NASA Contractor Report 165631

INTEGRATED APPLICATION OF ACTIVE CONTROLS (IAAC) TECHNOLOGY TO AN ADVANCED SUBSONIC TRANSPORT PROJECT–

CURRENT AND ADVANCED ACT CONTROL SYSTEM DEFINITION STUDY–VOLUME I

FINAL REPORT

BOEING COMMERCIAL AIRPLANE COMPANY
P.O. BOX 3707, SEATTLE, WASHINGTON 98124

CONTRACTS NAS1-14742 AND NAS1-15325
October 1981

FOR EARLY DOMESTIC DISSEMINATION

Because of their possible commercial value, these data developed under Government contracts NAS1-14742 and NAS1-15325 are being disseminated within the United States in advance of general publication. These data may be duplicated and used by the recipient with the expressed limitations that the data will not be published nor will they be released to foreign parties without prior permission of The Boeing Company. Release of these data to other domestic parties by the recipient shall only be made subject to these limitations. The limitations contained in this legend will be considered void after October 1983. This legend shall be marked on any reproduction of these data in whole or in part.

NASA
National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

NASA Contractor Report 165631

INTEGRATED APPLICATION OF ACTIVE CONTROLS (IAAC) TECHNOLOGY TO AN ADVANCED SUBSONIC TRANSPORT PROJECT–

CURRENT AND ADVANCED ACT CONTROL SYSTEM DEFINITION STUDY–VOLUME I

FINAL REPORT

BOEING COMMERCIAL AIRPLANE COMPANY
P.O. BOX 3707, SEATTLE, WASHINGTON 98124

CONTRACTS NAS1-14742 AND NAS1-15325
October 1981

NASA
National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

FOREWORD

This document constitutes the final report of the Current Technology ACT Control System Definition and the Advanced Technology ACT Control System Definition Tasks of the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project. The report covers work performed from July 1978 through October 1980 under Contracts NAS1-14742 and NAS1-15325.

Volume I contains the principal results of the study, and supplementary technical data are contained in Volume II.

The NASA Technical Monitors for these contract tasks were R. V. Hood and D. B. Middleton of the Energy Efficient Transport Project Office at Langley Research Center.

The work was accomplished within the Preliminary Design and the Engineering Technology Departments of the Vice President-Engineering organization of the Boeing Commercial Airplane Company. Key contractor personnel who contributed were:

G. W. Hanks	Program Manager
H. A. Shomber	IAAC Project Manager
H. A. Dethman	Design Integration
L. B. Gratzer	Technology Integration
A. Maeshiro	Task Manager (Current Technology ACT Control System Definition)
D. Gangsaas	Task Manager (Advanced Technology ACT Control System Definition)
J. D. Blight	Flight Controls Technology
S. M. Buchan	Flight Controls Technology
C. B. Crumb	Flight Control Design
R. J. Dorwart	Product Assurance
C. C. Flora	Flight Controls Technology
U. Ly	Flight Controls Technology
K. A. B. Macdonald	Product Assurance
D. C. Norman	Flight Controls Technology

E. T. Reiquam	Systems Technology
J. Shen	Flight Controls Technology
R. D. Smith	Flight Control Design
T. D. Verrill	Flight Control Design
T. B. Cunningham	Honeywell Systems and Research Center
J. C. Larson	Honeywell Avionics Division
E. R. Rang	Honeywell Systems and Research Center
R. K. Mason	Hydraulic Research Textron
O. A. Walkes	Hydraulic Research Textron

During this study, principal measurements and calculations were made in U.S. customary units and were converted to Standard International units for this document.

Use of trade names or names of manufacturers in this report does not constitute an official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration.

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Concluded)

Page

VOLUME II, APPENDICES

FIGURES

FIGURES (Continued)

FIGURES (Continued)

Page

FIGURES (Continued)

FIGURES (Continued)

Page

FIGURES (Continued)

FIGURES (Continued)

FIGURES (Concluded)

Page

TABLES

TABLES (Continued)

Page

TABLES (Continued)

TABLES (Concluded)

1.0 SUMMARY

This report documents both the Current and Advanced Active Controls Technology (ACT) System Definition Tasks of the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project.

The first part of the report discusses development of an ACT system architecture, based on current technology system elements, that best meets the reliability and availability requirements. The balance of the report discusses use of optimal control theory for analysis and synthesis of the ACT multivariable control laws and the implementation of the same ACT functions as the current technology work with advanced system components.

The objectives of the Current Technology ACT System design were to:

- Define a highly reliable, low technical risk ACT control system for the IAAC airplane configurations using current technology
- Support assessment of the benefit associated with the ACT airplane by evaluating reliability, cost, and weight of the current technology system
- Identify technical risk areas and recommend system development and testing

The system architecture work addressed implementation of all potentially beneficial ACT functions, not just those employed on a particular airplane configuration. The approach of the current technology work was to define and evaluate two extreme system architecture forms, then define a "Selected System" that used the best features of extreme forms. The principal differences among these systems are the number and organization of the digital computers. The Selected System employs three redundant computers to implement all of the ACT functions, four redundant smaller computers to implement the crucial pitch-augmented stability function, and a separate maintenance and display computer. The reliability objective of probability of crucial function failure of less than 1×10^{-9} per flight of 1 hr can be met with current technology system components, if the software is assumed fault free and coverage approaching 1.0 can be provided. There is no generally accepted method to prove the software to be error free. However, a disciplined approach

beginning with functional analysis and proceeding through requirements, design, coding, verification, validation, exhaustive testing, configuration control, and documentation has been shown, on space and aircraft programs, to be both essential and effective in producing highly reliable real-time control software.

The objectives of the Advanced Technology ACT System design were to:

- Synthesize the multivariable ACT control laws directly, using time-domain optimal control theory
- Evaluate the effects of actuation system nonlinearities on gust-load alleviation and flutter-mode control
- Determine an ACT control system architecture based on advanced technology (circa 1990)

The optimal control theory approach to ACT control law synthesis yielded comparable control law performance much more systematically and directly than the classical s-domain approach. The ACT control law performance, although somewhat degraded by the inclusion of representative nonlinearities, remained quite effective. Certain high-frequency gust-load alleviation functions may require increased surface rate capability. Finally, the use of advanced computers with bus architecture (both potentially available circa 1990) has the promise of significant savings in airplane weight and first cost.

2.0 INTRODUCTION

The Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project has three major objectives. The first objective is the credible assessment of the benefit to a commercial jet transport airplane of the full application of active controls designed into the airplane from the beginning of the airplane program. The second objective is identification of the risks associated with the use of Active Controls Technology (ACT). The third objective is reduction of these risks to a level commensurate with commercial practice, through test and evaluation, to the degree possible within funding limitations.

This project, a part of the NASA-Boeing Energy Efficient Transport (EET) Program, has been organized into three major elements as shown at the top of Figure 1. The first major element, as shown in the figure, includes establishment of the design criteria appropriate for an ACT airplane, design of an ACT airplane configuration to meet the selected criteria, design of an ACT control system based upon current technology, and selection and evaluation of a Final ACT Configuration. In parallel with these tasks, the Advanced Technology ACT Control System element included exploration of more direct control law synthesis methods and alternative means of implementing the ACT functions using advanced technology, as shown in Figure 2. The final major element of the IAAC Project will address the reduction of risk associated with implementation of ACT on a commercial transport through test and evaluation activities. Figure 3 shows this final element. A more detailed discussion of the IAAC Project Plan is contained in Reference 1.

This report covers those parts of the IAAC Project shown shaded in Figures 1 and 2. The Current Technology Task is discussed in Sections 4.0 through 9.0. The Advanced Technology Task is discussed in Sections 10.0 through 14.0.

Results of the Configuration/ACT System Design and Evaluation Tasks to date have shown that when ACT is integrated early into the design of a transport aircraft, significant fuel savings can be realized. It was previously reported (ref 2) under this project that 6% to 6.5% block fuel savings could be obtained with ACT at the design range of about 3700 km (2000 nmi) compared to the performance of a Conventional Baseline Airplane (ref 3) with the same wing planform (aspect ratio 8.7). When this wing was retailored (to aspect ratio 12.0 with unchanged area), the fuel savings increased to about 10%, accompanied by

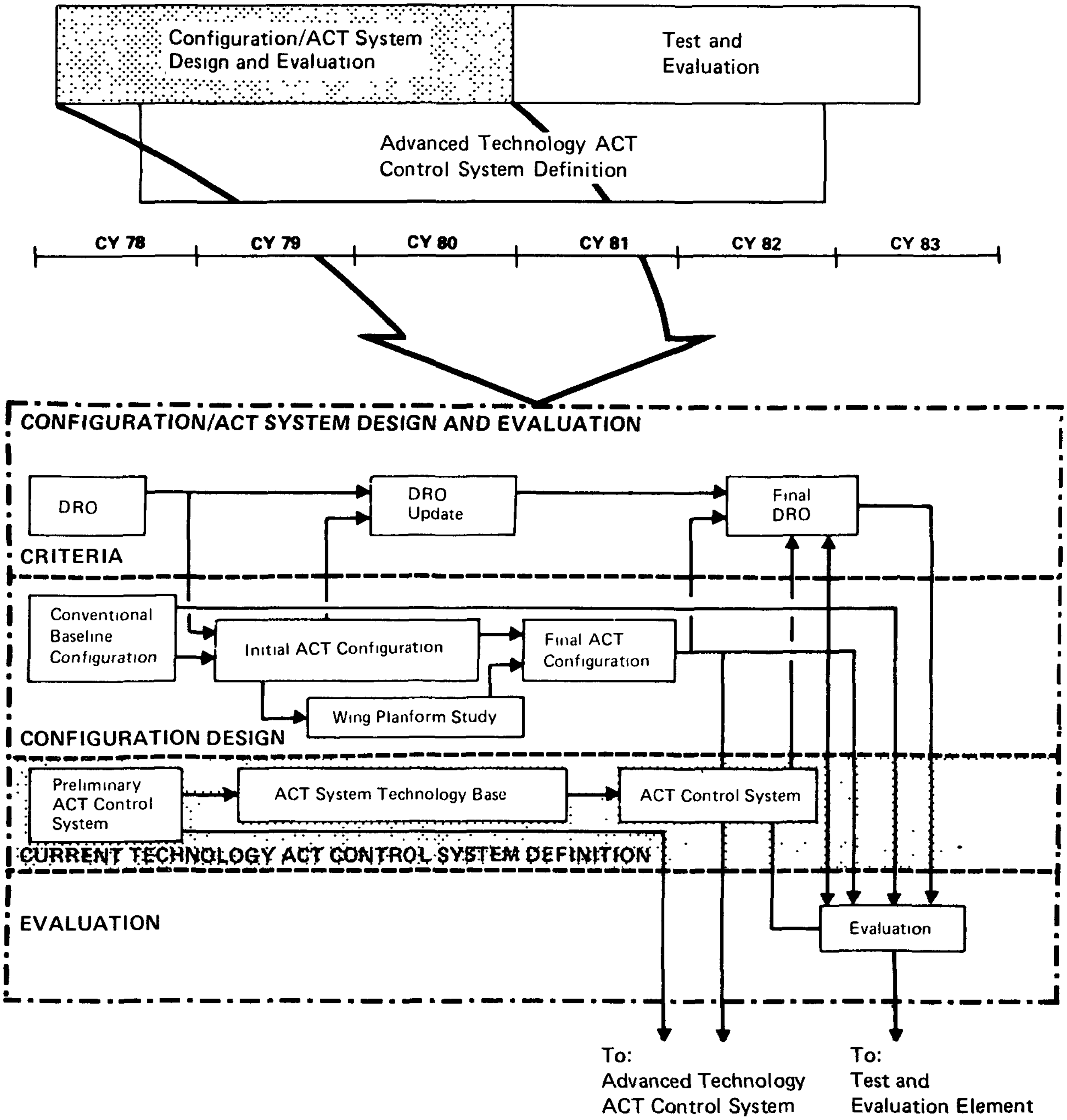


Figure 1. Configuration/ACT System Design and Evaluation Element

increased estimated manufacturing costs (ref 4). These benefits are all based on current technology ACT system implementation such as that discussed in Sections 4.0 through 9.0 of this document.

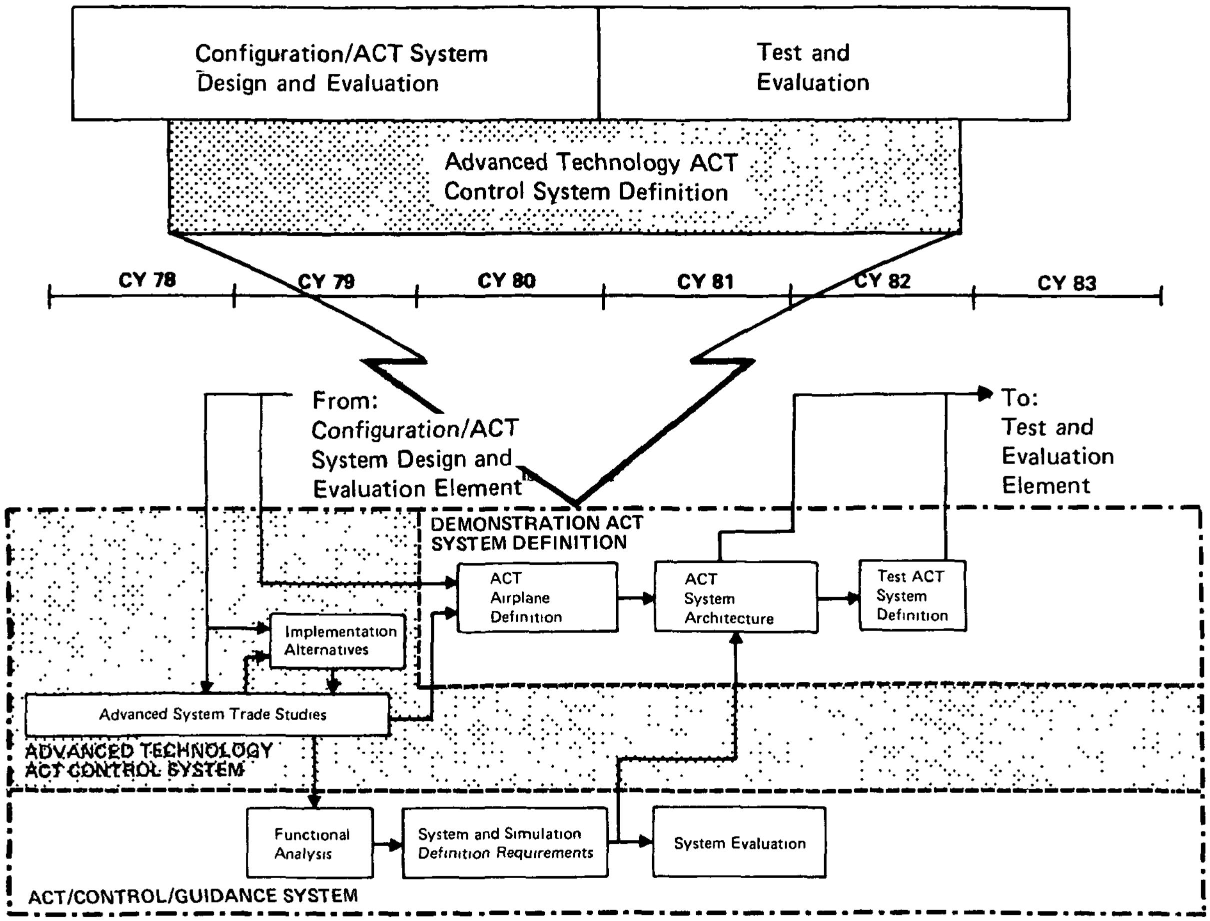


Figure 2. Advanced Technology ACT Control System Definition Element

The Current Technology ACT System Definition work proceeded under the important constraint of using only currently available control system technology. There was to be no dependence upon technological features that have not been demonstrated as feasible. The approach to the current technology system definition has been to perform a preliminary design of an ACT system that can provide all of the considered functions with appropriate reliability. The configuration definition started with two extreme architectural forms chosen to represent the apparent limits of system organization—first, with the functions integrated into one set of computers and, second, with the functions segregated into separate computer sets. This was followed with comparative evaluation of those two systems and the use of that comparison to derive the basic form of the Selected System. This third system was then evaluated in the same fashion as the other

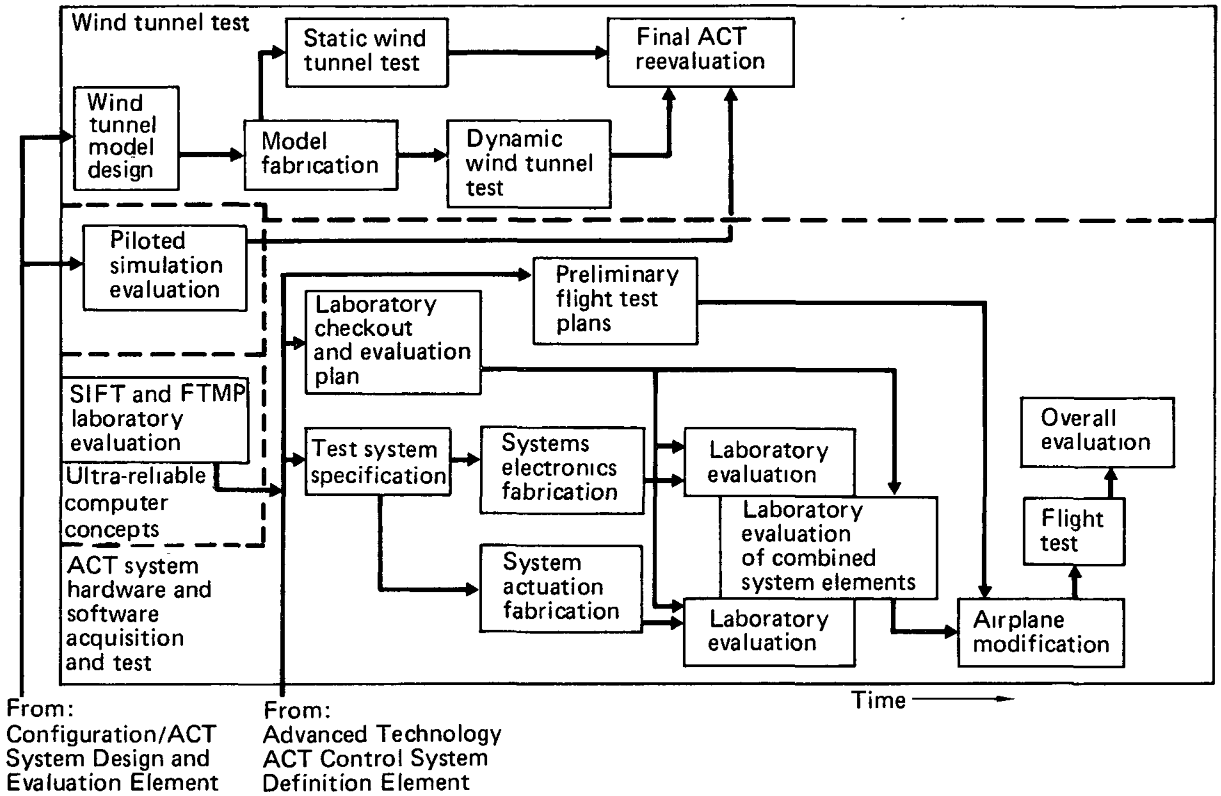


Figure 3. Test and Evaluation Element

two systems. In parallel, the feasibility of certain key assumptions was checked in computer laboratory work, simulating the digital control computers in single-channel and redundant hookups.

The advanced technology work reported in this document focused on two areas: the use of modern optimal control law synthesis techniques and an alternative implementation of the same ACT functions treated in the Current Technology ACT System Definition Task. The results of the optimal synthesis work are compared with that of the classical s-domain work done in support of the Configuration/ACT System Design and Evaluation element. The alternative ACT function implementation identified three approaches using varying degrees of technical readiness; these approaches are characterized as having low, medium, and high technical risk associated with a circa 1990 implementation. A derivative of the medium-risk system was selected for further evaluation and cost-of-ownership analysis.

This document is the complete report on IAAC control systems. Volume I covers the current and advanced technology system work accomplished to date. Volume II contains appendices to the material of Volume I.

3.0 SYMBOLS AND ABBREVIATIONS

This section contains five subsections: General Abbreviations, Subscripts, Superscripts, Operators, and Symbols. Each subsection is arranged in alphabetical order. For ease of reference, Subsection 3.2 is further divided into two parts–velocity and Mach number subscripts (3.2.1) and general subscripts (3.2.2).

3.1 GENERAL ABBREVIATIONS

ac	alternating current
app	appendix
$a_i, b_i,$	control law transfer function parameters
$c_i, d_i,$	control law transfer function parameters
p_i	control law transfer function parameters
A	ampere; piston area; system state matrix
AAL	angle-of-attack limiter
AB	address bus
ACES	airline cost-estimating system (program)
ACT	Active Controls Technology
A/D	analog to digital
AEC	Atomic Energy Commission
AFCS	automatic flight control system
Ah	ampere-hour
AIL	aileron
ALU	arithmetic logic unit
AP	autopilot
APB	auxiliary power breaker
APD	avalanche photodiodes
APFDC	autopilot flight director computer

APU	auxiliary power unit
AR	analog reversion
ARCS	Airborne Advanced Reconfigurable Computer System
ARINC	Aeronautical Radio Incorporated
ASYM	asymmetry detection electronics
ATDP	air-turbine-driven pump
ATE	automatic test equipment
A_a	airplane state matrix
A_u	actuator state matrix
A_w	wind state matrix
A_O	steady aerodynamic stiffness matrix
$\bar{A}_O$	steady aerodynamic stiffness matrix associated with control surface deflection
A_1,A_2	unsteady aerodynamic stiffness matrix
$\bar{A}_1,\bar{A}_2$	unsteady aerodynamic stiffness matrix associated with control surface deflection
bps	bits per second
brkr	breaker
b_i	unsteady aerodynamic lag coefficients
B	control input distribution matrix
B′	transformed control distribution matrix
BIT	built-in test
BTB	bus tie breaker
BPCU	bus power control unit
B_a	airplane control distribution matrix
B_i	bending moment at the ith station
B_u	actuator input distribution matrix

B_w	wind input distribution matrix
B_1	input to system matrix for full-state feedback system
B_2	command to input matrix for full-state feedback system
c	characteristic chord length
cg	center of gravity
cm	centimeter
$\bar{c}$	mean aerodynamic chord
c_i	Kussner scaling coefficients
C	Celsius; state to output distribution matrix
C′	transformed state to output distribution matrix
CARE	computer-aided reliability estimates
CARSRA	computer-aided redundant system reliability analysis
CAS	control augmentation system
CB	circuit breaker
CCDL	cross-channel data link
CCDT	cross-channel data transfer
CDC	Control Data Corporation
CLB	climb
CLP	control law processor
CMOS/SOS	complementary metal-oxide semiconductor/silicone on sapphire
CONT	continuous
CPU	central processing unit
CQ	servovalve spool gain
CRT	cathode-ray tube
CRZ	cruise
CSEU	control system electronic unit

CSU	computer service unit
CV	control valve
CWS	control wheel steering
CY	calendar year
C_a	airplane output distribution matrix
C_d	plant process noise covariance matrix
$\bar{C}_d$	fictitious input noise covariance matrix
C_p	state to performance output distribution matrix
C_u	actuator output distribution matrix
C_v	measurement noise covariance matrix
C_w	state to wind output distribution matrix
C_O	quasi-steady aerodynamic stiffness matrix
$\bar{C}_O$	quasi-steady aerodynamic stiffness matrix associated with control surface deflection
C_1	quasi-steady aerodynamic damping matrix
$\bar{C}_1$	quasi-steady aerodynamic damping matrix associated with control surface deflection
$\mathcal{C}$	controllability matrix
d	plant disturbance vector; control law transfer function denominator
$\bar{d}$	fictitious input noise
dB	decibel
dc	direct current
deg	degree
d_i	Kussner lag coefficients
D	control to output distribution matrix; drag
D/A	digital to analog
DADC	digital air data computer

DAST	drone for aerodynamic and structural testing
DC	don't care
DITS	digital information transfer system
DMA	direct memory access
DN	down
DRO	design requirements and objectives
D_i	unsteady aerodynamic matrices to model lift growth effects associated with rigid and elastic modes
$\bar{D}_i$	unsteady aerodynamic matrices to model lift growth effects associated with control surfaces
D_w	direct transmission matrix from wind to output
e	base for Napierian logarithms; system state estimate error
E	process noise distribution matrix; exponent; wind to output distribution matrix
EASY	environmental control analysis system
EDP	engine-driven pump
EET	Energy Efficient Transport (Program)
EHV	electrohydraulic valve
ELEV	elevator
EM	electric motor
EMA	electromechanical actuator
EMI	electromagnetic interference
EMP	electric-motor-driven pump; electromagnetic pulse
E(p)	fit error function
EPC	external power contactor
f	longitudinal correlation function
fig.	figure
ft	feet

F	Fahrenheit; FMC; force; measurement noise distribution matrix
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FBW	fly by wire
FCC	flight control computer
FD	failure detection
FH	flight hour
FMC	flutter-mode control
FMEA	failure mode and effect analysis
FO	fiber optic
FTAP	fault tree analysis program
FTMP	fault-tolerant multiple processor
FTREE	fault tree computer program
g	acceleration due to gravity; structural damping coefficient for neutral stability; transverse correlation function
g-dump	normal acceleration autopilot disconnect
g_i	unsteady aerodynamic lag coefficients
gal	gallon
gen	generator
G	feed forward gain; feedback gain matrix
GCB	generator circuit breaker
GCU	generator control unit
GG	device type identifier
GLA	gust-load alleviation
GMPS	general-purpose multiplex system
GPIB	general-purpose interface bus
GPM	gallons per minute

GPS	global positioning system
G(s)	transfer function
G_i	constants associated with representation of unsteady aerodynamic forces
hr	hour
H	feedback gain; Hamiltonian
HDC	device type identifier
HDP	device type identifier
HMOS	high-performance metal-oxide semiconductor
HPU	hydraulic power unit
Hz	hertz
in	inch
ips	inches per second
I	identity matrix; input
IAAC	Integrated Application of Active Controls Technology to an Advanced Subsonic Transport Project
IAP	integrated actuator package
IAS	indicated airspeed
IB	input bus
IC	integrated circuit
IDG	integrated drive generator
IEEE	Institute of Electrical and Electronics Engineers
ILD	injection laser diodes
ILS	instrument landing system
INBD	inboard
INOP	inoperable
INS	inertial navigation system
I/O	input/output

IOC	input/output controller
IRS	inertial reference system
ISO	International Standards Organization
I^2L	integrated injection logic
I_A, I_B, I_C	input current
J	cost function
JPL	Jet Propulsion Laboratory
kg	kilogram
km	kilometer
kn	knot
kPa	kilopascal
kVA	kilovoltampere
kW	kilowatt
k_i	gain at the ith control input
K	counter value; gain; gain matrix; state estimate error covariance matrix; thousand
KA	amplifier gain; angle-of-attack gain
KAM	model amplifier gain
KC	servovalve monitor gain
KD	demodulator gain
KE	elevator gain
KF	flutter-mode control gain
KFB	feedback transducer gain
KG	gust-load alleviation gain
KL	feedback gain; lever ratio
KM	maneuver gain
KMA	maneuver-load control aileron gain

KME	maneuver-load control elevator gain
K ops	thousand operations per second
KQ	pitch-rate gain
KSV	servovalve gain
KSVM	model servovalve gain
KTD	transducer gain
KU	speed gain
K_f	filter gain
K_{ss}	steady-state solution to matrix Riccati equation
$K_{1/3}(\,)$	modified Bessel functions of the second kind of order 1/3
$K_{2/3}(\,)$	modified Bessel functions of the second kind of order 2/3
K_θ	pitch angle scaling constant
lb	pound
lbf	pound-force
lb/in	pounds per inch
lb/in^2	pounds per square inch
L	dynamic load vector; length; lift; transverse turbulence scale length
LAS	lateral/directional-augmented stability
LAT	lateral
LE	leading edge; left elevator
LED	light-emitting diode
LIA	left inboard aileron
LIF	left inboard flaperon
LOAI	left outboard aileron, inboard
LOAO	left outboard aileron, outboard
LOF	left outboard flaperon

LR	lower rudder
LRU	line replaceable unit
LSI	large-scale integration
LSIC	large-scale integrated circuit
LVDT	linear variable differential transformer
L_i	three-component vector of L consisting of shear, bending moment, and torsion of the ith station
L_u	longitudinal turbulence scale length
$L_{v,w}$	transverse turbulence scale lengths
ℓ	liter
$\mathcal{L}$	Laplace transform
m	meter
mA	milliampere
max	maximum
min	minute
mm	millimeter
ms	millisecond
μs	microsecond
M	feedforward gain matrix; Mach; mega; motor
MARG	marginal, one failure away from function loss
MCU	modular control unit (ARINC dimension specification)
MCV	main control valve
MEL	minimum equipment list
MG	main gear
MHD	magnetohydrodynamic
MLC	maneuver-load control
MMU	memory management unit

MOS	metal-oxide semiconductor
M/R	maximum range to its resolution
MTBF	mean time between failures
MUX	multiplexer
MVL	midvalue logic
MW	megawatt
MZFW+F	maximum zero fuel weight plus fuel (including full reserve tanks)
M_q	dimensional variation of pitching moment with pitching rate
M_u	dimensional variation of pitching moment with speed
M_{α}	dimensional variation of pitching moment with angle of attack
$M_{\dot{\alpha}}$	dimensional variation of pitching moment with angle-of-attack rate
M_{δ_E}	dimensional variation of pitching moment with elevator angle
nmi	nautical mile
ns	nanosecond
$_n\lambda_{mp}$	Markov transition rate, stage n between states m and p
n_i	control law transfer function numerators
n_z	vertical acceleration
N	dummy vector; newton; ultimate normal load factor
NAV	navigation (mode)
NDP	numerical data processor
NG	nose gear
NMR	nuclear magnetic resonance
No.	number
N_1	speed of the No. 1 rotor
ops	operations per second
oz	ounce
o_B	body axis coordinates

o_I	inertial axis coordinates
O	output
OAI	outboard aileron (inboard section)
OB	output bus
OEM	original equipment manufacture
OMP	output monitor processor
OUTBD	outboard
$\mathcal{O}$	observability matrix
$p, \hat{p}$	Lagrange's multiplier
psi	pounds per square inch
pwr	power
P	inertial-to-body transformation matrix; probability; pump
Pa	pascal
PAS	pitch-augmented stability
PBW	power by wire
PCU	power control unit
PCM	power conditioning module
PF	pump and filter
PIN	p-layer intrinsic n-layer
PLIM	nonlinear actuator position limit
PM	permanent magnet
PROM	programmable read-only memory
P/S	parallel/serial
PSD	power spectral density
P_R	pressure, return
P_S	pressure, supply
P_1	hydraulic supply pressure, hydraulic system 1

P_2	hydraulic supply pressure, hydraulic system 2
q	dynamic pressure; perturbation value of pitch rate; rigid and flexible modal coordinates
$\dot{q}$	rigid and flexible mode rates
$\ddot{q}$	rigid and flexible mode accelerations
$\bar{q}_i$	unsteady aerodynamic states associated with q
Q	pitch rate
QA	device type identifier
QSAE	quasi-static aeroelastic
Q, Q_1, Q_2	cost weighting matrices for performance variables
r	yaw rate
rad	radian
ref	reference
rms	root mean square
r_i	ith gust input reference coordinate vector
R	cost weighting matrix for control inputs; receiver
RADC	Rome Air Development Center
RAM	random-access memory
RAT	ram air turbine
RE	right elevator
RIA	right outboard aileron
RIF	right inboard flaperon
RLIM	nonlinear rate limit
ROAD	right outboard aileron, outboard
ROAI	right outboard aileron, inboard
ROF	right outboard flaperon
ROI	return on investment

ROM	read-only memory
RPS	rotor position sensor
RT	remote terminal
RTS	real-time counter
R_i	unsteady aerodynamic force matrix associated with wind disturbance
R_{ij}	cross-correlation function between gust states i and j
$\bar{R}_{ij}$	Laplace transform of R_{ij}
R_x, R_y, R_z	rotations about x, y, z axes
R_O	steady aerodynamic force matrix associated with wind disturbance
R_1	hydraulic return pressure, system 1
R_2	hydraulic return pressure, system 2
$\mathcal{R}(\tau)$	cross-correlation matrix with time lag
s	Laplace variable; second (same as sec)
sec	second (same as s)
subsec	subsection
S	Kalman filter gain matrix; standby
SAS	stability augmentation system
S/C	short circuit
SDEU	servodrive electronics unit
S/H	sample and hold
SIFT	software-implemented fault tolerance
SKC	Singer-Kearfott Corporation
S/P	serial/parallel
SRI	Stanford Research Institute
SS	signal selection
SSFD	signal selection and failure detection

SVDED	dead band
SYNC	synchronization
S_i	shear force at the ith station
t	time limit; time setting; time variable
t_f	final time
t_i	ith column of the transformation matrix T
t_o	initial time
T	cycle time; sampling period; similarity transformation matrix; threshold; transistor
TD	Teledyne
TE	trailing edge
T/O	takeoff
T-R	transformer-rectifier
TRU	transformer-rectifier unit
TTL	transistor-transistor logic
TX/RCV	transmitter-receiver
T_i	torsion at the ith station
T_u	control effectiveness scaling matrix
T_x,T_y,T_z	translations along x, y, z directions
u	incremental value of forward-speed component; control input vector
$u^*,\hat{u}^*$	optimal control solutions
u_c	control input command
u_g	longitudinal turbulence (output of Dryden model)
u_n	white noise process for longitudinal turbulence (input to Dryden model)
UART	universal asynchronous receiver/transmitter
UPI	device type identifier

UR	upper rudder
USART	universal synchronous/asynchronous receiver/transmitter
UTIL-1,-2	utility bus
U,V,W	positive integer
v	measurement noise vector
v_i	ith system eigenvector
v_{ij}	cross-variance between the ith and jth output variables
$v_{jj/i}$	variance of jth output response to ith control input
V	steady-state airspeed; true airspeed; variable displacement; velocity; volt
VA	volt-ampere
V ac	volt alternating current
VC	actuator position command voltage; voltage, common; volts, command
V dc	volt direct current
VFB	volts, feedback
VHSIC	very-high-speed integrated circuits
VLSI	very-large-scale integrated
VLSIC	very-large-scale integrated circuit
V/N	volts per Newton
VOR	very-high-frequency omnidirectional radio range
VPB	volts, pitch, channel B
VPC	volts, pitch, channel C
V/V	verification and validation
VWRS	vibrating wire rate sensor
VYRO	pitch-rate sensor (trade name)
V_{bias_A}	bias voltage of channel A

V_{bias_B}	bias voltage of channel B
V_{bias_C}	bias voltage of channel C
V_A, V_B, V_C	voltage of channels A, B, C
V_1	forward velocity
w, wg	wind input vector
wps	words per second
w_c	white noise wind input
w_g	transverse turbulence (output of Dryden model)
$\bar{w}_g$	unsteady gust states
W	vertical-speed component; watt
WLA	wing-load alleviation
W_n	white noise process for transverse turbulence (input to Dryden model)
x	system state estimate vector; system state vector
$\hat{x}$	estimated state vector
x_a	airplane state vector
x_o	initial state vector
x_u	actuator state vector
x_w	wind state vector
x_B	state vector, body-fixed axis coordinates
x_I	state vector, moving-inertial axis coordinates
X	index
XAH	actuator displacement
XBIAS	null bias
XDED	feedback dead band
X_q	dimensional variation of X force with pitch rate
X_u	dimensional variation of X force with speed

X_{wz} intermediate state variable for transverse turbulence in Dryden model

X_{α} dimensional variation of X force with angle of attack

X_{δ_E} dimensional variation of X force with elevator angle

X_{δ_i} state vector for standard controllable form

$\underline{\bar{X}}$ state covariance matrix

$\underline{\bar{X}}'$ covariance matrix for $\dot{x}(t)$

$\underline{\bar{X}}_o$ initial state covariance matrix

y output; output positions; output vector

$\dot{y}$ output rates

$\ddot{y}$ output accelerations

$\hat{y}$ estimated sensor output vector

y_i component of y

$\underline{\bar{Y}}$ output covariance matrix

$\underline{\bar{Y}}'$ covariance matrix for $\dot{y}(t)$

z system modal vector

$\dot{z}$ vertical velocity

z_1 unsteady aerodynamic states associated with $\dot{z}$

Z Z transform variable; modal response covariance matrix

$\ddot{Z}_{cg}$ vertical acceleration at body station 922.7 (cg)

Z_q dimensional variation of Z force with pitch rate

Z_u dimensional variation of Z force with speed

Z_{α} dimensional variation of Z force with angle of attack

$Z_{\dot{\alpha}}$ dimensional variation of Z force with angle-of-attack rate

Z_{δ_E} dimensional variation of Z force with elevator angle

3.2 SUBSCRIPTS

3.2.1 Subscripts Related to Velocity V or Mach Number M

B	gust penetration
D	dive
e	equivalent airspeed
MO	maximum operating

3.2.2 General Subscripts

a	airplane model
A	aileron; amplifier
c	command inputs
cg	(at) center of gravity
com	command
COL	control column
D	demodulator
E	elevator
f	final time
FB	feedback
g	gust model state
i	initial time
m	implicit or explicit model
max	maximum of
MU	minimum unstick speed condition
n	white noise
OAI	outboard aileron (inboard section)
OAO	outboard aileron (outboard section)

R	reduced-order model
ss	steady-state value of
SCS	steady aero control surfaces
SM	steady aero model
SV	servovalve
SW	steady aero wind gusts
u	control actuator model
UCS	unsteady aero control surfaces
UM	unsteady aero model
UW	unsteady aero wind gusts
w	gust model

3.3 SUPERSCRIPTS

T	transpose of
-1	inverse of
^	auxiliary variable; Kalman filter estimated quantity
′	auxiliary variable
‾	auxiliary variable

3.4 OPERATORS

Det(-)	determinant of
E(-)	expected value of
exp(-)	exponential function
Im(-)	imaginary part of
Re(-)	real part of
sgn(-)	signum or sign function

$\delta(-)$	impulse function
$\cdot$	derivative with respect to time or rate of change (superscript)
$\cdot\cdot$	acceleration or second derivative with respect to time (superscript)

3.5 SYMBOLS

0	zero matrix
℄	centerline
α	angle of attack; prescribed degree of stability
α_{wc}	difference between airplane angle of attack with respect to the air and ideal model angle of attack
β	sideslip angle
Γ	disturbance distribution matrix, gamma function
Γ, Γ_a	gust distribution matrix
δ	control surface command; control surface vector
$\bar{\delta}$	steady aerodynamic states associated with δ
δ_A	commanded aileron angle
δ_{A_c}	outboard aileron command
δ_c	column angle; control column deflection
δ_{c_i}	ith control surface command
δ_E	commanded elevator angle
$\delta_{E'}$	intermediate state variable elevator actuator
δ_{E_c}	elevator deflection command
δ_i	*ith control surface position*
$\dot{\delta}_i$	ith control surface rate
$\ddot{\delta}_i$	ith control surface acceleration
δ_{kl}	Kronecker delta

Δ	change in quantity
ΔP	difference in pressure
ϵ	state estimate error vector
ζ	damping ratio
η	fraction of semispan (2 y/b)
θ	incremental pitch angle; input matrix in modal coordinates (discrete time); pitch attitude; pitch-rate sensor output; surface angular position
θ_i	phase at the ith control input
$\bar{\theta}_i$	unsteady aerodynamic states associated with $\dot{\theta}$
λ	failure rate
λ_i	ith system eigenvalue
Λ, Λ_a	diagonal or block diagonal state matrix
μ	micro
ξ	spatial separation vector
ξ_i	flexible mode displacements
σ	mean rms turbulence intensity
σ_d	discrete gust intensity
σ_i	real part of the complex eigenvalue λ_i
σ_u	longitudinal rms gust intensity
σ_{vw}	transverse rms gust intensity
τ	time lag; time constant
τ_f	time constant of filter
ϕ	roll attitude
Φ	mode shape matrix at ith station; output mode shape matrix; state transmission matrix in modal coordinate (discrete time)
Φ_L	load distribution matrix

ω	frequency, radians
ω_i	imaginary part of the complex eigenvalue λ_i

4.0 CURRENT TECHNOLOGY ACT SYSTEM DEFINITION: TASK OVERVIEW

Section 2.0 described the relationship of the Current Technology Active Controls Technology (ACT) System Definition to the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project as a whole. This section provides a brief overview of the current technology control system work, showing its foundation, organization, approach, and progress. Sections 5.0 through 9.0 cover that same work in greater detail. Section 5.0 states the rules under which the study was performed. Section 6.0 describes the Integrated and Segregated Systems and their evaluation leading to the Selected System. The Selected System is defined in more detail in Sections 7.0 and 8.0, and its evaluation is reported in Section 9.0.

4.1 OBJECTIVES

The objectives of the Current Technology ACT System Definition are as follows:

- Define a highly reliable and low technical risk ACT control system for the ACT airplane configuration using current technology
- Support the assessment of the benefit associated with the ACT airplane by evaluating reliability, cost, and weight of the current technology system
- Identify technical risk areas and recommend hardware development and testing

The ACT functions are designed to be transparent to the flight crew except for mode control in the presence of ACT system faults. Certain functions are critical and some may be crucial, as those terms are defined in Subsection 4.3. These safety factors and the consequent dispatch implications make reliability a key requirement of this equipment.

The system design was aimed at incorporating all active control functions that could contribute significantly to fuel savings in a transport airplane design. Concurrent with this "worst case" system evolution, specific current technology systems were defined to support the Initial ACT Configuration (ref 2) and the Wing Planform Study Configuration (ref 4).

4.2 GUIDELINES

The following guidelines were established early in the current technology system work:

- Assume a set of ACT functions and control surfaces
- Provide maximum function survivability
- Isolate hardware and software by criticalities, where feasible
- Test all dispatch-critical ACT functions prior to takeoff
- Provide in-flight operation similar to conventional airplanes
- Use on-condition maintenance

In identifying ACT functions for this project, all active control functions that would add to airplane efficiency were considered. Thus the control system definition work identified a broad list of functions, including some that could require new control surfaces. These assumed functions are listed in Subsection 4.3.

Reliability is the key feature. The ACT functions are of varying criticality; isolation of any function requiring very high reliability is very desirable. All functions must be tested at preflight. The problem is to keep the incidence of delays and cancellations below an acceptable maximum level while meeting the reliability standards corresponding to the criticality of the functions.

Because active controls are transparent to the crew, normal in-flight operation will involve no special attention to the active controls, thus permitting pilots to fly the airplane in the customary manner.

The policy of on-condition maintenance (that which arises from "flight squawk" or other fault report, as opposed to scheduled maintenance) is essential to minimize the ongoing maintenance cost.

An important corollary to these guidelines is illustrated in Figure 1, which shows the time-phased relationship of the tasks of the Configuration/ACT System Design and Evaluation element of the IAAC Project. In normal control system design, airplane performance characteristics are the basis for determining the control system requirements. As Figure 1 shows, in the IAAC Project the current technology control system work began before there was an ACT airplane configuration and therefore had to proceed substantially independent of the airplane design work. For that reason, the control system work has **not** been tied to any specific airplane configuration; rather, it is aimed at providing all the active control function capability that might reasonably be required by a most-demanding ACT airplane design, which is one that can use all of the ACT functions assumed.

4.3 ACTIVE CONTROL FUNCTIONS

The active control system will use all of the Baseline Airplane primary control surfaces, as shown in Figure 4. In some instances, changes will be made to adapt them to active control use while retaining their function in primary control. An example is the split outboard ailerons, in which the Baseline Airplane outboard ailerons each become two panels, with the smaller inboard panel serving the high-frequency active control function of flutter-mode control (FMC), in addition to normal lateral control and wing-load alleviation (WLA).

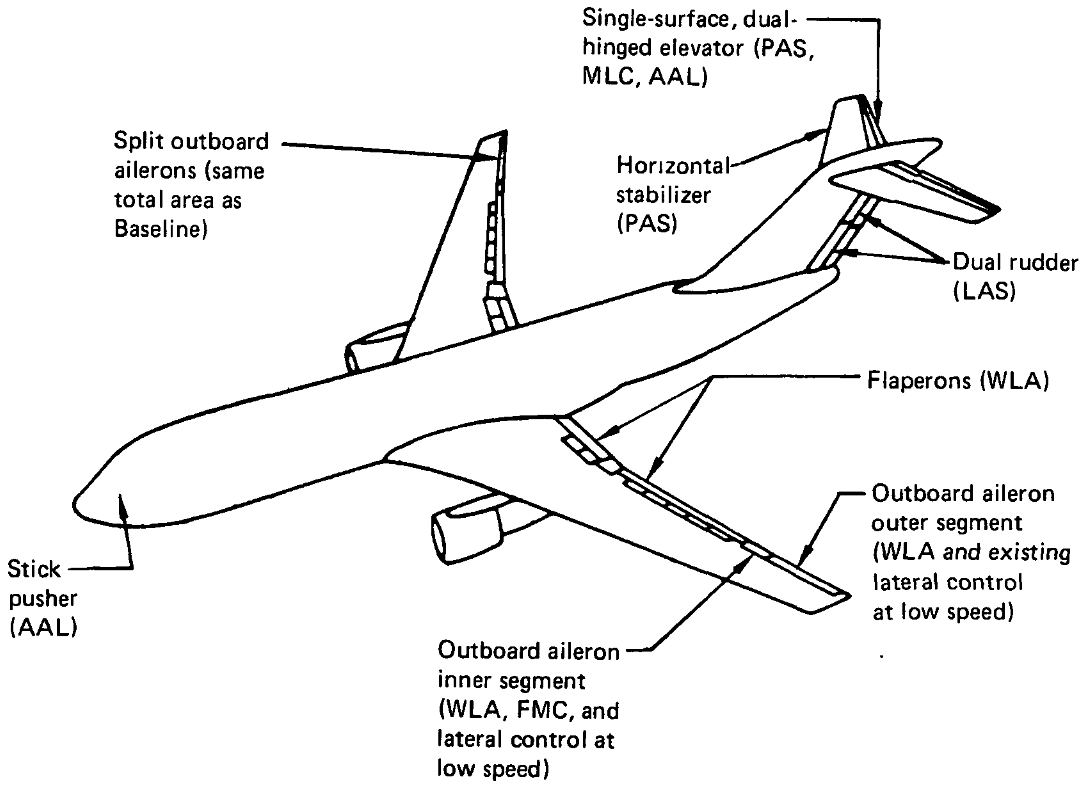


Figure 4. Assumed Active Control Surfaces

Flaperons also are shown as a feature of the active control system. They were assumed necessary for the ACT control system task, whether or not they will ultimately be required.

The active control functions are assumed to be required in the current technology ACT system to meet the specification cited in Subsection 4.2.

The two categories of function criticality represented in Table 1 are defined as follows: a **crucial** function is one whose complete loss results in an immediate unconditional flight safety hazard. A **critical** function is one whose complete loss results in a potential safety hazard that can be averted through appropriate pilot action. Using these definitions, reliability requirements for each of the ACT functions have been set based on statistical data on airplane system failures and the FAA Advisory Circular (ref 5).

Table 1. Assumed ACT Function Criticality and Reliability Requirements

	Criticality	Reliability requirement, probability of failure per 1-hr flight
Pitch-augmented stability, short-period (PAS_{SHORT})	Crucial	10^{-9}
Pitch-augmented stability, speed (PAS_{SPEED})	Critical	10^{-5}
Angle-of-attack limiter (AAL)	Critical	10^{-5}
Lateral/directional-augmented stability (LAS)	Critical	10^{-5}
Gust-load alleviation (GLA)	Critical	10^{-5}
Maneuver-load control (MLC)	Critical	10^{-5}
Flutter-mode control (FMC)	Critical	10^{-5}

The allowable failure rate of a crucial function makes its reliability essentially equivalent to that of primary flight controls in present-day aircraft. The critical function probability of failure of 10^{-5} per 1-hr flight has been fixed by consideration of dispatchability and in-flight operability requirements.

4.4 ORGANIZATION OF WORK

Work on the ACT control system has included a number of tasks; Figure 5 shows them as they relate to one another. The central horizontal row of boxes may be thought of as the mainstream effort, with the smaller boxes above and below representing supporting operations. The arrows are only partly indicative of the interrelationships; in particular, the evaluation work and the boxes representing the supporting tasks apply to all three of the system studies and not just to the Selected System.

The choice of two extremes of control system form (fig. 6) is based upon the recognition that this would identify the limits of most of the major variables that must be considered in forming the system concept. The Integrated System depends upon a single set of control computers to perform all active control functions. Digital flight control computers that have speed and capacity adequate for this central computer task are in production now.

The Segregated System depends upon multiple individual sets of control computers, one set for each active control function. This results in more computers, although they can be lower performance machines using microprocessors.

The third system is the Selected System, a merger of features of the Integrated and Segregated Systems. It represents the best compromise of advantages and disadvantages of the first two systems studied.

4.5 STATUS AND PLAN

The system work accomplished to date shows that it is feasible to implement these ACT functions using current technology system components and meet the reliability requirements. Considerable development and test work must be completed before the perceived

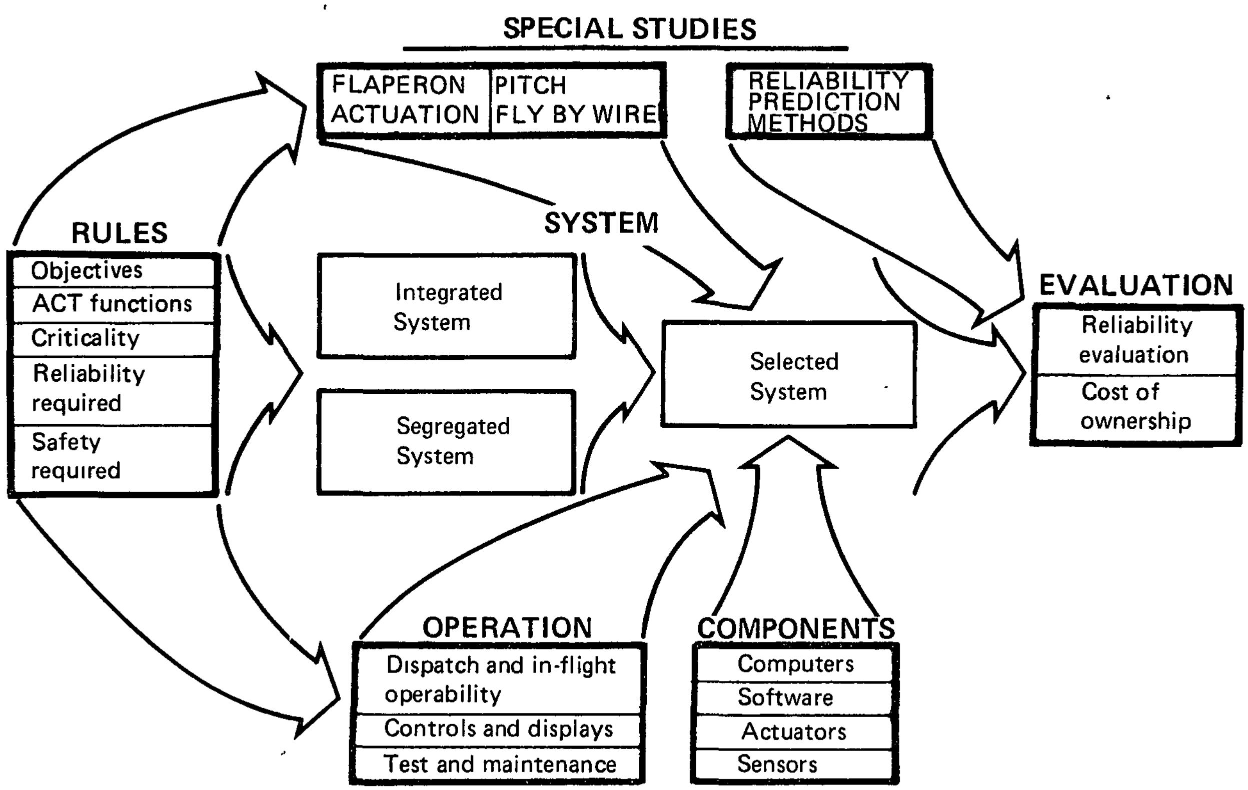


Figure 5. Current Technology System Task Overview

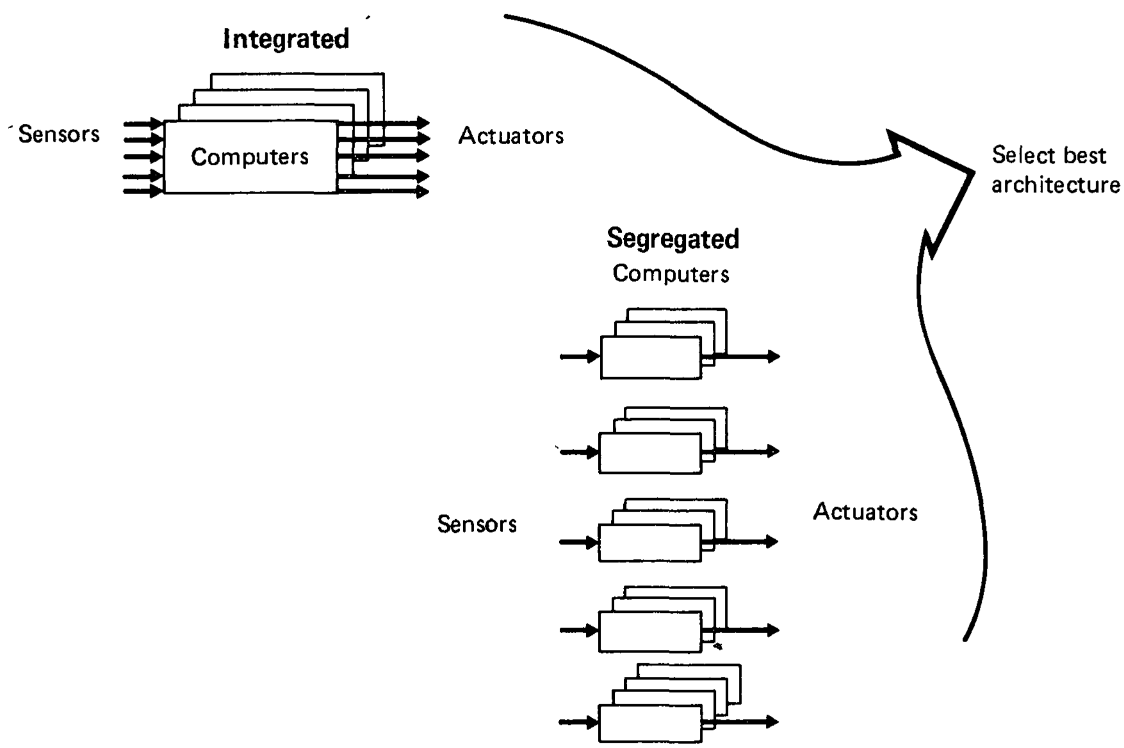


Figure 6. Approach

risks of such systems are reduced to a level commensurate with current commercial practice. The major unresolved issue with respect to digital implementation of these critical systems is software verification and validation. There is no generally accepted method to prove the software to be error free. However, a disciplined approach beginning with functional analysis and proceeding through requirements, design, coding, verification, validation, exhaustive testing, configuration control, and documentation has been shown to be both essential and effective in producing high-reliability real-time control software.

During the course of this project, a number of risk areas have been identified that require specific development and test for resolution. The final element of the IAAC Project (previously shown in fig. 3) will address this work.

It is expected that piloted simulations of the ACT airplane will be required to ensure that these systems are appropriate for a commercial transport and exhibit acceptable characteristics in the presence of failures.

Technical risk concerns may well surface in many different areas of an ACT system, and some of these concerns may not be resolved without flight test. Boeing intends to proceed toward that goal with the developmental activities identified for this project element to the maximum extent permitted by available resources.

5.0 ASSUMPTIONS AND DESIGN GROUND RULES

This section presents the ground rules imposed on the system design task and the assumptions made for the Active Controls Technology (ACT) airplane performance capability. The first ground rule of this task was to design the ACT system using only currently available hardware technology. Other ground rules were derived from Appendix A of Reference 2.

In addition to the ground rules, this section presents the various assumptions made for the characteristics of the ACT airplane. Knowledge of the detailed airplane characteristics is a prerequisite of normal flight control system design. However, because this system design proceeded in parallel with the airplane configuration design, it was necessary to estimate many airplane characteristics to support the system design. This was done early in the project history by a group of experienced engineers representing the several technical disciplines involved. Their collective engineering judgment of the effect of losing ACT functions, individually and in combinations, became the basis for the configuration-dependent portions of this section.

5.1 ASSUMED ACT FUNCTIONS

This subsection describes the active flight control functions assumed to be implemented in the ACT control system. These functions rather naturally group themselves into airplane mode control and structural mode control. Airplane mode control tends to be low frequency and involves traditional stability augmentation functions. Structural mode control may include both low-frequency functions (maneuver-load alleviation) and higher frequency functions (flutter-mode control and gust-load alleviation). Functions included to protect the airplane from entering stall condition are considered a part of ACT, although they do not operate in a continuous closed-loop mode.

The functions selected for the system design are described in Subsections 5.1.1 and 5.1.2. Subsection 5.1.3 presents the system organization.

5.1.1 STABILITY AUGMENTATION

Inclusion of augmented stability functions allows relaxation of the inherent airplane stability, which results in reduced empennage surface areas with attendant benefits in performance. The following such functions are incorporated:

- Pitch-Augmented Stability (PAS)—For the design study, it is assumed that the horizontal stabilizer surface area is reduced substantially below Baseline size, and center-of-gravity range is chosen such that the unaugmented airplane would exhibit negative stability characteristics throughout the flight envelope. A PAS function is therefore required to augment airplane longitudinal stability to provide the desired safety and handling quality. PAS includes both short-period and speed stability modes defined in Subsection 6.1.

- Lateral/Directional-Augmented Stability (LAS)—The Baseline Airplane lateral/directional characteristics are such that a yaw damper is required for good handling quality. It is assumed that there will be little or no reduction in the ACT airplane vertical fin size because the size is determined by the low-speed asymmetric power control requirement. Consequently, the handling quality improvement required above the unaugmented airplane characteristics will be similar to that required for the Baseline Airplane. The LAS function required for the ACT airplane will include the yaw damper and turn coordination functions.

- Angle-of-Attack Limiter—The ACT airplane is assumed to have "locked in" stall characteristics at high post-stall angle of attack because of the reduced T-tail horizontal control surface area, a factor not characteristic of the Baseline Airplane. The AAL function consists of two subfunctions:

    - Stick Shaker—The stick shaker warns the pilots by shaking the column and providing audio warning when the airplane approaches a stall.

    - Stick Pusher—The stick pusher prevents the ACT airplane from entering a deep stall condition by sensing angle of attack and adding an airplane nose-down force on the control column.

5.1.2 STRUCTURAL LOAD CONTROL

Control of structural loads in the ACT airplane is limited to the wing loads. There is no plan to incorporate systems to control fuselage or empennage structural loads.

- Wing-Load Alleviation (WLA)—The WLA functions have the objective of redistributing and/or reducing loads by suitable deflection of control surfaces. The resultant benefit is wing weight reduction through reduced structural design loads. WLA consists of two submodes:

    - Maneuver-Load Control (MLC)—load redistribution in response to maneuvering flight and flight through atmospheric turbulence, principally in the low-frequency spectrum

    - Gust-Load Alleviation (GLA)—alleviation of loads generated by flight through atmospheric turbulence, principally in the high-frequency spectrum

- Flutter-Mode Control (FMC)—FMC suppresses wing flutter from dive speed V_D/M_D to $1.2V_D/M_D$ by commanding appropriate control surface motions. The ACT airplane will be flutter free to V_D/M_D without FMC. The FMC system increases modal damping at the flutter frequency and actively suppresses the flutter mode.

5.1.3 ACT SYSTEM ORGANIZATION

A block diagram of the system incorporating the preceding ACT functions is shown in Figure 7. The control laws upon which it is based were developed as part of the airplane design using traditional design technique.

5.2 GENERAL DESIGN GROUND RULES

5.2.1 DEFINITIONS

This subsection defines those terms concerned with criticality of function and system failures.

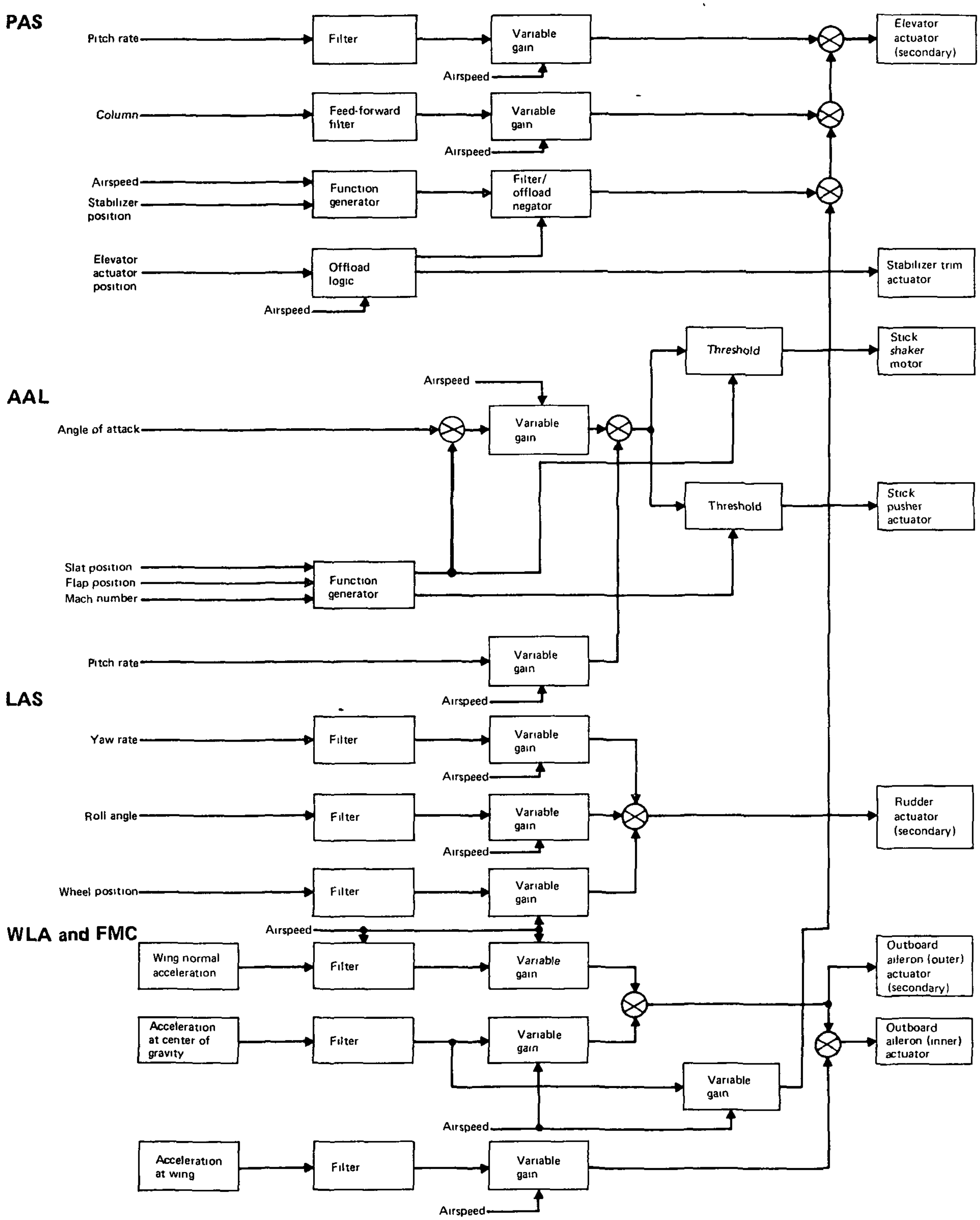


Figure 7. ACT System Block Diagram

5.2.1.1 Flight Criticality

The following terms define flight criticality.

- Flight Crucial—that function whose complete loss inevitably results in loss of the aircraft. The consequence of complete function loss cannot be averted by procedure change or flight envelope restriction.

- Flight Critical—that function whose complete loss in a specific portion of flight could result in loss of aircraft, but such loss could be averted by proper flight crew action.

- Nonflight Critical—that function whose complete loss has no impact on flight safety, but the function is considered necessary for some other requirement such as passenger comfort in rough air.

- Dispatch Critical—that function without which an airplane cannot legally be dispatched on a revenue flight.

- Workload Relief—that function that impacts neither flight dispatch status nor flight plan but that has convenience value to flight crews. Loss of function may affect precision or economy of flight but has no significant effect on safety.

5.2.1.2 Failure Survivability

This subsection defines the terms concerned with system failures.

- Fail-Operational/Fail-Operational—A fail-operational/fail-operational configuration is designed to withstand at least two independent failures and continue to function at the specified level of performance.

- Fail-Operational—A fail-operational configuration is designed to withstand any single failure and continue to function at the specified level of performance.

- Fail-Safe—A fail-safe configuration is designed such that any failure, or any combination of failures not shown to be extremely improbable, will not cause transients that exceed airplane structural limits or conditions from which a pilot with average skill cannot safely recover. Control surfaces will maintain a safe position after failure. The affected function(s) may no longer be available.

5.2.1.3 Reliability Requirement

The criticality defined in Subsection 5.2.1.1 must be interpreted in terms of reliability figures prior to the system design. The reliability requirement of a crucial function is explained in Federal Aviation Regulation (FAR) 25, paragraph 25, 1309(b), as follows:

> The airplane system and associated components, considered separately and in relation to other systems, must be designed so that the occurrence of any failure condition which would prevent the continued safe flight and landing of the airplane is "extremely improbable."

Figure 8, from the FAA Advisory Circular (ref 5), indicates that "extremely improbable" should be interpreted as less than 1×10^{-9} probability of failure per 1-hr flight.

Defining the reliability requirement of critical ACT functions is more difficult. A detailed criticality analysis is needed for the complete flight envelope under various conditions to determine that reliability requirement. For this study, the reliability requirement of each ACT function was determined by engineering judgment considering the functional criticality and the data in Figure 8. In general, the failure consequence of a critical ACT function can be defined in the range between "concern" and "emergency procedures" in Figure 8. In some specific portion of the flight envelope, the function may be in the "catastrophic total loss" category (e.g., FMC for airspeed above V_D). For this study, the reliability requirement of a critical function is defined as follows:

- General reliability requirement for critical ACT functions: failure probability $= 1 \times 10^{-5}$ per 1-hr flight

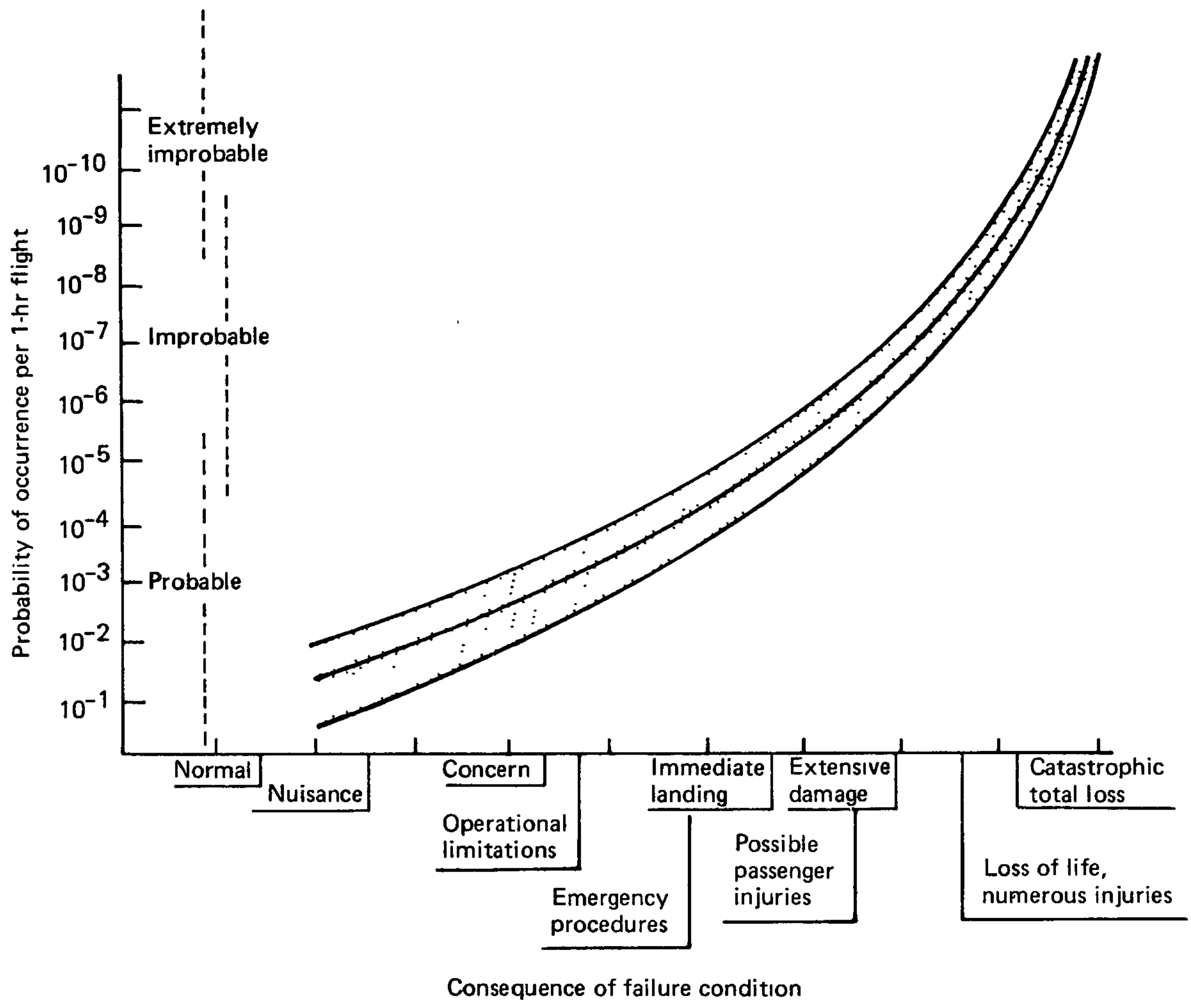


Figure 8. Relationship Between the Consequence of Failure and the Probability of Occurrence

- Reliability requirement for critical ACT functions at specific crucial conditions: (probability of function loss) x (probability of encountering specific condition) = 1 x 10^{-9} per 1-hr flight

5.2.2 GROUND RULES FOR SYSTEM DESIGN

The following items are ground rules for system design.

- The system shall be designed to reconfigure automatically in case of component failures to retain ACT functions as long as possible.

- In general, a minimum of two channels is required for operation of the ACT functions; i.e., the cross-channel comparison technique is used to detect failures. Single-channel operation may be permitted only under the following conditions:

    - Adequate self-monitor can be provided in the system.

    - In case of self-monitor failure, the pilot is able to detect system failure by other cues such as airplane response time, damping, etc.

    - Without degrading system performance, authority and rate limits can be applied to the actuators to prevent airplane structural damage in case of hardover or oscillatory failures.

- As an objective, the ACT functions shall be isolated in hardware and software in accordance with their criticalities to prevent failure propagation and to avoid extensive requalification test of functions other than those repaired.

- Redundant hardware and software elements in the system (serving the same ACT functions) shall be interchangeable.

- As an objective, fly-by-wire techniques shall be used wherever benefits may be realized by the integration of primary flight control and the ACT control system.

- It shall be an objective to integrate the ACT system with other dispatch-required flight control systems wherever benefits may be realized in terms of weight, reliability, and cost.

- The system shall be designed for the most demanding ACT airplane configuration anticipated from the system design point of view. Data from appendix A of the Initial ACT Configuration (ref 2) and Wing Planform Study (ref 4) shall be used for the system design where appropriate.

- If all ac power is lost, sufficient storage battery capacity shall be provided for 30 min of crucial function operation, and the computers shall operate properly without their cooling system for that time.

5.2.3 GROUND RULES FOR OPERATION

This subsection lists ground rules for operation of the ACT airplane.

- In-flight operation of the ACT airplane shall be similar to that of a conventional jet transport airplane; i.e., the ACT systems are transparent to flight crews except for flight mode control following function loss.

- In case of system failures in flight, the failure status and operational status in accordance with minimum equipment list (MEL) restrictions shall be displayed to the flight crew. To assist the pilot in making his decision in response to failure, advisory information shall be displayed in an appropriate manner.

- Degradation of system functions that requires immediate crew response shall result in a warning display and aural signal from the baseline aircraft warning system.

- All dispatch-critical ACT functions shall be tested prior to every dispatch. The preflight test shall detect and identify failures to the line replaceable unit (LRU). Time required to complete the preflight test shall be minimized. As an objective, the time for preflight test shall be less than 2 min.

- All fault data detected during flight and preflight test shall be stored in nonvolatile memory and be available to support maintenance.

- The results of the preflight test shall be displayed in a manner that shows the failure and operational status of the ACT system in accordance with MEL restrictions to assist the pilot in making his decision regarding dispatch.

- The ACT system shall be maintained in accordance with the "on condition" concept. Maintenance shall be performed on only those failures identified by preflight test or flight squawk (log book).

- Through-flight maintenance shall be restricted to only those items reported failed and required for dispatch. As an objective, through-flight maintenance shall be completed within a 20-min period. Any other maintenance may be deferred to an overnight stop.

5.3 OPERATIONAL CONSIDERATIONS

A primary design goal of the ACT control system is to configure a system that can be operated safely and economically in flight and on the ground in the commercial transport environment. This subsection describes design considerations related to safety and operability. Subsection 5.3.1 deals with safety, where system criticality and reliability requirements become very important. Subsection 5.3.2 describes ACT airplane operability, which includes dispatch and flight restrictions when ACT functions become inoperative.

5.3.1 SAFETY AND ASSUMED FUNCTIONAL CRITICALITY

The safety of the ACT airplane is greatly influenced by the characteristics of the basic airplane with the ACT functions inoperative. Knowledge of structural margins and stability and control characteristics with the ACT functions inoperative is essential to the ACT system design. Because the system design in the IAAC Project proceeded in parallel with the basic airplane design, airplane characteristics were estimated as noted in Section 5.0 using collective engineering judgment. Based on these estimated characteristics, the functional criticality and reliability requirements were defined and are described in the following subsections.

The ACT airplane is assumed to have an aft center of gravity (cg) and a small horizontal tail to improve lift-to-drag ratio and to reduce weight. The static and dynamic longitudinal instability attributed to the aft cg and the small tail necessitate a PAS function to provide acceptable handling qualities. The criticality of PAS is dependent upon how far the cg is moved aft and how much the tail size is reduced. For this study, the airplane is assumed to be configured so that the short-period-mode PAS is crucial; i.e., the airplane will be lost if the short-period PAS fails completely. Speed control PAS is critical; i.e., loss of the speed PAS may result in some safety hazard, but it can be averted by proper pilot action or flight envelope change.

In addition to the pitch axis instability, the ACT airplane is assumed to have "locked-in deep stall" that is attributed to the reduced horizontal tail. The AAL function is provided to prevent the airplane from getting into a locked-in stall condition. Once the airplane is in a locked-in stall, the pilot would be unable to recover from that condition; i.e., the

AAL becomes a crucial function in locked-in condition. However, because the simultaneous occurrence of complete AAL loss and excessive angle of attack because of either a pilot's inadvertent maneuver or atmospheric turbulence is extremely improbable, the AAL function is designated as critical. Inadvertent activation of the AAL system could result in a disastrous condition; therefore, AAL series devices (subsec 6.1) that prevent uncalled-for operation are designated as crucial.

Lateral control of the ACT airplane is assumed to be similar to conventional airplanes. Thus, the LAS function of the ACT airplane is similar to the conventional yaw damper in criticality. Complete loss of the LAS in critical flight condition could result in loss of the aircraft, but such loss can be averted by proper crew action to restrict the airplane flight envelope. Therefore, LAS is designated as critical.

The WLA function will redistribute and/or reduce the wing loads caused by pilot maneuver or atmospheric disturbance. Presence of the WLA function makes it possible to reduce wing-box structural material. However, the primary structure of the ACT airplane is designed to have sufficient ultimate strength to sustain the design limit load even when the WLA function becomes inoperative. This will allow continuation of the normal flight schedule after complete WLA loss in the air. The airplane, however, cannot be dispatched if the WLA is inoperative. WLA is defined as a critical function.

The basic ACT airplane is designed with sufficient flutter margin for safe flight to V_D/M_D without FMC. FMC will keep the airplane flutter free from V_D/M_D to $1.2V_D/M_D$. Thus, FMC is a crucial function for airspeed above V_D/M_D. However, simultaneous occurrence of complete FMC loss and airspeed in excess of V_D/M_D is extremely improbable. Therefore, FMC is defined as a critical function.

5.3.2 AIRPLANE OPERABILITY WITH ACT FUNCTION LOSS

Failures of the ACT system may delay or cancel airplane dispatch or restrict flight or necessitate diversion to another airport depending upon the nature of the system failure. The effect of ACT system failures on airplane operability is one of the important economic factors to be considered in the system design. As part of the system design, the effect of system failure on airplane operation was analyzed.

The assumed airplane performance capability when a single ACT function is completely inoperative is described as follows:

- Short-Period PAS Loss—The airplane will be lost if short-period PAS is lost in the air. If the function is lost on the ground, the airplane cannot be dispatched until the system is repaired.

- Speed PAS Loss—The airplane must be operated within a restricted flight envelope when speed PAS is lost in the air. If the function is inoperative on the ground, the airplane can be dispatched with flight restriction.

- LAS Loss—The airplane must be operated within a restricted flight envelope when LAS is lost in the air. If the function is totally inoperative on the ground, the airplane cannot be dispatched because LAS is required for limiting structural loads.

- AAL Loss—The airplane can continue normal flight schedule after AAL is lost in the air; such loss does not degrade airplane handling qualities. However, the pilot will be informed of system status and will continue the flight with special caution. If AAL is inoperative on the ground, the airplane cannot be dispatched because of loss of safety margin.

- WLA Loss—The airplane can continue normal flight schedule after WLA is lost in the air because the airplane structure ultimate strength exceeds the design limit load. If WLA is inoperative on the ground, the airplane cannot be dispatched because the airplane structural strength is less than the design ultimate load.

- FMC Loss—Airplane speed must be reduced upon loss of FMC to provide an adequate speed margin for upset. If the function is inoperative on the ground, the airplane can be dispatched with flight restriction.

The ACT airplane must meet dispatch requirements as follows:

- Airplane schedule reliability must be at least 98.7%. Not more than 5% of the unreliability shall be attributable to ACT system failure.

- The airplane shall be dispatchable with any one ACT system component failed.

Subsection 8.4.1 contains a detailed development of the airplane operability based upon the preceding consequences of ACT function loss; it includes multiple-function loss effects and effects of some partial-function losses that reduce operability.

6.0 COMPARISON OF SYSTEM DESIGN CONCEPTS

In preparation for selecting an Active Controls Technology (ACT) system design concept, two system architectures were studied using the ground rules and assumptions described in Section 5.0. The first step of the study was to design an Integrated System, where all ACT functions are processed in the same computer set. This system was designed using currently available flight control computer hardware with minimum modification of the basic mechanical primary control system. The second step was to design a Segregated System, where each ACT function is processed in a separate dedicated computer set. The Segregated System computers were designed using currently available large-scale integrated (LSI) circuit components. In addition to computer design, an effort was made to improve the actuation system. The third step of the study was to define the Selected System design concept by combining good features of the Integrated and Segregated Systems and the results of the actuation system design work. Finally, a brief study was conducted to investigate the benefit of pure fly-by-wire (FBW) pitch axis control, because airplane safety had already been made dependent upon the short-period pitch augmentation, an FBW control function.

All the systems studied sense airplane motion and deflect appropriate control surfaces to perform the ACT functions. In airplane design, selection of sensor signals and control surfaces is an important task, requiring many iterative cycles between the airplane design and system design. In the IAAC Project, the sensor signals and the control surfaces needed to perform each ACT function were defined in another task, Configuration/ACT System Design and Evaluation. The sensor signals, control surfaces, and system organization selected in that task are shown in Figure 7. Major tasks of the Current Technology ACT System Definition include definition of system configuration, selection of sensor hardware, definition of computers, and definition of actuation systems to implement the ACT functions and meet the requirements of system safety and economical operation.

Subsection 6.1 describes the Integrated System, and Subsection 6.2 describes the Segregated System. Subsection 6.3 compares these two contrasting concepts and describes the Selected System design concept. Subsection 6.4 briefly describes the Selected System. Finally, Subsection 6.5 describes the pitch axis FBW system and compares this system with a conventional cable system.

6.1 INTEGRATED SYSTEM

The Integrated System employs a single computer set to process the five ACT functions: pitch-augmented stability (PAS), angle-of-attack limiter (AAL), lateral/directional-augmented stability (LAS), wing-load alleviation (WLA), and flutter-mode control (FMC). PAS consists of short-period and speed (phugoid) mode control. It was assumed that the short-period PAS is a crucial function (i.e., loss of the function results in loss of the airplane) and that the speed PAS is critical (i.e., loss of the function may result in a flight safety hazard, but it can be avoided by appropriate flight crew actions). The crucial function must meet the failure rate requirement of less than 1×10^{-9} per 1-hr flight. A reliability analysis was conducted on the short-period PAS to select appropriate redundancy for each element to meet the reliability requirement. The results indicated that the short-period PAS requires four channels to meet the specific reliability requirement; this established the requirement of four central computers.

Figure 9 is a block diagram of the Integrated System. Figure 10 shows the location of system sensors.

The system shares the triple digital air data computer (DADC) and the triple inertial reference system (IRS) with the autopilot and display functions. These two sensors provide multiple signals for the ACT functions via Aeronautical Radio Incorporated (ARINC) 429 digital communication links. A dedicated pitch-rate sensor, used in conjunction with the triple IRS, serves to implement the quadruply redundant short-period PAS function. The remaining dedicated sensors; i.e., accelerometers at several wing locations, provide triply redundant analog signals for WLA and FMC functions. The sensors are coupled to their assigned digital computers, where data are then transmitted to other computers over cross-channel communication links to satisfy the redundancy requirement. These links are dedicated one-way, high-speed data buses. It should be noted that all four computers receive all sensor signals via the cross-channel communication link even though one of the computers is not directly connected to a sensor. Triply redundant critical sensors are connected to the four computers in a manner that minimizes the probability of multiple-function loss with two computer failures.

Each computer consolidates all input signals (analog, digital, and discrete) in a signal selection and failure detection (SSFD) process. The signal selection process uses midvalue

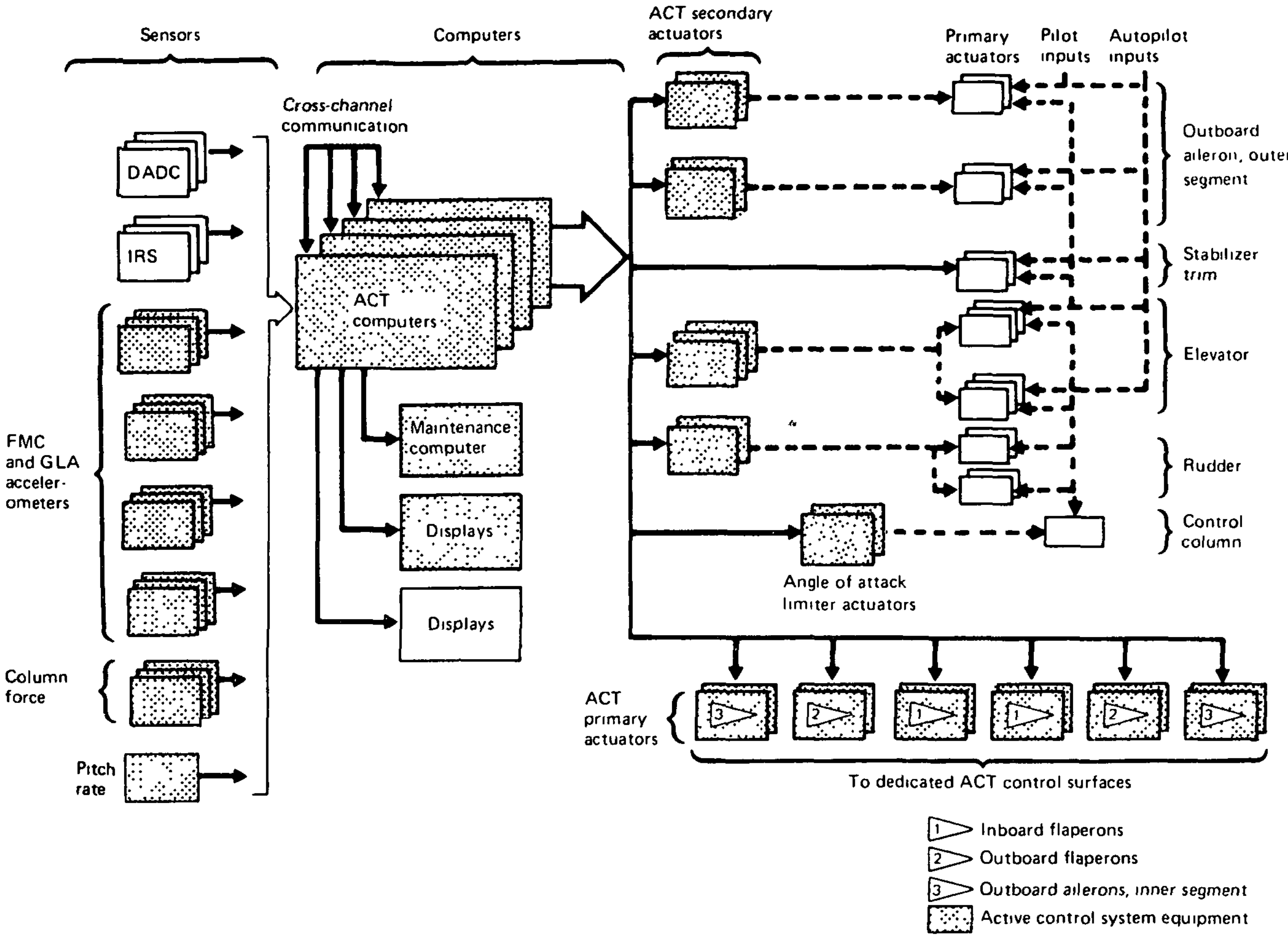


Figure 9. Integrated System Configuration

selection for three sensor signals and average value selection for two sensor signals. A four-sensor set is treated as three with one operating standby. The failure monitor is a dynamic and static comparison of inputs and selected signal to single out any value that is inordinately different from the selected value. Because the Integrated System computers process many ACT functions that require different redundancy levels depending upon function criticality and failure conditions, the SSFD process is varied as necessary to handle the different types of sensor signals.

The Integrated System computers are frame synchronized such that each executes the same function essentially at the same time. Using the SSFD process and frame

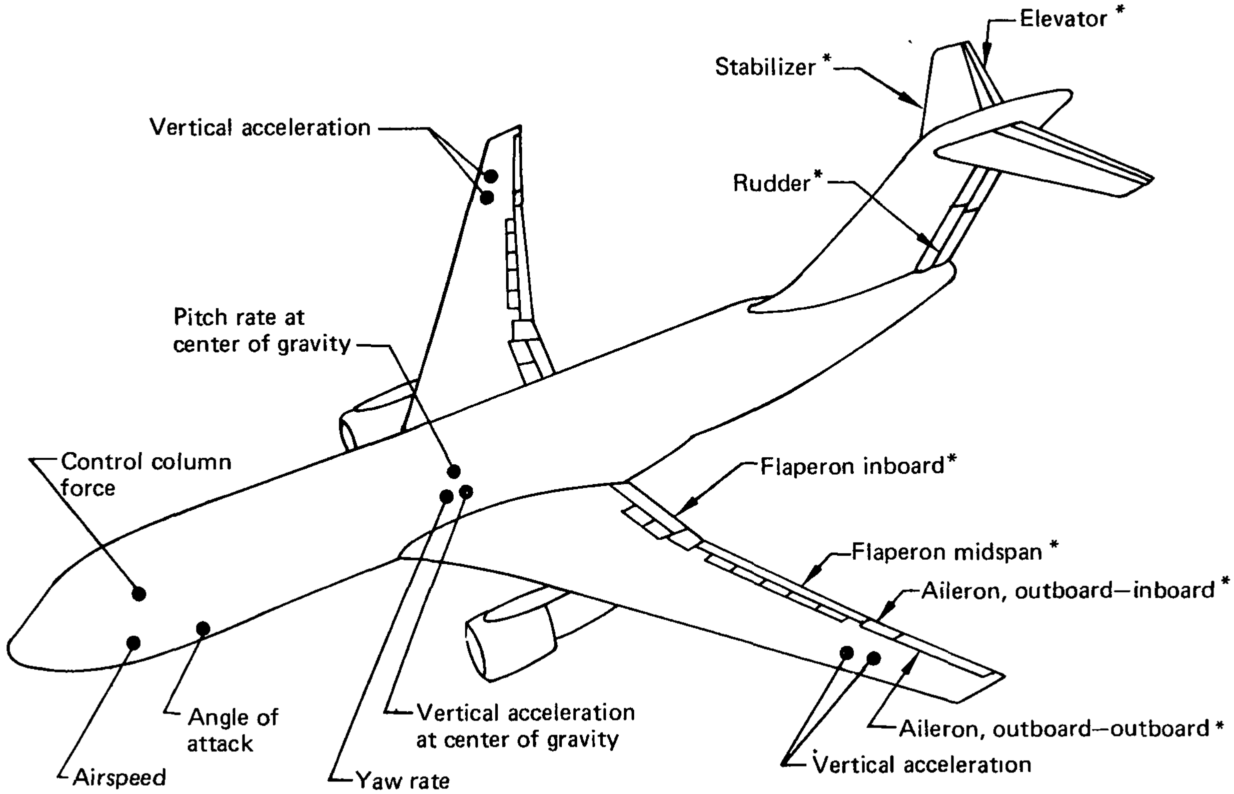


Figure 10. ACT Sensor Placement

synchronization, the quadruple computers transmit nearly identical command signals to the ACT actuators, reducing force fighting in the actuation system and simplifying the failure detection algorithm for passive failures.

The architecture of the Integrated System computer retains many of the Airborne Advanced Reconfigurable Computer System (ARCS) (ref 6) features, such as the bus-oriented structure, autonomous input/output operation, and microprogrammed control processing. The basic change from the ARCS to the integrated computer is in partitioning crucial and critical functions. This change is essential because of the extremely high reliability required of the crucial function. Comprehensive self-testing capability is provided to meet flight safety requirements and support maintenance. Self-test features include monitoring of internal power supplies, machine timing, processor capability, memory operation, and input/output operation. Specific tests include program memory sum checks, parity checks, arithmetic overflow detection, and wraparound testing of analog and discrete input/output.

The redundant command signals generated in the quadruple computers are sent to the various control surfaces through the ACT actuation systems. The signals are consolidated at the actuators to perform mechanical voting. Three actuator configurations are used in the Integrated System design: ACT secondary actuator configuration, ACT dedicated force-displacement summed actuator configuration, and stick pusher actuator configuration.

ACT Secondary Actuator Configuration—A side-by-side force-summed secondary actuator concept (fig. 11) was chosen for the elevator, rudder, and outboard aileron (outer section) actuation systems. The output of the secondary actuator is series summed with the pilot's mechanical control signal to form a command input to the power control unit (PCU). The PAS and maneuver-load control (MLC) functions use the ACT secondary elevator actuator to control the elevator. The command signals of these two functions are summed in each of the quadruple computers before sending the signal to the actuator. Three actuators provide sufficient reliability to meet the requirements of crucial functions when comparison of valve spool position to that predicted by a mathematical model in the computer is used to augment cross-channel comparison for failure detection. Similarly, two actuators are sufficient for critical functions. Each actuator has a conventional two-stage, low-pressure-gain electrohydraulic servovalve operating a single ram. Valve spool and ram position are fed back to each ACT computer for servo-loop control and failure detection. The mathematical actuator model in each computer receives the summed ram position feedback and combines this with the command signal to compute servovalve position.

ACT Dedicated Force-Displacement Summed Actuator Configuration—The force-displacement summed actuator configuration (fig. 12) was chosen as the PCU for the dedicated ACT control surfaces, which include inboard and outboard flaperons and the inner section of the outboard aileron. The force-displacement summed actuator was designed to remain fully operational with decreased dynamic performance after one electric and one hydraulic failure. The actuator configuration is an FBW implementation in that electric signals from the computers directly command the control surface. The command signals from the computers are magnetically flux summed in the four first-stage electrohydraulic servovalves, two per hydraulic system. Each group of two first-stage valve outputs is mechanically position summed by a linkage.

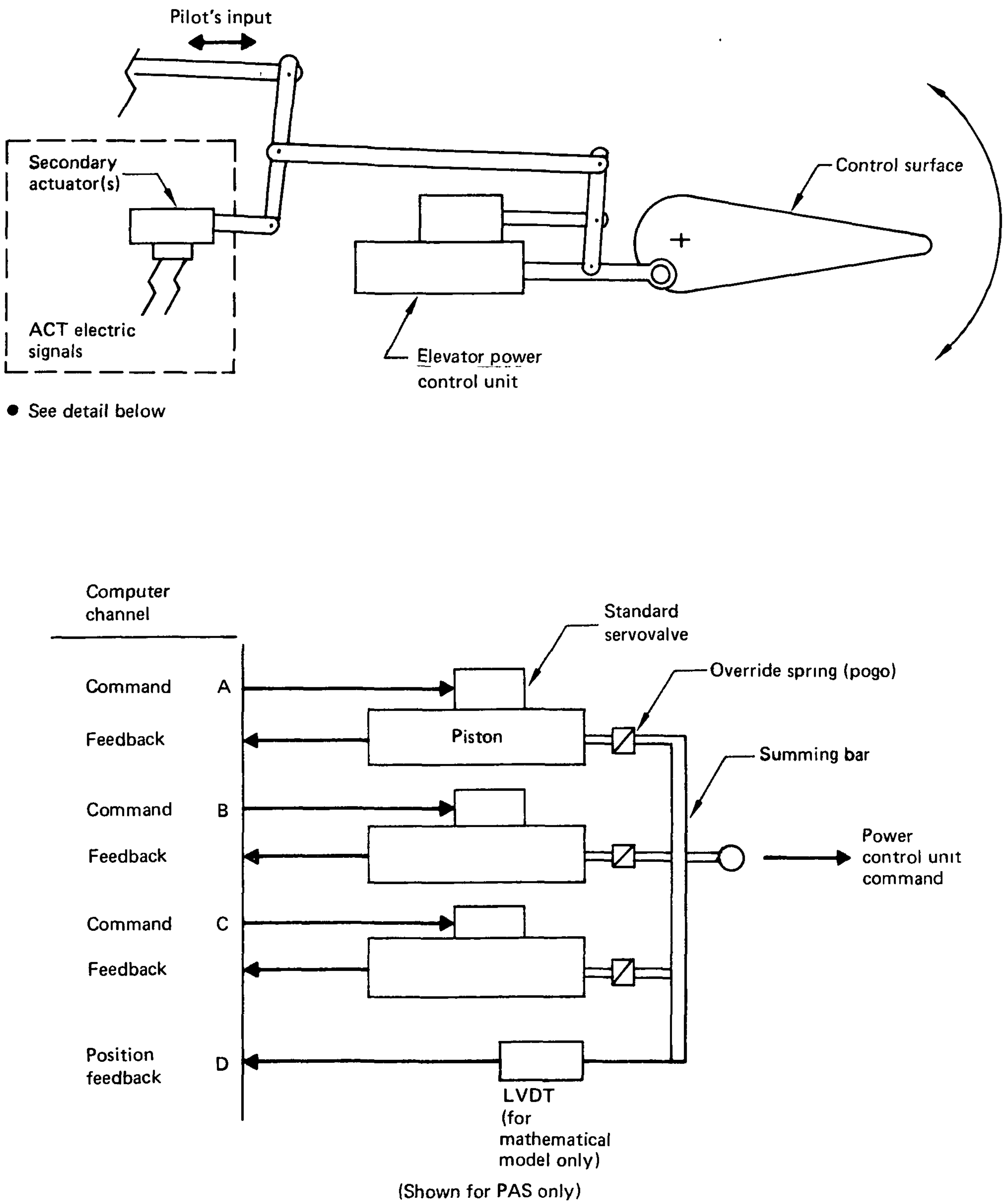


Figure 11. ACT Secondary Actuation Configuration

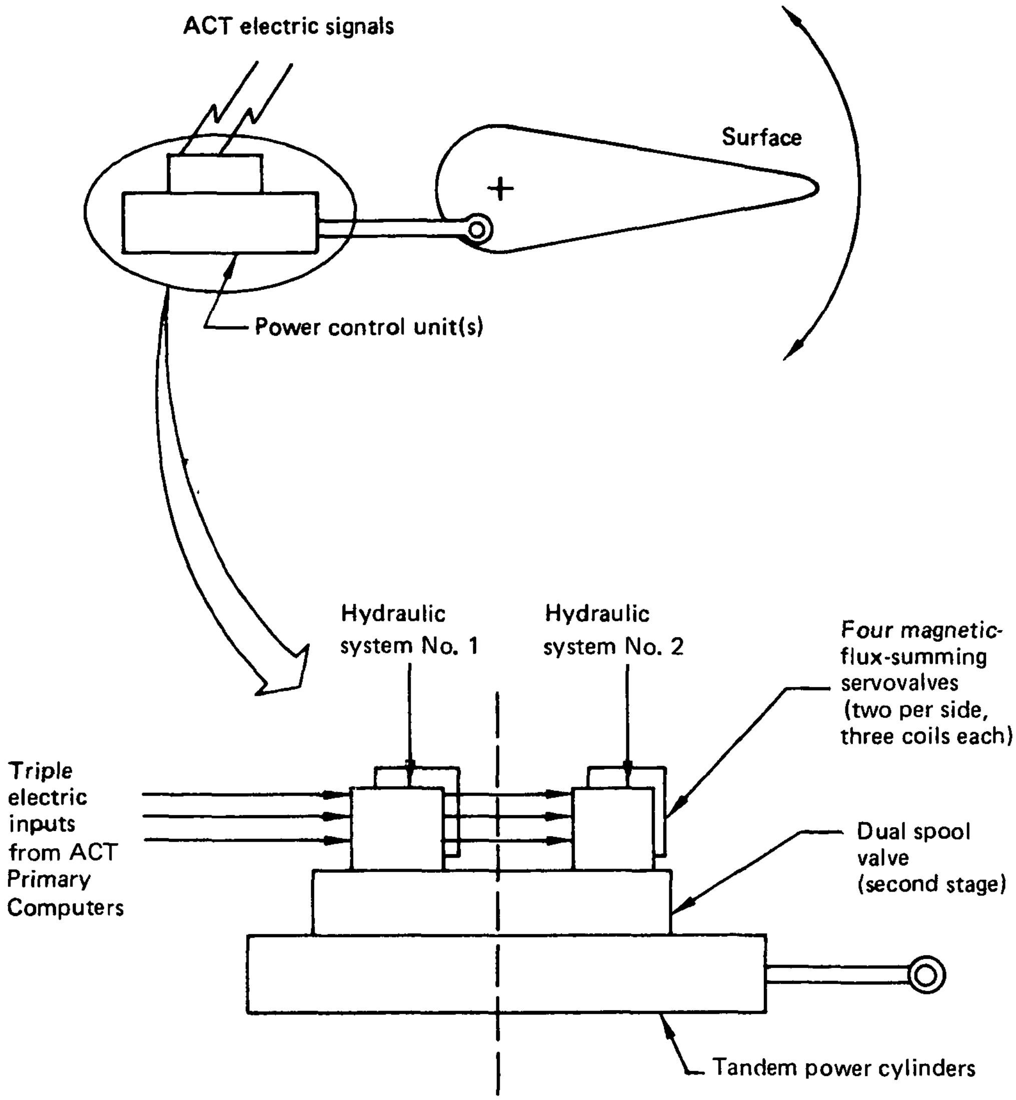


Figure 12. ACT Dedicated Force-Displacement Summed Actuator Configuration

The second-stage valve spool is controlled by force summing the resultant mechanical output of each first-stage linkage. The second-stage valve-to-main-output ram power application is the same as in conventional dual-tandem actuation.

Stick Pusher Actuator Configuration—The stick pusher (fig. 13) uses three sensors, four computers, a dual-tandem floating actuator, and two pneumatic power sources. The actuator exerts the same force when pressurized by either one or both sides. The installation linkage is such that the force exerted on the control column continuously decreases as it travels forward; 356N (80 lbf) exerted at the full aft position reduces to 178N (40 lbf) at the full forward position. Each dual-pneumatic power source consists of a nitrogen bottle at 13 788 kPa (2000 lbf/in^2) and a regulator that reduces the pressure to 3447 kPa (500 lbf/in^2) required for actuation. Two series solenoids, each signaled by an ACT computer, must be opened before either actuator is operated.

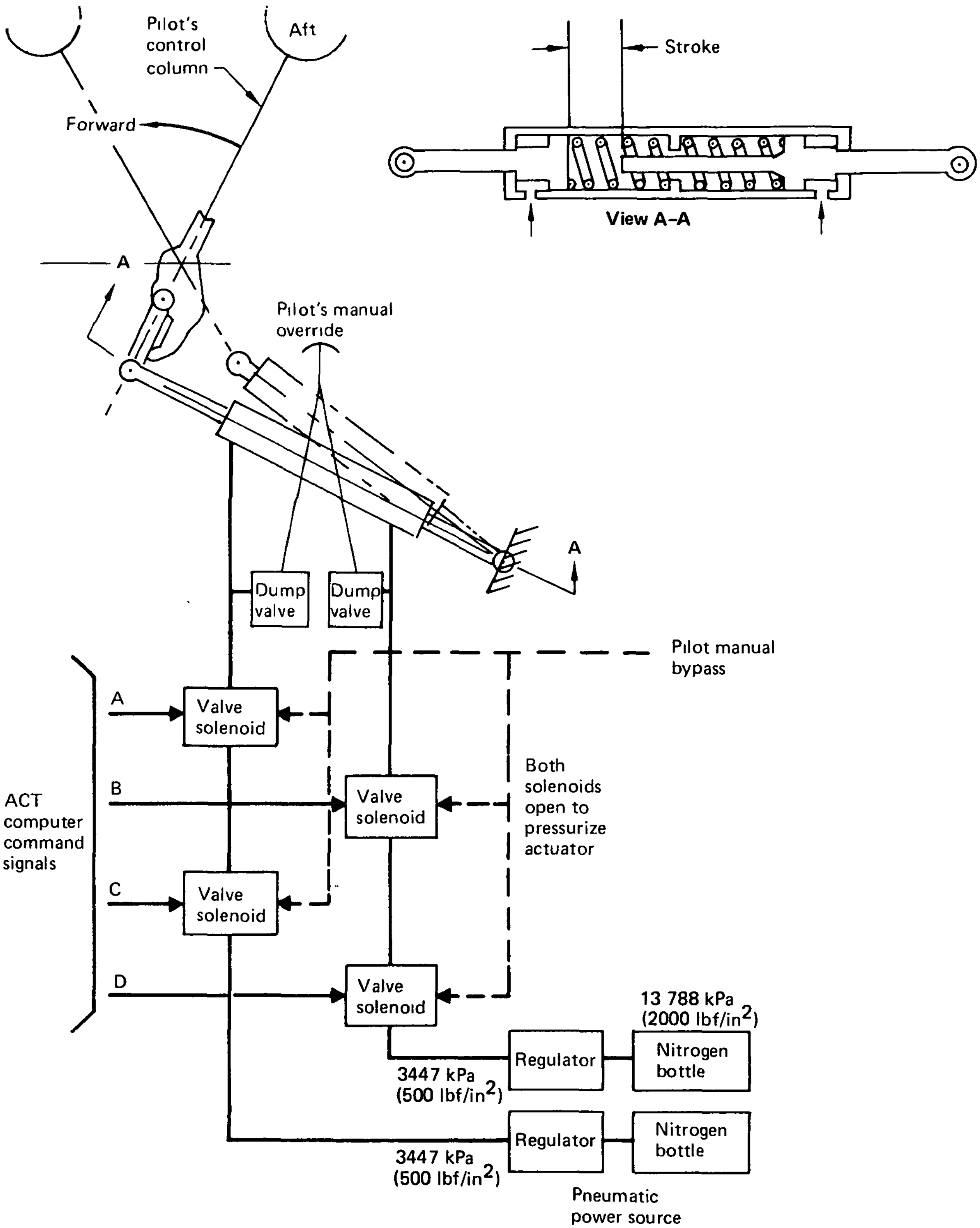


Figure 13. Stick Pusher Actuator Concept

6.2 SEGREGATED SYSTEM

The Segregated System design concept employs a separate dedicated computer set to implement each ACT function. For the Segregated System to be cost effective, a simple, inexpensive, and reliable computer is needed. An attempt was made to use LSI technology extensively to achieve this goal. The majority of the sensors and all the actuators defined for the Integrated System will remain unchanged for the Segregated System. Figure 14 is a block diagram of the Segregated System. Nineteen computers are used to compute control laws, provide system testing, and monitor for failures of the ACT functions. Two ACT Management Computers are added to process crew communication logic and to store data on detected faults.

The crucial short-period PAS is mechanized in dedicated quadruple computers to meet the fail-operational/fail-operational requirement. Four dedicated ACT pitch-rate sensors are provided to serve the crucial short-period control law. The dedicated pitch-rate sensors and dedicated short-period PAS computer essentially separate the crucial function from the remaining ACT functions. This separation is desirable considering maintenance, testing of the crucial function, and protection from potential propagation of noncrucial function failures to the crucial short-period functions.

The remaining sensors shown in Figure 14 are identical to those of the Integrated System. The interface between these sensors and the Segregated System computers is different from that of the Integrated System. The triple IRS and DADC sensors are shared by many ACT functions, as in the case of the Integrated System. Their signals are sent to each segregated computer set via ARINC 429 digital communication links. The triple digital signals from the DADC and IRS are dedicated to their respective digital computers in each set, where data are transmitted to other channels via cross-channel communication links. Each analog sensor is cross strapped to computers; i.e., each analog sensor is directly connected to all of the redundant computers. The input signals (digital, analog, and discrete) are processed by the signal selection algorithm to create a signal for computing control laws and processing other functions. The algorithms are similar to those of the Integrated System. Primary failure detection in the Segregated System is provided by cross-channel comparison. Cross-channel monitors are provided for sensor input, digital computer output, and actuator position, as in the case of the Integrated System.

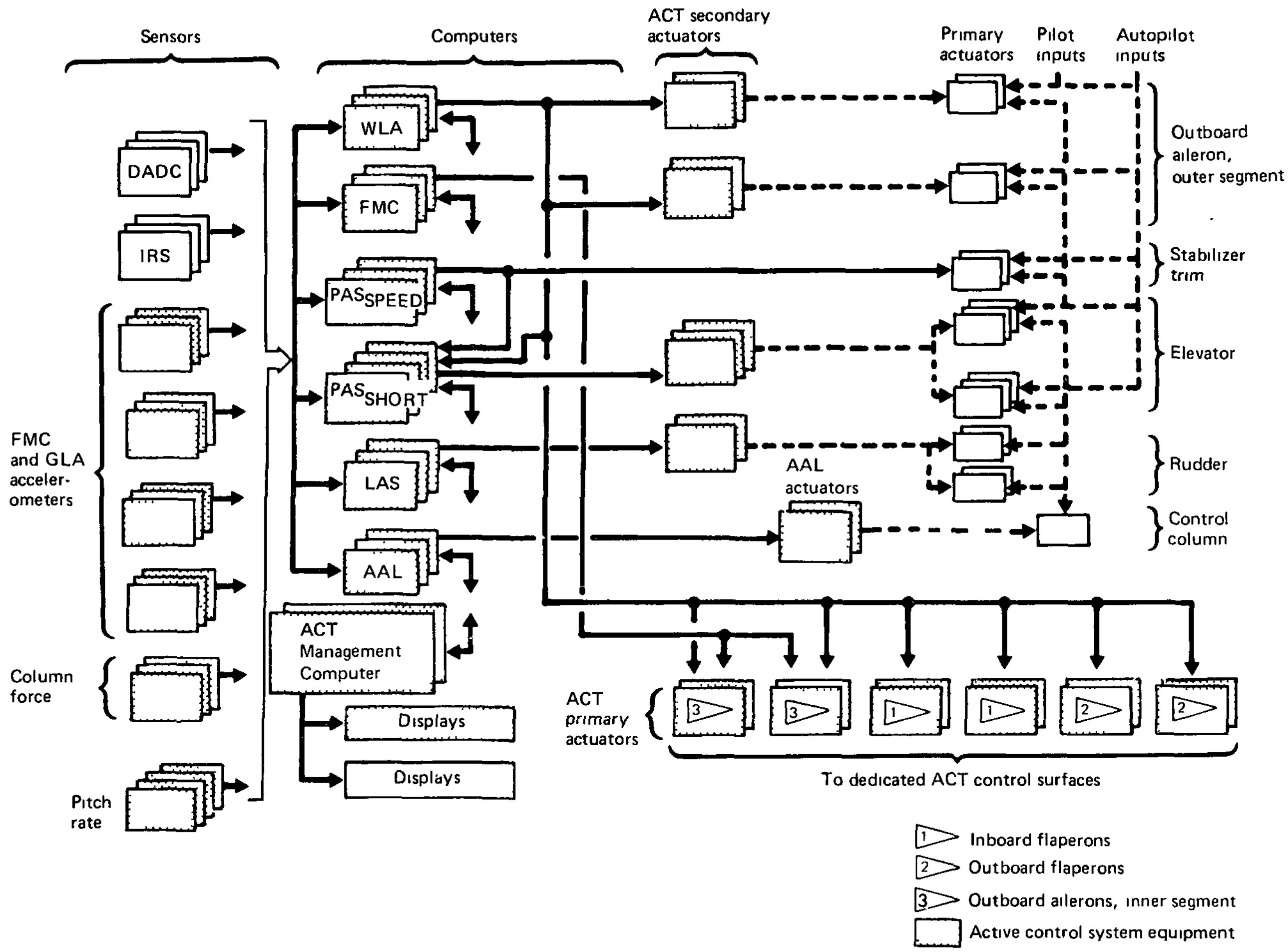


Figure 14. Segregated System Configuration

To keep the Segregated System computers as simple as possible, monitoring is largely done in software and additional hardware for inline monitoring is kept to a minimum. The computers of the crucial short-period PAS system run asynchronously. Because the computers are not synchronized, each samples the input data and computes the control laws at its own time. Thus, outputs of the redundant computers may differ due to time skew between them. Feasibility of this asynchronous ACT computer operation was studied by simulating the Segregated System in the laboratory. Test results indicate no appreciable difference between the outputs of the synchronous and asynchronous systems. It should be noted that the ACT functions described in Subsection 5.1 do not have any integrating functions, a key factor in successful asynchronous operation.

The FMC computer requires the highest sampling rate to process the high-frequency filters with acceptable tolerance in phase and gain margin. The WLA computer has the most computation to perform because the function has the largest number of sensors and control surfaces among the ACT functions. The short-period PAS is a crucial function and requires the highest reliability among the ACT functions. A study was conducted to define a microcomputer with speed, word size, and memory suitable for implementing the ACT functions. Some of the functions could be mechanized using an 8-bit microprocessor. However, to standardize the hardware and thus reduce the overall cost, a 16-bit single-chip microprocessor was selected as the Segregated System computer. Table 2 shows the requirements of each ACT function computer. The segregated computer has the following self-tests:

- Power monitor
- Parity check
- Sum check

Table 2. Segregated System Computer Requirements

Function	Memory requirements		Sampling rate, ms	Minimum CPU performance required, K ops
	Real time	Test and maintenance, words ROM		
PAS_{SHORT}	3100 words ROM 128 words RAM	5700	10	75
PAS_{SPEED}	2950 words ROM 128 words RAM	5350	10	60
LAS	3750 words ROM 128 words RAM	5400	10	75
FMC	3750 words ROM 128 words RAM	5350	5	125
AAL	3170 words ROM 64 words RAM	5350	10	55
WLA	4750 words ROM 512 words RAM	5500	10 GLA 20 MLC	120

6.3 COMPARISON OF INTEGRATED AND SEGREGATED SYSTEMS

The Integrated and Segregated Systems described in the previous subsections are compared to identify advantages and disadvantages of the two contrasting design concepts. From the results of the comparison, the Selected System design concept was defined.

The following subsections describe the comparison results.

6.3.1 PERFORMANCE

Laboratory tests were conducted to evaluate the performance of the Integrated and Segregated Systems. These two schemes were simulated in the laboratory using quadruple General Electric MCP-701A flight control computers. Airplane and actuation systems were simulated using a Boeing-owned ECLIPSE digital computer and an analog computer. The tests included cross-channel communication, synchronization, and SSFD of the quadruple Integrated System. These functions were tested open loop and closed loop with the airplane and actuator simulator and showed satisfactory results. A study was conducted in the laboratory to verify that the integrated computer can process all ACT functions without degrading the response. The integrated computers serve all ACT functions at their specified sampling rates with margins.

Asynchronous operation of PAS and FMC was studied using the MCP-701A to simulate the segregated computers. Special attention was paid to computer drift during 1-hr flights. There was no appreciable difference between the synchronized and asynchronized computer outputs after computing the PAS and FMC control laws for 1 hr. If the control laws had an integration algorithm, the asynchronous system might drift and need some equalization between the channels. The survey of available microprocessors concluded that current models are acceptable for segregated ACT application.

6.3.2 WEIGHT ANALYSIS

Because 19 computers are used to implement the segregated ACT functions, the weight of the Segregated System increased considerably as shown in Table 3. Four dedicated pitch-rate sensors were used in the Segregated System, but their weight is insignificant compared with that of the computers.

Table 3. Comparison of Integrated and Segregated Systems

Item \ System	Integrated System	Segregated System
Computer weight	4 control computers at 11.3 kg (25 lb) each Total: 45.4 kg (100 lb)	19 control computers at 6 kg (13.25 lb) each 2 management computers at 2.7 kg (6.0 lb) each Total: 119.6 kg (263.8 lb)
Memory requirement	32K ROM 2K RAM (128-word 8-bit non-volatile memory for maintenance information)	16K ROM and 64- to 512-word RAM for control computers plus 32K ROM and 1K RAM and 128-word 8-bit nonvolatile for management computer
Computer reliability (assumed)	6800-hr MTBF	6800-hr MTBF
Probability of restricted flight	2.5×10^{-3} per 1-hr flight	3.3×10^{-3} per 1-hr flight
Probability of flight diversion	7.1×10^{-4} per 1-hr flight	2.6×10^{-4} per 1-hr flight
Probability of dispatch delay	3.8×10^{-4} per dispatch	1.5×10^{-4} per dispatch
Incremental cost per airplane (to airline)	$274 000	$390 000
Expected return on investment	25.1%	22.8%

6.3.3 RELIABILITY ANALYSIS

6.3.3.1 Flight Restriction and Diversion

Subsection 8.4.1 describes the functional failure conditions that lead to various flight restrictions. Because there are so many elements in the Segregated System, it is extremely difficult to assess the probability of occurrence of a flight restriction. The computer-aided redundant system reliability analysis (CARSRA) program developed during a previous NASA-Boeing program (vol. II, app B, subsec B.1.0, and ref 6) could not deal with the complex problems of predicting flight restriction of the ACT airplane. Instead, a computer program for fault-tree analysis developed at Boeing was used for the reliability analysis of the ACT system. Subsection 9.2 presents the results of the reliability analysis in more detail.

Table 3 shows the results of predicting flight restrictions and diversions for the Integrated and Segregated Systems, assuming the same computer reliability for both.

Flight restrictions are caused by loss or near loss of individual ACT functions (see subsec 8.4.1). The Segregated System, having six sets of computers, has a higher probability that a single set will become inoperable and hence a higher probability of flight restriction.

Diversions are caused only by the Essential PAS being one component failure away from function loss; or by speed PAS, LAS, and WLA functions all lost; or that same group being one component failure away from all lost (see subsec 8.4.1). Because integrated ACT functions share one set of computers, multiple computer failures must cause multiple function loss. This explains the higher probability of diversion for the Integrated System.

6.3.3.2 Dispatch Delay and Cancellation

The method used to predict dispatch delay and cancellation probability is to examine dispatch experience of airplanes with components similar to those of the ACT system. If the ACT system uses off-the-shelf components, experience delay rates are used. Where new hardware, such as the Segregated System microprocessor, is incorporated, experience data are modified to account for expected reduction in failure rate.

Table 3 shows the prediction of dispatch delays and cancellations caused by component failures in the Integrated and Segregated Systems. The Integrated System has higher dispatch delay and cancellation rates than the Segregated System.

Another requirement is that the ACT airplane must be dispatchable with any one component failed. This requirement is not presently met by either the Integrated or the Segregated System. When the system has failures in the crucial function, the airplane cannot be dispatched unless the remaining system meets the failure probability requirement of less than 10^{-9} per 1-hr flight. When the Integrated System loses one of the quadruple computers, or the Segregated System loses one of the quadruple short-period PAS computers, the remaining system fails to meet the 10^{-9} probability of failure per 1-hr flight requirement. The Integrated System requires quintuple redundancy to meet the dispatch requirement with one system element failure. The quintuply redundant ACT system was considered impractical because of its design complexity. For the Segregated System, the problem can be solved if the short-period PAS function is duplicated in the speed PAS computers. This approach mixes two different levels of criticality in defining the dispatch requirement.

Another proposed solution is to allow single-channel operation of the crucial short-period PAS function. To do this, the system must meet the requirements of Subsection 5.2.2. Short-period PAS cannot meet those requirements.

6.3.3.3 Individual Functional Reliability

The ACT system must meet the individual functional reliability requirements defined in Subsection 5.1. Subsection 9.2 contains the results of reliability analysis of individual ACT functions. All functions meet the requirements except WLA, where the requirement is 1×10^{-5} probability of failure per 1-hr flight and the prediction is 1.2×10^{-5}.

6.3.4 COST OF OWNERSHIP

The Boeing-developed airline cost-estimating system (ACES) computer program was used to assess cost-of-ownership differences between the Integrated and Segregated Systems when they are installed on the Initial ACT Airplane (ref 6). The program calculates the return on investment to the airline considering the system purchase cost, maintenance cost, spares inventory cost, fuel burn per hour, weight, drag, etc. Table 3 compares the return on investment and incremental cost per aircraft as calculated using the ACES program. The Integrated System shows lower ownership cost. Subsection 9.3 presents a more detailed analysis of the cost of ownership.

6.3.5 CONCLUSIONS

The Segregated System employs separate microcomputers for processing each ACT function in the belief that the separate computers would reduce the probability of simultaneous function loss caused by computer failure. The comparison study indicated that the benefit in reduction of simultaneous function loss was small compared with the increase of purchase cost and maintenance cost, resulting in a lower return on investment for the Segregated System, as shown in Table 3.

It is concluded that the Integrated System concept is the better concept for the ACT control system design. There are, however, reservations concerning the Integrated System approach (some are common to both systems); these concerns are discussed in the following paragraphs.

Software Design—Computer channels that perform identical ACT functions employ identical hardware and software to allow interchangeability. Thus, it is essential to have error-free software especially in the crucial short-period PAS function. The approach adopted in this task to develop error-free crucial software is to use conventional techniques; i.e., simplify the function, adopt appropriate software design techniques such as top-down structured programming, conduct rigorous testing including laboratory and flight test, and conduct rigorous software error analysis. To conduct rigorous analysis and thorough testing, it is very desirable to isolate the crucial hardware and software. This approach is less appropriate for the Integrated System, in which both crucial and critical software are executed in one set of computers.

Failure Propagation—Any system that includes flight-crucial functions should be designed to prevent noncrucial function failure propagating to crucial functions. This criterion is not met by the Integrated System, which mechanizes both critical and crucial functions in one set of hardware with some common software. The probability of failure propagation is reduced by the separation of crucial computing from critical computing in the Segregated System.

Requalification Test—Extensive and rigorous testing is required after any repair or modification whenever crucial functions may be affected. This means any repair or modification of the Integrated System computer hardware or software would result in a long test time because a crucial function may be affected. This supports using separate hardware and software to implement crucial functions.

6.4 SELECTED SYSTEM

The Selected System configuration is shown schematically in Figure 15. It combines features of the Integrated System with the advantages of separating the crucial function from critical functions. The system uses a triple set of ACT Primary Computers, similar to the Integrated System computers, to implement all ACT functions under normal conditions. An additional quadruple set of Essential PAS Computers commands and monitors the crucial elevator servos and provides backup calculation of the crucial short-period PAS control law. These are microprocessor-based computers similar to those used in the Segregated System. An additional computer, the ACT Maintenance and Display Computer, provides management and communication functions.

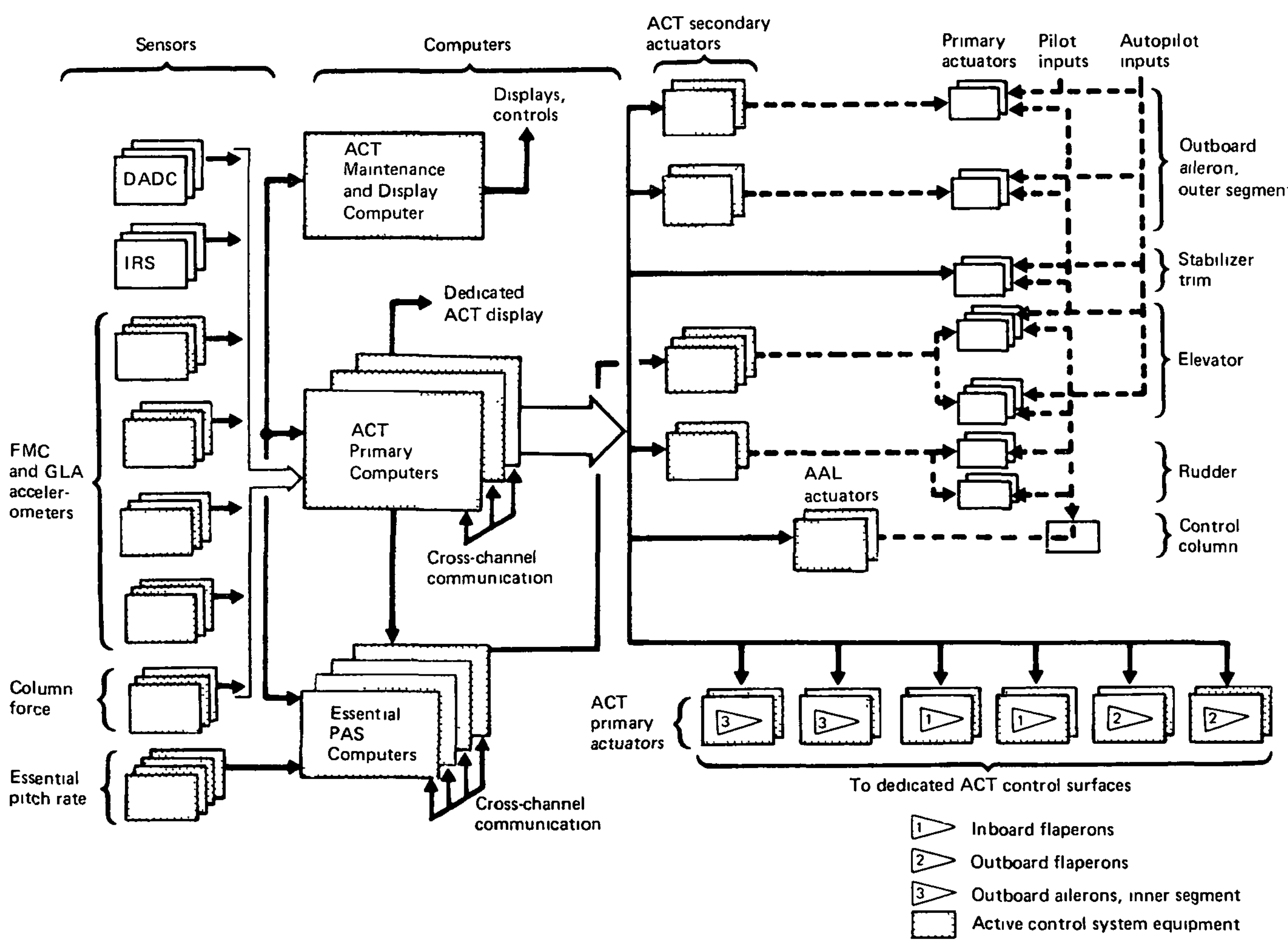


Figure 15. Selected System Configuration

The sensor set for the Selected System is the same as that for the Integrated System, with the addition of four dedicated pitch-rate sensors. The Selected System uses new actuators for the flaperon and AAL. Actuators for other surfaces are the same as in previous systems. A detailed description of the Selected System is contained in Section 8.0.

The Selected System was analyzed and compared to the Integrated and Segregated Systems. The results are shown in Table 4. The Selected System yields slightly lower return on investment than the Integrated System due primarily to higher initial cost, but the Selected System overcomes some of the objections to the Integrated System by separating crucial hardware and software from critical.

The Selected System still does not meet the requirement of dispatchability with any one ACT system component failed. After failure of either an Essential PAS Computer or an elevator secondary actuator, the short-period PAS loss probability rises above 1×10^{-9} per 1-hr flight.

6.5 PITCH AXIS FLY BY WIRE

A brief study was conducted to assess the benefit of replacing the pitch axis primary cable control system with an FBW system. Because pitch axis stabilization by the short-period PAS function is crucial, the airplane cannot be flown with the mechanical pitch axis control system alone. This makes airplane safety dependent upon an electric control system and suggests the change to a full FBW pitch axis to obtain the benefits of weight and cost reduction due to eliminating the mechanical pitch system. Figure 16 shows a block diagram of the FBW system defined in the study. Volume II, Appendix C, describes a pitch FBW servoactuator and its redundancy concept.

The primary cable system is replaced by quadruple electric paths. Pilot inputs are converted into quadruple electric signals and routed to the Essential PAS Computers. These signals are processed to generate a pilot elevator command that is summed with the ACT signals: Full PAS (normal condition) or Essential PAS (Full PAS failure), and MLC elevator command. Quadruple redundancy provides sufficient reliability; no backup system for the FBW by either direct electric link or mechanical link is included because

Table 4. Comparison of Integrated, Segregated, and Selected Systems

Item / System	Integrated System	Segregated System	Selected System
Computer weight	4 control computers at 11.3 kg (25 lb) each Total: 45.4 kg (100 lb)	19 control computers at 6 kg (13.25 lb) each 2 management computers at 2.7 kg (6.0 lb) each Total: 119.6 kg (263.8 lb)	3 ACT Primary Computers at 11.3 kg (25 lb) each 4 Essential PAS Computers at 6 kg (13.25 lb) each 1 ACT Maintenance and Display Computer at 2.7 kg (6 lb) Total: 60 kg (134 lb)
Memory requirement	32K ROM 2K RAM (128-word 8-bit non-volatile memory for maintenance information)	16K ROM and 64- to 512-word RAM for control computers plus 32K ROM and 1K RAM and 128-word 8-bit nonvolatile for management computer	24K ROM and 2K RAM per ACT Primary Computer 11K ROM and 256-word RAM per Essential PAS Computer 32K ROM and 1K RAM and 128-word, 8-bit nonvolatile
Computer reliability (assumed)	6800-hr MTBF	6800-hr MTBF	6800-hr MTBF for ACT Primary Computer 1200-hr MTBF for Essential PAS Computer
Probability of restricted flight	2.5×10^{-3} per 1-hr flight	3.3×10^{-3} per 1-hr flight	1.7×10^{-3} per 1-hr flight
Probability of flight diversion	7.1×10^{-4} per 1-hr flight	2.6×10^{-4} per 1-hr flight	4.0×10^{-4} per 1-hr flight
Probability of dispatch delay	3.8×10^{-4} per dispatch	1.6×10^{-4} per dispatch	1.2×10^{-4}
Increment cost per airplane (to airline)	$274 000	$390 000	$297 100
Expected return on investment	25.1%	22.8%	24.6%

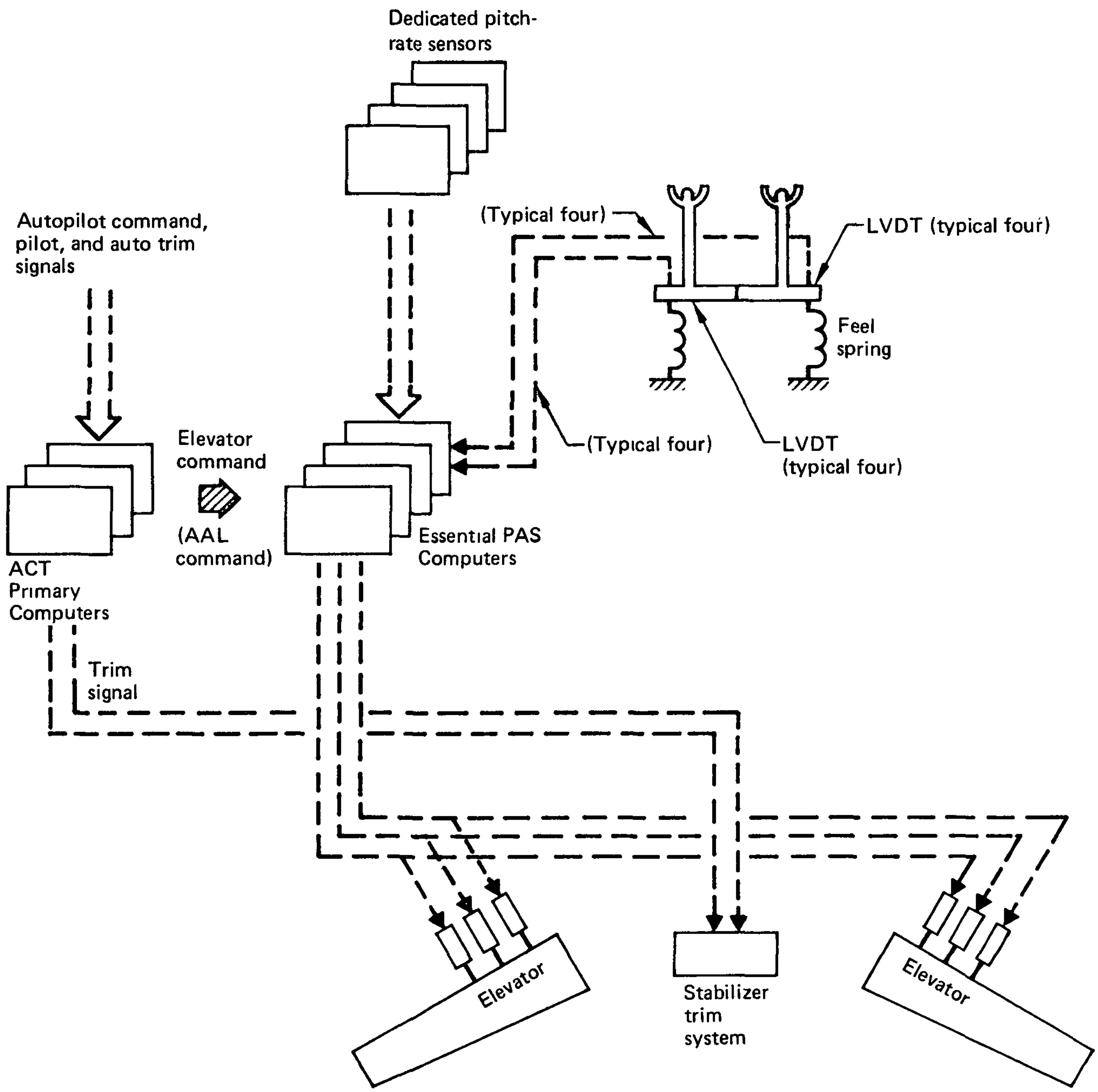


Figure 16. Pitch Axis Fly-by-Wire System

loss of the FBW loses the crucial short-period function that by itself would result in loss of the airplane. The FBW and the crucial short-period PAS share the computers and the actuation system. The conventional artificial feel computer that provides a fixed-stick force per g is replaced by a simple spring. By an appropriate gain schedule in the computers, the spring system provides a fixed displacement per g. The stick pusher of the Selected System is replaced by a pitch system signal that commands nose-down elevator directly without moving the control column. The autopilot will be a series system as shown in Figure 16.

Table 5 shows the decidedly favorable changes in Selected System cost and weight parameters resulting from substitution of pitch axis FBW for the conventional mechanical system. The lateral and directional control systems also have very complex mechanical assemblies and should be redesigned in FBW form to obtain maximum benefit from the concept.

Table 5. Changes Due to Pitch Axis Fly by Wire

Purchase cost	-$90 100
System weight	-157 kg (-345 lb)
Fuel burn per flight hour	-12 kg (-27 lb)
Incremental return on investment	+3.0%

7.0 SELECTED SYSTEM GENERAL DESCRIPTION

This section describes the Active Controls Technology (ACT) system concept resulting from the Current Technology ACT System Design Task. The concept, introduced in Subsection 6.4, was selected by conducting trade studies of system configurations, computer architecture, and actuation systems.

Subsection 7.1 is a concise description of the overall ACT system. Subsection 7.2 is an overview of the flight control system of the ACT airplane and includes general descriptions of the primary flight control system, secondary flight control system, and automatic flight control system. Subsection 7.3 describes the general configuration and functional configuration of the Selected System. Finally, operation of the Selected System is presented in Subsection 7.4.

7.1 SELECTED SYSTEM CONCEPT

The Selected System normally implements all ACT control laws in a single triplex set of computers and provides a separate quadruplex set of computers for backup crucial function implementation. The ACT Primary Computers compute all control laws and drive all ACT servos except the elevator servos. The Essential Pitch-Augmented Stability (PAS) Computers compute a simple fixed-gain short-period PAS control law and drive the elevator servos. A single ACT Maintenance and Display Computer performs general system management, communication, and data storage functions.

The sensor configuration is the same as in the Integrated System except for the addition of four dedicated pitch-rate sensors as in the Segregated System. A new actuator concept has been adopted for the flaperons, with electrically commanded power control units (PCU) force summed by the flaperon torque box structure. The angle-of-attack limiter (AAL) actuators have been modified to operate from low-pressure engine bleed air rather than the high-pressure nitrogen system used in previous configurations. Figure 17 illustrates the Selected System configuration.

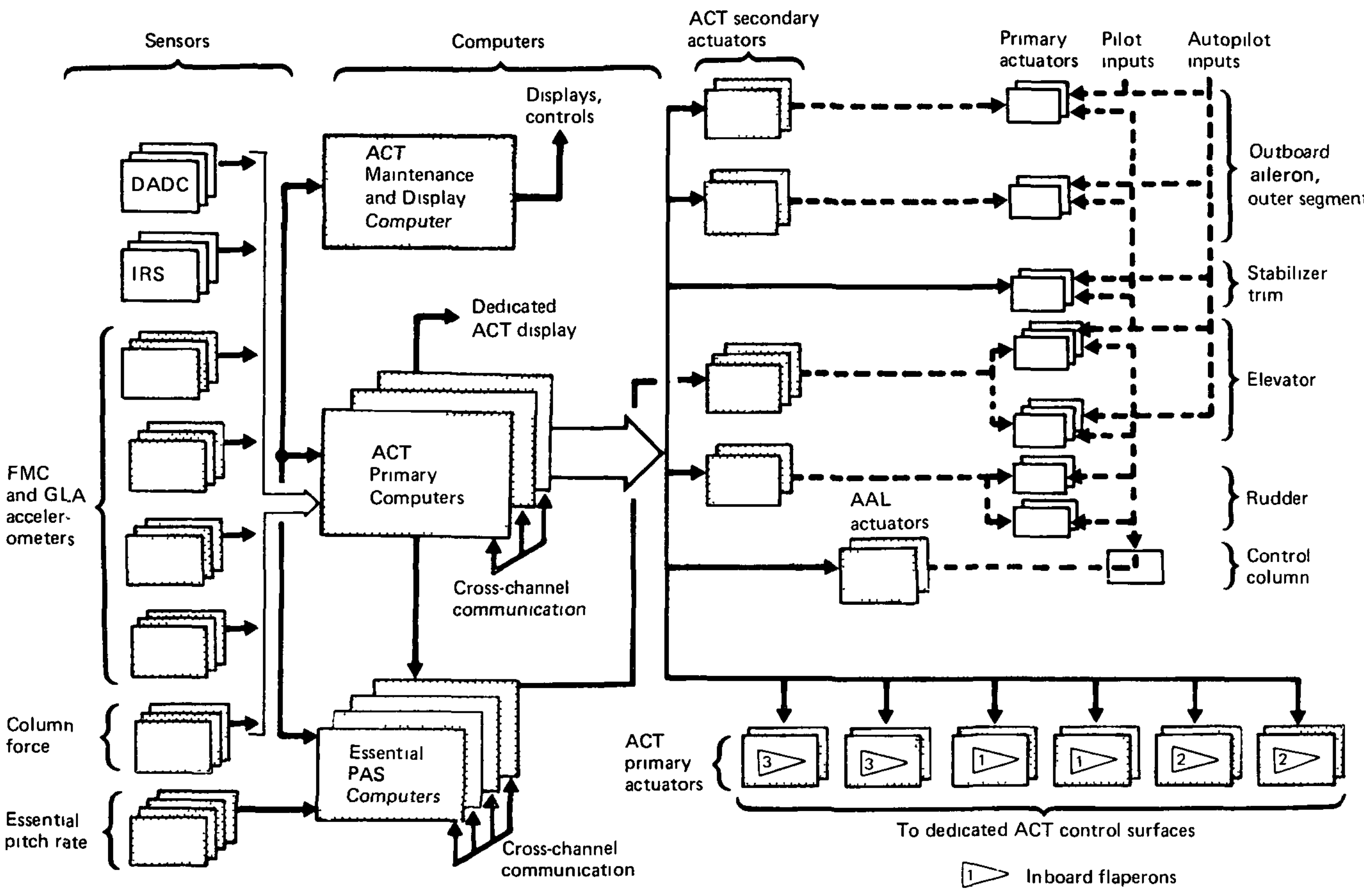


Figure 17. Selected System Configuration

7.1.1 SELECTED SYSTEM COMPUTERS

The ACT Primary Computers calculate control laws for Full PAS, AAL, lateral/directional-augmented stability (LAS), wing-load alleviation (WLA), and flutter-mode control (FMC) functions. Full PAS includes short-period PAS and speed PAS to give the airplane good longitudinal handling qualities. The ACT Primary Computers receive air data and inertial reference signals from the airplane triplex digital air data computers (DADC) and inertial reference systems (IRS) over Aeronautical Radio Incorporated (ARINC) 429 digital buses. Dedicated ACT sensors provide wing accelerations for FMC and gust-load alleviation (GLA) functions and column force for maneuver-load control (MLC). Other sensors include flap and slat position and servo position feedback. Dynamic pressure and center-of-gravity acceleration are taken directly from sensors in the DADC and IRS and transmitted over analog lines, bypassing the computing portions of those two sensor systems.

Each ACT Primary Computer is directly connected to only one sensor in each triplex set. Incoming sensor data are transmitted cross channel by an autonomous input/output (I/O) controller so all these computers see data from all sensors. The central processing unit (CPU) is not required in this cross-channel transmission; if a CPU fails, the other two computers still receive full sets of sensor data.

Sensor signals are monitored for failures and processed to select a single value for each signal that is used in control law calculations. Computer outputs are monitored and all but elevator signals are sent to servoelectronics to drive ACT servos. The ACT Primary Computers monitor servo operation (except elevators) and shut down a servo if the servo or the computer driving it fails. Failure detection is primarily accomplished through cross-channel comparison augmented by inline monitoring. Because inline monitoring is not considered 100% accurate, single-channel operation is permitted only for the LAS function, which meets special safety requirements stated in Subsection 5.2.2.

The Essential PAS Computers drive the elevator servos and calculate a simple fixed-gain short-period PAS control law capable of providing minimum acceptable longitudinal handling qualities. The Essential PAS Computers receive elevator commands from the ACT Primary Computers over ARINC 429 buses. Each of the three Essential PAS Computers receives data directly from an ACT Primary Computer, but after cross-channel communication all four Essential PAS Computers have data from all three ACT Primary Computers. CPU action is required for this transfer. In addition, each Essential PAS Computer receives signals from the four dedicated pitch-rate sensors. Each sensor is directly connected to all four computers. In normal operation, an Essential PAS Computer selects the midvalue of the ACT Primary Computer elevator commands (or average value after one failure) and uses that to drive an elevator servo. If the PAS signal from the ACT Primary Computers fails, the Essential PAS Computers substitute their own short-period PAS command, calculated using the signals from the dedicated pitch-rate sensors.

The ACT Maintenance and Display Computer coordinates testing of the system and collects fault data from the ACT Primary and Essential PAS Computers. These data are stored to be used later by maintenance personnel; advisories and warnings are displayed to the flight crew as required.

7.1.2 SELECTED SYSTEM ACTUATORS

The Selected System uses four different types of actuators; these actuators are described in the following paragraphs.

Secondary Force-Summed Actuator (With Mechanical Primary Control System)—The secondary force-summed actuator scheme is used for elevators, rudders, and the outer segment of the outboard ailerons where the ACT functions share the control surfaces with pilot and autopilot inputs. The output of the ACT secondary actuator is series summed with the pilot or autopilot mechanical input to form the total surface command. Detailed description of the secondary actuator is contained in Subsection 8.1.3. Figure 11 is a simplified diagram illustrating the concept. The secondary actuation concept is used for control of the rudder (LAS), outer section of outboard aileron (WLA), and elevator (PAS and MLC). Outputs from the ACT Primary Computers control the secondary actuator of the rudder and the outboard aileron outer section directly.

ACT Dedicated Force-Summed Actuator—This fly-by-wire force-summed actuation scheme was developed to meet the requirements of the flaperons, which are commanded by WLA signals from the ACT Primary Computers. The WLA function requires two actuators plus a mathematical model to meet the fail-operational requirement. Normally two hydraulic systems are required to implement dual actuation systems. A major concern of this approach is that a separated flap would cause the simultaneous loss of both hydraulic systems. Because of this, only one hydraulic power system is directly connected to the flaperon actuator. A hydraulic motor-pump unit is used to connect the second hydraulic power system as backup to the normal power supply system for power redundancy. Figure 18 illustrates this scheme. Subsection 8.1.3 describes this actuation system in more detail.

ACT Dedicated Force-Displacement Summed Actuator—The third actuation scheme, the dedicated force-displacement summed actuator, is shown in Figure 12. It is used to

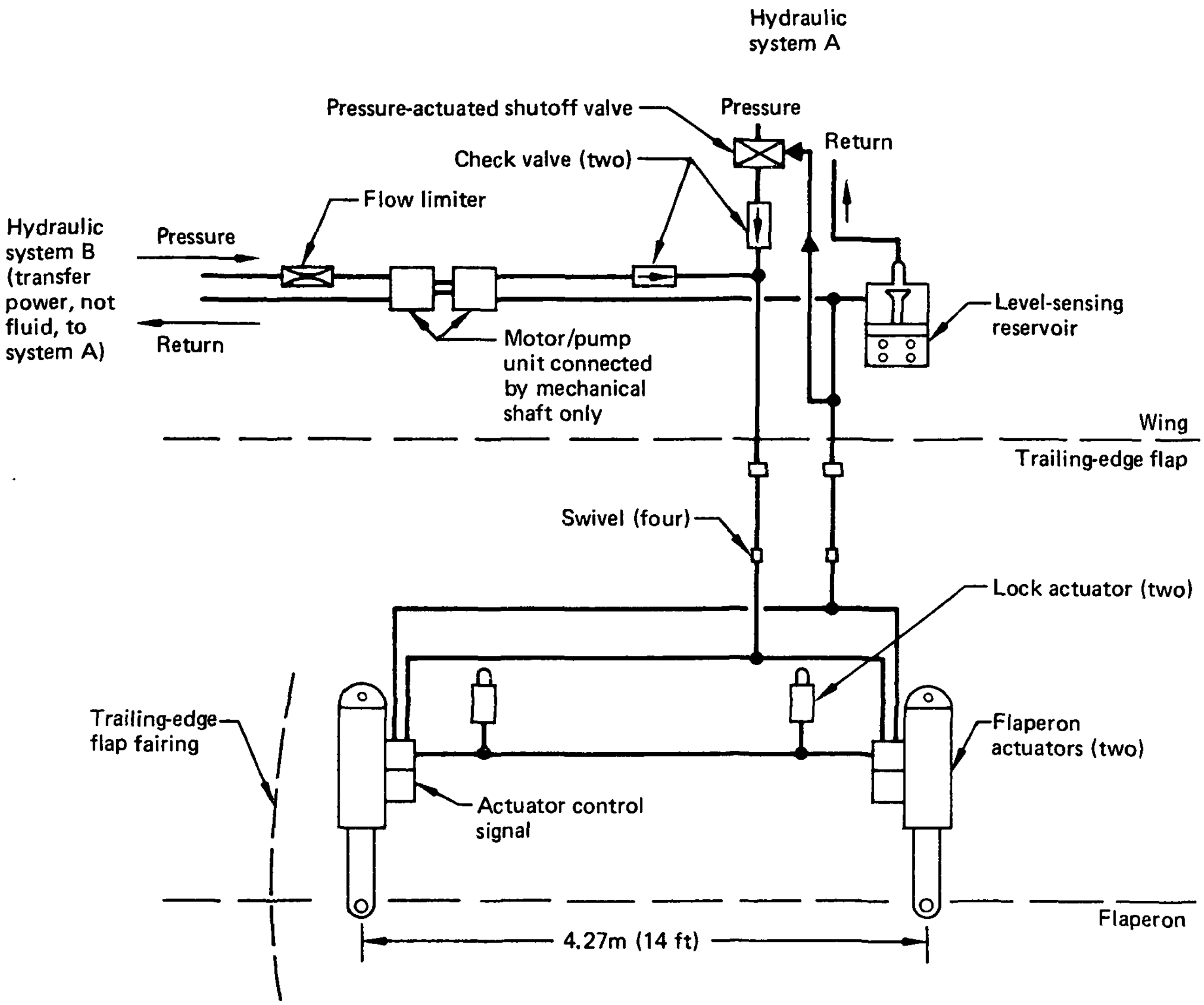


Figure 18. Flaperon Hydraulic Actuation System

control the inner section of the outboard aileron, which is shared by the pilot, autopilot, and ACT functions (FMC and WLA). It was designed to remain fully operational after one electric failure and one hydraulic failure. The pilot input is converted into electric signals by position transducers and summed with the FMC and WLA commands in the ACT Primary Computers. The autopilot controls the inner section of the outboard aileron by parallel actuators that move the cables and back drive the control wheels. Subsection 8.1.3 describes this in more detail.

Dual Pneumatic Actuator—The fourth actuator scheme, the dual pneumatic actuator, is used as the stick pusher, which provides positive stall prevention by strong, rapid forward motion of the control column at the incipient stall point. Figure 13 is a simplified diagram of the concept. This concept was developed to replace the high-pressure pneumatic stick pusher actuator defined for the Integrated System, which used high-pressure bottled gas, causing special problems in system test, maintenance, and servicing. The new scheme uses low-pressure air from the engine bleed system instead of nitrogen from a high-pressure bottle that must be frequently checked and refilled. Subsection 8.1.3 describes the dual pneumatic actuator in detail.

7.1.3 ESSENTIAL PAS SYSTEM

The Essential PAS System is a highly reliable system that provides minimum acceptable handling quality in pitch axis control. A simple fixed-gain pitch-rate feedback is implemented in quadruple redundancy to meet the fail-operational/fail-operational requirement. Figure 17 illustrates this system. Each of the dedicated analog pitch-rate sensors is connected to each of the quadruple computers; i.e., cross-strapped channel communication is used. The analog pitch-rate signals are processed in each computer by the signal selection and failure detection (SSFD) algorithm to produce a single value for computation of the fixed-gain short-period PAS control law. Normally, the output of the Essential PAS Computers is not coupled to the elevator actuators. The elevator commands from the ACT Primary Computers normally control the pitch axis. These elevator signals are sent to the Essential PAS Computers, which continuously monitor their uniformity. When the primary short-period PAS has failed, the fixed-gain pitch-rate feedback takes over the pitch axis control.

The computers of the Essential PAS System are operated asynchronously as in the case of the Segregated System. Feasibility of fixed-gain PAS has been verified in the laboratory. Outputs of the Essential PAS Computers command the elevator through the secondary force-summed actuators. The triple force-summed actuators plus mathematical models implemented in the computers meet the fail-operational/fail-operational requirement.

7.2 FLIGHT CONTROL SYSTEM OVERVIEW

The flight control system is divided into primary, secondary, and automatic flight control systems based upon function. The primary control system provides the basic capability of maneuvering, stabilizing, and trimming the airplane. The secondary control system manipulates wing trailing-edge flaps, leading-edge slats, and spoilers to control airplane lift and drag. The automatic flight control system provides the autopilot, flight director, and thrust management to reduce pilot's workload.

The general arrangement of the flight control surfaces of the current technology ACT airplane is shown in Figure 19.

7.2.1 PRIMARY FLIGHT CONTROLS

The primary flight controls control the airplane in the longitudinal (pitch) axis, lateral (roll) axis, and directional (yaw) axis. The following subsections describe controls for these three axes and the primary flight control electronics.

7.2.1.1 Longitudinal Controls

Longitudinal controls consist of elevator control and horizontal stabilizer trim. The elevator controls illustrated in Figures 20 and 21 position two single-segment dual-hinge elevators in accordance with pilot, ACT, and AFCS inputs. Each elevator is powered by three side-by-side PCUs. The pilots position the elevator (via PCU) by control columns and cables. The pilots' control columns are interconnected by a torque shaft incorporating a jam override spring and a manual disconnect mechanism. The control column is also controlled by an AAL to prevent the airplane from exceeding a limiting angle of attack. The electric elevator commands from the Essential PAS Computers are converted into mechanical signals through the ACT secondary elevator actuators and then summed in series with the pilots' mechanical input. To meet the redundancy requirement of crucial short-period PAS, quadruple sensors and computers are provided in the Essential PAS System. Three side-by-side force-summed secondary actuators are used for ACT elevator control. In addition to the triple ACT elevator actuators, mathematical actuator models are programmed in the backup computers to meet the fail-operational/fail-operational requirement of short-period PAS. The ACT Primary Computers process the control laws,

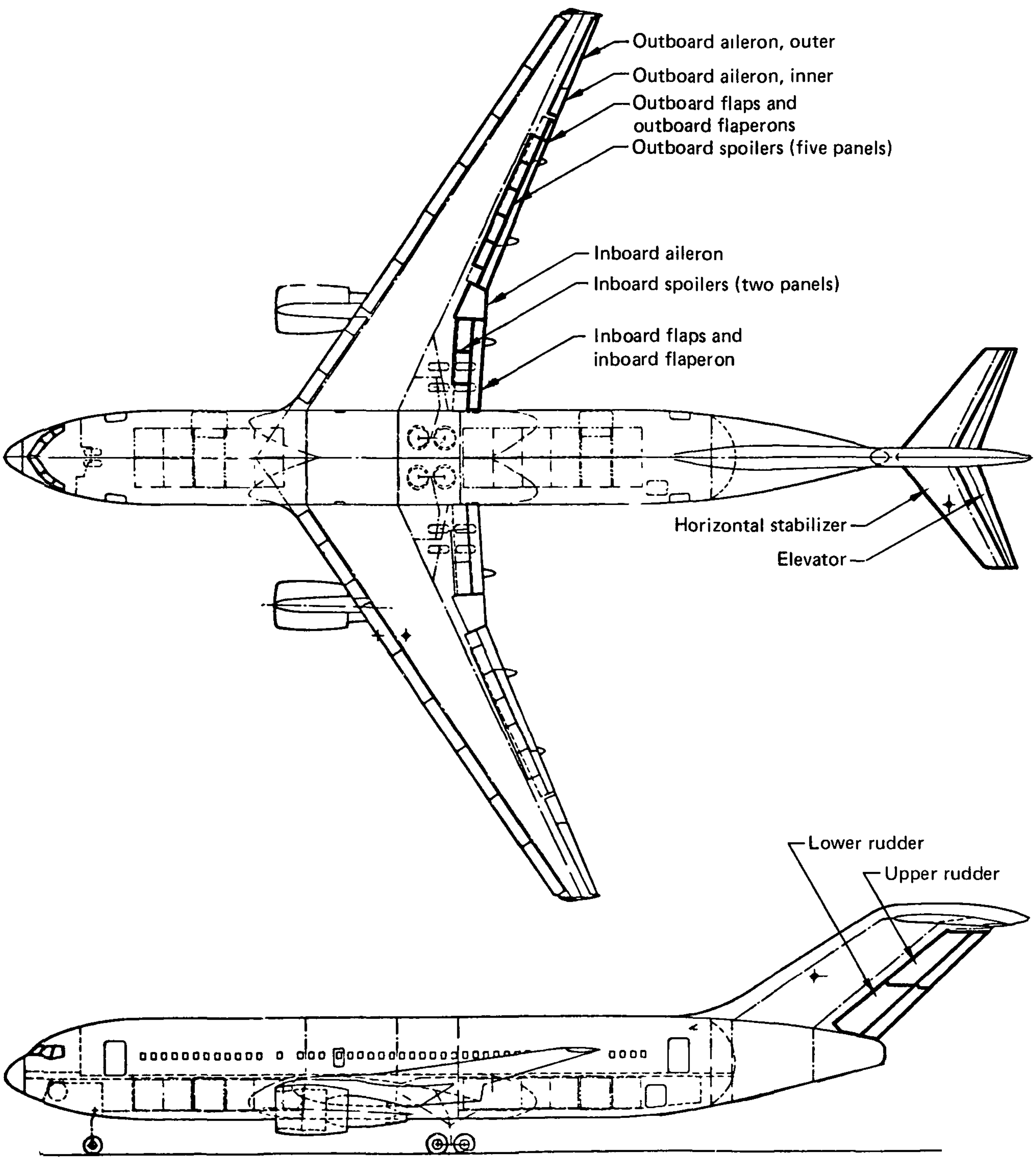
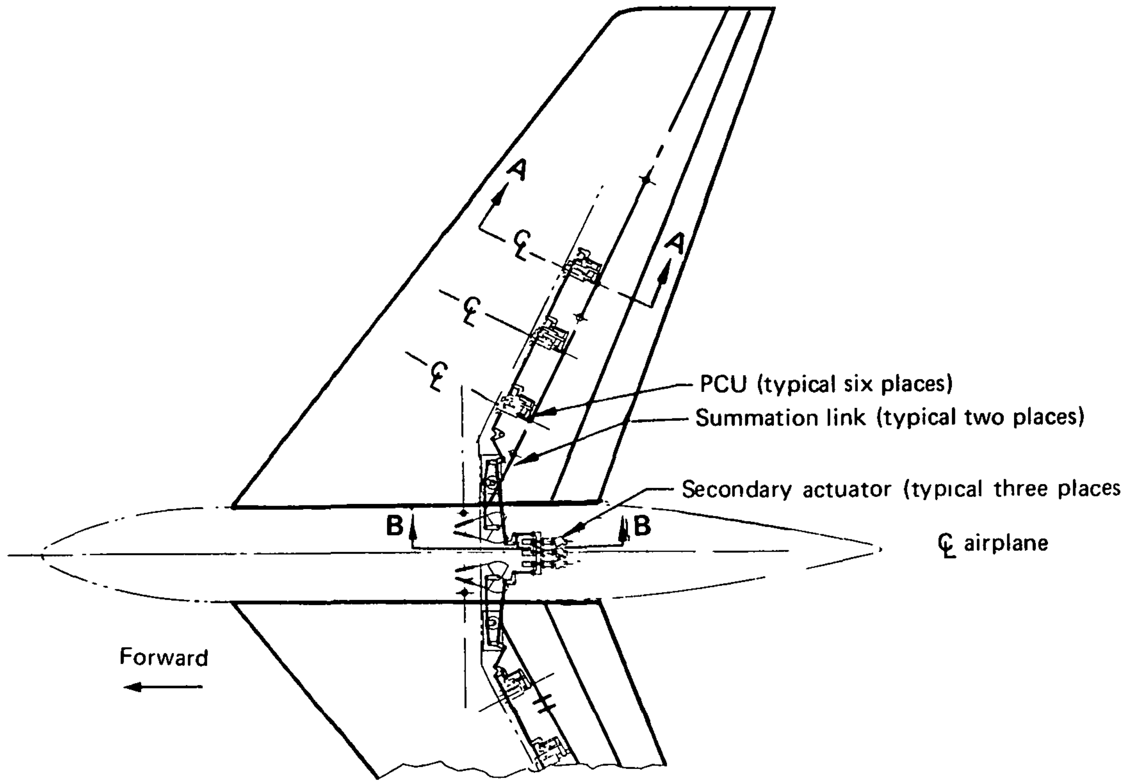
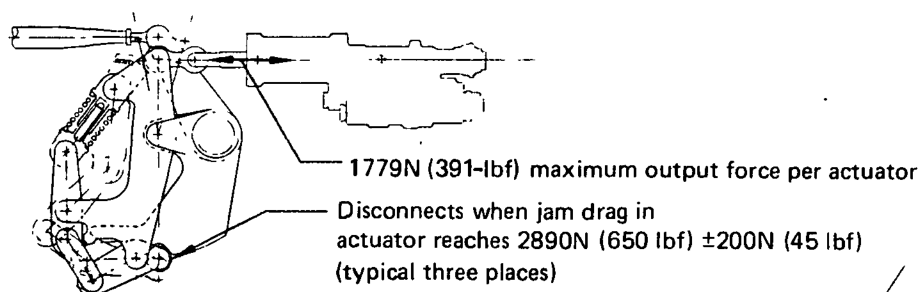


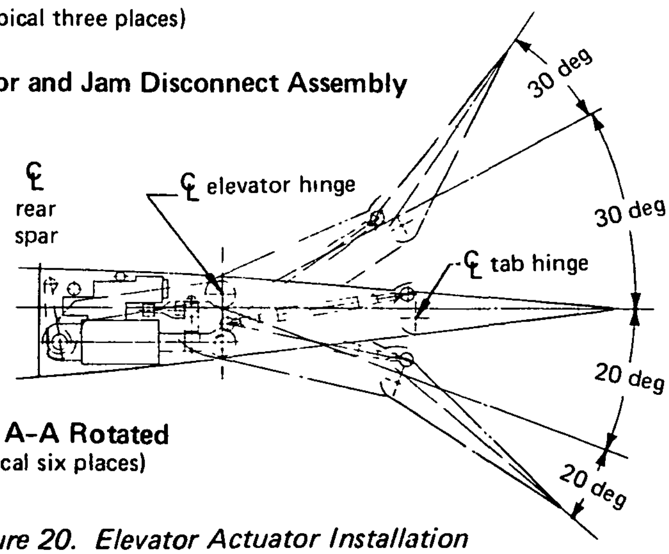
Figure 19. Aerodynamic Control Surfaces



Plan View of Horizontal Stabilizer



Section B-B Secondary Actuator and Jam Disconnect Assembly



Section A-A Rotated
(typical six places)

Figure 20. Elevator Actuator Installation

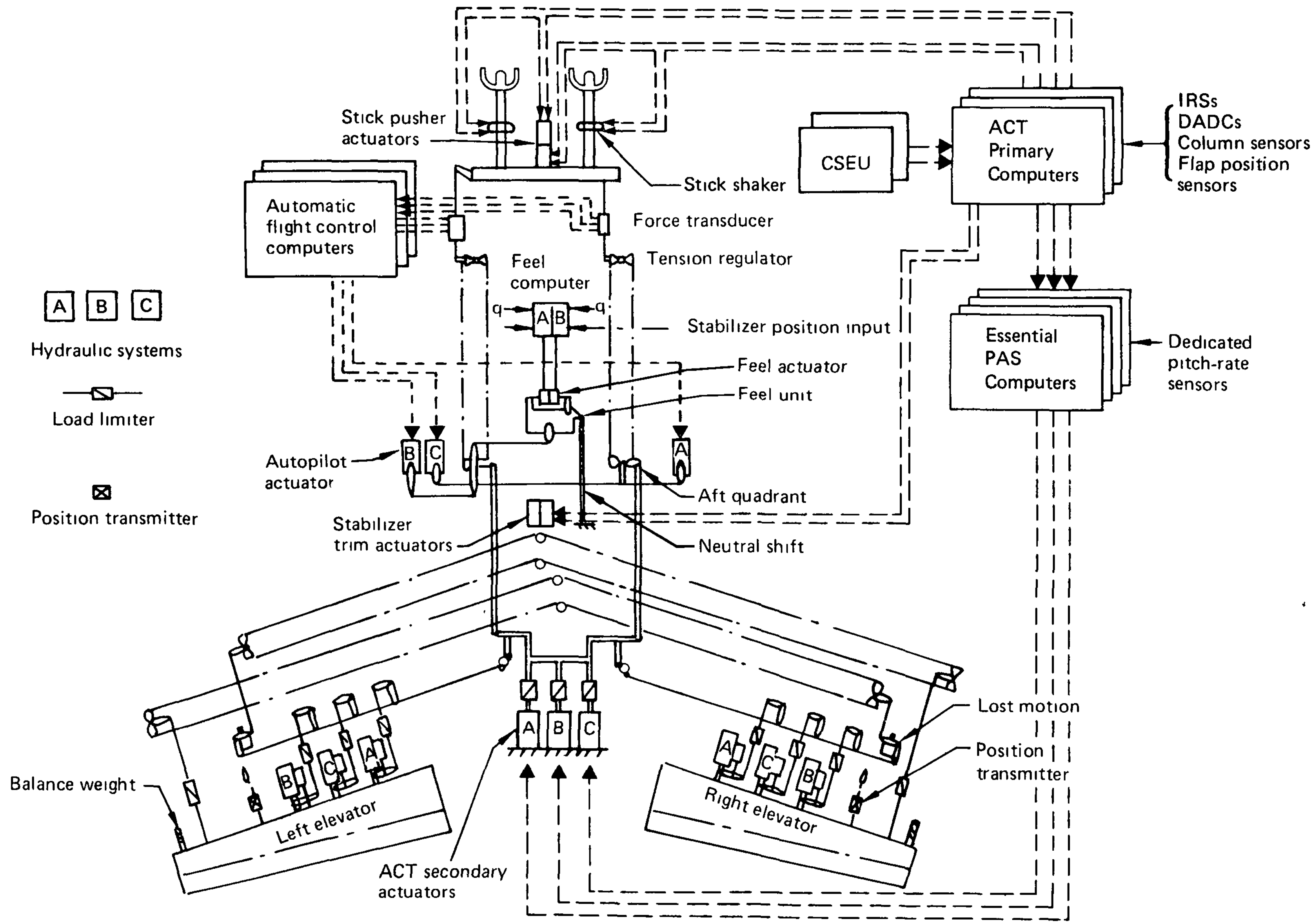


Figure 21. Longitudinal Control System

redundancy management, and testing function of PAS and MLC and send the elevator command to the Essential PAS Computers. Triple sensors and computers are provided in the ACT Primary System to meet the fail-operational requirement.

The autopilot elevator signals are converted into mechanical signals by the parallel autopilot secondary actuators to control the elevators. To meet the category III-B automatic landing requirement, triple electronics and force-summed secondary actuators are provided.

The elevator control cable systems are connected to a common artificial feel mechanism that is modulated by variable hydraulic pressure controlled by dual pressure modules.

The horizontal stabilizer trim control is illustrated in Figure 22. The stabilizer trim actuator consists of a ballscrew and nut, upper and lower gimbals, primary no-back brake, reduction gearing, and two hydraulic motors that drive the reduction gearing via a differential gear assembly. Each motor is also connected to a hydraulic pressure-released brake, controlled by a separate control module, and powered by a separate hydraulic system. Each control module contains two valves: one to arm the module by turning on hydraulic power, and one to control flow to one motor. The arm and control valves are commanded either electrically or mechanically. The electric commands are controlled by dual trim interface units. Manual electric trim commands originate at two dual switches, one on each pilot's control wheel. The trim interface units prohibit electric trim commands in opposition to elevator control inputs. Manual mechanical trim commands are transmitted to the control modules by dual control cables connected to dual levers on the pilots' aisle stand. All electric commands may be overridden by mechanical commands. Each control module contains a stabilizer rate-limiting valve that reduces hydraulic fluid flow as a function of airspeed. Dual electric stabilizer position indicators are mounted on the aisle stand and connected to dual stabilizer position sensors.

7.2.1.2 Lateral Controls

Lateral controls position the inboard aileron, outboard aileron, inboard flaperon, outboard flaperon, and five spoiler panels on each wing as shown in Figures 23 and 24. The outboard aileron is split into two sections. The pilot and autopilot use the ailerons and spoilers to maneuver the airplane in the lateral axis. The FMC of the ACT uses only the

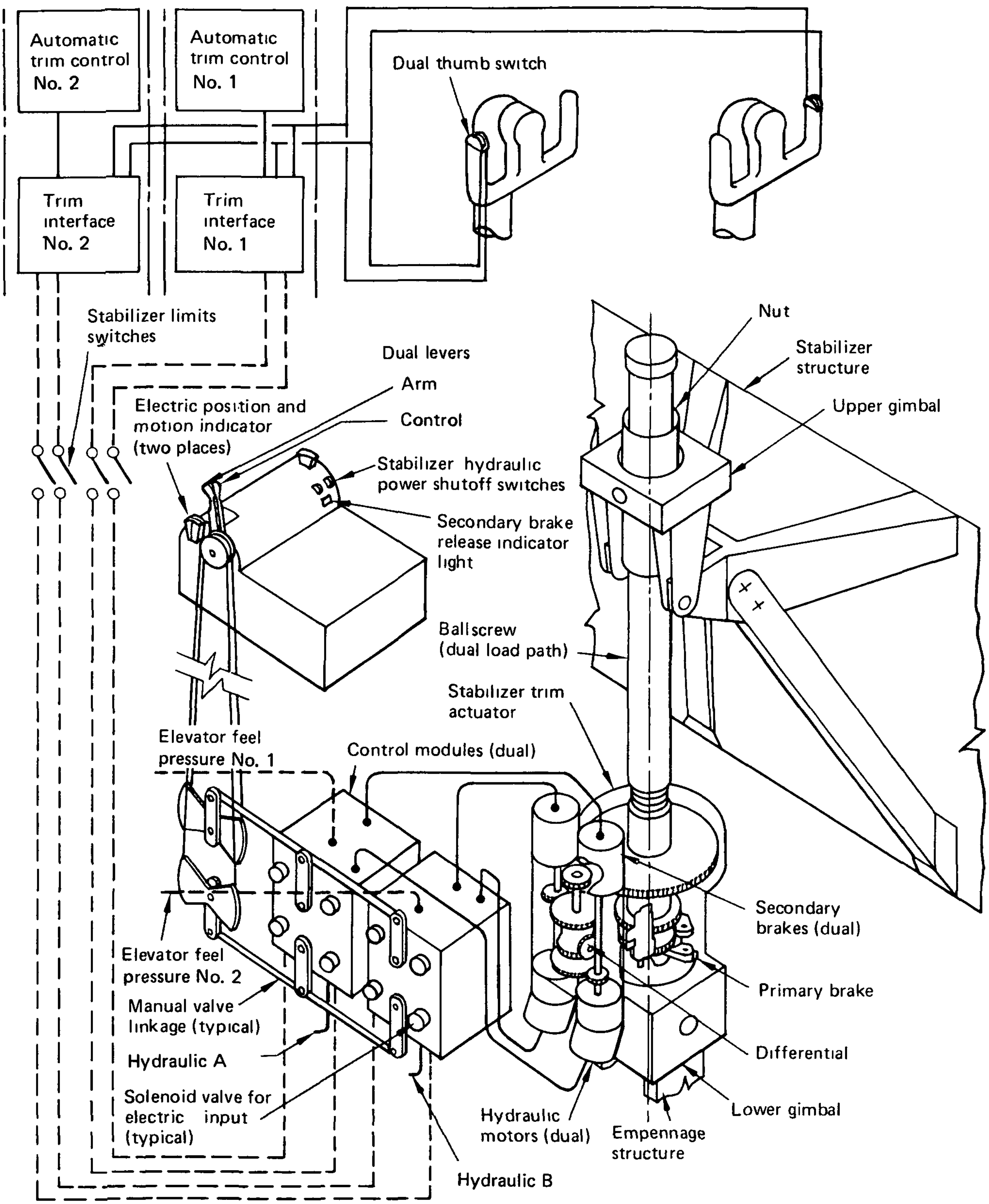


Figure 22. Stabilizer Trim System

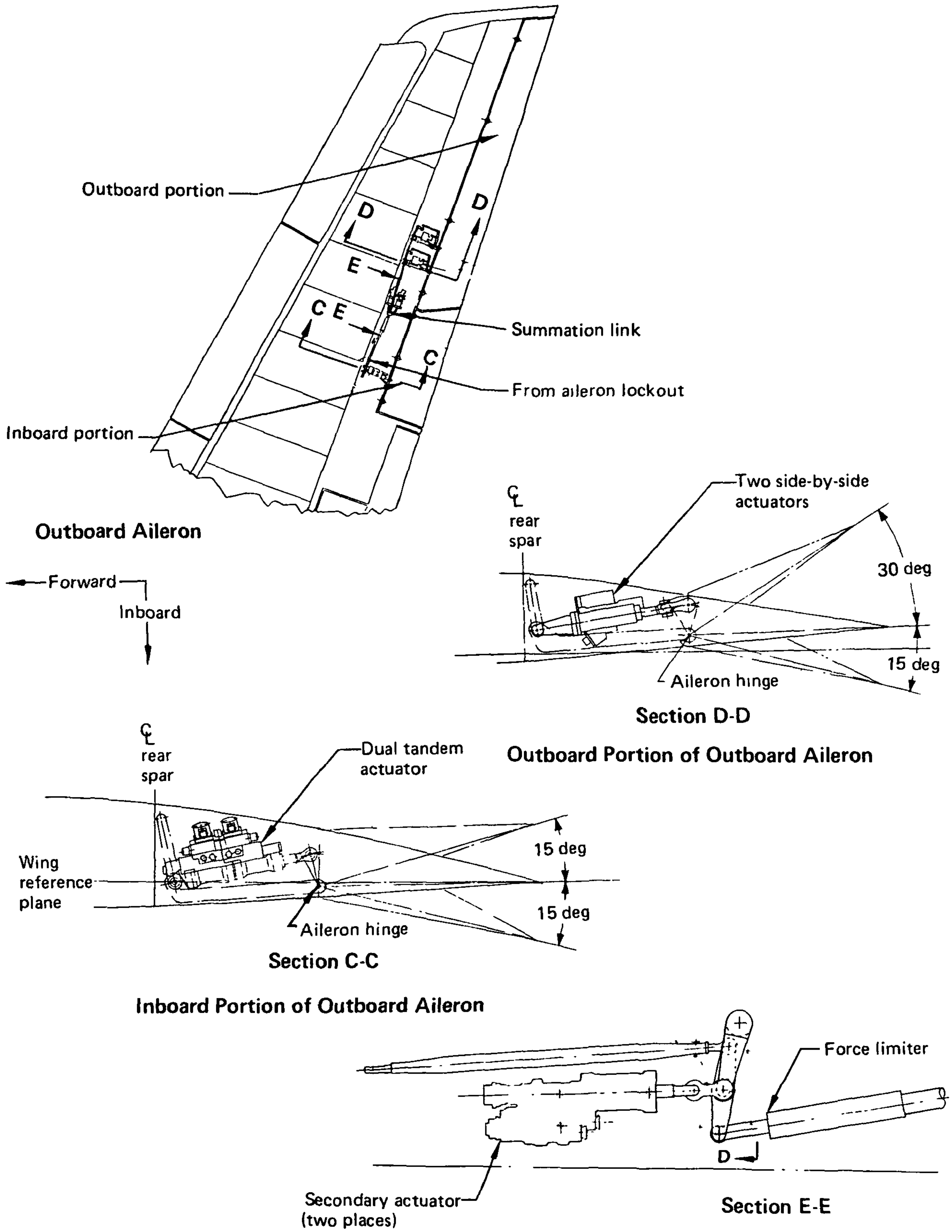


Figure 23. Outboard Aileron Actuator Installation

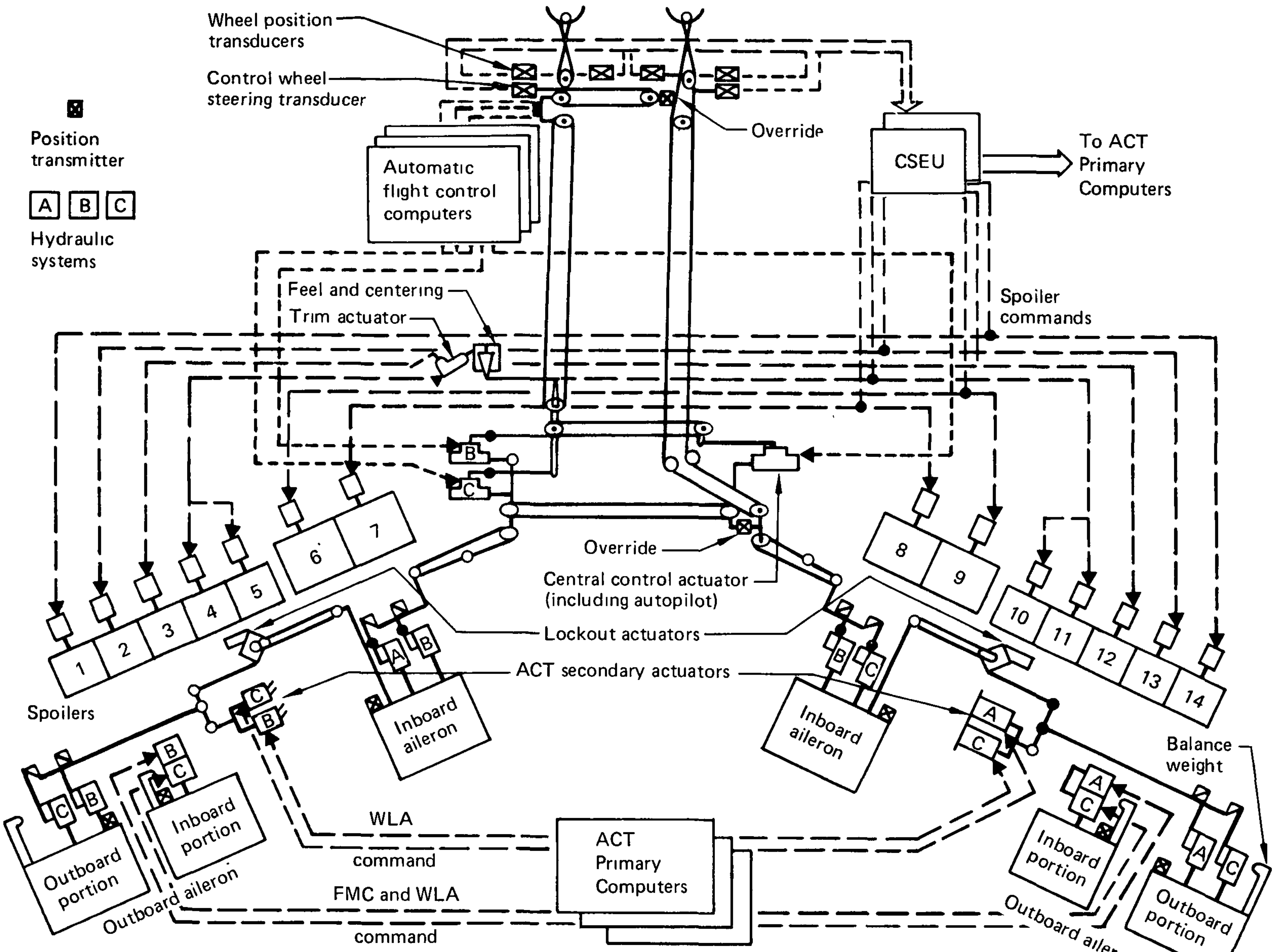


Figure 24. Aileron Control System

small inner section of the outboard aileron to enable high-frequency response. The WLA of the ACT uses the outboard aileron (including both inner and outer sections), inboard flaperon, and outboard flaperon to control the wing loads induced by pilot maneuver and atmospheric turbulence.

The pilots' input for lateral control is provided by two control wheels. The output of the control wheels, through a dual cable system, positions the ailerons except for the inner section of the outboard aileron. The pilots control the inner section of the outboard aileron electrically through the triple ACT Primary Computers. The actuators of the outboard aileron outer section receive their input command via cables from the inboard aileron motion. The control inputs of the outboard aileron (including both sections) from both pilot and autopilot are locked at neutral at high airspeed and unlocked at low speed. However, the inner sections of the outboard aileron are controllable by the ACT Primary Computers at cruise speed for FMC and WLA control. The control cables are connected to a dual feel and centering mechanism at each forward quadrant.

The electric command for the outer section of the outboard aileron is computed in the triple ACT Primary Computers and converted into a mechanical signal through the ACT secondary actuators and then summed with the pilot and autopilot in series. Dual side-by-side force-summed secondary actuators and the mathematical models implemented in the triple ACT Primary Computers provide a fail-operational capability for WLA. The electric command signal for the inner section of the outboard aileron is generated in the triple ACT Primary Computers by summing the FMC, WLA, and pilot inputs. To meet the fail-operational requirement of FMC and WLA, dual side-by-side force-summed dedicated PCUs plus mathematical models are provided for control of the inner section of the outboard aileron.

In the system selected in the Current Technology ACT System Definition Task, GLA is assumed to use the outboard flaperon and MLC is assumed to use the inboard flaperon. Dual force-summed actuators plus mathematical models are used to meet the fail-operational requirement. The flaperon actuator receives hydraulic power through swivel joints. To avoid the simultaneous loss of two hydraulic systems in the flaperon actuation system, only one hydraulic system is directly connected to the dual actuators.

The spoilers are controlled electrically through the control system electronic unit (CSEU). Flight spoilers may be used as speed brakes while continuing to provide lateral control.

The lateral autopilot manipulates the outboard and inboard ailerons and spoilers to control the airplane during roll. To meet the category III-B automatic landing requirement, triple redundancy is provided in sensors, computers, and actuators. Parallel force-summed actuators are used to drive the ailerons and to back drive the control wheels.

Lateral trim is accomplished using electric switches located on the aisle stand to control an electronically operated actuator that connects to the trim feel and centering mechanism at the forward quadrant bus and changes the neutral position of the lateral control system.

7.2.1.3 Directional Controls

Directional controls position the two double-hinged rudder surfaces as shown in Figures 25 and 26. Each rudder is controlled by dual PCUs. The actuators are signaled by the pilots' rudder pedals through a common cable system coupled by separate pushrods and linkages to PCUs in the vertical tail. The LAS of the ACT system performs yaw damper and turn coordination originally computed in the CSEU of the Baseline system. The electric LAS commands generated in the triple ACT Primary Computers are converted into mechanical signals through the secondary actuators and then summed in series with the pilots' mechanical input. Dual side-by-side force-summed actuators plus mathematical actuator models implemented in the ACT Primary Computers provide fail-operational capability in the LAS function. Lateral control includes rollout guidance, which is a part of category III-B automatic landing. Triple rollout guidance actuators are provided to meet the fail-operational requirement.

Separate rudder ratio changers are installed in the upper and lower rudder control linkages. A fixed-gradient, artificial feel mechanism is connected in parallel to the control cable system common to the upper and lower rudders. The neutral position of the feel mechanism is varied by an electric trim actuator.

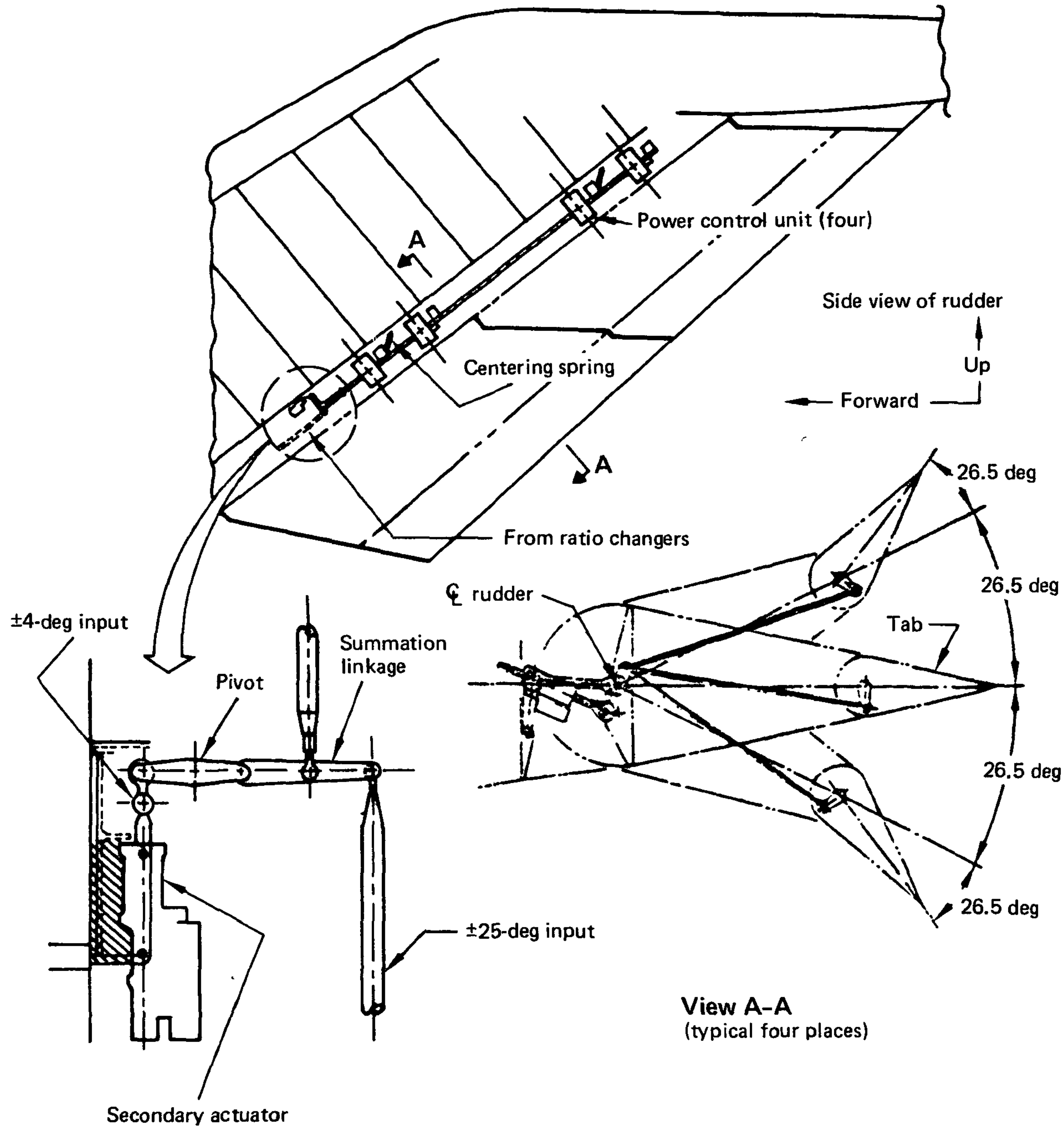


Figure 25. Rudder Actuator Installation

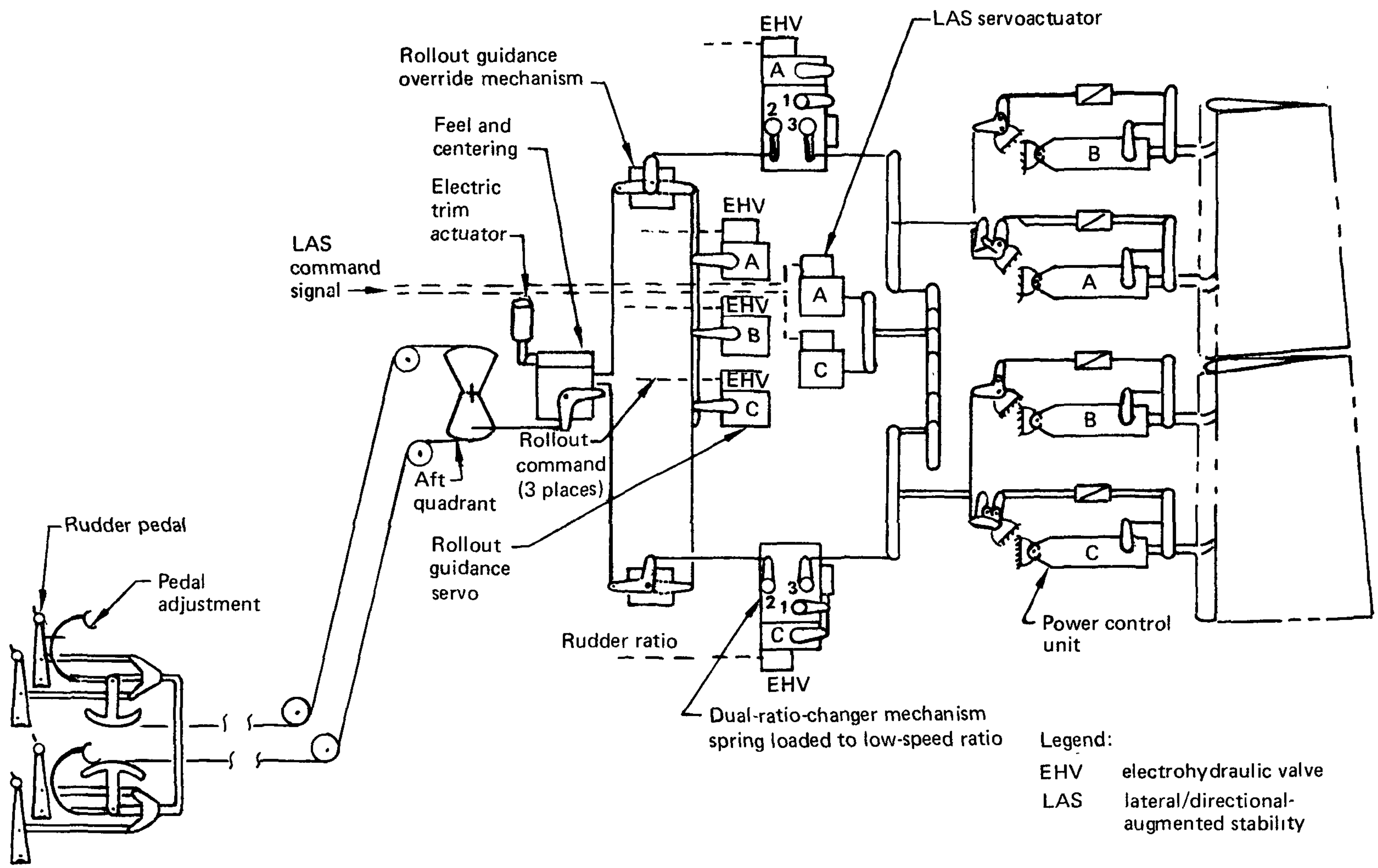


Figure 26. Directional Control System

7.2.1.4 Primary Flight Control Electronics

The primary flight control electronics include ACT electronics and the CSEU. The ACT electronic system is described in Subsections 7.3 and 8.0. Only the CSEU function is described in this subsection. The CSEU function is shown in Figure 27 and includes rudder ratio changer, spoiler control, stabilizer trim interface, and outboard aileron lockout. These functions are described in the following paragraphs.

Rudder Ratio Changer—The rudder ratio changer provides a variable schedule of rudder deflection versus rudder pedal input as a function of dynamic pressure. The ratio changer electronics are online replaceable cards in the CSEU.

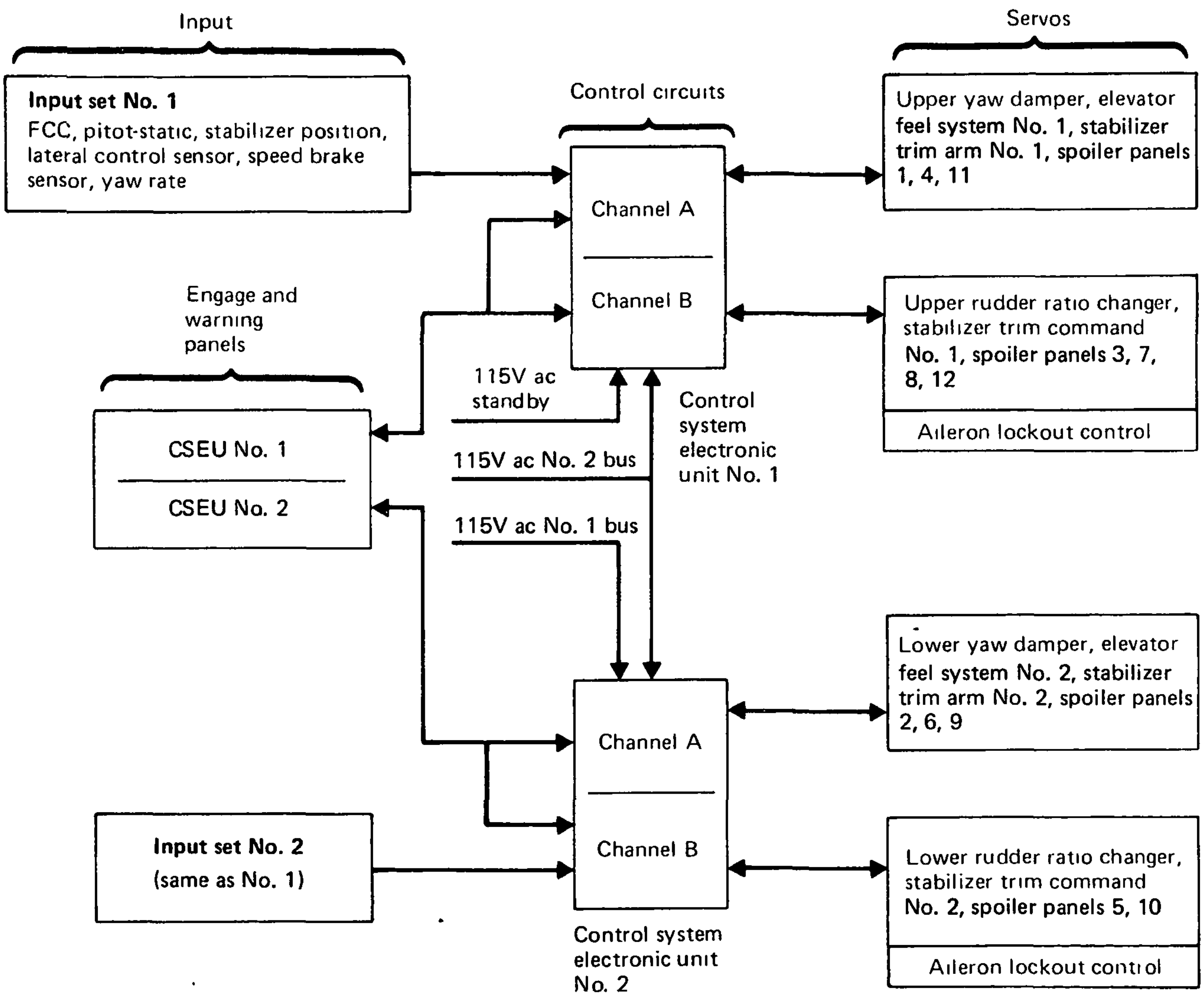


Figure 27. Control System Electronic Unit

Spoiler Control—Spoiler control causes the spoiler panels to deploy as a function of control wheel deflection, speed lever position, and air-to-ground logic. The 10 spoiler panels are programmed in symmetric pairs to provide speed brakes, ground spoilers, and, in conjunction with ailerons, linearization of rolling moment to wheel deflection. Spoiler control includes control modules installed in the CSEU. Each control module commands a symmetrical pair of spoiler panels. One electrohydraulic power control actuator drives each spoiler panel.

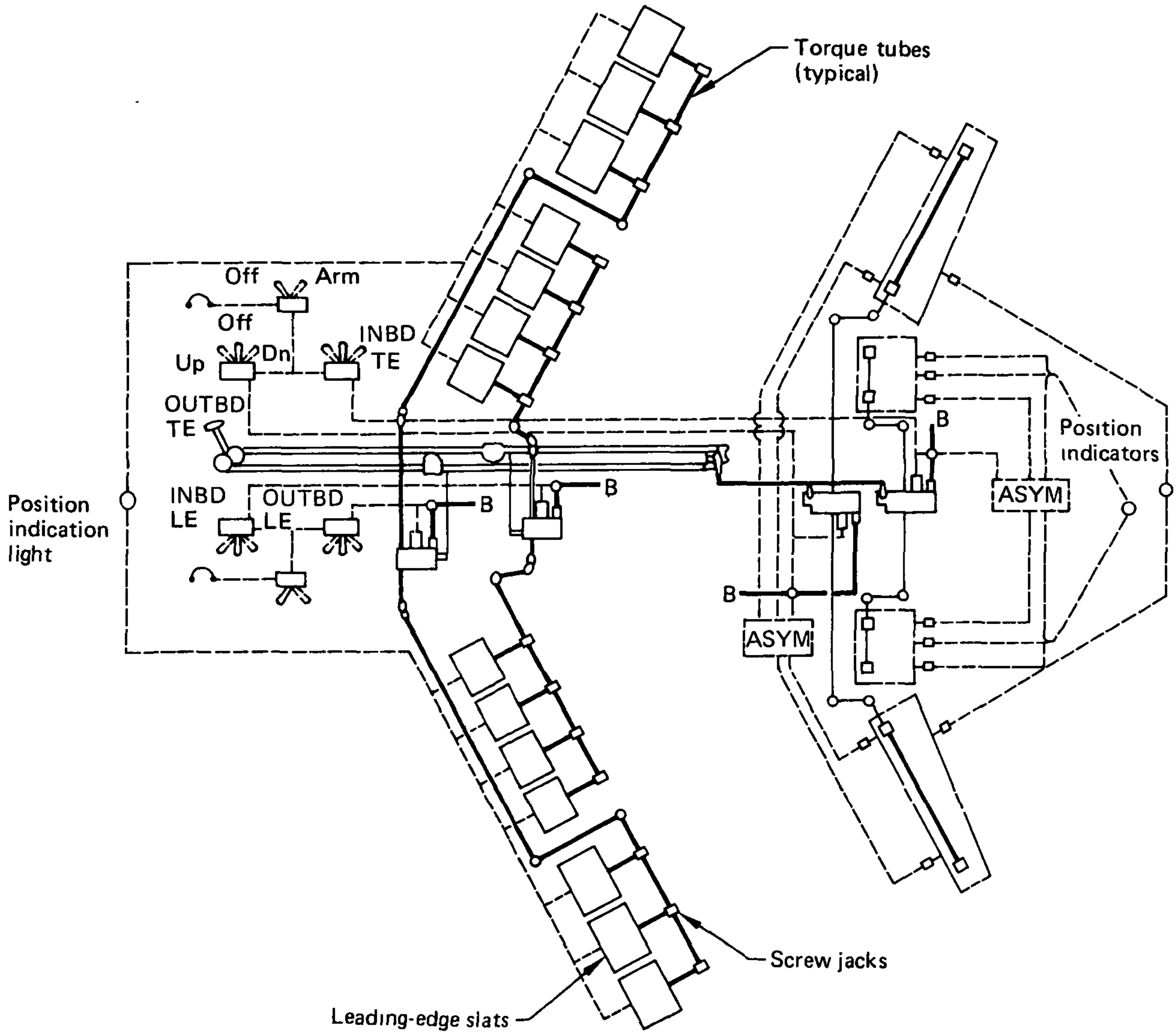
Stabilizer Trim—The stabilizer trim function will maintain the airplane in a trimmed condition per various alternative sources of trim command signals. Two independent actuation systems position the stabilizer in response to the source commands. An electric module in the CSEU selects a particular trim signal source and also energizes one or the other actuation system. The three sources of electric trim command are the flight control computers (FCC), the pilot electric trim switches, and the ACT Primary Computers. Selection of the active trim source is accomplished by logic circuits in the CSEU.

Outboard Aileron Lockout—Outboard aileron lockout logic provides discrete signals to the left and right outboard aileron lockout actuators and to lockout logic in the ACT Primary Computers. The lockout actuators disable the pilot inputs to the outer section of the outboard aileron. The lockout logic in the ACT Primary Computers will disconnect the pilot input to the inner section of the outboard aileron by software.

7.2.2 SECONDARY FLIGHT CONTROLS

Secondary flight (high lift) controls position the wing trailing-edge flaps and the wing leading-edge slats in a coordinated manner. The trailing-edge flaps and the leading-edge slats are controlled to three positions (cruise, takeoff, and landing). The wing spoiler panels are controlled symmetrically as a secondary flight control function for in-flight drag modulation and landing deceleration.

Wing Trailing-Edge Flap Controls—The wing trailing-edge flap surfaces (see fig. 28) are driven by two centrally located power drive units, one driving the inboard trailing-edge flaps and the other driving the outboard trailing-edge flaps. Each power drive unit is powered by a hydraulic motor with alternate electric motor operation. The drive units



Legend:

Systems:
— Mechanical
--- Electric
▬ Hydraulic

⊗ Bypass valve

Power drive unit:
Electric motor / Hydraulic motor

ASYM	asymmetry detection electronics
DN	down
INBD	inboard
LE	leading edge
OUTBD	outboard
TE	trailing edge

Figure 28. Leading- and Trailing-Edge High-Lift System

are connected to torque-tube systems in each wing to drive mechanical rotary actuators on each inboard and outboard flap segment. A torque-limiting brake is provided in each rotary actuator, which also incorporates a no-back brake. Asymmetry detection systems shut off the respective power drive units if the inboard or outboard flaps become asymmetric between left and right wings or between actuators on a flap segment. Flap position indication is provided to the flight deck indicator from each flap panel.

Wing Leading-Edge Slat Controls–The wing leading-edge slat controls shown in Figure 28 use two power drive units located in the inboard wing leading edges, one to drive the inboard leading-edge slats and the other to drive the outboard leading-edge slats.

The controls are programmed to extend partially the leading-edge slats before trailing-edge flap extension and to retract the slats after trailing-edge flap retraction. Each power drive unit is powered by a hydraulic motor with an electric motor alternate. The drive units are normally controlled by a dual cable system from the flight deck control lever. If hydraulic power is lost, the electric alternate system can be controlled by flight deck switches. The drive units are connected to torque-tube systems that run along the front spar of each wing and drive the screw jacks mounted on each slat segment. Each screw jack on each slat segment incorporates a no-back brake and a torque-limiting brake. Indication is provided to the flight deck when the leading-edge slats are in transit or fully extended. Failure of any slat surface to fully extend or fully retract is indicated by the intransit lights.

7.2.3 AUTOMATIC FLIGHT CONTROLS

The automatic flight control system (AFCS) provides the following functions: autopilot, automatic stabilizer trim, flight director, and thrust management. These functions will reduce the pilots' workload in flight by automatically controlling the flight path and engine thrust or by providing flight guidance signals to pilots.

Figure 29 and Table 6 show subsystem partitioning and summary definition of the major components of the AFCS.

Automatic flight controls are shown in Figures 30 and 31 and provide the following major functions: autopilot, flight director, and thrust management. These functions are described in the following subsections.

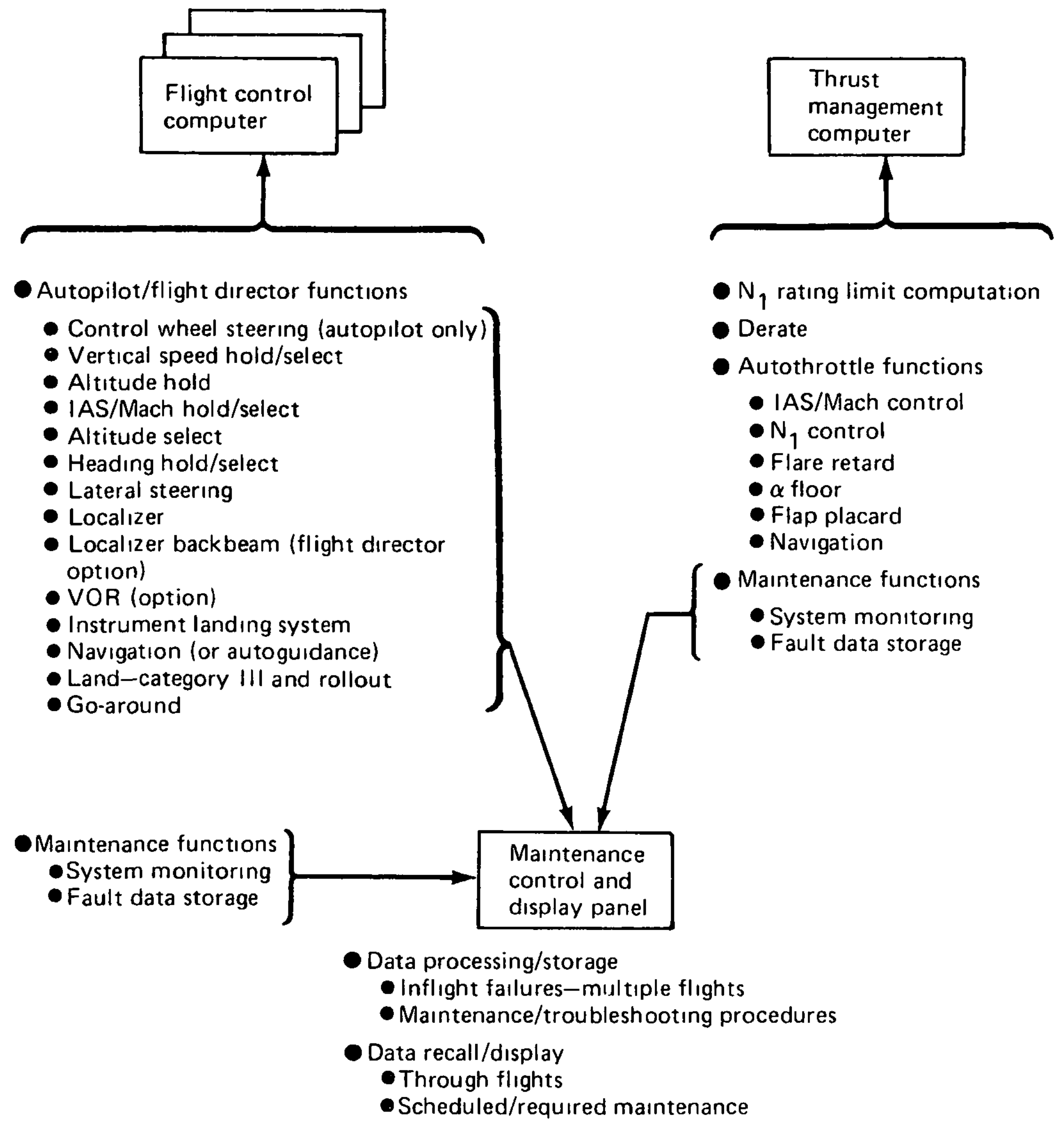


Figure 29. Automatic Flight Control (Functional Partitioning)

7.2.3.1 Autopilot

The autopilot function is computed in the FCC. Three FCCs are provided for those autopilot modes that require redundancy. The following autopilot modes are provided.

Control Wheel Steering Mode—Control wheel steering (CWS) is a selectable mode of operation for the autopilot. It provides (1) pitch and roll maneuver control for pilot inputs and (2) pitch attitude hold and roll or heading hold when there are no pilot inputs.

Table 6. Major Components of Automatic Flight Controls

Description	Quantity
Autopilot controls	
Flight control computer	3
Integrated autopilot/flight director autothrottle/mode control panel	1
Maintenance control and display panel	1
Remote-mounted maintenance panel	1
Thrust management	
Thrust management computer	1
Rating limit select panel	1
Limit display and mode annunciation panel	1
Miscellaneous flight control electronics	
Autopilot interface unit	3
Flight mode annunciator	2
Barometric altitude rate unit	1
Sensor flag warning annunciator	1

Altitude Select Mode—Altitude select is the basic pitch mode and is armed when the autopilot is placed in the command mode and a glide slope has not been captured.

Vertical Speed Select and Hold Mode—Vertical speed select and hold is the basic pitch transition mode between selected altitudes. It is engaged synchronously when the autopilot is initially engaged into the command mode of operation or when the vertical speed mode is selected.

Vertical Profile Mode—Vertical profile mode is a selectable altitude transition mode that uses combined autothrottle and elevator control. This mode provides airspeed and Mach control through the elevators during altitude transitions coupled with either selected rated (or derated) N_1 (low-pressure compressor speed) control during climb or idle thrust during descent. When maintaining altitude, airspeed and Mach control is automatically transferred to the autothrottle control section. Two selectable submodes of the vertical profile mode are available. One submode is the automatic mode in which target altitudes and speeds are automatically commanded by the flight management computer. The other submode is manual in which the target altitudes and speeds are manually entered through the mode control panel.

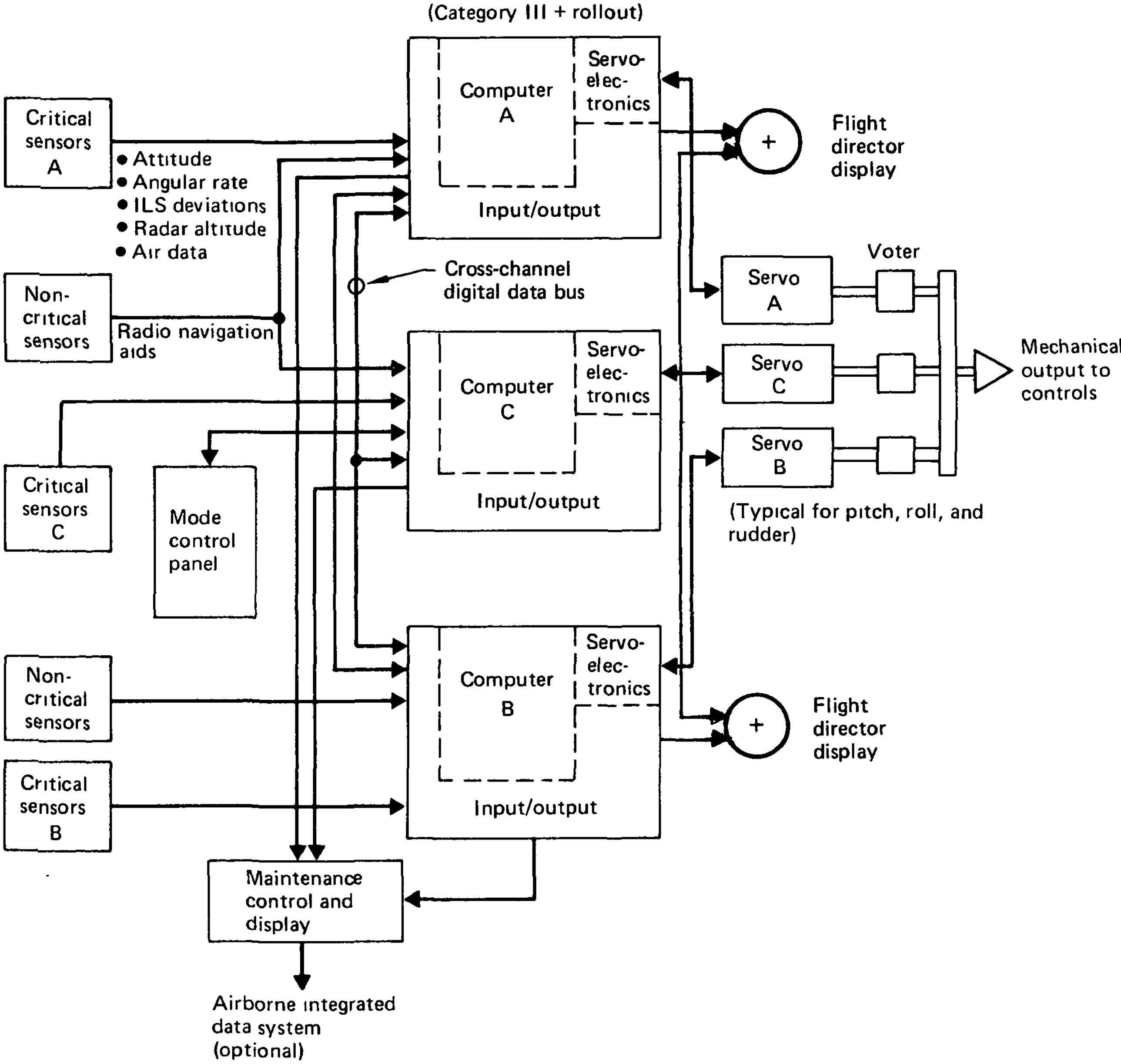


Figure 30. Automatic Flight Controls (Autopilot/Flight Director)

Heading Select Mode—Heading select is the basic lateral mode and is engaged synchronously when the autopilot is initially engaged into the command mode of operation. Heading preselect capability is available whenever the mode is selected from any other engaged lateral command mode.

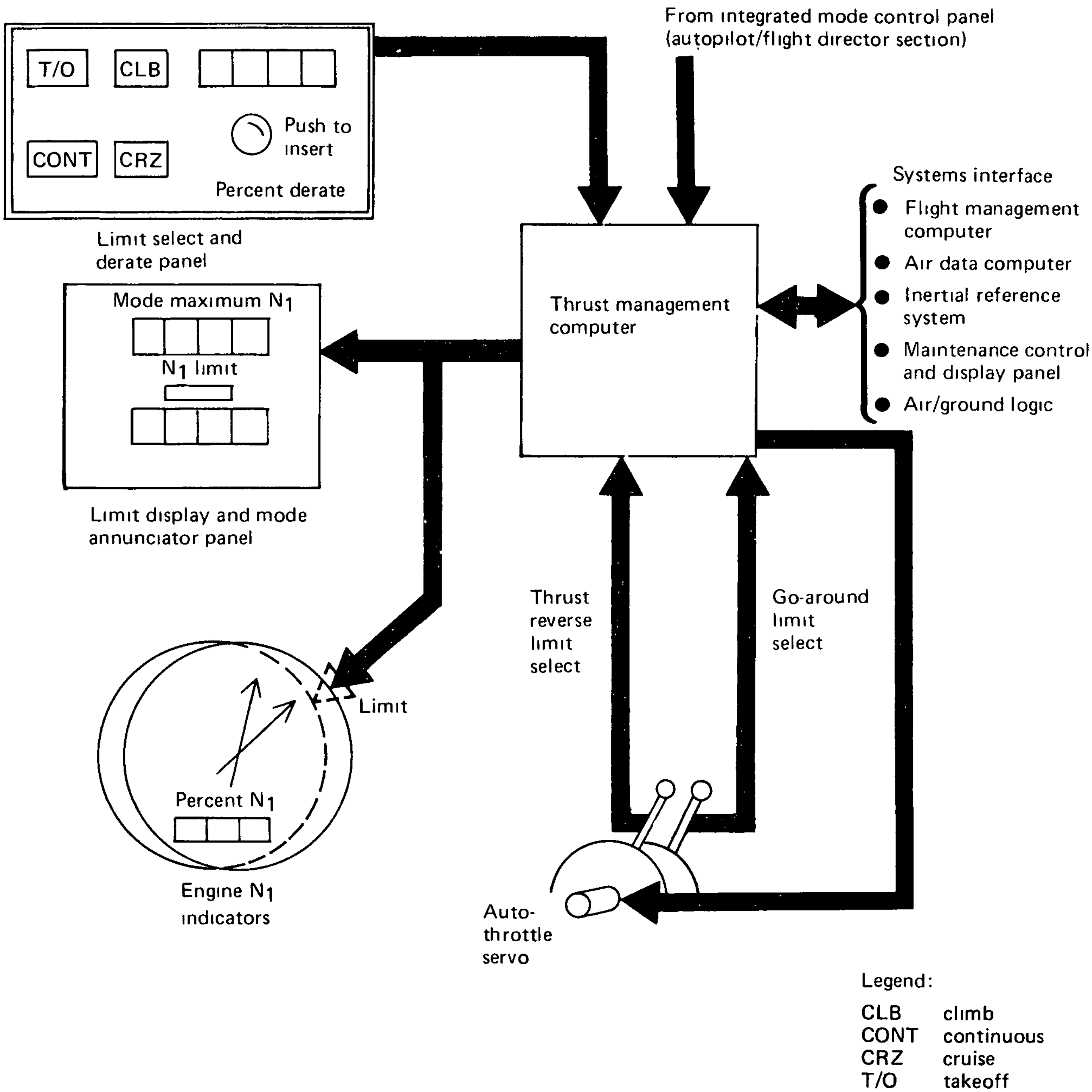


Figure 31. Automatic Flight Controls (Thrust Management)

Navigation Mode—Navigation is a selectable lateral mode that uses the flight management computer to provide lateral steering commands to the autopilot function in the FCC. The flight management computer commands can be based on very-high-frequency omnidirectional radio range (VOR) information or waypoint guidance.

Localizer Mode—This selectable lateral mode allows the lateral autopilot function to be coupled to a localizer independent of the glide slope.

Approach Mode—The approach mode is a selectable mode that provides automatic glide slope and localizer control and manually initiated automatic go-around control. This mode can be operated multichannel through touchdown and automatic rollout. The mode meets the fail-operational requirement with triple redundancy and meets fail-safe with dual redundancy.

VOR Mode—The VOR mode is optional if the flight management computer is deleted. In this configuration, the navigation (NAV) mode of autopilot operation couples the VOR receiver outputs into the FCC to provide VOR control. This mode is not part of the basic certification of the airplane.

7.2.3.2 Flight Director

The flight director function is also computed in the FCC and has the same modes as the autopilot functions plus one additional mode, localizer back beam. The flight director command displays can be turned on and off, independent of whether or not an autopilot function is selected. Provisions are included to allow the flight director command bars to be automatically biased out of view whenever the autopilot function is engaged in modes other than CWS.

The control mode select functions for autopilot, flight director, and autothrottle are integrated into a single panel. Select knobs are provided for indicated airspeed and Mach, altitude, vertical speed, heading, and course.

7.2.3.3 Thrust Management

The thrust management system includes the thrust rating limit and autothrottle functions. These computations are performed in the thrust management computers located in the electronic equipment bay. Space provisions are made for a second thrust management computer. The thrust rating limit and autothrottle functions are described in the following paragraphs.

Thrust Rating Limit—The thrust management computer selects, computes, and displays engine-rated thrust limits. Figure 31 is a simplified diagram illustrating the thrust rating limit. Controls for selecting thrust rating limits and commanding derated limits are provided on the thrust limit selection panel located near the engine instruments; the limits are displayed on the limit display and mode annunciation panel shown in Figure 31. Selection limits include takeoff and go-around, climb, maximum continuous, and cruise. Actuation of automatic go-around switches at the thrust levers automatically causes go-around thrust limit to be selected and displayed. Unlocking the thrust reverse automatically causes reverse limit to be selected and displayed. The thrust management computer interfaces with the engine N_1 instruments to position an indicator to show the value of the selected thrust limits or derated limit.

Autothrottle—The thrust management computer controls the thrust levers, via the autothrottle servomotor, to provide full-range thrust control (within the thrust rating limits) during climb, cruise, approach, and landing.

7.3 SELECTED SYSTEM CONFIGURATION

Subsections 6.4 and 7.1 introduced the general configuration of the Selected System. This subsection presents the same configuration in more detail. Subsystem functions and components of the Selected System are described in Section 8.0.

The system consists of the ACT Primary System, Essential PAS System, and ACT Maintenance and Display Computer as shown in Figure 17. The ACT Primary System implements five ACT functions—Full PAS, AAL, LAS, WLA, and FMC—and provides all active controls in the normal system condition. Essential PAS implements a simple short-period PAS control law and is engaged when the short-period PAS of the Full PAS has failed. These functions are described in the following subsections.

7.3.1 ESSENTIAL PAS SYSTEM

The Essential PAS System implements a highly reliable, fixed-gain short-period PAS that provides minimum acceptable handling quality in the pitch axis. Figure 32 shows a simplified block diagram of the Essential PAS System. To meet the reliability requirement, the system provides quadruple pitch-rate sensors, quadruple computers, and triple actuators, plus actuator mathematical models. The pitch-rate sensors are simple analog devices dedicated to the Essential PAS System. Each of the quadruple analog pitch-rate sensors is directly connected to all of the computers. The consolidated pitch-rate signals are processed by the SSFD algorithm to create a signal for control law computation and to monitor system failure status. The output of the Essential PAS control law is normally disconnected from the servocommand summing circuits. Full PAS failure status is monitored as shown in Figure 32. When the short-period PAS of the ACT Primary System fails, the fixed-gain short-period PAS is introduced into the loop using easy-on logic. The PAS and MLC elevator commands of the ACT Primary System are processed by the SSFD algorithm to form a summed elevator command and to monitor their failure status. To provide functional independence, elevator commands of short-period PAS, speed PAS, and MLC are processed separately by the SSFD algorithm. Outputs of the triple ACT Primary Computers are connected to Essential PAS Computers A, B, and C, where the signals are routed to other channels via digital cross-channel communication links. The computers are operated asynchronously; this feature of Essential PAS has been verified in laboratory testing. Three of the quadruple computers command the triple forced-summed elevator

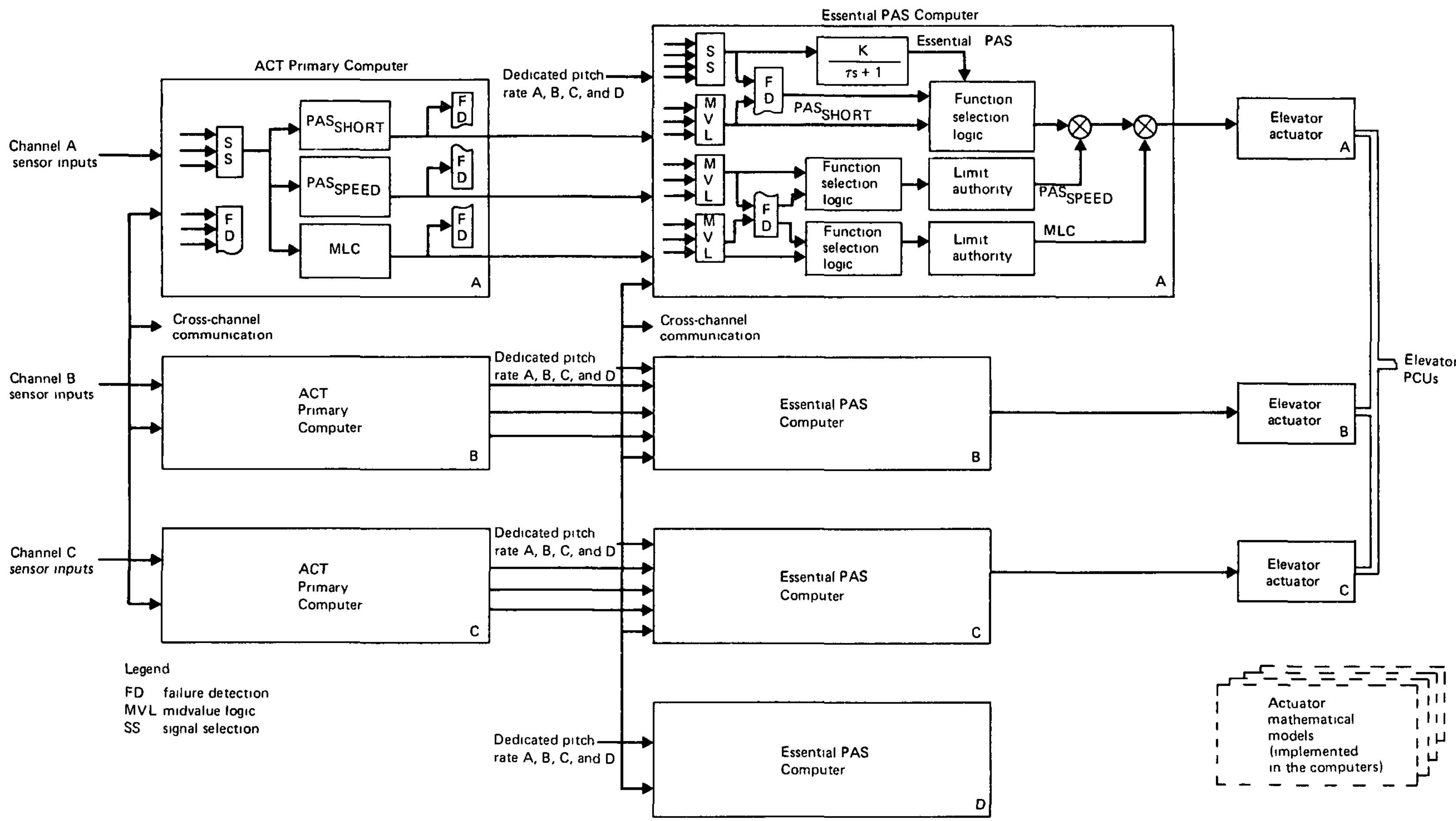


Figure 32. Pitch-Augmented Stability Block Diagram

actuators. Figure 11 shows the side-by-side force-summed secondary actuation concept used for the elevator control. Output of the secondary ACT actuator is series summed with the autopilot's or the pilot's mechanical control signal to form a command input to the elevator PCU. Each channel contains a conventional two-stage electrohydraulic servovalve that converts the input electric signal into hydraulic flow. The hydraulic flow displaces the actuator piston against the center spring. Valve spool and ram positions are fed back to each computer for servo-loop control and failure detection. A force detent is provided to serve as an antijam device.

7.3.2 ACT PRIMARY SYSTEM

The ACT Primary System implements five ACT functions in triple redundancy. The following paragraphs describe each function.

Full PAS—Full PAS includes both short-period and speed PAS to provide good handling quality in the pitch axis equivalent to or better than in a conventional airplane. Figure 33 shows a block diagram of Full PAS. Short-period PAS uses pitch-rate signals from the triple IRS sensors and control column signals to generate an elevator command. The pitch-rate and column signals are sent to the dedicated primary computers where the signals are routed to other primary computers via cross-channel communication. The feedforward and feedback gains of the short-period PAS are scheduled as a function of airspeed. In case of airspeed sensor (DADC) failures, the gain schedule becomes dependent upon flap position signals. If the IRS signals are lost, pitch axis control deteriorates to a minimum acceptable level. However, the pilot will be able to control the airplane with the combination of the remaining speed PAS and the Essential PAS System. The primary computer must forward status information on its short-period PAS to the essential computer to enable timely engagement of the Essential PAS control law.

The output of the elevator actuators is used to relieve a steady-state elevator trim deflection. When the elevator deflection exceeds a certain threshold value for more than a predetermined time, the elevator offload logic acts to adjust pitch trim by moving the horizontal stabilizer through the stabilizer trim interface in the CSEU. The CSEU trim function receives the autopilot, pilot, and ACT trim inputs and selects an appropriate signal to command the stabilizer position. Figure 34 shows a simplified block diagram of the elevator offload function.

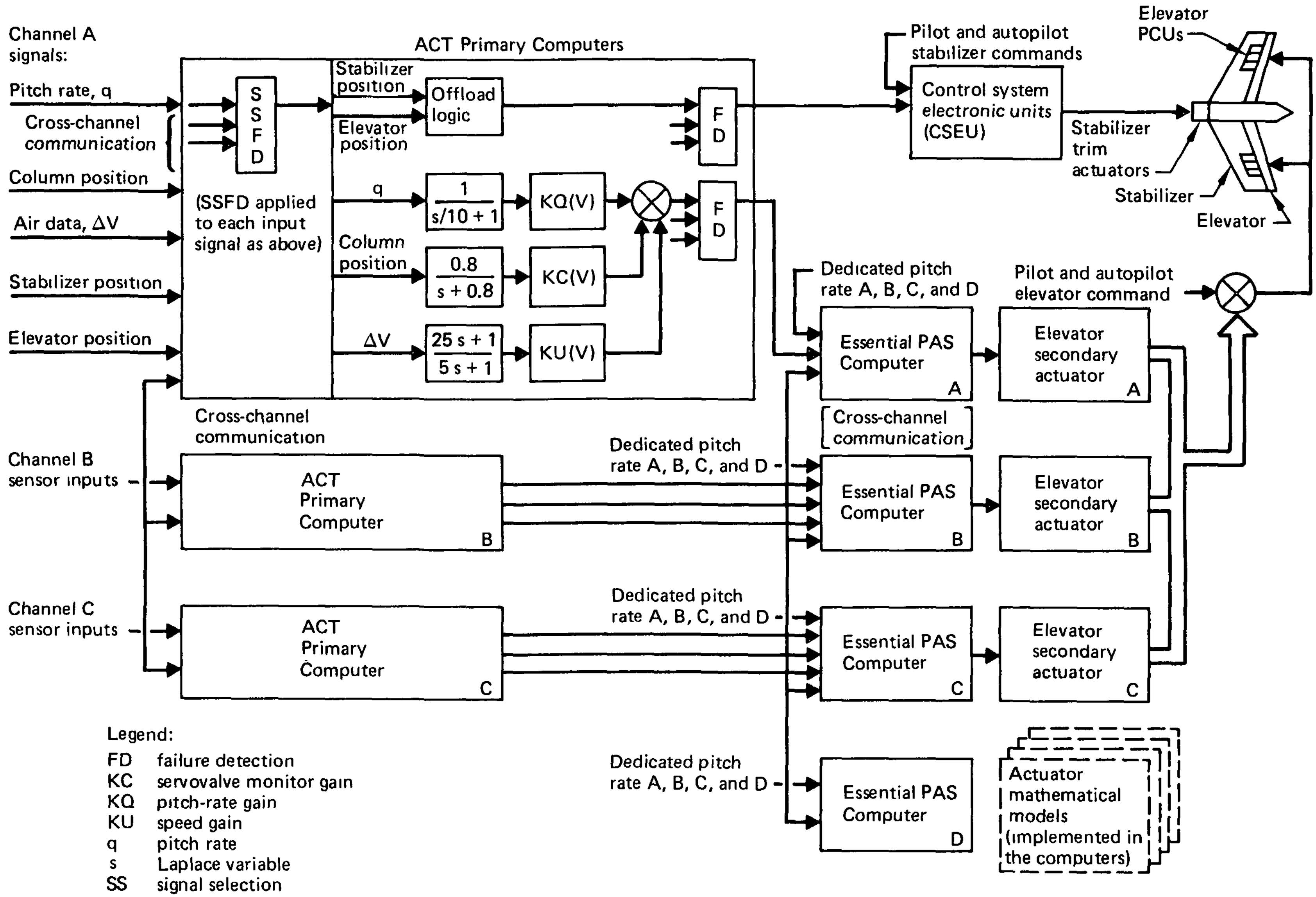


Figure 33. Full Pitch-Augmented Stability Block Diagram

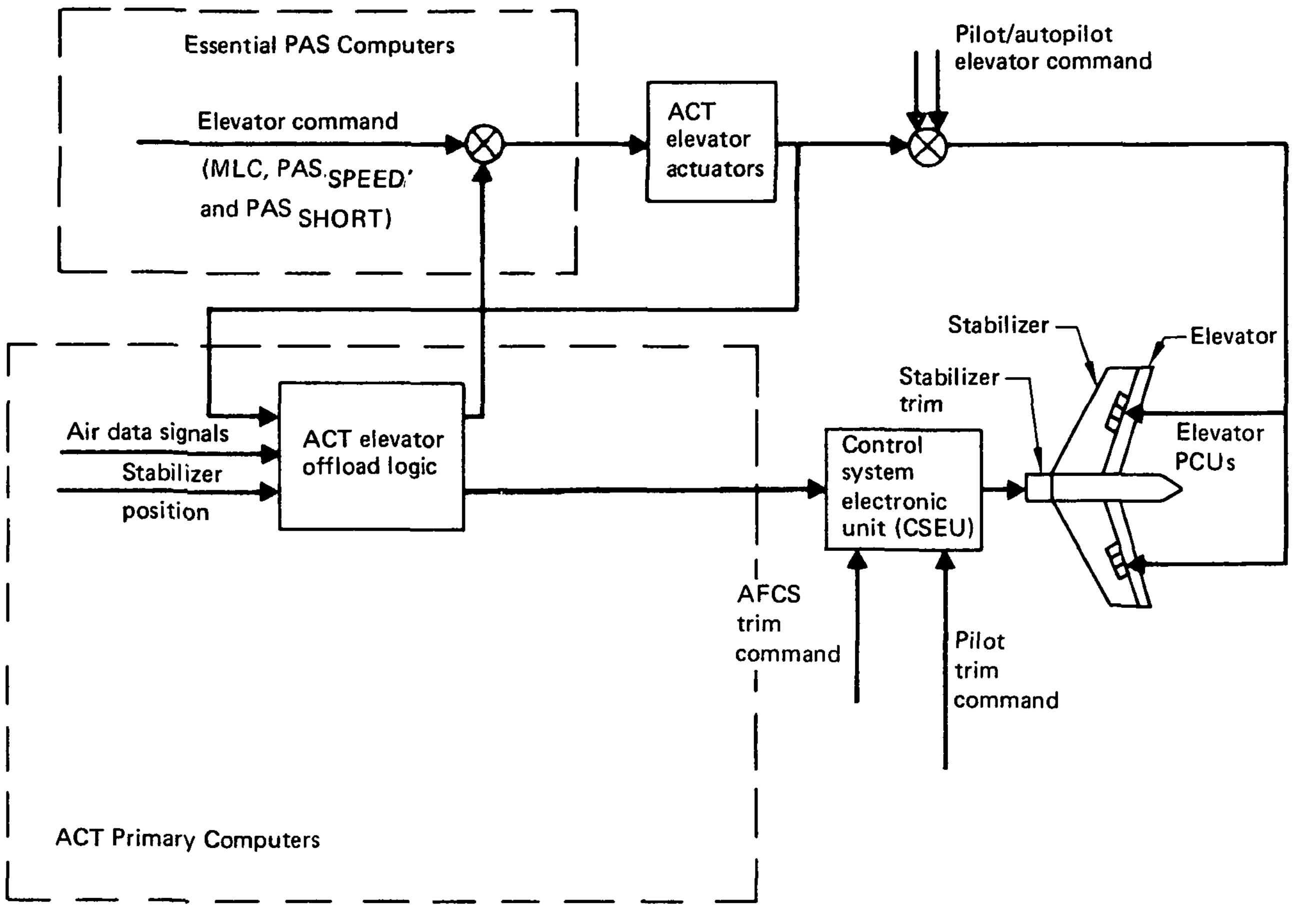


Figure 34. ACT Elevator Actuator Offload Function

Angle-of-Attack Limiter (AAL)—The AAL function includes the stick shaker and the stick pusher. Figure 35 is a block diagram of the AAL subsystem. Maximum allowable angle of attack, the AAL reference point, is defined as a function of leading-edge slat and trailing-edge flap position and airspeed. When the difference between the actual angle of attack and the reference angle declines below a certain threshold, the stick shaker is activated to provide stall warning. Pitch-rate signals shown in Figure 35 provide anticipation; i.e., if the airplane rapidly approaches the stall condition, the stick shaker will be activated sooner by the pitch-rate signal. If no pilot action results from the stick shaker warning, the stick pusher operates to apply an airplane nose-down force on the control column. Figure 36 shows the AAL actuation system of the Selected System. The actuator uses low-pressure air from the basic airplane engine bleed system and pressure from an accumulator as dual power supplies to meet the fail-operational requirement. Actuation

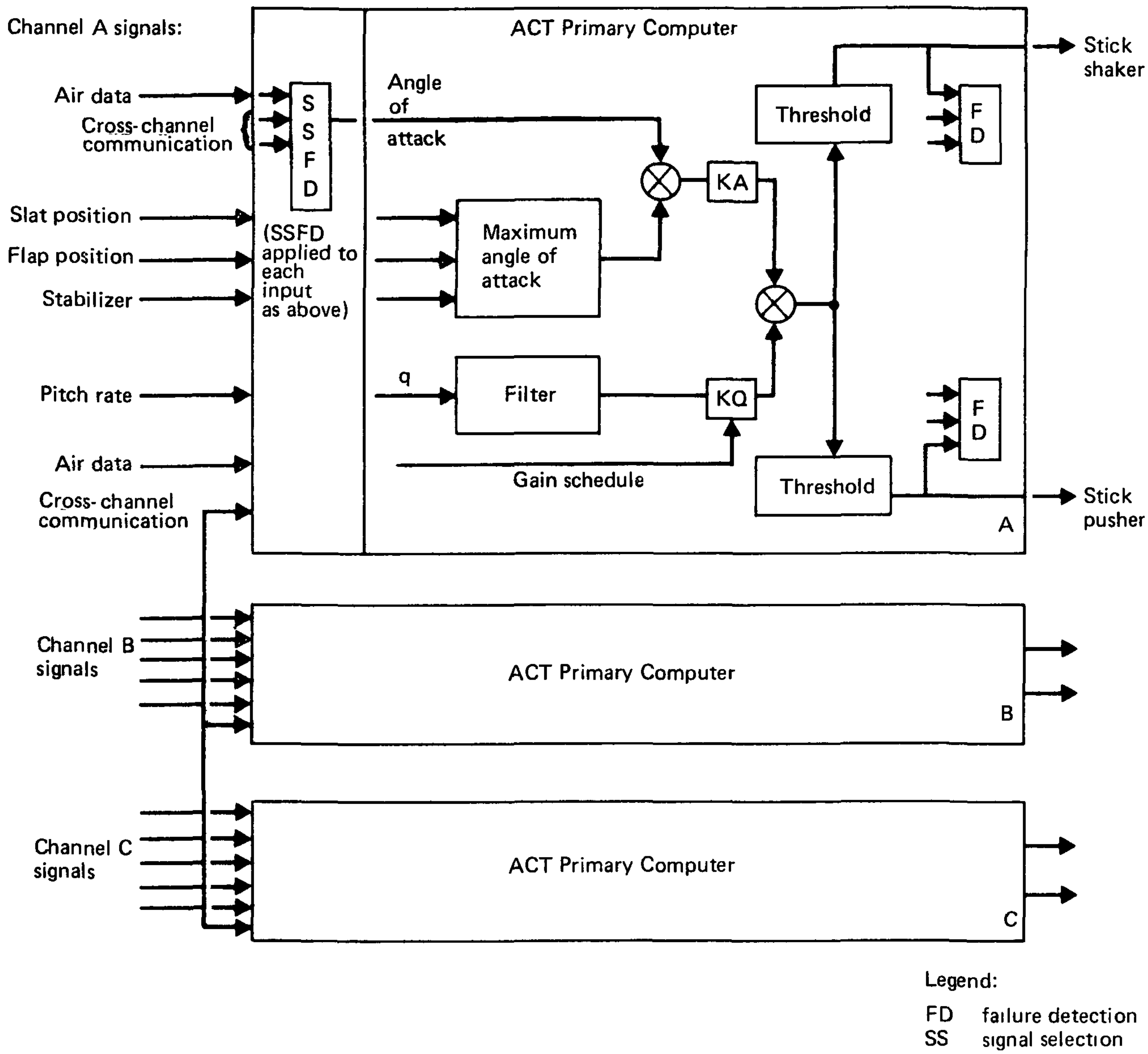


Figure 35. Angle-of-Attack Limiter Block Diagram

time of the stick pusher actuator is approximately 0.2 sec. The pilot may override the pusher at any time by exerting sufficient force on the column or by operating a manual dump that directly vents the actuator to ambient.

Lateral/Directional-Augmented Stability (LAS)—The LAS function, which includes the conventional yaw damper and turn coordination originally implemented in the Baseline CSEU and automatic flight control computer, is implemented here in the triple ACT

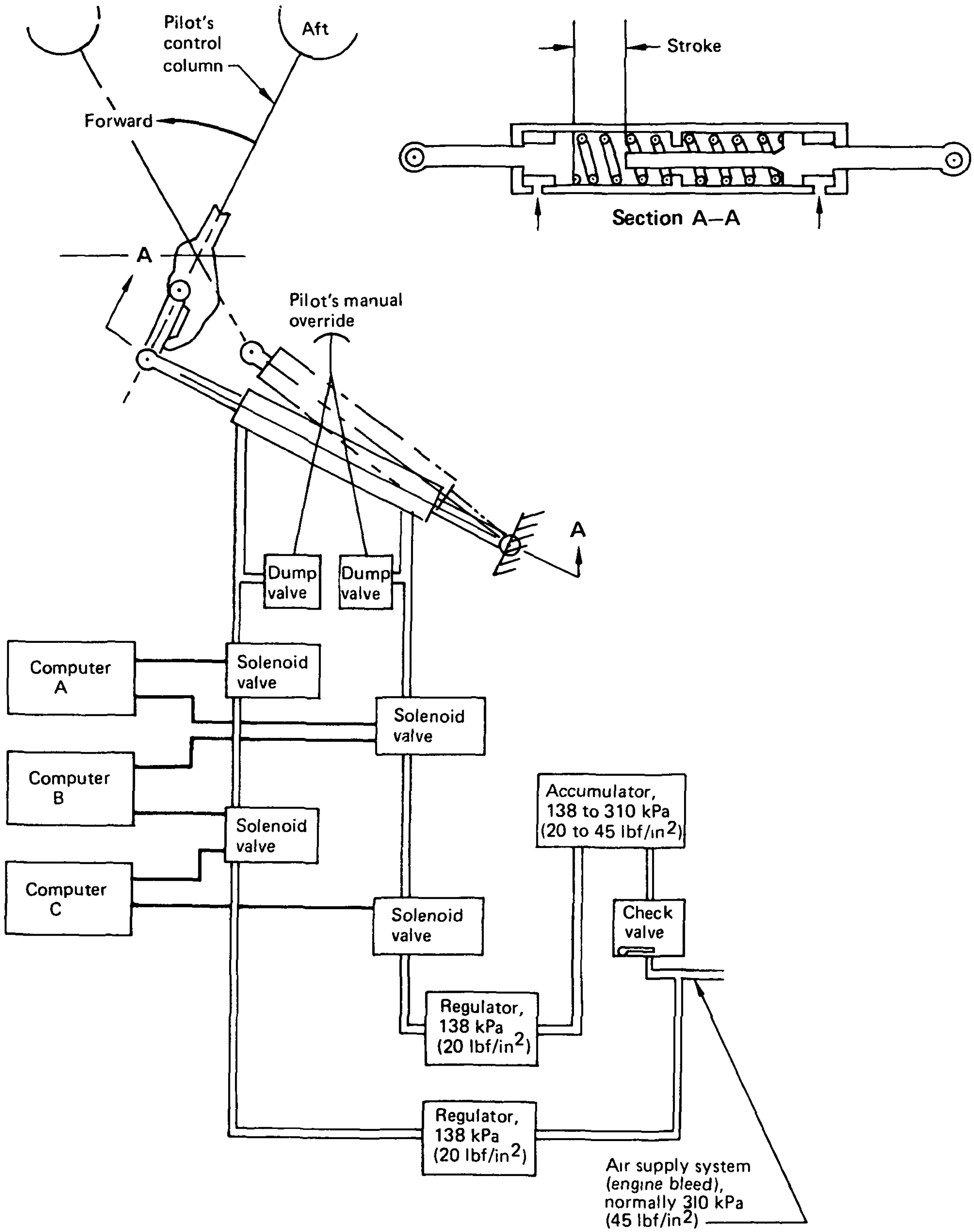


Figure 36. Stick Pusher Actuation Concept

Primary System. Figure 37 is a block diagram of the LAS function. The yaw rate and bank angle from the triple IRSs and control wheel signals are used to improve dutch roll damping and reduce side-slip angle. Air data signals from the triple DADCs are used as gain schedule inputs to provide good handling quality equivalent to or better than that of a conventional airplane. In case of DADC loss, LAS will use flap position as a backup gain schedule input. The ground rule of Subsection 5.1 states that single-thread operation is permitted if the function meets the following three conditions:

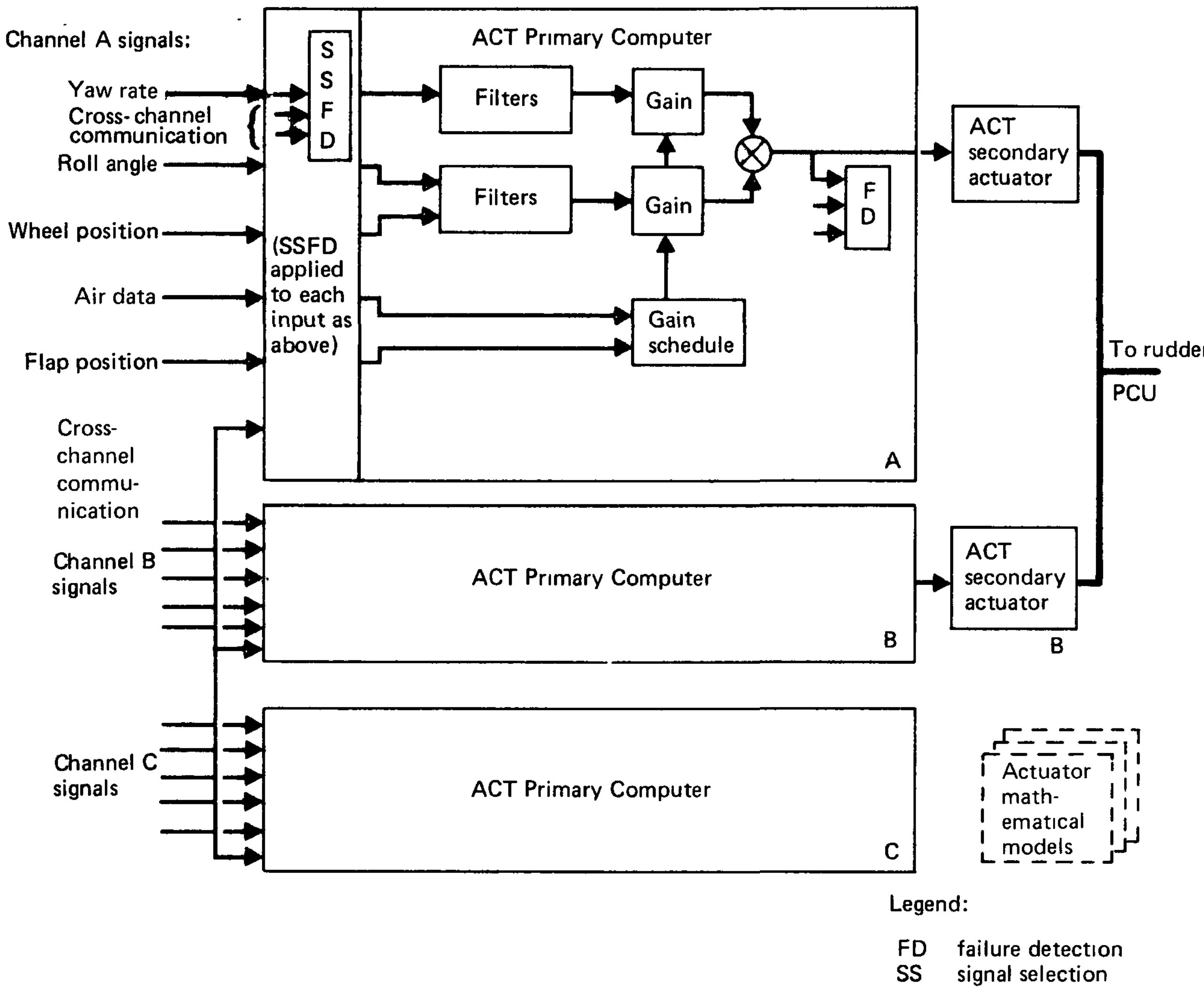


Figure 37. Lateral/Directional-Augmented Stability Block Diagram

- Adequate self-monitor can be provided.

- Pilot is able to detect system failure by observing airplane performance or other cues.

- Authority and rate limit can be provided in the actuators to prevent hardover or oscillatory failures causing structural damage.

LAS can meet these three conditions. Thus the airplane can dispatch with restriction when a single channel (one success path) is available in the LAS system.

Two of the triple ACT Primary Computers command the rudder control surfaces through secondary force-summed actuators similar to the elevator actuator. Mathematical actuator models are implemented in the triple computers to meet the fail-operational requirement.

Wing-Load Alleviation (WLA)—The WLA function, which has the most complex I/O interfaces because of the large number of airplane control surfaces used, is implemented in the triple ACT Primary Computers. The function consists of maneuver-load control (MLC) and gust-load alleviation (GLA) subfunctions. By moving the outboard aileron, MLC controls wing loads induced by pilot maneuver and reduces low-frequency gust loads. To compensate for the pitching moment caused by symmetric aileron motion, an elevator command is sent to the Essential PAS Computer, where the total ACT elevator command is formed by summing command inputs from these functions. In addition to the elevator and aileron, the WLA function uses inboard and outboard flaperons.

Figure 38 is a block diagram of WLA showing its sensors and control surfaces. The function uses center-of-gravity and wing acceleration signals as inputs to control law computation. Air data signals from the triple DADCs are used to gain schedule the control laws. WLA shares the inner segment of the outboard aileron with FMC and manual pilot control. The control wheel input is converted into triple electric signals that are routed to the ACT Primary Computers. Command signals from WLA, FMC, and the pilot are there summed by software before being sent to the inner segment of the outboard aileron.

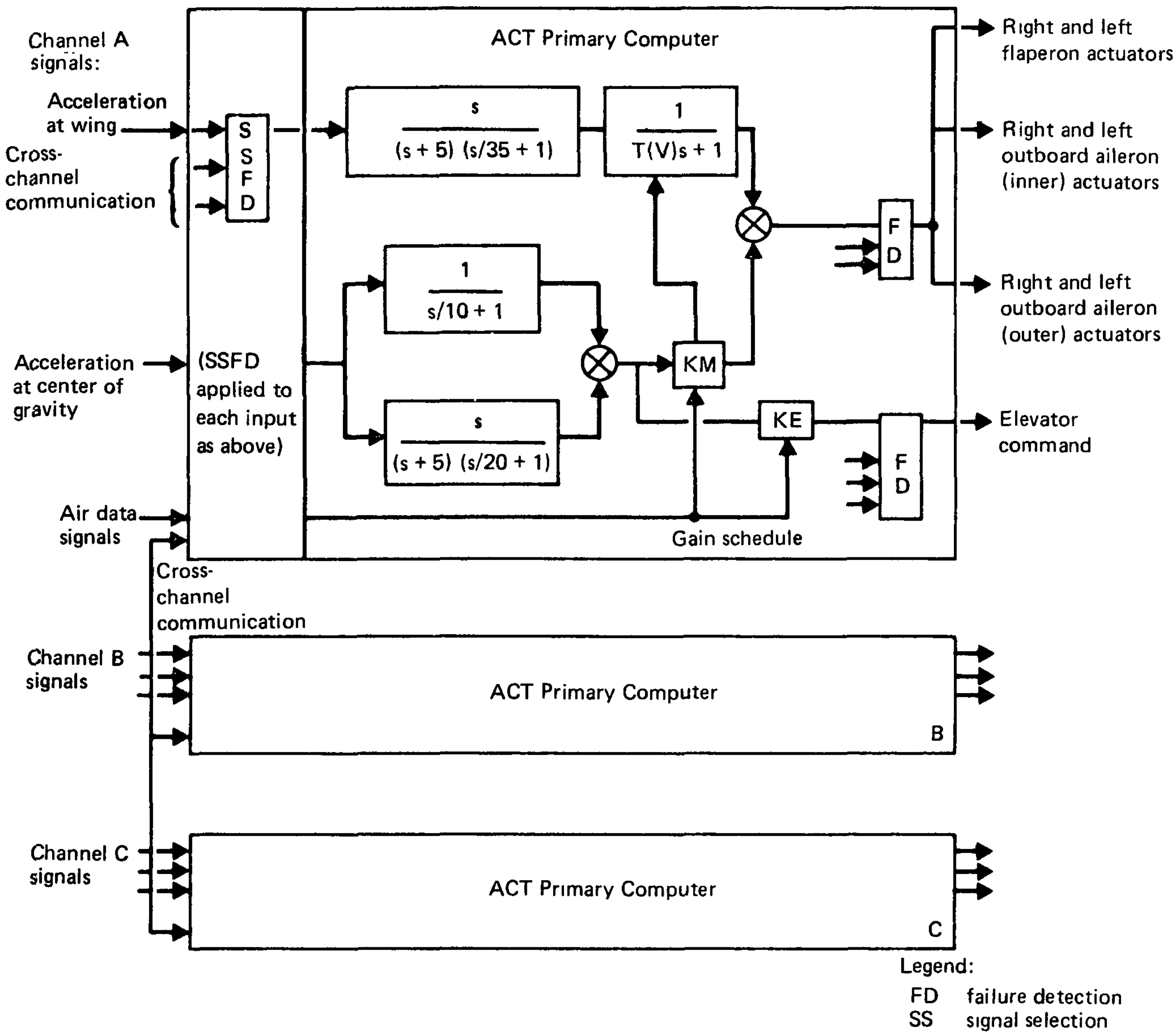


Figure 38. Wing-Load Alleviation Block Diagram

A force-displacement summed actuation concept is used in the outboard aileron inner segment actuator. Figure 12 is a simplified block diagram of this actuation concept. The command signals from each of three ACT Primary Computers are magnetically flux summed in the four first-stage electrohydraulic servovalves, two per hydraulic system. Output of the valves is mechanically position summed by a linkage to control the power valve. The dual pistons are force summed to control the inner section of the outboard aileron.

WLA also uses the trailing-edge flaperons as control surfaces. Two hydraulic systems are required to meet the fail-operational requirement of the WLA function. Hydraulic power must be supplied to the flaperons through swivel joints. The plan for ensuring hydraulic system safety in the event of swivel joint failure or flap separation is described in Subsection 7.1 and illustrated in Figure 18.

Flutter-Mode Control (FMC)—The FMC function suppresses flutter modes at airspeed between V_D and $1.2V_D$ by sensing wing acceleration and driving the inner segment of the outboard aileron. Figure 39 is a block diagram of the FMC function. Triple accelerometer outputs from right and left wings are processed by the SSFD algorithm to generate a signal for control law computation and sensor monitoring.

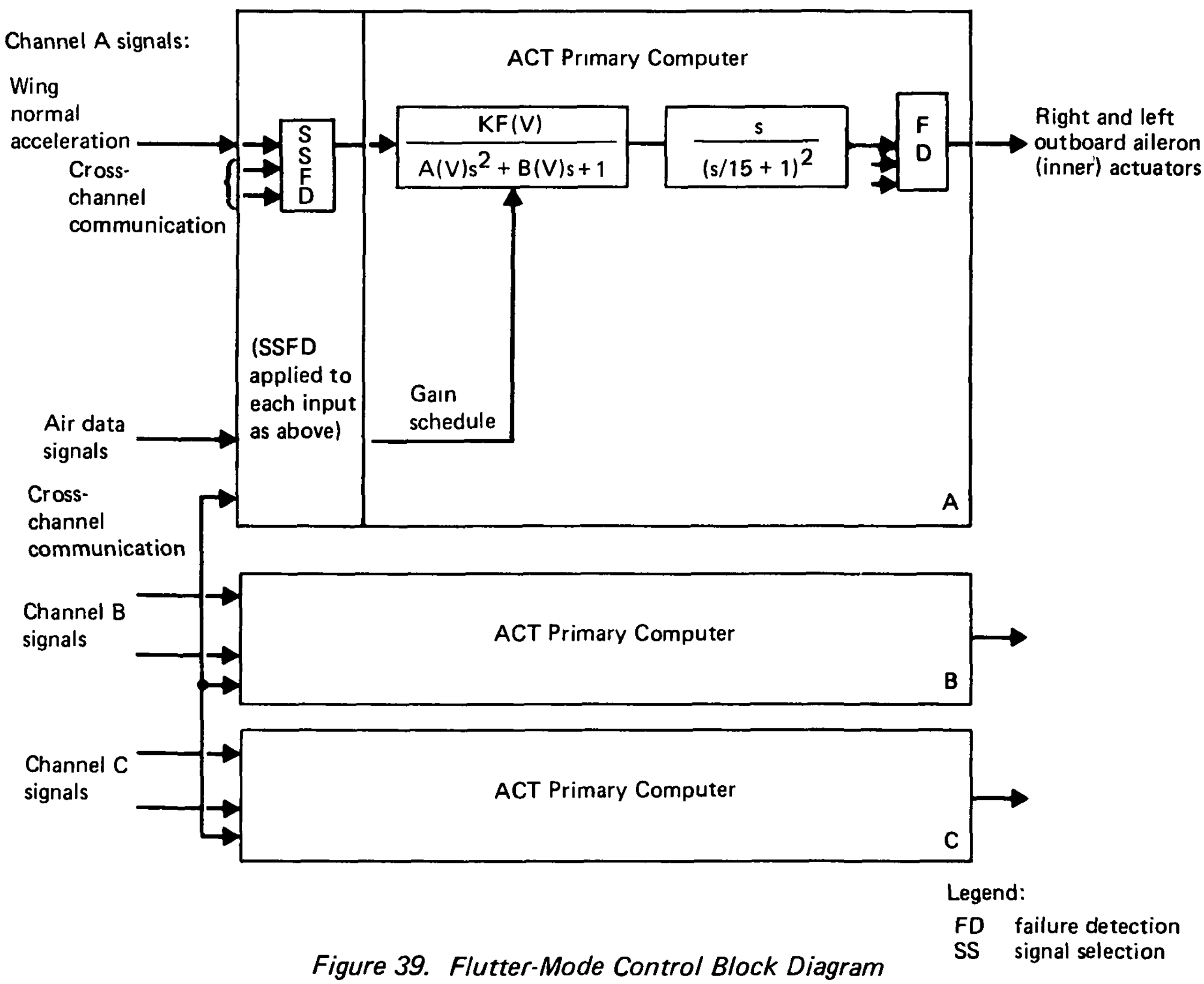


Figure 39. Flutter-Mode Control Block Diagram

7.3.3 ACT MAINTENANCE AND DISPLAY COMPUTER

Use of two separate sets of control computers presents a need for coordination of their communication functions in a separate unit. These functions are:

- Start test
- Enunciate significant failures
- Select appropriate operations advisory messages
- Retain fault information for maintenance use

The ACT Maintenance and Display Computer meets these needs. Its primary functions of output to the crew are shown in Figure 40.

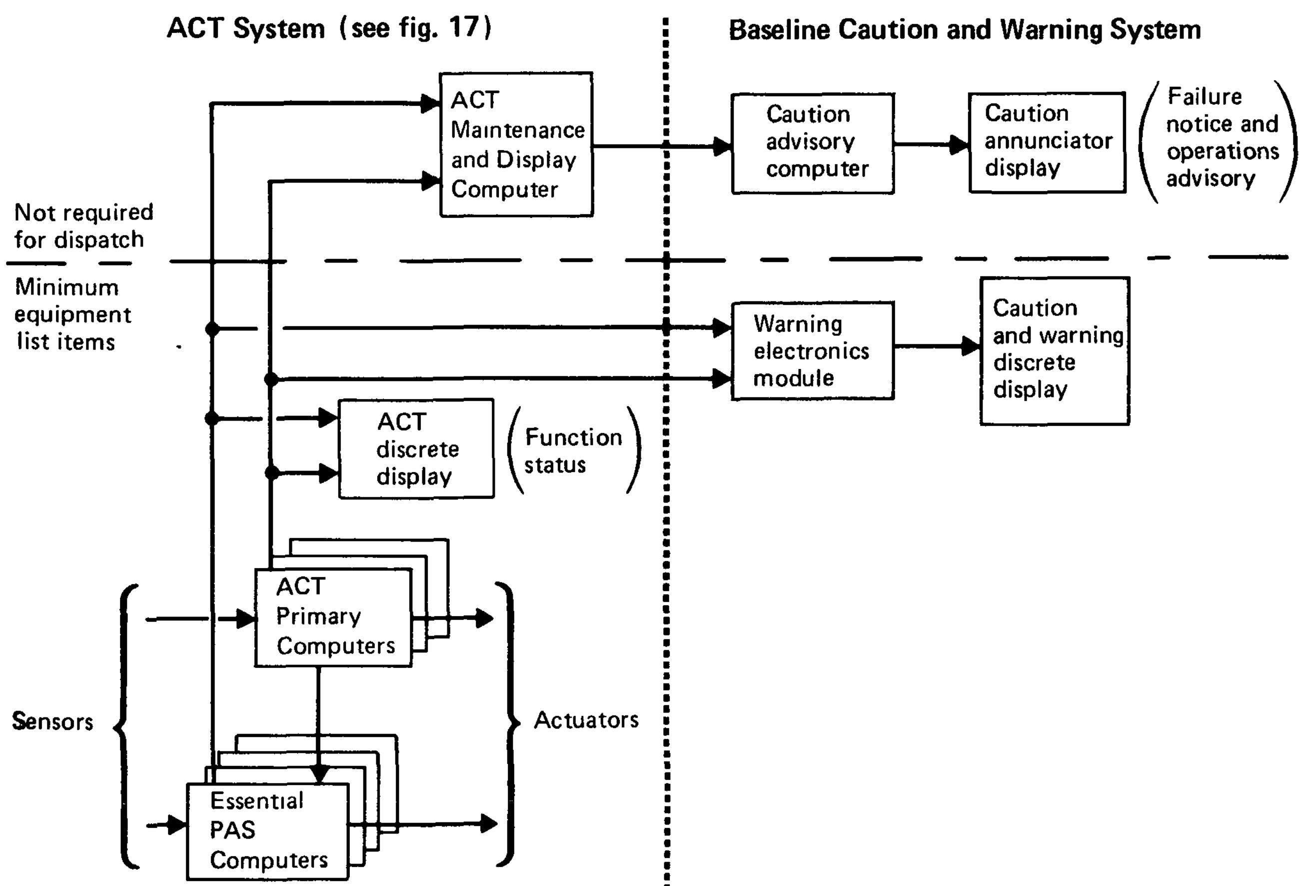


Figure 40. ACT Displays

In its normal mode of operation, the ACT Maintenance and Display Computer accepts fault information from the ACT Primary Computers and the Essential PAS Computers. This information is processed in the ACT Maintenance and Display Computer microprocessor to determine status of both functions and the resulting advisory message if any is required. By using this processing and the caution and warning system of the Baseline Airplane, the crew is presented with essential fault information and the resulting operation change requirements. Thus, when the ACT Maintenance and Display Computer is active, the pilots do not need to refer to an operations manual for response to ACT system failure.

A more detailed picture of the functions of this unit is contained in Figure 86.

7.4 ACT SYSTEM OPERATION CONCEPT

In normal operation, the ACT system is entirely automatic and transparent to the flight crew and requires no crew attention. It also affects safety of flight, and hence the crew must be kept apprised of its condition. These facts lead to the system operation concept that follows.

The crew communication functions are held to the minimum in both directions, crew to system and system to crew. The controls provided are only those needed for the crew to choose the time for preflight testing and for emergency manual disconnect. The displays provided are only those needed to communicate essential failure information and to (1) enable the crew to make appropriate dispatch and flight plan adjustment decisions under failure conditions and (2) supply data for maintenance essential to dispatch. These provisions are described in Subsection 8.4.3.

With this system operating unattended by the crew, self-testing and self-monitoring become essential for tracking soundness of the system. The extensive digital computation capacity shown in Figure 17 makes that testing and monitoring practical. As stated in Subsection 5.3.1, one ACT function is crucial and all others are critical. All must be operating for dispatch without any flight restriction, and lack of any one of five of the ACT functions will prevent dispatch. Therefore, preflight test must be performed to verify that the airplane, from the ACT system point of view, is dispatchable. Similarly, in-flight monitoring must be done to keep track of ACT control system soundness and enable retreat to safe operating conditions if failures make that necessary. Finally, the effectiveness of ACT system maintenance operations must be rapidly verifiable; e.g., when a "no go" ACT airplane undergoes maintenance action at the flight line, such action must be quickly verified to restore dispatch status as early as possible.

The preceding paragraphs refer to communication from the ACT system to the air crew in the event of some fault in the system. Figure 41 is a block diagram illustrating the system response to such faults; it shows a two-part parallel communication process that is effective during flight and preflight test. In the first part, the ACT Primary and/or Essential PAS Computers detect a line replaceable unit (LRU) fault, determine the corresponding ACT function status, and report that status to a dedicated ACT display. Then, from that information, possible prior faults, and current operating conditions, the

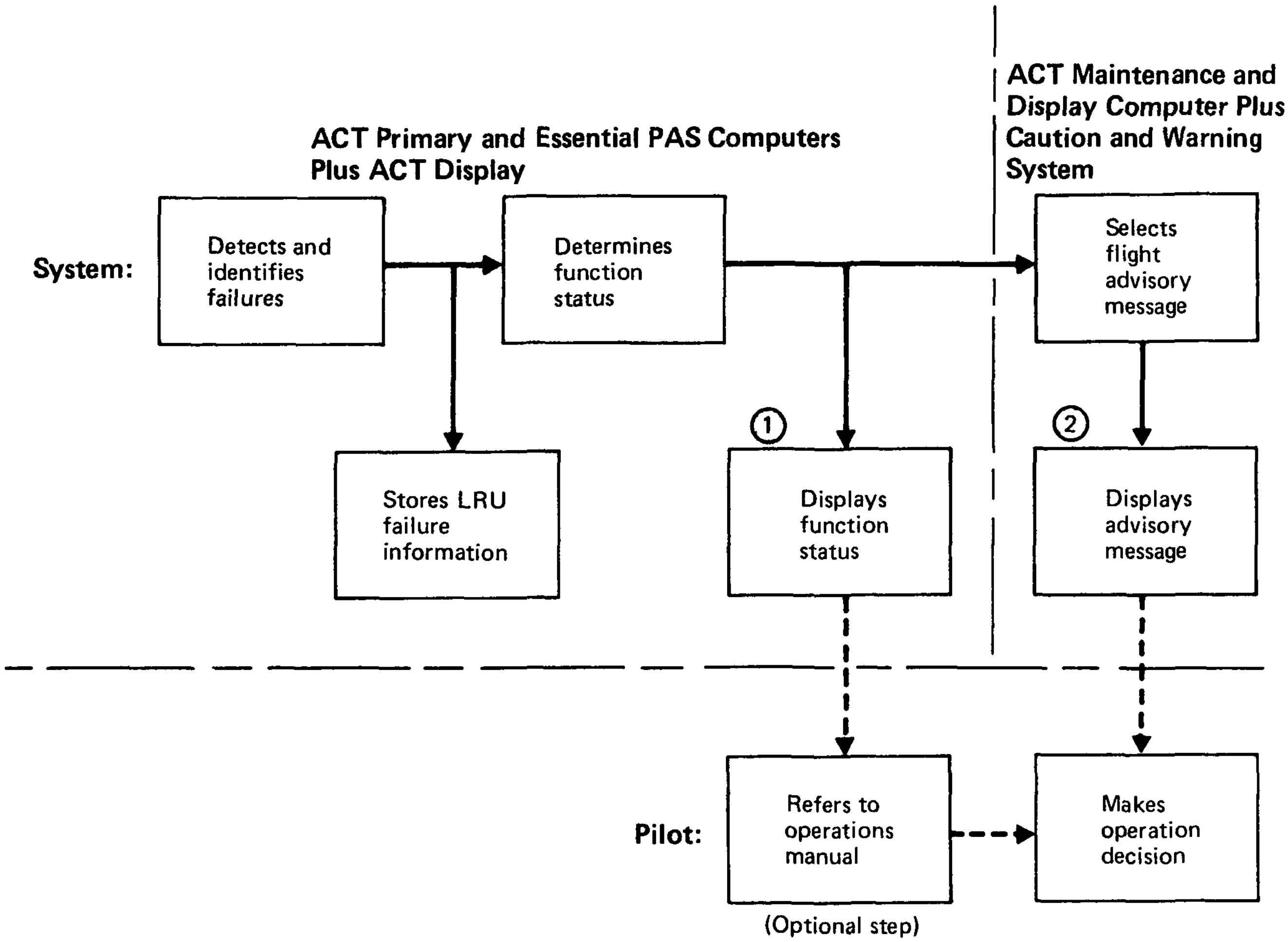


Figure 41. Response to ACT System Failures

ACT Maintenance and Display Computer determines the operations advisory message (if any is needed) for transmission to the crew. Thus, the pilot can make his dispatch or flight plan change decision in either of two ways:

- Observe the ACT status display and refer to an operations manual, the "manual option" shown at 1 in Figure 41

- Observe the operations advisory, the "automatic option" shown at 2 in the figure

Subsection 8.4.4 treats the fault response processes in greater detail.

8.0 SELECTED SYSTEM DETAILED DESCRIPTION

This section describes, in detail, the system selected from studies described in Section 6.0. The description covers physical components such as sensors, computers, and actuators and also the design concepts involved in software, redundancy management, and system operation.

Subsection 8.1 describes those sensors, computers, and actuators that are a part of the Active Controls Technology (ACT) system. It discusses the way the components of the Baseline are changed in form or use to accommodate the ACT functions. Subsection 8.2 describes redundancy management and failure protection aspects of the ACT system. Subsection 8.3 is concerned with the software used to control the computers and with the memory requirements. Subsection 8.4 discusses system operation, test and maintenance, communications to and from the crew, monitors, and fault response. Subsection 8.5 describes how the hydraulic system is changed from the Baseline Airplane to accommodate the needs of the ACT system. Subsection 8.6 describes changes to the electric system.

8.1 SYSTEM COMPONENTS

8.1.1 SENSORS

The Selected System shares sensors with the automatic flight control system (AFCS) and display functions where appropriate. The Baseline Airplane provides many of the sensors required for the ACT functions; some special sensors must be added to meet ACT system standards of performance and redundancy. Figure 10 shows general locations of the ACT sensors. Table 7 lists all required sensors and associates them with the ACT functions that they serve. Table 8 is a condensed table of sensor specifications.

The crucial short-period pitch-augmented stability (PAS) function must have quadruple redundancy to meet the reliability requirement. The Baseline Airplane senses the pitch rate in triplex by using the inertial reference systems (IRS). Addition of a fourth IRS is not economical. Furthermore, the Baseline IRS has a high failure rate (as shown in subsec 9.2.1), which is a severe drawback in a sensor for the crucial PAS control law. It is

Table 7. Sensors for ACT Systems

Sensed quantity \ ACT function	PAS_{SHORT}	PAS_{SPEED}	Wing-load alleviation		LAS	FMC	AAL	
			MLC	GLA				
Vertical acceleration at center of gravity			IRS,* table A					Input sensors
Vertical acceleration (wing)				Accelerometer,** table B		Accelerometer,** table B		
Pitch rate (body)	IRS, table C VYRO,* table C						IRS, table C	
Yaw rate and roll angle (body)					Inertial reference system, table D			
Airspeed/Mach number			Digital air data computer, table E					
Control column force			Force transducer,** table F					
Angle of attack							DADC, table G	
Flaperon servo position			LVDT,** table J (inboard) (outboard)					Servo feedback sensors
Outboard aileron, outboard segment servo position			LVDT,** table J (included in secondary actuator)					
Outboard aileron, inboard segment servo position			LVDT,** table J (included in dedicated actuator)			LVDT,** table J		
Elevator servo position	LVDT,** table J (included in secondary actuator)							
Rudder servo position					*LVDT,*** *table J* (included in secondary actuator)			
Stabilizer servo position		LVDT,** table J						

*"Table" refers to Table 8.
**Sensors added for ACT

IRS inertial reference system
DADC digital air data computer
LVDT linear variable differential transformer
VYRO pitch-rate sensor (trade name)

Table 8. Sensor Specifications

	Sensed quantity	Instrument	Range	Sensitivity or accuracy	Excitation
A	Vertical acceleration at center of gravity	Inertial reference system (IRS)	±4g	±0.01g	115V, 400 Hz, 28V dc
B	Vertical acceleration (wing)	*Accelerometer*: cg, front spar*	±5g	1V dc/g	28V dc
		Accelerometer*: rear spar	±20g	0.25V dc/g	28V dc
C	Pitch rate (body)	IRS	±1.22 rad/s	0.0017 rad/s or 1%	115V, 400 Hz, 28V dc
		VYRO*	±1.22 rad/s	0.012 rad/s or 1%	12V dc
D	Yaw rate (body)	IRS	±0.7 rad/s	0.0017 rad/s or 1%	115V, 400 Hz, 28V dc
E	Airspeed	Digital air data computer (DADC)	±1024 kn	±1 to 4 kn, depending on speed	115V, 400 Hz
F	Control column force	Linear variable differential transformer (LVDT)*	±529N	0.0058 V/N	26V, 400 Hz
G	Angle of attack	Digital air data computer	±1.05 rad, electrical ±2.1 rad, mechanical	±1.5 V/rad	26V, 400 Hz
H	Model channel position feedback	LVDT**	±0.019m	±0.5%	26V, 400 Hz
J	Surface servo position feedback	LVDT**	±0.019m	±0.05%	26V, 400 Hz
K	Hydraulic pressure failure detector	LVDT**	±0.005m	±1%	26V, 400 Hz

*Sensors added for ACT
**Typical of several; used in various functions

essential to have a small and reliable source of pitch-rate signal for the ACT system. The VYRO, a small, long-life, vibrating beam sensor designed by General Electric, is one of the acceptable sensors that can supply the quadruple pitch-rate signal.

The airspeed variables shown in Table 7 are needed for gain variation schedules in several control loops. The table also shows the control surface servo linear variable differential transformers (LVDT) that are used to close the servo loops and also to monitor failures.

8.1.2 COMPUTER ARCHITECTURE

The Selected System uses three different types of computers: the ACT Primary Computers, the Essential PAS Computers, and the ACT Maintenance and Display Computer. Each shares the following characteristics:

- Digital implementation for design flexibility and self-test capabilities

- Fault-tolerant design for flight-crucial and flight-critical function implementation, including capability to recover from transient failures

- Fault detection and identification capability to enhance maintainability of the overall system

8.1.2.1 ACT Primary Computers

The ACT Primary Computer provides commands for flight-critical ACT functions. This computer shares many of the architectural features of the Airborne Advanced Reconfigurable Computer System (ARCS) computer described in Reference 6 and is similar to flight control computers used in the current generation of commercial transports. The ACT Primary Computers are used in a triply redundant set, with each computer having identical hardware and software. The computers are loosely synchronized on a 20-ms major frame.

The major sections of the ACT Primary Computer are central digital processing, input/output (I/O), servoelectronics, discrete output voters, and power supplies. Figure 42 is a block diagram of the computer. The digital processing section includes the central

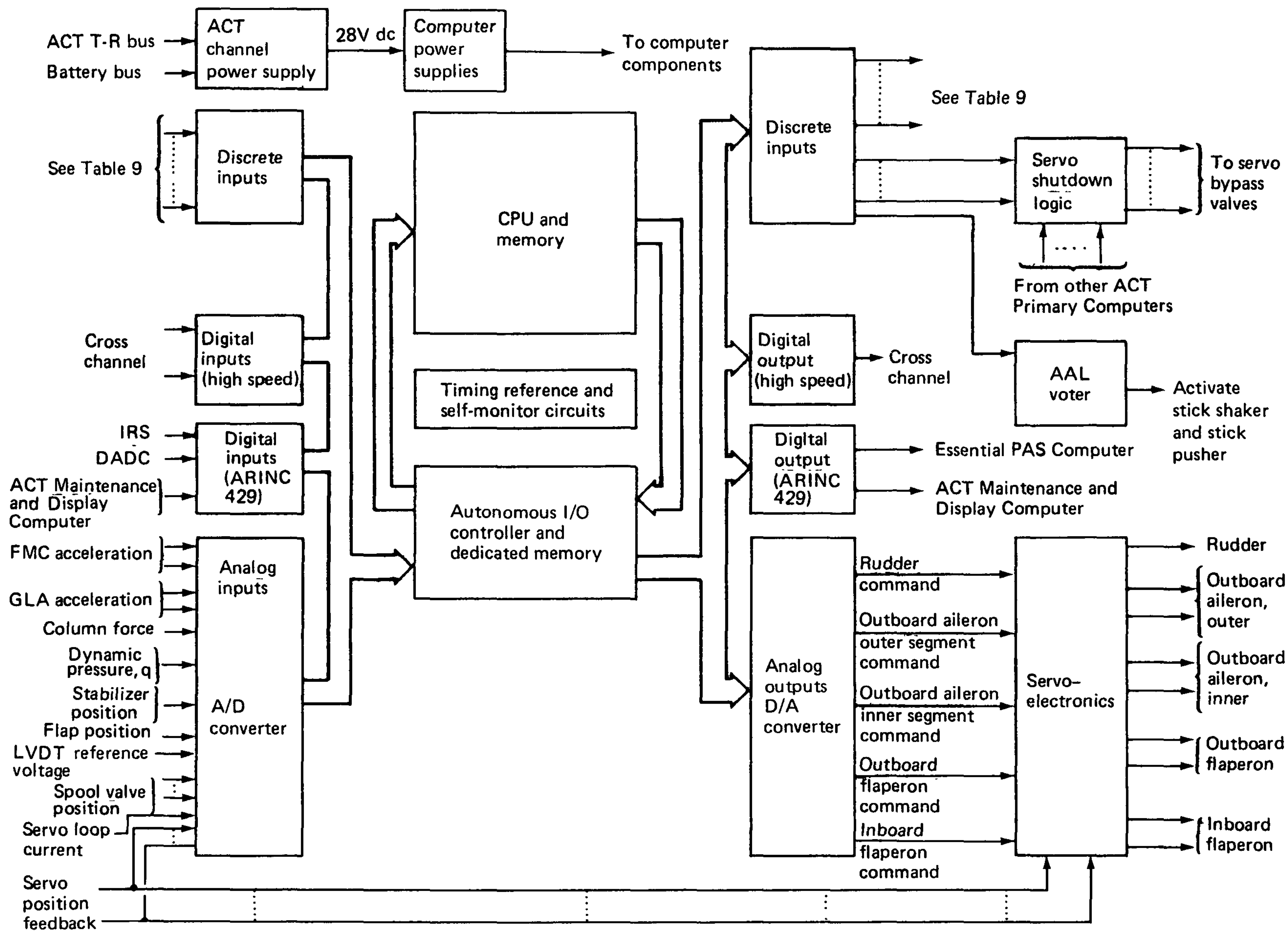


Figure 42. ACT Primary Computer Block Diagram

processing unit (CPU), program memory, variable memory, iteration reference timer, and self-monitor circuits. The CPU is a general-purpose, 16-bit parallel processor with an instruction set structured to accommodate flight-critical control functions. The memory includes a minimum of 20K read-only memory (ROM) for program storage, with expansion capability to at least 32K, and 2K of read and write random-access memory (RAM) for variable storage. The iteration reference timer generates an interrupt signal to the CPU to provide a 5-ms minor frame. The reference timer also provides signals to the I/O section. Self-monitor circuits include a watchdog timer, power monitor, arithmetic overflow monitor, memory parity monitor, and digital communication monitors. The watchdog timer confirms that the CPU is responding to reference timer interrupts and can perform a series of tasks. It also checks the operation of the reference timer. The power monitor checks voltage levels and generates interrupts when voltage drops below required levels and when it returns to proper levels. Power monitor interrupts are nonmaskable. The arithmetic overflow monitor generates an interrupt whenever an operation is performed that results in an overflow. The memory parity monitor checks validity of data received from memory by generating a fault interrupt if parity is incorrect; i.e., the data word contains the wrong number of ones. Digital communication monitors check for errors in data transmitted over serial bus lines. These monitors are physically located in the I/O section of the computer in digital I/O interfaces.

Data flow within the digital processing section of the computer is via internal parallel buses.

The I/O section of the computer consists of analog, digital, and discrete interfaces, plus an autonomous controller and dedicated memory. This section provides communication between the digital processing section and external devices such as sensors and servodrives. Table 9 summarizes the I/O requirements of the ACT Primary Computers.

Each I/O interface has a dedicated variable memory associated with it. Data are placed into or taken directly from this memory, which can also be addressed by the CPU. The analog input section includes signal conditioners, antialiasing filters, demodulators, and analog-to-digital (A/D) converters. The analog output section includes digital-to-analog (D/A) converters and amplifiers. The discrete I/O section provides buffers for system discretes at required voltage levels. The digital I/O section includes Aeronautical Radio

Table 9. ACT Primary Computer Inputs and Outputs

	Inputs	Outputs
Power	ACT channel 28V dc power	
Digital	Air data (ARINC 429) Inertial reference (ARINC 429) ACT Maintenance and Display Computer (ARINC 429) Essential PAS Computer (ARINC 429)	ACT Maintenance and Display Computer Essential PAS Computer (ARINC 429) Cross channel (high speed)
Analog	FMC acceleration, left and right GLA acceleration, left and right Column force Dynamic pressure Stabilizer position Flap position Wheel position Center-of-gravity acceleration Outboard aileron outer segment servo spool valve position, left and right Rudder servo spool valve position Outboard aileron inner segment servo loop current, left and right Outboard aileron outer segment servo position, left and right Outboard aileron inner segment servo position, left and right Outboard flaperon servo position, left and right Outboard flaperon servo position, left and right Rudder servo position LVDT reference voltage	Rudder position command Outboard aileron outer segment position command Outboard aileron inner segment position command Outboard flaperon position command Inboard flaperon position command
Discrete	Air and ground logic (2) Test initiate (3) Electric power monitor (4) Hydraulic pressure monitor (3) Pneumatic pressure monitor (2) Solenoid valve (AAL) position (4) Dump valve position (2) Outboard aileron inner segment servo failure (4) Slat position (3) Spare (13)	Warning displays (12) Self-test (6) Stick pusher activate Stabilizer drive (2) Shutdown actuators—servo failure (27) Shutdown foreign actuators—computer failure (15) PAS_{SHORT} failed PAS_{SPEED} failed MLC failed Spare (30)

Incorporated (ARINC) 429 serial data receivers and transmitters for communication with digital sensors and other system computers and high-speed serial data transmitters and receivers for cross-channel communication with the other ACT Primary Computers. A signal from the reference timer causes the autonomous I/O controller to load data from the various input sections into the cross-channel transmitter. No CPU action is required, thereby allowing transmission of sensor data to other channels after failure of the CPU. The controller also transfers computer output data to the cross-channel transmitter on a signal from the CPU. The I/O controller is designed to prevent a failed CPU from disabling the transmission of sensor data. The cross-channel transmission of sensor data is inhibited only when the computer power supplies or the computers are lost.

The servoelectronics section of the computer provides servo-loop closure and drive current for the electrohydraulic servos of the ACT system.

Discrete output voters are used to provide output discretes to shut down servos or to provide angle-of-attack limiter (AAL) actuation. The servo shutdown logic votes on outputs from each ACT Primary Computer and provides a signal to activate a servo bypass valve for the force-summed actuators or to electrically null the servodrive output for the force-displacement actuators. The servo shutdown logic receives signals on the status of computers and servos, and if a majority vote indicates either is bad, it shuts down the servo. Status signals from computers that are judged to be failed are ignored. Majority vote is normally required to shut down the servo, but if only two good computers remain and a disagreement occurs between these two, the actuator will be shut down. Each computer contains a servo shutdown logic voter for each of the surfaces driven by the ACT Primary Computers.

The AAL voter works in a manner similar to the servo shutdown logic, except when a disagreement occurs between two remaining computers; then the output is zero (i.e., the AAL actuator is not activated). Because there are only three ACT Primary Computers and four AAL actuators, each line replaceable unit (LRU) contains two AAL voters, although only one is used on two of the three ACT Primary Computers.

The power supply section provides power to all sections of the computer. Input power is supplied from a 28V dc ACT system bus, which is supplied from a main dc bus and from a standby battery bus. LVDT sensor excitation is provided by the ACT system ac buses, which are supplied from the dc buses. This same voltage is provided as a demodulation reference. Computer power outputs can sustain a short circuit without causing failure to internal voltage supplies.

8.1.2.2 Essential PAS Computers

The Essential PAS Computers provide crucial pitch augmentation. These computers require far less processing and I/O capability than the ACT Primary Computers but require higher reliability. The objective of the design of these computers is to keep them simple to enhance reliability. These computers are used as a quadruply redundant set and operate asynchronously.

Like the ACT Primary Computers, the Essential PAS Computers include a digital processing section, I/O section, servoelectronics, discrete output voter, and power supplies. Figure 43 is a block diagram of the Essential PAS Computers. The CPU requirements are such that they can be met by currently available 16-bit microprocessors. Use of a microprocessor saves size, weight, and cost and should provide improved reliability. It adds some constraints, as it is not feasible to design a microprocessor tailored to this application for a current technology system. This may lead to some simplification of the self-monitor circuits. The arithmetic overflow monitor may not produce an interrupt but instead may require a flag to be read under software control. The memory parity monitor is deleted. All other monitors are functionally the same as those in the ACT Primary Computers. Memory is reduced to 11K of ROM and approximately 256 words of RAM. Real-time programs are contained within approximately 2K words of the 11K of ROM, and this memory is physically separate from the non-real-time program memory.

The Essential PAS Computers require significantly less I/O than the ACT Primary Computers. Table 10 summarizes the I/O for the Essential PAS Computers. Analog sensors are cross strapped (i.e., each of the four computers receives a signal directly from all analog sensors), so the autonomous I/O controller has been deleted from the Essential PAS Computers. All I/O is done with CPU control. As a result of the reduced I/O requirement, the system requires relatively low cross-channel data rates and a high-speed cross-channel data link is not required. An ARINC 429 standard bus may be used. This provides uniformity of digital communication interfaces.

Servoelectronics, discrete output voters, and power supplies are functionally the same as in the ACT Primary Computers, except the power supply must provide a $\pm 12V$ dc excitation for the pitch-rate sensor. One discrete output voter is required in the Essential PAS Computer to provide servo shutdown logic for the elevator servos.

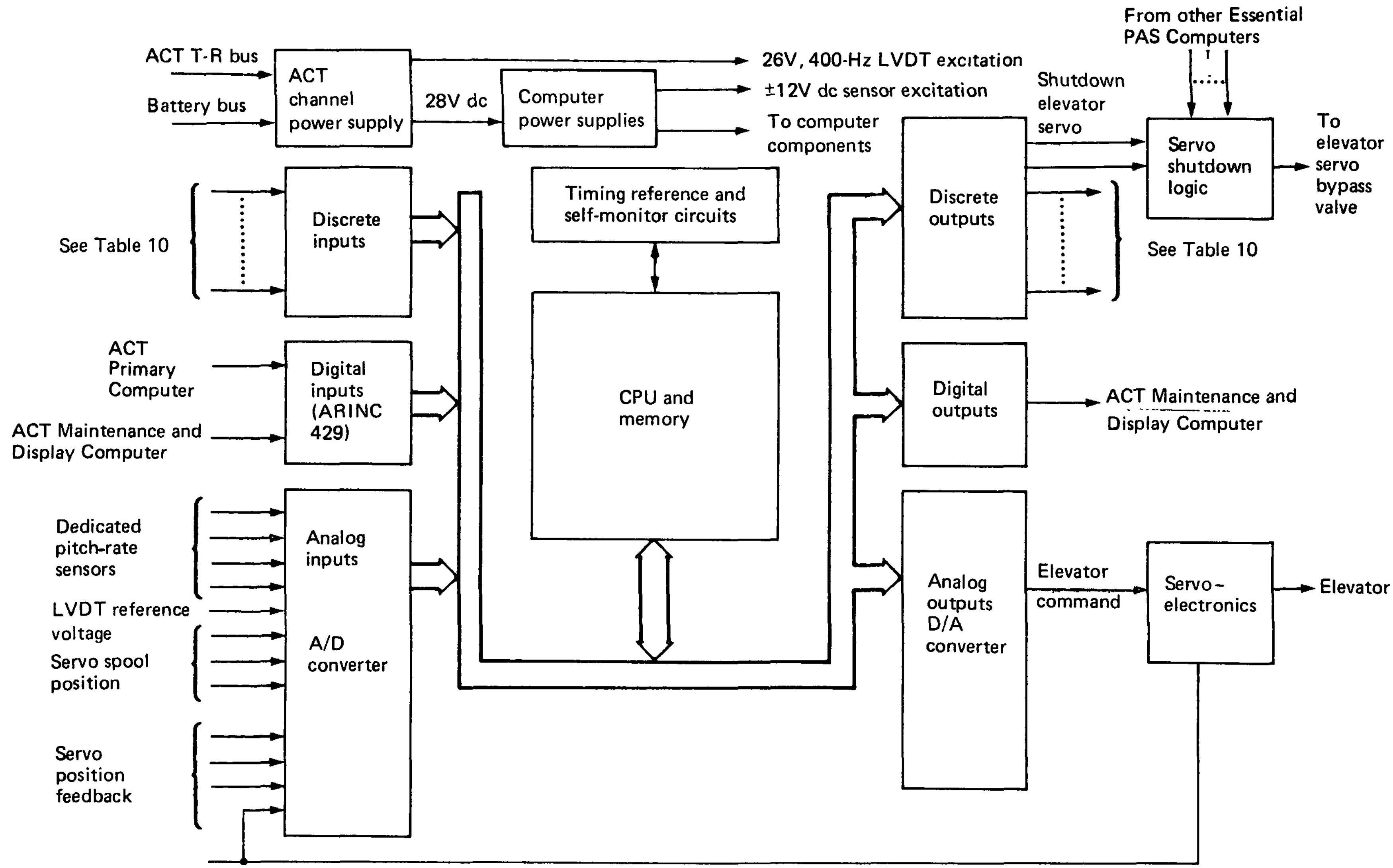


Figure 43. Essential Pitch-Augmented Stability Computer Block Diagram

Table 10. Essential Pitch-Augmented Stability Computer Inputs and Outputs

	Inputs	Outputs
Power	ACT channel 28V dc power	±12V dc sensor excitation
Digital (ARINC 429)	ACT Primary Computer ACT Maintenance and Display Computer Cross channel (3)	ACT Maintenance and Display Computer Cross channel
Analog	Pitch rate (4) Elevator servo spool valve position (3) Elevator servo position (4) LVDT reference	Elevator position command
Discrete	Air and ground logic (2) Test initiate (3) Electric power monitor (4) Hydraulic pressure monitor (3) PAS_{SHORT} fail (3) PAS_{SPEED} fail (3) MLC fail (3) Spare (11)	Warning displays (2) Self-test (6) Shutdown actuator—servo failure Shutdown actuator—computer failure (4) Spare (17)

8.1.2.3 ACT Maintenance and Display Computer

The ACT Maintenance and Display Computer collects data from the ACT Primary and Essential PAS Computers and analyzes it to provide fault identification for maintenance purposes and also to provide flight crew advisory and warning messages. The ACT Maintenance and Display Computer also provides access to the other system computers for maintenance testing. These tasks do not require high processing speeds and could be performed by an 8-bit microprocessor. Detailed requirements for this noncritical computer have not been determined, but 32K of program memory plus a 512-word nonvolatile variable memory for fault-table storage have been specified for cost and weight estimating purposes. This provides sufficient margin to incorporate any foreseen function into the ACT Maintenance and Display Computer.

8.1.2.4 General Characteristics

Table 11 summarizes general characteristics of the ACT Primary and Essential PAS Computers. These characteristics are discussed further in the following paragraphs.

Table 11. Selected System Computer Characteristics

Item \ Computer	ACT Primary	Essential PAS
Processing speed	400K ops	75K ops
Memory requirements	24K ROM 2K RAM	11K ROM 256 RAM
Inputs and outputs	3 ARINC 429 digital inputs 2 high-speed digital inputs 25 analog inputs 40 discrete inputs 2 ARINC 429 digital outputs 1 high-speed digital output 5 analog outputs 96 discrete outputs	5 ARINC 429 digital inputs 12 analog inputs 32 discrete inputs 2 ARINC 429 digital outputs 1 analog output 32 discrete outputs
Interrupts	Minimum 4-level priority	Minimum 4-level priority
Reference timing	5-ms minor frame 20-ms major frame	10-ms frame
Weight	11.3 kg (25 lb)	6 kg (13 lb)
Reliability	6800-hr MTBF	12 000-hr MTBF

Processing Speed and Memory Requirements—Estimates of processing speed and memory requirements are based largely on laboratory test development. These estimates are discussed in more detail in Subsection 8.3.3.

Input/Output—The I/O data in Table 11 summarize the detailed information in Tables 9 and 10. The I/O sections constitute much of the physical size and weight of the computers and have a major impact on cost.

Interrupts—Both the ACT Primary and the Essential PAS Computers are largely interrupt driven. The computers require a minimum of four priority interrupts to provide power, reference timer, computer fault, and externally generated I/O interrupts. All interrupts except the power interrupt are to be software maskable.

Reference Timing–Both computers perform real-time control operations using signals from a reference timer for scheduling. The ACT Primary Computers are multirate scheduled to provide real-time iteration rates with periods of 5, 10, and 20 ms. Figure 44 shows typical scheduling of control activities for the ACT Primary Computer. The Essential PAS Computers operate at a single rate with real-time operations scheduled at 10-ms intervals.

Physical Size and Weight–The ACT Primary and Essential PAS Computers have been sized to conform to the ARINC 600 standards for packaging of digital avionic equipment. While this is not a specific requirement for active controls computers, it provides a comparison with other Baseline equipment and offers a configuration that would be readily accepted for installation reasons.

Reliability–ACT system reliability is discussed in detail in Subsection 9.2.

8.1.3 ACTUATION SYSTEMS

8.1.3.1 Hydraulic Actuators

Two basic actuation configurations are being considered for the multiple-function ACT actuation system: the secondary actuation configuration and the fly-by-wire (FBW) power control unit configuration. In the secondary actuation configuration, the ACT control signals control secondary actuators to produce a mechanical signal that is series summed with the pilot's mechanical input. The combined mechanical input controls the servovalve of the power control unit, which, in turn, drives the control surface. In the FBW power control unit configuration, the ACT control signals are fed directly to the power control unit. For most of the control surfaces used to perform ACT functions as well as basic flight controls, the secondary actuation configuration is used. The FBW power control unit configuration is used to drive the inboard portion of the outboard aileron and the flaperons. Figures 11 and 12 show both configurations, and Table 12 shows which type is assigned to each surface. In either configuration, system-failure-induced transients should be held to a minimum and hardover surface failures must be prevented. Thus, certain redundancy concepts, such as the active, standby type that usually induces large failure transients and takes much time for switching, are not acceptable for ACT

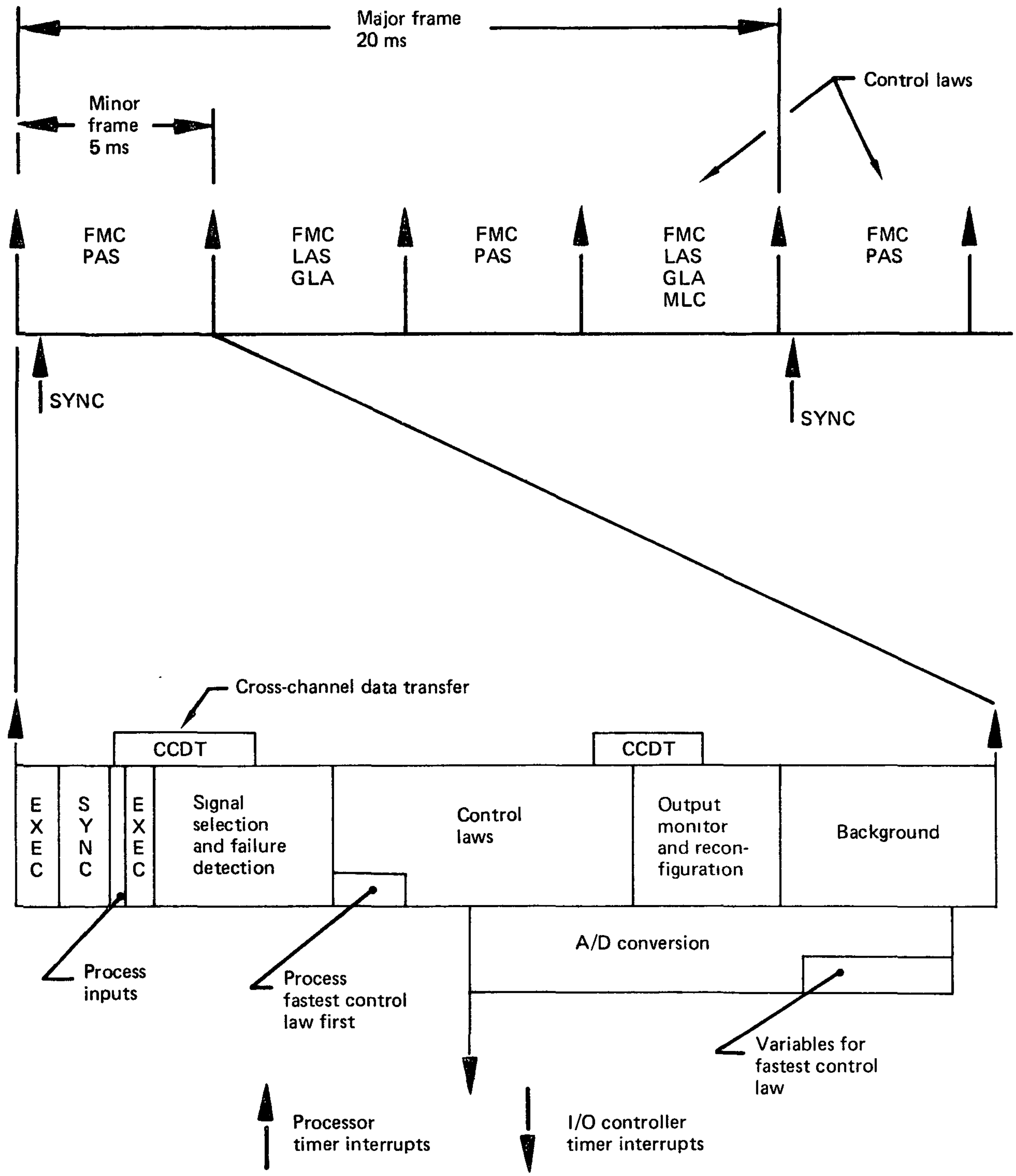


Figure 44. ACT Primary Computer Scheduling Diagram

Table 12. ACT Actuator Characteristics Summary

	Surface actuator [1]							Secondary actuator [1]					
	Type	Number per airplane	Maximum output, N·m	Average rate, deg/s	Maximum deflection, deg	Maximum no-load rate, deg/s	Open-loop gain, rad/s	Type	Number per airplane	Design rate, deg/s	Open loop gain, sec	Authority, deg	Configuration
Outboard aileron, outboard portion	[2]	4	1 500	85	+15 -30	120	20	[6]	4	[7]	80	+15 -30	Secondary actuator
Outboard aileron, inboard portion	[3]	2	1 200	100	+15 -30	130	40	No secondary actuator used					
Outboard flaperon	[4]	4	1 240	85	+15 -30	120	40	No secondary actuator used					
Inboard flaperon	[4]	4	3 400	85	+15 -30	120	40	No secondary actuator used					
Elevator	[5]	6	7 344	40	+20 -30	55	20	[6]	3	[7]	80	+12 -12	Secondary actuator
Rudder	[2]	4	20 902	40	±25	55	20	[6]	2	[7]	80	+4 -4	Secondary actuator

[1] Surface actuator controls surface, secondary actuator controls surface actuator
Hydraulic power requirements
- Proof pressure 37 233 kPa, high pressure 20 700 kPa, low pressure 350 to 690 kPa
- Extreme temperature· -54° to 125°C
- Operating temperature -40° to 71°C

[2] Side-by-side actuator, two for each surface, mechanical input/mechanical feedback

[3] Dual-tandem electrohydraulic actuator

[4] Two side-by-side electrohydraulic actuators

[5] Same as 2 except three for each surface

[6] Side-by-side force-summed secondary actuators—each actuator contains LVDT, bypass filter, and centering spring with maximum force of 230N Maximum output force is limited to 1780N

[7] Each secondary actuator has maximum rate of 127 mm/s and 38-mm stroke with linkage and mechanism. Stops make authority differences This rate exceeds maximum no-load rate of the surface actuator, except rudder rate is 35 deg/s

application. Several actuation redundancy schemes that satisfy the design criteria have been considered. Based on practical design considerations, two of the most promising concepts have been selected for the ACT actuation system.

Force-Summed Concept—As shown in Figure 45, the first concept is a force-summed, multiple-channel, detection-correction system. Each actuation channel contains a two-stage electrohydraulic servovalve that converts the input electric signal into hydraulic flow. The hydraulic flow displaces the actuator piston against the centering spring. A position transducer (LVDT) is used to close the position loop. A load limiter that limits the pressure difference across the actuator piston is used to limit the maximum output force to 1800N (400 lbf). This force is available to prevent minor jams. For normal operation, the force output required is on the order of 90N (20 lbf). For a three-actuator system, a pogo (force detent) is also provided to serve as an additional antijam device. The pogo load is set to exceed the maximum output force of one actuator but be below

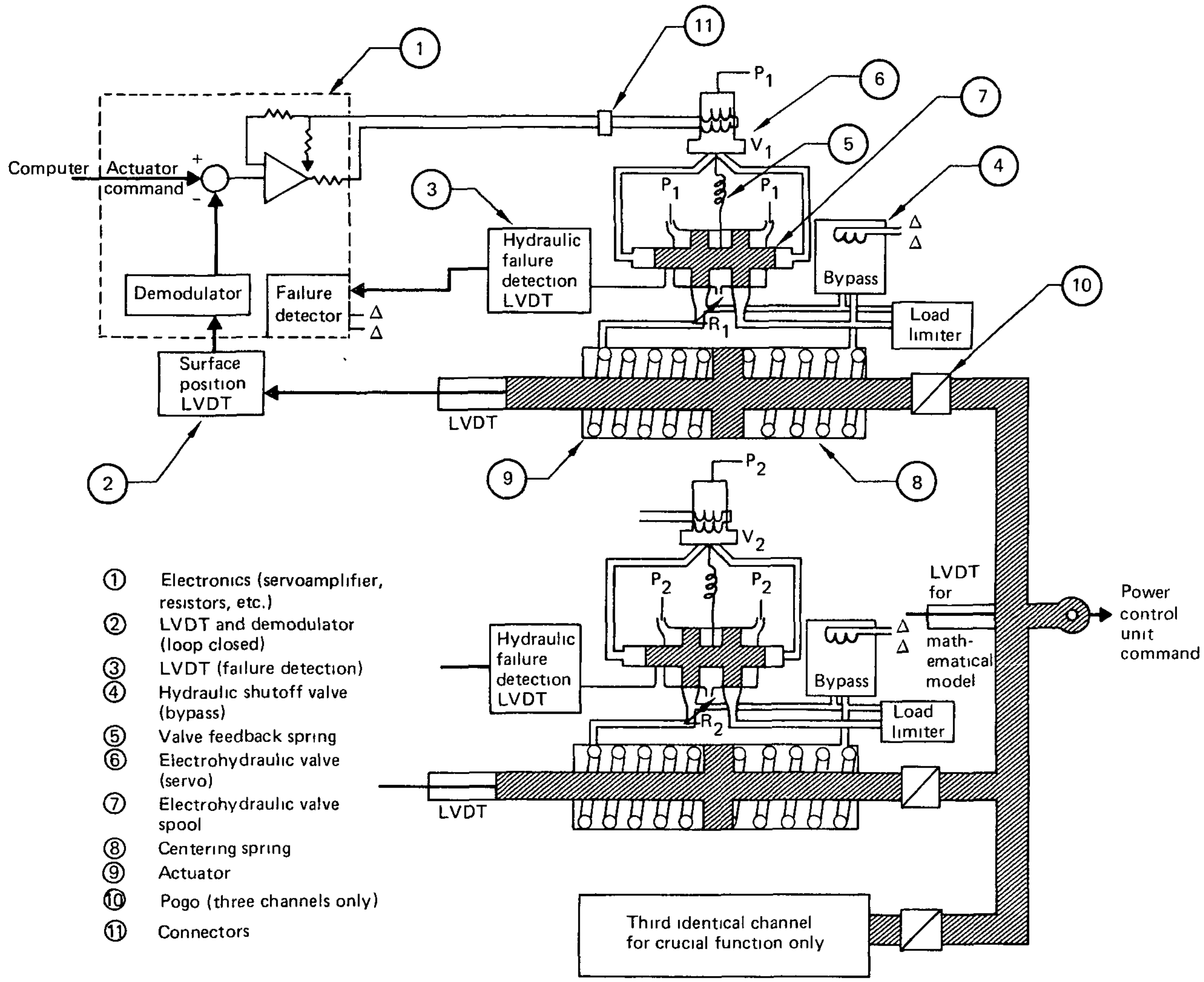


Figure 45. Force-Summed Actuation Concept

the combined maximum output force of two actuators. Thus, if one actuator completely jammed, the combined force of the other two actuators would collapse the pogo and the system would remain fail-operational. Hardware used in this application is a lightweight, off-the-shelf secondary actuator with performance proven in other Boeing programs. Two or three redundant actuators are used for each ACT function, depending on the redundancy requirements of the particular function. The two-actuator system with mathematical model provides a single fail-operational capability, and the three-actuator system with mathematical model provides fail-operational/fail-operational capability.

For failure monitoring and fault detection, an additional LVDT and electronics are used to provide a mathematical model of valve spool position. Each servovalve spool position is compared with the other and with the mathematical model to determine system integrity.

LVDTs are used to measure valve spool positions. The inner servo loop may be closed electrically if there is a design advantage. Normally a mechanical spring is used as feedback for the valve spool. The piston position transducers (LVDTs) are used to close the actuator position loop. Because the LVDTs are tied together, any component failure in one channel would result in a large valve spool position disagreement and would be easily detected. Arm and interlock logic are also provided in the computers to prevent faulty detection and correction during system startup.

The servovalve failure detection circuit can be straightforward. For a valve failure like a stuck or jammed valve, oscillatory failure, or a hardover valve, sufficiently large position errors between spools will be produced to allow easy detection. An elaborate spool dynamic model is not necessary to sense these kinds of failures. The mathematical model is expected to be a simple gain change or first-order lag. This is combined with comparators, thresholds, and time delays to denote failures. A low-pass filter may be needed to smooth spool position LVDTs to match the mathematical model. A simulation study of the force-summed actuator system with a mathematical model and detection circuits has been done to verify fully the feasibility of the system and to find the requirements of the mathematical model, the failure threshold level, filtering, and timing requirements of the failure logic. The study results are described in Subsection 9.1.2.

The force-summed secondary actuator traditionally suffers some inherent force-fighting problem among channels. This problem is caused by component tolerances in each channel, resulting in each channel seeking its own position in a high-gain servo loop. Force fighting increases system hysteresis and causes a "dead zone" when an even number of actuation channels are active. Force-fighting problems can be resolved by using a low-pressure-gain servovalve or by using channel equalization circuits. These methods would either reduce system performance or add additional hardware. Typical channel mismatch conditions that cause force fights are as follows:

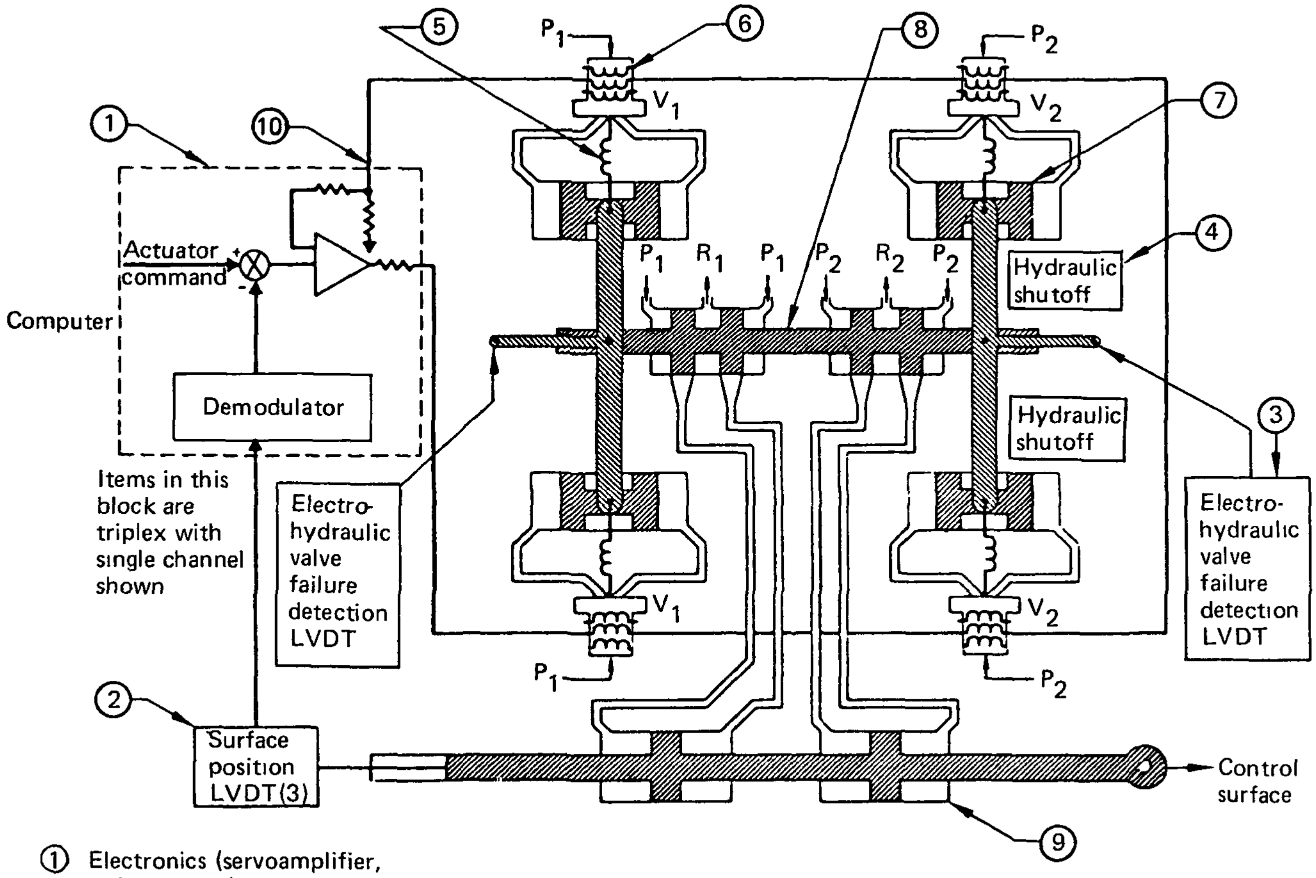
- Input mispatch 5%
- Position feedback 1%
- Servovalve null shift 1%
- Servoamplifier tolerance 1%
- Equalization loop mismatch 1%

Notice that the equalization loop itself causes 1% mismatch. The major portion is obviously caused by input mismatch (presumably in an analog circuit system). The ACT computers use signal selection and failure detection (SSFD) and frame synchronization, and they produce essentially identical command signals to the surface actuators. Input mismatch is not a problem, and a little force fight between channels will not significantly degrade system performance. By using high-quality, low-gain servovalves, servo-amplifiers, and LVDTs to minimize tolerance mismatches, it is believed that equalization loops between channels are not required. Assuming the force-fight problem can be reduced, then the associated problems such as increased hysteresis and dead zone become less significant and the force-summed secondary actuators will perform satisfactorily.

This forced-summed, multiple-channel, detection-correction system concept covered herein applies specifically to the ACT secondary actuators. Force-summed FBW power control units are covered in Subsection 8.1.3.3.

Force-Displacement Hybrid Concept—The second concept is shown in Figure 46. This concept was developed by the Parker-Hannifin Corporation of Irvine, California, intended for advanced FBW applications. A similar prototype unit has been laboratory tested by the Boeing Flight Control Research Group with acceptable results.

Four two-stage electrohydraulic servovalves with mechanical feedback spools are used. Two valves are grouped to each hydraulic system and position summed by a linkage. The two linkages are force summed and connected to the main control valve spool. Each first-stage valve positions the corresponding spool (fig. 46) proportionally to the sum of currents through the multiple coils. Because each valve receives the same combination of currents, the four spools should track each other within the accuracy of the mechanical servo loop. Any significant rotation of either summing linkage would indicate abnormal operation. An LVDT is used to measure this rotation, and the two valves associated with the rotating linkage will be shut off if the measurement exceeds a threshold. In addition, the actuators are monitored by computers using a cross-channel monitor scheme. The unit will provide fail-operational capability. The main feature of this unit is the electrical and mechanical channel independence; i.e., a failure of an electric component will not affect any mechanical component in the same channel or vice versa.



1. Electronics (servoamplifier, resistors, etc.)
2. LVDT and demodulator (loop closed)
3. LVDT (failure detection)
4. Hydraulic shutoff valve (bypass)
5. Valve feedback spring
6. Electrohydraulic valve (servo)
7. Electrohydraulic valve spool
8. Main valve spool
9. Actuator
10. Connectors

Figure 46. Force-Displacement Actuation Concept

The actuator will be used without gain compensation under failure conditions. Without gain compensation, an electric channel disconnection would reduce actuation loop gain by one-third. Acceptable dynamic performance is expected under failure conditions.

Both concepts (i.e., force summed and force-displacement hybrid) would be compatible with the ACT design philosophy, computer redundancy arrangement, and total system complexity. However, for the secondary actuation configuration to achieve the same design goal, the force-displacement concept requires more hardware. In the pitch axis, for example, to achieve a fail-operational/fail-operational capability requires eight electrohydraulic valves while the force-summed concept requires only three. The simple system should provide comparable system reliability with less cost and weight impact. The force-summed concept is selected for the secondary actuation configuration.

For the FBW power control unit configuration, the force-displacement concept was chosen to drive the inboard portion of the outboard aileron. The Boeing laboratory test of a prototype unit demonstrated that for the required loads, the rate, frequency response, hysteresis, and failure transients of this actuator concept are compatible with the ACT flutter-mode control (FMC) and gust-load alleviation (GLA) control laws for this control surface. (The somewhat simpler force-summed FBW power control unit concept was chosen to drive the less demanding flaperons.) Table 12 lists parameters of all control surface actuation systems in the technology base ACT airplanes.

8.1.3.2 AAL Pneumatic Actuator

The AAL system senses an impending stall condition and provides the pilot not only aural and tactile warning (stick shaker) but also applies forward torque to the pilot's and copilot's control column (stick pusher). This is accomplished by employing a dual tandem floating actuator to pull the control column forward when the actuator is pressurized. Figure 47 shows a block diagram of the system. Three electric channels and two pneumatic channels are used to ensure fail-operational capability against either inadvertent actuation or failure to actuate when needed. The actuator will provide a starting force of 356N (80 lbf) when pressurized by either one or both sides. As shown in Figure 47, the installation linkage is such that the force exerted on the control column is continuously reduced as it travels forward.

8.1.3.3 Flaperon Actuation System

The flaperon actuation system poses a difficult design problem. Although operation will likely be required only when the trailing-edge flaps are fully retracted, the flaperon

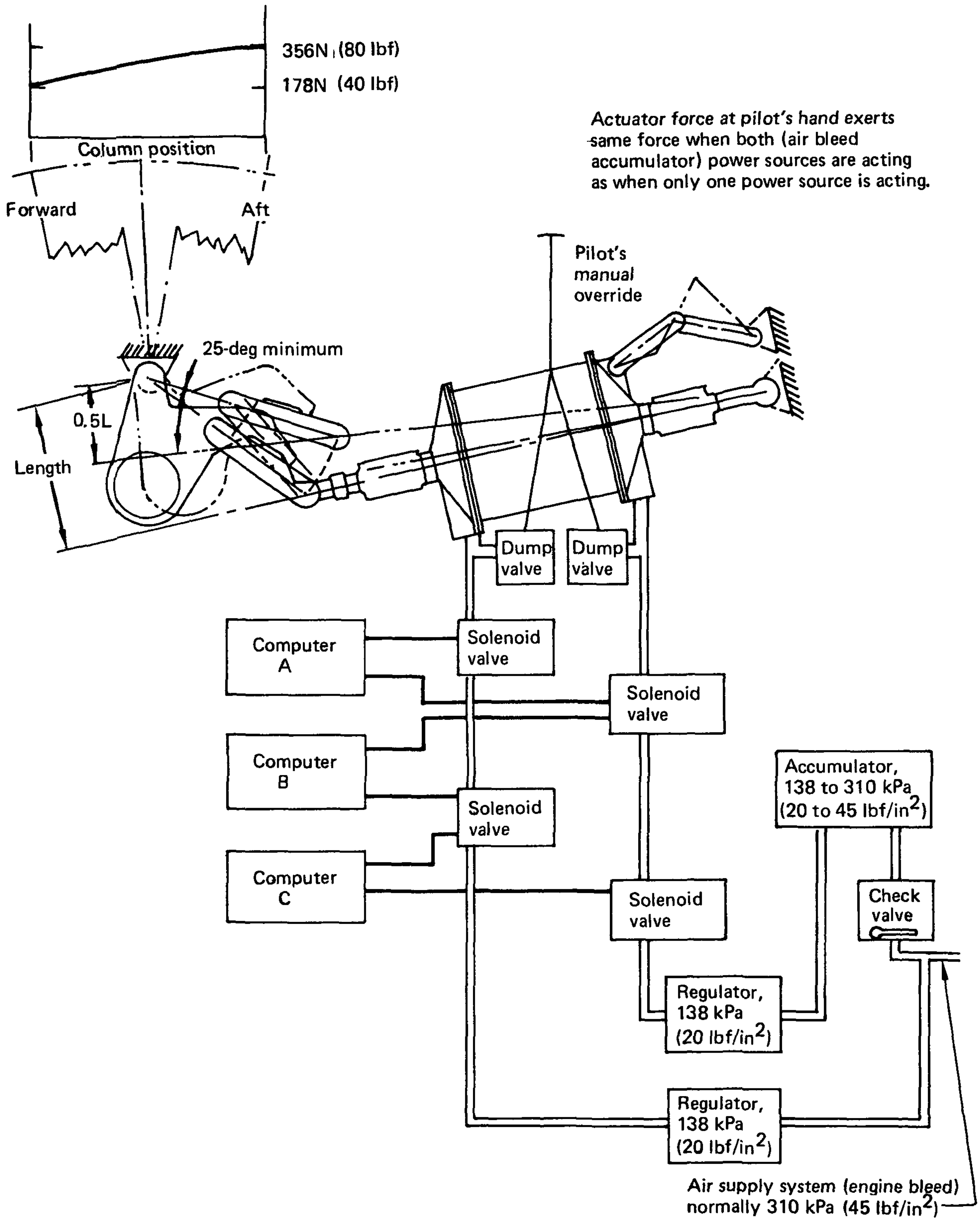


Figure 47. Stick Pusher Actuation Concept

actuation installation must accommodate the large flap motion during extension. At least two actuators and two hydraulic power systems are required for each flaperon to meet the redundancy requirements. Major structural damage could conceivably cause the loss of two hydraulic systems. Alternative designs are available by using power-by-wire (PBW) techniques. A PBW system uses electric wiring to transmit power in lieu of hydraulic lines. The advantage of a PBW system is that local damage or failure to the wiring and equipment will not cause the loss of the affected electric power system(s). Currently, there are two different PBW approaches under development. The first is called an electromechanical actuator (EMA), which uses electric power directly. The second is called integrated actuator package (IAP). Electric power is transmitted by wire and converted to hydraulic power at the actuators.

The following three viable flaperon actuation systems were studied:

- Hydromechanical actuation system
- Electromechanical actuation system
- Integrated actuator package

Following completion of the study, an assessment based on performance, weight, cost, and reliability was made on these systems. The results of the assessment indicated that the optimum system is the hydromechanical actuation system. The hydromechanical system is described in the following paragraphs. The other two configurations are covered in Volume II, Appendix D.

Hydromechanical System—The hydromechanical actuation system consists of two actuators and two flaperon lock systems powered by aircraft hydraulic power and electric power. The hydraulic power and ACT electric control signals are supplied to the flaperon as shown in Figure 48. Hydraulic power is transmitted to the actuators through hydraulic lines and swivel joints. These hydraulic lines and swivel joints are well shielded from the runway and tire debris by the flap support fairing. The swivel joints possess the same high degree of reliability as the swivel joints that provide flow to the Boeing spoiler actuators on the 727 and 747.

As shown in Figure 18, two actuators and two hydraulic power systems are required for each flaperon to meet its redundancy requirement. A major concern is that a flap loss

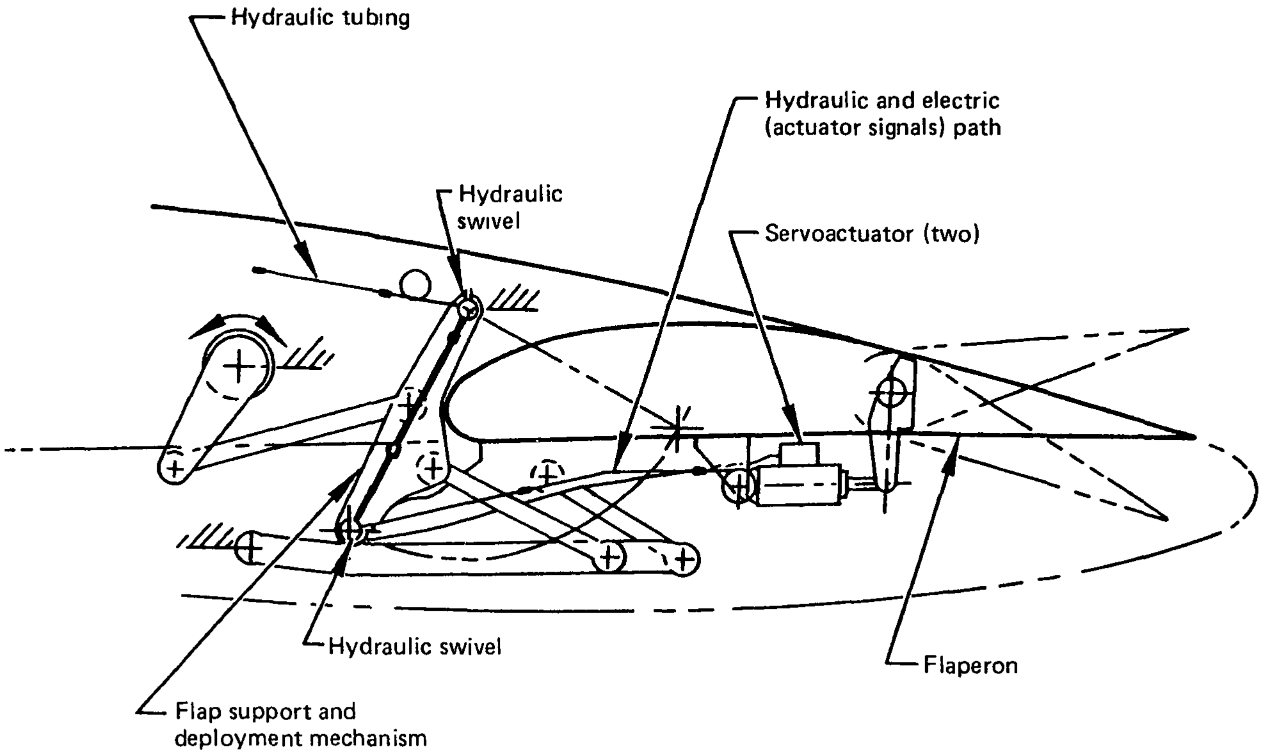


Figure 48. Flaperon Actuation (Hydraulic Power Through Swivel Joints)

would cause the simultaneous loss of two hydraulic systems. Because of this, the proposed design provides power capability from two hydraulic systems, but only one hydraulic power system is directly connected to the flaperon actuators. Hydraulic power to the actuators is normally supplied by hydraulic system A. Only one set of hydraulic lines is brought to the actuators through swivel joints. A motor-pump unit is used to connect hydraulic system B to hydraulic system A for power redundancy. In normal operation the motor-pump unit is stalled and is therefore inactive. Should hydraulic system A fail, the hydraulic motor in system B will automatically provide power to the pump in system A. The pump in system A will pressurize the hydraulic fluid in the local flaperon area with makeup fluid from the level-sensing reservoir. If a major fluid leakage occurs in the local area or if the flaperon is lost, hydraulic systems A and B will remain operational. System B will remain operational because it is not directly connected to the flaperon. System A will remain operational because the level-sensing reservoir and the normally closed shutoff valve will respond to block the path of the fluid flow to the flaperon.

Flaperon Servoactuator System—The actuators shown in Figure 18 are force-summed actuators. Each actuator possesses the full force and rate capability required to drive the flaperon. Figure 49 shows a schematic of the servoactuator. Hydraulic pressure to the servoactuator and flaperon lock system is controlled by a solenoid valve driving a bypass valve. The solenoid valve is a normally closed valve driven by dual coils. The bypass valve has the following modes of operation:

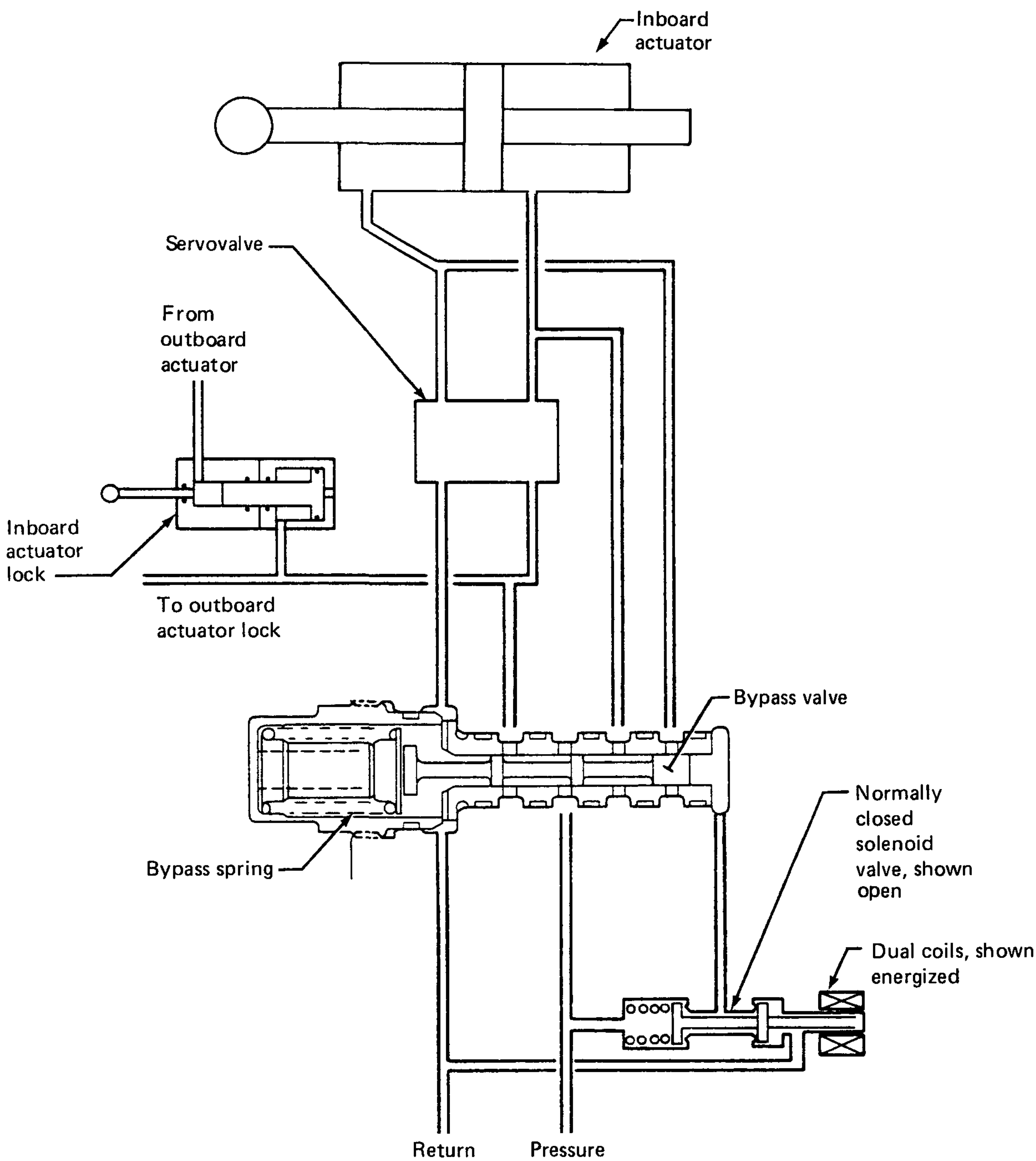
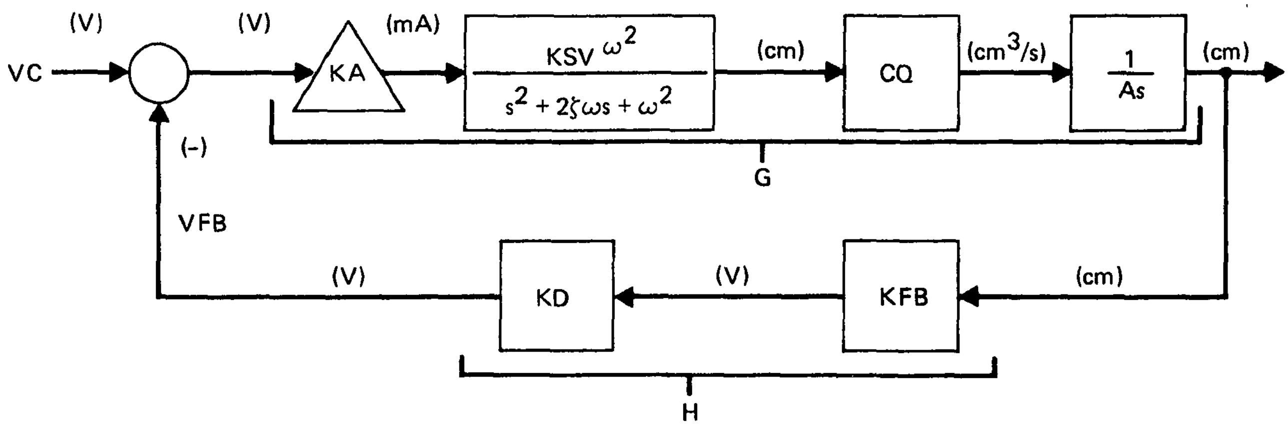


Figure 49. Flaperon Servoactuator System Schematic

- Normal Mode—When the solenoid valve is energized, it allows hydraulic pressure to actuate the bypass valve to a position that ports hydraulic fluid to the servoactuator and the flaperon lock system. In this mode the hydraulic fluid unlocks the flaperon, thus allowing the servoactuator to control the position of the flaperon. It should be noted that the failure of one solenoid coil will not affect operation of the solenoid valve.

- Failed Mode—When the solenoid valve is deenergized, it allows the bypass spring to drive the bypass valve to a position that blocks the path of the hydraulic fluid to the servoactuator, interconnects the actuator cylinder ports (thus eliminating its capability to induce force fights), and depressurizes the lock system. This action also can be triggered by a hydraulic system failure or failure of both solenoid coils.

The actuator is an equal-area actuator that is driven by a two-stage jet pipe servovalve. An LVDT is used for loop closure as shown in Figure 50.



Legend:

A	area of actuator piston
cm	centimeter, actuator position
CQ	servovalve spool gain
G	feed forward gain
H	feedback gain
KA	amplifier gain
KD	demodulator gain
KFB	feedback transducer gain
KSV	servovalve gain
mA	milliamperes
V	volts
VC	actuator position command voltage
VFB	volts, feedback
ζ	damping ratio
ω	frequency, rad/s
s	Laplace operator

Figure 50. Flaperon Control Loop Diagram

Flaperon Lock System–Surface locking is used in the event of total hydraulic power loss to the flaperon actuation system. Upon shutdown, it is also desirable that the flaperon be locked in the neutral position to prevent asymmetry and loss of lift on landing. The flaperon lock system is shown in Figure 51. The flaperon lock and unlocking system uses a spring force to lock the flaperon at the 0-deg position following failure of both servoactuator loops or both hydraulic systems. The flaperon is unlocked by a dual hydraulic piston. The area of the hydraulic piston is sized so that either hydraulic system can unlock and hold the flaperon in the unlocked position. With power loss to the flaperon actuation system, this locking system, with its 3560N (800-lbf) force spring in conjunction with the cam and lever-roller arrangement, can drive the flaperon to neutral (and lock) from the maximum free-float up angle against the maximum holddown hinge moment (two required per flaperon).

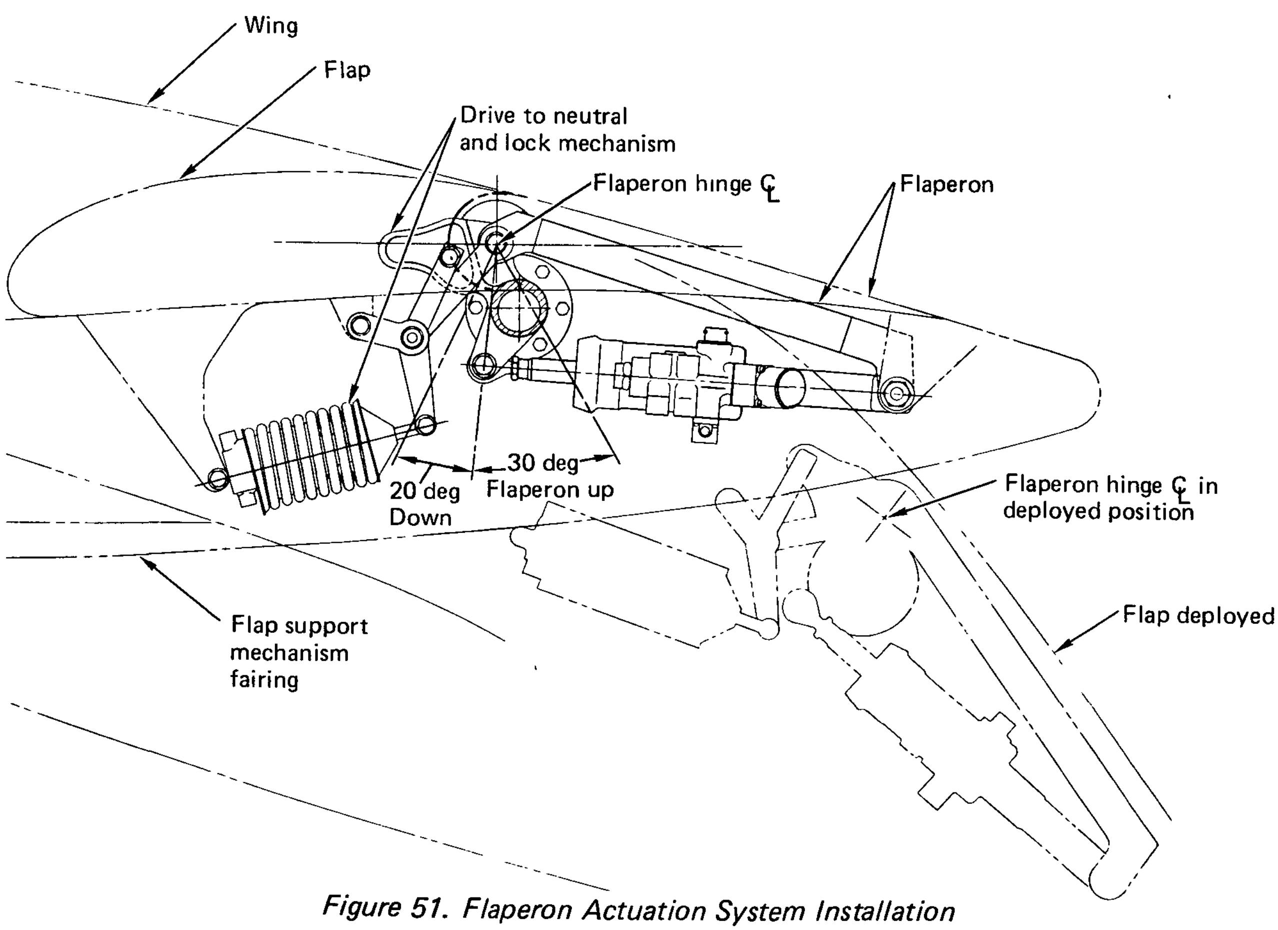
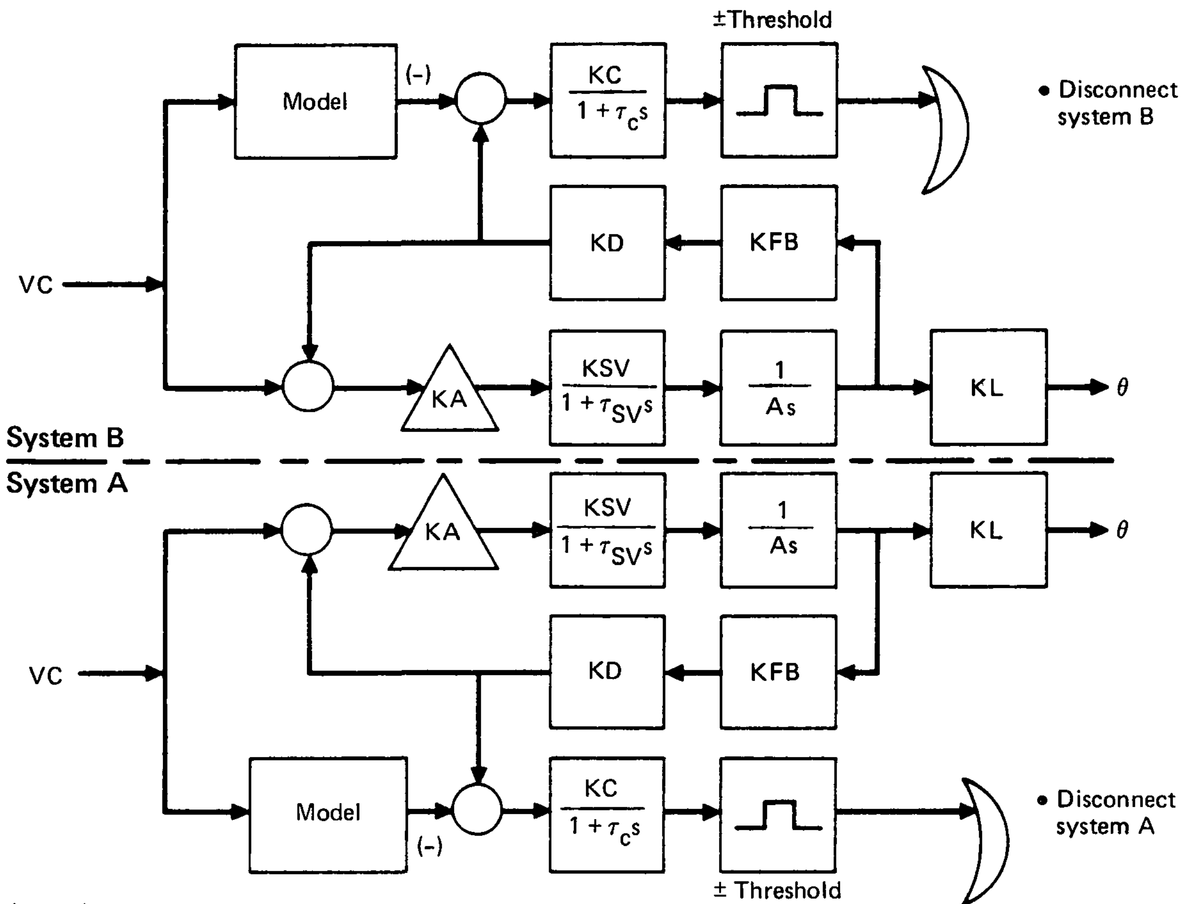


Figure 51. Flaperon Actuation System Installation

Failure Detection Scheme—The flaperon is a flight-critical surface driven by two force-shared actuators that are located 4.3m (14 ft) apart. The distance between the two actuators allows the structure to absorb the mismatches of the two actuators. If the actuators are driven in opposite directions at maximum force, the windup in the shaft would be 4.7 deg. It should be noted that the torsional stress induced by the windup will not cause the shaft material to exceed its torsional stress limit. The windup capability of the shaft allows selection of a failure detection scheme that monitors the complete servoactuator closed-loop performance.

The failure detection scheme is shown in Figure 52. A command signal from the controller drives the servoactuator control loop and a model of the servoactuator control loop. The position output of the servoactuator control loop is compared to the position output of the model. If one of the components in the control loop fails, then the outputs of the control loop and model will deviate. The actuator is considered failed when this differential error exceeds 6% of actuator full stroke. Once the differential error exceeds 6%, the computer will immediately deactivate the bypass valve on the failed actuator. The same magnitude of error or a failure of the second hydraulic system will cause the second actuator to switch to the bypass mode, thus allowing the spring lock system to lock the flaperon in its 0-deg position.

System Installation—The servoactuator and the flaperon lock system installed in the flaperon fairing are shown in Figure 51.



Legend:

A	area
KA	amplifier gain
KC	servovalve monitor gain
KD	demodulator gain
KFB	feedback transducer gain
KL	lever ratio
KSV	servovalve gain
s	Laplace operator
θ	surface angular position
VC	actuator position command voltage
τ_{SV}	time constant for servovalve
τ_c	time constant for monitor filter

Figure 52. Flaperon Failure Detection Schematic

8.2 REDUNDANCY MANAGEMENT

Redundancy management is the process that enables continued operation of the Active Controls Technology (ACT) system in the presence of transient or permanent failures of the control system hardware. Critical to this operation is the concept of coverage, which is defined here as the probability that the system correctly detects component failure(s) and successfully reconfigures the other components to maximize function success paths. This philosophy for the ACT system includes the following:

- System elements are monitored for faults using a combination of hardware and software.
- Faults are primarily detected by cross-channel comparison. Inline and inchannel monitoring supplements cross-channel comparison and provides fault isolation.
- Detected faults are isolated and switched out where possible to prevent any detrimental effect on airplane performance. Critical systems will survive at least single failure, except actuator jam, without affecting performance. Crucial systems will survive at least two similar failures, including a jam.
- Following clearance of a transient fault, the system will recover to the redundancy level in effect prior to the failure.
- Following a failure that leaves only two success paths for a function, a disagreement in those paths will lead to shutdown of the function or reconfiguration to a degraded mode. Single-thread operation will be allowed only if it can be shown that the function satisfies conditions stated in Subsection 5.2.2.

Redundancy management goals can be achieved by system monitoring and reconfiguration and additional failure protection through design features such as physical and functional isolation in redundant channels, supplying of electric power from redundant sources, and hydromechanical voting. System monitoring and reconfiguration are performed under the control of the ACT Primary Computers and Essential Pitch-Augmented Stability (PAS) Computers on three separate planes: sensors, computers, and servoactuators. Figures 53 and 54 illustrate the redundancy management processes performed by each computer. These processes are discussed in the following subsections.

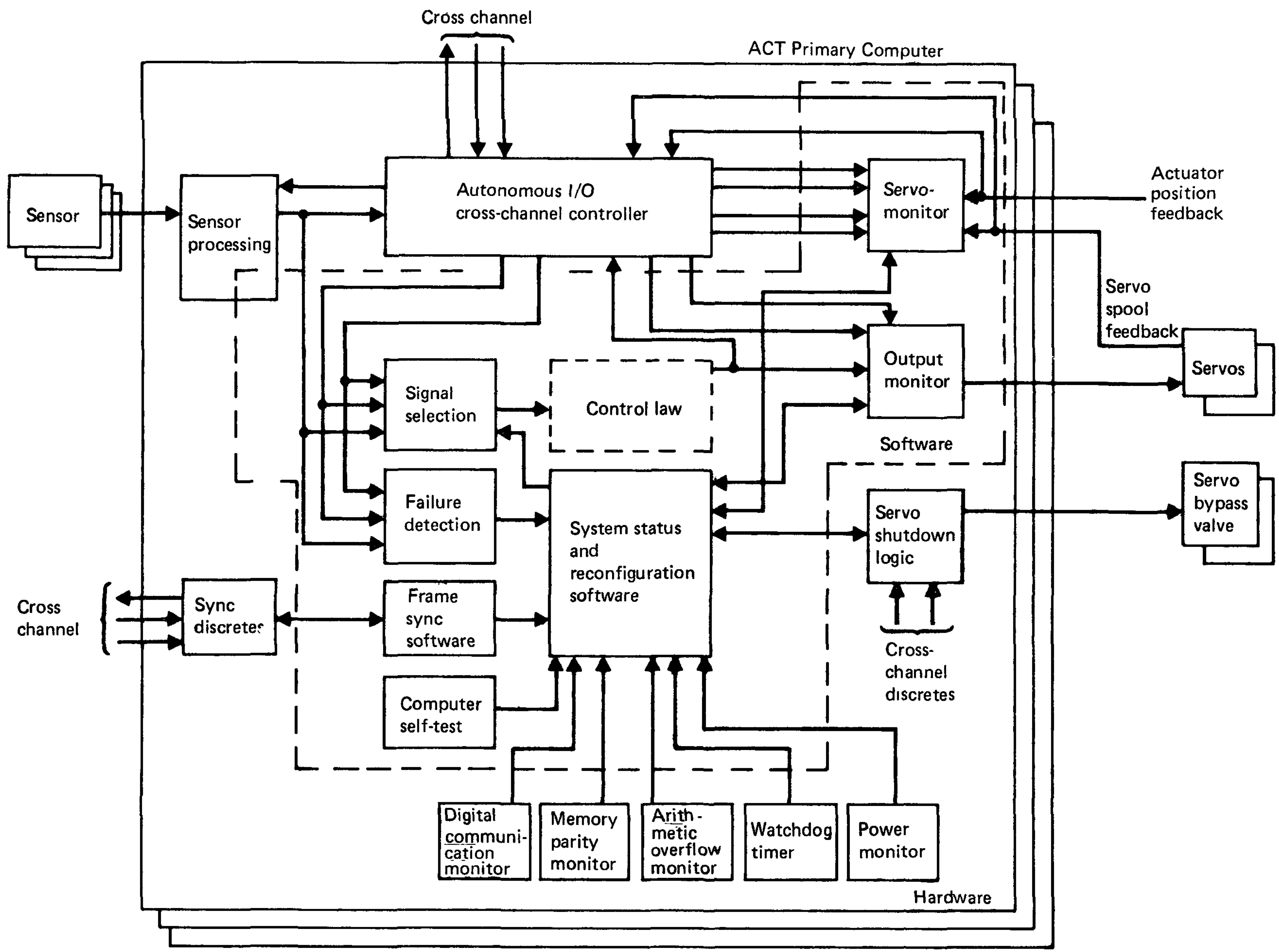


Figure 53. Redundancy Management Processes, ACT Primary Computer

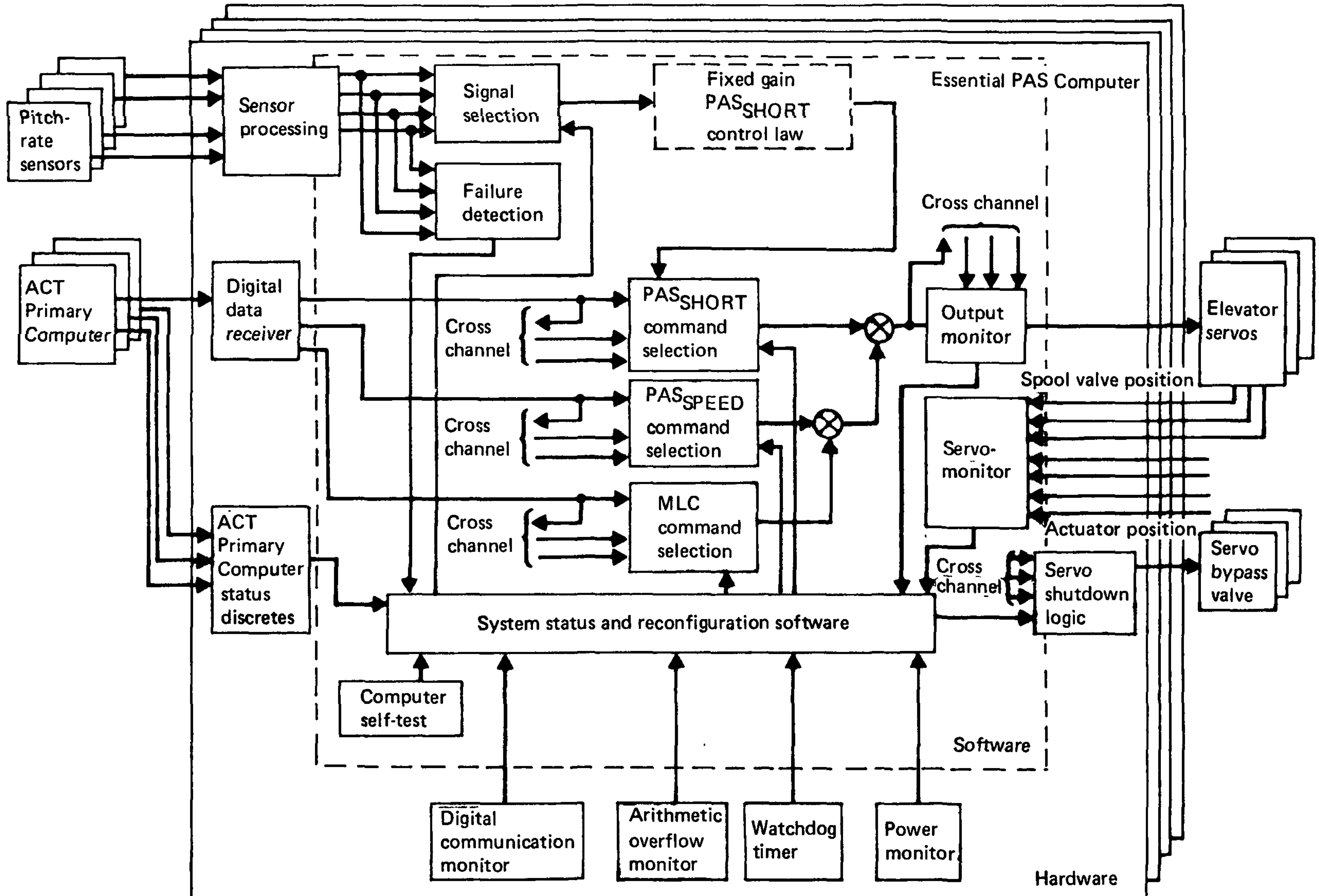


Figure 54. Redundancy Management Processes, Essential Pitch-Augmented Stability Computers

8.2.1 SIGNAL SELECTION AND FAILURE DETECTION

Protection against failures of the sensors, wiring, and input sections of the computer is provided by the signal selection and failure detection (SSFD) processes. Each computer uses signal selection to select a single value from each set of redundant sensors, and it uses failure detection and reconfiguration processes to isolate failed signals. Because sensor configuration differs between the ACT Primary Computers and Essential PAS Computers, they will be discussed separately. In both cases, SSFD is performed primarily by software.

8.2.1.1 ACT Primary Computer SSFD

Each ACT Primary Computer sees identical sensor information received from direct connection to the sensor or from the cross-channel data link. The signal selection takes two basic forms, depending on the type of sensor, as illustrated in Figure 55. Equalization is used on most signals. For those sensors that normally have dc components (such as airspeed, angle of attack, etc.), the last "good" value is held when two of three sensors fail. For those sensors with values that are normally zero (such as pitch rates), the selected value is set to zero when the sensors fail. Some sensors are filtered through a washout upstream of the signal selection (e.g., wing acceleration for gust-load alleviation and flutter-mode control), and the equalization is deleted. These signals are set to zero in the event of sensor failure.

Equalization is used to minimize transients due to sensor failure. Equalization takes the error signal, which is the difference between the equalized local signal and the selected signal, and passes it through a high-gain lag filter to form an equalization signal. The equalization signal is then subtracted from the local raw signal to form an equalized local signal. The signal selector acts on the equalized signals. This method does not provide exact equalization but can be used to reduce transients to levels that will not affect airplane performance.

Failure detection is done by comparing input signals. For signals that have been equalized, the error signal represents the difference between the equalized signal and the selected signal, which is used as a reference. Because the error signal represents differences between the equalized signal and the reference, steady-state errors are

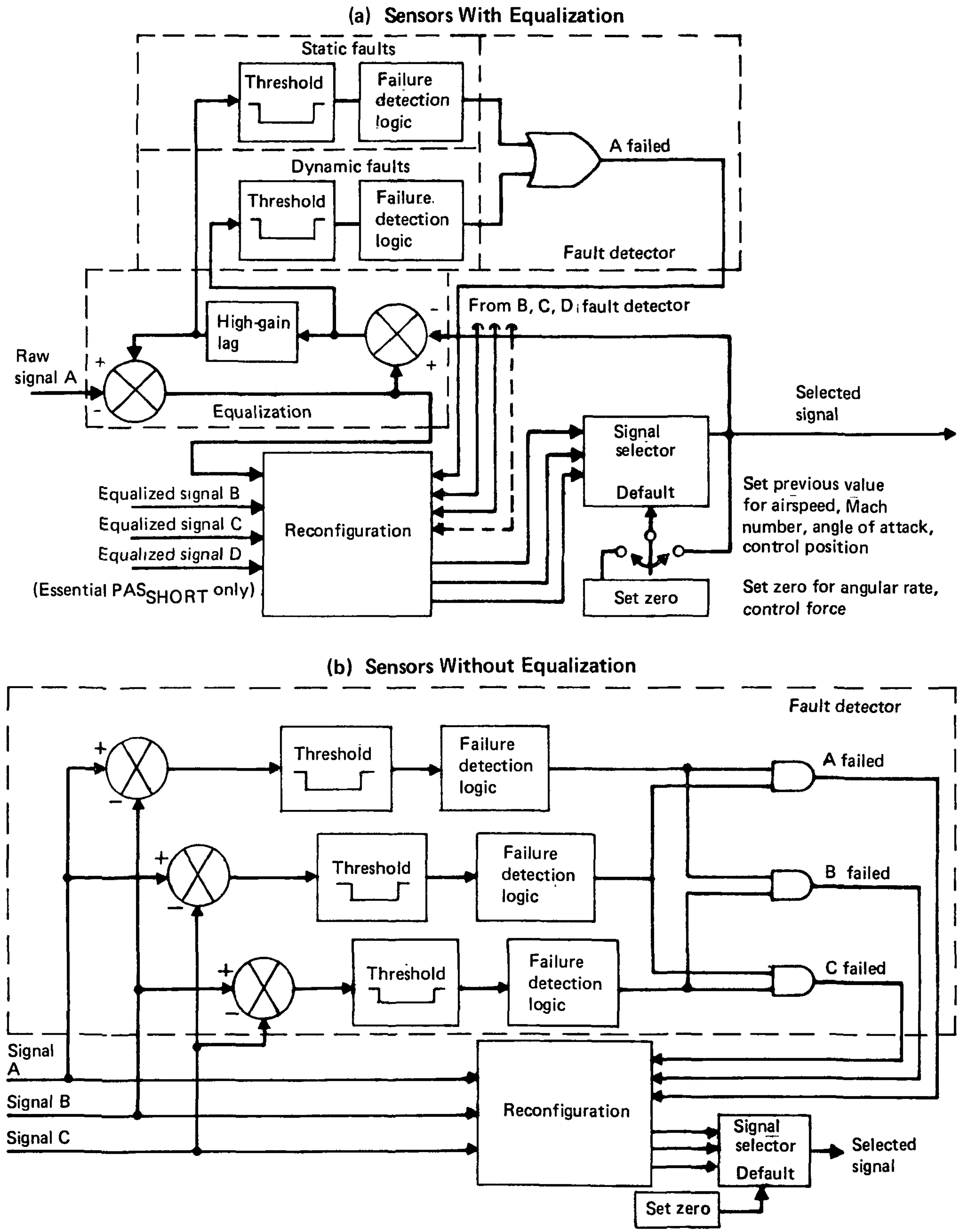


Figure 55. Signal Selection and Failure Detection for Sensors With and Without Equalization

masked and this signal may be used to detect dynamic faults. This gives rapid response to step failures and detection of high-frequency oscillatory failures. The equalization error represents the difference between the raw and the equalized signal, which in steady state is approximately the same as the difference between the raw signal and the selected signal, so the equalization error may be used to detect static failures. The two separate error detectors allow different thresholds and detect times for dynamic failures than for static errors such as drift and bias errors. For signals that are not equalized, there is only a single detector, which compares differences between raw signals. Because these signals are washed out upstream, this is essentially a dynamic type of fault detector.

Each detector uses a counter scheme to prevent nuisance trips and provide oscillatory failure protection. If the threshold is exceeded, the counter counts up. If the threshold is not exceeded and the counter is not zero, the counter counts down. If the counter exceeds a threshold level, the signal is considered failed. The counter is counted up at a faster rate than it is counted down so that an oscillatory failure will be detected. Figure 56 shows typical flow charts for the failure detection schemes.

Signal selection operates on those signals that have not exceeded thresholds. When a signal has been determined to be failed, sensors are reconfigured to exclude that signal from the selection process, but the fault detector continues to monitor all signals. The selection process selects the median signal, when none of the signals is considered failed, and averages the two unfailed signals when one of the signals fails. When the second signal fails, the selector uses a default value, which may be either the last good value or zero, depending on the type of signal.

A further reconfiguration occurs when two of the three sensors have failed. A default value is selected that may provide degraded performance operation. For example, a default value selected for a variable used to schedule gains results in a fixed gain being used in calculations. If operation with a default value does not provide adequate performance, two options are available. The control law may be reconfigured (i.e., replaced with another control law not requiring the failed signal), or if no other control laws are available, the function will be shut down.

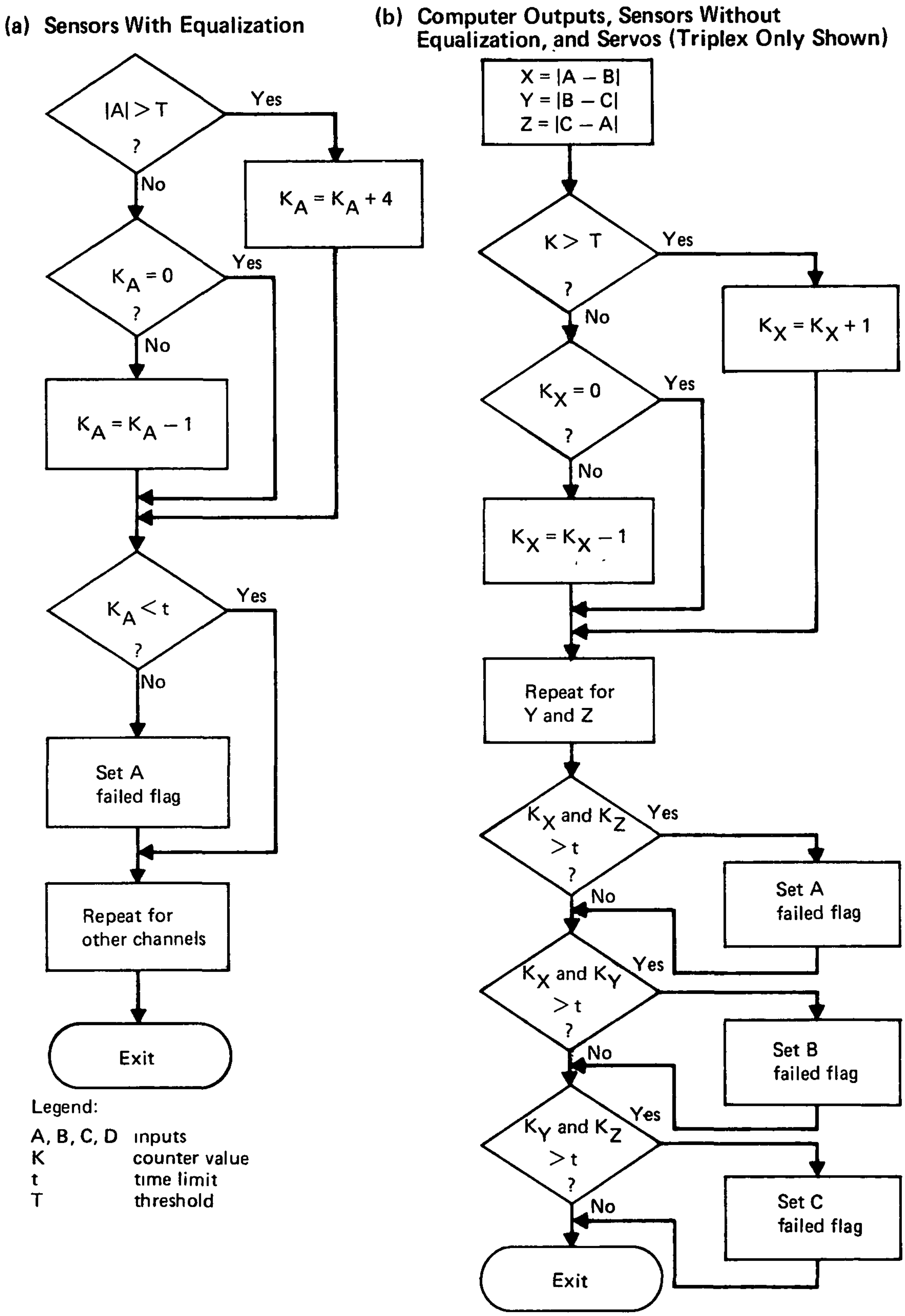


Figure 56. Failure Detection Logic

8.2.1.2 Essential PAS Computer SSFD

Sensor selection is slightly more complex for the Essential PAS Computers as there are now four rather than three computers. For the pitch-rate signal with four sensors, an active, online selection process is used. Three signals are considered active and feed directly into the selector. All four signals are monitored for failure, but the standby signal is not used unless one of the active signals fails. The selection process becomes (1) median of active channels with no failures, (2) median of active channels with standby channel replacing failed channel after first failure, (3) average of unfailed channels with two failures, (4) and a best-value selection with three or more failures. Best-value selection uses inline monitoring to select the remaining good signal following the third failure. Because inline monitoring does not provide 100% failure detection, it may not be possible to determine which sensor has failed. Loss of sensor data for the Essential PAS Computer can result in loss of pitch augmentation, which is unacceptable. Therefore, it is not reasonable to ignore all sensors. One of the signals is arbitrarily selected if inline monitoring does not provide enough information to determine which of the sensors has failed. However, reliability analyses indicate that loss of three signals in the Essential PAS System is extremely improbable.

The Essential PAS Computers also select the Full PAS signal from the ACT Primary Computers to drive the elevator. In this case there are only three signals; therefore, there is no standby channel. The SSFD algorithm is altered to account for simultaneous failures, as redundancy management of the ACT Primary Computers masks the effect of sensor failures on the output. A single sensor failure will not affect the output, but a second similar failure can cause all three computers to shut down their outputs simultaneously. The Essential PAS Computer cannot obtain sufficient information about the PAS command from the ACT Primary Computers by comparison monitoring alone to properly determine the status of that command. Discrete status bits from the ACT Primary Computers will be used to augment the monitor information. If a signal fails a comparison, or a vote of the status bits indicates a signal is failed, that signal is no longer used by the selector. If two or more signals fail, the PAS command from the ACT Primary Computers is disregarded and the output from the Essential PAS control law computation is substituted.

For both ACT Primary and Essential PAS Computers, the sensors are continuously monitored. If a failed signal returns to within tolerances for a prescribed period of time and inline testing indicates it is good, the sensor is considered recovered and once again becomes active.

8.2.1.3 Discrete Input SSFD

Signal selection for discrete inputs (fig. 57) is based upon majority logic voting. Monitoring is done by comparing input value to a selected value. If a disagreement occurs and persists for more than the specified time, the signal is considered failed. If a failure occurs while there are only two remaining signals, the output will revert to a safe state, if such a state exists. For example, failure of the air-to-ground logic results in an "in the air" state to be assumed, thus preventing entry to a self-test mode in flight. If a safe state does not exist, the discrete will default to local input state.

8.2.2 COMPUTER MONITORING

Protection against computer faults is provided by computer monitoring and reconfiguration. Failures are detected by both cross-channel comparison and inline monitoring techniques.

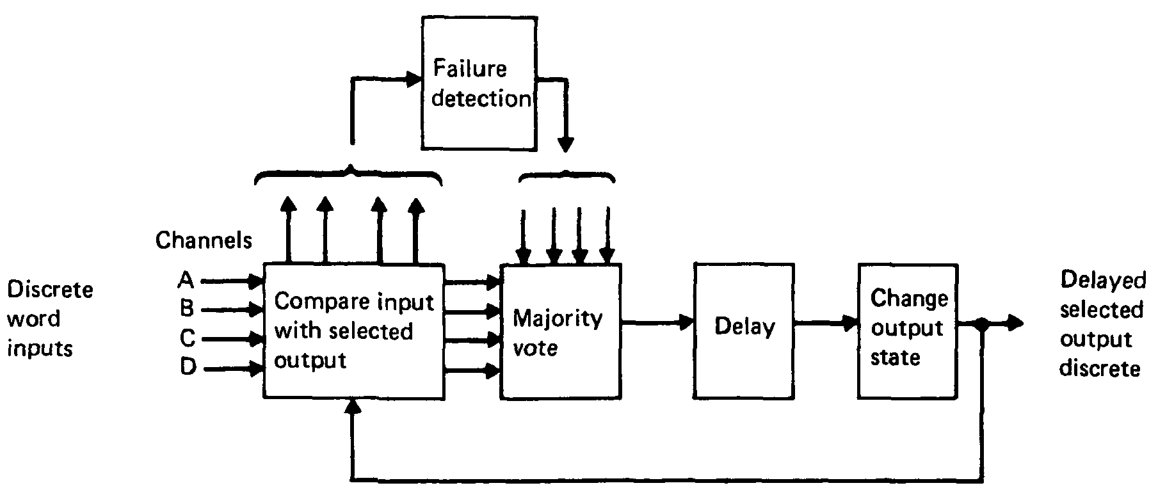


Figure 57. Signal Selection Concept for Discrete Inputs

8.2.2.1 Synchronization

The ACT Primary Computers are loosely synchronized on a frame basis, primarily to aid fault detection. This synchronization is not essential but would be necessary if integral feedback were incorporated. Synchronization aids fault detection and reconfiguration, but it is not a requirement for these processes. The system is designed to operate asynchronously as well as synchronously so that loss of synchronization does not shut down the system.

The ACT Primary Computers are synchronized through the exchange of sync bits. The synchronization concept is based on a "wait" algorithm. Each computer sets its sync bit and waits until the others set theirs or until a time limit is exceeded. The computer then clears its sync bit and checks to see that the other sync bits have also been cleared. If a computer fails to achieve sync (i.e., it does not set and clear its sync bit in the prescribed manner), that information is recorded for fault isolation use and possible maintenance action, but it is not considered as a system fault by itself.

Because the computers can operate asynchronously, a simple "slow-resync" algorithm may be used. If an out-of-sync condition exists, each computer varies its frame starting time slightly, so that the computers drift back into sync. Performance during this out-of-sync period is not significantly degraded.

The Essential PAS Computer implements a single fixed-gain control law. Synchronization is not required for this function, and the Essential PAS Computers are designed to operate asynchronously.

8.2.2.2 Output Monitor

The computer output monitor compares the outputs from the different computers. Comparison of the digital outputs is made before conversion to analog form. Failures of the digital-to-analog converters are not detected by the computer output monitor but are tested by the servomonitor. The computer output monitor compares output differences to thresholds and uses a counter algorithm similar to the sensor monitor (fig. 56) to determine failures. Thresholds for the primary computer can be low, as all computers operate on identical data and are normally synchronized. The Essential PAS Computers require higher thresholds to allow analog-to-digital tolerances and slight time differences due to asynchronous operation.

If a computer fault is indicated by the output monitor, the computers will send output discretes to the servo shutdown logic indicating that the computer output is failed and the affected servos will be shut down. If only two computers are considered good when the output monitor detects an error, both servos will be shut down.

The good computers will continue to monitor the faulted computer output. If the faulted computer retains its ability to execute its programs and the program memory has not been altered and input/output (I/O) capability is retained so that the computer can receive information essential to processing, it will attempt to recover. The recovery procedure used is a "warm restart." This is a simplified power-up routine that reinitializes control function variables and checks to see if the failed output commands return to within tolerances of the good computer outputs. The time constants of the control law filters for the ACT airplane are such that this will occur within a short time if the fault has cleared. If the output returns to within tolerance and stays there for a prescribed period of time, the computer will be considered recovered and its actuators put back on line. Servo output synchronization routines will be used as required to reduce transients at recovery.

8.2.2.3 Self-Monitors and Self-Tests

In addition to the cross-channel comparison monitors, failure protection is provided by computer self-monitors and self-tests. These functions are described in Subsections 8.1.2 and 8.3, respectively. Self-monitoring and self-testing provide extensive coverage of latent failures and aid in failure isolation for maintenance purposes. It is expected that self-test and self-monitoring can provide at least 95% coverage of computer failures.

If one of these monitors indicates a fault but the central processing unit (CPU) is still able to execute instructions and the program memory is intact, the computer will attempt to correct the error. For a watchdog monitor trip, the affected computer will attempt a "warm start" recovery as in the previous case, but the servo will not be shut down unless the output is out of tolerance for a period long enough to trip the output monitor.

Coverage by inline methods is not considered adequate to allow single-channel operation if a potentially catastrophic failure mode exists. Single-thread operation is permitted only for the lateral/directional-augmented stability (LAS) function, which meets requirements as described in the ground rules (subsec 5.2.2), and the crucial PAS function, as there is no reasonable alternative to single-thread operation if a failure occurs when system redundancy for this function has been reduced to two.

8.2.3 SERVOMONITORING

Faults in the servos, as well as in analog output and servoelectronic portions of the computers, are handled by the servomonitor. The servomonitor provides cross-channel comparison of servo outputs, and in some cases it is compared with an output predicted by a mathematical model. There are three different kinds of servos to be monitored: force-summed secondary servos, force-summed flaperon power control units (PCU), and force-displacement magnetic flux-summed servos. Each servo requires a slightly different monitor scheme, and Figure 58 illustrates the monitoring scheme for both kinds of force-summed actuators. The positions of the servo spool valves are compared for the secondary servos. The flaperon actuator forces are not summed by a rigid link but rather by the torque box of the flaperon surface, which allows sufficient windup to compare actuator ram position output and determine if a servoactuator has failed. Reliability of the servos is such that only two servos are required for critical functions and three for crucial functions when single-thread operation of actuators is allowed. Cross-channel comparison alone does not provide adequate fault isolation for single-thread operation, as a disagreement between two servos only indicates one has failed and not which one has failed. A mathematical model in the computer provides an additional channel for comparison. The servo spool valve position, or the actuator ram position, is predicted by the model, and this prediction is compared to the measured position. This mathematical prediction is used primarily for failure identification rather than failure detection, and a simple mathematical model that ignores high-order dynamics of the system may be used. Aerodynamic forces on the flaperon may also be ignored. Thresholds are set high enough to avoid nuisance trips.

A counting algorithm similar to the SSFD algorithm described in the preceding subsection can prevent nuisance trips caused by transients but retain detection of oscillatory failures. Once a failure is detected and determined not to be a short transient, the servo

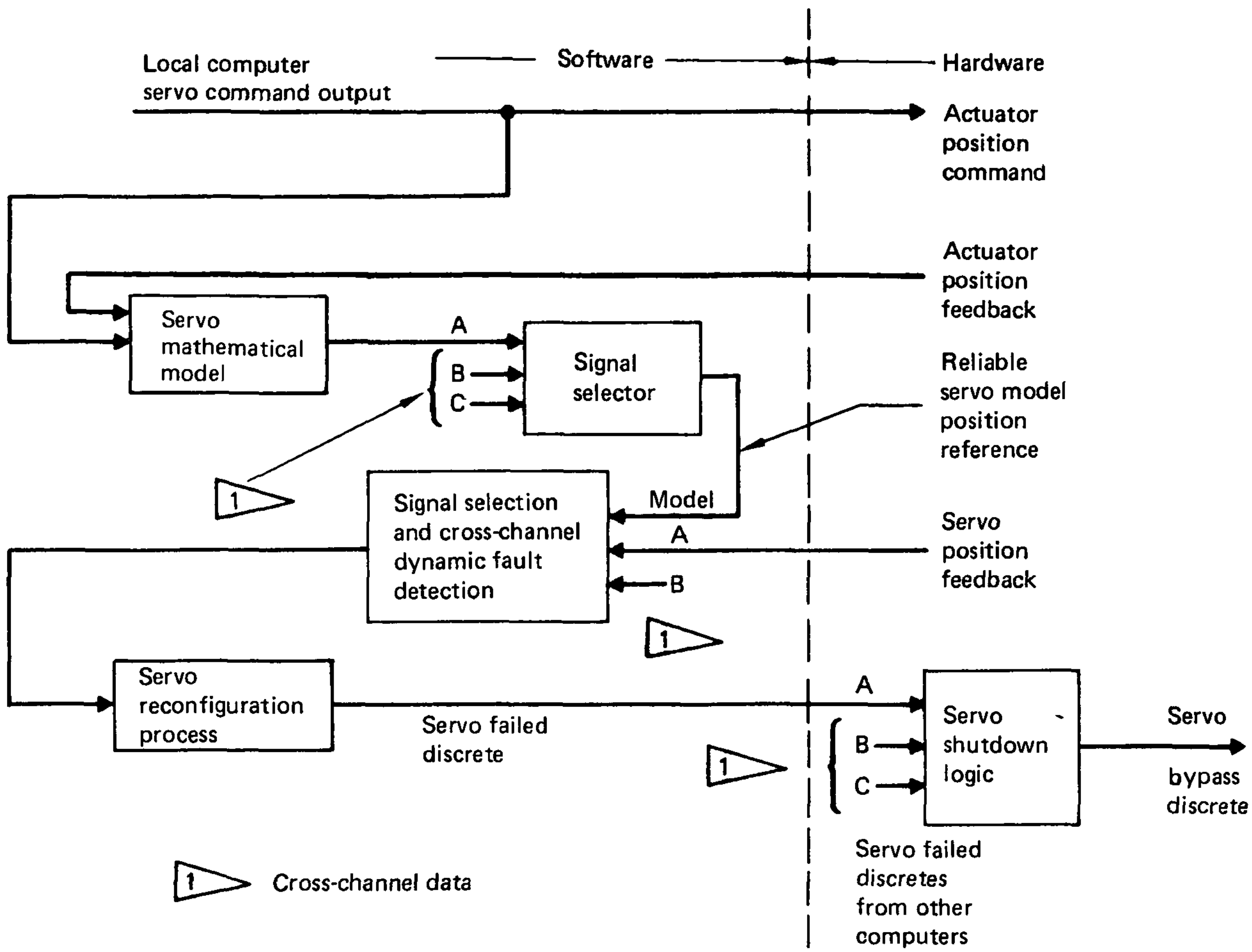


Figure 58. ACT Force-Summed Actuation Fault Detection

is shut down by sending a signal to a voter whose output activates a hydraulic bypass valve to shut down the servo. Once a servo has been shut down due to servo failure, it cannot be monitored to see if the failure clears, so the servo remains shut down for the duration of the flight.

The magnetic flux-summed servos of the force-displacement actuators use three coils in each electrohydraulic valve so each valve receives signals from all three ACT Primary Computers. This provides three channels for the electric portions of the actuation system even though only two hydraulic channels are used. The electric and hydraulic portions of the servo are monitored separately. Figure 59 illustrates monitoring for the electric channels. The force-displacement actuators have four electrohydraulic valves, each

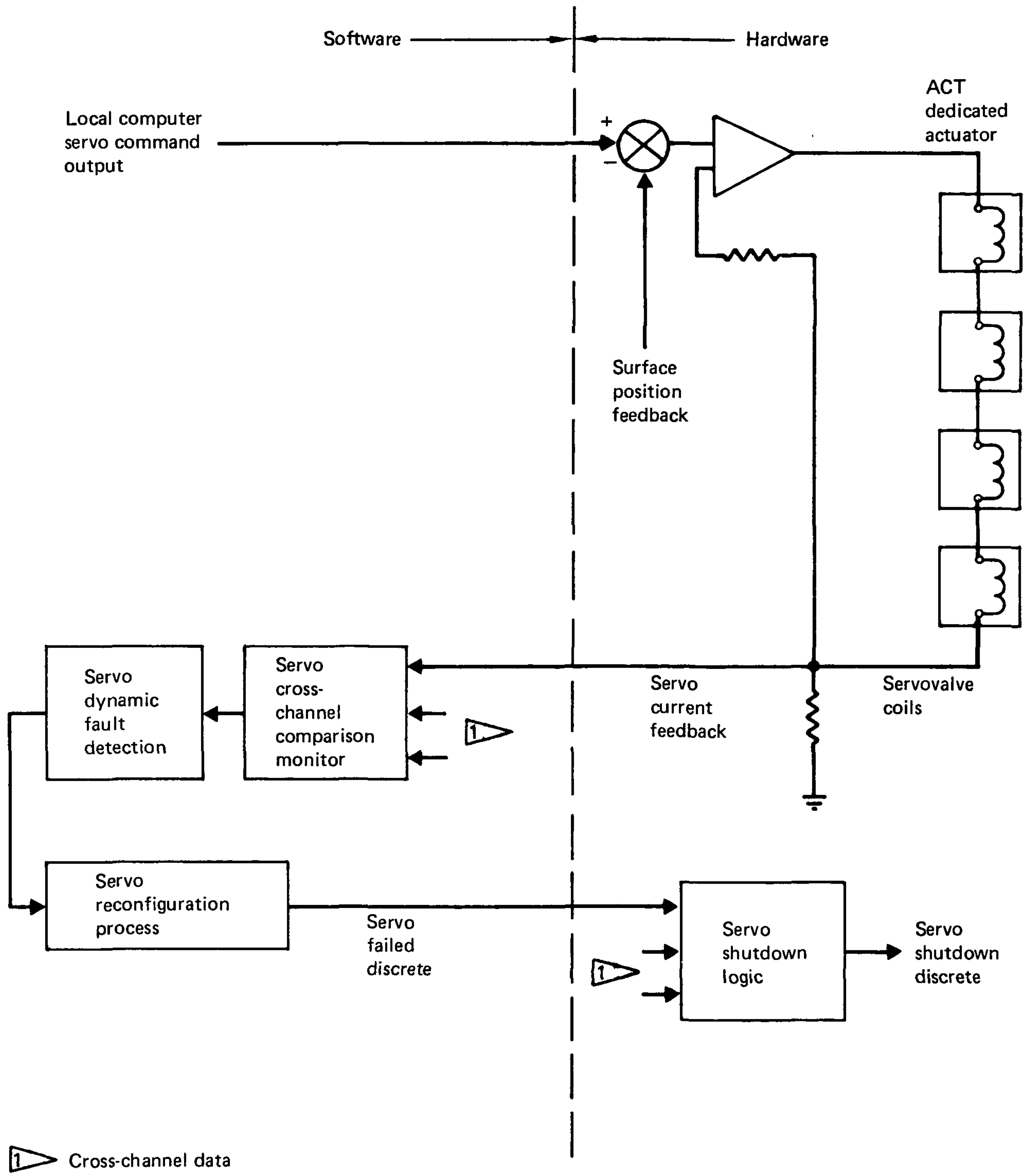


Figure 59. ACT Force-Displacement Actuation Fault Detection

containing three coils. The servodrive from each ACT Primary Computer is connected in series with one of the coils in each of the four valves. The currents flowing in each of the servodrive loops are compared to check the analog output and servoelectronic portions of the computer as well as the coils themselves. The differences between the currents in the three loops are compared to thresholds and the counting algorithm used to determine failures. If a failure is detected, the servodrive current is electrically nulled by a signal from the servo shutdown logic.

Hydraulic failures are monitored by using pressure switches in hydraulic pressure lines and by measuring rotation in the summing linkages for each pair of electrohydraulic valves, as described in Subsection 8.1.3. If a failure is detected, the failed valves are hydraulically bypassed.

8.2.4 ADDITIONAL FAILURE PROTECTION

Additional failure protection is provided for the actuation and for the electric systems. Figure 60 summarizes the failure protection features. Actuator outputs are voted through mechanical force summing or through a combination of magnetic flux summing and force summing. These techniques are described in Subsection 8.1.3.

Electric power for each ACT channel is derived from two independent sources: one of the aircraft main dc transformer-rectifier (T-R) buses and one of the standby batteries.

Power supply energy storage is sufficient to maintain voltage levels under maximum load until battery backup can take over if a T-R bus fails. The probability of temporary loss of power is therefore very low (see subsec 9.2.1), and no special procedures have been designated for reconfiguration when power is lost. This would be treated as a simultaneous failure of all parts of the affected channel. Reconfiguration and recovery would follow procedures described in the preceding subsections.

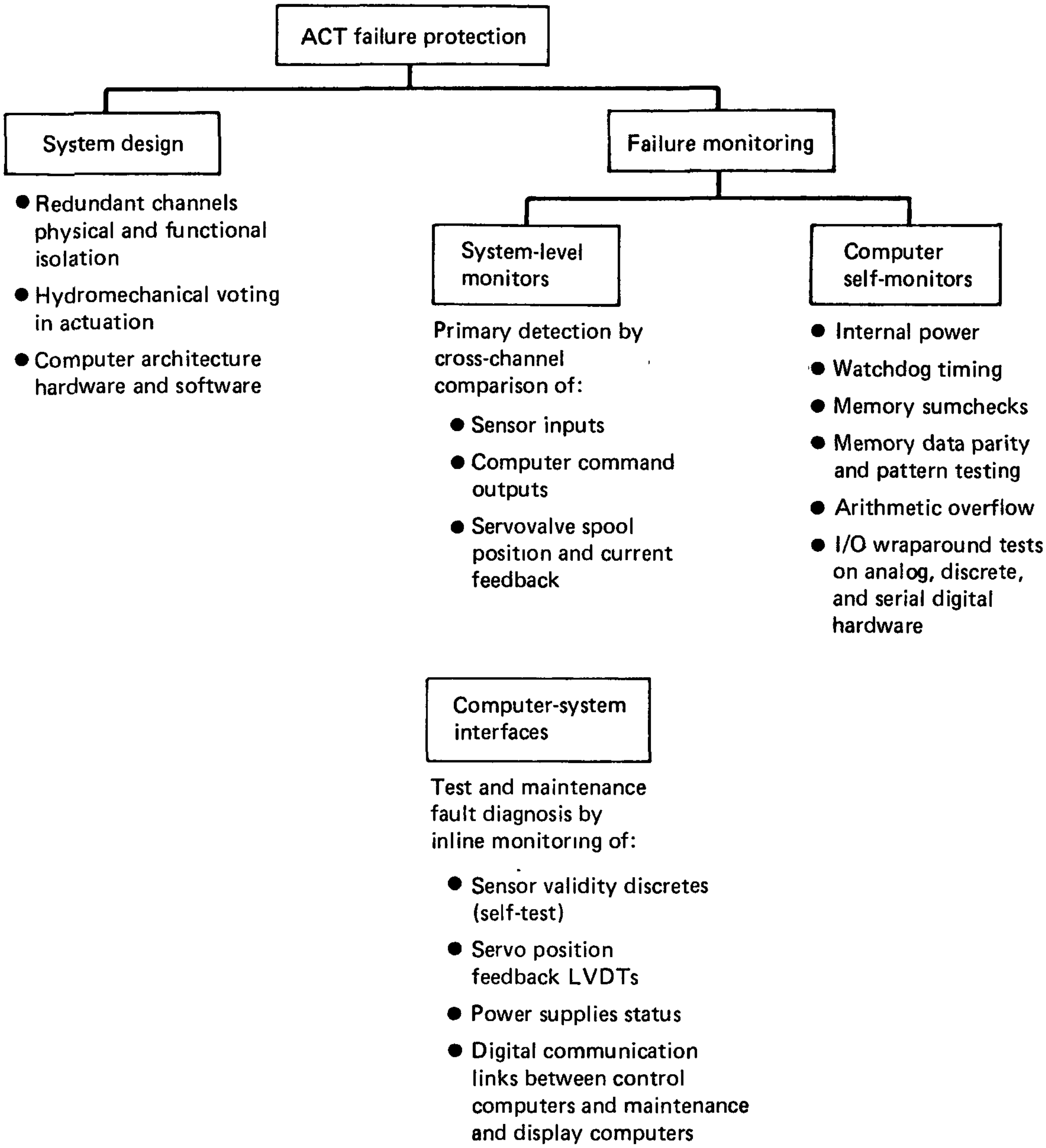


Figure 60. ACT Failure Protection Summary

8.3 ACT SYSTEM SOFTWARE

Software is an essential part of any digital system. Software for the Active Controls Technology (ACT) system has been designed with the objective of maintaining visibility and simplifying verification and validation.

8.3.1 SOFTWARE DESIGN GUIDELINES

The design methodology requires using the "top-down" technique with limited application of "bottom-up" techniques. Both top-down and bottom-up techniques are organizational processes or ordered approaches used to attack a task. The top-down technique starts at the "top" or highest level of consideration and uses partitioning designed to separate the lower level parts required. The bottom-up technique is the opposite of top-down, in that it first considers the parts available and then designs at the next higher level to integrate the parts into a unit.

For a design problem using the top-down approach, the highest starting level is the total conceptual statement of the problem and the requirements the solution must meet. Then, the next lower level of the solution and the interrelation of its parts are determined. This process is repeated for each part on a new level to design the next lower level until the lowest level parts of the solution are defined.

The bottom-up approach first takes the lowest level parts (e.g., a computer instruction set) and integrates them into a higher level unit. This process may be repeated at each successively higher level, with the lower level integrated units being further integrated into higher level units. Successive bottom-up integration is not allowed in the ACT design. Design of a bottom-level component is permitted when it is known that it will be required even though the top-down design has not proceeded down to the level where it would be specified. Control laws are examples of this type of component. There must still be a top-down design and specification of the manner in which these components are used.

Programs will be developed using the rules of structured programming, which define organization and enforce discipline in the programming process. Programming rules are made to ensure uniformity in style and to maximize clarity and visibility of how the program performs its intended task, regardless of how many programmers are involved. Modularity is a main concern; it includes the organization of sets of instructions and data. Standard control structures are established and program blocks are arranged sequentially so that the flow of a program has maximum visibility. A classic objective of structured programming is elimination of the GO-TO concept.

The entire computer resident software should be structured to maximize visibility, flexibility, and efficiency. Items to be considered in program design include partitioning, program module boundaries, hardware and software interfaces, program module control and data interfaces, functional and safety criticality, implementation constraints, and software testing.

8.3.2 SOFTWARE STRUCTURE

The objective of the overall ACT software task is to provide real-time implementation of active controls and adequate failure detection and self-test to ensure safe and reliable operation of the ACT system. Using a top-down design methodology, this overall task has been divided functionally into progressively lower levels. The resulting functional design trees are described in the following subsections.

8.3.2.1 ACT Primary Computer Software

Figure 61 shows the upper levels of the design tree for the ACT Primary Computers. The overall task of active control is divided into two parts: real-time control, which performs all tasks required for safe flight of the aircraft, and ground operations, which performs preflight and maintenance testing while the aircraft is on the ground. The real-time control software module is further divided into three tasks: real-time executive, foreground tasks, and background tasks.

Real-Time Control–The real-time executive (fig. 62) provides interrupt processing capability necessary for operating the control software in real time. There are four kinds of interrupts: power interrupt, timer interrupt, input/output (I/O) interrupt, and fault interrupt. Power interrupt software supplies system initialization following application of power to the system and performs required tasks when loss of power is imminent.

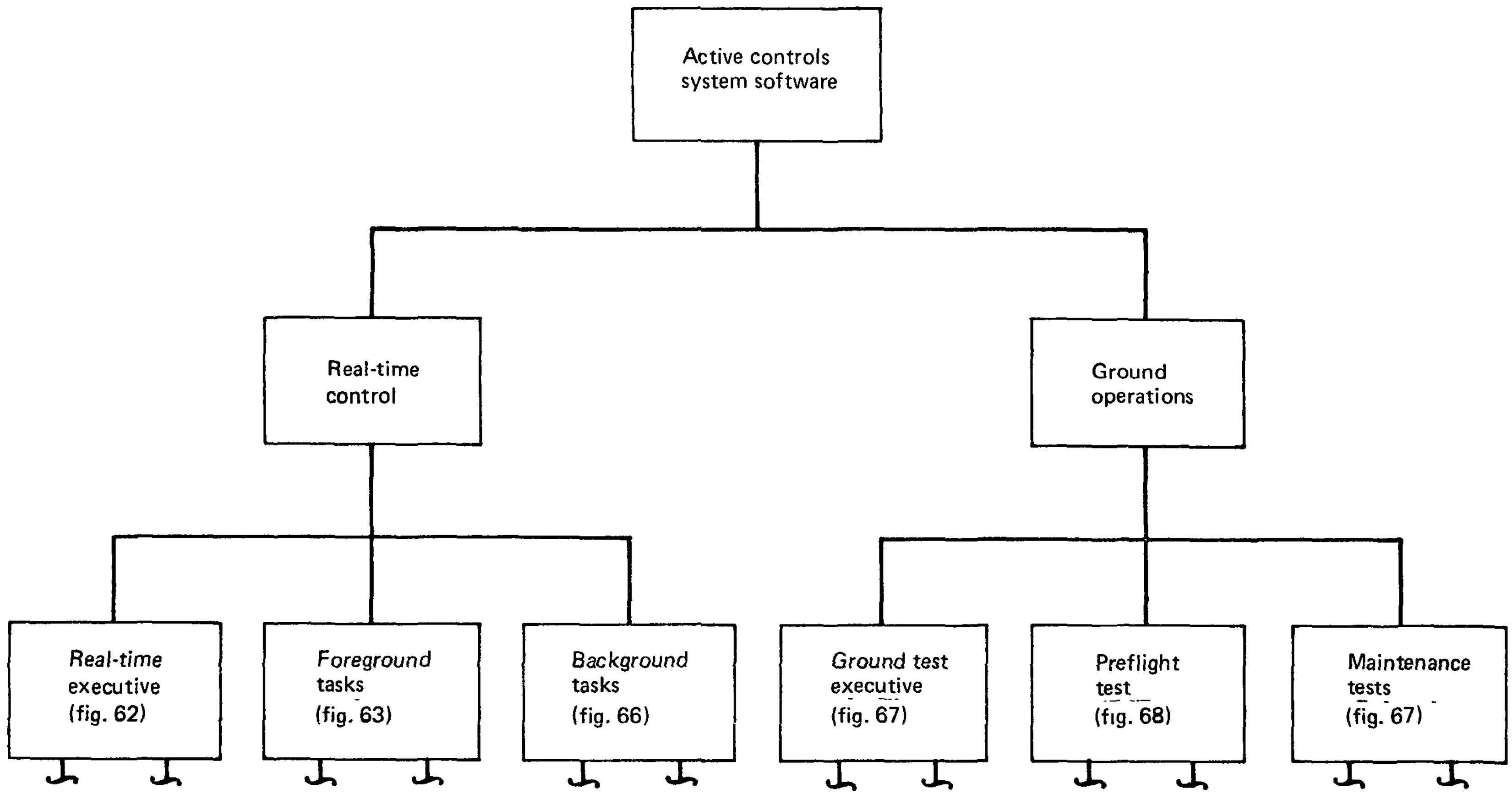


Figure 61. ACT Primary Computer Software Design Tree, Upper Levels

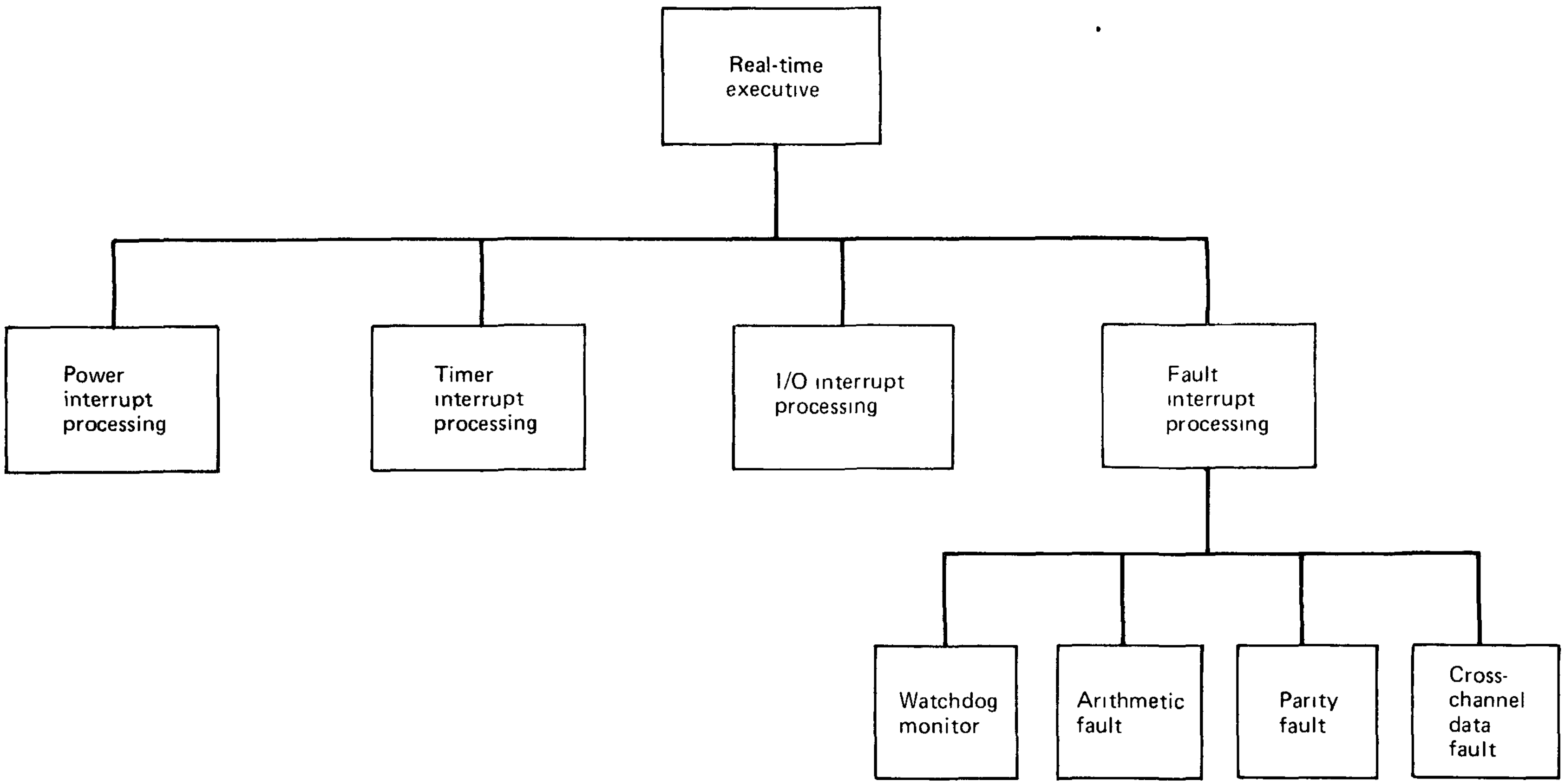


Figure 62. ACT Primary Computer Software, Real-Time Executive

An event timer provides real-time information to the system. The timer interrupt software begins processing for a new minor frame following a signal from the timer.

I/O interrupt processing software facilitates communication with external devices, including communication with other computers via the cross-channel data link.

Fault interrupt software furnishes software processing for four different kinds of faults. Cross-channel data link (CCDL) fault software indicates an error in data from another computer and makes that information available to the reconfiguration software. Arithmetic fault software provides overflow protection. Parity fault software provides information about memory failures to the reconfiguration software. Watchdog monitor software attempts recovery following a computer failure that prevents reset of the watchdog monitor timer.

Foreground tasks are those control tasks that must be performed in real time. These include tasks relating to redundancy management, control law calculation, and, because different control laws run at different sampling rates, a multirate scheduler. The breakdown for these tasks is shown in Figure 63.

The synchronization software minimizes the time difference between the start of a major frame on different computers. Synchronization ensures that sensor data are sampled at nearly the same time by all computers. Normal synchronization causes all computers to start a major frame at the same time when the computers are operating within tolerance. Resynchronization software acts to synchronize operation of the computers when one or more is operating at a different point in a frame than the others. Cross-channel status determines which computers are operating and in which minor frame they are operating. This information is used to determine if resynchronization is necessary.

The multirate scheduler schedules control law software. Each control law calculation is done only as often as required to get the proper sampling rate. Control laws and the associated redundancy management are scheduled at different rates by the scheduler.

The signal selection and failure detection (SSFD) software (fig. 64) provides sensor failure detection, signal selection, and transient suppression. Equalization is used on some sensor signals to reduce transients by forcing all inputs to be close to the selected input value.

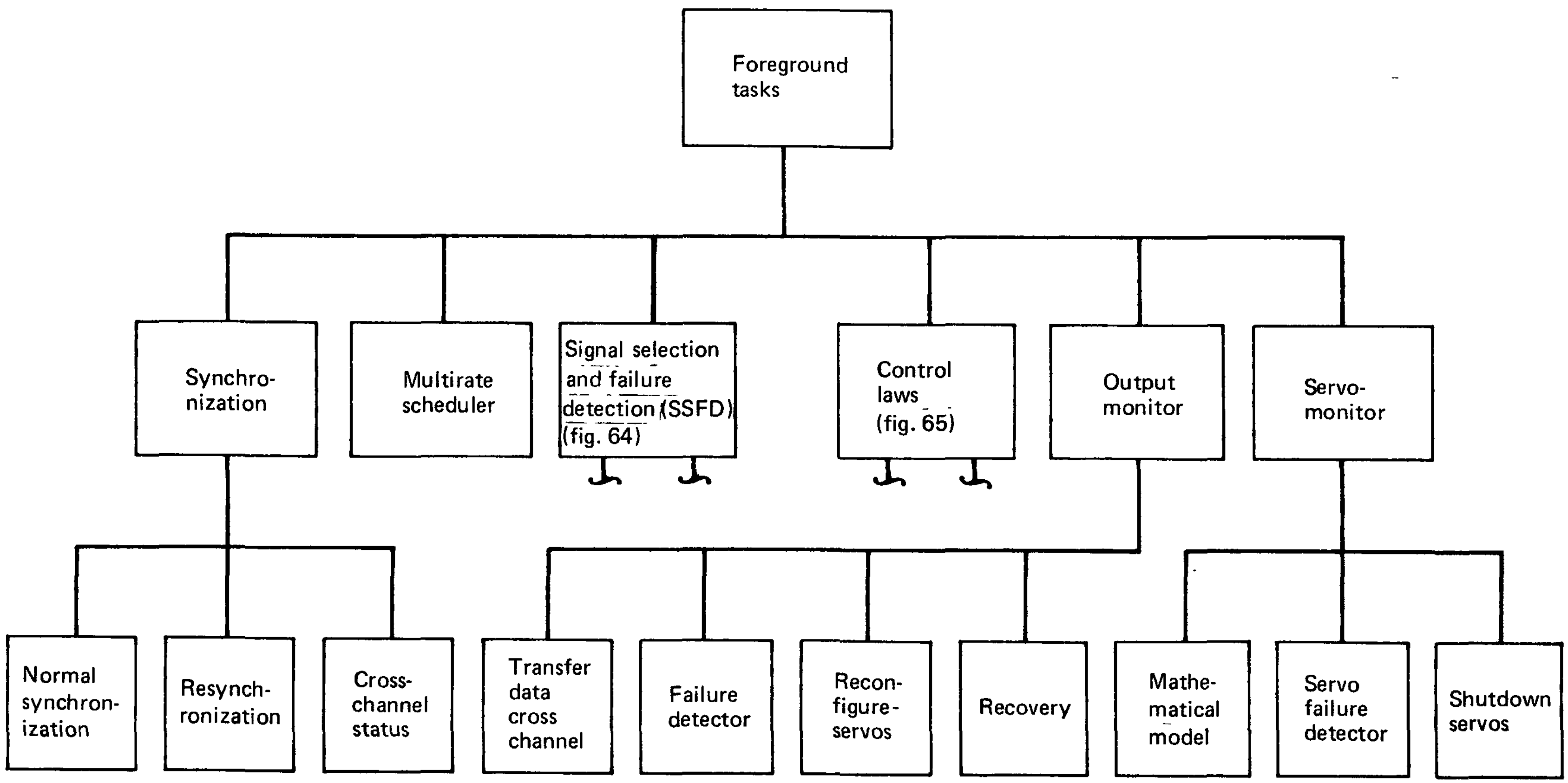


Figure 63. ACT Primary Computer Software, Foreground Tasks

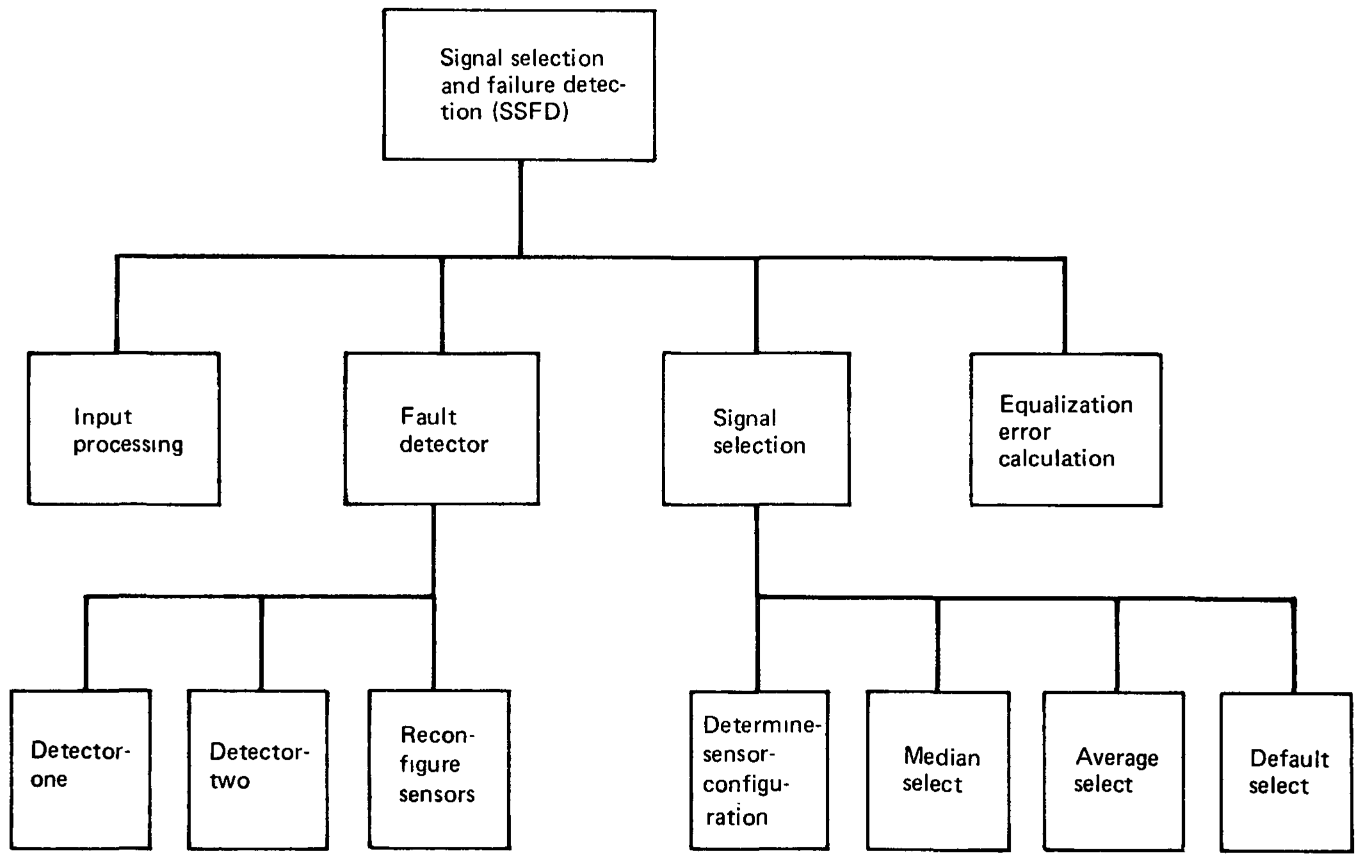


Figure 64. ACT Primary Computer Software, Signal Selection and Failure Detection

The input processing module determines an equalized input value from the raw signal and an error signal based on data from the previous frame. Fault detector software determines which sensors are functioning correctly and changes the configuration to exclude failed sensors. Two different failure detectors are implemented in software. Detector-one compares the difference between the signal and a reference value to a threshold. Signals that use equalization use detector-one to determine both high-frequency and low-frequency failures. The error signal, which is the difference between the input signal and the selected signals, is used for high-frequency detection. The equalization signal, which is the filtered error signal, is used for low-frequency and bias errors. Detector-two is used when no reference is available and compares differences between signals to determine errors. Additional logic determines from the threshold comparisons which sensor has failed. Both detectors use a threshold detector plus an increment/decrement counter algorithm to detect oscillatory failures. The reconfigure-sensors software uses fault detector information to declare temporary and permanent faults. Sensors that have been declared failed, either temporarily or permanently, are not used in the signal selection process. A temporarily failed sensor is declared good again if it returns to proper operation, as confirmed by indicating the correct signal for a period of time. This time limit is chosen so that changes in the measured quantity over that time period are presumed large enough that the error would be detected even if a sensor failed to the nominal signal value. A permanent failure cannot be recovered until maintenance action has been taken.

Signal selection software determines a single value for each parameter from the redundant sensor signals. The determine-sensor-configuration module determines how many values the selector operates upon. Median select software produces an output equal to the median of three input values. The average select module averages the values of the inputs when only two sensors remain. The default select module produces an output equal to some default value when there are fewer than two valid input signals. This default value may be a constant or the previously selected output.

Following signal selection, the equalization error calculation routine calculates the error signal, which is the difference between the raw signal and selected signal. The error signal is passed through a high-gain lag filter to form the equalization signal. The equalization signal is subtracted from the raw signal in the next frame during input processing to form the equalized input. This routine is bypassed for signals that do not require equalization.

Control law software (fig. 65) calculates surface position commands using selected sensor input data to provide the following functions: Full Pitch-Augmented Stability (Full PAS), angle-of-attack limiter (AAL), wing-load alleviation (WLA), lateral/directional-augmented stability (LAS), and flutter-mode control (FMC). The Full PAS function includes both short-period PAS and phugoid PAS modes. WLA includes maneuver-load control (MLC) and gust-load alleviation (GLA).

The output monitor software (fig. 63) detects computer failures that result in erroneous digital output. Surface position commands calculated using the control laws are transmitted digitally to other channels by the transfer data cross-channel software module. The failure detector compares these output values and detects any that are out of tolerance using the same algorithm as used by detector-two of the SSFD software (fig. 64). Reconfigure-servos software shuts down the servos of the affected channel when a failure is detected and reengages servos following successful recovery, providing reengagement synchronization as required to prevent undesirable transients. Recovery software reinitializes the computer and checks to see if proper operation resumes.

The servomonitor software checks the proper operation of the servos. The force-summed servoactuators use comparison with a mathematical model to determine which servo has failed in the event of a disagreement between the two servos, thus allowing operation with a single servo. Mathematical model software is used to predict the spool valve position for forced-summed secondary servos and to predict the actuator position for the flaperon power control units (PCU). The servo failure detector software compares servo parameters to detect failures using the same algorithm used in detector two of the SSFD software. Spool valve positions, both measured and predicted by the mathematical model, are compared for the secondary force-summed servos. Actuator positions, again including a mathematical model prediction, are compared for the flaperon actuators. Servo-loop currents in the electrohydraulic valve coils are compared for the dedicated force-displacement servos. If a failure is detected, the shutdown servos software acts to hydraulically bypass the servo in the case of force-summed actuators or to electrically null the output command to the force-displacement summed servos.

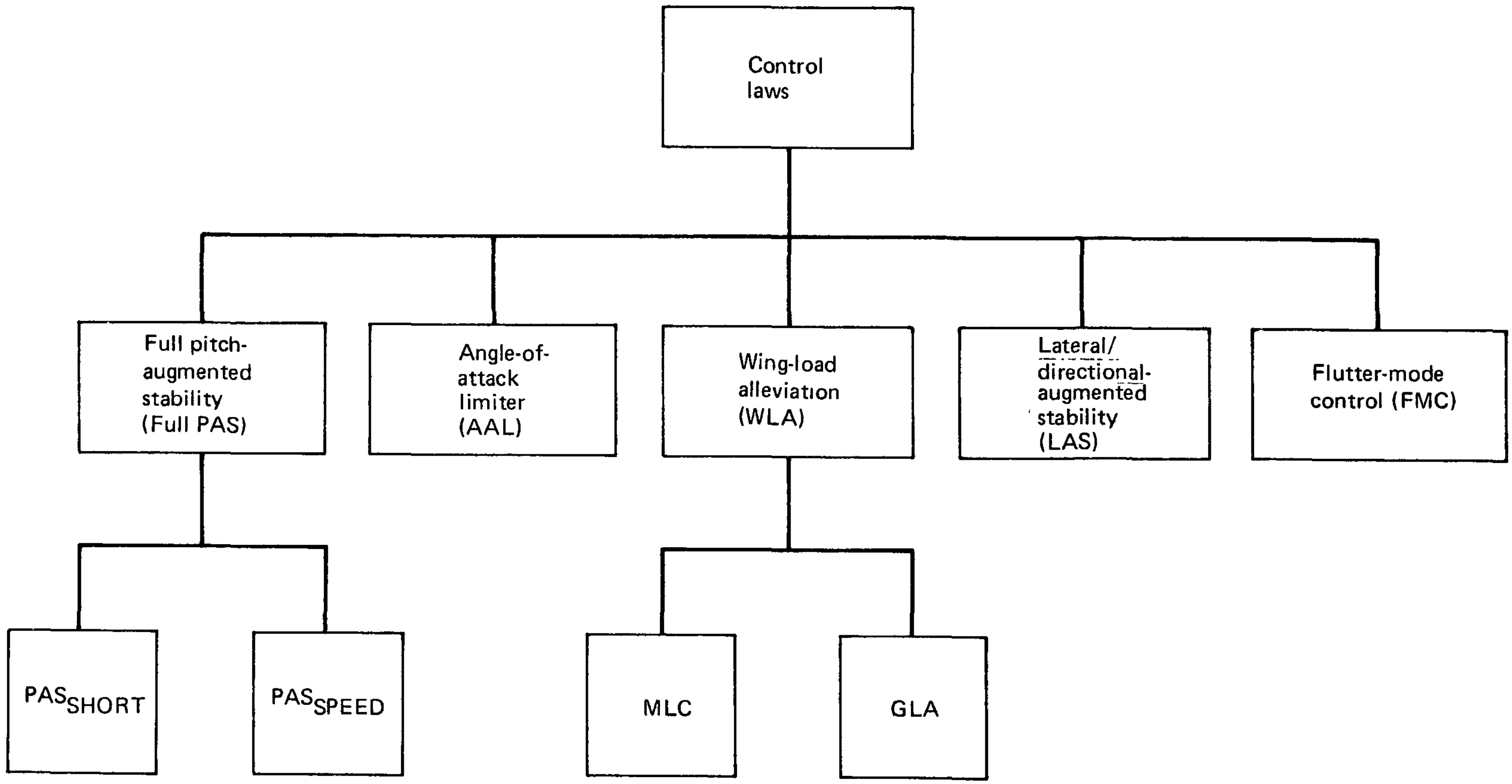


Figure 65. ACT Primary Computer Software, Control Laws

Background tasks software (fig. 66) performs tasks in flight to provide information about the state of the ACT systems. These tasks are not time critical and are performed on a time-available basis following completion of the foreground tasks. However, because information on the state of the system is required to be up to date for reliability purposes, sufficient time must be allowed for background tasks such that all may be completed within a reasonable amount of time.

The ACT Maintenance and Display Computer interface software provides system status information to the maintenance and display computer. The determine-LRU-status software consolidates information supplied by system monitors and self-test. The LRU status and monitor status is checked to determine functional status by the determine-functional-status software, and the result is supplied to the ACT Maintenance and Display Computer.

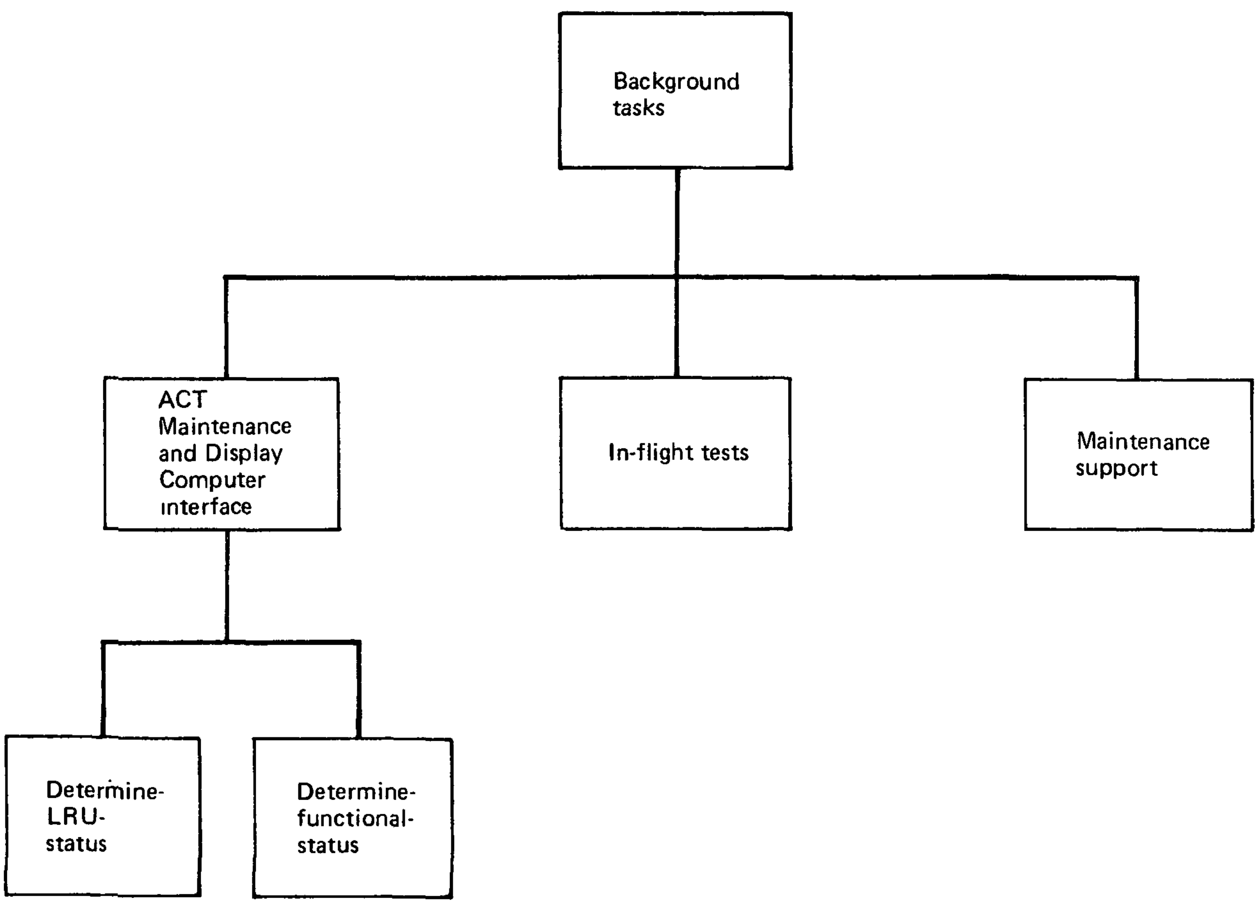


Figure 66. ACT Primary Computer Software, Background Tasks

In-flight tests software provides up-to-date information on the soundness of the system. Test results are used to supplement information from the real-time monitors. The in-flight tests include reasonableness tests on sensor data, program memory sum checks, cross-channel integrity tests, etc., that do not interfere with real-time operation.

Maintenance support software supplies information about the system so that failures may be located and repaired quickly and easily by maintenance personnel. This primarily consists of collecting and formatting data from monitors and tests so that data may be transmitted to the ACT Maintenance and Display Computer.

Ground Operations—Ground operations (fig. 67) is the second major task of the software. It provides software to perform preflight and maintenance testing. This module is active only when certain conditions are met; one condition is that the airplane must be on the ground. The software is divided into three parts: ground test executive, preflight test, and maintenance test.

The ground test executive software verifies test conditions and interrupt processing. Verify-test-conditions software checks to ensure that the airplane is on the ground and then runs the proper test when other conditions are met. The executive includes fault interrupt processing software and I/O interrupts processing software. The test scheduler software determines which tests are to be run.

The bulk of the ground operations software is the preflight test (fig. 68), which performs tests to determine the state of the system before flight. System components and functions must be tested before takeoff to ensure that safety requirements are met. These same tests may also be performed at other times for maintenance testing. The preflight consists of three different kinds of tests: computer self-tests, system monitor tests, and end-to-end tests.

Computer self-tests software (fig. 69) includes tests to verify the ability of the computer to perform correctly. These tests are performed solely by the computer and require no external input. The instruction repertoire test software checks the computer central processing unit (CPU) to verify that each legitimate operation code executes properly. Monitor-checks software tests each computer hardware monitor and verifies that they trip when an error occurs. The watchdog monitor test software checks that the watchdog

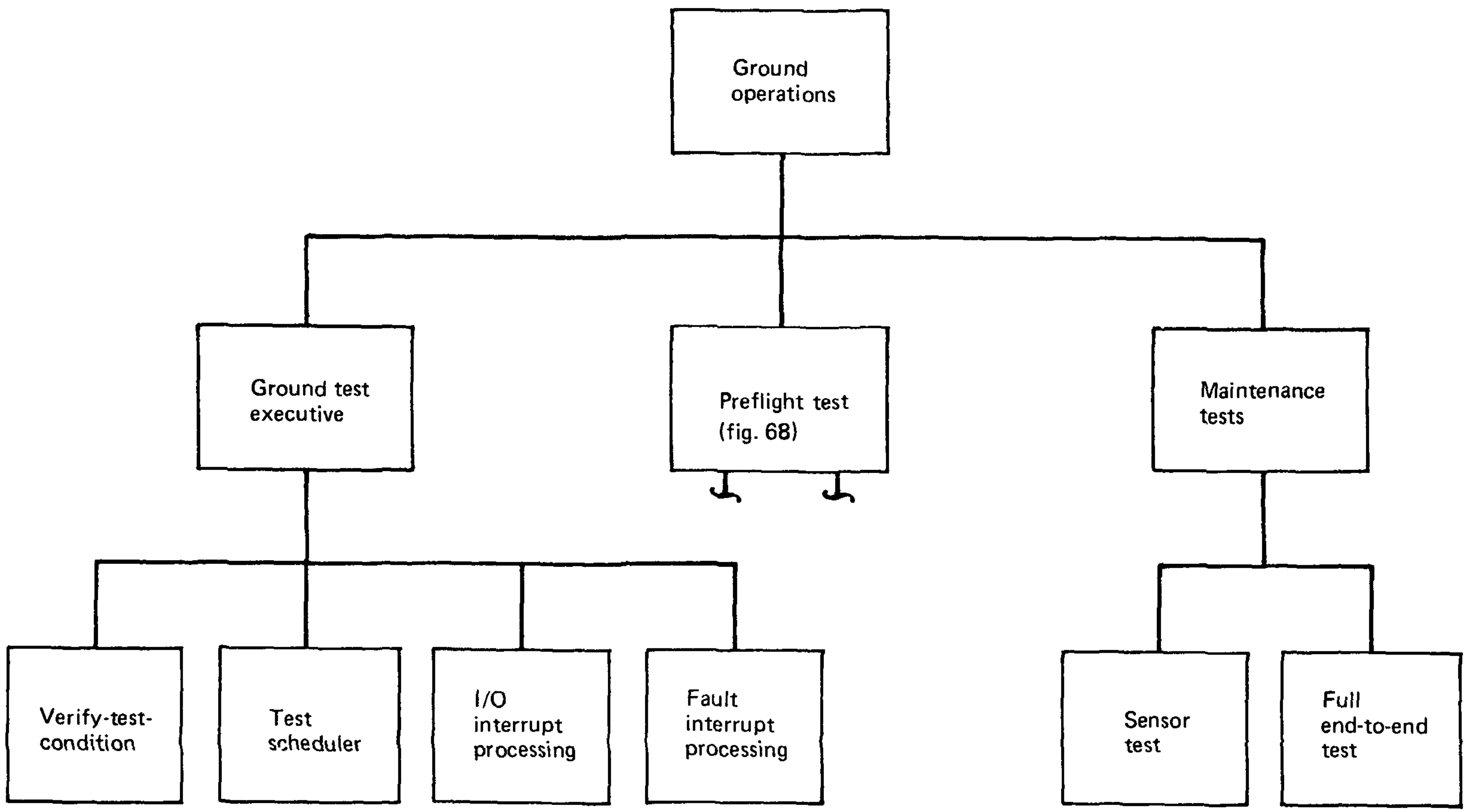


Figure 67. ACT Primary Computer Software, Ground Operations

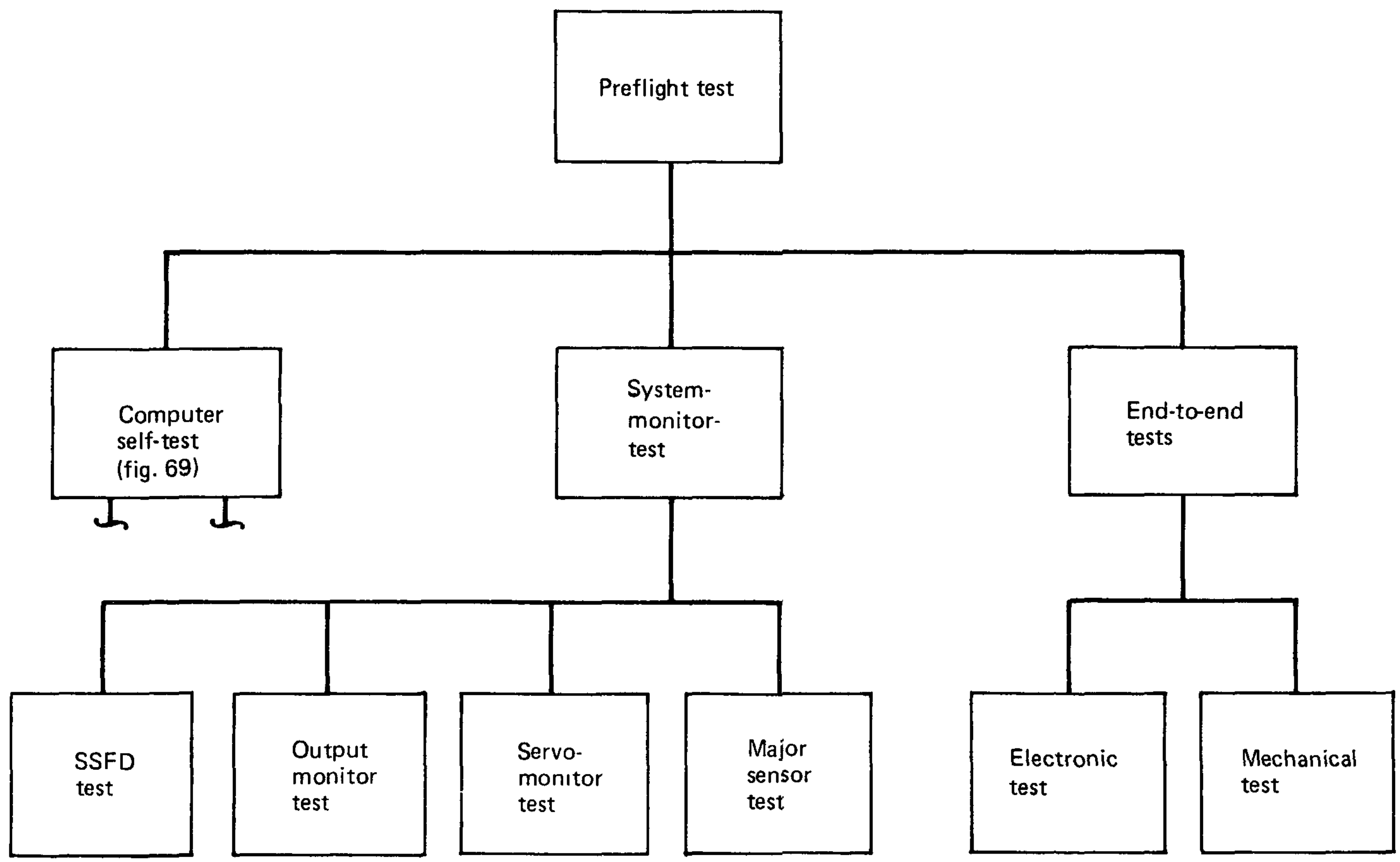


Figure 68. ACT Primary Computer Software, Preflight Test

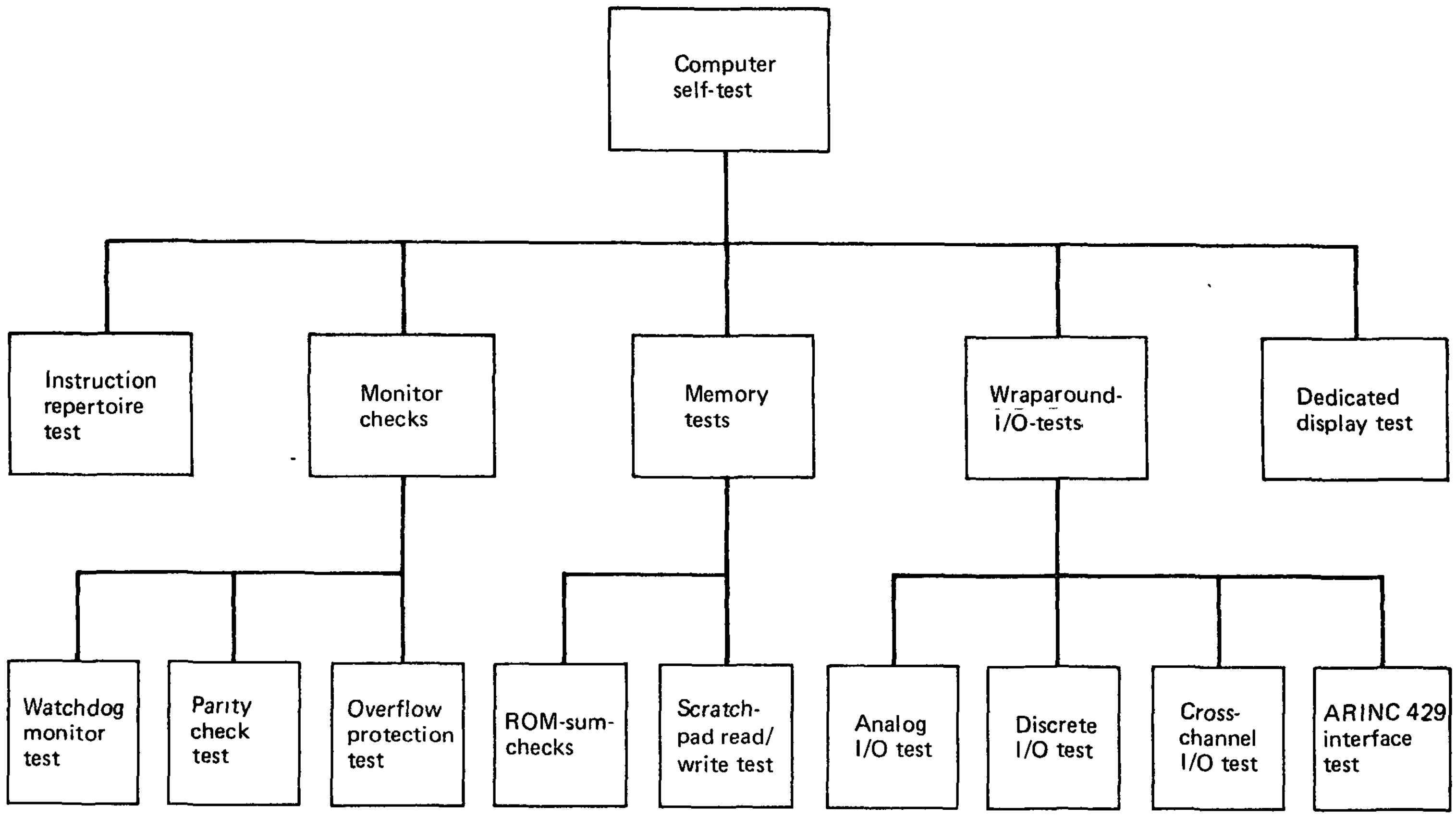


Figure 69. ACT Primary Computer Software, Computer Self-Test

monitor trips if not reset by the CPU. It also checks the length of the watchdog monitor timing windows. The parity check test software generates words with improper parity and verifies that the parity check hardware functions properly. The overflow protection test software performs arithmetic operations that result in an overflow and verifies that an overflow interrupt is generated.

Memory-tests software checks that all computer memory is functioning without error. ROM-sum-checks software verifies that errors have not occurred in program memory. This is done by adding all the operation codes in each block of memory and comparing that to a sum check code. Scratch pad read/write test software writes different patterns into each scratch pad memory location and then reads it back to verify that proper data are stored in memory. Patterns are chosen to show that each bit will set to both a "1" and a "0."

Wraparound-I/O-tests software tests the I/O circuits of the computer by closing a relay and wrapping outputs around to the computer inputs. Data are output, and inputs are checked after a suitable delay to verify that these same data have been received at the input. Four such tests are made, one for each type of I/O. These are analog I/O test, discrete I/O test, cross-channel I/O test, and Aeronautical Radio Incorporated (ARINC) 429 interface test.

The final computer self-test software module is the dedicated display test module, which checks the computer message display capability. The discrete warning lights are tested such that the crew can determine if the displays are functioning.

System-monitor-test software (fig. 68) tests major system functions. All computers must run the test simultaneously and depend on data from other computers. This software tests the system-level monitors and the major sensors. The major sensor test software asks the digital air data computer (DADC) and the inertial reference system (IRS) to perform internal tests and then checks the results. The system-level monitor test software provides data for normal and failure cases and checks that the monitor correctly determines the status. Three software modules perform these tests: the SSFD test module, output monitor test module, and servomonitor test module.

End-to-end tests software checks that the system is functioning properly from sensor input to actuator output. This is done in two steps: Electronic end-to-end test software checks all components from the sensors to the computer output. The mechanical end-to-end test software checks actuator operation when hydraulic power and ground clearance are available. This test checks servoelectronics and hydraulic actuators. The two parts of the end-to-end test verify that the system is functional.

Maintenance tests software (fig. 67) isolates faults and provides information for corrective maintenance. Although each test in the preflight may also be used for this purpose, the maintenance tests software is defined to include only those tests not also included in the preflight. This includes sensor test software, which tests dedicated ACT sensors, and full end-to-end test software, which combines the electronic and mechanical end-to-end tests of the preflight.

8.3.2.2 Essential PAS Computer Software

Software for the Essential PAS Computers is similar to the ACT Primary Computer software, except the real-time software is simplified because only one control law is implemented in the Essential PAS Computer. Figures 70 through 77 show the design tree for the Essential PAS Computer software. The task of the software is to maintain pitch stability augmentation and to provide adequate self-test and failure detection. This task is divided into two main tasks: real-time control and ground operations (fig. 70). All software modules may be considered to be identical in function to similarly named ACT Primary Computer software modules except as noted.

The real-time executive software (fig. 71) performs the same function as described earlier, except the Essential PAS Computer does not have a separate autonomous I/O controller to initiate analog-to-digital (A/D) conversion, so this must be performed by the CPU. The timer interrupt software must now perform two tasks: Analog-to-digital converter cycle software signals the conversion hardware to begin producing new values before the beginning of a frame. The frame initiate software begins processing a new frame. In addition, the fault processing software performs the same function as the fault interrupt processing module in the ACT Primary Computers, but the submodules are not necessarily interrupt driven.

It is in the foreground tasks module (fig. 72) that the Essential PAS Computer software differs most from the ACT Primary Computer software. The SSFD software (fig. 73)

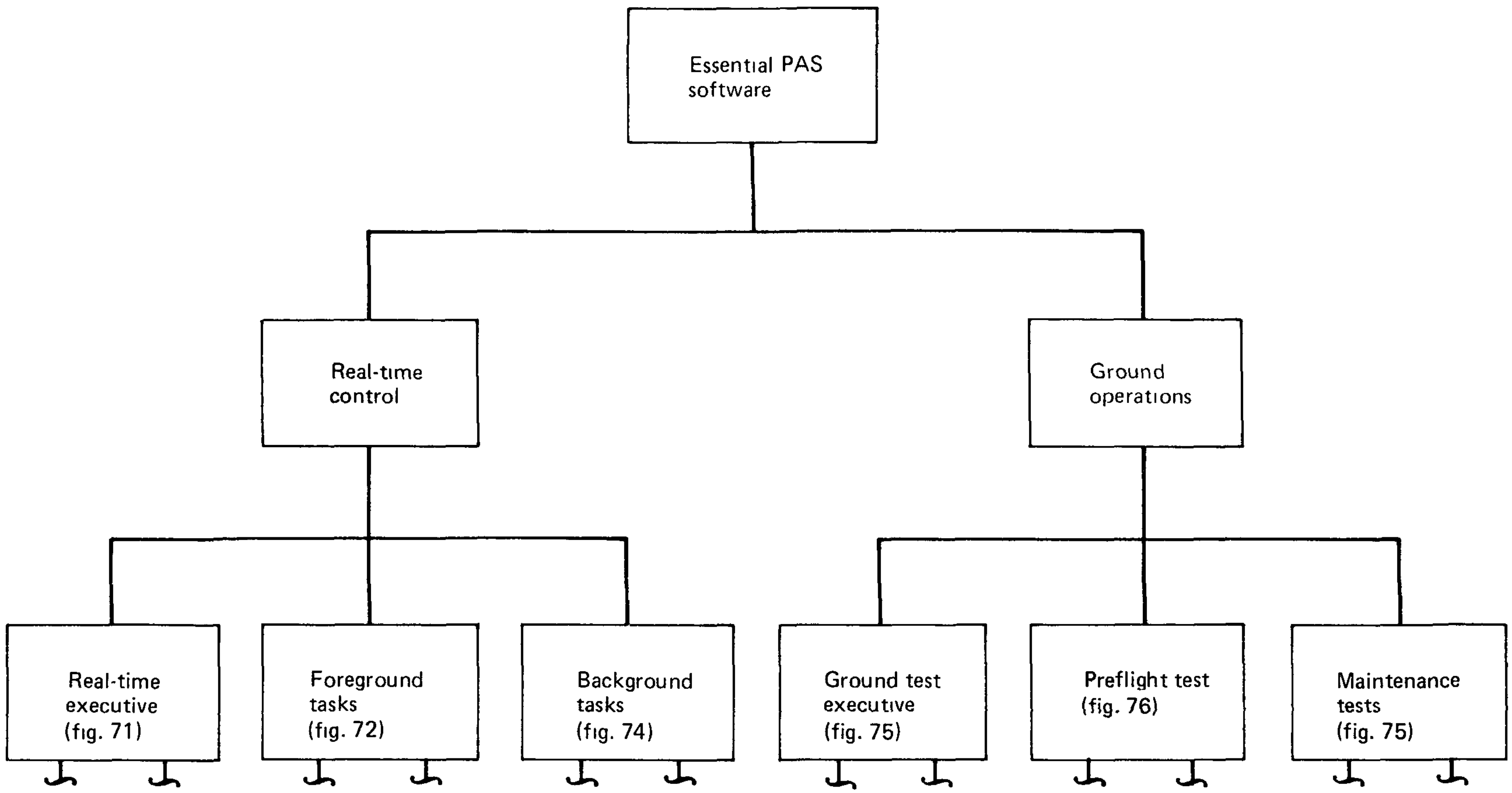


Figure 70. Essential Pitch-Augmented Stability Software Design Tree, Upper Levels

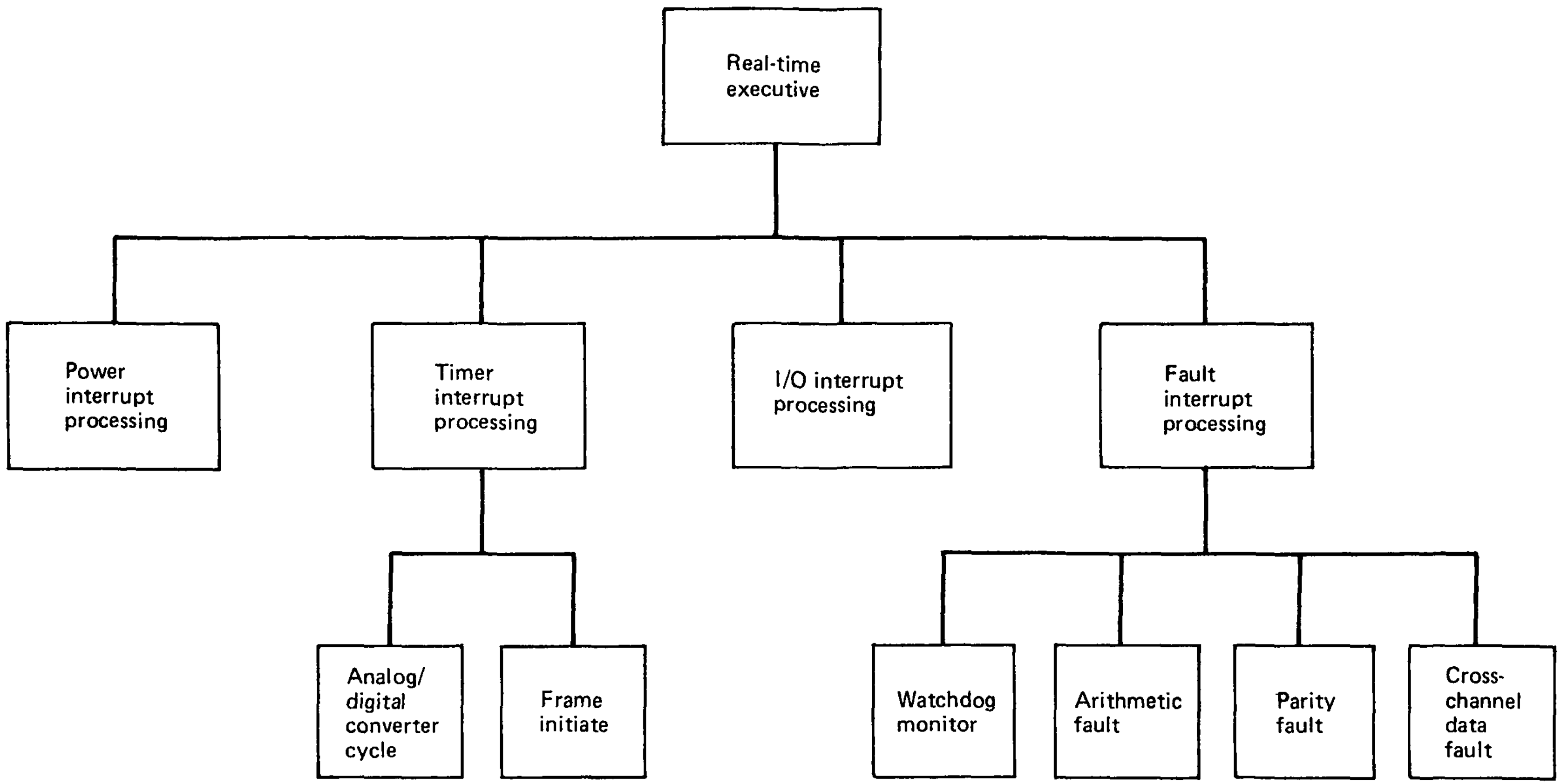


Figure 71. Essential Pitch-Augmented Stability Software, Real-Time Executive

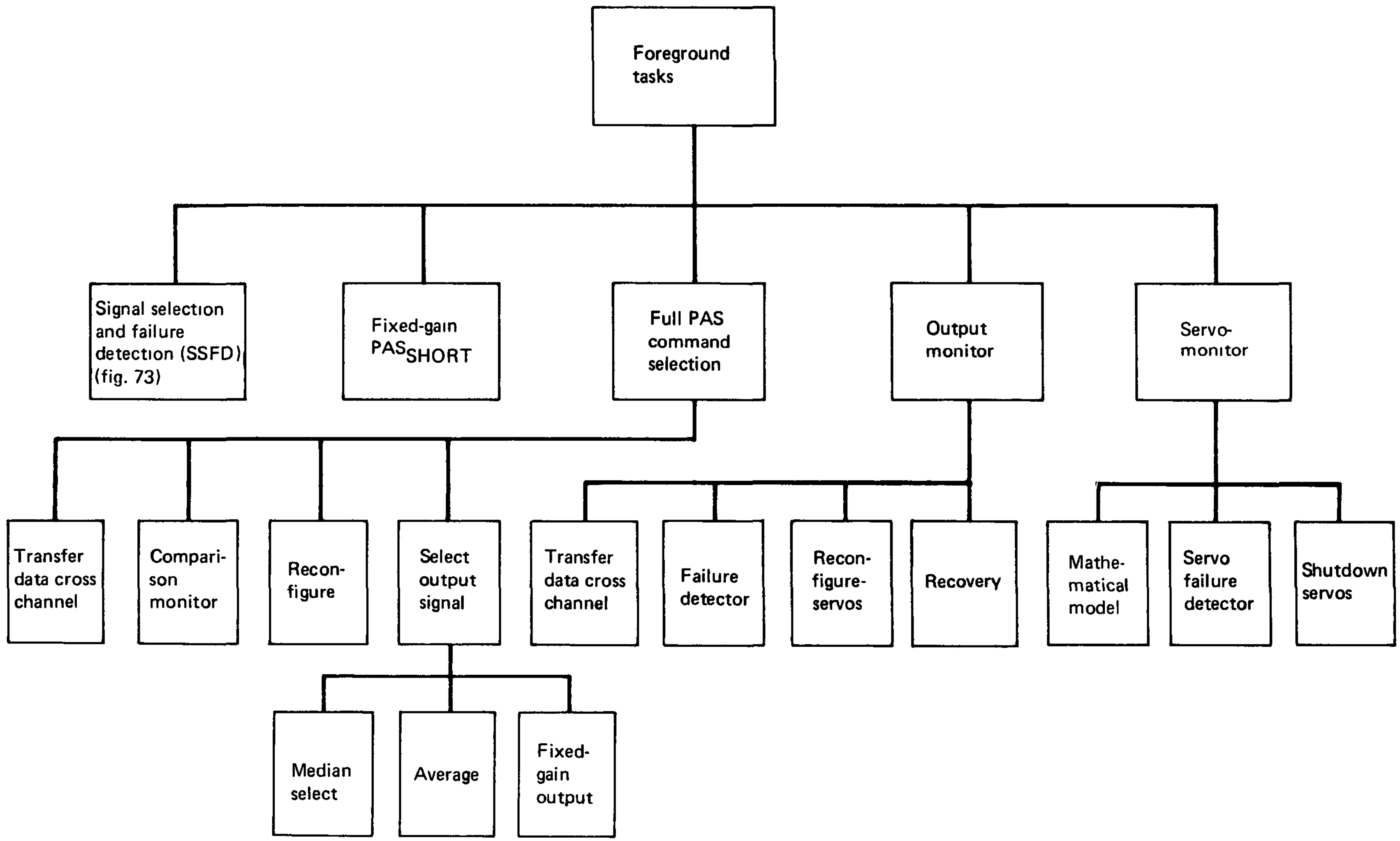


Figure 72. Essential Pitch-Augmented Stability Software, Foreground Tasks

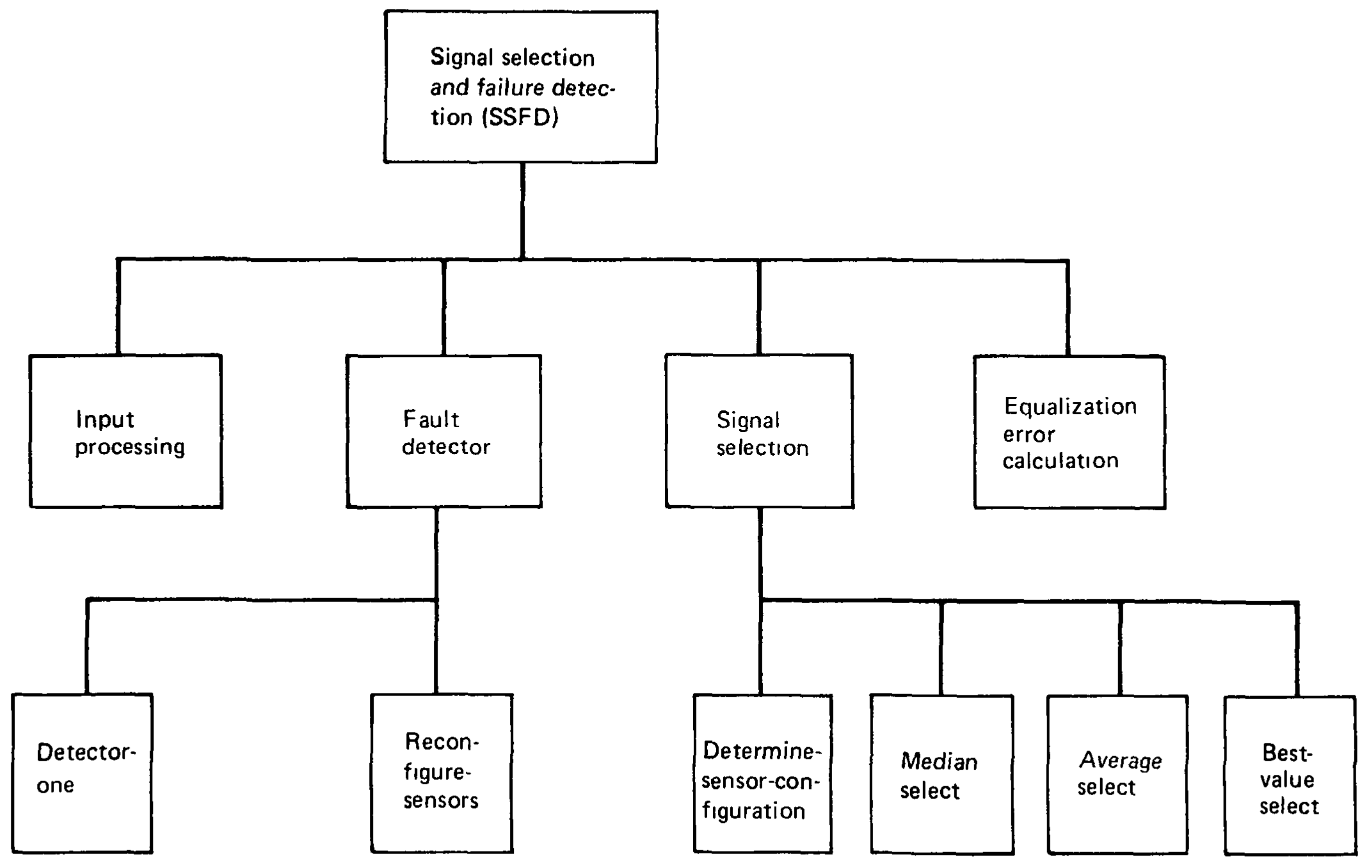


Figure 73. Essential Pitch-Augmented Stability Software, Signal Selection and Failure Detection

operates on sensor input to be used by the fixed-gain short-period PAS module. The SSFD software is similar to the ACT Primary Computer software with one major exception. The Essential PAS Computer SSFD must deal with four redundant sensors rather than three. All four signals are monitored by fault detector software at all times, but a maximum of three signals is used by the signal selection software; one signal is designated a standby and is not used until one of the other signals fails. If one of these signals fails while the standby is good, the median select software is still used, but with the standby signal used in place of the failed signal. If two failures occur, the average select software acts on the two remaining good signals. If a third sensor failure occurs, the fault detector provides insufficient data to determine which of the last two sensors has failed. In the ACT Primary Computer, this would result in choosing a default value. Selecting a default value for the pitch-rate signal, which is the only control variable processed by the Essential PAS Computer, would result in potential loss of the short-period PAS function and, given the ground rules of this study, loss of the aircraft. The best-value select software applies reasonableness tests and checks the results of sensor self-tests to determine which signal is correct. If no probable best value is indicated, one of the signals is arbitrarily selected.

Fixed-gain short-period PAS software is similar to that of the ACT Primary Computer. The Essential PAS foreground tasks software includes an additional module, the Full PAS command selection (fig. 72). This software selects the output signal to be sent to the elevator from the Full PAS commands received from the ACT Primary Computers and the command calculated by the fixed-gain short-period PAS software. Because each Essential PAS Computer receives data on only one bus from the ACT Primary Computers, these data must be transferred cross channel before any voting may take place. The Essential PAS Computers have no autonomous I/O controller; hence, the transfer must be done by the CPU. Transfer data cross-channel software performs this task. These data are then checked by the comparison monitor software to detect failures in the ACT Primary Computers or in the data link. The reconfigure software examines the results of this comparison along with status outputs from the ACT Primary Computers to determine validity of these data. The select output signal software selects a signal from the valid outputs. If all three redundant signals from the ACT Primary Computers are good, the median select software determines the median signal for use as the output. If only two signals are valid, the average software selects the mean of those two signals. If less than two valid signals are available, the fixed-gain output software selects the output. Three separate signals are transmitted over the bus from the ACT Primary Computers, one for a variable-gain short-period PAS, one for speed PAS, and one for the pitch compensation component of MLC. If either of the latter two fails, the fixed-gain output software selects zero for that part of the output. If the short-period PAS signal from the ACT Primary Computers fails, the fixed-gain output software selects the output calculated by the fixed-gain short-period PAS module of the Essential PAS Computers. The three parts of the elevator output are then summed. To prevent large step changes on the output, the fixed-gain output module includes easy-on software to smooth transients.

The output monitor software functions the same as that of the ACT Primary Computers, except that it compares four outputs rather than three. The servomonitor software also functions similar to the ACT Primary Computer software, except it deals only with force-summed secondary servos and compares three servo spool valve positions and a mathematical model.

Background tasks software (fig. 74) in the Essential PAS Computer is functionally identical to that of the ACT Primary Computer. Ground operations software (fig. 75) is also very similar to that of the ACT Primary Computer. The only significant differences occur in preflight test software (fig. 76), where the major sensor test module is deleted because the Essential PAS Computers do not receive data from the IRS or DADC. With cross strapping of pitch-rate sensor data, each computer communicates directly with each sensor and can isolate sensor failures by comparing results from the end-to-end tests.

Computer self-test (fig. 77) software in the Essential PAS Computers is simplified from that of the ACT Primary Computers due to reduced complexity of the computers. The major simplification is deletion of wraparound testing. Cross strapping of sensor data and coordination of testing provide comparable fault detection with only slightly degraded fault isolation, and deleting wraparounds aids computer simplification.

In addition to ACT Primary Computers and Essential PAS Computers, the system also requires an ACT Maintenance and Display Computer, which coordinates testing, provides fault information storage to aid maintenance personnel, and furnishes cockpit warning and advisory messages. This computer is not flight critical, and no effort has yet been made to detail software for it.

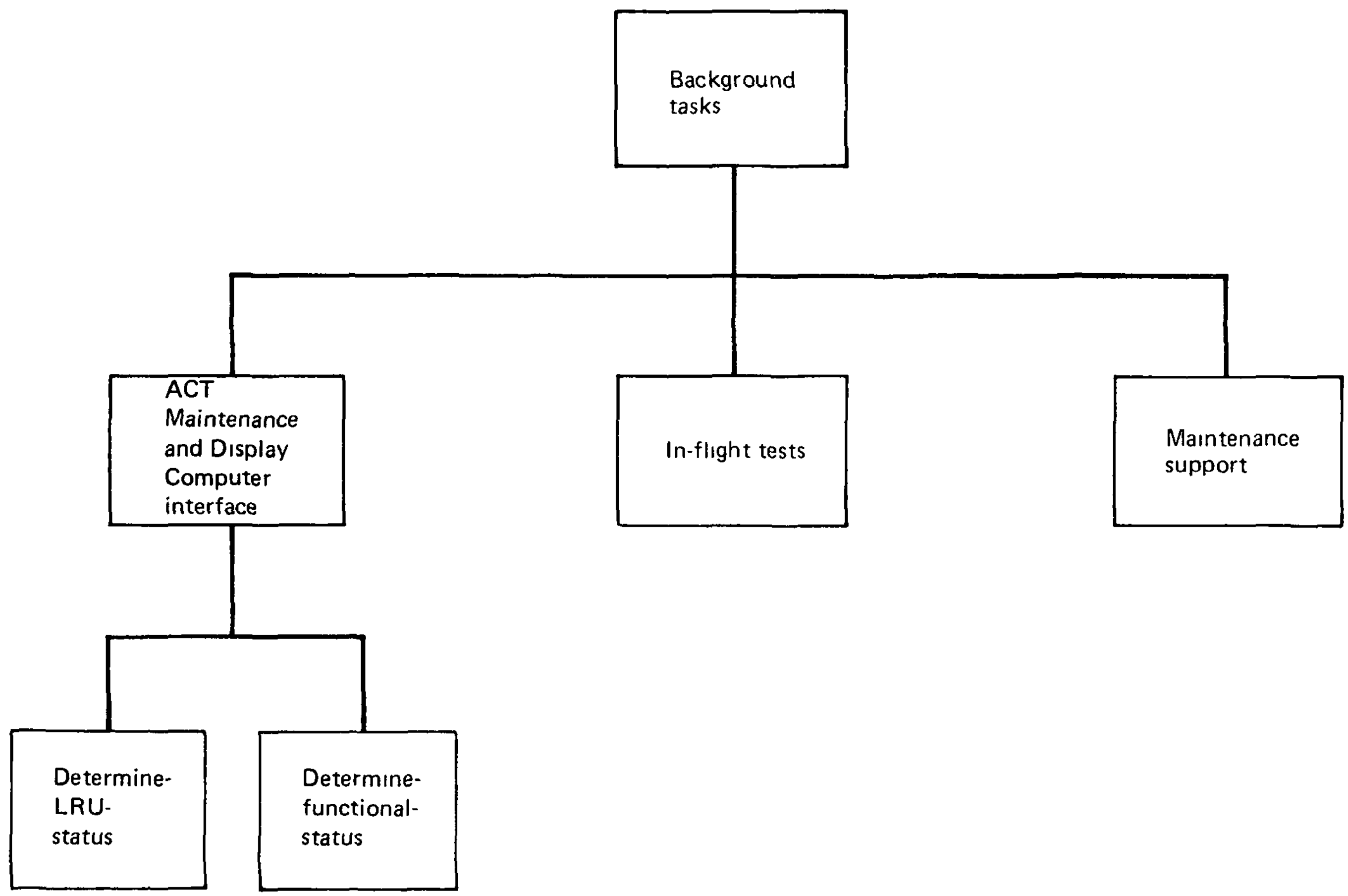


Figure 74. Essential Pitch-Augmented Stability Software, Background Tasks

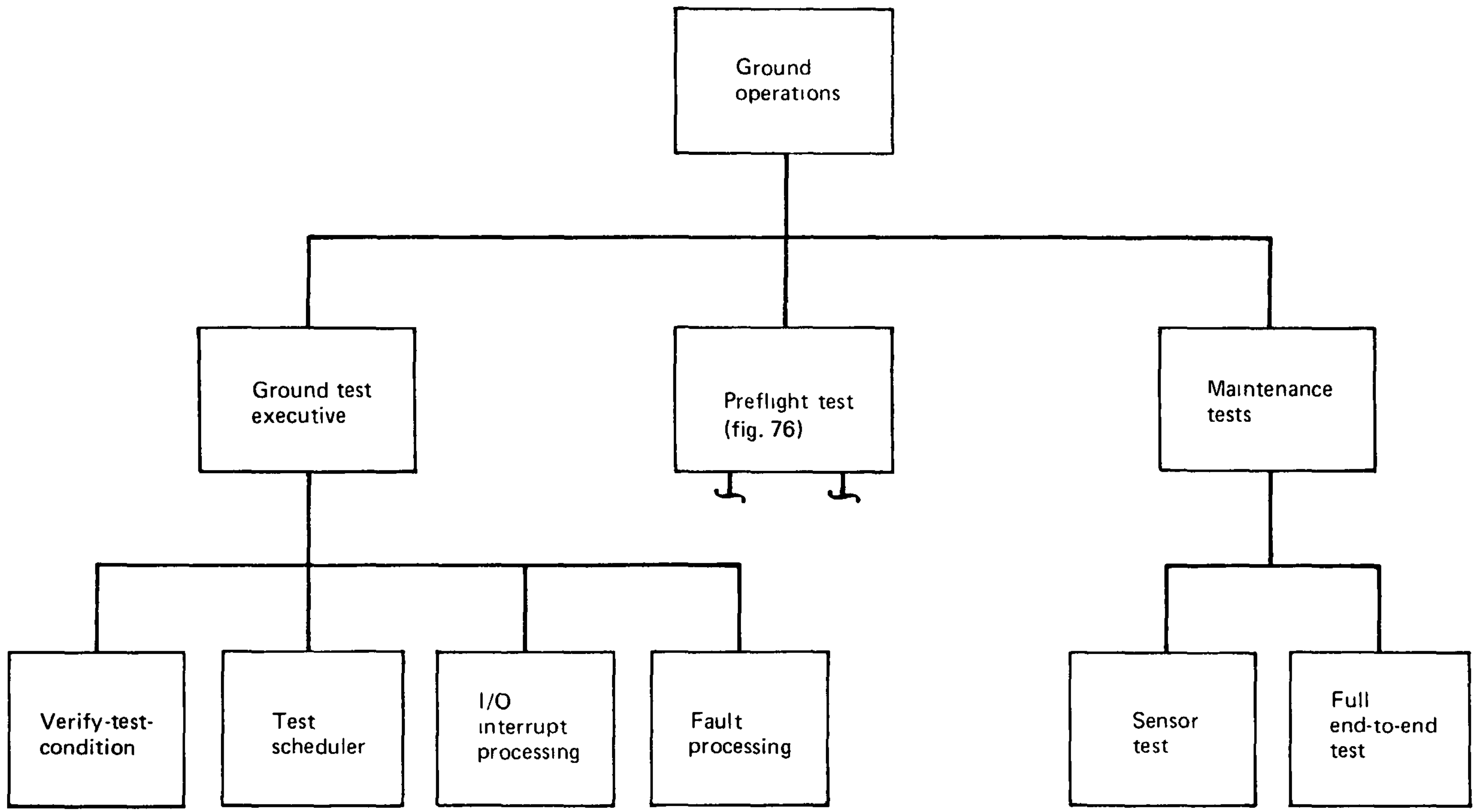


Figure 75. Essential Pitch-Augmented Stability Software, Ground Operations

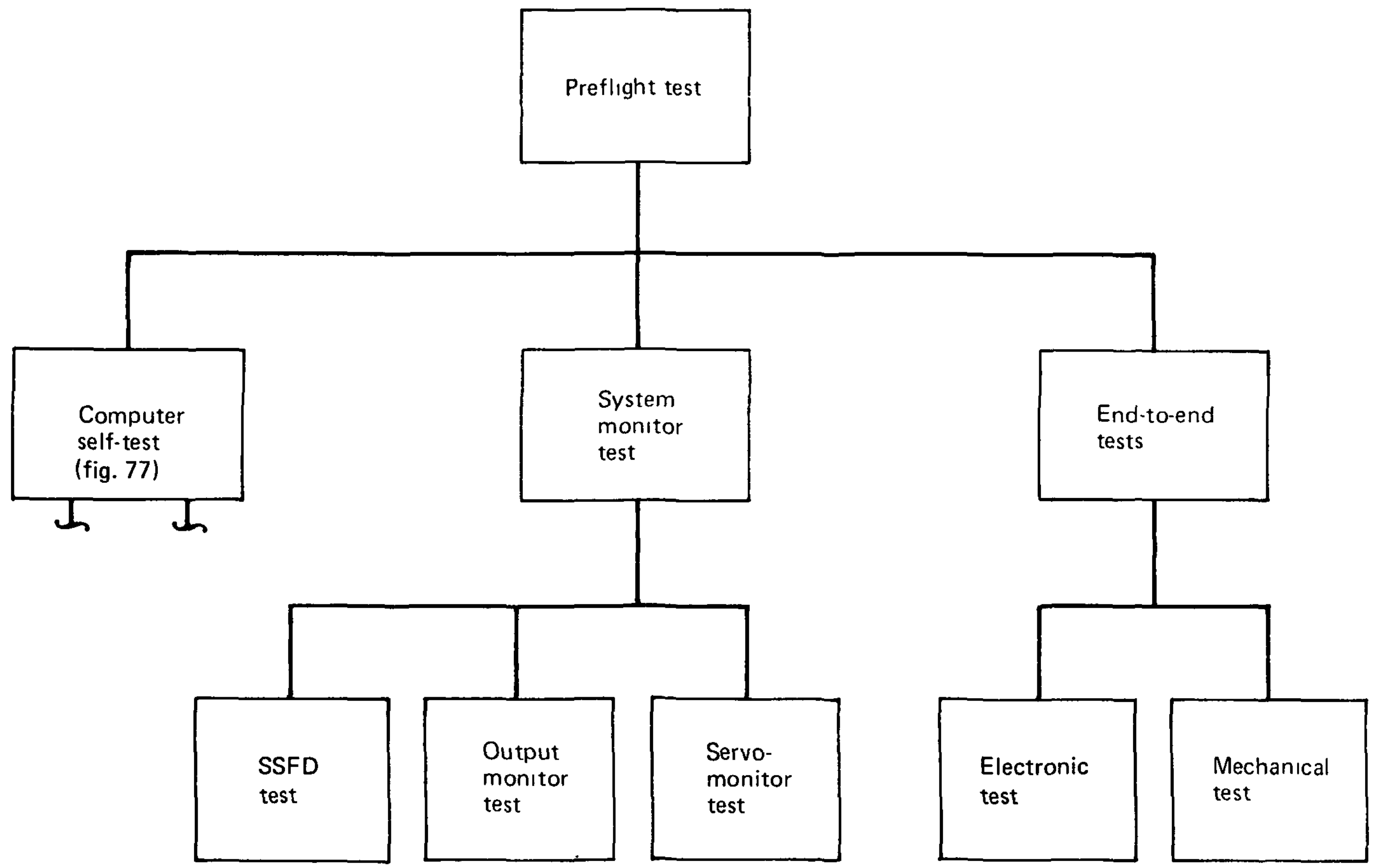


Figure 76. Essential Pitch-Augmented Stability Software, Preflight Test

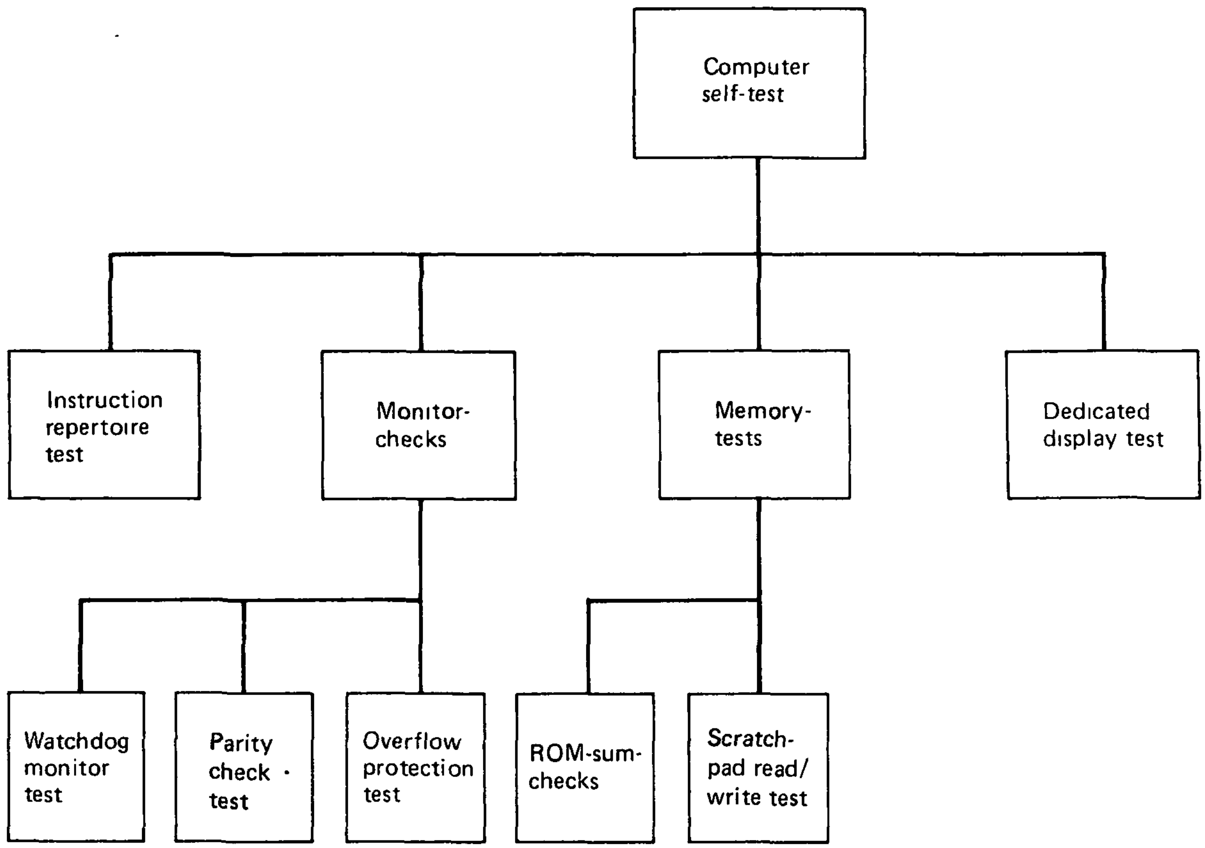


Figure 77. Essential Pitch-Augmented Stability Software, Computer Self-Test

8.3.3 MEMORY AND EXECUTION TIME ASSESSMENT

Based upon the software design described in Subsection 8.3.2, estimates of memory size and execution time have been made for major software modules. Table 13 summarizes these estimates. Wherever possible, estimates are based on laboratory test software written for the General Electric MCP-701A flight control computers, which represent performance typical of current-generation flight control computers.

Total memory required for the ACT Primary Computers is approximately 15.1K words. Allowing for 50% growth yields a requirement for a minimum of 22K words. Execution time for the real-time functions is estimated to be 14.8 ms in each 20-ms major frame. This allows a 34% margin for time growth. The growth margin may be inadequate if another machine with somewhat lower performance is considered. Even the 34% margin

Table 13. ACT Control System Memory Requirements and Timing

Software module	ACT Primary Computers		Essential PAS Computers	
	Memory, words	Execution time (per major frame),ms	Memory, words	Execution time (per major frame),ms
Real-time executive	750	0.2	400	0.2
Foreground tasks				
Synchronization	200	0.6	–	–
Scheduler	100	0.1	–	–
SSFD	800	4.5	325	2.3
Control laws	1 200	2.4	100	0.1
Output monitor	400	2.5	100	0.3
Servomonitor	500	4.5	200	0.5
Background tasks				
Control and display communication	1 900		300	
In-flight tests	500		300	
Maintenance support	200		100	
Ground operations				
Test executive	1 200		600	
Preflight tests	6 000		4500	
Maintenance test	1 000		400	
Total	15 100	14.9	7325	3.4

available for time growth with the MCP-701A is considerably below the preferred margin of 100%. However, the software is estimated based on a full-feature aircraft with all possible functions and surfaces. This represents a worst case with respect to execution time. For an actual system, the number of functions and surfaces is expected to be fewer; therefore, less execution time is expected. With this in mind, the relatively small growth margin is deemed acceptable. Background task execution time is not critical, with all functions completed in less than 5 sec.

The Essential PAS Computers require approximately 7325 words of memory, 1625 of which are for programs necessary for safe flight of the aircraft. The rest are active only on the ground for preflight or maintenance testing. A 50% memory growth allowance results in a requirement for a minimum of 11K words. Execution time for real-time software is estimated to be 3.4 ms per 10-ms frame, based upon execution of laboratory test programs on the MCP-701A. The PAS backup computer is based on a 16-bit microprocessor and would take longer to execute the problem. The margin available should be adequate even with the expected slower execution time. Timing of background tasks is not critical.

The ACT Maintenance and Display Computer software has not been defined in detail. It is expected to require up to 32K words, which could be 8-bit words. Timing is not critical for this computer.

8.4 ACT SYSTEM OPERATION

The Selected System was introduced in Subsection 6.4 and described more fully in Section 7.0. The basic plan of system operation is given in those sections.

This subsection will first detail the operability of the Active Controls Technology (ACT) system. Subsection 5.3.2 contains a narrative description of airplane performance capability with complete loss of each active control function considered separately. Subsection 8.4.1 presents that data in tabular form and develops the effect upon airplane operability of various combinations of functions lost and functions whose loss could result from one additional line replaceable unit (LRU) failure.

Subsections 8.4.2, 8.4.3, and 8.4.4, respectively, detail the means by which:

- System and component failures are detected.
- Failure and system status information is communicated to flight and ground crews.
- Flight plan change advisory messages are selected and communicated to the flight crew.

8.4.1 AIRPLANE DISPATCHABILITY AND IN-FLIGHT OPERABILITY

The effect of single function loss in the ACT system is described in Subsection 5.3.2 and reproduced in Table 14. Based on that airplane operability with a single function loss, a study was conducted to define operability with multiple ACT function failures. The results are shown in Table 15. The following list contains the assumptions made for the operability assessment shown in Table 15.

- Only those functions listed in the table are assumed to be failed completely; i.e., other functions are assumed to be operating normally.
- The table does not treat the period of transition from normal to restricted flight.

Table 14. (Assumed) ACT Airplane Operability and Dispatchability With a Single Function (Complete) Loss

Case No.	Complete function loss	In-flight operability	Dispatchability	Remarks
1	Short-period pitch-augmented stability (PAS)	Loss of airplane	No go	PAS consists of crucial and critical PAS. The crucial PAS loss will cause an immediate safety hazard.
2	Speed PAS	Flight restriction A	Restricted flight	Flying quality after the critical PAS loss is assumed to be acceptable for the restricted flight.
3	Angle of attack (AAL)	Continue normal flight schedule	No go	The ACT airplane has "locked in" stall at low-speed condition. The airplane can continue normal flight schedule if the AAL failure (complete function loss) occurs in flight. The airplane may divert to other airport if severe turbulence exists at destination. No go is for safety reasons.
4	Lateral/directional-augmented stability (LAS)	Flight restriction B	No go	Flying quality after the complete LAS loss is assumed to be acceptable for restricted flight. No go exists because of structure dispatch requirement.
5	Wing-load alleviation (WLA)	Continue normal flight schedule		The ACT airplane structure exceeds design limit load strength requirements without WLA system.
			No go	The ACT airplane structure does not satisfy design ultimate load strength requirements without WLA.
6	Flutter-mode control (FMC)	Flight restriction C	Restricted flight	The ACT airplane will retreat to a lower cruise speed to allow adequate speed margin for upset.

Table 15. (Assumed) ACT Airplane Operability and Dispatchability With Combination of Function Losses

Case No.	Crucial PAS	Critical PAS	AAL	LAS	WLA	FMC	In-flight operability	Dispatch-ability	Remarks
7		X	X				Flight restriction A	No go	Combination of cases 2 and 3
8		X		X			Flight restriction D	No go	Combination of cases 2 and 4; assume flight restriction D exists
9		X			X		Flight restriction A	No go	Combination of cases 2 and 5
10		X				X	Flight restriction E	Go with flight restriction	Combination of cases 2 and 6; assume flight restriction E exists
11			X	X			Flight restriction B	No go	Combination of cases 3 and 4
12			X		X		Continue normal flight	No go	Combination of cases 3 and 5
13			X			X	Flight restriction C	No go	Combination of cases 3 and 6
14				X	X		Flight restriction B	No go	Combination of cases 4 and 5
15				X		X	Flight restriction F	No go	Combination of cases 4 and 6; assume flight restriction F exists
16					X	X	Flight restriction C	No go	Combination of cases 5 and 6
17		X	X	X			Flight restriction D	No go	Combination of cases 2, 3, and 4
18		X	X		X		Flight restriction A	No go	Combination of cases 2, 3, and 5
19		X	X			X	Flight restriction E	No go	Combination of cases 2, 3, and 6
20		X		X	X		Flight diversion	No go	Loss of critical PAS, LAS, and WLA may result in an unacceptable risk in handling quality or performance capabilities under some abnormal conditions
21		X		X		X	Flight restriction G	No go	Combination of cases, 2, 4, and 6

X indicates function loss

Table 15. (Assumed) ACT Airplane Operability and Dispatchability With Combination of Function Losses (Continued)

Case No.	Crucial PAS	Critical PAS	AAL	LAS	WLA	FMC	In-flight operability	Dispatch-ability	Remarks
22		X			X	X	Flight restriction E	No go	Combination of cases 2, 5, and 6
23			X	X	X		Flight restriction B	No go	Combination of cases 3, 4, and 5
24			X	X		X	Flight restriction F	No go	Combination of cases 3, 4, and 6
25			X		X	X	Flight restriction C	No go	Combination of cases 3, 5, and 6
26				X	X	X	Flight restriction F	No go	Combination of cases 4, 5, and 6
27		X	X	X	X		Flight diversion	No go	Flight diversion for the same reasons as case 20
28		X	X	X		X	Flight restriction G	No go	AAL loss does not affect handling quality; same as case 21
29		X	X		X	X	Flight restriction E	No go	Loss of WLA and AAL does not affect handling quality; same as case 22
30		X		X	X	X	Flight diversion	No go	Flight diversion for the same reason as case 20
31			X	X	X	X	Flight restriction F	No go	Loss of AAL and WLA does not affect handling quality; flight restriction for the same reason as case 26
32		X	X	X	X	X	Flight diversion	No go	Flight diversion for the same reason as cases 20, 27, and 30

X indicates function loss

- Flight restrictions A, B, C, D, E, F, or G indicate that some type of restriction exists for given system failures. Lacking detailed airplane performance characteristics, these restrictions cannot be specified. Experience indicates that when these restrictions can be determined, there will probably be not seven different restrictions but three or four.

Based on the airplane operability in Tables 14 and 15, the ground rules were established as described in Subsections 8.4.1.1 and 8.4.1.2 for prediction of in-flight operability and dispatchability of the ACT airplane. Results of these predictions are discussed in detail under Reliability Assessment in Subsections 9.2.2 and 9.2.3.

8.4.1.1 In-Flight Operability

In-flight schedule changes include flight restriction and flight diversion. Flight restriction may force the pilot to change the flight envelope or alter normal operating procedures to avoid a potential safety hazard. In case of flight diversion, the pilot must land the airplane as soon as possible to avoid the hazard. In some cases, the pilot will initiate flight restriction with sufficient redundancy remaining prior to complete function loss. The following ground rules are established for assessment of schedule change reliability:

- The pilot shall continue a normal flight schedule when angle-of-attack limiter (AAL) or wing-load alleviation (WLA) or both functions become inoperative in the air, assuming other ACT functions are operative with sufficient redundancy.

- The pilot shall initiate a flight restriction when subsystem failures occur in such a way that one more LRU failure would result in the flight restriction in Table 15.

- The pilot shall initiate a flight diversion when subsystem failures occur in such a way that one more LRU failure would require flight diversion shown in Table 15.

The last two "one more failure" stipulations make the operability tables much more complicated. Adding those statements to Table 15 yields Tables 16 and 17.

8.4.1.2 Dispatchability

The following general rules govern the influence of the ACT control system on airplane dispatch.

- <u>No Go</u>

  - The airplane shall not be dispatched, regardless of other function status, when the system has complete loss of one or a combination of AAL, lateral/directional-augmented stability (LAS), and WLA functions.

Table 16. Dispatchability

Case No.	System failures[a,b]	Dispatchability			Remarks[c]
		No go	Go with flight restriction	Go without flight restriction	
1	Crucial PAS	Crucial PAS(X)[d,e]	Not applicable	Not applicable	–
2	Critical PAS	Not applicable	Critical PAS(X or 0)	Not applicable	Flight restriction A
3	AAL	AAL(X)	Not applicable	AAL(0)	
4	LAS	LAS(X)	LAS(0)	Not applicable	Flight restriction B
5	WLA	WLA(X)	Not applicable	WLA(0)	
6	FMC	Not applicable	FMC(X)	FMC(0)	Flight restriction C
7	Critical PAS, AAL	AAL(X) and DC[f]	PAS(X or 0) and AAL(0)	Not applicable	Flight restriction A
8	Critical PAS, LAS	LAS(X) and DC	PAS(X or 0) and LAS(0)	Not applicable	Flight restriction D
9	Critical PAS, WLA	WLA(X) and DC	PAS(X or 0) and WLA(0)	Not applicable	Flight restriction A
10	Critical PAS, FMC	Not applicable	PAS(X or 0) and FMC(X)	Not applicable	Flight restriction E
			PAS(X or 0) and FMC(0)	Not applicable	Flight restriction A
11	AAL, LAS	AAL(X) or LAS(X) and DC	AAL(0) and LAS(0)	Not applicable	Flight restriction B
12	AAL, WLA	AAL(X) or WLA(X) and DC	Not applicable	AAL(0) and WLA(0)	
13	AAL, FMC	AAL(X) and DC	AAL(0) and FMC(X)	AAL(0) and FMC(0)	Flight restriction C
14	LAS, WLA	LAS(X) or WLA(X)	LAS(0) and WLA(0)	Not applicable	Flight restriction B

[a]Only those functions shown in this column are assumed to be failed. (Other functions are operative with sufficient redundancy.)

[b]Single and combination of failures

[c]Indicates type of flight restriction

[d]No go if remaining crucial PAS has reliability lower than 10^{-9} failure after 1 hr of flight

[e](X) indicates complete function loss
(0) indicates one failure away from complete function loss
Example 1: LAS(X) = complete LAS loss
LAS(0) = one failure away from complete LAS loss
Example 2: LAS(X or 0) = LAS(X) or LAS(0)

[f]DC indicates don't care
Example: (Case 7) no go under AAL(X) and DC = no go if AAL is lost regardless of other functions (don't care other functions)

Table 16. Dispatchability (Continued)

Case No.	System failures[a,b]	Dispatchability			Remarks[c]
		No go	Go with flight restriction	Go without flight restriction	
15	LAS, FMC	LAS(X) and DC	LAS(0) and FMC(X) LAS(0) and FMC(0)	Not applicable Not applicable	Flight restriction F Flight restriction B
16	WLA, FMC	WLA(X) and DC	WLA(0) and FMC(X)	WLA(0) and FMC(0)	Flight restriction C
17	Critical PAS, AAL, LAS	AAL(X) or LAS(X) and DC	PAS(X or 0) and LAS(0) and AAL(0)	Not applicable	Flight restriction D
18	Critical PAS, AAL, WLA	AAL(X) or WLA(X) and DC	PAS(X or 0) and WLA(0) and AAL(0)	Not applicable	Flight restriction A
19	Critical PAS, AAL, FMC	AAL(X) and DC	PAS(X or 0) and AAL(0) or FMC(X) PAS(X or 0) and AAL(0) or FMC(0)	Not applicable Not applicable	Flight restriction E Flight restriction A
20	Critical PAS, LAS, WLA	(PAS·LAS·WLA) (0) or LAS(X) or WLA(X)	Not applicable	Not applicable	
21	Critical PAS, LAS, FMC	LAS(X) and DC	PAS(X or 0) and LAS(0) and FMC(X) PAS(X or 0) and LAS(0) and FMC(0)	Not applicable Not applicable	Flight restriction G Flight restriction D
22	Critical PAS, WLA, FMC	WLA(X) and DC	PAS(X or 0) and WLA (0) and FMC(X) PAS(X or 0) and WLA(0) and FMC(0)	Not applicable Not applicable	Flight restriction E Flight restriction A
23	AAL, LAS, WLA	AAL(X) or LAS(X) or WLA(X) and DC	AAL(0) and LAS(0) and WLA(0)	Not applicable	Flight restriction B
24	AAL, LAS, FMC	AAL(X) or LAS(X) and DC	AAL(0) AND LAS(0) and FMC(X) AAL(0) and LAS(0) and FMC(0)	Not applicable Not applicable	Flight restriction F Flight restriction B
25	AAL, WLA, FMC	AAL(X) or WLA(X) and DC	AAL(0) and WLA(0) and FMC(X)	AAL(0) and WLA(0) and FMC(0)	Flight restriction C
26	LAS, WLA, FMC	LAS(X) or WLA(X) and DC	LAS(0) and WLA(0) and FMC(X) LAS(0) and WLA(0) and FMC(0)	Not applicable Not applicable	Flight restriction F Flight restriction B
27	Critical PAS, AAL, LAS, WLA	(PAS·LAS·WLA) (0) or LAS(X) or WLA(X) and DC·AAL	Not applicable	Not applicable	

Table 16. 'Dispatchability (Concluded)

Case No.	System failures[a,b]	Dispatchability			Remarks[c]
		No go	Go with flight restriction	Go without flight restriction	
28	Critical PAS, AAL, LAS,FMC	AAL(X) or LAS(X) and DC	PAS(X or 0) and AAL(0) and LAS(0) and FMC(X)	Not applicable	Flight restriction G
			PAS(X or 0) and AAL(0) and LAS(0) and FMC(0)	Not applicable	Flight restriction D
29	Critical PAS, AAL, WLA, FMC	AAL(X) or WLA(X) and DC	PAS(X or 0) and AAL(0) and WLA(0) and FMC(X)	Not applicable	Flight restriction E
			PAS(X or 0) and AAL(0) and WLA(0) and FMC(0)	Not applicable	Flight restriction A
30	Critical PAS, LAS, WLA, FMC	PAS(X or 0) and LAS(X or 0) and WLA(X or 0) and DC-AAL	Not applicable	Not applicable	
31	AAL, LAS, WLA, FMC	AAL(X) or LAS(X) or WLA(X) and DC-FMC	AAL (0) and LAS(0) and WLA (0) and FMC(X)	Not applicable	Flight restriction F
			AAL(0) and LAS(0) and WLA (0) and FMC(0)	Not applicable	Flight restriction B
32	Critical PAS, AAL, LAS, WLA, FMC	PAS(X or 0) and LAS (X or 0) and WLA(X or 0) and DC-FMC and DC-AAL	Not applicable	Not applicable	

- The airplane shall not be dispatched, regardless of other function status, when subsystem failures occur in such a way that the crucial pitch-augmented stability (PAS) function does not meet the reliability requirement of 10^{-9} failure probability per 1-hr flight.

- Go With Flight Restriction

The airplane shall be dispatchable with a flight restriction when subsystem failure results in one of the following failure combinations:

- Loss of critical PAS function
- One failure from complete critical PAS loss
- One failure from complete LAS loss
- Loss of flutter-mode control (FMC) function

Table 17. : In-Flight Operability

Case No.	System failures[a,b]	Flight diversion	Flight restriction	Continuation of normal flight schedule	Remarks[c]
1	Crucial PAS	PAS(0)[d]	Not applicable	Other crucial PAS failures	Complete loss of crucial PAS will lose aircraft
2	Critical PAS	Not applicable	Critical PAS(X or 0)	Not applicable	Flight restriction A
3	AAL	Not applicable	Not applicable	AAL (X or 0)	
4	LAS	Not applicable	LAS(X or 0)	Not applicable	Flight restriction B
5	WLA	Not applicable	Not applicable	WLA(X or 0)	
6	FMC	Not applicable	FMC(X)	FMC(0)	Flight restriction C
7	Critical PAS, AAL	Not applicable	PAS(X or 0) and DC-AAL[e]	Not applicable	Flight restriction A
8	Critical PAS, LAS	Not applicable	PAS(X or 0) and LAS(X or 0)	Not applicable	Flight restriction D
9	Critical PAS, WLA	Not applicable	PAS(X or 0) and DC-WLA	Not applicable	Flight restriction A
10	Critical PAS, FMC	Not applicable	PAS(X or 0) and FMC(X)	Not applicable	Flight restriction E
			PAS(X or 0) and FMC(0)	Not applicable	Flight restriction A
11	AAL, LAS	Not applicable	LAS(X or 0) and DC-AAL	Not applicable	Flight restriction B
12	AAL, WLA	Not applicable	Not applicable	AAL(X or 0) and WLA (X or 0)	
13	AAL, FMC	Not applicable	FMC(X) and DC-AAL	FMC(0) and DC-AAL	Flight restriction C
14	LAS, WLA	Not applicable	LAS(X or 0) and DC-WLA	Not applicable	Flight restriction B

[a]Only those functions shown in this column are assumed to be failed.
(Other functions are operative with sufficient redundancy.)

[b] Single and combination of failures

[c] Indicates type of flight restriction

[d] (X) indicates complete function loss
(0) indicates one failure away from complete function loss
Example 1: LAS(X) = complete LAS loss
LAS(0) = one LRU failure away from complete LAS loss
Example 2: LAS(X or 0) = LAS(X) or LAS(0)

[e] DC indicates don't care
Example: (Case 7) No go under AAL (X) and DC = No go if AAL is lost regardless of other functions
(don't care other functions)

Table 17. In-Flight Operability (Continued)

Case No.	System failures[a,b]	Flight diversion	Flight restriction	Continuation of normal flight schedule	Remarks[c]
15	LAS, FMC	Not applicable	LAS(X or 0) and FMC(X)	Not applicable	Flight restriction F
			LAS(X or 0) and FMC(0)	Not applicable	Flight restriction B
16	WLA, FMC	Not applicable	FMC(X) and DC-WLA	FMC(0) and DC-WLA	Flight restriction C
17	Critical PAS, AAL, LAS	Not applicable	PAS(X or 0) and LAS(X or 0) and DC-AAL	Not applicable	Flight restriction D
18	Critical PAS, AAL, WLA	Not applicable	PAS(X or 0), and DC-AAL and DC-WLA	Not applicable	Flight restriction A
19	Critical PAS, AAL, FMC	Not applicable	PAS(X or 0) and FMC(X) and DC-AAL	Not applicable	Flight restriction E
			PAS(X or 0) and FMC(0) and DC-AAL	Not applicable	Flight restriction A
20	Critical PAS, LAS, WLA	(PAS·LAS·WLA) (X or 0)	Not applicable	Not applicable	
21	Critical PAS, LAS, FMC	Not applicable	PAS(X or 0) and LAS(X or 0) and FMC(X)	Not applicable	Flight restriction G
			PAS(X or 0) and LAS(X or 0) and FMC(0)	Not applicable	Flight restriction D
22	Critical PAS, WLA, FMC	Not applicable	PAS(X or 0) and DC-WLA and FMC(X)	Not applicable	Flight restriction E
			PAS(X or 0) and DC-WLA and FMC(0)	Not applicable	Flight restriction A
23	AAL, LAS, WLA	Not applicable	LAS(X or 0) and DC-AAL and DC-WLA	Not applicable	Flight restriction B
24	AAL, LAS, FMC	Not applicable	LAS(X or 0) and DC-AAL and FMC(X)	Not applicable	Flight restriction F
			LAS(X or 0) and DC-AAL and FMC(0)	Not applicable	Flight restriction B
25	AAL, WLA, FMC	Not applicable	FMC(X) and DC-AAL and DC-WLA	FMC(0) and DC-AAL and DC-WLA	Flight restriction C
26	LAS, WLA, FMC	Not applicable	LAS(X or 0) and DC-WLA and FMC(X)	Not applicable	Flight restriction F
			LAS(X or 0) and DC-WLA and FMC(0)	Not applicable	Flight restriction B

Table 17. In-Flight Operability (Concluded)

Case No.	System failures[a,b]	Flight diversion	Flight restriction	Continuation of normal flight schedule	Remarks[c]
27	Critical PAS, AAL, LAS, WLA	(PAS-LAS-WLA) (X or 0) and DC-AAL	Not applicable	Not applicable	
28	Critical PAS, AAL, LAS, FMC	Not applicable	PAS(X or 0) and DC-AAL and LAS(X or 0) and FMC(X) and	Not applicable	Flight restriction G
			PAS(X or 0) and DC-AAL and LAS(X or 0) and FMC(0)	Not applicable	Flight restriction D
29	Critical PAS, AAL, WLA, FMC	Not applicable	PAS(X or 0) and DC-AAL and DC-WLA and FMC(X) and	Not applicable	Flight restriction E
			PAS(X or 0) and DC-AAL and DC-WLA and FMC(0)	Not applicable	Flight restriction A
30	Critical PAS, LAS, WLA, FMC	(PAS-LAS-WLA) (X or 0) and DC-FMC	Not applicable	Not applicable	
31	AAL, LAS, WLA, FMC	Not applicable	LAS(X or 0) and FMC(X) and DC-AAL and DC-WLA	Not applicable	Flight restriction F
			LAS(X or 0) and FMC(0) and DC-AAL and DC-WLA	Not applicable	Flight restriction B
32	Critical PAS, AAL, LAS, WLA, FMC	(PAS-LAS-WLA) (X or 0) and DC-FMC and DC-AAL	Not applicable	Not applicable	

- <u>Go With No Flight Restriction</u>

  The airplane shall be dispatchable without any restriction when subsystem failures occur in such a way that one more failure would result in loss of FMC function.

Again, inclusion of the "one failure from" statements expands Table 15 to the much more inclusive Table 17 form.

8.4.2 SYSTEM TEST AND MAINTENANCE

Figure 78 shows the organization of ACT testing in three modes. Although routine preflight testing is undesirable, the system criticality described in Subsection 5.3.1 makes preflight essential, and the primarily electronic character of the system, coupled with computer automation, combines to make preflight test time very short. These tests are indicated in the second column at the left in Figure 78. The same tests and others can be called up as needed for maintenance operations represented at the right in the figure. In the IAAC ground test concept, all of these tests are accomplished without special test equipment; the active control computers do the whole job.

The ACT Primary Computers have sufficient capacity such that they can perform multiple control law and redundancy management calculations during flight and also monitor their own soundness and that of the system as well. The in-flight monitor mode is indicated in the middle of Figure 78.

The computer memory required for system test is presently estimated at 8200 words in the ACT Primary Computer and 5500 words in the Essential PAS Computer. Figure 79 shows the logic fundamental to the selection of test and maintenance modes and the basic control character of each mode. One of the large blocks will be detailed in a later figure.

Figure 80 is a bar chart that compares time available to time required for routine ACT testing. The ACT Design Requirements and Objectives Document states the objective that "preflight test time shall be less than 2 minutes." ACT Primary and Essential PAS Computer sets will test themselves simultaneously; the ACT Primary self-test series is slightly longer than the Essential PAS, both remaining below 30 sec by current estimate.

The electronic preflight will be performed simultaneously with other cockpit preparation operations typically needing 10 min or more. The mechanical part of preflight must exercise the ACT servos. This requires operating pressure in the hydraulic system and means that either the main propulsion engines or the auxiliary power unit (APU) must be running. The servo test also results in control surface motions. These conditions prerequire that the aircraft have ground clearance for personnel safety reasons. Thus, this part of preflight will be done just before taxiing or during taxi out. At least in the

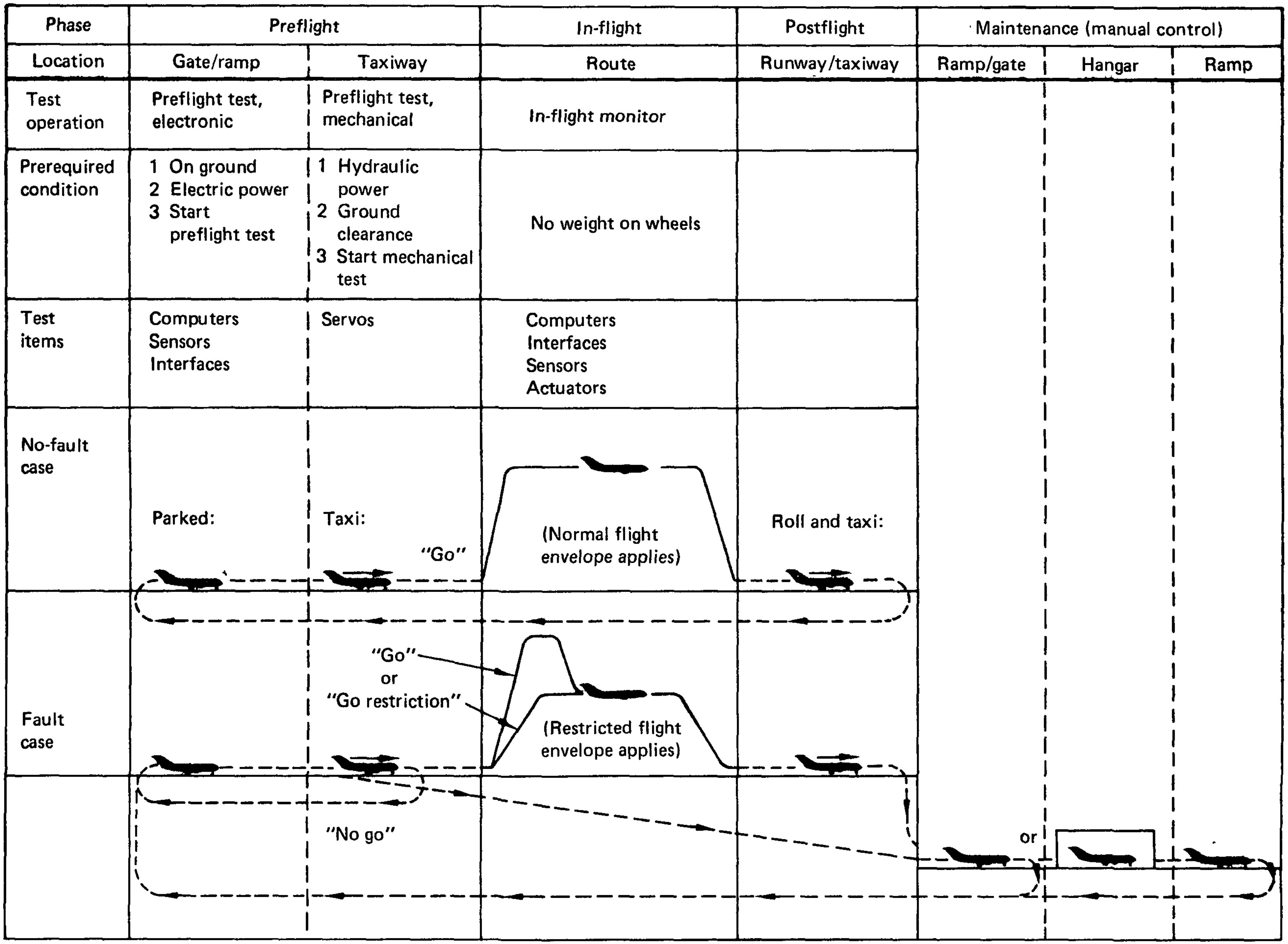


Figure 78. Test and Maintenance Overview

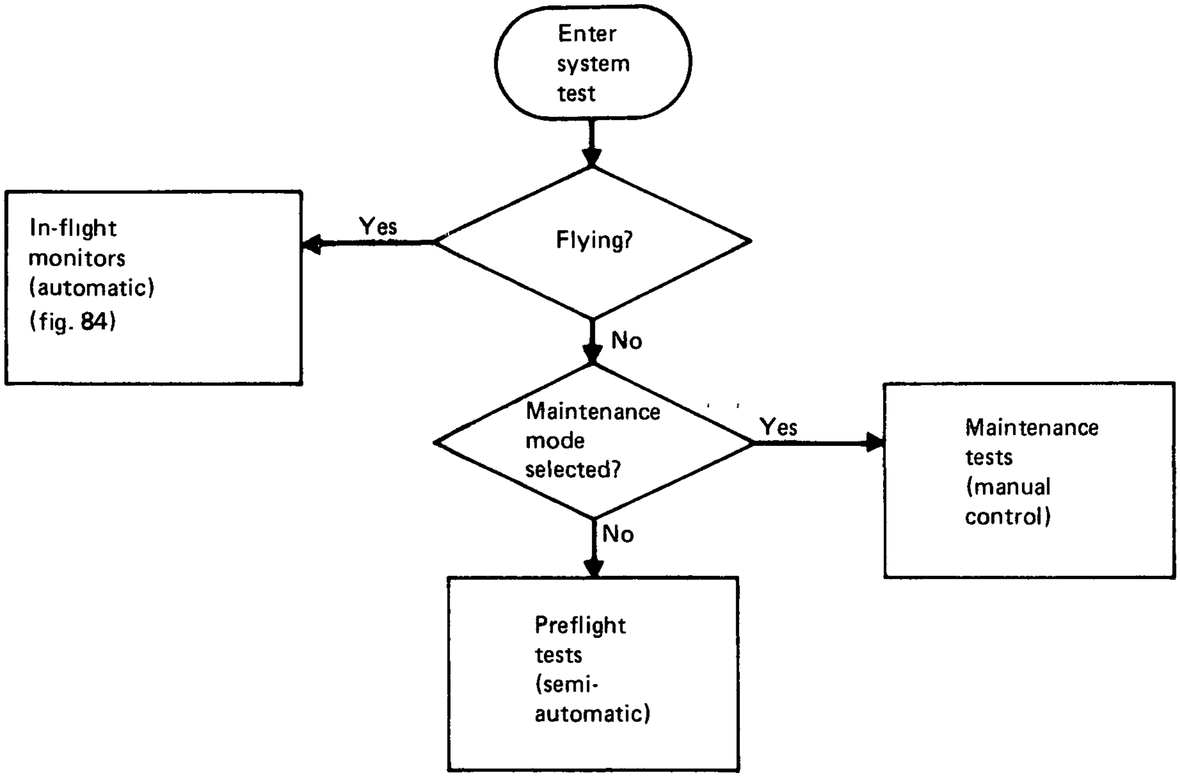


Figure 79. System Test Functional Flow

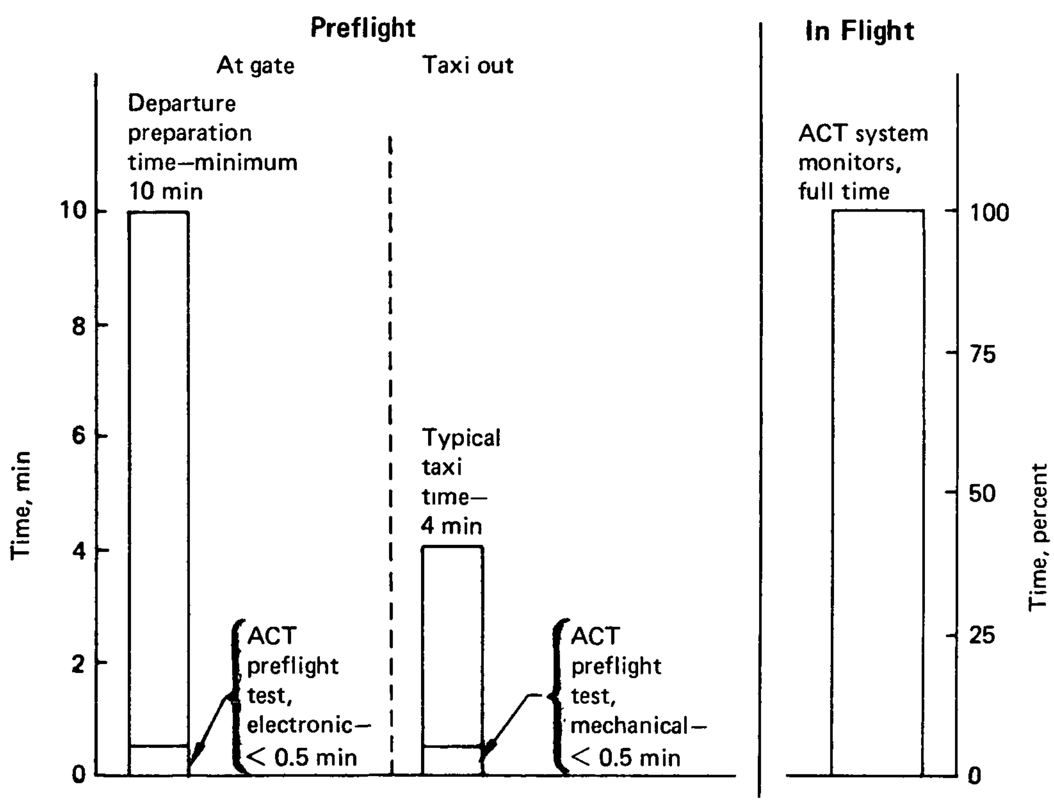


Figure 80. System Test Time Requirements

latter case, the duration of less than 0.5 min will not extend airplane preflight time. The last bar at the right in Figure 80 indicates the full-time character of the in-flight monitors.

8.4.2.1 Preflight Testing

The objectives of preflight testing are to:

- Provide a factual basis for the part of the pilot's dispatch decision representing the ACT control system

- Identify the failed LRUs if the ACT system indication is no go, and verify system integrity after replacement

The first objective requires that the control system operating status be made known to the pilot.

System faults determined during preflight test will have one of three results:

- No effect upon the flight plan
- Takeoff into a restricted flight plan
- "No go," dispatch denied

Fault information is presented to the flight crew only in the latter two cases. Subsection 8.4.4 details how the system and air crew respond to such cases.

Figure 17 shows the organization of the ACT control system as a whole, including the provisions for all six active control functions. Figure 81 is an idealized form of the prior diagram and will be used to illustrate the preflight test series. Each computer set performs the same test and monitor functions again with respect to its own input and output quantities. The one exception is the singular interface between the two computer sets; this must be treated as a special case.

Although the ACT Maintenance and Display Computer is not dispatch required, it can be undergoing test during the ACT preflight test period simultaneously with the two control

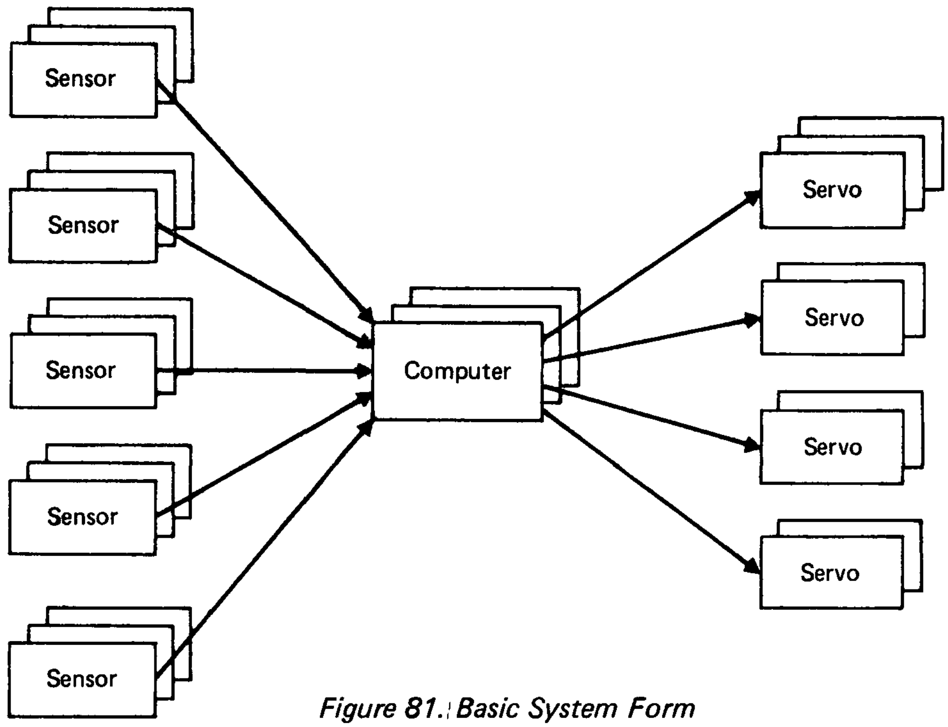


Figure 81. Basic System Form

computer sets. This saves time and allows the crew to know before takeoff the availability of the "automatic option" information output illustrated at 2 in Figure 41.

Preflight Test–Electronic–The basic scheme of the preflight test series is the "inside-out" plan, in which the computers first test themselves and then, employing their recently verified hardware and software, test the operation of other components and combinations. This sequence is illustrated in Figure 82. In the first preflight test phase, the computers will perform the following self-tests under software control:

- Program sum checks
- Hardware monitor test
- Digital receiver and transmitter test
- Scratch pad memory read/write test
- Discrete input/output test
- Analog input/output test
- Instruction repertoire

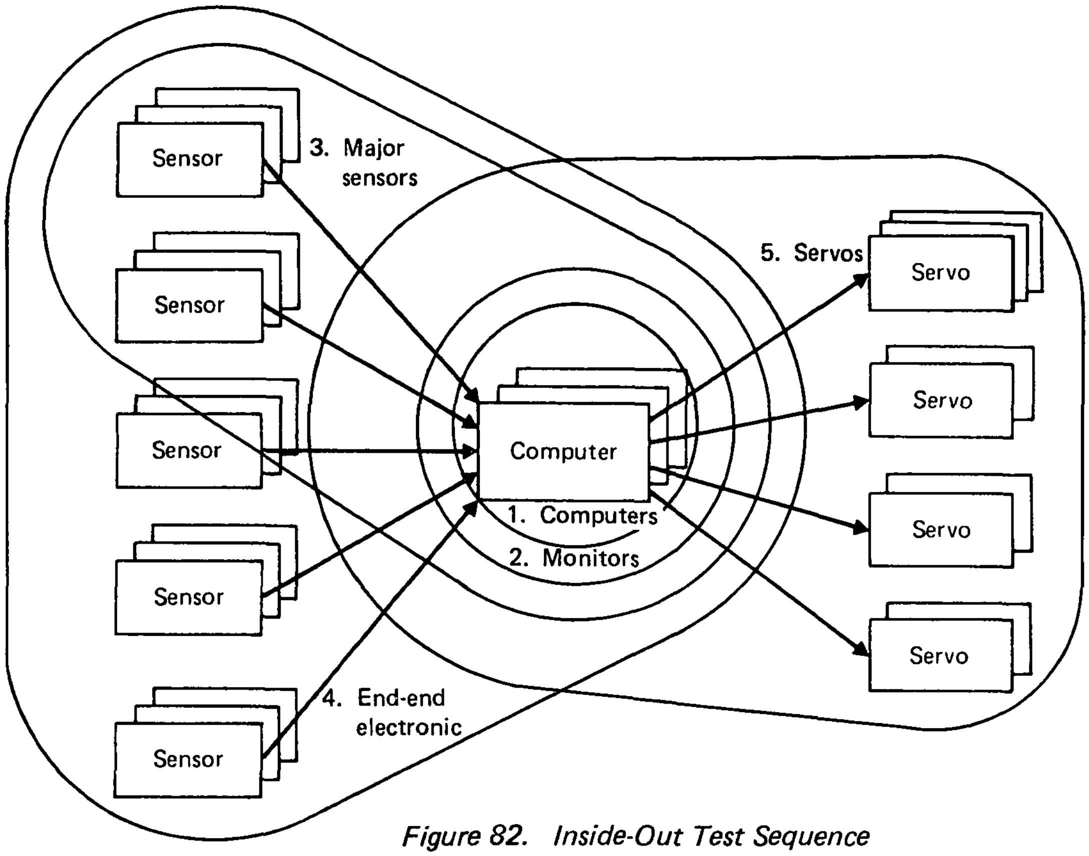


Figure 82. Inside-Out Test Sequence

The Essential PAS Computers will not require all the tests of this list; e.g., discrete input/output is not needed in these computers.

Next, the computers test the control system monitors as indicated at 2 in Figure 82. The three sets of cross-channel monitors are as follows:

- Signal selection and failure detection (SSFD) is the sensor comparison monitor. It selects a midvalue from among the sensor signals gathered by cross-channel communication and then derives and tests differences to detect sensor failure. Selected midvalues are input to the computer control law process.

- The computer output monitor compares the results of the multiple parallel control law computations that are performed based upon the one selected sensor signal. Differences detected there indicate faulty computer operation.

- The servomonitor is a computer operating on signals fed back from active control servos to the computer. In the redundant secondary servos, the second-stage valve spool position is sensed by a linear variable differential transformer (LVDT); after being digitized, these signals and the mathematical models of spool position are compared to enable detection of servo faults.

These monitors, illustrated in Figure 53, are used in flight operations as full-time overseers of system operation and detectors of local faults. They also serve the same functions in ground test phases; hence their proper operation must be verified early in the ground test sequence.

The foregoing tests will be run separately but simultaneously by the ACT Primary and Essential PAS Computers.

The "special case" cited early in this subsection is the test of the decision process in which the short-period pitch augmentation responsibility is transferred from the ACT Primary Computers to the Essential PAS Computers. As part of its software, the latter computer set has a continuous monitor function on the elevator command output of the primary computers. This midvalue logic cross-channel monitor must be tested using both computer sets; therefore, this test occurs at the end of the control system monitor testing.

Next, as indicated at 3 in Figure 82, the primary computers call for reports on the soundness of the two major sensor sets: the digital air data computers (DADC) and the inertial reference systems (IRS). Of the six active control functions, five are served by the DADCs and four by the IRSs. Thus, those two sets are the most complex and widely used sensors. Both have their own digital computers and their own self-test routines. In the preflight test, the ACT Primary Computers will call upon the DADCs and the IRSs to report individually upon their availability for service in the upcoming flight.

To complete the electronic preflight test, the ACT Primary and Essential PAS Computers perform an end-to-end electronic exercise of the various functions without actuator operation. This is shown at 4 in Figure 82 and at B in Figure 83. The airplane presumably still lacks ground clearance and hydraulic power; hence, there will be no actuator response. However, the servocommand output is still available for comparison with stored

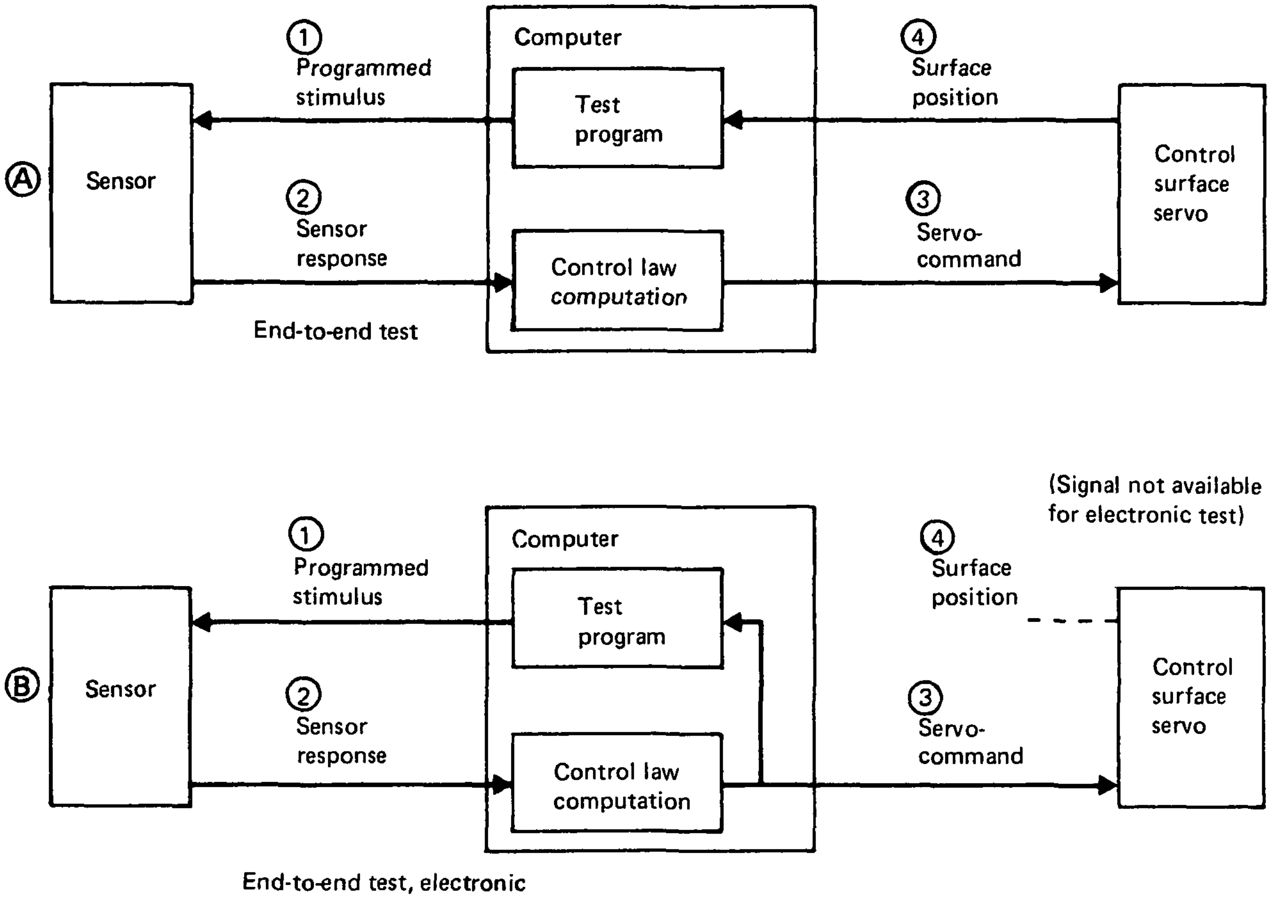


Figure 83. ACT Function Loop Tests

reference values. For the control laws for low-frequency ACT functions, a static output represents an adequate test criterion. For high-frequency mode control laws, the dynamic response must be observed to check for proper damping. This requires much greater test data and sampling rate than the static output case.

In operation, the test program injects a sensor stimulus to cause a sensor output that will be processed in the control law computation to yield a servocommand. An out-of-tolerance response by itself would indicate only a fault somewhere in the loop represented in the sketch. The fact that (1) three or four such loops are being run simultaneously and (2) that cross-channel comparisons are being done at SSFD and computer output monitors enable the test system to identify the faulty LRU, excepting, of course, the servo.

Preflight Test–Mechanical–This final part of the preflight routine, shown at A in Figure 83, tests the servoactuators. It requires hydraulic power and ground clearance; hence, it is normally performed during taxi out from loading gate to takeoff runway. Requiring confirmation of mechanical motions, these tests are much slower than the electronic steps. Therefore, they are performed in parallel, with computer commands synchronized to each servo. Entirely satisfactory servo response testing can be achieved using small excursion commands such as 20% of full-scale deflection. Using such commands limits the hydraulic power demand such that all ACT servos can be tested simultaneously as long as both engine-driven hydraulic pumps are operating.

The servomonitor and the ability to deactivate failed channels (functions illustrated in fig. 53) are also checked in the mechanical test phase.

8.4.2.2 In-Flight Monitors

ACT control system in-flight monitoring has these important objectives:

- Continuously check on system operations
- Detect and report failures
- Reconfigure the failure-affected function control chains to minimize redundancy loss

The in-flight monitors are shown in block diagram form in Figure 53. The monitor functional flow is illustrated in Figure 84, which, as shown in the small key diagram at the right, is a part of Figure 79. Figure 41 illustrates system response to in-flight faults as detected by the monitor network, the same as the preflight fault response activity.

As illustrated in Figure 78, at the moment of ACT airplane lift-off, the condition of "no weight on wheels" signals the start of ACT system operation and the beginning of routine in-flight monitor operations in the computer background mode. In the operating condition, the computer watchdog and overflow monitors operate continuously to guard against timing errors and overflow. Parity and sum checks are also cyclically performed. These are sources of information on the soundness of the operating computers.

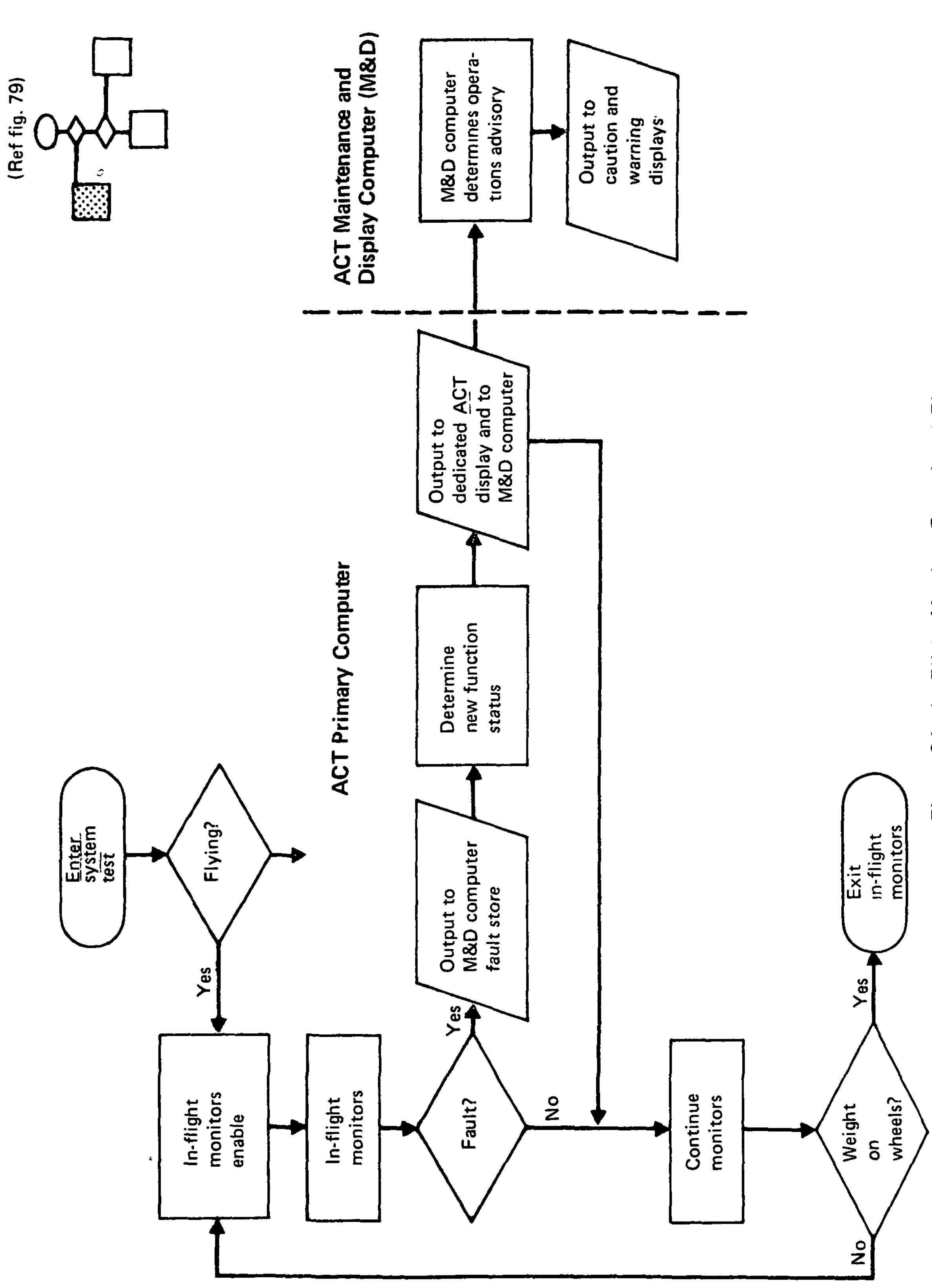


Figure 84. In-Flight Monitor Functional Flow

In this phase, sensors and servos cannot safely be given artificial stimuli. The major sensors, DADC and IRS, are within their computers performing continuous self-test and reporting their soundness via "status monitor" bits in each data word transmitted. This comes from inline testing and can be relied on for about 90% fault detection. Thus, continuous operation of ACT cross-channel monitors in flight is essential if the system is to achieve very high failure detection probability.

Information on an in-flight fault that would prevent next dispatch will be displayed to enable call ahead for maintenance action at the next stop. All fault information also is retained in the ACT Maintenance and Display Computer memory and maintained there for several flights in chronological order.

8.4.2.3 Maintenance Testing

The maintenance test plan for the active control system is based upon these concepts:

- Fault information for several flights is immediately available to the maintenance technician via the ACT Maintenance and Display Computer.

- Individual component and subsystem tests are run automatically under control of the ACT computers.

- Selection of tests is accomplished by the technician via the ACT Maintenance and Display Computer.

All tests of the preflight series are available individually for maintenance use. Those involving the DADC and the IRS require advance power up of those systems. In addition, the maintenance test spectrum includes individual sensor and servo tests and full end-to-end tests, as shown at A in Figure 83. In these, the computer test program sends an appropriate stimulus to the sensor or servo and compares the returned signal with its stored range of acceptable values.

Any of these tests is callable at will by the technician to enable (1) verifying stored fault data, (2) diagnosing failed functions, and (3) checking results of maintenance action.

8.4.3 CREW COMMUNICATION

The communication requirements of the active control system, both from system to crew and from crew to system, are described in terms of system operation; i.e., what is required of the crew to monitor and manage the ACT system in its almost entirely automatic operation.

8.4.3.1 Preflight Testing

Preflight testing required of the active control system occurs in two automatic sequences: electronic preflight and mechanical preflight. Although both test sequences proceed without crew intervention, the crew must select the time they are performed. Thus, there must be a "start test" switch for each of these two operations. These momentary contact pushbutton switches will be illuminated as they are pushed and the test begins. The light behind the button will be extinguished at completion of the test sequence.

The preceding information describes crew requirements for a normal preflight test in which no fault is found. Where faults are found, the communication requirement is not as simple. The system recognizes and stores information on faults of three levels:

- Faults that have no effect upon aircraft dispatchability
- Faults that allow dispatch but with some restriction to the flight plan
- Faults that prevent dispatch

Figure 41 illustrates response to faults detected in preflight test.

The first category of fault is not reported to the flight crew. If a fault found in preflight test affects dispatchability of the airplane, it is next necessary to determine the status of the ACT function or functions affected by the failure and display that status for the flight crew. At this point, the crew has the option of referring to an operations manual to determine the flight plan effect specified for the particular status. Alternatively, the crew can use the same information, determined as a flight advisory message by the active control system and displayed subsequent to the function status message. The pilot's decision may be made based upon the same information derived either way, from the manual or from the information display.

The operations are illustrated in Figure 41. System operations are shown at the top of the figure, and the pilot actions are represented in the lower portion. Communication from the control system to the crew shows in that diagram at two places indicated by the circled numbers 1 and 2. Function status is reported by the dedicated ACT discrete display; the operations advisory message is shown by the baseline caution and warning system caution annunciator display. The block diagram in Figure 40 illustrates this scheme. The two sets of active control computers determine function status as indicated in the top line in Figure 41 and then relay that information to the ACT Maintenance and Display Computer. This computer selects the corresponding flight advisory message, if any is required, and passes it to the caution annunciator display.

The preceding constitutes the primary mode of operation of system communications in the preflight test process. Only one ACT Maintenance and Display Computer is included in the system, and it is not required for dispatch. Therefore, there must be a backup communication alternative to determine dispatch information about the control system itself if the ACT Maintenance and Display Computer is not available. The provision for that is a set of dedicated active control system indicator lights, capable of displaying function status as represented at circled number 1 in Figure 41. Those indicators, combined with the operations manual, enable the dispatch decision to be made whether or not the caution annunciator display is showing information from the ACT Maintenance and Display Computer.

The only additional crew communication provisions required are the active control function disconnect switches. These switches are accompanied by reconnect switches to accommodate possible erroneous disconnect.

All crew communication provisions represented in the foregoing discussion are shown in Figure 85 and are intended to illustrate requirements rather than design selection. The concept represented there is based upon these assumptions:

a. Fail-operative communication from the ACT system to the flight crew can be achieved by using the baseline caution and warning system plus dedicated discrete indicators showing the status of all ACT functions.

	Function	Indicators (1)	Switches (momentary contact, illuminated)	Remarks
Manual	Start preflight, electronic	→	Electronic Test	(A) Switch illuminated during test, off at end
	Start preflight, mechanical	→	Test Mechanical	See (A) above
Active control	PAS $_{SHORT}$	(4) Red Divert		Requires immediate diversion, function is never disconnected
	PAS $_{SPEED}$	Marg (2) Amber Inop Amber (4)	Off (3) On	(B) Switch pushed to disable function remains illuminated as long as function is disconnected
	LAS	Marg Amber Inop Amber (4)	Off On	See (B) above
	WLA	Marg White Inop Amber (4)	Off On	See (B) above
	FMC	Inop Amber	Off On	See (B) above (C) "Marg" is not significant
	Stall warning and AAL	Inop Amber	Off On	See (B) above See (C) above
	Failure information display	Master indicator: Warning Red, Caution Amber Each end of glare shield (Alphanumeric color CRT, 11 lines, 16 characters/line) Cancel BRT Recall	Caution annunciator display 1 Pilot's main panel 2 Flight engineer's panel	Annunciators and master indicators are parts of baseline caution and warning system Annunciator will display (1) Failed dispatch-required LRU (2) Operations advisory messages

Notes
(1) If flaperons are required, flaperon position indication must be provided
(2) Marg = one failure away from function loss Inop = function inoperative, auto or manual disconnect
(3) Disconnect switches (five) allow manual disconnect of individual functions "On" switches enable reconnecting functions
(4) Signaled by PAS $_{SHORT}$ lost or combination of PAS $_{SPEED}$, LAS, and WLA lost

Figure 85. ACT Cockpit Controls and Displays Requirements

b. In the primary operating mode, the caution annunciator display will report all system information essential to the flight crew. This includes:

- Identification of LRU failures if they are required for dispatch
- Any change in flight plan made necessary by system failures

c. In the backup mode for the loss of the caution annunciator panel, the crew will use the dedicated discrete status indicators. Flight manual data will be provided to interpret those indicators in terms of required operations changes.

d. Manual controls will be provided only for:

- Initiating preflight tests
- Emergency disconnect

In normal operation, system failures are followed by automatic reconfiguration or automatic function disconnect.

e. Information displayed by the caution annunciator will be supplied by the ACT Maintenance and Display Computer. Backup discrete displays, both ACT dedicated and caution and warning system discrete indicators, will be signaled directly from the ACT control computers.

f. Level A warnings (requiring immediate action) and level B cautions (crew attention and future action required) issued from the ACT system will activate the pilots' master lights on the glare shield.

g. Dedicated ACT indicators and controls will appear on a dedicated ACT panel.

In Figure 85, under the heading "Function," are the two "start test" operations needed for preflight test and the six active control functions. These individual items require the discrete indicators and controls appearing in the next two columns. Everything above the double line is there specifically for the associated functions. Below the double line is the baseline caution and warning system, which shows the items listed under assumption b. This system also provides crew communication for other onboard systems.

The logic of assumption b is that different kinds of failures in the ACT control system have widely varying consequences. They may:

- Have no effect upon operations
- Have no effect upon flight in progress, but prevent dispatch
- Require change in current flight plan

For the first failure category, no information is needed by the crew. When the second category is encountered in flight, corrective maintenance must be done at the next way station. This will often require active preparation before the airplane lands, to the extent of flying in parts from another base. Such a situation would need the maintenance call-ahead function by the flight crew, and for that, identification of the failed component must be announced. For the last case, the crew must be notified of the flight plan change.

8.4.3.2 In-Flight Monitors

Operation of the active control system in-flight monitor and self-test functions is completely automatic. The response of the system and flight crew to faults that these monitors discern in flight is again illustrated in Figure 41 and is almost entirely parallel to the system test and communication operations of the preflight period. The three response levels to faults required of the system and crew are as follows:

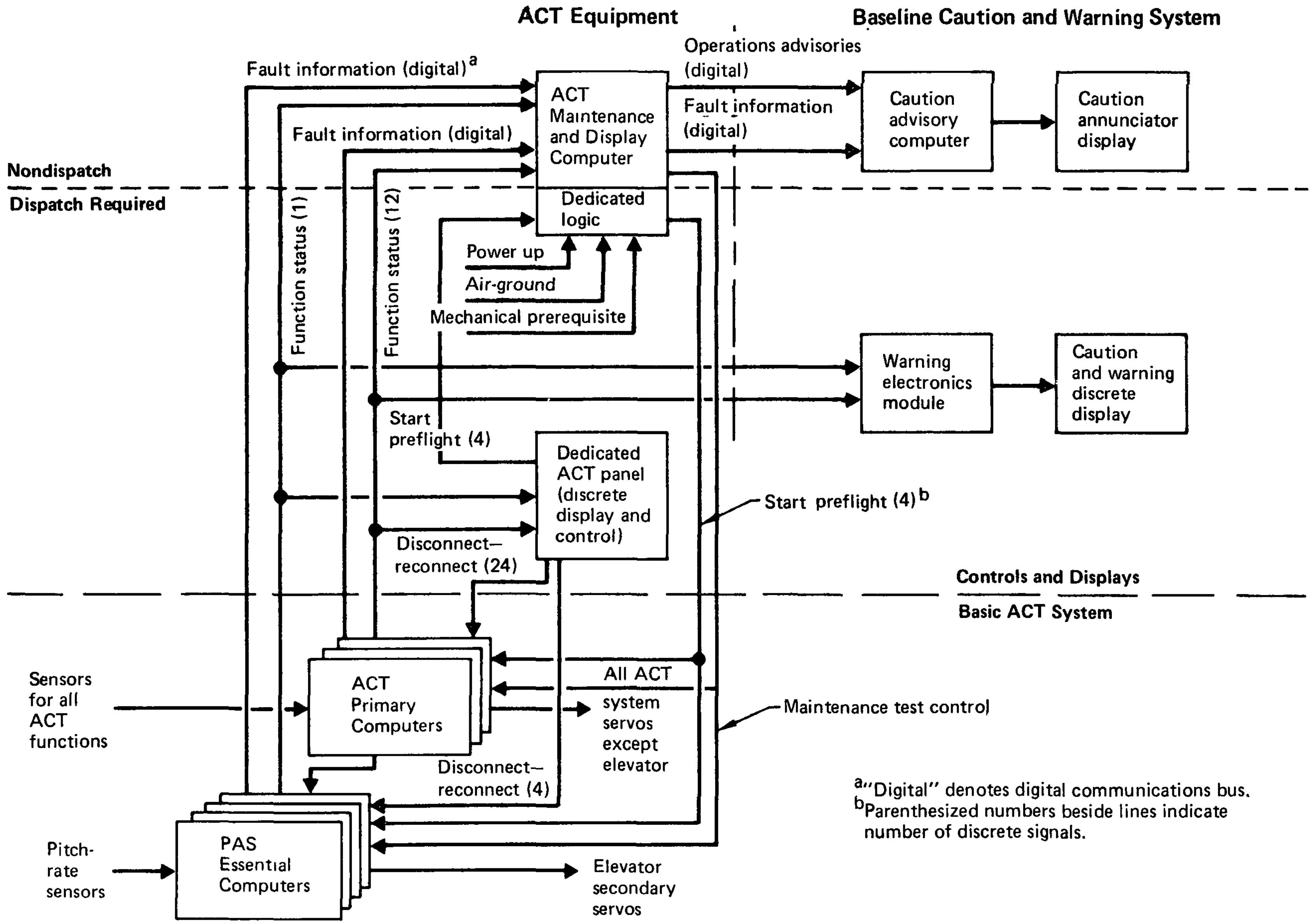
- For a fault that does not affect flight operations, the system merely stores the component fault data; no notification is provided to the flight crew.

- A system fault that does not affect current flight operation but will prevent the next dispatch must be identified so that the crew can arrange corrective maintenance at the next airport.

- A fault that changes the function status so as to require a change in flight plan must be identified and the flight plan change announced to the crew.

The responses just enumerated represent the normal mode of operation of the system when in-flight failures occur. Where the ACT Maintenance and Display Computer and caution and warning system chain is not operating, communication depends upon the set of discrete function status lights. Here again, as shown in Figure 41, the essential information is supplied by the discrete indicators and operations manual, enabling the pilot to determine proper response to the faults through the use of the manual. This dual display capability is shown in Figure 86, a more detailed version of Figure 40.

A special case of in-flight change of plan is represented by the "Divert" indicator shown in Figure 85 in the short-period pitch-augmentation function line. The appearance of this red light, accompanied by the master warning lights on the glare shield and a warning aural signal, calls for immediate diversion of the flight to a landing at the nearest adequate runway. This emergency condition is produced either by reduction of the crucial pitch-augmentation function to only two success paths or by loss of the combination of three critical functions (speed PAS, LAS, and WLA), as defined in Tables 15 and 17.



a"Digital" denotes digital communications bus.
bParenthesized numbers beside lines indicate number of discrete signals.

Figure 86. ACT Displays and Controls

8.4.4 FAULT RESPONSE

The fault response function in the ACT control system uses failure information derived by the control computers to first determine the status of active control functions and then, from that information, to determine the correct change in operations, if any is required, and to communicate that to the pilots. This function must not be confused with the automatic response of the ACT computers to reconfigure active control functions after component failures. That operation is described in Subsection 8.2.

The basic scheme in this fault response technique may be represented by the following sequence:

Component failure(s) → Logic A → Function status → Logic B → Advisory message

Thus, when a component failure is detected in the active control system, it is entered into logic process A and the resulting output will be the operating status of the ACT function or functions affected. Similarly, that information injected into logic process B will result in output of the proper operations advisory message if any is needed. Logic A is implemented in the control computers, ACT Primary and Essential PAS; and logic B is implemented in the ACT Maintenance and Display Computer. This process is represented in Figure 41. There, where the control system determines function status, logic A is being applied; and, in the box labeled "Selects Flight Advisory Message," logic B is being implemented. As described in the preceding subsection, the ACT Maintenance and Display Computer is simplex equipment not required for dispatch; therefore, occasionally the ACT airplane will be operating without that computer. In that instance, logic A and the sequence indicated by the circled number one in Figure 41 apply, using the crew's reference to an operations manual.

The simpler step shown in the preceding paragraph is logic B. This logic is expressed in the Boolean equations shown in version 1 of logic B, as listed in Table 18. Those equations are derived from the rules for dispatchability and operability of the ACT airplane as shown in Tables 15 and 16. Those tables, involved as they are, can be reduced to version 1 of Table 18 with no assumption on the character of flight restrictions that might result from failure of various ACT functions and function combinations. Using a reasonable

Table 18. Selection of Advisory Messages From ACT Function Status (Logic B, Boolean Equations)

Key:

First letter	Second letter
P = PAS_{SPEED} A = AAL L = LAS W = WLA F = FMC	L Function lost 0 One failure away from function lost Exceptions: PS PAS_{SHORT} PSN PAS_{SHORT} $\lambda > 1 \times 10^{-9}$/hr PSO PAS_{SHORT} < three success paths

Version 1: No assumptions on character of flight restrictions

Advisory message	Conditions requiring message: At dispatch	Conditions requiring message: In flight
Restriction A	PL + PO	PL + PO
Restriction B	LO	LL + LO
Restriction C	FL	FL
Restriction D	(PL + PO)•LO	(PL + PO)•(LL + LO)
Restriction E	(PL + PO)•FL	(PL + PO)•FL
Restriction F	LO•FL	(LL + LO)•FL
Restriction G	(PL + PO)•LO•FL	(PL + PO)•(LL + LO)•FL
NO GO	PSN + AL + LL + WL + [(PL + PO)•(LL + LO)•(WL + WO)]	
DIVERT		PSO + [(PL + PO)•(LL + LO)•(WL + WO)]

Version 2: Assuming flight restrictions,and function status scanned in this order: PS, A, W, L, P, F

Advisory message	Conditions requiring message: At dispatch	Conditions requiring message: In flight
Restriction B	LO	LL + LO
Restriction A (if B does not apply)	PL + PO	PL + PO
Restriction C (if neither B nor A applies)	FL	FL
Dispatch "NO GO" and in-flight "DIVERT" are the same as in version 1.		

configuration for the ACT airplane and assuming reasonable flight restrictions based upon current commercial jet transport practice, the number of flight restrictions applicable to these operations is reduced to only three plus the "Divert" command. With those additional assumptions, the Boolean equations for logic B reduce to the very simple form shown as version 2 in Table 18. Because the current technology control system work assumes no specific airplane configuration, version 2 does not represent a segment of the selected ACT configuration description. It does, however, represent a probable simplification of the version 1 list of equations when they are actually applied to a real airframe. Such determination of an actual airframe and actual logic B equations will be done in ensuing stages of the IAAC Project.

Logic A Boolean equations are less simple. They must accept the output of the entire fault determination process in both sets of active control computers operating on all components of the ACT system. With such a large number of elements as input items, the logic A equations become very long. One of the shorter equations, which determines flutter-mode control failure, has 60 terms in 15 variables. Including both function failure conditions and conditions where a function or a combination of functions is "one failure away from function loss," there must be 12 such logic A equations, some of them much larger than the sample described. Although these equations are subject to reduction techniques normal in Boolean algebraic operations, they will still require greater computer capacity than is needed for logic B. The derivation, reduction, and implementation planning for logic A equations will be subjects of the next phase of IAAC control system work.

8.5 HYDRAULIC SYSTEM

8.5.1 BASELINE CONFIGURATION

The general arrangement of the hydraulic power system is shown in Figure 87. The system is very similar to that of the Baseline Airplane.

The Baseline hydraulic power generation system consists of three continuous-duty 20 700-kPa (3000-lbf/in^2) systems that use phosphate ester fluid. The systems are identified as A, B, and C. In systems A and C, hydraulic power is generated by an engine-driven pump (EDP) installed parallel with an electric-motor-driven pump (EMP). System B hydraulic power is generated by two ac EMPs and one air-turbine-driven pump (ATDP). The bleed

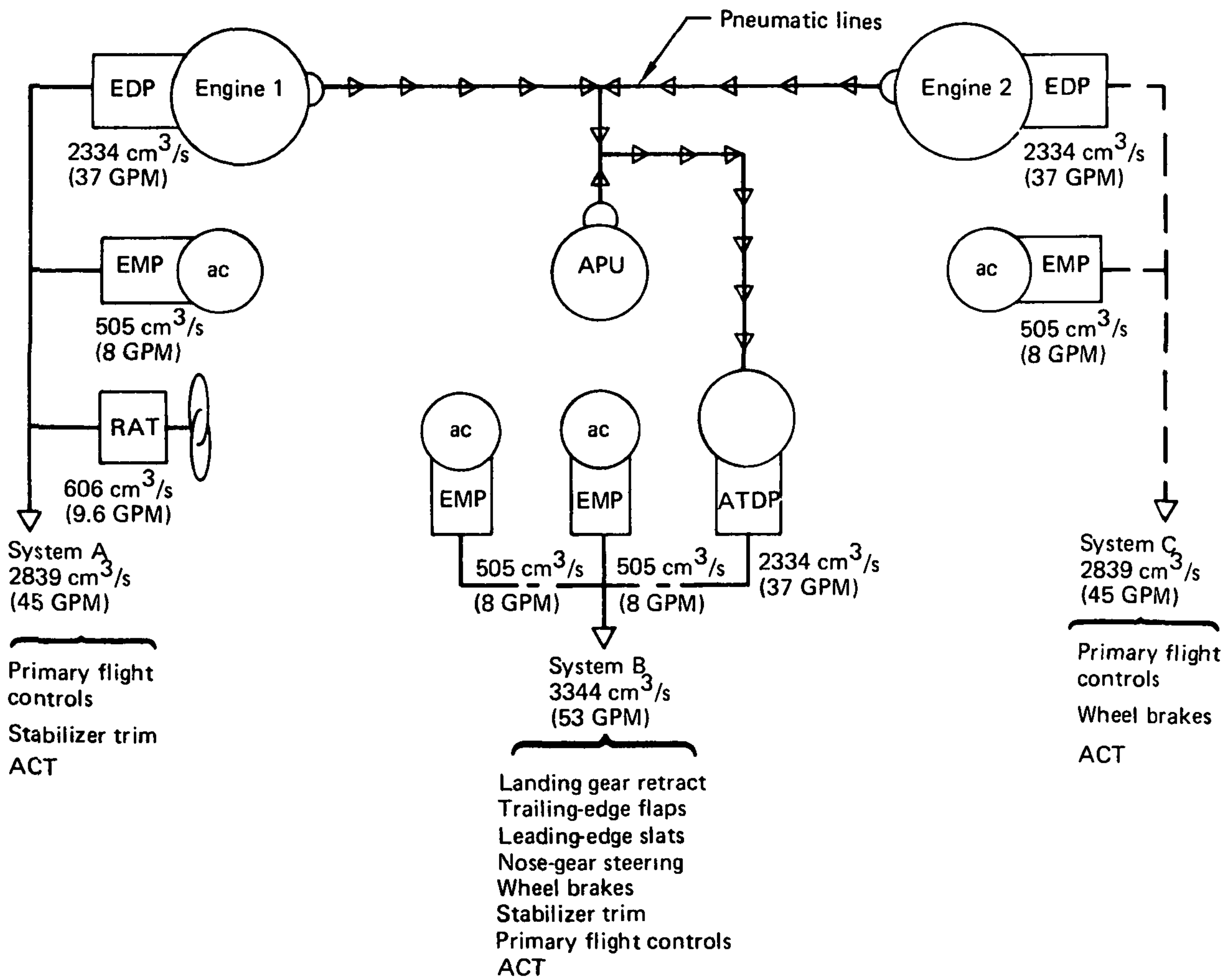


Figure 87. Baseline Hydraulic Power System

air manifold serves as the pneumatic source. Emergency hydraulic power is derived from the wind-milling engines rotating the EDPs. System A is augmented by a ram-air-turbine-driven (RAT) hydraulic pump. Ground hydraulic power is available from the ATDP, powered by the auxiliary power unit (APU) or pneumatic ground cart; the ac pumps, also energized by ground cart or APU; or by an external hydraulic power supply source. Flight deck controls and displays consist of depressurization switches for the EDP, shutoff switches for the ATDP and the EMPs, low-pressure and low-fluid warning lights, and selectable readout for system pressure and fluid quantity.

8.5.2 MODIFICATIONS FOR ACT CONFIGURATION

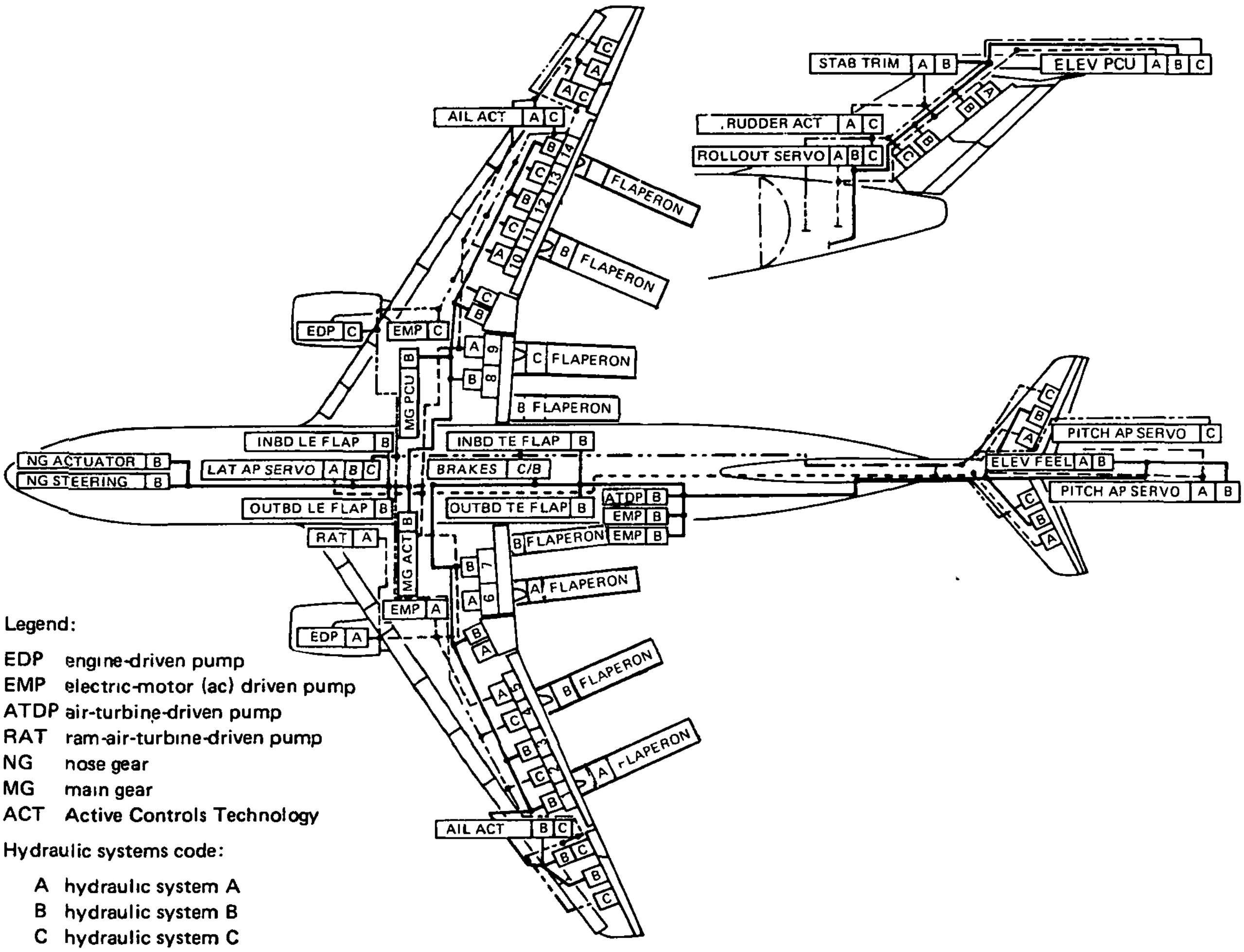
The maximum additional hydraulic flow for each of the three ACT hydraulic systems is as follows:

- 126-cm^3/s (2-GPM) (additional servovalves) leakage flow
- 126-cm^3/s (2-GPM) flow due to the higher rates for the ACT outboard ailerons (inboard and outboard segments)
- 63 cm^3/s (1 GPM) for secondary ACT actuators
- 126 cm^3/s (2 GPM) to drive outboard flaperons
- 315 cm^3/s (5 GPM) to drive inboard flaperons

There is no difference in the hydraulic demand for the ACT versus Baseline yaw and pitch functions.

A hydraulic flow load analysis indicates that, if the flaperons are excluded, all the other ACT functions can be accommodated by the Baseline system. When inboard and outboard flaperons are included in the ACT configuration, the 2334-cm^3/s (37-GPM) capacity of the three hydraulic pumps has to be increased by 20%, resulting in a total hydraulic system weight increase of approximately 4.5 kg (10 lb).

The ACT technology base configuration hydraulic distribution system is shown in Figure 88.



Legend:

EDP engine-driven pump
EMP electric-motor (ac) driven pump
ATDP air-turbine-driven pump
RAT ram-air-turbine-driven pump
NG nose gear
MG main gear
ACT Active Controls Technology

Hydraulic systems code:

A hydraulic system A
B hydraulic system B
C hydraulic system C

Figure 88. ACT Hydraulic System Distribution

8.6 ELECTRIC SYSTEM

Redundant crucial and critical systems, such as the active controls system, require power sources with the same degree of redundancy to avoid losing more than one channel if a single power source fails. Furthermore, reliability of the power sources must be significantly better than the power utilization systems if the overall reliability is not to be compromised by the power sources.

The electric system for the Baseline Airplane is discussed in Subsection 8.6.1; the modifications necessary for IAAC are described in Subsection 8.6.2.

8.6.1 BASELINE CONFIGURATION

8.6.1.1 Primary Electric Power

The Baseline and ACT electric systems are shown in Figure 89. Primary three-phase, 115V, 400-Hz power is supplied by two engine-driven 90-kVA integrated drive generators (IDG) that cannot be paralleled, so the system operates as two separate isolated channels. A third 90-kVA APU-driven generator is provided for ground maintenance operations and as an in-flight backup for the two main engine-driven generators. The APU can be started at any altitude up to 7620m (25 000 ft) and can provide full electric power up to 10 670m (35 000 ft). The APU generator control unit is interchangeable with those used for the engine-driven generators. Any single generator can supply all flight-essential loads. Two of the three generators must be operative for airplane dispatch with no load reduction or for a category III landing.

During ground operations, electric power can be provided from either the APU generator or from a ground power cart through the 90-kVA external power receptacle. Ground power can be used to energize all main power buses or only those electric loads required for normal maintenance, servicing, and cargo handling. On the ground or in flight, utility and galley loads will be automatically shed when the system is overloaded.

Airplane 28V dc power is provided by two 120A unregulated transformer-rectifier (T-R) units. Each of the two main ac buses supplies its own T-R unit. The dc system operates

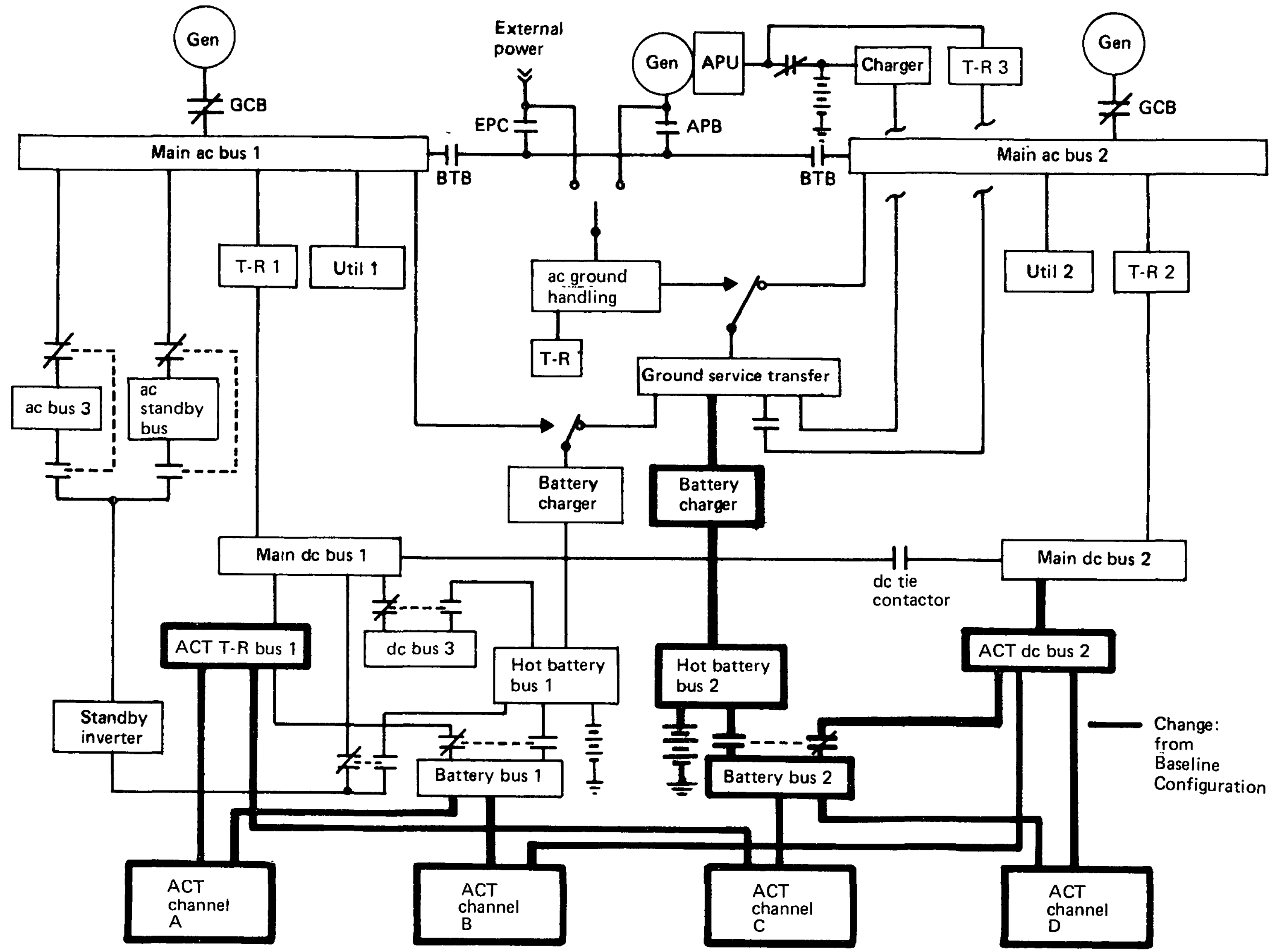


Figure 89. ACT Airplane Electric Power System

isolated only. If a T-R unit fails, a dc bus tie contactor enables the remaining T-R unit to supply both main dc buses. During ground operation, a 20A T-R unit provides dc power for ground handling loads.

A T-R unit is provided for normal starting of the APU when ac power is available from either the main generators or external power. When ac power is not available, a dedicated APU battery is used; a dedicated APU battery charger operates from either the main buses or external power.

8.6.1.2 Standby Electric Power

Backup power to flight-critical loads is supplied by a 40-Ah nickel-cadmium battery and a 1000-VA static inverter. A battery charger provides controlled recharge of the battery and operates as a T-R unit to supply the standby loads if the main dc source is lost but ac power is still available. Standby bus transfer is automatic.

The third power source for the category III autoland system is provided by the standby system. During category III landing, third-channel autoland dc loads will be supplied by the standby battery charger in the T-R mode. Autoland ac loads will be supplied from the standby inverter.

8.6.2 MODIFICATIONS FOR ACT CONFIGURATION

The electric system for the ACT Configuration is shown in Figures 89 and 90. A second standby battery, with charger, with the same capacity as the Baseline Configuration standby battery is added to the Baseline electric system to provide the necessary redundancy to support the ACT system. The standby battery capacity requirement for the ACT airplane is approximately double that in the Baseline Airplane. Therefore, the additional standby battery has the same capacity as that in the Baseline Airplane. Battery 1 supplies channels A and B, and battery 2 supplies channels C and D. For dissimilar redundancy, no two buses share the same T-R and battery. Thus, ACT T-R 1 supplies buses A and C, while T-R 2 supplies buses B and D (fig. 89). It is assumed that the Baseline Airplane standby battery load can be redistributed between the two ACT system battery buses to place about the same load on each battery. The individual T-R 1 and battery loads are shown in Table 19.

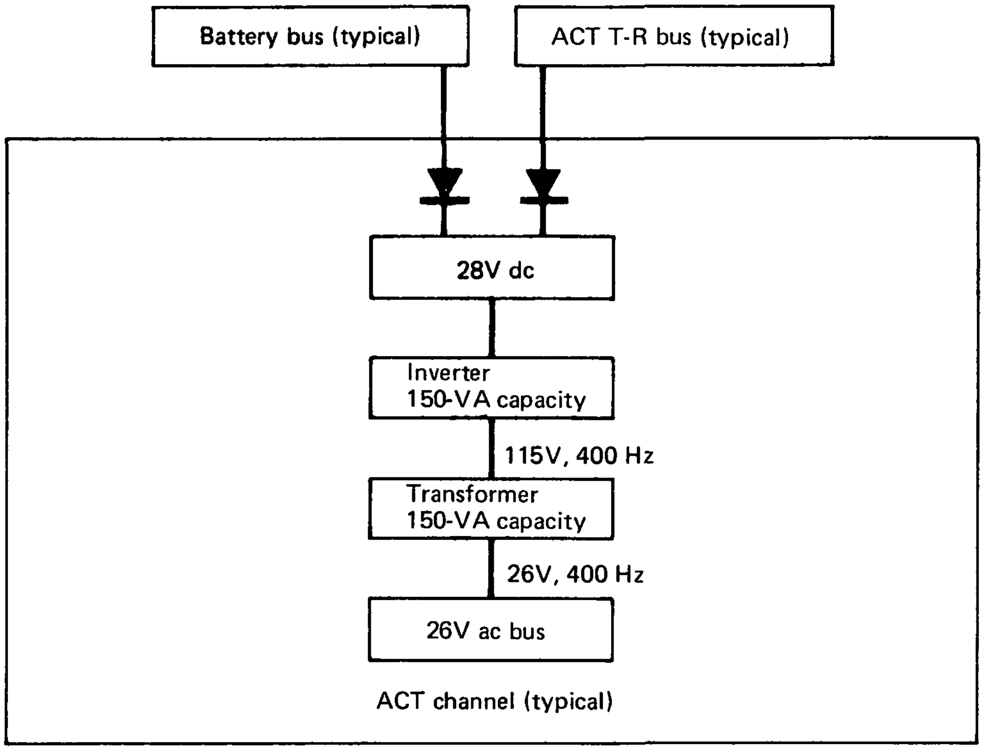


Figure 90. Detail of ACT Channel Power Supply (Typical)

Table 19. Individual Transformer-Rectifier and Battery Loads

Power supply	Channels	Amperes		
		ACT system	Baseline	Total
T-R 1	A and C	39.3	40.8	80.1
T-R 2	B and D	36.1	43.5	79.6
Battery 1	A and B	39.0	18.4	57.4
Battery 2	C and D	36.4	22.9	59.3

Assuming the total ACT system battery load and the Baseline standby loads all are supplied by the batteries for 30 min during emergency operation, the battery energy requirements are:

- Battery 1 57.4A x 0.5 hr = 28.7 Ah
- Battery 2 59.3A x 0.5 hr = 29.7 Ah

The Baseline Airplane standby and APU starting batteries are 40-Ah units. To maintain commonality between standby and APU batteries, the same 40-Ah battery is recommended for the ACT airplane.

The T-R capacity to supply the ACT system equipment is too small to justify additional dedicated T-Rs. The Baseline Airplane T-Rs, however, do not have growth potential to absorb the ACT system loads and still retain sufficient reserve capacity to supply all loads with one T-R inoperative. Therefore, the modification is to replace the baseline T-Rs with larger units (150A) and supply the ACT system T-R buses from the main dc buses. In effect, the ACT system T-R buses are extensions of the main buses (fig. 89).

The load totals for each power source are summarized in Table 20. Tables 21 and 22 list the ACT system dc and ac bus loads, respectively. Table 23 shows electric system equipment from Baseline specifications that satisfies ACT system needs.

Table 20. ACT Load Totals for Each Power Source

T-Rs operating			T-Rs not operating			
TRU	Channel	A	Battery	Channel	A	Ah*
1	A and C	80.1	1	A and B	57.4	28.7
2	B and D	79.6	2	C and D	59.3	29.7

*30 min

Table 21. ACT Direct-Current Bus Loads (Amperes)

Load unit		Current per unit		Units per channel				Current per channel				Comments
								Maximum steady state				
		Maximum steady state	Maximum surge	A	B	C	D	A	B	C	D	
MCP-701A computer		3.6		1	1	1	–	3.6	3.6	3.6	–	
Control/display panel		0.5		1	1	1	1	0.5	0.5	0.5	0.5	One panel with four channels
Elevator servo bypass solenoid		0.6 (hold)	1.0 (pull-in)	–	1	1	1	–	0.6	0.6	0.6	Mathematical model in D channel
Failure warning display		0.1		1	1	1	1	0.1	0.1	0.1	0.1	One panel with four channels
303T rear-spar accelerometer		0.04		1	1	1	–	0.04	0.04	0.04	–	Three each wing
303T front-spar accelerometer		0.04		1	1	1	–	0.04	0.04	0.04	–	Three each wing
Outboard aileron (inner segment) bypass solenoid		0.6 (hold)	1.0 (pull-in)	2	–	2	–	1.20	–	1.20	–	Two each wing, mathematical model in D channel
Outboard aileron (outer segment) bypass solenoid		0.6 (hold)	1.0 (pull-in)	2	2	–	–	1.20	1.20	–	–	Two each wing, mathematical model in C channel
Stick pusher solenoid (AAL actuator)		0.6 (hold)	1.0 (pull-in)	1	1	1	1	0.6	0.6	0.6	0.6	
Rudder bypass valve solenoid		0.6 (hold)	1.0 (pull-in)	1	–	1	–	0.6	–	0.6	–	
Flaperon bypass valve solenoid	Outboard	0.6 (hold)		–	2	2	–	–	1.20	1.20	–	
	Inboard			2	2	–	–	1.20	1.20	–	–	
PAS backup computer		3.6		1	1	1	1	3.60	3.60	3.60	3.60	
Dedicated pitch-rate sensors		0.03		1	1	1	1	0.03	0.03	0.03	0.03	
ACT Maintenance and Display Computer		3.6		–	–	–	1	–	–	–	3.60	
Total dc				–	–	–	–	12.71	12.71	12.11	9.03	

Table 22. ACT Alternating-Current Bus Loads (Amperes at 26V)

Load unit		Current per unit: Maximum steady state	Current per unit: Maximum surge	Units per channel: A	B	C	D	Current per channel, Maximum steady state: A	B	C	D	Comments
Elevator servo LVDT		0.25		2	4	4	4	0.50	1.00	1.00	1.00	Only a mathematical model in one channel
Outboard aileron servo LVDT	(Outer segment)	0.25		4	4	–	2	1.00	1.00	–	0.50	
	(Inner segment)	0.25		–	2	4	4	–	0.50	1.00	1.00	
Flaperon servo LVDT	Outboard	0.25		2	–	4	4	0.50	–	1.00	1.00	
	Inboard	0.25		4	4	2	–	1.00	1.00	0.50	–	
Rudder servo LVDT		0.25		2	–	2	1	0.50	–	0.50	0.25	
Total transformer load at 26V ac (amps)								3.50	3.50	4.00	3.75	
Transformer load at 115V ac								0.79	0.79	0.90	0.85	
Transformer output, VA								91.0	91.0	104.0	97.5	
Transformer losses, VA (at 80% efficiency)								22.8	22.8	25.2	24.4	
Transformer input, VA (= inverter output)								113.8	113.8	130.0	121.9	
Inverter input, VA (at 60% efficiency)								189.6	189.6	216.7	203.2	
Inverter input (amps at 28V dc)								6.77	6.77	7.74	7.26	

Table 23. ACT Power Supply Equipment

Item	ACT		Baseline	
	Number	Rating	Number	Rating
Battery[a]	2	40 Ah	1	40 Ah
Battery charger[a]	2		1	
Transformer-rectifier	2	150A	2	120A
Static inverter	4	100 VA		
Transformer, 115/26V	4	150 VA		
Battery-switching relay	b			

[a] ACT same as Baseline except quantity.
[b] Weight and cost insignificant compared to major equipment.

9.0 SELECTED SYSTEM EVALUATION

9.1 LABORATORY PERFORMANCE TESTS

This section describes several laboratory experiments made using a typical digital flight control computer (the General Electric MCP-701A) to examine potentially critical mechanization questions. The results were encouraging in that no significant problem in achieving digital mechanization of the Active Controls Technology (ACT) control laws was found.

9.1.1 CONTROL LAW IMPLEMENTATION EVALUATION

The ACT control functions were designed to be implemented using currently available hardware technology. With the application of digital computers to airborne flight control systems, such control functions will be implemented in a sampled-data environment. Control laws for these functions were developed in the continuous domain using root locus synthesis and analysis techniques, and the same functions were to be translated into a sampled-data representation. Some discrepancies in digital system performance were expected because of either the programming technique used or associated hardware design aspects such as sample rate, analog input prefiltering, and computation delay. Laboratory tests were conducted to evaluate the performance of four representative ACT control laws implemented using the General Electric MCP-701A digital flight control computer. Digital control law implementation was developed beginning with an analysis of the digital system basic hardware performance characteristics. Next, the control laws were implemented and evaluated from a frequency response performance viewpoint.

9.1.1.1 Laboratory Setup

The basic hardware characteristics of the General Electric MCP-701A were investigated using a single channel of the quadruple-channel system. Laboratory tests were conducted to establish the hardware input/output bandpass characteristics.

The basic General Electric MCP-701A hardware can be represented as a prefilter, computation delay, and zero-order hold, as shown in Figure 91. The prefilter attenuated the analog input signal high-frequency components to suppress "aliasing" during the

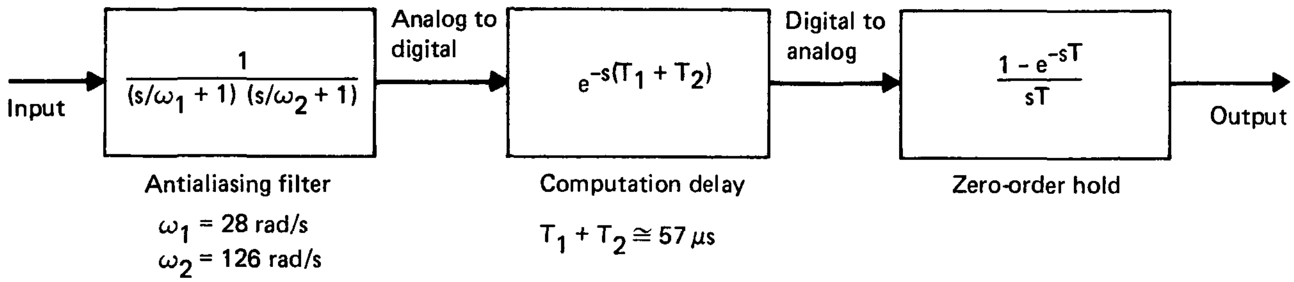


Figure 91. Mathematical Model of Laboratory Computer

analog-to-digital (A/D) conversion. Aliasing is the characteristic of an A/D process in which a high-frequency analog signal is sampled (any signal frequency above one-half the sampling rate) that results in false low-frequency signal in the discrete time domain. The laboratory computer prefilter was mechanized by two single-lag filters with fixed break frequencies of 28 and 126 rad/s as shown in Figure 91. Computation delay is made of two parts, the A/D conversion cycle time, T_1, plus the time, T_2, required to execute the program and produce a result. For the MCP-701A, the value of $T_1 + T_2$ was determined to be approximately 57 μs, which is negligible relative to the sampling rate and antialiasing filter contributions. The zero-order hold function shown in Figure 91 is the mathematical expression for the computer sample-hold characteristic; its time constant is the sampling period T.

Frequency response of the basic General Electric MCP-701A hardware was obtained in the laboratory to compare the mathematical model with laboratory results. Figure 92 presents the Bode plot of the laboratory frequency response and mathematical model for four different sampling intervals of 5, 10, 20, and 40 ms. The plots show that the mathematical model represents the actual response very closely. It further shows that the predominant frequency response characteristic of the hardware comes from the prefilter element. These results show that the digital system basic hardware characteristics can be mathematically modeled for analysis purposes.

9.1.1.2 Control Law Implementation

Four representative ACT control laws were implemented using the laboratory computers for system performance evaluation. The evaluation was limited to system dynamic

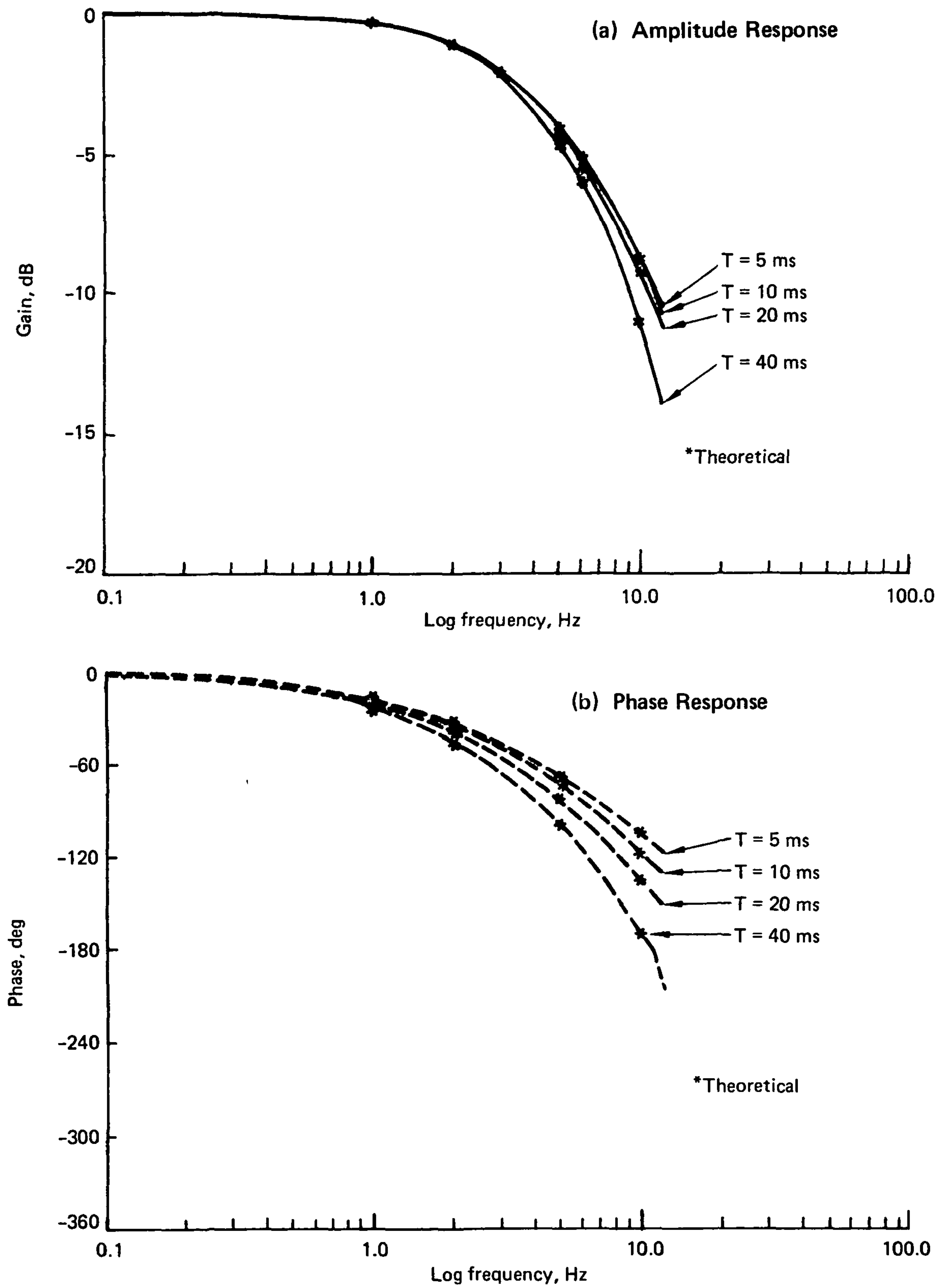


Figure 92. Effect of Sampling Rate on Computer Hardware

performance. The ACT control laws selected for laboratory performance evaluation were:

- Pitch-augmented stability (PAS)
- Maneuver-load control (MLC)
- Gust-load alleviation (GLA)
- Flutter-mode control (FMC)

Figure 93 shows the functional block diagram of the ACT control laws. These represent an early system configuration and may not be consistent with control laws given elsewhere in this document. It should be noted that these control laws were developed assuming that perturbation values of the input variables were available. No detailed consideration has been given to the signal processing required to handle steady-state signal values due to trim conditions, biases, etc. The control filters were designed in the continuous s-domain and mechanized in the discrete (digital) domain using the bilinear transformation (Tustin's substitution). The procedure is based on substituting $\frac{2}{T}\frac{Z-1}{Z-1}$ for s in the analog filter equation and writing difference equations to develop algorithms for software coding.

Frequency response data were taken for the control filters implemented using a frequency analyzer across the MCP-701A associated input and output channels. Figures 94 through 99 show frequency responses of the preceding filter cases with sampling periods of 5, 10, 20, and 40 ms and also the theoretical predicted continuous responses. The plots clearly show errors in gain and phase between the measured and theoretical continuous responses. The measured responses have faster gain rolloff and significantly more phase lag due to the basic hardware characteristics discussed previously. The responses also show that the gain and phase differences from the theoretical decrease as the sampling rate is increased. Test results show that the analog input signal prefilter values must be included in the hardware model and, depending on the sample rate selected, can influence system performance. These values can be selected to change basic hardware characteristics, and software implementation can be used to compensate for certain hardware limitations. After the hardware system has been fully integrated, careful analysis and laboratory testing will be required to ensure that the optimum performance has been obtained.

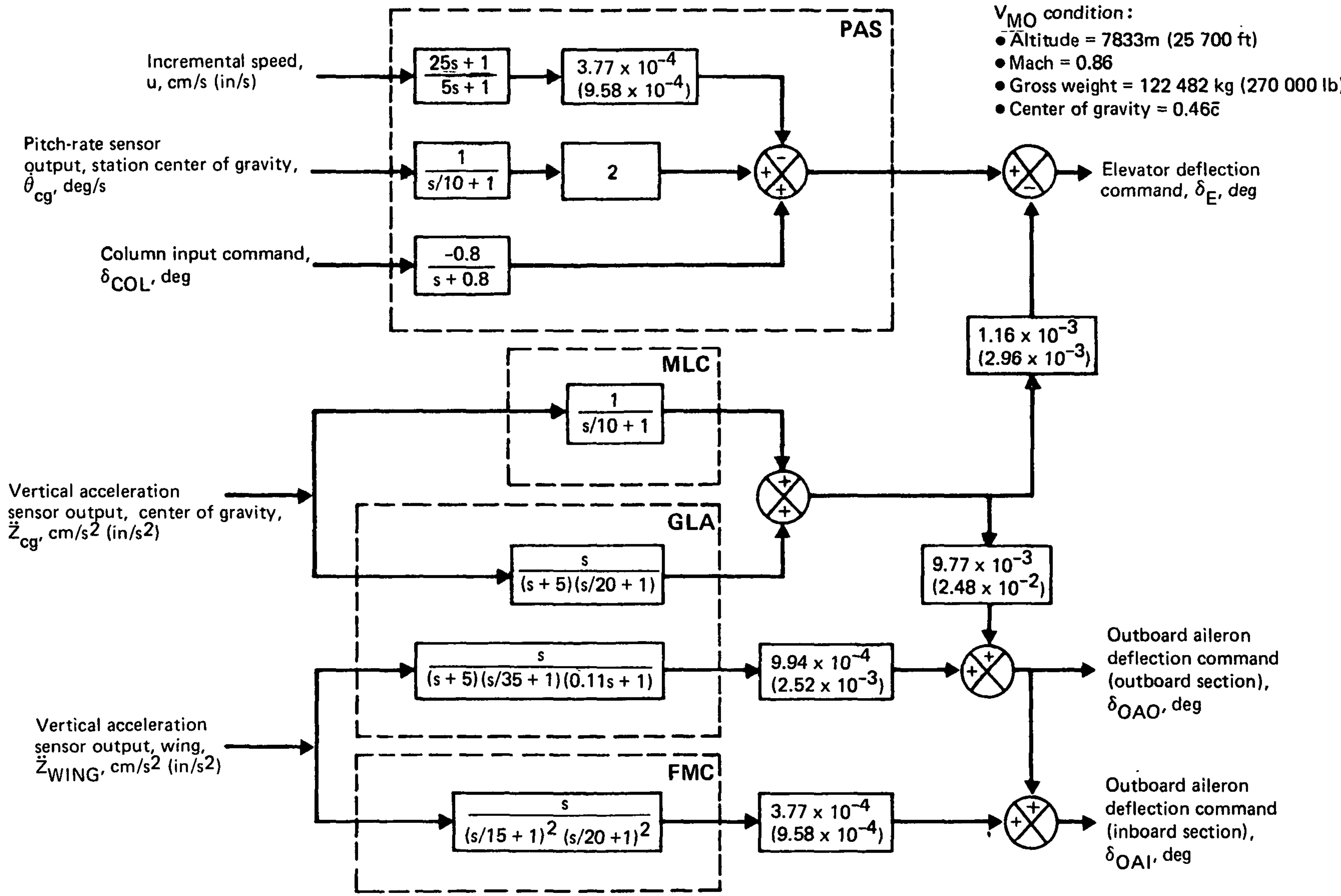


Figure 93. ACT Control Law Functional Block Diagram

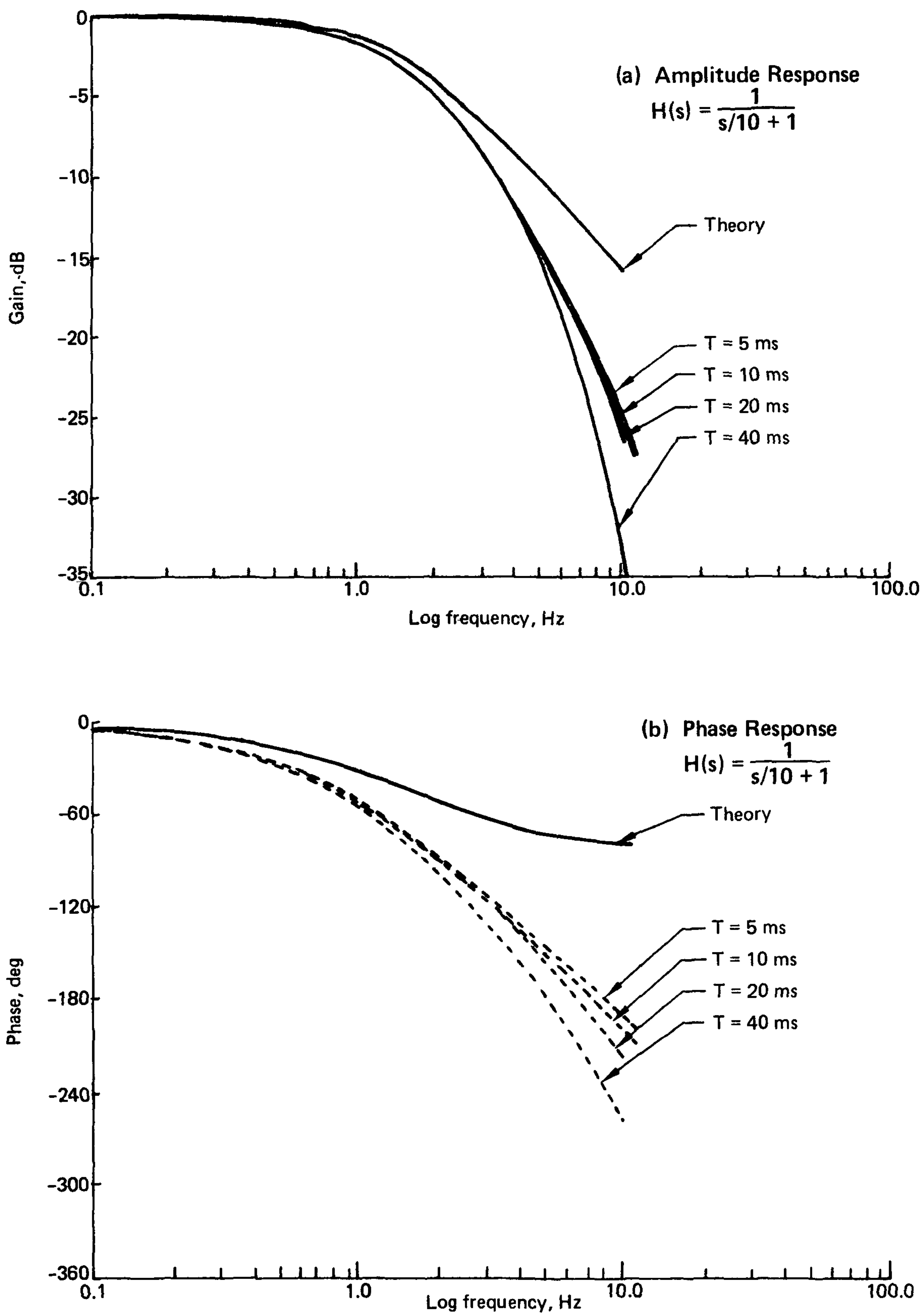


Figure 94. Maneuver-Load Control Frequency Response

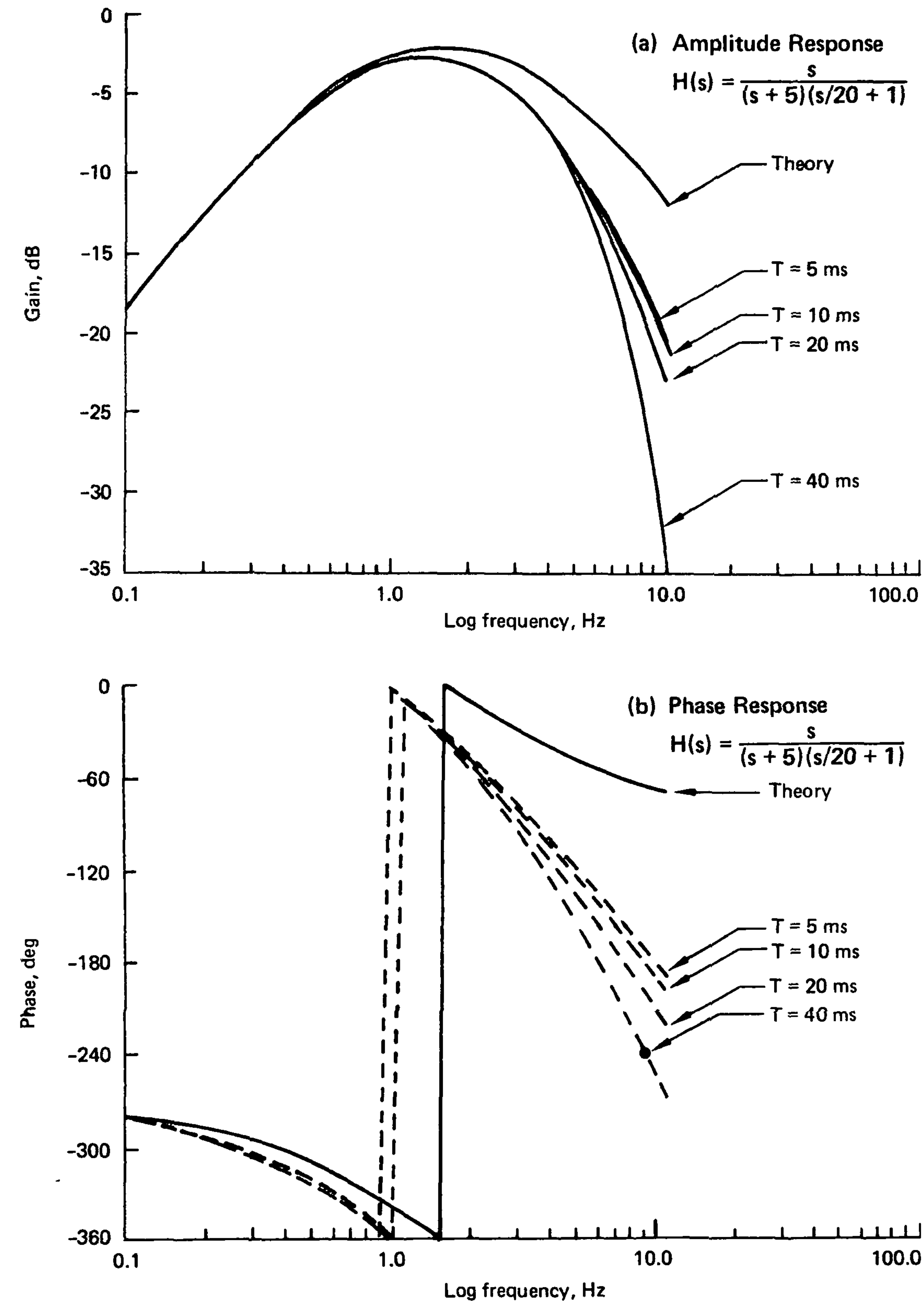


Figure 95. Gust-Load Alleviation Frequency Response, Second Order

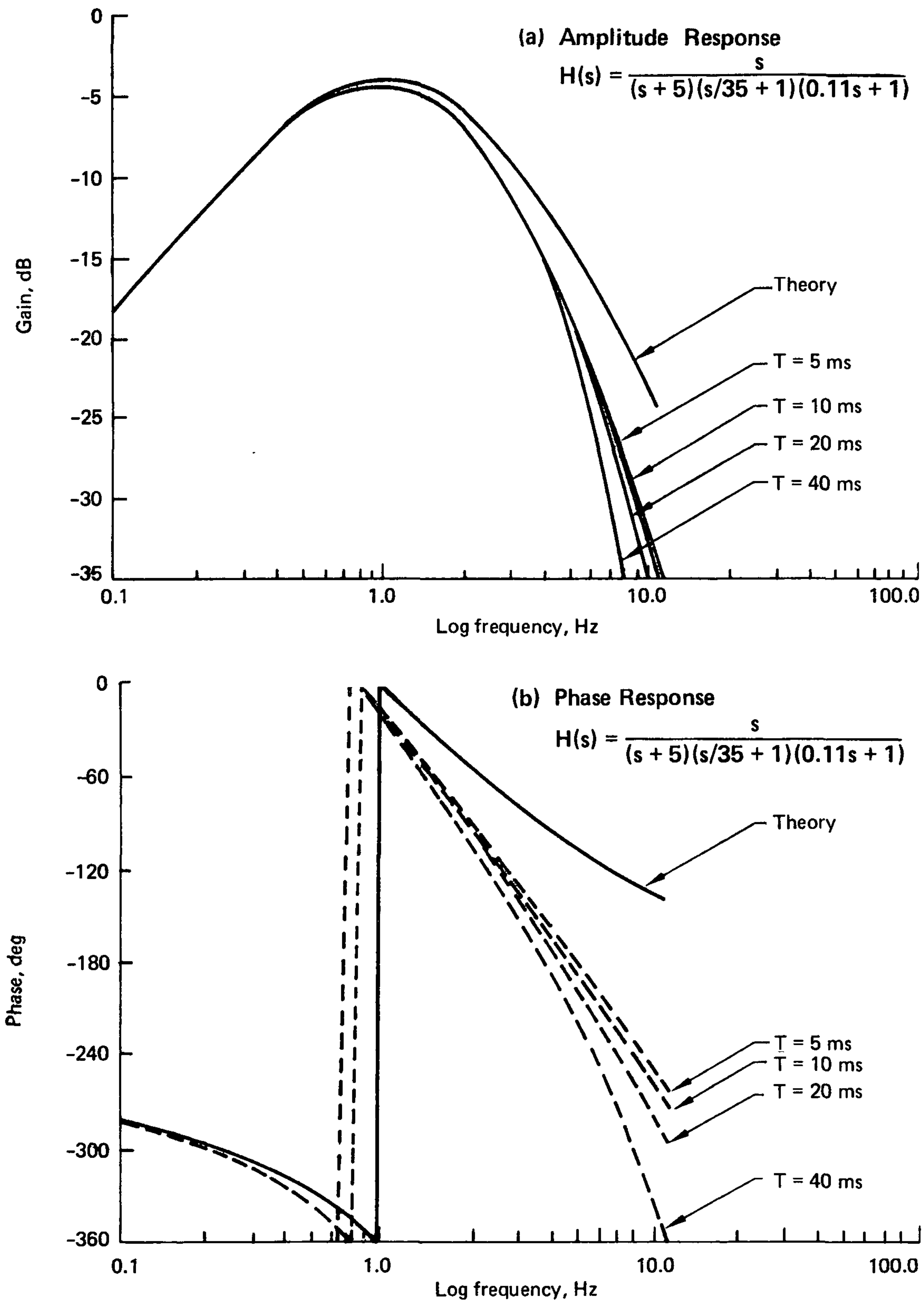
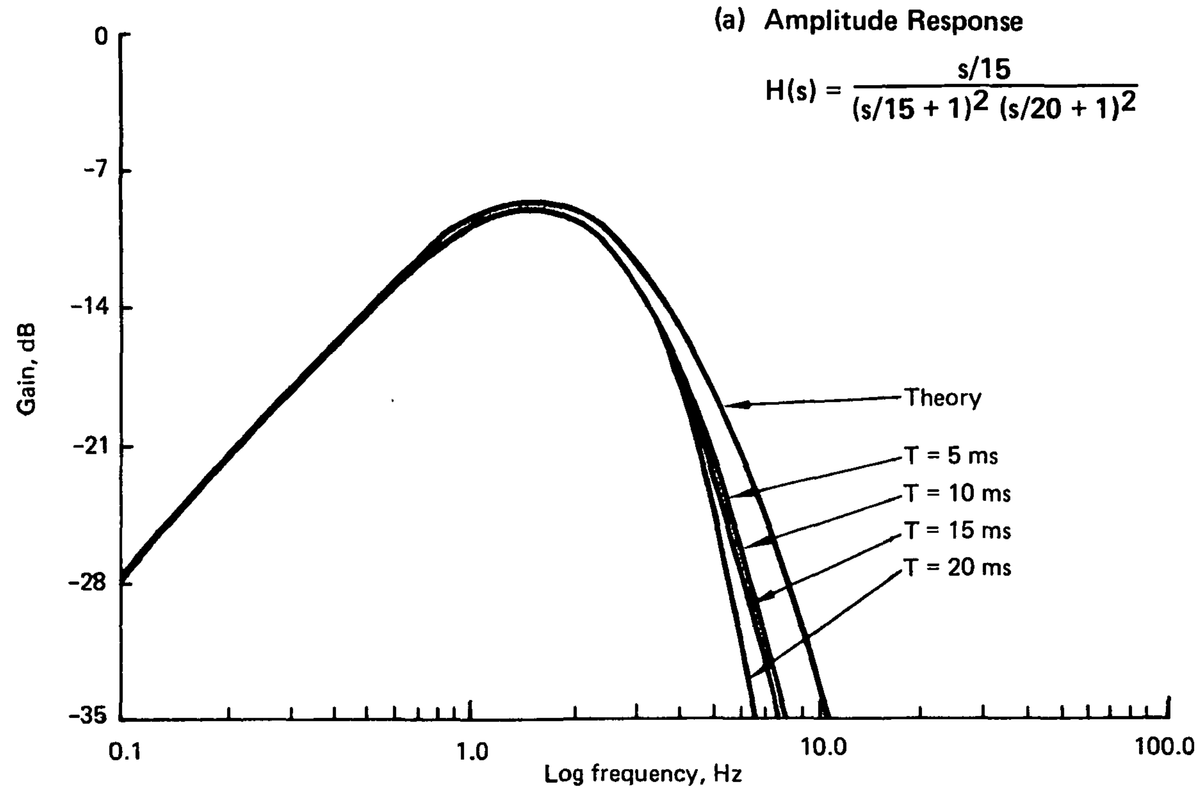


Figure 96. Gust-Load Alleviation Frequency Response, Third Order



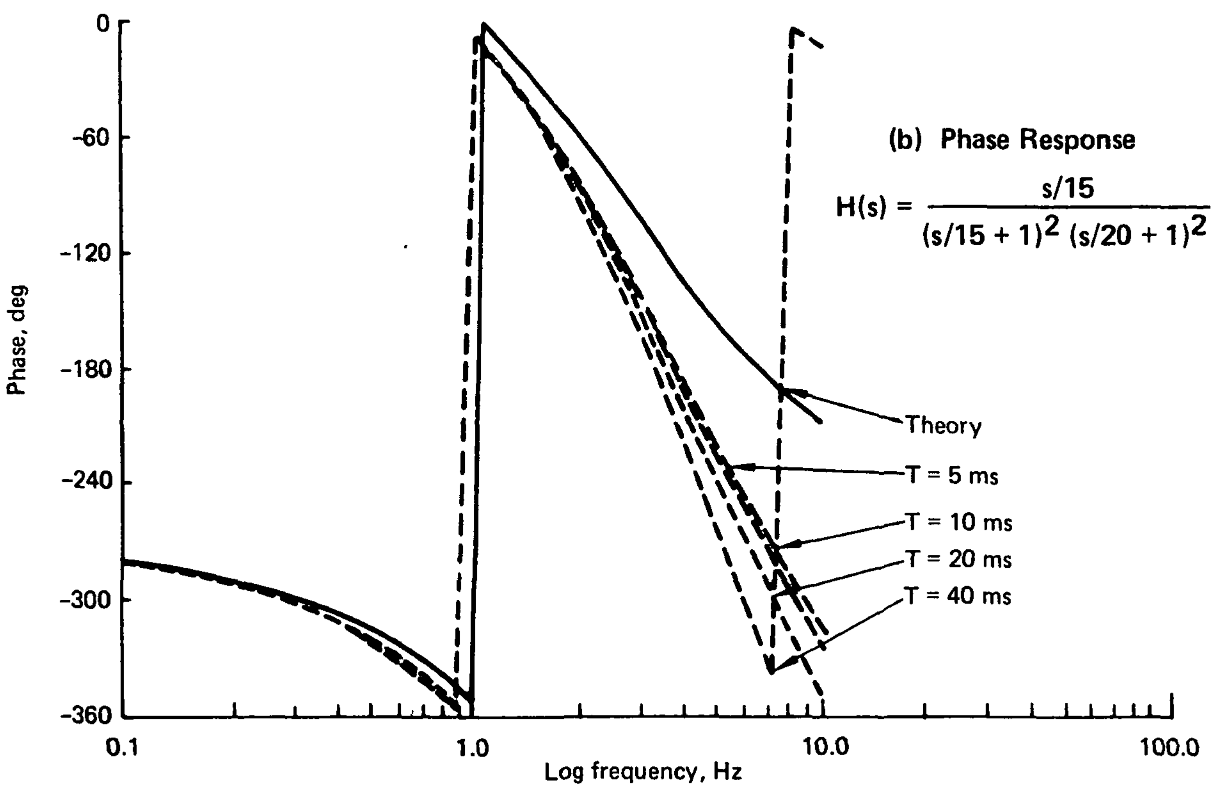
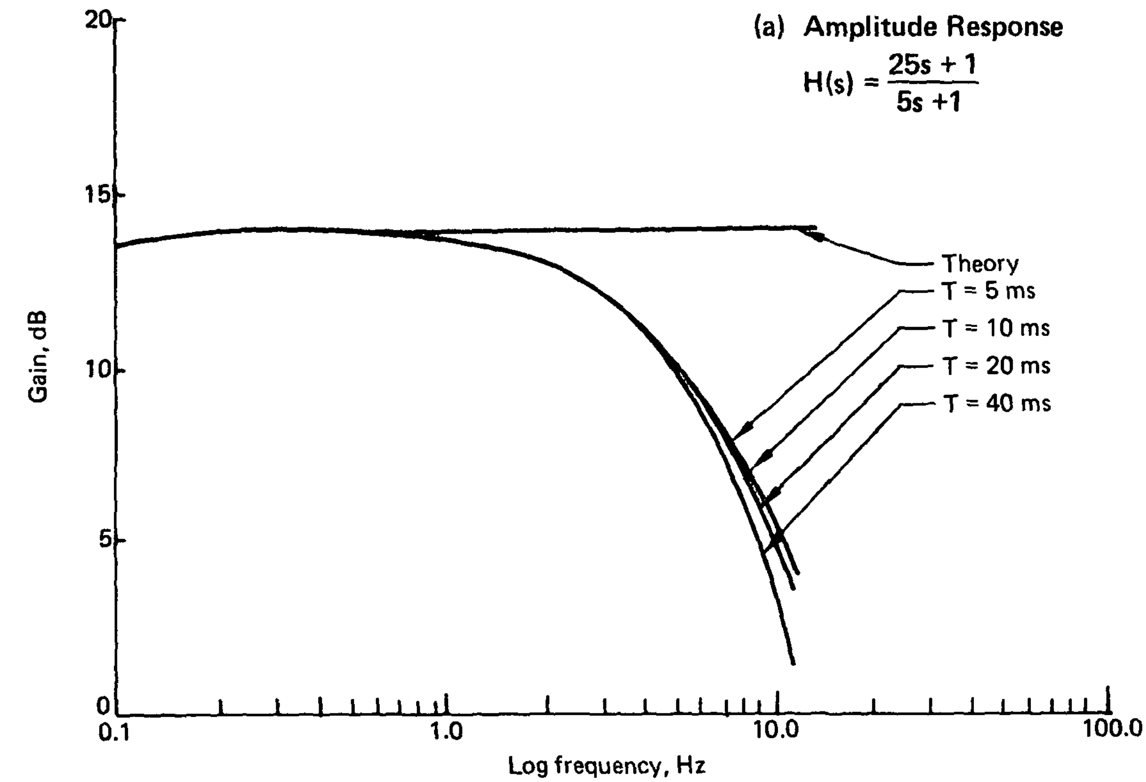


Figure 97. Flutter-Mode Control Frequency Response



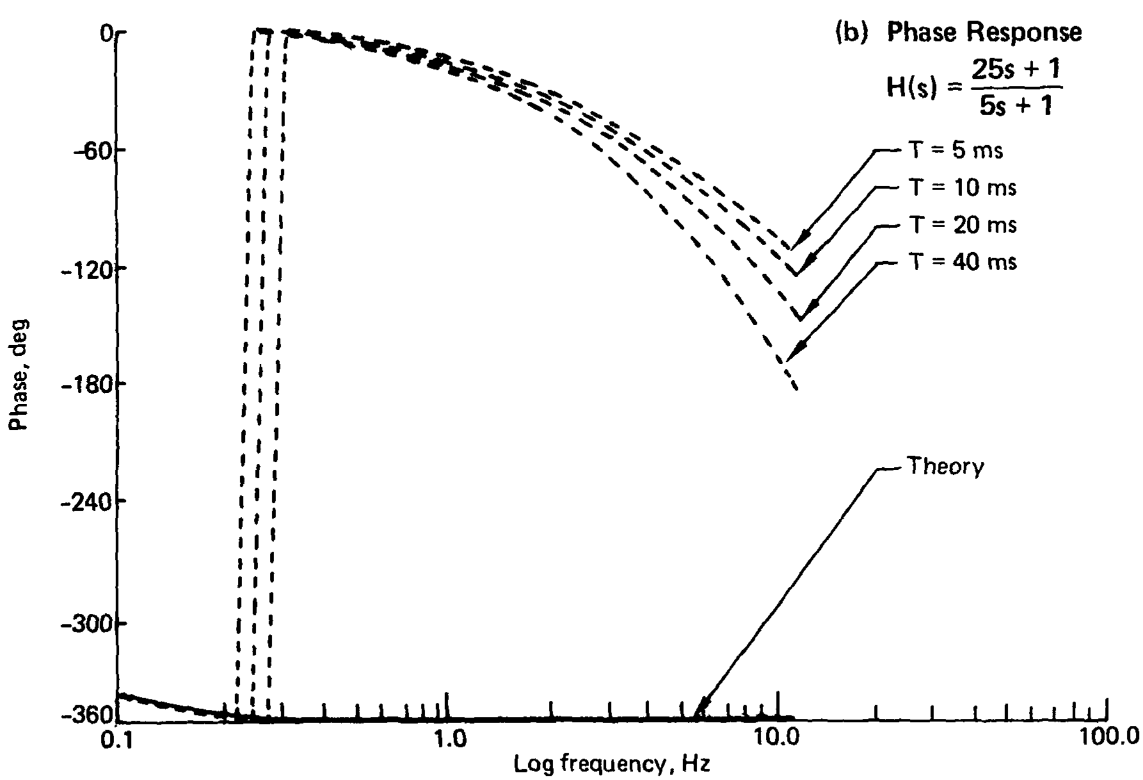


Figure 98. Pitch-Augmented Stability Frequency Response, With Lead

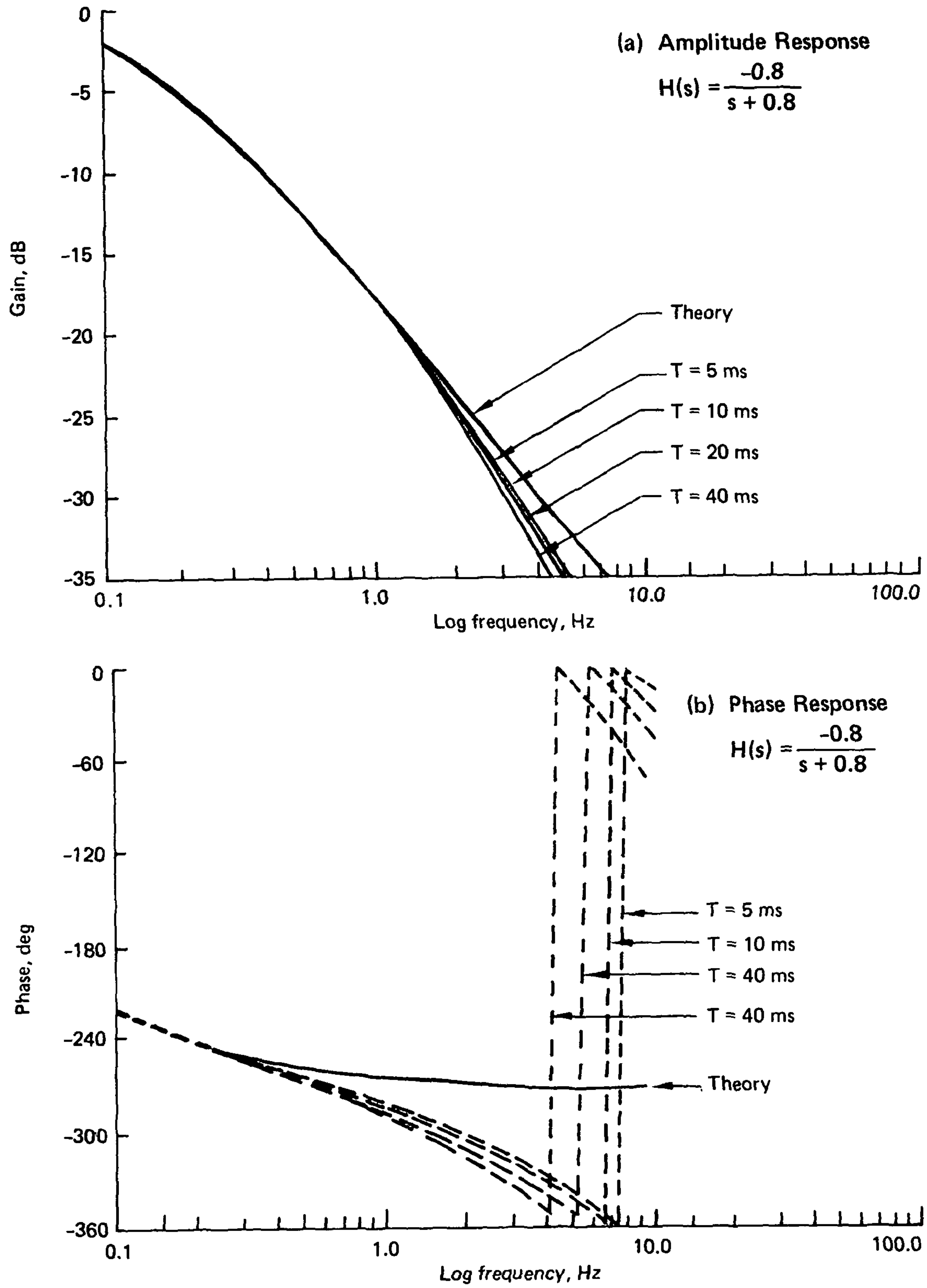


Figure 99. Pitch-Augmented Stability Frequency Response, No Lead

9.1.2 SYSTEM CONCEPT EVALUATION

The tests described in the preceding subsections essentially illustrate that the ACT control functions can be realized using digital technology. This subsection deals with special investigations conducted to evaluate various functional elements that are part of a redundant system configuration.

9.1.2.1 Laboratory Setup

The basic laboratory equipment consisted of the following elements:

- A digital computer simulating the ACT airplane
- A digital redundant flight control system mechanized on flight control hardware
- An analog computer simulating the actuators

A block diagram of these elements, as they were interconnected for closed-loop tests, is shown in Figure 100. Figure 101 is a photograph of this setup, showing the four flight control computers in the center foreground. A Boeing-owned ECLIPSE computer was used to simulate the longitudinal equations of motion of the ACT airplane at several flight

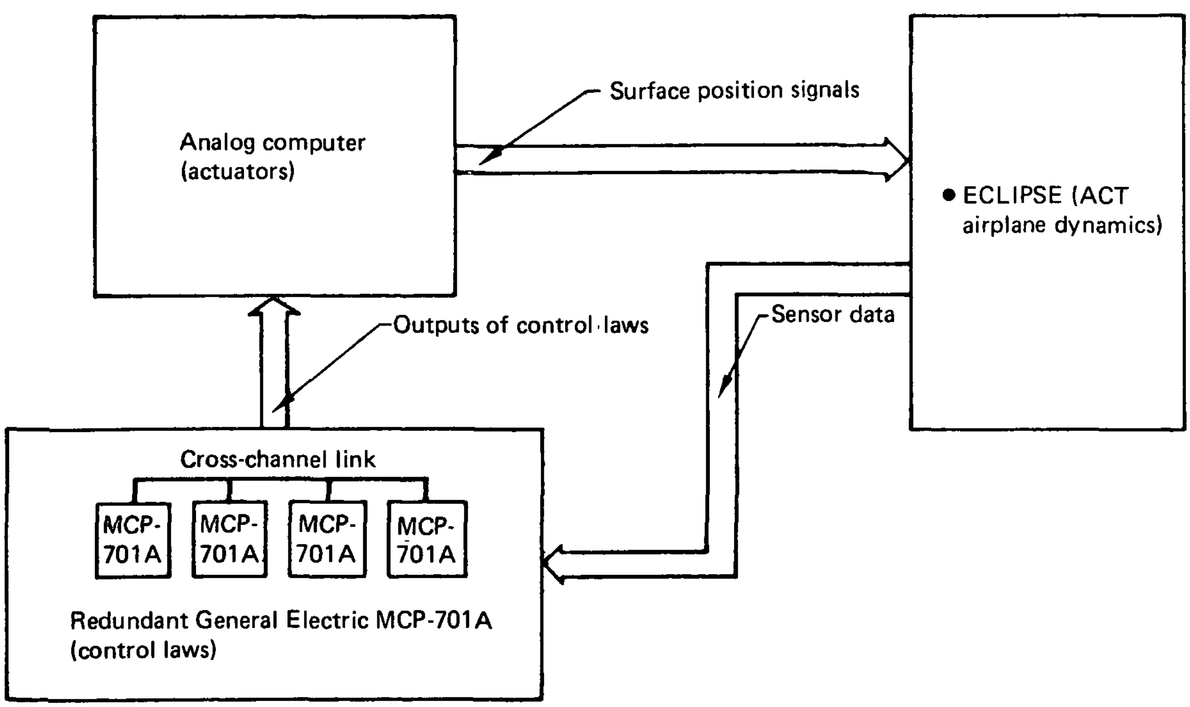


Figure 100. Laboratory Setup for Closed-Loop Simulation

Figure 101. Computer Laboratory Setup

conditions. State variable techniques were used to implement the dynamics of the longitudinal axis in four rigid and six elastic modes for real-time execution. Minimum real-time execution for the airplane model was set at 40 iterations per second (25-ms sampling period). Although different flight conditions (i.e., V_B, V_{MO}, and V_D) were programmed, only the V_{MO} flight condition was tested as a closed-loop system in the laboratory. Sensor data were simulated in the ECLIPSE computer and fed to the General Electric MCP-701A computers.

General Electric MCP-701A flight control computers were used to implement ACT control laws and redundancy management. ACT control laws were programmed using an algorithm derived from incremental difference equations. In addition to the software required to realize the ACT control functions, extensive redundancy management software routines were coded and tested in the laboratory. A cross-channel data link connected the flight control computers to make up a redundant digital system configuration. Outputs of the computers were fed to an analog computer.

An EAI-PACER 500 analog computer was used to simulate servoactuators. Simple lag filters represented the ACT actuation system. Surface position signals from the analog computer were fed back into the airplane simulation to close the loop.

9.1.2.2 Synchronous and Asynchronous Computer Operation

The Segregated System mechanizes each ACT control law in a separate dedicated computer set, whereas the Integrated System implements all ACT control laws using one computer set. Because both system concepts were designed to operate in a redundant system configuration, additional multiple computer hardware and software packages would be required in the Segregated System. To minimize hardware and software complexity, the dedicated computers of the Segregated System and the Essential PAS Computers of the Selected System were designed to be operated asynchronously. A series of laboratory closed-loop tests was conducted to verify that implementation of each ACT control law in a redundant system is feasible in computers operating either synchronously or asynchronously. The testing involved simulation of the quadruply redundant PAS and triply redundant FMC control functions, each operating in closed-loop configuration, as described in Subsection 9.1.2.1.

The tests were conducted separately with each redundant control system. The simulated pilot input continuously disturbed the simulated ACT airplane with a 0.1-Hz sinusoidal column input command for a period of 1 hr while the control computers operated synchronously. The same test procedure was repeated with the control computers operating asynchronously.

A high-amplitude pilot input command (on the order of 15 deg of elevator deflection command) was provided to investigate control computer drift. Control computers drifting with respect to each other would each sample similar input signals at different times and produce dissimilar output signals. For the worst case, these differences in output signals will be more pronounced if the drift is large and the input signals contain high-frequency components.

During the tests, the aircraft parameters were continuously recorded to note any response disparity between the two modes of computer operation. Further, synchronous and asynchronous operations were verified by monitoring the clock signals on a four-channel oscilloscope. The test results obtained were compared, and no differences in response were apparent. Computer drift did not affect the response; drift time was negligible compared with the sampling period. Further, these input signals are processed by a signal selection and failure detection (SSFD) algorithm that selects an appropriate signal for the control law computations. This SSFD concept, discussed in Subsection 9.1.2.3, is used in the ACT system design to isolate input signal failures and produce identical computer output commands in a redundant system configuration.

Laboratory tests indicate that each ACT control function implemented in control computers operating asynchronously can achieve the same performance as the control computers operating synchronously.

9.1.2.3 Signal Selection and Failure Detection

All redundant systems require some form of SSFD if fault isolation is required. Digital computers can be effectively multiplexed to provide SSFD functions for processing incoming variable signals. The following discusses SSFD functions that process triple sensor output information by a median selection or an averaging algorithm if a failure is detected. Median selection simply provides a signal output that is the median signal of three incoming sensor signals. If two signals are identical, the median will be that signal. If one failure is detected, the averaging algorithm provides a signal output that is the average of the remaining two valid sensor signals. Sensor output data are first converted to digital form and then cross fed to the other remaining channels via a cross-channel data link. Each individual computer channel of the redundant system collects the sensor output data and then derives and tests differences to detect sensor failures. Two SSFD functions were coded for real-time tasks: one that processes pitch-rate sensor outputs and one for acceleration sensor outputs.

Tests were performed to simulate sensor output failures, both "zero" failures (sensor output goes to zero) and "hardover" failures (sensor output goes to a plus or minus maximum value), and evaluate the two SSFD functions for the quadruply redundant PAS and triply redundant FMC configurations, each operating in a closed-loop system (subsec 9.1.2.1). Evaluations were made by observing the simulated ACT airplane time history for "normal" system operation (no failures) followed by various combinations of sensor output failures.

Both the quadruply redundant PAS and the triply redundant FMC achieved performance requirements during this test. Following two pitch-rate sensor failures for PAS or one acceleration sensor failure for FMC, the observed aircraft parameters showed normal performance. Following a third pitch-rate sensor failure or second acceleration sensor failure, the redundant control system automatically shuts down, resulting in an open-loop airplane. It should be noted that the open-loop ACT airplane is stable at the V_{MO} flight condition tested. The following paragraph discusses the SSFD process for the quadruply redundant PAS system. SSFD is described in more detail in Section 8.0.

The quadruply redundant PAS system design must be operational following any two sensor failures, providing all required functions with no performance degradation. The signal selection concept for the quadruply redundant configuration uses an active-standby, online technique. Three out of four incoming sensor signals are designated active and processed by the SSFD algorithm; the fourth is on standby. When the first active signal failure is detected, the fourth input is switched into an active mode.

If no fault is detected, the median algorithm is selected. If one fault is detected, the standby signal switches to an active mode and the median algorithm is again selected. If two faults are detected, the averaging algorithm is selected. Following a third failure, third-failure information is generated to shut down the PAS control computers. Figure 102 illustrates the process just described. The slightly biased pitch-rate sensor outputs and the elevator deflection commands shown were obtained by disturbing the simulated ACT airplane with a 0.1-Hz sinusoidal pilot input command.

The median-selection process input signal can exhibit an abrupt change when one signal fails to a hardover state and that signal was in, or passed through, the middle region of the three signals. Figure 102 shows that suppression of the transient associated with hardover failures can be achieved. These abrupt responses are softened by inserting an equalization loop around the median-selection function. The selected output is the median signal of the equalized input signals, where the input signals are continously being equalized toward the selected output. The equalization technique forces the inputs to the median-selection algorithm to be nearly identical.

The two SSFD algorithms were evaluated using the simulated ACT airplane having PAS and FMC control laws, respectively. Various types of sensor signal failures were

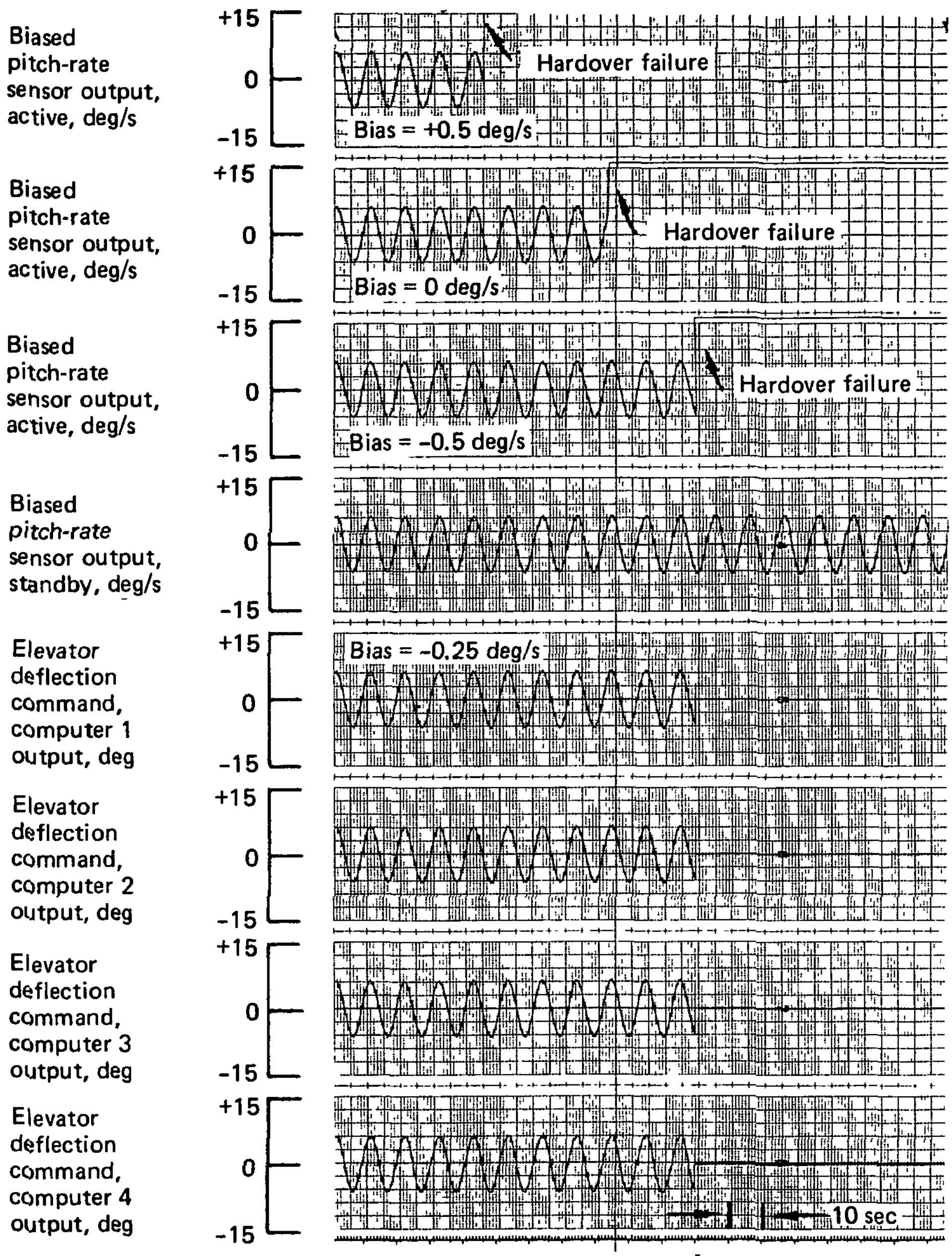


Figure 102. Quadruply Redundant Pitch-Augmented Stability Pitch-Rate Sensor Failures

introduced (e.g., hardover and zero failures) with satisfactory test results. These SSFD functions can be used to prevent sensor output signal failures from disturbing the airplane response and to extend the operational reliability of the flight control system.

9.1.2.4 Actuator Recovery From Computer Transient

Many computer shutdowns can be attributed to transient failures of software or hardware. Laboratory testing was conducted to establish that an ACT computer actuator system can recover from such transient failures.

The dual force-summed actuator used in the laboratory test is described in detail in Subsection 8.1.3. Because each servoactuator channel is directly commanded by an associated processor, failure of a computer will temporarily shut down its associated servo to prevent deterioration in performance or possible structural damage. The temporarily disconnected actuator must then be brought back into the system smoothly when the transient failure of the computer disappears. A hydraulic bypass valve is incorporated into the actuator design to shut down the actuator when it or the computer driving it fails. The same bypass valve is also used to reactivate the actuator if the system recovers to normal operation.

Testing involved simulation of the dual force-summed actuator model and the PAS system. Figure 103 shows the laboratory setup for closed-loop tests. The ECLIPSE computer, described in Subsection 9.1.2.1, implements the longitudinal equations of motion of the ACT airplane at different flight conditions. The dual force-summed actuator model was implemented in one MCP-701A digital computer operating in real time at 200 iterations per second (5-ms sampling period). The bypass valve was simulated by a digital first-order lag filter with a time constant of 0.01 sec. The output signal of the lag filter falls exponentially to zero when bypass is requested and returns to its nominal value when bypass is no longer requested. The bypass signal is activated when a failure is detected. Computer failure was simulated by shutting down one of the computers driving one actuator, then an appropriate signal was introduced from the analog computer to simulate the output of the failed computer. The actuator was programmed to detect the failure 0.5 sec after the computer failure had occurred. Identical PAS control laws were contained in three MCP-701A digital computers operating asynchronously in real time at 200 iterations per second (5-ms sampling period). A cross-channel data link connected the three digital computers. The PAS control laws were integrated with the redundancy management software to realize a triplex redundancy scheme.

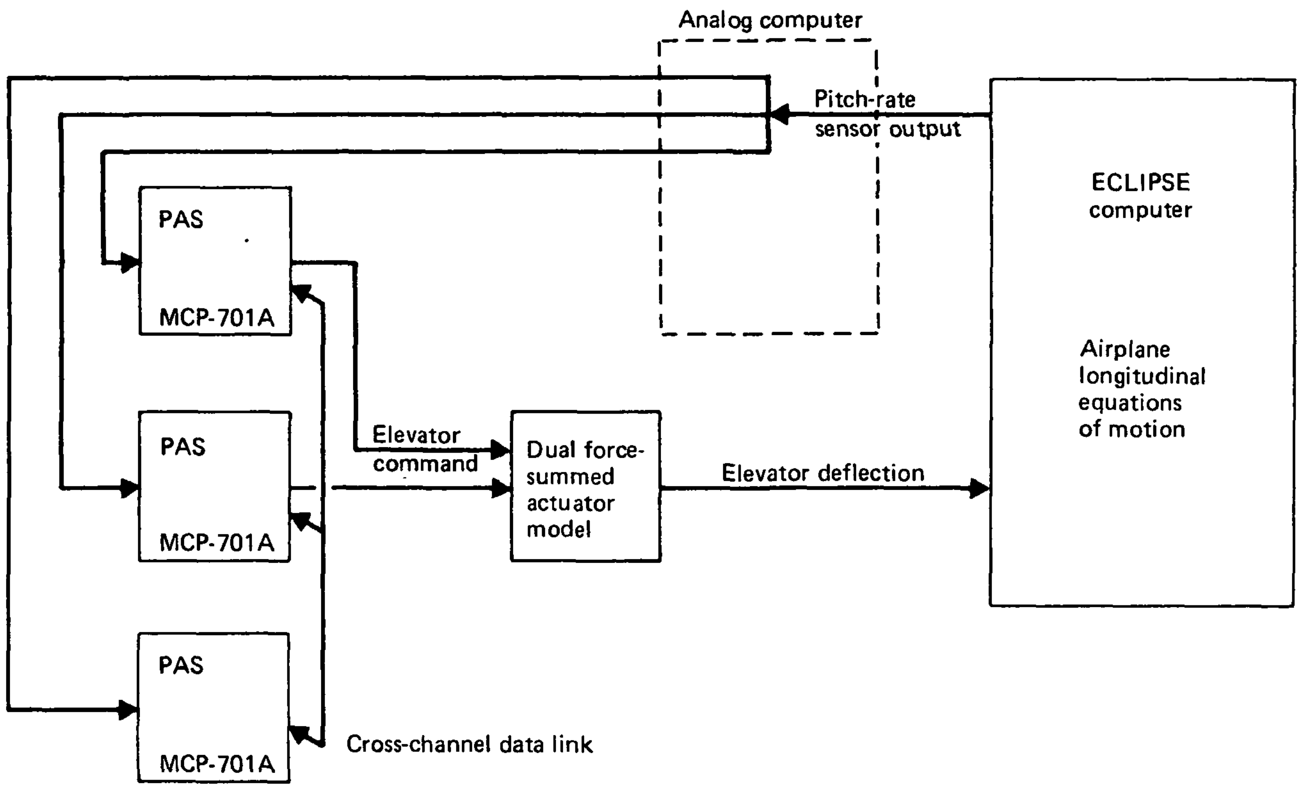


Figure 103. Laboratory Setup for Pitch-Augmented Stability Closed-Loop Simulation

The control law equation implemented was

$$\delta_E = \frac{1.5}{0.1\,s + 1}\ \dot{\theta}_{1202} + 3.77 \times 10^{-6}\ \Delta u - \frac{0.8}{s + 0.8}\delta_{COL}$$

where

$\dot{\theta}_{1202}$	=	pitch-rate sensor output, station 1202, deg/s
Δu	=	incremental horizontal velocity, body axis, m/s
δ_{COL}	=	column input signal, deg
δ_E	=	elevator deflection command, deg

The analog computer was used to add three voltage biases to the pitch-rate sensor signals from the ECLIPSE computer and to provide a method of introducing sensor failures. All data used in the simulation correspond to flight condition V_{MO}.

Figure 104(a) shows the closed-loop short-period response to a 6-deg step column input with PAS control law implemented on a digital computer. Figure 104(b) shows the same system responses to the same input with PAS control law implemented on an analog computer. Responses from the two test methods are closely matched.

Tests were performed using a sinusoidal δ_{COL} command signal at 0.1 Hz. Figure 105(a) shows the closed-loop system responses to a hardover failure (e.g., computer output command goes to maximum value). When the failure is removed and the failed computer recovers to its normal operation (e.g., back on line), a time delay occurs before the PAS computer outputs command the same actuator positions. This is due to reinitialization of software parameters following computer startup and the 1.25-sec lag on the δ_{COL} command signal path. Figure 105(b) shows the same system responses for the case where the 1.25-sec lag filter was removed. The PAS computer outputs are identical. Figure 106(a) and (b) shows the closed-loop system responses for both "zero" failure (computer output goes to zero value) and oscillatory failure (computer output goes to $\pm$ maximum value), respectively, with a 0.1-Hz sinusoidal δ_{COL} command signal input, including the lag filter on its path.

This study indicates that if the ACT system has integrating functions or very-low-frequency filters, the transient response during recovery will be large and may not be acceptable. For this case, the system must incorporate a more sophisticated recovery technique, such as exchange of filter data between the computers, so that the transient response will be within acceptable levels. The control laws synthesized during the IAAC Project, however, have fairly high frequency, and the simple recovery mode is adequate.

Figure 106(c) shows the closed-loop system responses to a hardover computer failure with no δ_{COL} input command. The failures excited a lightly damped 1.0-Hz oscillatory mode of the basic ACT airplane response.

Laboratory tests verified the reconfiguration method of the dual force-summed actuator operating in a closed-loop configuration.

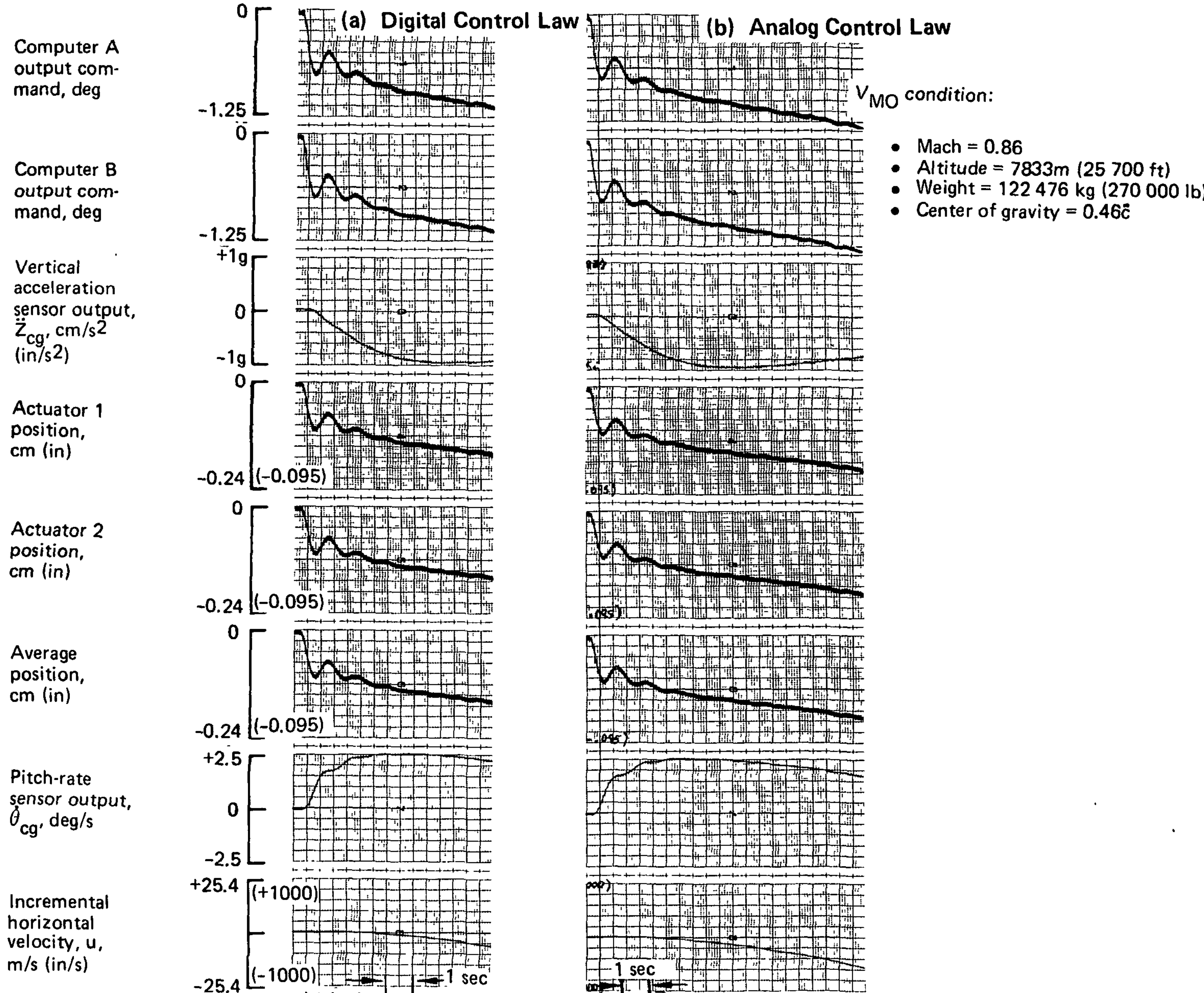


Figure 104. Short-Period Response to Step Column Input

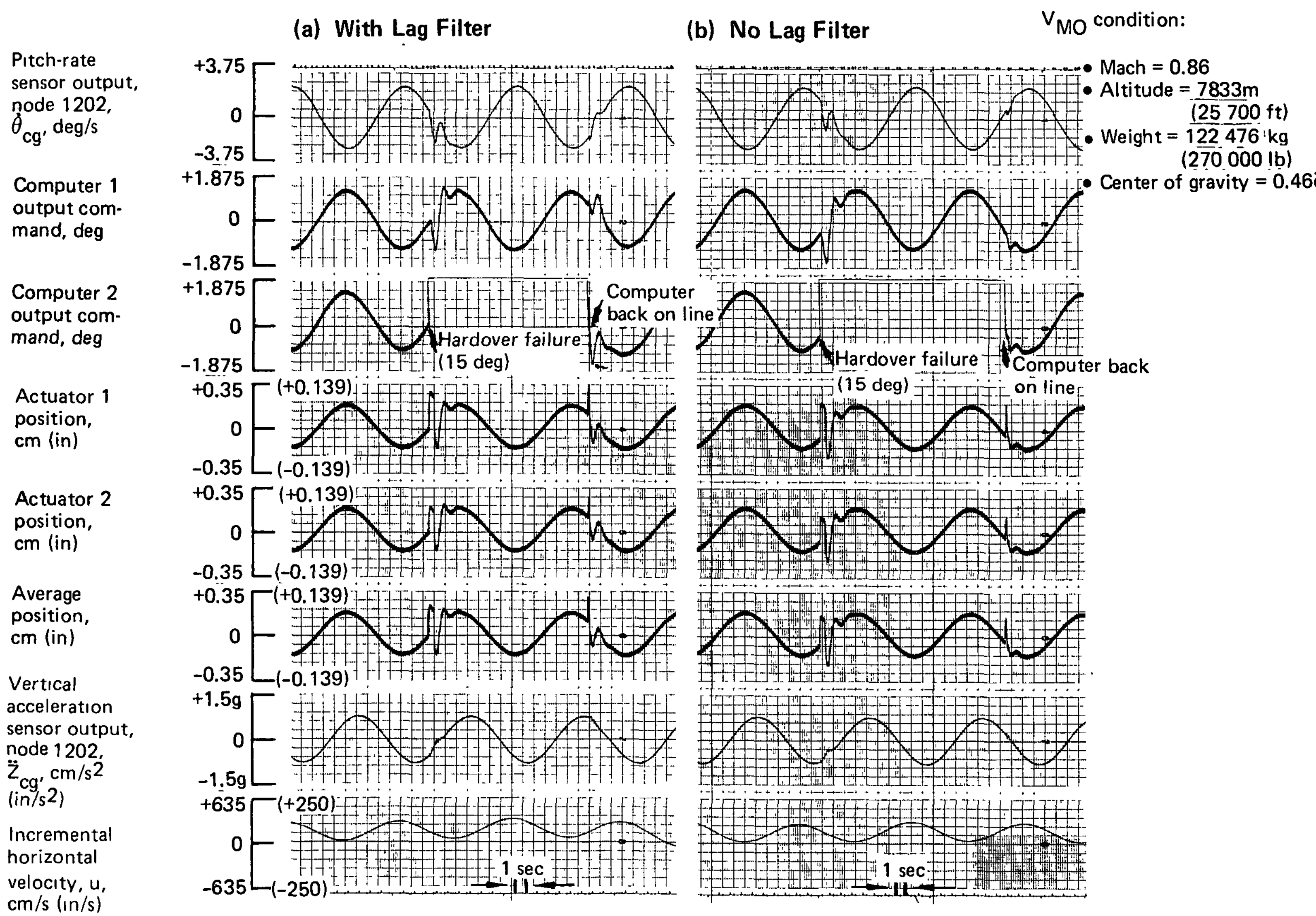


Figure 105. System Response to Hardover Failure

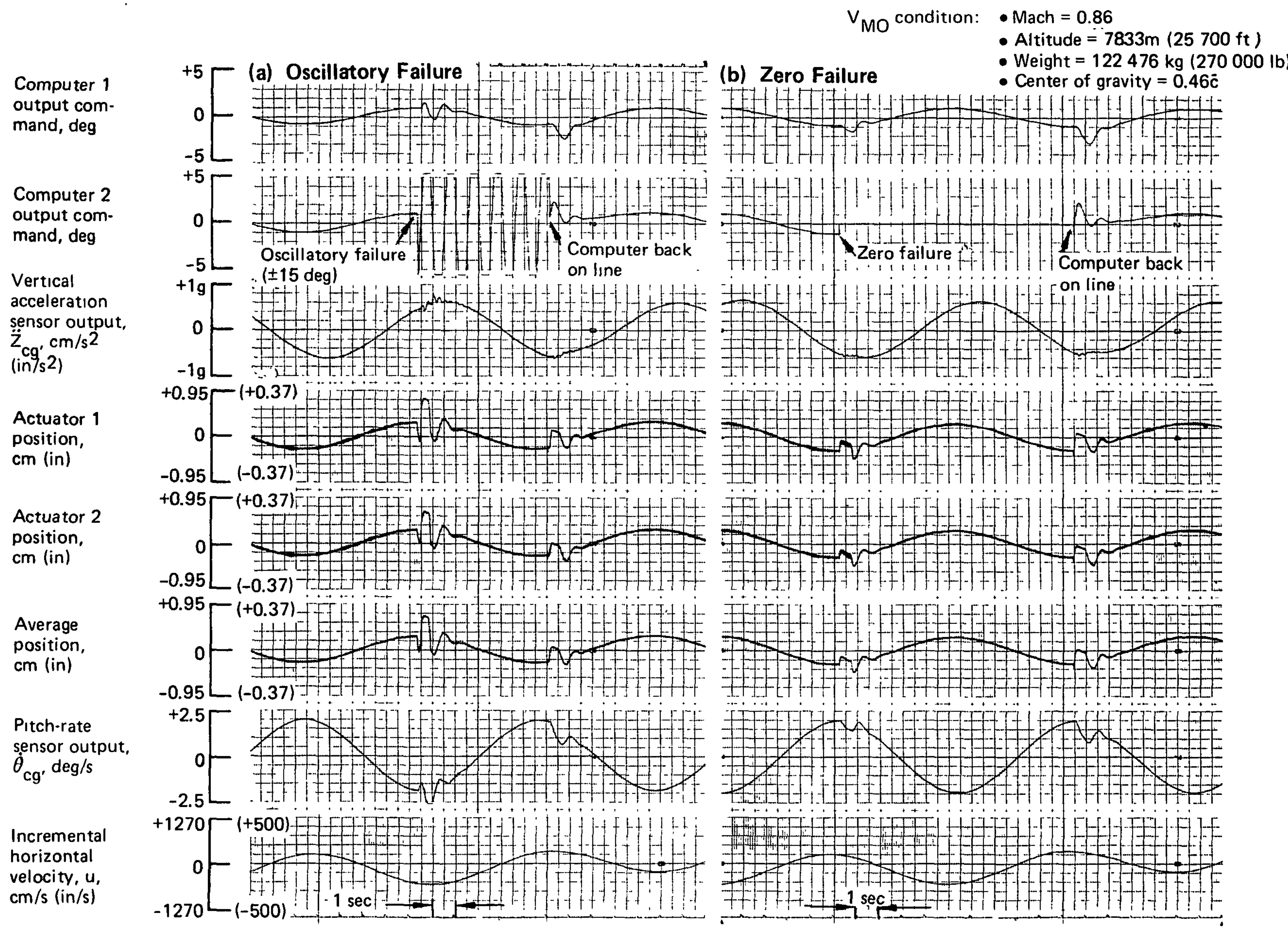
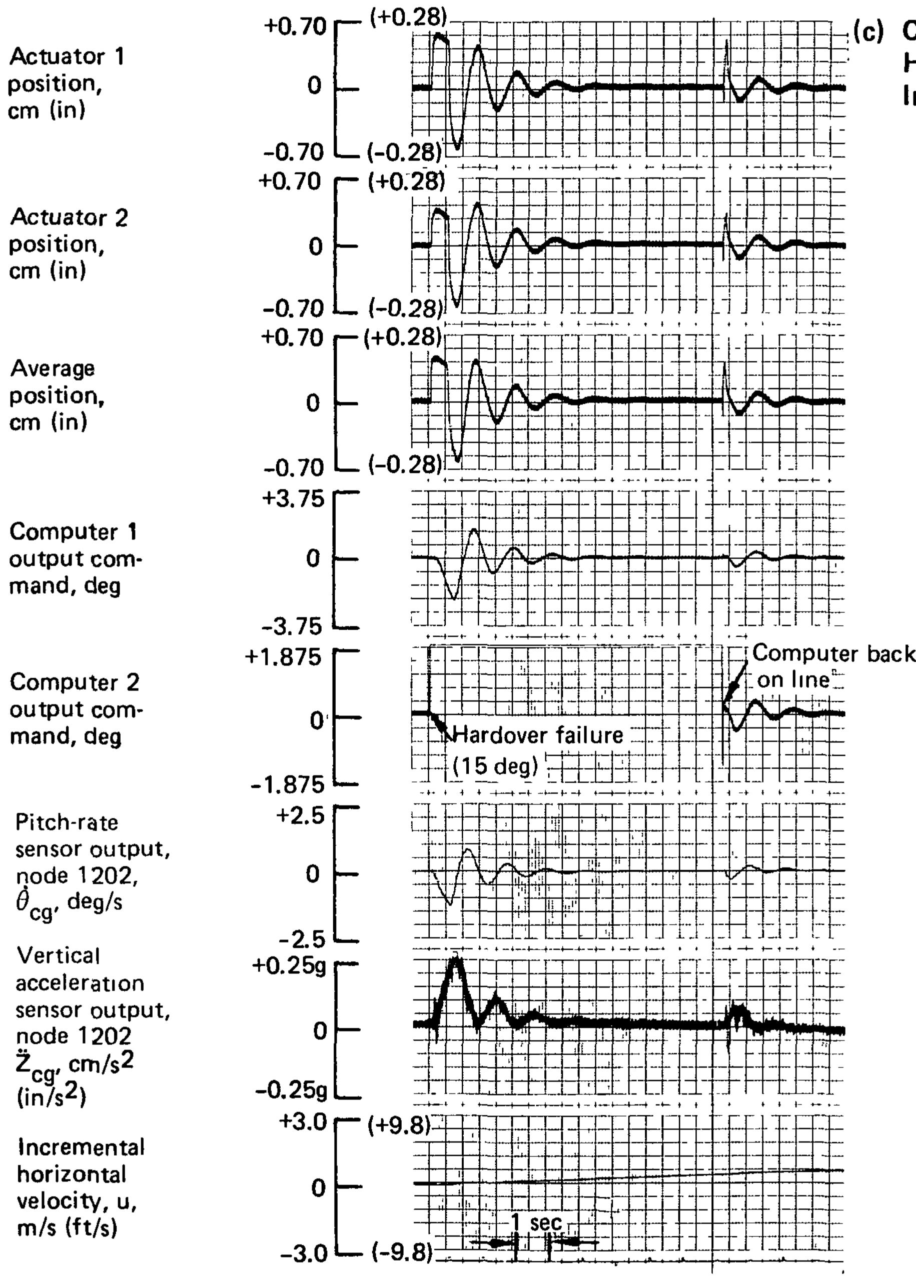


Figure 106. System Response to Computer Failure



(c) Closed-Loop System Response to Hardover Computer Failure (No Input Command)

Figure 106. System Response to Computer Failure (Continued)

9.1.3 ACTUATOR EVALUATION

9.1.3.1 Actuator Failure Mode and Effect Analysis

An actuator failure mode and effect analysis (FMEA) has been conducted for the ACT secondary actuation and ACT fly-by-wire (FBW) power control unit actuation systems described in Subsection 8.1. The analysis was conducted based on major component breakdowns as shown in Figures 45 and 46.

The results are shown in Tables 24 and 25. The following can be concluded from the analysis:

- For a crucial ACT function:
    - The actuator meets the fail-operational/fail-operational requirement against any two component failures, excluding jamming failure.
    - The actuator meets the fail-operational requirement against any single jamming failure.
- For a critical ACT function:
    - The actuator meets the fail-operational requirement against any single component failure, excluding jamming failure.
    - The actuator meets the fail-safe requirement after a second component failure.
    - The actuator will be shut down after a jamming failure.

9.1.3.2 Performance Evaluation of Force-Summed Actuators

Subsection 8.1.3 and Figure 45 describe the force-summed secondary actuator concept for application in the ACT control system. A simulation study has been conducted to verify the design concept. The following questions were answered in the simulation study:

Table 24. Force-Summed Secondary Actuation System Failure Mode and Effect Analysis

Item identification name [a]	Item function	Type of failure or error	Detection method	Failure effects subsystem operation
Electronics ①	Computer command signals, close servo loop, current feedback, etc	Fail, no output due to power loss, loose connections, fracture, overheat	Failure detector or preflight test	Lose one channel
		Fail, hardover due to hot short or overheat	Failure detector or preflight test	Lose one channel, small failure transient
LVDT ②	To close actuation position loop	Fail, no output or power loss, loose connection or short	Failure detector or preflight test	Lose one channel, small failure transient
LVDT ③	To measure electrohydraulic valve spool position and use it to detect failure by comparing with other channels	Fail, no output due to open circuit or power loss	Failure detector or preflight test	None
		Fail, hardover due to jam or short	Failure detector or preflight test	Inadvertent channel shutoff
Bypass valve ④	Solenoid-operated on off valve used to bypass a failed channel	Failure to open due to jam or electrical open or loose connection	Preflight test	None
		Fail, open inadvertently due to short circuit	Failure detector or preflight test	Inadvertent channel shutoff
Valve feedback spring ⑤	To close electrohydraulic valve position loop	Disconnect or break causes valve to go hardover	Failure detector or preflight test	Lose one channel, small failure transient
First stage electrohydraulic valve ⑥	To transfer electric signal to valve position	No output due to leakage or break	Failure detector or preflight test	Lose one channel, small failure transient
		Hardover due to clog or jam in the fluid nozzle	Failure detector or preflight test	Lose one channel
Second stage electrohydraulic valve ⑦	To control fluid flow to the actuator	No output due to excessive leakage	Preflight test	Lose one channel
		Offcenter due to jam or leakage that causes actuator hardover	Failure detector or preflight test	Lose one channel, small failure transient
Centering spring ⑧	To provide hard point for pilot's mechanical input	Fail, break, or jam	Maintenance inspection	None, unless both channels fail and both spring breaks
Actuator ⑨	To provide force voting and mechanical input to PCU	Leak due to break or seal failure	Preflight test	Lose one channel
		Jam	Failure detector or preflight test	Lose one channel for triple actuator, lose function for dual actuator
Pogo ⑩	To disconnect a jammed mechanical path	Spring break	Failure detector or preflight test	Lose one channel
Connectors ⑪	To provide electric power signal for electrohydraulic valve, shutoff valve, LVDT	Break or short	Failure detector or preflight test	Lose one channel
Two failures in different channels	For two-actuator system, any two independent failures or an actuator jam will cause inoperative system For three-actuator system, any three independent failures or any two actuator jams will cause inoperative system			

[a]Numbers identify the component shown in Figure 45

[b]This device is used in three-actuator PAS function only If one of the three actuators is jammed and full hydraulic pressure or bypass fails to free it, the combined force of the other two will break the pogo loose There is no pogo in two-actuator system

Table 25. Force-Displacement Actuator Failure Mode and Effect Analysis

Item identification name*	Item function	Type of failure or error	Detection method	Failure effects subsystem operation
Electronics ①	Amplify input signals, close servoloop, current feedback, etc.	Fail, no output due to power loss, loose connections, fracture, overheat	Failure detector or preflight test	Lose one electrical channel
		Fail, hardover due to hot short or overheat	Failure detector or preflight test	Lose one electrical channel, small fail transient
LVDT ②	To close actuation position loop	Fail, no output or hardover due to power loss, loose connection, or short	Failure detector or preflight test	Lose one electrical channel, small fail transient
LVDT ③	To detect hydraulic channel failure	Fail, no output due to power failure or loose connection	Preflight test	None
		Fail, hardover due to short or jam	Failure detector or preflight test	Inadvertent shutoff of one hydraulic channel
Hydraulic shutoff valve ④	To shut off electro-hydraulic valves in the failed channel	Failure to close due to jam or electrical open or loose connection	Preflight test	None
		Fail, closed inadvertently due to short circuit	Failure detector or preflight test	Inadvertent shutoff of one hydraulic channel
Valve feedback spring ⑤	To close electrohydraulic valve position loop	Disconnect or break causes valve hardover	Failure detector or preflight test	Lose one hydraulic channel, small fail transient
First-stage electrohydraulic valve ⑥	To transfer electric signal to valve position	No output due to leakage or break	Failure detector or preflight test	Lose one hydraulic channel
		Hardover due to clog or jam in the fluid nozzle	Failure detector or preflight test	Lose one hydraulic channel
Second-stage electrohydraulic valve ⑦	To maintain valve position and to control the main spool	No output due to excessive leakage	Failure detector or preflight test	Lose one hydraulic channel
		Offcenter due to jam or break	Failure detector or preflight test	Lose one hydraulic channel
Main valve spool ⑧	To control hydraulic flow to the actuator	Excessive leakage due to erosion	Failure detector or preflight test	Lose one channel
		Jam	Failure detector or preflight test	May cause total loss of function
Actuator ⑨	To control surface position	No force output due to excessive leakage or crack	Preflight test	Lose one hydraulic channel
		Jam	Failure detector	Inoperative system
Connectors ⑪	Provide electric power, signal for electrohydraulic valve, shutoff valve, LVDT	Break or short	Failure detector or preflight test	Lose one electrical channel

Double failures: Any two electric or any two mechanical failures, one in each channel, or a single jam in the main spool or actuator, will cause the inoperative actuator system

*Numbers identify the components shown in Figure 46.

- In normal operation, how high is the force-fight level caused by various component tolerances?

- How can the failure transient be predicted? How does one design the failure logic, threshold, and timing?

- What kind of mathematical model is needed to track valve spool motion?

Computer Simulation Construction–The dual force-summed actuator configuration was selected for the simulation study because it represents typical even-numbered-channel force-summed actuator characteristics. The force-summed actuator tends to present a dead-band characteristic when operated in an even-number-of-channels configuration. Figure 107 shows a block diagram of the computer simulation, which includes two hardware actuator channels and one mathematical model used as a failure monitor. The parameters have been collected from vendor catalogs, test data, or other documents and represent typical servoactuator parameters. A simple first-order lag (840 rad/s) was used to compute valve spool position. The mathematical model used in the failure monitor is represented by spool position; i.e., no lag. Major position loop gain was 80 rad/s, as was the actual hardware tested. Because a prior laboratory test of a single-channel actuator showed sufficient stability margin, no attempt was made to use the computer simulation for stability check or compensation. Most of the parameters associated with force fight and failure transients were well represented; i.e., valve dead zone, null shift, feedback spring rate, etc. Force-fight level was set to be proportional to the difference between actuator positions and combined structure compliance. The position output, which is also the feedback signal of the valve spool "mathematical model," is the average of the two actuator positions.

Cross-channel comparators between the valve spool positions and the mathematical model were used for failure logic design. Figure 108 shows the flow diagram of the failure logic. Every time the valve spool position difference exceeds the threshold, the detector counts up four; every time valve spool position is within the threshold, it counts down one. This device effectively detects high-frequency oscillatory failure. Threshold was set at 25% of the full stroke, and average detect time was about 50 ms. The threshold was large enough and the timing was long enough so that chances of false alarms were minimized (several trial runs verified this). These numbers can be changed easily if necessary, even after the airplane is in operation. The failure logic used should be able to detect all hardover- or

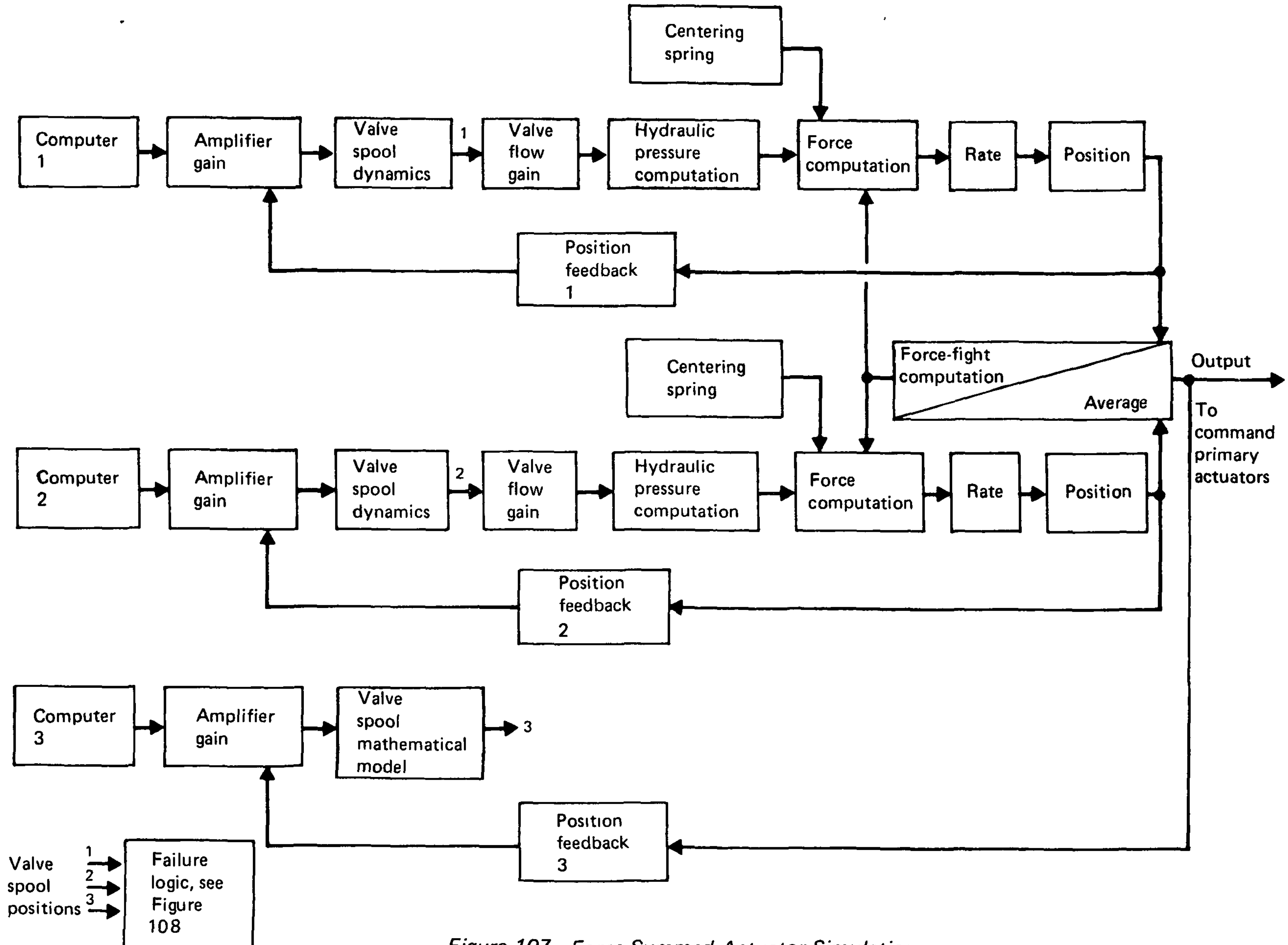
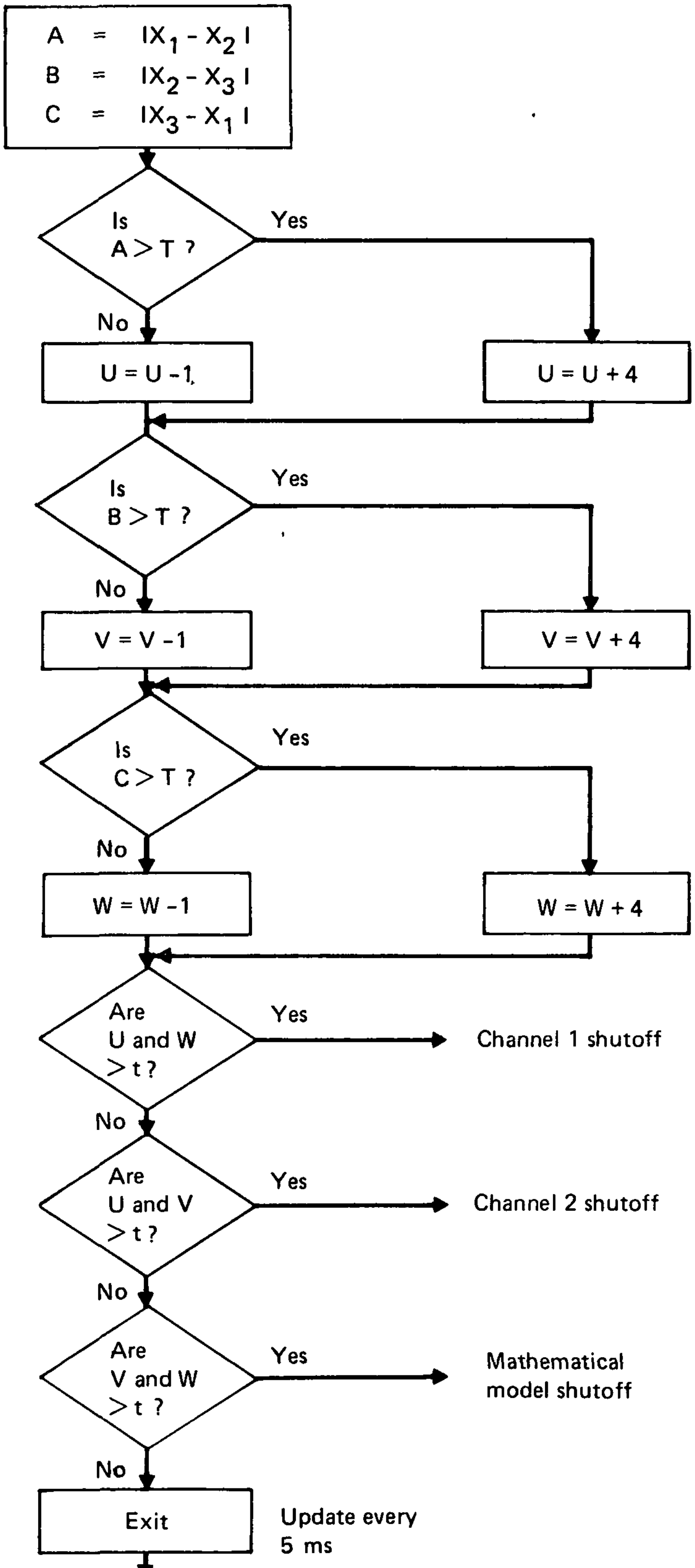


Figure 107. Force-Summed Actuator Simulation



X_1, X_2 } Valve spool LVDT position

X_3 Mathematical model position

V, U, W } Positive integer

T Threshold

t Time setting

Recommended value:

T = 25% full stroke
(for LVDT of ± 10V
T = 5V)

t = 25

Figure 108. Failure Detector

oscillatory-type failures. Passive failures will be detected when the error signal exceeds the threshold by an amount equivalent to a large airplane motion. The preflight procedure will detect passive failures.

Simulation Results—Figure 109 shows three identical input signals fed into the two identical actuation channels (possible only mathematically) and one simplified mathematical model. The simplified model is representative of the model that would be part of the actual monitor implementation. Both actuators have identical outputs and no force fight. The mathematical model, which represents spool position without lag, tracked the real spool positions closely.

Figure 110 shows a computer run with a reasonable combination of component tolerances in each channel. The position difference between the actuator outputs is small enough so that both actuator positions still look identical. This is evidenced by the appearance of a small force fight between actuators.

Figure 111 simulates a hardover failure. The trace shows that at 0.11 sec, the No. 1 valve suddenly fails hardover. Then, as the No. 2 valve and mathematical model go in the opposite direction to keep the No. 1 actuator from running away, a maximum force fight exists between the two actuators. At approximately 0.14 sec, the failure is detected and the No. 1 actuator is bypassed. Hydraulic pressure difference across the piston drops to zero. As soon as the failed actuator is bypassed, the good actuator takes over and the output follows the computer input the same as before the failure occurred. The failure transient is very small. Before the failed channel is bypassed, the output to the power control unit is the average of the two actuator positions that essentially still follows the input. After the failed channel is bypassed, a large dip occurred. In an actual case, it would take a much longer time to bypass the hydraulic pressure, and the dip would be much smaller.

Figure 112 shows that the failure detector is also able to detect oscillatory-type failures at high frequencies. The mathematical model and the good valve spool position are tracking each other, and a large position error exists between the good and the failed valve spools, sufficient to trigger the failure detectors. Further investigations into actuator responses to computer transients were conducted in the laboratory and are covered in Subsection 9.1.2.4. The simulation study shows that secondary actuators with mathematical model are a feasible and acceptable approach for the ACT system design.

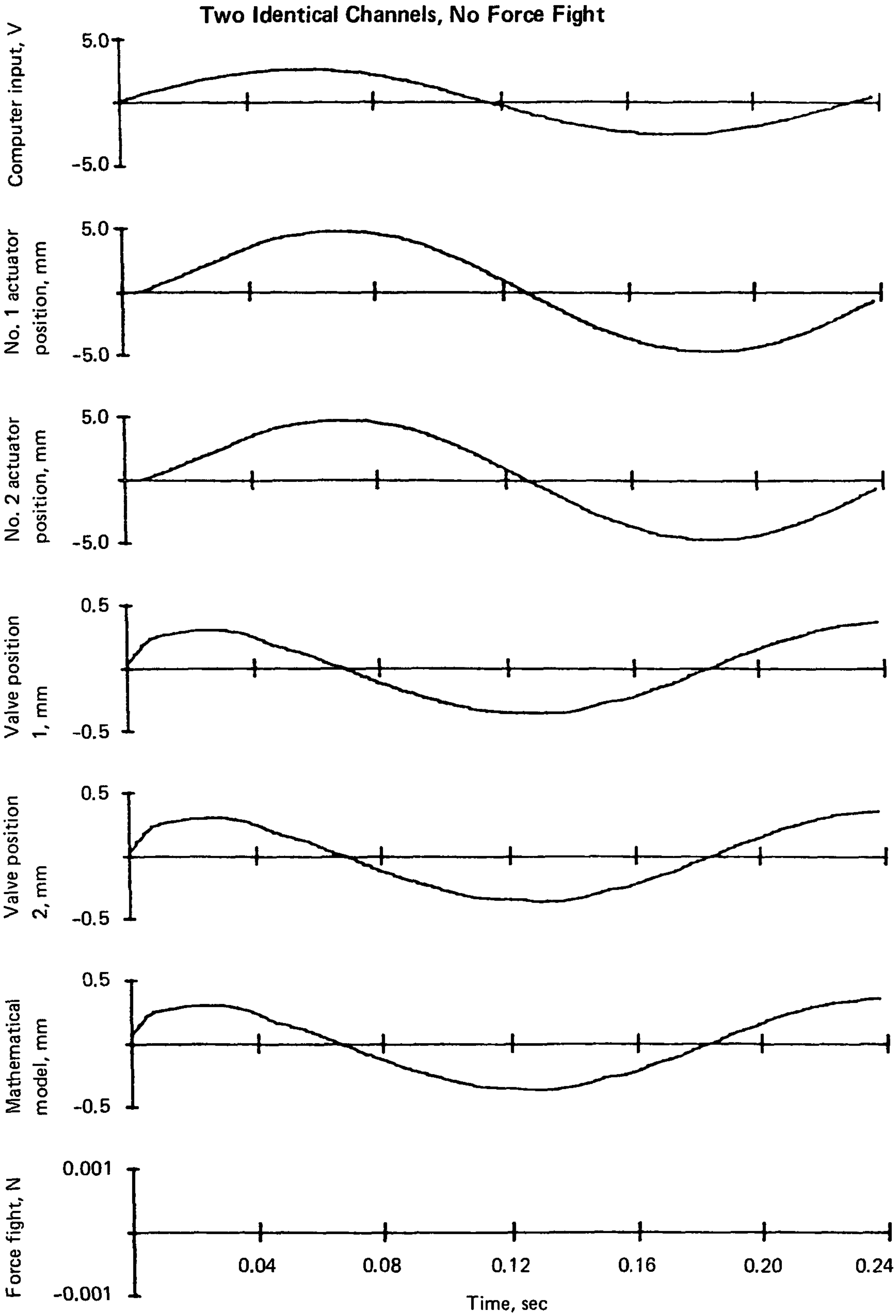


Figure 109. Force-Summed Actuator Simulation, Identical Input

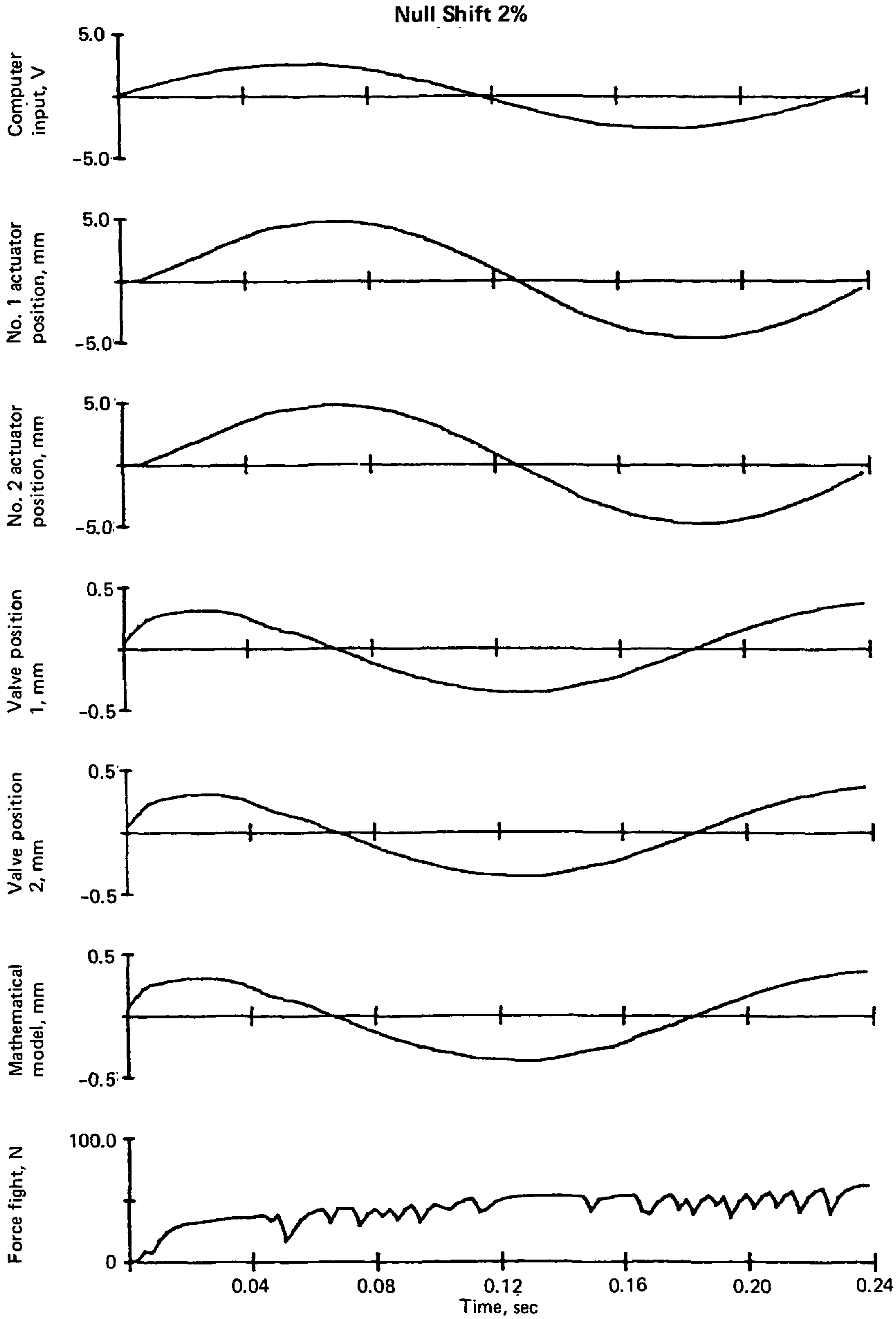


Figure 110. Force-Summed Actuator, Component Tolerances Included

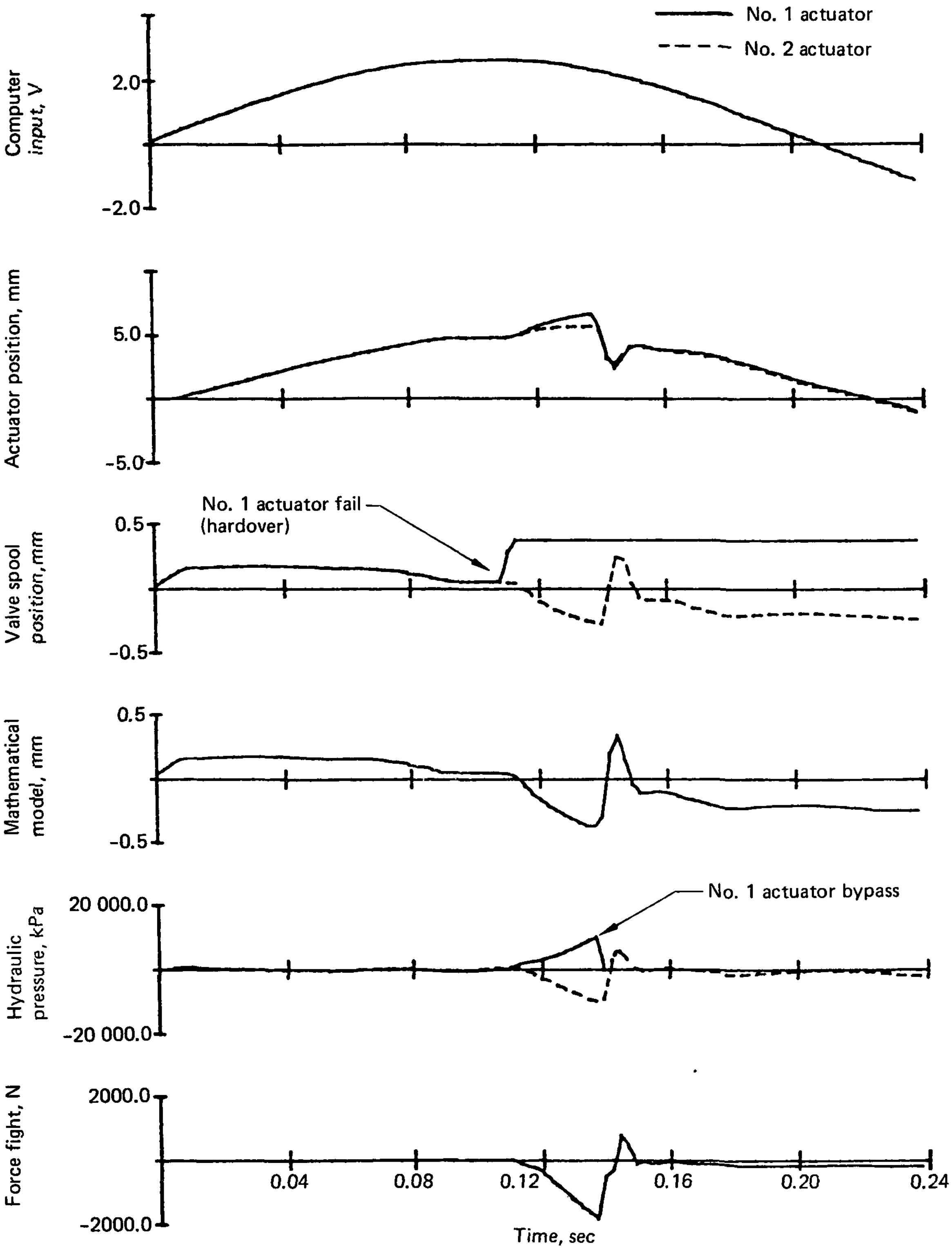


Figure 111. Force-Summed Actuator Simulation, Hardover Failure

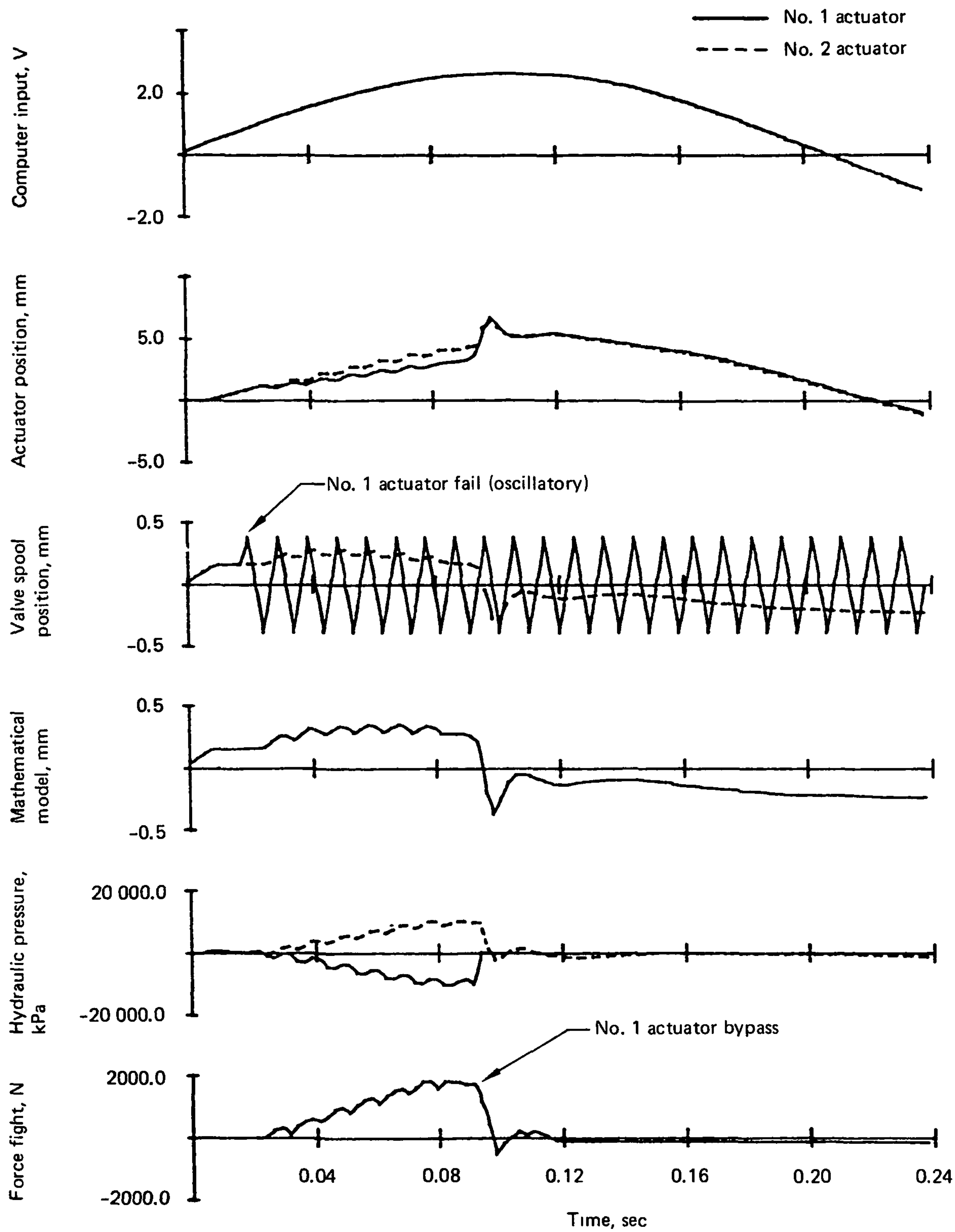


Figure 112. Force-Summed Actuator Simulation, Oscillatory Failure

9.2 RELIABILITY ASSESSMENT

9.2.1 COMPONENT RELIABILITY DATA

Reliability of the individual ACT functions and of combinations of those functions depends upon the failure rate of their components and upon their configuration. The best source for reliability data was experience with the components in commercial service. If that information was not available, the manufacturer's quoted failure rate or an analysis by subcomponents from MIL-HDBK-217C was used. Values used and their sources are listed in Table 26.

9.2.1.1 Partitioning of Line Replaceable Unit Failure Rates

The reliability studies of the Integrated and Segregated Systems assumed that any failure of a line replaceable unit (LRU) represented total failure of the unit. Actually, a multiple data source such as the digital air data computer (DADC) or the inertial reference system (IRS) can partially fail and still provide some of the data correctly. Benefits can be realized by extracting (when possible) the desired information where it is first produced. Thus, the pitch and yaw rate and normal acceleration will be extracted from the IRS in analog form without being converted to digital form and operated upon by the IRS computer. The DADC dynamic pressure and angle-of-attack outputs will also be extracted as analog signals. Similarly, the entire primary computer need not always be used. Table 27 shows how the LRUs were broken down into subcomponents to select the necessary subcomponents and subcomponent failure rates for each application.

This ability to intercept an analog signal before it becomes subject to the potential failures of other parts of the LRU does improve system reliability, but some of the differences noted in Subsection 9.2.2 is only a consequence of the improved method of calculating reliability.

9.2.1.2 Failure Rates of Supporting Systems

The ACT functions require support from the airplane electric, hydraulic, and pneumatic systems. The pneumatic system is used only in the angle-of-attack limiter (AAL) and

Table 26. Failure Rates of ACT Components

System elements	MTBF, flight hours	Failure rate per 10^6 flight hours	Data source
ACT secondary actuator by parts:			
Power piston and servovalve	6.25×10^5	1.6	NAS1-14742
T-valve	10^5	10.0	Boeing data
LVDT servovalve	1.43×10^5	7.0	NASA CR-145271
LVDT power piston	1.43×10^5	7.0	NASA CR-145271
Solenoid bypass valve	1.66×10^5	6.0	Boeing data
Total secondary actuator	31 600	31.6	Sum of above
Single-wire segment	5×10^6	0.2	Boeing data
Connectors:			
10-pin stanchion connector (for Essential PAS)	11.0×10^6	0.09	MIL-HDBK-217B
20-pin stanchion	7.14×10^6	0.14	MIL-HDBK-217B
10-pin production break (uninhabited environment)	5.09×10^6	0.196	MIL-HDBK-217B
20-pin production break (uninhabited environment)	3.29×10^6	0.304	MIL-HDBK-217B
200-pin rack and panel computer connector	2.78×10^5	3.6	MIL-HDBK-217B
Secondary actuator mechanical voter	–	0	Multiple mechanical failures required to cause malfunction
Hydraulic power			
A	11 500	87.0	Boeing data
B	71 400	14.0	Boeing data
C	71 400	14.0	Boeing data
Pneumatic power (for AAL)	–	0	Supplied by both engines and accumulator
Electric power (per channel)	–	0.001	Subsection 9.2.1.2
Actuator, triple electric and duplex hydraulic:			
Electrohydraulic valves (four)	46 900	21.3	Boeing data on comparable parts
Power piston and servovalve (two)	6.25×10^5	1.6	NAS1-14742
Hydraulic bypass valve (two)	1.67×10^5	6.0	Boeing data
Total actuator, aileron, outboard, inner segment	5.55×10^5	1.8	A combination of the above, allowing for redundancy

Table 26. Failure Rates of ACT Components (Continued)

System elements	MTBF, flight hours	Failure rate per 10^6 flight hours	Data source
Actuator, aileron, outboard (inner segment)	5×10^5	2.0*	From above components
Actuator, secondary, outboard aileron outer segment	27 900	35.9*	Boeing data
Actuator flaperon	33 800	29.6*	Vendor MTBF
Actuator, secondary, elevator	25 900	38.6*	Boeing data
Actuator, secondary, rudder	26 700	37.4*	Boeing data
Actuator, stick pusher	20 000	50.0*	Boeing data on similar 727 thrust reverser actuator
Actuator, stick shaker	10^6	1.0	727 experience
Computer, essential (see subsec 9.2.1.3 for breakdown)	12 000	83.0	Vendor data for similar component
Computer, primary (see subsec 9.3.1.3 for breakdown)	6 620	151.0*	Vendor data
Computer, management	6 620	151.0*	Vendor data
Sensor, accelerometer, FMC	18 400	54.35*	Boeing data
Sensor, accelerometer, GLA	18 400	54.35*	Boeing data
Digital air data computer (DADC) (see subsec 9.2.1.3 for breakdown)	11 765	85.0*	Vendor MTBF
Sensor, inertial reference system	2 392	418.0*	Vendor MTBF
Sensor, angle of attack (used with DADC, see subsec 9.2.1.3)	15 400	65.0*	727 experience
Sensor, flap position (LVDT)	91 000	11.0*	NASA CR-145271
Sensor, wheel position (LVDT)	91 000	11.0*	NASA CR-145271
Valve, solenoid, stick pusher	65 400	15.3*	Boeing data

*Including connections and wires

provides bleed air from both engines. An additional backup is the low-pressure accumulator. Because this system is a simple adjunct to the necessary engines, it was felt that any additional airplane unreliability caused by the AAL use of pneumatic power would be negligible.

Hydraulic power is essential to safe flight of the Baseline Airplane. A ground rule has been that the ACT unreliabilities are to be the incremental unreliabilities beyond those of the Baseline Airplane. However, because the three hydraulic systems are used in many places in ACT, the reliability calculations are made more meaningful if the contributions

Table 27. Partitioned Failure Rates

Line replaceable unit and its component functions	Failure per 10^6 flight hours	MTBF, flight hours
Primary computer	151.0*	6 620
CPU and memory	40.6*	24 600
Servoamplifiers	17.8*	56 200
Input set, including:		
Cross-channel communication	66.6*	15 000
Common parts	26.0*	38 500
Digital air data computer	85.0*	11 800
α sensor (analog) (includes α vane at 65)	70.8*	14 100
q sensor (analog)	44.6*	22 400
Common parts	18.2*	55 000
Digital (computer) parts	18.2*	55 000
Inertial reference system	418.0*	2 392
Yaw-rate sensor (analog)	20.3*	49 300
Pitch-rate sensor (analog)	20.3*	43 900
Normal acceleration (analog)	31.7*	31 500
Computer parts	118.5*	8 440
Common parts	31.0*	32 200

*Including connections and wires

from the three hydraulic systems are included; hence, the differing hydraulic system loss rates of Table 26 are included in the fault trees as multiple-occurring events.

The electric system has been especially designed to provide an extremely low probability of loss of all four electric channels and of loss of individual channels. Each channel is supported from both an airplane dc bus and a battery. No two channels have the same two sources. The airplane dc buses support each other and are themselves powered from ac buses that have multiple sources of supply. A fault tree illustrating the failure paths necessary to produce channel failure is shown in Figure 113. Input events shown in double circles are sets of components combined to simplify the drawing. Figure 114 shows components of these sets, and Table 28 lists failure rates used for the components. Analysis of the fault tree by the FTREE program (vol. II, app B) yielded the failure rates in Table 29 for several different takeoff conditions. It should be understood that when

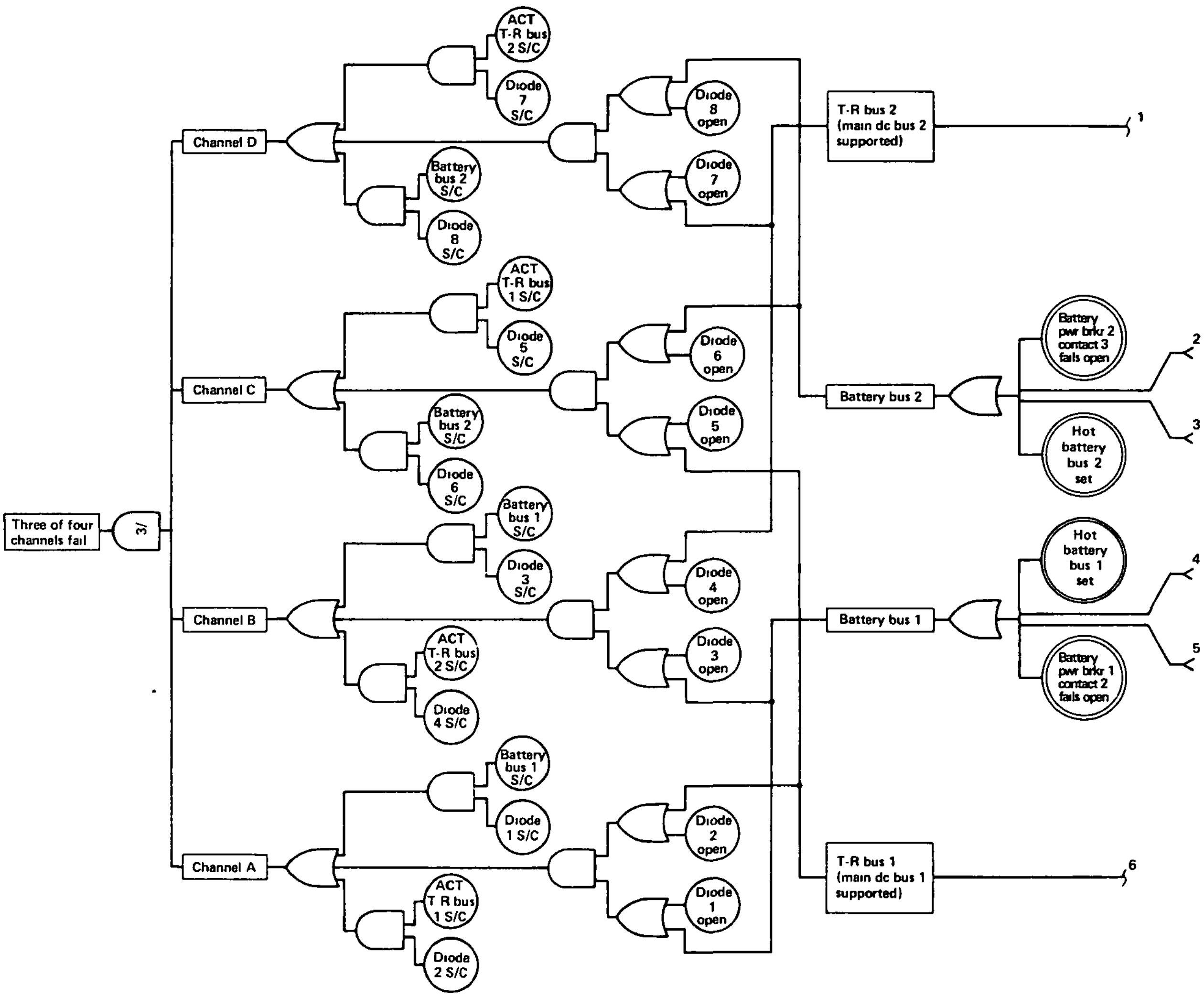


Figure 113. Fault Tree for Electric System

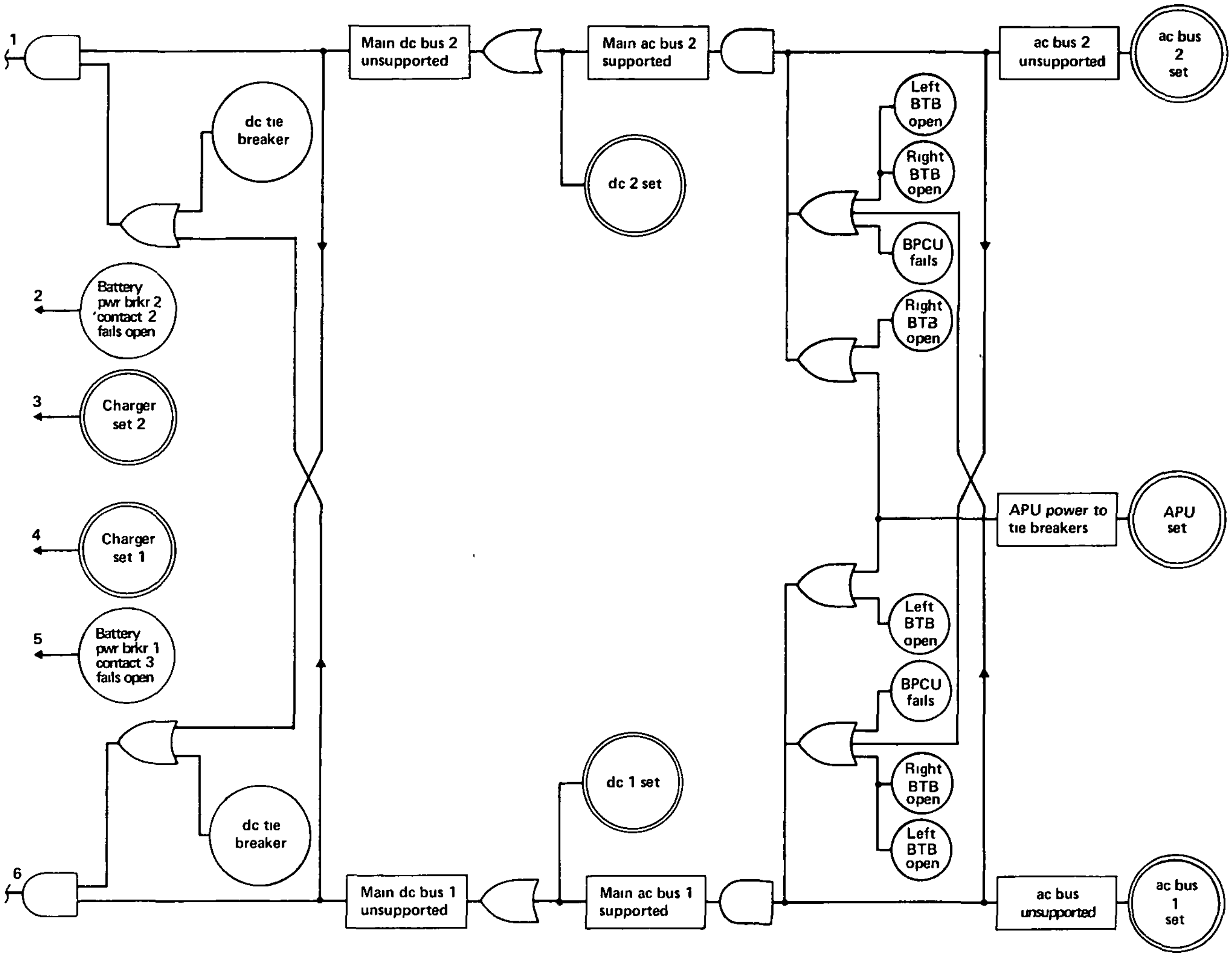


Figure 113. Fault Tree for Electric System (Continued)

failure rates such as 10^{-14} or 10^{-16} appear, they represent only "less than 10^{-10}." The numbers shown are presented for comparison but should not be interpreted as absolute values.

The electric system enters into the ACT systems in many places and in very involved ways. To consider this dependency when calculating the ACT function would have been extremely costly in time and computer use. Because the failure rates are small, the electric system does not contribute significantly to the ACT failure rate. Accordingly, it was not necessary to enter electric failure rates into the fault trees.

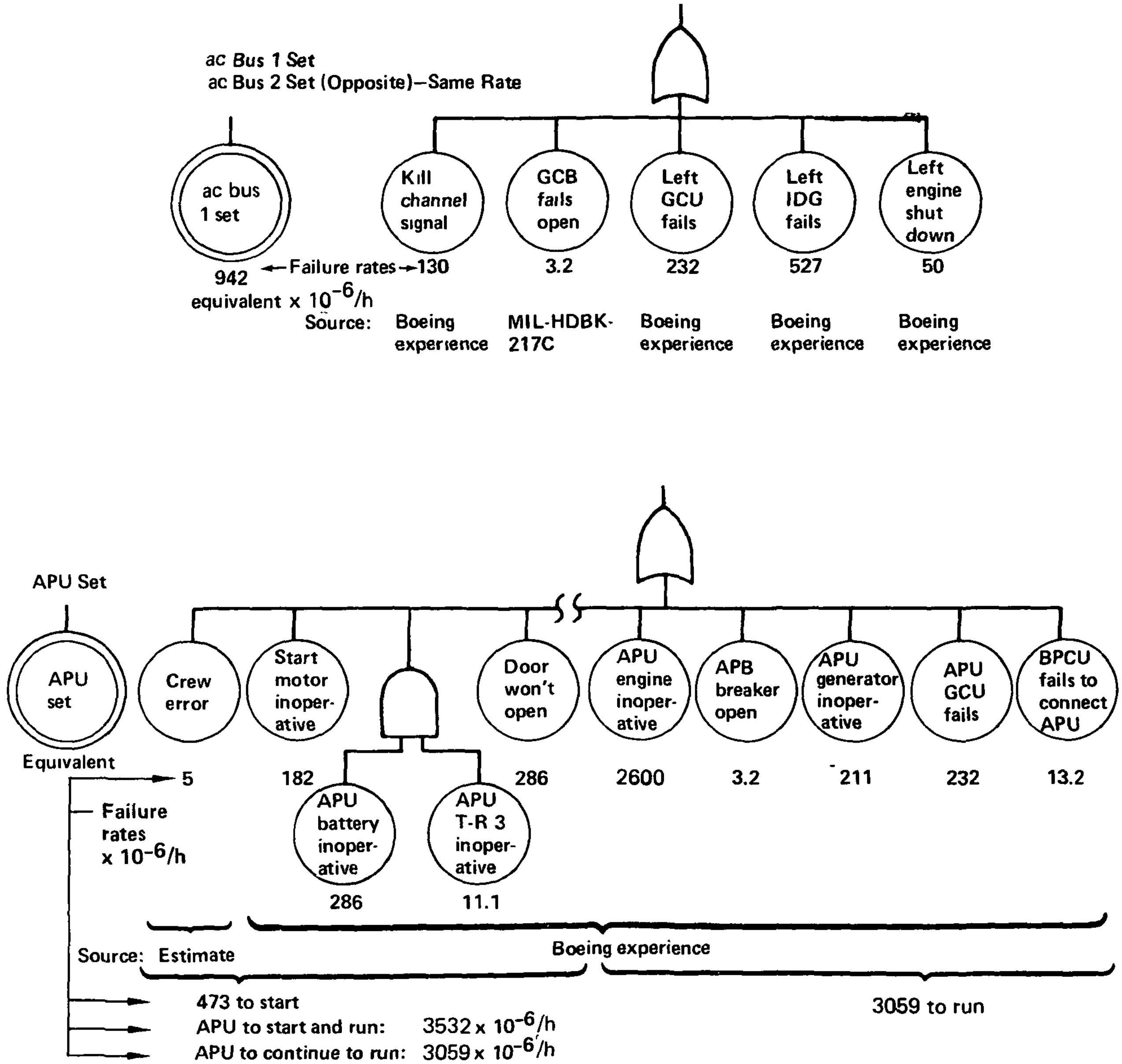


Figure 114. Sets of Components Reducible to One Component

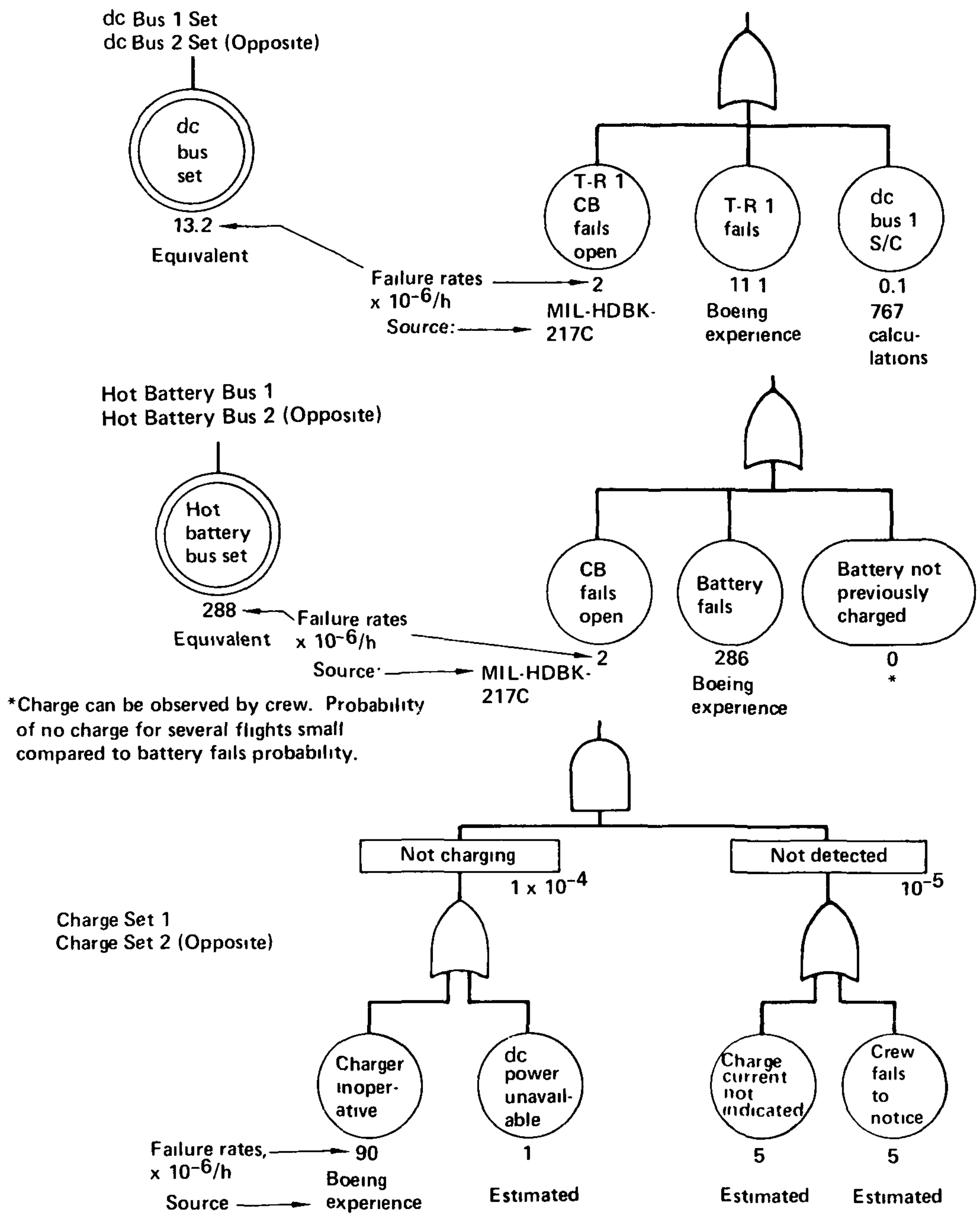


Figure 114. Sets of Components Reducible to One Component (Continued)

Table 28. Electric Power System Component Failure Rates

Failure probability, x 10^{-6}/h	Source of failure rate	Name of event
942.0	See Figure 114, ac bus set	ac bus 1 set
942.0	See Figure 114, ac bus set	ac bus 2 set
3532.0	See Figure 114, APU set	Auxiliary power unit—fail to start and run
3059.0	See Figure 114, APU set	Auxiliary power unit—fail to continue to run
40.0	Boeing experience and Rome Air Development Center (RADC)	Over 90-kVA load, switch to one generator, excess not shed
3.2	Boeing experience and RADC	Left bus tie breaker—fails open
3.2	Boeing experience and RADC	Right bus tie breaker—fails open
93.0	Boeing calculation	Bus power control unit—all failure modes
13.2	See dc bus set	dc bus 1 set
13.2	See dc bus set	dc bus 2 set
3.2	Boeing experience and RADC	dc tie breaker—fails open
288.0	See Figure 114, hot battery 2 set	Hot battery bus 1 set
288.0	See Figure 114, hot battery 2 set	Hot battery bus 2 set
3.2	Boeing experience and RADC	Battery power breaker 1 contact 3—fails open
3.2	Boeing experience and RADC	Battery power breaker 1 contact 2—fails open
3.2	Boeing experience and RADC	Battery power breaker 2 contact 3—fails open
0.2	Boeing experience and RADC	Battery power breaker 2 contact 3—fails open
1.0	MIL-217—all modes	Computer power input diode 1—fails open
1.0	MIL-217—all modes	Computer power input diode 2—fails open
1.0	MIL-217—all modes	Computer power input diode 3—fails open
1.0	MIL-217—all modes	Computer power input diode 4—fails open
1.0	MIL-217—all modes	Computer power input diode 5—fails open
1.0	MIL-217—all modes	Computer power input diode 6—fails open
1.0	MIL-217—all modes	Computer power input diode 7—fails open
1.0	MIL-217—all modes	Computer power input diode 8—fails open
0.1	Boeing estimate	ACT T-R bus 1—short circuit to ground
0.1	Boeing estimate	ACT T-R bus 2—short circuit to ground
0.1	Boeing estimate	Battery bus 1—short circuit to ground
0.1	Boeing estimate	Battery bus 2—short circuit to ground
1.0	MIL-217C—all modes	Computer power input diode 1—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 2—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 3—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 4—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 5—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 6—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 7—short circuit across junction
1.0	MIL-217C—all modes	Computer power input diode 8—short circuit across junction
1.0	Arbitrary value to examine sensitivity	(Probability the charger failed in previous flights and was
1.0	Arbitrary value to examine sensitivity	not noticed by crew)

Table 29. Electric System Failure Rates

Takeoff condition	Propability of all ac channels inoperative (i. e., probability of being dependent upon battery alone)	Probability of one channel inoperative	Probability of three of four ACT channels inoperative (system failure)
Two engine-driven systems on line and APU operable but not running	3.3×10^{-9}	2.97×10^{-10}	2.91×10^{-16}
Two engine-driven systems on line and APU inoperable	–	4.63×10^{-10}	4.95×10^{-14}
No. 2 engine-driven system and APU on line; other engine-driven system inoperable	–	8.35×10^{-10}	1.6×10^{-13}

9.2.2 IN-FLIGHT SCHEDULE RELIABILITY

Reliability of the Selected System during in-flight operation has been predicted using the BCAC fault tree (FTREE) model as described in Volume II, Appendix B (subsec B.4.2). The results are presented in Tables 30 through 32. This assessment differs in two significant ways from those conducted for the Integrated and Segregated Systems. First, the selected crucial short-period PAS electronics is mechanized in the separate, four-channel, independent hardware and software of an Essential PAS backup function, while Full PAS is performed by triple ACT Primary Computers. Second, the major sensors and the computer LRU failure rates have been allocated down to the section or module level to obtain more realistic function failure probabilities (subsec 9.2.1.1).

9.2.2.1 ACT System Reliability

Table 30 shows the probability of occurrence of flight restriction and flight diversion for the Selected System and provides data previously developed for the Integrated and

Table 30. Probability of Occurrence of Flight Diversion and Flight Restriction

	Probability of event occurrence in 1-hr flight		
Flight envelope change	Integrated System	Segregated System	Selected System
Flight restriction	2.51×10^{-3}	3.27×10^{-3}	1.73×10^{-3}
Flight diversion	7.08×10^{-4}	2.56×10^{-4}	4.00×10^{-4}
Flight diversion not including contribution from crucial PAS	Not applicable	Not applicable	4.00×10^{-4}

Table 31. Probability of Failure of Individual ACT Functions

	Probability of failure in 1-hr flight		
ACT function	Integrated System	Segregated System	Selected System
PAS (short or Essential)	3.49×10^{-10}	2.46×10^{-11}	5.56×10^{-12}
PAS (speed)	1.67×10^{-7}	1.67×10^{-7}	–
PAS (Full)	–	–	1.73×10^{-7}
LAS	1.13×10^{-6}	3.88×10^{-6}	2.93×10^{-8}
FMC	3.93×10^{-6}	3.88×10^{-6}	2.86×10^{-7}
WLA	1.21×10^{-5}	1.21×10^{-5}	5.15×10^{-7}
AAL	4.26×10^{-7}	1.13×10^{-6}	2.97×10^{-7}
AAL, inadvertent operation	–	–	5.96×10^{-10}
AAL, both stick pusher and stick shaker	–	–	2.55×10^{-7}

Segregated Systems. The Selected System has lower probability of occurrence than the comparable Integrated System because failure rate allocations have been made at the module level rather than at the LRU level. The gain schedule signal in the ACT PAS and lateral/directional-augmented stability (LAS) functions is furnished by either the DADC or by flap position, and single-thread operation is permitted for the LAS function.

Table 32. Selected Configuration Studies

Condition different from Selected System being evaluated	Probability of event at left per 1-hr flight	Probability of Selected System for comparison*
ACT system evaluated without flap position as backup		
Probability of flight restriction	2.10×10^{-3}	1.73×10^{-3}
Probability of flight diversion	7.68×10^{-4}	4.00×10^{-4}
ACT system evaluated without flap position and IRS as backup		
Probability of Full PAS loss	3.11×10^{-7}	1.73×10^{-7}
Probability of flight restriction	2.15×10^{-3}	1.73×10^{-3}
Probability of flight diversion	7.68×10^{-4}	4.00×10^{-4}
LAS single-channel operation		
Probability of function loss without single-thread operation capability (baseline)	3.62×10^{-7}	2.93×10^{-8}
Probability of function loss using servo output channel switching	3.55×10^{-8}	2.93×10^{-8}

*Selected System incorporates flap position as backup for DADC for gain scheduling, IRS pitch rate in Full PAS is backed up by four highly reliable pitch-rate instruments in crucial PAS, and LAS is permitted to degrade to single-channel operation.

9.2.2.2 Individual Function Reliability

The ACT system individual function reliability requirements defined in the ground rules and this subsection have been met by the Selected System. Table 31 shows the individual ACT function reliability analysis results. The crucial ACT functions of Full PAS and inadvertent stick pusher operation show failure probabilities of less than 1×10^{-9} per 1-hr flight. All other individual function failure probabilities are also satisfactory. An assessment of the combined Baseline Airplane stick shaker and the ACT system AAL with stick pusher resulted in an essentially unchanged probability value. This is not an unexpected outcome, as both the stick shaker and the stick pusher actuators receive commands from the same computer and sensor combination. The probability that the stick pusher will not operate when required must also be less than 1×10^{-9} per 1-hr flight. The combination of probabilities that would result in this condition is (1) the probability that the airplane is at an airspeed and aircraft attitude that represents incipient stall, (2) the probability that the stall warning system has failed or the flight crew does not act upon the stall warning, and (3) the probability that both stick pusher actuators fail to operate. This potential system failure probability is assessed as extremely remote.

9.2.2.3 System and Function Reliability for Selected System

A significant part of the analysis conducted on the Selected System addressed potential reliability improvements available from reconfiguration. The question of single-channel operation (single thread) in LAS was investigated because of the potential improvement in dispatch reliability if dispatch were to be permitted with one channel inoperative (subsec 9.2.3). Function failure probabilities for three computer and actuator configurations were assessed. First, the basic three-computer, two-actuator model requiring two operating channels was evaluated to establish a baseline. Next, the LAS function was modeled to evaluate single-thread operation by adding computer and sensor failure rates modified to evaluate covered as well as noncovered component failures. Coverage as used herein is the probability that the system will correctly detect a component failure and then successfully reconfigure the system. This became the Selected System. Finally, LAS was modeled to permit switching either computer servo output channel upon computer failure by incorporating a computer servo output logic and a switching circuit. Results of the single-channel operation study are presented in Table 32. As expected, the failure probabilities of the two computers hardwired to the actuators and the two actuator failure probabilities dominate the LAS function failure probability. The improvement in failure rate achieved by use of the servo output switch is tenfold. The full single-channel configuration is less than 18% better. Subsection 9.2.3 discusses the benefits in dispatch reliability accruing from single-channel operation.

Also investigated were the reliability improvements obtainable from providing backup sensor signals for the DADC outputs. Table 32 shows results of these studies. Probability of occurrence increased by 21% for flight restriction and 92% for flight diversion without flap position signals as backup inputs. With both the IRS pitch attitude and flap position signals deleted as backup inputs in the models, probabilities were calculated for the Full PAS function loss and for flight restriction and flight diversion. The Full PAS loss probability increased by 79%, and the flight restriction probability increased by 24%. There was no change in flight diversion probability over the calculation without flap position signals as backup inputs alone, because the failure rate contribution of the IRS pitch attitude signal in this model is much smaller than either the DADC with flap position or primary computer failure rates. However, the results of these configuration studies verify the benefit of including backup signals for the major system sensors to attain lower flight schedule interruption rates.

9.2.2.4 Assumptions of Reliability Analysis

Because no adequate modeling technique was available to accommodate all logic complexities of ACT, the following simplifying assumptions were made:

- Software reliability equals 1.0.

- All circuits are verified serviceable prior to each flight.

- Coverage in going from quadruple to triple and from triple to dual redundancy equals 1.0.

Such assumptions are questionable in the design of crucial digital flight hardware. Boeing is collaborating with Raytheon to assess whether the computer-aided reliability estimates (CARE III) model could properly model digital computers driving highly complex, critical flight control systems. The Boeing fault tree model (vol. II, app B, sec B.2.0) can list and predict the independent probability of each minterm. It appears that this information can be used by CARE III to account for latent failures and degrading coverage, so Boeing is presently transferring the FTREE model to Raytheon to provide the input end of CARE III. If CARE III is available during the next phase of the IAAC Project, this combined program may provide a more realistic prediction for the Selected System and for FBW modifications. At the present time, it appears that the Selected System will be able to meet the "extremely improbable" criterion for catastrophic failure because of the independent hardware and software provided by the Essential PAS backup function and because of the very short failure recovery time inherent in the Selected System architecture. This reinitialization capability should provide the answer to a large part of the transient hardware and software problems of present digital computers. Over 90% of computer failures are not traceable to hardware defects (ref 7), and the impact of such interruptions on ACT operability can be avoided by rapid, automatic recovery from temporary faults.

9.2.3 DISPATCH RELIABILITY

The prediction of dispatch reliability (the probability that the airplane may be dispatched without delay in excess of 15 min) can be made by either of two methods:

a. By an analysis based on the probability of a required ACT function being made inoperable by failure of a component

b. By comparison to delay rates experienced in commercial service resulting from failure of components that are similar to the ACT components

Although the FTREE program (vol. II, app B, sec B.2.0) can compute the probability as in item a (that an airplane would be in an undispatchable condition upon landing), the program cannot assess the impact of the time required to troubleshoot, repair, or replace; nor does it account for the different maintenance time available for a through-stop, a turnaround, or an overnight. All of these data and more are required to determine whether there will be a dispatch delay, and such data are not readily available.

The second method requires extensive actual airline experience data on the number and duration of delays charged to particular components similar to ACT hardware in both function and minimum equipment list (MEL) requirements. The MEL identifies those components that can be inoperable without precluding dispatch. Previous experience has shown that only the second method provides accurate predictions, and it is therefore used here.

A simplifying assumption used in the calculations was that if any component is part of a redundant set, not all of which is needed for dispatch, it does not contribute significantly to dispatch delays or cancellations. This is because the probability of two failures is so much lower than that of a single failure. These calculations cover only the increment in dispatch delays and cancellations produced by incorporation of the ACT system into the Baseline Airplane.

Table 33 shows the ACT component to be analyzed, then lists hardware, currently in airline service, chosen to approximate the ACT component and the airplane type in which it is used and from which the actual delay and cancellation rates experienced were calculated. The following correction factors are used as indicated in the table:

- Number per Airplane—The ratio of the number of components in the ACT airplane to the number of similar components in the airplane in service.

- Flight Length Factor—The ratio of 1 hr, assumed as the ACT airplane duration, to the average flight duration of the inservice airplane.

- Removal Rate Factor—The ratio of anticipated removal rate of an ACT component to the experienced removal rate of an inservice component. This value may be estimated by the ratio of failure rates.

The sum of the interruption rates for each of the listed components represents the LRU total; i.e., the total interruptions traceable to the particular ACT LRUs. Experience has shown that the total airplane interruptions for an automatic flight control system are about 1.3 times as great as the sum of all interruptions traceable to particular LRUs

Table 33. Tabular Solution to Dispatch Reliability

			A	B	C	D	E	F	G = A D E F	H = B D E F	J = C D E F	
Name of part added to (deleted from) Baseline Airplane	Comparison part assigned serial numbers and part names	Comparison airplane	Baseline (per 1000 departures)			Number per airplane factor	Flight-length factor	Removal rate factor	ACT airplane (per 1000 departures)			Remarks and MEL requirements
			Inter-ruptions	Delay hours	Cancel-lations				Inter-ruptions	Delay hours	Cancel lations	
Primary ACT computers (three)[a]	34-12-130-011 Digital air data computer	DC-10	0 116	0 097	0 0	3/2	0 58	1 10	0 111	0 093	0 0	Assumes no MEL dispatch for DADC or ACT Primary Computers
Essential PAS computers (four)	34-12-130-011 Digital air data computer	DC-10	0 116	0 097	0 0	4/2	0 58	0 60	0 081	0 068	0 0	Assumes no MEL dispatch for DADC or Essential PAS Computers
Elevator secondary actuators (three)	27-31-675-051 Elevator PCU	747	0 0277	0 095	0 0	3/4	0 36	0 91	0 007	0 023	0 0	No MEL dispatch
WLA primary actuators (two)	27-11-675-011 Aileron actuator	747	0 0235	0 0434	0 0149	2/4	0 36	1 0	0 004	0 008	0 003	No MEL dispatch
Elevator PCUs (two)	27-31 675-051 Elevator PCU	747	0 0277	0 095	0 0	2/4	0 36	0 91	(0 005) (Negative	(0 016) Negative	0 0	Net reduction
ACT hydraulic lines	27-21-280-191 Rudder hydraulic lines	727	0 0021	0 0094	0 0	4/1	1 0	1 0	0 008	0 038	0 0	No MEL dispatch
DADC on ACT (three)[b]	34-12-130-011 Digital air data computer	DC-10	0 116	0 097	0 0	3/2	0 58	1 0	0 101	0 084	0 0	No MEL dispatch[b]

[a]If LAS function is configured for single-thread operation, it is possible to dispatch with one primary computer inoperative

[b]When DADC is backed up with pitch angle and flap position, it is possible to dispatch with one DADC inoperative

because of interface problems. Therefore, the ACT system totals are computed as 1.3 times the LRU totals.

Delay and cancellation rates are significantly impacted by the following hardware options:

- Pitch Angle and Flap Position Signals Substitute for DADC—With this system configuration, dispatch is allowed with one DADC failed.

- Single-Thread Operation of LAS—Because single-thread LAS is allowed under flight restrictions, dispatch is possible with one primary computer failed. This implies the possible need to exchange primary computer positions during preflight preparation, and this is considered acceptable.

Three combinations of the preceding options for the Selected System were evaluated. The results are shown in Table 34 and compared with similar versions of the Integrated System and Segregated System dispatch reliability. Figures in parentheses show the interruptions, delay hours, and cancellations as a percentage of those of the basic Selected System without the beneficial impact of such additional dissimilar redundancy. The delay rate allowed under the design requirements and objectives (DRO) for the Baseline Airplane was 1.3%. The ACT system is not to add more than 5% to that. Thus, the allowable limit is 0.65 delays per thousand departures (5% of 1.3% of 1000). All combinations shown in Table 34 fall below this limit.

Data presented for interruptions and delay hours show that the Segregated System has the lowest schedule interruption times. This reflects the fact that most delays, like diversions, result from a fault that would cause a combination of ACT functions to be inoperable. The Segregated System, with different computers for each ACT function, is less prone to simultaneous component failures than the other systems. The smaller difference between the Integrated and the Selected System delays results from using four high-reliability dedicated pitch-rate sensors in short-period PAS instead of inputs from the IRSs.

The Selected System has an important advantage over the Integrated System that cannot be quantified because of lack of comparable commercial transport experience. It is anticipated that rigorous retest will be required after any maintenance work on any

Table 34. Predicted Delay Rates

Systems and assumptions	Per 1000 departures					
	LRU totals			System totals (LRU total x 1.3)		
	Interruptions	Delay hours	Cancellations	Interruptions	Delay hours	Cancellations
Basic Selected System but: •No DADC backup •No single-thread LAS*	0.307 (100%)	0.298 (100%)	0.003 (100%)	0.399 (100%)	0.387 (100%)	0.009 (100%)
Selected System •With DADC backup •No single-thread LAS	0.206 (67%)	0.214 (72%)	0.003 (100%)	0.268 (67%)	0.278 (72%)	0.004 (100%)
Selected System •With DADC backup •With single-thread LAS**	0.095 (31%)	0.121 (41%)	0.003 (100%)	0.124 (31%)	0.157 (41%)	0.004 (100%)
Integrated System •With DADC backup •No single-thread LAS	0.292 (95%)	0.265 (89%)	0.0055 183%	0.380 (95%)	0.344 (89%)	0.0072 183%
Segregated System •With DADC backup •No single-thread LAS	0.122 (40%)	0.128 (43%)	0.0055 183%	0.159 (40%)	0.167 (43%)	0.0072 183%
Limit under DRO	0.5	–	–	0.65	–	–

* Figures in parentheses show the interruption, delay, or cancellation rates as a percentage of those for the basic Selected System that has no DADC backup and no single-thread LAS.

**The Selected System with DADC backup and with single-thread LAS was used in calculations of cost of ownership.

portion of the crucial short-period PAS system. Because the Integrated System shares computers and pitch-rate sources between the crucial and critical systems, it is probable that any maintenance or removal of the computers or IRSs resulting from other ACT function maintainance would also require this time-consuming retest. The Selected System should avoid this penalty.

9.3 COST OF OWNERSHIP

Cost-of-ownership analysis predicts the incremental return on investment (ROI) that may be expected by the airlines for the Integrated, Segregated, and Selected Systems. This analysis enables present dollar values per flight hour to be calculated for economic parameters such as fuel cost saving, maintenance cost, spares inventory cost, and system purchase cost. This avoids the need for intuitive weighting factors (previously inherent in trade matrices) and removes subjective judgment from the design decision process. The ACT functions considered in this analysis were PAS, LAS, WLA, FMC, and AAL. Flaperons were not included in WLA.

9.3.1 COST-OF-OWNERSHIP MODEL

The Boeing-developed airline cost-estimating system (ACES) computer program was used in this analysis. For each future year, this program calculates the airline profit or loss that may be expected from the add-on ACT, then calculates the ROI to the airline based on the present equivalent value method. ACES considers the expected inflation rate, investment tax credit, depreciation credit, income tax, operating cost, etc., and shows parameters that have the greatest impact on ROI. It also establishes the payback point after which a positive cash flow (profit) to the airline may be expected.

9.3.2 COST-OF-OWNERSHIP ANALYSIS ASSUMPTIONS

The economic analysis is based on the following cost-of-ownership ground rules that are consistent with those used by Boeing for in-house studies on aircraft of similar size:

- Fleet size = 30 aircraft
- Airplane production run = 300 airplanes
- Minimum attractive ROI = 15%
- Tax depreciation life = 10 years
- Fleet life = 15 years
- Investment tax credit = 10%
- Cost per delay hour = $1400
- Cost per cancellation = $5100
- Spares holding cost = 10% of spares cost

- Yearly utilization = 3000 hr
- Average trip = 1.25 flight hours and 863 km (466 nmi)
- Yearly general inflation rate = 10%; yearly fuel inflation rate = 15%
- Insurance = 1% of purchase cost
- All costs in 1978 dollars

9.3.3 COST-OF-OWNERSHIP PARAMETERS

Table 35 shows the cost-of-ownership parameters used to calculate ROI to the airline. The calculated ROI and years to payback are also shown. These data are based on the ACT system configurations described in this document, except that flaperons are not included. This simplification was necessary to enable using the Initial ACT weight, drag, and fuel burn predictions when evaluating the configurations described herein (see ref 2).

Table 35. Cost-of-Ownership Results for Various ACT Systems

Parameter incremented	ACT technology base			
	Integrated	Segregated	Selected	Selected, pitch FBW
Aircraft purchase cost per aircraft	$274 000	$390 200	$297 100	$207 000
Maintenance manual cost per 30-airplane fleet	$21 000	$31 400	$26 100	$26 100
Test equipment cost per 30-airplane fleet	$22 500	$44 900	$33 600	$33 600
Spare inventory initial cost per 30-airplane fleet	$250 000	$356 000	$271 100	$271 100
Maintenance cost per aircraft flight hour	$4.18	$4.91	$4.22	$3.98
Departure delay and cancellation cost per aircraft flight hour	$0.54	$0.45	$0.19	$0.12
Change in system weight relative to Integrated System	0	+114 kg (+252 lb)	+14 kg (+30 lb)	-157 kg (-345 lb)
Fuel saving per flight hour at 863 km (466 nmi)	160 kg (352 lb)	146 kg (322 lb)	160 kg (352 lb)	172 kg (379 lb)
Payback period in years	2.83	4.14	2.98	2.02
Incremental return on investment to airline	25.1%	22.1%	24.6%	27.6%*

*Assumes 1985 introduction with fuel cost of $0.555/ℓ ($2.10/gal) and that fuel cost inflates at 15% per year against a general inflation rate of 10% per year.

Significant variations in cost-of-ownership parameters among ACT systems are discussed in the following items.

- Incremental Aircraft Purchase Cost—The Segregated System showed the highest cost because of the 21 small segregated computers used as compared with the 4 larger computers in the Integrated System. The Selected System cost is higher than the Integrated System cost because of four additional small computers and three extra dedicated pitch-rate sensors used to provide the standby, independently redundant, short-period PAS system. The Selected System with FBW in the pitch axis was the lowest in incremental cost because of the high parts count of the mechanical pitch signaling system, which was deleted. The parts deleted included the elevator feel mechanism, elevator pressure control modules, elevator feel computer, cables, bell cranks, push rods, pulleys, etc.

- Maintenance Manual Cost for 30-Airplane Fleet—This parameter was based on typical autopilot cost prorated for increasing electronic complexity. Thus, the Segregated System cost is the highest.

- Incremental Test Equipment Cost for 30-Airplane Fleet—This assumed that the airline would already possess the basic digital test equipment for the remainder of the digital electronic suite. Cost increment thus was based on the cost of additional adapters and test software required for ACT. The more electronically complex the ACT system, the greater the cost.

- Incremental Spares Inventory Initial Purchase Cost per 30-Airplane Fleet—This parameter was affected primarily by the higher cost, higher-removal-rate electronic equipment. The Segregated System cost was highest and the Selected System was next.

- Maintenance Cost per Flight Hour—This was calculated based on past experience and recent predictions at the significant LRU level. As expected, the 21-computer Segregated System cost was highest. The cost of the Selected System with pitch FBW was lowest because of the removal of elevator mechanical controls. The additional maintenance cost for ACT is less than that for a current autopilot system. This is because rate gyros, which account for up to 70% of maintenance cost of current systems, are eliminated, and vertical reference and rate data are provided by the IRS.

- Incremental Delay and Cancellation Cost per Flight Hour—Integrated System cost was highest because the loss of a single flight control computer or IRS prevented dispatch. The Segregated System showed some improvement because of the separation of functions and the adoption of the four reliable, dedicated pitch-rate sensors for the short-period PAS function. Hence, the aircraft could now dispatch with one IRS down. The Selected System with pitch FBW shows the lowest delay and cancellation cost because of the adoption of the flap position switch to back up the DADCs for the speed PAS function and the removal of the mechanical connections, feel computer, etc., between the pilot's control column and the elevator.

- ACT System Weight and Fuel Saving—Detailed performance data were derived only for the Initial ACT Integrated System because the aircraft of the later systems remained externally the same and used the same active control surfaces. Thus, only ACT system weight relative to the Integrated System would impact block fuel. The weight delta for the Selected System had an insignificant impact on fuel burn, but fuel savings for weight changes in the Segregated System and Selected System with pitch FBW were adjusted as indicated. The weight penalty for the Segregated System derived from its 21 computers and the addition of an extra battery to supply emergency dc power. The weight saving for the Selected System with pitch FBW resulted from deletion of mechanical components in the elevator pitch controls.

- Payback Period and Incremental ROI—The payback period for all systems was short, as desired, and the ROI to the airline exceeded the 15% minimum attractive rate of return established for this study. The ROIs shown in Table 35 are based on a service entry date of 1985 and assume that fuel inflates at 15% per year and other cost-of-ownership cost parameters inflate at 10% per year. The 1985 starting fuel price used was \$0.555/ℓ (\$2.10/gal), and all dollars are 1978 dollars to keep the present studies in line with the Initial ACT ROI reported in Reference 2. ROI is highly dependent on the relationship between average yearly fuel inflation rate and the inflation rates of aircraft cost, maintenance cost, etc. This is shown in Figure 115, where the fuel yearly inflation rate is varied between 15% and 25% while maintaining other inflation rates at 10%. It is apparent that ROIs much greater

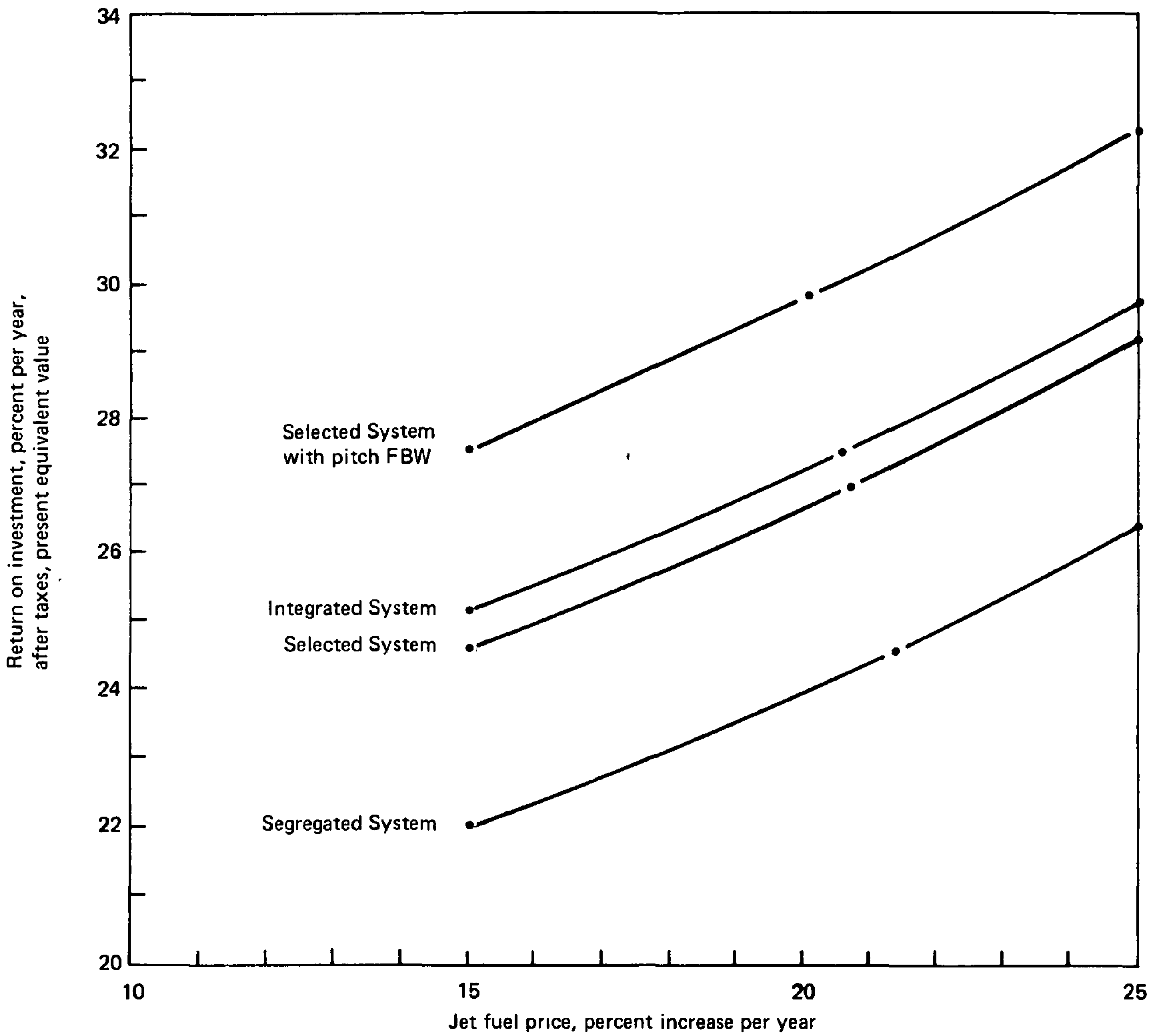


Figure 115. Return on Investment Versus Inflation Rate of Fuel

than those shown in Table 35 will be obtained if historical (since 1972) average annual fuel inflation rates persist into the future.

Figure 116 plots ROI against years to payback for the various ACT systems. Given equal technical risk for these current technology systems, the least financial risk will be provided by the system that gives the shortest payback period and the highest ROI; namely, the Selected System with pitch FBW at 2.02 years and 27.6% ROI.

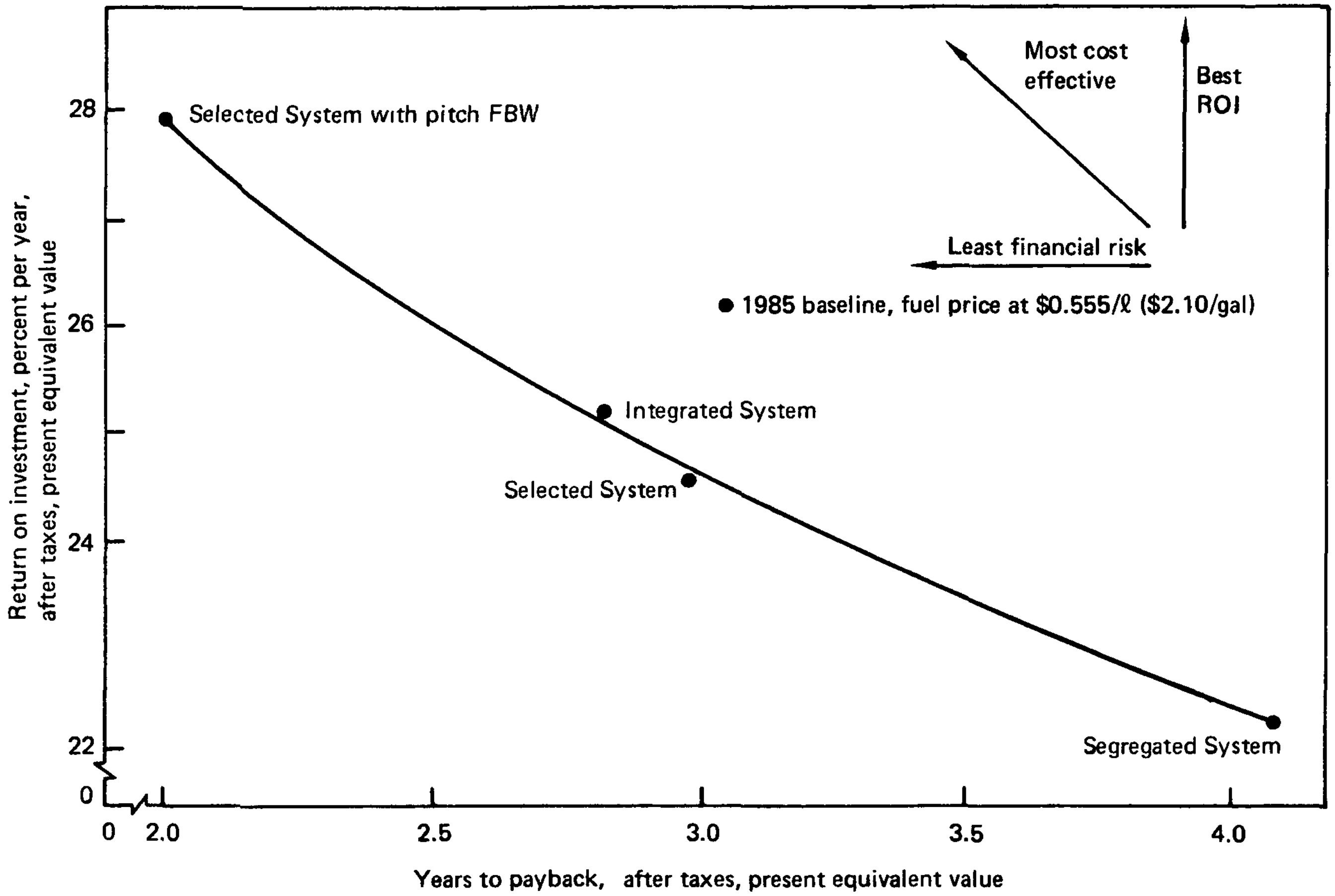


Figure 116. Time Required To Pay Back ACT Investment

9.3.4 HIGH-LEVERAGE COST-OF-OWNERSHIP PARAMETERS

Figure 117 shows the impact on incremental ROI of varying the incremental aircraft price and the fuel savings or fuel cost by 50% above and below nominal. It is apparent that the highest leverage exists for decreases in fuel cost and incremental aircraft cost.

9.3.5 COST-OF-OWNERSHIP CONCLUSIONS

The incremental ROI and the short payback period obtainable from the active control systems studied make them a highly attractive investment at expected fuel-price inflation rates.

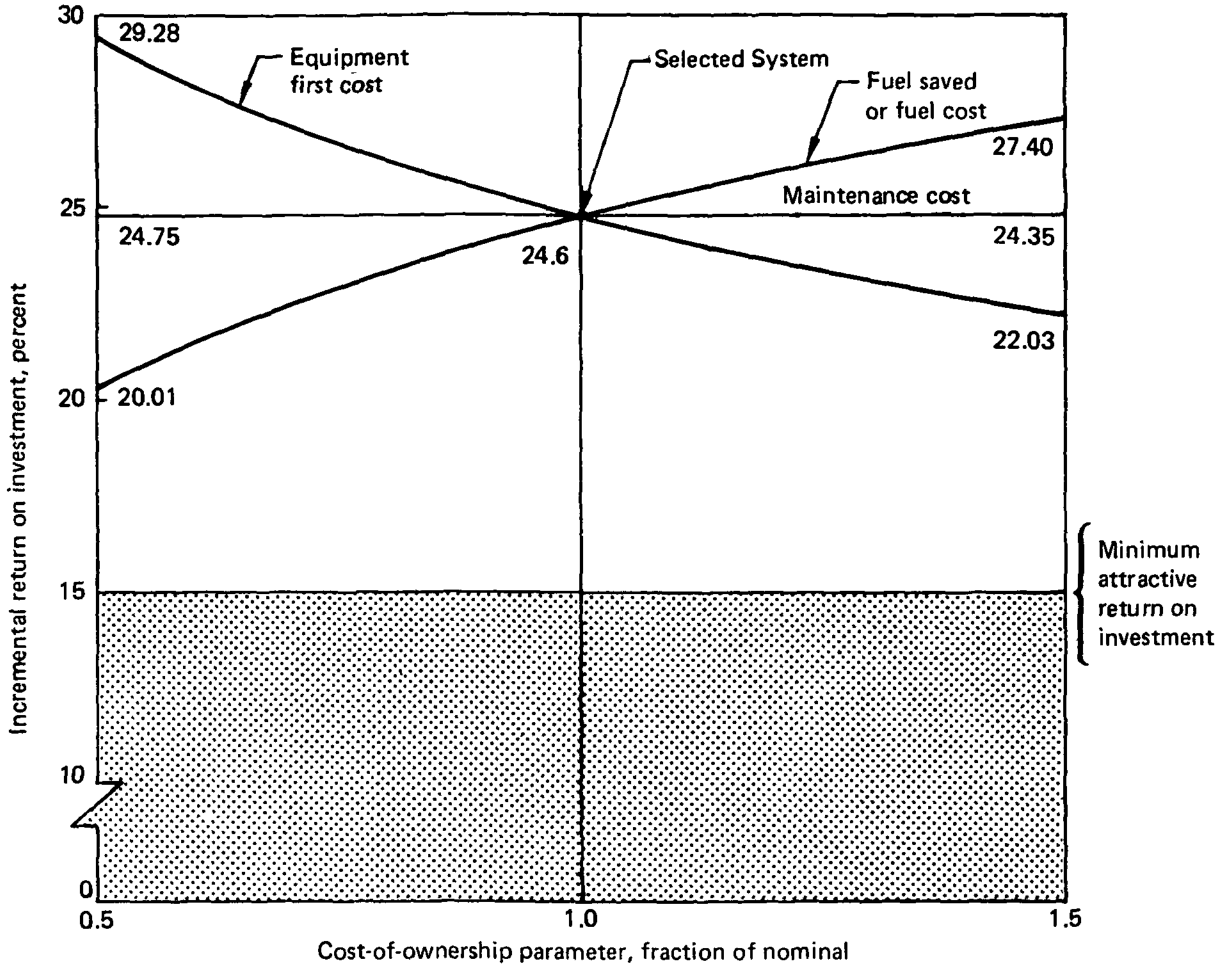


Figure 117. Effect of Changes in Cost Parameters on Incremental Return on Investment

It should be noted that the preceding statement applies to the Integrated, Segregated, and Selected Systems as a whole, because no attempt has yet been made to assess the incremental ROI for individual functions. Thus, it has been shown (ref 2) that PAS, by permitting a reduction in horizontal stabilizer area (weight and drag), accounts for about 90% of the reduction in fuel burn. Whether the addition of FMC and WLA would be cost effective has not yet been established. However, if they are not cost effective and therefore are excluded from the Initial ACT Configuration, the ROIs of the various systems can only increase.

Advanced technology studies indicate that FBW in all axes is highly cost effective. From this it can be concluded that if total FBW is adopted, most active control functions that may prove necessary can be provided at minimum additional cost.

10.0 ADVANCED TECHNOLOGY TASK OVERVIEW

The overall objective of the work performed under the advanced technology task was to define advanced control systems for future commercial transports. It is part of the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project, which is one element of NASA's Energy Efficient Transport (EET) Program. Figure 118 shows the Advanced Technology ACT Control System Definition Task breakdown. The work consisted of two tasks: Advanced System Trade Studies and Implementation Alternatives. The combined output from these two tasks constituted definition of representative advanced control systems.

The objectives of the Advanced System Trade Studies were to develop control law analysis and synthesis methods suitable for coupled multiloop systems and to demonstrate the

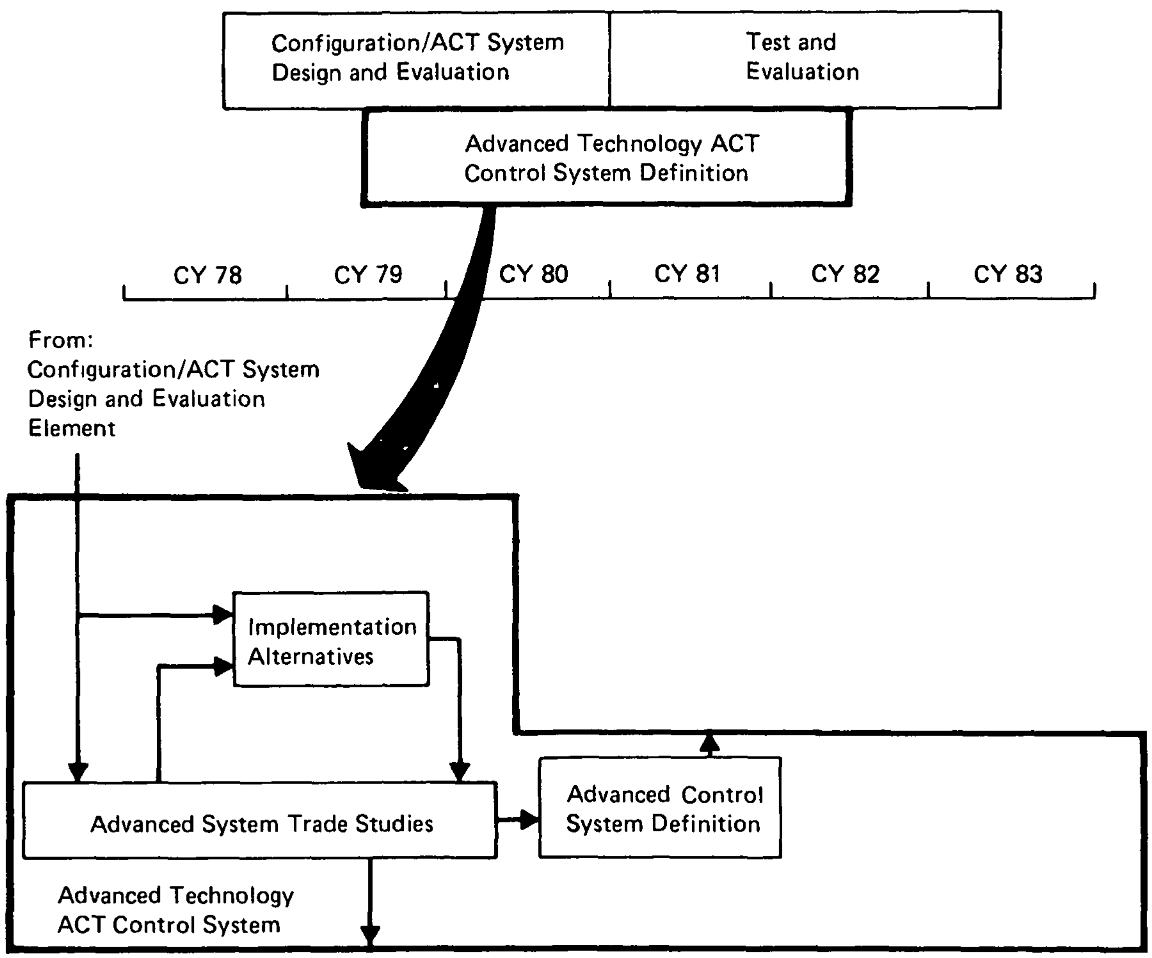


Figure 118. Advanced Technology Control System Task Breakdown

potential benefits of these by evaluating the closed-loop performance of a set of control laws. Gust-load alleviation (GLA), flutter-mode control (FMC), and rigid-body pitch stability and command augmentation control laws were synthesized. The methods employed were based on modern optimal control and estimation theory.

The GLA and FMC control law performance was evaluated based on wing-load and control surface responses to continuous random vertical turbulence, discrete vertical gusts, and control loop stability margins. The pitch augmentation control laws were evaluated based on pitch-rate and load factor responses to elevator commands.

The technical approach for the control law synthesis and performance evaluation is summarized in Section 11.0 and described in more detail in Volume II, Appendix E. The GLA and FMC control laws and the corresponding results from the performance analyses are summarized in Section 12.0, with a more extensive collection of performance data presented in Volume II, Appendix F. The pitch augmentation control laws and the results of the performance evaluations are presented in Section 13.0.

The objective of the Implementation Alternatives work was to identify advanced system alternatives to the current technology implementation. The total control system, including sensors, computers, servos, actuators, and data transmission, was considered. The work comprised two major tasks. First, sensors, computers, servos, actuators, and data transmission technology that might be expected to be available for a circa 1990 commercial airplane were projected. Second, the corresponding flight control system architecture system provides the same functions as the current technology system and was designed to provide improved performance and reliability with cost of ownership equal to or better than the conventional system. Alternative implementation is described in Section 14.0, with more details provided in Volume II, Appendix G.

11.0 ADVANCED TECHNOLOGY CONTROL LAW SYNTHESIS AND ANALYSIS METHODS

The following subsections summarize the technical approach used for control law synthesis and open-loop and closed-loop performance evaluations. Volume II, Appendix E, contains a more detailed description.

The complexity of the control task and the dynamic characteristics of a typical flexible transport airplane dictate the solution of a coupled multiloop control problem. The classical approach of synthesizing one loop at a time is not well suited to deal directly and efficiently with coupled multiloop systems or to take advantage of favorable interactions between the control loops.

The design was accomplished using a set of experimental computer programs on the CDC 6600 system. These programs are particularly suited for the analysis and synthesis of multivariable controllers for Active Controls Technology (ACT) airplanes and are based on time-domain modern control theory. Key elements are state-space representation of dynamic systems, modal analysis, and optimal control and observer synthesis. Figure 119 indicates the design process flow. The main elements are model generation, linear analysis, optimal controller design, and simulation.

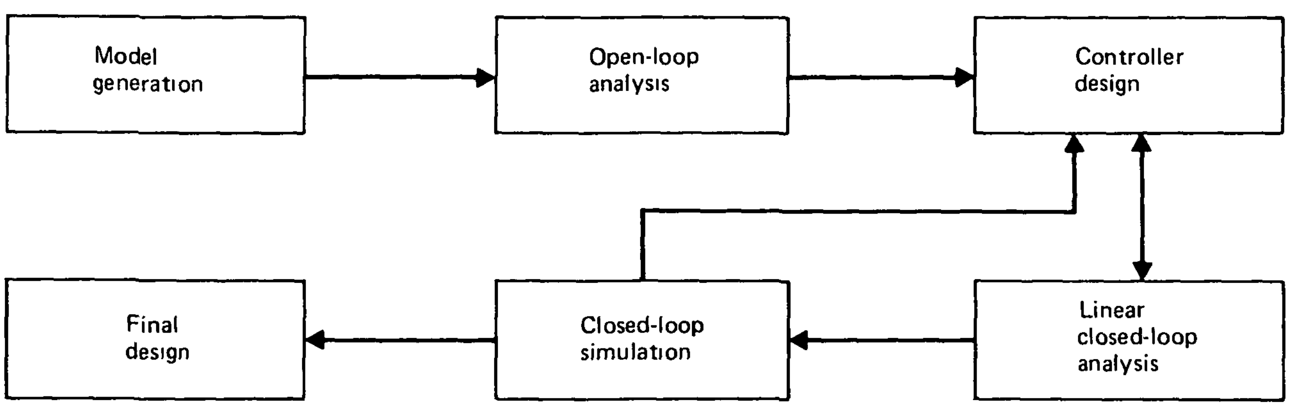


Figure 119. Control Design Procedure

11.1 DYNAMIC MODELS

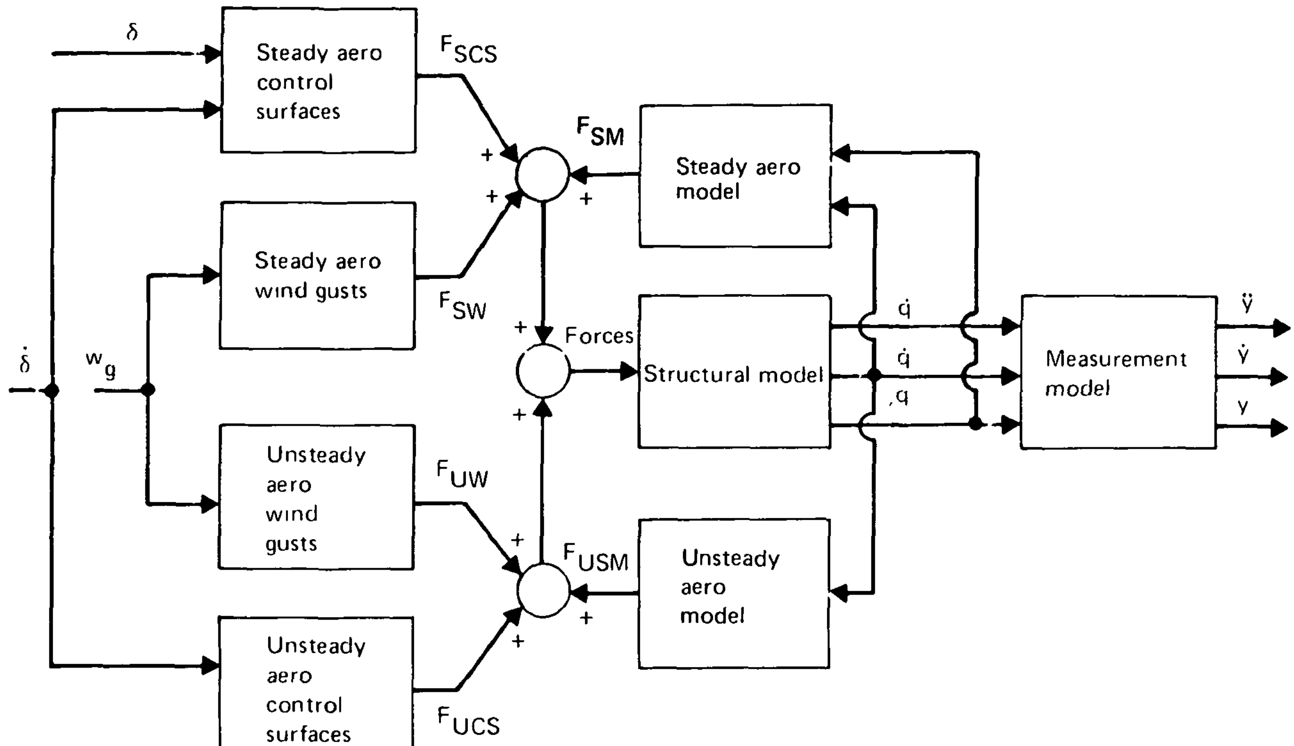
The control law synthesis and analysis require models of the flexible airplane, measurements and performance parameters, actuation system, and wind disturbances. These models are connected to perform open-loop analysis, control law synthesis, and, when combined with a control law, closed-loop performance evaluation.

11.1.1 FLEXIBLE AIRPLANE MODEL

The airplane at each flight condition is represented by a set of constant coefficient linear second-order differential equations modified by the addition of first-order lag terms. Figure 120 is a corresponding block diagram description, with blocks representing the steady and unsteady aerodynamic forces and the structural model. This diagram graphically describes how the model is put together in the computer using an automated model generation program. After the input-to-output relations of each block and the block connections are specified, a precompiler program generates Fortran subroutines that are combined to represent the complete model in program form. The individual blocks may represent nonlinear relationships. These models are used to perform static trim calculations, conduct simulations, and generate linear state models at specified operating points. These particular formulations produce a well-structured state vector consisting of $q(t)$, the rigid and elastic modal coordinates; $\dot{q}(t)$, the corresponding rates; and $\bar{w}_g(t)$, the unsteady aerodynamic states due to wind.

In modeling the airplane, an approximate transformation from frequency- to time-domain representations of the unsteady aerodynamic forces, except for forces due to gusts, is accomplished with a least-square fit of a second-order polynomial in the Laplace variable s, resulting in steady and unsteady aerodynamic forces as functions of displacements and the corresponding first- and second-order time derivatives. The unsteady effects associated with gust inputs are approximated with Kussner lift-growth functions. The outputs of these become states of the system. The state model of the flexible airplane takes the form

$$\dot{x}_a(t) = A_a x_a(t) + B_a u(t) + \Gamma_a w_g(t) \tag{1}$$



States:

q	Rigid- and flexible-mode deflections
$\dot{q}$	Corresponding rates
$\bar{q}$	Unsteady aerodynamic states associated with q
$\bar{\delta}$	Steady aerodynamic states associated with δ
$\bar{w}_g$	Unsteady gust states

F_{SCS}	Steady aero control surface forces
F_{SM}	Steady aero forces
F_{UCS}	Unsteady aero control surface forces
F_{USM}	Unsteady aero forces
F_{UW}	Unsteady aero wind gust forces

Figure 120. Model of the Flexible Airplane

where

$$x_a(t) = \begin{bmatrix} q(t) \\ \hdashline \dot{q}(t) \\ \hdashline \bar{w}_g(t) \end{bmatrix}$$

is the airplane state vector, u(t) is the control vector, and $w_g(t)$ is the gust velocity input vector.

11.1.2 OUTPUT MODEL

The needed outputs are the displacements, velocities, accelerations, and loads at various stations on the airframe. The first three items are related to the modal coordinates, q(t), and the mode shape matrix, ϕ.

The displacement, velocity, and acceleration measurements and the loads are all expressed in a general form of the state model output equation

$$y(t) = C_a x_a(t) + D_a u(t) + E_a w_g(t) \tag{2}$$

where y(t) is a vector of measurements and loads. The detailed development of the output equations is shown in Volume II, Appendix E, Subsection E.1.3.

11.1.3 ACTUATOR MODEL

Only linear actuator models have been considered in this study. They are expressed as state models in the standard controllable form

$$\dot{x}_{\delta_i}(t) = \begin{bmatrix} 0 & 1 & \ldots 0 \\ 0 & 0 & \ldots 0 \\ \vdots & \vdots & 1 \\ -a_0 & -a_1 & \ldots -a_{n-1} \end{bmatrix} x_{\delta_i}(t) + \begin{bmatrix} 0 \\ 0 \\ \cdot \\ \cdot \\ a_0 \end{bmatrix} \delta_{c_i}(t) \tag{3}$$

where $\delta_{c_i}(t)$ is the ith control surface command, $\delta_i(t)$ is the ith control surface position, $x_{\delta_i}(t)$ is the state vector of the actuator, and $a_0, a_1, a_2 \ldots , a_{n-1}$ are the coefficients of the actuator transfer function.

The general form of the model for the complete actuation system is

$$\dot{x}_u(t) = A_u x_u(t) + B_u u_c(t) \tag{4}$$

$$u(t) = C_u x_u(t) \tag{5}$$

where x_u is the state vector for all actuators, $u_c(t)$ is the command vector for all actuators, and u(t) is the control vector for all surfaces.

11.1.4 WIND MODEL

Because of its simplicity, the Dryden turbulence model for vertical gusts was selected to represent gust disturbances. In state variable form this becomes

$$\dot{x}_w(t) = A_w x_w(t) + B_w w_c(t) \tag{6}$$

$$w_g(t) = C_w x_w(t) \tag{7}$$

where x_w is the wind state vector, $w_c(t)$ is a white noise input vector, and $w_g(t)$ is the vector of vertical gust velocities at various airframe stations. The input is white noise, and the outputs are correlated gust velocities having the Dryden power spectral density.

11.2 MODAL ANALYSIS

The stability and response characteristics of a linear aeroelastic system represented by a state model are completely described by the associated eigenvalues, eigenvectors, and the input and output distribution matrices. From Subsection 11.1, the state model general form is

$$\dot{x}(t) = Ax(t) + Bu(t) + \Gamma\, w_g(t) \tag{8}$$

$$y(t) = Cx(t) + Du(t) + Ew_g(t) \tag{9}$$

The eigenvalues of the system are the values of λ_i (i = 1,n) that satisfy the equation

$$\det(\lambda I - A) = 0$$

where I is the identity matrix and det(-) means the determinant of the argument.

The eigenvectors v_i (i = 1,n) of the system are defined by the relation

$$Av_i = \lambda_i v_i \quad (i = 1,n)$$

If λ_i is a complex eigenvalue, then the corresponding eigenvector v_i is also complex. Because the state matrix A is real, it can be shown that complex eigenvalues and eigenvectors always occur in conjugate pairs. The complete eigensystem consists of n eigenvalues and the corresponding n eigenvectors.

It can be shown that there exists a similarity transformation

$$x(t) = Tz(t)$$

such that the preceding system of equations can be reduced to block diagonal form in terms of the modal coordinate vector z(t):

$$\dot{z}(t) = (T^{-1}AT)z(t) + (T^{-1}B)u(t) + (T^{-1}\Gamma)w_g(t) \tag{10}$$

$$y(t) = (CT)z(t) + Du(t) + Ew_g(t) \tag{11}$$

11.3 RESPONSE CALCULATIONS

11.3.1 COVARIANCE ANALYSIS

The frequency-domain power-spectral-density technique has been widely used to compute steady-state gust responses. This requires determination of a complex frequency response matrix relating gust-velocity inputs to output response variables and computation and integration of a large number of power and cross spectra. For a flexible airplane with a large number of lightly damped modes that are subjected to distributed random-gust inputs, these calculations may be costly and inaccurate. In this work, this technique has only been used to compute the output power spectral density for a limited number of parameters. For most gust response calculations, the total root-mean-square (rms) responses are of primary interest. These are obtained by using a new approach for computing the steady-state response correlation matrices of a dynamic system subjected to random inputs. This method avoids the computational difficulties and inaccuracies associated with lightly damped modes, with approximate gust penetration effects, and with the finite frequency range of integration. The calculations are performed using time-domain state-space representation of the airplane model. A transformed covariance matrix is obtained by computation of convolution integrals. The values of the integrals can be evaluated in closed form for white and Dryden spectra, among others. Gust penetration effects as the airplane traverses the field of atmospheric turbulence are modeled by pure time delays, avoiding the use of Pade approximations. The algorithm computes the steady-state gust response correlation matrices for the states and outputs and the corresponding rates with the state model in block diagonal form.

11.3.2 LINEAR SIMULATION

Simulations are performed using a block diagonal discrete-time state transition model. This model is derived from continuous-time state variable form

$$\dot{x}(t) = Ax(t) + Bu(t) \tag{12}$$

$$y(t) = Cx(t) + Du(t) \tag{13}$$

where y(t) is the vector of output response variables, and the state vector x(t) consists of the rigid and flexible airplane modes, actuator states, and controller states. The elements of the input vector, u(t), are the pilot commands and gust inputs.

The system represented by these equations is transformed to block diagonal form using the similarity transformation as described in Subsection 11.2. Assuming a constant value for input u(t) over an interval t to t + Δt, the transition of the system model response in that time interval can be expressed as

$$z(t + \Delta t) = \phi z(t) + \bar{\theta} u^* \tag{14}$$

where u* is a value of u(t) in the interval $[t, t + \Delta t]$, ϕ is the modal state transition matrix, and $\bar{\theta}$ is the integral of the modal state transition matrix over the interval $[t, t + \Delta t]$ multiplied by the modal input distributions matrix B. The output vector is expressed as

$$y(t) = C' z(t) + Du^* \tag{15}$$

If the responses of the original state vector x(t) are required, they are simply included in the output vector. For an nth-order system, the total number of operations required at each integration step is at most 2n multiplications and additions plus n multiplications and additions for each input. This compares with n^2 and n, respectively, for a system that is not block diagonalized. For high-order systems where the number of states is substantially higher than the number of outputs, there is a significant reduction in the computational time.

11.4 MODEL REDUCTION

The open-loop dynamic model of a flexible airplane must be simplified before the design of a practical controller. Also, any high-order Kalman filter that has been synthesized based on either a full-order or reduced-order open-loop model must, in most cases, be simplified before it is implemented in the flight computers. In both cases, the purpose is to reduce the order to a level consistent with computational capabilities while preserving the significant dynamic characteristics relative to the control objectives. Many techniques are available, but none will consistently produce accurate and meaningful results without a good understanding of the inherent physical relations behind the control task. The most suitable approach to the design of a low-order controller has not been resolved in this work. However, two methods and their particular application will be described.

11.4.1 DELETION OF NONESSENTIAL STATES

Figure 121 illustrates this method. The dynamic model is reduced in size by deleting rows and columns from the state model matrices. If the full model is of order n, and m states are deleted, the reduced model is of order n - m. This method is suitable for deleting nonessential states, such as airplane rigid-body displacement states that do not produce forces and moments or states associated with weakly coupled modes. For example, the phugoid mode can be eliminated from the model with insignificant impact on the remaining short-period and structural modes by deleting the forward velocity and pitch angle states.

11.4.2 MODAL RESIDUALIZATION

This method for model reduction is suitable for systems that include fast dynamics that are not significant with respect to the control task or that have uncontrollable or unobservable modes. This is typical for flexible airplanes that may have a large number

- Full-order state model.

$$\begin{bmatrix}\dot{x}_1\\ \dot{x}_2\\ \vdots\\ \dot{x}_k\\ \vdots\\ \dot{x}_n\end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & \cdots & A_{1k} & \cdots & A_{1n}\\ A_{21} & A_{22} & \cdots & A_{2k} & \cdots & A_{2n}\\ \vdots & \vdots & & \vdots & & \vdots\\ A_{k1} & A_{k2} & \cdots & A_{kk} & \cdots & A_{kn}\\ \vdots & \vdots & & \vdots & & \vdots\\ A_{n1} & A_{n2} & \cdots & A_{nk} & \cdots & A_{nn}\end{bmatrix}\begin{bmatrix}x_1\\ x_2\\ \vdots\\ x_k\\ \vdots\\ x_n\end{bmatrix} + \begin{bmatrix}B_1\\ B_2\\ \vdots\\ B_k\\ \vdots\\ B_n\end{bmatrix} u + \begin{bmatrix}\Gamma_1\\ \Gamma_2\\ \vdots\\ \Gamma_k\\ \vdots\\ \Gamma_n\end{bmatrix} w_g$$

$$y = \begin{bmatrix} C_1 & C_2 & \cdots & C_k & \cdots & C_n\end{bmatrix}\begin{bmatrix}x_1\\ x_2\\ \vdots\\ x_k\\ \vdots\\ x_n\end{bmatrix} + Du + Ew_g$$

- Delete rows and columns of matrix A
- Delete rows of matrix B
- Delete columns of matrix C

Figure 121. Deletion of Nonessential States

of stable high-frequency modes and a number of weakly controllable or weakly observable modes. The high-frequency modes are generally not modeled very accurately and in most cases need not be controlled to meet the closed-loop design objectives, except that the control loop gain at high frequency must be sufficiently low so as not to destabilize any high-frequency mode.

The flexible airplane is represented by a linear time-invariant system described by equations (10) and (11) in modal form. The state model is partitioned into two sets of modes, $z_1(t)$ and $z_2(t)$, as shown in Figure 122. Having ordered the modes so that the upper partition contains low-frequency modes and all unstable modes and the lower partition contains high-frequency stable modes, it is assumed that the modes $z_2(t)$ respond much faster than the modes $z_1(t)$ and that only the dynamics of $z_1(t)$ are important with respect to the control task. If $z_2(t)$ consists of i modes, the original nth-order system has been reduced to an (n - i)th-order system. The eigenvalues of the system are simply those of the retained modes, and the controllability and observability of these are unchanged from the original full-order model. The steady-state effects from the deleted modes $z_2(t)$ are included in the outputs y(t) through additional input terms.

- Full-order model (in block diagonal form)

$$\dot{z} = \begin{bmatrix} \dot{z}_1 \\ \dot{z}_2 \end{bmatrix} = \begin{bmatrix} \Lambda_1 & 0 \\ 0 & \Lambda_2 \end{bmatrix} \begin{bmatrix} z_1 \\ z_2 \end{bmatrix} + \begin{bmatrix} B'_1 \\ B'_2 \end{bmatrix} \mu + \begin{bmatrix} \Gamma'_1 \\ \Gamma'_2 \end{bmatrix} w_g$$

$$y = \begin{bmatrix} C'_1 & C'_2 \end{bmatrix} \begin{bmatrix} z_1 \\ z_2 \end{bmatrix} + D_a u + E_a w_g$$

- Assumption:

$$\dot{z}_2 \simeq 0$$

- Reduced-order model·

$$\dot{z}_1 = \Lambda_1 z_1 + B'_1 u + \Gamma'_1 w_g$$
$$z_2 = -\Lambda_2^{-1} B'_2 u - \Lambda_2^{-1} \Gamma'_2 w_g$$
$$y = C'_1 z_1 + (D_a - C'_2 \Lambda_2^{-1} B'_2) u + (E_a - C'_2 \Lambda_2^{-1} \Gamma'_2) w_g$$

Figure 122. Modal Residualization

11.4.3 LEAST-SQUARE ERROR MINIMIZATION

This method is used to approximate the frequency response characteristics of a low-order system to that of a high-order system. The technique is based on curve-fitting single-loop frequency responses against a specified model system over a finite range of frequencies. For example, suppose it is desired to approximate the ijth control loop transfer functions G(s) using a low-order model with transfer function $\hat{G}(s,p)$. The vector p contains the parameters of the low-order system. A fit error function, E(p), is defined as the integral of the squared error between the actual system and the modeled system. The transfer function parameters p are determined from the minimization of this error function. This is accomplished by a modified conjugate gradient search to minimize the error function. Constraints are imposed on the parameters to ensure consistent phase and gain characteristics and to maintain stability requirements.

This procedure can be applied to integrate filters of different order. It is achieved by curve-fitting the filter frequency responses at different design points with a common filter form whose parameters are scheduled as a function of flight condition.

11.5 OPEN-LOOP ANALYSIS

For the open-loop analysis, state models of the airplane and actuation systems are needed. All models are full order except for the deletion of nonessential states. These two models are combined and are expressed in the form

$$\dot{x}(t) = A_o x(t) + B_o u_c(t) + \Gamma_o w_g(t) \tag{16}$$

$$y(t) = C_o x(t) + E_a w_g(t) \tag{17}$$

where the matrices and vectors are defined in Volume II, Appendix E, Subsection E.5.1.

11.5.1 OPEN-LOOP STABILITY

The stability of the airplane rigid and flexible modes is determined by computing the eigenvalues of the dynamic models. This is done for several flight conditions and airplane mass distributions. The effect of selected parameters, such as dynamic pressure, altitude, or true airspeed, is evaluated by determining the range of values for which the system is stable.

11.5.2 OPEN-LOOP RESPONSE

For flight conditions at which the open-loop airplane is stable, the steady-state gust response correlation matrices for the state, modal state, measurements and performance parameters (such as bending moments, torsional moments, accelerations, etc.), and the output power spectra of selected performance parameters are computed. The transient responses due to discrete gusts and elevator commands are obtained by simulations. Because the load equations are based on a truncated set of modal coordinates, the absolute magnitudes of the loads are not correct. However, because all modes that are significant with respect to the control task are included, these approximate load calculations are sufficient for evaluating the relative merits of various control laws.

For the various correlation matrices, the diagonal elements represent the variances of the gust response and the offdiagonal elements represent the cross-variances of the gust response. The output power spectral density of selected gust loads is used to determine the significant frequency content.

11.5.3 CONTROLLABILITY

The initial open-loop airplane model contains a selection of several possible control surfaces that may be suitable for the control task. One or more of these will be selected for the final design. Two criteria are used in this selection process: mode controllability and performance parameter controllability.

The relative controllability of a given mode by the various elements of the control vector can be obtained by transforming the state equations into block diagonal form and by appropriate scaling of the control vector.

Inspection of the columns of the transformed control matrix

$$B'_o = T^{-1} B_o T_u$$

will show the coupling of each control into the various modes. T_u is a diagonal scaling matrix with elements corresponding to the maximum allowable control deflections. To assess the relative effectiveness of the controls in controlling a particular mode, it is necessary only to examine the corresponding row (or two rows, if the mode is oscillatory) of the matrix B'_o. The column with the largest absolute value will identify the most effective control.

The concern is also with control of certain performance parameters such as bending moments, torsional moments, accelerations, etc., at various airplane stations. These are represented by the state model output equation (17). The relative controllability of these can be assessed by computing the steady-state output response correlation matrices given an appropriately scaled white noise input at the individual control actuators. Consider the state model given by equation (16) with a single input $u_{c_i}(t)$ where $u_{c_i}(t)$ is a scalar input corresponding to the ith control and is stationary white noise with intensity corresponding to the appropriate element in the matrix T_u.

The steady-state response correlation matrix is computed for the output vector y(t), equation (17). This is repeated for all control inputs $u_{c_i}(t)$ $(i = 1,m)$. For given performance parameter $y_j(t)$, there will be a set of m variances (one for each control)

$$V_{jj}\Big|_i = \lim_{t \to \infty} E\left[y_j(t)y_j(t)\right]\Big|_i \quad (i = 1,m)$$

where m is the number of controls. The most effective control, $u_{c_i}(t)$, with respect to the performance parameter, $y_j(t)$, is identified by the largest variance $V_{jj}\big|_i$. The mode controllability and performance parameter controllability described previously only assess relative effectiveness and do not guarantee that the selected controls are adequate to perform the control task. However, evaluating the closed-loop control surface responses of a full-state design will determine whether or not a given choice of controls is adequate for the control task.

11.5.4 OBSERVABILITY

The initial open-loop model contains measurement equations for sensors placed at a number of possible locations. One or more of these will be selected for the final design. Two criteria are used for this selection: mode observability and performance parameter observability.

The relative observability of the rigid and flexible modes from measurements at the various airplane stations can be obtained by transforming equation (17) into block diagonal form. Consider a set of like measurements, y(t) at all possible sensor locations, and expressed in terms of the block diagonal coordinate, z(t), as

$$y(t) = C_o T z(t) + E_a \dot{w}_g(t) \tag{18}$$

Inspection of the columns of the transformed measurement matrix

$$C_o' = C_o T$$

shows the relative observability of the system modes from measurements at the various locations. The row containing the largest absolute value identifies the most suitable location for that particular type of sensor.

The concern is also with the observation of certain performance parameters, such as bending moments, torsional moments, accelerations, etc., that are excited by the random-gust inputs but that cannot be measured directly. Consider the state model represented by equations (16) and (17). In this case the output vector, y(t), consists of various performance parameters that are not directly measurable, as well as a set of measurements at all possible sensor locations. The steady-state gust response correlation matrix for the complete output vector, y(t), is computed. For a given performance parameter, $y_j(t)$, a set of p cross-variances is obtained

$$V_{ji} = \lim_{t \to \infty} E\left[y_j(t) y_i(t)\right] \quad (i = 1,p)$$

where p is the number of sensor locations. The most suitable sensor location is identified by the largest absolute value of the cross-variance V_{ji}. For a given performance, magnitudes of the cross-variances depend on the correlation between $y_j(t)$ and the measurement $y_i(t)$ and the magnitude of the measurement variance V_{ii}. Thus, using the magnitude of the cross-variance as a basis for sensor selection ensures the best combination of sensor-to-performance criteria correlation, as well as sensor output signal-to-noise ratio.

11.6 CONTROL LAW SYNTHESIS

Control law synthesis consists of formulation of a linear small-perturbation state model for synthesis, modified linear quadratic regulator design, modified Kalman state estimator design, and controller simplification. The first three items have been addressed extensively during this study and will be described here in some detail. Controller simplification will be the subject of future work; however, a general approach will be described here.

The state model for control law synthesis comprises the airplane dynamic model, actuation system model, and the wind disturbance model and is expressed by

$$\dot{x}(t) = A_s x(t) + B_s u_c(t) + \Gamma_s w_c(t) + B_s d(t)$$

$$y(t) = C_s x(t) + F\,v(t)$$

where the matrices and vectors are defined in Volume II, Appendix E, Subsection E.6.1. This model represents the dynamic characteristics of the airplane for small-perturbation motion around a trim point. In the absence of external disturbances, the airplane is in steady unaccelerated flight at the trim point.

The linear regulator provides optimum closed-loop response with respect to release from initial conditions and with respect to random input disturbances that have a flat power spectrum over the range of frequencies characteristic of the airplane. The power spectrum of turbulence is not flat over the range of rigid and structural mode frequencies of a transport airplane. Thus, it is necessary to augment the synthesis model with a model of the atmosphere that has white noise as inputs and the gust velocities with the desired power spectra as outputs. This implies that for an optimal gust-load alleviation (GLA) control law, it is necessary to feed back the gust states. These states are observable from acceleration sensors.

11.6.1 LINEAR REGULATOR DESIGN

The application of optimal control theory furnishes direct synthesis of the structure and gains of a control law. Optimal control is based on the minimization of a cost functional, subjected to the constraints of the equations of motion. To meet the closed-loop requirements of an active control transport, three methods have been adopted for directly incorporating specific design criteria in the optimal control law synthesis. The first method is the usual quadratic cost penalty on specific performance criteria such as deflection, velocity, acceleration, or load. The second method is implicit model-following, which is used to structure the cost function so that the dynamic response of the closed-loop system approaches that of the model. This is a suitable method for incorporating handling qualities criteria or other transient and steady-state response specifications. The third method is specification of a minimum degree of stability. This will ensure that all closed-loop eigenvalues will be placed to the left of a line parallel to the imaginary axis.

Apart from implicit model-following, another approach for incorporating command response criteria into the linear regulator design is explicit model-following. This method was found to be very useful in the synthesis of control laws that produced good pitch-rate and normal load factor responses. The method consists of placing an ideal model of the airplane to be controlled in the forward path of the control loop.

The mathematical development of the linear regulator design and the appropriate equations are presented in Volume II, Appendix E, Subsection E.6.2.

11.6.2 MODIFIED KALMAN FILTER DESIGN

After solving the control problem using the modified linear quadratic regulator design outlined previously, it is necessary to construct a state estimator. The usefulness of state estimators has been limited by the sensitivity of the closed-loop performance to parameter variations. The modeling of a process is never exact, and because the design of a system is based on an approximate model, the design must be insensitive to modeling uncertainties, in particular with respect to the stability of the system. Optimal control with full-state feedback offers a gain of -6 to $+\infty$ dB and $\pm$60-deg phase margins in all control loops. When a Kalman filter is inserted into the loop to estimate state variables, the stability margins shrink, sometimes drastically. To alleviate this problem, a method has been used that allows the increase of stability margins in each control loop at the expense of filter performance when parameters are at their nominal values. A brief description is given in Volume II, Appendix E, Subsection E.6.3.

11.6.3 CONTROLLER SIMPLIFICATION

The Kalman filter will have the same dynamic order as that of the open-loop model used for the synthesis. In the case of a flexible airplane model that contains a large number of structural modes, the high order of the filter imposes an excessive and unnecessary computational burden on flight computers. The most suitable approach to the design of a low-order suboptimal filter has not been determined during this study. However, a preliminary approach will be outlined here. The first task is to establish the minimum bandwidth of the controller. The actuation bandwidth is set by the highest frequency mode that must be controlled. In the case of flutter suppression, it is set by the highest frequency mode that contributes significantly to the gust loads. The latter is easily determined from cumulative power density plots of the appropriate performance parameters such as bending moments, torsional moments, accelerations, etc., at various airplane stations. A third factor that must be considered is the increasing uncertainty in the dynamic model with increasing frequency. Above a certain frequency, the modeled dynamics in terms of frequencies and mode shapes are at best suspect. The controller bandwidth must be limited such that at these higher frequencies the closed-loop system has sufficiently large stability margins.

The modal residualization technique can be used to eliminate filter modes that are outside the required actuation bandwidth. The reduced filter may still be too complex for practical implementation on flight computers. Further reduction may still be possible without any significant loss in closed-loop performance. Again, the modal residualization technique can be used to eliminate filter modes that are within the actuation bandwidth but that are associated with weakly observable or weakly controllable airplane modes or with airplane modes that are not observable from the cost function.

In the preceding discussion, it was assumed that the Kalman filter was synthesized using a full-order airplane model and that the lower order suboptimal filter was obtained by the reduction of this full-order filter. However, another approach would be to reduce the open-loop model using the modal residualization technique, leaving only the modes considered essential to the control task. A suboptimal filter (with respect to the full-order model) would then be synthesized using the lower order airplane open-loop model.

Still another approach would be to use the full-order Kalman filter to define the required control-loop frequency responses over the actuation bandwidth and to design lower response characteristics. This could involve least-square fitting, other optimization techniques, or simply trial-and-error design.

11.7 CLOSED-LOOP ANALYSIS

The closed-loop analysis consists of evaluating the performance of full- and partial-state feedback designs and full- and reduced-order Kalman filter designs in terms of gust response and stability margins. This analysis is an important part of an iterative design procedure that is divided into two parts: the control problem is solved by synthesizing and analyzing the closed-loop performance of full- or partial-state feedback designs. After the proper cost function and associate state feedback gain matrix have been determined, the Kalman filter is synthesized and inserted in the control loop, and the closed-loop performance is evaluated. The performance of various reduced-order filters is evaluated until one is found that gives close to the optimum closed-loop performance with adequate stability margins and without imposing excessive computational burden on flight computers. The formulation of the models used for closed-loop analyses is given in Volume II, Appendix E, Subsection E.7.1.

11.7.1 CLOSED-LOOP STABILITY

The closed-loop stability analysis consists of computing eigenvalues, gain and phase margins in all control loops, and the range of values of key parameters such as dynamic pressure, for which the closed-loop system remains stable. The various closed-loop control laws are evaluated based on location of closed-loop poles and the margins of stability as a function of frequency.

If the synthesis model has full order, then the full-state feedback system is optimal with respect to the cost function, stable if all unstable open-loop modes are controllable, and robust with respect to parameter variations in the control loops. Similarly, the closed-loop system with the full-order Kalman filter is always stable provided that all unstable open-loop modes are controllable and observable. With the Kalman filter inserted into the control loop to estimate the states, the good stability margins of the full-state feedback design may shrink, sometimes drastically. For the closed-loop system with reduced-order filter, there is no guarantee that the system is stable even at the nominal gain and phase.

11.7.2 CLOSED-LOOP RESPONSE

The steady-state gust response correlation matrices for the states, modal states, measurements and performance parameters (such as bending moments, torsional moments, accelerations, etc.), and the output power spectral density of selected performance parameters are computed. The transient responses due to discrete gusts and elevator commands are obtained by simulations. Because the load equations are based on a truncated set of modal coordinates, the absolute magnitudes of the loads are not correct. However, because all modes that are significant with respect to the control task are included, and the same truncated model has been used to compute the gust loads of the open-loop airplane, these approximate load calculations are sufficient for evaluating the relative merits of various control laws. The closed-loop gust response is evaluated in terms of the relative reduction in the related performance parameters and the peak and rms deflections and rates of the control surfaces.

11.7.3 EVALUATION OF STATE FEEDBACK DESIGNS

The closed-loop analysis of the state feedback design is part of the iterative design cycle to solve the control task. Full-state feedback designs are evaluated until the proper cost function and control surfaces have been selected. The evaluations are based on gust-load reductions, control surface activities, and closed-loop eigenvalues.

Because the control law includes feedback of control surface states, the optimal linear regulator can be used to determine whether or not the control surface actuators have sufficient bandwidth. If there is a significant change in the actuator closed-loop eigenvalues from their nominal open-loop values, it will be necessary to increase the actuation bandwidth.

The trade between closed-loop performance and actuation bandwidth can be determined by considering the cumulative power-spectral-density plots of the open-loop and full-state closed-loop gust responses of the various performance parameters. The effects of eliminating modes from the feedback can be determined by evaluating the closed-loop performance with the appropriate columns in the optimal gain matrix set equal to zero.

11.7.4 EVALUATION OF KALMAN FILTER DESIGNS

The closed-loop analysis of the Kalman filter designs is part of the iterative design cycle to solve the state estimation problem. Full-order Kalman filters are evaluated until the closed-loop performance and stability margins meet or exceed the design requirements. The key design parameters evaluated are types, numbers, and locations of sensors and the trade between gust response and stability margins. This same iterative analysis is used to evaluate reduced-order filters.

12.0 CONTROL LAWS FOR FMC AND GLA

Constant gain control laws for suppression of a symmetric flutter mode and reduction of loads due to vertical gusts have been synthesized at eight flight conditions for the Initial ACT Airplane. They correspond to four speed conditions identified as V_B, V_{MO}, V_D, and $1.2V_D$ and two mass distributions as indicated in Table 36. The first four are used for gust-load alleviation (GLA) design, while the latter four are used for flutter-mode control (FMC) design.

12.1 DYNAMIC MODEL DESCRIPTION

At each flight condition, the flexible airplane is represented by a set of constant coefficient second-order differential equations modified by the addition of first-order lag terms that represent the effects of unsteady aerodynamics associated with gust inputs.

Table 36. Flight Conditions for Dynamic Models of the Initial ACT Airplane

Flight condition	Speed condition	Equivalent airspeed, m/s (kn)	Mass condition	Altitude, m (ft)	Pressure, N/m^2 (lb/ft^2)
1	V_B	142 (276)	0.8F	10 668 (35 000)	12 344 (258)
2	V_B	142 (276)	MZFW+F	10 668 (35 000)	12 344 (258)
3	V_{MO}	176 (341)	0.8F	7 833 (25 700)	18 846 (394)
4	V_{MO}	176 (341)	MZFW+F	7 833 (25 700)	18 846 (394)
5	V_D	214 (416)	0.8F	4 968 (16 300)	28 049 (586)
6	V_D	214 (416)	MZFW+F	4 968 (16 300)	28 049 (586)
7	$1.2V_D$	262 (508)	0.8F	1 890 (6 200)	41 887 (875)
8	$1.2V_D$	262 (508)	MZFW+F	1 890 (6 200)	41 887 (875)

- Symmetric model 768-103
- Mach number M = 0.86
- Mass = 122 470 kg (270 000 lb)
- 0.8F = 80% fuel (including full reserve tanks) plus payload to maximum gross weight resulting in aft center of gravity ($0.46\bar{c}$)
- MZFW+F = Maximum zero fuel weight plus fuel (including full reserve tanks) to maximum gross weight resulting in forward center of gravity ($0.22\bar{c}$)

The development of these equations is described in Section 11.0 and in more detail in Volume II, Appendix E. The model has 2 rigid and 19 flexible body modes. The latter range in frequency from approximately 11 to 302 rad/s. The lag and scaling constants in the three Kussner lift-growth functions (fig. 120) were computed assuming an infinite aspect-ratio wing (ref 8) and expressed as

$$\begin{aligned} d_1 &= 0.058 \left(\frac{2V}{c}\right) & c_1 &= 0.236 \\ d_2 &= 0.364 \left(\frac{2V}{c}\right) & c_2 &= 0.513 \\ d_3 &= 2.42 \left(\frac{2V}{c}\right) & c_3 &= 0.171 \end{aligned} \tag{19}$$

where V is the true airspeed (see table 36) and c is the characteristic chord length c = 6.03m (237.4 in).

The dynamic models of the actuators for the elevator and the outboard aileron are shown in Figure 123. Originally, these were represented by second-order dynamics; however, an additional lag filter with a time constant of 0.001 sec was added. This was done to avoid the explicit dependence of acceleration measurements on actuator commands. Another approach would be to include sensor dynamics in the model.

The output models for measurements and loads are described in Subsection 11.1.2. The elements of the mode shape matrix for pitch-rate measurement at the center of gravity and vertical acceleration measurement at the wing tip are listed in Volume II, Appendix F,

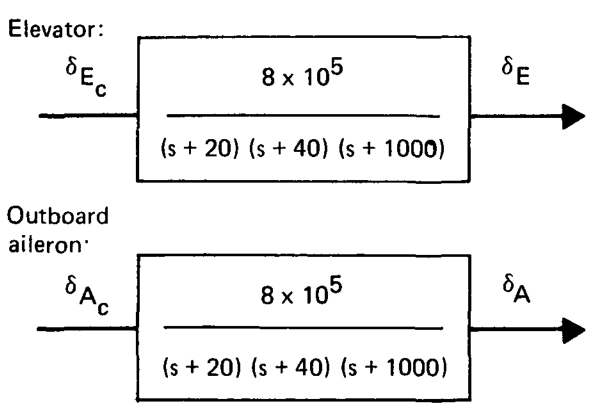


Figure 123. Control Surface Actuator Dynamics

for the two airplane mass conditions. Table 37 lists the rigid- and flexible-body free-vibration modes. The measurements were transformed from inertial axes to body-fixed axes. Elements of the load distribution matrix are also referenced to body-fixed axes. The six load equations represent the bending moment, torsional moment, and shear at an inboard (η = 0.25) and an outboard (η = 0.75) wing station. Details are shown in Volume II, Appendix F.

The wind disturbance model is based on the Dryden spectrum for a vertical gust with a turbulence scale length of 533.4m (1750 ft).

Table 37. Description of Airplane Rigid and Flexible Modes

Modes	Symbol	Units	Description	Sign convention
q_1	u	in/s	Fore and aft velocity of center of gravity	+ forward
q_2	w	in/s	Vertical velocity of center of gravity	+ downward
q_3	q	rad/s	Pitch rate	+ nose up
q_4			Forebody vertical bending deflection	
q_5			First aft-body vertical bending deflection	
q_6			Second aft-body vertical bending deflection	
q_7			First vertical tail vertical bending deflection	
q_8			Second vertical tail vertical bending deflection	
q_9			First horizontal tail vertical bending deflection	
q_{10}			Second horizontal tail vertical bending deflection	
q_{11}			Horizontal tail torsion deflection	
q_{12}			First wing vertical bending deflection	
q_{13}			Second wing vertical bending deflection for $0.46\bar{c}$ condition First wing fore and aft bending deflection for $0.22\bar{c}$ condition	
q_{14}			First wing fore and aft bending deflection for $0.46\bar{c}$ condition Second wing vertical bending deflection for $0.22\bar{c}$ condition	
q_{15}			First wing torsion deflection	
q_{16}			Third wing vertical bending deflection	
q_{17}			Second wing fore and aft bending deflection	
q_{18}			Second wing vertical bending deflection	
q_{19}			Second wing torsion deflection	
q_{20}			Nacelle side bending deflection	
q_{21}			Nacelle vertical bending deflection	
q_{22}			Nacelle roll deflection	

The elements of the state vector for the combined airplane actuation system and wind model are listed in Table 38.

Table 38. Description of the System State Vector

Modes	Symbol	Units	Description	Sign convention
X_1	u	in/s	Fore and aft velocity of center of gravity	+ forward
X_2	w	in/s	Vertical velocity of center of gravity	+ downward
X_3	q	rad/s	Pitch rate	+ nose up
X_4		rad	Pitch angle	+ nose up
X_5			Forebody vertical bending rate	
X_6			Forebody vertical bending deflection	
X_7			First aft-body vertical bending rate	
X_8			First aft-body vertical bending deflection	
X_9			Second aft-body vertical bending rate	
X_{10}			Second aft-body vertical bending deflection	
X_{11}			First vertical tail vertical bending rate	
X_{12}			First vertical tail vertical bending deflection	
X_{13}			Second vertical tail vertical bending rate	
X_{14}			Second vertical tail vertical bending deflection	
X_{15}			First horizontal tail vertical bending rate	
X_{16}			First horizontal tail vertical bending deflection	
X_{17}			Second horizontal tail vertical bending rate	
X_{18}			Second horizontal tail vertical bending deflection	
X_{19}			Horizontal tail torsion rate	
X_{20}			Horizontal tail torsion deflection	
X_{21}			First wing vertical bending rate	
X_{22}			First wing vertical bending deflection	
X_{23}			Second wing vertical bending rate	
X_{24}			Second wing vertical bending deflection	
X_{25}			First wing fore and aft bending rate	
X_{26}			First wing fore and aft bending deflection	
X_{27}			First wing torsion rate	
X_{28}			First wing torsion deflection	

Table 38. Description of the System Vector (Continued)

Modes	Symbol	Units	Description	Sign convention
X_{29}			Third wing vertical bending rate	
X_{30}			Third wing vertical bending deflection	
X_{31}			Second wing fore and aft bending rate	
X_{32}			Second wing fore and aft bending deflection	
X_{33}			Second wing vertical bending rate	
X_{34}			Second wing vertical bending deflection	
X_{35}			Second wing torsion rate	
X_{36}			Second wing torsion deflection	
X_{37}			Nacelle side bending rate	
X_{38}			Nacelle side bending deflection	
X_{39}			Nacelle vertical bending rate	
X_{40}			Nacelle vertical bending deflection	
X_{41}			Nacelle roll rate	
X_{42}			Nacelle roll deflection	
X_{43}			First Kussner lift-growth state	
X_{44}			Second Kussner lift-growth state	
X_{45}			Third Kussner lift-growth state	
X_{46}	$\ddot{\delta}_E$	rad/s^2	Elevator acceleration	
X_{47}	$\dot{\delta}_E$	rad/s^2	Elevator rate	
X_{48}	δ_E	rad	Elevator deflection	
X_{49}	$\ddot{\delta}_A$	rad/s^2	Outboard aileron acceleration	
X_{50}	$\dot{\delta}_A$	rad/s	Outboard aileron rate.	
X_{51}	δ_A	rad	Outboard aileron deflection	
X_{52}			First Dryden gust model state	
X_{53}			Second Dryden gust model state	

12.2 SELECTION OF CONTROL SURFACES

Seven control surfaces have been considered for the control task: inboard and outboard elevators, complete elevator, inboard and outboard segments of the outboard aileron, complete outboard aileron, and inboard aileron. The suitability of these control surfaces has been analyzed as outlined in Subsection 11.5.3. The relative controllability of the flutter mode at flight conditions 5, 6, 7, and 8 is shown in Volume II, Appendix F, Table F-3. Among the surfaces considered and with the specified control authorities, the outboard aileron is most suitable for FMC. However, the outboard portion of the outboard aileron is also suitable for this task, having relative controllability of about 0.70 at these flight conditions.

Tables F-4 through F-7 (vol. II, app F) show the relative root-mean-square (rms) load responses to white noise inputs at the actuators for flight conditions 1, 2, 3, and 4, respectively. The relative rms load response is defined as the ratio between the actual rms load response and the maximum rms load response (from the set of all control surfaces). If the relative rms load response is 1.0, then the associated surface is the most effective surface for controlling that particular load (shear, bending, or torsion). Among the surfaces considered and with the specified control authorities, the outboard aileron and elevator are most suitable for controlling the wing loads.

In addition, safety, actuator installation, and other design requirements must be considered. A small control surface with limited control authority that is dedicated to automatic control functions will have obvious advantages with regard to failures and actuator bandwidth and resolution. The location of the control surface may affect other design requirements that have not been considered in the analysis of the open-loop airplane. For example, using the elevator to control the loads at the wing root may adversely affect the ride and handling qualities of closed-loop airplanes. Thus, the final selection of control surfaces should only be done after the analysis of the closed-loop performance of various control surface configurations. The open-loop analysis described in Subsection 11.5.3 should primarily be used to identify all control surface configurations that are possible candidates.

The control law synthesis performed in this study used the combination of the elevator and the outboard aileron for control.

12.3 SELECTION OF SENSORS

Sensor selection criteria are described in detail in Subsection 11.5.4. The two criteria used are mode observability and performance parameter observability. The first is important in the selection of sensors for FMC, and the second is important in the selection of sensors for GLA. Three sensors have been selected: one pitch-rate gyro at the center of gravity will be used to observe the short-period mode and two vertical accelerometers located symmetrically (one in each wing) will be used to observe the structural modes of the wing. The subsequent analysis was only concerned with finding the best location for the wing-mounted accelerometers.

The sensor locations considered in this study are shown in Figure 124. Each location is identified by a number between 1 and 27. The locations are distributed spanwise along the elastic axis and the rear spar of the wing. The relative observability of the two modes involved in the flutter of the wing was determined for flight conditions 5, 6, 7, and 8. The sensor-to-flutter mode coupling is explained in Subsection 11.4. Plots presented in Volume II (app F) show these values normalized to the coupling at sensor location 27. Sensor location 27, on the rear spar and at the wing tip, is most suitable for flutter suppression using vertical accelerometers.

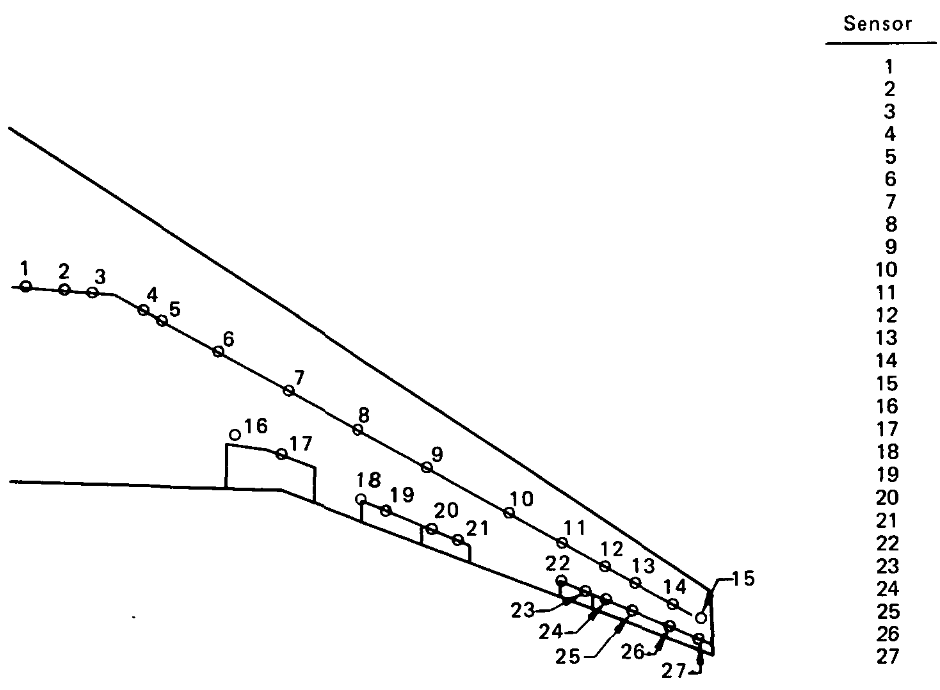


Figure 124. Candidate Wing Accelerometer Locations

The output gust response correlation matrices for the open-loop airplane were computed as described in Subsections 11.3 and 11.5. These data illustrate the relative cross-variances between accelerometer outputs and bending moments, the results being normalized with respect to the rms level of the bending moments. Similar data were obtained for the closed-loop airplane. In this case, a full-state feedback control law is assumed. Detailed data are presented in Volume II, Appendix F.

Sensor position 27 was selected for the subsequent Kalman filter designs for the following reasons: high flutter-mode observability, high rms output, and good correlation with the inboard wing bending moment, which was selected as the main gust-load parameter to be reduced by feedback control (see subsec 12.4).

12.4 GLA DESIGNS

Several intermediate designs were synthesized before selecting a final design for gust-load reduction. Table 39 lists designs used, and Figure 125 contains a flow chart of the approach followed in the design for each flight condition. Figure 126 shows how the eight flight conditions were integrated into the final design. Results at each of the intermediate design steps will be discussed to show how the final design was reached.

Full-state feedback design (design A) is the initial step in the design process for gust-load reduction. The design is determined by the values selected for the parameters in the cost function

$$J = \int_0^{\infty} \left[(Q_1 y_1^2 + Q_2 y_2^2 + Q_3 y_3^2 + Q_4 y_4^2) \ + \ (R_1 \delta_{E_c}{}^2 + R_2 \delta_{A_c}{}^2) \right] dt \tag{20}$$

where

$$Q_1 = K_1 / y^2{}_{1max}$$

$$Q_2 = K_2 / y^2{}_{2max}$$

$$Q_3 = K_3 / y^2{}_{3max}$$

$$Q_4 = K_4 / y^2{}_{4max}$$

Table 39. Design Types

Type	Name
A	Full-state feedback
B	Nominal Kalman filter
C	Robust Kalman filter
D	Reduced-order Kalman filter
E	Reduced-order Kalman filter with gain reduction in one or both control loops
F	Integrated 8th-order filter for flight conditions 1, 2, 3, and 4
G	Integrated 10th-order filter for flight conditions 5, 6, 7, and 8
H	Final 8th-order filter integrated for all flight conditions

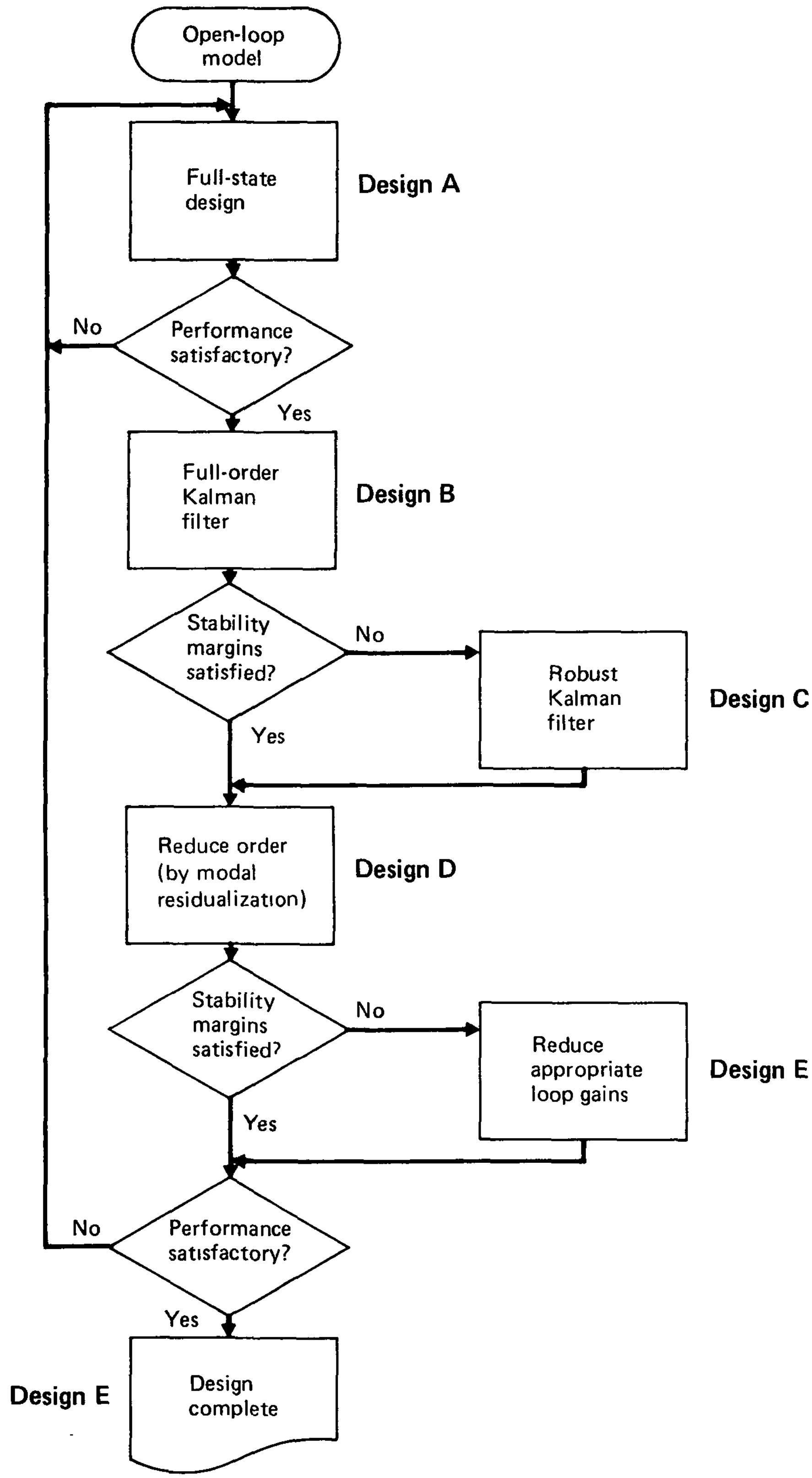


Figure 125. Design Process for All Flight Conditions

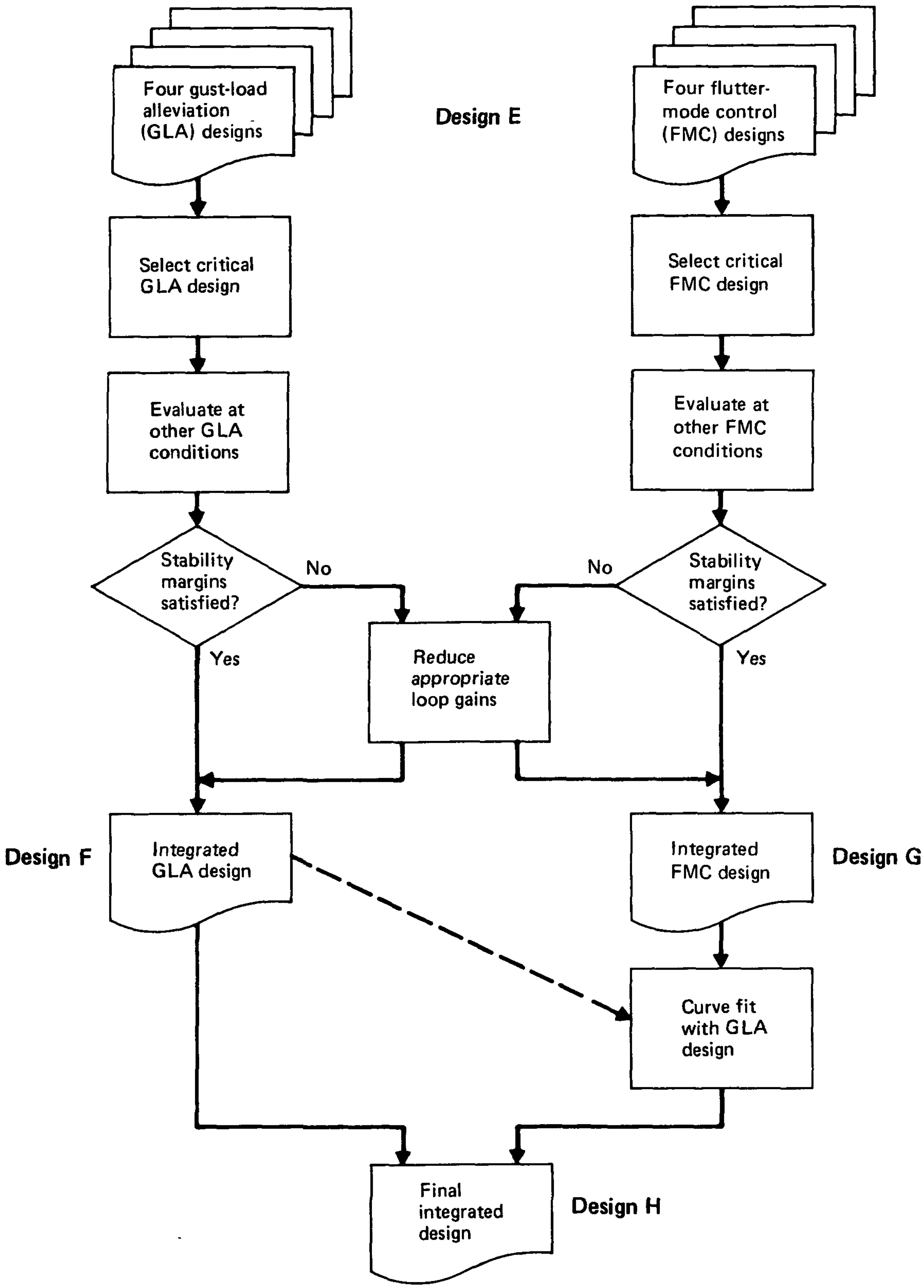


Figure 126. Integration of Designs From All Flight Conditions

$R_1 \quad = \quad 1.0/\delta_{E}^2{}_{max}$

$R_2 \quad = \quad 1.0/\delta_{A}^2{}_{max}$

and

$y_1 \quad =$ wing bending moment at the inboard station ($\eta = 0.25$)

$y_2 \quad =$ wing torsion at the inboard station ($\eta = 0.25$)

$y_3 \quad =$ wing bending moment at the outboard station ($\eta = 0.75$)

$y_4 \quad =$ wing torsion at the outboard station ($\eta = 0.75$)

$\delta_{E_c} \quad =$ command input to the elevator

$\delta_{A_c} \quad =$ command input to the outboard aileron

In selecting appropriate values for the cost function parameters, their relative weighting is of prime consideration. However, to achieve a reasonable first guess, the following values were assumed as representative:

$\delta_{E_{max}} \quad = \quad 3$ deg

$\delta_{A_{max}} \quad = \quad 15$ deg

$y_{i_{max}} \quad =$ corresponding open-loop load ($i = 1,4$)

$K_i \quad = \quad 1$ ($i = 1,4$)

In all the full-state designs, $\delta_{E_{max}}$ and $\delta_{A_{max}}$ were not changed. Although the correlation between changes in particular loads and control surface activity varies, generally an increase in value of one or more of the K_i results in a decrease of the corresponding load and an increase in the remaining loads and control surface activity.

Full-state design objectives included reducing bending moment loads and avoiding an increase in torsion loads while keeping the aileron rate within approximately 1.5-deg/s rms for a gust intensity of 0.305 m/s (1 ft/s). Gust response calculations used the Dryden spectrum with a turbulence scale length of 533.4m (1750 ft).

Gust loads were reduced in design A for flight conditions 1, 2, 3, and 4 by incorporating a combination of four cost function parameters: inboard bending moment, inboard torsion, outboard bending moment, and outboard torsion. The combination and relative weighting varied with flight conditions, although the penalty on inboard bending moment predominated. However, to illustrate the effects of changes in the cost function, flight condition 3 was considered with different penalty weights on inboard bending moment and inboard torsion. Figures 127 and 128 show the changes in loads and control surface activity as the penalty on both loads is increased proportionately.

Note that although the control surface activity increases as the penalty on loads increases, the changes in loads are less predictable. The design corresponding to a factor of μ = 4 was selected because it provided good reduction in the bending moments with reasonably low control surface activity.

Table 40 shows the weighting factors of the cost functions selected for the final full-state designs of each flight condition. The resultant performance in load reduction and control surface activity for flight conditions 1, 2, 3, and 4 is given in Tables 41 through 44. Power-spectral-density (PSD) plots illustrating the load reduction and corresponding control activity provided by design A are given in Volume II, Appendix F. Note that, as expected, the short period is predominant in all loads and that inboard torsion is influenced by the flutter mode and first structural mode due to engine inertia response. The first structural mode also contributes to the outboard bending moment. Elevator activity is primarily in response to the short period and first structural mode, while aileron activity is dominated by the first structural and flutter modes. The open-loop and closed-loop poles tabulated in Volume II, Appendix F, for each of the four flight conditions show that damping ratios for phugoid, short-period, and first structural modes were significantly increased by design A.

Design B resulted from combining a Kalman state estimator with the full-state feedback design. The process and sensor noise characteristics used in the estimator design are

Cost function			Symbol
	Parameter	Weighting factor	
Q_1	Inboard bending moment	$(3.17 \times 10^{-14}) \times \mu/(N{\cdot}m)^2$ $((2.48 \times 10^{-12}) \times \mu/(in\text{-}lb)^2)$	●
Q_2	Inboard torsion	$(1.79 \times 10^{-13}) \times \mu/(N{\cdot}m)^2$ $((1.40 \times 10^{-11}) \times \mu/(in\text{-}lb)^2)$	○
Q_3	Outboard bending moment	0	▲
Q_4	Outboard torsion	0	△
R_1	Elevator deflection	$364.76/(rad/s)^2$	
R_2	Aileron deflection	$14.59/(rad/s)^2$	

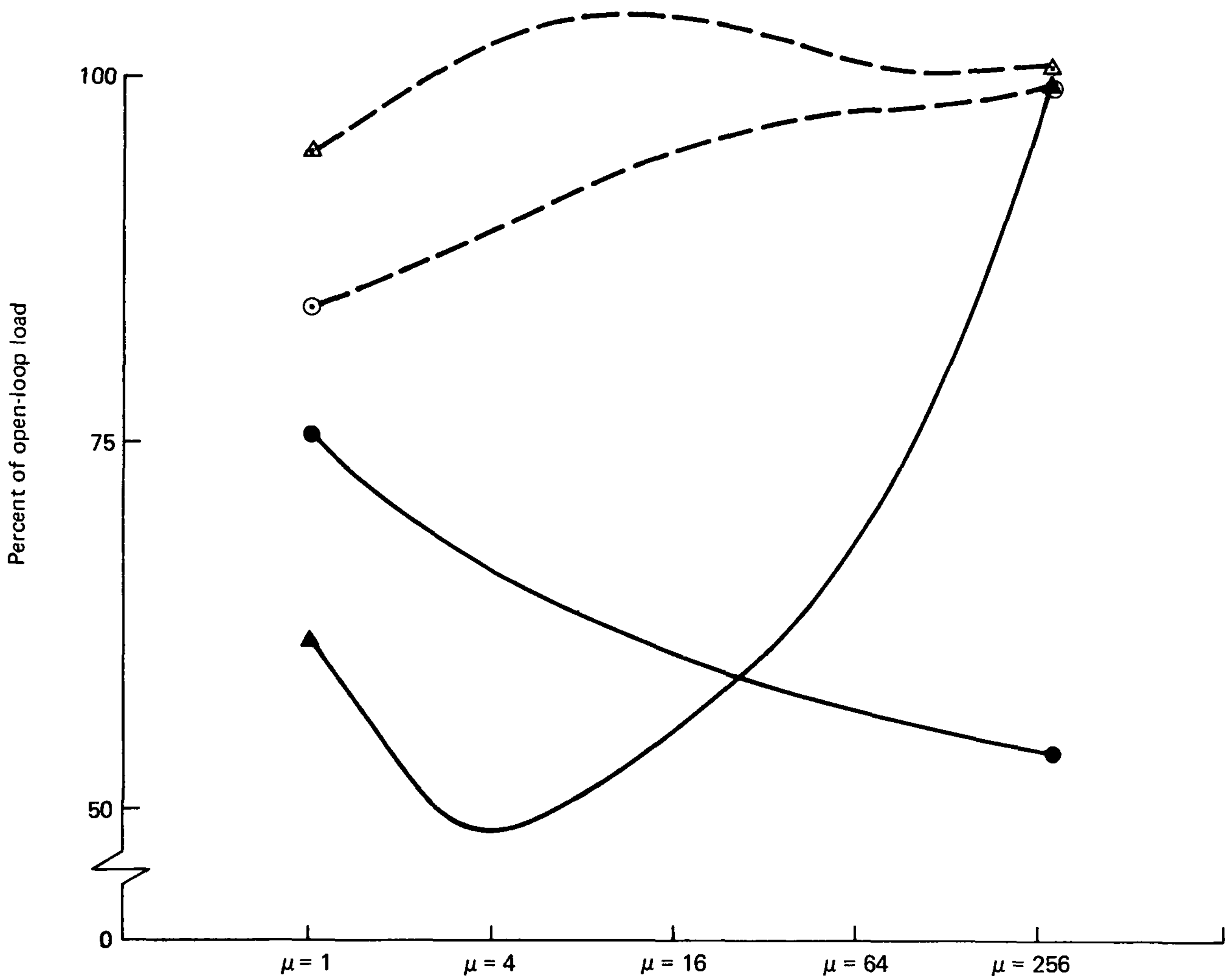


Figure 127. Load Reduction for Full-State Feedback Designs at Flight Condition 3

Cost function		Symbol
Parameter	Weighting factor	
Q_1 Inboard bending moment	$(3.17 \times 10^{-14}) \times \mu/(N{\cdot}m)^2$ $((2.48 \times 10^{-12}) \times \mu/(in\text{-}lb)^2)$	
Q_2 Inboard torsion	$(1.79 \times 10^{-13}) \times \mu/(N{\cdot}m)^2$ $(1.40 \times 10^{-11}) \times \mu/(in\text{-}lb)^2)$	
Q_3 Outboard bending moment	0	
Q_4 Outboard torsion	0	
R_1 Elevator deflection	$364.76/(rad/s)^2$	●
R_2 Aileron deflection	$14.59/(rad/s)^2$	▲
Elevator rate	–	○
Aileron rate	–	△

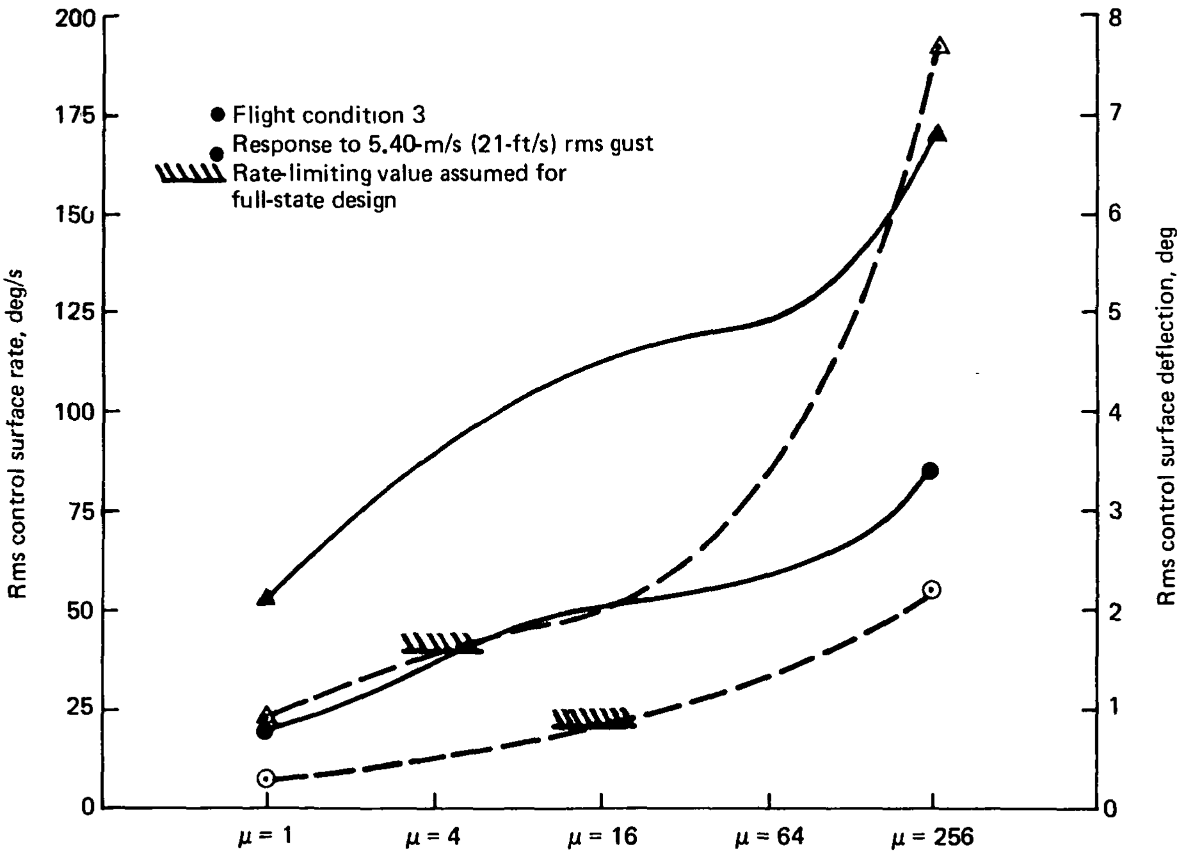


Figure 128. Control Surface Activity for Full-State Feedback Designs

Table 40. Cost Function Parameters in Full-State Designs

Flight condition	Inboard		Outboard	
	Bending moment Q_1	Torsion Q_2	Bending moment Q_3	Torsion Q_4
1	0.904	12.6	23.8	0
2	0.859	0.675	0	177.0
3	0.990	5.58	0	0
4	0.616	0	0	49.5
5	0.390	1.38	0	0
6	0.0167	1.64	0	30.6
7	0.376	1.20	0	24.5
8	0	0	0	16.7

All values above multiplied by 10^{-13}

R_1 = 364.8 and R_2 = 14.59 for all flight conditions

Table 41. Root-Mean-Square Gust Response, Flight Condition 1

Design	Filter order	Load reduction, percent of open loop						Rms control surface activity [1]			
		Inboard			Outboard			Elevator		Outboard aileron	
		Shear	Bending moment	Torsion	Shear	Bending moment	Torsion	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
A	–	84.2	57.1	97.2	21.0	29.3	93.4	1.20	8.40	5.14	43.2
B [2]	53	87.0	67.5	91.5	40.5	33.0	92.3	0.888	7.20	4.06	45.0
C [2]	53	87.1	68.0	92.5	41.3	33.6	93.0	0.798	6.26	4.02	44.0
D [2]	8	87.4	68.2	99.6	40.9	32.5	93.2	0.738	7.48	4.08	43.4
E [3]	8	89.0	80.9	92.8	72.1	68.5	87.3	0.688	5.80	1.60	16.6
F, H [4]	8	85.6	75.9	89.7	65.6	62.6	84.4	0.950	7.76	1.79	17.9

[1] Response to 8.53-m/s (28-ft/s) rms gust intensity

[2] DRO gain and phase margin requirements not satisfied

[3] δ_E gain reduced to 0.56, δ_A gain reduced to 0.26

[4] Design for flight condition 3 with δ_A loop gain reduced to 0 63

Table 42. Root-Mean-Square Gust Response, Flight Condition 2

Design	Filter order	Load reduction, percent of open loop						Rms control surface activity [1]			
		Inboard			Outboard			Elevator		Outboard aileron	
		Shear	Bending moment	Torsion	Shear	Bending moment	Torsion	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
A	–	74.0	56.2	94.3	48.6	40.7	99.7	1.38	9.90	3.60	33.4
B [2]	53	76.7	63.5	87.7	57.9	43.7	93.3	1.16	8.36	3.06	36.6
D [2]	8	77.7	63.7	96.0	58.2	43.3	93.7	1.10	8.54	3.04	38.1
E [3]	8	79.1	71.8	91.8	69.7	67.1	83.5	1.11	8.66	1.55	17.8
F, H [4]	8	76.6	68.5	87.4	65.6	61.9	84.7	1.27	8.82	1.79	20.0

[1] Response to 8.53-m/s (28-ft/s) rms gust intensity

[2] DRO gain and phase margin requirements not satisfied

[3] δ_E gain reduced to 0.79; δ_A gain reduced to 0.37

[4] Design for flight condition 3 with δ_A loop gain reduced to 0 63

Table 43. Root-Mean-Square Gust Response, Flight Condition 3

Design	Filter order	Load reduction, percent of open loop						Rms control surface activity [1]			
		Inboard			Outboard			Elevator		Outboard aileron	
		Shear	Bending moment	Torsion	Shear	Bending moment	Torsion	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
A	–	91.3	66.7	89.8	34.4	48.5	103	1.25	10.4	3.65	33.8
B [2]	53	91.9	75.3	81.4	50.9	49.9	96.3	0.980	9.23	2.76	34.8
C [2]	53	92.6	78.4	83.5	56.5	52.3	96.6	0.758	7.06	2.48	30.3
D [2]	8	92.3	78.9	92.9	55.0	47.1	96.9	0.684	6.78	2.52	29.7
E [3]	8	90.7	79.5	91.4	62.2	56.5	91.6	0.799	7.71	1.91	23.2
F, H [4]	8	90.6	79.5	91.3	62.8	57.4	91.2	0.788	7.79	1.86	22.7

[1] Response to 6.40-m/s (21-ft/s) rms gust intensity

[2] DRO gain and phase margin requirements not satisfied

[3] δ_A gain reduced to 0.71

[4] δ_A gain reduced to 0.69 for compatibility with flight condition 4

Table 44. Root-Mean-Square Gust Response, Flight Condition 4

Design	Filter order	Load reduction, percent of open loop						Rms control surface activity [1]			
		Inboard			Outboard			Elevator		Outboard aileron	
		Shear	Bending moment	Torsion	Shear	Bending moment	Torsion	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
A	–	83.1	68.8	95.4	62.0	53.9	99.2	1.03	8.51	2.34	25.1
B [2]	53	84.7	73.9	90.6	68.7	55.5	93.4	0.869	7.29	2.07	26.9
D [2]	8	86.4	74.1	95.3	68.9	55.4	93.5	0.854	7.58	2.18	29.7
E [3]	8	86.3	74.8	96.9	70.8	63.1	89.8	0.914	8.13	1.74	23.7
F, H [4]	8	84.8	73.7	92.0	69.3	59.8	90.8	1.02	9.05	1.86	18.0

[1] Response to 6.40-m/s (21-ft/s) rms gust intensity
[2] DRO gain and phase margin requirements not satisfied
[3] δ_A gain reduced to 0.75
[4] Design for flight condition 3 with δ_A loop gain reduced to 0.69

given in Table 45. The Kalman filter introduced more reduction in performance than any other step in the design process. For flight conditions 1, 2, 3, and 4, inboard bending moment increases averaged 8%; outboard bending moment increases averaged over 2%. Torsion loads decreased nearly 4%. Elevator rates decreased slightly with moderate increase in the aileron rates when compared with design A. Results of these comparisons are summarized in Tables 41 through 44 for flight conditions 1, 2, 3, and 4. Closed-loop controller poles are tabulated in Volume II, Appendix F.

Design B introduced deficiencies in phase and gain margin performance predominately in the aileron loop, at all four flight conditions. Margin requirements, which vary with flight condition, are summarized in Table 46. These phase and gain requirements are displayed graphically on a Bode plot by a band around -180 deg and 0 dB, respectively. For example, a phase margin of 60 deg is defined by lines at -120 deg and -240 deg. The phase margin is defined at points of 0-dB gain to be the difference between the actual phase and -180 deg. Consequently, if the phase curve is within the region bounded by -120 deg and -240 deg at a point of 0-dB gain, the phase margin of 60 deg is not met. An analogous definition applies to gain margin.

Table 45. Plant Process Noise and Sensor Noise Characteristics Used in Kalman Estimator Designs

Wind disturbance intensity	Pitch-rate sensor noise intensity	Wing-tip accelerometer sensor intensity
4.267 m/s (14.0 ft/s)	0.15 deg/s	0.05g

Note: Gaussian power density function, one standard deviation

Table 46. Stability Margin Requirement

Frequency range, rad/s	Flight condition							
	1,2		3,4		5,6		7,8	
Low frequency	$0 \leqslant \omega < 0.5$		$0 \leqslant \omega < 0.5$		$0 \leqslant \omega < 0.5$		$0 \leqslant \omega < 0.5$	
	±4 dB	±15 deg	±4 dB	±15 deg	0 dB	0 deg	0 dB	0 deg
Medium frequency	$0.5 \leqslant \omega < 11$		$0.5 \leqslant \omega < 13.4$		$0.5 \leqslant \omega < 14.3$		$0.5 \leqslant \omega < 13.4$	
	±6 dB	±45 deg	±6 dB	±45 deg	±4 dB	±20 deg	0 dB	0 deg
High frequency	$11.0 \leqslant \omega$		$13.4 \leqslant \omega$		$14.3 \leqslant \omega < 39.5$		$13.4 \leqslant \omega < 40.6$	
	±12 dB	±180 deg	±12 dB	±180 deg	±12 dB	±60 deg	±6 dB	±45 deg
					$39.5 \leqslant \omega$		$40.6 \leqslant \omega$	
					±12 dB	±180 deg	±6 dB	±180 deg

Points at which gain or phase margin is checked are indicated by vertical bars on the Bode plot. A vertical bar intersects the phase region at points of 0-dB gain and the gain region at points of -180-deg phase. If the vertical bar fails to intersect the gain or phase curve at one of these critical frequencies, the stability margin is satisfied.

Stability margin deficiencies for design B in flight conditions 1 and 3 were considered serious enough to warrant a robust Kalman filter (design C). Values of input noise covariance used for these robust designs are shown in Table 47. Stability margins improved with a slight degradation in performance. All low- or mid-frequency margin deficiencies are eliminated by the robust design procedure.

Table 47. Input Noise Covariance in Robust Kalman Filter Designs (Design C)

Flight condition	Elevator, rad^2	Aileron, rad^2
1	0	7.84
2	0	0
3	3.25	25.8
4	0	0
5	0.197	6.80
6	0	40.3
7	0	0
8	0	0

Modal residualization was used to reduce the 53rd-order Kalman filters of flight conditions 1, 2, 3, and 4 to 8th-order filters (design D). Modes selected for retention in the reduced filter were those with frequencies below 10 rad/s (which is within the control design bandwidth). Attempts to use this technique for further reduction resulted in unstable closed-loop designs for some flight conditions. By eliminating the dynamic contribution at the higher frequency modes, gain rolloff was experienced in the high-frequency mode for the aileron loop, which resulted in a significant improvement in the stability margins for flight conditions 1, 2, and 3. The inboard torsion load (which has been significantly reduced in design A at the flutter-mode frequency) increased an average of over 7% due to this gain rolloff. Little change resulted in other performance parameters (see tables 41 through 44).

Design D did not satisfy stability margins in either loop for flight conditions 1 and 2 nor in the aileron loop for flight conditions 3 and 4. Table 48 gives the values used to reduce the appropriate loop gains to satisfy the margin requirements. Although the gain reductions for flight conditions 1 and 2 resulted in bending moment increases of 12% and 8%, respectively, the reduction prior to these changes in loop gains was much greater for flight conditions 1 and 2 than 3 and 4.

Results were similar for outboard bending moment loads, although the changes were more dramatic. Tables 41 through 44 give the performance data for design E. Bode plots showing the stability margins for the preceding controller designs are included in Volume II, Appendix F (figs. F-47 through F-81).

Table 48. Loop-Gain Reduction Factors in Design Cases E, F, and H at All Gust-Load Conditions

	Design E		Designs F and H	
Flight condition	Elevator loop	Aileron loop	Elevator loop	Aileron loop
1	0.56	0.26	1.0	0.63
2	0.79	0.37	1.0	0.63
3	1.0	0.71	1.0	0.69
4	1.0	0.75	1.0	0.69

The final design step for GLA was to integrate the four E designs representing flight conditions 1, 2, 3, and 4. Although this could have been accomplished using the curve-fitting technique described in Subsection 11.4.3, it was found that design E from flight condition 3 actually reduced both inboard and outboard bending moment loads at other gust-load conditions more than the level provided by the design E developed individually for each of those flight conditions.

In addition, control surface activity and torsion loads were generally lower. Slight gain reductions in the aileron loop were required to meet margin requirements (table 48). The lower of the loop gains required for flight conditions 1 and 2 (and separately for flight conditions 3 and 4) was used so parameter scheduling in the final design would only be a function of speed and independent of mass distribution. Tables 41 through 44 show the performance changes resulting from the integrated design F. Note the final integrated design H is the same as design F for flight conditions 1, 2, 3, and 4, but it also integrated flight conditions 5, 6, 7, and 8.

The unexpected load reduction phenomena can be partially explained by the significant performance degradation experienced between designs D and E (tables 41 and 42) resulting from the loop gain reductions shown in Table 48. The improved performance of design F over design E is greatest in flight conditions 1 and 2 because of the major changes in loop gain at these conditions.

PSD plots of design H for the bending moment and torsion at flight conditions 1, 2, 3, and 4 are shown in Volume II, Appendix F (figs. F-14 through F-29). Although the rms inboard torsion loads are significantly reduced from design A to design H, the reduction near the flutter mode (approximately 20 rad/s) is less for design H because control filter modes above 10 rad/s were eliminated in the reduction to an 8th-order filter. This is also reflected in the PSD plots (also in vol. II, app F), which show the aileron activity. Satisfactory stability margins were also achieved.

12.5 FMC DESIGNS

Development of the FMC designs paralleled that of the GLA designs discussed in Subsection 12.4. For the full-state design, the cost function used was similar to that given in equation (20). FMC is achieved by satisfying the structural damping requirements of $\zeta = 0.015$ for flight conditions 5 and 6 and $\zeta = 0$ for flight conditions 7 and 8. These requirements reflect an allowance for structural damping of $\zeta = 0$ at flight conditions 5 and 6 and $\zeta = 0.015$ at flight conditions 7 and 8. Other full-state design objectives included moderate rms aileron rate limits of 50 deg/s (flight conditions 5 and 6) and 75 deg/s (flight conditions 7 and 8) for a gust intensity of 4.27-m/s (14-ft/s) rms.

As reflected in the cost function parameters shown in Table 40, torsion was more of a problem than it was in the gust-load designs. Tables 49 through 52 list control surface activity and the damping ratio of the flutter mode.

The flutter mode, caused primarily by coupling between wing vertical bending, wing torsion, and nacelle strut vertical bending, occurs at approximately 20.1 rad/s.

For flight conditions 5, 6, and 7, design A significantly increases the short-period frequency while it decreases the phugoid frequency. In addition to stabilizing the flutter mode in all four flight conditions, damping ratio also increases significantly for a

Table 49. Root-Mean-Square Gust Response, Flight Condition 5

Design	Filter order	Rms control surface activity [1]				Flutter-mode characteristics	
		Elevator		Outboard aileron			
		Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s	Damping ratio	Frequency, rad/s
Open loop						-0.002	19.8
A		0.665	5.21	2.08	31.6	0.080	19.2
B	53	0.536	4.77	1.89	28.1	0.080	19.2
C [2]	53	0.513	4.50	1.81	26.6	0.080	19.2
D [2]	10	0.524	5.25	1.51	20.1	0.023	19.9
G	10	0.309	3.18	1.36	24.7	0.020	20.6
H	8	0.013	0.165	1.36	24.5	0.019	20.5

[1] Response to 4.27-m/s (14-ft/s) rms gust intensity

[2] DRO gain and phase margin requirements not satisfied

Table 50. Root-Mean-Square Gust Response, Flight Condition 6

Design	Filter order	Rms control surface activity [1]				Flutter-mode characteristics	
		Elevator		Outboard aileron			
		Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s	Damping ratio	Frequency, rad/s
Open loop						-0.005	19.8
A		0.615	4.44	1.77	40.7	0.074	19.8
B [2]	53	0.537	4.26	1.49	27.9	0.074	19.8
C [2]	53	0.506	3.98	1.43	25.9	0.074	19.8
D, G	10	0.477	3.69	1.46	27 3	0 072	20.0
H	8	0.057	0.209	1.49	27.2	0.027	20.3

[1] Response to 4.27-m/s (14-ft/s) rms gust intensity

[2] DRO gain and phase margin requirements not satisfied

Table 51. Root-Mean-Square Gust Response, Flight Condition 7

Design	Filter order	Rms control surface activity *				Flutter-mode characteristics	
		Elevator		Outboard aileron			
		Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s	Damping ratio	Frequency, rad/s
Open loop						-0.014	20.5
A		0.192	1.85	2.64	76.3	0 055	21.3
B	53	0.168	2.21	2.46	58.9	0.055	21.3
D	10	0.162	2.13	2 45	54.6	0.016	20.5
G	10	0 326	4 27	1.93	50.6	0.014	20.2
H	8	0.017	0 416	1.87	45.5	0.011	20 2

* Response to 4.27-m/s (14-ft/s) rms gust intensity

Table 52. Root-Mean-Square Gust Response, Flight Condition 8

Design	Filter order	Rms control surface activity [1]				Flutter-mode characteristics	
		Elevator		Outboard aileron			
		Deflection, deg	Rate, deg/s	Deflection, deg	Rate deg/s	Damping ratio	Frequency, rad/s
Open loop						-0.011	20.3
A		0.091	3.34	2.07	74.2	0.022	20.2
B [2]	53	0.094	3.30	2.15	69.2	0.022	20.2
D	10	0.081	2.57	2.00	58.7	0.025	20.1
G	10	0.530	5.40	2.14	57.3	0.040	20.0
H	8	0.057	0.437	2.11	52.7	0.032	19.9

Flight condition 8

[1] Response to 4.27-m/s (14-ft/s) rms gust intensity

[2] DRO phase and gain margin requirements not satisfied

structural mode of 25.7 and 27.8 rad/s of flight conditions 7 and 8, respectively. This mode appears to consist primarily of the first two wing bending modes and the aft-body bending mode. Open-loop poles for these conditions are tabulated in Volume II, Appendix F (figs. F-17 through F-20).

The nominal Kalman filter (design B) resulted in a significant reduction in aileron rate for flight conditions 6 and 7 and a slight decrease in damping ratio for flight conditions 5 and 7. Due to weaker requirements (table 46), stability margins were less constraining than in flight conditions 1, 2, 3, and 4. However, it might be expected that stability margins would be inadequate near the flutter mode because of the high gains required to stabilize the airplane. Although this was the case for flight conditions 5 and 6, design B provided satisfactory phase and gain margins for flight conditions 7 and 8. Bode plots and closed-loop controller pole tabulations are shown in Volume II, Appendix F, for these conditions.

Because design B provided satisfactory stability margins for flight conditions 7 and 8, the robust Kalman filter was used only for flight conditions 5 and 6. Design C provided some improvement in phase and gain margins with a slight reduction in surface activity.

The procedure to reduce the order of the Kalman filter was identical to that used in the gust-load designs. However, in addition to the lowest frequency modes that provided an eighth-order filter, it was also necessary to retain a complex filter mode at the flutter frequency. The resulting 10th-order filter (design D) satisfied the stability margin requirements at flight conditions 5, 6, 7, and 8. The improved phase margins in the aileron loop were due to a combination of two tendencies at the flutter frequency caused by the filter reduction: sharper peaks in the gain curve and less lag in the phase. Design D also resulted in some reduction in the aileron rate and damping ratio for flight conditions 5, 7, and 8, while the opposite effect was observed for flight condition 6. The damping ratio was significantly reduced in flight conditions 5 and 7 but showed little change in flight conditions 6 and 8.

Because all stability margins were satisfied, it was not necessary to reduce the loop gains as was done in flight conditions 1, 2, 3, and 4. The four flutter designs were integrated by selecting the critical design. Only design D at flight condition 6 provided satisfactory flutter damping at flight conditions for which it was not designed; thus it was used as the integrated flutter design G for flight conditions 5, 6, 7, and 8.

The final step in the design procedure was to integrate the 8th-order design F (which provided satisfactory performance for flight conditions 1, 2, 3, and 4) with the 10th-order design G for flight conditions 5, 6, 7, and 8. These designs were integrated by curve-fitting an eighth-order filter with selected poles and zeros fixed from the gust-load design in each of the four control loops. Three pairs of poles were taken from the gust-load design; the fourth pair (at a frequency of 19.4 rad/s) was included to control flutter. Approximately one-half of the zeros were free to minimize the error between the original curve and the fitted curve. Figures 129 through 132 show the frequency response in each sensor to control surface loop of the 10th-order filter (design G) and the 8th-order fitted filter. The resulting curve fit is good, especially in aileron loop near the flutter frequency.

The integrated eighth-order filter resulting from the curve-fitting technique was expressed as a matrix transfer function. Realization in the time domain was necessary to compute rms performance. This resulted in a 16th-order filter, which was reduced to 8th order by retaining only the static portion of the pitch rate to elevator loop and the wing-tip accelerometer to aileron loop. Elimination of nonessential control loops decreased elevator activity and slightly decreased damping ratio of the flutter mode.

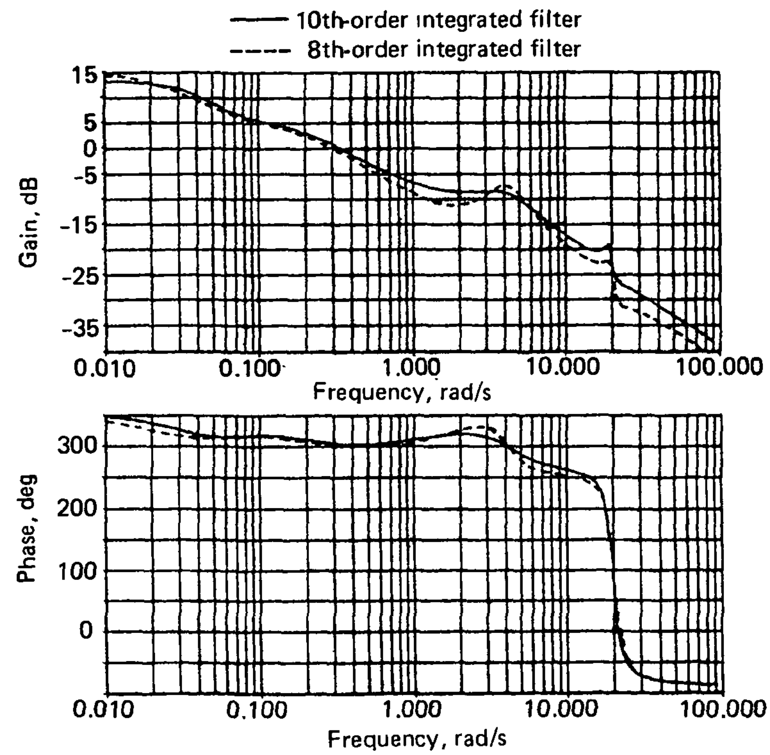


Figure 129. Integrated Flutter-Mode Control Design, Pitch Rate to Outboard Aileron Loop

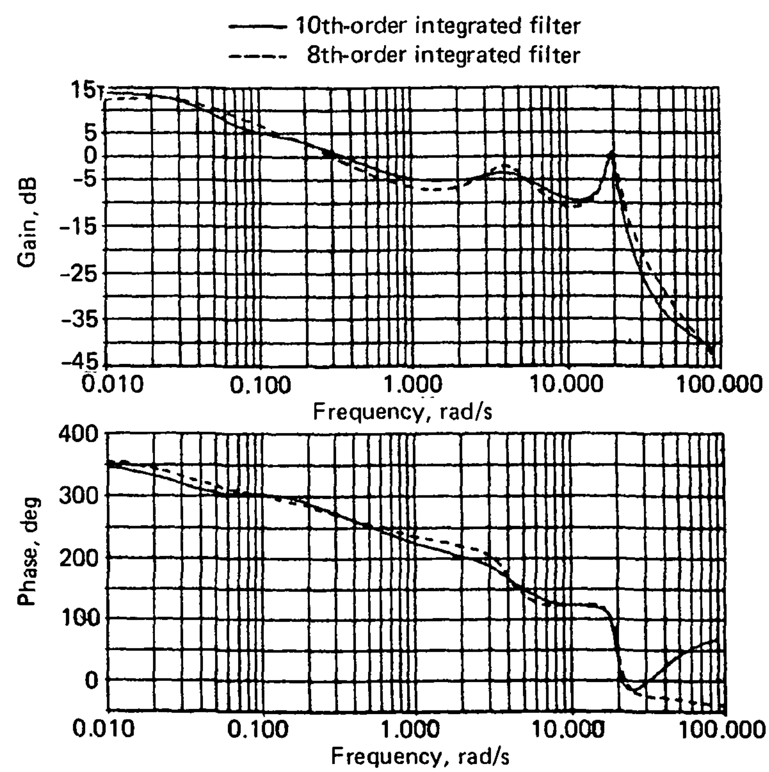


Figure 130. Integrated Flutter-Mode Control Design, Pitch Rate to Elevator Loop

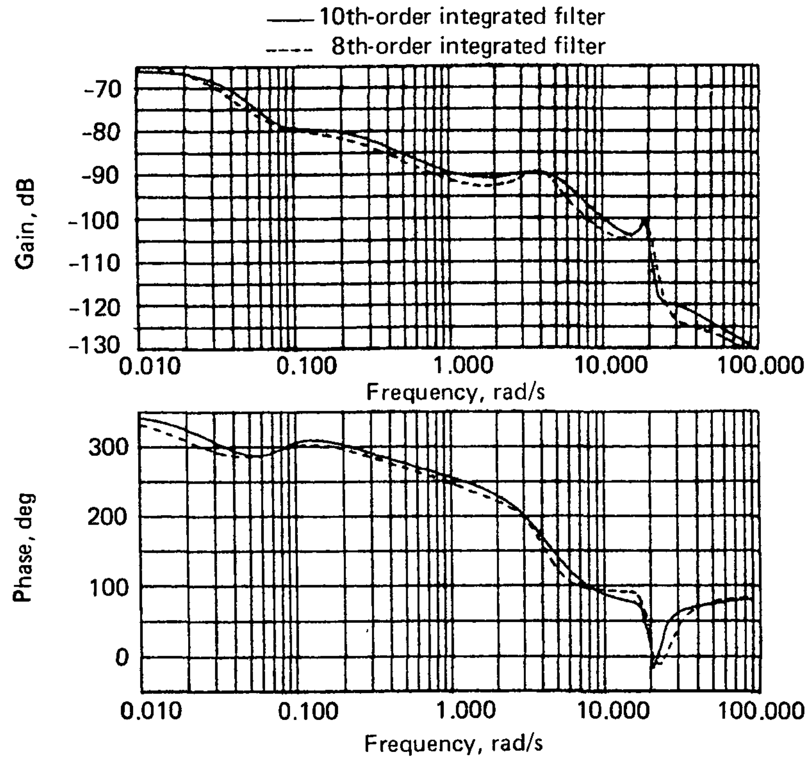


Figure 131. Integrated Flutter-Mode Control Design, Wing-Tip Acceleration to Elevator Loop

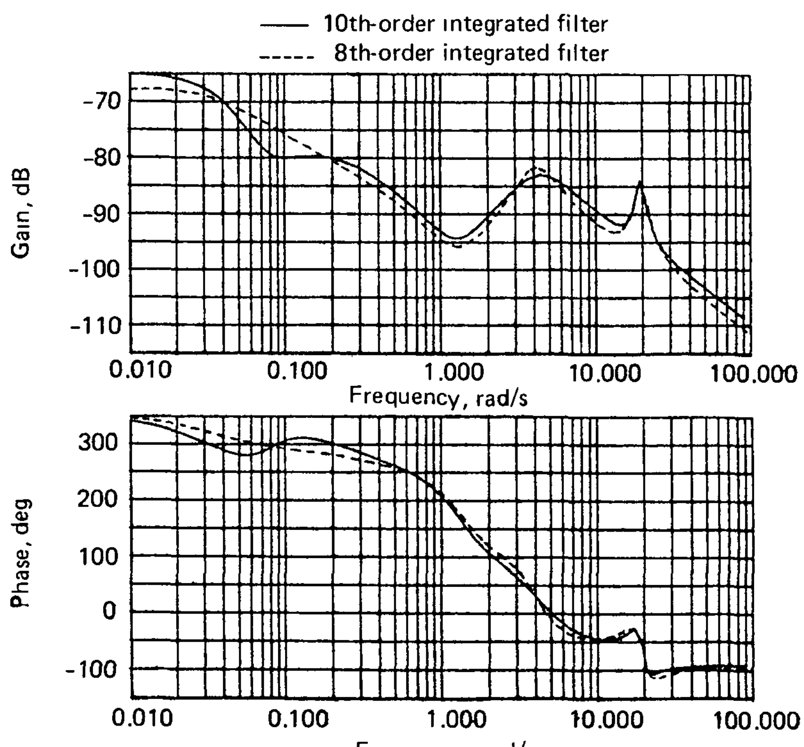
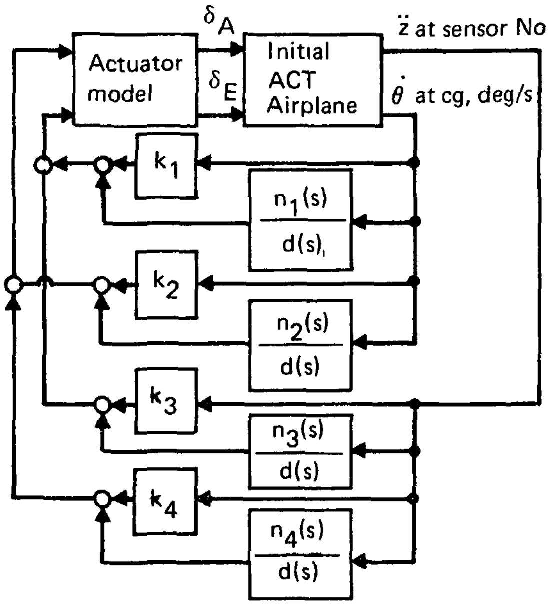


Figure 132. Integrated Flutter-Mode Control Design, Wing-Tip Acceleration to Outboard Aileron Loop

The final integrated design satisfied stability margin requirements at all four flutter flight conditions. Tables 49 through 52 show a slight improvement in control surface activity from design G, as well as a further decline in damping ratios. However, all remained well above minimum acceptable levels and wing loads were essentially unchanged. Activity for both elevator and aileron predominates at a frequency of 20 rad/s for all flight conditions. In addition, the aileron has significant activity above 20 rad/s for flight conditions 7 and 8.

Figure 133 is a block diagram of the final integrated filter and gives the parameter gain schedule as a function of speed. Note that only four parameters change between V_B and V_{MO} speeds.



$$n_1(s) = k_5(s^2 + 0.091s + 0.0027)\,(s + 0.35)\,(s^2 + a_1 s + a_2)\,(s^2 + a_3 s + a_4)$$

$$n_2(s) = k_6(s + 0.33)(s + 0.33)\,(s + 0.018)\,(s^2 + b_1 s + b_2)\,(s + b_3)\,(s + b_4)\,(s + b_5)$$

$$n_3(s) = k_7(s + 5.2)\,(s + 0.315)\,(s^2 + 0.09s + 0.005)\,(s^2 + c_1 s + c_2)\,(s + c_3)$$

$$n_4(s) = k_8(s + 0.022)\,(s^2 + d_1 s + d_2)\,(s^2 + d_3 s + d_4)\,(s + d_5)\,(s + d_6)$$

$$d(s) = (s^2 + 2.5s + 16.7)\,(s^2 + 0.48s + 0.06)\,(s^2 + 0.055s + 8.5 \times 10^{-4})\,(s^2 + p_1 s + p_2)$$

Parameter	Flight condition		Parameter	Flight condition	
	1, 2, 3, 4	5, 6, 7, 8		1, 2, 3, 4	5, 6, 7, 8
p_1	10.69	2.638	b_5	1.20	0.1217
p_2	30.69	376.4	c_1	16.93	10.61
a_1	13.64	4.127	c_2	89.3	674.2
a_2	50.98	5.789	c_3	-0.284	-1.515
a_3	2.053	-3.22	d_1	10.14	11.96
a_4	8.18	432.6	d_2	51.12	324.4
b_1	19.6	14.59	d_3	0.5335	-1.139
b_2	121.0	45.67	d_4	0.07673	1.765
b_3	-0.156	-126.0	d_5	5.38	0.2025
b_4	-1.24	-0.909	d_6	-0.0468	0.2025

Parameter	Flight condition			Parameter	Flight condition		
	1, 2	3, 4	5, 6, 7, 8		1, 2	3, 4	5, 6, 7, 8
k_1	38.8	38.8	3.78	k_5	36.5	36.5	0
k_2	7.37	8.03	0	k_6	-10.9	-11.9	0
k_3	-1.42×10^{-7} (-4.65×10^{-7})	-1.42×10^{-7} (-4.65×10^{-7})	0	k_7	-3.01×10^{-6} (-9.88×10^{-6})	-3.01×10^{-6} (-9.88×10^{-6})	
k_4	5.01×10^{-7} (1.64×10^{-6})	5.46×10^{-7} (1.79×10^{-6})	2.77×10^{-7} (9.08×10^{-7})	k_8	9.24×10^{-6} (3.03×10^{-5})	1.01×10^{-5} (3.30×10^{-5})	6.64×10^{-6} (2.18×10^{-5})

Note: Dimensions are in units of m/s²(ft/s²) and deg/s

Figure 133. Block Diagram of Final Integrated Active Control Law Design

12.6 RESPONSE DUE TO DISCRETE GUSTS

Time history simulation was performed to evaluate the effects of actuator nonlinearities on gust-load reduction. Ordinarily, substantially more computation is required for nonlinear simulation. However, due to a unique formulation of the nonlinear actuator model, it was possible to use the efficient linear simulation algorithm of Subsection 11.3.2 for the nonlinear as well as the linear simulation.

12.6.1 LINEAR SIMULATION MODELS

The models used for linear simulation are derived from the continuous time models used for gust covariance analysis. The closed-loop linear equations consisting of the airplane, actuator, and eighth-order control law are

$$\begin{bmatrix} \dot{x}(t) \\ \dot{\hat{z}}(t) \end{bmatrix} = \begin{bmatrix} A + BF_R C & BG_R \\ S_R C & \Lambda_R \end{bmatrix} \begin{bmatrix} x(t) \\ \hat{z}(t) \end{bmatrix} + \begin{bmatrix} \Gamma + BF_R E \\ S_R E \end{bmatrix} w_g(t)$$

where x(t) is the airplane state vector and $\hat{z}(t)$ is the filter state vector.

The random gust w_g is replaced by a discrete (1-cos) gust, which is given by

$$w_g(t) = \frac{\sigma_d}{2}\left(1-\cos\frac{2\pi tV}{25c}\right) \qquad (21)$$

where σ_d is the discrete gust intensity, t is time, V is true airspeed, and c is the characteristic chord length. Table 53 defines the discrete gust parameters at each flight condition.

The output equation is given by

$$[C \,\vdots\, 0] \begin{bmatrix} x(t) \\ \hat{z}(t) \end{bmatrix} + Dw_g(t) \qquad (22)$$

Table 53. Discrete Gust Characteristics

Flight condition	Gust intensity, σ_d, true airspeed	Duration
1, 2	28.4 m/s (93.4 ft/s)	0.591 sec
3, 4	20.9 m/s (68.4 ft/s)	0.568 sec
5, 6	9.8 m/s (32.2 ft/s)	0.547 sec

As described in Subsection 11.3.2, the system is block diagonalized and subsequently converted to a set of difference equations of the form

$$z(t+\Delta t) = \bar{\phi} z(t) + \bar{\theta} w_g(t) \tag{23}$$

$$y(t) = \bar{C} z(t) + \bar{D} w_g(t) \tag{24}$$

where z(t) is a set of modal coordinates. Note the gust input is assumed to be constant in the time interval between t and t + Δt. This model is then used by the algorithm of Subsection 11.3.2 to compute time histories of selected performance criteria.

12.6.2 NONLINEAR SIMULATION MODELS

The airplane, control law, and discrete gust models used for nonlinear simulation are identical to those of Subsection 12.6.1. However, the actuator model for each control surface was removed from the system matrix and modified to include the nonlinearities of position limit, rate limit, and hysteresis.

The nonlinear actuator was represented by two linear models. The linear actuator of Figure 123 and given in block diagram form by Figure 134 is used when the actuator rate is below the specified rate limit. Whenever the rate exceeds the rate limit, the actuator model is changed from that shown in Figure 134 to the model shown in Figure 135. The latter is a linear model that approximates the characteristics of a rate-limited actuator. The effects of the rate limit are obtained by feeding back the actuator rate with a very high gain (1000 times that of the nominal model).

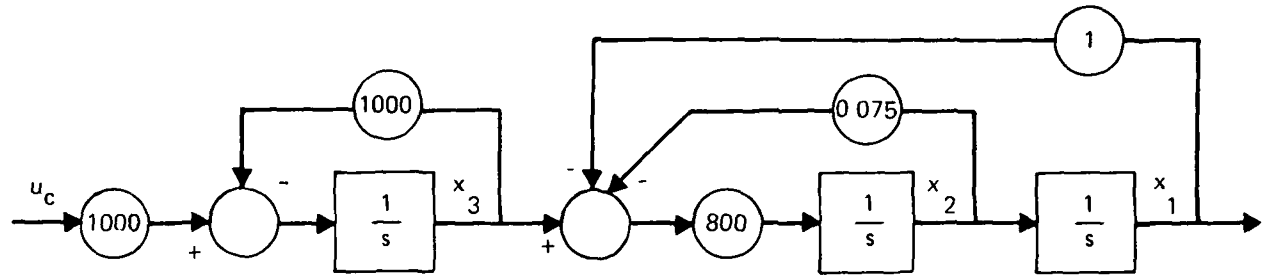


Figure 134. Actuator in Linear Range

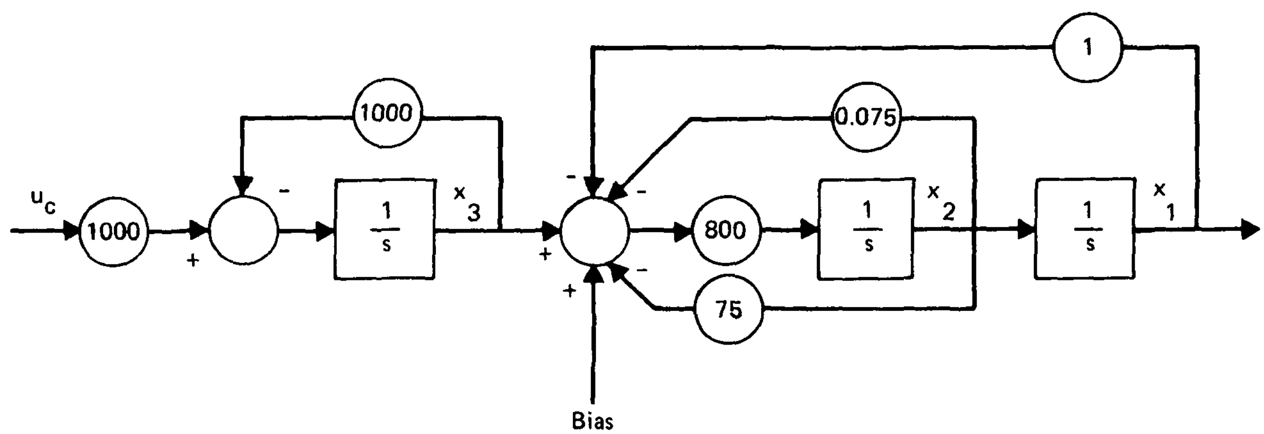


Figure 135. Actuator in Rate-Limited Range

To ensure a smooth transition from one model to another, a bias input is used as shown in Figure 135. This bias is simply the value of the rate limit multiplied by the gain in the additional rate feedback loop. When a transition is made from the rate-limited actuator (fig. 135) to the linear actuator (fig. 134), the bias is removed.

A position limit and hysteresis were included at the command path to the actuator. The position limit is incorporated by simply limiting the command from the controller to the actuator. Figure 136 is a block diagram of the algorithm that implemented the model of the nonlinear actuator. The hysteresis effects in deflection (±0.25 deg) were modeled

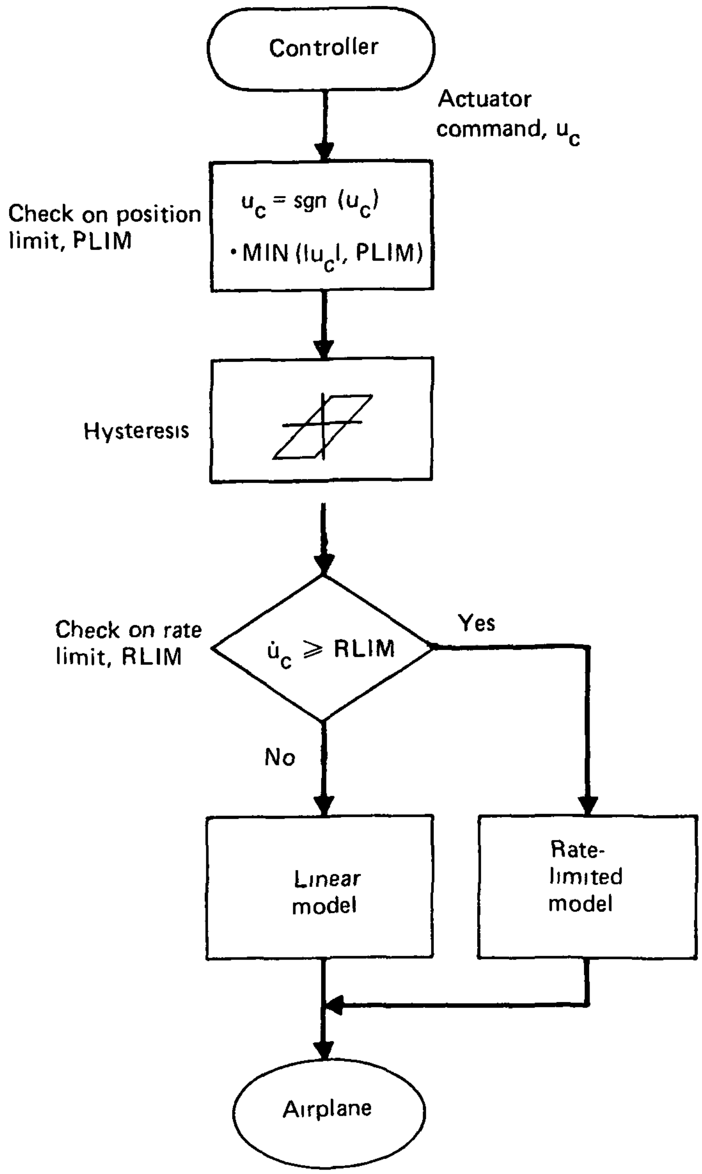


Figure 136. Nonlinear Actuator Model

by a typical hysteresis algorithm. Table 54 lists the nonlinear parameters used for each actuator during the nonlinear simulation.

The two nonlinear actuators were then combined with the airplane and control law as shown in Figure 137. The complete nonlinear simulation functions as three separate coupled linear simulations with a delay between the airplane-controller simulation and the actuator simulations. A small time step ensures that the error introduced by this temporal decoupling is insignificant.

Table 54. Nonlinear Actuator Parameters

	Elevator	Outboard aileron
Position limit	±10.0 deg	±7.5 deg
Rate limit	40.0 deg/s	55.0 deg/s
Hysteresis	±0.25 deg	±0.25 deg

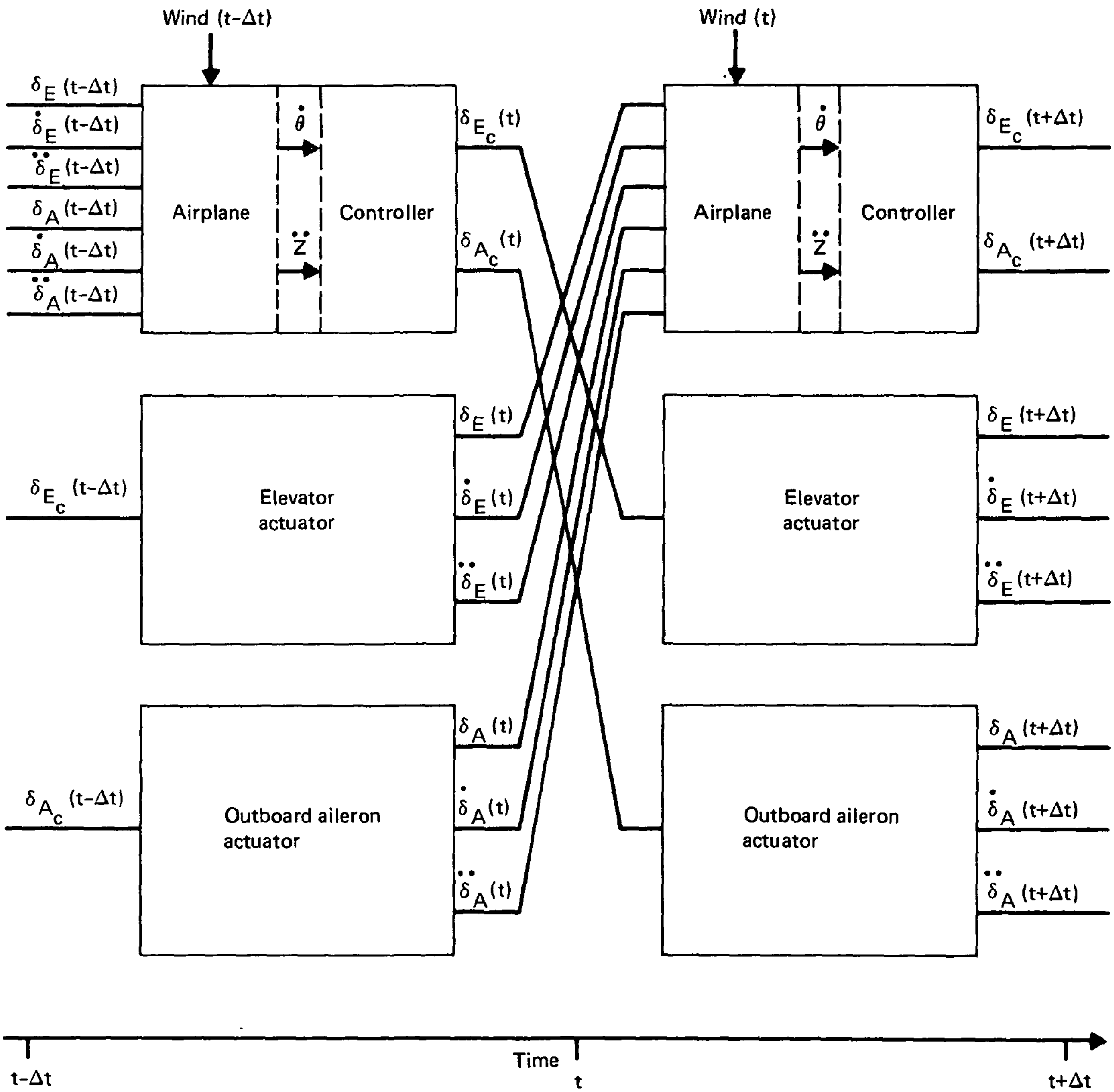


Figure 137. Nonlinear Simulation Diagram

12.6.3 SIMULATION RESULTS

Time history simulations were performed to evaluate the wing loads and control surface activity in response to a discrete (1-cos) wind gust at flight conditions 1, 2, 3, 4, 5, and 6. (Table 36 defines the various flight conditions.) Wing loads include shear, bending moment, and torsion at inboard ($\eta = 0.25$) and outboard ($\eta = 0.75$) stations. All loads evaluated are incremental from the nominal steady-state loads in 1g flight. Control surface activity includes the elevator and outboard aileron rate and deflection.

At flight conditions 1, 2, 3, and 4, the airplane is stable and the primary consideration in the design process was load reduction. Incremental wing loads and control surface activity are shown for these flight conditions. At flight conditions 5 and 6, the unaugmented airplane is unstable and only control surface activity is shown.

Time histories of the wing-load response parameters and control surface activity are shown in Volume II, Appendix F, for all conditions investigated. Each chart superimposes open-loop airplane response, response of the closed-loop system with a linear actuator, and closed-loop response with a nonlinear actuator.

The actuator nonlinearity had little effect on inboard loads; however, outboard bending moment reduction was significantly affected. Load reduction benefits were reduced from approximately 32% for the linear actuator to approximately 25% with the nonlinearities for flight conditions 1 and 2 and from approximately 40% to 30% for flight conditions 3 and 4. Effects on outboard shear were less pronounced. The torsion load increases that accompany bending and shear load reduction were, as would be expected, slightly reduced by the actuator nonlinearity.

For flight conditions 1, 2, 3, and 4, elevator peak command did not exceed the limit of 10 deg. The elevator peak rate command varied from 49 to 59 deg/s compared to the limit of 40 deg/s. The outboard aileron peak command was approximately 3 deg above the limit of 7.5 deg, while the corresponding command peak rate was nearly double the limit of 55 deg/s.

In flight conditions 5 and 6, the nonlinear actuator limits imposed a very slight constraint on system performance. Only the outboard aileron peak rate command exceeded the specified limit. With such low elevator activity, hysteresis was a more significant nonlinearity than the aileron rate limit. Performance data for the six flight conditions are summarized in Tables 55 through 60.

To better study the effects of actuator nonlinearities on system performance, nonlinear simulations were performed for a range of gust intensities. Even with gust intensities at three times the nominal level, some load relief was experienced. Inboard at approximately 35% to 45% of the bending moment, reduction experienced at the nominal gust intensity was lost. Outboard, where the load reduction was much greater at the nominal level, approximately 65% of the benefit was lost. The decreased benefit of higher gust levels caused torsion loads to decrease toward open-loop levels.

Although load reduction was not the primary objective of the flutter suppression designs for flight conditions 5 and 6, the system robustness was checked at these flight conditions to determine the effect of actuator nonlinearities on stability. At greater than three times the nominal gust intensity, no adverse stability effects were observed.

Table 55. Discrete Gust Response for Flight Condition 1

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Actuator	Inboard, $\eta = 0.25$			Outboard, $\eta = 0.75$		
	Shear	Bending moment	Torsion	Shear	Bending moment	Torsion
Linear	105.0	91.1	106.0	73.1	68.7	108.0
Nonlinear	105.0	92.1	105.0	78.4	75.4	107.0

(b) Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
Linear	6.52	55.6	10.3	104.0
Nonlinear	6.53	40.0*	7.5*	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 28.5 m/s (93 4 ft/s)

*Limit value

Table 56. Discrete Gust Response for Flight Condition 2

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Actuator	Inboard, $\eta = 0.25$			Outboard, $\eta = 0.75$		
	Shear	Bending moment	Torsion	Shear	Bending moment	Torsion
Linear	102.0	90.2	107.0	84.5	67.0	111.0
Nonlinear	102.0	91.5	105.0	87.1	74.7	109.0

(b) Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
Linear	8.29	58.7	10.6	106.0
Nonlinear	8.06	40.0*	7.5*	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 28.5 m/s (93.4 ft/s)

*Limit value

Table 57. Discrete Gust Response for Flight Condition 3

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Actuator	Inboard, $\eta = 0.25$			Outboard, $\eta = 0.75$		
	Shear	Bending moment	Torsion	Shear	Bending moment	Torsion
Linear	109.0	95.7	110.0	67.2	60.9	115.0
Nonlinear	108.0	96.0	108.0	73.6	69.5	113.0

(b) Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
Linear	5.07	49.3	10.1	104.0
Nonlinear	5.38	40.0*	7.5*	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 20.9 m/s (68.4 ft/s)

*Limit value

Table 58. Discrete Gust Response for Flight Condition 4

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Actuator	Inboard, $\eta = 0.25$			Outboard, $\eta = 0.75$		
	Shear	Bending moment	Torsion	Shear	Bending moment	Torsion
Linear	106.0	95.0	112.0	86.9	60.1	118.0
Nonlinear	105.0	95.5	110.0	89.2	69.6	115.0

(b) Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection,deg	Rate, deg/s
Linear	6.73	52.7	10.4	107.0
Nonlinear	6.88	40.0*	7.5*	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 20.9 m/s (68.4 ft/s)

*Limit value

Table 59. Discrete Gust Response for Flight Condition 5

Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
Linear	1.21	12.3	5.12	58.6
Nonlinear	1.06	11.1	5.04	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 9.8 m/s (32.2 ft/s)

*Limit value

Table 60. Discrete Gust Response for Flight Condition 6

Peak Control Surface Activity

Actuator	Elevator		Outboard aileron	
	Deflection, deg	Rate, deg/s	Deflection, deg	Rate, deg/s
Linear	1.71	14.3	4.73	60.2
Nonlinear	1.56	13.2	4.64	55.0*

- Gust shape = (1 - cos)
- Gust intensity = 9.8 m/s (32.2 ft/s)

*Limit value

12.7 COMPARISON WITH CURRENT TECHNOLOGY CONTROL LAWS

Classical techniques were also used to synthesize control laws for flight conditions 1 through 8. Performance data will be shown to compare the classical and optimal designs for flight conditions 2, 4, and 6. The results used for comparison are based on rms response to a von Karman gust and simulation with a discrete (1-cos) gust.

Several distinctions should be noted regarding the two designs:

- The optimal controller is designed as an integrated multiloop filter that addresses the functions of GLA, FMC, and pitch-augmented stability (PAS). The classical approach relies on single-loop design techniques, which results in a separately designed filter for each function. In addition to GLA, FMC, and PAS, the classical design also incorporated maneuver-load control (MLC).

- The optimal design used the outboard aileron as one surface, together with the elevator, for all control tasks. In the classical design, the outboard aileron was split into an inboard section and an outboard section. FMC was accomplished by the inboard section, while the outboard section was dedicated to load reduction. The elevator was used primarily for PAS.

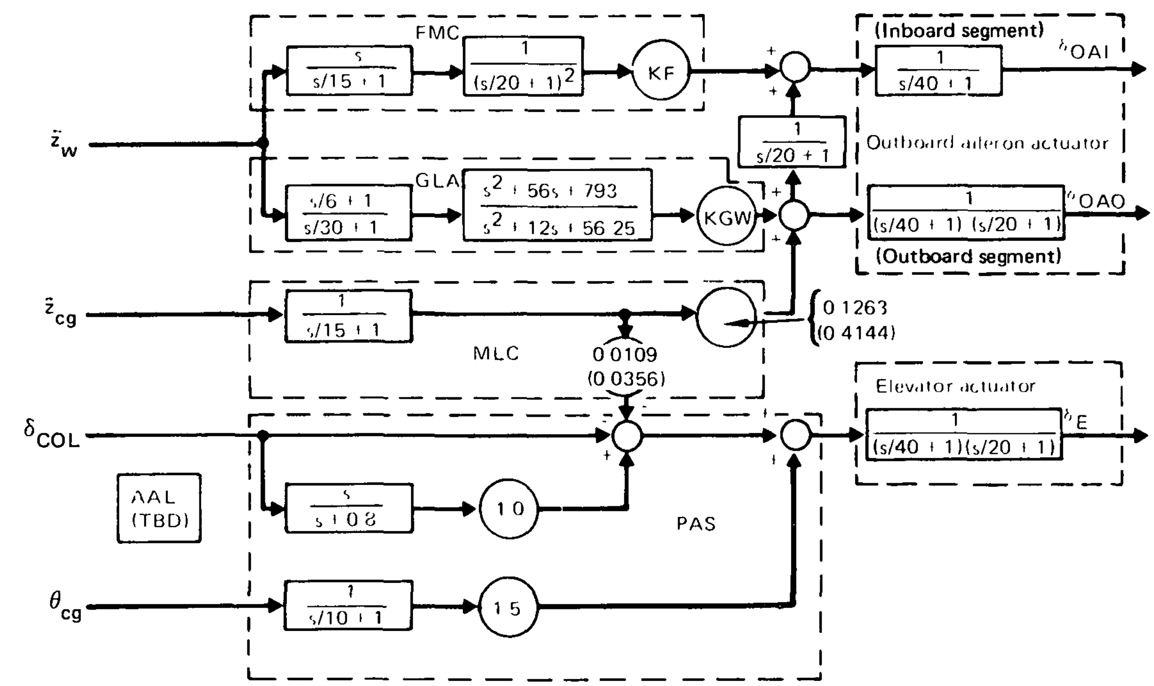
- Both designs use two sensors: a pitch-rate gyro at the airplane center of gravity and a wing accelerometer. The classical design used sensor 11, which is somewhat inboard from the wing-tip accelerometer (sensor 27) of the optimal design (fig. 124).

- The classical design was evaluated using the von Karman turbulence model. The optimal controller was designed to provide optimum performance to Dryden turbulence and was evaluated using both Dryden and von Karman models.

The classical design is shown in Figure 138. For comparison, the optimal design is shown in Figure 133.

The performance of both designs is summarized for flight conditions 2, 4, and 6. Tables 61, 62, and 63 show the response to a random gust. The classical control law performance is shown in response to a von Karman wind model. The response of optimal design to both Dryden and von Karman turbulence is shown. Had the optimal controller been designed for the von Karman spectrum, its performance would likely be better.

In achieving similar load reduction for flight conditions 2 and 4, the classical design required significantly greater aileron activity (particularly in aileron rate) and somewhat less elevator activity than the optimal design. For both designs, the elevator activity is modest when compared to the aileron activity. The FMC design for flight condition 6 showed similar results for control surface activity. Both designs exceeded the damping ratio requirement for the flutter mode of $\zeta \geq 0.015$.



Legend

KF	FMC gain
KGW	GLA gain
δ_{COL}	control column deflection, deg
δ_E	elevator deflection, deg
δ_{OAI}	outboard aileron deflection, inboard segment, deg
δ_{OAO}	outboard aileron deflection, outboard segment, deg
$\dot{\theta}_{cg}$	pitch rate, deg/s
$\ddot{z}_{cg}$	sensor normal acceleration at cg, m/s^2 (ft/s^2)
$\ddot{z}_w$	wing normal acceleration at sensor No. 11, m/s^2 (ft/s^2)

Parameter	Flight condition		
	2	4	6
KGW	0.00189 (0.006188)	0.00105 (0.003438)	0.0
KF	0.00349 (0.01146)	0.00349 (0.01146)	0.0125 (0.04097)

Figure 138. Block Diagram of Classical Control Law Design

Table 61. Random Gust Response Comparison, Flight Condition 2

(a) Incremental Load Reduction (Percent of Open Loop)

Design	Gust	Inboard bending moment, $\eta = 0.25$	Outboard bending moment, $\eta = 0.75$	Inboard torsion, $\eta = 0.25$
Optimal	Von Karman	71.2	65.6	91.2
	Dryden	68.5	61.9	87.4
Classical	Von Karman	70.4	61.9	101.0

(b) Control Surface Activity

Design	Gust	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron deflection, deg	Outboard aileron rate, deg/s
Optimal	Von Karman	1.25	9.90	1.88	24.1
	Dryden	1.27	8.88	1.79	20.0
Classical	Von Karman	1.00	6.16	5.29[a] 4.53[b]	71.0[a] 33.6[b]

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard aileron
• Gust intensity = 8.5 m/s (28 ft/s)

Table 62. Random Gust Response Comparison, Flight Condition 4

(a) Incremental Load Reduction (Percent of Open Loop)

Design	Gust	Inboard bending moment, η = 0.25	Outboard bending moment, η = 0.75	Inboard torsion, η = 0.25
Optimal	Von Karman	76.7	63.2	92.7
	Dryden	73.7	59.8	92.0
Classical	Von Karman	81.3	63.4	89.3

(b) Control Surface Activity

Design	Gust	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron, deflection, deg	Outboard aileron rate, deg/s
Optimal	Von Karman	1.03	10.5	1.99	31.0
	Dryden	1.02	9.05	1.86	25.2
Classical	Von Karman	0.802	6.32	5.26[a] 4.13[b]	79.9[a] 31.1[b]

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard aileron
- Gust intensity = 6.4 m/s (21 ft/s)

Table 63. Random Gust Response Comparison, Flight Condition 6

(a) Control Surface Activity

Design	Gust	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron deflection, deg	Outboard aileron rate, deg/s
Optimal	Von Karman	0.438	4.23	1.68	33.0
	Dryden	0.443	3.70	1.44	27.0
Classical	Von Karman	0.318	2.84	11.0[a] 3.70[b]	250.0[a] 33.0[b]

(b) Damping Ratio of Flutter Mode

Optimal design	0.027
Classical design	0.022

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard aileron
- Gust intensity = 4.3 m/s (14 ft/s)

Comparisons provided in Tables 64, 65, and 66 show the response of each design to a discrete gust. As in the case of random gust response, the classical design requires considerably higher aileron deflection and rate to achieve comparable load reduction. Similarly, the optimal design requires greater elevator activity. However, in both cases the control demand on the elevator is considerably less than the demand on the aileron.

Table 64. Discrete (1-cos) Gust Response Comparison, Flight Condition 2

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Design	Inboard bending moment, $\eta = 0.25$	Outboard bending moment, $\eta = 0.75$	Inboard torsion, $\eta = 0.25$
Optimal	90.2	67.0	107.0
Classical	85.4	70.3	116.0

(b) Control Surface Activity

Design	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron deflection, deg	Outboard aileron rate, deg/s
Optimal	8.29	58.7	10.6	106.0
Classical	5.30	35.1	21.9[a] 21.6[b]	175.0[a] 142.0[b]

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard

- Gust intensity = 28.5 m/s (93.4 ft/s)

Table 65. Discrete (1-cos) Gust Response Comparison, Flight Condition 4

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Design	Inboard bending moment, η = 0.25	Outboard bending moment, η = 0.75	Inboard torsion, η = 0.25
Optimal	95.0	60.1	118.0
Classical	96.7	69.2	120.0

(b) Control Surface Activity

Design	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron deflection, deg	Outboard aileron rate, deg/s
Optimal	6.73	52.7	10.4	107.0
Classical	4.36	33.6	18.4[a] 18.8[b]	174.0[a] 112.0[b]

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard aileron

- Gust intensity = 20.9 m/s (68.4 ft/s)

Table 66. Discrete (1-cos) Gust Response Comparison, Flight Condition 6

(a) Incremental Load Reduction at Peak (Percent of Open Loop)

Design	Inboard bending moment, η = 0.25	Outboard bending moment, η = 0.75	Inboard torsion, η = 0.25
Optimal	104.0	81.3	111.0
Classical	109.0	65.1	149.0

(b) Control Surface Activity

Design	Elevator deflection, deg	Elevator rate, deg/s	Outboard aileron deflection, deg	Outboard aileron rate, deg/s
Optimal	1.71	14.3	4.73	60.2
Classical	1.01	6.73	32.6[a] 11.7[b]	120.0[a] 75.9[b]

[a]Inboard segment of outboard aileron
[b]Outboard segment of outboard aileron

- Gust intensity = 9.8 m/s (32.2 ft/s)

13.0 PITCH AUGMENTATION CONTROL LAWS

The pitch-augmented stability (PAS) system was designed using modern optimal control theory. Full-state feedback systems were designed for two flight conditions. Subsection 13.1 describes the state models for the airplane, actuator, Dryden wind model, and ideal airplane model used for the control design and then describes the formation of the composite state model incorporating all four of the separate state models.

In Subsection 13.1, time responses for step elevator commands are shown for the unstable open-loop airplane at the two flight conditions. The full-state design process using linear quadratic regulator theory with explicit model-following is described in Subsection 13.3. This subsection presents closed-loop analysis of selected feedback controllers and the resulting closed-loop poles. Time responses to step elevator commands as well as gust covariances are shown. Subsection 13.4 compares the performance of the optimal control design with the classical control design.

13.1 DYNAMIC MODEL DESCRIPTION

This subsection describes the state models and output models used for analysis and simulation.

13.1.1 STATE MODELS FOR AIRPLANE

Full-state feedback designs were completed for two flight conditions designated as flight conditions 58 and 97, respectively. These flight conditions and the corresponding state models are defined as follows.

Flight Condition 58—This is at high altitude with an aft center of gravity and is defined by the following parameters:

Altitude, h	=	11 125m (36 500 ft)
Mach number	=	0.65
Speed, V	=	192 m/s (629 ft/s)
Gross mass (weight)	=	122 000 kg (270 000 lb)
Center of gravity	=	$0.46\bar{c}$

The state model, in stability axes, is given by the following equation:

$$\begin{bmatrix} \dot{u} \\ \dot{\alpha} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} -5.04 \times 10^{-3} & 12.85 & 0.422 & -32.2 \\ -1.85 \times 10^{-4} & -0.388 & 0.996 & 1.88 \times 10^{-5} \\ -5.39 \times 10^{-4} & 0.413 & -0.276 & 0 \\ 0 & 0 & 1.0 & 0 \end{bmatrix} \begin{bmatrix} u \\ \alpha \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} 6.31 \times 10^{-3} \\ -2.2 \times 10^{-2} \\ -1.16 \\ 0 \end{bmatrix} \delta_E \tag{25}$$

where δ_E is the elevator angle.

The eigenvalues are:

0.367
$-1.1 \times 10^{-3} \pm i3.17 \times 10^{-2}$
-0.973

Flight Condition 97—This is a flaps-down approach condition with the following parameters:

Altitude, h	=	10 668m (35 000 ft)
Mach number	=	0.378
Speed, V	=	112 m/s (368 ft/s)
Gross mass (weight)	=	90 720 kg (200 000 lb)
Center of gravity	=	$0.46\bar{c}$

The state model, in stability axes, is given by the following equation:

$$\begin{bmatrix} \dot{u} \\ \dot{\alpha} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} -2.16 \times 10^{-2} & 20.87 & -2.3 \times 10^{-2} & -32.2 \\ -4.76 \times 10^{-4} & -0.348 & 0.994 & -5.87 \times 10^{-6} \\ -7.13 \times 10^{-5} & 1.95 \times 10^{-2} & -0.256 & 0 \\ 0 & 0 & 1.0 & 0 \end{bmatrix} \begin{bmatrix} u \\ \alpha \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} 6.88 \times 10^{-4} \\ -2.21 \times 10^{-2} \\ -0.56 \\ 0 \end{bmatrix} \delta_E \tag{26}$$

The eigenvalues are:

0.0763
-0.128 $\pm$ i0.126
-0.447

13.1.2 ACTUATOR MODEL

The model for the actuator is given in transfer function form as

$$G(s) = \frac{1}{(s/20 + 1)(s/40 + 1)} \tag{27}$$

This is a second-order model with poles at 20 and 40 rad/s and is essentially the same actuator used for the flutter and gust analyses (see fig. 123). The state model selected to represent this transfer function in the time domain is

$$\begin{bmatrix} \dot{\delta}_{E'} \\ \dot{\delta}_E \end{bmatrix} = \begin{bmatrix} -40 & 0 \\ 20 & -20 \end{bmatrix} \begin{bmatrix} \delta_{E'} \\ \delta_E \end{bmatrix} + \begin{bmatrix} 40 \\ 0 \end{bmatrix} \delta_{E_c} \tag{28}$$

where

δ_E = elevator angle

$\delta_{E'}$ = intermediate state variable

δ_{E_c} = elevator command to actuator

A block diagram for this model is shown in Figure 139. The nomenclature that will be used to describe this model as part of the combined state model is

$$x_u = \begin{bmatrix} \delta_{E'} \\ \delta_E \end{bmatrix} \tag{29}$$

$$A_u = \begin{bmatrix} -40 & 0 \\ 20 & -20 \end{bmatrix} \tag{30}$$

$$B_u = \begin{bmatrix} 40 \\ 0 \end{bmatrix} \tag{31}$$

so that equation (28) can be written compactly as

$$\dot{x}_u = A_u x_u + B_u \delta_{E_c} \tag{32}$$

The matrix C_u referred to in the simulation sections is defined as

$$C_u = \begin{bmatrix} 0 & 1 \end{bmatrix} \tag{33}$$

13.1.3 DRYDEN WIND MODEL

This subsection briefly describes the Dryden wind model used for flight conditions 58 and 97. The wind model is described in greater detail in Volume II, Appendix E, Subsection E.1.5.

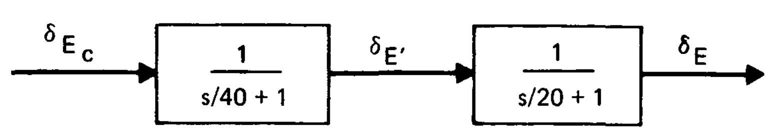


Figure 139. Cascade Representation of Transfer Function

For flight condition 58, the model is

$$\begin{bmatrix} \dot{u}_g \\ \dot{x}_{w2} \\ \dot{w}_g \end{bmatrix} = \begin{bmatrix} -0.629 & 0 & 0 \\ 0 & 0 & -0.396 \\ 0 & 1 & -1.26 \end{bmatrix} \begin{bmatrix} u_g \\ x_{w2} \\ w_g \end{bmatrix} + \begin{bmatrix} 1.12 & 0 \\ 0 & 0.499 \\ 0 & 1.37 \end{bmatrix} \begin{bmatrix} u_n \\ w_n \end{bmatrix} \tag{34}$$

where

u_g = longitudinal gust velocity
x_{w2} = intermediate state for vertical gust velocity
w_g = vertical gust velocity
u_n = white noise input for horizontal gust
w_n = white noise input for vertical gust

For flight condition 97, the model is

$$\begin{bmatrix} \dot{u}_g \\ \dot{x}_{w2} \\ \dot{w}_g \end{bmatrix} = \begin{bmatrix} -0.377 & 0 & 0 \\ 0 & 0 & -0.142 \\ 0 & 1.0 & -0.754 \end{bmatrix} \begin{bmatrix} u_g \\ x_{w2} \\ w_g \end{bmatrix} + \begin{bmatrix} 0.434 & 0 \\ 0 & 0.231 \\ 0 & 1.06 \end{bmatrix} \begin{bmatrix} u_n \\ w_n \end{bmatrix} \tag{35}$$

The wind model equation can be written compactly as

$$\dot{x}_w = A_w x_w + B_w w \tag{36}$$

$$y_w = C_w x_w \tag{37}$$

where

$$x_w^T = \begin{bmatrix} u_g \; x_{w2} \; w_g \end{bmatrix} \tag{38}$$

$$w^T = \begin{bmatrix} u_n \; w_n \end{bmatrix} \tag{39}$$

$$y_w^T = \begin{bmatrix} u_g \; w_g \end{bmatrix} \tag{40}$$

$$C_w = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{41}$$

13.1.4 IDEAL AIRPLANE MODEL FOR EXPLICIT MODEL-FOLLOWING

The philosophy and procedure of explicit model-following is described in Subsection 13.3.1.

For full-state feedback design, the model equations can be summarized in the form

$$\dot{x}_m = A_m x_m \tag{42}$$

where A_m is the 4 x 4 coefficient matrix. The elements of A_m are discussed in Subsection 13.3.1.

The state vector x_m is

$$x_m = \begin{bmatrix} u_m \\ \alpha_m \\ q_m \\ \theta_m \end{bmatrix} \tag{43}$$

where

u_m = incremental forward velocity of model
α_m = incremental angle of attack of model
q_m = incremental pitch rate of model
θ_m = incremental pitch angle of model

13.1.5 COMBINED STATE MODEL

The airplane model, actuator model, wind model, and ideal airplane model are combined to form one large state model with 13 state variables. The state vector, x, is defined as

$$
\begin{aligned}
x^T &= \left[x_a{}^T, x_u{}^T, x_w{}^T, x_m{}^T\right] \\
x^T &= \left[u, \alpha, q, \theta, \delta_E', \delta_E, u_g, x_{w2}, w_g, u_m, \alpha_m, q_m, \theta_m\right]
\end{aligned}
\tag{44}
$$

The state equations can be written compactly as

$$\dot{x} = Ax + B_1 u_c + B_2 w \tag{45}$$

The system coefficient matrix, A, is defined as

$$A = \begin{bmatrix} A_a & B_a C_u & \Gamma_a C_w & 0 \\ 0 & A_u & 0 & 0 \\ 0 & 0 & A_w & 0 \\ 0 & 0 & 0 & A_m \end{bmatrix} \tag{46}$$

where **0** represents a zero matrix, and $\boldsymbol{\Gamma}_a$ is a matrix of coupling coefficients representing the effect of wind on the airplane states. Also,

$$B_1{}^T = \begin{bmatrix} 0 & B_u^T & 0 \end{bmatrix} \tag{47}$$

$$B_2{}^T = \begin{bmatrix} 0 & 0 & B_w^T \end{bmatrix} \tag{48}$$

The A matrix for flight condition 58 is

$$
A = \left[\begin{array}{cccc:cc:ccc:cccc}
-5.04 \times 10^{-3} & 12.85 & 0.422 & -32.2 & 0 & 6.31 \times 10^{-3} & -5.04 \times 10^{-3} & 0 & 2.04 \times 10^{-2} & & & & \\
-1.85 \times 10^{-4} & -0.388 & 0.996 & 1.88 \times 10^{-5} & 0 & -2.2 \times 10^{-2} & -1.85 \times 10^{-4} & 0 & -6.17 \times 10^{-4} & & & & \\
-5.39 \times 10^{-4} & 0.413 & -0.276 & 0 & 0 & -1.16 & -5.39 \times 10^{-4} & 0 & 6.57 \times 10^{-4} & & \mathbf{0} & & \\
0 & 0 & 1.0 & 0 & 0 & 0 & 0 & 0 & 0 & & & & \\
\hdashline
 & & & & -40 & 0 & 0 & 0 & 0 & & & & \\
 & & & & 20.0 & -20.0 & 0 & 0 & 0 & & & & \\
\hdashline
 & & & & & & -0.629 & 0 & 0 & & & & \\
 & \mathbf{0} & & & & & 0 & 0 & -0.396 & & & & \\
 & & & & & & 0 & 1.0 & -1.26 & & & & \\
\hdashline
 & & & & & & & & & -5.04 \times 10^{-3} & 12.85 & 0.422 & -32.2 \\
 & & & & & & & \mathbf{0} & & -1.85 \times 10^{-4} & -0.388 & 0.996 & 0 \\
 & & & & & & & & & -1.57 \times 10^{-4} & -0.464 & -0.863 & 0 \\
 & & & & & & & & & 0 & 0 & 1.0 & 0
\end{array}\right] \tag{49}
$$

The A matrix for flight condition 97 is

$$
A = \left[\begin{array}{cccc:cc:ccc:cccc}
-2.16 \times 10^{-2} & 20.87 & -2.3 \times 10^{-2} & -32.2 & 0 & 6.88 \times 10^{-4} & -2.16 \times 10^{-2} & 0 & 5.67 \times 10^{-2} & & & & \\
-4.16 \times 10^{-4} & -0.348 & 0.994 & -5.87 \times 10^{-6} & 0 & -2.2 \times 10^{-2} & -4.76 \times 10^{-4} & 0 & -9.46 \times 10^{-4} & & & & \\
-7.13 \times 10^{-5} & 1.95 \times 10^{-2} & -0.256 & 0 & 0 & -0.56 & -7.13 \times 10^{-5} & 0 & 5.3 \times 10^{-5} & & \mathbf{0} & & \\
0 & 0 & 1.0 & 0 & 0 & 0 & 0 & 0 & 0 & & & & \\
\hdashline
 & & & & -40.0 & 0 & 0 & 0 & 0 & & & & \\
 & & & & 20.0 & -20.0 & 0 & 0 & 0 & & & & \\
\hdashline
 & & & & & & -0.377 & 0 & 0 & & & & \\
 & \mathbf{0} & & & & & 0 & 0 & -0.142 & & & & \\
 & & & & & & 0 & 1.0 & -0.754 & & & & \\
\hdashline
 & & & & & & & & & -2.16 \times 10^{-2} & 20.87 & -2.3 \times 10^{-2} & -32.2 \\
 & & & & & & & \mathbf{0} & & -4.76 \times 10^{-4} & -0.384 & 0.994 & 0 \\
 & & & & & & & & & -7.13 \times 10^{-5} & -0.148 & -1.152 & 0 \\
 & & & & & & & & & 0 & 0 & 1.0 & 0
\end{array}\right] \tag{50}
$$

13.1.6 OUTPUT MODEL

The output model serves two purposes: (1) provides variables to be used as criteria in the quadratic cost performance function and (2) provides output variables needed for simulation.

The variable selected for the cost performance index was α_{wc}, defined as

$$
\alpha_{wc} = \alpha + \frac{w_g}{V} - \alpha_m \tag{51}
$$

The output variable used in the simulation was the normal acceleration, n_z, defined as

$$
\begin{aligned}
n_z &= \dot{w} - Vq \\
n_z &= V(\dot{\alpha} - q)
\end{aligned}
\tag{52}
$$

The ideal model vertical acceleration, n_{zm}, was also used for simulation. n_{zm} is defined as

$$n_{zm} = V(\alpha_m - q_m) \tag{53}$$

The C matrix is the matrix of coefficients such that

$$y = \begin{bmatrix} \alpha_{wc} \\ n_z \\ n_{zm} \end{bmatrix} = Cx \tag{54}$$

For flight condition 58

$$C = \begin{bmatrix} 0 & 1.0 & 0 & 0 & 0 & 0 & 0 & 0 & 1.59 \times 10^{-3} & 0 & -1.0 & 0 & 0 \\ -0.116 & -244.0 & -2.7 & 1.18 \times 10^{-2} & 0 & -13.9 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & -0.116 & -244.0 & -2.7 & 1.18 \times 10^{-2} \end{bmatrix} \tag{55}$$

For flight condition 97

$$C = \begin{bmatrix} 0 & 1.0 & 0 & 0 & 0 & 0 & 0 & 0 & 2.72 \times 10^{-3} & 0 & -1.0 & 0 & 0 \\ -0.175 & -128.0 & -2.32 & -2.16 \times 10^{-3} & 0 & -8.11 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & -0.175 & -128.0 & -2.32 & -2.16 \times 10^{-3} \end{bmatrix} \tag{56}$$

13.2 OPEN-LOOP ANALYSIS

This subsection presents the characteristics of the unaugmented airplane at flight conditions 58 and 97. Root locations and time responses are shown.

13.2.1 ROOT LOCATIONS

The eigenvalues for the unaugmented airplane presented in Subsection 13.1.1 are shown graphically in Figures 140 and 141.

13.2.2 TIME RESPONSES

A block diagram of the unaugmented airplane is shown in Figure 142. The response to a step column command at flight condition 58 is shown in Figures 143 through 148. Note the rapid divergence due to the unstable short-period root. The response to a step elevator command at flight condition 97 is shown in Figures 149 through 154. Note the slow divergence due to the unstable root.

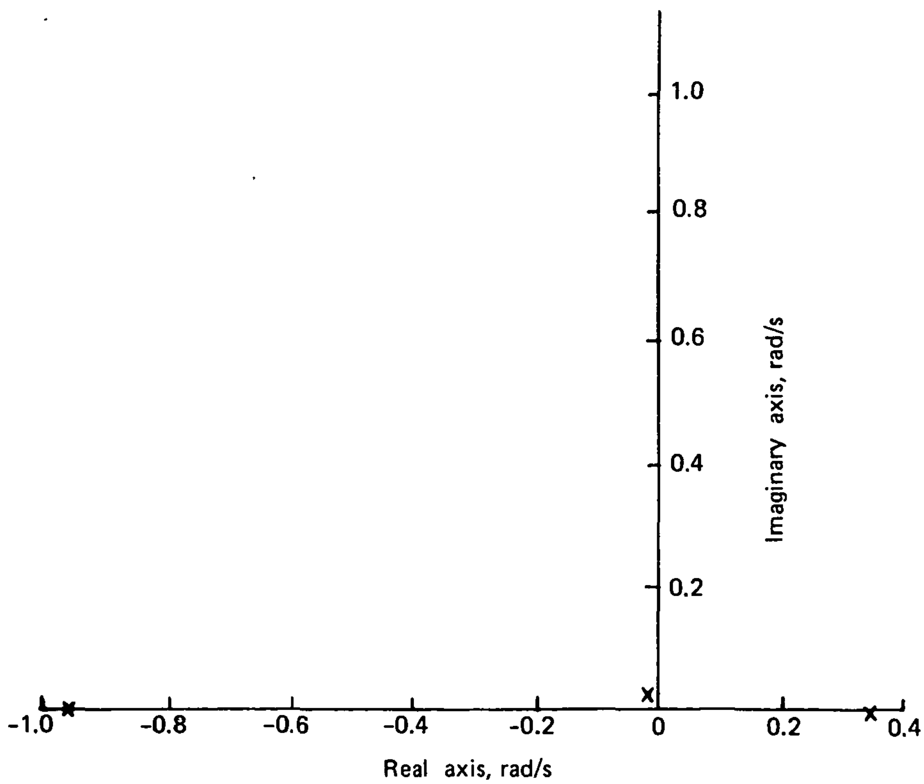


Figure 140. Eigenvalues of Unaugmented Airplane, Flight Condition 58

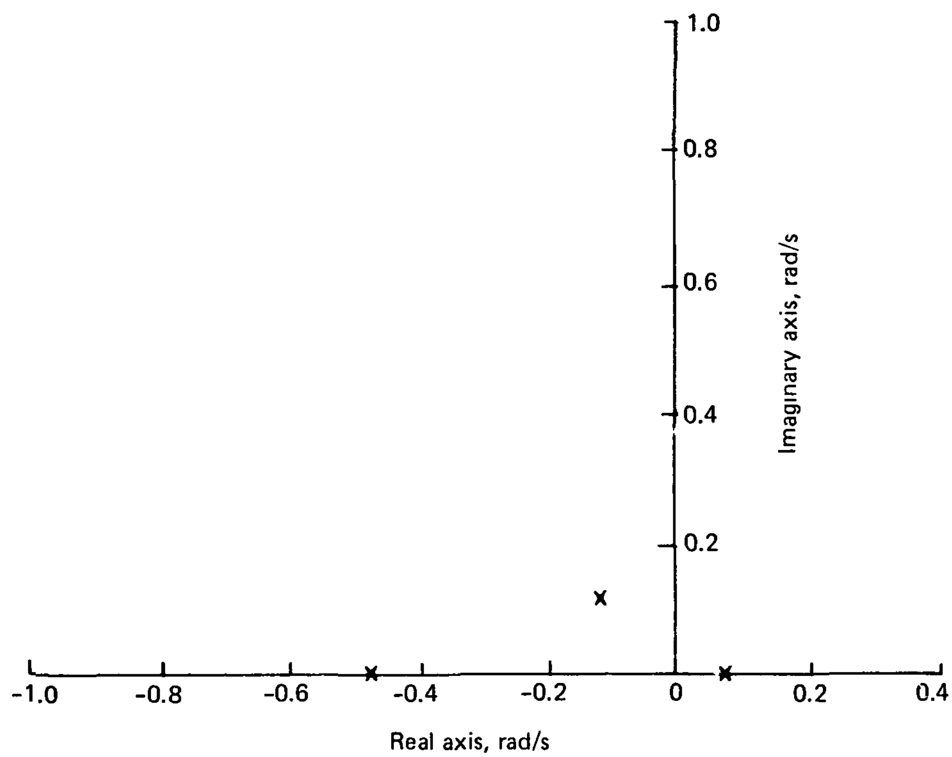


Figure 141. Eigenvalues of Unaugmented Airplane, Flight Condition 97

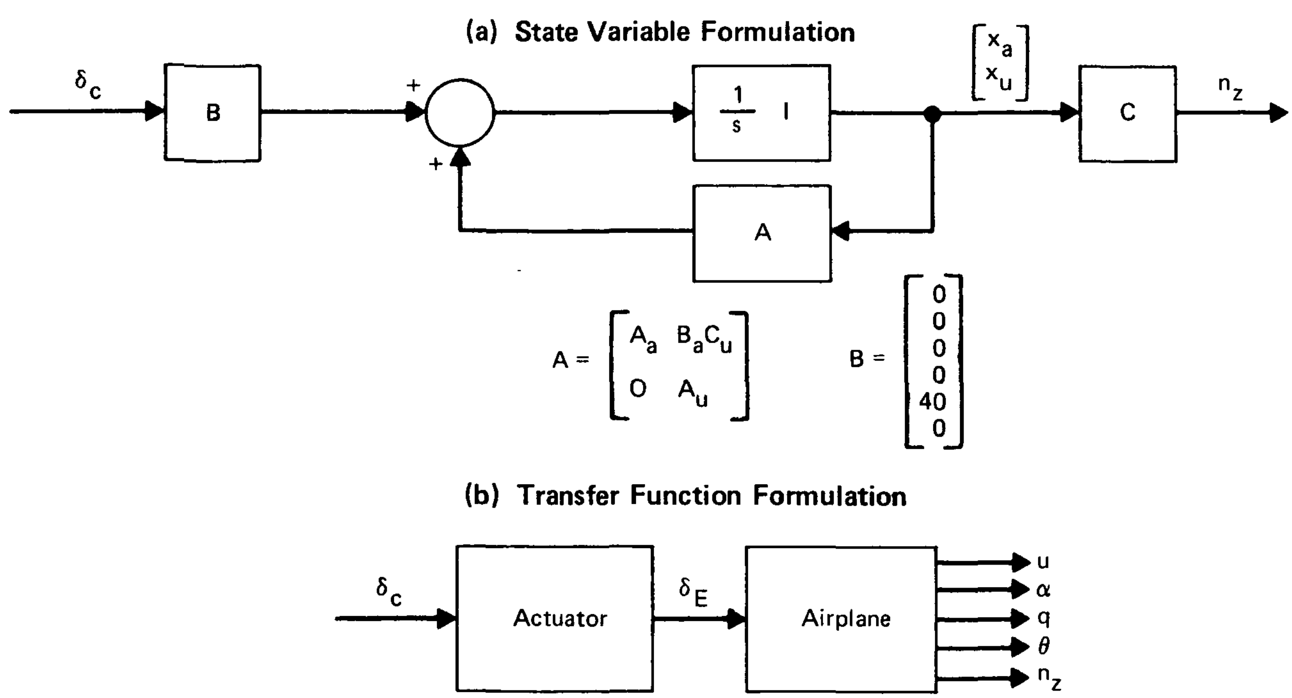


Figure 142. Block Diagram of Unaugmented Airplane for Simulation

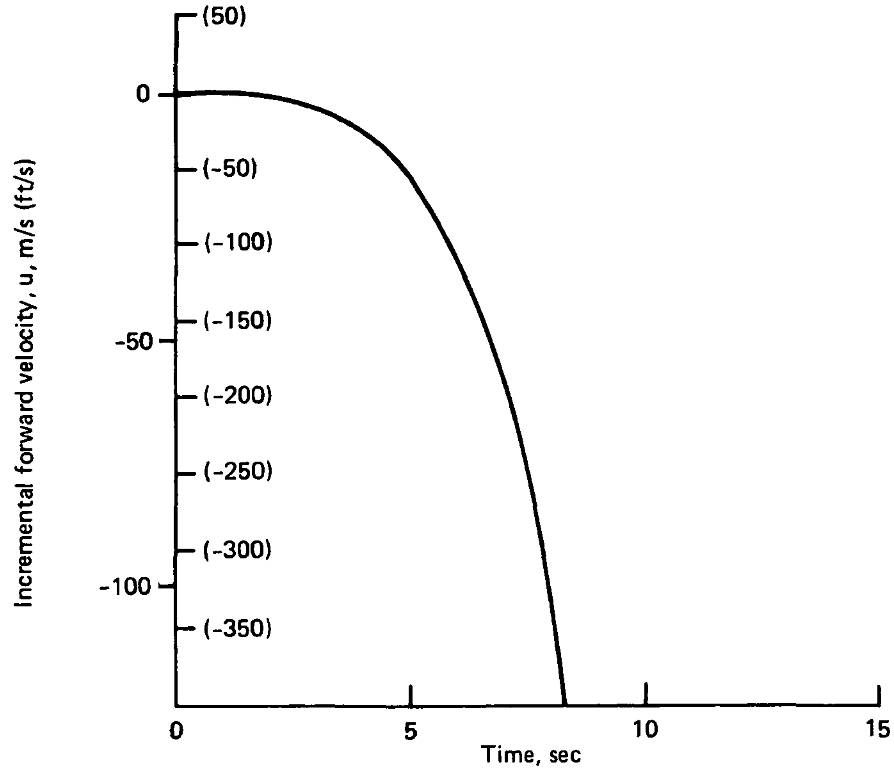


Figure 143. Velocity Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

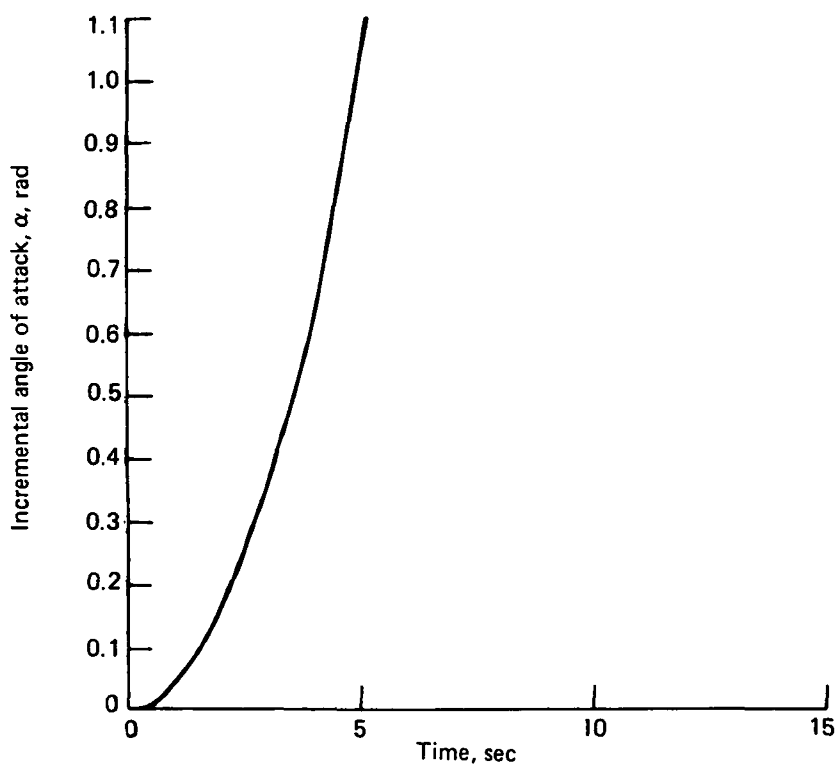


Figure 144. Angle-of-Attack Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

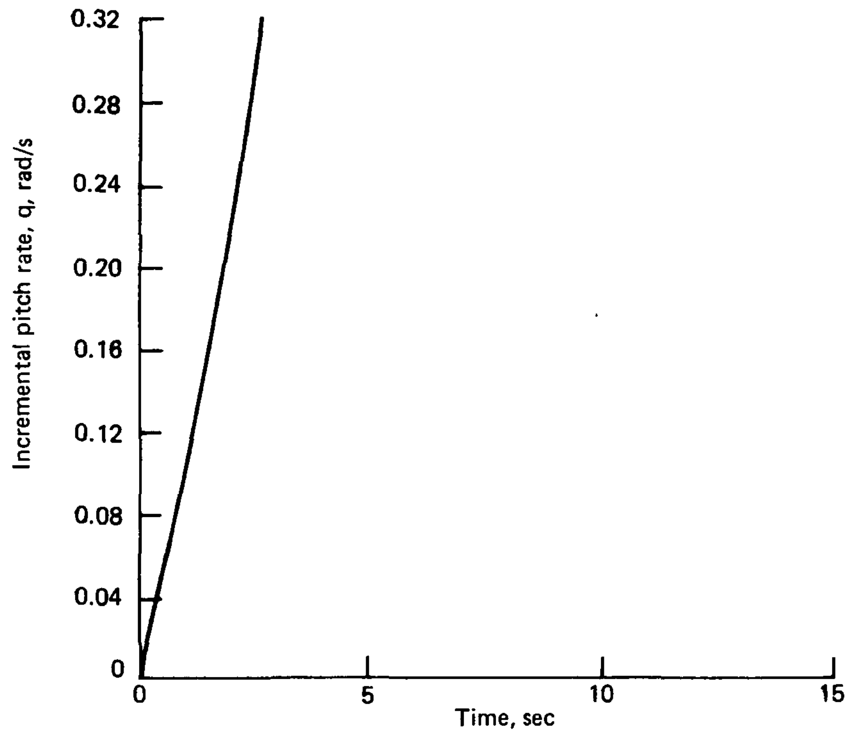


Figure 145. Pitch-Rate Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

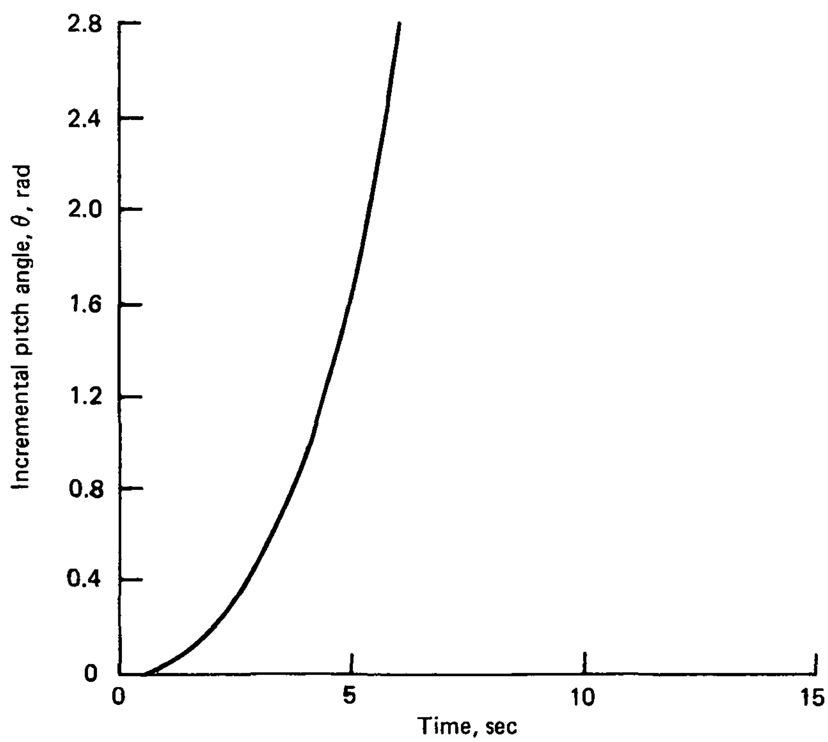


Figure 146. Pitch-Angle Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

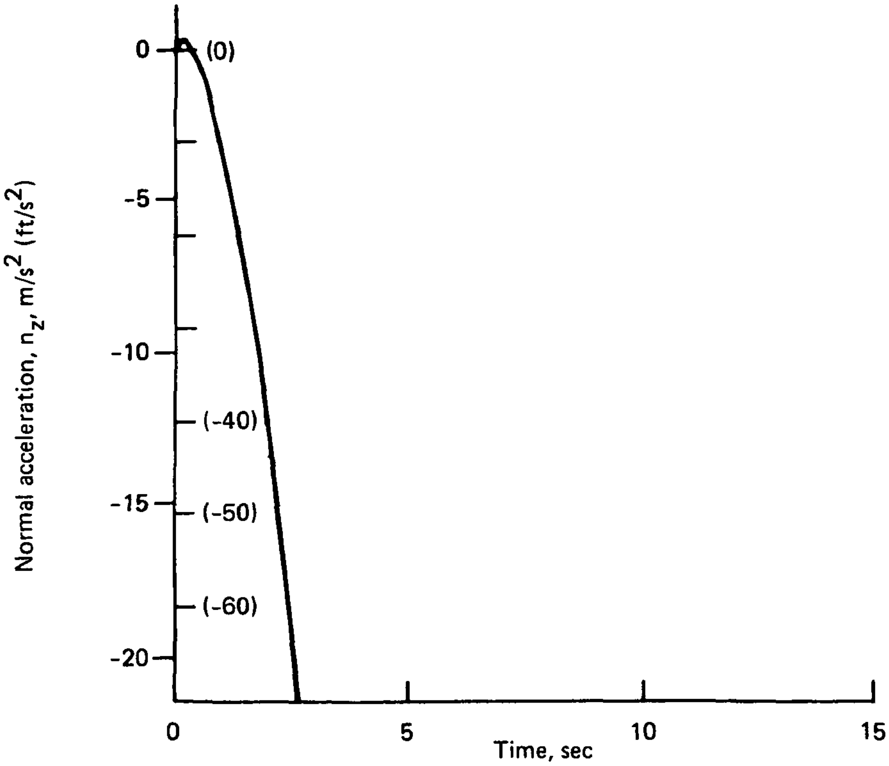


Figure 147. Normal Acceleration Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

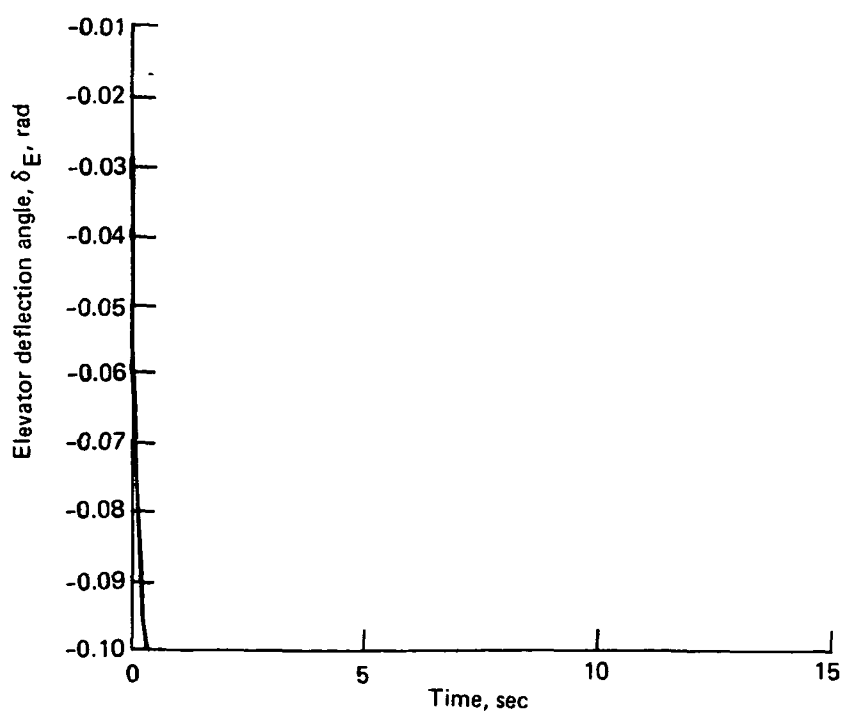


Figure 148. Elevator-Angle Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Open Loop

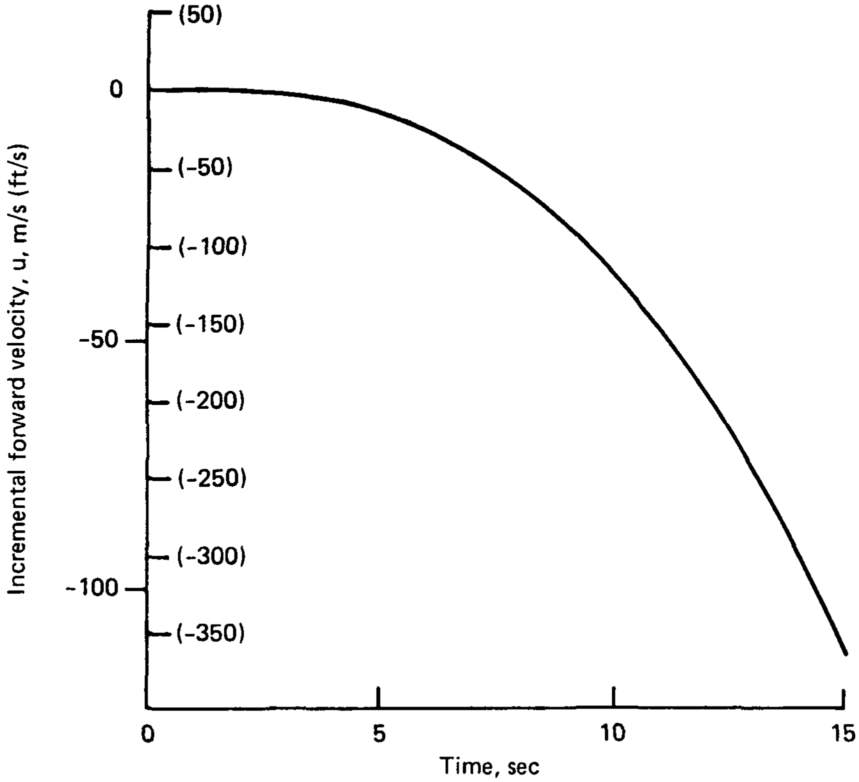


Figure 149. Velocity Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Open Loop

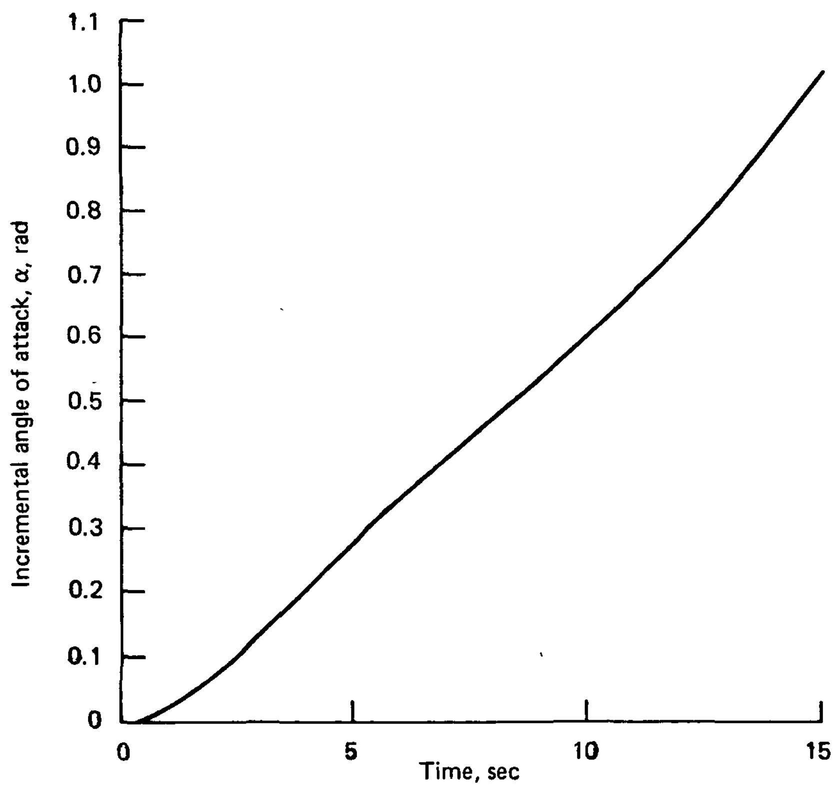


Figure 150. Angle-of-Attack Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Open Loop

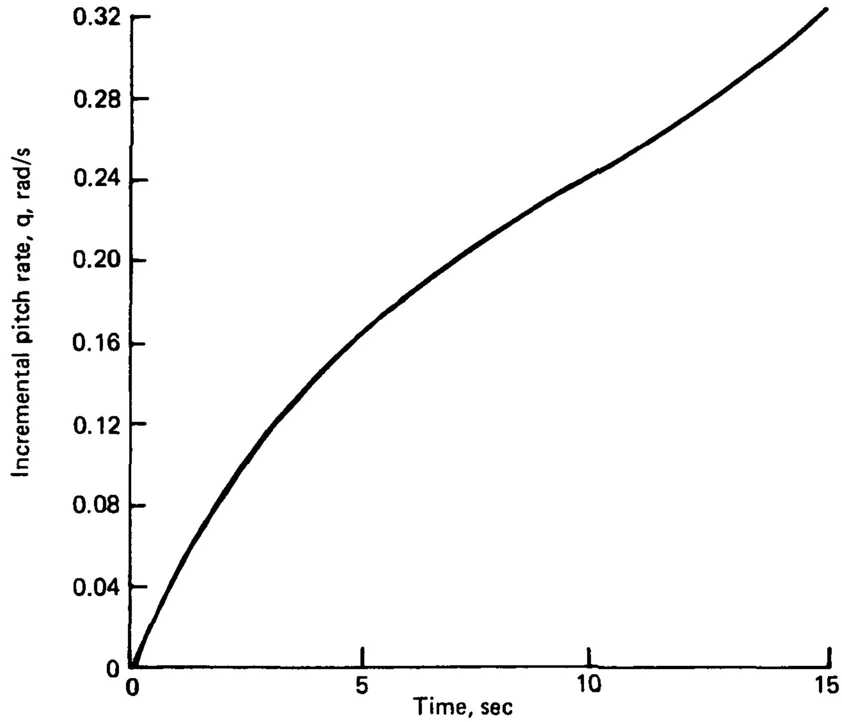


Figure 151. Pitch-Rate Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Open Loop

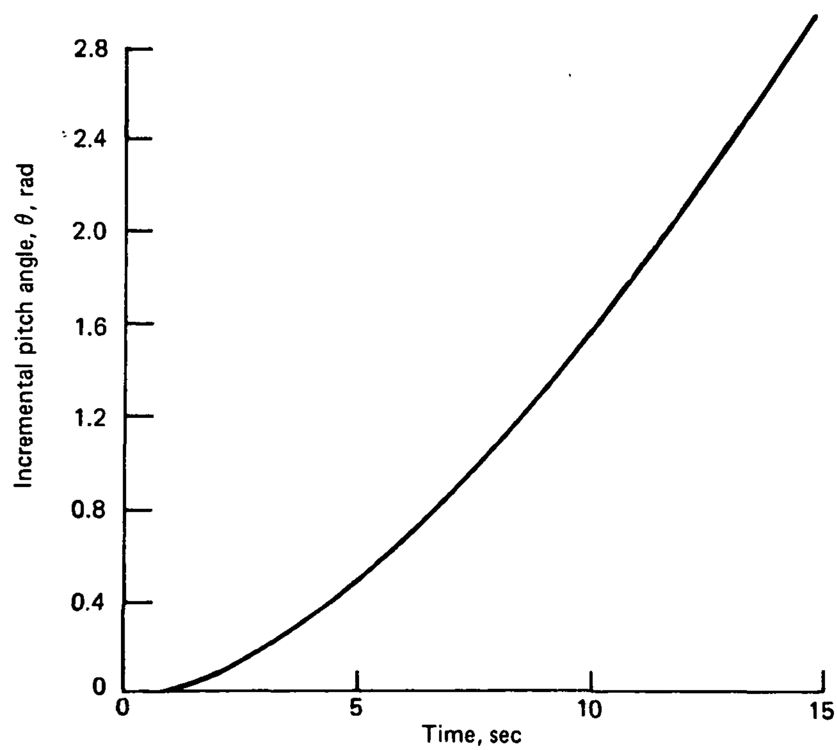


Figure 152. Pitch-Angle Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Open Loop

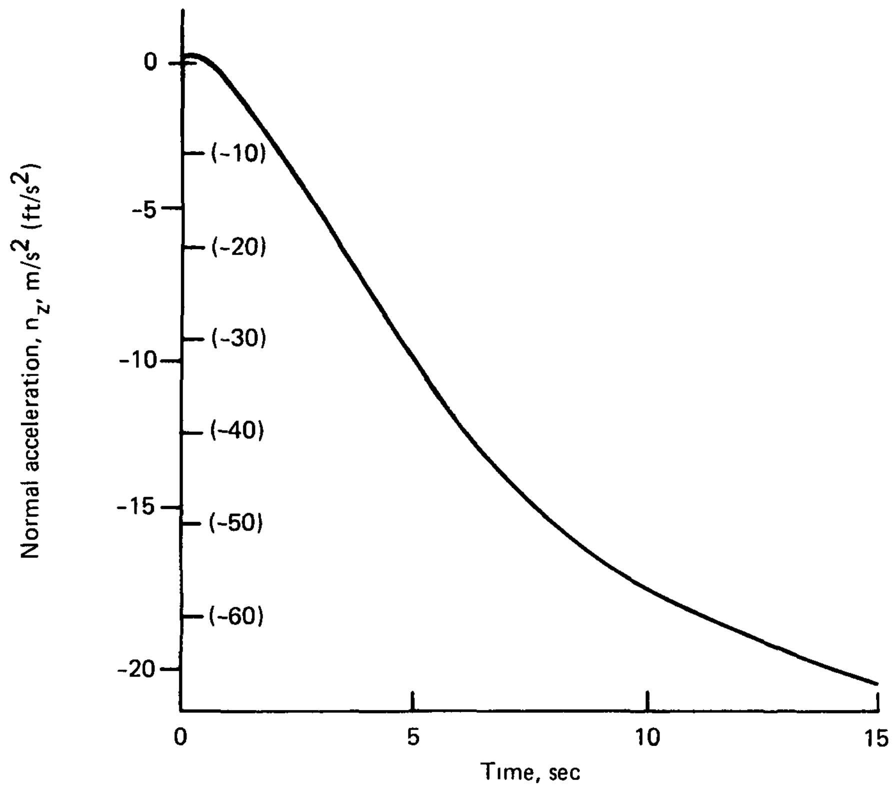


Figure 153. Normal Acceleration Response to Step Column Command, $\delta_C = 0.1$ rad, Flight Condition 97, Open Loop

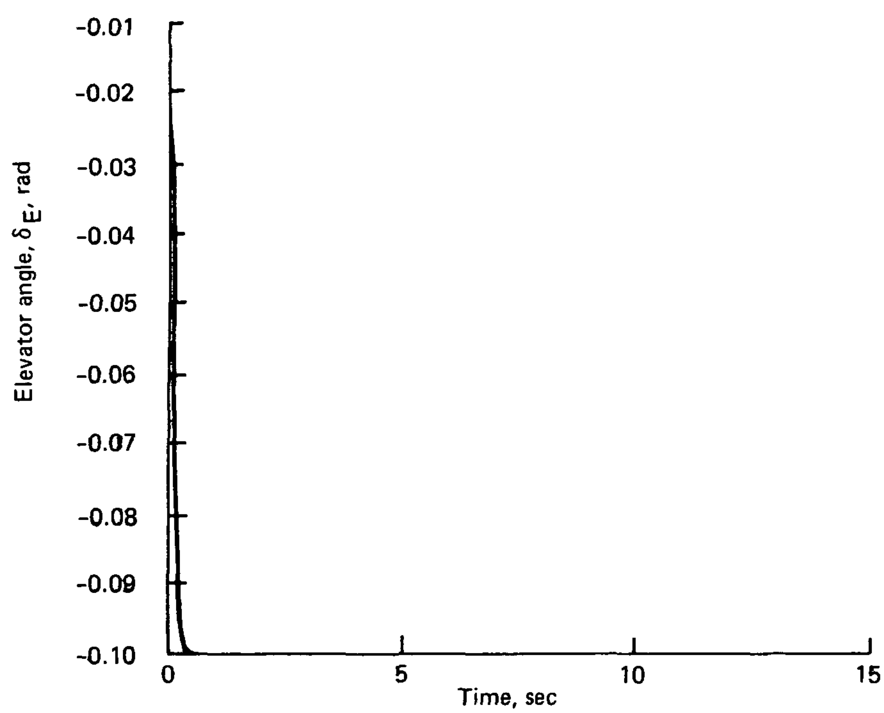


Figure 154. Elevator-Angle Response to Step Column Command, $\delta_C = 0.1$ rad, Flight Condition 97, Open Loop

13.3 CONTROL LAW SYNTHESIS

Linear quadratic regulator theory was used to produce a full-state feedback design. The methodology of linear quadratic design, discussed in Volume II, Appendix E, Section E.6.0, requires the selection of penalty weighting matrices Q and R in the cost function

$$J = \int_0^{\infty} \left[y_c{}^T Q y_c + u^T R u \right] dt \qquad (57)$$

where y_c = vector of criteria variables. Selecting as criteria variables states or linear combinations of states such as normal acceleration may not always produce the best system designs. In the design process for the PAS system, traditional criteria did not produce acceptable designs for the two flight conditions in question. For that reason, explicit model-following was used to obtain acceptable performance.

13.3.1 EXPLICIT MODEL-FOLLOWING

The technique of explicit model-following not only produces criteria for the quadratic cost function but also directly produces a feedforward control law.

The method consists of placing an ideal model of the plant to be controlled in the forward path of the control loop, as shown in Figure 155.

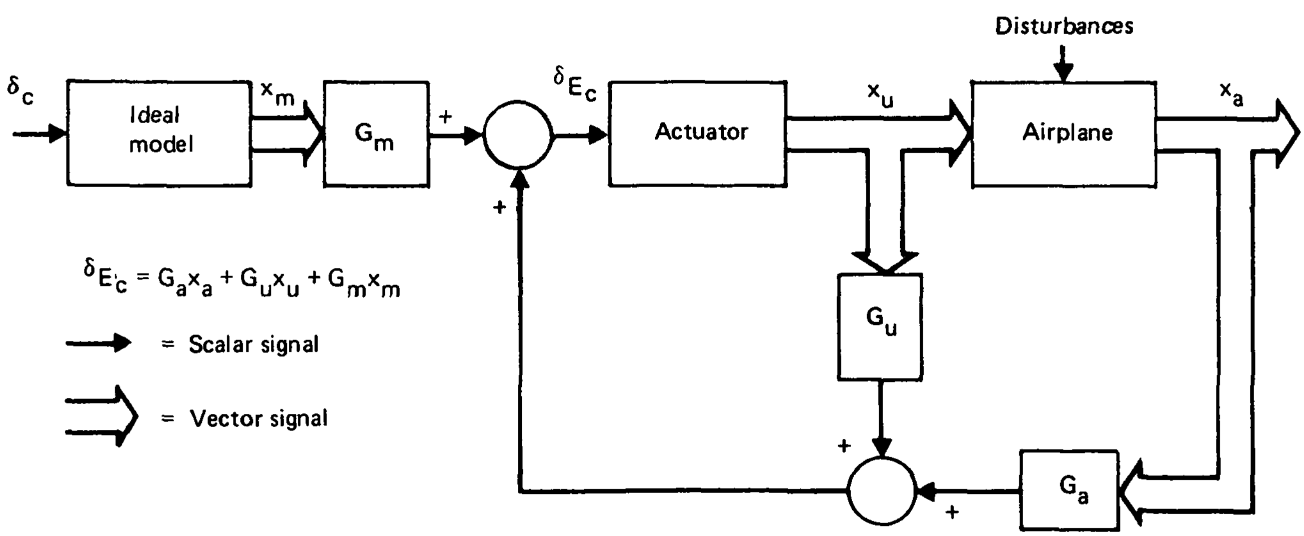


Figure 155. System Using Explicit Model-Following

The ideal model is a fourth-order model of the airplane with the desired response characteristics. Its state vector, x_m, is given by

$$x_m = \begin{bmatrix} u_m \\ \alpha_m \\ q_m \\ \theta_m \end{bmatrix} \tag{58}$$

and its state equation is

$$\dot{x}_m = A_m x_m + B_m \delta_c \tag{59}$$

The result of this formulation is that a control command at the column is fed to a filter that produces an ideal response, x_m. The rest of the system attempts to follow this response by minimizing the difference between x_a and x_m. In effect, the system is taking x_m as a command and is minimizing certain components of $(x_a - x_m)$ depending upon how the cost criteria are defined. The ideal model, shown in Figure 155 and defined by equation (59), acts as a feedforward filter in the control system.

Formulating the system in this way also reduces sensitivity of the airplane to turbulence, as the ideal model is not disturbed by the wind; the regulator action of the system thus acts to keep the components of x_a near their nominal values.

The criterion variable used in the quadratic cost function was α_{wc}, defined as

$$\alpha_{wc} = \alpha + \frac{w_g}{V} - \alpha_m \tag{60}$$

The sum $\alpha + w_g/V$ is the actual airplane angle of attack relative to the air, and α_m is the angle of attack of the ideal model. Linear quadratic regulator design produces a controller that will attempt to minimize α_{wc} during flight. The two state variables that primarily affect the short-period response are α and q. It was found that penalizing the difference between the actual α and α_m produced the best time responses.

The ideal model consists of the equations

$$\begin{bmatrix} \dot{u}_m \\ \dot{\alpha}_m \\ \dot{q}_m \\ \dot{\theta}_m \end{bmatrix} = \begin{bmatrix} a_{m11} & \cdots & a_{m14} \\ \cdot & \cdot & \\ \cdot & \cdot & \\ a_{m41} & \cdot & a_{m44} \end{bmatrix} \begin{bmatrix} u_m \\ \alpha_m \\ q_m \\ \theta_m \end{bmatrix} + \begin{bmatrix} b_{m1} \\ b_{m2} \\ b_{m3} \\ b_{m4} \end{bmatrix} \delta_c \tag{61}$$

or, in vector notation

$$\dot{x}_m = A_m x_m + B_m \delta_c \tag{62}$$

The input matrix B_m is taken to be the same as B_a in equation (25) to have the model respond to the command input in a manner analogous to the response of the airplane to the elevator angle. In other words, the lags produced by the elevator actuator are ignored in the model. This is no real limitation, as the elevator bandwidth is much higher than that of the airplane.

The design process concentrated on the subsystem

$$\begin{bmatrix} \dot{\alpha}_m \\ \dot{q}_m \end{bmatrix} = \begin{bmatrix} a_{m22} & a_{m23} \\ a_{m32} & a_{m33} \end{bmatrix} \begin{bmatrix} \alpha_m \\ q_m \end{bmatrix} + \begin{bmatrix} b_{m2} \\ b_{m3} \end{bmatrix} \delta_c \tag{63}$$

to optimize the short-period response.

The short-period approximation has a transfer function with a first-order numerator and a quadratic denominator. Because the pitch-rate response to column commands is of primary concern, the pitch-rate transfer function was developed from equation (63) and is

$$\frac{Q_m(s)}{\Delta_c(s)} \quad \frac{(s-a_{m22})b_{m3}}{s^2-(a_{m22}+a_{m33})\,s+(a_{m22}a_{m33}-a_{m23}a_{m32})} \tag{64}$$

where $Q_m(s)$ and $\Delta_c(s)$ are the Laplace transforms of $q_m(t)$ and $\delta_c(t)$, respectively. Comparing this transfer function with a general one of the form

$$G(s) = \frac{(\omega_n^2/a)(s+a)}{s^2 + 2\zeta\omega_n s + \omega_n^2} \tag{65}$$

it develops that

$$a_{m22} = -a \tag{66}$$

$$a_{m23} = 1 \tag{67}$$

$$a_{m32} = \omega_n{}^2 \left[\zeta^2(2\beta - \beta^2) - 1\right] \tag{68}$$

$$a_{m33} = (\beta - 2)\zeta\omega_n \tag{69}$$

where

$$\beta = a/\zeta\omega_n \tag{70}$$

The selection of $a_{m23} = 1$ does not arise from equation (65) but arises from flight dynamics considerations. In fact, a_{m23} was taken to be equal to the actual values of a_{23} for the two flight conditions, these values being very nearly 1. The purpose of this formulation is to select physically reasonable values of a_{m22}, a_{m23}, a_{m32}, and a_{m33} that produce a desirable pitch-rate response.

Values of β, ζ, and ω_n were selected from standard curves relating these parameters to percent overshoot of the response to a step input and damping and natural frequency requirements of the design requirements and objectives (DRO).

The remaining elements of A_m, relating to the phugoid response, were taken equal to those of A_a.

13.3.2 IDEAL MODEL FOR FLIGHT CONDITION 58

Flight condition 58 is a high-altitude, cruise-phase portion of the flight envelope. Under these conditions, it is considered desirable for the pitch-rate response to be underdamped and for the normal acceleration to be approximately critically damped. The pilot perceives the response of the airplane to column commands in terms of normal accelerations at this type of flight condition.

The ideal model was chosen to produce approximately a 50% overshoot for q_m. Parameters that produce this response are

$$\beta = 0.62 \tag{71}$$

$$\zeta = 0.7 \tag{72}$$

$$\omega_n = 0.89 \tag{73}$$

The values of ζ and ω_n fall within the DRO requirements

$$\zeta \geqslant 0.35 \tag{74}$$

$$0.8 \leqslant \omega_n \leqslant 5.0 \tag{75}$$

for this flight condition.

The ideal model matrix for flight condition 58 is

$$A_m = \begin{bmatrix} -5.04 \times 10^{-3} & 12.85 & 0.422 & -32.2 \\ -1.85 \times 10^{-4} & -0.388 & 0.996 & 0 \\ -1.57 \times 10^{-4} & -0.464 & -0.863 & 0 \\ 0 & 0 & 1.0 & 0 \end{bmatrix} \tag{76}$$

The responses of q_m and n_{zm} to a step command $\delta_c = 0.1$ rad are shown in Figures 156 and 157.

13.3.3 DESIGN FOR FLIGHT CONDITION 58

A large portion of the design process consisted of the analysis described in Subsection 13.3.2. The remainder of the design process consisted of selecting the cost weighting matrices Q and R to be used in the cost function given in equation (57).

The value for R was arbitrarily taken to be 1.0, and $Q(\alpha_{wc})$ was varied to produce different feedback designs. The criteria used to judge the quality of the designs were:

- Closed-loop root locations
- Feedback gain magnitudes
- Response to step column command
- Intensity of airplane response to transverse random turbulence

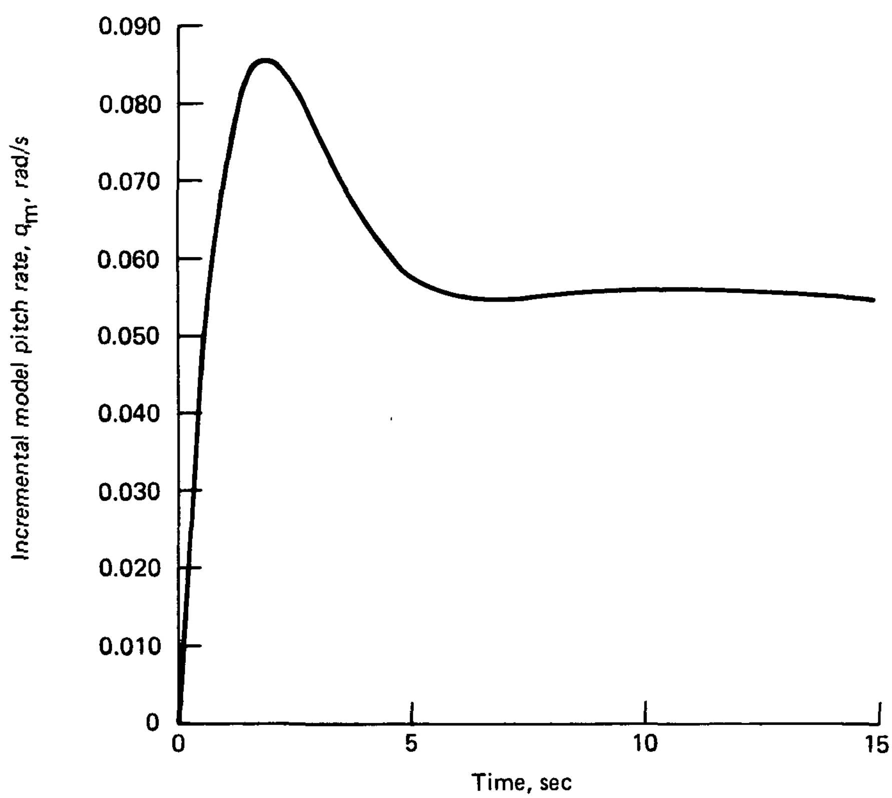


Figure 156. Pitch-Rate Response of Ideal Model, $\delta_c = 0.1$ rad, Flight Condition 58

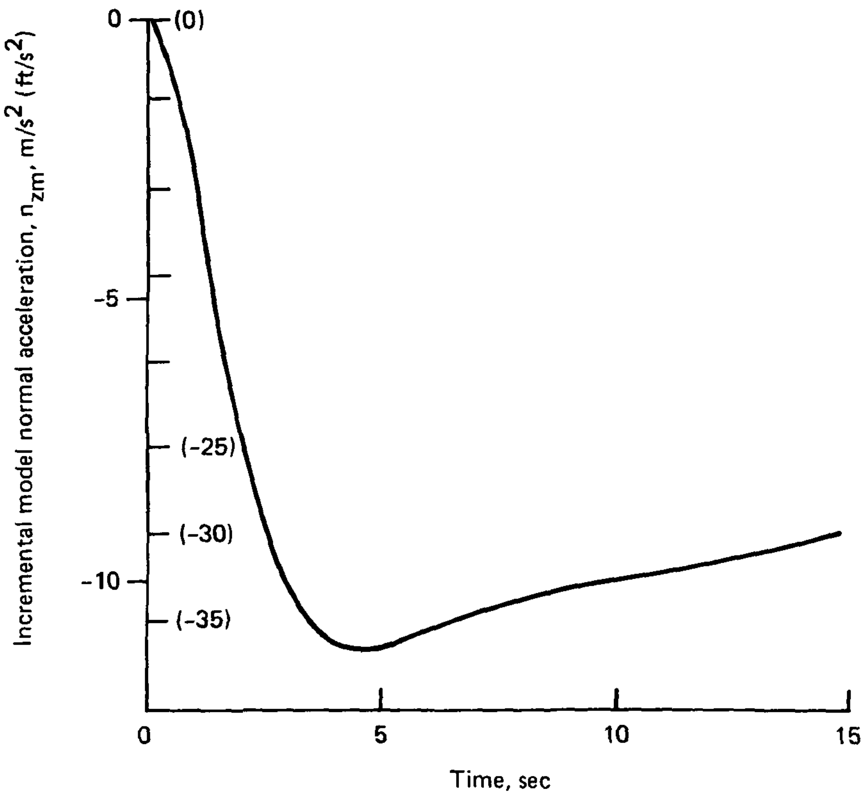


Figure 157. Normal Acceleration Response of Ideal Model, δ_c = 0.1 rad, Flight Condition 58

Figure 158 shows a root locus of the closed-loop phugoid roots, and a locus of the closed-loop short-period roots is given in Figure 159.

Gain matrix elements for different values of $Q(\alpha_{wc})$ are shown in Table 67. Table 68 shows the responses of system variables to 3.28-m/s (1.0-ft/s) root-mean-square transverse random turbulence for the different values of $Q(\alpha_{wc})$.

The design selected for flight condition 58 is that for which $Q(\alpha_{wc}) = 5$. The responses of pitch rate and normal acceleration to a step column command of 0.1 rad are shown in Figures 160 and 161, respectively. The elevator response is shown in Figure 162. It can be seen that the shapes of the pitch-rate and acceleration responses are almost identical to those of the ideal model shown in Figures 156 and 157.

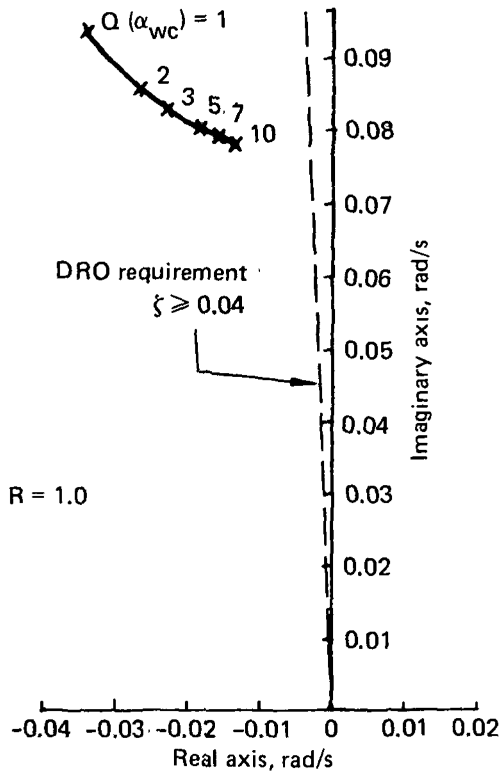


Figure 158. Phugoid Roots as a Function of Q (α_{wc}), Flight Condition 58

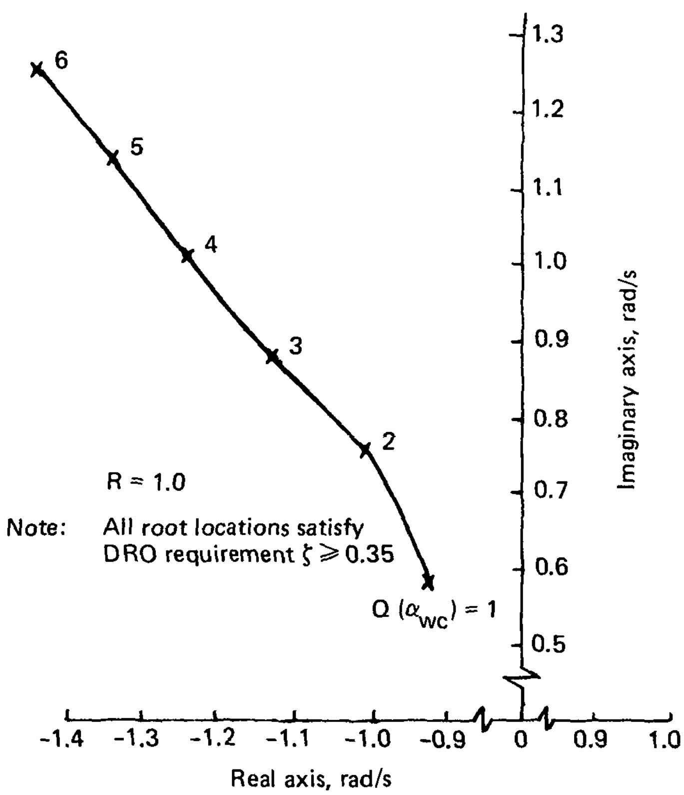


Figure 159. Short-Period Roots as a Function of Q (α_{wc}), Flight Condition 58

Table 67. Gain Matrix Elements as a Function of Q (α_{wc}), Flight Condition 58

Q (α_{wc})	G (u), rad/m/s (rad/ft/s)	G (α), rad/rad	G (q), rad/rad	G (θ), rad/rad	G (δ_E'), rad/rad	G (δ_E), rad/rad	G (δ_E), rad/rad	G (x_{w2}), rad/m/s^2 (rad/ft/s^2)	G (w_g), rad/m/s (rad/ft/s)	G (u_m), rad/m/s (rad/ft/s)	G (α_m), (rad/rad)	G (q_m), rad/rad/s	G (θ_m), rad/rad
1	$-3\ 378 \times 10^{-3}$ (-1.03×10^{-3})	0.829	1 122	0.171	-3.15×10^{-2}	$-6\ 39 \times 10^{-2}$	-1.722×10^{-3} ($-5\ 25 \times 10^{-4}$)	2.342×10^{-3} (7.14×10^{-4})	7.71×10^{-4} (2.35×10^{-4})	2.22×10^{-4} (6.77×10^{-5})	-0 322	-0.364	$-4\ 75 \times 10^{-2}$
2	$-3\ 35 \times 10^{-3}$ (-1.02×10^{-3})	1 126	1.327	0.180	$-3\ 7 \times 10^{-2}$	$-7\ 53 \times 10^{-2}$	-1.89×10^{-3} ($-5\ 75 \times 10^{-4}$)	3.2×10^{-3} (9.77×10^{-4})	$1\ 24 \times 10^{-3}$ ($3\ 78 \times 10^{-4}$)	4.1×10^{-4} (1.25×10^{-4})	-0.561	-0.573	-8.0×10^{-2}
3	-3.35×10^{-3} ($-1\ 02 \times 10^{-3}$)	1 37	1 47	0 183	$-4\ 09 \times 10^{-2}$	-8.35×10^{-2}	$-2\ 00 \times 10^{-4}$ (-6.1×10^{-4})	$3\ 84 \times 10^{-3}$ ($1\ 17 \times 10^{-3}$)	$1\ 7 \times 10^{-3}$ (5.12×10^{-4})	$5\ 64 \times 10^{-4}$ (1.72×10^{-4})	-0.767	-0.727	-0.101
5	$-3\ 38 \times 10^{-3}$ (1.03×10^{-3})	1.78	1 69	0.185	-4.68×10^{-2}	-9.57×10^{-2}	-2.17×10^{-3} ($-6\ 63 \times 10^{-4}$)	$4\ 82 \times 10^{-3}$ ($1\ 47 \times 10^{-3}$)	$2\ 5 \times 10^{-3}$ ($7\ 6 \times 10^{-4}$)	8.07×10^{-4} ($2\ 46 \times 10^{-4}$)	-1.12	-0.959	-0.127
7	-3.44×10^{-3} ($-1\ 05 \times 10^{-3}$)	2.12	1.86	0.184	-5.11×10^{-2}	-0.1049	-2.3×10^{-3} ($-0\ 02 \times 10^{-4}$)	$5\ 61 \times 10^{-3}$ ($1\ 71 \times 10^{-3}$)	$3\ 25 \times 10^{-3}$ ($9\ 9 \times 10^{-4}$)	9.91×10^{-4} (3.02×10^{-4})	-1.43	-1.14	-0.141
10	$-3\ 51 \times 10^{-3}$ (-1.07×10^{-3})	2.55	2.06	0 181	$-5\ 63 \times 10^{-2}$	-0.116	$-2\ 45 \times 10^{-3}$ ($-7\ 48 \times 10^{-4}$)	$6\ 53 \times 10^{-3}$ (1.99×10^{-3})	4.3×10^{-3} ($1\ 31 \times 10^{-3}$)	$1\ 2 \times 10^{-3}$ (3.67×10^{-4})	-1.83	-1 34	-0.152

Table 68. Root-Mean-Square Responses to Transverse Random Turbulence of 3.28-m/s (1.0-ft/s) RMS, Flight Condition 58

$Q'(\alpha_{wc})$	u, m/s (ft/s)	α, rad	q, rad/s	θ, rad	δ_E, rad	$\dot{\delta}_E$, rad/s
1	1.3×10^{-1} (4.26×10^{-1})	5.57×10^{-4}	2.56×10^{-4}	1.26×10^{-3}	3.62×10^{-4}	1.66×10^{-3}
2	1.51×10^{-1} (4.95×10^{-1})	5.89×10^{-4}	2.59×10^{-4}	1.34×10^{-3}	4.69×10^{-4}	2.44×10^{-3}
3	1.65×10^{-1} (5.42×10^{-1})	6.15×10^{-4}	2.80×10^{-4}	1.42×10^{-3}	5.62×10^{-4}	3.11×10^{-3}
5	1.84×10^{-1} (6.05×10^{-1})	6.54×10^{-4}	3.29×10^{-4}	1.54×10^{-3}	$7\ 20 \times 10^{-4}$	4.29×10^{-3}
7	1.97×10^{-1} (6.47×10^{-1})	6.83×10^{-4}	3.75×10^{-4}	1.62×10^{-3}	8.54×10^{-4}	5.32×10^{-3}
10	2.11×10^{-1} (6.92×10^{-1})	7.16×10^{-4}	4.36×10^{-4}	1.72×10^{-3}	1.03×10^{-3}	6.69×10^{-3}

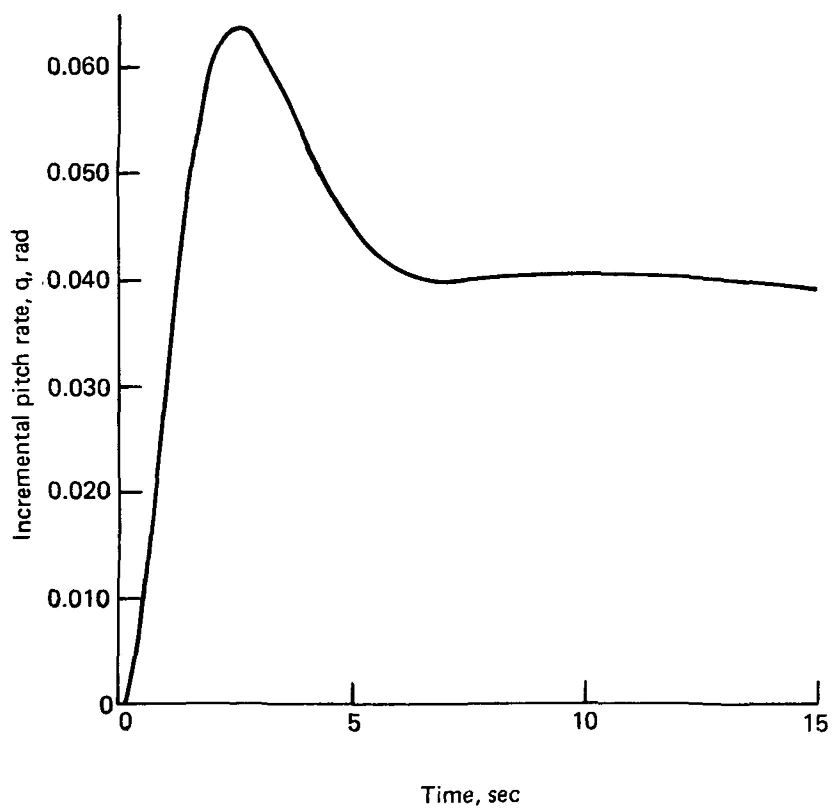


Figure 160. Pitch-Rate Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 58, Closed Loop

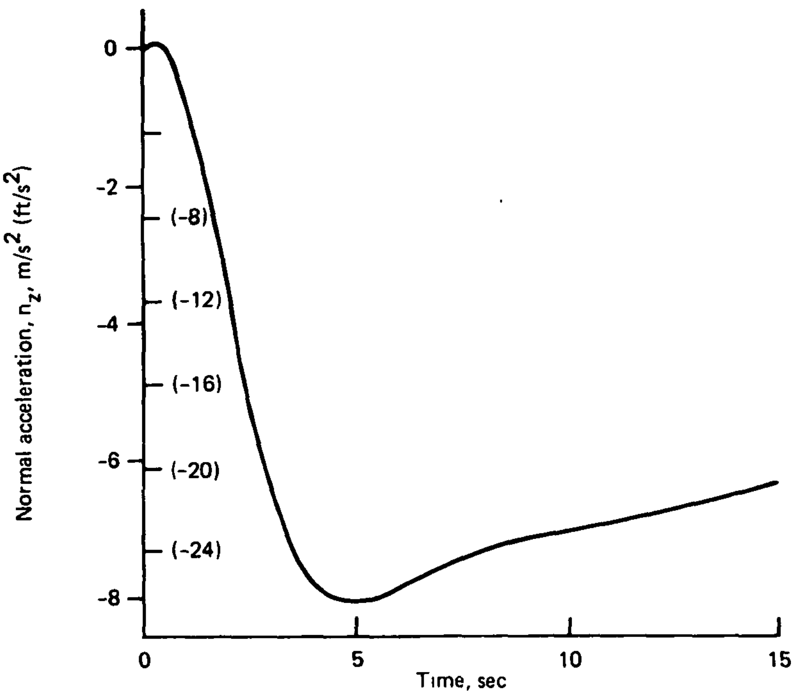


Figure 161. Normal Acceleration Response to Step Column Command, δ_c = 0.1 rad, Flight Condition 58, Closed Loop

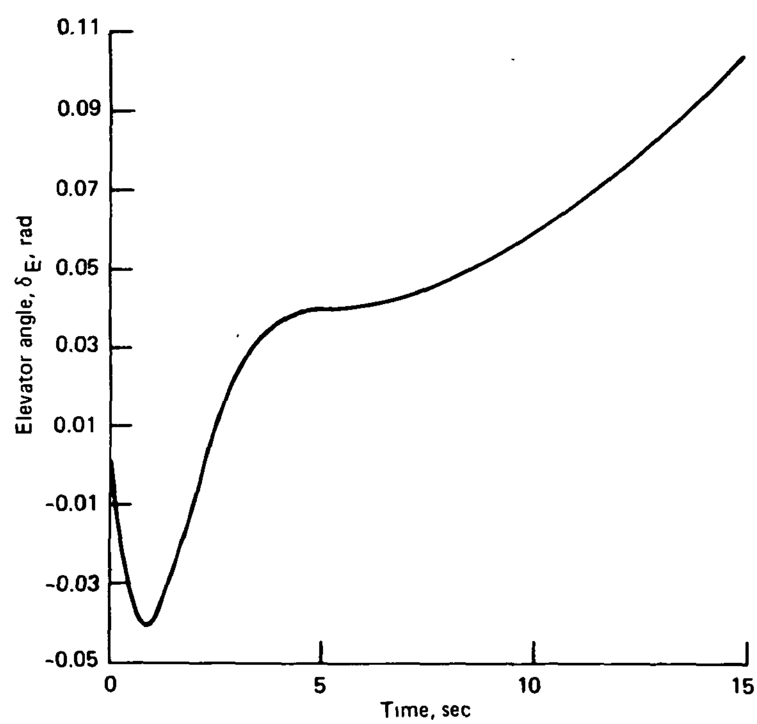


Figure 162. Elevator-Angle Response to Step Column Command, δ_c = 0.1 rad, Flight Condition 58, Closed Loop

13.3.4 IDEAL MODEL FOR FLIGHT CONDITION 97

Flight condition 97 is a flaps-down approach portion of the flight envelope. Under these conditions, it is considered desirable for the pitch-rate response to be much less underdamped than for a cruise condition. The pilot perceives the response of the airplane to column commands in terms of pitch rate at this type of flight condition.

The ideal model was chosen to produce approximately a 10% overshoot for q_m. Parameters that produce this response are

$$\beta = 0.5 \tag{77}$$

$$\zeta = 1.0 \tag{78}$$

$$\omega_n = 0.768 \tag{79}$$

This low value of ω_n was required to produce the desired response. Even though this value for the model is below that required for the airplane, it produces acceptable short-period roots for the closed-loop airplane.

The ideal model matrix for flight condition 97 is

$$A_m = \begin{bmatrix} -2.16 \times 10^{-2} & 20.87 & -2.3 \times 10^{-2} & -32.2 \\ -4.76 \times 10^{-4} & -0.384 & 0.994 & 0 \\ -7.13 \times 10^{-5} & -0.148 & -1.152 & 0 \\ 0 & 1.0 & 0 & 0 \end{bmatrix} \tag{80}$$

The responses of q_m and n_{zm} to a step command $\delta_c = 0.1$ rad are shown in Figures 163 and 164.

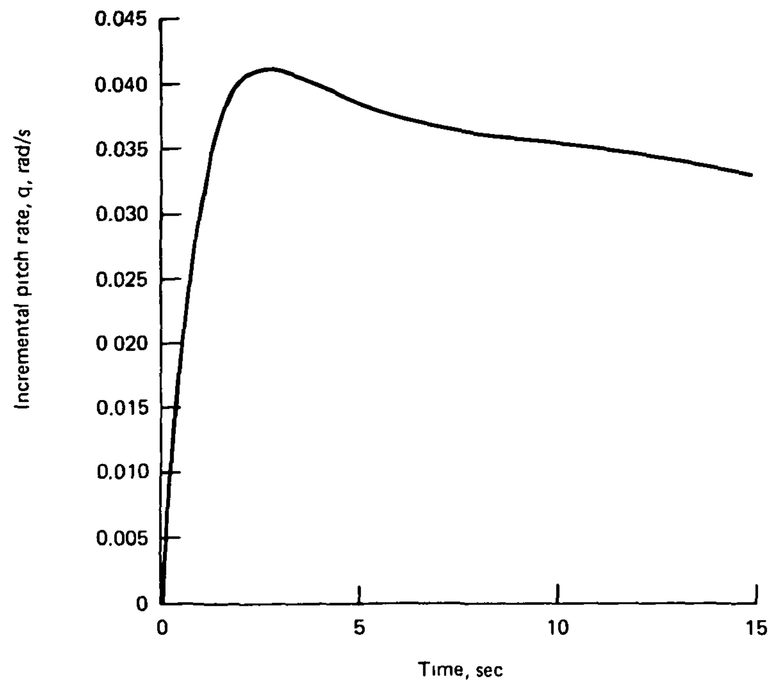


Figure 163. Pitch-Rate Response of Ideal Model, $\delta_c = 0.1$ rad, Flight Condition 97

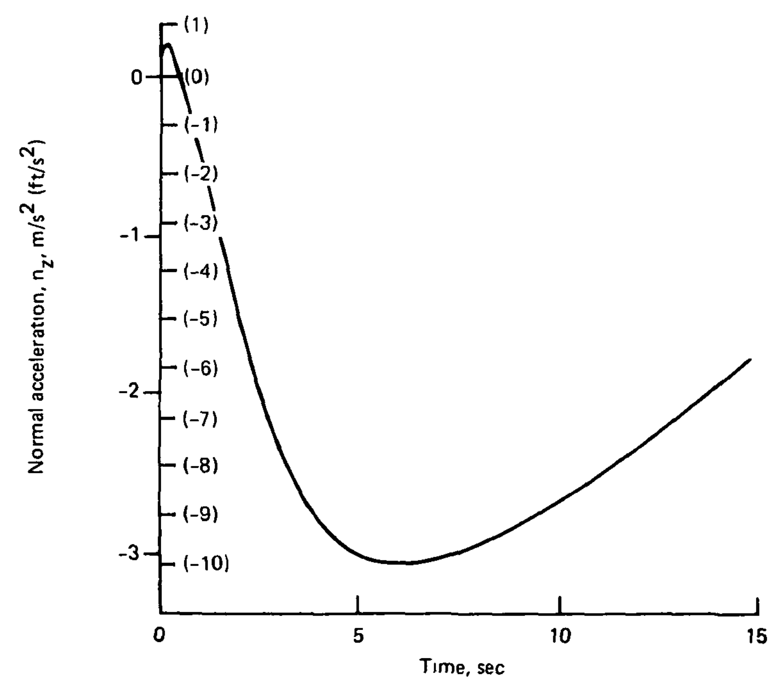


Figure 164. Normal Acceleration Response of Ideal Model, $\delta_c = 0.1$ rad, Flight Condition 97

13.3.5 DESIGN FOR FLIGHT CONDITION 97

As for flight condition 58, the value of R was taken to be 1.0, and $Q(\alpha_{wc})$ was varied to produce different feedback designs. A root locus of the closed-loop phugoid roots is given in Figure 165, and a locus of the closed-loop short-period roots is given in Figure 166.

Gain matrix elements for different values of $Q(\alpha_{wc})$ are shown in Table 69. Table 70 shows the responses of system variables to 3.28-m/s (1.0-ft/s) root-mean-square transverse random turbulence for the different values of $Q(\alpha_{wc})$.

The design selected for flight condition 97 is that for which $Q(\alpha_{wc}) = 10$. The responses of pitch rate and normal acceleration to a step column command of 0.1 rad are shown in Figures 167 and 168, respectively. The elevator response is shown in Figure 169.

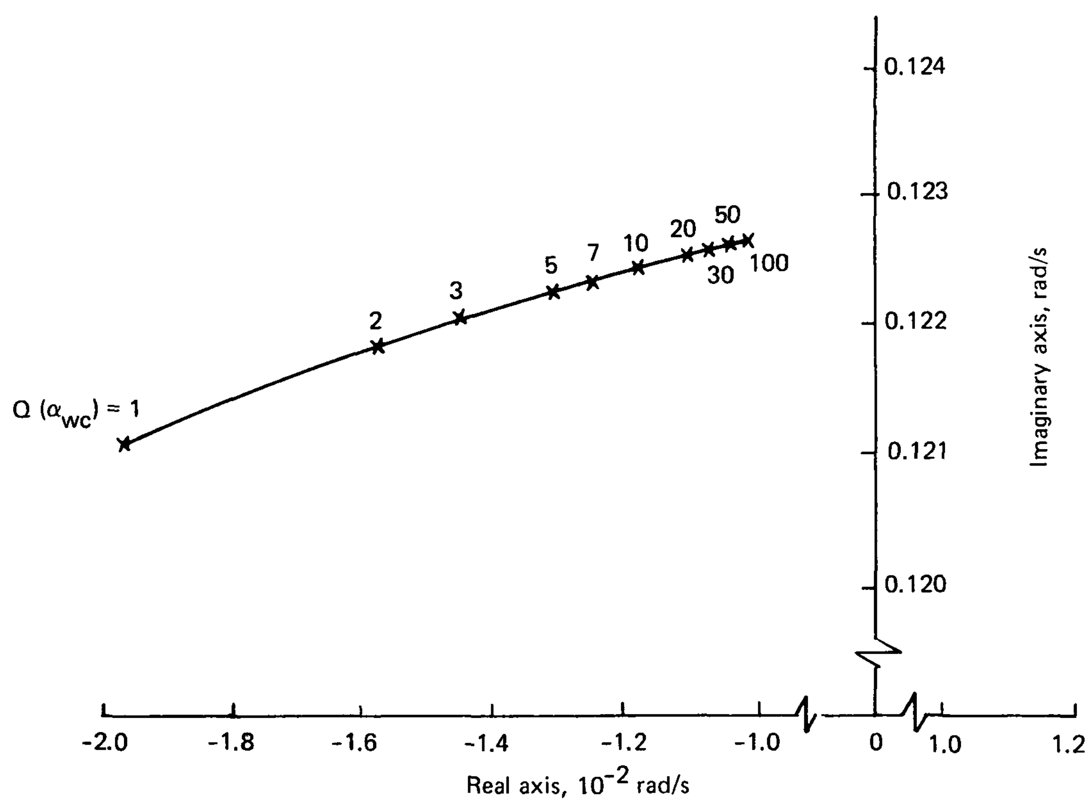


Figure 165. Phugoid Roots as a Function of Q (α_{wc}), Flight Condition 97

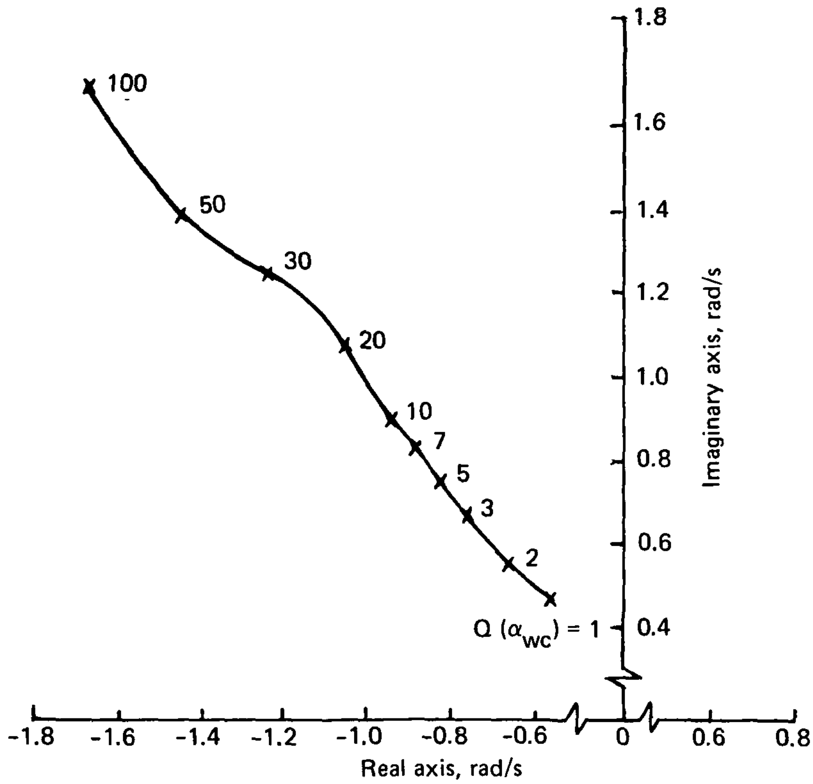


Figure 166. Short-Period Roots as a Function of Q (α_{wc}), Flight Condition 97

Table 69. Gain Matrix as a Function of Q (α_{wc}), Flight Condition 97

Q (α_{wc})	G (u), rad/rad	G (α), rad/rad	G (q), rad/rad/s	G (θ), rad/rad	G (δ_E'), rad/rad	G (δ_E), rad/rad	G (u_g), rad/m/s (rad/ft/s)	G (x_{w2}), rad/m/s^2 (rad/ft/s^2)	G (w_g) rad/m/s (rad/ft/s)	G (u_m), rad/m/s (rad/ft/s)	G (α_m), rad/rad	G (q_m), rad/rad/s	G (θ_m), rad/rad
1	-2.50×10^{-3} (-7.61×10^{-4})	0.489	1.05	8.04×10^{-2}	-1.47×10^{-2}	-2.95×10^{-2}	-1.42×10^{-3} (-4.33×10^{-4})	4.23×10^{-3} (1.29×10^{-3})	1.69×10^{-3} (5.51×10^{-5})	1.87×10^{-3} (5.71×10^{-4})	-0.368	-0.375	-6.83×10^{-2}
2	-3.07×10^{-3} (-9.35×10^{-4})	0.748	1.37	7.18×10^{-2}	-1.90×10^{-2}	-3.84×10^{-2}	-1.96×10^{-3} (-5.98×10^{-4})	6.53×10^{-3} (1.99×10^{-3})	8.36×10^{-4} (2.55×10^{-4})	2.50×10^{-3} (7.62×10^{-4})	-0.627	-0.562	-6.3×10^{-2}
3	-3.44×10^{-3} (-1.05×10^{-3})	1.02	1.59	6.65×10^{-2}	-2.21×10^{-2}	-4.47×10^{-2}	-2.34×10^{-3} (-7.12×10^{-4})	8.20×10^{-3} (2.50×10^{-3})	1.54×10^{-3} (4.71×10^{-4})	2.92×10^{-3} (8.90×10^{-4})	-0.841	-0.705	-6.01×10^{-2}
5	-4.00×10^{-3} (-1.22×10^{-3})	1.41	1.91	6.02×10^{-2}	-2.65×10^{-2}	-5.37×10^{-2}	-2.87×10^{-2} (-8.76×10^{-4})	1.07×10^{-2} (3.27×10^{-3})	2.94×10^{-3} (9.86×10^{-4})	3.51×10^{-3} (1.07×10^{-3})	-1.20	-0.929	-5.55×10^{-2}
7	-4.4×10^{-3} (-1.34×10^{-3})	1.73	2.15	5.63×10^{-2}	-2.98×10^{-2}	-6.04×10^{-2}	-3.27×10^{-3} (-9.97×10^{-4})	1.27×10^{-2} (3.87×10^{-3})	4.26×10^{-3} (1.30×10^{-3})	3.94×10^{-3} (1.20×10^{-3})	-1.50	-1.11	-5.25×10^{-2}
10	-4.85×10^{-3} (-1.48×10^{-3})	2.15	2.44	5.26×10^{-2}	-3.36×10^{-2}	-6.84×10^{-2}	-3.71×10^{-3} (-1.13×10^{-3})	1.50×10^{-2} (4.57×10^{-3})	6.13×10^{-3} (1.87×10^{-3})	4.43×10^{-3} (1.35×10^{-3})	-1.89	-1.32	-4.96×10^{-2}
20	-5.90×10^{-3} (-1.80×10^{-3})	3.23	3.08	4.65×10^{-2}	-4.23×10^{-2}	-8.63×10^{-2}	-4.76×10^{-3} (-1.45×10^{-3})	2.04×10^{-2} (6.21×10^{-3})	1.16×10^{-2} (3.53×10^{-3})	5.54×10^{-3} (1.69×10^{-3})	-2.93	-1.84	-4.47×10^{-2}
30	-6.63×10^{-3} (-2.02×10^{-3})	4.09	3.52	4.38×10^{-2}	-4.81×10^{-2}	-9.85×10^{-2}	-5.48×10^{-3} (-1.67×10^{-3})	2.41×10^{2} (7.34×10^{-3})	1.63×10^{-3} (4.97×10^{-3})	6.26×10^{-3} (1.91×10^{-3})	-3.75	-2.21	-4.24×10^{-2}
50	-7.62×10^{-3} (-2.33×10^{-3})	5.46	4.15	4.10×10^{-2}	-5.64×10^{-2}	-0.116	-6.46×10^{-3} (-1.97×10^{-3})	2.95×10^{-2} (8.99×10^{-3})	2.44×10^{-2} (7.43×10^{-3})	7.31×10^{-3} (2.23×10^{-3})	-5.09	-2.76	-4.00×10^{-2}
100	-9.24×10^{-3} (-2.82×10^{-3})	8.03	5.16	3.82×10^{-2}	-6.95×10^{-2}	-0.144	-8.07×10^{-3} (-2.46×10^{-3})	3.84×10^{-2} (1.17×10^{-2})	4.07×10^{-2} (1.24×10^{-2})	8.95×10^{-3} (2.73×10^{-3})	-7.60	-3.66	-3.77×10^{-2}

Table 70. Root-Mean-Square Responses to Transverse Random Turbulence of 3.28-m/s (1.0-ft/s) RMS, Flight Condition 97

Q (α_{wc})	u, m/s (ft/s)	α, rad	q, rad/s	θ, rad	δ_E, rad	$\dot{\delta}_E$, rad/s
1	1.44×10^{-1} (4.73×10^{-1})	1.76×10^{-6}	1.89×10^{-7}	6.99×10^{-6}	1.67×10^{-7}	8.42×10^{-7}
2	1.94×10^{-1} (6.35×10^{-1})	1.76×10^{-6}	2.72×10^{-7}	9.71×10^{-6}	3.40×10^{-7}	3.44×10^{-6}
3	2.18×10^{-1} (7.14×10^{-1})	1.83×10^{-6}	3.26×10^{-7}	1.11×10^{-5}	5.35×10^{-7}	7.39×10^{-6}
5	2.39×10^{-1} (7.83×10^{-1})	1.95×10^{-6}	4.05×10^{-7}	1.23×10^{-5}	9.79×10^{-7}	1.83×10^{-5}
7	2.47×10^{-1} (8.09×10^{-1})	2.05×10^{-6}	4.68×10^{-7}	1.29×10^{-5}	1.47×10^{-6}	3.22×10^{-5}
10	2.50×10^{-1} (8.19×10^{-1})	2.17×10^{-6}	5.48×10^{-7}	1.32×10^{-5}	2.27×10^{-6}	5.71×10^{-5}
20	2.44×10^{-1} (8.01×10^{-1})	2.46×10^{-6}	7.64×10^{-7}	1.32×10^{-5}	5.21×10^{-6}	1.64×10^{-4}
30	2.37×10^{-1} (7.76×10^{-1})	2.66×10^{-6}	9.42×10^{-7}	1.30×10^{-5}	8.30×10^{-6}	2.92×10^{-4}
50	2.24×10^{-1} (7.36×10^{-1})	2.92×10^{-6}	1.23×10^{-6}	1.27×10^{-5}	1.46×10^{-5}	5.91×10^{-4}
100	2.07×10^{-1} (6.79×10^{-1})	3.29×10^{-6}	1.78×10^{-6}	1.22×10^{-5}	3.01×10^{-5}	1.46×10^{-3}

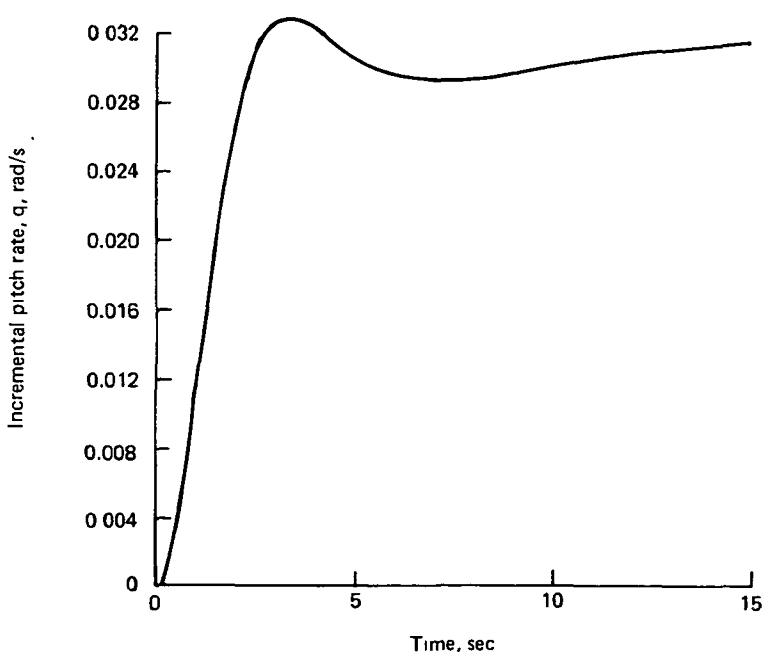


Figure 167. Pitch-Rate Response to Step Column Command, $\delta_C = 0.1$ rad, Flight Condition 97, Closed Loop

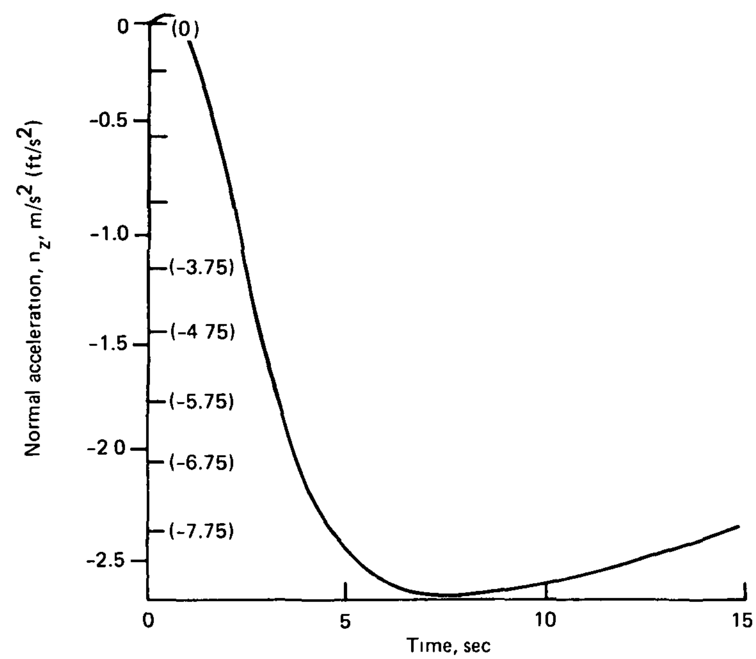


Figure 168. Normal Acceleration Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Closed Loop

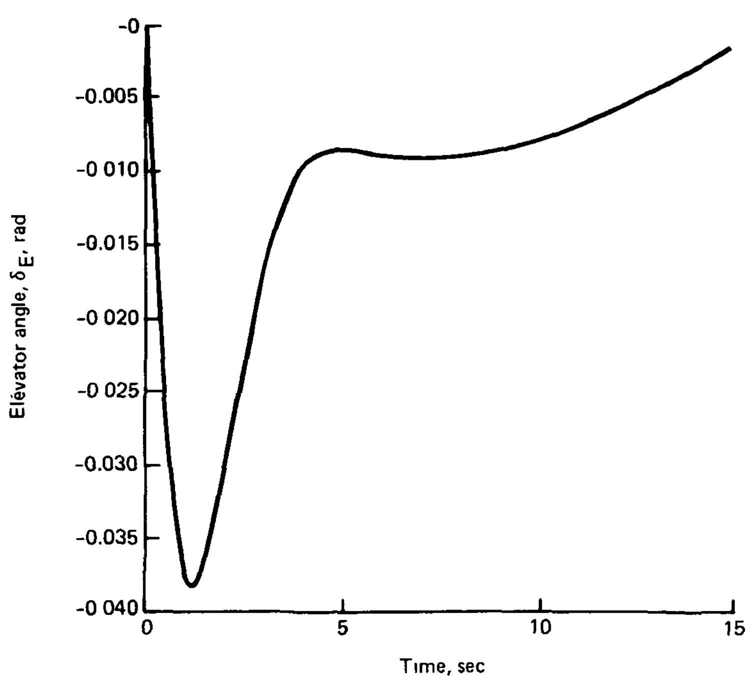


Figure 169. Elevator-Angle Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Closed Loop

13.4 COMPARISON WITH CURRENT TECHNOLOGY CONTROL LAWS

Comparison of the system designed using modern control theory and that designed using classical methods shows that both methods produce acceptable responses to column commands. The responses of the classical system for flight conditions 58 and 97, shown in Figures 170 and 171, are rapid and have no overshoot. Comparison of Figure 170 with Figure 160 indicates that the optimal system has a faster rise time and reaches steady state more rapidly for flight condition 58. The optimal system has about a 50% overshoot, which was designed to produce rapid normal acceleration response.

Comparison of Figure 171 with Figure 167 indicates that the two systems have approximately the same speed of response at flight condition 97; the optimal system has a small overshoot.

An advantage of the optimal control method is that it allows the type of response desired to be directly selected by specifying an ideal model having that response. The optimal control design process then directly produces the feedback gains required to cause the augmented airplane to behave similar to the ideal model. The method allows the designer to design for acceptable responses to command inputs as well as turbulence inputs.

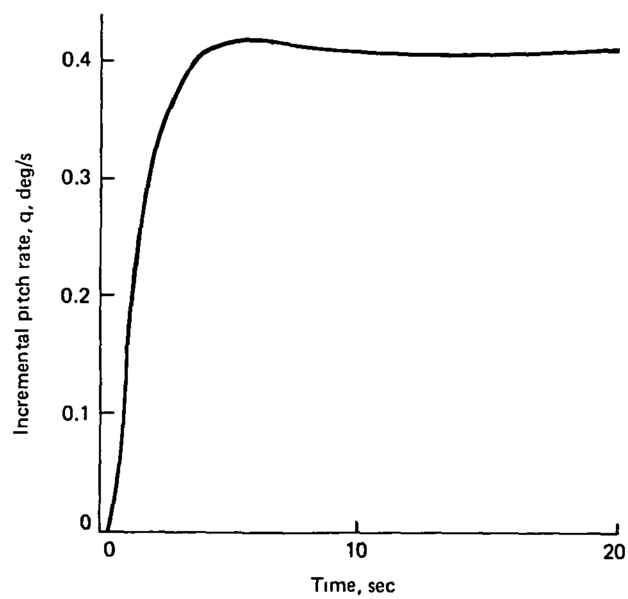


Figure 170. Pitch-Rate Response to Step Column Command, $\delta_c = 0.1$ rad, Flight Condition 97, Classical Feedback

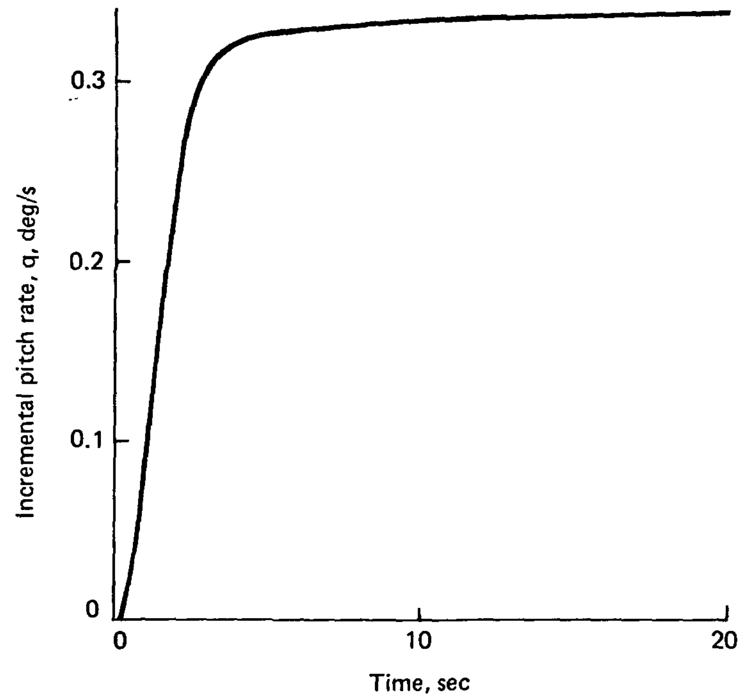


Figure 171. Pitch-Rate Response to Step Column Command, $\delta_C = 0.1$ rad, Flight Condition 58, Classical Feedback

14.0 1990 ACT SYSTEM STUDY SUMMARY

This section assesses the effect of advancing technology in the electronic and automatic control areas on the cost of ownership of an Active Controls Technology (ACT) airplane. More specifically, the effects of the technology advances associated with the implementation of an ACT system that embodies properties and characteristics expected to be available for a circa 1990 commercial airplane are evaluated. Additional information is contained in Volume II, Appendix G, "Alternative Implementation of ACT."

Figure 172 shows the study tasks that comprise the advanced technology ACT system definition. A familiarization phase was followed by a flight control system element study. This study examined the technology developments in various automatic flight control areas and projected the developmental status to the late 1980s. Based on these projections, a study of alternative systems was begun. Three alternative systems were defined and examined qualitatively for their advantages. Actuation studies were conducted separately to define actuators that would be used with the final system.

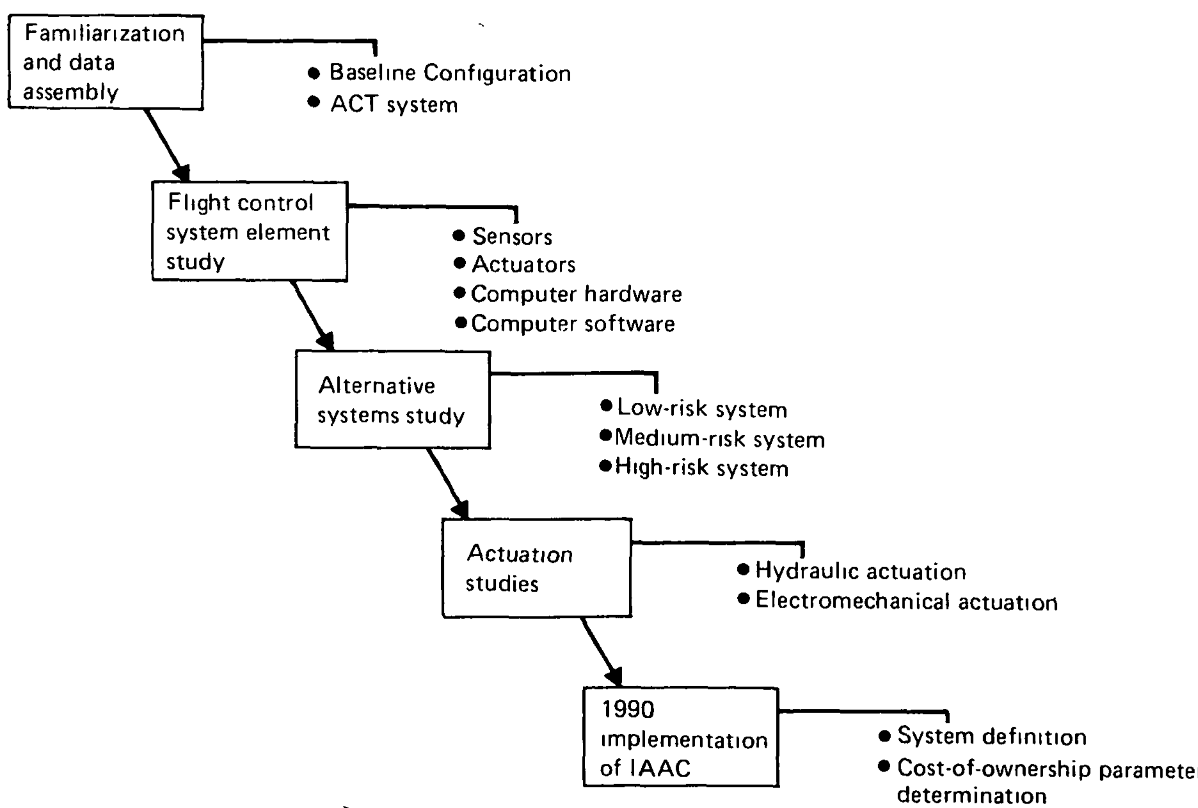


Figure 172. Advanced Technology ACT System Study Elements

Following the alternative systems study, a 1990 implementation of ACT was defined based on the previous work. Flight safety reliability was predicted, and cost-of-ownership parameters were estimated. These parameters were then entered into the Boeing cost model, and fuel savings (during the system payback period) and return on investment (ROI) were calculated. The advanced technology system provides an ROI to the airlines of 34.9% compared to a maximum ROI of 27.6% for current technology systems (see subsec 14.5.4).

14.1 FAMILIARIZATION AND DATA ASSEMBLY PHASE

A familiarization phase was required for the Honeywell engineers involved in the project so that they could learn how past work applied to the Advanced Technology ACT Control System Definition. ACT functions were defined during this phase, including input signal requirements, control laws, and surfaces to be controlled. The control-surface hinge moments, maximum deflections, and rates were defined as were the dynamic response requirements for the actuators. Slats, flaps, and spoiler control and actuation were separated from the advanced technology ACT study because no change is required of these characteristics from the Baseline definition. Fly-by-wire (FBW) control was defined as an option for the 1990 ACT system, but any associated effect on cockpit controls, including the angle-of-attack limiting function actuator, was not addressed to avoid dilution of the basic 1990 ACT study effort. The aircraft electric and hydraulic systems, as modified for ACT (subsecs 8.5 and 8.6), were prescribed as applicable to the 1990 ACT system.

14.2 FLIGHT CONTROL SYSTEM ELEMENT STUDY

Applicable developments were surveyed, and forecasts were prepared to better identify elements appropriate to a 1990 operational system. Sensors, computers, servoactuators, and software design and validation were the areas surveyed.

14.2.1 SENSOR SURVEY

The sensor survey included air data, angular rate, and acceleration sensors. Based on the sensor requirements of the ACT system, it was concluded that air data information should be obtained from the aircraft air data system and that three digital air data computers (DADC) should be required. Data obtained in this manner are adequate in the area of performance. Triplex sources of air data information provide adequate functional reliability. This is the most economical manner of acquiring the required air data because the DADCs also are required for other aircraft avionic subsystems.

When angular rate sensors were surveyed, many present and evolving concepts were found to be satisfactory in performance and reliability. The ring laser gyro is recommended for the ACT application because there is no mechanical wear associated with the gyro and because this type of gyro will be required aboard the aircraft for the triplex inertial reference system (IRS) function.

The IRS accelerometer output signals are recommended for the required normal acceleration signal. The piezo-resistive strain gage is recommended for wing-mounted accelerometers required for flutter-mode control (FMC) and wing-load alleviation (WLA). These accelerometers are relatively low cost and have a high dynamic response. Filters can be applied to limit the bandwidth of the signals.

The roll attitude signal required for lateral/directional-augmented stability (LAS) is satisfactorily obtained from the triplex IRSs.

14.2.2 AIRBORNE COMPUTER TECHNOLOGY

A dramatic increase of capability will be experienced in computing over the next several years. This will be due mainly to very-large-scale and very-high-speed integrated circuit developments.

Ultrareliable central processing components will be available in the late 1980s to meet the needs of digital calculations for flight controls. This reliability will be the result of increased integration. Chip counts and connections between chips will be significantly reduced. Fault-tolerant and self-testing mechanisms will be included in the chips. The size, weight, and power requirements of the systems will no longer be significant considerations, and costs will be reduced to relatively unimportant levels. Standardization of instruction sets will permit accumulation of software support systems and development tools so that software development for flight control calculations will be economical and certifiably reliable. This will be enhanced by the trend to use hardware rather than complicated software for fault tolerance and self-checking.

The input/output (I/O) functions, already the main part of the system design problem, will become even more dominant as the central computing structures become routine. Multiplexed busing of sensor data and actuator commands will present the critical technical problems. Fault-tolerant components, which will provide more capability than is necessary for flight controls, will be used in these systems because of large production runs and experience in similar systems.

14.2.3 ACTUATORS

Actuation concepts were reviewed and compared to requirements, which resulted in the conclusion that conventional hydraulic actuation concepts should be applied for the 1990 ACT system. The alternatives included various higher efficiency techniques such as servopumps, which were questionable in performance for the applications of the ACT configuration.

Electromechanical actuators (EMA) were identified as the type to be used for flaperon control. They use electric input power and thus minimize the difficulty of transmitting power across the flap-wing interface. EMAs were not considered for other applications because so much effort would have to be expended to properly consider them due to their overall effect on the aircraft electric and hydraulic power systems. Although it is beyond the scope of this study, the impact that EMAs have in these specific areas is being studied in other programs.

14.2.4 SOFTWARE DESIGN

As a result of examining software design developments, it is projected that methodologies will be available by 1990 so that software can be designed and validated for flight control applications and can be certified to be error free. Flight controls have a regular structure that can be the basis for specification, design, coding, verification, and validation. Many variants of a systematic methodology can be shaped about this structure. It will be possible to develop flight control software that is certifiably free of technical software errors and performs according to precisely defined specifications.

14.3 ALTERNATIVE SYSTEM CONCEPTS

14.3.1 INTRODUCTION

Three alternative advanced technology ACT system configurations were developed. They differ in the assumed technology risk, but the ACT requirements and functions are identical for all three configurations. The alternatives are characterized as having low, medium, and high risk associated with system implementation by 1990. Three alternatives of varying risk were formulated so that a single alternative could be selected with awareness of the likelihood of the implementation being advanced in concept and yet realizable by 1990.

The three systems have different approaches to the computational elements of the systems and the data busing concepts. The sensor sets and actuator approaches are nearly the same for each with the exception of an additional pitch-rate sensor requirement for the low-risk system.

The low-risk system follows the developments of the 1970s in that redundant computers are run in a macrosynchronized manner. Data are exchanged between the redundant computers via dedicated serial buses. Computations are bit-for-bit identical between computers because of the redundancy management approach used and the data exchange qualities. Sensor and servo interfaces are primarily analog. Only moderate technology growth is assumed, and minimal use of integrated circuit advances is required.

The medium-risk system uses multiple microprocessors in each computing channel. Computer channels operate asynchronously. Serial digital data busing is used extensively for both sensor and actuator interfaces. Objectives were to create an increased number of success paths for flight safety and dispatch reliability and to reduce software complexity and preparation costs.

The high-risk system capitalizes on recent and projected advances in self-testing digital circuitry and in integrated circuit technology. The computational segment of the system builds on, and extends, the concepts used in the fault-tolerant multiple processor (FTMP) and software-implemented fault tolerance (SIFT) architectures. It consists of four self-checking computer modules composed of multiple microprocessors. Each module is 100%

self-checking; it does not require cross-channel comparison of the computers. The computers run asynchronously, and the system relies on ultrareliable self-checking bus adapters and controllers.

A derivative of the medium-risk system was selected for further evaluation and cost-of-ownership analysis. The system is discussed in Subsection 14.5. Additional information on the ACT system configurations is in Volume II, Appendix G.

14.3.2 DESCRIPTION AND COMPARISON OF ALTERNATIVE SYSTEMS

This subsection describes the three alternative forms of the advanced technology ACT system and presents comparative data in tabular form.

14.3.2.1 Low-Risk System

The low-risk system is based on the principle of digital FBW control developed during the 1970s. Moderate extension of large-scale integrated circuit technology is projected.

The low-risk system is an integrated one in that the crucial pitch-augmented stability (PAS) functions are accomplished by the same computers as the critical functions. This is also true of the medium- and high-risk systems. Because the pitch axis is already dependent upon crucial electronics due to the short-period PAS function, pitch axis control can be made FBW without additional risk.

Figure 173 shows the essentials of the low-risk system architecture. The quadruply redundant computers are state-of-the-art uniprocessors. All processing is run in the ACT computers. Sensor data are input to the ACT computer in digital and analog form. The IRS and DADC inputs are via a serial digital Aeronautical Radio Incorporated (ARINC) 429 bus. All other sensor inputs and the pitch-rate gyro signals from the IRS are hardwired analog. Outputs to both the triple and quadruple servos and actuators are also hardwired analog. Servo-loop electronics are contained in the computer.

The sensor set consists of rates and accelerations from the strapdown IRS, air data from the DADC, accelerometers mounted in the wings for gust-load alleviation (GLA) and FMC, the pilot input transducer, and an additional pitch-rate sensor. All sensors are tripled with the exception of pilot inputs and pitch rate, which are quadrupled. The quadruple pitch-rate configuration is achieved by using three IRS gyros and an additional, separate pitch-rate sensor.

The computers are frame synchronized. Each cycle of computation is initiated simultaneously by all computers as a result of the "halt-release" implementation used. Data are exchanged between computers via a dedicated serial data bus. Sensor data are exchanged and monitored. Signals are related in such a way that control law computations are identical in each computer. Each computer has an analog reversion mode that allows continued safe flight after loss of all digital computing capability. Failure management is accomplished with comparison monitoring. The required reliability for the crucial PAS system is achieved through the additional (fourth) pitch-rate sensor.

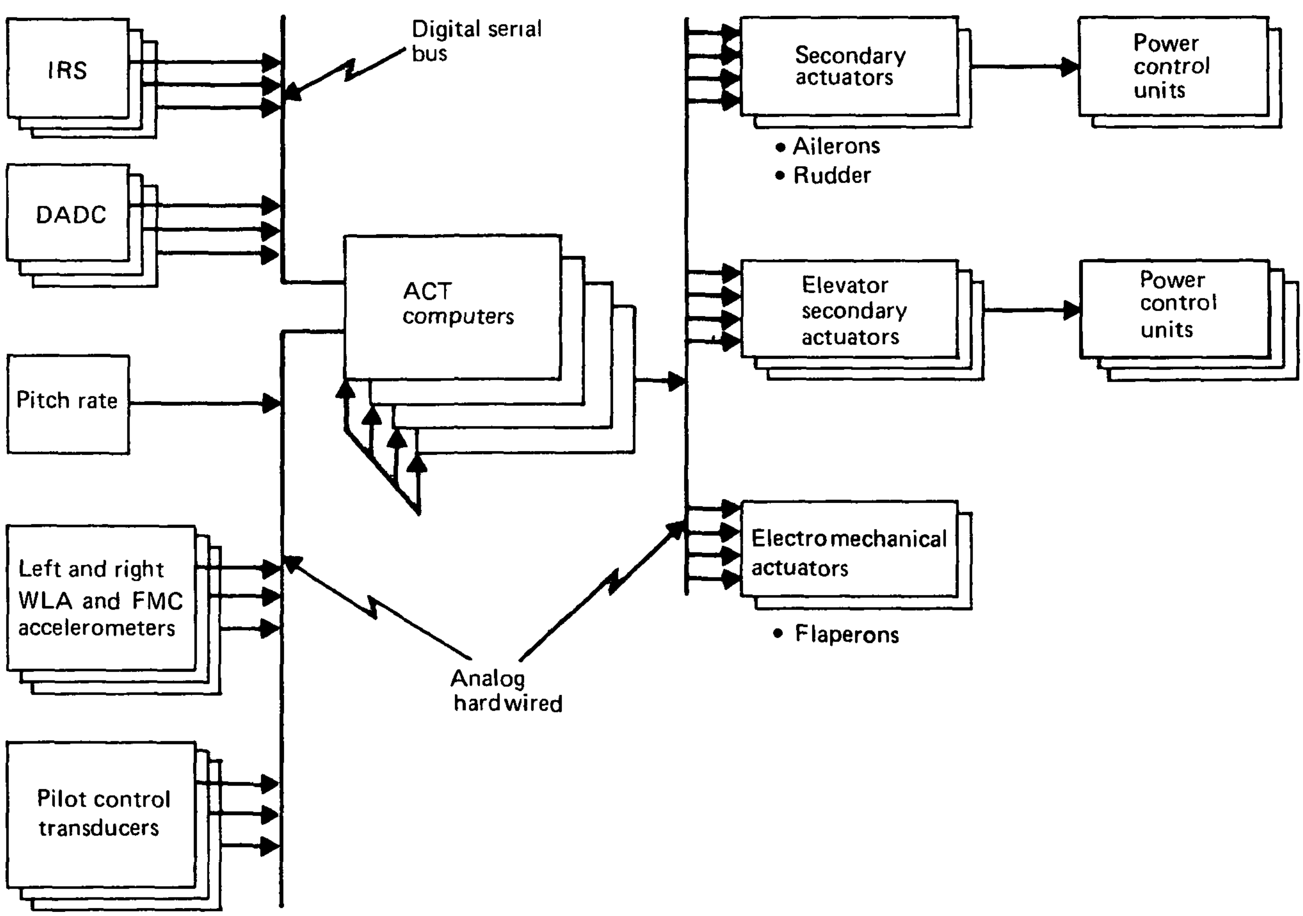


Figure 173. Low-Risk System

14.3.2.2 Medium-Risk System

The medium-risk system (fig. 174) uses extensive busing and multiple microprocessors. All axes are FBW. The system assumes projections of software and integrated circuit technologies that have a reasonable probability of being available for system realization by 1990.

The sensor set is the same as for the low-risk system except that no separate pitch-rate sensor is provided. A Luenberger observer is used to estimate pitch rates if two of the three identical IRS rate gyros fail. Thus, the fourth pitch-rate sensor required for the low-risk system can be eliminated.

The computer architecture differs significantly from the low-risk system in that it is a multiple-microprocessor-based system. Each of the four channels, which operate asynchronously, has several microprocessors. Each processor performs a specific function and each has low loading and simplified software. The sensor processors, located one to each sensor, perform analog-to-digital (A/D) conversion and data preparation for insertion

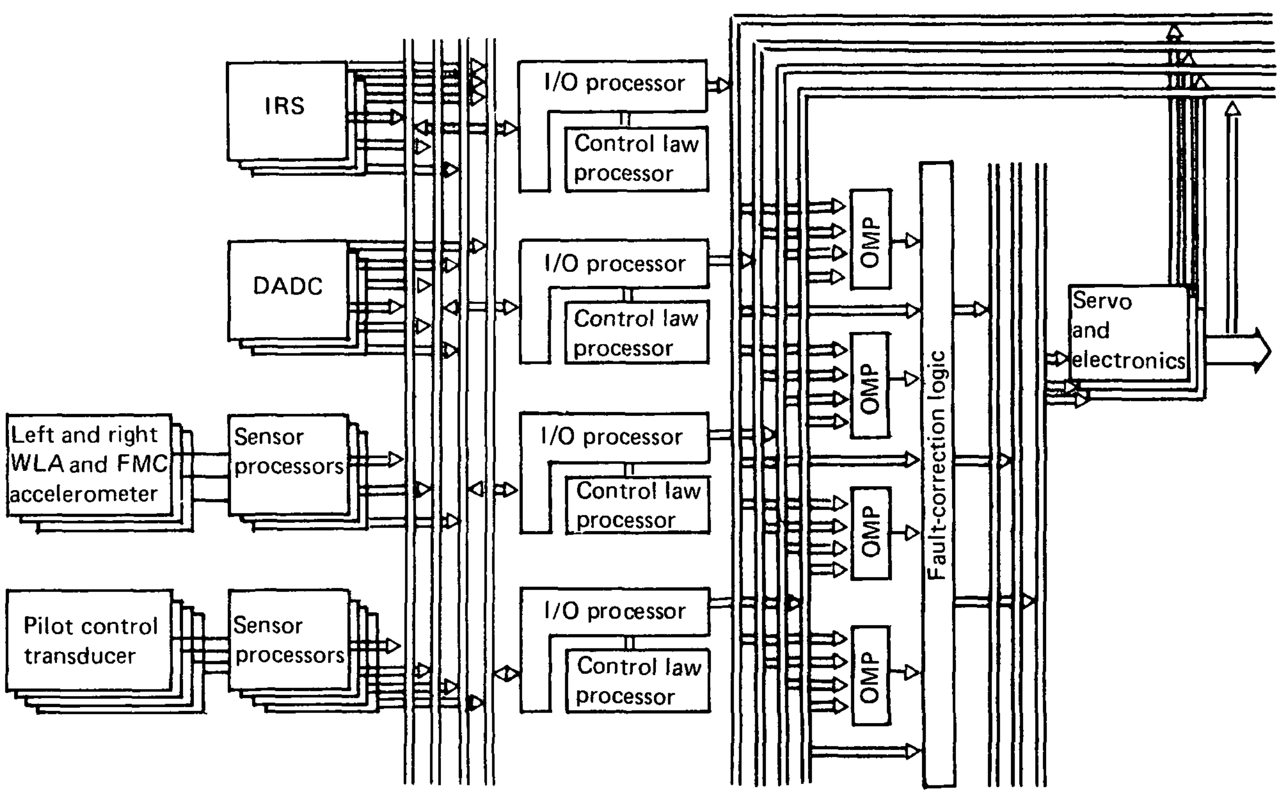


Figure 174. Medium-Risk System

on the digital bus. The I/O processor (one to each ACT computer) performs the basic sensor failure management. The I/O processor also outputs the control commands on the output data buses. The control law processor does the control computations, and the output monitor processor (OMP) performs downstream checks on control command validity and failure management of the servos.

The architecture uses digital serial data buses extensively. There is a set of four buses from the sensor processors to the I/O processor, four buses from the I/O processors to the OMPs, and three buses from the OMP to the servoprocessors.

14.3.2.3 High-Risk System

The high-risk system, shown in Figure 175, is characterized by a quadruple bus set that carries both sensor and actuator data.

All the circuits within the ACT computer are 100% self-checking. With the projected processor speed, there is adequate throughput in each computer to perform the ACT functions. Redundancy management is simplified as a result of the self-checking capability.

The low-, medium-, and high-risk systems are compared in Table 71.

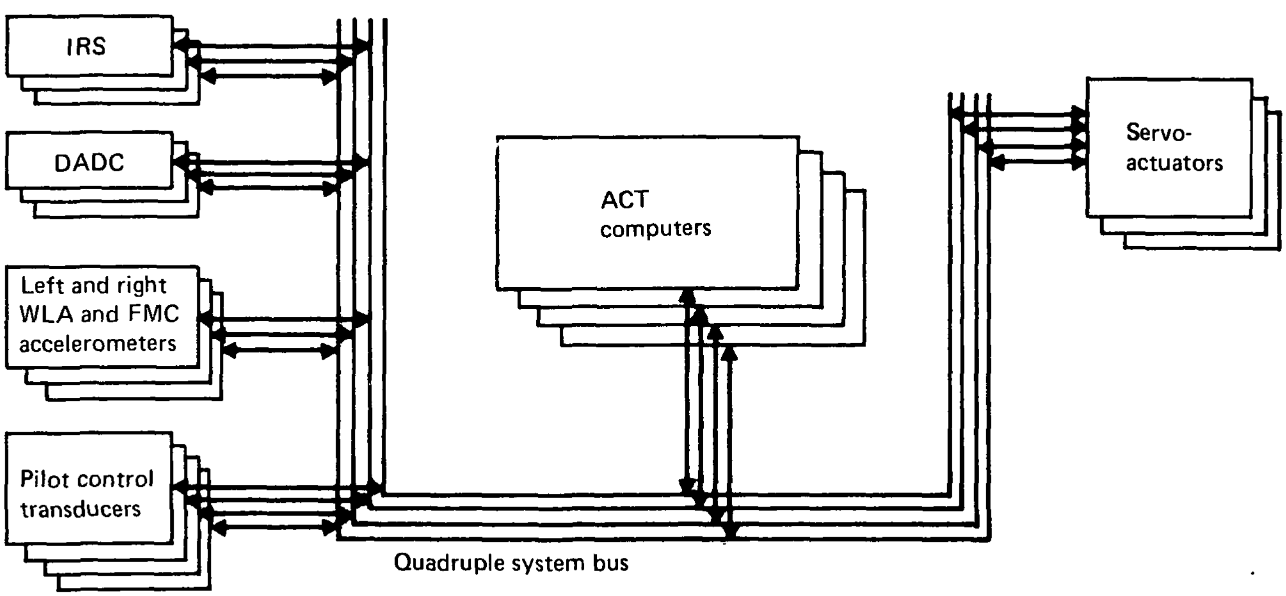


Figure 175. High-Risk System

Table 71. Alternative System Comparison

Characteristics	System features		
	Low risk	Medium risk	High risk
• Sensor set	• Three IRSs • Three DADCs • Three sets of accelerometers for WLA and FMC • Four pilot input transducers • One pitch-rate sensor	• Same as the low-risk system without pitch-rate sensor	• Same as the medium-risk system
• Sensor input approach • IRS, • DADC	• Serial digital bus to the ACT computer	• Serial digital bus to I/O processor	• On common serial digital bus
• Others	• Hardwired analog to the ACT computer		
• Failure management • Critical functions	• Majority vote and comparison monitoring	• Same as the low-risk system	• Same as the low-risk system
• Crucial functions	• Same with fourth pitch-rate sensor	• Same with Luenberger observer to estimate q from n_z and other signals	• Same as the medium-risk system
• Bus structure	• Two bus systems • ARINC 429 from IRS and DADC to ACT computer • Serial digital data exchange between computers	• Three bus systems • Quadruple sensors to I/O processor • Quadruple I/O processor to output monitor processor • Triplex, output monitor processor to servos	• One universal quadruple bus system • Self-checking
• Computer system • Redundancy • Architecture	• Quadruple • Uniprocessors	• Quadruple • Multimicroprocessors • Sensor • I/O • Control law • Output monitor • Servo	• Quadruple • Self-checking computing modules composed of multiple processors
• Synchronization	• Frame synchronized	• Asynchronous	• Asynchronous
• Failure management	• Self-check and bit-by-bit comparison monitor	• Output monitor processor, comparison	• Completely self-checking, no comparison
• Analog backup	• Yes	• No	• No
• Servos and actuators • Servo-loop electronics	• In ACT computers	• In dedicated servo microprocessor	• Incorporated in multiprocessor
• Command output approach	• Hardwired analog	• Serial digital buses • Quadruple to OMP • Triplex OMP to servo	• On common serial digital bus
• Failure management	• Monitored in ACT computer • Hardwired fault correction	• Monitored in OMP • Fault correction via serial bus	• Monitored in ACT computer • Fault correction via bus
• Software characteristics	• Complex, 1980 technology	• Simplified, segmented into microprocessors by function, reduced redundancy management required	• More simple because of self-checking autonomous channels, highly reliable through advanced verification and validation
• Reliability assessment (probability of failure per 1-hr flight for sensing and computing, actuators neglected)	• 4×10^{-12}*	• $< 10^{-12}$*	• Not assessed

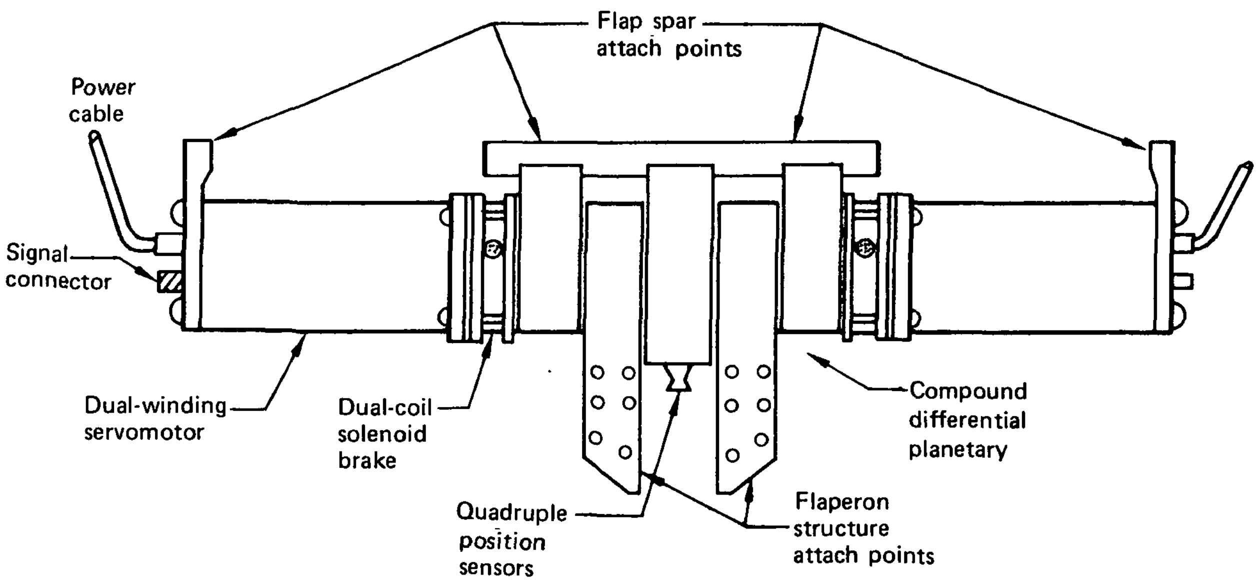
*Assumes software reliability and coverage generally equal to 1 0

14.4 ACTUATION STUDIES

Studies were conducted to define the actuators to be used with the 1990 ACT system. An EMA was defined for control of the flaperons. An active, online form of integrated hydraulic actuator was defined for each of the other actuation requirements.

14.4.1 ELECTROMECHANICAL ACTUATOR

Figure 176 shows the electromechanical flaperon hinge-line actuator. Two dual-wound motors power the actuator. The two dual-wound, brushless, permanent magnet motors drive a differential gear set that sums the positions of each motor to determine the output.



Design characteristics

- 0.76m (30-in) long, 0.1m (4-in) diameter
- 22-kg (48-lb) actuator assembly
- Dual-dual motors, velocity summed
- Dual, power off, brake
- Passive thermal mass temperature control
- 270V dc power
  - 12A (6 each) at stall
  - 1A (0.5 each) at no-load maximum rate

Performance capability

- 4240-N·m (37 500-in-lb) minimum stall torque with two of four channels
- 80-deg/s no load with two motors
- Capability to hold stall continuously
- 40-rad/s minimum bandwidth
- Two-channel fault tolerance
- +120° to -68°C (+250° to -90°F) operational environment

Figure 176. Electromechanical Flaperon Hinge-Line Actuator

The actuator is controlled by two servodrive electronics units (SDEU) that are sized so they may be located within the flap. The SDEUs:

- Interface with actuator buses
- Compare actuator deflection with that commanded
- Commutate the field windings to achieve the desired response
- Convert 115V, 3-phase ac power to 270V dc power
- Monitor the actuator performance and correct for faults
- Report fault status to ACT computer via actuator buses

The redundancy management enables two failures in any one actuator to be tolerated without loss of function so that reliability in excess of the minimum required is achieved.

14.4.2 ACTIVE, ONLINE INTEGRATED HYDRAULIC ACTUATOR

The active, online integrated actuator is shown in Figure 177. An integrated actuator was selected instead of a secondary actuator and power control unit combination because of lower cost, lighter weight, and higher reliability for this FBW application. The active, online approach was preferred to alternative integrated actuator concepts for the same reasons. Figure 177 illustrates the simplicity of the actuation concept. Three actuators operate each elevator control surface where fail-operational/fail-operational controls are desired. One of the three is designated as the active cylinder. The other two also follow the position command. However, the difference in the cylinder pressures between the active cylinder and the online cylinders is fed back to equalize the force outputs. Because the equalizing feedbacks are limited to small values, any failure transients are almost immediately opposed. Computer monitoring of the actuators bypasses the faulty cylinder and redesignates the active channel as necessary.

Other surfaces are driven by two appropriately sized actuators. The operation is similar except that only fail-operational performance is achieved.

14.5 1990 ACT SYSTEM IMPLEMENTATION

A derivative of the medium-risk system described in Subsection 14.3.2.2 was selected for further evaluation and cost-of-ownership analysis. It is called the 1990 ACT system and has the following features:

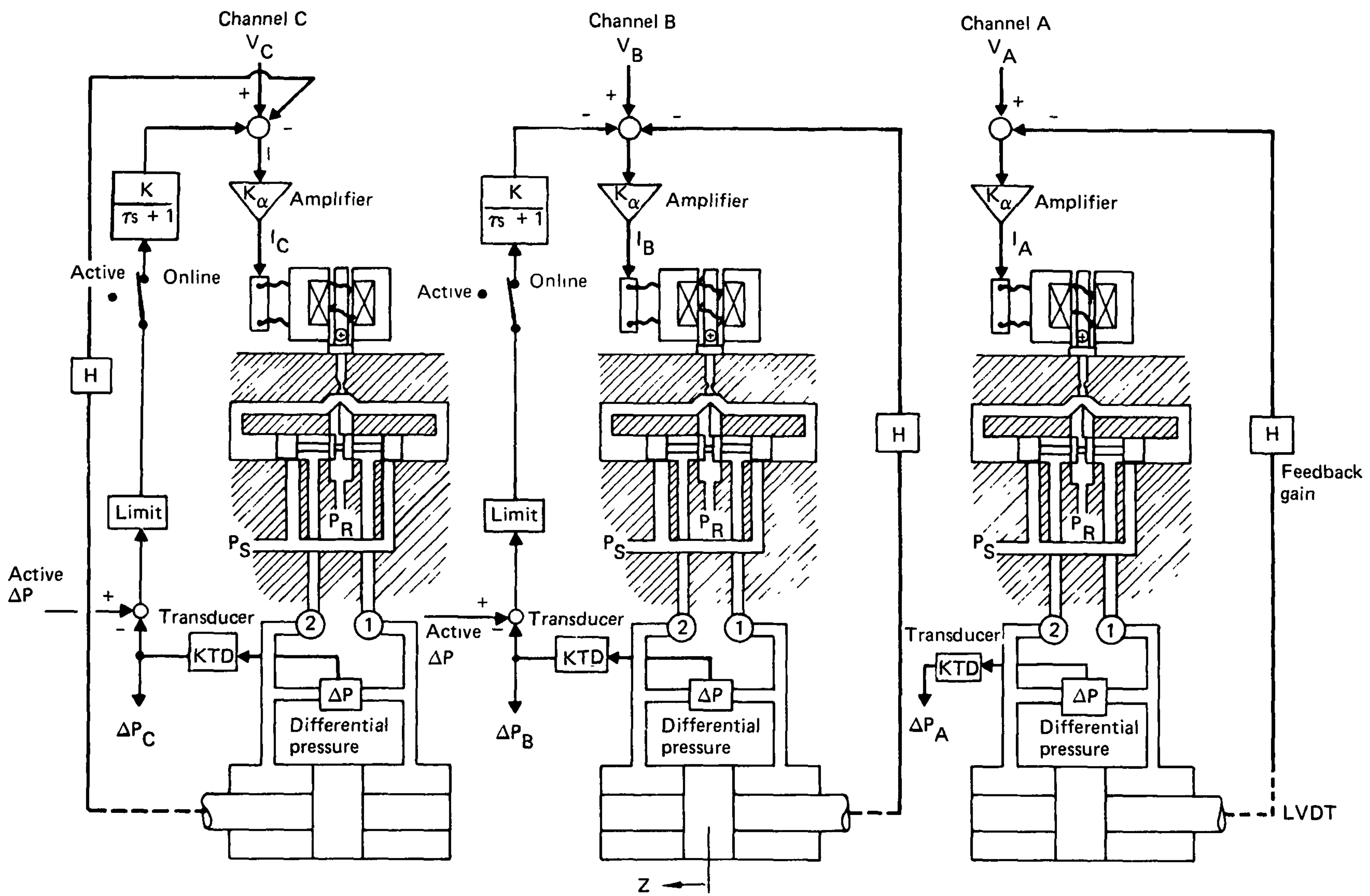


Figure 177. Active, Online Integrated Actuator

- Redundant buses are used for sensor-computer and computer-actuator interfaces. This saves weight and cost by marked reduction of aircraft wiring and makes all sensor data available to all computing channels.

- Computing is asynchronous between channels and is compartmented such that separate microcomputers perform I/O processing, control law computations, and redundancy management. This avoids monolithic software structure and results in lower costs of software design, validation, and verification.

- Sensors and actuators have self-contained electric power supplies and bus interface circuits.

- Crucial control law computation load is assumed by the I/O microcomputer if the control law microcomputer fails. This adds redundancy and reliability to the crucial functions.

14.5.1 SYSTEM ARCHITECTURE

The 1990 ACT system is integrated; all functions are performed by each of a central set of four ACT computers. Sensors and control surface actuators are shared between functions to the extent allowed by the control laws.

Airplane primary control is FBW; all control surface actuators are signaled electrically, and there is no mechanical control system.

The system architecture is shown in Figure 178. A set of four buses interfaces the sensors with the ACT computers. Similarly, a set of four buses interfaces the ACT computers with the surface actuators. The ACT Maintenance and Display Computer, warning electronics module, and dedicated ACT panel also interface with the computers via the same set of four buses associated with the surface actuators.

The DADC and IRS are airplane sensors that interface with the ACT computers via buses, as defined for the aircraft as intersystem buses. The IRS can interface with the ACT system via the quadruple intrasystem buses and the intersystem buses. This is necessary because the pitch-rate and acceleration signals, used for crucial functions, need to be

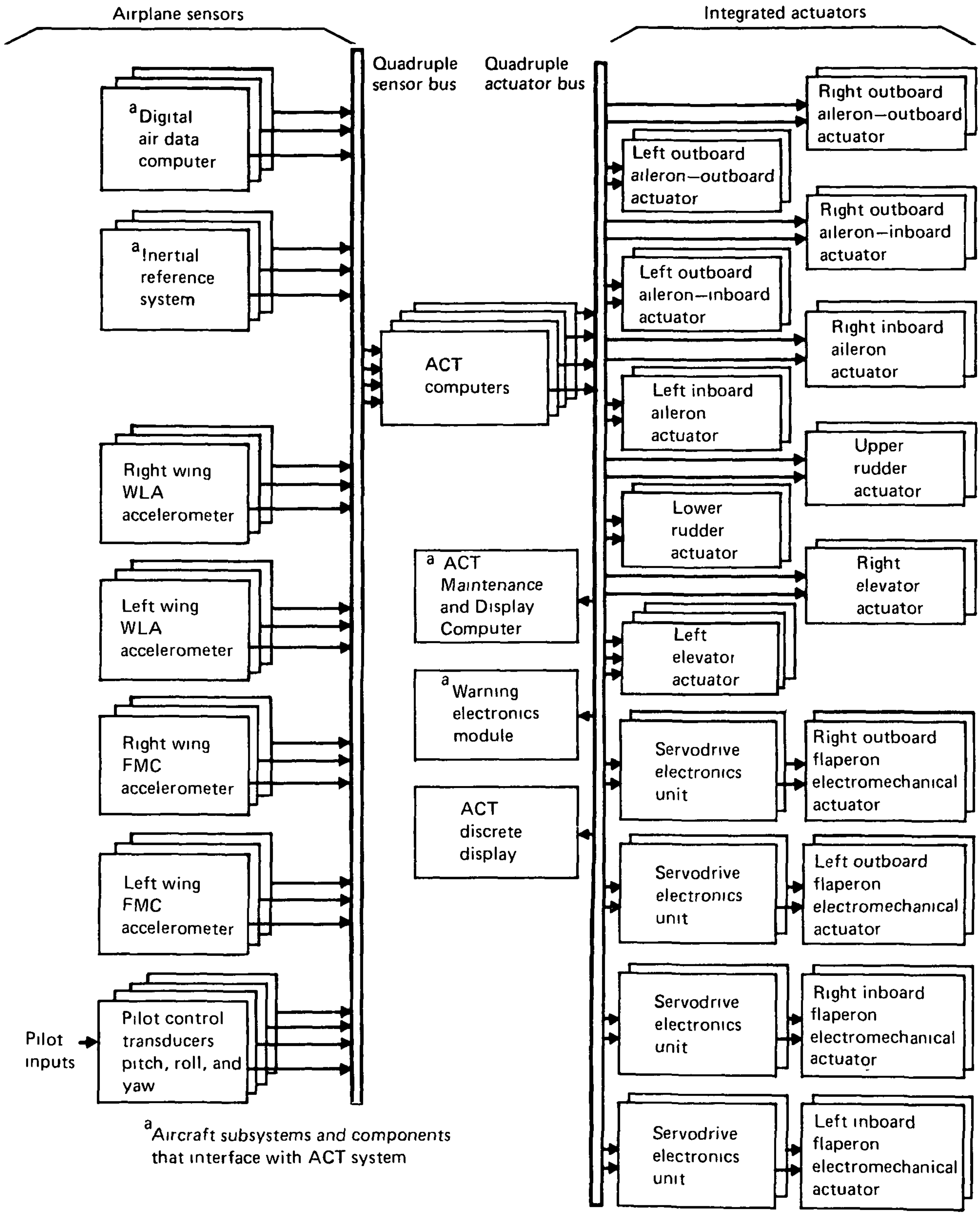


Figure 178. 1990 ACT System Architecture

obtained more reliably and faster than is possible with an intersystem bus presumed to be of the ARINC 429 type.

Each sensor contains bus interface electronics. The interface electronics include an A/D converter, an asynchronous serial I/O communications circuit, and the logic required to recognize a data request and format the data response for transmission to the computers on the sensor bus. Each response includes data, label, and a parity bit.

Each hydraulic servoactuator contains electronics to receive and decode the serial data, convert the command to an analog signal, demodulate the feedback signals required for servo control, and close the servo loop. The servoelectronics also transmit signals on the bus in response to ACT computer requests, which include the feedback signals required for monitoring of the servo.

The EMAs associated with the flaperons similarly interface with the ACT computers. However, each EMA has two SDEUs. Two are needed because the control electronics required for the EMAs are considerably more elaborate than those required for the hydraulic servos. Much of the EMA monitoring and redundancy management is done in SDEUs.

The ACT Maintenance and Display Computer interfaces with the actuator buses. It receives data from the ACT computers indicating the failure state of the active control system and provides fault annunciation messages and signals to the caution and warning system.

The ACT discrete display also interfaces with the actuator buses and provides an independent source of system failure status information to the crew. The caution and warning system discrete displays also interface with the actuator buses.

14.5.2 FAILURE MANAGEMENT

Figure 179 is a simplified diagram illustrating the redundancy management used in the 1990 ACT system. Quadruple (or triple) sensors interface with quadruple ACT computers through a set of sensor data buses so that each ACT computer has access to all the redundant sensor data. The sensors are monitored, and the actuator commands are computed in ACT computers operating asynchronously with one another. The actuator commands are output on quadruple actuator buses to hydraulic actuators and EMAs at the control surfaces. Monitor processors contained within the ACT computers monitor actuator commands and actuator performance and effect fault correction as appropriate.

The sensor bus transmits multiplexed sensor data to the four ACT computers. There are approximately 20 data inputs to these buses. The transfer rate will approximate 500 samples per second per sensor.

Each of the ACT computers includes a bus controller that controls one of the buses. The bus controller is dependent only on the computer power supply and clock for its proper operation. Loss of a bus or of a controller results in loss of data from sensors on that bus. Therefore, highest reliability for each bus and bus controller is of paramount importance.

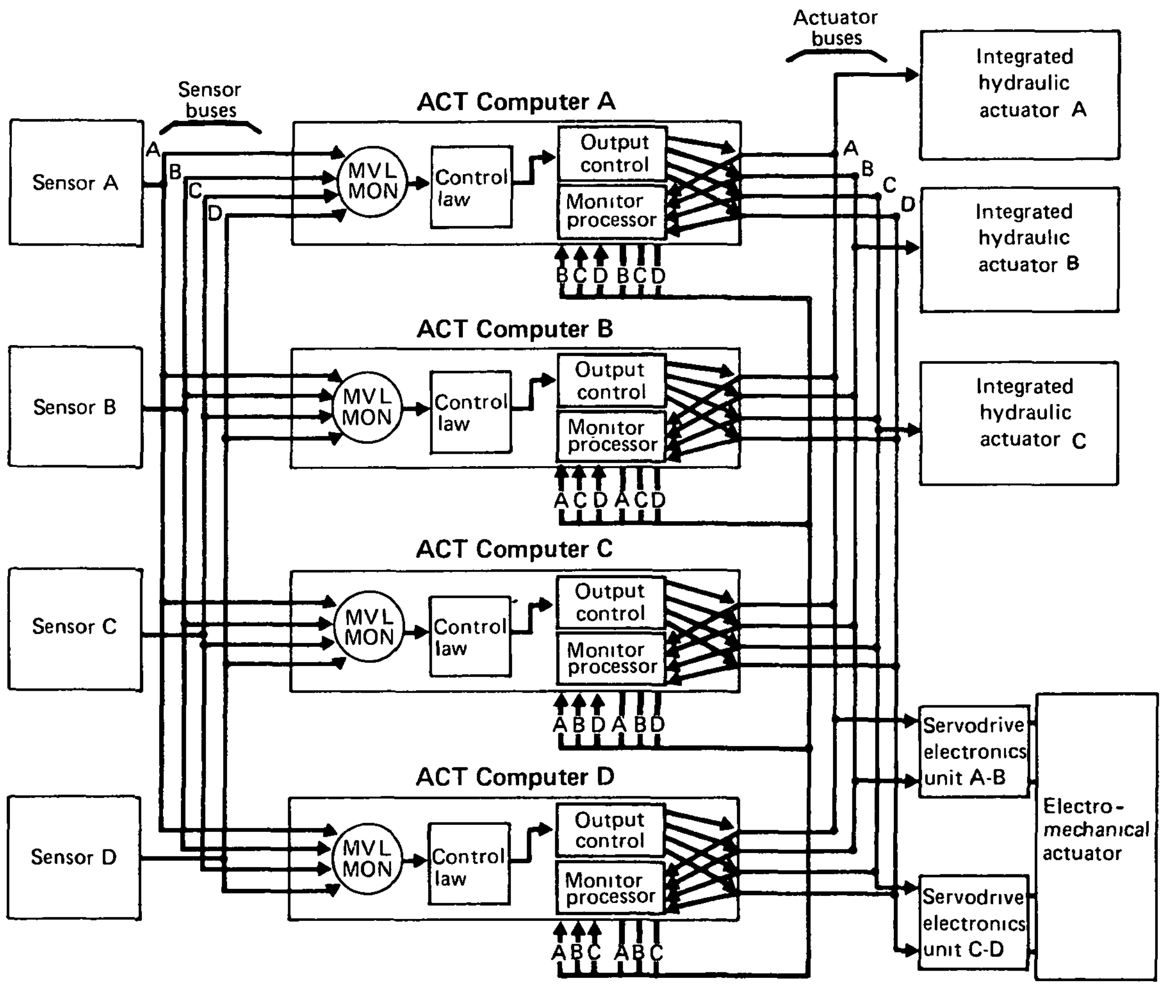


Figure 179. 1990 ACT System Simplified Redundancy Management Block Diagram

A dead controller or a dead terminal (located with each sensor) will be detected by sensor comparison monitoring within the ACT computers and by watchdog timer checks associated with the bus controller.

Parity bit errors will be used to reject specific faulty data transmissions.

Certain sensor data, typified by air data signals, are transmitted to the ACT computers by ARINC 429 buses. Each of the A, B, and C computers receives data from one of the three DADCs in this manner at a relatively low sample rate. These data are output on the sensor buses by the ACT computers at a rate of 500 samples per second so that each computer has access to all three sets of air data.

The midvalue of the redundant sensor data is determined in each of the four ACT computers. The midvalue of four signals is the average of the two middle signals. When only two sensors are valid, their average is determined in lieu of a midvalue.

These sensor data are compared across channels. Discrepancies are used to detect sensor failures and isolate the fault whenever possible. When a discrepancy between two valid sensors occurs, fault isolation is not attempted in the case of most sensors. Both sensors are presumed failed, and the ACT system continues to function in a degraded state, normally discontinuing the function associated with the failed sensor type. The pilot command transducer and pitch-rate signals, which are essential to safe flight, are treated somewhat differently.

If faults occur in the quadruply redundant pilot command transducers so that a discrepancy between the only two remaining signals occurs, a sensor validity signal is used to identify the good sensor. Safe flight is continued using this single good sensor. The sensor validity signal is obtained by means of circuitry that takes the sum and difference signals formed when the linear variable differential transducer (LVDT) secondary center tap is grounded. The difference signal is proportional to the transducer signal; the sum remains nearly constant for a properly functioning transducer.

The pitch-rate signal is compared not only with the other pitch-rate signals but, following a sequence of failures, is compared with an estimated pitch-rate signal based on normal acceleration.

The control laws compute the servoactuator commands in each ACT computer. Because an asynchronous computation is involved, the servocommands computed by the four ACT computers will not agree precisely. Channels are equalized by noting the difference between a particular ACT computer servocommand and that associated with the active channel.

Each ACT computer is assigned a particular bus on which it outputs the servocommands. The monitor processor compares the servocommands as issued on each bus. Faulty servocommands are detected and isolated by the monitor processor using comparison monitoring techniques. If three computer failures ever occur, the third computer failure is isolated by the computer self-test. The computer self-test confidence is greater than 95%. Single-channel operation is permitted following a third computer failure because there is no apparent better alternative.

14.5.3 FUNCTIONAL RELIABILITY

The probability of the FBW, crucial PAS system failing to function during a 1-hr flight was calculated as 1.7×10^{-12}. The probability of loss of any critical ACT function was calculated as 2.7×10^{-7}.

Both estimates are more than an order of magnitude less likely than the functional failure probability goals (less than 10^{-9} for crucial functions and less than 10^{-5} for critical functions). However, these predictions assume that software reliability and coverage are generally equal to 1.0.

14.5.4 COST-OF-OWNERSHIP EVALUATION

Cost of ownership for the 1990 ACT system was analyzed and compared with the current technology systems. Because cost of ownership depends upon the airplane configuration, and the most complete data are available for the Initial ACT Configuration (ref 2), the external aerodynamic characteristics and active control functions and surfaces of that airplane were assumed. This means that flaperons were not included in the cost-of-ownership evaluation. Analysis for the 1990 ACT system was performed based on increments from the current technology Selected System, which was analyzed earlier.

The 1990 ACT system differs basically from the Selected System as follows:

- All mechanical connections and components connecting the cockpit flight controls to the actuator servovalves are deleted.

- Mechanical servo feedback to hydraulic power control unit (PCU) servovalves is deleted.

- All ACT secondary servos and two FMC servos are deleted, and their functions are assumed by the generally more expensive (but lighter weight) FBW PCUs listed in Table 72.

- Four flight control computers (see table 72) replace the three primary and four standby short-period PAS computers of the Selected System.

Table 72. Advanced Technology Fly-by-Wire Line Replaceable Unit Nomenclature, Weight, and Quantity Per Aircraft

Component	Quantity per shipset	LRU weight, kg (lb)
ACT computer	4	5.4 (12.0)
Sensors		
• Control position transducer		
• Pitch	2	0.9 (2.0)
• Roll	2	0.9 (2.0)
• Yaw	2	0.9 (2.0)
• Accelerometer, wing FMC	2	0.9 (2.0)
• Accelerometer, wing WLA	2	0.9 (2.0)
Servos		
• Electrohydraulic		
• Outboard aileron inboard	2	5.1 (11.2)
• Elevator	6	4.7 (10.4)
• Rudder	4	8.0 (17.6)
• Outboard aileron outboard	4	2.8 (6.1)
• Inboard aileron	4	3.4 (7.6)

- Sensors peculiar to the Selected System (and not included in the Baseline) are replaced by the more expensive sensors shown in Table 72, which can communicate directly to the digital data bus used for signal transmission. The digital sensors added for stabilizer, flap, and slat position have not been included because this information is hardwired to the electronics bay in the Baseline Aircraft.

- All autopilot and yaw damper actuators are replaced by the primary FBW actuators.

- Because insufficient definition was available and data buses are used for all signal communication, a wiring weight estimate was not included.

The cost-of-ownership parameters in Table 73 were based on estimates of line replaceable unit (LRU) cost, weight, and maintenance cost for the 1990s time frame. Computer analysis showed some significant differences from the Selected System, which are discussed as follows:

- Incremental Airplane Cost—Most of the cost reduction is due to the deletion of the thousands of parts inherent in the mechanical transmission of pilot's control signals. Deleting all autopilot actuators also contributes.

- Maintenance Manual Cost—This cost was estimated differently than the Selected System cost. Total cost of developing and printing maintenance manuals was estimated to be $547 000. Cost for a 30-aircraft fleet was determined by dividing the total cost by the estimated production run of 300 airplanes and multiplying by 30. This cost is more than the Selected System cost (based on typical autopilot cost) because of FBW in all axes. Some reduction might be expected from deleting mechanical controls, but this has not been considered in this preliminary analysis. Maintenance manual cost is, in any case, a low-leverage parameter.

- Maintenance Cost per Flight Hour—Major reductions here stem from the deletion of all mechanical connections between the cockpit and the control surface servos. Deletion of all autopilot actuators is a contributor of almost equal importance.

Table 73. Cost-of-Ownership Results for Various ACT Systems

Parameter incremented	Current technology				Advanced technology
	Integrated	Segregated	Selected	Selected + pitch FBW	1990 ACT
Aircraft purchase cost per aircraft (in dollars)	274K	390 2K	297.1K	207K	98.8K
Maintenance manual cost per 30-aircraft fleet (in dollars)	21K	31.4K	26.1K	26 1K	54.7K
Test equipment cost per 30-aircraft fleet (in dollars)	22.5K	44.9K	33.6K	33.6K	10.0K
Spare inventory initial cost per 30-aircraft fleet (in dollars)	250K	356K	271.1K	271 1K	221 1K
Maintenance cost per aircraft flight hour (in dollars)	4.18	4.91	4.22	3.98	-0.59
Departure delay and cancellation cost per aircraft flight hour (in dollars)	0.54	0.45	0.19	0.12	0.54
Change in system weight relative to integrated ACT	0	+114 kg (+252 lb)	+14 kg (+30 lb)	-156 kg (-345 lb)	-629 kg (-1386 lb)
Fuel saving per flight hour at 863 km (466 nmi)	160 kg (352 lb)	146 kg (322 lb)	160 kg (352 lb)	172 kg (379 lb)	212 kg (468 lb)
Payback period in years	2.83	4.14	2.98	2.02	< 2.0
Incremental return on investment to airline	25.1%	22.1%	24 6%	27.6%	34.9%

- Maintenance manual cost = \$547K total, which gives $\frac{547K}{300}$ x 30 = \$54.7K for 30-airplane fleet

- Departure Delay and Cancellation Cost per Flight Hour—Because inadequate data and resources were available to perform a detailed analysis, the 1990 FBW system was judged to be the same as the Integrated System, as it is a four-computer integrated system. However, FBW in all axes will require careful reliability analysis before an acceptable minimum equipment list can be defined and a detailed departure reliability predicted.

- Change in System Weight—This very significant [628.7 kg (1386 lb)] weight reduction (compared to Integrated) was largely due to the deletion of mechanical flight controls, but deletion of all autopilot actuators and lower weight for the FBW surface control actuators [112.9 kg (249 lb)] also contributed.

- Fuel Savings per Flight Hour—This significant change is due to the weight savings of 628.7 kg (1386 lb).

- Return on Investment and Payback Period—The large improvements for full FBW are largely due to the reduction in first cost and fuel burn obtained from weight reduction.

15.0 CONCLUDING REMARKS

The results to date of the IAAC Project work indicate that the concept of an Active Controls Technology (ACT) airplane, designed as such from the beginning, will indeed yield an important saving in fuel over a similar commercial transport airplane without active controls. The results of the current and advanced technology ACT control system work presented in this report indicate that it is feasible to support such an active controls airplane with a control system that meets all reliability and availability requirements, assuming fault-free software and coverage approaching 1.0. The results also indicate that this can be done at a cost that enables a suitable return on investment for the operators.

The Current Technology ACT Control System Definition Task had two primary objectives. The first objective was to define a digital ACT control system architecture, using digital flight control system elements currently being used or considered in commercial jet transports, that exhibits high reliability and low cost. The second objective was to identify the major concerns and unresolved issues relative to the use of such a control system. The technical approach was to develop two extreme forms of system architecture and then select a system architecture for the ACT control system that uses the best features of these extreme forms. The major conclusions of this work are as follows:

- All three of the control systems studied meet the reliability requirements of the design requirements and objectives, if software reliability and coverage are assumed equal to 1.0. The Segregated System is predicted to be the most reliable, followed in order by the Selected and Integrated Systems.

- The Integrated System, with its single set of redundant digital control computers, is the most efficient of the three systems. It satisfies function and reliability requirements at the lowest cost.

- The Segregated System failed to show the expected major improvement in reliability. Overall system reliability is principally a multiple-function loss probability and is dependent primarily upon the digital air data computers and inertial reference systems.

- The Segregated System cost is unacceptable, being 30% greater than the Selected System cost and 40% greater than the Integrated System cost. Although the individual control computers specified for the Segregated System are based upon microprocessor technology, the resulting reduction in unit cost, compared to the Integrated System computers, is not sufficient to compensate for the large number (19) of computers required for the Segregated System.

- The Selected System shows a decided reliability improvement over the Integrated System with a small increase in cost.

The major concerns that arise from review of these results center around system complexity and the ever-present question of system reliability in the operational environment. The hardware reliability predictions are based on the assumption of software reliability and coverage equal to 1.0. Although the absolute values of the resulting reliability predictions may be suspect, their use as one relative figure of merit is considered well founded. There is no generally accepted method to prove software reliability equal to the required level. However, 15 years of Boeing experience in engineering real-time digital control systems has shown that a process that begins with careful functional analysis and leads through requirements, design, coding, verification, validation, exhaustive testing, configuration control, and careful documentation can produce highly reliable real-time control software. Therefore, it is concluded that the ACT systems can be implemented using currently available technology and software design processes.

The objectives of the Advanced Technology ACT Control System Definition Task were to (1) determine the benefits of synthesizing ACT control laws using optimal control and estimation theory, (2) determine the effects of actuation system nonlinearities on gust-load alleviation and flutter suppression effectiveness, and (3) identify advanced flight control system implementation concepts as an alternative to the current technology implementation of ACT control functions.

The technical approach to the synthesis evaluation objective was to synthesize control laws, using these techniques, to equal or exceed the performance of the systems synthesized using classical techniques. The actuator nonlinearities were examined by introducing representative position and rate limits and output hysteresis into the mathematical model and comparing load reduction in the presence of discrete gusts. The alternative implementation objective was met by determining the system components (e.g., sensors, computers, actuators, and fiber optics) that would probably be available in 1990 for use on commercial transports. Using these components, three systems—with various risk levels—were formulated to meet the same design requirements and objectives that the current technology system was designed to meet. The best of these systems was examined further.

The major conclusions of this work are as follows:

- The analysis procedures presented in this document offer systematic methods for selecting the proper control surfaces, actuation bandwidths, and sensor locations and for evaluating open- and closed-loop stability and gust responses for an ACT airplane.

- Design procedures based upon time-domain optimal control theory offer a direct and systematic method to derive multiloop control laws that satisfy typical active control design requirements.

- The synthesis procedure using optimal control theory created a pitch-augmented stability control law that produced augmented airplane behavior similar to an ideal model directly and systematically.

- The medium-risk 1990 technology ACT system design is a quadruply redundant integrated system combined with fly-by-wire (FBW) primary controls in all three axes. Its architecture is strongly oriented toward digital buses and multiple microprocessors, using quadruple input buses coupling digital sensors to the four central control computers and quadruple output buses feeding servoelectronics units incorporating digital-to-analog conversion. The resulting predicted system reliability is appropriate for an ACT flight control system, even though fault-free software and coverage approaching 1.0 were assumed.

The encouraging results of the control system work, and important unresolved issues, emphasize the desirability of proceeding with the planned laboratory tests and flight demonstration. Thus the plan is to continue the NASA-Boeing IAAC Project activity according to the project plan.

16.0 REFERENCES

1 Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport–Project Plan. NASA CR-3305, Boeing Commercial Airplane Company, February 1981.

2 Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project–Initial ACT Configuration Design Study. NASA CR-159249, Boeing Commercial Airplane Company, July 1980.

3 Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project–Conventional Baseline Configuration Study. NASA CR-159248, Boeing Commercial Airplane Company, June 1980.

4 Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project–Wing Planform Study and Final Configuration Selection. NASA CR-165630, Boeing Commercial Airplane Company, June 1981.

5 "Airplane System Design Analysis." Advisory Circular, Federal Aviation Administration, AC-25.1309-X (to be published in the Federal Register October 1981).

6 Bjurman, B. E., G. M. Jenkins, C. J. Maereliez, K. L. McClellan, and J. E. Templeman. Airborne Advanced Reconfigurable Computer System. NASA CR-145024, 1976.

7 McConnel, S. R., D. P. Siewiorek, and M. R. Tsao. "The Measurement and Analysis of Transient Errors in Digital Computer Systems." Proceedings of the 9th Annual International Conference on Fault Tolerant Computing, Madison, Wisconsin, June 20-22, 1979.

8 Bisplinghoff, R. L., H. Ashley, and R. L. Halfman. Aeroelasticity. Addison-Wesley, 1965.

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-165631		

4. Title and Subtitle	5. Report Date
Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project—Current and Advanced ACT Control System Definition Study, Volume I	October 1981
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
Boeing Commercial Airplane Company Preliminary Design Department	D6-48673
9. Performing Organization Name and Address	10. Work Unit No.
Boeing Commercial Airplane Company P. O. Box 3707 Seattle, Washington 98124	11. Contract or Grant No. NAS1-14742 and NAS1-15325
12. Sponsoring Agency Name and Address	13. Type of Report and Period Covered Contractor Report July 1978 to October 1980
National Aeronautics and Space Administration Washington, D. C. 20546	14. Sponsoring Agency Code

15. Supplementary Notes

Technical Monitors: D. B. Middleton and R. V. Hood
NASA Langley Research Center

16. Abstract

This report documents the Current and Advanced ACT Control System Definition Study Tasks of the Integrated Application of Active Controls (IAAC) Technology Project within the Energy Efficient Transport Program. These system definitions support the Initial ACT Configuration, Wing Planform Study, and Final ACT Airplane Configuration with data to validate the assessment of their energy efficiency. Study ground rules required the current technology system to use only elements fully demonstrated and available in 1980; the advanced technology system represents technology of the 1990s era. The systems mechanize six active control functions: pitch-augmented stability, angle-of-attack limiting, lateral/directional-augmented stability, gust-load alleviation, maneuver-load control, and flutter-mode control. The redundant digital control systems defined meet all function requirements with required reliability and declining weight and cost as advanced technology is introduced. They indicate the advisability of demonstrating key system elements in laboratory and flight test.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Energy Efficient Transport, Active Controls Technology, Redundant Digital Control System, Augmented Stability, Wing-Load Alleviation, Flutter-Mode Control, Angle-of-Attack Limiting, Control Configured Vehicle	FEDD Distribution

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
Unclassified	Unclassified	439	

Available: NASA's Industrial Applications Centers

N 305