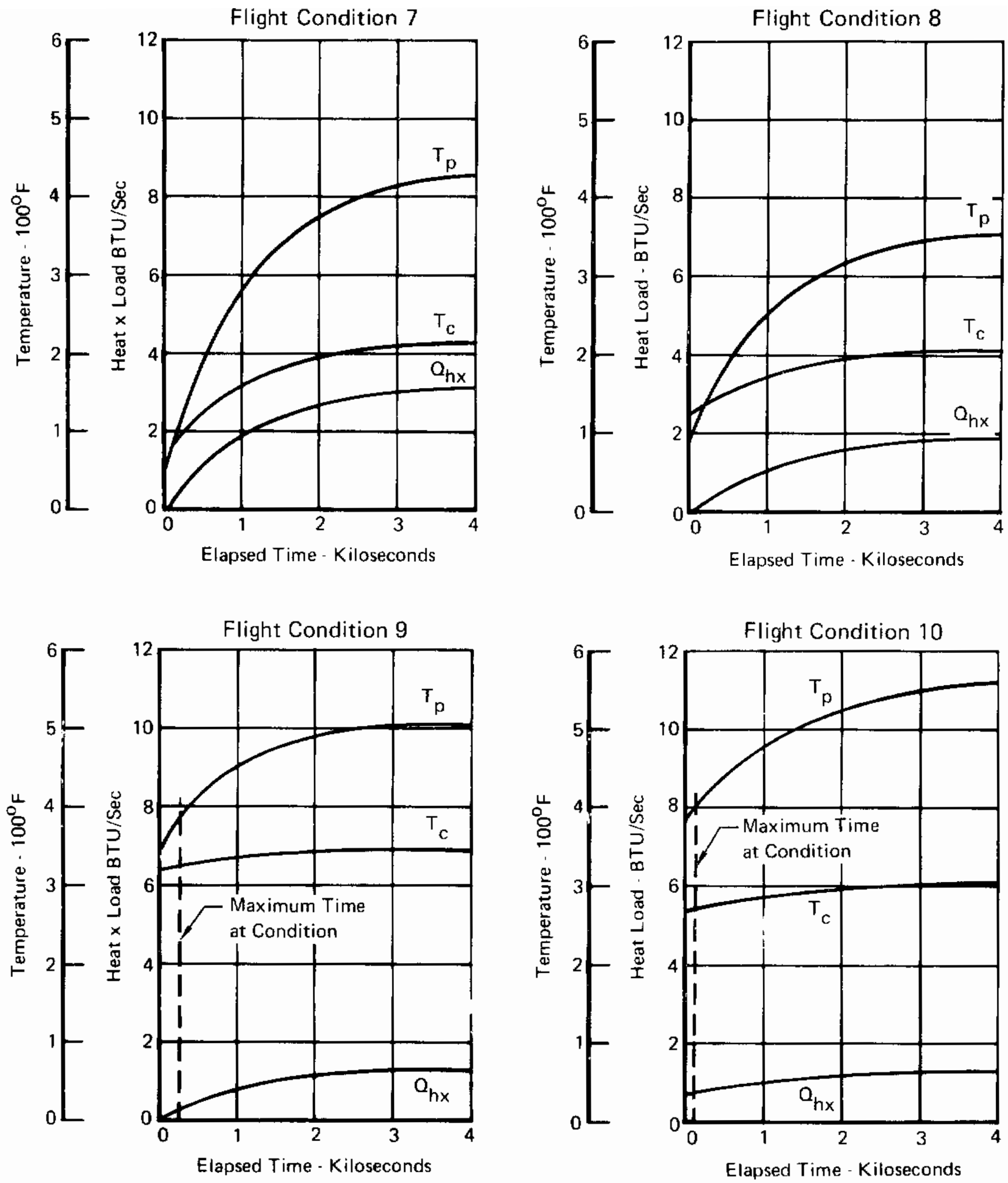
T_p = Package Temperature

T_c = Compartment Temperature

Q_{hx} = Heat Exchanger Heat Flow

FIGURE 95

SSAP TRANSIENT TEMPERATURE PERFORMANCE



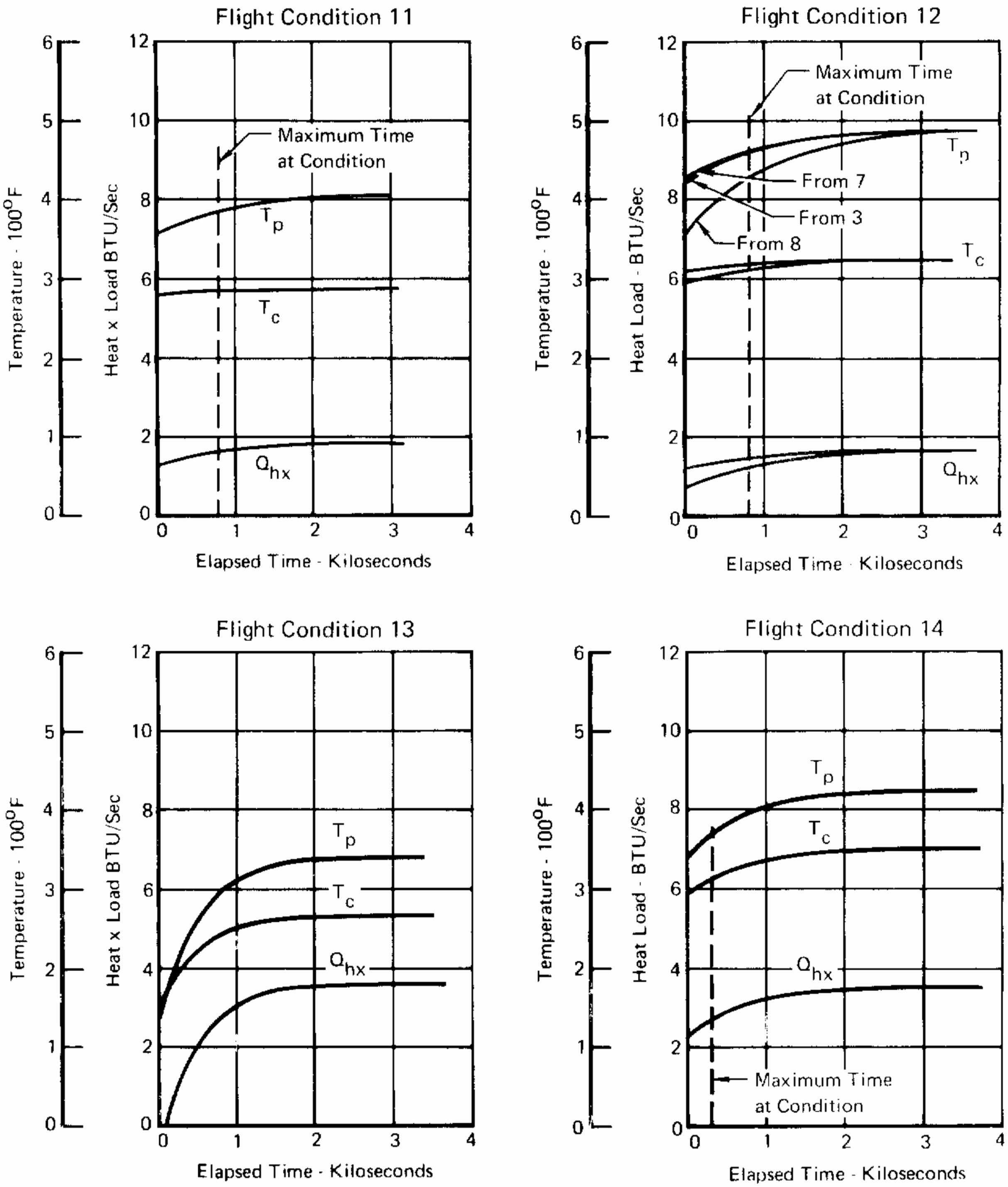
T_p = Package Temperature

T_c = Compartment Temperature

Q_{hx} = Heat Exchanger Heat Flow

FIGURE 96

SSAP TRANSIENT TEMPERATURE PERFORMANCE



T_p = Package Temperature

T_c = Compartment Temperature

Q_{hx} = Heat Exchanger Heat Flow

FIGURE 97

SSAP TRANSIENT TEMPERATURE PERFORMANCE

the aircraft goes into Flight Condition 12 from 7. In reality, a gradual increase would occur and a reasonable estimate of maximum package temperature on this basis would be 455°F.

(6) SSAP Heat Exchange Loads and Provisions

The total power lost by the SSAP is shown in Table XXXII for the various flight conditions defined as design requirements. A detailed breakdown of pump motor and hydraulic losses are contained in Tables XXXIII and XXXIV, respectively. The SSAP hydraulic pump power losses in terms of the pressure-flow characteristics of the pumps, which determine the hydraulic power output, and the estimated efficiency of the pump at several fluid temperatures are shown in Figure 98. Calculated electric motor efficiency is shown in Figure 99.

TABLE XXXII

SSAP TOTAL POWER LOSSES

Flight Conditions	→	1	2	3	4	5	6	7
1 Hyd. Losses	BTU/Sec	0.467	0.177	2.16	1.95	1.42	1.86	2.06
2 Pump Losses	BTU/Sec	1.607	←					
3 Motor Losses	BTU/Sec	0.526	0.482	0.710	0.691	0.607	0.649	0.691
4 Blower Losses	BTU/Sec	0.382	0.398	0.327	0.359	0.355	0.341	0.111
5 Total/Side	BTU/Sec	2.98	2.66	4.80	4.61	3.99	4.46	4.47
6 Total/SSAP	BTU/Sec	5.96	5.32	9.60	9.22	7.98	8.92	8.94

Flight Conditions	→	8	9	10	11	12	13	14
1 Hyd. Losses	BTU/Sec	0.75	0.785	0.828	0.602	0.828	2.24	2.24
2 Pump Losses	BTU/Sec						→	1.607
3 Motor Losses	BTU/Sec	0.554	0.539	0.543	0.530	0.543	0.727	0.727
4 Blower Losses	BTU/Sec	0.095	0.085	0.061	0.137	0.081	0.341	0.324
5 Total/Side	BTU/Sec	3.01	3.02	3.04	2.88	3.06	4.92	4.90
6 Total/SSAP	BTU/Sec	6.02	6.04	6.08	5.76	6.12	9.84	9.80

Note: Electromechanical secondary actuator not included

TABLE XXXIII

PUMP MOTOR LOSSES

Flight Condition	→	1	2	3	4	5	6	7	8	9	10	11	12	13	14
Small Amp Condition															
1 Actuator Vel (Peak)	In/Sec	0	0	3.14	3.14	2.2	2.83	3.14						→	3.14
2 Pump Flow (Peak)	In^3/Sec			18.1	18.1	12.7	16.3	18.1						→	18.1
3 Pump Flow (Ave)	In^3/Sec			11.51	11.51	8.08	10.37	11.51						→	11.51
4 Motor Loss (Ave)	Kin-Lb/Sec			6.25	6.25	5.6	6.0	5.25						→	6.25
5 Active Time/Total Time				30/34	60/64	60/61	60/61.5	60/64	1/5			→	1/5	20/24	20/24
6 Motor Losses (Active Ave)	Kin-Lb/Sec	0	0	5.51	5.86	5.51	5.85	5.86	1.25			→	1.25	5.21	5.21
Maneuver Condition															
7 Actuator Vel (Peak)	In/Sec	6.28	0	6.28	→	6.28	5.66	6.28	6.28	5.33	6.28	5.65	6.28	→	6.78
8 Pump Flow (Peak)	In^3/Sec	36.18		36.18	→	36.18	32.6	36.18	36.18	30.7	36.18	32.6	36.18	→	36.18
9 Pump Flow (Ave)	In^3/Sec	23.01		23.01	→	23.01	20.73	23.01	23.01	19.53	23.01	20.73	23.01	→	23.01
10 Motor Loss (Ave)	Kin-Lb/Sec	9.5		9.5	→	9.5	8.8	9.5	9.5	8.4	9.5	8.8	9.5	→	9.5
11 Active Time/Total Time		4/49		4/34	4/69	1/61	2.5/61.5	4/64	4/64	4/94	4/94	2/92	4/94	4/24	4/24
12 Motor Losses (Active Ave)	Kin-Lb/Sec	0.78	0	1.12	0.59	0.16	0.21	0.59	0.59	0.36	0.40	0.19	0.40	1.58	1.58
Static Condition															
13 Motor Loss	Kin-Lb/Sec	4.5												→	4.5
14 Active Time/Total Time		45/49	1/1	0			→	0	0.74	0.76	0.76	0.78	0.76	0	0
15 Motor Loss (Ave)	Kin-Lb/Sec	4.13	4.5	0			→	0	3.33	3.42	3.42	3.51	3.42	0	0
Total															
16 Motor Losses	Kin-Lb/Sec	4.91	4.5	6.63	6.45	5.67	6.06	6.45	5.17	5.03	5.07	4.95	5.07	6.79	6.79
	BTU/Sec	0.526	0.482	0.710	0.691	0.607	0.649	0.691	0.554	0.539	0.543	0.530	0.543	0.727	0.727

TABLE XXXIV

HYDRAULIC LOSSES

Flight Condition	→	1	2	3	4	5	6	7	8	9	10	11	12	13	14
Small Amp Cond															
1 Ave Hyd Flow	In^3/Sec	0	0	11.51	11.51	8.08	10.37	11.51	--					→	11.51
2 Pump Pressure	PSI			1580	1580	1600	1590	1580						→	1580
3 Hydraulic Power	Kin-Lb/Sec			18.19	18.19	12.93	16.49	18.19						→	18.19
4 Active Time/Total Time				30/34	60/64	60/61	60/61.5	60/64	1/5			→	1/5	20/24	20/24
5 Pwr Loss (Active Ave)	Kin-Lb/Sec	0	0	16.05	16.05	12.72	16.09	17.05	3.64			→	3.64	15.15	15.16
Maneuver Condition															
6 Ave Hyd Flow	In^3/Sec	23.01	0	23.01	→	23.01	20.73	23.01	23.01	19.53	23.01	20.73	23.01	→	23.01
7 Pump Pressure	PSI	1510		1510	→	1510	1520	1510	1510	1530	1510	1520	1510	→	1510
8 Hydraulic Power	Kin-Lb/Sec	34.75		34.75	→	34.75	31.51	34.75	34.75	29.88	34.75	31.51	34.75	→	34.75
9 Active Time/Total		4/49		4/34	4/64	1/61	1.5/61.5	4/64	4/64	4/49	4/49	2/92	4/49	4/24	4/24
10 Pwr Loss (Active Ave)	Kin-Lb/Sec	2.84	0	4.09	2.19	0.57	0.77	2.17	2.17	2.44	2.84	0.69	2.84	5.79	5.79
Static Condition															
11 Ave Hyd Flow	In^3/Sec	1.0												→	1.0
12 Pump Pressure	PSI	1650												→	1650
13 Hydraulic Power	Kin-Lb/Sec	1.65												→	1.65
14 Active Time/Total Time		45/49	1/1	0			→	0	0.74	0.76	0.76	0.78	0.76	0	0
15 Power Loss	Kin-Lb/Sec	1.52	1.65	0			→	0	1.22	1.25	1.25	1.29	1.25	0	0
Total															
16 Hyd Power Loss	Kin-Lb/Sec	4.36	1.65	20.14	18.24	13.29	16.86	19.22	7.03	7.33	7.73	5.62	7.73	20.95	20.95
	BTU/Sec	0.467	0.177	2.16	1.95	1.42	1.81	2.06	0.75	0.785	0.828	0.602	0.828	2.24	2.24

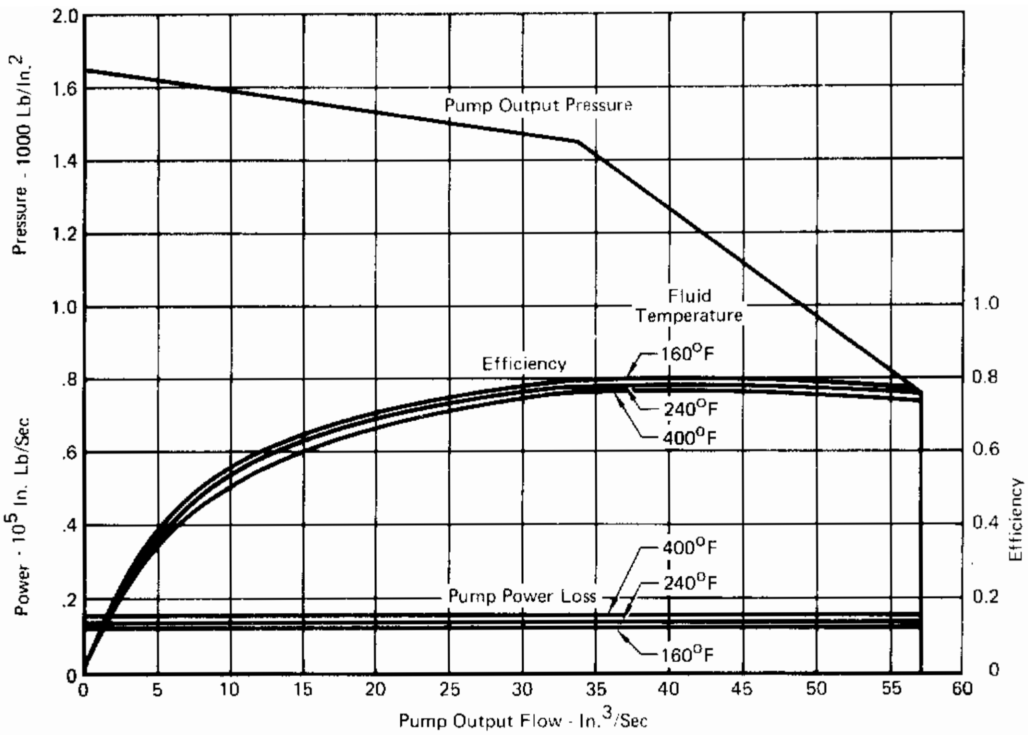


FIGURE 98
SSAP HYDRAULIC PUMP CHARACTERISTICS

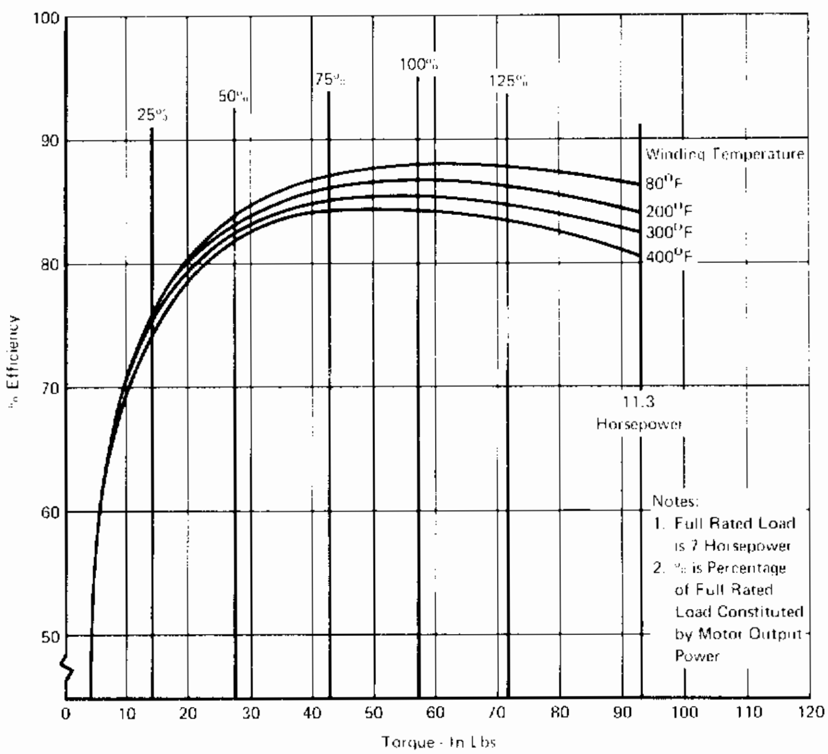


FIGURE 99
CALCULATED MOTOR EFFICIENCIES

Return fluid from the switching valves and main control valve is mixed with the case drain fluid and is circulated through the heat exchanger where its heat is rejected to a mixture of aft fuselage compartment and ram air. The air is drawn through the heat exchanger by an electric blower-motor unit.

Initially, it was planned to circulate compartment air through the heat exchanger using a blower at the inlet. The established heat exchanger requirements were as shown in Table XXXV. With the resultant heat exchanger size, the available envelope was insufficient. By routing the ram air to the vicinity of the heat exchanger, the inlet air temperature could be reduced to the average of the compartment air and ram air temperatures as the air supplies mixed. Using this approach, and drawing the air through the heat exchanger with a motor-blower unit, a significant reduction in required heat exchanger size was accomplished.

TABLE XXXV
HEAT EXCHANGER REQUIREMENTS

Flight Condition		1	2	3	4	5	6	7	8	9	10	11	12	13	14
1 Pressure Altitude	K Ft	S.L	←		→		S/L	45	50	50	70	35	[illegible]	[illegible]	
2 Air Temperature	°F			350	155	190	245	215	205	345	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
3 Oil Flow	In.³/Sec			14.8	14.1	10.2	12.5	14.1	6.4	5.8	6.0	5.5	[illegible]	[illegible]	[illegible]
4 Package Temp	°F			420	255	250	310	430	355	400	400	[illegible]	460	340	[illegible]
5 Return ΔT	°F			6	6	9	7	6	14	16	15	17	15	6	[illegible]
6 Hot Oil Temp	°F			426	261	259	317	436	369	416	415	[illegible]	475	346	376
7 Heat Flow	BTU/Sec			3.5	3.5	2.9	3.2	3.1	1.9	0.6	0.7	1.6	1.5	3.6	2.8
7a Heat Flow Min	BTU/Sec			0.35	0.45	0.25	0.4	0.5	0.6	[illegible]	0.8	0.75	0.82	[illegible]	[illegible]

Note: Condition 7a based on pump losses of 1.6 hp pump, duty cycle [illegible] no stabilator motion, and steady state operation

Table XXXV line 7 presents the design heat flow for the heat exchanger. Line 7a presents the equivalent heat load assuming a duty cycle with no stabilator motion, a pump with more optimistic pump losses, and steady state operation. At most flight conditions the value in line 7 far exceeds the value in line 7a. From this observation and a belief that the specified duty cycles and air temperatures are somewhat conservative, it is concluded that the design heat rejection from the SSAP to the ram/compartment air combination, is more than adequate during most flight conditions and is adequate at all flight conditions.

The power loss of the blower motor units is dumped into the compartment and was taken into account in the heat exchanger design. Data describing the input power required for the blower is presented in Figure 100. Estimates of heat exchanger blower motor inputs are presented in Table XXXVI. The thermal analysis of this arrangement is not yet complete.

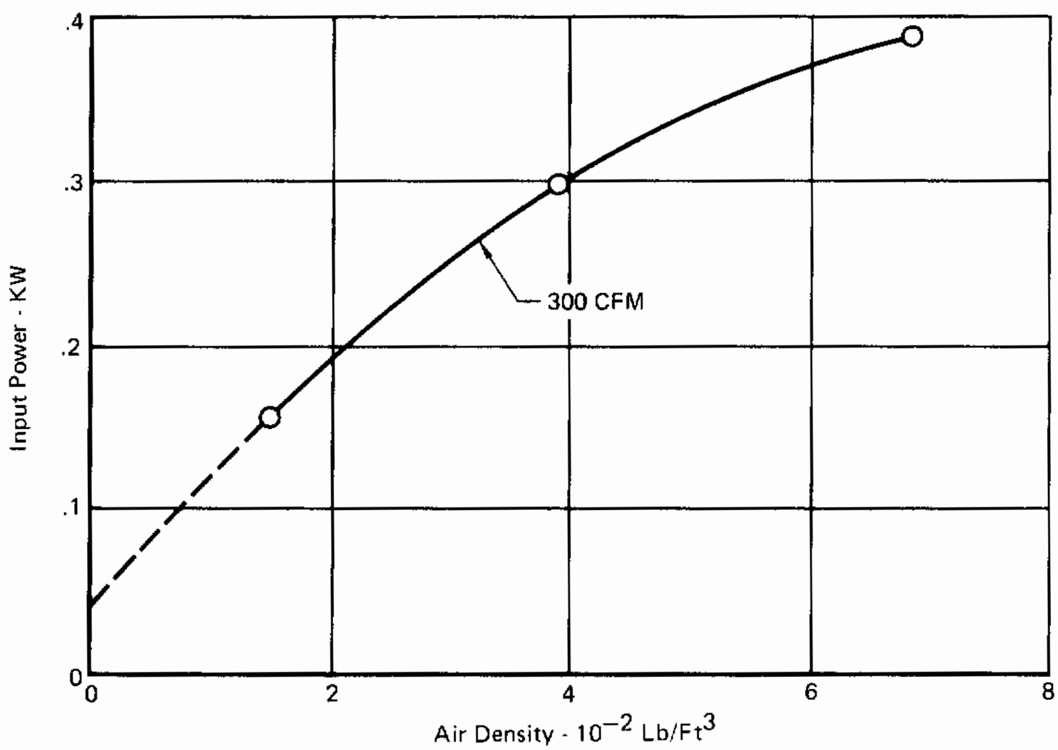


FIGURE 100

SSAP HEAT EXCHANGER BLOWER INPUT POWER

(7) Electromechanical Secondary Actuator

The heat generated by the secondary actuator under normal (unstalled) loading conditions is approximately 197.0 watts (.187 BTU/sec), very small compared to the heat generated by the remaining parts of the SSAP. Each of the four servo motors is to be cooled by an individual blower which forces compartment air across the servo motor housing. The resulting winding temperatures are expected to be well below the 425°F temperature recommended by LTV-E as the maximum allowable for short durations.

Under stall conditions of excessive duration, however, the recommended maximum winding temperature could be exceeded at the compartment temperatures expected. When stall conditions are sustained, the resulting allowable stall durations presented in

TABLE XXXVI
HEAT EXCHANGER BLOWER - MOTOR INPUT

Flight Condition		1	2	3	4	5	6	7	8	9	10	11	12	13	14
Ram Air Temperature Rise	^{o}F/BTU/Sec	N/A	5.0	50.0	13.3	7.72	5.0	32.2	38.4	16.7	51.5	12.5	28.6	29.4	50.0
Ram Air Temperature Rise to HX Inlet	^{o}F		6.9	123	32	16	11	73	59	25	76	19	43	74	125
Ram Air Temperature	^{o}F		51	128	120	160	222	35	95	318	185	250	250	135	129
Air Temp in HX	^{o}F		44	251	152	176	233	108	154	343	261	269	293	209	254
Air Flow	Lb/Min Lb/Sec			19 0.317	23 0.383	23 0.383	21 0.350	4.2 0.070	3.5 0.058	2.4 0.040	1.2 0.020	4.3 0.072	2.6 0.043	21 0.350	19.5 0.32
Heat Load	BTU/Sec			3.5	3.5	2.9	3.2	3.1	1.9	0.6	0.7	1.6	1.5	3.6	2.8
Δt_{HX}	^{o}F			46	38	32	38	184	136	63	146	93	145	43	36
Blower Air Temp	^{o}F			297	190	207	271	292	290	405	407	362	438	252	290
Pressure Altitude	K Ft	S/L	←			→	S/L	45	50	50	70	35	53	S/L	S/L
Pressure Ratio (δ)		1.0	←			→	.30	0.146	0.115	0.115	0.044	0.235	0.099	1.0	1.0
Air Density at Blower	Lb/Ft^{3}			0.053	0.061	0.060	0.054	0.0077	0.0061	0.0053	0.0020	0.0114	0.0044	0.0559	0.05

Figure 101 result. When stall loads are intermittently but repeatedly applied, the allowable duration of each period of stall is presented in Figure 102. The data presented are based on an estimate by the manufacturer that the servo motor thermal time constant is 15 minutes.

The capability of the secondary actuator servo motors to withstand sustained or repeated stalls is estimated to be beyond any in-flight requirement that can reasonably be foreseen at this point.

4. CONCLUSIONS

Based upon the studies discussed above, the following conclusions may be drawn.

a. The SFCS T/Rs and batteries are not expected to exhibit unacceptable performance or reliability degradation because of their thermal environment.

b. The fourth hydraulic system is expected to operate at acceptable temperature levels for the SFCS program.

c. The temperatures predicted to be encountered by the SFCES are within the specified operating temperature range of the equipment and are not expected to have an excessively adverse effect on the operation, safety, or reliability of the SFCS.

d. The heat rejection from the SSAP is not predicted to produce equipment or airframe thermal problems during the SFCS test program.

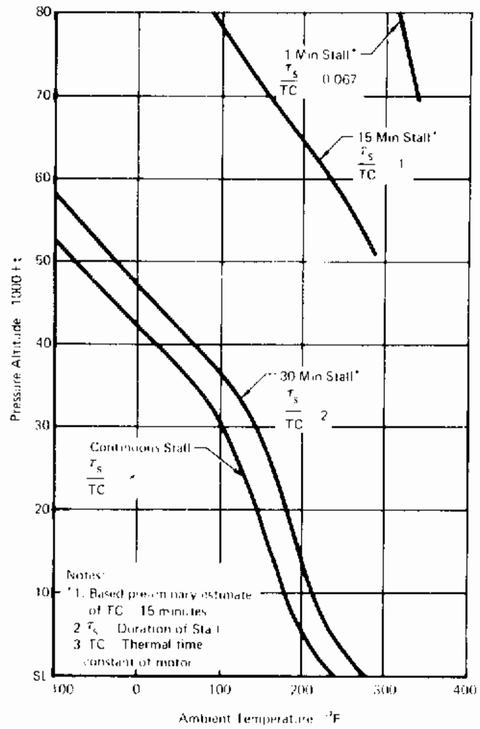


FIGURE 101

ALLOWABLE SUSTAINED FULL STALL DURATIONS FOR ELECTROMECHANICAL SECONDARY ACTUATOR

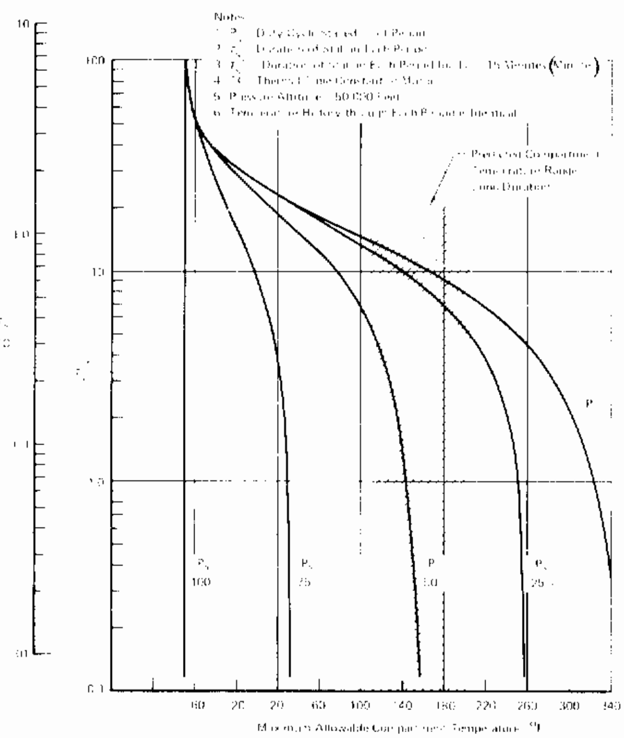


FIGURE 102

ALLOWABLE REPEATED STALL DURATIONS FOR ELECTROMECHANICAL SECONDARY ACTUATOR

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX II

ABBREVIATIONS:

alt - Altitude

amp - Amplitude

amps - Amperes

ave - Average

BTU - British Thermal Units

CFM - Cubic Feet per Minute

Cont. - Continuous

ECS - Environmental Control System

HX - Heat Exchanger

ICAO - International Civil Aviation Organization

K in-lb/sec - Power in thousand inch pounds per second

KW - Kilowatts

Kw - Kilowatts of Heat Dissipated by SSAP into Aft Fuselage Compartment

Max - Maximum

Min. - Minutes, minimum

N/A - Not Applicable

No. - Number

Ps - Duty Cycle Stalled, Percent of Period

Pwr - Power

Qhx - Heat Exchanger Heat Flow

Recirc ΔT - Amount Oil Temperature into Heat Exchanger Exceeds Package Temperature

Reqd. - Required

Taw - Temperature of Adiabatic Wall at Aircraft Boundary

TC - Thermal Time Constant

Tc - Temperature of Compartment

Tp - Temperature of the SSAP Package

Tr - Temperature of Ram Air

Vel. - Velocity

Wr or Wram - Ram Air Mass Flow Rate

SYMBOLS:

δ	Ratio of air pressure to standard atmospheric pressure at sea level
τs	Duration of stall in each period
ΔT	Increment of temperature
ΔT_{HX}	Air temperature rise across heat exchanger
°F	Degrees Fahrenheit, temperature
°C	Degrees Centigrade, temperature

APPENDIX III

ELECTRICAL SYSTEM DESIGN STUDIES

1. INTRODUCTION AND SUMMARY

Three separate study and analysis tasks were performed as a means of selecting the most reliable electrical system for the SFCS test aircraft. The studies include investigations of and conclusions pertinent to:

a. Electrical Power Generation, Distribution, and Load Analysis

This study provides a comparison of electrical generation and distribution systems. It develops the rationale leading to selection of a modified split bus system for the SFCS test aircraft. Finally an analysis is made of the electrical loads imposed on the aircraft Alternating Current (AC) generating system.

b. Wiring Techniques

Investigations and evaluations were made which provide the rationale leading to the particular wiring methods selected for the SFCS test aircraft.

c. Wiring Dispersion

This study pertains to the particular problems of providing adequate dispersion of the SFCS test aircraft electrical wiring. Using the intended dispersion criteria, an investigation was made of areas in the aircraft where special attention is required due to existing routing path limitations, and the results define the planned physical and electrical protective measures to be used to provide safe and reliable operation of the SFCS.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. ELECTRICAL POWER GENERATION, DISTRIBUTION, AND LOAD ANALYSIS

Safety, reliability, and survivability of the entire aircraft electrical system are essential considerations in the studies leading to the development of the SFCS electrical power systems. Candidate electrical power systems including the power sources, distribution, and utilization aspects have been evaluated. Design modifications and additions to the test aircraft electrical power distribution system were investigated. Variations of the basic bus systems were also evaluated. The results of this study provide the rationale leading to the test aircraft electrical power distribution system design. In addition, an analysis has been performed to show that the electrical loads expected to be imposed by the added SFCS equipment do not exceed the power available under any normal or emergency operating condition.

a. AC Power

Two basic types of aircraft AC electric generating and distribution systems are in general use: parallel bus systems and split bus systems. There are many variations of either of the basic systems. These variations generally consist of different bus switching schemes. Bus switching may be fully automatic or manually controlled. Usually a combination of automatic and manual control is used. The particular arrangement depends on many factors such as the degree of control and switching flexibility desired, safety and reliability considerations, and the intended use of the aircraft.

(1) Parallel Bus System

A parallel bus system consists of two or more AC generators connected to a common load bus system, with suitable switchgear and circuit protective devices. Each generator must be operating at nearly the same frequency and in phase with the other generators feeding the bus when it is connected to the common bus. Important characteristics of a parallel bus system are summarized as follows:

- o All generators are synchronized in frequency and phase; therefore, there will be no "beat" frequencies which might be sources of low frequency Electromagnetic Interference (EMI).

- o Equipment faults (shorts) are "cleared" faster when compared with a split bus system, because circuit breakers will see at least twice the fault current capacity of a single generator.

- o Bus and/or feeder-fault currents are high since the fault current capacity is at least twice that of a single generator. The heat developed at the fault would be at least 4 times as great as that developed with a split bus system.

- o System is sensitive to load surges. All generators and their control subsystems will be influenced by heavy inrush loads.

- o Full capacity of the complete system cannot be used, since the system must be derated 10% to allow for paralleling losses.

- o System control is more complex than with a split bus, and requires the addition of a frequency and load controller.

(2) Split Bus System

A split bus system consists of two or more AC generators, each connected to a separate load bus. The number of main load buses is no less than the number of AC generators installed. Switchgear and circuit protective devices are used to permit feeding any load bus from one of the operating generators if the generator that normally feeds that bus becomes inoperative. Important characteristics of a split bus system are summarized as follows:

- Full capacity of system is usable.
- Simpler system control with fewer components, since auto-paralleling and real and reactive load division control circuits are not required.
- Only one generator "feels" the inrush surges imposed by any single load.
- Bus and feeder fault current capacity is limited to the fault capacity of only one generator.
- Fault currents do not immediately affect the other generator(s).
- Loads must be assigned to each main bus carefully to assure that they do not exceed the rating of the corresponding generator.
- No synchronization is required in a split bus system; therefore, induced beat frequencies can exist.

(3) SFCS Split Bus System

The decision to use a split bus electric generating and distribution system for the SFCS test aircraft is based primarily on improved safety and reliability.

- Faults are isolated to the specific bus affected. Likewise each Survivable Stabilator Actuator Package (SSAP) motor pump load is isolated to one bus.
- The split bus system prevents the loss of both generators due to a fault on only one bus.
- The split bus system is more reliable. An analysis of MCAIR records show that split bus systems have an overall Mean Time Between Failure (MTBF) of 220 hours compared to a MTBF of 170 hours for parallel bus systems.

The split bus system to be used in the SFCS test aircraft consists of two 30 Kilovolt-Amperes (KVA) oil cooled AC generators, each connected to a separate main load bus. The two SSAP motor driven hydraulic pumps for the Phase IIC program are to be individually powered from these sources; one from the generator driven by the left engine and the other from the generator driven by the right engine. The motor driven pumps are not switched from one generator to the other during single-generator operation because one generator cannot supply enough power for both motor pump units and the remainder of the aircraft loads. The SFCS test aircraft electrical system is designed to provide four redundant sources of Direct Current (DC) power for the SFCS flight control circuits. Power to drive the 4th hydraulic system is also derived from the AC generators.

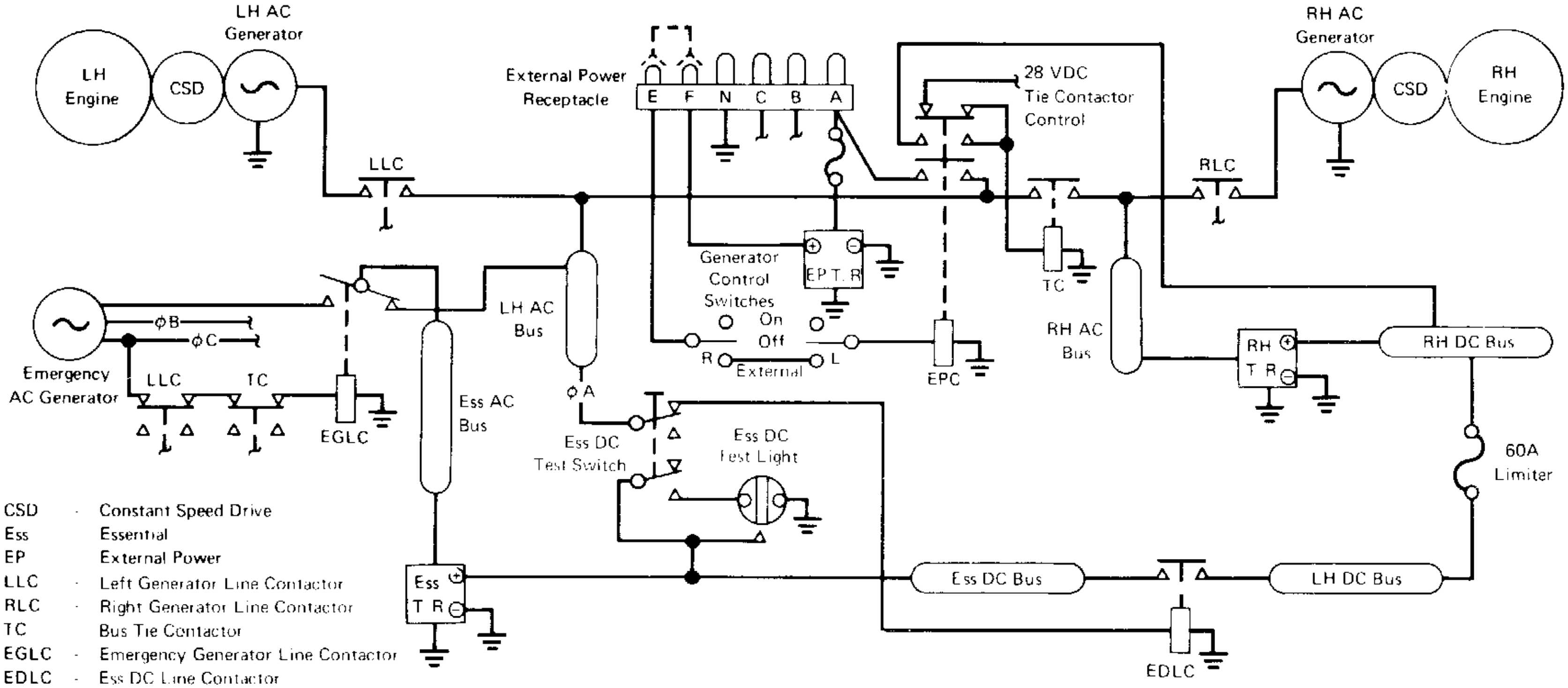
(a) Variations of the basic split bus system were studied. Figure 103 represents the F-4J split bus system, which was used as the baseline from which the SFCS system will evolve. Circuit design changes are to be made in this system to provide improved system safety and reliability, and to prevent overloads during single-generator operation. Other changes planned to be incorporated in the AC electrical system are associated with the addition of the motor driven hydraulic pumps for the SSAP and the 4th hydraulic system.

(b) The initial SFCS concept included a Hydraulic Driven Electric Generator (HDEG) to supply electric power to operate one channel of the SFCS and to drive one motor pump unit on the SSAP. The HDEG was also intended to provide power for a limited time during windmilling conditions following engine flameout. Further development of the SFCS concept eliminated the HDEG from the SFCS. A discussion of the studies related to the HDEG is provided in Supplement 3 to this report.

(c) The potential problems associated with the beat frequencies that may exist in a split bus electrical system have been given careful consideration in the evaluations and studies leading to the decision to use a split bus system for the SFCS test aircraft. This beat frequency condition occurs due to a variation between the frequencies of the AC produced by the two generators. Each generator produces 3Ø power at between 380 Hz and 420 Hz. They may, at any particular time, have a difference frequency of from zero up to a maximum of 40 Hz. The EMI effects of these beat frequencies depends on factors such as shielding, wire routing, proximity of sensitive equipment to power wires, etc. These effects tend to produce low amplitude, low frequency fluctuations in the susceptible circuits. Such minor perturbations have been encountered in previous split bus F-4's and successfully eliminated or reduced to within acceptable limits.

b. Aircraft DC Power

The SFCS test aircraft's basic DC power system is expected to be the same as that of the F-4J except for revisions to the Essential DC, Essential DC test, and battery circuits. All F-4 aircraft have a means of checking for the presence of DC on the Essential DC Bus. A typical F-4 Essential DC test circuit consisting of an Essential DC test switch and indicator light, is shown in Figure 103. An



CSD	-	Constant Speed Drive
Ess		Essential
EP		External Power
LLC	-	Left Generator Line Contactor
RLC	-	Right Generator Line Contactor
TC		Bus Tie Contactor
EGLC	-	Emergency Generator Line Contactor
EDLC	-	Ess DC Line Contactor
EPC	-	External Power Contactor
T/R		Transformer Rectifier
φ	-	Phase

FIGURE 103

F-4J SPLIT BUS ELECTRICAL SYSTEM

examination of this circuit reveals that if the Essential T/R is inoperative, Essential DC power will be lost when the test switch is depressed. Circuit design changes expected to be made in the DC portion of the SFCS test aircraft electrical system, as shown in Figure 104, include:

- The "Essential DC test" circuit is to be redesignated "Essential T/R test". This circuit is redesigned to test for the presence of DC at the output of the Essential T/R without isolating the Essential DC Bus from the LH and RH DC Buses.

- A means of connecting the aircraft battery to the Essential DC Bus, in case the Essential T/R becomes inoperative when the emergency generator is supplying power to the system, is being incorporated.

c. SFCS Electrical Power

The presently planned SFCS concept requires both AC and DC electrical power. Since the final system design of the SFCS must meet the Phase IIC requirements, the electrical power system design must be developed to meet the needs of that configuration. The requirement for AC power for the motor pump units of the SSAP imposes the more severe load condition between the Phase IIA and IIC configurations. Phase IIB SFCS changes have no significant effect on the power system design.

Figure 105 illustrates the Phase II SFCS Split Bus Electrical System design.

(1) SFCS DC Power System

The SFCS requires four independent sources of DC power. The original concept included a 28 VDC source of power which was derived from an HDEG. Two other DC sources were to be derived from transformer-rectifiers (T/R) individually powered from the aircraft's main AC buses. The fourth source was to be supplied by an added battery. The originally proposed DC power source configuration, when subjected to closer scrutiny, had several shortcomings which rendered the overall concept unattractive in the areas of redundancy and reliability. Some of the problems associated with the originally proposed configuration would be:

- The T/Rs, which would be powered from the main AC buses, would exhibit fluctuations in output voltage in direct proportion to any disturbances (surges, outages, etc.) present on the main AC buses. A primary cause of these disturbances is the switching and control functions of the main AC generating systems.

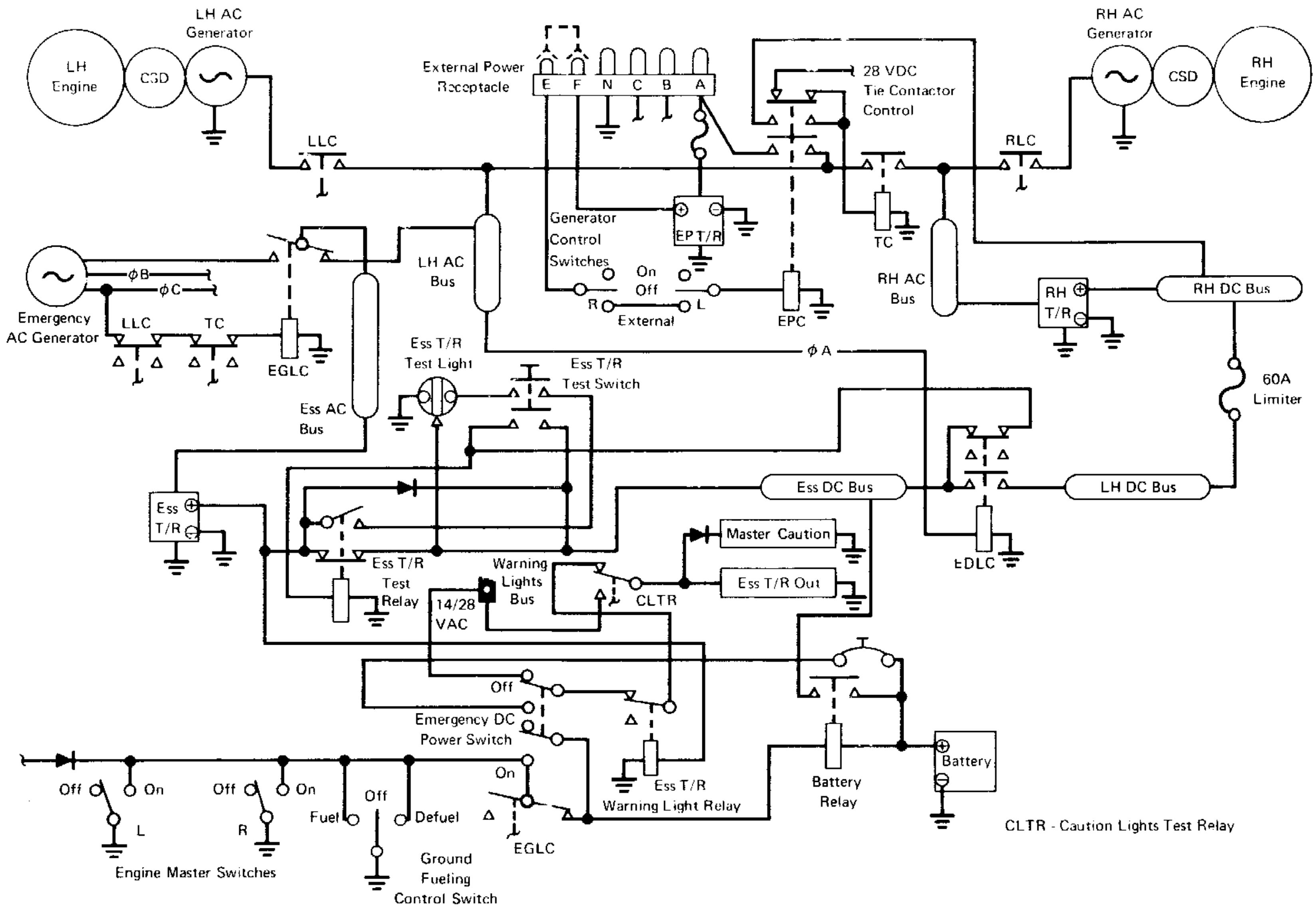
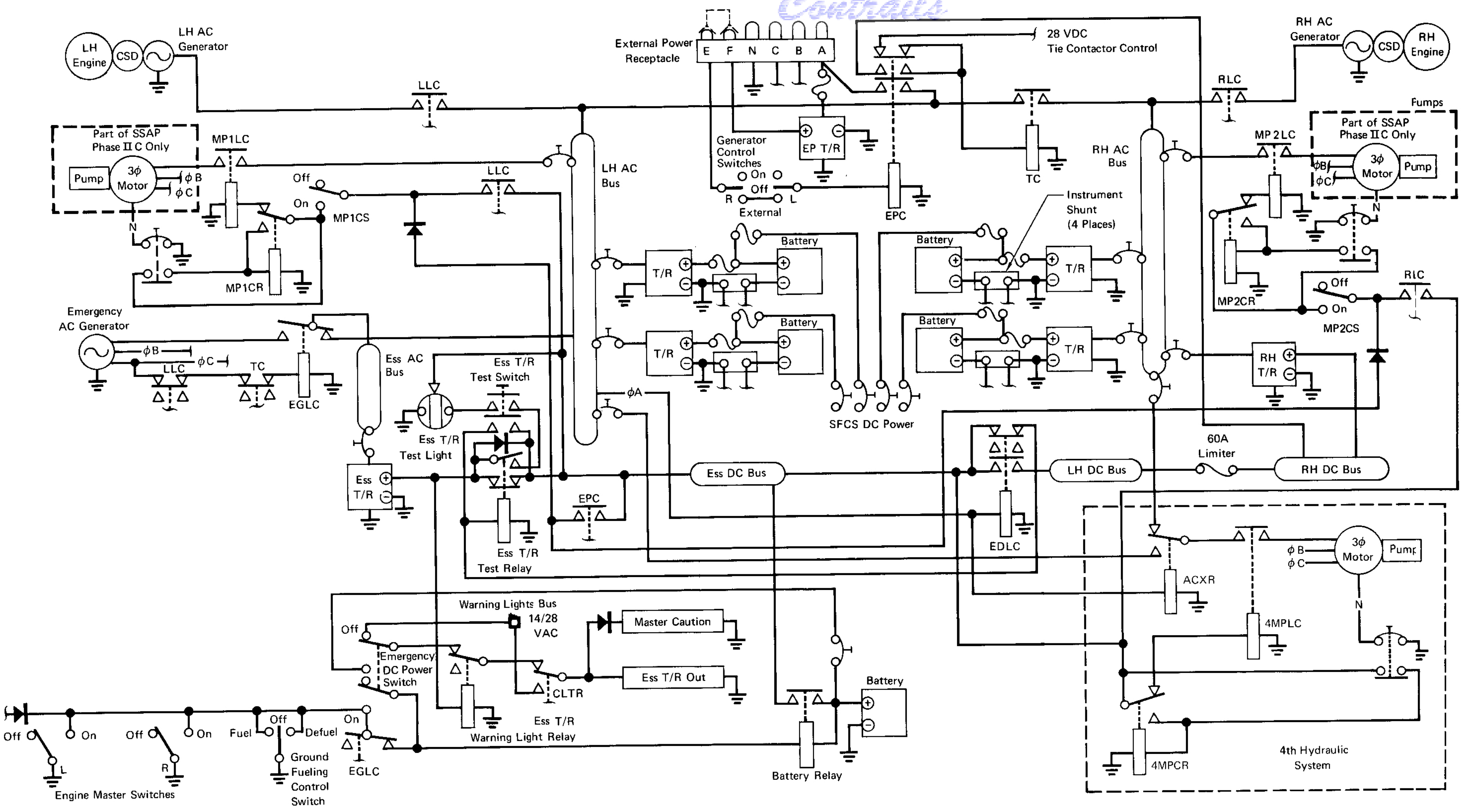


FIGURE 104
SPLIT BUS ELECTRICAL SYSTEM
(WITH IMPROVED DC TEST AND BATTERY CIRCUIT)



ACXR – AC Source Transfer Relay
CLTR – Caution Light Test Relay
CSD – Constant Speed Drive
EDLC – Essential DC Line Contactor
EGLC – Emergency Generator Line Contactor
EPC – External Power Contactor
EP – External Power
Ess – Essential
LLC – Left Generator Line Contactor
MP1CR – Motor Pump 1 Two-Phase Cutout Relay
MP1CS – Motor Pump 1 Control Switch
MP1LC – Motor Pump 1 Line Contactor
MP2CR – Motor Pump 2 Two-Phase Cutout Relay
MP2CS – Motor Pump 2 Control Switch
MP2LC – Motor Pump 2 Line Contactor
RLC – Right Generator Line Contactor
SSAP – Survivable Stabilator Actuator Package
TC – Bus Tie Contactor
T/R – Transformer Rectifier
4 MPLC – 4th Hyd. System Motor Pump Line Contactor
4 MPCR – 4th Hyd. System Two-Phase Cutout Relay
ϕ - Phase
N – Neutral

FIGURE 105
SFCS SPLIT BUS ELECTRICAL SYSTEM - PHASE II

- It would be difficult to determine the proper source from which to charge the added battery; i.e., various schemes would be available involving the present AC and DC buses.

- A failure of the HDEG would cause a failure of the SFCES channel using the DC power derived from it.

Several other DC power configurations were analyzed with the intent of finding one suitable for the purpose without unduly compromising the overall electrical system. Any of the schemes which involved one battery charged from multiple sources became so complex with monitoring and switching components that any gains in reliability were far outweighed by complexity. This also applies to multiple T/Rs whose inputs were transferred from one main AC bus to another.

A means of providing four redundant, highly reliable sources of DC power for the SFCS has evolved from the studies. The planned SFCS Phase II electrical system, as represented by Figure 105, is to include four T/Rs, which normally supply the SFCS DC power, and four batteries to provide back-up power. Each of the four added batteries is kept charged by one of the T/Rs. If any T/R should fail due to either an internal fault or a loss of input power, the battery connected to it is expected to supply adequate power to the SFCS for a sufficient length of time to enable the pilot to complete the mission. The battery also acts as a transient suppressor. Transients produced at the load terminals of the T/R due to DC load changes or T/R input transients are reduced by the capacitive filtering action of the battery.

The satisfactory functioning of the SFCS for a reasonable length of time after a failure of a channel primary T/R power source is important to successful demonstration of the SFCS concept. A reasonable length of time for operation of this test aircraft was assumed to be about 0.75 hours after failure of an electrical generator. The added batteries will be 22 ampere-hour nickel cadmium type. The battery size was selected to provide approximately one hour of operation of the SFCES and the SFCS secondary actuators, plus the SSAP electromechanical secondary actuator in Phase IIC. This time was calculated by assuming the power required at the input to each SFCS channel as 400 watts at any input voltage between 20.0 and 28.3 VDC. The current required to provide 400 watts of power at the lowest specified voltage of 20.0 volts is equal to:

$$\frac{400 \text{ watts}}{20 \text{ volts}} = 20.0 \text{ Amperes}$$

The total average voltage drop in the SFCS DC power supply circuits is computed to be 0.82 volts. This voltage drop must be added to the 20.0 volt minimum specified SFCES input voltage to arrive at the minimum usable battery voltage, i.e.:

20.0 + 0.82 = 20.82 Volts

Figure 106 illustrates typical discharge characteristics of a MS24497-5 battery. Note that the capacity rating in Ampere-hours has been derated to 90% of the manufacturer's rating to provide a safety factor. Using this curve, the minimum usable battery voltage of 20.82 volts intersects the 20 Ampere discharge current line at 22.5 ampere-hours. Therefore:

$$\frac{22.5 \text{ Ampere-hours}}{20.0 \text{ Amperes}} = 1.125 \text{ hours}$$

This figure is conservative since the battery will begin operating at a voltage higher than 20.82, drawing less current, and extending its operating time accordingly.

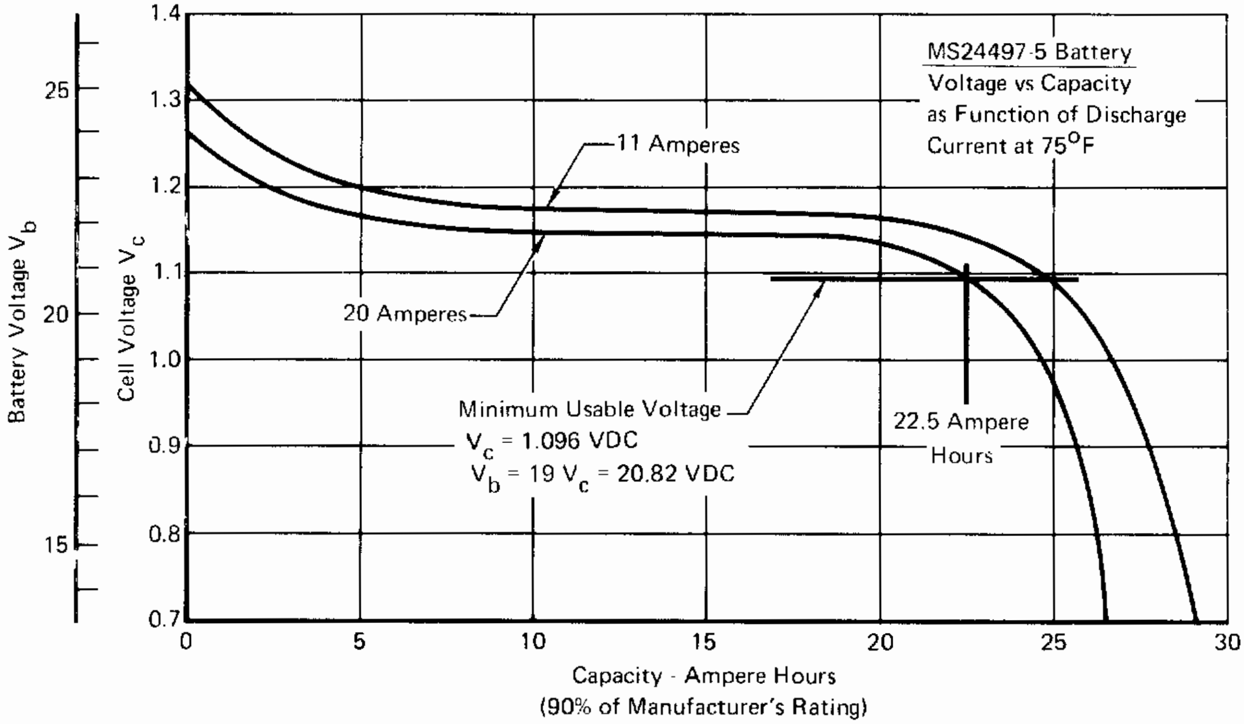


FIGURE 106
BATTERY DISCHARGE CHARACTERISTICS

(2) 4th Hydraulic System Electrical Power

Figure 105 also shows the 4th Hydraulic System motor pump and related control and power circuitry. The 4th Hydraulic System motor pump normally receives its power from the LH AC Bus. Should the LH AC Bus become de-energized for any reason, the AC Source Transfer Relay will drop out and transfer the 4th Hydraulic System motor pump load to the RH AC Bus. The system design also provides two-phase protection circuitry to prevent excessive winding current. The neutral current of a 3-phase, 4-wire, Y-connected AC motor will normally not exceed about 10% of the line current. However, should one of the three motor windings open or become disconnected from the power source, the line current in the remaining two windings and the neutral current may be expected to increase to as much as 170% of the line current present in each winding prior to the start of the two-phase operation. If a failure should occur, this increased neutral current will be used to trip a circuit breaker. The circuit breaker selected for this application has one set of auxiliary contacts which close when the main contacts open. These auxiliary contacts will be used to energize and lockup a relay. This action is planned to interrupt the control circuit to the motor pump line contactor which, in turn, will shut down the motor.

(3) SSAP Motor Pump AC Power

The planned Phase IIC SFCS Split Bus Electrical System design is essentially the same as that planned for Phase IIA except for the addition of the SSAP motor pumps and their associated power and control circuitry.

Each SSAP motor pump will be connected to one main AC bus and energized only when: (1) the generator that normally feeds that bus is operating, (2) the corresponding motor pump control switch is "on", and (3) when essential DC is available. This arrangement is required to prevent overloading of the remaining generator during single-generator operation.

The SSAP pump motors are also protected against two-phase operation in a manner similar to the 4th Hydraulic System pump motor. Should single generator operation occur, an aircraft central hydraulic power system will be switched into the portion of the SSAP normally powered by the inoperative motor pump.

d. Power Distribution

An aircraft electrical power distribution system must be able to deliver the current required under all reasonable operating conditions at a voltage within specified limits for satisfactory electrical equipment operation. The SFCS power distribution system design is expected to adhere to the above, which also is the established practice of current F-4 production systems.

(1) Location of Main AC Switchgear

Figure 105 represents the SFCS Split Bus Electrical System. The main AC switchgear components and their basic functions are:

- Left Generator Line Contactor - when closed, connects the LH AC generator to the LH AC Bus.
- Right Generator Line Contactor - when closed, connects the RH AC generator to the RH AC Bus.
- Bus Tie Contactor - when closed, connects the LH AC Bus to the RH AC Bus.
- External Power Contactor - when closed, connects external power directly to the LH AC Bus. Whenever the External Power Contactor is closed, the Bus Tie Contactor is also closed.

All of the interconnecting power wiring between these components and any branch circuits constitute the main AC Buses of the system.

The overcurrent protection of this circuitry for the SFCS test aircraft is planned to be accomplished by sensing the occurrence of a fault and clearing of the fault after a short period of time by interrupting the source of power which is supplying the fault current. The Voltage Regulator/Supervisory Panels (VR/SP) contain undervoltage and overvoltage sensing circuits. When a bus fault is detected, this unit disconnects the generator from its main bus, thus interrupting the fault current source. This method of fault clearing has successfully been employed in all F-4 aircraft.

The latest configuration of the AC Power Control Panel used in the F-4J contains fast operating contactors, and is qualified to more stringent environmental requirements than an earlier panel. This latest AC Power Control Panel is to be used as the central bus switching unit for the SFCS test aircraft. It will replace the early configuration panel that is presently installed in the test aircraft.

(2) Main Load Buses

The main AC load buses consist of the various bus bars on the line side of the load circuit breakers in the aircraft Circuit Breaker Panels.

(3) Auxiliary Load Buses

Auxiliary load buses will be contained within a circuit breaker panel which will be added to the forward cockpit. The circuit breakers located within this panel will be fed from the main load bus through circuit breakers used to protect the distribution wiring between these two panels. This configuration is expected to provide an SFCS Phase II electrical power distribution system design which will meet the program objective.

e. Load Analysis

An electrical load analysis has been made to determine the adequancy of the two 30 KVA AC generators and to confirm that the electrical loads are properly apportioned to the various AC and DC buses. Proper load distribution is that which results in nearly equal loading of the generators.

The SSAP motor pump starting inrush current requirement represents the most critical single loading condition applied to either of the aircraft generators. The starting and running current limitations of these motor pump units imposed on the SSAP Supplier are as follows:

- o The maximum starting peak line current of each motor pump unit shall not exceed 150 amperes RMS with rated voltage and frequency applied to the motor terminals.
- o The RMS starting line current shall decay to less than 100 amperes within 2.0 seconds after initial power application.
- o The RMS full load (nameplate rating) line current shall not exceed 30 amperes with rated voltage and frequency applied to the motor input terminals.

These limitations were augmented with data derived in tests performed on other motors used in similar applications to develop the "estimated SSAP motor pump starting KVA versus time" characteristics shown as Curve 1 on Figure 107. Curve 1 was integrated over the first 5 second period to arrive at the 5 second average load of 27.1 KVA shown as Curve 2. The normal maximum steady state single generator load without an SSAP pump motor load is calculated to be 14.70 KVA.

The highest total 5 second average load imposed on either of the aircraft AC generators is 41.8 KVA which is well within the 60 KVA, 5 second rated generator capacity. This maximum load condition could be imposed on the LH generator only if the SSAP motor pump powered from the LH generator were started during the first 5 seconds of the "takeoff and climb" operating period. The LH generator loads are higher than the RH generator loads because the 4th Hydraulic System is normally powered from the LH generator.

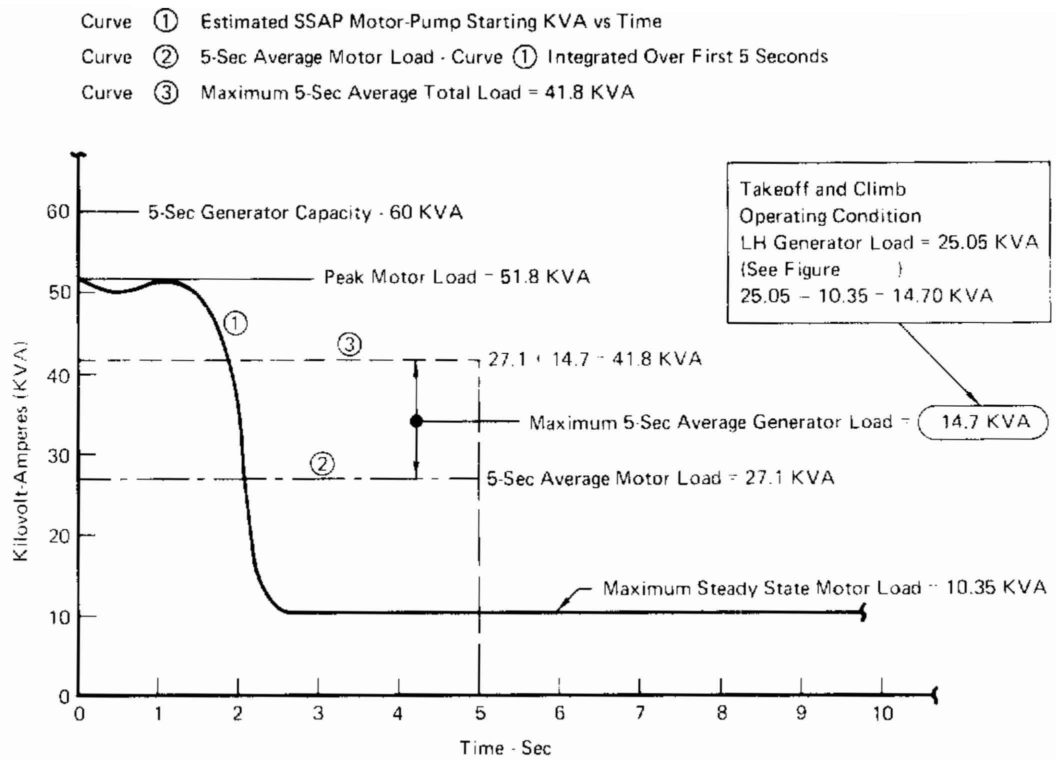


FIGURE 107
SSAP MOTOR-PUMP STARTING LOADS

The 41.8 KVA value was derived as follows:

Max. 5 second average load - LH Generator-----------	25.05 KVA
Subtract steady-state motor load------------------	-10.35
LH generator load before motor start---------------	14.70 KVA
Add 5 second average motor start load-------------	+27.10
Total 5 second average LH generator load-----------	41.80 KVA

Figure 108 shows the LH and RH generator capacities and anticipated electrical loads vs. time. It also shows the ram-air-turbine powered emergency generator capacity and anticipated loads during emergency operating conditions.

Figure 109 shows the single-generator in-flight operating loads vs. time. This figure also indicates the proper action to be taken by the pilot should one generator become inoperative. Table XXXVII shows the electrical load analysis summary.

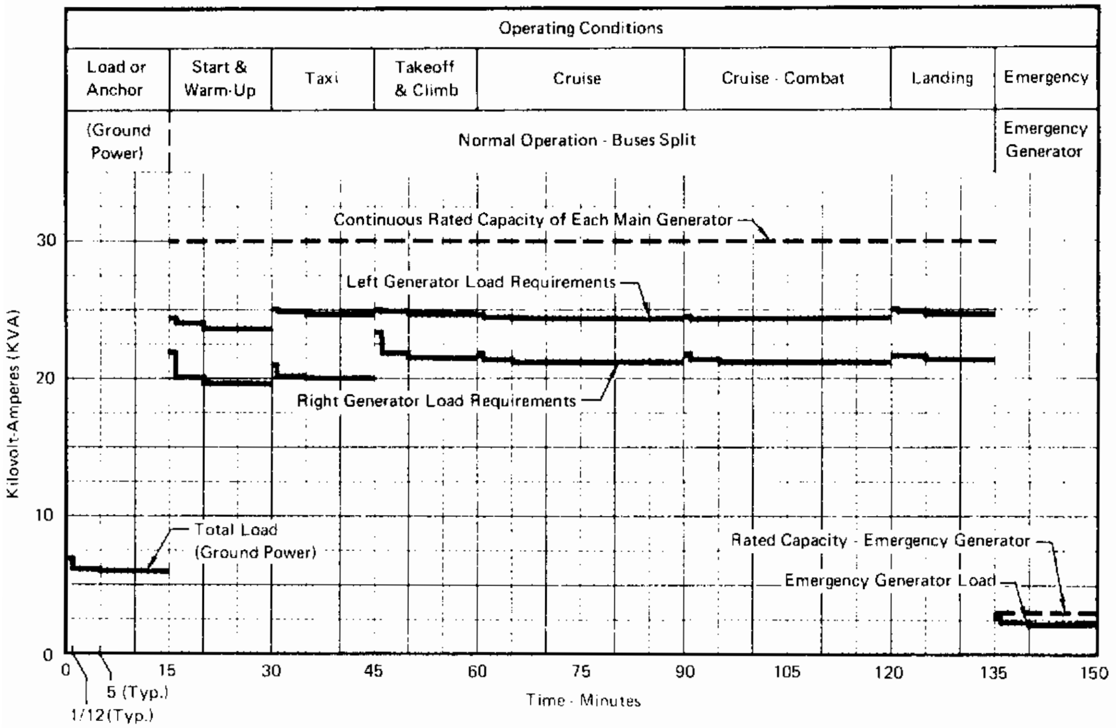


FIGURE 108

ELECTRICAL LOADS vs TIME (NORMAL AND EMERGENCY)

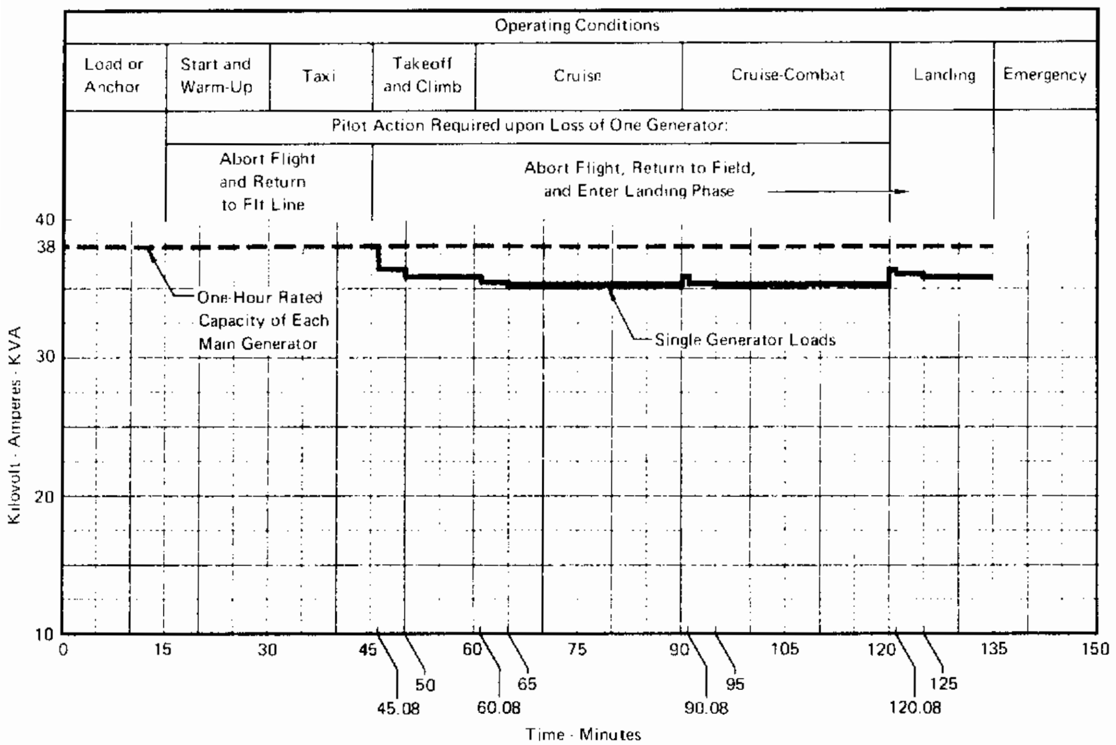


FIGURE 109

ELECTRICAL LOADS vs TIME - SINGLE GENERATOR OPERATION (BUSES TIED)

TABLE XXXVII
ELECTRICAL LOAD ANALYSIS SUMMARY

Power Source	Operating Conditions (2)											
	Load or Anchor (Grd Pwr Only)			Start and Warm-Up			Taxi			Takeoff and Climb		
	5 Sec Avg	5 Min Avg	15 Min Avg	5 Sec Avg	5 Min Avg	15 Min Avg	5 Sec Avg	5 Min Avg	15 Min Avg	5 Sec Avg	5 Min Avg	15 Min Avg
Ground Pwr	6.81	6.07	6.05									
LH Generator				24.43	24.03	23.57	24.99	24.88	24.68	25.05	24.91	24.74
RH Generator				21.93	20.17	19.62	20.97	20.18	19.98	23.28	21.80	21.54
Emer Generator												
Single-Generator Operation (1)										38.03	36.41	35.98
Total	6.81	6.07	6.05	46.36	44.20	43.19	45.36	45.06	44.66	48.33	46.71	46.28

Power Source	Operating Conditions (2)											
	Cruise			Cruise-Combat			Landing			Emergency		
	5 Sec Avg	5 Min Avg	30 Min Avg	5 Sec Avg	5 Min Avg	30 Min Avg	5 Sec Avg	5 Min Avg	15 Min Avg	5 Sec Avg	5 Min Avg	15 Min Avg
Ground Pwr												
LH Generator	24.61	24.50	24.44	24.51	24.43	24.41	24.96	24.88	24.68			
RH Generator	21.71	21.31	21.20	21.71	21.34	21.18	21.62	21.62	21.43			
Emer Generator										2.73	2.29	2.24
Single-Generator Operation (1)	36.00	35.49	35.33	35.90	35.45	35.27	36.27	36.19	35.80			
Total	46.32	45.81	45.64	46.22	45.77	45.59	46.58	46.50	46.11	2.73	2.29	2.24

All Entries are Kilovolt-Amperes (KVA)

(1) In-Flight Single-Generator Operating Conditions only (Not Included in Total)

(2) In accordance with MIL-E-7016D

3. WIRING TECHNIQUES

Components and installation techniques of various wiring systems have been considered. In order to provide an electrical installation meeting the stated program objectives, the SFCS will be wired with MCAIR developed COMPACT wire bundles.

Early investigations considered flat cable and Hard Harness as well as conventional open wiring and COMPACT wiring. These investigations indicated that flat cable and Hard Harness are not physically compatible with the installation requirements of rewiring an existing aircraft. In addition, available qualified termination systems for flat cable are limited. "Hard Harness" is the wiring system developed and used by LTV-Aerospace, on the F-8 and the A-7 aircraft. Hard Harness is rigid encapsulated wire bundles requiring tooling to fabricate and special structural consideration when designing the airframe.

Consideration of conventional open wiring reiterates MCAIR's past experience with the following disadvantages:

- Open wiring requires an abrasion resistant outer jacket over the insulation of each individual conductor, resulting in a large overall bundle diameter.
- Wires are easily damaged by chafing, by oils and fumes, and from rocks, ice, water, mud, and other environmental effects.
- Routing is difficult because of the larger wire bundles, large bend radius requirements, and increased bundle stiffness.
- The stiffness of the jacketed wire increases the possibility of wire damage at wire terminations due to the lack of flexibility when removing and installing equipment and from aircraft vibration.
- Comparative service records indicate the superiority of the F-4 COMPACT wire in both reliability and maintainability.

The COMPACT wiring system uses high temperature Teflon insulated wire assembled into bundles which are then overbraided with a Dacron jacket. This creates a flexible multi-conductor jacketed bundle which will provide maximum mechanical protection for the SFCS wiring throughout the test aircraft. This type of wire bundle has the following advantages:

- Teflon insulated wire is used, resulting in a small overall diameter. The assembly of this wire develops a small wire bundle requiring less routing space.
- The wire bundles are covered with Dacron braid and impregnated with Kel-F, providing protection from chafing, oils and fumes, from rocks, ice, water, mud, and other environmental conditions.
- Metallic shielded bundles may be fabricated in the same manner as standard COMPACT bundles by adding metallic braid to the standard construction.

- Routing is simplified because of the smaller wire bundles, smaller bend radii, and increased flexibility.

- Increased bundle flexibility provides greater reliability at wire terminations when removing and installing equipment.

Figure 110 shows a typical COMPACT wire bundle.

The service record of the braided COMPACT wire bundles used on the F-4 has been outstanding with long Mean Time Between Maintenance Action (MTBMA) and low Maintenance Manhours per Flight Hour (MMH/FH). Data from the Naval Maintenance Material Management System (3M), the USAF Maintenance Management System (66-1), and the Air Transport Association Data System (ATA-100) provide a comparison of the F-4 COMPACT wiring versus open laced wiring which substantiates this record. Table XXXVIII summarizes the above data.

COMPACT wire has been analyzed to determine if any specialized components or procedures are required. Special consideration has been given to the following:

- Channeling to avoid maintenance damage

- Fire Protection

- Overheat Protection

- Contraction and Expansion of Components

a. Channeling to Avoid Maintenance Damage

The braid on the COMPACT wire bundles provides good mechanical protection. In addition, a study of the planned wire bundle routing has been made to provide maximum protection of the bundles from being stepped on or damaged by tools or service equipment. Metallic or fiberglass protective channel or conduit will be installed over the bundles in specific areas where the wire bundles are exposed to potential damage.

b. Fire Protection

A study of the SFCS electrical system has been made to determine the wiring and protective devices required to safeguard against overloads, shorts, and mechanical breakdowns which may cause an aircraft fire. In particular:

- Wire of adequate size will be selected to carry the electrical loads and to prevent overheating and insulation breakdown from electrical overloads. Typical SFCES signal wires will be #22 gage. Typical DC power wiring to each CVU will be three #10 gage wires in parallel.

- The circuit breakers will be selected with an ampere rating below the current carrying rating of the associated wiring. An enclosed

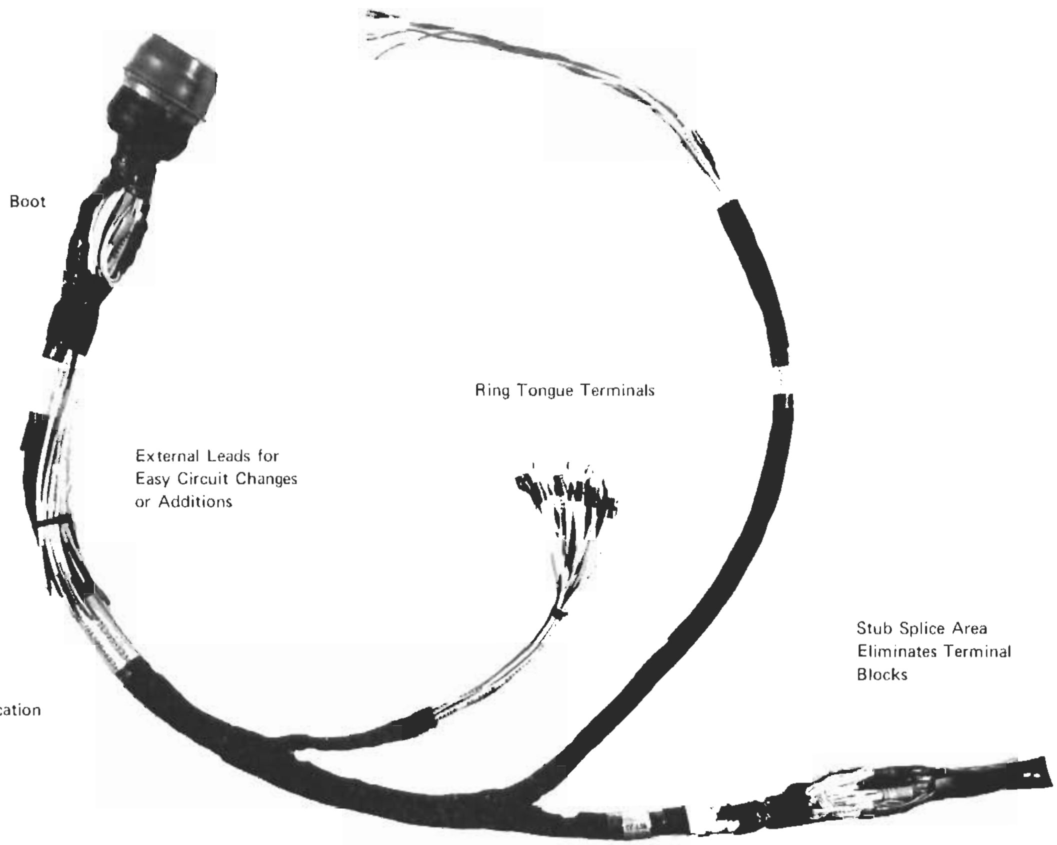


FIGURE 110
TYPICAL COMPACT WIRE BUNDLE

TABLE XXXVIII
COMPACT WIRE SERVICE RECORD

Type of Aircraft	Type of Wiring	MTBMA ①	MMH/FH ②	Flight Hours ③
F/RF-4B/J	MCAIR Compact	813.0	0.00830	208,952
DC-8	Open, Laced Wiring	342.8	0.00643	155,995
737	Open, Laced Wiring	314.9	0.00520	26,448
720	Open, Laced Wiring	278.1	0.00769	49,772
727	Open, Laced Wiring	258.9	0.00520	163,895
C-123	Open, Laced Wiring	97.2	0.02396	45,862
C-141	Open, Laced Wiring	67.4	0.04367	259,747
F-106A/B	Open, Laced Wiring	21.1	0.19937	56,425
F-105B/D	Open, Laced Wiring	13.0	0.15903	107,336

① Mean time between maintenance actions (Hours)

② Maintenance manhours per flight hour

③ Flight hours on which MTBMA and MMH/FH are based type will be selected which will contain the arc from a short or overload.

- All relays and switches will be a sealed type to contain switching arcs.

- The relays and switches selected will have adequate current carrying capacity to meet the SFCS requirements.

- The spacing of switches, terminal strips, relays and circuit breakers will be adequate to prevent flashover between terminals.

- Adequate clearances will be maintained from all oxygen and hydraulic lines.

c. Overheat Protection

A study has been made of the planned SFCS wire bundle routing to determine which areas may require high temperature protection. The standard Dacron covered COMPACT wire bundle has a temperature rating of 300°F, which is sufficient for most applications. When the ambient temperature may be expected to exceed the limits of the Dacron braid, one of the following methods of heat protection will be used:

- In locations where the maximum ambient temperature approaches 392°F, the wire bundle will be covered with a shrinkable high temperature Teflon tubing or may be braided with Nomex, a high temperature nylon, as an alternate to Dacron.

- In locations where temperatures exceed 392°F, the affected branch of the wire bundle will be fabricated with high temperature Rockbestos insulated wire and covered with Viton impregnated Nomex braid, providing a bundle which may be used to 500°F.

d. Contraction and Expansion of Components

A study has been made of the contraction and expansion effects of the wire bundles and their components. These conditions do not present a problem for the installation of the SFCS wiring. The wiring will be installed in compliance with standard MCAIR practices which will provide sufficient slack in the bundle routing to prevent contraction or expansion from causing a strain on the wire terminations. All plugs and receptacles will be either environmental type or potted type to prevent moisture from reaching the pins which would cause short circuits and freeze damage.

4. WIRING DISPERSION

Dispersion of electrical wire bundles is an SFCS requirement to demonstrate the practicability of the principle of increased survivability through the use of dispersed redundancy. The quadruplex SFCS is expected to utilize the existing wire routing paths on both sides of the aircraft. An investigation of the airframe has been made from which additional routing paths may be established to provide wire bundle dispersion. However, there are locations in the aircraft where dispersion of the redundant circuits may not be practical. Bundles routed through areas where they cannot be dispersed without a major structural rework will have adequate mechanical and electrical protection to provide safe, reliable system operation.

a. Redundancy and Dispersion

The safety of the electrical wiring system for the SFCS test aircraft will be enhanced by application of the concepts of redundancy and dispersion. This is to be accomplished by isolating the four channels from each other, electrically and physically, throughout the aircraft. It is intended that failure of any single component in the aircraft electrical wiring system will not result in the loss of more than one channel. Physical isolation of SFCS wiring is to be accomplished by the COMPACT wiring technique where each channel will be isolated from each other channel and from all present aircraft wiring. This jacketed bundle will provide maximum mechanical protection for the SFCS wiring throughout the test aircraft. It is anticipated that all wiring between LRUs will be end-to-end hard wired with no breaks or connectors. The SFCS DC power and signal returns will be hard wired to their respective power and signal sources. AC power grounds, equipment chassis grounds, and certain aircraft electrical equipment returns such as relay coils, solenoid coils, and lighting devices will be connected to aircraft structure at the closest available previously established or added ground points. Twisting, shielding, and separation of wiring is expected to minimize electromagnetic interference within the SFCS and with existing aircraft systems. Strict separation of SFCS wiring from exterior lighting wiring, pitot heater wiring, and antenna feed lines is planned to minimize possible damage to the SFCS from lightning strike.

In general, the SFCS electrical installation is to consist of four separate routing paths throughout the aircraft. However, ideal dispersion of wiring could be incorporated into the entire aircraft only if the concept had been included in the aircraft's initial design. Since the F-4 airframe was not initially designed to include quadruplex wire routing paths, the SFCS electrical wiring installation will adhere to the following order of precedence:

- Four separate paths through an area.
- Two separate paths through an area with each bundle clamped separately.
- Two or four bundles "butterfly" clamped to the same support point.

In addition to the above precedence, the following criteria will be used:

- o Bundles routed, tied, or clamped together are to be acceptable only if one of the above choices cannot be met without major modification to the aircraft structure or equipment installation.

- o In no case will wiring from more than one channel or cross-channel wiring be mixed in one bundle where the complete failure of the bundle could cause the failure of more than one channel of the SFCS.

- o Twisting, shielding, and separation will be adhered to in accordance with good practices for electromagnetic compatibility.

- o Strict separation of SFCS wiring from exterior lighting wiring, pitot heater wiring, and antenna feed lines will be maintained.

b. Wire Routing

The general routing paths planned for the SFCS electrical wiring are shown in Figure 111. These routing paths and their limitations are described in the following paragraphs.

(1) Nose

The electrical wiring in the nose is expected to route from the SFCES Computer Voter Units, the Adaptive Gain and Stall Warning Computer, and the Maintenance Test Panel through the nose in four separate paths to bushings leading into the forward cockpit and holes leading into the lower forward fuselage area.

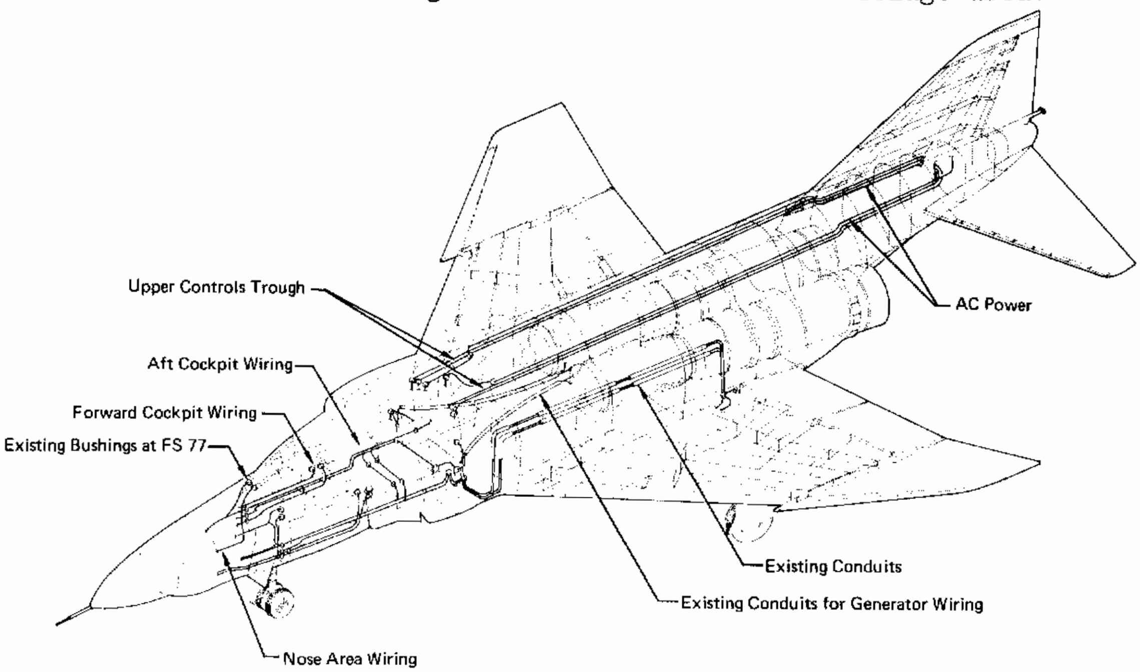


FIGURE 111
SFCS ELECTRICAL WIRE ROUTING

(2) Forward Cockpit

The SFCS bundles from the nose area are to enter the forward cockpit through four bushings in the forward bulkhead. They will then route aft through both right and left hand consoles. Due to the space limitations in the console areas, caused by the limited depth of the consoles, the size of the control panels, the cooling duct in the RH console, and the throttle quadrant and linkage in the LH console, it may be necessary to use the same clamps that the present aircraft wiring uses. The SFCS bundles that go to the Master Control and Display Panel are to be routed on either side of the instrument panel structure with existing aircraft bundles to the main instrument panel. The wiring to the SFCS instruments in the pedestal panel must be routed along with existing wiring through a fiberglass guard located on the lower RH side of the instrument panel structure. The fiberglass guard is used to keep the bundle from interfering with the pilot's feet, the RH rudder pedal at maximum travel, and the rudder pedal adjustment cable.

(3) Aft Cockpit

The SFCS bundles within the aft cockpit must be routed on each side of the aircraft with the present wiring. The SFCS wiring to the SFCES Secondary Control and Display Panel is expected to be routed separately from the present aircraft wiring. The SFCS bundles which exit from the aft cockpit into the upper controls trough and into the splitter area must route with present aircraft wiring through existing bushings.

The test aircraft's main electrical power switching and distribution system consisting of the main buses, AC contactor panel, and circuit breaker panels are located on the right hand side of the aft cockpit. The SFCS split bus electrical power switching and distribution system will retain the same component locations as the present aircraft; however, the wiring is to be all new. The large terminal blocks previously used as main buses are to be replaced with permanent splices. MIL-W-5088 establishes a preference of permanent splices over bolted connections. MCAIR experience has also substantiated this preference in terms of safety and reliability. Particular emphasis will be devoted in this area to enhance the reliability, electromagnetic compatibility, and safety of the SFCS electrical power switching and distribution system.

(4) Forward Fuselage

The SFCS bundles from the nose will route through both Side Looking Radar (SLR) antenna and equipment bays in separate paths. A new forward fuselage crossover path is planned below the cockpit floor for SFCS bundles. The SFCS bundles must route from the right and left hand SLR equipment bays into the splitter area with present aircraft wiring. In each splitter area they will route aft through one large hole in the aft bulkhead with the present aircraft bundle to the lower fuselage. The SFCS wiring then must route through two conduits on each side of the aircraft with present aircraft wiring to the center fuselage mold line conduits. It will be necessary to have two SFCS bundles in each conduit in addition to present aircraft wiring.

(5) Center Fuselage

The SFCS bundles will route via the upper controls troughs above and outboard of the fuel cells to the aft fuselage. At the aft end of the controls trough, the SFCS bundles must route through the same clamps with present aircraft wiring. There are two existing conduits on each side of the aircraft at the mold line skin for bundles routing from the forward fuselage through the center fuselage to the wings. All four SFCS channels must use this routing, two SFCS channels per conduit along with present aircraft wiring. The wiring exits from these conduits to the wings through two holes on each side of the aircraft.

(6) Aft Fuselage

The SFCS wiring is expected to route via four separate paths to the SFCS components in the aft fuselage.

(7) Inner Wing

The installation of the SFCS lateral Secondary Actuators requires a new routing for all wiring aft of the rear spar. The SFCS bundles are expected to be routed separately from the relocated aircraft system wiring.

c. AC System Test

Production F-4 aircraft have a single test receptacle for ground check-out of the electrical system. Test circuit leads from both AC generators and their controls are normally terminated in this receptacle. A "jumper" plug is mated with this receptacle except when ground check-out operations are being performed. For the SFCS, this single receptacle is expected to be replaced with two dissimilar receptacles. One receptacle will terminate the test circuits from the LH generating system and the other will terminate the test circuits from the RH system. A "patch" cable will be required to connect the two new receptacles to the single plug on the test equipment cable. A plug containing the necessary jumper circuitry will cover each of test receptacles. These plugs will be installed and suitably protected to prevent circuit interruptions and malfunctions after ground check-out operations are completed.

5. CONCLUSIONS

The modified split bus electrical system and the quadruplex SFCS DC power supplies are expected to be entirely capable of satisfying the program requirements.

The superiority of the MCAIR-developed COMPACT wiring has been proven by long, trouble-free usage. Its adaptability to various environmental extremes is a function of the materials used. The use of COMPACT wiring is therefore expected to meet the needs of the SFCS electrical wiring installation.

The electrical wiring in the SFCS test aircraft will be dispersed, end-to-end hard wired, and separated from other aircraft circuits. Dispersion and separation are particularly important and advantageous in minimizing EMI and helping to prevent catastrophic damage to the SFCS due to single-point electrical failures.

These studies provide a high degree of confidence that the presently planned SFCS electrical system will meet the SFCS program objectives. Primary emphasis has been placed on survivability through safety, redundancy, dispersion, and reliability.

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX III

ABBREVIATIONS:

A - Ampere

ACXR - AC Source Transfer Relay

ATA - Air Transport Association

CLTR - Caution Lights Test Relay

CSD - Constant Speed Drive

EDLC - Essential DC Line Contactor

EGLC - Emergency Generator Line Contactor

Emer - Emergency

EP - External Power

EPC - External Power Contactor

Ess - Essential

Flt. - Flight

LLC - Left Generator Line Contactor

Max - Maximum

Min - Minutes

MP1CR - Motor Pump 1 Two-Phase Cutout Relay

MP1CS - Motor Pump 1 Control Switch

MP1LC - Motor Pump 1 Line Contactor

MP2CR - Motor Pump 2 Two-Phase Cutout Relay

MP2CS - Motor Pump 2 Control Switch

MP2LC - Motor Pump 2 Line Contactor

N - Neutral

Pwr - Power

RLC - Right Generator Line Contactor

TC - Bus Tie Contactor

Typ - Typical

V_b - Battery Voltage

V_c - Cell Voltage (battery)

4MPCR - 4th Hyd. System Two-Phase Cutout Relay

4MPLC - 4th Hyd. System Motor Pump Line Contactor

SYMBOL:

ϕ Phase

APPENDIX IV

STABILATOR DYNAMIC AND AEROELASTIC TRENDS

1. INTRODUCTION AND SUMMARY

Changes in the stabilator elastic restraint and effective stabilator pitch inertia resulting from the installation of the SSAP could significantly affect stabilator aeroelastic characteristics. It was desirable, therefore, to generate trend curves, using these factors as parameters, which could be readily utilized with Ground Vibration Test (GVT) results to establish flight placards based on aeroelastic considerations.

Stabilator dynamic frequency and aeroelastic trends utilizing stabilator restraint stiffness and effective inertia as parameters are presented in Figure 112. In the following paragraphs the derivative of these curves is described and the method of use in conjunction with GVT results to establish aeroelastic margins is explained.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. AEROELASTIC CONSIDERATIONS

a. Vibration Modes and Instability Mechanisms

Five analytical normal vibration modes were considered in the aeroelastic investigations - stabilator 1st bending, fuselage 1st vertical bending, stabilator rotation, stabilator 2nd bending, and stabilator 1st torsion. Conventional velocity versus damping (V-g) flutter solutions were computed by using component flow aerodynamic theory with incompressible subsonic aerodynamics for sea level and revealed two mechanisms of instability: (1) a dynamic instability (flutter) involving the coupling of the stabilator 1st bending mode with the stabilator rotation mode, and (2) a static instability (divergence) involving only the stabilator rotation mode. For all values of stabilator restraint and stabilator inertia used in the studies, the flutter speed was significantly lower than the divergence speed. Therefore, it is assured that flight envelopes based on adequate flutter margins will also satisfy stability margin requirements for divergence.

b. Aspect Ratio and Compressibility Effects

Flutter speeds were computed using infinite aspect ratio incompressible aerodynamic theory for sea level. Aspect ratio and compressibility effects were incorporated into these results through use of a correction factor. This factor was determined as the ratio of the flight test flutter speed predicted by Reference 9 to the analytically determined flutter speed for the F-4 production Slotted Leading Edge (SLE) stabilator configuration.

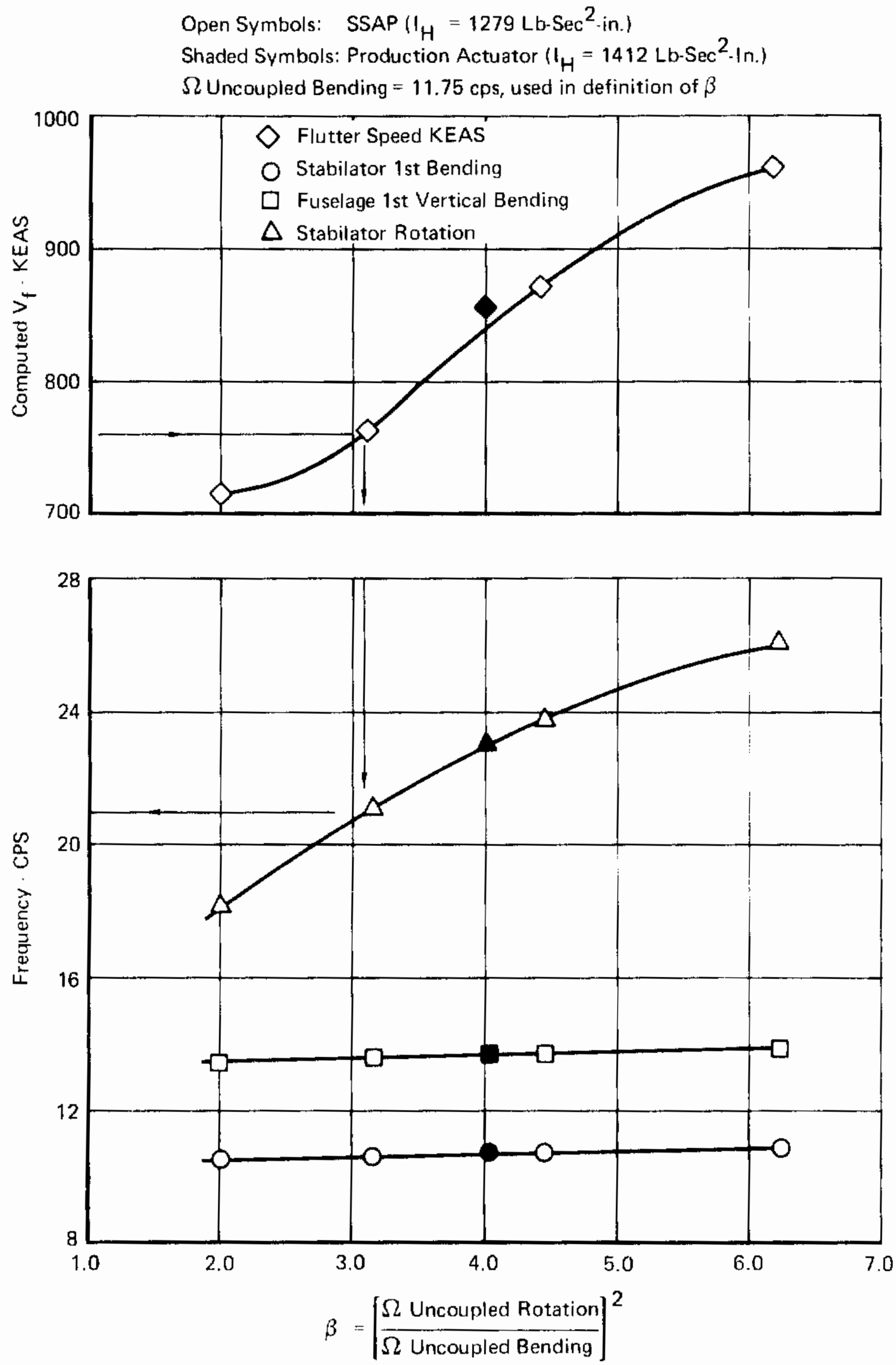


FIGURE 112
F-4 SLOTTED LEADING EDGE STABILATOR FREQUENCY AND AEROELASTIC TRENDS

c. SLE Stabilator Dynamic Frequency and Aeroelastic Trends

(1) Configuration Analyzed and Results

Figure 112 presents stabilator modal frequencies and flutter speeds plotted versus β, where β is the ratio of uncoupled stabilator rotation frequency to uncoupled stabilator bending frequency, squared. For each value of β considered, elastic modes were computed and utilized to obtain flutter solutions. A total of four stabilator restraint springs and two variations in stabilator pitch inertias were considered to establish flutter speed trends. Stabilator flutter speed trends with the SSAP installed were established using a stabilator pitch inertia of 1279.0 $lb\text{-}sec^2\text{-}in$ about the hinge line while varying stabilator pitch restraint. As a basis for determining a correction factor for aspect ratio and compressibility as discussed in Paragraph 2.b above, the production SLE stabilator configuration was also analyzed using the experimentally determined stiffness of 3.00×10^7 in-lb/radian and a stabilator pitch inertia of 1412.0 $lb\text{-}sec^2\text{-}in$. Results show that the coupled frequency trends for the SSAP and production actuators virtually coincide when plotted versus β. Therefore, the stabilator coupled rotation frequency can be used as the independent variable to specify the approximate value of β in Figure 112 for the range of stabilator pitch inertia being considered.

(2) Stabilator Rotation Frequency Requirements

For an unrestricted flight envelope including a 15% minimum velocity margin of safety requirement per Reference 10, it is sufficient to show a theoretical flutter speed of 760 KEAS for sea level altitude, the most critical Mach number - altitude combination as shown in Figure 9 of Reference 9. Referring to the directed line segments of Figure 112, it is apparent that the SSAP requires a minimum stabilator rotation frequency of 21.0 cps for unrestricted flight from the standpoint of aeroelastic stability considerations.

3. UTILIZATION OF GVT RESULTS

Prior to Phase IIC flight testing, a Ground Vibration Test (GVT) will be conducted on the SLE stabilator when powered by the SSAP to determine a flight envelope based on aeroelastic considerations. A flutter speed can be determined directly from Figure 112 using GVT frequency results by locating the stabilator coupled rotation frequency and referring to the appropriate flutter trend curve. However, in order to use the results of Figure 112 directly, the observed GVT stabilator bending frequency must also agree closely with the computed coupled stabilator bending frequency at the stabilator rotation frequency. Should this not be the case, a change in β is computed from the equation

$$\Delta\beta = \left[\left(\frac{f_c}{f_o}\right)^2 - 1.0\right]\beta \qquad (1)$$

where f_c and f_o are the computed and GVT stabilator 1st bending frequencies, respectively.

Then, using the flutter speed correction curve presented in Figure 113 the corrected flutter speed is given by

$$V_c = V_f + \frac{\partial V_f}{\partial \beta}\,\Delta\beta \qquad (2)$$

where V_f and $\frac{\partial V_f}{\partial \beta}$ are obtained from Figures 112 and 113, respectively.

4. CONCLUSIONS

Based on the results of studies presented herein, it has been shown that:

- The installation of the SSAP can significantly affect stabilator aeroelastic characteristics by changing the stabilator rotational restraint and effective pitch inertia.

- The effect of changes in the stabilator rotational restraint and effective pitch inertia on stabilator aeroelastic characteristics can be readily assessed using established trend curves in conjunction with GVT results.

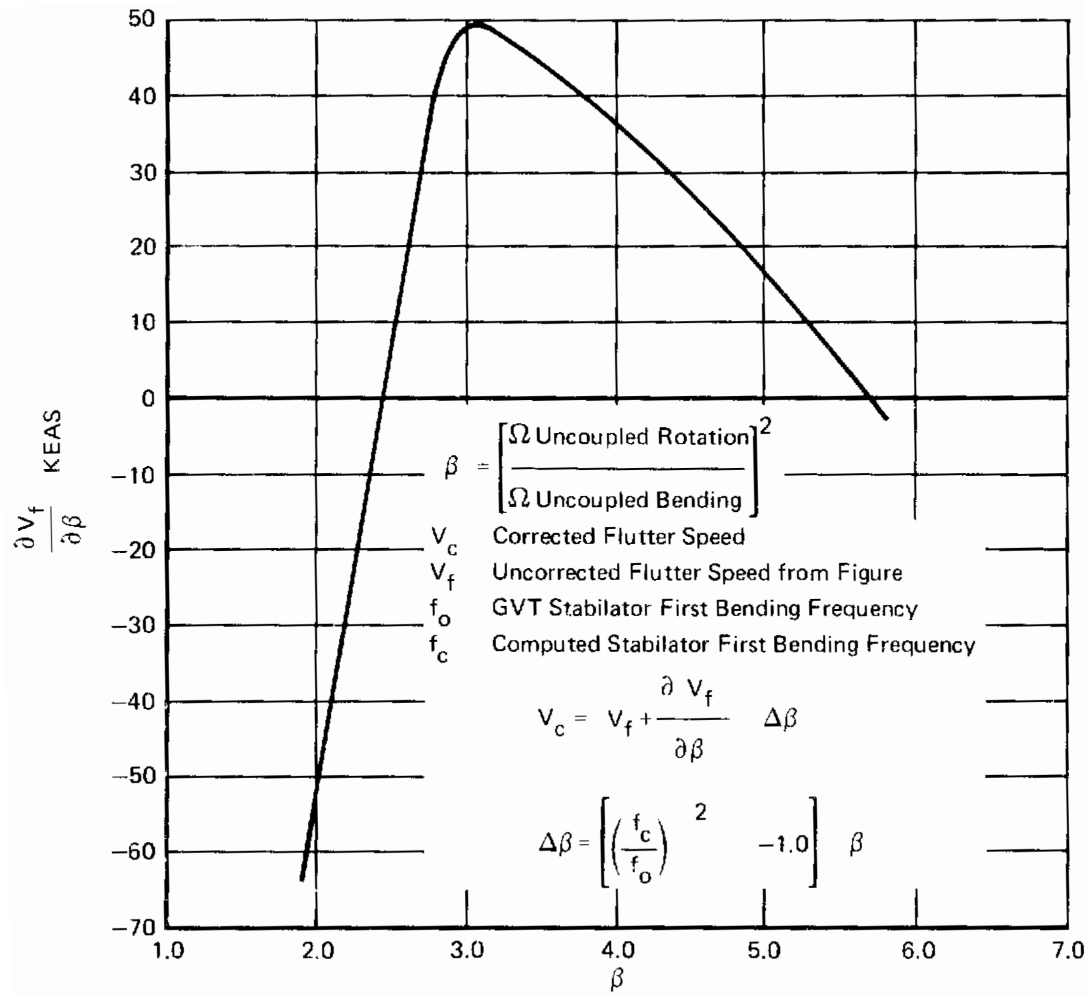


FIGURE 113
CORRECTION OF FLUTTER SPEED FOR SMALL VARIATIONS IN STABILATOR BENDING FREQUENCY

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX IV

ABBREVIATIONS:

GVT - Ground Vibration Test

KEAS - Equivalent Air Speed in Knots

SLE - Slotted Leading Edge

SYMBOLS:

f_c	Computed stabilator first bending frequency
f_o	Ground vibration test stabilator first bending frequency
g	Structural damping parameter - nondimensional
I_H	Effective pitch inertia about stabilator hinge line due to stabilator and actuator - lb-sec^2-in
V_c	Flutter speed corrected for differences between computed and ground vibration test stabilator first bending frequencies
V_f	Flutter speed
β	$[\Omega$ uncoupled rotation$/\Omega$ uncoupled bending$]^2$
Ω uncoupled rotation	Uncoupled stabilator rotation frequency defined as $\sqrt{k_H/I_H}$
Ω uncoupled bending	Uncoupled stabilator bending frequency defined as 73.827 - rad/sec
$\Delta\beta$	Incremental change in β
$\frac{\partial V_f}{\partial \beta}$	Partial derivative of flutter speed with respect to β resulting from a perturbation in the stabilator first bending frequency

APPENDIX V

STABILITY DERIVATIVES FOR LONGITUDINAL EQUATIONS OF MOTION

1. INTRODUCTION AND SUMMARY

Aeroelastic transfer functions were utilized in the Phase II longitudinal control system analysis and design. This was accomplished by the generation of flexible aircraft stability derivatives for significant elastic modes which were subsequently used to compute flexible vehicle transfer functions. This appendix describes the general approach and the equations of motion used; it also gives the results obtained for transfer functions, response characteristics, and stability characteristics.

Starting with a general discussion of the structural feedback phenomena encountered in flight control system analysis, this appendix proceeds from problem definition to the final formulation of the equations of motion. Stability derivatives and sensor feedback derivatives were presented for elastic degrees of freedom. Finally, it concludes with a discussion of the applicability of the selected approach.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. PROBLEM DEFINITION

The aircraft longitudinal control system senses motion using a system of normal accelerometers and rate gyros in pitch. Structural feedback coupling will exist if these devices detect structural motions and cause the generation of undesirable feedback signals. It is desirable to minimize these effects without severely compromising the control system response characteristics. This requires the consideration of aeroelastic transfer functions in the control system analysis and design.

3. GENERAL CONSIDERATIONS AND APPROACH

a. Degrees of Freedom

Two considerations were involved in the selection of elastic degrees of freedom to be used in the analysis. First, since the objective of the analysis was to include the effect of structural modes in longitudinal control system response characteristics, modes involving strong fuselage-stabilator coupling were chosen because these modes are excited by stabilator control inputs and contribute significantly to sensor responses. Secondly, vibration modes which adequately define the stabilator static as well as dynamic aeroelastic stability characteristics were chosen. Based on these considerations, three elastic degrees of freedom, which are aircraft normal modes, were included in the analysis. These three modes are stabilator first bending, fuselage first vertical bending, and stabilator rotation, respectively. The aircraft rigid degrees of freedom of vertical translation and pitch are excited by stabilator inputs, and were therefore, considered in the analysis. Since stabilator rigid rotation is the control surface input, the total dynamic system consisted of five degrees of freedom forced by the rigid stabilator control input.

b. Inertial Forces

In the course of the development of the F-4, it was discovered that some aircraft exhibited control system limit cycle instabilities on the ground with the stability augmentation system engaged. This problem was traced to structural feedback through the pitch rate gyro, and was fixed by the addition of a structural filter. As evidenced by the above incidents, the inertial forces and moments resulting from stabilator mass unbalance distribution along the span and the large mass moment of inertia excite fuselage elastic modes to a significant degree when the stabilator is subjected to a linear or angular acceleration, and have, therefore, been included in the analysis.

c. Aerodynamic Force Coefficients

(1) General

Aerodynamic forces considered in the analysis include those due to both displacement and rate and may be classified into two types - those which result from perturbations in the degrees of freedom themselves, and those which result from a rigid control input. The first type is involved in defining the dynamic characteristics (eigenvalues) of the system. Since longitudinal transfer functions are to be computed, the second type is required in defining forces in each degree of freedom resulting from the rigid stabilator control input.

(2) Rigid Aircraft Degree of Freedom Coefficients

The rigid aircraft degree of freedom coefficients, including static aeroelastic corrections, normally used to compute longitudinal rigid aircraft modes were utilized in the analysis herein. Because of the wide frequency separation between the fuselage and stabilator elastic modes considered and the rigid aircraft modes, the rigid aircraft degree of freedom coefficients do not include any significant static aeroelastic effects from the elastic modes considered. This means that these coefficients may be used with a flexible aircraft problem formulation including elastic degrees of freedom without introducing fuselage and stabilator elastic effects twice.

(3) Rigid Stabilator Force and Moment Coefficients

Using lift coefficient data consistent with the measured coefficients of Reference 11 for the rigid stabilator, component flow aerodynamics theory was used for computing generalized forces in the elastic degrees of freedom.

(4) Elastic Degree of Freedom Coefficients

Generalized forces were computed for the elastic degrees of freedom using measured coefficients as discussed above with aerodynamic forces applied to the stabilator only, since contributions from the wing and fuselage are considered to be negligible. For subsonic cases, the aerodynamic center was considered to be located at the aerodynamic section quarter-chord, and for supersonic cases, at the section mid-chord. This approach was also used in determining aerodynamic coupling effects between rigid and elastic degrees of freedom.

d. Structural Forces

The total oscillatory displacement of a structure can be obtained by superimposing the contributions of each of the elastic modes. Using a normal coordinate formulation, this is expressed as

$$Y(x,t) = \Sigma \Phi_i(x)\, \eta_i(t) \qquad (1)$$

where $\Phi_i(x)$ and $\eta_i(t)$ are the mode shape and normal coordinate, respectively. The corresponding unforced damped equation of motion is given by

$$\ddot{\eta}_i + 2\,\zeta_i \omega_i \dot{\eta}_i + \omega_i^2 \eta_i = 0 \qquad (2)$$

For each of the elastic modes considered, the damping ratio ζ_i and the undamped circular frequency ω_i were determined from ground vibration test logarithmic decrement data and resonant frequencies, respectively.

4. EQUATIONS OF MOTION

a. Introduction

The equations of motion have been formulated in the stability axis coordinate system shown in Figure 114. Since aircraft oscillatory displacements are relative to the rigid airframe reference, the elastic degree of freedom equations of motion retain their usual form.

b. Rigid Body Equations of Motion

The rigid body translational and rotational equations of motion can be written, respectively, as

$$M_{00}\,(\ddot{q}_0 - V\ \dot{q}_2) + M_{06}\,\ddot{q}_6 = Q_0(t) \qquad (3)$$

and

$$M_{22}\,\ddot{q}_2 + M_{26}\,\ddot{q}_6 = Q_2(t) \qquad (4)$$

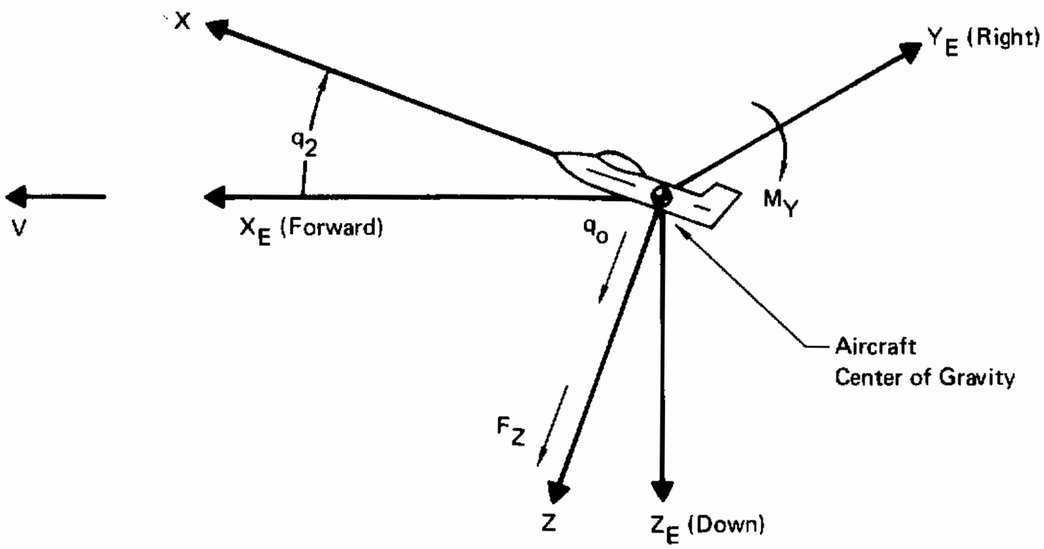


FIGURE 114
STABILITY AXIS COORDINATE SYSTEM FOR LONGITUDINAL EQUATIONS OF MOTION

Dividing both equations by their respective masses and substituting $\dot{q}_1$ for $\dot{q}_0/V$, these equations become

$$\dot{q}_1 - \dot{q}_2 + M_{06}/(V\ M_{00})\ddot{q}_6 = Q_0(t)/(V\ M_{00}) \qquad (5)$$

and

$$\ddot{q}_2 + M_{26}/M_{22}\ \ddot{q}_6 = Q_2(t)/M_{22} \qquad (6)$$

c. Flexible Airframe Equations of Motion

The equations of motion for airframe elastic degrees of freedom can be written as

$$\sum_j \{M_{ij}\ \ddot{q}_j + C_{ij}\ \dot{q}_j + K_{ij}\ q_j\} + M_{i6}\ \ddot{q}_6 = Q_i(t) \qquad (7)$$

where $\sum_j$ indicates summation over the j subscript. If the elastic modes are normal modes of vibration, then $M_{ij} = C_{ij} = K_{ij} = 0$ for $i \neq j$ where i, j = 3,4,5, and Equation (7) can be written as

$$\ddot{q}_i + 2\ \zeta_i \omega_i\ \dot{q}_i + \omega_i^{\ 2}\ q_i + M_{i6}/M_{ii}\ddot{q}_6 = Q_i(t)/M_{ii} \qquad (8)$$

As explained in Paragraph 3 .d., ζ_i and ω_i represent the structural damping ratio and circular frequency for the i^{th} mode at zero airspeed, respectively.

d. Aerodynamic Forces

When using unsteady incompressible potential flow theory, the generalized aerodynamic forces $Q_i(t)$ are generally given in the form

$$Q_i(t) = -\bar{q} \sum_j [\{A_{S_{ij}} + A_{C_{ij}} C(k)\} q_j + \{B_{S_{ij}} + B_{C_{ij}} C(k)\} \dot{q}_j/U] \qquad (9)$$

The term C(k) is the Theodorsen function which depends on the reduced frequency $k = \omega b/U$ and is, in general, a complex number. In order to keep the equations of motion in a mathematically tractable form while permitting the assessment of system dynamic characteristics at arbitrary airspeed with adequate accuracy, quasi-steady theory has been introduced. In this theory, C(k) = 1.0, permitting Equation (9) to be written for any Mach number as

$$Q_i(t) = -\bar{q} \sum_j [A_{ij} q_j + B_{ij} \dot{q}_j/U] \qquad (10)$$

where A_{ij} and B_{ij} represent, respectively, the matrices of the aerodynamic spring and damping obtained in each case by summing noncirculatory and circulatory contributions. The A_{ij} and B_{ij} coefficients are based on the rigid stabilator wind tunnel measurements presented in Reference 11, and in addition, depend on airfoil geometry and degree of freedom constraints.

e. Coupled Equations of Motion

Based on the considerations presented in Paragraphs 4.b through 4.d, the total system equations of motion can be written as

$$\begin{bmatrix} \dot{q}_1 - \dot{q}_2 \\ \ddot{q}_2 \\ \ddot{q}_3 \\ \ddot{q}_4 \\ \ddot{q}_5 \end{bmatrix} + [C] \begin{bmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \\ \dot{q}_4 \\ \dot{q}_5 \\ \dot{q}_6 \end{bmatrix} + [K] \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \\ q_5 \\ q_6 \end{bmatrix} + [A]\, \ddot{q}_6 = 0 \qquad (11)$$

where the [A], [C], and [K] matrices are defined by Table XXXIX and include both structural and aeroelastic effects. They are the stability derivatives for the flexible aircraft.

TABLE XXXIX

TABULATION OF ELEMENTS DEFINING THE [A], [C], AND [K] MATRICES FOR THE LONGITUDINAL EQUATIONS OF MOTION

$$
\left[C_{ij}\right] = \begin{bmatrix}
0.0 & -Z_{\dot{\theta}} & \frac{\bar{q}\ B_{03}}{U\ V\ M_{OO}} & \frac{\bar{q}\ B_{04}}{U\ V\ M_{OO}} & \frac{\bar{q}\ B_{05}}{U\ V\ M_{OO}} & \frac{\bar{q}\ B_{06}}{U\ V\ M_{OO}} \\
-M_{\dot{\alpha}} & -M_{\dot{\theta}} & \frac{\bar{q}\ B_{23}}{U\ M_{22}} & \frac{\bar{q}\ B_{24}}{U\ M_{22}} & \frac{\bar{q}\ B_{25}}{U\ M_{22}} & \frac{\bar{q}\ B_{26}}{U\ M_{22}} \\
0.0 & \frac{\bar{q}\ B_{32}}{U\ M_{33}} & 2\zeta_3\,\omega_3 + \frac{\bar{q}\ B_{33}}{U\ M_{33}} & \frac{\bar{q}\ B_{34}}{U\ M_{33}} & \frac{\bar{q}\ B_{35}}{U\ M_{33}} & \frac{\bar{q}\ B_{36}}{U\ M_{33}} \\
0.0 & \frac{\bar{q}\ B_{42}}{U\ M_{44}} & \frac{\bar{q}\ B_{43}}{U\ M_{44}} & 2\zeta_4\,\omega_4 + \frac{\bar{q}\ B_{44}}{U\ M_{44}} & \frac{\bar{q}\ B_{45}}{U\ M_{44}} & \frac{\bar{q}\ B_{46}}{U\ M_{44}} \\
0.0 & \frac{\bar{q}\ B_{52}}{U\ M_{55}} & \frac{\bar{q}\ B_{53}}{U\ M_{55}} & \frac{\bar{q}\ B_{54}}{U\ M_{55}} & 2\zeta_5\,\omega_5 + \frac{\bar{q}\ B_{55}}{U\ M_{55}} & \frac{\bar{q}\ B_{56}}{U\ M_{55}}
\end{bmatrix}
$$

$$
\left[K_{ij}\right] = \begin{bmatrix}
-Z_{\alpha} & 0.0 & \frac{\bar{q}\ A_{03}}{M_{OO}\ V} & \frac{\bar{q}\ A_{04}}{M_{OO}\ V} & \frac{\bar{q}\ A_{05}}{M_{OO}\ V} & \frac{\bar{q}\ A_{06}}{M_{OO}\ V} \\
-M_{\alpha} & 0.0 & \frac{\bar{q}\ A_{23}}{M_{22}} & \frac{\bar{q}\ A_{24}}{M_{22}} & \frac{\bar{q}\ A_{25}}{M_{22}} & \frac{\bar{q}\ A_{26}}{M_{22}} \\
\frac{\bar{q}\ V\ B_{30}}{U\ M_{33}} & 0.0 & \omega_3^2 + \frac{\bar{q}\ A_{33}}{M_{33}} & \frac{\bar{q}\ A_{34}}{M_{33}} & \frac{\bar{q}\ A_{35}}{M_{33}} & \frac{\bar{q}\ A_{36}}{M_{33}} \\
\frac{\bar{q}\ V\ B_{40}}{U\ M_{44}} & 0.0 & \frac{\bar{q}\ A_{43}}{M_{44}} & \omega_4^2 + \frac{\bar{q}\ A_{44}}{M_{44}} & \frac{\bar{q}\ A_{45}}{M_{44}} & \frac{\bar{q}\ A_{46}}{M_{44}} \\
\frac{\bar{q}\ V\ B_{50}}{U\ M_{55}} & 0.0 & \frac{\bar{q}\ A_{53}}{M_{55}} & \frac{\bar{q}\ A_{54}}{M_{55}} & \omega_5^2 + \frac{\bar{q}\ A_{55}}{M_{55}} & \frac{\bar{q}\ A_{56}}{M_{55}}
\end{bmatrix}
$$

$$
\left[A_i^T\right] = \begin{bmatrix}
\frac{M_{06}}{V\ M_{OO}} & \frac{M_{26}}{M_{22}} & \frac{M_{36}}{M_{33}} & \frac{M_{46}}{M_{44}} & \frac{M_{56}}{M_{55}}
\end{bmatrix}
$$

f. Angular Rate and Normal Acceleration Equations

Total displacement equations provide the basis for transfer function computation. Multiplying each of the normal coordinates with its respective mode shape and superimposing rigid and elastic contributions, the equations for total pitch rate and normal acceleration are

$$\dot{\theta} = \dot{q}_2 + \sum_{i=3}^{5} \frac{\partial \theta(x)}{\partial q_i} \dot{q}_i \qquad (12)$$

and

$$N_z = V(\dot{q}_1 - \dot{q}_2) - x\ddot{q}_2 + \sum_{i=3}^{5} \frac{\partial h(x)}{\partial q_i} \ddot{q}_i \qquad (13)$$

g. Transfer Function Computation

Taking the Laplace transform of Equations (11), (12), (13) with initial displacements and velocities equal to zero, the equations are rewritten in the form presented in Table XL. After assuming sinusoidal motion and substituting $S = i\omega$, the transfer functions $\dot{\theta}(\omega)/q_6(\omega)$ and $N_z(\omega)/q_6(\omega)$ can be directly computed.

TABLE XL
TRANSFORMED LONGITUDINAL EQUATIONS OF MOTION

$$\begin{bmatrix}
(1.0+C_{11})S + K_{11} & (C_{12}-1.0)S + K_{12} & C_{13}S + K_{13} & C_{14}S + K_{14} & C_{15}S + K_{15} & 0.0 & 0.0 \\
C_{21}S + K_{21} & S^2 + C_{22}S + K_{22} & C_{23}S + K_{23} & C_{24}S + K_{24} & C_{25}S + K_{25} & 0.0 & 0.0 \\
C_{31}S + K_{31} & C_{32}S + K_{32} & S^2 + C_{33}S + K_{33} & C_{34}S + K_{34} & C_{35}S + K_{35} & 0.0 & 0.0 \\
C_{41}S + K_{41} & C_{42}S + K_{42} & C_{43}S + K_{43} & S^2 + C_{44}S + K_{44} & C_{45}S + K_{45} & 0.0 & 0.0 \\
C_{51}S + K_{51} & C_{52}S + K_{52} & C_{53}S + K_{53} & C_{54}S + K_{54} & S^2 + C_{55}S + K_{55} & 0.0 & 0.0 \\
0.0 & -S & -S\frac{\partial\theta(x)}{\partial q_3} & -S\frac{\partial\theta(x)}{\partial q_4} & -S\frac{\partial\theta(x)}{\partial q_5} & 1.0 & 0.0 \\
-SV & xS^2 + VS & -S^2\frac{\partial h(x)}{\partial q_3} & -S^2\frac{\partial h(x)}{\partial q_4} & -S^2\frac{\partial h(x)}{\partial q_5} & 0.0 & 1.0
\end{bmatrix}
\begin{pmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \\ q_5 \\ \dot{\theta}(S) \\ N_z(S) \end{pmatrix}
= -
\begin{pmatrix} A_{16}S^2 + C_{16}S + K_{16} \\ A_{26}S^2 + C_{26}S + K_{26} \\ A_{36}S^2 + C_{36}S + K_{36} \\ A_{46}S^2 + C_{46}S + K_{46} \\ A_{56}S^2 + C_{56}S + K_{56} \\ 0.0 \\ 0.0 \end{pmatrix} q_6$$

h. Data Package

(1) Configuration Analyzed

Stability derivatives were computed for the 41,001 pound combat-gear up configuration using the inertial properties presented in Reference 12. Pertinent data for the configuration are summarized in Table XLI.

TABLE XLI
AIRCRAFT INERTIAL CHARACTERISTICS FOR LONGITUDINAL STABILITY DERIVATIVES

Combat - Gear Up

Weight: 41,001 Lb

Center of Gravity Location

Horizontal: Fuselage Station 315.61

Vertical: Water Line 26.50

Pitch Moment of Inertia

About Center of Gravity: 152,495 Slug - Ft^2

(2) General Applicability of Stability Derivatives

Changes in aircraft mass, pitch moment of inertia, and center of gravity must be considered when computing stability derivatives for each different mass configuration. Since the aircraft elastic modes considered herein do not change appreciably with moderate changes in these parameters, these changes are reflected only in the vertical translation and pitch equations of motion. Observing that the aircraft mass and pitching moment of inertia appear in the denominators of the stability derivatives for the first two equations of motion defined by Equation (11), it is apparent that a change in mass or inertia is effected by multiplying the stability derivatives of the appropriate equations by the ratio of the referenced weight (or inertia) to the new weight (or inertia). The coefficients $M_{\dot{\alpha}}$, $Z_{\dot{\theta}}$, $M_{\dot{\theta}}$, M_{α} , and Z_{α} are affected to a greater extent by a slight shift in aircraft center of gravity. Therefore, changes in center of gravity must be taken into account in calculating these coefficients for various aircraft weight and flight conditions when using the data of Reference 11.

(3) Stability Derivatives

The stability derivatives for the longitudinal equations of motion are presented in Supplement 2 in terms of the notation used therein. Table XLII provides definitions of the [A], [C], and [K] matrices using Supplement 2 notation.

(4) Sensor Feedback Derivatives

Table XLIII presents sensor feedback derivatives for all of the sensor locations considered. These data are utilized with Equations (12) and (13) to compute total aircraft response at a desired location. Figure 115 presents the same data as Table XLIII only on a continuous basis between aircraft fuselage stations (FS) 300 and 450. These data were utilized to study the effects of shifts in sensor location in the vicinity of F.S. 383.00 in order to determine an optimum location.

TABLE XLII
DEFINITION OF [A], [C], AND [K] MATRICES FOR LONGITUDINAL EQUATIONS OF MOTION IN TERMS OF SUPPLEMENT 2 NOTATION

$$[C_{ij}] = \begin{bmatrix} 0.0 & -Z_{\dot{\theta}} & -Z_{\dot{\eta}_1} & -Z_{\dot{\eta}_2} & -Z_{\dot{\eta}_3} & -Z_{\dot{\delta}} \\ -M_{\dot{\alpha}} & -M_{\dot{\theta}} & -M_{\dot{\eta}_1} & -M_{\dot{\eta}_2} & -M_{\dot{\eta}_3} & -M_{\dot{\delta}} \\ 0.0 & -F_{\dot{\theta}} & -F_{\dot{\eta}_1} & -F_{\dot{\eta}_2} & -F_{\dot{\eta}_3} & -F_{\dot{\delta}} \\ 0.0 & -G_{\dot{\theta}} & -G_{\dot{\eta}_1} & -G_{\dot{\eta}_2} & -G_{\dot{\eta}_3} & -G_{\dot{\delta}} \\ 0.0 & -H_{\dot{\theta}} & -H_{\dot{\eta}_1} & -H_{\dot{\eta}_2} & -H_{\dot{\eta}_3} & -H_{\dot{\delta}} \end{bmatrix}$$

$$[K_{ij}] = \begin{bmatrix} -Z_{\alpha} & 0.0 & -Z_{\dot{\eta}_1} & -Z_{\dot{\eta}_2} & -Z_{\dot{\eta}_3} & -Z_{\delta} \\ -M_{\alpha} & 0.0 & -M_{\dot{\eta}_1} & -M_{\dot{\eta}_2} & -M_{\dot{\eta}_3} & -M_{\delta} \\ -F_{\alpha} & 0.0 & -F_{\dot{\eta}_1} & -F_{\dot{\eta}_2} & -F_{\dot{\eta}_3} & -F_{\delta} \\ -G_{\alpha} & 0.0 & -G_{\dot{\eta}_1} & -G_{\dot{\eta}_2} & -G_{\dot{\eta}_3} & -G_{\delta} \\ -H_{\alpha} & 0.0 & -H_{\dot{\eta}_1} & -H_{\dot{\eta}_2} & -H_{\dot{\eta}_3} & -H_{\delta} \end{bmatrix}$$

$$[A_i^T] = \begin{bmatrix} -Z_{\ddot{\delta}} & -M_{\ddot{\delta}} & -F_{\ddot{\delta}} & -G_{\ddot{\delta}} & -H_{\ddot{\delta}} \end{bmatrix}$$

TABLE XLIII
SUMMARY OF LONGITUDINAL SENSOR FEEDBACK DERIVATIVES

i	$\left(\frac{\partial h}{\partial q_i}\right)$ - Inches FS 24 (FS 77)	$\left(\frac{\partial \theta}{\partial q_i}\right)$ - Radians FS 179	$\left(\frac{\partial \theta}{\partial q_i}\right)$ - Radians FS 313	$\left(\frac{\partial \theta}{\partial q_i}\right)$ - Radians FS 383
3	−0.0733 (−0.0573)	0.000158	0.000233	0.00
4	0.2850 (0.1760)	0.001070	0.000264	0.000711
5	0.0100 (0.0100)	0.000114	0.00	−0.000114

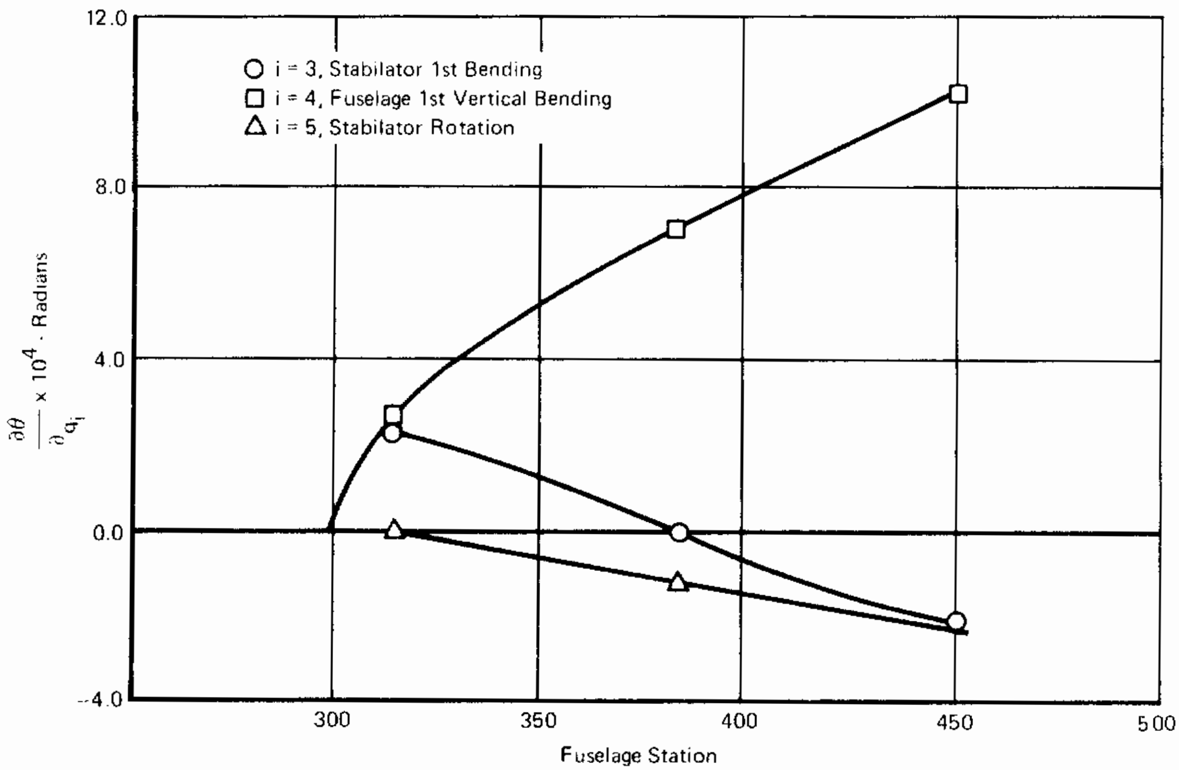


FIGURE 115
SENSOR FEEDBACK DERIVATIVES vs FUSELAGE STATION
FOR LONGITUDINAL EQUATIONS OF MOTION

5. RESULTS

a. Computed Transfer Functions

As a basis for discussion, Figures 116 and 117 present $\dot{\theta}(\omega)/q_6$ and $N_z\ (\omega)/q_6$ transfer functions computed using 0.9 Mach stability derivatives modified as explained in Paragraph 4.h.(2) for a 38,732 pound aircraft configuration. The pitch rate and normal accelerometer response locations are at FS 383 and FS 77, respectively.

b. Low Frequency Response Characteristics

The response characteristics for frequencies below the fuselage and stabilator elastic modes are in close agreement with previous rigid body results obtained using experimentally verified effective aeroelastic coefficients and ignoring elastic degrees of freedom. This agreement implies that the appropriate elastic modes and aerodynamics have been included in the analysis and, from this one standpoint alone, gives confidence in the adequacy of the high frequency transfer function characteristics.

c. High Frequency Response Characteristics

For frequencies above the rigid body modes, confidence in theoretical response characteristics is based on use of a good definition of airframe dynamics and excellent agreement with previous results in the low frequency region as discussed above. It should be remembered that the aeroelastic transfer function is an open loop transfer function and does not approach zero with increasing frequency if stabilator rate and acceleration forces are included in addition to the stabilator position forces in the analysis. However, had the stabilator actuator and control system been included in the transfer computation, the closed loop response would have approached zero with increasing frequency since actuator rate and displacement approach zero as command input frequency increases.

d. Dynamic Aeroelastic Stability Characteristics

As a further check to gain confidence in theoretical response characteristics, the stability derivatives were used in a two degree of freedom flutter analysis of the stabilator. The stabilator first bending and rotation modes were considered in this analysis. A flutter speed of 750 KEAS was computed using Mach 0.9 displacement stability derivatives for sea level. After applying an oscillatory flow correction, this speed becomes 825 KEAS which is in good agreement with the 855 KEAS predicted by the Reference 9 flight flutter test report. A similar analysis of supersonic cases show that no dynamic aeroelastic instabilities exist regardless of flight velocity.

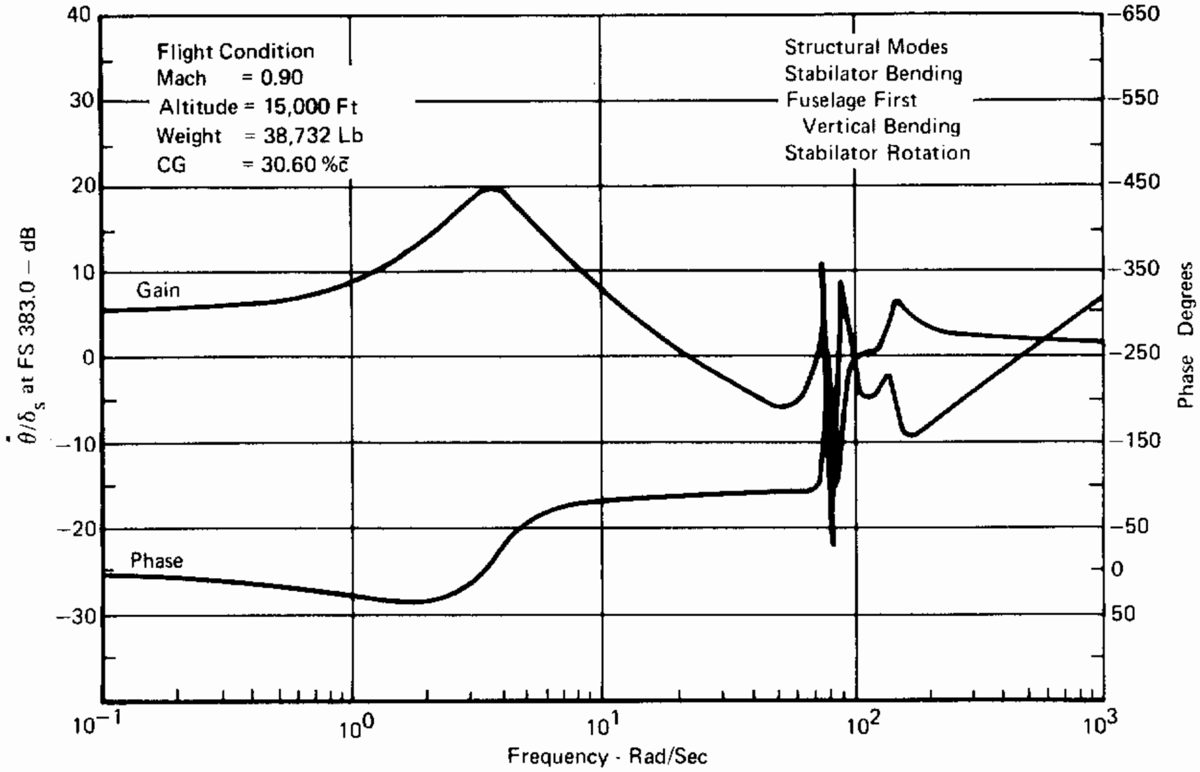


FIGURE 116
PITCH RATE FREQUENCY RESPONSE

e. Pitch Rate Gyro Location

The pitch rate response characteristics for the selected pitch rate gyro location at FS 383 are of particular interest because the location corresponds to a zero pitch slope for the stabilator first bending mode. Therefore, if the pitch rate gyro is located at that point, it will not sense any direct contribution from the stabilator first bending mode. However, examination of the pitch rate response plot of Figure 116 reveals contributions from the stabilator first bending, fuselage first vertical bending, and stabilator rotation modes. The peak at approximately 78.0 radians/second results from excitation of the fuselage first vertical bending mode by means of aeroelastic coupling with the stabilator first bending mode. This explains why the point of zero pitch slope of the one vibration mode

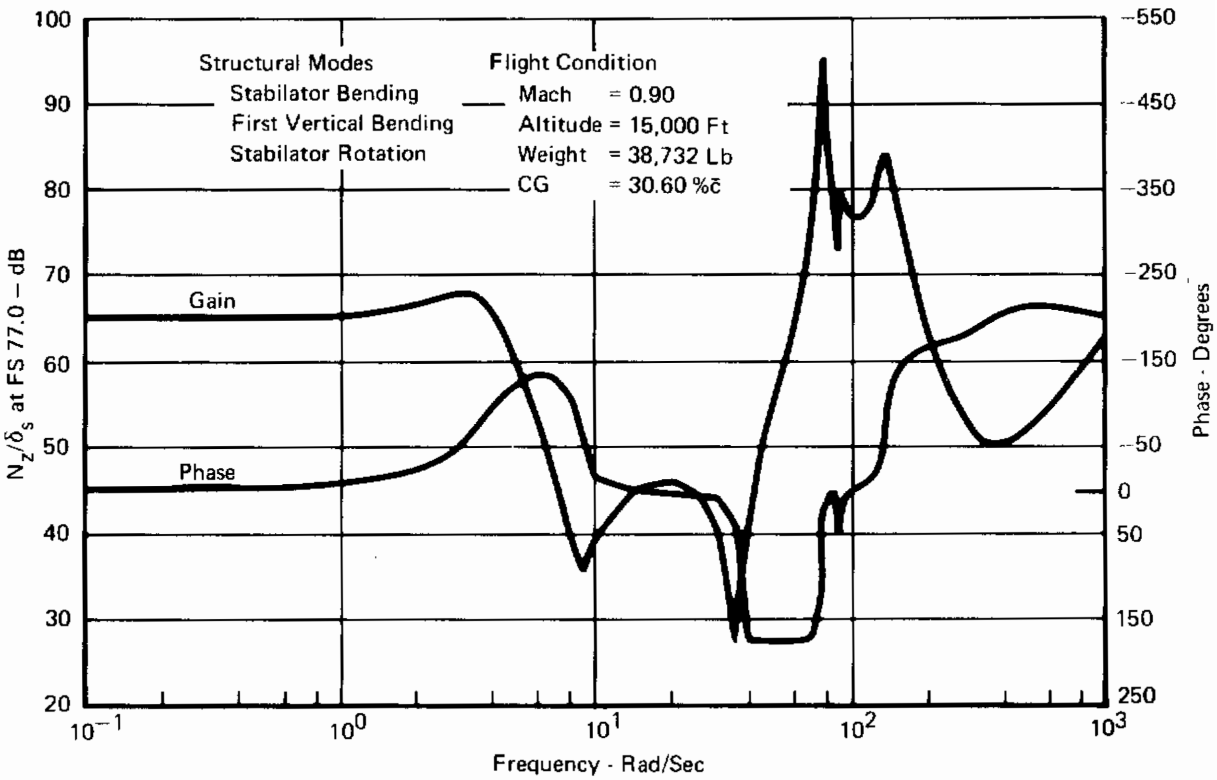


FIGURE 117

NORMAL ACCELERATION FREQUENCY RESPONSE

expected to be most strongly coupled is not valid. As can be seen from the curves of Figure 115 the stabilator first bending mode response can be directly introduced by locating the sensor either forward or aft of FS 383. Furthermore, relocating the rate gyro will either amplify or attenuate the resonant peaks. Thus, it is readily seen that undesirable airframe response characteristics can be minimized by optimizing sensor locations. Coupling of the higher frequency structural resonances shown in Figures 116 and 117 into the system is prevented by notch filters incorporated in the forward loop. A more detailed discussion is provided in Supplement 2.

6. DISCUSSION

a. Aerodynamics

In deriving stability derivatives for computing forced response, it is important to have the aerodynamic force distribution defined with regard to magnitude and aerodynamic center. For this reason, the computations should be based on rigid model wind tunnel measurements. In the absence of measured center of pressure data, it is advisable to use the aerodynamic section quarter-chord as aerodynamic center for subsonic cases and the mid-chord for supersonic cases.

b. Structural Damping

Since aeroelastic transfer function characteristics are very sensitive to system damping, it is essential to include structural damping coefficients determined from ground vibration test logarithmic decrement data. For a stable aeroelastic system, the structural damping contribution for fuselage and empennage modes is generally large compared with the aerodynamic damping.

c. Mathematical Approach

(1) Quasi-Steady Theory

As explained in Paragraph 4.d. quasi-steady aerodynamic theory was utilized in the computation of stability derivatives. In addition to being mathematically expedient, this approach seems highly justified when being applied to a stable aeroelastic system where damping is primarily determined by structural characteristics. It has been shown in Reference 13 to yield good flutter results for surfaces of low aspect ratio. Furthermore, the flexible airframe stability derivatives were computed with the primary objective of quantitatively evaluating various sensor locations rather than providing absolute airframe - control system stability margins.

(2) Indicial Lift

As explained in Reference 14, the aeroelastic equations of motion may be solved using the indicial lift approach which transforms the aerodynamics into a nondimensional time domain. This procedure provides a mathematical solution for the equations of motion, but is unwieldy because it requires solving a characteristic equation with complex coefficients. Its use might be justified when attempting to provide servo-stabilization for an aeroelastic system.

d. Head or Tail Wind versus Still Air

The stability derivatives presented herein were computed assuming an aircraft flying through still air. In this case, the speed of the airfoil relative to the air stream is equal to the absolute velocity of the aircraft, or

$$U = V \tag{14}$$

However, should the aircraft be flying against a head wind or with a tail wind, this equality becomes, respectively,

$$U > V \quad \text{or} \quad U < V \tag{15}$$

where the dynamic aeroelastic stability characteristics are primarily controlled by U, and the transfer function response levels by both U and V. The distinction between U and V could become important when providing servo-stabilization for an aeroelastic instability.

7. CONCLUSION

The general approach for computing longitudinal stability derivatives has been treated in detail. The resulting stability derivatives were utilized in Phase II control system design and analysis, as described in Supplement 2.

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX V

SYMBOLS:

$[A]$	Matrix defined per Table XXXIX
A_{ij}	$A_{S_{ij}} + A_{C_{ij}}$, summation of circulatory and noncirculatory aerodynamic stiffness derivatives
$A_{C_{ij}}$	Generalized circulatory aerodynamic stiffness derivative
$A_{S_{ij}}$	Generalized noncirculatory aerodynamic stiffness derivative
$[A_i^T]$	Matrix defined per Table XXXIX
b	Wing semi-chord - in
B_{ij}	$B_{S_{ij}} + B_{C_{ij}}$, summation of circulatory and noncirculatory aerodynamic damping derivatives
$B_{C_{ij}}$	Generalized circulatory aerodynamic damping derivative
$B_{S_{ij}}$	Generalized noncirculatory aerodynamic damping derivative
$[C]$	Matrix defined per Table XXXIX
C.g.(CG)	Center of gravity
C_{ij}	Structural damping coefficient
$C(k)$	Theodorsen function
F_z	Force along Z axis in stability axis coordinate system - positive down
F_α	Stabilator bending mode acceleration due to angle of attack - $1/sec^2$
$F_{\dot{\theta}}$	Stabilator bending mode acceleration due to pitch rate - 1/sec
F_δ	Stabilator bending mode acceleration due to stabilator deflection - $1/sec^2$
$F_{\dot{\delta}}$	Stabilator bending mode acceleration due to stabilator rate - 1/sec
$F_{\ddot{\delta}}$	Stabilator bending mode acceleration due to stabilator acceleration - dimensionless
$F_{\dot{\eta}_1}$	Stabilator bending mode acceleration due to stabilator bending rate - 1/sec

$F_{\dot{\eta}_2}$	Stabilator bending mode acceleration due to first vertical bending rate - 1/sec
$F_{\dot{\eta}_3}$	Stabilator bending mode acceleration due to stabilator rotation mode rate - 1/sec
G_{α}	First vertical bending mode acceleration due to angle of attack - 1/sec^2
$G_{\dot{\theta}}$	First vertical bending mode acceleration due to pitch rate - 1/sec
G_{δ}	First vertical bending mode acceleration due to stabilator deflection - 1/sec^2
$G_{\dot{\delta}}$	First vertical bending mode acceleration due to stabilator rate - 1/sec
$G_{\ddot{\delta}}$	First vertical bending mode acceleration due to stabilator acceleration - dimensionless
$G_{\dot{\eta}_1}$	First vertical bending mode acceleration due to stabilator bending rate - 1/sec
$G_{\dot{\eta}_2}$	First vertical bending mode acceleration due to first vertical bending mode - 1/sec
$G_{\dot{\eta}_3}$	First vertical bending mode acceleration due to stabilator rotation mode rate - 1/sec
H_{α}	Stabilator rotation mode acceleration due to angle of attack - 1/sec^2
$H_{\dot{\theta}}$	Stabilator rotation mode acceleration due to pitch rate - 1/sec
H_{δ}	Stabilator rotation mode acceleration due to stabilator deflection - 1/sec^2
$H_{\dot{\delta}}$	Stabilator rotation mode acceleration due to stabilator rate - 1/sec
$H_{\ddot{\delta}}$	Stabilator rotation mode acceleration due to stabilator acceleration - dimensionless
$H_{\dot{\eta}_1}$	Stabilator rotation mode acceleration due to stabilator bending rate - 1/sec
$H_{\dot{\eta}_2}$	Stabilator rotation mode acceleration due to first vertical bending rate - 1/sec
$H_{\dot{\eta}_3}$	Stabilator rotation mode acceleration due to stabilator rotation mode rate - 1/sec

[K]	Matrix defined per Table XXXIX
KEAS	Equivalent airspeed - knots
K_{ij}	Structural spring coefficient
M_{ij}	Inertia derivative
M_Y	Moment about Y axis in stability axis coordinate system positive nose up
M_α	Rigid aircraft aerodynamic coefficient for q_1 in rotational equation of motion
$M_{\dot{\alpha}}$	Rigid aircraft aerodynamic coefficient for $\dot{q}_1$ in rotational equation of motion
M_δ	Pitching angular acceleration due to stabilator deflection - $1/sec^2$
$M_{\dot{\delta}}$	Pitching angular acceleration due to stabilator rate - 1/sec
$M_{\ddot{\delta}}$	Pitch angular acceleration due to stabilator acceleration - dimensionless
$M_{\dot{\theta}}$	Rigid aircraft aerodynamic coefficient for $\dot{q}_2$ in rotational equation of motion
$M_{\dot{\eta}_1}$	Pitching angular acceleration due to rate of stabilator bending mode - 1/sec
$M_{\dot{\eta}_2}$	First vertical bending mode due to rate of stabilator bending mode - 1/sec
$M_{\dot{\eta}_3}$	Stabilator rotation mode due to rate of stabilator bending mode - 1/sec
N_z	Total flexible airframe acceleration at accelerometer location - in/sec^2
$Q_i(t)$	Generalized force for i^{th} equation of motion
q_j	Generalized coordinate for j^{th} degree of freedom
q_o	Vertical displacement of rigid aircraft center of gravity - inches, positive down
q_1	Defined as $\dot{q}_o/V_T$, where the dot denotes differentiation with respect to time and V_T the true airspeed - in/sec
q_2	Rigid aircraft pitch perturbation with respect to inertial reference - radians, positive nose up

q_3	Normal coordinate representing stabilator first symmetric bending mode - rad
q_4	Normal coordinate representing fuselage first vertical bending mode - rad
q_5	Normal coordinate representing stabilator rotation mode - rad
q_6	Independent variable (control input) representing rigid stabilator with respect to rigid aircraft fuselage - radians, positive leading edge up
S	Laplace transform variable
U	True airspeed - in/sec
V	Absolute speed of aircraft center of gravity - in/sec
X, Y, Z	Reference axes in the stability axis reference system
x	Distance in inches positive forward of aircraft center of gravity
X_E, Y_E, Z_E	Axes in earth reference coordinate system
Z_q, $Z_{\dot{\theta}}$	Flight path angular velocity due to unit pitch rate - dimensionless
Z_α	Rigid aircraft aerodynamic coefficient for q_1 in translational equation of motion
Z_δ	Flight path angular velocity due to stabilator deflection - 1/sec
$Z_{\dot{\delta}}$	Flight path angular velocity due to stabilator rate - dimensionless
$Z_{\ddot{\delta}}$	Flight path angular velocity due to stabilator acceleration - sec
$Z_{\dot{\theta}}$	Rigid aircraft aerodynamic coefficient for $\dot{q}_2$ in translational equation of motion
$Z_{\dot{\eta}_1}, Z_{\dot{\eta}_2}, Z_{\dot{\eta}_3}$	Flight path angular velocity due to deflection rates of stabilator bending mode, first vertical bending mode, and stabilator rotation mode, respectively - dimensionless
∂	Character denoting partial differentiation
$\partial h(x)/\partial q_i$	Bending Displacement for i^{th} mode x inches forward of aircraft center of gravity - inches, positive down

$\partial\theta(x)/\partial q_i$	Pitch slope for i^{th} mode at x inches forward of aircraft center of gravity - radians, positive nose up
ζ_i	Damping ratio associated with i^{th} elastic mode
$\eta_i(t)$	Normal coordinate to represent i^{th} elastic mode - rad
$\dot{\theta}$	Total flexible airframe pitch rate at gyro location - rad/sec
$\sum_i$	Denotes summation over subscript i
$\Phi_i(x)$	Mode shape of i^{th} mode, either displacement or slope
ω_i	Undamped natural frequency for i^{th} elastic mode - rad/sec
$\%\bar{c}$	Location of the center of gravity expressed as a percent of the mean aerodynamic chord

APPENDIX VI

STABILITY DERIVATIVES FOR LATERAL-DIRECTIONAL EQUATIONS OF MOTION

1. INTRODUCTION AND SUMMARY

Aeroelastic transfer functions were utilized in the Phase II lateral-directional control system analysis and design. This was accomplished with the generation of flexible aircraft stability derivatives for significant elastic modes which were subsequently utilized to compute flexible vehicle transfer functions. This appendix documents the stability derivative computational effort for the elastic modes.

Starting with a general discussion of the structural feedback phenomena encountered in flight control system analysis, this appendix proceeds from problem definition to the final formulation of the equations of motion. Stability derivatives and sensor feedback derivatives are presented for elastic degrees of freedom. Particular attention is devoted to defining aileron, spoiler, and rudder control input forces. Finally, it concludes with a discussion of the applicability of the selected approach.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. PROBLEM DEFINITION

The aircraft lateral-directional control system senses motion using a system of lateral accelerometers and rate gyros in roll and yaw. Structural feedback coupling will exist if these devices detect structural motions and cause the generation of undesirable feedback signals. It is desirable to minimize these effects without severely compromising the control system response characteristics. This requires the consideration of aeroelastic transfer functions in the control system analysis and design.

3. GENERAL CONSIDERATIONS AND APPROACH

a. Required Transfer Functions

In the SFCS, lateral-directional control is implemented by feeding back roll rate signals to the aileron-spoiler channel, and airframe lateral acceleration and yaw rate to the rudder channel. The required transfer functions are, therefore, roll rate per unit aileron-spoiler input, lateral acceleration per unit rudder input, and yaw rate per unit rudder input. These are denoted, respectively by $\dot{\Phi}_T/q_7$, N_y^A/q_8, and $\dot{\psi}_T/q_8$.

b. Degrees of Freedom

(1) Aircraft Degrees of Freedom

The motion at the aircraft center of mass was defined by three degrees of freedom - lateral translation, roll, and yaw. In order to define primary surface and control surface flexibilities as well as the airframe response at sensor locations, it is desirable to include all airframe elastic modes through the control system cutoff frequency. To keep the equations of motion from becoming unwieldly for computer mechanization of the system, only modes characterized by significant fuselage displacements were chosen. These modes included fuselage first torsion, wing first asymmetric bending, and fuselage first lateral bending. Since these modes did not include all the basic primary surface and control surface modes required to adequately define control surface effectiveness, this aspect of the analysis was handled by using control input coefficients which include static aeroelastic effects in the rigid body equations of motion as discussed in Paragraph 3.d.(1).

(2) Rigid Control Surface Inputs

Rigid aileron, spoiler, and rudder degrees of freedom were the independent variables considered in the analysis. Spoiler and aileron effects were subsequently combined, since the spoiler nominally deflects 1.5 degrees for each degree of aileron deflection.

c. Control Surface Inertial Forces

Inertial forces generated by control surface angular accelerations result from either mass unbalance about the hinge line or mass moment of inertia. These factors do work, respectively, on modes having either displacement or slope at the control surface location. Since the rudder is very nearly statically mass balanced about its hinge line and has a relatively small mass moment of inertia, its inertial effects were not included in the analysis. Similarly, aileron and spoiler inertial effects were excluded because the modal displacements at the control surface locations were small. The adequacy of this approach is further corroborated by the fact that no structural feedback through the lateral-directional control system has ever been observed in the course of F-4 development, either on the ground or in flight.

d Aerodynamic Forces

(1) Types

Forces proportional to both rate and displacement have been considered in the analysis. These were based on the wind tunnel data presented in Reference 11 and may be classified as being either rigid or effective aeroelastic forces. Control surface aerodynamic forces in the rigid body equations of motion included the effects of both airframe and control surface compliances, and hence, are referred to as effective aeroelastic forces. Since these forces essentially account for the combined effects of airframe elastic modes, they were adequate to define the motion at the aircraft center of mass. In contrast, the control surface aerodynamic forces in the aeroelastic equations of motion were computed for discrete elastic modes, and therefore, were based on rigid model wind tunnel data. This procedure allowed modes to retain their identity for utilization in computing sensor responses.

(2) Application

Generalized forces were computed for the aeroelastic equations of motion with aerodynamic forces applied to the F-4 wing and vertical fin. Fuselage contributions to generalized forces in elastic degrees of freedom were considered negligible. The local aerodynamic centers were considered to be located at the aerodynamic section quarter-chord for subsonic cases, and at mid-chord for supersonic cases.

e. Structural Forces

The total oscillatory displacement of an elastic structure can be obtained by superimposing the contributions of each of the elastic modes. Using a normal coordinate formulation, this is expressed as

$$Y(x,t) = \sum_i \Phi_i(x)\eta_i(t) \tag{1}$$

where $\Phi_i(x)$ and $\eta_i(t)$ represent mode shape and normal coordinate, respectively. The corresponding damped equation of motion is given by

$$\ddot{\eta}_i + 2\,\zeta_i\omega_i\dot{\eta}_i + \omega_i^2\eta_i = 0 \tag{2}$$

For each of the elastic modes considered in the analysis, the damping ratio ζ_i and the undamped circular frequency ω_i were obtained from ground vibration test logarithmic decrement data and resonant frequencies, respectively.

4. EQUATIONS OF MOTION

a. Rigid Body Equations of Motion

Referring to the stability axis coordinate system shown in Figure 118 the rigid body translation, roll, and yaw equations can be written, respectively, as

$$M_{11}(\ddot{q}_o + V\,\dot{q}_3) = Q_o(t) \tag{3}$$

$$I_{xx}\,\ddot{q}_2 - I_{xz}\,\ddot{q}_3 = Q_2(t) \tag{4}$$

and

$$-I_{xz}\,\ddot{q}_2 + I_{zz}\,\ddot{q}_3 = Q_3(t) \tag{5}$$

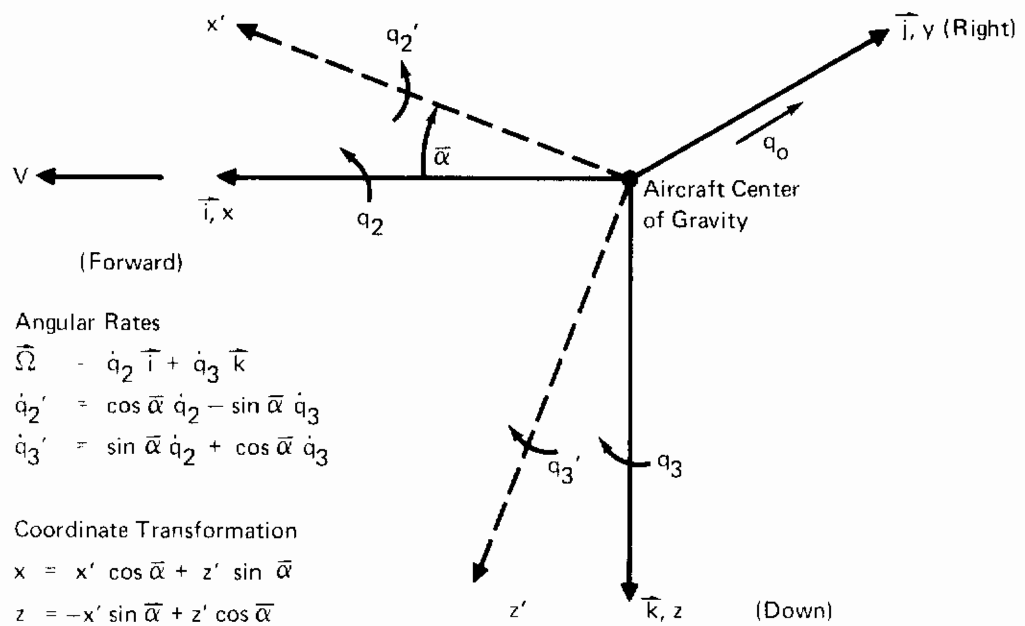


FIGURE 118

STABILITY AXIS COORDINATE SYSTEM
FOR LATERAL-DIRECTIONAL EQUATIONS OF MOTION

Dividing Equation (3) by $M_{11}V$ and replacing $\ddot{q}_o/V$ by $\dot{q}_1$, one obtains

$$\dot{q}_1 + \dot{q}_3 = Q_o'(t)/(M_{11}V) = Q_1'(t) \tag{6}$$

Solving Equations (4) and (5) for q_2 and $\ddot{q}_3$, one obtains

$$\ddot{q}_2 = \frac{I_{zz}\,Q_2(t) + I_{xz}\,Q_3(t)}{I_{xx}\,I_{zz} - (I_{xz})^2} = Q_2'(t) \tag{7}$$

and

$$\ddot{q}_3 = \frac{I_{xx}\,Q_3(t) + I_{xz}\,Q_2(t)}{I_{xx}\,I_{zz} - (I_{xz})^2} = Q_3'(t) \tag{8}$$

Equations (6) through (8) are in a conventional format with the generalized forces $Q_1'(t)$, $Q_2'(t)$ and $Q_3'(t)$ being defined in Paragraph 4.d.(1).

b. Flexible Airframe Equations of Motion

The equations of motion for airframe elastic degrees of freedom can be written as

$$\sum_j \{M_{ij}\,\ddot{q}_j + C_{ij}\,\dot{q}_j + K_{ij}\,q_j\} = Q_i(t) \tag{9}$$

where $\sum_j$ indicates summation over the j subscript. If the elastic modes are normal modes of vibration, then $M_{ij} = C_{ij} = K_{ij} = 0$ for $i \neq j$ where $i,j = 4,5,6$, and Equation (9) can be written as

$$\ddot{q}_i + 2\,\zeta_i\omega_i\dot{q}_i + \omega_i^{\,2}\,q_i = Q_i(t)/M_{ii} \tag{10}$$

As explained in Paragraph 3.e, ζ_i and ω_i represent the structural damping ratio and circular frequency, respectively, for the i^{th} mode at zero airspeed.

c. Aerodynamic Coefficients

(1) Sectioning

Aerodynamic sections were idealized as shown in Figure 119 to determine rigid surface aerodynamic coefficients utilizing the rigid model wind tunnel data of Reference 11. Accordingly, the F-4 wing and vertical fin planforms were sectioned as shown in Figures 120 and 121. Basic section data for the wing and vertical fin are tabulated in Table XLIV.

(2) Computation

The aerodynamic coefficients are based on rigid model wind tunnel measurements obtained for steady flow conditions. For low frequency unsteady motion, quasi-steady theory is applicable, permitting aerodynamic forces to be expressed as

$$[F_i] = -\bar{q}[\bar{A}_{ij}][q_j] \tag{11}$$

where $[\bar{A}_{ij}]$ is defined by Table XLV with $i,\ j = h,\ \alpha,\ \beta$.

Differentiating Equation (11) with respect to q_j and expanding, one obtains

$$\frac{\partial F_h}{\partial q_\alpha} = -2\;\bar{q}\;b_r\Delta y_r\;\frac{\partial C_L}{\partial \alpha} \tag{12}$$

$$\frac{\partial F_h}{\partial q_\beta} = -2\;\bar{q}\;b_r\Delta y_r\;\frac{\partial C_L}{\partial \beta} \tag{13}$$

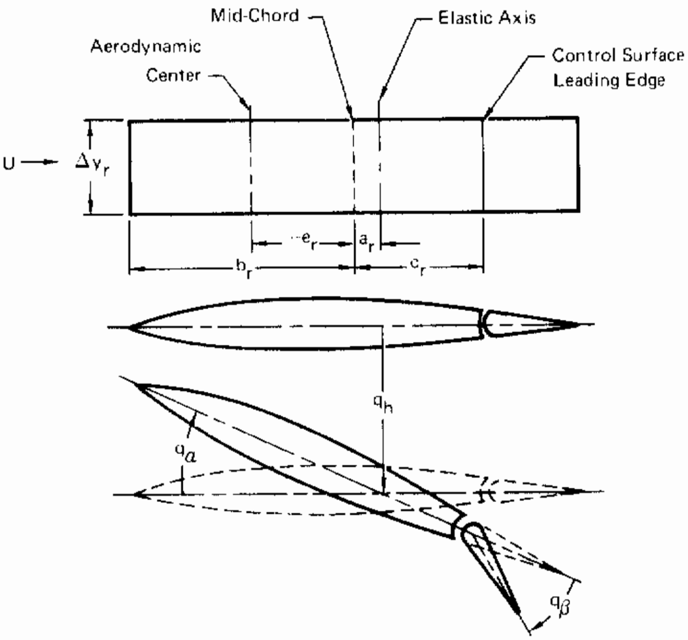


FIGURE 119

AERODYNAMIC SECTION IDEALIZATION FOR WING AND VERTICAL FIN

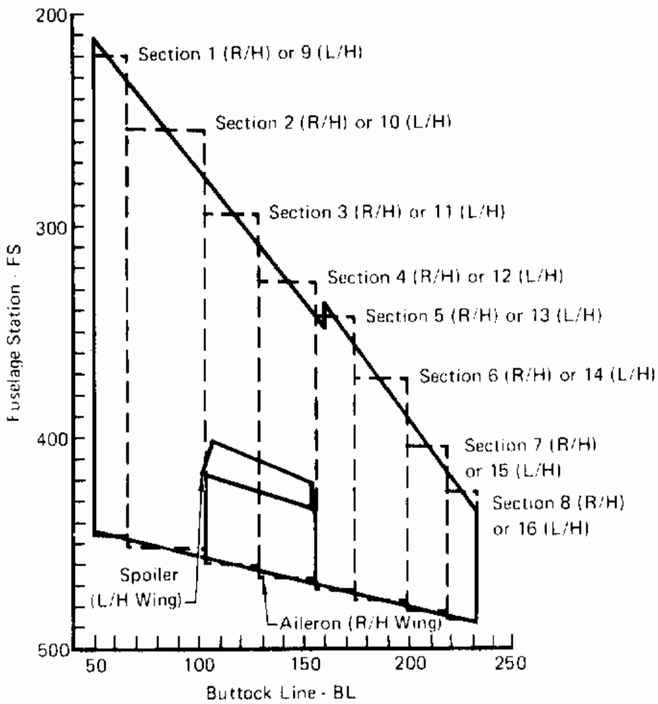


FIGURE 120

F-4 WING AERODYNAMIC SECTIONING

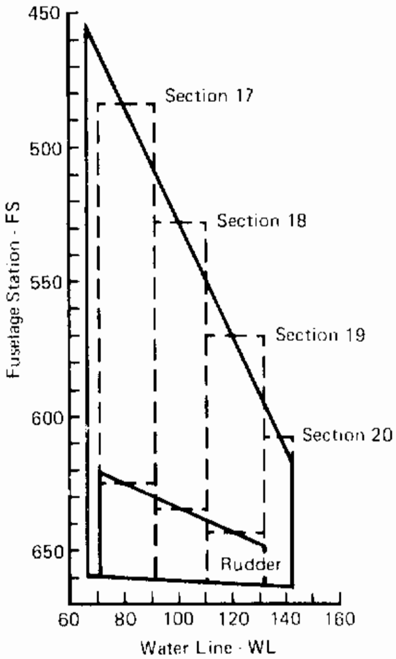


FIGURE 121

F-4 VERTICAL FIN AERODYNAMIC SECTIONING

TABLE XLIV
BASIC F-4 DATA FOR AERODYNAMIC SECTIONS

Section Number	b_r	a_r	e_r (1)	Δy_r	c_r
1	112.0	−112.5	−56.0	16.5	
2	98.5	−98.5	−49.3	37.2	
3	80.0	−80.0	−40.0	26.1	43.0
4	69.0	−69.0	34.5	26.1	34.0
5	63.8	−63.8	−31.9	18.1	
6	51.5	−51.5	−25.8	25.5	
7	39.3	39.3	−19.6	19.0	
8	29.5	−29.5	−14.8	14.0	
9	112.0	−112.0	−56.0	16.5	
10	98.5	−98.5	−49.3	37.2	
11	80.0	−80.0	−40.0	26.1	26.0
12	69.0	−69.0	−34.5	26.1	21.0
13	63.8	−63.8	−31.9	18.1	
14	51.5	−51.5	−25.8	25.5	
15	39.3	−39.3	−19.6	19.0	
16	29.5	29.5	−14.8	14.0	
17	88.0	−88.0	−44.0	20.0	52.0
18	67.0	−67.0	−33.5	20.0	39.0
19	46.0	46.0	−23.0	20.5	26.5
20	29.0	−29.0	14.5	11.0	

Note
(1) Values listed are for subsonic speeds only. All e_r values for supersonic speeds are equal to zero.

TABLE XLV
TABULATION OF ELEMENTS DEFINING THE [A] MATRIX

$\left[\bar{A}_{ij}\right] =$

i \ j	h	α	β
h	0	$2 b_r \Delta y_r \dfrac{\partial C_L}{\partial \alpha}$	$2 b_r \Delta y_r \dfrac{\partial C_L}{\partial \beta}$
α	0	$-2 b_r \Delta y_r (-e_r + a_r) \dfrac{\partial C_L}{\partial \alpha}$	$4 b_r^2 \Delta y_r \dfrac{\partial C_{MAC}}{\partial \beta}$ $-2 b_r \Delta y_r (-e_r + a_r) \dfrac{\partial C_L}{\partial \beta}$
β	0	$2 b_r^2 \Delta y_r \dfrac{\partial C_{H_\beta}}{\partial C_L} \dfrac{\partial C_L}{\partial \alpha}$	$4 b_r^2 \Delta y_r \dfrac{\partial C_{H_\beta}}{\partial \beta}$ $+2 b_r^2 \Delta y_r \dfrac{\partial C_{H_\beta}}{\partial C_L} \dfrac{\partial C_L}{\partial \beta}$

$$\frac{\partial F_\alpha}{\partial q_\beta} = -2\ \bar{q}\ b_r \Delta y_r\ [2b_r \frac{\partial C_{MAC}}{\partial \beta} - (a_r - e_r) \frac{\partial C_L}{\partial_\beta}] \tag{14}$$

$$\frac{\partial F_\beta}{\partial q_\beta} = -2\ \bar{q}\ b_r{}^2 \Delta y_r\ [2 \frac{\partial C_{H_\beta}}{\partial \beta} + \frac{\partial C_{H_\beta}}{\partial C_L} \frac{\partial C_L}{\partial \beta}] \tag{15}$$

$$\frac{\partial F_\beta}{\partial q_\alpha} = -2\ \bar{q}\ b_r{}^2 \Delta y_r \frac{\partial C_{H_\beta}}{\partial C_L} \frac{\partial C_L}{\partial \alpha} \tag{16}$$

After summing Equations (12) through (16) over appropriate sections using the sectional data of Table XLIV, and evaluating the left sides of the equations using the wind tunnel data of Reference 11, the equations were then utilized to solve for average sectional values of the coefficients $\partial C_L/\partial \alpha$, $\partial C_L/\partial \beta$, $\partial C_{H_\beta}/\partial C_L$, $\partial C_{H_\beta}/\partial \beta$, and $\partial C_{MAC}/\partial \beta$.

(3) Results

Average sectional values of the required aerodynamic coefficients for the wing-aileron, wing-spoiler, and vertical fin-rudder are summarized in Tables XLVI, XLVII, and XLVIII, respectively.

TABLE XLVI

SUMMARY OF RIGID AERODYNAMIC COEFFICIENTS FOR F-4 WING AND AILERON

Mach Number	$\frac{\partial C_L}{\partial \alpha}$	$\frac{\partial C_L}{\partial \beta}$	$\frac{\partial C_{H_\beta}}{\partial C_L}$	$\frac{\partial C_{H_\beta}}{\partial \beta}$	$\frac{\partial C_{MAC}}{\partial \beta}$
0.5	3.1154	1.5876	0.0157	0.0235	0.3684
0.9	3.6167	1.4112	0.0212	0.0264	0.6364
1.2	3.6526	1.1937	0.0379	0.0377	0.4494
1.5	2.9722	0.6468	0.0434	0.0368	0.3856
1.8	2.2381	0.5586	0.0510	0.0307	0.3641
2.15	2.0053	0.5527	0.0475	0.0254	0.3518

TABLE XLVII

SUMMARY OF RIGID AERODYNAMIC COEFFICIENTS FOR F-4 WING AND SPOILER

Mach Number	$\frac{\partial C_L}{\partial \alpha}$	$\frac{\partial C_L}{\partial \beta}$	$\frac{\partial C_{H_\beta}}{\partial C_L}$	$\frac{\partial C_{H_\beta}}{\partial \beta}$	$\frac{\partial C_{MAC}}{\partial \beta}$
0.5	3.1154	0.4246	0.0	0.0048	−0.006
0.9	3.6167	0.4125	0.0	0.0068	0.1682
1.2	3.6526	0.3882	0.0	0.0109	0.0729
1.5	2.9722	0.2912	0.0	0.0091	0.0771
1.8	2.2381	0.2729	0.0	0.0076	0.0860
2.15	2.0053	0.2729	0.0	0.0063	0.0977

TABLE XLVIII

SUMMARY OF RIGID AERODYNAMIC COEFFICIENTS FOR F-4 VERTICAL FIN AND RUDDER

Mach Number	$\frac{\partial C_L}{\partial \alpha}$	$\frac{\partial C_L}{\partial \beta}$	$\frac{\partial C_{H_\beta}}{\partial C_L}$	$\frac{\partial C_{H_\beta}}{\partial \beta}$	$\frac{\partial C_{MAC}}{\partial \beta}$
0.5	2.0048	1.2709	0.0041	0.0088	0.5189
0.9	2.0048	1.1218	0.0041	0.0091	0.4186
1.2	2.5059	0.8592	0.0071	0.0297	0.1040
1.5	3.2076	0.7697	0.0153	0.0212	0.0384
1.8	2.8568	0.6802	0.0166	0.0156	0.0977
2.15	2.3055	0.5609	0.0162	0.0093	0.0836

d. Generalized Forces

(1) Generalized Forces for Rigid Aircraft Equations of Motion

The generalized forces Q_1', Q_2', and Q_3' for the rigid body equations of motion including both static aeroelastic and inertial contributions are presented below. They are defined as

$$Q_1' = (YP)\dot{q}_2 + (YB)q_1 + (YPHI)q_2 + (YDA)q_7 + (YR)\dot{q}_3 + \dot{q}_3$$
$$+ (YDA)q_7 + (YDR)q_8 \tag{17}$$

$$Q_2' = (LP)\dot{q}_2 + (LR)\dot{q}_3 + (LB)q_1 + (LDA)q_7 + LDR\ q_8 \tag{18}$$

and

$$Q_3' = (NP)\dot{q}_2 + (NR)\dot{q}_3 + (NB)q_1 + (NDA)q_7 + NDR\ q_8 \tag{19}$$

The coefficients including static aeroelastic effects are based on data presented in Reference 11, and are those normally used in rigid body analysis for the F-4 aircraft.

(2) Generalized Forces for Aeroelastic Equations of Motion

When using unsteady incompressible potential flow theory, the generalized aerodynamic forces $Q_i(t)$ are generally given in the form

$$Q_i(t) = -\bar{q} \sum_j [(A_{S_{ij}} + A_{C_{ij}}\ C(k))q_j + (B_{S_{ij}} + B_{C_{ij}}\ C(k))\dot{q}_j/U] \tag{20}$$

The term $C(k)$ is the Theodorsen function which depends on the reduced frequency $k = \omega b/U$, and is, in general, a complex number. In order to keep the equations of motion in a mathematically tractable form and to permit the assessment of system dynamic characteristics at arbitrary airspeeds, quasi-steady theory has been introduced. In this theory, $C(k) = 1.0$, permitting Equation (20) to be written for any Mach number as

$$Q_i(t) = -\bar{q} \sum_j [A_{ij}\ q_j + B_{ij}\ \dot{q}_j/U] \tag{21}$$

where A_{ij} and B_{ij} represent, respectively, the aerodynamic spring and damping matrices obtained in each case by summing noncirculatory and circulatory contributions. The A_{ij} and B_{ij} matrices were computed using the control surface coefficients presented in Tables XLVI, XLVII, and XLVIII.

e. Equations of Motion

Combining Equations (6) through (8) with Equations (17) through (19), respectively, gives the rigid body equations of motion. Similarily, combining Equation (10) with Equation (21) gives the elastic degree of freedom equations. These equations can be compactly summarized in matrix notation as

$$\begin{bmatrix} \ddot{q}_1 \\ \ddot{q}_2 \\ \ddot{q}_3 \\ \ddot{q}_4 \\ \ddot{q}_5 \\ \ddot{q}_6 \end{bmatrix} = [C] \begin{bmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \\ \dot{q}_4 \\ \dot{q}_5 \\ \dot{q}_6 \\ \dot{q}_7 \\ \dot{q}_8 \end{bmatrix} + [K] \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \\ q_5 \\ q_6 \\ q_7 \\ q_8 \end{bmatrix} \tag{22}$$

where the [C] and [K] matrices are defined by Tables XLIX and L, respectively. The generalized coordinates q_1 through q_6 represent the system degrees of freedom forced by the independent variables q_7 and q_8.

f. Rate and Acceleration Equations

(1) Lateral Acceleration Equation

The total rigid airframe velocity at an arbitrary point can be written as

$$\vec{V}_T = V\vec{i} + (x\,\dot{q}_3 - z\dot{q}_2 + \dot{q}_o)\,\vec{j} \tag{23}$$

Differentiating with respect to time, the rigid body lateral acceleration with respect to inertial reference is

$$N_y{}^R = V\,(\dot{q}_1 + \dot{q}_3) + x\,\ddot{q}_3 - z\,\ddot{q}_2 \tag{24}$$

The acceleration due to elastic degrees of freedom is

$$N_y{}^E = \sum_{i=4}^{6} \frac{\partial h_T}{\partial q_i}\,\ddot{q}_i \tag{25}$$

Combining rigid and elastic degree of freedom accelerations, the total acceleration with respect to inertial reference is

$$N_y{}^T = V\,(\dot{q}_1 + \dot{q}_3) + x\ddot{q}_3 - z\ddot{q}_2 + \sum_{i=4}^{6} \frac{\partial h_T}{\partial q_i}\,\ddot{q}_i \tag{26}$$

TABLE XLIX
TABULATION OF ELEMENTS DEFINING THE [C] MATRIX FOR THE LATERAL-DIRECTIONAL EQUATIONS OF MOTION

$[C_{ij}]=$

0.0	YP	YR	0.0	0.0	0.0	0.0	0.0
0.0	LP	LR	0.0	0.0	0.0	0.0	0.0
0.0	NP	NR	0.0	0.0	0.0	0.0	0.0
0.0	$\frac{-\bar{q}\,B_{42}}{U\,M_{44}}$	$\frac{-\bar{q}\,B_{43}}{U\,M_{44}}$	$-2\zeta_4\omega_4 \frac{-\bar{q}\,B_{44}}{U\,M_{44}}$	$\frac{-\bar{q}\,B_{45}}{U\,M_{44}}$	$\frac{-\bar{q}\,B_{46}}{U\,M_{44}}$	$\frac{-\bar{q}\,B_{47}}{U\,M_{44}}$	$\frac{-\bar{q}\,B_{48}}{U\,M_{44}}$
0.0	$\frac{-\bar{q}\,B_{52}}{U\,M_{55}}$	$\frac{-\bar{q}\,B_{53}}{U\,M_{55}}$	$\frac{-\bar{q}\,B_{54}}{U\,M_{55}}$	$-2\zeta_5\omega_5 \frac{-\bar{q}\,B_{55}}{U\,M_{55}}$	$\frac{-\bar{q}\,B_{56}}{U\,M_{55}}$	$\frac{-\bar{q}\,B_{57}}{U\,M_{55}}$	$\frac{\bar{q}\,B_{58}}{U\,M_{55}}$
0.0	$\frac{-\bar{q}\,B_{62}}{U\,M_{66}}$	$\frac{-\bar{q}\,B_{63}}{U\,M_{66}}$	$\frac{-\bar{q}\,B_{64}}{U\,M_{66}}$	$\frac{-\bar{q}\,B_{65}}{U\,M_{66}}$	$2\zeta_6\omega_6 \frac{-\bar{q}\,B_{66}}{U\,M_{66}}$	$\frac{\bar{q}\,B_{67}}{U\,M_{66}}$	$\frac{-\bar{q}\,B_{68}}{U\,M_{66}}$

TABLE L
TABULATION OF ELEMENTS DEFINING THE [K] MATRIX FOR THE LATERAL-DIRECTIONAL EQUATIONS OF MOTION

$[K_{ij}]=$

YB	YPHI	0.0	0.0	0.0	0.0	YDA	YDR
LB	0.0	0.0	0.0	0.0	0.0	LDA	LDR
NB	0.0	0.0	0.0	0.0	0.0	NDA	NDR
$\frac{-V\,\bar{q}\,B_{40}}{U\,M_{44}}$	0.0	0.0	$-\omega_4^2 \frac{-\bar{q}\,A_{44}}{M_{44}}$	$\frac{-\bar{q}\,A_{45}}{M_{44}}$	$\frac{-\bar{q}\,A_{46}}{M_{44}}$	$\frac{\bar{q}\,A_{47}}{M_{44}}$	$\frac{-\bar{q}\,A_{48}}{M_{44}}$
$\frac{-V\,\bar{q}\,B_{50}}{U\,M_{55}}$	0.0	0.0	$\frac{-\bar{q}\,A_{54}}{M_{55}}$	$-\omega_5^2 \frac{-\bar{q}\,A_{55}}{M_{55}}$	$\frac{-\bar{q}\,B_{56}}{M_{55}}$	$\frac{-\bar{q}\,B_{57}}{M_{55}}$	$\frac{-\bar{q}\,B_{58}}{M_{55}}$
$\frac{-V\,\bar{q}\,B_{60}}{U\,M_{66}}$	0.0	0.0	$\frac{-\bar{q}\,A_{64}}{M_{66}}$	$\frac{-\bar{q}\,A_{65}}{M_{66}}$	$-\omega_6^2 \frac{-\bar{q}\,A_{66}}{M_{66}}$	$\frac{-\bar{q}\,A_{67}}{M_{66}}$	$\frac{-\bar{q}\,A_{68}}{M_{66}}$

The acceleration due to aerodynamic forces alone is obtained by subtracting the gravitational contribution to the total acceleration. Thus,

$$N_y^A = V\,(\dot{q}_1 + \dot{q}_3) + x\,\ddot{q}_3 - z\,\ddot{q}_2 + \sum_{i=4}^{6} \frac{\partial h_T}{\partial q_i} - V(YPHI)q_2 \qquad (27)$$

Equation (27) provides the basis for computing lateral acceleration as sensed by the F-4 lateral-directional flight control system.

(2) Flexible Aircraft Roll and Yaw Rates

After superimposing rigid and elastic contributions, the total aircraft roll and yaw rates are given, respectively, by

$$\dot{\Phi}_T = \cos\bar{\alpha}\dot{q}_2 - \sin\bar{\alpha}\,\dot{q}_3 + \sum_{i=4}^{6} \frac{\partial \Phi_T}{\partial q_i}\,\dot{q}_i \qquad (28)$$

and

$$\dot{\psi}_T = \sin\bar{\alpha}\dot{q}_2 + \cos\bar{\alpha}\,\dot{q}_3 + \sum_{i=4}^{6} \frac{\partial \psi_T}{\partial q_i}\,\dot{q}_i \qquad (29)$$

These equations provide the basis for computing roll and yaw rates as sensed by on-board gyros.

g. Transfer Function Computation

After taking the Laplace transform of Equations (22), (27), (28), and (29) with initial conditions set to zero and replacing the Laplace variables by $i\omega$, the transfer functions $\dot{\Phi}_T(\omega)/q_7$, $\dot{\psi}_T(\omega)/q_8$, and $N_Y^A(\omega)/q_8$ can be computed directly.

h. Data Package

(1) General Applicability of Stability Derivatives

Stability derivatives were computed for the 41,000 pound combat-gear up configuration using the inertial characteristics presented in Reference 12. Pertinent inertia data are summarized in Table LI. Since the airframe dynamic characteristics do not change significantly with variation in internal fuel for the elastic modes considered, the stability derivatives presented herein for elastic degrees of freedom are applicable for any internal fuel configuration. Changes in mass, inertia, and center of gravity location are reflected in the effective aeroelastic stability derivatives for the rigid body equations of motion. These stability derivatives are indicated symbolically by the [C] and [K] matrices of Equation (22).

TABLE LI

AIRCRAFT INERTIAL CHARACTERISTICS FOR LATERAL-DIRECTIONAL STABILITY DERIVATIVES

Combat - Gear Up

Weight: 41,001 Lb

Center of Gravity Location	
Horizontal:	Fuselage Station 315.61
Vertical:	Water Line 26.50
Roll Moment of Inertia:	24,873 Slug-Ft^2
Yaw Moment of Inertia:	169,824 Slug-Ft^2
Product of Inertia:	4,800 Slug-Ft^2

(2) Stability Derivatives

The stability derivatives for the lateral-directional equations of motion are presented in Supplement 2. Table LII provides definitions of the [C] and [K] matrices using Supplement 2 notation.

(3) Sensor Feedback Derivatives

Figures 122 through 126 present curves of sensor feedback derivatives versus aircraft fuselage station which are utilized with Equations (27), (28), and (29) to compute total airframe response. It should be noted that the distance x of Equation (27) must be consistent with the fuselage station location at which sensor feedback derivatives are read.

TABLE LII

DEFINITION OF [C] AND [K] MATRICES FOR LATERAL-DIRECTIONAL EQUATIONS OF MOTION IN TERMS OF APPENDIX V NOTATION

$$
\left[C_{ij}\right] = \begin{bmatrix}
0.0 & Y_p & Y_r & 0.0 & 0.0 & 0.0 & 0.0 & 0.0 \\
0.0 & L_p & L_r & 0.0 & 0.0 & 0.0 & 0.0 & 0.0 \\
0.0 & N_p & N_r & 0.0 & 0.0 & 0.0 & 0.0 & 0.0 \\
0.0 & A_p & A_r & A_{\dot{\eta}_4} & A_{\dot{\eta}_5} & A_{\dot{\eta}_6} & A_{\dot{\delta}} & A_{\dot{\delta}_R} \\
0.0 & B_p & B_r & B_{\dot{\eta}_4} & B_{\dot{\eta}_5} & B_{\dot{\eta}_6} & B_{\dot{\delta}} & B_{\dot{\delta}_R} \\
0.0 & C_p & C_r & C_{\dot{\eta}_4} & C_{\dot{\eta}_5} & C_{\dot{\eta}_6} & C_{\dot{\delta}} & C_{\dot{\delta}_R}
\end{bmatrix}
$$

$$
\left[K_{ij}\right] = \begin{bmatrix}
Y_\beta & Y_\phi & 0.0 & 0.0 & 0.0 & 0.0 & Y_\delta & Y_{\delta_R} \\
L_\beta & 0.0 & 0.0 & 0.0 & 0.0 & 0.0 & L_\delta & L_{\delta_R} \\
N_\beta & 0.0 & 0.0 & 0.0 & 0.0 & 0.0 & N_\delta & N_{\delta_R} \\
A_\beta & 0.0 & 0.0 & A_{\eta_4} & A_{\eta_5} & A_{\eta_6} & A_\delta & A_{\delta_R} \\
B_\beta & 0.0 & 0.0 & B_{\eta_4} & B_{\eta_5} & B_{\eta_6} & B_\delta & B_{\delta_R} \\
C_\beta & 0.0 & 0.0 & C_{\eta_4} & C_{\eta_5} & C_{\eta_6} & C_\delta & C_{\delta_R}
\end{bmatrix}
$$

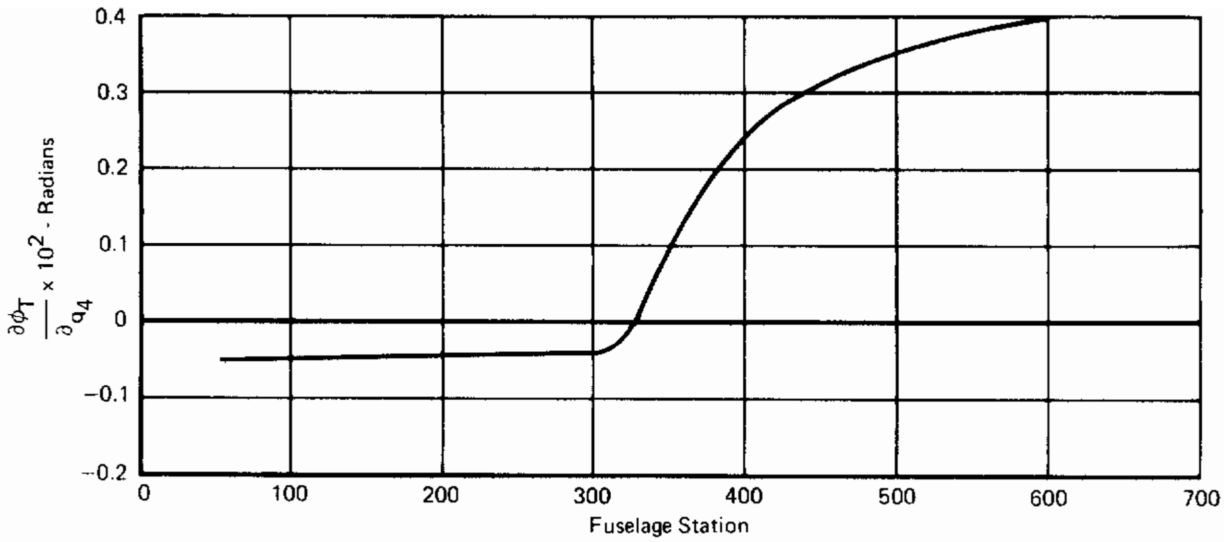


FIGURE 122
LATERAL-DIRECTIONAL SENSOR FEEDBACK DERIVATIVES

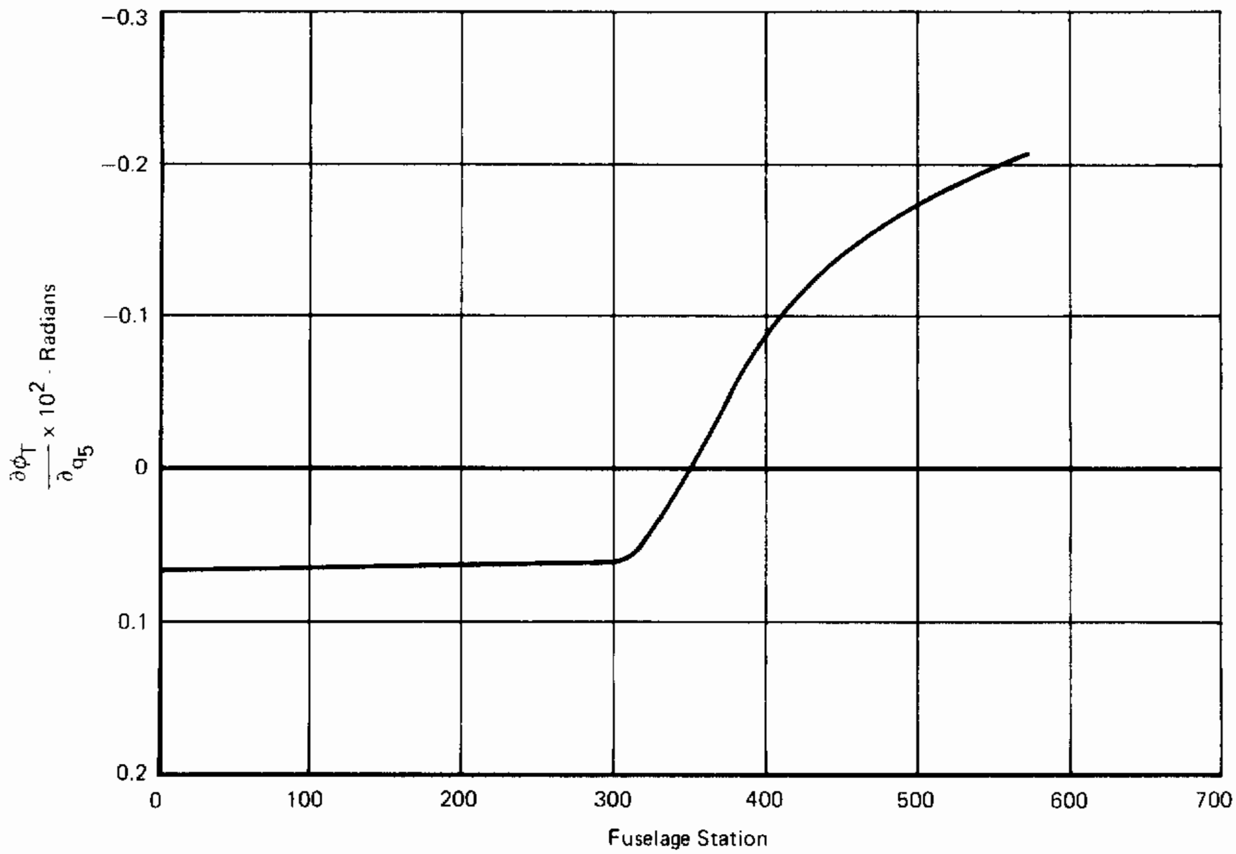


FIGURE 123
LATERAL-DIRECTIONAL SENSOR FEEDBACK DERIVATIVES

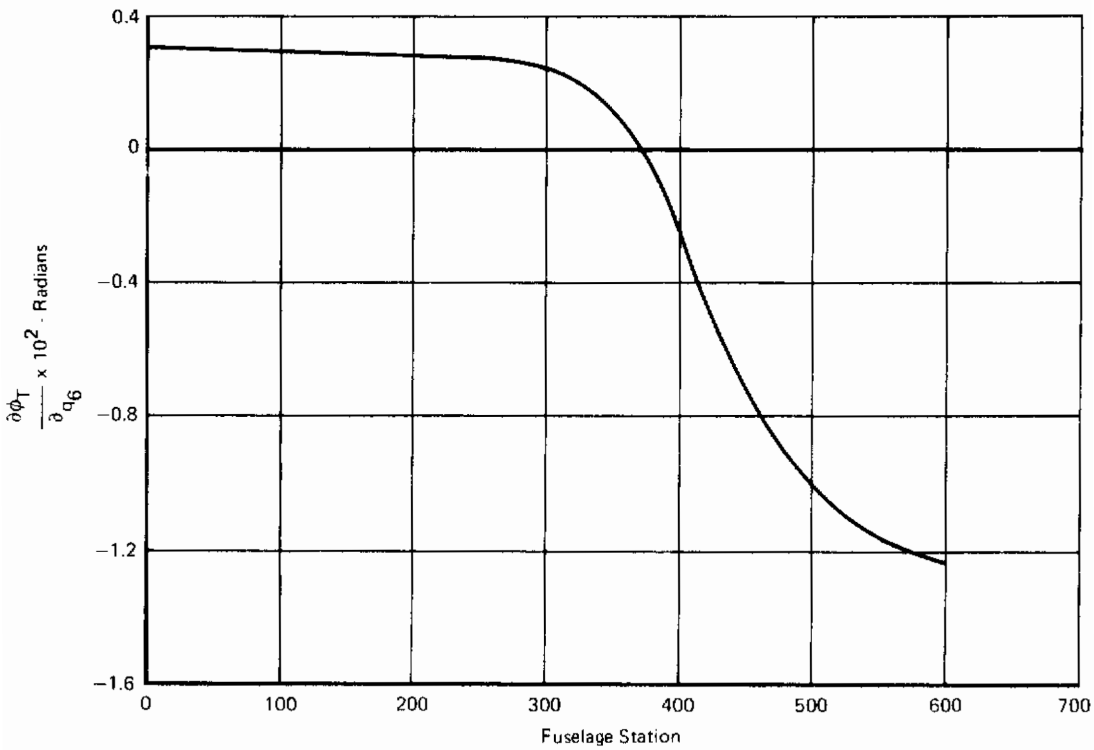


FIGURE 124

LATERAL-DIRECTIONAL SENSOR FEEDBACK DERIVATIVES

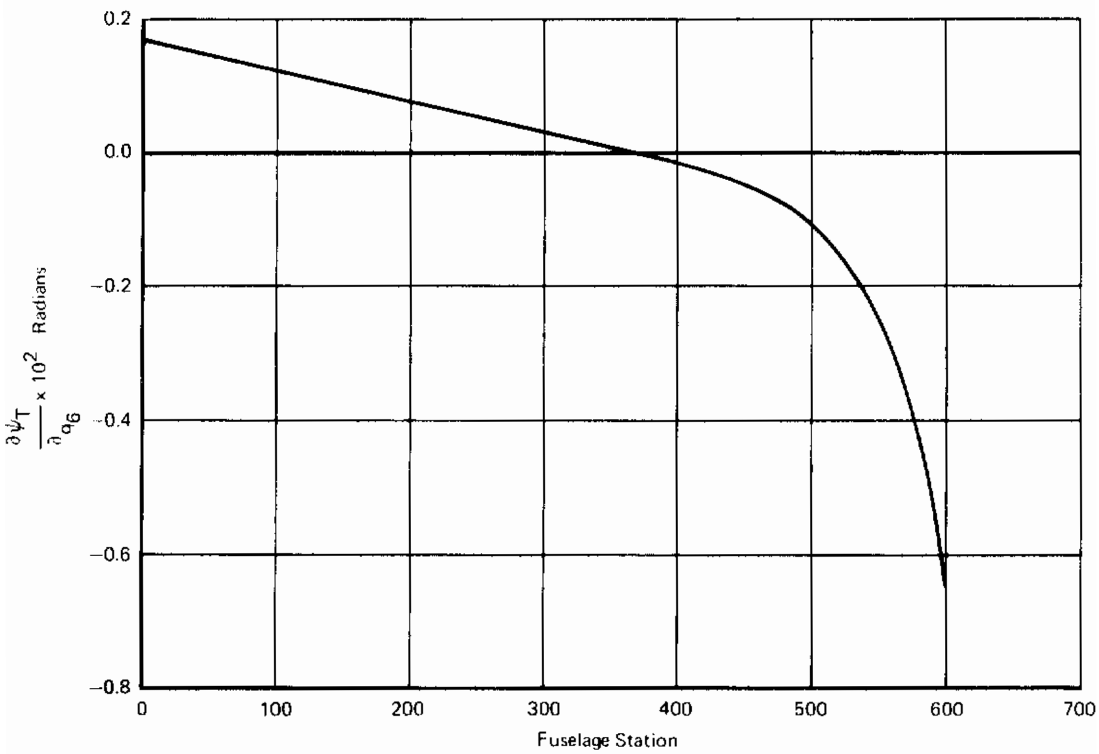


FIGURE 125

LATERAL-DIRECTIONAL SENSOR FEEDBACK DERIVATIVES

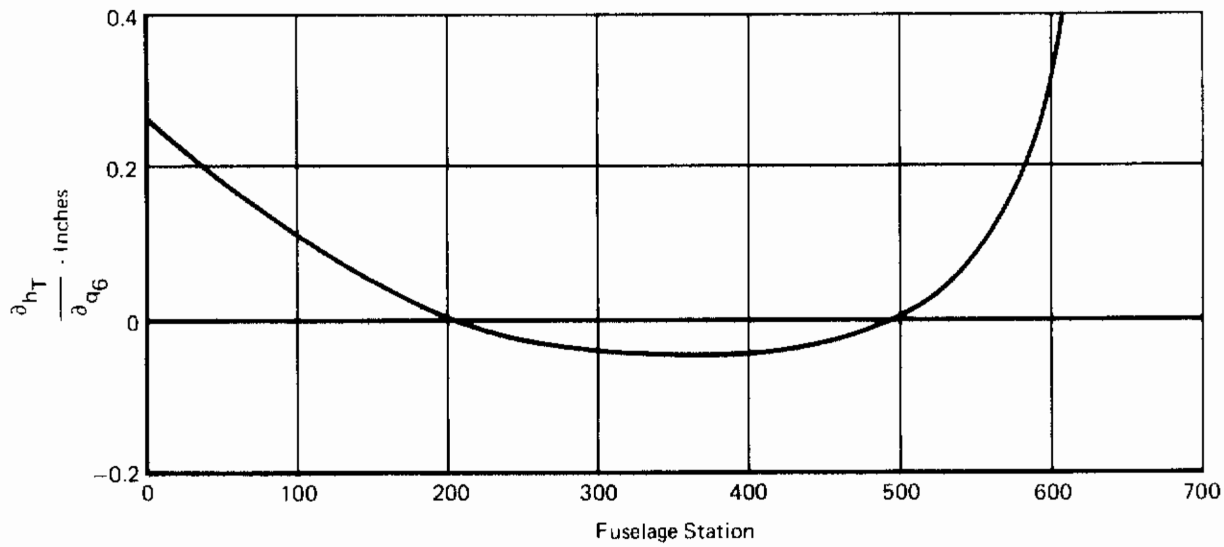


FIGURE 126
LATERAL-DIRECTIONAL SENSOR FEEDBACK DERIVATIVES

5. TRANSFER FUNCTION RESULTS

a. Low Frequency Characteristics

The low frequency response characteristics of transfer functions computed using data presented herein agree well with those normally obtained in a rigid body analysis using effective aeroelastic coefficients. This is to be expected, since the contribution due to wing and fuselage elastic modes is relatively small at low frequencies.

b. High Frequency Characteristics

The response characteristics at high frequencies are governed by the dynamics of the elastic modes included in the analysis. Confidence in the results obtained is based on having utilized good definitions of both airframe structural modes and aerodynamic forcing functions obtained from measured vibration and aerodynamic data.

6. DISCUSSION

a. Effective Aeroelastic Versus Rigid Coefficients

Providing that adequate frequency separation between rigid body and elastic modes exist, effective aeroelastic coefficients may be utilized in the rigid body equations of motion in lieu of considering elastic airframe degrees of freedom. This approach also requires including static aeroelastic effects in control surface forcing coefficients for the rigid body equation of motion. Total airframe response is then obtained by combining rigid and elastic contributions, where elastic responses are calculated using aerodynamic lift distributions based on rigid surface aerodynamic coefficients. However, should strong coupling between aircraft rigid body and structural modes exist, it is necessary for modal coefficients to retain their identity in the equations of motion so that correct system eigenvalues and responses can be computed. Thus, airframe flexibility can be accounted for by including elastic coupling effects in rigid body equations of motion and using rigid control surface inputs, or by lumping elastic effects in a single effective control surface coefficient. Previous experience with the longitudinal control system indicated that both approaches yield essentially identical results.

b. Mathematical Approach

As in the case of the longitudinal stability derivative analysis presented in Appendix V, two approaches were possible: quasi-steady and indicial lift. Here, again, the quasi-steady approach was chosen for mathematical expediency.

c. Primary Surface Aerodynamic Lift Coefficients

Lift coefficients for primary surfaces such as the wing and vertical fin are generally based on wind tunnel data obtained by varying the rigid model angle of attack and angle of yaw rather than that of the surface alone. This type of wind tunnel data does not permit the separation of incremental lift due to lifting surface and fuselage. The result is that the primary surface coefficients include fuselage effects which make these coefficients larger than would be obtained for a rigid surface alone. It would be desirable, therefore, to obtain incremental life coefficients based on wind tunnel data obtained by varying primary surface angle of attack relative to the aircraft fuselage. Incremental lift coefficients so established would be due to the primary surface alone in the flow field of the aircraft fuselage and would be more direct-y applicable to aeroelastic derivative computation.

7. CONCLUSION

The general approach used for computing lateral-directional stability derivatives has been treated in detail. The resulting stability derivatives are presented in Supplement 2 and were utilized for Phase II control system design and analysis.

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS FOR APPENDIX VI

ABBREVIATIONS:

BL - Buttock Line

WL - Water Line

SYMBOLS:

$[\bar{A}_{ij}]$	Aerodynamic coefficient matrix for rigid section, defined per Table XLV
A_{ij}	$A_{S_{ij}} + A_{C_{ij}}$, Summation of circulatory and noncirculatory aerodynamic stiffness derivatives
$A_{C_{ij}}$	Generalized circulatory aerodynamic stiffness derivative
a_r	Distance from mid-chord to elastic axis for r^{th} section, positive aft - in
$A_{S_{ij}}$	Generalized noncirculatory aerodynamic stiffness derivative
b	Aerodynamic section semi-chord
B_{ij}	$B_{C_{ij}} + B_{S_{ij}}$, Summation of circulatory and noncirculatory aerodynamic damping derivatives
$B_{C_{ij}}$	Generalized circulatory aerodynamic damping derivative
b_r	Semi-chord of r^{th} aerodynamic section - in
$B_{S_{ij}}$	Generalized noncirculatory aerodynamic damping derivative
$[C]$	System damping matrix, defined per Table XLIX
C_{ij}	Structural damping coefficient
$C(k)$	Theodorsen function
C_L	Lift coefficient
c_r	Distance of control surface hinge line aft of mid-chord for r^{th} aerodynamic section
e_r	Distance from mid-chord to aerodynamic center for r^{th} section, positive aft - in
F	Force
F_h	Aerodynamic force normal to rigid airfoil section, positive down or left

F_{α}	Aerodynamic moment about rigid airfoil section elastic axis, positive leading edge up or left
F_{β}	Aerodynamic moment about rigid control surface hinge line, positive leading edge up or right
i	Index of summation
I_{xx}	Aircraft roll inertia about center of gravity, defined as $\int(y^2 + z^2)dm$
I_{xz}	Aircraft product of inertia, defined as $\int(xz)dm$
I_{zz}	Aircraft yaw inertia about center of gravity, defined as $\int(x^2 + y^2)dm$
[K]	System spring matrix, defined per Table L
k	Reduced frequency, defined as $\omega b/U$
K_{ij}	Structural spring coefficient
LDA	Effective aeroelastic aileron - spoiler coefficient in roll equation of motion
LDR	Effective aeroelastic rudder coefficient in roll equation of motion
LP	Effective aeroelastic coefficient for $\dot{q}_2$ in rigid body roll equation of motion
LR	Effective aeroelastic coefficient for $\dot{q}_3$ in rigid body roll equation of motion
Lβ	Effective aeroelastic coefficient for q_1 in roll equation of motion
M_{ij}	Generalized mass
NB	Effective aeroelastic coefficient for q_1 in yaw equation of motion
NDA	Effective aeroelastic aileron - spoiler coefficient in yaw equation of motion
NDR	Effective aeroelastic rudder coefficient in yaw equation of motion
NP	Effective aeroleastic coefficient for $\dot{q}_2$ in rigid body yaw equation of motion
NR	Effective aeroelastic coefficient for $\dot{q}_3$ in yaw equation of motion

N_y^E	Acceleration due to elastic degrees of freedom
N_y^R	Rigid airframe lateral acceleration
N_y^T	Total acceleration with respect to inertial reference
$Q_i(t)$ or Q_i	Generalized force for i^{th} equation of motion
$Q_o(t)$ or Q_o	Generalized force for aircraft lateral translation, positive right
$Q_2(t)$ or Q_2	Generalized moment for aircraft roll, positive right wing down
$Q_3(t)$ or Q_3	Generalized moment for aircraft yaw, positive nose right
$Q_1'(t)$ or Q_1'	Defined as $Q_o/(M_{11}V_T)$
$Q_2'(t)$ or Q_2'	Defined as $(I_{zz}\ Q_z + I_{xz}Q_3)/(I_{xx}I_{zz} - I_{xz}{}^2)$
$Q_3'(t)$ or Q_3'	Defined as $(I_{xx}Q_3 + I_{xz}Q_2)/(I_{xx}I_{zz} - I_{xz}{}^2)$
q_i	Generalized coordinate for the i^{th} degree of freedom, where $i = 4$, through 6
q_h	Rigid translation of aerodynamic section elastic axis in inches, positive down or positive left
q_α	Rigid pitch or yaw of aerodynamic section in radians, positive, respectively, leading edge up or right
q_β	Rigid control surface angular rotation relative to section chord in radians, positive trailing edge down or left
q_j	Generalized coordinate for j^{th} degree of freedom, where $j = 0$, through 8
q_o	Rigid aircraft lateral translation - inches, positive right
q_1	Defined as $\dot{q}_o/V$, where the dot denotes differentiation with respect to time and V the true airspeed
q_2	Rigid aircraft roll - radians, positive right wing down
q_3	Rigid aircraft yaw - radians, positive nose right
q_2'	Defined by Figure 118
q_3'	Defined by Figure 118
q_7	Rigid aileron-spoiler control input referenced to aileron rotating, positive for right wing down rolling moment - rad

q_8	Rigid rudder rotation about hinge line - radians, positive trailing edge left
U	True airspeed
V_T	Total rigid airframe velocity
x	Distance forward of aircraft center of gravity in the stability axis coordinate system ($=x' \cos\bar{\alpha} + z' \sin\bar{\alpha}$)
x'	Spacial coordinate along aircraft Fuselage Station (FS) axis, positive forward of aircraft center of gravity
y	Lateral distance from aircraft center of gravity in the stability axis coordinated system, positive right ($=y'$)
y'	Spacial coordinate along aircraft Buttock Line (BL) axis, positive right
Y(x,t)	Total oscillatory displacement of an elastic structure
YB	Effective aeroelastic coefficient for q_1 in rigid body lateral equation of motion
YDA	Combined effective aeroelastic aileron - spoiler coefficient in rigid body lateral equation of motion
YDR	Effective aeroelastic rudder coefficient in rigid body lateral equation of motion
YP	Effective aeroelastic coefficient for $\dot{q}_2$ in rigid body lateral equation of motion
YPHI	Effective aeroelastic coefficient for q_2 in rigid body lateral equation of motion ($=g/v$)
YR	Effective aeroelastic coefficient for $\dot{q}_3$ in rigid body lateral equation of motion
Δy_r	Width of r^{th} aerodynamic section - in
z	Vertical distance from aircraft center of gravity in the stability axis coordinate system positive down ($= -x' \sin\bar{\alpha} + z' \cos\bar{\alpha}$)
Z'	Spacial coordinate along aircraft Water Line (WL) axis, positive down from aircraft center of gravity
α	Incremental angle of attack for rigid aerodynamic section - rad
$\bar{\alpha}$	Angle between aircraft forward velocity and longitudinal axis, positive nose up

∂	Denotes partial differentiation
$\partial C_L/\partial\alpha$	Wing lift coefficient per unit wing angle of attack
$\partial C_L/\partial\beta$	Wing lift coefficient per unit control surface rotation
$\partial C_{MAC}/\partial\beta$	Noncirculatory moment coefficient about the aerodynamic center per unit control surface rotation
$\partial C_{H_\beta}/\partial\beta$	Noncirculatory control surface hinge moment coefficient per unit surface rotation
$\partial C_{H_\beta}/\partial C_L$	Coefficient relating control surface circulatory hinge moment to wing lift
$\partial h_T/\partial_{q_i}$	Lateral bending mode shape for i^{th} mode at x' inches forward of aircraft center of gravity - inches, positive right
$\partial\Phi_T/\partial_{q_i}$	Roll slope for i^{th} mode at x' inches forward of aircraft center of gravity - radians, positive right wing down
$\partial\psi_T/(x')/\partial_{q_i}$	Yaw slope for i^{th} mode at x' inches forward of aircraft center of gravity - radians, positive nose right
β	Incremental rigid control surface excursion - rad
ζ_i	Damping ratio for i^{th} elastic mode
$\eta_i(t)$	Normal coordinate for i^{th} elastic mode
$\Phi_i(x)$	Mode shape for i^{th} elastic mode
Φ_T	Total airframe roll rate
$\dot{\psi}_T$	Total airframe yaw rate
$\bar{\Omega}$	Rigid airframe angular rate
ω_i	Undamped natural frequency for i^{th} elastic mode - rad/sec
$\sum\limits_i$	Denotes summation over subscript i
$\vec{i},\vec{j},\vec{k}$	Unit vectors defined in Figure 118

APPENDIX VII

SIDE STICK CONTROLLER DEVELOPMENT STUDY

1. INTRODUCTION

This Side Stick Controller (SSC) development study established the design and performance criteria which were incorporated into the SSC Procurement Specification for the SFCS. Use was made of existing side-stick technology in a concerted effort to obtain the simplest design which would meet the requirements of the SFCS. Thorough evaluations of current SSC designs were made and previous Air Force SSC programs were reviewed. Cockpit mockups were constructed and evaluations were conducted. MCAIR pilots received practical SSC experience by flying aircraft which had SSC's installed.

The following paragraphs describe various parts of the study and give results in terms of SSC design and performance criteria.

A list of specialized abbreviations and symbols is found at the end of this appendix.

2. CURRENT SSC DESIGNS

The SSC evaluation included a review of the available SSC data to determine applicability to the SFCS program. Items and features relevant to system needs were selected and refined for incorporation into the SSC design for the SFCS. The following paragraphs provide a summary of the individual evaluations.

a. USAF B-47 Fly-By-Wire Aircraft

The USAF B-47 Fly-By-Wire program was conducted at Wright-Patterson Air Force Base. An SSC was utilized to provide command inputs to the pitch and roll axes of the aircraft.

(1) General Description

The SSC is mounted under the left sill. Removing an unlocking pin allows the SSC to extend inboard to a position under the crew member's left arm. When in the extended position the SSC is in the ejection envelope and it must be manually stowed before ejection. The left forearm is supported by an adjustable tray, lined with thin rubber padding. Pitch and roll control verniers are located on the top of the grip and an engage-disengage button is located on the inboard side of the grip. The throttles are located on the right hand console.

(2) Flight Report

The B-47 aircraft was flown by MCAIR test pilots to evaluate flight characteristics using the SSC for control. The following pertinent comments were taken from the pilot's flight report.

- The breakout force was one pound and the maximum force at full fore and aft was two pounds.

- The maximum force at full lateral travel was 1.75 pounds.

- Unfortunately, when engaged, the Fly-By-Wire exhibited degraded damping characteristics due to a polarity reversal of the pitch rate gyro. The degraded characteristics of the pitch channel precluded a complete evaluation of the flight characteristics.

- The task of precisely holding attitude with the SSC was difficult. This was attributed to the sensitivity of the controller and to the low feel force gradients and the low damping of the aircrafts' Dutch Roll Mode.

- When monitoring the SSC inputs on an oscilloscope it was noted that the airplane was being controlled through a series of step inputs, i.e. by control position instead of control force. This was attributed to the very low feel forces supplied by the spring.

b. Cornell Aeronautical Laboratory's B-26 Aircraft

The USAF Aerospace Research Pilot School (ARPS), Edwards Air Force Base, California, uses Cornell Aeronautical Laboratory's B-26 aircraft equipped with a Dual Sidearm Controller, for SSC familiarization flight training.

(1) General Description

This Dual Sidearm Controller is an integral part of the co-pilot's seat frame. One controller is located on each side of the seat on the arm rest. The aircraft can be flown using either the left or right hand controller and whenever one moves the other one tracks in the same direction. Only the right hand controller is equipped with a trim button and intercom switch. The artificial feel forces are nonadjustable during flight. The following information was considered pertinent:

- The Dual Sidearm Controller was designed expressly for use with a Fly-By-Wire flight control system employing heavy augmentation.

- A light weight hybrid linkage system was used to provide slaved left-hand and right-hand controller grips. The linkage path was designed so that if one system breaks the other grip will continue to operate.

- The mechanical feel system provides a choice of force gradients by spring replacement.

- o Breakout forces are provided via adjustable spring preloading.

- o Controller damping is provided via an adjustable compensated viscous damper.

(2) Flight Report

The B-26 aircraft was flown by MCAIR test pilots to evaluate controllability using the Dual Sidearm Controller. The following pertinent comments were taken from the pilot's flight report.

- o The controllers are located too far aft in relation to the body.

- o The performance restrictions on the aircraft made detailed investigation impossible. To get a comprehensive evaluation one must look at all the tasks that will be performed with a given stability under realistic fighter type environment.

c. Edwards AFB F-104D Aircraft

The USAF Aerospace Research Pilot School (APRS), Edwards AFB, California, uses a F-104D aircraft equipped with a side stick controller for familiarization flight training.

(1) General Description

The SSC is located on the right hand console in each cockpit. The following information is pertinent:

- o Pitch trim is available in the forward cockpit only. The trim switch is mounted on the SSC grip. This switch provides "beeper" trim control in the same manner as on the basic aircraft. The SSC pitch trim switch is active only when the SSC is engaged.

- o The SSC Disengage Trigger Switch opens an electrical control circuit which de-energizes the solenoid operated hydraulic valves. The valves remove hydraulic power from the system.

- o SSC force gradient adjustments are accessible through slotted openings in the left and top sides of the controller housing. These adjustments allow the pilot to select any combination of the three force gradients available in each axis.

- o SSC damping adjustments provide adjustable viscous damping of the SSC in each axis.

- o Handgrip position adjustment controls are located at the base of the hand grip. These controls allow the pilot to position the grip to the most comfortable operating position.

- o Arm rest adjustment can be made by means of controls located at the rear of the controller. The arm rest may be raised or lowered to the most comfortable operating position.

(2) Flight Report

The F-104D aircraft was flown by MCAIR test pilots to evaluate flight characteristics using the SSC for control. The following pertinent comments were taken from the pilot's flight report.

- o Initial reaction after taking control with the SSC, after takeoff on center stick, was one of amazement at the increase in aircraft response in pitch and roll.

- o Adapting to the SSC was very easily accomplished.

- o The flight control workload was considerably reduced with the SSC.

- o The SSC location on the right-hand console was very comfortable, and became second nature almost immediately.

- o The pitch trim control switch is awkward to operate and should be located on the inboard side of the grip.

- o The seat must be raised or lowered for optimum SSC flight operation. This sometimes results in a pilot sitting lower in the cockpit than the normal flying position.

- o Shifting from center stick to side stick gave the impression that the cockpit became considerably larger. The cramped feeling essentially disappeared.

- o Precise pilot tasks involving capture appeared as though they would be easier on the side stick than on the center stick.

d. Mercury and Gemini

The Mercury spacecraft hand controller incorporated mechanical push rod outputs for the actuating devices. This is not applicable to the SFCS which requires electrical signal outputs. The Gemini controller provided electrical output signals but the configuration is not compatible with the required F-4 installation.

e. Tactical Weapon Delivery Program (F-4)

The Tactical Weapon Delivery (TWeaD) flight control system does not have a sidestick. However, it utilizes vernier pitch and yaw controls similar to those required for the SFCS. As a result of the TWeaD experience and the MCAIR simulation programs, the following vernier characteristics will be implemented:

- o Vernier control from the SSC is limited to the pitch axis.
- o A combination of normal acceleration and pitch rate is commanded by the pitch vernier.
- o The acceleration which can be commanded by the pitch vernier is 0.3g at 360 knots true air speed.
- o The yaw vernier will be located on the forward trim panel.

3. MCAIR SSC MOCKUP

MCAIR designed and fabricated an SSC mockup in order to evaluate SSC interface with F-4 cockpit geometry, determine SSC artificial feel forces, and establish arm rest and grip configuration. The mockup shown in Figure 127 was installed in an F-4 forward fuselage mockup and evaluated by MCAIR pilots.

a. Cockpit Installation Criteria

The location of the SSC in the right console of both cockpits was first selected by using anthropometry data from WADC Technical Report 52-321 for forearm lengths, shoulder heights, etc., for crewmembers in the range of 5 through 95 percentiles. The evaluation of the installed mockup unit by MCAIR pilots resulted in relocation of the SSC further forward for optimum controllability. The critical flight conditions which determined location were landing approaches with the seat raised and the normal flying position with the seat lowered.

Preliminary cockpit installation criteria were selected for forward and aft cockpits such as:

- o the arm-rest angle for the 5 through 95 percentile man, and
- o the distances from the seat reference point forward and up to the SSC neutral reference point.

b. SSC Design Criteria

Various artificial feel forces, grip configurations, and arm rest configurations were investigated and recommendations for breakout force, total force gradient, SSC trigger actuating force, adjustment range of the arm rest, location of the pitch vernier control, and grip pivot geometry were developed.

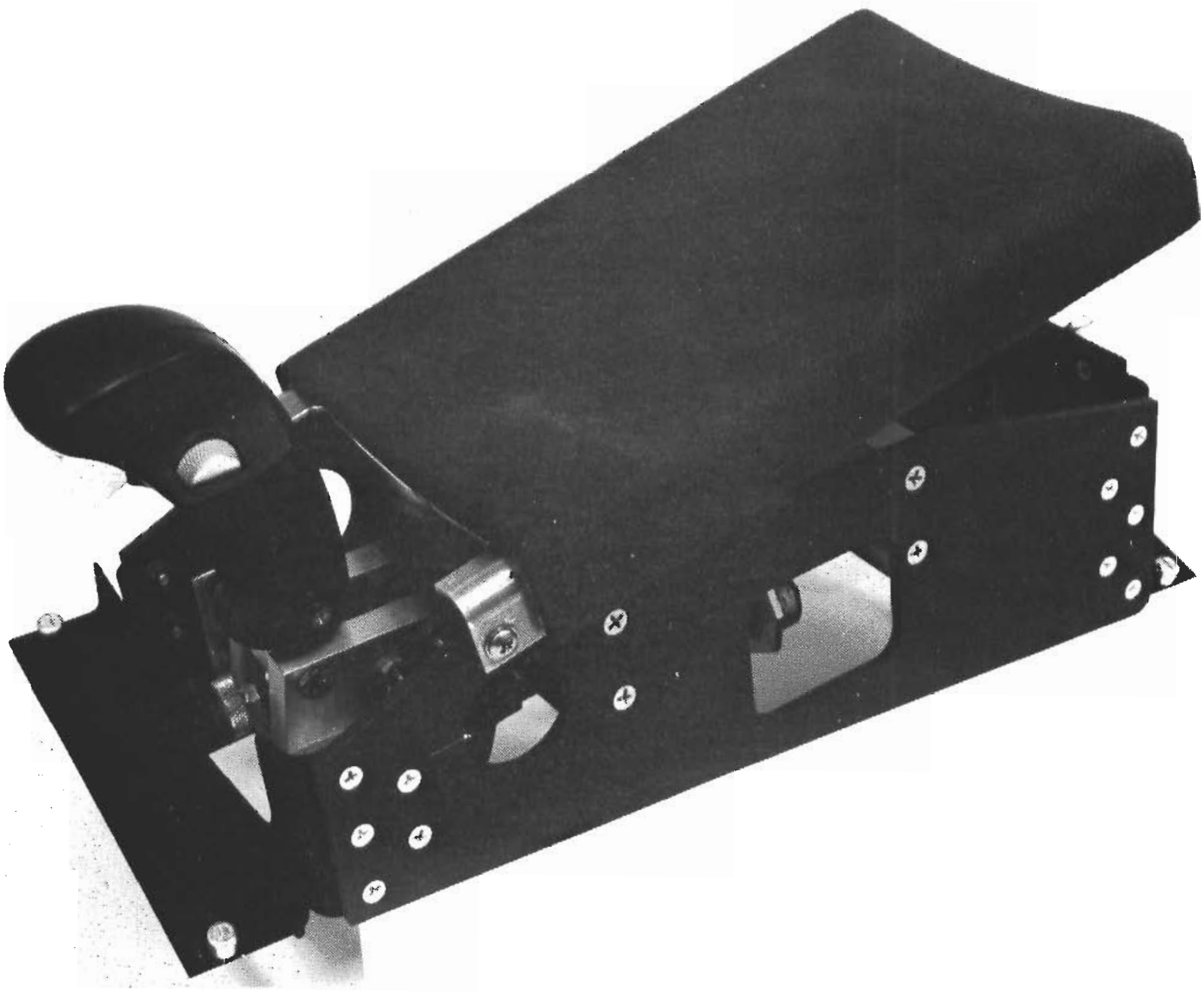


FIGURE 127
MCAIR SIDE STICK CONTROLLER MOCKUP

c. Arm Restraint

An elbow cup restraint was considered during the mockup evaluation. The decision was made that it would not be included for the following reasons:

- Acceleration g restraint is provided by the inclined angle of the arm rest.
- Hand grip movement due to g inputs can be substantially reduced by gripping the grip near the pitch pivot point.
- The elbow cup would cover the oxygen regulator panel and a portion of the compass control panel located directly aft of the SSC.
- The elbow cup could restrict normal movement of the arm on the arm rest.

d. Additional Requirements

The SFCS test aircraft will have two SSC's installed, one in the forward and one in the aft cockpit. Both SSC's are to be mechanically and electrically independent from each other. A neutral position grip lock is required to help prevent inadvertent actuation when the controller is not in use. The grip neutral position is required to be adjustable. Grip neutral position, range and travel are shown in Table LIII.

The SSC is required to provide output signals for primary pitch and roll control functions and pitch vernier control function, and a trigger initiated signal to engage the electrical back-up mode. These signals are to be quadruplex. It was not practicable to provide a quadruplex yaw axis vernier within the confines of the grip shape.

4. SSC DEVELOPMENT MOCKUP

The request for proposal to prospective SSC Suppliers required the submittal of an engineering mockup representative of their proposed design. Subsequently the contract required the selected SSC Supplier to upgrade his mockup to be representative of the SSC design as the design effort progressed. The purpose of this mockup was to:

- provide a tool for verification of human engineering design criteria.
- provide a tool for evaluating mechanical design concepts.
- Provide continuous MCAIR "exposure" to the design and development of the SSC.
- provide for timely incorporation of knowledge obtained during the evaluation effort prior to design finalization.

Figure 128 illustrates the current configuration of the Lear-Siegler Inc. Side Stick Controller development mockup.

Several mockup evaluations were made beginning with the SSC unit submitted as part of the Supplier's proposal package. As a result of these evaluations, MCAIR was afforded the opportunity of guiding the design and development of the SSC as it progressed through evolutionary design stages. The human engineering verification effort expended by LSI coupled with MCAIR's evaluation provided a double engineering critique of the system as it evolved. Some of the modifications resulting from this effort were:

- Grip neutral adjustment was changed from a friction locked device to a semi-gimbal arrangement with thumbscrews for adjusting the fore and aft and lateral neutral positions.

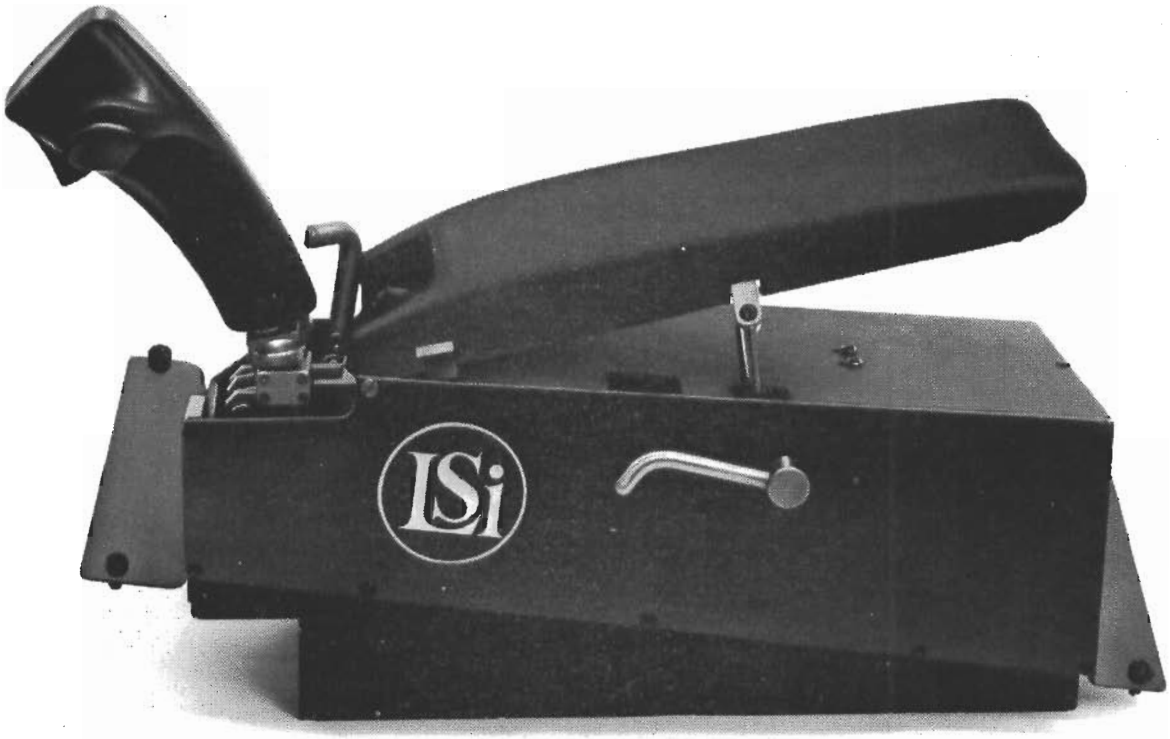


FIGURE 128
SSC DEVELOPMENT MOCKUP

- Arm rest geometry was revised for improved arm and wrist support.
- Arm rest adjustment was improved by changing from a rotary no-back arrangement to a positive locking detent design including an upward spring-loaded feature for easy and quick adjustment.
- Grip configuration was optimized.
- Damping characteristics were improved.
- Grip neutral position adjust controls were relocated for improved pilot access.
- Breakout force levels were revised as a result of evaluating the SSC in the MCAIR simulation program.

5. RESULTS

Results of the SSC development study and evaluations of the SSC developmental mock-up provided the following design and performance criteria for the SSC flight units:

- Arm rest adjustment angle is to be 0° to 31°.
- The vernier control is to be located on the inboard side of the grip.
- The vernier control is to be spring-loaded to the neutral position. Control travel will be ±90° rotation from neutral.
- Grip neutral position adjustment range is to be 10 1/2° forward, 7 1/2° aft and ±5° laterally from nominal, which is 5° inboard from vertical.
- Grip travel is to be ±20° from neutral in the fore and aft direction and ±15° from neutral in the lateral direction.
- The artificial feel forces to be initially mechanized are shown in Table LIII.

TABLE LIII
ARTIFICIAL FEEL FORCES AND GRIP GEOMETRY

	Pitch Axis	Roll Axis	Pitch Vernier	Trigger Switch
Breakout Force	1.75 lb	1.75 lb	3.5 oz	15 lb
Total Force at Full Travel	9.45 lb	3.55 lb	7.5 oz	N/A
Damping (In. Lb/Rad/Sec)	12 ⚠1	2 ⚠1	N/A	N/A
Travel	20° Fwd and Aft	15° Inb'd and Outb'd	±90°	N/A
Grip Neutral Adjust	10° 30′ Fwd 7° 30′ Aft	5° Inb'd and Outb'd	N/A	N/A
Grip Neutral Adjust Knob Location	Outb'd of Grip	Inb'd of Grip	N/A	N/A

⚠1 At Ambient Room Temperature

LIST OF SPECIALIZED ABBREVIATIONS AND SYMBOLS
FOR APPENDIX VII

ABBREVIATIONS:

AFB - Air Force Base

ARPS - Aerospace Research Pilot School

g - Acceleration due to Gravity

In - Inch

lb - Pound

LSI - Lear Siegler, Incorporated

MCAIR - McDonnell Aircraft Company

N/A - Not Applicable

oz - Ounce

Rad - Radian

Sec - Second

SFCS - Survivable Flight Control System

SSC - Side Stick Controller

TWeaD - Tactical Weapon Delivery

USAF - United States Air Force

WADC - Wright Air Development Center

SYMBOLS:

$^\circ$ Degree

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13. ABSTRACT

The Survivable Flight Control System (SFCS) Program is an advanced development program of which the principal objective is the development and flight test demonstration of an SFCS utilizing Fly-By-Wire and Integrated Actuator Package techniques. The studies and analyses conducted to date have sufficiently defined the system requirements to provide a definition of an approach to the implementation of the SFCS. The results of these studies and the definition of the approach are presented herein. The details of the Control Criteria, Control Law Development, and Hydraulic Power and Actuation Studies are presented in report supplements 1, 2, and 3, respectively.

The SFCS Program is based on the principle of dispersed redundant control elements providing improved control performance and a very stable weapons delivery platform. The SFCS equipment includes:

o A quadruplex, three-axis, two-fail operational Survivable Flight Control Electronics Set which provides the computations for fly-by-wire control. Preflight built in test, in-flight monitoring of SFCS equipment, an adaptive gain and stall warning function, and provisions for in-flight failure simulation are included.

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13. ABSTRACT (Continued)

- Quadruplex force-summing electrohydraulic secondary actuators to convert the fly-by-wire flight control signals into the physical motion required to command the existing power actuators of the F-4 test aircraft.

- A quadruplex, two-axis sidestick controller in each cockpit. The front cockpit will also provide fly-by-wire center stick to allow direct comparison.

- A duplex power-by-wire actuator termed the Survivable Stabilator Actuator Package (SSAP). The SSAP will be capable of full-time operation throughout the F-4 flight envelope while receiving only electrical power. The SSAP incorporates a quadruplex velocity-summing electromechanical secondary actuator, and is controlled solely by the fly-by-wire system.

Unclassified
Security Classification

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Aircraft Equipment						
Aircraft Handling Qualities						
Control System Analysis						
Controllers						
Electrohydraulic Actuators						
Electromechanical Actuators						
Flight Control Systems						
Fly-By-Wire						
Hydraulic Systems						
Integrated Actuator Packages						
Power-By-Wire						
Redundant Systems						

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Unclassified
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