NASA Contractor Report 172221

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

TEST ACT SYSTEM DESCRIPTION

FINAL REPORT

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

CONTRACT NAS1-15325
December 1983

NASA
National Aeronautics and
Space Administration

Langley Research Center
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National Aeronautics and
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Langley Research Center
Hampton, Virginia 23665

N86-14277#

FOREWORD

This document constitutes the final report of the design, acquisition, and test planning phases of the Test ACT System, a portion of the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project. The report covers work performed from June 1981 through June 1983 under Contract NAS1-15325.

The NASA Technical Monitor for this task was D. B. Middleton of the Energy Efficient Transport Project Office at Langley Research Center.

The work was accomplished by people of (1) the Preliminary Design department of the Vice President–Engineering organization of Boeing Commercial Airplane Company and (2) the Collins Air Transport Division of Rockwell International, operating under purchase order Y-405143-0935 F.

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During this work, principal measurements and calculations were made in 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.

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1.0 SUMMARY

1.0 SUMMARY

This report covers a portion of the final program element in the Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport Project, a part of the NASA Energy Efficient Transport Program. It documents the design and acquisition of the Test ACT System; planned continuation of that project includes laboratory testing and, later, flight test.

The work reported here was done by an integrated engineering and fabrication organization drawn from the Preliminary Design department of Boeing Commercial Airplane Company and Collins Air Transport Division of Rockwell International. In the period November 1981 to June 1983, this organization accomplished the following:

- Selected the system concept and the test airplane
- Performed the system design
- Designed and analyzed the control laws and tested them by piloted simulation
- Designed, fabricated, and bench tested the computer hardware, both digital and analog
- Designed, integrated, and verified the digital system software
- Selected and procured the system sensors
- Designed, fabricated, and bench tested the man-machine interface equipment for the flight crew and the test engineering personnel
- Designed modifications to the test aircraft, adding redundant secondary servos for elevator position commands
- Planned laboratory and flight test programs
- Documented all of the above steps

The end product of this work is an active controls system composed of pitch-augmented stability, pitch fly by wire, and wing-load alleviation, including both maneuver-load control and gust-load alleviation, for the Boeing 757-200 flight test airplane. The electronic equipment is mounted in consoles so it can be readily tested in the laboratory and then moved into the airplane with a minimum of dismantling.

The system is now installed in a prepared position at the Boeing Digital Avionics Flight Controls Laboratory for a series of detailed tests. This laboratory testing began with

hardware and software open-loop testing and will progress through failure detection, system integration, and finally into closed-loop testing, with increasing fidelity in simulation of flight operations, all in preparation for later installation and flight test in a 757-200 airplane.

2.0 INTRODUCTION

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2.0 INTRODUCTION

2.1 IAAC PROJECT OVERVIEW

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

This project was organized into three major elements, as shown at the top of Figure 1 (ref. 1). The first major element, Configuration/ACT System Design and Evaluation, included establishment of the design criteria appropriate for an ACT airplane (designed from the outset to utilize active controls), design of an ACT airplane configuration to meet the selected criteria, design of an active controls system based upon current technology, and selection and evaluation of a Final ACT airplane configuration. The results of these studies are documented in References 2, 3, 4, 5, 6, and 7.

In parallel with these tasks, the second major element, Advanced Technology ACT Control System Definition (fig. 2), included exploration of optimal control synthesis methods, alternative means of implementing the ACT functions using advanced technology, and an examination of the integration of ACT, control, and guidance functions. The results of these studies are documented in References 2, 3, and 8.

The final major element, Test and Evaluation, is shown in Figure 3. The components of this element, first conceived in the development of the Project Plan (ref. 1) and also shown in the figure, address reduction of the risk associated with implementation of active controls on a commercial transport. Subsequent to publication of the Project Plan, it was concluded that it would be inappropriate to conduct the wind tunnel tests described in that plan under NASA funding. The software-implemented fault tolerance (SIFT) and fault tolerant multiprocessor (FTMP) projects sponsored by NASA Langley Research Center and currently under test in the Langley AIRLAB have been considered throughout the IAAC Project, but did not directly influence the Test ACT system architecture.

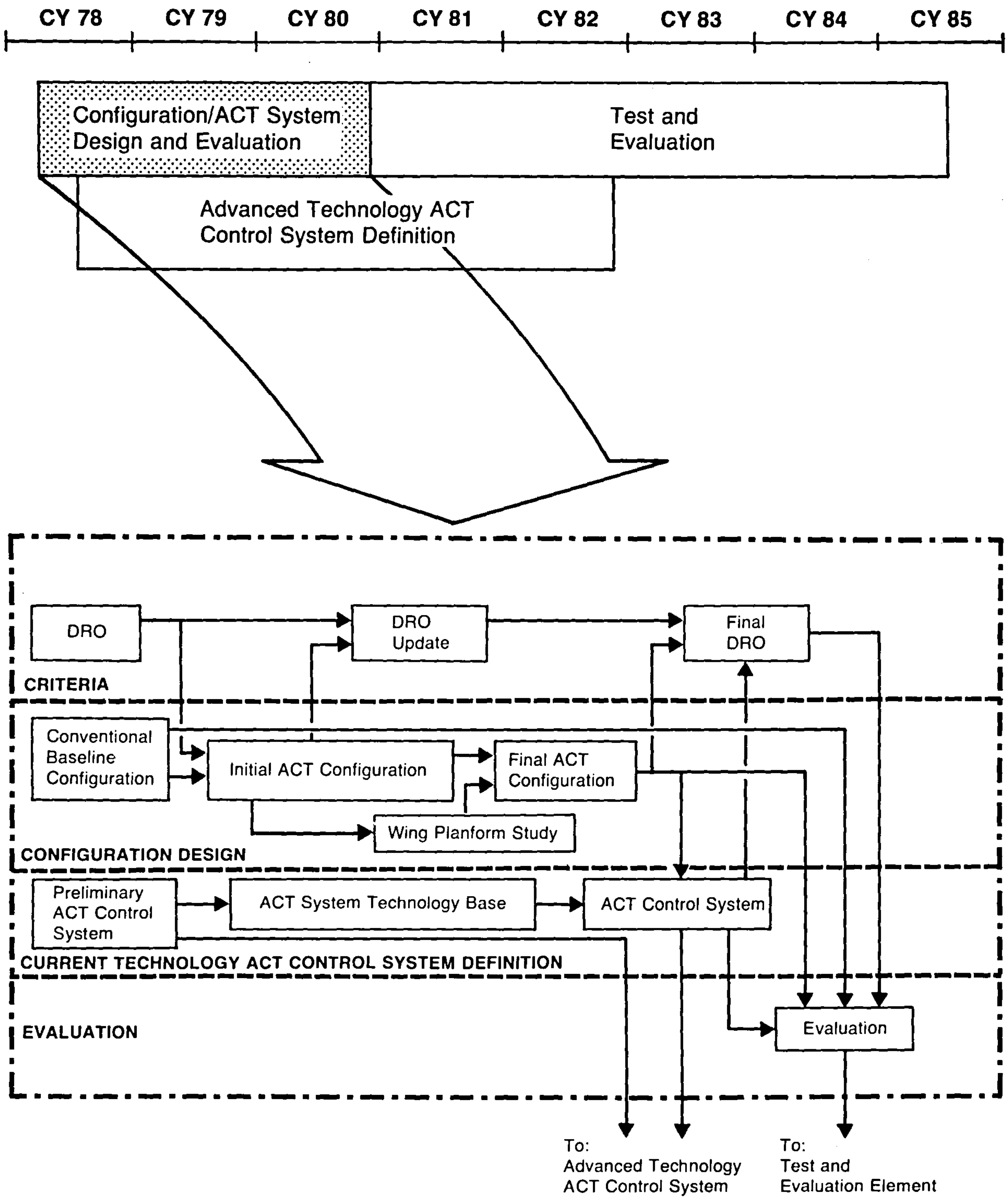

Figure 1. Configuration/ACT Design and Evaluation Element

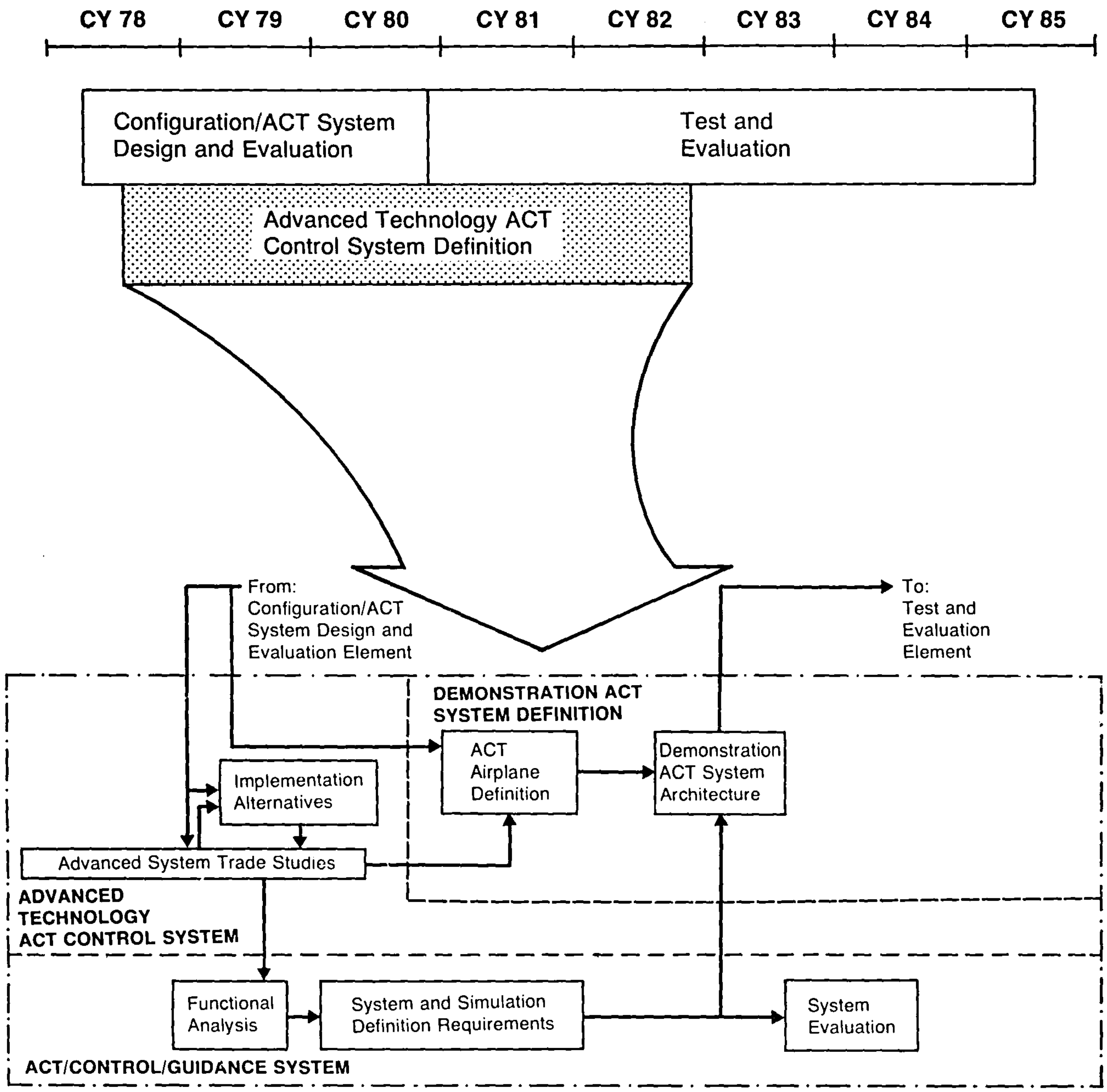

Figure 2. Advanced Technology ACT Control System Definition Element

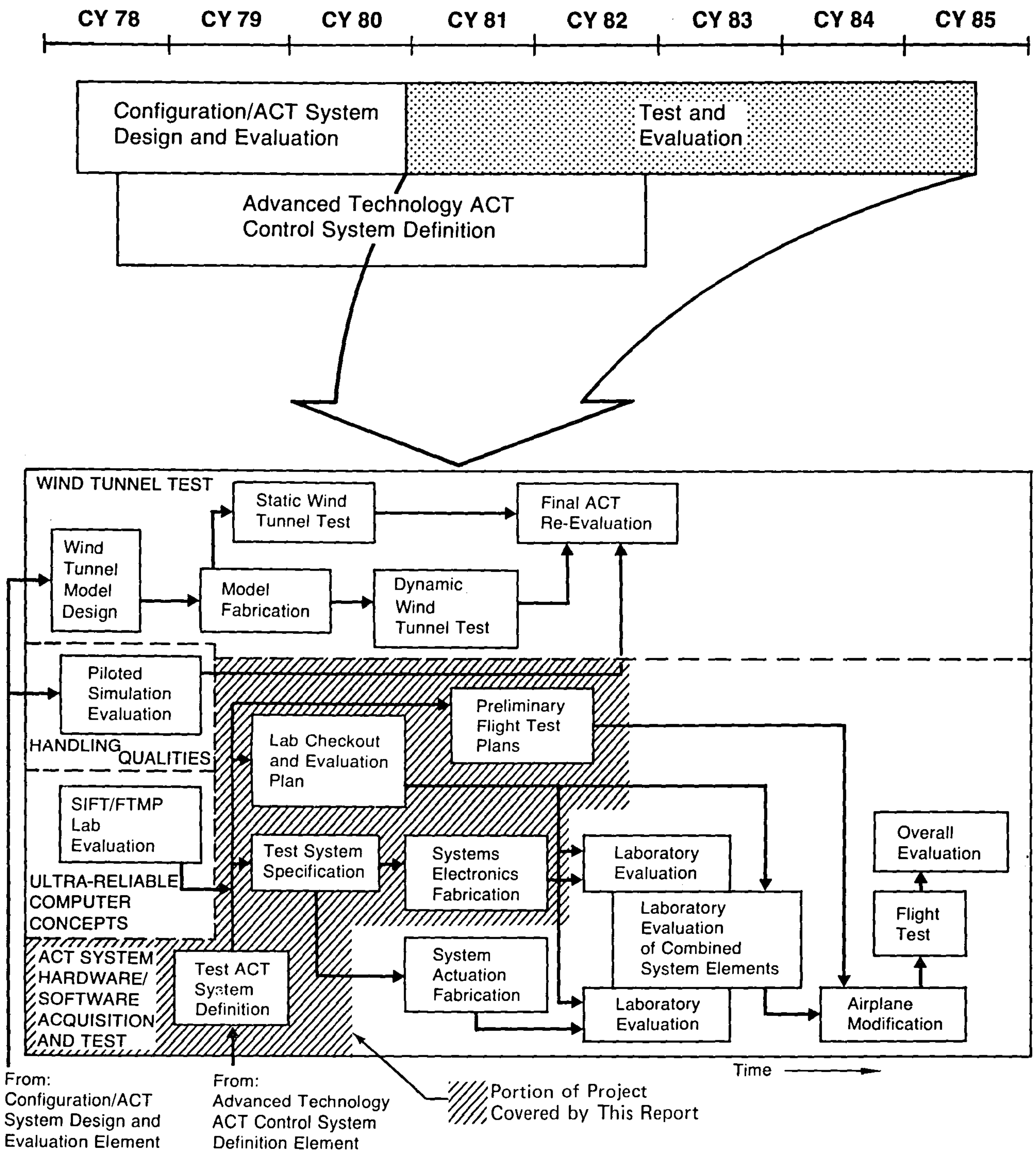

Figure 3. Test and Evaluation Element

A piloted simulation evaluation has been completed, the results of which are reported in Reference 9. The work covered by this report is shown cross-hatched in the figure and includes selection of a test airplane and system concept, design and fabrication of the Test ACT System electronic elements, and initial laboratory and flight test planning.

2.2 TEST ACT TASK OVERVIEW

The approach to this element of the IAAC Project was to (1) develop the design requirements and objectives for a Production ACT System intended to be certifiable in the 1990s, (2) develop the system requirements from the design requirements and objectives, (3) identify the specific exceptions to the Production ACT System requirements that are necessitated by the guidelines in the following paragraph, (4) design and build a flightworthy active control system (Test ACT), with fly-by-wire (FBW) implementation of pitch axis manual control, and (5) conduct laboratory and flight tests of the Test ACT System, with the aim of resolving the technical risks associated with a commercial application of this technology.

The Test ACT System development proceeded under the following ground rules:

a. The Test ACT System shall implement:

- Pitch-augmented stability (PAS)
- Wing-load alleviation (WLA)
- Fly-by-wire primary pitch control (elevator)

b. The Test ACT System shall be designed for flight test in the Boeing-owned 757-200 (NA001).

c. No change shall be made to primary control surface actuators.

d. Capability for inflight reversion to mechanical pitch control shall be retained.

e. The Test ACT System electronics shall be installed in consoles so the equipment can be tested in the laboratory and then installed in the flight test airplane with minimum disassembly.

In the early stages of work on the Test ACT System, Boeing prepared a request for proposal on the electronic parts (computers and dedicated sensors) of the system and submitted it to the following:

- Bendix Corporation*
- General Electric Company*
- Honeywell Incorporated*
- Hydraulic Research
- Parker Bertea Aerospace
- Rockwell-Collins Air Transport Division*
- Sperry Flight Systems
- Teledyne Controls

* Submitted proposal.

That was followed by discussions with potential subcontractors, preparation of proposals, and selection of the subcontractor, Rockwell International Corporation, Collins Air Transport Division. Boeing retained responsibility for the overall system architecture, mechanical modifications, and additions to the test airplane; installation of the system in the test airplane; laboratory test; and flight test. The electronics system design and evaluation were shared between Boeing and the subcontractor. In the accomplishment of this work, Boeing and Collins have operated as an integrated team; this report describes the product of that team effort, with no special attention given to the division of responsibilities between the prime contractor and the subcontractor.

This report covers the design, development, and fabrication of the system, up to the start of laboratory testing at Boeing.

3.0 SYMBOLS AND ABBREVIATIONS

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3.0 SYMBOLS AND ABBREVIATIONS

3.1 DEFINITIONS

In this report certain common words are given special meaning that is not contained in their dictionary definitions. These singular usages are defined below. The first two are adjectives that categorize the flight safety implications of control functions.

Critical–any function whose loss can result in a potential hazard, avoidable by appropriate pilot action.

Crucial–any function whose loss can result in an immediate, unconditional flight safety hazard.

The following two terms are used here as names of function categories and of corresponding controllers and control elements, such as computers.

Essential–those functions and control elements that are crucial; they must be operating if safe flight is to continue.

Primary–those functions and control elements that are critical; their loss is not necessarily threatening to flight safety but would normally require revision to the flight plan.

The following definitions distinguish two similar control systems treated in this document.

Production ACT System–the Active Controls Technology (ACT) control system having the same functions as Test ACT and of such design and redundancy as to have predicted dispatch and inflight reliability that shall meet the IAAC requirements stated herein.

Test ACT System–the ACT System that is the subject of this report. Although its Primary sensor redundancy is not sufficient to meet the IAAC reliability standards, it is sufficiently reliable for flight test and evaluation of anticipated Production ACT problem areas.

3.2 ABBREVIATIONS

A	ampere
ac	alternating current
ACC	Active Controls Computer
ACL	accelerometer
ACT	Active Controls Technology
AED	ALGOL Extended for Design
AFDS	Autopilot/Flight Director System
ALGOL	algorithmic-oriented language
alt	altitude
A/P	autopilot
APP	approach
ARINC	Aeronautical Radio Incorporated
C	Celsius
CAPS	Collins Adaptive Processing System
CAS	Control Augmentation System; computed airspeed
CDR	Critical Design Review
CFT	column force transducer
cg	center of gravity
CPU	central processing unit
CRT	cathode ray tube
CRU	cruise
CSEU	Control System Electronics Unit
CY	calendar year
D/A	digital to analog
DADC	Digital Air Data Computer

DAFCL	Digital Avionics Flight Controls Laboratory
dB	decibel
dc	direct current
deg	degree of arc
DID	Design Implementation Document
DOF	degree of freedom
DRO	Design Requirements and Objectives
DTP	Detailed Test Procedure
EAS	equivalent airspeed
EHSV	electrohydraulic servovalve
EICAS	Engine Indication and Crew Alerting System
EMI	electromagnetic interference
ESS	Essential
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FBW	fly by wire
FCC	Flight Control Computer
FCTR	Flight Control Test Rig
f_m	mode frequency
FMEA	failure mode and effect analysis
FSEU	Flap/Slat Electronics Unit
FTMP	fault tolerant multiprocessor
FTP	Flight Test Programmer
FY	fiscal year
g	acceleration due to gravity
gal	gallon

GLA	gust-load alleviation
GW	gross weight
h	height
HLL	high-level language
H/W	hardware
IAAC	Integrated Application of Active Controls Technology to an Advanced Subsonic Transport Project
I/O	input/output
IRS	Inertial Reference System
IRU	Inertial Reference Unit
K	gain
kn	knot
kN	kilonewton
lbf	pound-force
LRU	line replaceable unit
LVDT	linear variable differential transformer
M	Mach
MAC	mean aerodynamic chord
MCU	modular control unit (ARINC dimension specification)
MEL	minimum equipment list
MLC	maneuver-load control
M_{MO}	maximum operating Mach number
ms	millisecond
MTBF	mean time between failures
n	load factor
N	newton
Pa	pascal

PACS	Pitch Augmentation Control System
PAS	pitch-augmented stability
PCU	power control unit
PDR	Preliminary Design Review
PFTP	Preflight Test Panel
Q	pitch rate
Q_{SS}	steady-state pitch rate
q_c	calibrated impact pressure
RAM	random-access memory
RFCSHL	Renton Flight Control Systems Hydromechanical Laboratory
RFSC	Renton Flight Simulation Center
rms	root mean square
RSS	reduced static stability
RVDT	rotary variable differential transformer
SAM	Stabilizer Trim/Elevator Asymmetry Limit Module (part of CSEU)
SAT	System Acceptance Test
SCD	Specification Control Drawing
SIFT	software-implemented fault tolerance
SIMCON	simulation console
SPM	Stabilizer Position Module
SSFD	signal selection and fault detection
STCM	Stabilizer Trim Control Module
S/W	software
TAC	Test ACT Console
TACP	Test ACT Control Panel
TED	trailing edge down

TEU	trailing edge up
UAT	Unit Acceptance Tests
V_C	calibrated airspeed; computed airspeed
V_{GND}	speed when on ground
V_{MO}	maximum operating airspeed
V_S	stall speed
V_T	true airspeed
WLA	wing-load alleviation
WSI	work station interface

3.3 SYMBOLS

δ_E	elevator deflection
δ_{EC}	elevator command
δ_S	stabilizer deflection
Δ	change in quantity
ζ	root-mean-square turbulence intensity
θ	pitch attitude
λ	failure rate
σ	real part
σ_{GUST}	rms isotropic gust level
τ	time constant
ϕ	roll angle
ω	imaginary part

4.0 REQUIREMENTS

4.0 REQUIREMENTS

Early in the Test ACT Program, system requirements were established to govern the engineering work. These included requirements of reliability and dispatchability and also limitations on what could be done to modify the proposed test airplane. This section is a brief summary of those requirements. More detailed requirements are contained in "ACT System Requirements" (ref. 10) and a Test ACT Specification Control Drawing. Section 4.1 of this document states requirements for a Production ACT System. Section 4.2 cites the waivers from Production ACT requirements that apply to Test ACT and lists appropriate Test ACT implementation requirements.

4.1 PRODUCTION ACT SYSTEM REQUIREMENTS

4.1.1 PITCH-AUGMENTED STABILITY

The prime objective of the Production ACT System is to enable an airplane to be flown with reduced or negative longitudinal static stability. The pitch-augmented stability (PAS) function of the Production ACT System shall enable flight with Level 1 flying qualities (fig. 4) throughout the flight envelope and design center-of-gravity range.

4.1.2 WING-LOAD ALLEVIATION

The wing-load alleviation (WLA) function consists of maneuver-load control (MLC) and gust-load alleviation (GLA). Maneuver-load control uses symmetrical deflection of outboard ailerons to shift wing loads in the inboard direction and thus reduce wing bending moments that result from loads generated during controlled maneuvers. Gust-load alleviation generates aileron deflections to reduce wing loads produced by atmospheric disturbances. The Production ACT System shall incorporate both forms of WLA.

4.1.3 PITCH FLY BY WIRE

Pitch axis fly by wire (FBW) shall be provided in the Production ACT System to enable a pilot to control the elevators to maximum positive and negative deflections, with no mechanical column-to-elevator coupling and comfortable column feel forces.

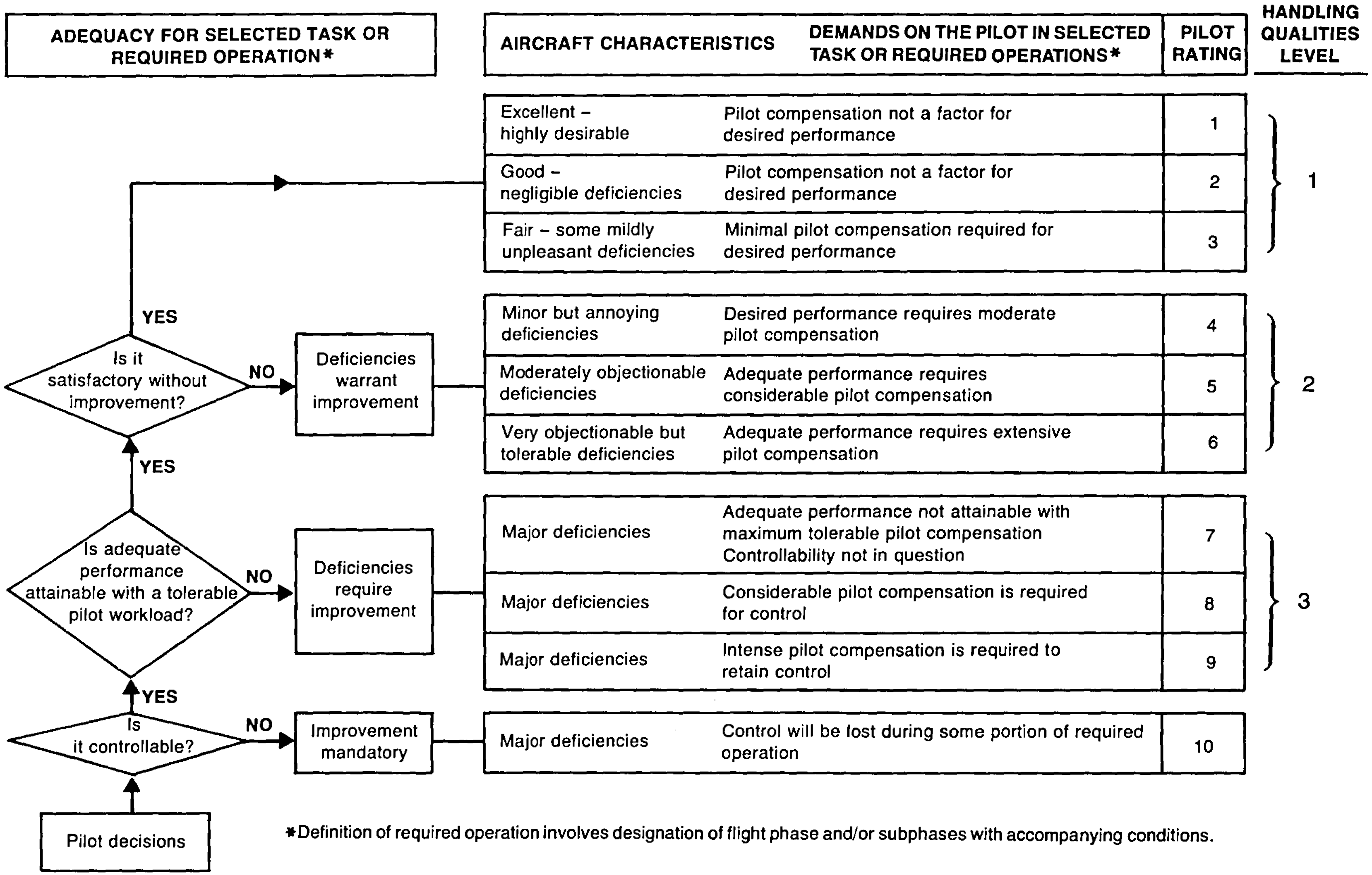


Figure 4. Revised Cooper-Harper Rating Scale, Annotated

4.1.4 RELIABILITY AND SAFETY

The Production ACT System shall be designed in accordance with the failure survival specifications of the Federal Aviation Administration (FAA) Advisory Circular 25.1309b (ref. 11) as follows:

- Any condition that can prevent the continued safe flight and landing of the airplane shall be extremely improbable. This is interpreted to be loss of the Essential System. Probability of such a condition will be shown by analysis to be less than 10^{-9} during a 1-hr flight.

- The occurrence of any other failure condition that can reduce the capability of the airplane or the ability of the crew to cope with adverse operating conditions shall be improbable. This is interpreted to be loss of Primary PAS. Such a probability will be shown by analysis to be less than 10^{-5} during a 1-hr flight.

- No single Production ACT System failure shall preclude continued safe flight and landing.

4.1.5 SCHEDULE RELIABILITY

Schedule reliability is defined as the probability of starting and completing a scheduled revenue flight without an interruption chargeable to an aircraft system or component primary malfunction (not secondary or consequential) involving cancellations, air turnbacks, diverted landings, and delays greater than 15 min. Dispatch reliability includes only cancellations and delays greater than 15 min.

The Production ACT System shall meet the following schedule reliability requirement:

- Schedule interruptions caused by ACT equipment shall not exceed 65 per 100 000 departures.

4.2 TEST ACT SYSTEM REQUIREMENTS

The overriding requirements applying to these systems are cited in Section 4.1.4. The extreme reliability required of a full authority elevator controller gives rise to subordinate requirements, some of which are stated in the following paragraphs as they apply to the Test ACT System. The complete statement of requirements for Test ACT is contained in the System Requirements document (ref.10) and several procurement specifications. Some of the more fundamental requirement items are given in the following paragraphs.

4.2.1 SYSTEM DESIGN REQUIREMENTS

The system design applicable to the Test ACT System shall be such that the performance requirements of Section 4.1 can be met. The system requirements shall include, but not be limited to, the following:

- The control computers shall provide the system monitoring, redundancy management, and reconfiguration required to meet the performance specifications of Section 4.1.

- The system shall have an automated preflight test capability that determines, in less than 3 min, the dispatch status of Test ACT and indicates it to the crew.

- Preflight test coverage shall be sufficient to meet the inflight reliability and safety requirements stated in Section 4.1.

- System faults detected by automatic tests and monitors shall be automatically stored in system memory and readily recalled by maintenance personnel.

- The system shall incorporate control and display panels enabling the flight crew to:

    - Exert necessary control over the operation and testing of the system

    - Monitor system status and all adjustments caused by faults and automatic reconfiguration

    - Make preplanned changes to the system for flight test investigations

- The system shall be designed such that a generic software error cannot result in a hardover elevator command.

- The electronics shall be contained in two consoles, each limited to the following dimensions:

  - Height 1.3m (53 in.)
  - Length 1.1m (42 in.)
  - Depth 0.8m (32 in.)

  The maximum weight shall be 181 kg (400 lb).

4.2.2 MECHANICAL DESIGN REQUIREMENTS

The mechanical backup control requirement and the problems of installing Test ACT into an existing airplane create special electrical and mechanical implementation requirements. Those of the mechanical category follow:

- Both mechanical and FBW controls shall be available during flight.

- The design shall be such that a single disconnect in the elevator linkage will not disable more than one elevator.

- The maximum hysteresis, deadband, and linearity between commanded and measured elevator positions shall be in accordance with Figure 5.

- The feel system parameters and tolerances shall be designed per Figure 6. Requirements for force transducers are defined in References 2 and 3.

- The FBW control column shall be mass balanced so that the column force resulting from inertial forces generated by accelerated flight within the design envelope will not exceed 13.3N (3.0 lb). The column and its feel system shall be damped to avoid column overshoot.

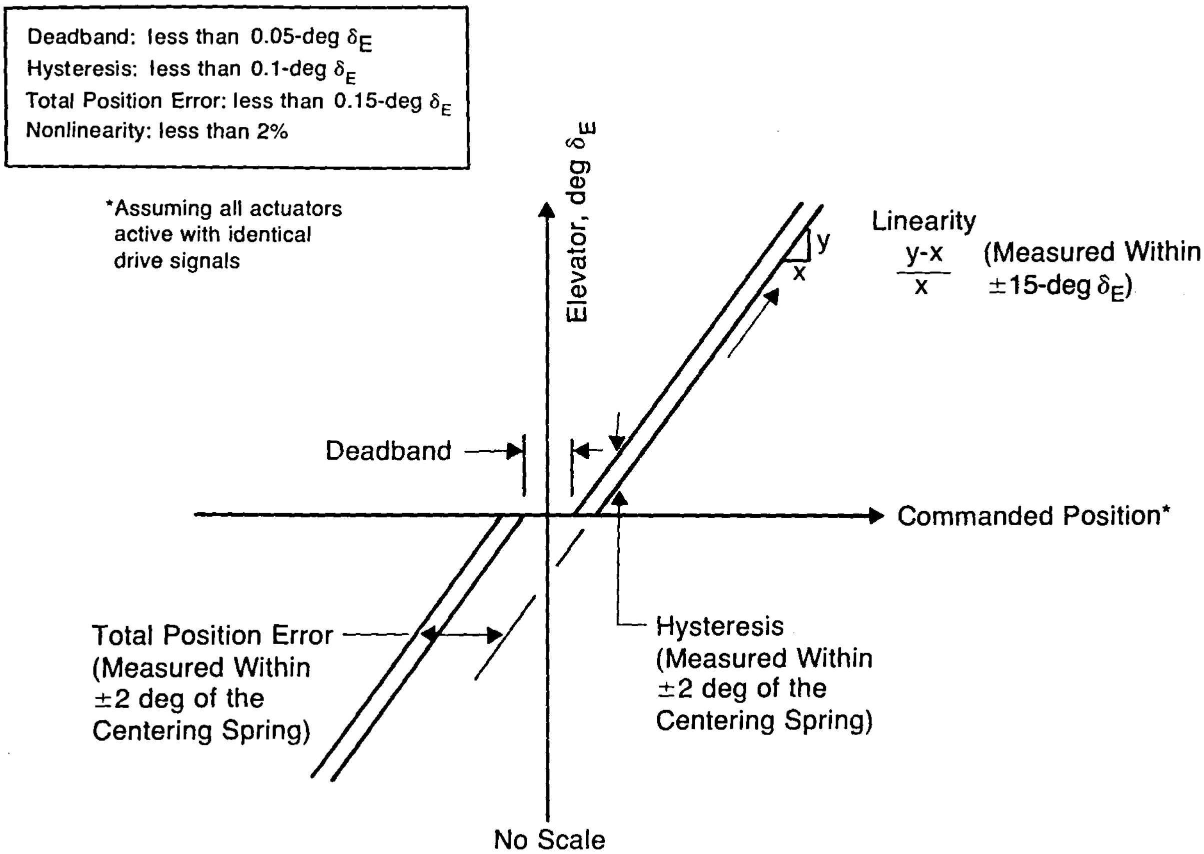


Figure 5. Hysteresis, Deadband, and Linearity Requirements–FBW Mechanical Path

Implementation of PAS and pitch FBW in accord with the above statements necessitates redundant elevator secondary servoactuators. Requirements applying to those servoactuators include the following:

- The actuators have the authority to drive the power control units (PCU) and the elevators to 30-deg trailing edge up and 20-deg trailing edge down.

- Maximum no-load surface rate is at least 55 deg/sec. Electrohydraulic valve characteristics are the same as those of the current 757 rollout guidance actuator, modified as required for increased flow.

- Torque output of each actuator is such that no more than two actuators are required to handle all anticipated loads, including jam override capability.

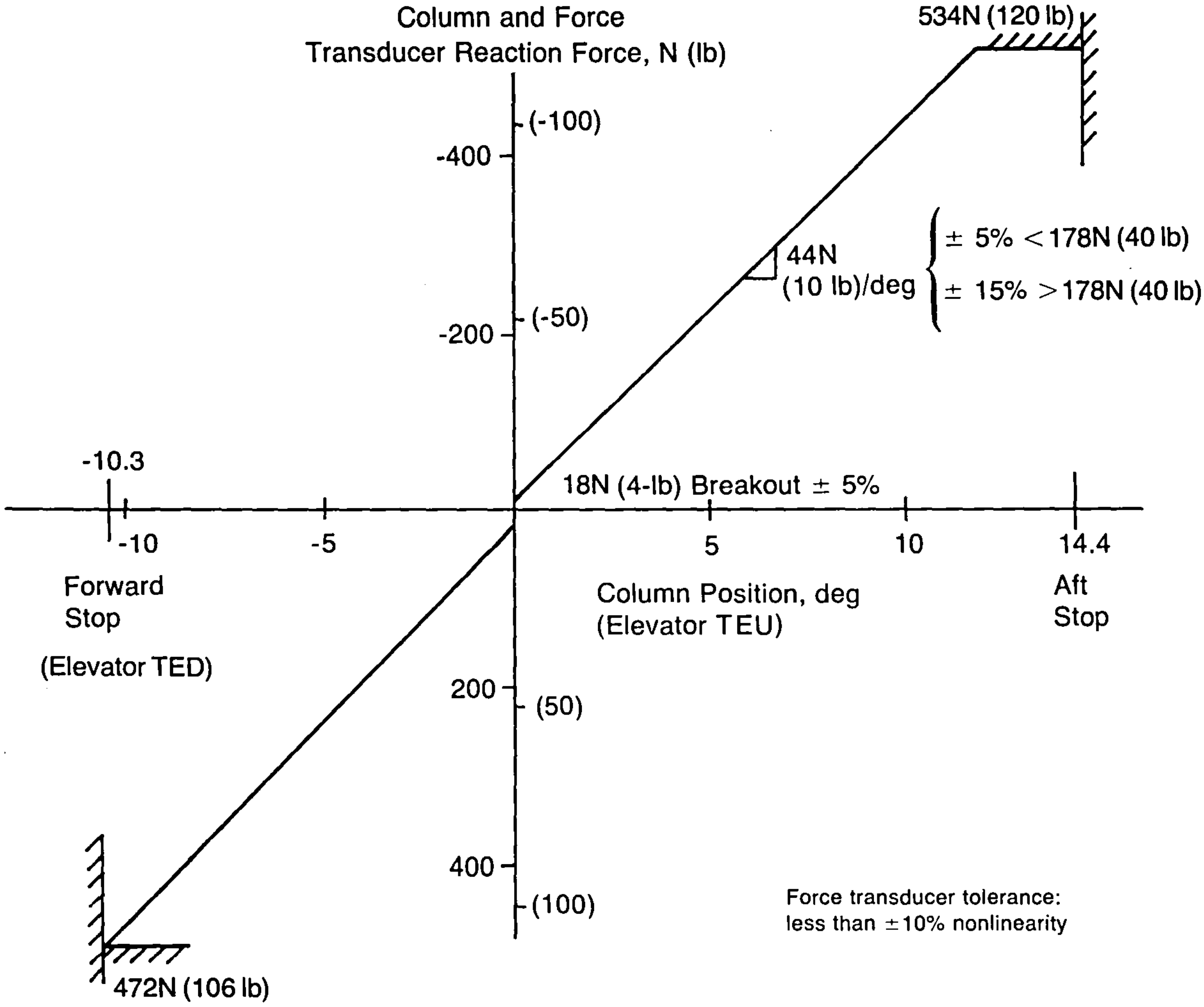


Figure 6. Control Column Feel System Requirements

- When connected in a control loop with a loop gain of 30 rad/sec, the secondary actuator meets the stability, frequency response, null shift, and hysteresis requirements of the existing 757 rollout guidance actuator.

Requirements are not stated for aileron secondary servoactuators. Only electronic WLA commands will be evaluated during flight testing; i.e., actuator commands will be monitored but not routed to control surfaces.

4.2.3 ELECTRONIC DESIGN REQUIREMENTS

The design and fabrication requirements of the electronic parts were imposed to ensure adequate reliability. These requirements are as follows:

- Materials and processes employed in the design of the equipment shall be consistent with those approved for the 757/767 Autopilot/Flight Director System.

- The design of the equipment shall be adequate to ensure operation within the design limits during and/or after exposure to altitude, temperature, vibration, and g loads applicable to 757/767 airplanes.

- The equipment shall operate properly when supplied with power of 757/767 standards.

- The computer power supply shall operate from either or both of two 28V dc power sources. The power supply shall provide seven dc-regulated voltages for logic circuits and two semiregulated single-phase 400-Hz ac sources of power required to excite system sensors.

- The power supply shall sustain all loads during zero power input from both sources for a period of 50 ms. Current sensing of each source to the power supply shall ensure both a limit on initial inrush of current and a current balance between the two sources.

- The system wiring shall be designed to minimize susceptibility to electromagnetic interference.

- Connector and wiring separation between channels and circuit segregation on the printed circuit boards shall be retained to prevent a short from resulting in the failure of the system to perform its design functions.

- The software design procedures shall be consistent with 707/727/737 software standards.

- The software shall be verified by functional test, functional walkthrough, and module inspection.

- The control computers shall have the speed and capability required to perform in real time all calculations required to implement simultaneously the control laws, monitors, and redundancy management functions.

5.0 SYSTEM ARCHITECTURE

Page

5.0 SYSTEM ARCHITECTURE

This section identifies the main features of the Test ACT system architecture and relates their choice to the critical issues faced in the system design.

The system mechanizes flight-crucial pitch axis stability augmentation and fly-by-wire (FBW) longitudinal control; hence the pitch control system must have a probability of total function loss of less than 10^{-9} in a flight of 1-hr duration. It also mechanizes wing-load alleviation (WLA), speed stability augmentation, and elevator offload functions requiring a function loss rate less than 10^{-5}. These facts yield the critical issues of the system architecture and result in the features described below.

5.1 ARCHITECTURE ISSUES

In the course of work by the Boeing-Collins team toward selection of the system architecture, the following items were identified as issues of prime importance:

- What redundancy management plan, system elements, and interfaces will serve to achieve a probability of function loss less than 1×10^{-9} in a 1-hr flight?

- What redundancy level is required to preserve airline schedule reliability?

- What system architecture will minimize susceptibility to generic hardware and software faults?

- What monitors can be allowed to shut down a channel of crucial function control?

- Assuming a two-level system, composed of Primary and Essential computer sets:

  - Is switching between levels allowable?
  - In which level is preflight test performed?
  - Are both levels full authority?
  - How should the Essential part be implemented:

- Digital or analog?
- Cross compared or "brick walled"?
- Sensors dedicated or shared with the other level?

- Can gain variation be allowed in the Essential set?

- Should digital computer operation be synchronous or asynchronous?

- Should preflight test be automatic or manual?

- What are the system voting planes?

These critical issues are discussed later in this section. A number of lesser issues arose in the system design process, including these:

- Should redundant Primary system sensors be connected one-for-one with the computers, or "cross strapped"?

- Shall the system when "down moded" (reduced in redundancy by apparent component fault) be allowed to "up mode" if component recovery occurs?

- How much redundancy is needed in the servoactuator shutdown function?

- Must there be an Essential channel oscillatory failure monitor?

- What servoactuator tests are needed?

- What down-mode strategy is employed?

- How is preflight test of the analog channels accomplished?

These and other design decisions are addressed in Section 5.1.

5.1.1 RESOLUTION OF CRITICAL ISSUES

Functional Partitioning—The challenge of the 10^{-9} function loss has been a key feature of the IAAC Project since its inception. Early in the project, it was determined that an integrated active control system, using a single set of redundant digital computers serving all functions, was better than a segregated one using a separate computer set for each function. The digital computer served admirably to combine in one machine the diverse operations of control law computation, self-test, monitoring, voting, and redundancy management. On the other hand, the computer's complexity raised serious doubts that it could be made to meet the specified reliability. In light of that, the principle of partitioning by criticality of functions was applied and has been retained in all project phases, including Test ACT. Since fly by wire is a beneficial function with the same criticality as that of pitch augmentation, it was added as a functional requirement; but the less critical WLA and speed stability augmentation are kept separate from the Essential pitch control system. This functionally partitioned approach ensures that a less critical function cannot compromise the safety of a crucial function and thereby reduces risk. Functional partitioning therefore was an early architectural decision.

Digital Primary—For the Primary system, the versatility and computation power of the digital computer were judged necessary. The interface with digital state sensors (air data and inertial) is easier and more reliable with digital computation. Also, redundancy management and built-in test requirements make digital capability almost mandatory. This left open the question of what was to serve as the ultrareliable backup system, now called the Essential system, for the functions requiring the less than 10^{-9} probability of function loss.

Backup System—The requirement for crucial function reliability is a probability of loss of function less than 10^{-9} in a 1-hr flight. Based on the projected reliability of components such as sensors, computers, and actuators, an Essential system that could survive two similar failures (fail op^2) was required. The next decision was that the fail op^2 fault tolerance requirement would be met with a four-channel (quadruple) system. While a three-channel system can be fail op^2 if sufficient inline monitoring is provided, that introduces the major risk of uncertainty in providing and proving coverage for the second fault. The quadruple concept avoids this risk by detecting all faults by either voting or comparison monitoring. This approach is widely accepted as providing essentially 100% coverage. The penalty of this approach is the cost of the fourth channel and supporting systems such as electric and hydraulic power.

Protection Against Generic Faults—A major issue to be addressed was the generic fault problem. How do you prove to the required extreme confidence level that a software error or latent fault cannot result in failure of all redundant channels? Solutions to this problem take two forms: (1) fault avoidance by vigorous design and test and (2) fault tolerance. The next major architectural decision was that Test ACT would be designed to tolerate generic faults. The basic way to survive a generic error is to provide dissimilar redundancy.

Several quadruple, dissimilar, standby computer concepts were developed using both digital and analog computers for the standby system. They used the basic principles of generic fault detection by reasonableness or "red line" monitoring (ref. 12) and also generic fault isolation by switching control from the failed digital computer set to a dissimilar system providing "get home" capability. Since the airplane may be longitudinally unstable, it was assumed that the pilots could not perform the switching operation; it would have to be computer controlled.

This concept was examined in depth. Aside from the obvious implementation complexity and some concern over generic faults such as lightning that could disable all digital computers, the primary technical issue emerged as a lack of confidence in the switching function. For example, what sort of monitor can, with extreme reliability, detect generic faults? And how do you know that the standby system is working when it is used only under extremely rare conditions?

The Collins proposal addressed the switching dilemma with a concept based on the following premises. First, simple analog systems can be designed and tested to be immune to generic faults. Second, the "simple" analog computer (supported with dedicated, high-reliability, feedforward and feedback sensors) can do all crucial functions operating continuously and provide at least "get home" flying qualities. The plan uses digital Primary computers with additional feedback variables and gain scheduling to provide additive control commands, yielding very good handling qualities for normal conditions.

Digital command authority is limited by the analog computer such that even a hardover generic digital fault is fail-safe (failure transients kept below passenger injury or structural damage levels). The digital computers also perform most of the test and monitor functions.

Essential System Monitors—The concept just described is founded upon simplicity of the analog Essential channel, a low channel failure rate (MTBF more than 10 000 hr), and careful avoidance of any path by which failure of one channel can contaminate another channel. Cross-channel communication was found to be incompatible with these objectives; the Essential system has to be "brick walled"; i.e., the channels are isolated from one another through all electric paths. This requires that all monitoring is done in line rather than cross channel; and all input voting must be performed individually in each channel. This leads to the questions of what monitors in the system are "executive" (i.e., able to shut down an Essential channel) and where are the voting planes of the system in general.

The monitor philosophy is based on the same rationale that led to the choice of brick-walled analog Essential control computers; namely any executive monitor must be simple, analog, and in line. Thus the executive monitors are all in either the power supplies or in the Essential computers and comprise the power monitor, the LVDT common mode monitors, the rate gyro spin monitor, and the servoactuator detent monitor.

Voting Planes—The choice of voting planes was aimed at promoting reliability by minimizing the impact of failed components. Inputs to the Primary digital system are obtained from autonomous input/output circuits and communicated cross channel such that a failed central processor does not deny its sensor input signal to other channels. Primary elevator commands are voted in the Essential analog computer, as are the three flap position discrete inputs, so that the brick-walled analog channels all see the same values for those inputs. The ultimate voting plane is the secondary servoactuator summing shaft—the final system output. Here the four Essential channel signals are force voted. The detent of a failed channel actuator is overcome by the other three actuators in the first failure case, or by the other two in case of a second failure. If the detent offset exceeds 2.5-deg equivalent elevator deflection for 3 sec, the servoactuator detent monitor disengages the failed channel.

Primary Redundancy—Earlier IAAC studies indicated that the critical function probability of loss rate less than 10^{-5} in a 1-hr flight can be met with a three-channel Primary system architecture. That requires that all three channels are available for dispatch, a stipulation which would affect schedule reliability. The preferred solution, given that four electric power supplies are already needed for the Essential system, is a quadruple Primary system that may be dispatched with one channel down.

Preflight Test—Meeting the prescribed reliability levels of the Primary and Essential systems requires a certain known availability level of redundant channels at takeoff. The Primary system must have three of the four channels available, and the Essential system all four. Knowing this requires a thorough preflight test before each departure. How should preflight test be performed? If it is automatic, where is its control lodged?

Early IAAC system studies yielded these conclusions:

- Preflight test should not add significantly to the time now required for the cockpit preparation routines of a commercial airliner.

- It should add little or nothing to the work required of the pilot.

Corollary conclusions were that preflight test should require no more than 2-min elapsed time and should be automatic to the maximum practical degree.

In proceeding with detailed specification of Test ACT preflight test, the design team observed that a small measure of flight crew participation is needed. The crew should initiate the test so that it may be done at a convenient and appropriate time; and, since a complete checkout includes control surface motion, the crew must ascertain airplane ground clearance before that part of the test occurs. These factors led to the design decision to automate preflight in two stages, each initiated by a pilot. The first is an all-electronic part, called "Passive"; the second, involving control surface motions, is called "Active" and calls for a pilot's application of fore and aft forces on the control column. In all other respects, preflight test is conducted automatically; assuming the pilots respond promptly to indications requesting "Start Active," "Pull Column," and "Push Column," the complete test requires less than 2 min.

The preflight test sequence is of sufficient complexity such that it could not be done in a simple analog computer; it must be controlled in the Primary system. This requires special care in testing Essential analog functions to preserve the "brick walled" character of Essential channels. This is achieved by allowing the crucial functions of an analog channel to be interrogated by only one Primary.

Essential Gain Variation—From the point of view of preserving simplicity of the Essential computer, it is clearly desirable that it operate in all flight modes at constant gain. It soon becomes evident that the Primary elevator command limit of 2.5 deg, chosen to limit hardover response to 1g normal acceleration in 3 sec under cruise condition, would not allow the system to produce Level 1 handling qualities at low speed. This made two limit levels, and a high-reliability means of switching between them, necessary in the Essential system. The switching requirement was met by means of voting among three available "flaps down" discrete signals, and using the voted discrete to change the limit from 7 deg at low airspeed to 2.5 deg at high airspeed and vice versa. Given this reliable speed change signal, it was practical to solve the low-speed/high-speed problem in the Essential control laws by switching gain in the feedforward and feedback paths while keeping the low-gain (high-speed) line continuous in both circuits. In the transition to low speed, the flap discrete brings in additional gain.

Asynchronous Primary Computers—Asynchronous operation of the redundant digital computers was chosen for this application. Low bandwidth inputs and high sampling rates minimize the time offset disadvantage of asynchronous operation. The wide fault threshold normally characteristic of asynchronous operation is not needed here because the digital signals are not isolated by comparison monitoring. Asynchronous operation also avoids the synchronizer as a possible single-point failure source. The option of synchronized computer operation for future tests was retained by designing and including a synchronization circuit card in each computer box.

5.1.2 SUMMARY

Architecture merits that resulted from the evolution described previously are as follows:

- The system can survive a worst case generic digital fault.

- The system can tolerate any two similar failures with no reduction in performance. (Because of the voting planes, even greater tolerance for dissimilar failures is provided.)

- The high-reliability analog portion of the system is always "on line" and requires no switching.

- Digital computers are used for their unique control law and redundancy management capability.

- The concept provides very high fault isolation coverage.

In summary, the selected system is hybrid and combines the safety of simple analog computers with the performance and versatility of digital computers.

5.2 TEST ACT CONFIGURATION

The Test ACT System Configuration is shown in Figure 7. Test ACT is separated by a heavy dashed line from airplane equipment with which it interfaces. The redundancy limitations of the Primary system, described in Section 4.2, are shown in the boxes representing 757 sensors at the left and the trim system at the bottom of the figure. Note that the Essential sensors for column force and dedicated pitch rate are quadruple. (The term "dedicated" distinguishes the quadruple pitch-rate gyro inputs from the lower reliability Primary pitch-rate signals coming from triple Inertial Reference Systems (IRS)).

The general arrangement of Figure 7 is conventional for control systems; i.e., in general sensors are situated at the left, the computers in the center, and the servoactuators at the right side. An exception is the placement of the dedicated pitch-rate sensors in the

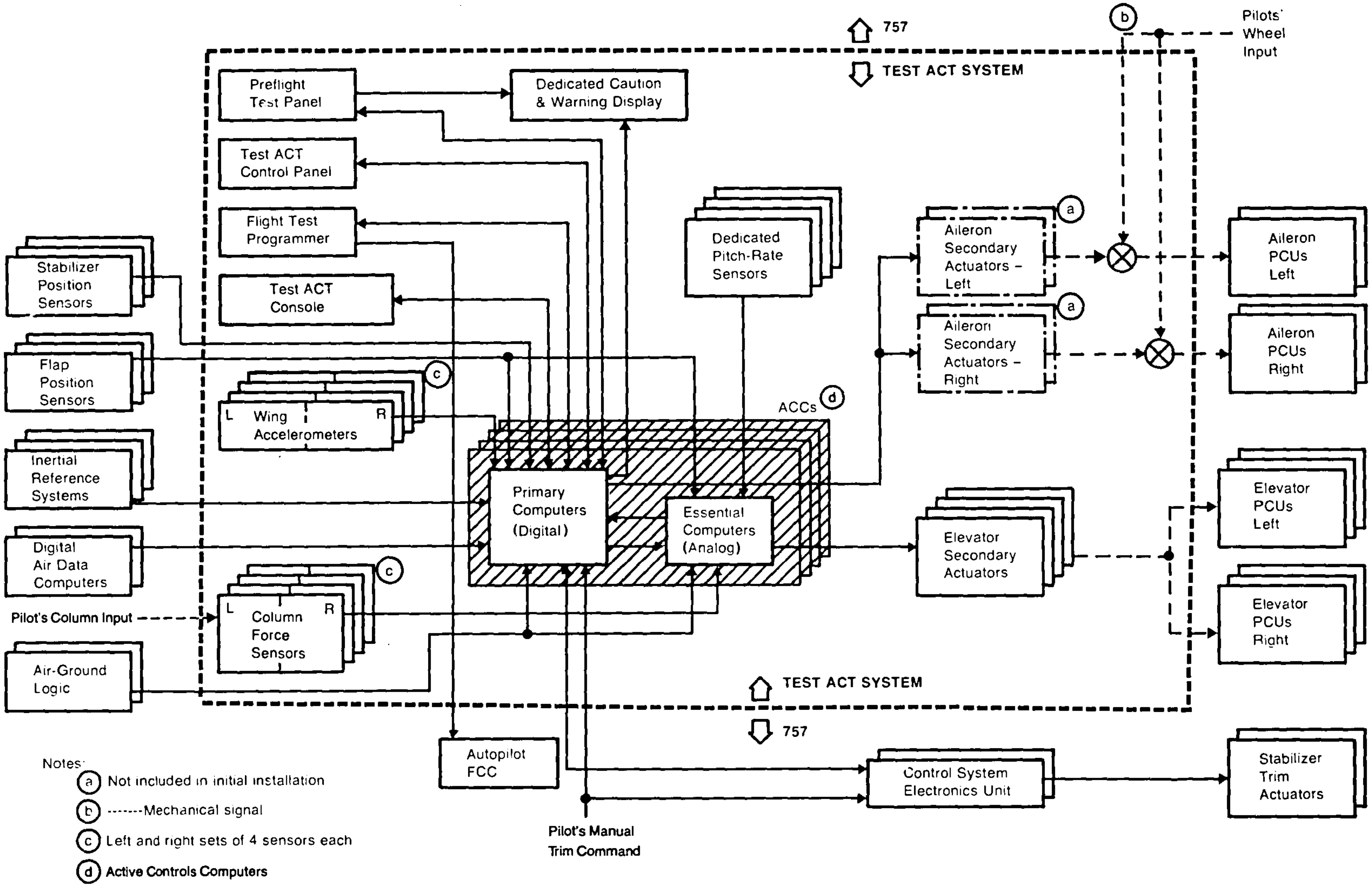


Figure 7. Test ACT System Block Diagram

middle of the Test ACT System part of the diagram, just above the Essential analog computers. The autopilot Flight Control Computer (FCC) and the Control System Electronics Unit at the bottom of the diagram are both control computers that are part of the existing test airplane equipment. Control and display panels and the system console are in the Test ACT System box at the upper left. For the sake of diagram clarity, all redundant connections, whether dual, triple, or quadruple, are represented by single lines.

5.2.1 ESSENTIAL SYSTEM

The part of the Test ACT System that must perform with extremely high reliability, as discussed in Section 5.0, is the Essential system, comprising the column force sensors, dedicated pitch-rate sensors, Essential analog computers, and elevator secondary actuators. This is the quadruple, simple, brick-walled, high-reliability system that always operates to provide acceptable airplane handling characteristics in the pitch axis, regardless of center-of-gravity location. The FBW function is generated by the column force sensors and a simple, dual-gain feedforward control law in the analog computers coupled to the elevator servos. Short-period pitch stability augmentation is provided by the pitch-rate gyros and a simple dual-gain feedback control law. Both of those control functions are available for safe flight if the entire Primary system fails. In normal operation those controls are supplemented by Primary system commands to provide Level 1 flying qualities in pitch. By keeping the Essential system very simple and free of elaborate gain schedules and reconfiguration provisions, the estimated reliability meets the requirement quoted in Section 4.1.4.

5.2.2 PRIMARY SYSTEM

The Primary system utilizes the airplane sensors shown at the left in Figure 7 plus the wing accelerometers, the quadruple Primary digital computers, and the airplane trim system to perform the functions of speed stability augmentation, elevator offload, and wing-load alleviation. As shown in the diagram, the wing-load alleviation function is carried through computation of the servo command, which is monitored and made part of the redundancy management process in the computer; but no aileron secondary servos are installed and hence there is no airplane response to that function. Since autonomous input processing is provided to preserve sensor redundancy in the event of Primary central processor failure, the sensor-to-computer coupling plan is fundamentally one-for-one.

The test airplane's shortage of Primary sensors requires that limited cross-strapping be employed to serve each computer with direct coupling to a complete sensor set; for example, the center IRS is connected to two Primary computers.

The Primary system also contributes to short-period pitch augmentation and the fly-by-wire feedforward control law; for the latter it requires column force sensor input. For the Primary computers, that column force signal is supplied from the Essential computers; this is done for reliability reasons. Column force is a buffered output from the Essential computers to the Primary, so that a catastrophic failure in the Primary input system cannot affect the Essential column force input. That scheme also saves one set of demodulators.

The Primary computer is a minicomputer derived from the Collins FCC 701, the Autopilot/Flight Director System computer for the Boeing 757 and 767 airplanes. Its high throughput and memory capacity enable it to be programmed for these diverse functions:

- Control laws for the active control and fly-by-wire functions listed above
- Primary system redundancy management and reconfiguration control
- Preflight test of the complete Test ACT system, including the Essential channels
- Self-test and self-monitor functions
- Sensor signal selection and failure detection
- Flight crew communication and control via three flight deck panels
- Simulated maintenance interface via the Test ACT Console

Redundancy management design issues include such questions as the down-mode strategy for sequential failures in sensor sets; the choice made for Test ACT is "4-3-2-0"; i.e., no operation on a sensor signal is allowed if only a single valid input is available. This relates also to the question of whether or not sensor upmode is allowed; the answer is a qualified "yes," requiring a renewed valid signal from both inline monitor and comparison monitor.

Preflight test control is another large assignment for the Primary computer. Since the ultimate voting plane is the detent vote at the actuator force summing shaft, proper detent operation is crucial and a "soft detent" test must be a part of preflight. This requires that three Primary computers disengage their respective Essential servoactuators while the fourth Primary tests its associated actuator detent. Two solenoid valves in series provide redundant capability for servo shutdown. The detent comparator includes a means of Essential channel oscillatory failure detection.

The "Test ACT Console" block in Figure 7 represents the Test ACT System structure. In the laboratory, the console is the housing and mounting for all of the system except the servos; in the test airplane it carries all of the system except the servos, the system sensors, and the three flight deck panels. This arrangement minimizes the interconnection changes needed in the move from laboratory to airplane.

The connection from the Flight Test Programmer to the autopilot FCC is a discrete signal to (1) disable the cruise autopilot computer when Test ACT is operating and (2) engage the autopilot actuator as a detent in the Test ACT series summing linkage. That signal passes through the ACT STATUS switch, the element by which the pilots can promptly disable Test ACT and revert to mechanical control of the pitch axis. In such case, the single-channel cruise autopilot also becomes available. Because Test ACT employs two autopilot servo positions in the test airplane, the multiple-channel autoland autopilot is not available.

A single line is employed in Figure 7 to indicate one of the system's most important features; i.e., when Test ACT is operating the elevator secondary actuators are <u>always</u> under the control of the analog Essential computers.

5.3 OPERATION MODES

The operational modes sequence (fig. 8) represents normal operation of a production system. A few seconds after electric power is applied to the system, the automatic power-up test sequence is completed, the "NO GO" status indicator on the preflight test panel is illuminated, and the "PRESS PASSIVE" message appears to the crew on the Engine Indication and Crew Alerting System (EICAS). A pilot then initiates the passive preflight test by pushing the "PASSIVE" button on the preflight test panel. (The "PASSIVE" term refers to the fact that there is no control surface motion during this first portion of preflight test.) The PASSIVE test sequence, requiring less than 30 sec, is described in detail in Section 6.3.1.2. In the absence of faults, the end of this sequence is signified by the appearance of an illuminated "ARM" indication in the pushbutton for the active test

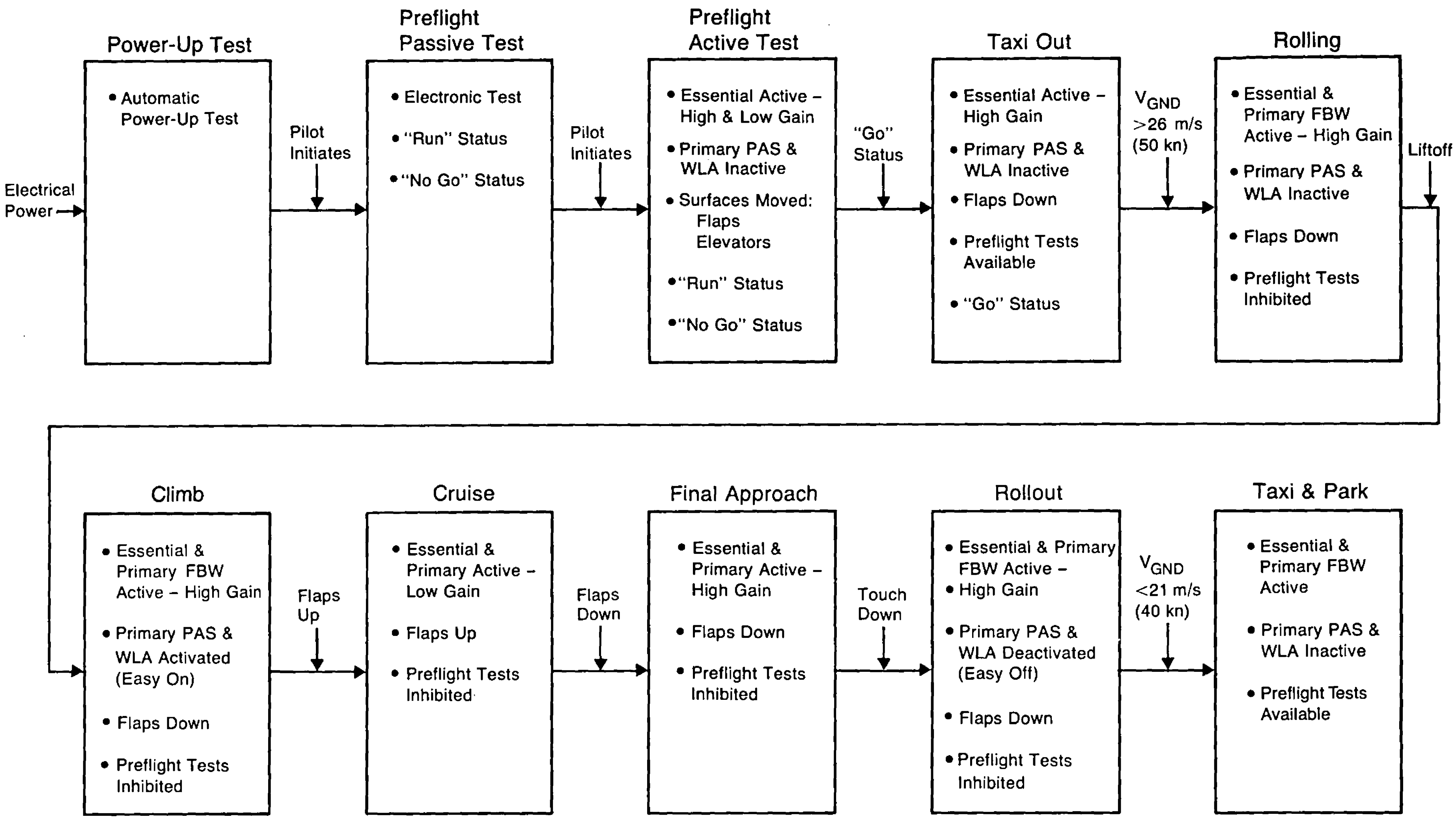


Figure 8. Test ACT System Operational Modes

sequence on the Preflight Test Panel. When the pilots have verified airplane ground clearance, they can initiate the active phase of preflight test involving the motion of control surfaces. This phase, also described in Section 6.3.1.2, requires a pilot to pull and push the control column in response to appropriate crew prompt messages displayed upon EICAS. Assuming no failure has been encountered, this test phase ends with the appearance of the green "GO" indication on the Preflight Test Panel. The complete preflight test series requires less than 2 min.

The system now is in takeoff status; the Essential system is in operation, but Primary PAS and WLA are inactive to ensure that they do not cause undesired response to accelerations encountered in ground taxi and rolling. Since the flaps are down in a takeoff position, the Essential system is on its high gain setting. During the takeoff roll, when airspeed exceeds 26 m/sec (50 kn), preflight test is locked out. At liftoff, Primary PAS and WLA are activated through "EASY ON" circuits such that they do not inject a step signal into the control surface actuators. During climb, when the flaps are fully retracted, the Essential and Primary systems are switched to the low gain setting; this status will be maintained until the approach to landing when the airplane is again at low speed, flying with flaps down. As the airplane touches down in the landing phase, Primary PAS and WLA are deactivated, again to prevent undesirable control surface motions in response to ground accelerations. As the airplane slows down to 21 m/sec (40 kn), preflight test becomes available so that the system can immediately be prepared for the next flight leg.

In the early stages of Test ACT flight test, takeoff and landing will be done with positive pitch stability and with the system deactivated. In such cases, preflight test will be run; then prior to takeoff, the servos will be disengaged by means of the ACT STATUS switch on the Flight Test Programmer (sec. 6.3.1.3).

5.4 CONTROL LAWS

The control laws implemented in both digital Primary and analog Essential computers are shown in Figures 9, 10, and 11. They are tailored to the test airplane and to the operations plan shown in Figure 8. Since the Primary computer is digital and the Essential computer analog, the column force detent is implemented in both forms and shown in Figure 9 in both machines. Analog-to-digital and digital-to-analog converters are not included in these diagrams. The "Easy On/Off" blocks shown are provided to ensure that

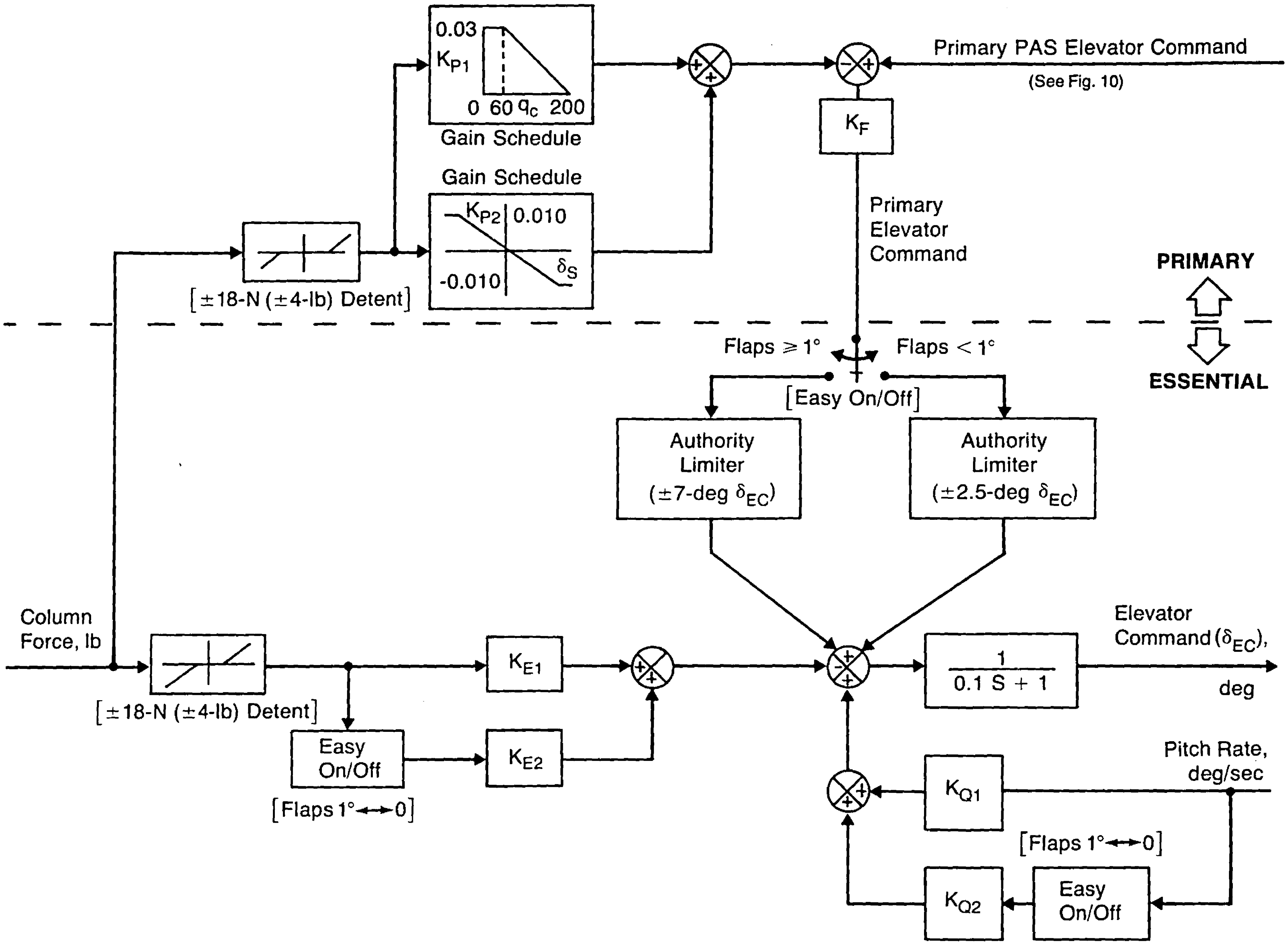


Figure 9. Essential PAS and FBW Control Law for the Test ACT System

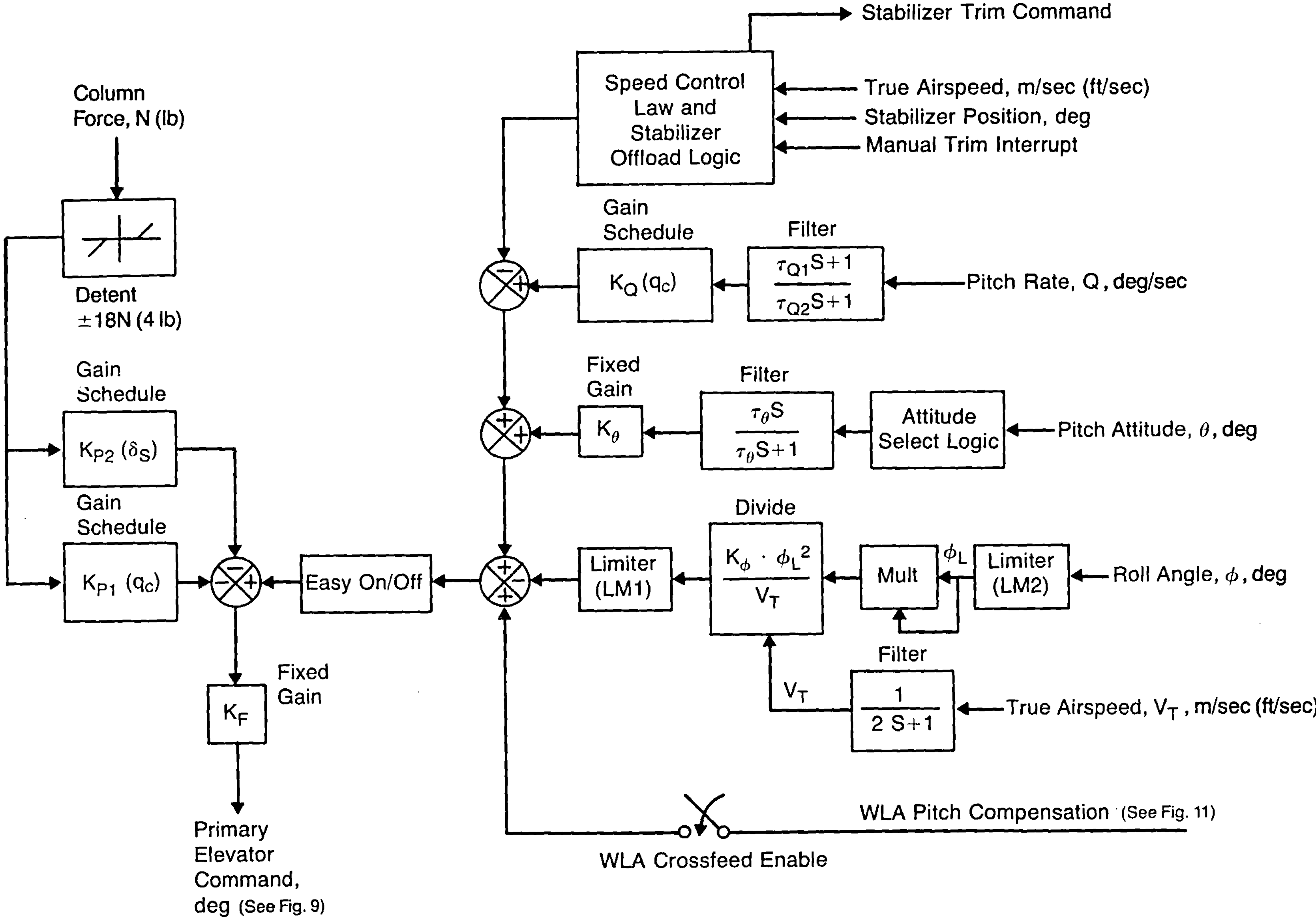
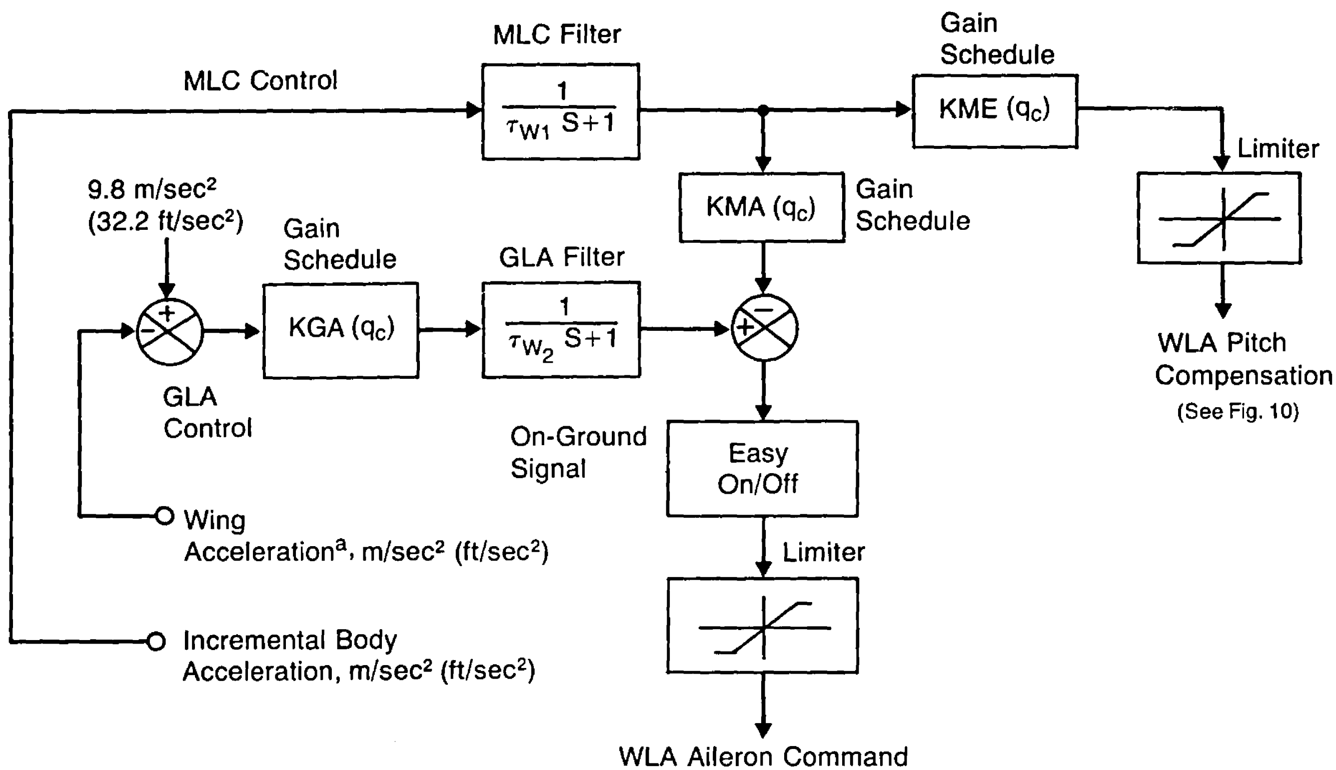


Figure 10. Primary Pitch-Augmentation System Block Diagram



[a]Average of left and right wing accelerometers

Figure 11. Wing-Load Alleviation Control Law

discrete switching functions occurring at operation mode changes such as lift-off, flaps up, and flaps down do not introduce step changes in elevator command.

Control law synthesis is described in Section 10.2.

5.5 HARDWARE CONFIGURATION

5.5.1 ELECTRONIC HARDWARE

The Test ACT System electronic hardware is shown in Figure 12. It includes all of the Collins-supplied elements of the actual control system except the wing accelerometers. On top of the computers in that photograph are three control and display panels that will be installed in the test airplane flight deck. Figure 13 provides a closer view of these panels. The Flight Test Programmer (FTP), occupying the center panel in that figure, is needed in the Test ACT System only; the Test ACT Control Panel (TACP) and the Preflight Test Panel (PFTP) would have counterparts in the Production ACT System installation.

Figure 14 shows the Test ACT Console (TAC), which houses the control system electronics, plus the equipment for controlling and communicating with the system in laboratory and flight test operations. The TAC is shown in its laboratory test configuration; the column force sensors are mounted on top of console No. 2, with a lever for force application. The three flight deck panels are installed at the upper right in console No. 1. The Active Controls Computers (ACC) occupy the left half of console No. 2, and the balance of the equipment provides means of instrumenting system conditions, loading and reading software, simulating faults, controlling power supplies, and conducting other test operations.

5.5.2 MECHANICAL HARDWARE

Mechanical modifications to the airplane affect two subsections of the pitch control system, located under the flight deck floor and aft of the pressurized cabin. Those modifications have been designed; they are described and illustrated in Section 8.0 and Appendix D.

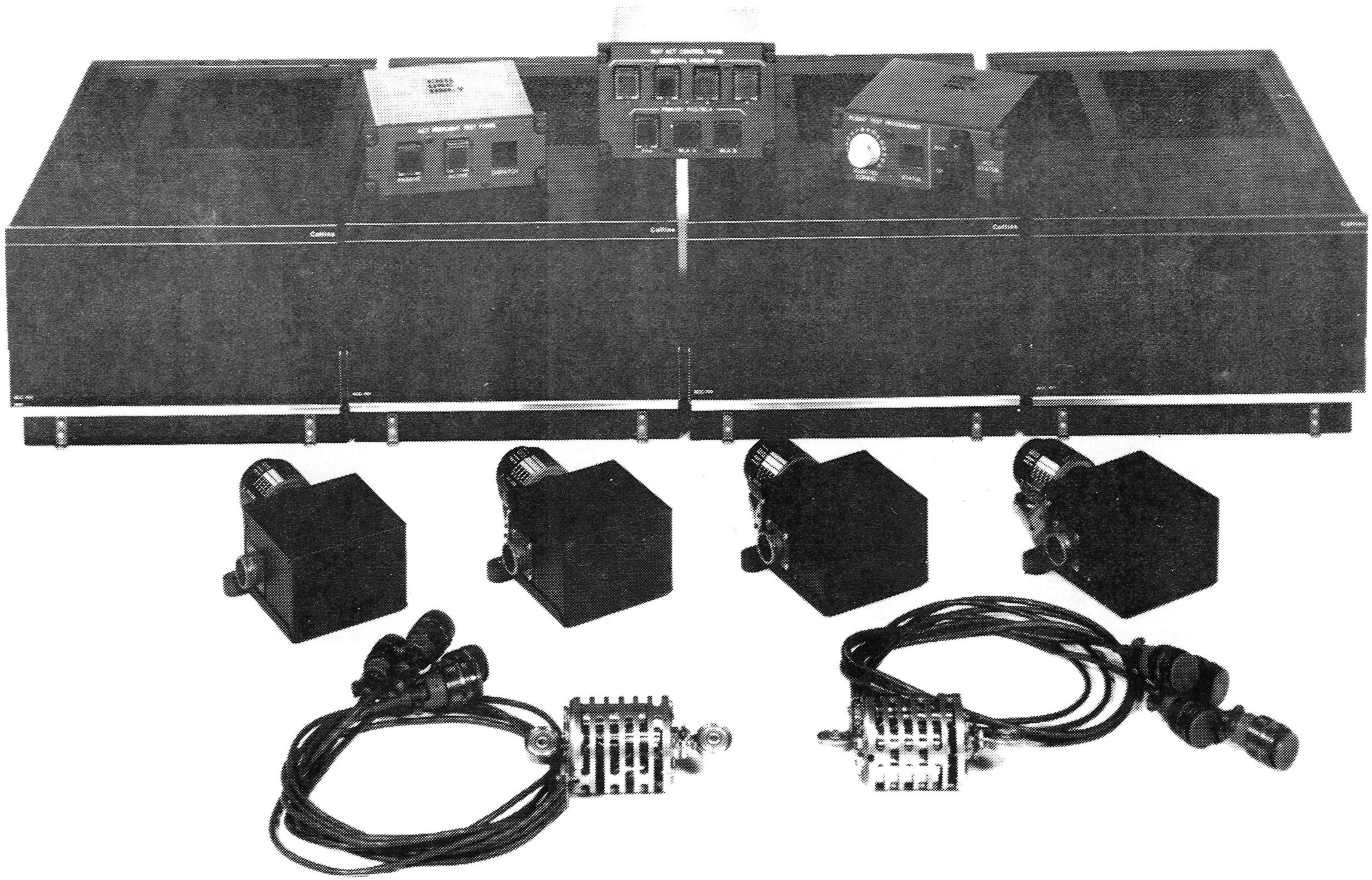


Figure 12. Test ACT Electronic Hardware

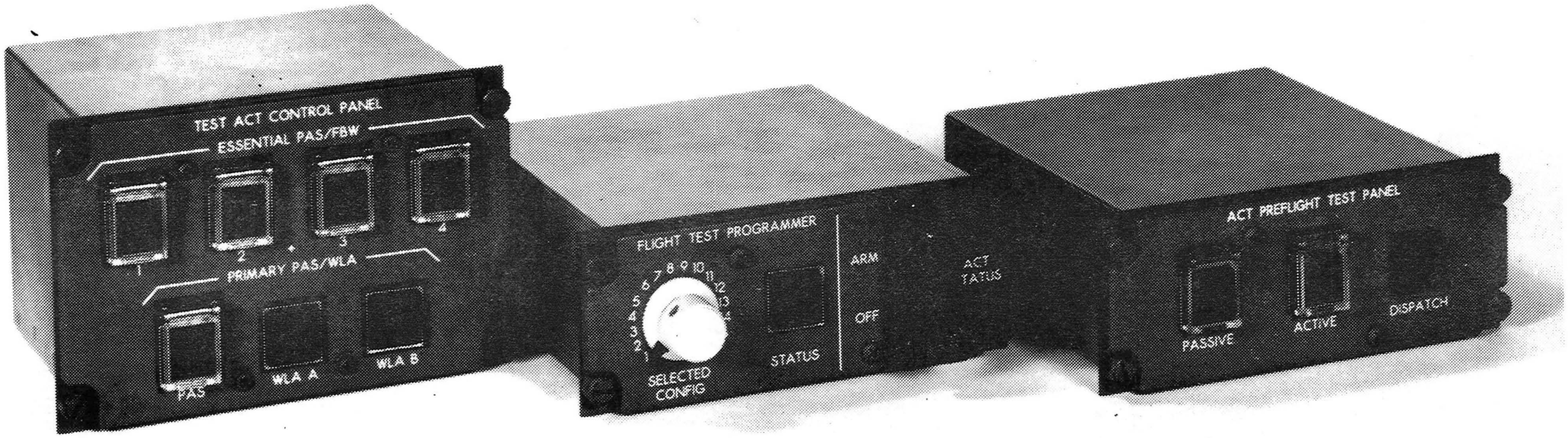


Figure 13. Flight Deck Panels

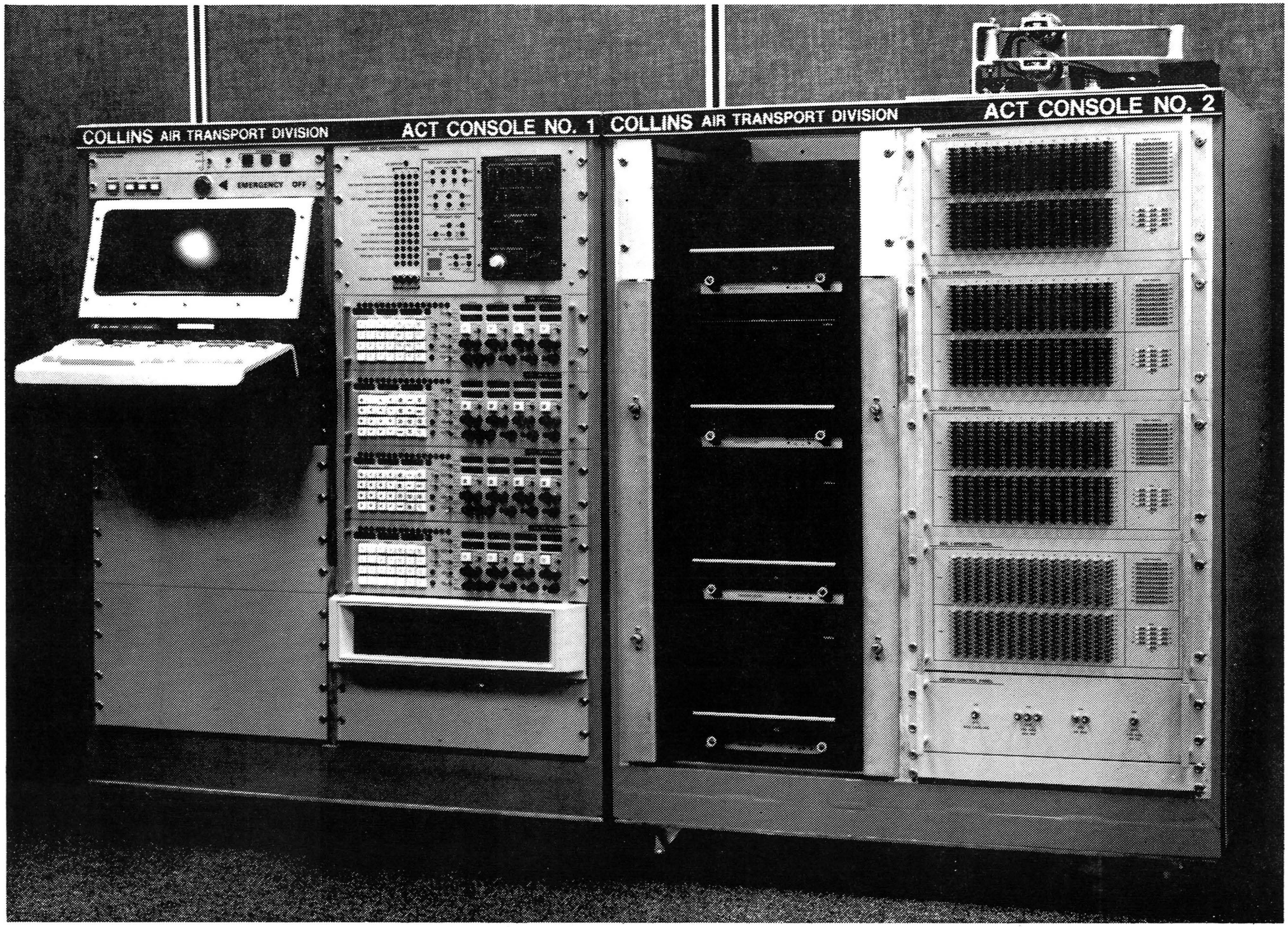


Figure 14. Test ACT Console

6.0 SYSTEM DESIGN: ELECTRONIC HARDWARE

Page

6.0 SYSTEM DESIGN: ELECTRONIC HARDWARE

Figure 7, the block diagram of the Test ACT System as a whole, is a good aid in recognizing the interrelationship of the diverse elements described in the following paragraphs.

6.1 COMPUTERS

The Test ACT System employs two control elements: the Primary digital computer and the Essential analog computer. These two computers are housed in one chassis to make use of existing equipment and to avoid the added cost and development time of a new chassis. The 757/767 Flight Control Computer (FCC-701) forms the framework, but the delivered unit is called an Active Controls Computer (ACC), as shown in Figure 15.

The ACC represents a high-technology design that provides for a flexible implementation of the Test ACT requirements. It is partitioned into 19 circuit cards, 3 interconnect cards, and a power supply packaged in an 8-MCU assembly, as shown in Figure 16. The hardware is divided into functional elements to meet the requirements of safety segregation, hardware tasks, and module testability.

Circuit segregation, a very important consideration in the hardware design, is accomplished by use of card boundaries and component isolation within the card. Major functional elements are located on individual cards, providing, when feasible, a natural separation. Where there are several redundant functions per card, components are physically segregated as much as possible.

6.1.1 PRIMARY COMPUTER

Figure 17 shows the partitioning of the Primary and Essential computers in the ACC. The Primary computer is designed around a dual-bus architecture, consisting of a high-speed computer bus (transfer bus) and a dedicated input/output (I/O) bus.

The I/O bus is managed by the I/O controller, which handles the transactions for serial input/output data, discrete inputs, and analog inputs. Five circuit cards interface with this bus: the I/O data path card, the I/O control card (both of which make up the I/O

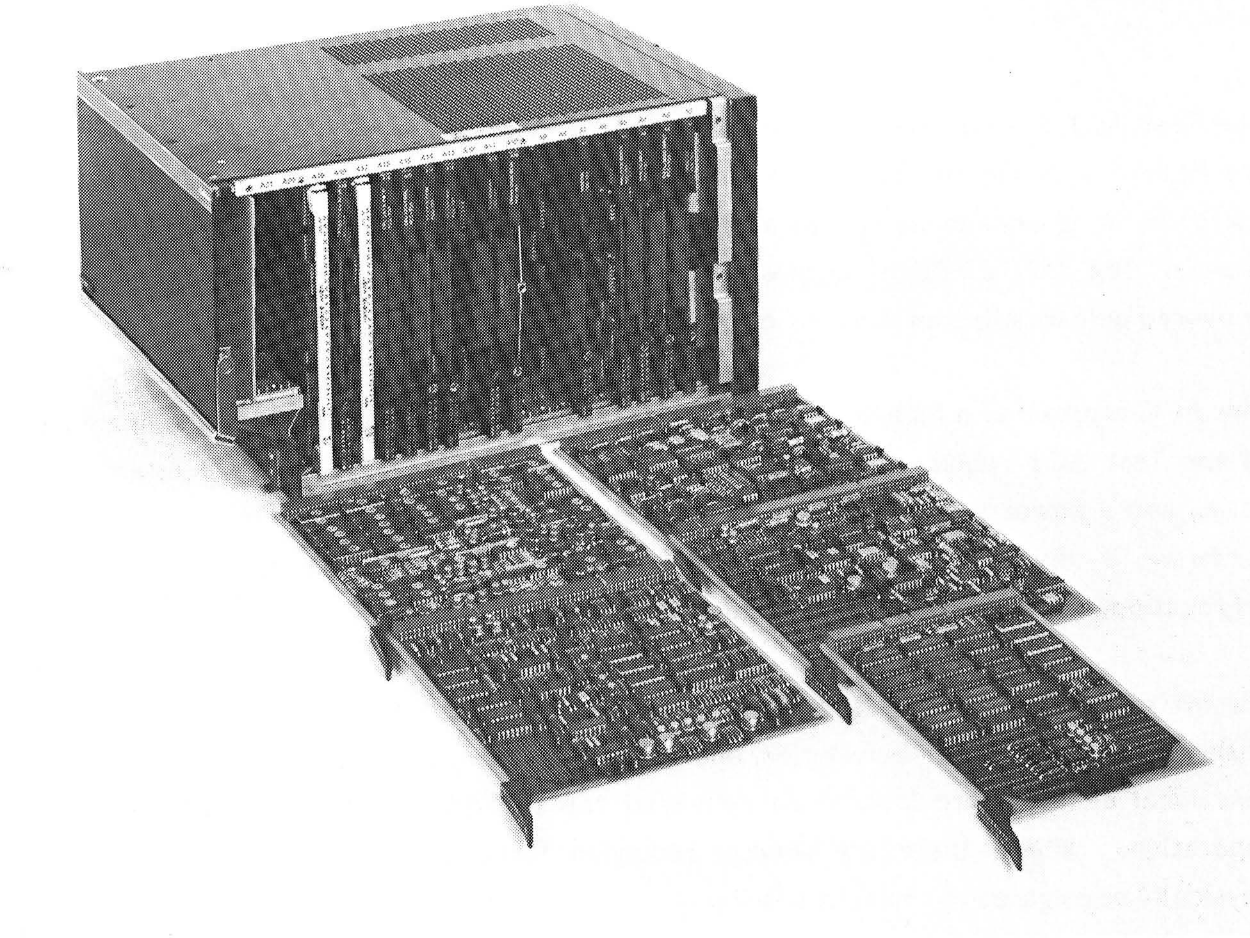


Figure 15. Active Controls Computer – Cards A6, A8, A9, A12, and A20 Removed

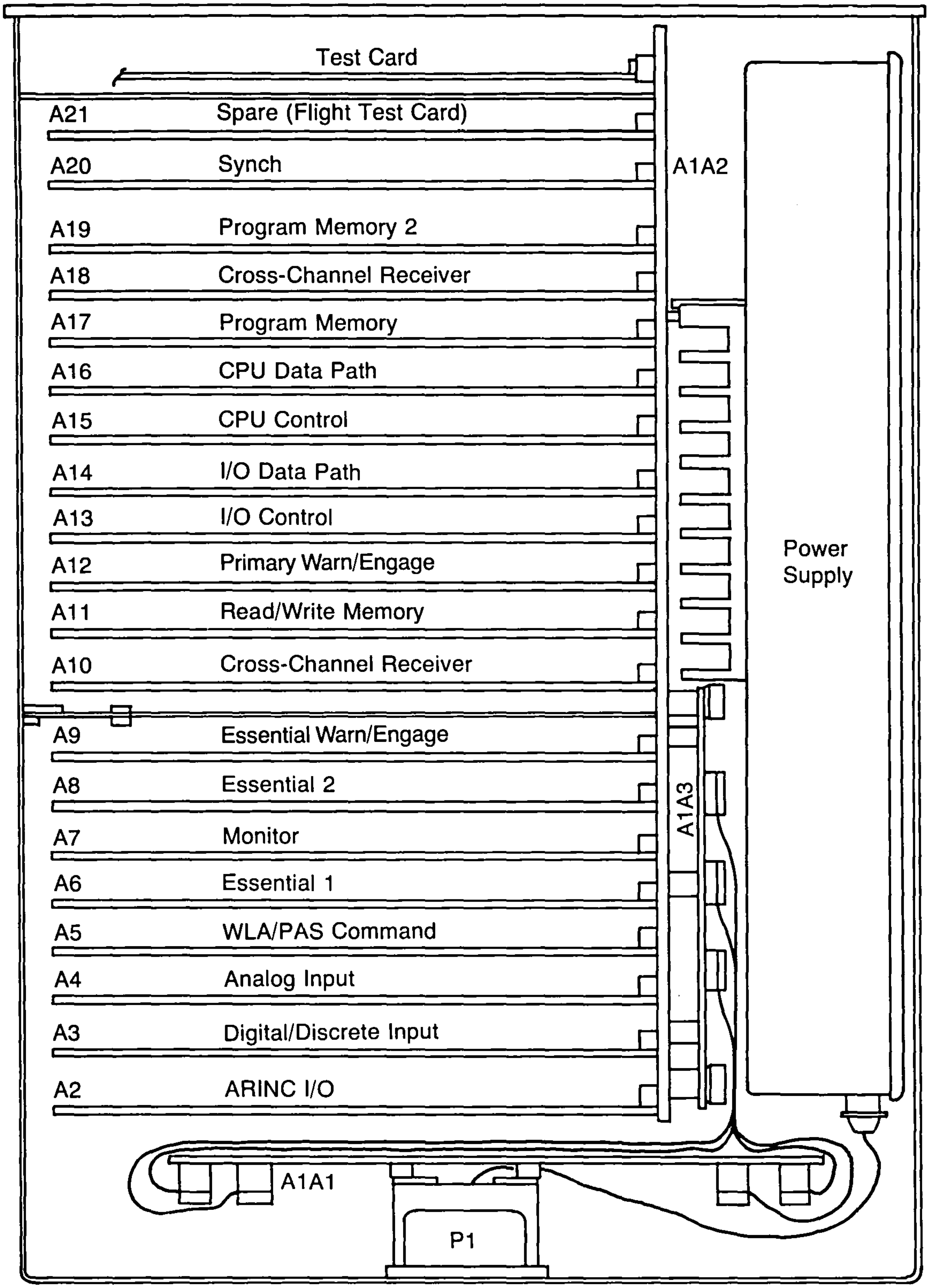


Figure 16. ACC Circuit Cards Placement–Top View

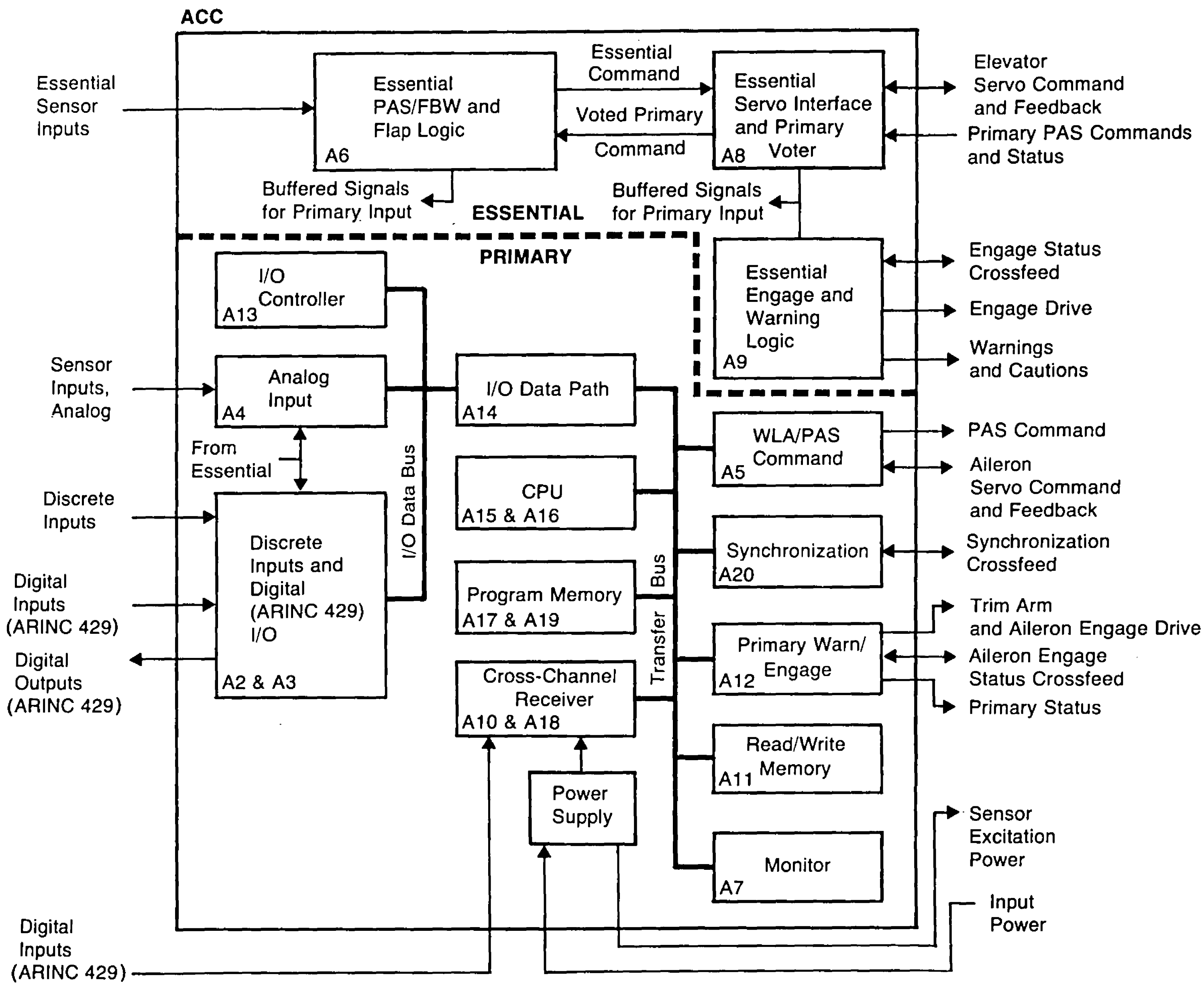


Figure 17. ACC Block Diagram–Partitioning Primary and Essential Functions

controller), the analog input card, the digital/discrete input card, and the Aeronautical Radio Incorporated (ARINC) 429 I/O card. Transactions that take place via the I/O bus are independent of the central processing unit (CPU).

However, the CPU, which consists of control and data path circuit cards, manages devices that interface with the transfer bus. Ten peripheral cards are tied to this bus and are under control of the CPU. They provide the functions of:

- Program memory
- Read-write memory
- Servo interfaces
- Computer timing
- Wing-load alleviation (WLA) engage arm and trim logic
- Pitch-augmented stability (PAS) and WLA caution and warn outputs
- System monitoring
- Cross-channel data reception
- Synchronization
- Bus termination

The basic architecture of the Primary computer is the same as that of the FCC-701. This is very important to the Test ACT Program in that it provides the low risk of a known, working digital system. It also allows the use of existing development tools, such as the Collins Adaptive Processing System (CAPS) Test Adapters, from the Autopilot/Flight Director System (AFDS) Program. Additional expense is avoided by using existing software tools, such as the ALGOL Extended for Design (AED) compiler and link editor, to change the AED source code to the CAPS relocatable object code.

6.1.2 ESSENTIAL COMPUTER

The Essential analog portion of the ACC has two main functions: computation of the Essential control laws and the warning and engage logic. As shown in Figure 17, the Essential control law functions are distributed between two cards. Essential No. 1 (card A6) contains the pitch-rate and column force input filtering and control law implementation, including flap position gain switching. Essential No. 2 contains the Primary command signal selection and the servo amplifier electronics. ACC card A9 contains the Essential warning and engage electronics.

The Essential control laws are implemented with analog circuitry. Certain gains and filters have been identified as candidates for change during the laboratory test and flight test program phases. For these, the mounting of the resistors and capacitors has been arranged to allow replacement of the components without damage to the printed circuit boards.

6.1.3 POWER SUPPLY

Early in the development of the ACCs, it was determined that the FCC power supply, with its single ac power source, did not have sufficient reliability to serve as the ACC power supply. In order to meet the Test ACT reliability requirements, dual airplane 28-V dc power sources for the power supply module were needed. This power supply module is designed to perform satisfactorily with either dc source unavailable.

The power supply module, supported by the aircraft power system, supplies all voltage necessary to power the ACC and to provide excitation and/or power for the system sensors of the Test ACT System.

Because of space limitations in the FCC chassis, it was not possible to incorporate independent power supplies for the Primary and Essential computers. The resulting single supply module was designed to serve both and still fulfill the requirements imposed on an independent Essential supply, which are:

- To provide regulated power to the computers
- To provide 26-V ac, 400-Hz sensor excitation
- To provide 115-V ac, 400-Hz power for the rate gyros

The FCC power supply, fed by 115V ac, 400 Hz, has two major sections: a primary ac-to-dc power converter and a secondary stepdown, dc-to-dc voltage regulator. For the ACC-regulated power requirement, the primary section was replaced by two step-up, dc-to-dc power converters; the secondary section remained the same. To fulfill the sensor excitation requirement, a new dc-to-ac power converter design was required.

Because power supplies of the "switching regulator" type are inherently noisy in terms of electromagnetic radiation, the power supply module design incorporates shielding and electromagnetic interference (EMI) filters to alleviate the problem. The completed design was subjected to Collins' standard qualification testing (based on Radio Technical Committee for Aeronautics document, DO-160, ref. 13) to ensure that radiation did not exceed acceptable limits. The design was also tested for electromagnetic susceptibility over the ambient temperature range and was found to be acceptable.

Avionics systems typically monitor the power supply and use power status information in engage, annunciation, and other decision logic. To aid in evaluation of system response to Primary and/or Essential power failures, the "power valid" signals for the Primary and Essential computers have been kept independent.

6.2 SENSORS

6.2.1 SYSTEM SENSORS

The Test ACT System employs redundant system sensors, shown inside the dashed box of Figure 7, as well as airplane sensors (sec. 6.2.2) already installed in the test airplane. The system sensors serving the Essential computers are pitch-rate gyros and control column force transducers; those for the Primary computers are wing accelerometers.

A rate gyro generates an analog output proportional to the angular rate of change about a specified axis. In the Test ACT application, the pitch-rate gyro measures the airplane angular rate about the lateral axis and provides a two-wire analog output used as feedback in the Essential PAS short-period stabilization loop.

The control column transducers issue an ac voltage proportional to the applied force used in the Essential fly-by-wire (FBW) function.

The accelerometers produce an analog output proportional to the acceleration along the axis that is perpendicular to the mounting surface. In Test ACT, the accelerometer measures the acceleration of the outboard wings normal to the reference plane and provides an input to the gust-load alleviation (GLA) function control law.

6.2.1.1 Pitch-Rate Gyros

In the Test ACT System, the pitch-rate gyro is a combination unit composed of a miniature rate gyro attached to a mounting base that contains electronics for self-monitoring and preflight test. This assembly is shown in Figure 18. The rate gyros are Smith's Industries 719S-9 air-bearing gyros. Their predicted mean time between failures (MTBF) is greater than 20 000 operating hours and has been confirmed in service.

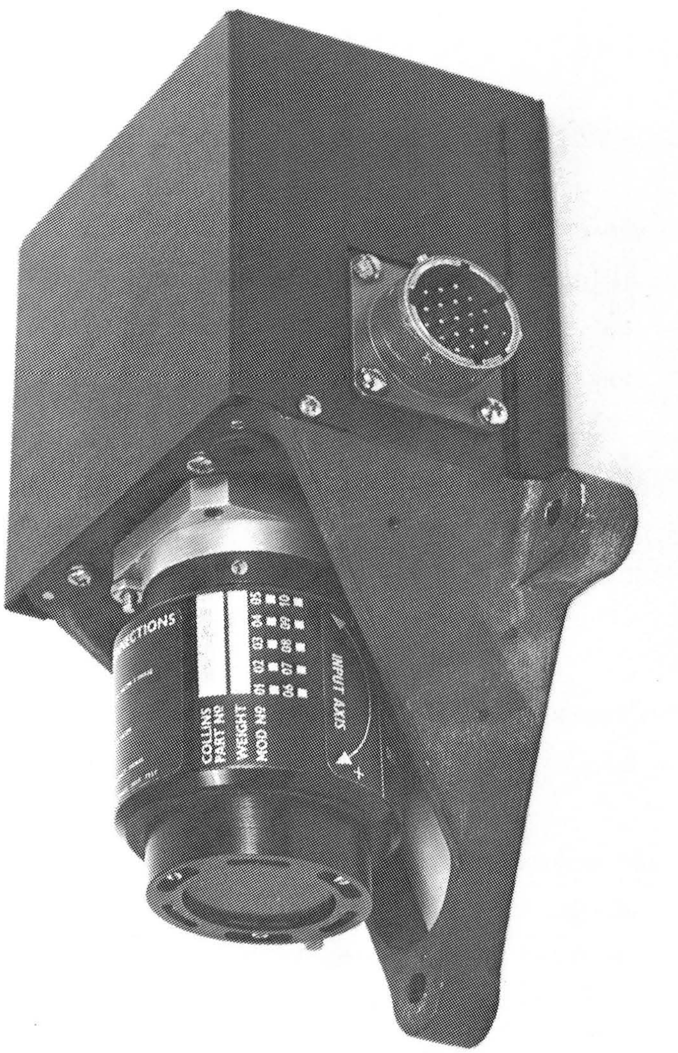


Figure 18. Pitch-Rate Gyro

Operating experience to date indicates that the most probable gyro failure mode is motor failure to start. To detect this mode, a dedicated spin monitor in the mounting base senses phase lock of the rotor and stator. A rotor that is not spinning at proper speed will be out of synchronism and will be detected. Because a slow or nonspinning rotor is a passive failure of the sensor (i.e., it is not observable at the analog output until a pitch event occurs), the monitor state is provided as discrete output. In the Test ACT Program, this discrete is used by the Essential computer as a condition in the Essential servo engage logic.

A torque coil is installed on the gyro to allow a ground check of the precession sensing mechanism. By means of a test circuit in the mounting base, the rate gyro may be commanded to produce an output equivalent to a known value of pitch rate. Proper response to the torque command indicates proper operation of the gyro output provisions downstream of actual gyroscopic precession. This response is not affected by rotor speed; it is no substitute for the spin monitor described above.

6.2.1.2 Control Column Force Transducers

The column force transducer (fig. 19) is constructed by Kavlico to Boeing 757 aircraft drawing S253T401, with changes that provide four independent electrical outputs per transducer.

Figure 19. Column Force Transducer

The linear variable differential transformer (LVDT) force transducers do not include monitoring or self-test provisions. Their outputs are three wire, which allow user monitoring of open or shorted signal paths. Check of the actual force sensing will be accomplished by a ground check force on the control column.

6.2.1.3 Wing Accelerometers

The wing accelerometer (fig. 20) is a dc analog unit purchased from the Systron-Donner Corporation and packaged per Collins specifications. It does not include monitoring or self-test provisions, but the nominal 1-g output due to gravity provides a convenient method of checking accelerometer operation. The unit has a predicted MTBF of 60 000 operating hours, supported by field data on 1 year of service.

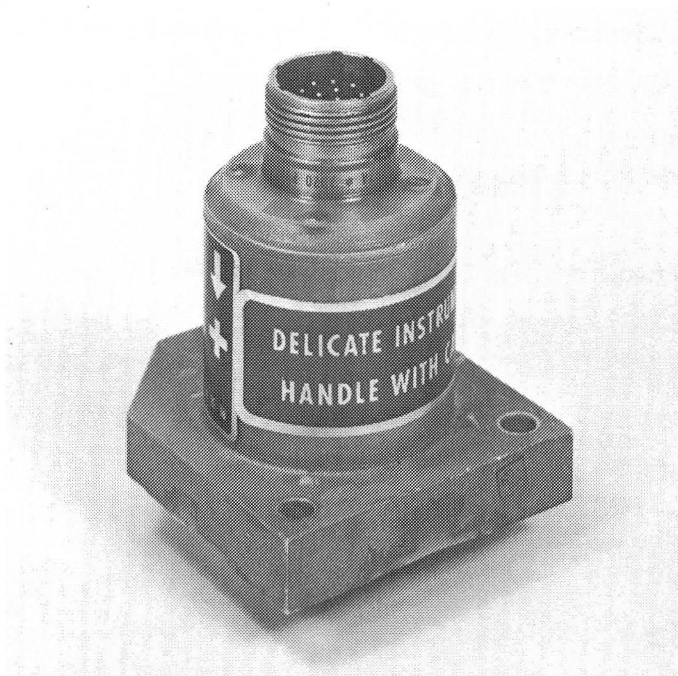


Figure 20. Accelerometer

6.2.2 AIRPLANE SENSORS

This section discusses those airplane sensors used in the 757-200 production configuration. The sensors are connected to the Test ACT computers as shown in Figures 21 through 24, and their signal and discrete information is utilized in the control laws, as shown in Figure 25.

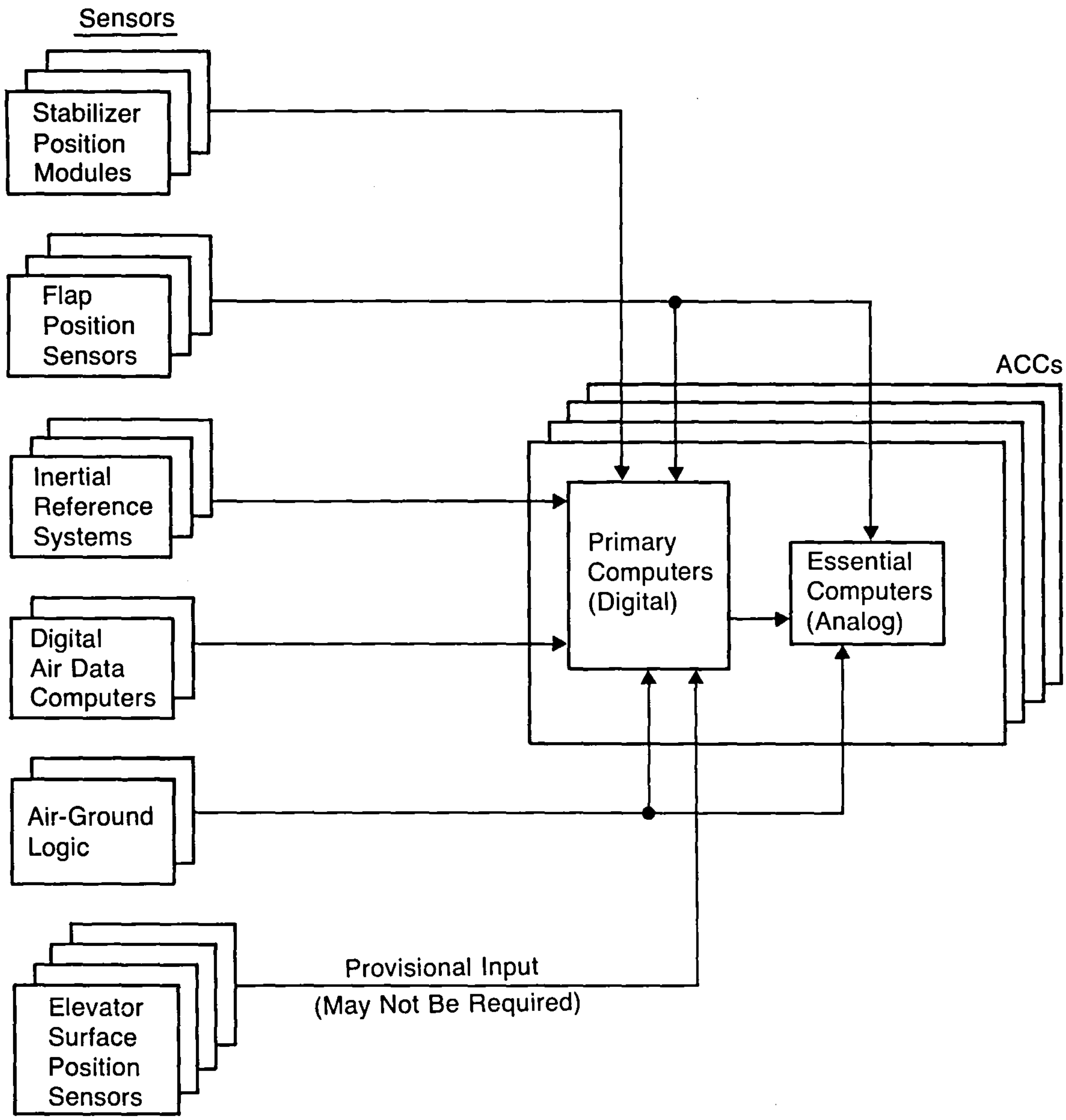


Figure 21. Airplane Sensors

6.2.2.1 Inertial Reference System

There are three Inertial Reference Systems (IRS) on the 757. These strapdown systems use laser gyros, accelerometers, and digital computation to determine the airplane pitch, yaw, and roll attitudes and rates; accelerations; and speeds. This information is transmitted over an ARINC 429 bus.

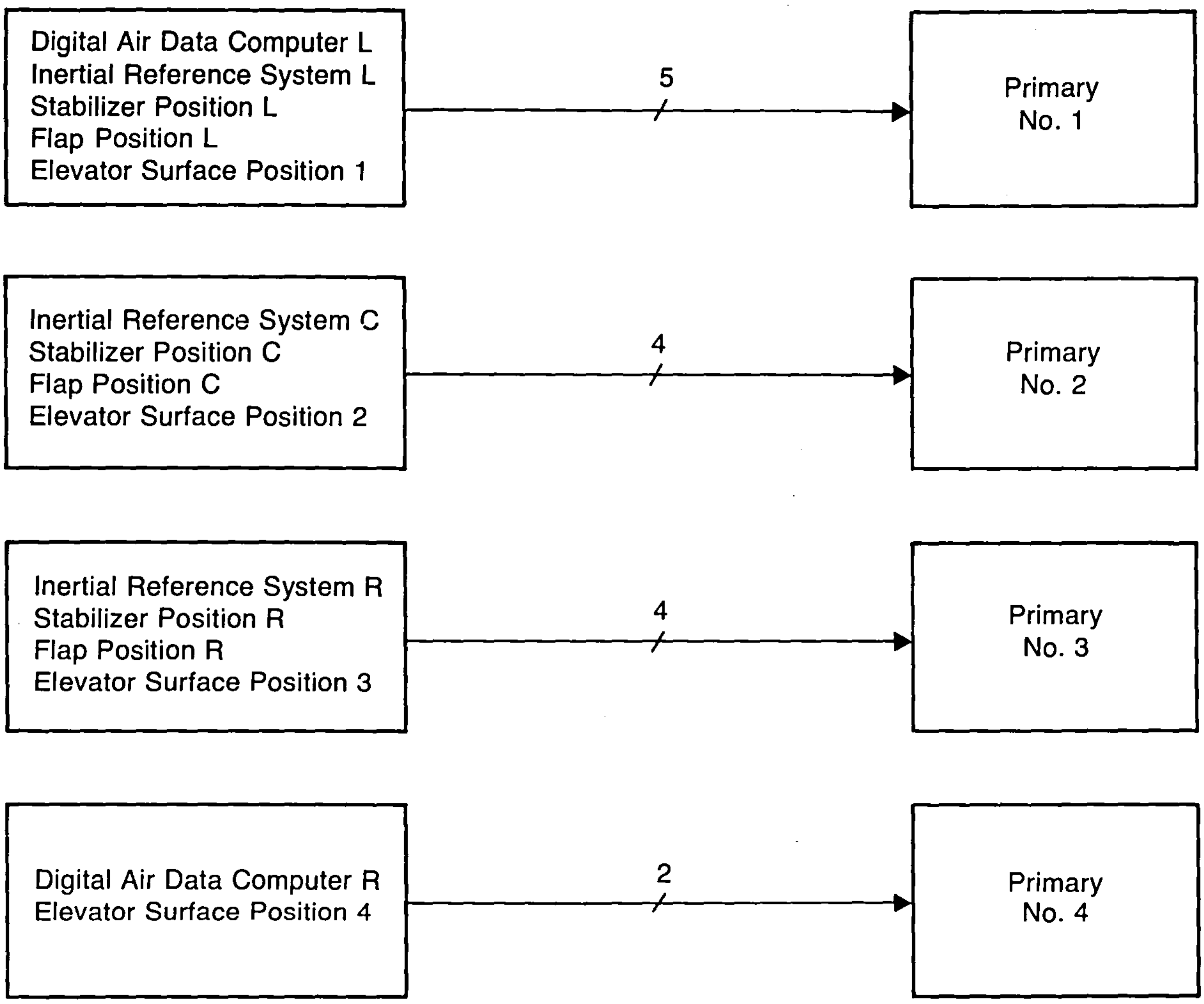


Figure 22. Primary Sensor Interface

Pitch, yaw, and roll attitude and pitch-rate information are fundamental parameters in the Primary control laws, whereas body normal acceleration is used in the WLA control laws. The information from all three IRSs is available to each of the four ACCs via the cross-channel buses.

IRS connections are shown in Figures 21 and 22; signal routing is shown in Figure 25.

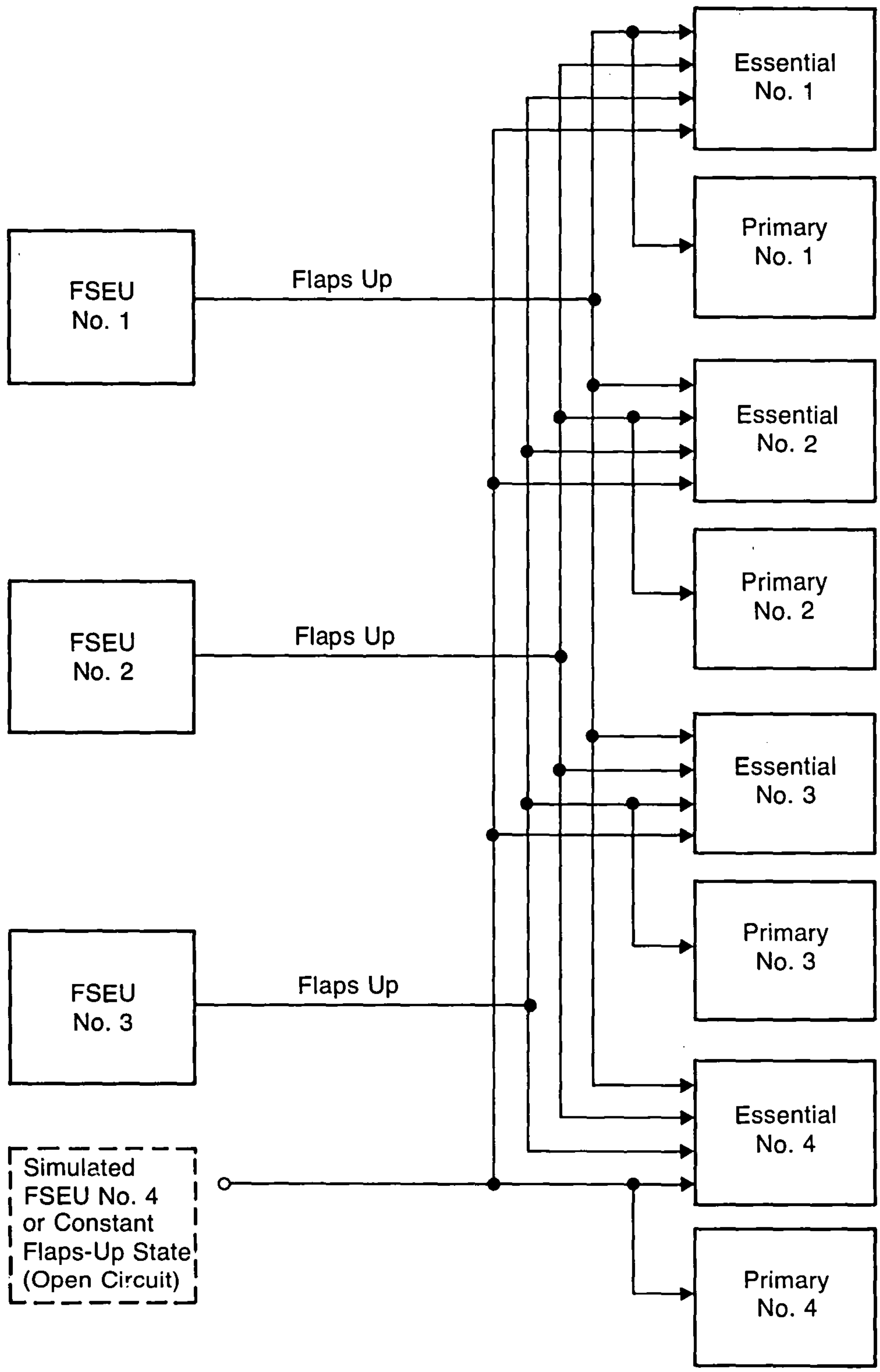


Figure 23. Flap Position Discrete Signal Interface

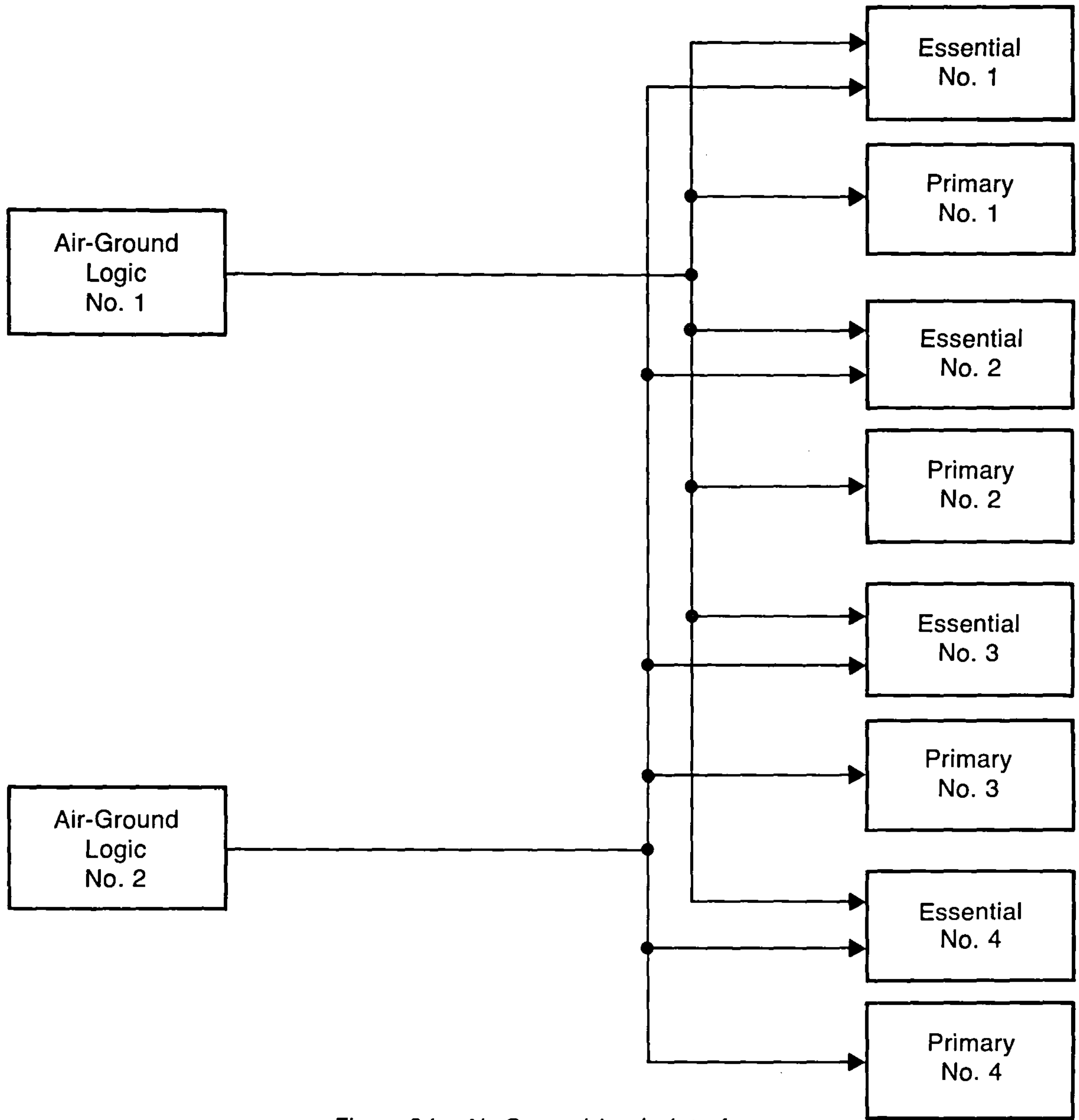


Figure 24. Air-Ground Logic Interface

6.2.2.2 Digital Air Data Computer

Impact pressure and static pressure are measured by means of pitot-static probes on the aircraft. The Digital Air Data Computer (DADC) then transforms the pressure signals into digital information and computes airspeed, Mach number, altitude, and rate of climb.

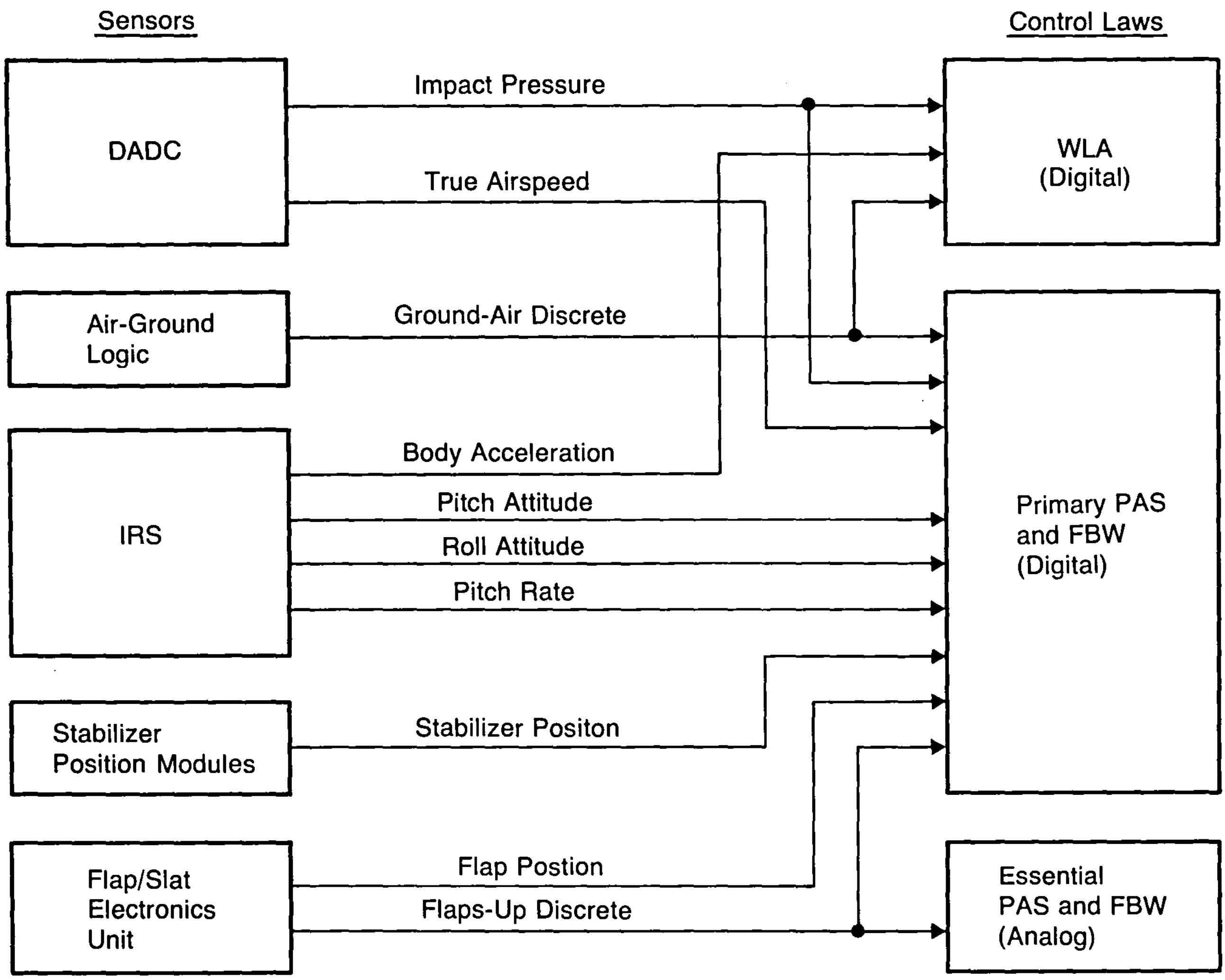


Figure 25. Airplane Sensor Signal Distribution

There are two DADCs on the test aircraft, information from which is transmitted via an ARINC 429 bus to two ACCs, as shown in Figure 22.

Impact pressure is used in the WLA and Primary PAS control laws for gain scheduling, while true airspeed is the feedback variable employed for speed stability in the Primary control laws.

6.2.2.3 Flap Position System

The flap position system is part of the 757 high-lift control system. Its information is derived from synchros and resolvers located on the inboard and outboard trailing-edge flaps. The Flap/Slat Electronics Units (FSEU) generate discrete and position information for use by other avionic components of the aircraft. The flap position signal is a dc voltage.

Flap position and flaps-up discrete information is used in Test ACT computations to schedule gains of the control laws for optimum stability. In the Test ACT Configuration, flaps-up discrete information from each of the three FSEUs goes into the Essential control laws of all four ACCs. A ground is provided on the console to simulate a fourth source of flap discrete information. In the event of a failure of one of the flap discretes, the system goes to the lower gain, flaps-up configuration.

Flap position sensor connections are shown in Figures 21 and 23.

6.2.2.4 Air-Ground System

The 757-200 air-ground system uses proximity switches that are activated by nose gear oleo compression, right landing gear truck tilt, and left landing gear truck tilt. Connection of the air-ground system to the Test ACT computer is indicated in Figures 21 and 24.

Air-ground logic is used to lock out WLA, Primary PAS, and automatic stabilizer trim on the ground and preflight test in flight. It also initializes the signal selection and fault detection (SSFD) logic.

Both on-ground signals are hardwired to each of the four Essential computers, and each Primary channel gets information from these relays via cross-channel buses.

6.2.2.5 Stabilizer Position Sensors

Stabilizer position is a fundamental parameter in the speed and elevator offload control laws. All computers receive this information by direct connection to a position sensor and/or by cross-channel buses, as indicated in Figure 22.

Stabilizer position signals are derived from the Stabilizer Position Module (SPM). Three identical SPMs are installed in the electric system card file, which is in the electronic equipment bay. They use an ac signal from a rotary variable differential transformer (RVDT) located in the position transmitter module to produce a dc voltage proportional to stabilizer position. Each SPM has both a stabilizer position and a monitor channel. The position channel output drives four output buffer amplifiers whose output is grounded if a fault is detected.

6.2.2.6 Elevator Surface Position Sensors

The elevator surface position sensors are dual synchros, one mounted on each elevator and hardwired to each Primary computer. The surface position signal will be derived from two windings of the synchro, and a third will be used for monitoring. Figures 21 and 22 show the surface position sensor connections.

Elevator position information is not presently used in the Test ACT control laws. It is a provisional signal that may be employed in flight test if unacceptable limit cycling develops.

6.3 CONTROL AND WARNING PANELS

6.3.1 TEST ACT PANELS

Two of the Test ACT flight deck panels, the Test ACT Control Panel (TACP) and the Preflight Test Panel (PFTP), are located in the pilots' overhead panel. The Flight Test Programmer (FTP) is used only during flight test and is located in the control stand.

6.3.1.1 Test ACT Control Panel

Physical Characteristics—The TACP (fig. 13, left) consists of seven switches and annunciators, hardwired to their associated connectors and circuitry, with discrete components mounted on a circuit board secured to the chassis.

Operation—Figure 26 shows the TACP layout with two functional groups: Essential PAS/FBW and Primary PAS/WLA advisories.

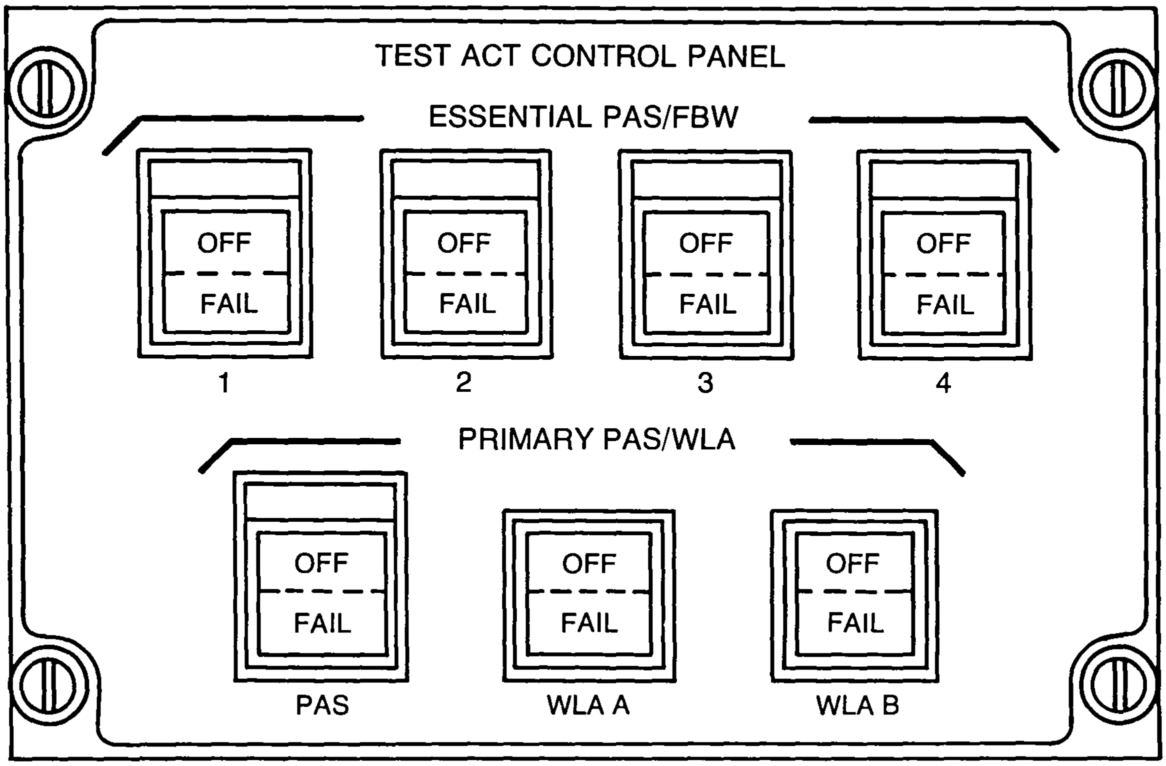


Figure 26. Test ACT Control Panel-Front Panel Layout

Essential PAS/FBW—The TACP contains a split-legend switchlight associated with each of the four Essential channels. While the switchlight is in the "on" position detent, a dedicated discrete output is provided to the respective ACC, indicating that engagement of that Essential channel is enabled. This switch is expected to remain in the "on" position, except when a failure has occurred; then it may be desirable to manually disable a channel or to make an attempt to reengage after an automatic disconnect. During normal operation the switchlights are not illuminated.

When an ACC computer detects an Essential channel fault, it provides an automatic disconnect of that channel. The ACC also outputs a fail discrete to the TACP, which is used to illuminate that channel's "fail" message, located in the lower half of the associated switchlight. This annunciation is advisory and does not require flight crew action.

Placing the Essential PAS switch in the "off" position removes the dedicated enable discrete output to the associated ACC, thus removing power from the arm and engage relays and disabling servo reengagement. When the switch is in the "off" position, it also locally extinguishes the associated "fail" annunciation and illuminates the "off" annunciation.

Cycling the switch back to the "on" position reapplies the enable discrete to the ACC and initiates Essential channel servo reengagement. This action also locally extinguishes the "off" annunciation and returns control of the "fail" annunciation to the ACC's dedicated fail discrete.

All Essential PAS/FBW "off" annunciations are white, while the "fail" annunciations are amber. The four switch functions of the system are guarded to preclude inadvertent manual disengagement of any Essential servo.

Primary PAS/WLA: Manual Primary PAS Disconnect Switch—This switchlight is provided to allow a manual disconnect of all four of the Primary PAS inputs to the Essential channels.

The PAS "fail" annunciation is driven directly by discrete outputs from any of the four ACC computers. This advisory may be used as a cue to the flight crew to override the Primary PAS system by pressing the switchlight. (As flight safety is provided by hardware limiters in each ACC, flight crew action is not a safety requirement.)

When the PAS switch is pressed, a dedicated PAS "on" discrete is removed from each ACC. Pressing the PAS switch also extinguishes the "fail" annunciation and illuminates the "off" annunciation. It is expected that this switch will normally be left in the "on" (detented) position. Because the PAS switch is guarded, inadvertent operation is unlikely.

Primary PAS/WLA: Primary WLA Status—The TACP provides independent status annunciation and manual disconnect provisions for each WLA servo system. Two ACC computers are associated with each servo system. In normal operation, a dedicated "on" discrete is output from the TACP to supply power to the WLA engage relay of each of the ACCs. Upon detection of a failure, the ACCs provide an automatic disconnect and output a fail discrete to the TACP. These fail discretes are inputs to a hardware "OR" gate such that either ACC may directly illuminate its associated "fail" advisory annunciation.

Manual disconnect of a failed WLA system is provided by pressing the associated WLA switchlight. This action removes the dedicated "on" discrete from the ACC, locally extinguishes the "fail" annunciation, and illuminates the associated "off" annunciation.

6.3.1.2 Preflight Test Panel

Physical Characteristics–The PFTP (fig. 13, right) consists of two switches and three annunciators, hardwired to their associated connectors and circuitry, with discrete components mounted on a circuit board secured to the chassis.

Preflight Test Operation–Figure 27 shows the PFTP layout. This panel provides preflight test initiation and status advisories, as well as a monitored discrete output to drive an external "no dispatch" warning.

To preclude the need for redundancy or monitored annunciations, preflight test steps occur in a specific order, with each step indicated by a pair of annunciation changes. Thus, the flight crew can verify correct test sequencing and detect annunciators that may be failed "on" or "off" due to system or PFTP failures.

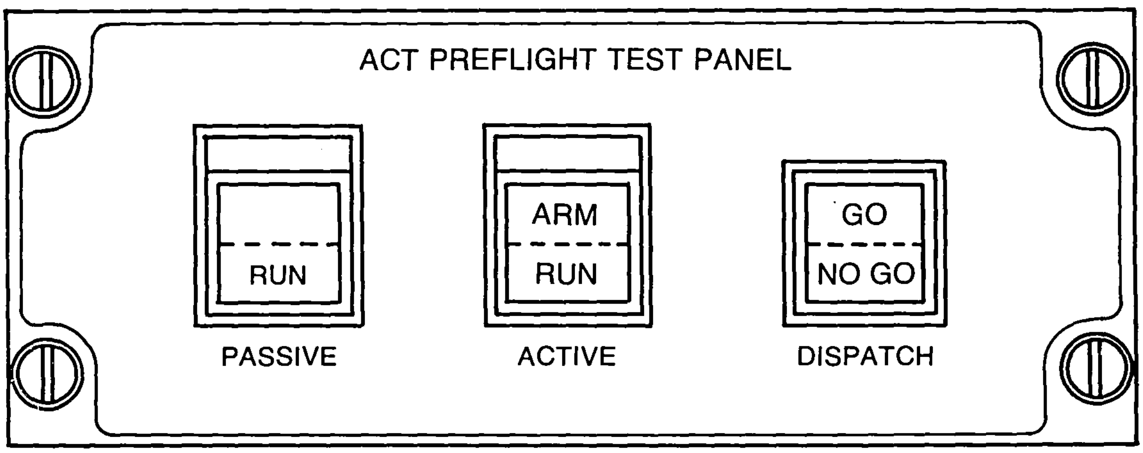


Figure 27. Preflight Test Panel–Front Panel Layout

The preflight test sequence is described in the following:

At the termination of a flight leg or at aircraft power-up, the PFTP "no go" status annunciator is illuminated to indicate that a preflight test has not been completed since the last flight.

The preflight test circuitry within the PFTP is enabled by the presence of two of four ACC discretes that indicate they are operating in the "on ground" mode. Preflight test availability is indicated by the first crew prompt message appearing on the Engine Indication and Crew Alerting System (EICAS) (sec. 6.3.2); testing is initiated by depression of the passive test switch, which operates a PFTP lamp test. The "no dispatch" output is monitored to verify its operation; assuming its warning is functional, the PFTP monitors discrete inputs from the ACCs. When three of four ACCs indicate their test is in process, the PFTP maintains the electrical position of the passive test switch via an internal solenoid and illuminates the passive test "run" annunciation. This same three-of-four logic is also used to remove the independent channel's "stim enable" lockout, thus enabling preflight test stimulation by the individual ACCs.

The PFTP continues to monitor the ACC test discretes. Upon indication that at least three ACCs have successfully completed their passive tests and that no ACC has detected a passive fault, the passive test "run" annunciation is extinguished and the active test "arm" annunciation illuminated. This signal and the second crew prompt message on EICAS serve as flight crew cues to initiate the active preflight test when external conditions allow control surface movement. The flight crew initiates the active test by depressing the "active" pushbutton. This results in the removal of the active test "arm" annunciation and illumination of the active test "run" annunciation. The sequence of active test elements and crew prompts is controlled by the ACCs.

Upon determination that three or four ACCs have successfully completed the preflight test, as indicated by the ACC test status discretes, the PFTP (1) releases the passive preflight test switch, (2) locks out the stim enable discretes, (3) illuminates the "go" dispatch annunciator, and (4) extinguishes the "no dispatch" warning output and "no go" dispatch status annunciation.

The dispatch status annunciation is interlocked to the solenoid-held "run" switch such that a "go" annunciation is only possible in the released position and a "no go," with remote "no dispatch," is always provided in the held position.

The failure of any two ACCs to complete their preflight test successfully will terminate that test and result in a continuous "no go" status annunciation; the "no dispatch" warning will remain illuminated.

No manual reset is provided for a "no go" indication. Clearing the "no go" or "no dispatch" annunciation requires reinitiation and successful completion of the entire preflight test. If takeoff is initiated without test completion, testing is aborted at ground speed greater than 26 m/sec (50 kn). An automatic reset of the "go" annunciation will be provided at liftoff (less than two of four ACCs indicate "on ground" mode).

As shown in Figure 27, the passive and active preflight test request switches are guarded to preclude inadvertent activation.

"No Dispatch" Warning Output—The PFTP provides combining logic and a current sink drive to control a remotely located red array (or equivalent) "no dispatch" warning. This warning will be annunciated during preflight test and will be extinguished upon its successful completion by three or more ACCs. Failure of two or more ACCs to complete preflight test successfully will result in a continuous "no dispatch" warning. No manual reset is provided for this annunciation. Resetting requires reinitiation and successful passing of preflight test. If takeoff is initiated without successful completion, the warning is extinguished at liftoff.

6.3.1.3 Flight Test Programmer

Physical Characteristics—The FTP (fig. 13, center) consists of a single pushbutton switch, a 16-position rotary switch, and a toggle switch, all hardwired to the FTP rear connector.

Operation—The FTP provides two functions within the Test ACT System: a capability to select and initiate preprogrammed flight test conditions and a Test ACT System arm and disconnect function. The FTP layout is shown in Figure 28.

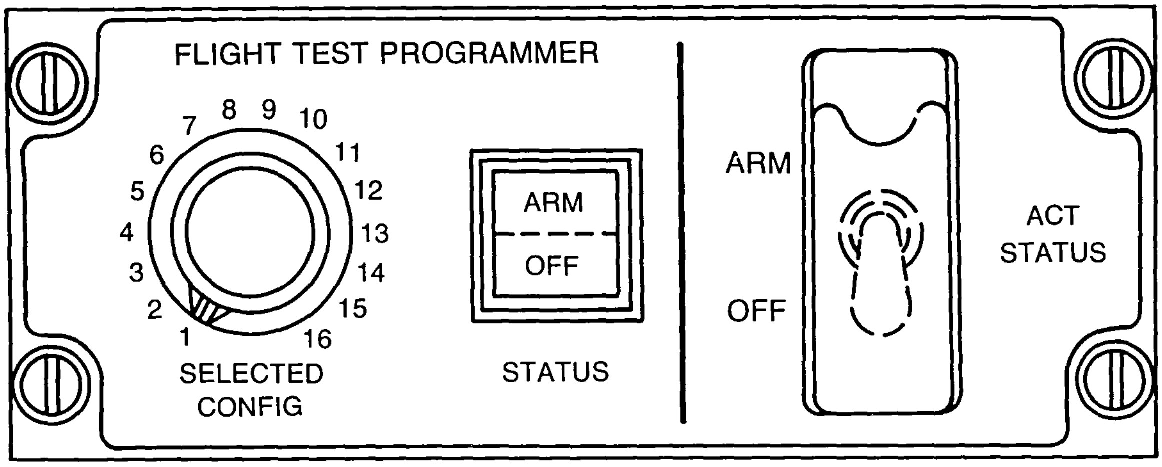


Figure 28. Flight Test Programmer–Front Panel Layout

Test Selection–Preprogrammed test conditions are chosen by placing the 16-position selected configuration switch in the position corresponding to the desired test. The position of this switch is continuously provided to the four ACCs by a binary code representation from four discretes. When the selected configuration switch is moved to a position other than position 1 (baseline), the "arm" annunciation in the status switch will illuminate, indicating a test configuration may be initiated.

Test Initiation–Initiation of the selected test is accomplished by depressing the status pushbutton switch. A common initiate test request is provided to all ACCs via redundant discrete interfaces. Initiation of a test extinguishes the "arm" status annunciation and illuminates an amber "on" annunciation that will remain until the test is terminated.

Test Termination–Test termination is provided by a second depression of the status switch or by rotation of the selected configuration switch to a different position. At termination, the "on" annunciation is extinguished, and the "arm" annunciation is illuminated if the selected configuration switch is in any position other than 1 (baseline).

Test ACT System Arm and Disconnect of FBW Functions–Emergency disconnect of the Test ACT System is provided by placing the ACT status switch in the "off" position. This simultaneously causes a servo disconnect of all PAS servos by removing the excitation to the individual servo arm and engage relays located within the ACCs. Dedicated discretes are also provided to transfer trim control to the AFDS when the Test ACT System is disconnected and to inhibit the autopilot when it is engaged.

The Test ACT System PAS servos are armed by setting the ACT status switch to the "arm" position. This enables an independent automatic engagement of all servos if the other manual inflight engage requirements are completely met. Arming the Test ACT System also sends a single discrete output to inhibit the AFDS.

The ACT status toggle switch is guarded to preclude inadvertent placement of the switch in the "arm" position.

6.3.1.4 Dedicated Warning Indicators

The Test ACT System will have two warning (red) indicators placed above the glare shield. These two warnings will be "total loss of Essential PAS/FBW" and "total loss of Primary PAS."

6.3.2 AIRPLANE SYSTEM

This section describes the Engine Indication and Crew Alerting System (EICAS) that will be used by Test ACT for annunciations to the pilots.

EICAS is an indication and alerting system that receives, processes, and displays all engine information required by the flight crew; it also receives and processes information from airplane subsystems and then displays necessary crew alerting messages (warnings, cautions, and advisories). For Test ACT, EICAS will be used to annunciate crew-prompt messages to the pilots during preflight test. The test phases are begun manually; for active preflight test, the crew must provide certain initial conditions, specific flap settings, and a column pullthrough. These crew actions are prompted with the first set of messages in Table 1. EICAS will also be used to annunciate system failures in the form of caution and advisory messages; they are listed in Table 1.

6.4 TEST ACT CONSOLE

The Test ACT Console (TAC) (fig. 14) consists of a pair of multibay racks designed to house the Test ACT computers and associated equipment for software loading and system evaluation. It will be used first for system test in the laboratory and later will be mounted in the test airplane for flight test operations.

Table 1. EICAS Messages for Test ACT

CREW PROMPTS

1. Press passive
2. Clear surfaces, hydraulics on, flaps zero, press active
3. Extend flaps
4. Pull column
5. Push column
6. Test complete

CAUTIONS

1. One failure from loss of Essential PAS/FBW
2. Total loss of WLA
3. No dispatch

ADVISORIES

1. Maintenance required
2. Full-time no dispatch

6.4.1 CONSOLE CONCEPT

The standard Boeing flight test rack affords convenient means of housing the equipment and enabling the shift from lab to airplane with minimal disassembly. Two such racks together form the Test ACT Console. The individual racks plus their contents are labeled console No. 1 and console No. 2.

All hardware in these consoles is designed to be rugged in order to retain a safe cabin environment in flight test.

6.4.2 SYSTEM HARDWARE CONTENT

For laboratory use, the console will house all the Collins-supplied Test ACT System hardware listed below:

LRU	Type No.	Quantity
Active Controls Computer	ACC-701	4
Test ACT Control Panel	TACP-701	1
Preflight Test Panel	PFTP-701	1
Flight Test Programmer	FTP-701	1
Pitch-rate gyro	345-A9	4
Column force transducer	CFT-100	2
Wing accelerometer	ACL-111	8

These units are powered and interconnected within the console. Interconnect points are provided to tie the console to the laboratory where remaining system components (such as servos, trim systems, IRSs, DADCs, etc.) are provided in either real or simulated form. Tie points for the console interconnect are contained in a junction box to which all sensor, control, and aircraft interfaces are connected. It provides military-type twist-lock connectors for the interfaces to allow mobility of the console and its components. Circuit breakers for the 28-V dc ACC input power are mounted in the box, providing protection either in the laboratory or in the aircraft. Wiring is physically arranged to meet safety segregation requirements of the Test ACT System. The junction box is mounted in console No. 2 behind the breakout panels.

When the console is moved to the airplane, sensors and control panels will be removed and installed in their appropriate locations; the ACCs will remain in the console. Other aircraft systems and sensors will be connected to the console via the same interface points used in the laboratory.

6.4.3 TEST SUPPORT HARDWARE

The Test ACT Console also contains equipment that is not required for the system to perform its functions but rather supports the testing of the system.

6.4.3.1 ACC Breakout Panels

A breakout panel for each of the four ACC units provides "break-in" and "break-out" test points between the ACC and other line replaceable units (LRU) in the Test ACT Console and lab interface. The front of the panel provides test points for every contact of each ACC connector. Plugs are installed to enable breaking of the signal path for fault simulation or alternate signal injection. During flight these plugs are positively retained by a clear plastic cover panel (entire breakout panel may be bypassed for safety in flight). The ACC breakout also has test points that are brought from inside the computer to the panel unit via the test access card.

6.4.3.2 Power Control Panel

In the laboratory environment, all power to the Test ACT Console is distributed by the power control panel. This panel provides breaker protection for 115-V ac, 60-Hz equipment and breaker protection and switching for the 28-V dc power to the ACCs, FTP, PFTP, and TACP. The latter is arranged to duplicate the aircraft installations, enabling valid testing of various power configurations. All power is routed through a normally open power relay that is electrically held closed to operate the console. An emergency disconnect button is provided to open the power relay and quickly remove power from console equipment.

During flight test, power to ACC breakers in the junction box will come directly from the aircraft through cockpit breakers, bypassing the power control panel. The power relay will no longer affect the computers, thus eliminating the possibility of a single-point total loss of power. Power will still be provided for the Hewlett-Packard terminal and the master processor. Because these will only be used for data gathering in flight, loss of their power would not affect Test ACT System performance.

6.4.3.3 CAPS Test Adapters

The Collins Adaptive Processing System (CAPS) Test Adapters are provided as a software development tool for CAPS processors. Each adapter interfaces with a CAPS processor and transfer bus inside an ACC via a test access slot and a transfer bus cable. The Test

Adapter may become a "master" on the bus and, as such, may command read and write operations by any "slave." It may also monitor all processor transactions on the bus. Because the adapter has control of the bus and can halt it as a result of operator instructions, the adapter will be disconnected during flight test to preclude inadvertent external interference with bus operations.

The CAPS Test Adapter provides the following functions:

- Halt–Stops all bus activity.
- Reset–Resets the CAPS processor.
- Run–Allows bus activity to continue.
- Bus step–While halted, executes single instructions.
- Exam–Reads contents of particular bus locations.
- Deposit–Loads data into particular bus locations.
- Monitor–Monitors reads and/or writes of particular locations.
- Breakpoint–Halts bus activity at the occurrence of a read and/or write of a preselected particular location; also, sets the halt to occur only if the information does or does not match a preselected value.
- History–Examines and continuously saves the last 16 bus transactions (address, data, and read or write) during a halt.
- Analog output–Four buffered digital-to-analog (D/A) converters provide analog outputs proportional to a digital value that is read from or written to four selected addresses.

The CAPS adapter also interfaces with the master processor via the master processor bus.

6.4.3.4 Master Processor

The master processor, with its operating system software, performs three major tasks: (1) interaction with the Hewlet-Packard terminal via an RS-232 bus, (2) interaction with the CAPS Test Adapters, and (3) collection of inflight fault information via ARINC 429 buses. The processor monitor program responds with printed messages to commands keyed into the terminal. Under control of the terminal, the monitor program handles various operations such as tape-to-memory load operations, memory verification operations, and many of the other functions provided by the CAPS Test Adapter panel (sec. 6.4.3.3).

As mentioned previously, the master processor functions as a data collection point for flight test fault information. The processor will receive data from each of the four computers via four ARINC 429 receivers. In a portion of the normal application program, the ACCs transmit preselected failure data to the master processor. These data are formatted and stored by the monitor program and are available for display.

6.4.3.5 Terminal

The Hewlett-Packard HP-2645A terminal provides I/O capability for the digital test equipment in the TAC. It consists of a keyboard, a cathode ray tube (CRT) display, and a dual minicassette magnetic tape drive. The keyboard is used to input manual commands to the master processor via RS-232 Standard. During laboratory test, software input comes from cassette cartridges; the CRT displays output data.

6.4.3.6 Line Replaceable Unit Mounting Racks and Forced Air Cooling System

The TAC has aircraft-type mounting shelves with rear connectors and hold-down mechanisms. Each shelf provides positive pressure cooling air to the bottom of the computers. The sources of this ambient temperature air are 400-Hz fans, one mounted on each shelf. The four computer shelves are mounted on a carriage that may be extended from the console, allowing access to the sides of the units for test purposes.

The TAC contains a Test ACT annunciation panel that indicates loss of computer cooling air. These indicators, whose information comes from air flow sensors, are easily visible to the test engineer in his normal test flight position. The panel is also the laboratory mounting fixture for the FTP, PFTP, and TACP. No forced air is provided to these units since the design allows sufficient free air flow for self-cooling.

To facilitate laboratory simulation, the flight sensors (accelerometers, rate gyros, and column force transducers) are mounted above the breakout panels. A mechanism is provided to allow the console operator to apply force to the column force transducers.

6.4.3.7 Flight Test Instrumentation Interfacing

By means of a circuit card in each ACC, the Test ACT Console supplies four high-speed ARINC outputs for instrumentation use. With appropriate programming of these cards, internal software variables are transmitted, and a buffered output of each ARINC cross-channel bus is provided.

6.4.3.8 Collection Panel

The collection panel provides stop, step, and start coordination among the CAPS Test Adapters. If the master adapter panel is armed in the "first stops all" mode, any CAPS adapter halting one CAPS processor will cause all four ACC CAPS processors to halt simultaneously. Once halted, all four processors may be stepped by pressing the "all step" switch on the collection panel. Conversely, they may be started simultaneously by pressing the "all start" switch. An "all stop" switch causes the halt of all four ACC CAPS processors.

7.0 SYSTEM DESIGN: PRIMARY SOFTWARE

Page

7.0 SYSTEM DESIGN: PRIMARY SOFTWARE

Primary software implements all Test ACT Primary control law functions, plus the testing and management of the Test ACT System components. The functions are executed both in preflight test and during flight and include test options that are selectable in flight.

7.1 SOFTWARE ORGANIZATION

The Primary software is composed of seven functions: executive, control law computation, redundancy management, failure detection, flight deck interface, Test ACT Console (TAC) outputs, and test option control (fig. 29). Each of these is described in subsequent paragraphs. Approximately 16 000 words of software are distributed among the functions, according to the percentages shown in Figure 30.

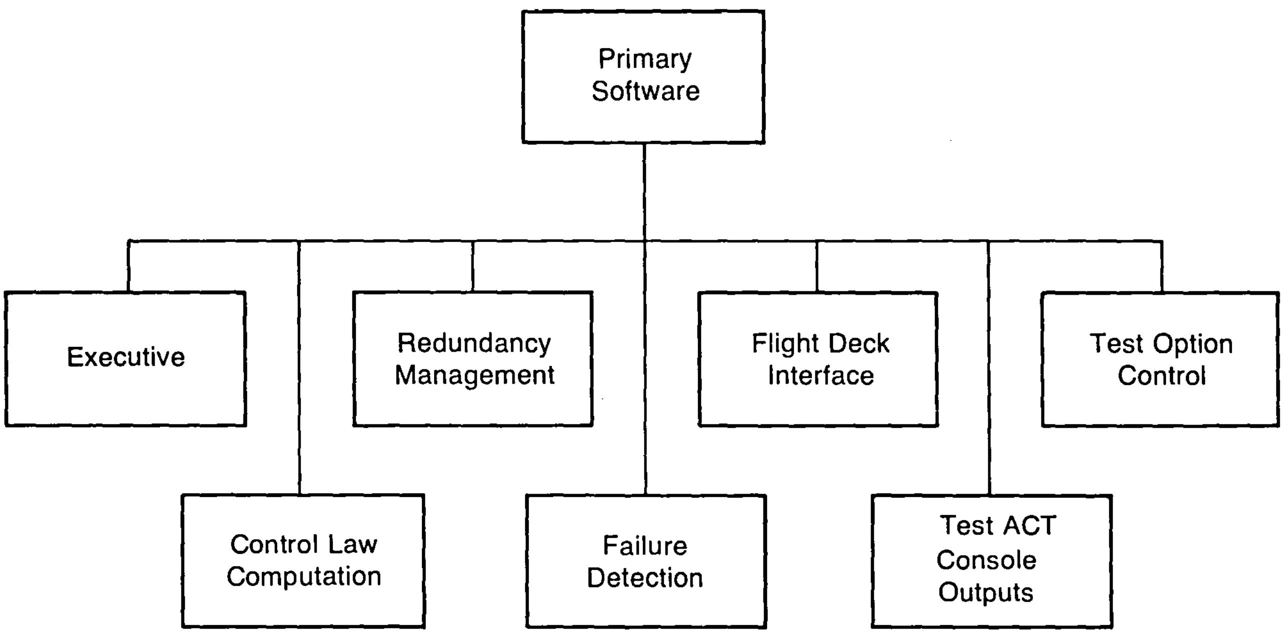


Figure 29. Primary Software Functions

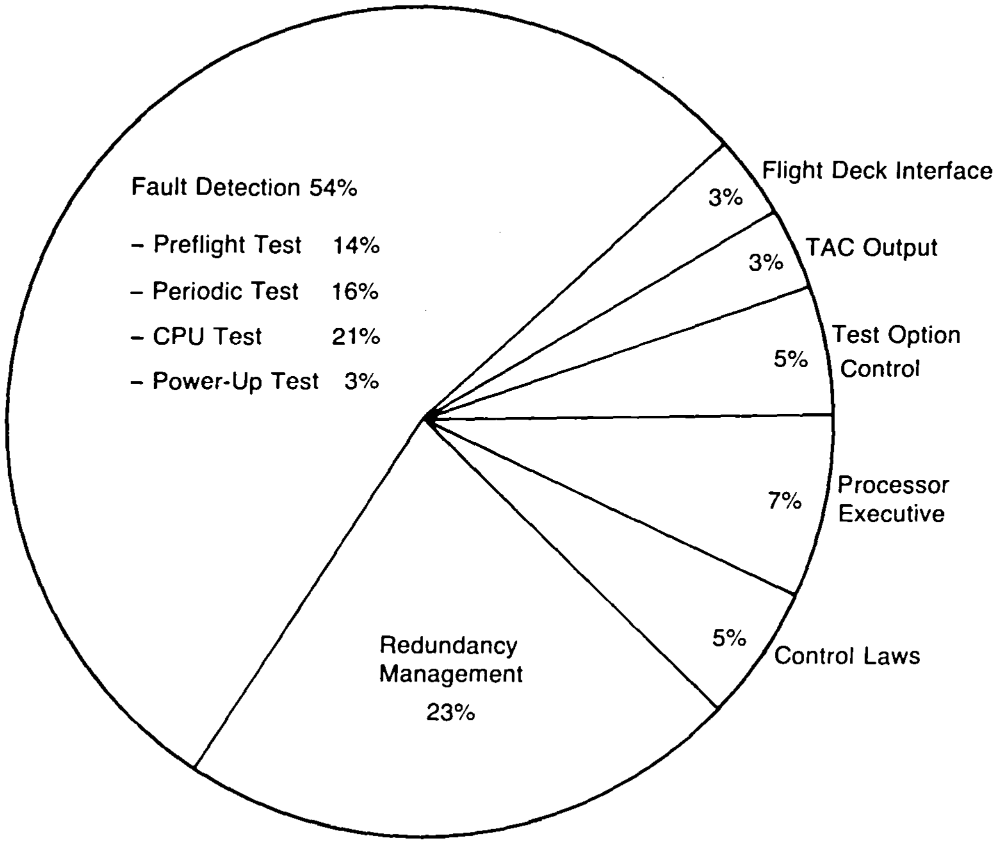


Figure 30. Test ACT Software Summary

7.1.1 EXECUTIVE

The executive function consists of the top-level Active Controls Computer (ACC) software and is broken down into the following:

- System states and task scheduling
- Hardware interrupt servicing
- Initialization

The executive establishes the multirate structure for foreground software functions and controls the execution of background computations. Executive processes fall into two categories: interrupt-mode processes, which include all operating system tasks required to service central processing unit (CPU) interrupts, and user-mode processes, including all tasks necessary to perform real-time and background scheduling and execution of operational tasks.

7.1.2 CONTROL LAW COMPUTATION

The control laws are divided into three major parts to implement the Primary control requirements:

- Aileron commands for wing-load alleviation (WLA), including both maneuver-load control (MLC) and gust-load alleviation (GLA)

- Elevator commands for fly-by-wire control augmentation and stability augmentation

- Stabilizer trim commands to avoid saturation of the limited-authority elevator command

Two subfunctions are used by each of the previously mentioned control law computations:

- Gain schedules and limiters that modify the control laws as a function of flight conditions

- Control law logic, providing mode switching and transient suppression during mode transitions

7.1.3 REDUNDANCY MANAGEMENT

The redundancy management function determines the utilization of redundant system resources. It incorporates two principal subfunctions: sensor management and output management. Sensor management (signal selection and fault detection, described in following paragraphs) combines redundant sensor inputs to produce high-integrity values for sensed parameters. The output management function manages associated output redundancy, determining ACC WLA and trim engage status and Primary pitch-augmented stability (PAS) validity.

The signal selection and fault detection (SSFD) function is designed to operate on a set of four continuous, redundant sensor signals. Only three of the four are required for normal operation of the SSFD. The fourth sensor signal is labeled as a "spare" if it is valid, otherwise the term "no spare" is used. However, there are sensor sets in the proposed 757 flight test airplane that have a maximum of two or three sensors. The SSFD algorithm operates on any of the variables listed in Table 2, provided at least two sensor signals are available. There are four major functional objectives of the SSFD algorithm: (1) to extract the most useful data from each redundant set of sensor signals, (2) to determine and isolate any sensor signal that is outside a preset tolerance limit, (3) to replace an isolated sensor signal with the "spare" or with the previously selected signal output of the SSFD if no spare exists, and (4) to upmode or reinstate an isolated signal to either the "active" or "spare" signal state when it reenters the preset tolerance limits for a sufficient period.

The previously mentioned objectives have been achieved through the use of four major subfunctions: (1) the signal selection function, (2) the fault detection function, (3) the sensor selection function, and (4) the SSFD initialization function.

The signal selection function utilizes the integral equalization technique on the redundant continuous-valued sensor signals to select the best signal. The selected signal is the mid-value of the equalized signals among the candidate inputs.

The fault detection function assesses the validity of the preceding sensor signals using sensor signal comparison and direct/inline monitoring of the sensor sets. Sensor signal monitoring is based on the instantaneous and long-term deviations of the sensor signals from the selected signal. Direct monitoring is based on such single-sensor inputs as direct activity, sign status matrix, common-mode monitors, and range.

The sensor selection function selects the candidate inputs to the signal selection function, which may be sensor or artificial signals. If fewer than two sensor signals are selected as candidate signals, the sensor selection function renders the corresponding sensor set invalid.

The SSFD initialization function establishes initial conditions of time-dependent computations for the SSFD subfunctions.

Table 2. Continuous-Valued Sensor Signals

Sensor Signal	Source
Elevator position (DE)	Elevator surface
Flap position (DF)	FSEU
Column force (FCOL)	Pilot's column
Stabilizer position (DS)	SPM
Normal wing acceleration (NZW)	Accelerometer
Body pitch rate (QB)	IRS
Pitch attitude (THETA)	IRS
Roll attitude (PHI)	IRS
Body normal acceleration (NZB)	IRS
Ground speed (VGND)	IRS
Flight path acceleration (VTDOT)	IRS
True airspeed (VTRU)	DADC
Impact pressure (QC)	DADC
Computed airspeed (CAS)	DADC
Mach number (MACH)	DADC

7.1.4 FAILURE DETECTION

Failure detection software assesses the ability of the Test ACT System to perform its design functions. Major subfunctions of failure detection are:

- Power-up tests
- Fault consolidation
- Periodic tests and monitors
- Preflight tests

Together these tests and monitors assess Primary computer performance and detect failures in the sensors, hardware monitors, Essential computers, actuators, and interfaces.

Power-up testing checks the operational capability of the ACC. Included are hardware interface checks, data and program memory tests, performance monitor tests, and error interrupt hardware checks. Power-up test execution occurs whenever power is applied to the ACCs.

The periodic tests and monitors routine is divided into the following four subfunctions:

- Software-interrogated hardware monitors
- Software-controlled hardware monitors
- Software monitors
- Software tests

Software-interrogated hardware monitors consist of those routines that periodically read the hardware monitor's state and set a flag if a monitor trips. This subfunction monitors Essential system sensors and Primary system buses. Software-controlled hardware monitors require periodic software stimulus and consist of Primary computer performance monitoring. Software monitors evaluate the health of normally operating system functions, whereas the software tests present data patterns that do not normally occur in flight to test data buffers and the CPU.

Preflight tests are subdivided into passive and active series; each series is initiated by engaging a switch on the Preflight Test Panel. Passive tests verify Primary and Essential system components that do not produce control surface movements. Active tests require hydraulic system pressure before testing control functions that may result in aileron, elevator, or stabilizer deflections.

Fault consolidation is a subfunction in terms of requirements only; i.e., the software implementation of requirements is distributed among various sections. Fault consolidation assesses the status of numerous monitors to determine:

- Primary PAS validity
- Executive monitor failures
- WLA validity
- Computer channel validity
- System dispatch status
- Assessment of maintenance requirements
- Local Essential interface validity

7.1.5 FLIGHT DECK INTERFACE

The flight deck interface functions manage the physical interfaces with flight deck annunciations, the Test ACT Control Panel (TACP), the Preflight Test Panel (PFTP), and the Flight Test Programmer (FTP). Flight deck interface functions consolidate inputs from the FTP, as well as inputs and outputs for the PFTP. They use inputs from the fault consolidation functions and other ACC software to drive maintenance and dispatch status annunciations.

7.1.6 TEST ACT CONSOLE OUTPUTS

Test ACT Console output functions provide processing to format and transfer fault and pertinent status data to the TAC for display, both on ground and during flight. Status data include the following information from each ACC: computer identification, trim and WLA control status, FTP state, and Primary computer state.

7.1.7 TEST OPTION CONTROL

Test option control provides the software interface between the ACC and the FTP. It responds to discretes provided by the FTP in selecting, initiating, and terminating implementation of any one of 16 preprogrammed test conditions within the ACC control law functions.

7.2 SOFTWARE METHODOLOGY

All Primary software was developed under Univac's EXEC-8 operating system on a Univac U1100 system, located at Rockwell's Scientific Computing Center in Seal Beach, California.

The Test ACT processor is a CAPS-6B model of the Collins Adaptive Processing System (CAPS) family that is designed to support embedded, real-time applications programmed in a high-level language (HLL). The Test ACT software was written in ALGOL Extended for Design (AED), a descendant of ALGOL-60.

7.3 SOFTWARE ADAPTABLE FROM 757/767 AUTOPILOT/FLIGHT DIRECTOR SYSTEM

Although most of the software for the Test ACT Program is new, two routines were transferred from the 757/767 autopilot virtually intact. These are the power-up tests and the CPU self-tests. The power-up tests check a number of hardware monitors, including the random-access memory (RAM) parity monitor, computer cycle monitor, and CPU self-test monitor. The CPU self-test verifies the functional capability of the CPU itself.

8.0 SYSTEM DESIGN: MECHANICAL

Page

8.0 SYSTEM DESIGN: MECHANICAL

8.1 MECHANICAL CONTROLS: TEST INSTALLATION

8.1.1 GENERAL

This section describes the design of the 757 mechanical pitch control system, as modified for flight testing of the Test ACT System. The modifications affect two parts of the airplane: the flight deck area below the pilots' floor and the empennage area near the horizontal stabilizer rear spar (fig. 31). Test ACT mechanical elements include two independent pilot pitch command paths: one for electrical flight and one for mechanical flight, plus quadruplex electrohydraulic servos whose outputs are linked differentially with the existing cable system to modulate the elevator surface actuator valves.

The design shown is intended to meet all requirements of Section 4.2.1. The selection of the arrangement and components has been directed toward maximum performance and safety of the Test ACT System. Particular emphasis has been placed on reduction of friction and freeplay in the mechanical signal paths to simulate the very low pitch control hysteresis that will be a primary requirement of reduced-stability airplanes. A major objective of the Test ACT flight program will be to evaluate the effect of varying levels of hysteresis on system performance, leading to standards for future mechanical control component design.

Hysteresis characteristics of the basic 757 pitch system have been determined, and design actions for improvement of the Test ACT System have been derived from a test program conducted on the 757 "Iron Bird." A test summary is presented in Appendix B.

To facilitate a simple and economical test installation in the 757 airplane, certain mechanical design ground rules were adopted:

- The normal 757 "neutral shift" function, in which the elevator is geared to the stabilizer for greater nose-down trim effectiveness, will be deactivated for the Test ACT Configuration. This removes the complication of trim point shift on the mechanical system side of the summing junction.

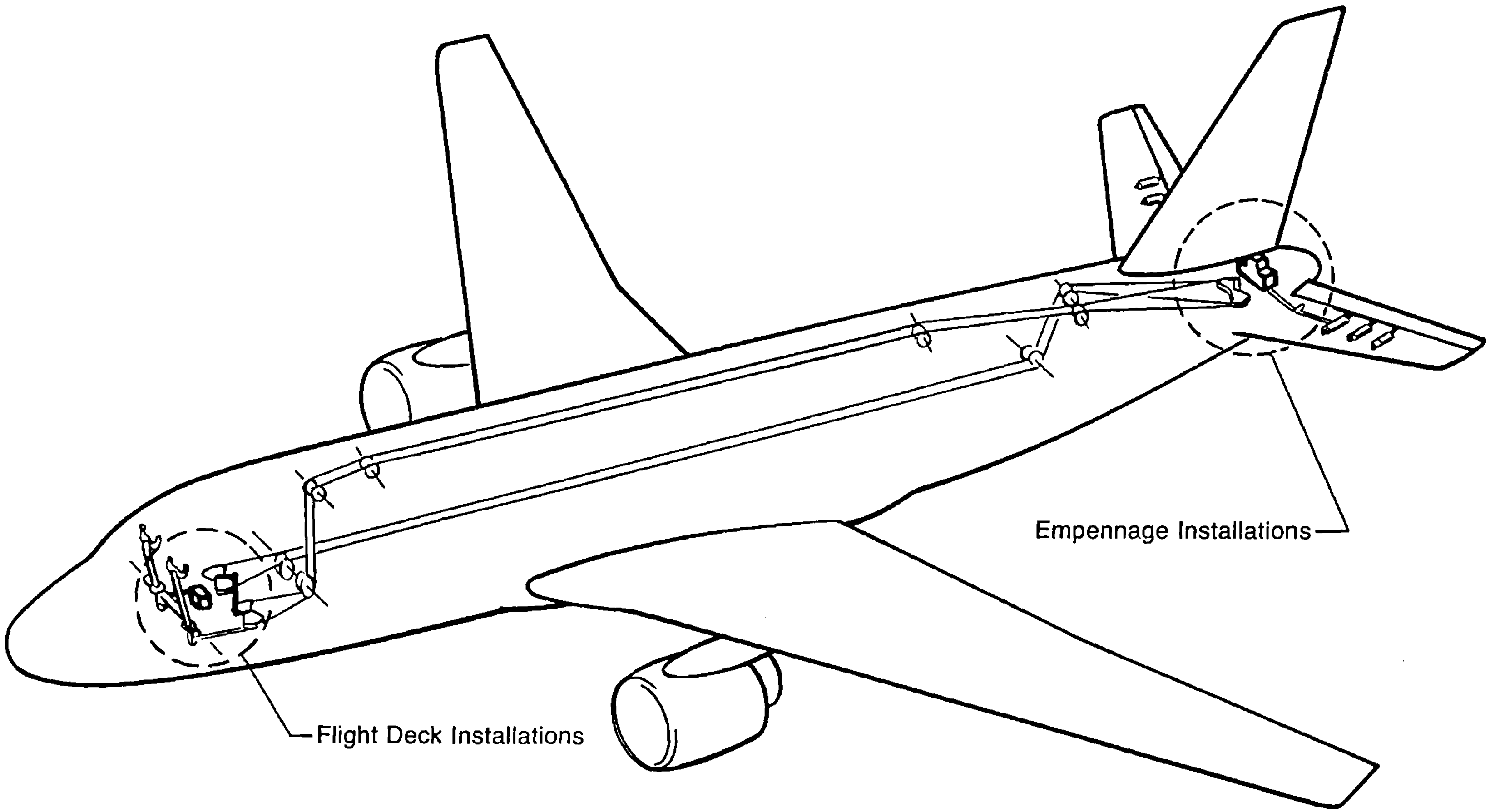


Figure 31. Test ACT Mechanical Controls–757

- The normal 757 flaps-up stabilizer nose-down electrical limit will be increased from +0.8-deg stabilizer angle to +3.8-deg stabilizer angle. This will ensure adequate nose-down trim authority when combined with an increased elevator downrig throughout the anticipated Test ACT flight regime.

- The Test ACT pilot station will be located at the first officer's (right-hand) seat position. The right-hand position offers better clearance and access for the under-floor Test ACT mechanism installation. Also, the existing 757 mechanical system rigging datum at the left-hand column is preserved, allowing for simpler conversion between Test ACT and conventional flight configurations.

Mechanical jam protection for elevator control is provided as follows for Test ACT:

- Electrical and mechanical control modes are each available to the other as backup for mechanical jams upstream of the summing junction.

- Overrides that allow separate control of right and left elevators are not provided for Test ACT.

- A jam of an elevator surface actuator valve will be overrideable by either electric or mechanical mode during Test ACT flight.

These provisions, plus appropriate preflight inspection and full-scale exercise of the Test ACT installations, will ensure a jam-free flight program while avoiding the complexity and performance compromises of a total jam isolation design policy.

8.1.2 SYSTEM ARRANGEMENT

Figure 32 shows the elevator system schematic for the unmodified 757 airplane. The system features conventional dual mechanical cable controls with a pilot/copilot breakout forward and a limited right/left breakout aft. A dual variable feel unit controlled by dual feel computers (not shown) attaches to the aft cable quadrants, as do triple autopilot servos. Linkage from the aft quadrants to right and left triple-parallel actuator installations provides valve input motion to position the single elevator surfaces on each side. Centering springs, valve pogo links, and shearouts protect against disconnects, valve jams, and freezeups at the surface actuators. Neutral shift linkage causes the feel unit to be reindexed to increase stabilizer effectiveness in the nose-down trim range.

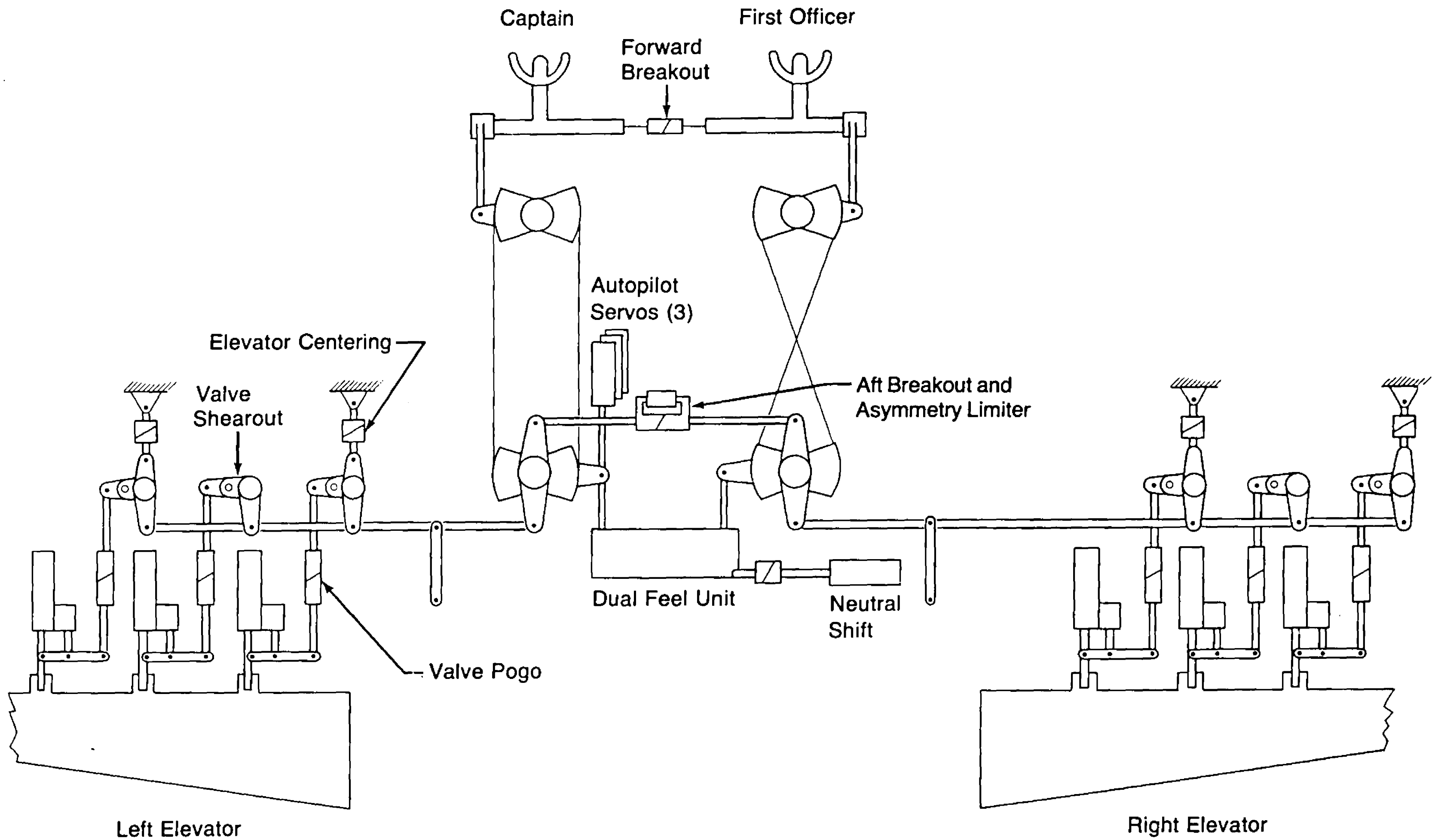


Figure 32. Elevator System–Basic 757

The elevator mechanical control schematic for Test ACT is shown in Figure 33. This arrangement reflects the following changes and additions to the basic 757 system:

- The first officer's column is disconnected from the mechanical system by disengaging the forward breakout and removing the right-hand body cable system input rod.

- A Test ACT pilot input mechanism is installed and connected to the first officer's column. The mechanism consists of dual four-channel force transducers, a feel cam and springs, and a rate damper.

- The forward right and left cable tension regulators are bused together to retain the stiffness of both cable paths for the mechanical control mode.

- A solid bus replaces the limited breakout between the aft cable quadrants. This provides a more rigid reaction path to ground during fly-by-wire (FBW) control.

- One pitch autopilot servo is retained in its 757 installation position, attached to the aft cable quadrant. In addition to mechanical-mode autopilot functions, this unit provides, through special modifications, an added increment of detent to react summing lever loads during FBW flight.

- The production pitch control feel and centering is retained unmodified, providing normal pilot feel functions in mechanical flight mode and, in parallel with the autopilot servo, a detent to react FBW loads.

- The neutral shift function is deleted. This involves removal of linkage between the stabilizer and the grounding point for the feel unit. The grounding point is then fixed at the desired rig position with a bolt to structure.

- The four Test ACT secondary servos, the servo summing shaft, and the servo voting detent mechanism are installed aft of the aft cable quadrant, using the support and space provisions of the upper two pitch autopilot servos. The new servos are 757 rollout guidance actuators, modified for increased rate and output path stiffness.

- Right and left Test ACT summing lever linkages are installed. These provide the summing function for the Test ACT servo and mechanical system inputs. It should be

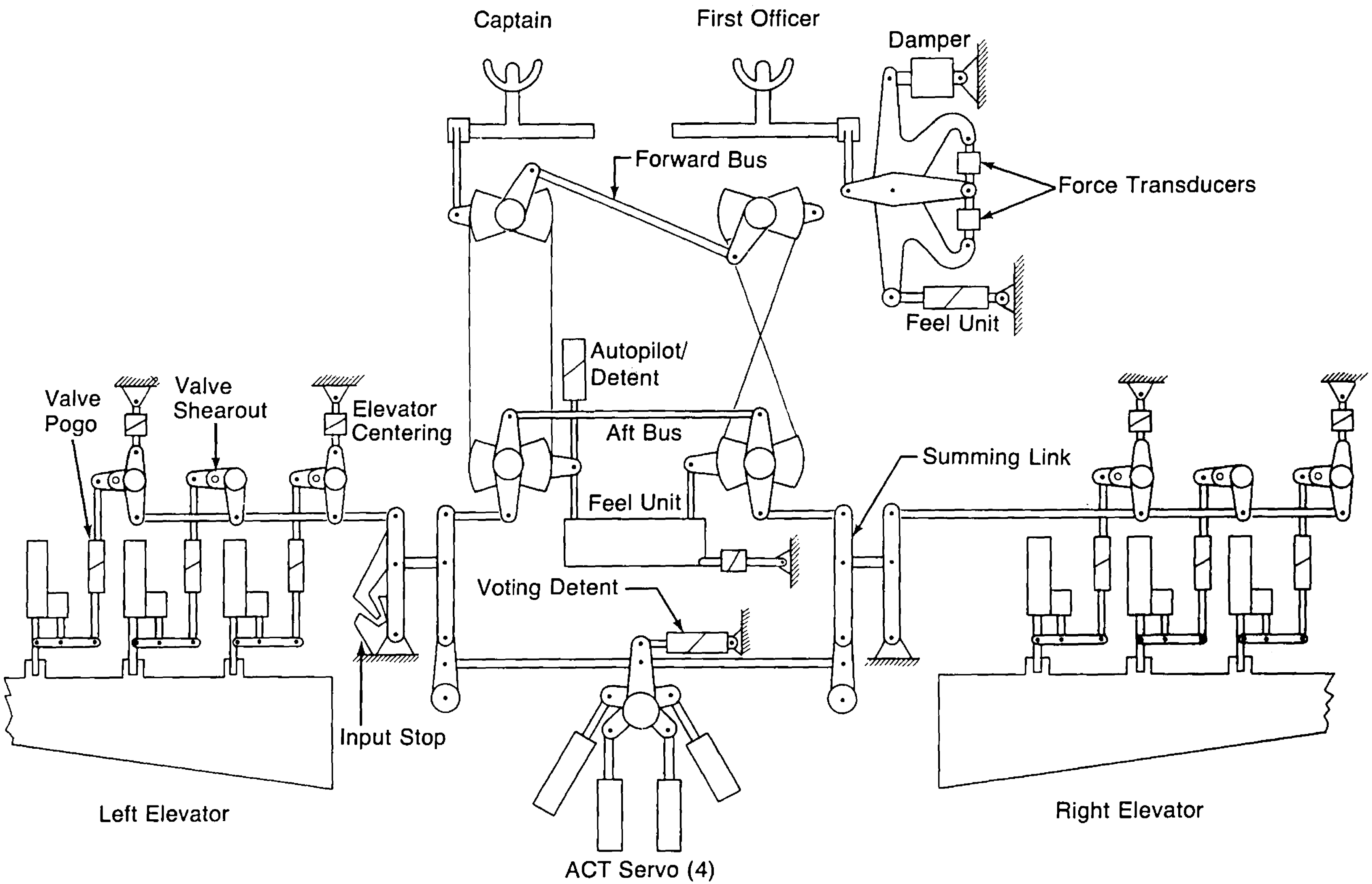


Figure 33. Elevator System–Test ACT

noted that normally only one input mode will be in control; however, the signals from one mode may be offset or augmented by inputs from the other, if necessary. Each mode has input authority of 100% of elevator. Limit stops in the mechanism prevent the summed transmitted signal from exceeding available surface actuator input travel.

- The surface actuators and actuator input linkages are unchanged, except for the addition of a light, antihysteresis spring behind the control valve spool in each actuator.

- Hydraulic systems are assigned to the various Test ACT actuators, as shown in Figure 34. Surface actuator hydraulic assignments, which provide for all three systems to each elevator, are unchanged from the basic 757.

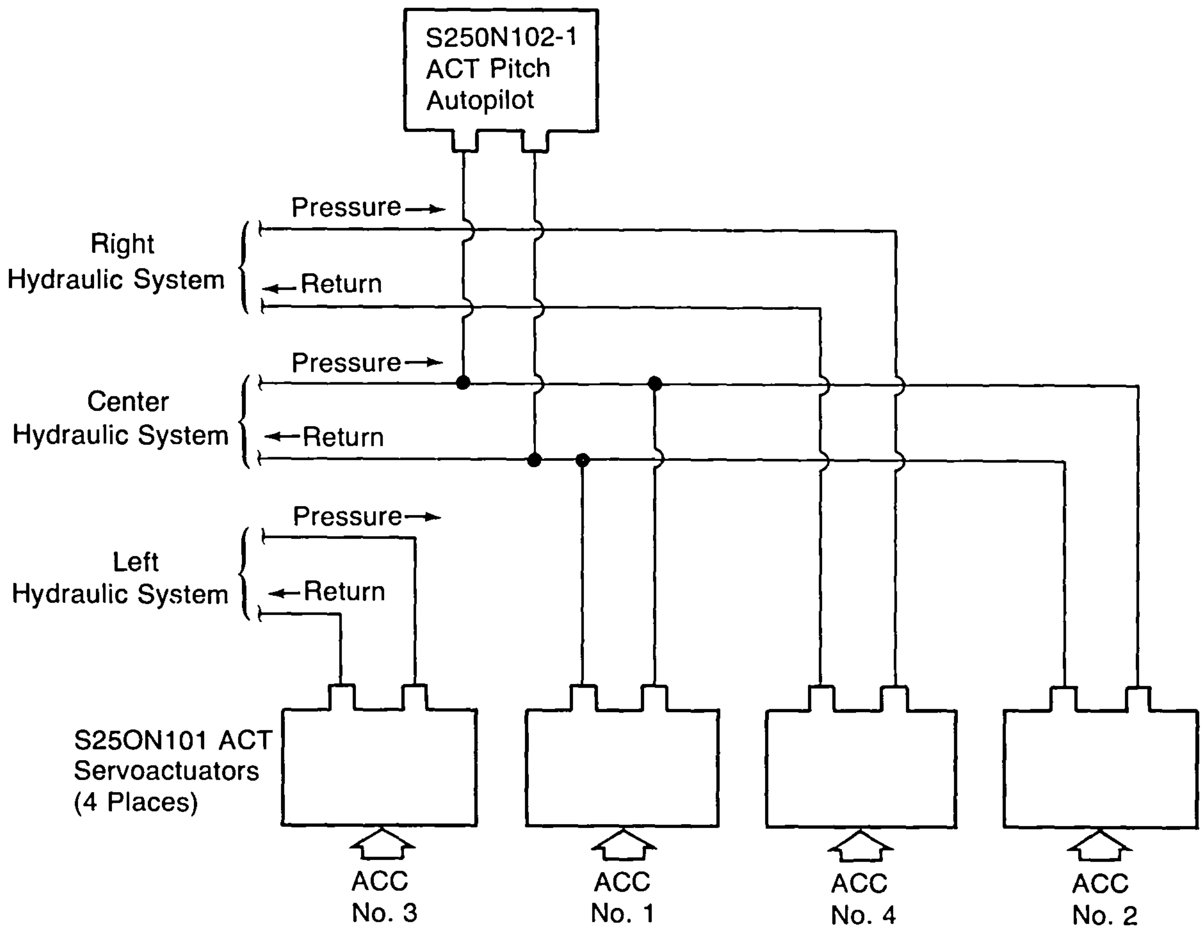


Figure 34. Test ACT System Hydraulic Schematic

8.1.3 SYSTEM DESIGN

Figure 35 shows the current 757 control column and forward tension regulator arrangement. The Test ACT flight deck installation is shown in Figure 36. A more detailed description of this installation is provided in Appendix D.

Figure 37 shows the current 757 empennage pitch control installation in isometric view.

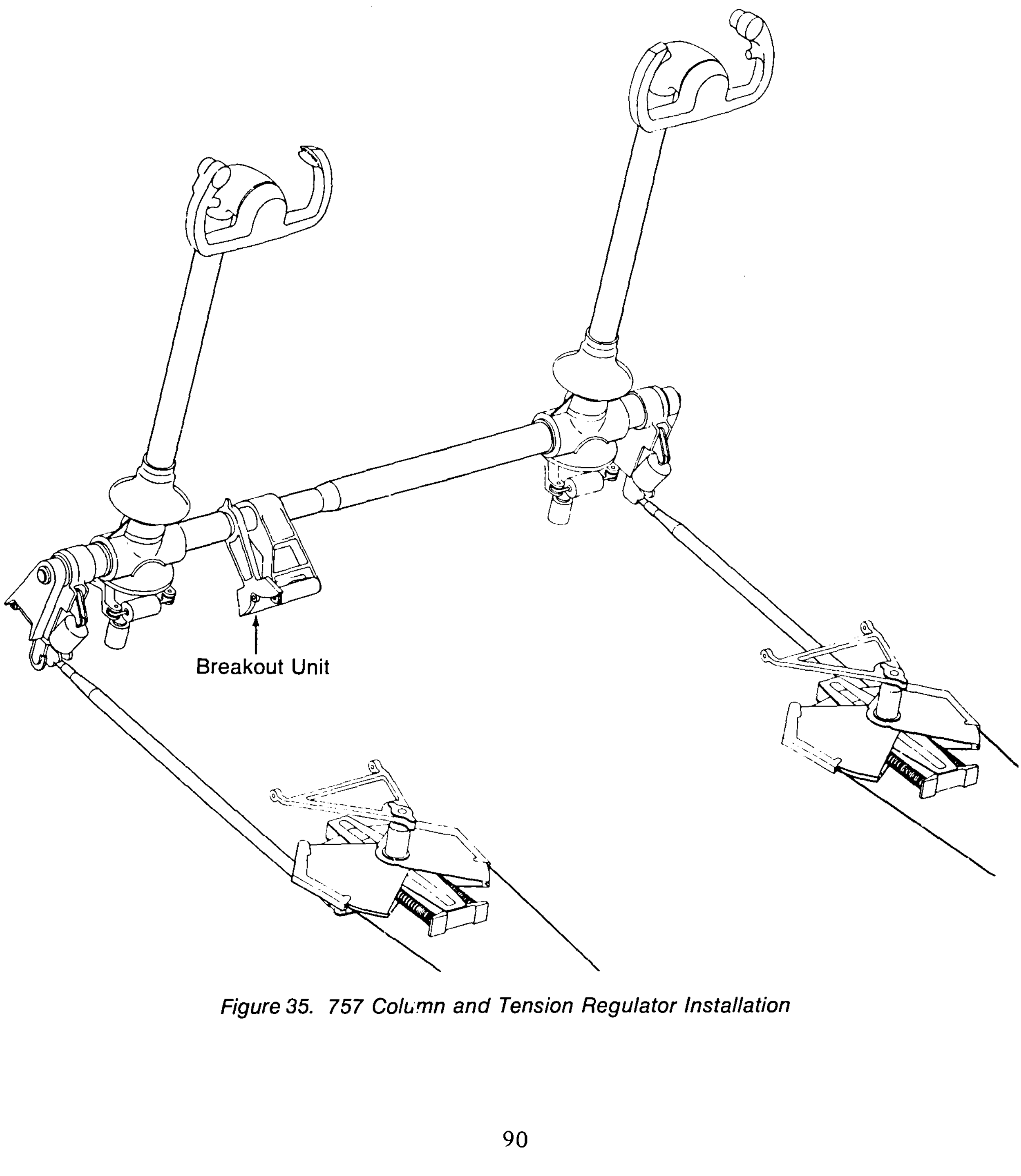


Figure 35. 757 Column and Tension Regulator Installation

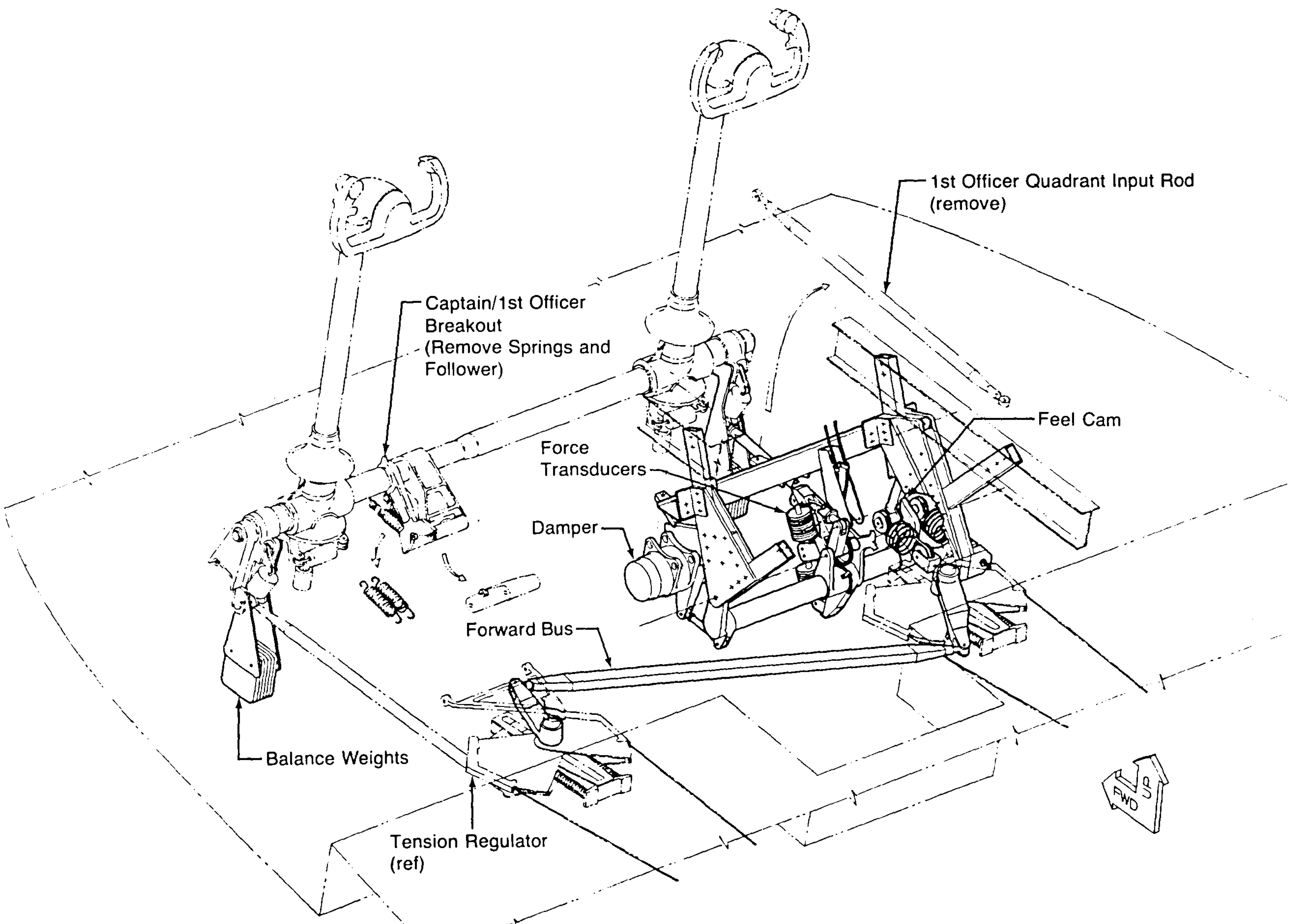


Figure 36. Test ACT Flight Deck Mechanical Installation

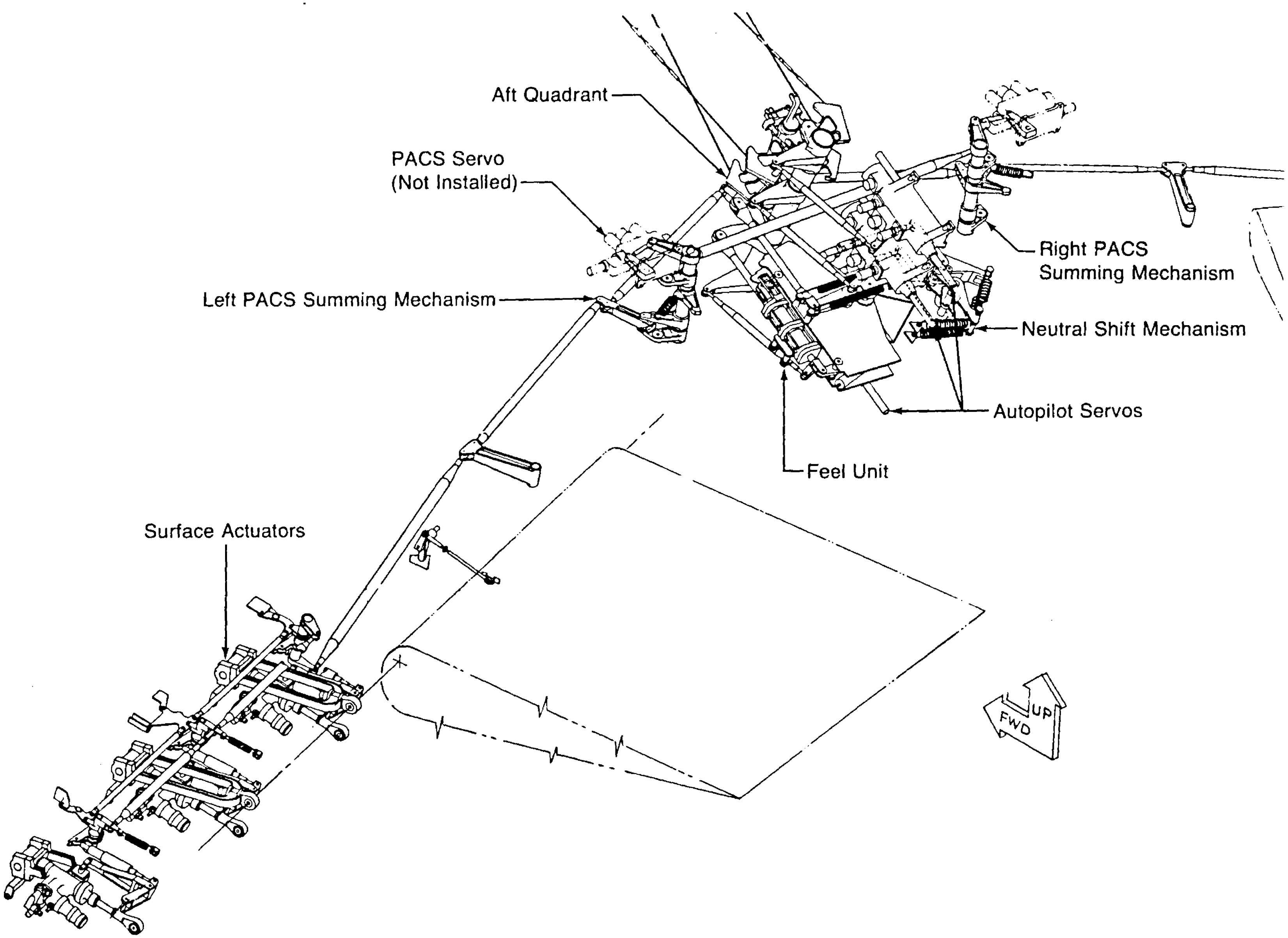


Figure 37. 757 Elevator Control System–Empennage Installation

The major changes effected by the Test ACT empennage installation are shown in Figure 38, in isometric form. A more detailed description of that installation may be found in Appendix D.

Requirements relating to the Test ACT secondary actuators, autopilot actuator, and primary actuators may be found in Section 4.2.1.

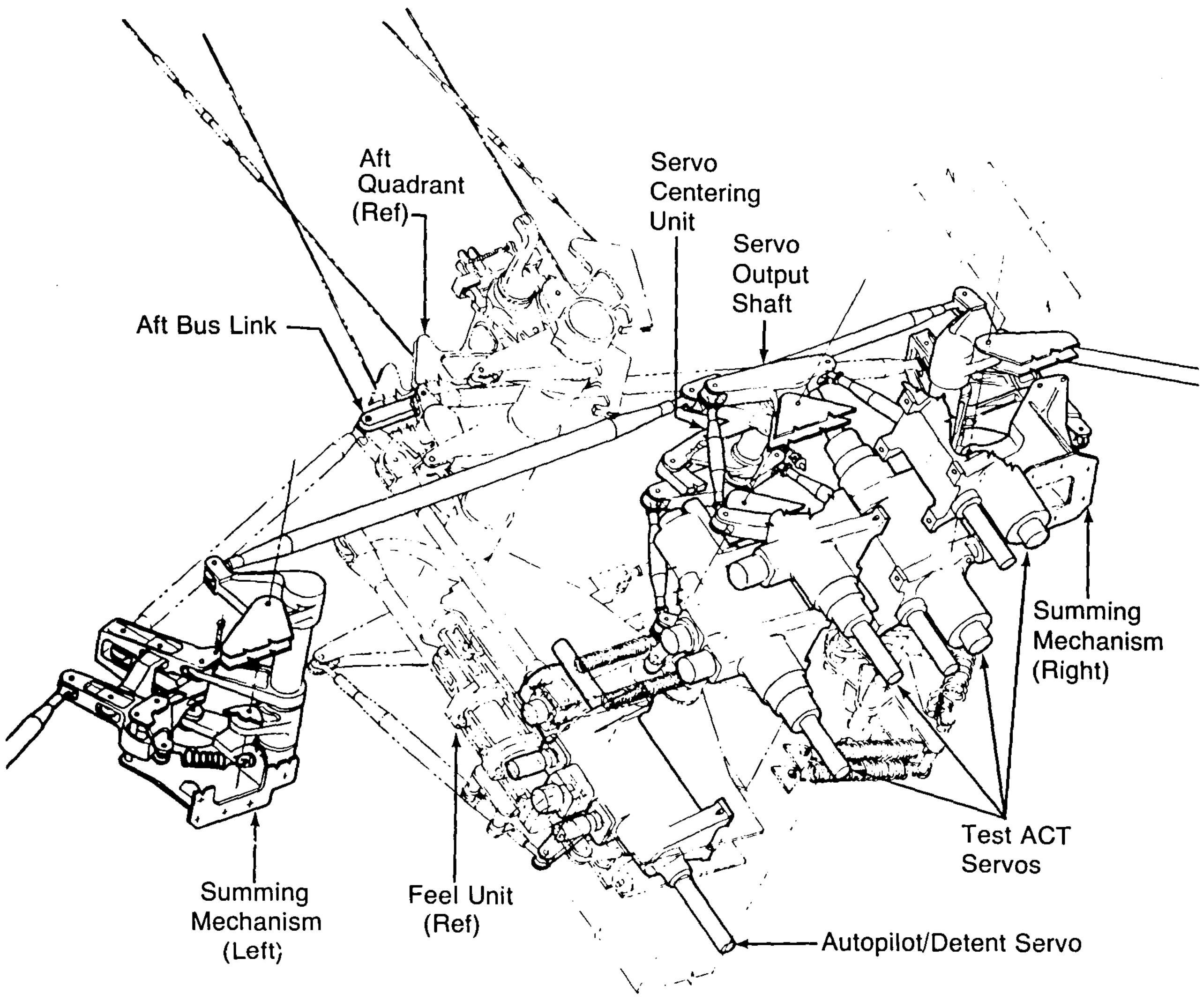


Figure 38. 757 Test ACT Elevator Control System–Empennage Installation

Actuators selected for use on the Test ACT airplane are units currently used on the 757, modified to comply with the requirements of Section 4.2.1. The selected actuators are:

- Test ACT servo (secondary actuator)–757 rollout guidance servo
- Test ACT autopilot/detent servo–757 pitch autopilot servo

In addition, the existing 757 elevator surface actuators are modified to add valve bias springs. A more detailed description of these actuator modifications is contained in Appendix D.

8.2 ELECTRIC EQUIPMENT INSTALLATIONS

Figure 39 is a plan view of the Test ACT airplane showing the general location of the equipment added to enable the Test ACT flight test. The following sections treat the installation of individual items.

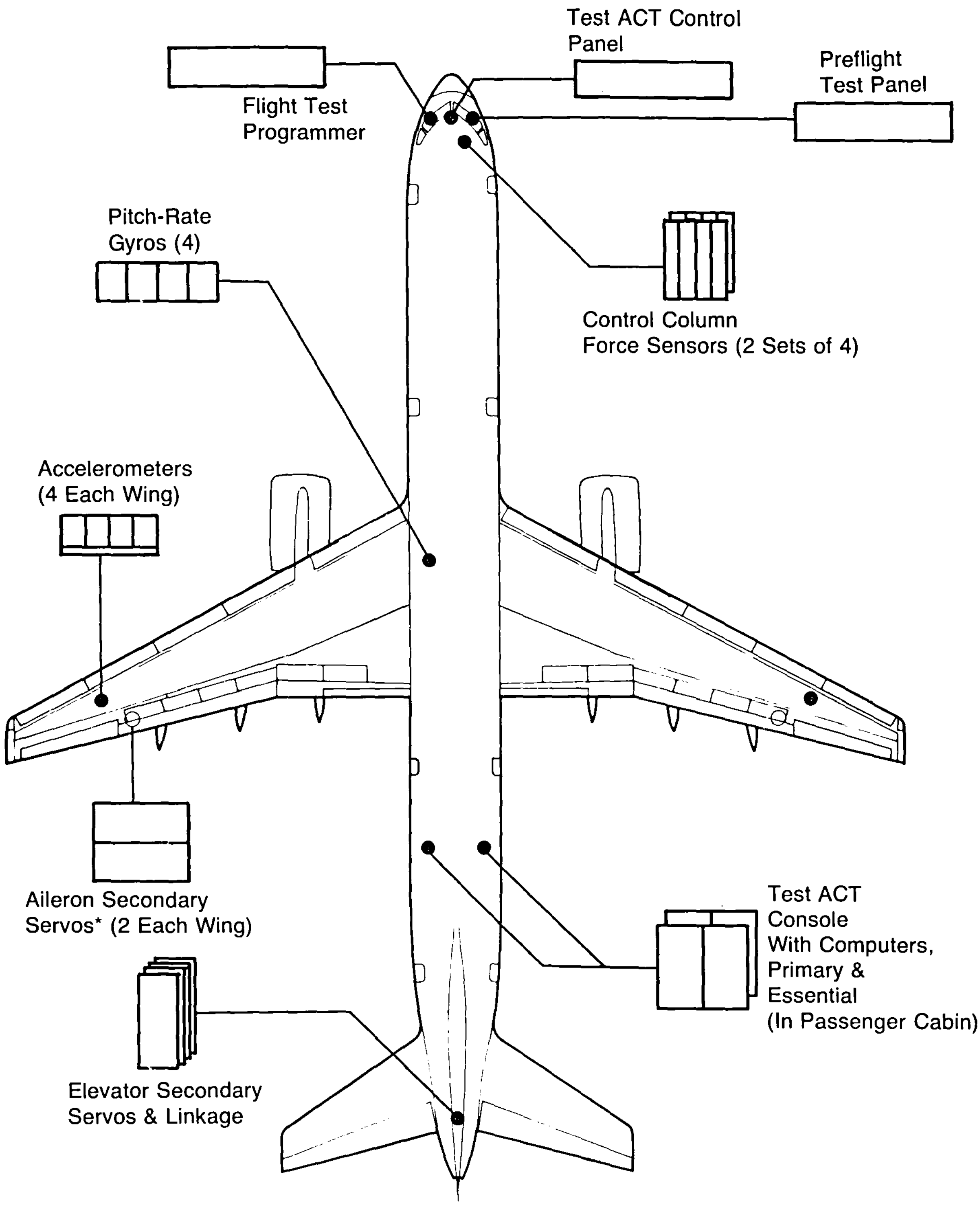
8.2.1 SENSORS

Installation of the three sets of sensors dedicated to the Test ACT System is as follows:

- Column force transducers–These force transducers are a part of the Test ACT modification of the mechanical flight control system and are treated in Sections 8.1.2 and 8.1.3.

- Pitch-rate gyros–The set of four pitch-rate gyros is mounted on the main cabin floor near the nominal airplane cg.

- Wing accelerometers–The installation of the wing accelerometers used in wing-load alleviation (WLA) instrumentation will be designed for the best compromise of high sensitivity to the first wing bending mode and suppression of the higher wing structural modes. Structural data derived in the 757 Airplane Project will be used to analyze wing bending and determine an optimum accelerometer location. The resulting position will probably be in the outboard wing at about 70% semispan on the front spar or rib. The accelerometer axis of sensitivity will be aligned perpendicular to the wing reference plane.



*Not part of initial installation

Figure 39. Components of Test ACT System–757

8.2.2 FLIGHT DECK PANELS

Figure 40 illustrates the location of the three Test ACT panels required in the flight deck. They will be accessible to both pilots from their normal seated position.

Two of the panels, the Test ACT Control Panel and the Preflight Test Panel (sec. 6.3.1), will be a part of the Production ACT System. They will be mounted in the pilots' overhead panel, as shown at the left in Figure 40. The Flight Test Programmer is applicable to Test ACT only and includes a "big red switch" that completely disables the system in case of emergency. This is located in the control stand just aft of the quadrant controls and opposite the pilots' elbows.

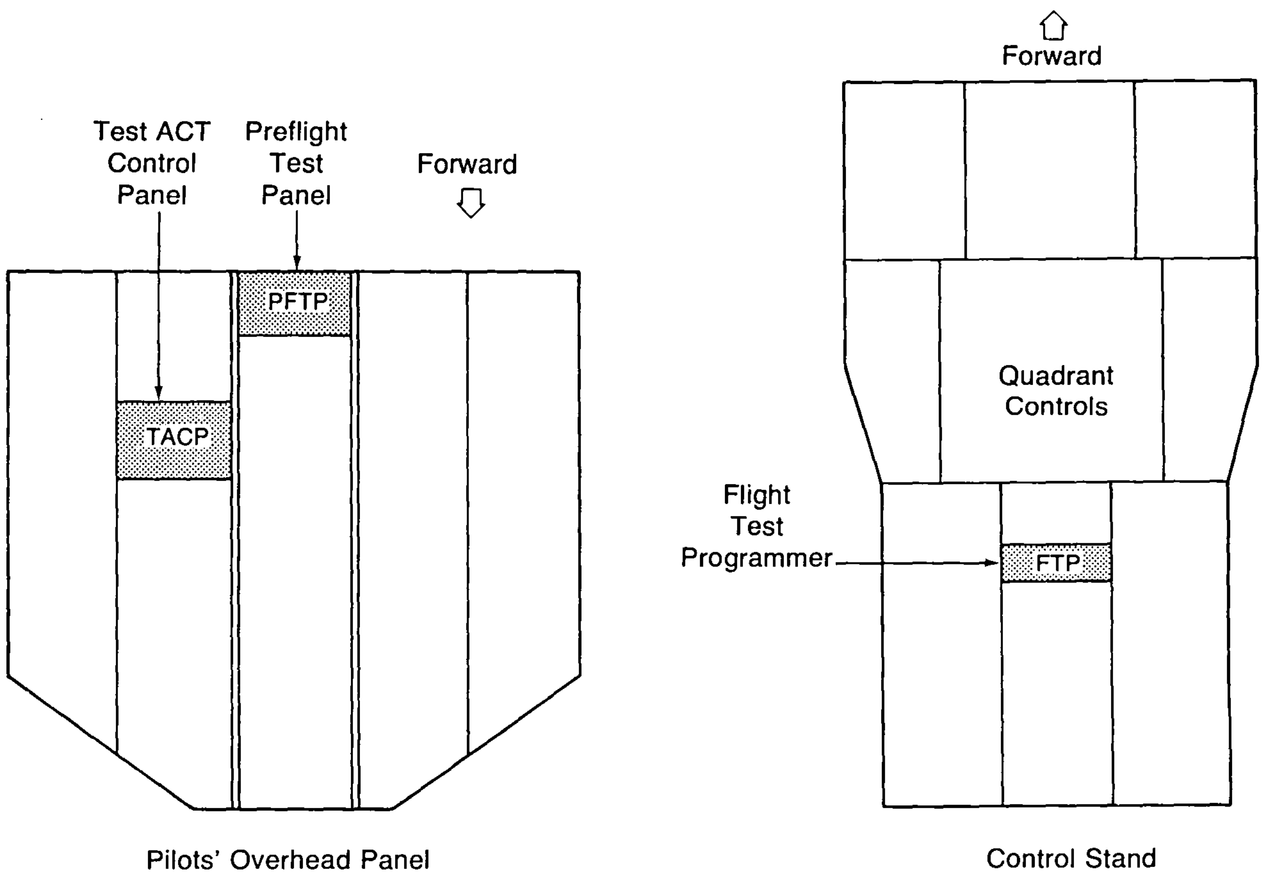


Figure 40. Control Panel Placement in the 757 Flight Deck

8.2.3 CONSOLE AND CABLES

Figure 41 illustrates the two possible arrangements of the Test ACT Console in the aft part of the main cabin. The location of console No. 1, on the left side of the cabin a short distance aft of the left emergency egress door, has been established. At this stage in the test aircraft operations planning, it is not possible to determine positively the location of console No. 2; two locations are shown in the diagram. The preferred location near console No. 1 and across the aisle will result in the shortest cable run and the best accessibility. The alternative location, labeled "worst case," will be provided for in the cable design.

The preferred position will have both consoles on the normal cabin floor level, making it necessary to provide the interconnecting cable with a sturdy protective cover. The cabling to the flight deck, the electronic equipment bay, the rate gyro box, and the wing accelerometers will all go forward from the consoles underneath the raised floor area, as is normal for cabling in the flight test airplane cabin.

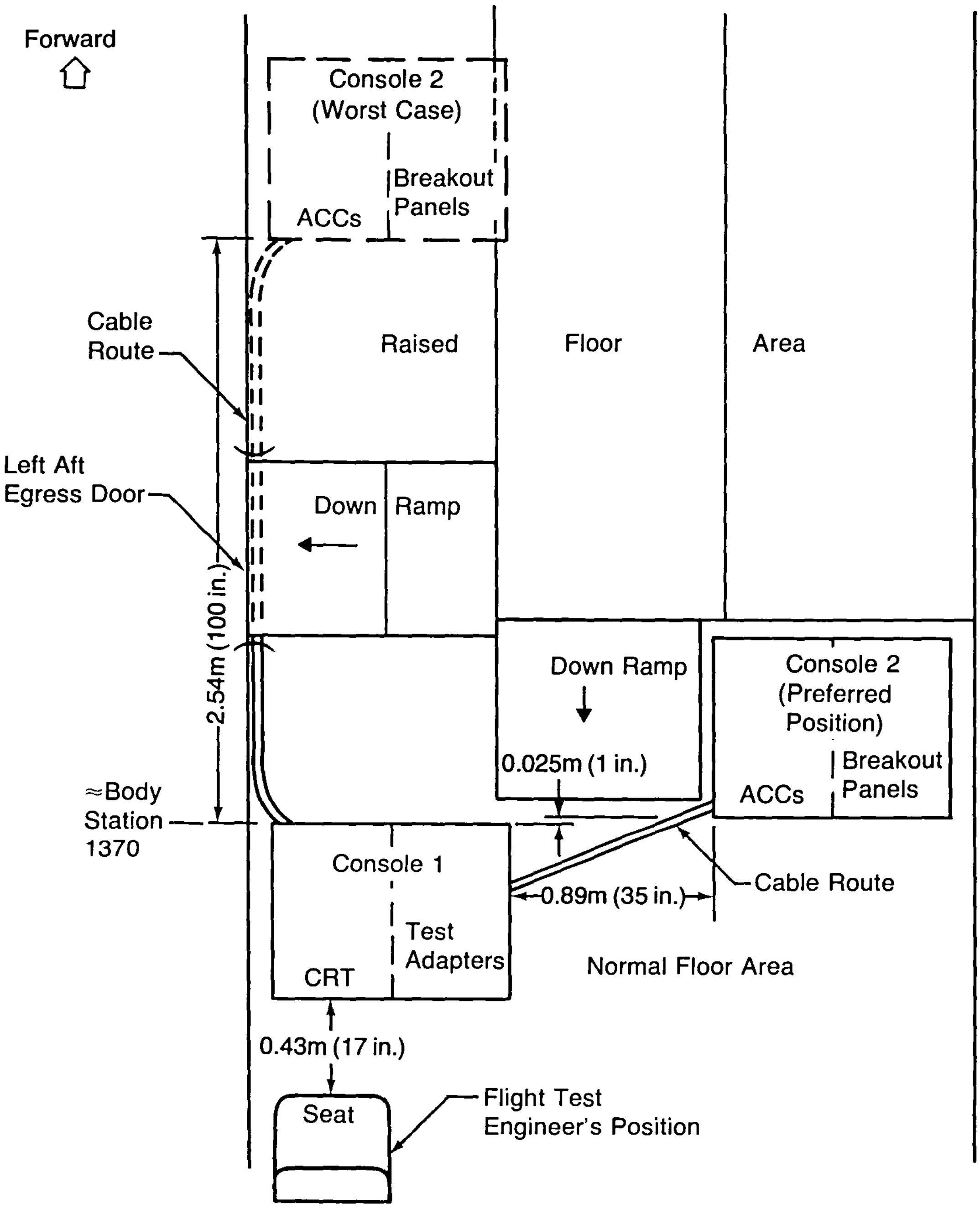


Figure 41. Test ACT Console Placement–757 Main Cabin

9.0 SYSTEM VERIFICATION

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9.0 SYSTEM VERIFICATION

System verification is the process used to determine whether or not the Test ACT System meets the requirements of the Specification Control Drawing (SCD), Volumes I and II.

9.1 VERIFICATION METHODS

Components of the Test ACT System were shown to meet the requirements of the SCD by means of five verification procedures (fig. 42). Application of this process demonstrated compliance with all SCD requirements, excluding the waivers noted in Section 9.6.2. Figure 42 also shows the documentation provided at hardware delivery to demonstrate compliance.

9.2 ANALYSES

Analytical methods were used to verify the following system features:

- Reliability, dispatchability, and safety
- Channel equalization
- Environmental impact
- Flightworthiness

Reliability analyses are treated in depth in Section 10.3. Environmental impact and flightworthiness were based largely upon similarity to the Autopilot/Flight Director System equipment for the 767 and 757 Programs.

9.3 INSPECTION

The Test ACT System hardware was inspected at appropriate steps in its manufacture by Quality Assurance representatives of both Collins and Boeing. The Boeing Quality Assurance department maintains a resident inspector at Collins in Cedar Rapids; he was available to inspect and approve the system equipment as it progressed through the system fabrication process.

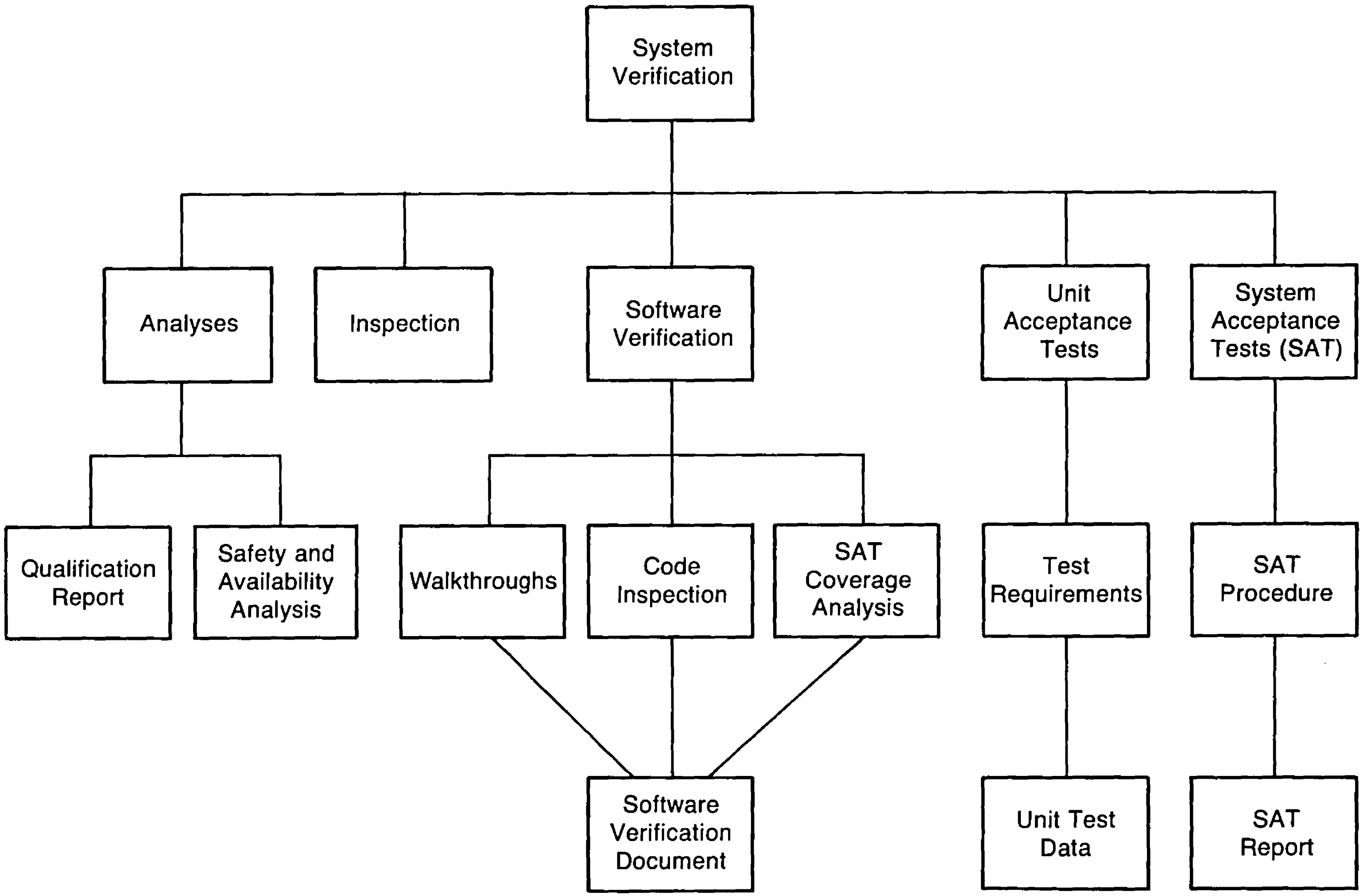

Figure 42. System Verification Plan

9.4 SOFTWARE VERIFICATION

The three software verification procedures included in Figure 42 were utilized to augment the Active Controls Computer (ACC) software verification, which was obtained in the System Acceptance Test (SAT). The following is a brief description of these procedures:

- Design "walkthroughs" verify that the software design, as documented in the Design Implementation document (DID), complies with the SCD and includes no unintended functions.

- The code inspections verify that the code faithfully implements the DID design.

- Analysis of the SAT coverage verifies that it executes all program branches.

The results of the software verification tasks were documented and submitted in the Software Verification document.

9.5 UNIT ACCEPTANCE TESTS

The Unit Acceptance Tests (UAT) were performed on selected line replaceable units (LRU) of the system and were designed to be capable of verifying that each LRU is fully functional. This testing was applied to the following units:

- Active Controls Computer
- Pitch-rate gyro
- Preflight Test Panel
- Wing linear accelerometer

The System Acceptance Test provided verification of the other LRUs.

9.6 SYSTEM ACCEPTANCE TESTS

The System Acceptance Tests verified the performance of the Test ACT equipment as a system and certain of the LRUs as units. These tests had the following characteristics:

- They emphasized end-to-end testing.
- Verification was based on easily observed system effects, such as servo disconnects and annunciations.
- Measurements required were limited to those that could be made through the Test ACT Console (TAC) breakout panel (which provides access to all the pins of the ACC rear connector) and the Collins Adaptive Processing System (CAPS) Test Adapters (which provide access to the transfer buses).
- Fault insertion was limited to power interrupts, disconnecting of equipment cables, simulated interface faults inserted at the breakout panel, and simulated processor faults inserted via the CAPS Test Adapters.

9.6.1 TEST CONFIGURATION

The System Acceptance Tests were performed with the test configuration shown in Figure 43. All of the airplane equipment and features necessary to provide inputs to and accept outputs from the Test ACT System are represented by the simulators that surround the ACCs in that figure.

9.6.2 ACCEPTANCE TEST RESULTS

The witnessed System Acceptance Test series was run at Rockwell-Collins in Cedar Rapids from 15 June through 24 June 1983. It included complete execution of the approved System Acceptance Test Procedure, plus some Unit Acceptance Tests on a complete ACC and phase and gain testing of an analog Essential computer. Those Unit Acceptance Tests were required to demonstrate that the system satisfies procurement specifications.

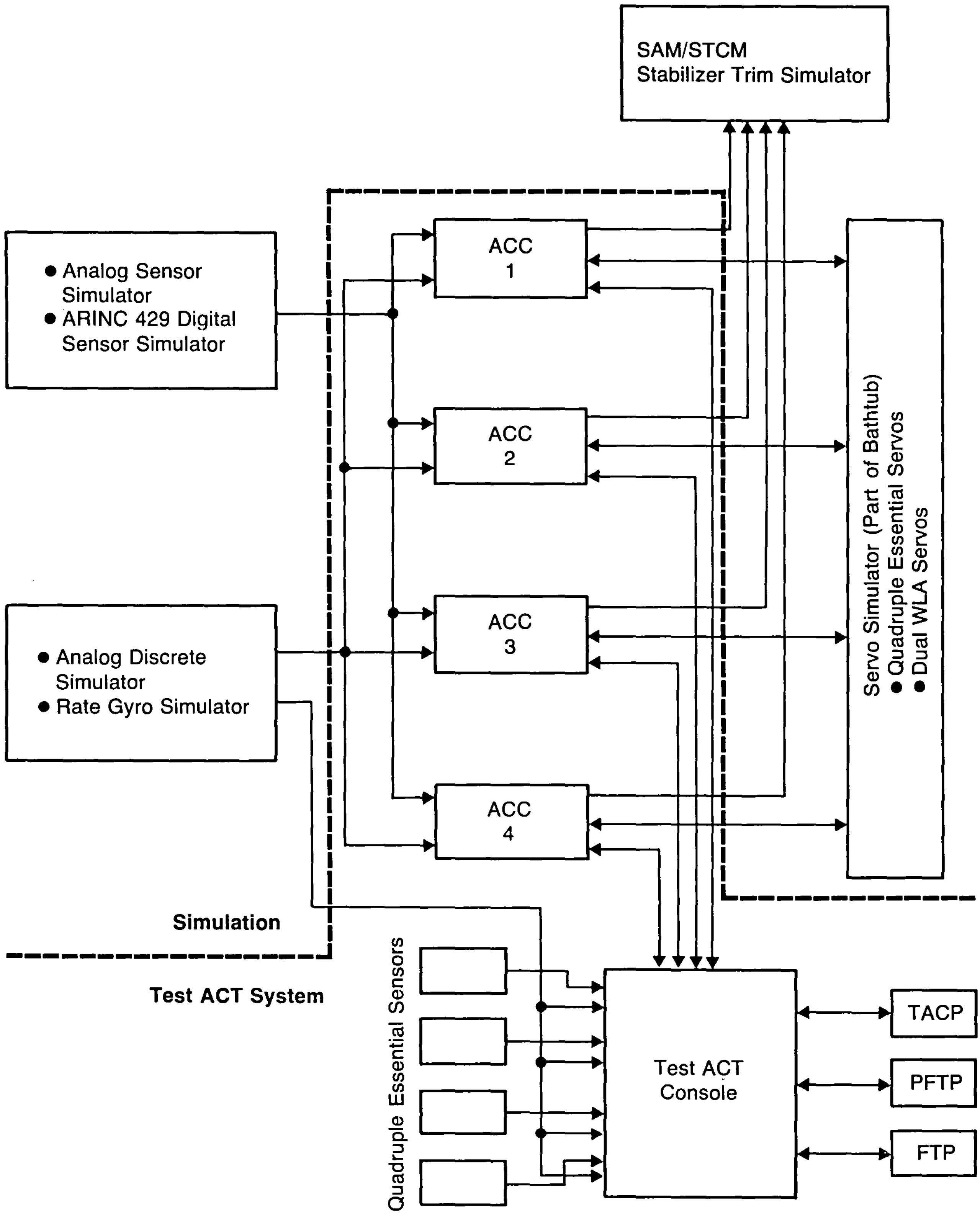

Figure 43. Configuration for System Acceptance Tests

Twenty-eight problem reports were recorded in the preceding procedure. Five of those were ascribed to minor specification shortcomings and hence were not charged to Collins. Several problems, most of them recorded in a single report, were traced to faulty jacks in the TAC breakout panels; their correction required rewiring of the panels, and that task carried over into the postdelivery period. The rest of the 28 problem reports were cleared before the second test period on 29 and 30 June 1983. In the same interim period, five additional problem reports were written; all of those were cleared except for one software change that will be handled by the Collins Service Center in Seattle.

In the second period of witnessed testing, the effectiveness of corrections to the problem reports per the previous listing was verified. Official acceptance notification was then transmitted to Collins, and the system was delivered to Boeing.

10.0 PERFORMANCE ANALYSIS

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10.0 PERFORMANCE ANALYSIS

10.1 TEST ACT AND PRODUCTION ACT

As configured for flight testing, the Test ACT System is not expected to meet production requirements for system and schedule reliability. This is because the test airplane has insufficient redundancy in certain equipment that cannot be augmented cost effectively. Test ACT includes a four-channel system of Primary and Essential computers and actuators, designed to function optimally with quadruple sensors and power supplies. The 757-200 test airplane has only two Digital Air Data Computers (DADC), three Inertial Reference Systems (IRS), and three hydraulic supplies. In addition, the Test ACT servo installation operates through existing mechanical linkage and surface actuators. This arrangement provides less jam protection and introduces greater hysteresis than a prospective Production ACT System, which would incorporate closely coupled servo control units and preserve complete isolation of left and right surface input paths.

In order to estimate the reliability of a Production ACT System designed for commercial operation, it was necessary to specify the assumed redundancy level of associated airplane systems. Because of the "brick-walled" architecture of four-channel Essential pitch-augmented stability (PAS), any equipment required by the Essential system, down to the summing bars, must necessarily be quadruplicated. This is not a requirement for the four-channel Primary system, which is not brick walled nor required for safety of flight.

During Test ACT development, alternative configurations were examined as candidates for a Production ACT System. These were:

- A system in which all Primary (as well as all Essential) components are quadruplicated. This approach increases system reliability, airplane schedule reliability, and safety but at the expense of higher initial and operating cost. As in the Test ACT System, the probability of erroneous control surface commands is minimized by disconnecting the Primary system if there is disagreement between the last two remaining operational channels.

- A system with triplex IRSs and DADCs and therefore lower cost than the all-quadruplex system. System reliability and schedule reliability are maintained by allowing the Primary system to function with only a single sensor of each type.

Protection against unacceptable surface transients, when degrading to single sensor operation, is provided by three independent techniques:

- Inline monitors are used to give a high degree of assurance that any single sensor selected will be the "correct" one.
- Hardware authority limiters in the Essential system limit the transient that can be induced by the Primary computers.
- Software authority limiters in the Primary computers limit the transient that can be caused by a malfunctioning sensor. Note that the signal selection algorithms in the Primary computers will eliminate all initial faults by comparison with the remaining "good" signals.

As shown in Table 3, the system with triplex IRSs and DADCs has been assumed as the Production ACT System for reliability estimation purposes. Section 10.2 deals with the Test ACT System; Section 10.3 is based upon the Production ACT System.

Table 3. Airplane Systems Redundancy for ACT

Item	Test ACT[a]	Production ACT
Digital Air Data Computers	2	3
Inertial Reference Systems	3	3
Flap position sensors	3	4
Stabilizer position sensors	3	3
28-V dc power sources	4	4[b]
Hydraulic systems	3	4[c]
Air-ground logic units	2	2
Mechanical surface input paths	1	2

[a]The redundancy shown in the Test ACT column is that of 757 NA001.

[b]The proposed fourth 28-V dc supply is a charger-floated battery.

[c]The proposed fourth hydraulic system is a small dedicated supply.

10.2 CONTROL LAW SYNTHESIS

10.2.1 OBJECTIVES

A general objective of the PAS control system is to enable an unstable airplane to meet the Federal Aviation Regulations (FAR), Part 25, Airworthiness Standards for Transport Category Airplanes (ref. 11). In addition, PAS was required to satisfy the same gain and phase margin requirements that have been imposed on the production 757 autopilot. These are:

Mode Frequency (f_m) (Hz)	Gain Margin (dB)	Phase Margin (deg)
$f_m < 0.06$	± 3.0	± 20
$0.06 \leqslant f_m \leqslant$ 1st aeroelastic mode	± 4.5	± 30
$f_m >$ 1st aeroelastic mode	± 6.0	± 45

The modal frequency of the first aeroelastic mode of the 757 is approximately 2 Hz.

Additional constraints were applied to the airplane transient response to a step column input, as shown in Figures 44 and 45. The resulting pitch rate, normalized with respect to the steady-state pitch rate, was required to be within the specified boundaries. When a steady-state pitch-rate response to a constant column input does not exist, a certain amount of "engineering judgment" has to be used with this criterion.

As described in Section 10.2.2, the normally operating PAS is referred to as the Primary PAS. A highly reliable submode, termed the Essential PAS, is also included to provide get-home capability in the event of Primary PAS failure. Both augmentation modes have been evaluated on a piloted moving-base simulator and were required to meet alternative flying quality standards, as expressed in Figure 4. The Primary PAS was expected to provide Level 1 flying qualities (defined in sec. 10.2.2.1) at all permissible center-of-gravity (cg) locations throughout the flight envelope. Similarly, the Essential PAS was required to be at least Level 3.

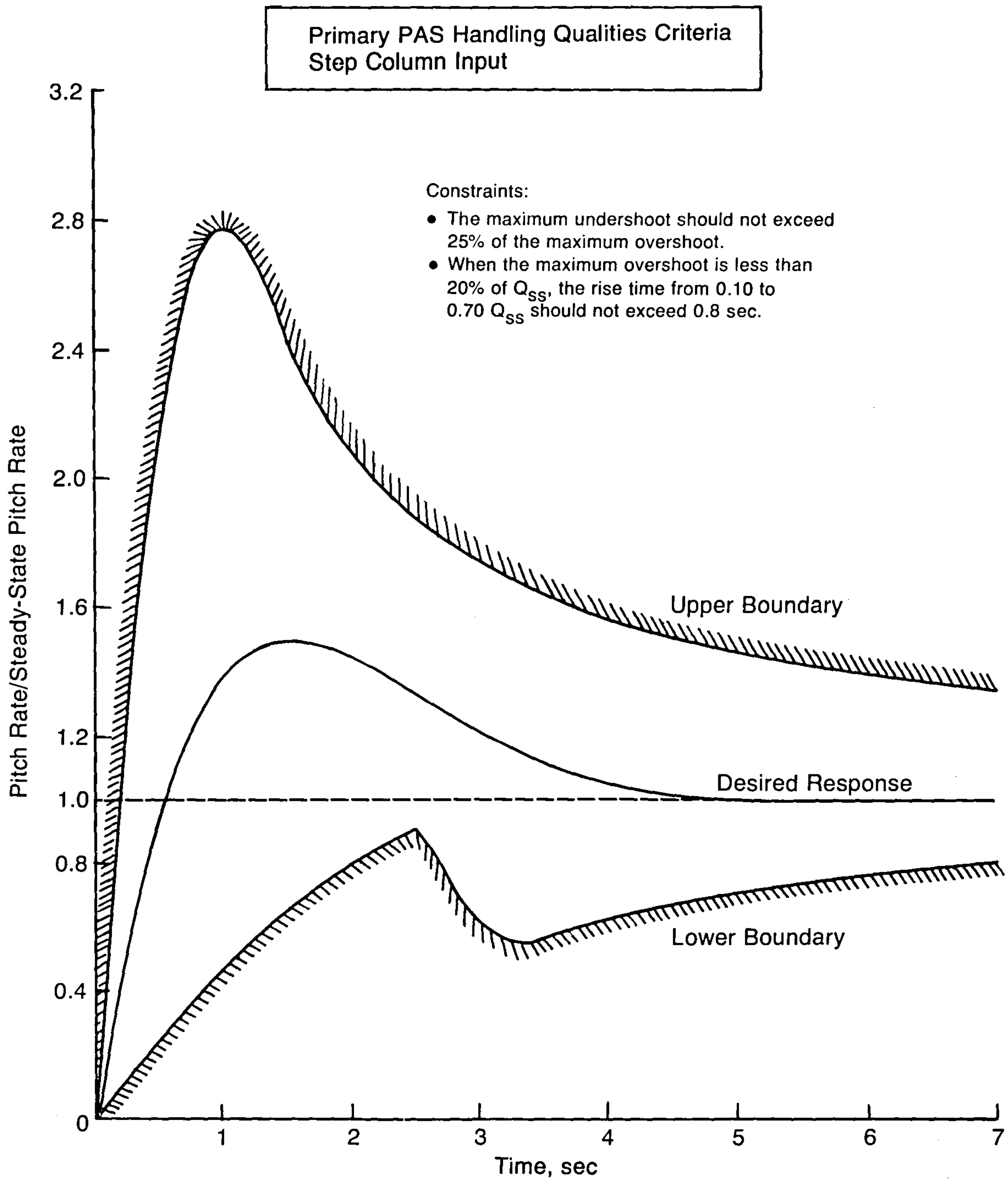


Figure 44. Handling Qualities Criteria–Low-Speed Conditions

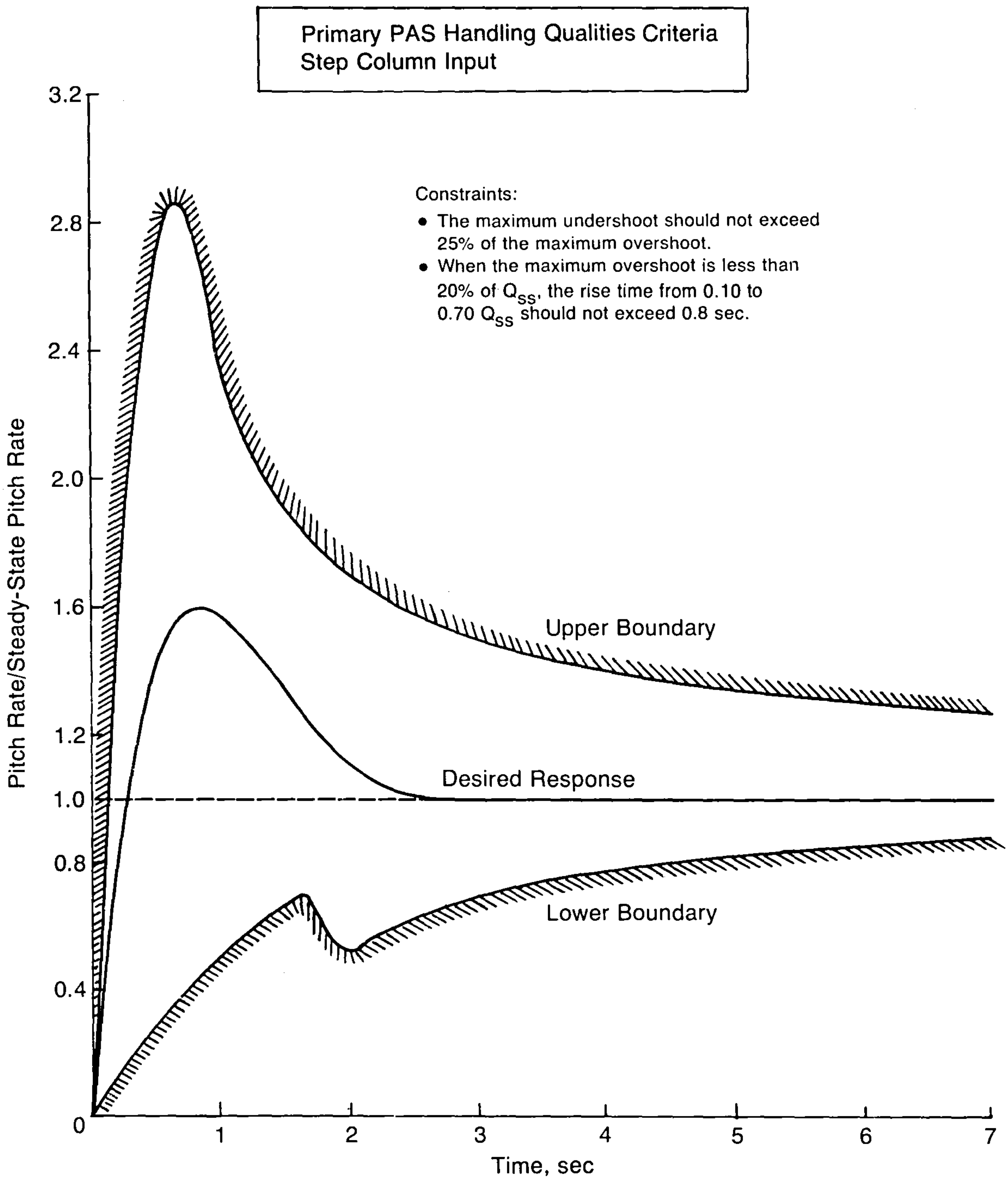


Figure 45. Handling Qualities Criteria–Cruise Conditions

Objectives for the wing-load alleviation (WLA) control system were less stringent. Whereas PAS was designed to be flight tested on a 757 airplane with artificially degraded stability, the WLA control law was developed principally to size the Test ACT hardware and software. Flight testing of WLA will be restricted to open-loop evaluation; i.e., servo commands will be derived from appropriate sensors but will not be used to drive control surfaces.

10.2.2 METHODS

10.2.2.1 Pitch-Augmented Stability Synthesis

The production 757 has a cg range from 7% to 39% mean aerodynamic chord (MAC). PAS synthesis was based on the assumption that a similar but aft-shifted cg range would be required of an ACT airplane. The control law was therefore developed for the 757 with an assumed cg range of 20% to 50% MAC. Table 4 indicates the flight conditions at which PAS was evaluated. Included is a cruise flight condition with the cg at 55% MAC. This was added since it is the probable aft limit achievable with the projected cg control system to be used during flight testing. It is also the most unstable flight condition of those evaluated.

PAS control laws were developed to provide the following features:

- Partitioning into critical and crucial functions
- Restoration of conventional stability
- Fly-by-wire (FBW) control
- Authority limiting of the critical elements
- Automatic offloading of long-term elevator commands to the stabilizer
- Compensation for excessive stick forces when in a banked turn

Partitioning Into Critical and Crucial Functions—Functional partitioning was selected to be compatible with the reliability requirements of Section 10.3. An underlying assumption was that it is not currently feasible to validate software with a confidence level that would satisfy an "extremely improbable" (ref. 11) failure requirement of 10^{-9} failures per flight hour. The crucial aspects of the stability augmentation, together with a minimal

Table 4. PAS Design Flight Conditions

Condition	Weight 10^3kg (10^3lb)	Altitude 10^3m (10^3ft)	Mach	q_c 10^3Pa (lb/ft^2)	cg (% MAC)	Remarks
HIGH-SPEED CONDITIONS						
CRU 1	83.5 (184)	10.7 (35)	0.82	13.26 (277)	20	
CRU 3	83.5 (184)	10.7 (35)	0.82	13.26 (277)	50	
CRU 4*	83.5 (184)	10.7 (35)	0.84	14.03 (293)	20	
CRU 6	83.5 (184)	10.7 (35)	0.84	14.03 (293)	50	
CRU 7	83.5 (184)	10.7 (35)	0.84	14.03 (293)	55	Probable flight test limit
CRU 11	74.8 (165)	3.0 (10)	0.46	10.68 (223)	50	Maximum thrust condition
CRU 14	83.5 (184)	7.6 (25)	0.50	6.51 (136)	50	1.2 V_S
CRU 16	83.5 (184)	12.8 (42)	0.82	9.48 (198)	50	1.2 V_S at maximum altitude
CRU 20	83.5 (184)	8.2 (27)	0.86	21.4 (447)	50	V_{MO}/M_{MO} corner
LOW-SPEED CONDITIONS			EAS m/sec (kn)			
APP 4	73.5 (162)	0.30 (1.0)	63.3 (123)	2.44 (51)	50	Landing flaps
APP 5	89.8 (198)	0.30 (1.0)	68.9 (134)	2.92 (61)	20	Landing flaps
APP 6	89.8 (198)	0.30 (1.0)	68.9 (134)	2.92 (61)	50	Landing flaps
APP 8	73.7 (162)	0.30 (1.0)	68.9 (134)	2.92 (61)	50	Go-around power
APP 11	99.8 (220)	0.30 (1.0)	74.6 (145)	3.40 (71)	50	Takeoff flaps

*Also WLA design flight condition.

level of FBW control, were therefore implemented in an analog Essential controller, with additional stability and control enhancement generated by a digital Primary control system.

Primary PAS commands to the elevator are first transferred to the Essential controller so authority limiting of the digital signals can be implemented independently in analog hardware.

Restoration of Conventional Stability—The principal stability augmentation requirements of the Test ACT System are restoration of acceptable short-period characteristics and speed stability. Pitch-rate feedback to the elevators is used to provide the former, with airspeed errors driving both elevators and the stabilizer to provide the latter. Control laws were developed using classical root locus techniques. The procedure is illustrated in Figures 46 through 48, which show the roots of the unaugmented airplane at a cruise flight condition and successive closures of pitch-rate and airspeed control loops. As shown in Figure 46, the principal result of shifting the aft cg limit from the production 757-200 value of 39% MAC to a Test ACT location of 50% to 55% MAC is to convert the pair of short-period oscillatory roots into a pair of roots on the real axis. At a cg location of 50% MAC, one of these roots is unstable, with a time-to-double amplitude of 8 sec. Included in the figure are root location boundaries that correspond to various levels of predicted handling qualities. These are shown as "Levels 1, 2, and 3" and are defined as follows:

- Level 1—Clearly adequate for the mission or flight phase evaluated

- Level 2—Adequate to accomplish the mission or flight phase but with objectionable deficiencies

- Level 3—Controllable but deficient for mission performance

As shown in Figure 46, two types of boundaries are imposed on an airplane which has roots that split into the classical short-period and phugoid complex pairs. It is evident that the production 757 at its aft cg limit of 39% MAC meets the Level 1 short-period frequency requirement of at least 1.2 rad/sec and satisfies the damping requirement of $\zeta > 0.35$. As the cg is moved aft, the criterion is not met, and pitch stability augmentation is required.

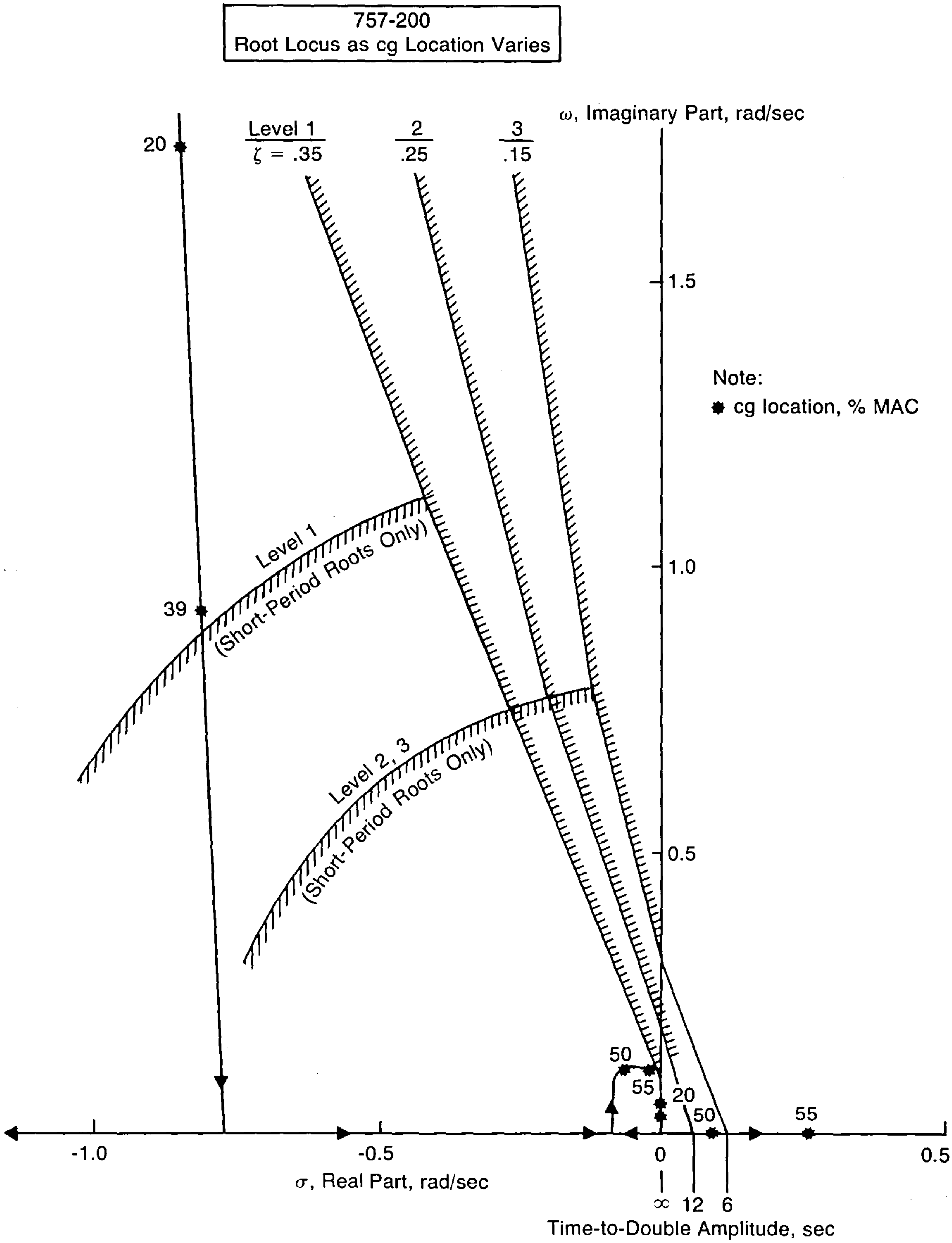


Figure 46. Effect of cg Location on Unaugmented Longitudinal Roots–Cruise

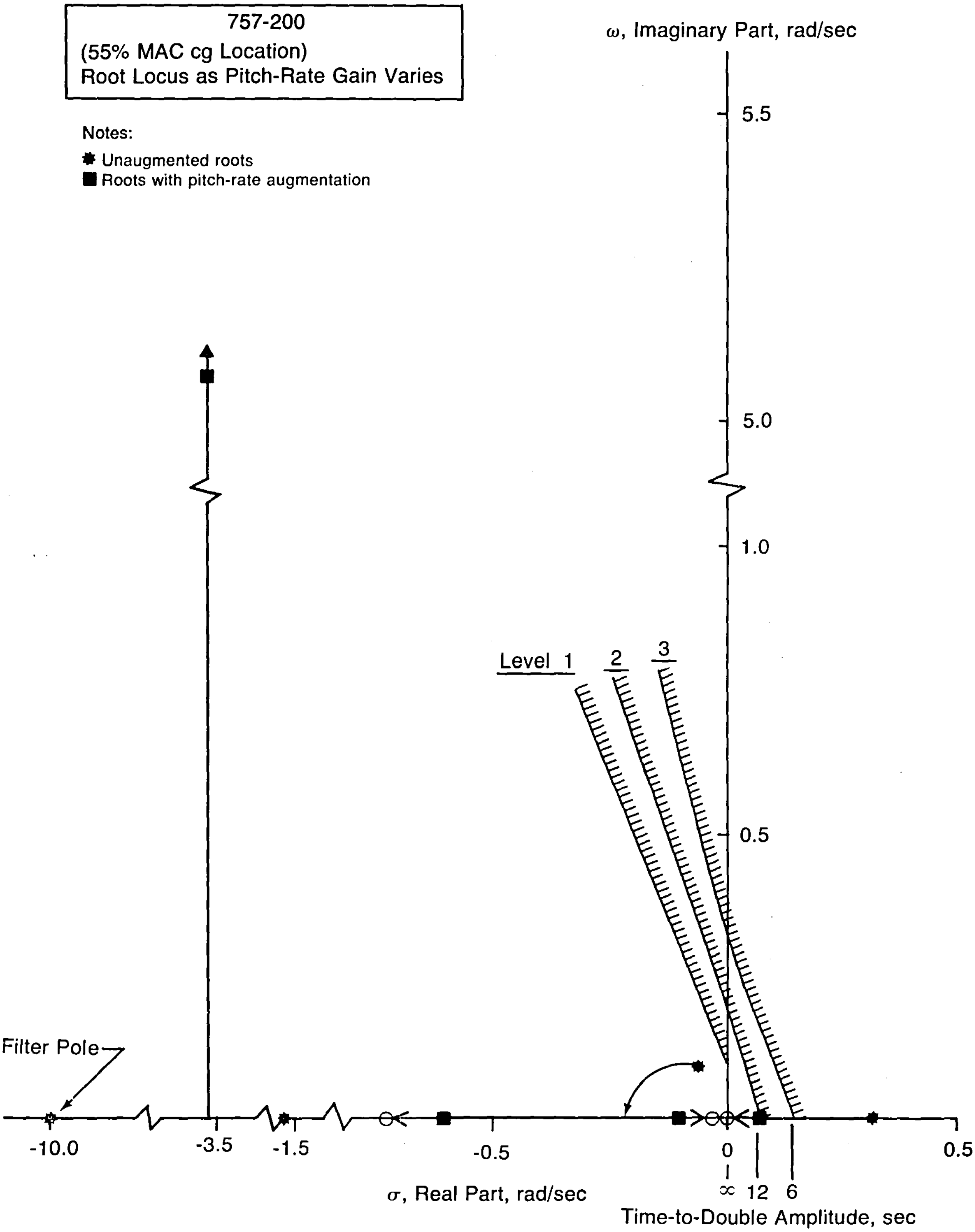


Figure 47. Effect of Pitch-Rate Augmentation at Aft cg Limit in Cruise

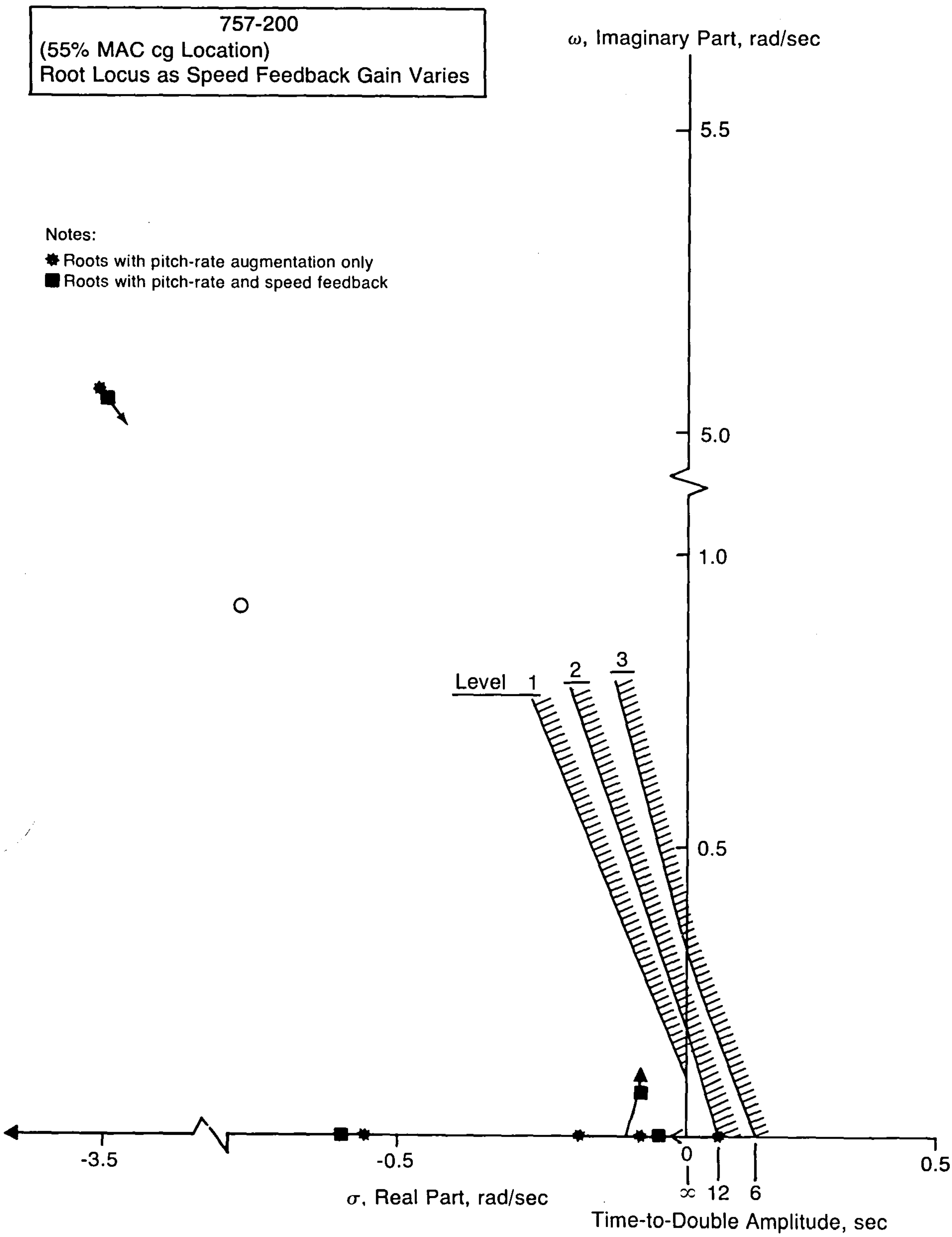


Figure 48. Addition of Speed Stabilization to Pitch-Rate Augmented System–Cruise

Figure 47 shows the effect of adding "lagged" pitch-rate feedback when the cg is at 55% MAC. In this case, the lag was provided by a first-order lag filter with a break frequency of 10 rad/sec. As pitch-rate feedback gain is increased, the unstable unaugmented root is driven toward the left half-plane but cannot be stabilized completely due to the "zero" at the origin. At the nominal pitch-rate feedback gain, K_{Q1} = 1.25 deg/deg/sec for flaps up, the time-to-double amplitude has changed from 2.5 sec (unaugmented) to 12 sec (augmented), thereby meeting the Level 2 stability requirement. Note that the roots normally associated with the short period have been relocated (at nominal pitch-rate feedback gain) such that their frequency is 6 rad/sec. Although this is considerably larger than the Level 1 requirement of Figure 46, the higher value is necessary since the short-period response is no longer dominated solely by the short-period roots; i.e., the roots on the real axis now have significant residues.

Figure 48 shows the effect of adding airspeed error feedback to the partially augmented system of Figure 47. As shown, the unstable root is stabilized at nominal gain, and a pair of complex roots, similar to the original phugoid roots, is restored. The airspeed error computation is derived in the digital control system and provides stability consistent with Level 1 handling qualities. The pitch-rate feedback of Figure 47 is provided by the Essential analog controller such that Level 2 handling qualities are ensured in the event of failure of the digital PAS.

Fly-by-Wire Control—Control inputs to the elevator are derived from force sensors on both the pilot's and copilot's control columns. Because these fly-by-wire commands are the only means whereby column deflections are routed to the elevator, the FBW system must necessarily have full authority. This requirement tends to conflict with the need (discussed in the next paragraph) to limit the authority of the digital Primary PAS. If all of the required FBW gain scheduling resided in the Essential controller, the high-reliability requirements of this analog PAS would be difficult to meet. As a minimum requirement, the Essential PAS must contain sufficient FBW gain scheduling to retain adequate control after Primary PAS failure. Piloted simulation studies indicate that the minimum Essential FBW gain scheduling should consist of a two-state gain change, with a low gain when flaps are up and a higher gain when flaps are extended. The remaining gain adjustments are provided by the Primary PAS. The flaps-up gain is 0.034 deg of elevator per newton (0.15 deg per pound) column force and this increases to 0.10 when flaps are down.

Authority Limiting of the "Critical" Elements—As mentioned previously, the control system architecture evolved on the assumption that protection must be provided against the possibility of generic digital control failures in which all channels simultaneously produce an erroneous hardover elevator command. Because the probability of this fault occurring cannot currently be quantified, the Primary PAS commands are limited in the Essential PAS. The criterion selected to size the authority limit was that at any flight condition and cg location within the normal operating flight envelope, the Primary PAS should not be capable of producing more than a 1g increment in normal load factor in the 3 sec following fault occurrence. It is obvious that Essential PAS reliability would be improved if this limiter could be implemented with a fixed value. Unfortunately, the limit value necessary for protection at high speeds is less than the maneuvering authority required at low speed. An acceptable compromise was found to be a fixed authority of ± 2.5 deg of elevator when flaps are up, and limits of ± 7 deg of elevator whenever the flaps are in any of the extended positions.

Automatic Offloading of Long-Term Elevator Commands to the Stabilizer—Speed stability is obtained from the Primary PAS by computing an elevator command that is proportional to true airspeed "error." This error is first computed during climb-out and is initially referenced to the climb speed. Subsequently, the airspeed reference is updated each time the pilot activates the manual trim button. During periods of infrequent trim switch usage, the possibility exists that the elevator command from airspeed error could exceed the authority limit of the Primary PAS. Because this would cause the Primary PAS to be temporarily ineffective, the airspeed error signal is automatically offloaded to the stabilizer. This offloading occurs whenever the elevator command from airspeed error exceeds ± 0.5 deg.

Compensation for Excessive Stick Forces When in a Banked Turn—One of the consequences of using pitch-rate feedback for short-period augmentation is that the steady-state pitch rate in a maneuver tends to be proportional to the applied control column force. Pitch rate associated with a symmetrical pullup maneuver is less than that for a banked turn maneuver of equal normal load factor. Therefore, to maintain identical stick-force-per-g characteristics in pullups and bank turns, pitch-rate compensation based on roll angle must be introduced.

10.2.2.2 Wing-Load Alleviation Synthesis

The WLA control laws were developed only to demonstrate the validity of the computer architecture and were not intended to confirm control law effectiveness. Nevertheless, sufficient analysis was done at a single cruise flight condition to verify the concept.

Maneuver-Load Control—The maneuver-load control (MLC) system was designed to drive the ailerons such that wing bending moments are reduced during maneuvers. Normal load factor, as measured by the IRS, is used to command aileron deflections so that wing loads are transferred inboard when the airplane is out of trim. Gains were selected to fully utilize the effective range of aileron deflection when maneuvering at limit load (1.5g incremental). A 10-rad/sec low-pass filter was included to avoid exciting the wing first bending mode (at approximately 12.5 rad/sec).

In a maneuver, the MLC-controlled ailerons deflect symmetrically to relieve wing outboard bending moments. With wings of high sweep, these aileron deflections can produce a significant pitching moment. To counteract this, the MLC aileron command is also used to apply elevator balancing deflections.

Gust-Load Alleviation—The gust-load alleviation (GLA) system was designed to reduce gust loads by suppressing the wing first bending mode. The system uses wing-mounted accelerometers that sense accelerations normal to the wing and commands aileron deflections that produce mode damping. The accelerometers are located to minimize the sensing of elastic modes other than the first bending mode. Isolation from higher frequency modes was also obtained by including a 20-rad low-pass filter in the command path.

10.2.3 RESULTS

Table 4 indicates the flight conditions at which the PAS control law was evaluated. Because the WLA control law is presently a conceptual study only (as opposed to the PAS, which is intended to be flight proven), WLA was evaluated at just a single flight condition—condition CRU 4 of Table 4.

10.2.3.1 Pitch-Augmented Stability

Figures 49 through 51 summarize the Primary PAS stability margins at the flight conditions evaluated. Margins were calculated for each of the individual feedback loops—airspeed, pitch rate, roll and pitch attitude—and for the loops in combination. Only the most critical stability margins are plotted on the figures. Also included in the figures are the required gain and phase margin boundaries specified in Section 10.2.1. As shown, the Primary PAS provides adequate stability at all flight conditions, except for conditions APP 8 and APP 11. At APP 8, the low-frequency gain margin is -2.4 dB rather than the required -3 dB. Thus, if the loop gain at APP 8 were reduced by 3 dB, the airplane would have an unstable root with a time-to-double amplitude of 177 sec instead of being neutrally stable, as implied by the gain margin requirement. For a trimmed flight condition, this would be unacceptable. However, since APP 8 is an untrimmed "go-around-power" condition, the margin deficiency is not significant.

APP 11 is a maximum weight takeoff condition. The configuration has speed stability but with only a -0.2-dB gain margin instead of the required -3 dB. This will not compromise the projected flight test program since takeoffs will be constrained to the nominal cg range of 7% to 39% MAC. Furthermore, it is questionable whether or not speed stability is meaningful at this flight condition, since at takeoff thrust the aircraft is untrimmed and accelerating. If required, the margin could be met for a production system by increasing airspeed error feedback by 35%, at the expense of a slight degradation in the pitch-rate response to column commands.

Stability margins were not evaluated for the Essential PAS, since this PAS did not completely stabilize the airplane at most of the extreme aft-cg flight conditions. Because the Essential PAS is required to provide only "get-home" capability, the acceptability of this mode was determined by piloted simulation rather than analytical considerations. Results of the Essential PAS piloted simulation study are documented in Reference 9.

10.2.3.2 Transient Response to Pitch Commands

The pitch-rate responses to step column inputs were within the acceptable bounds of Figures 44 and 45 at all flight conditions, when the airplane is augmented with the

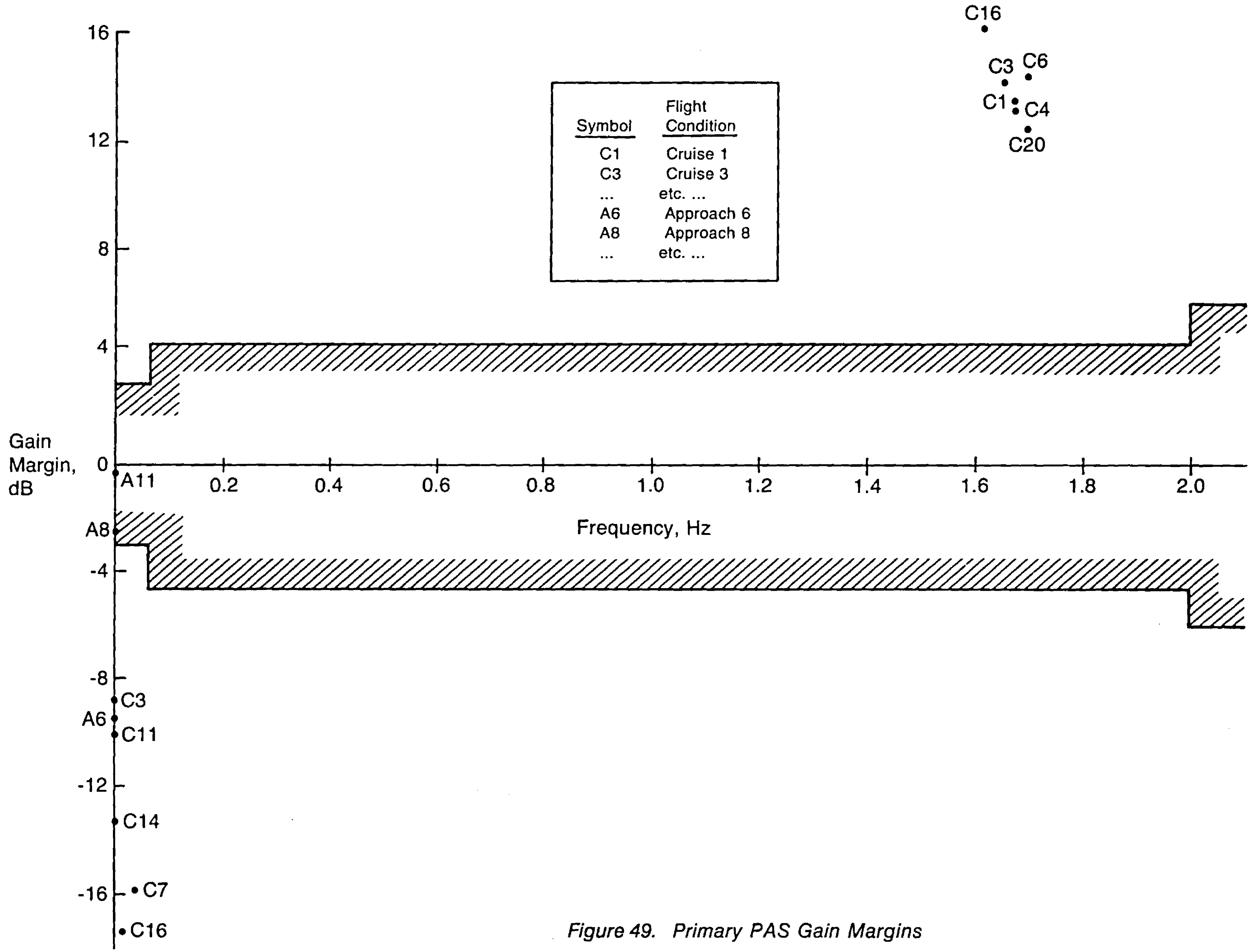


Figure 49. Primary PAS Gain Margins

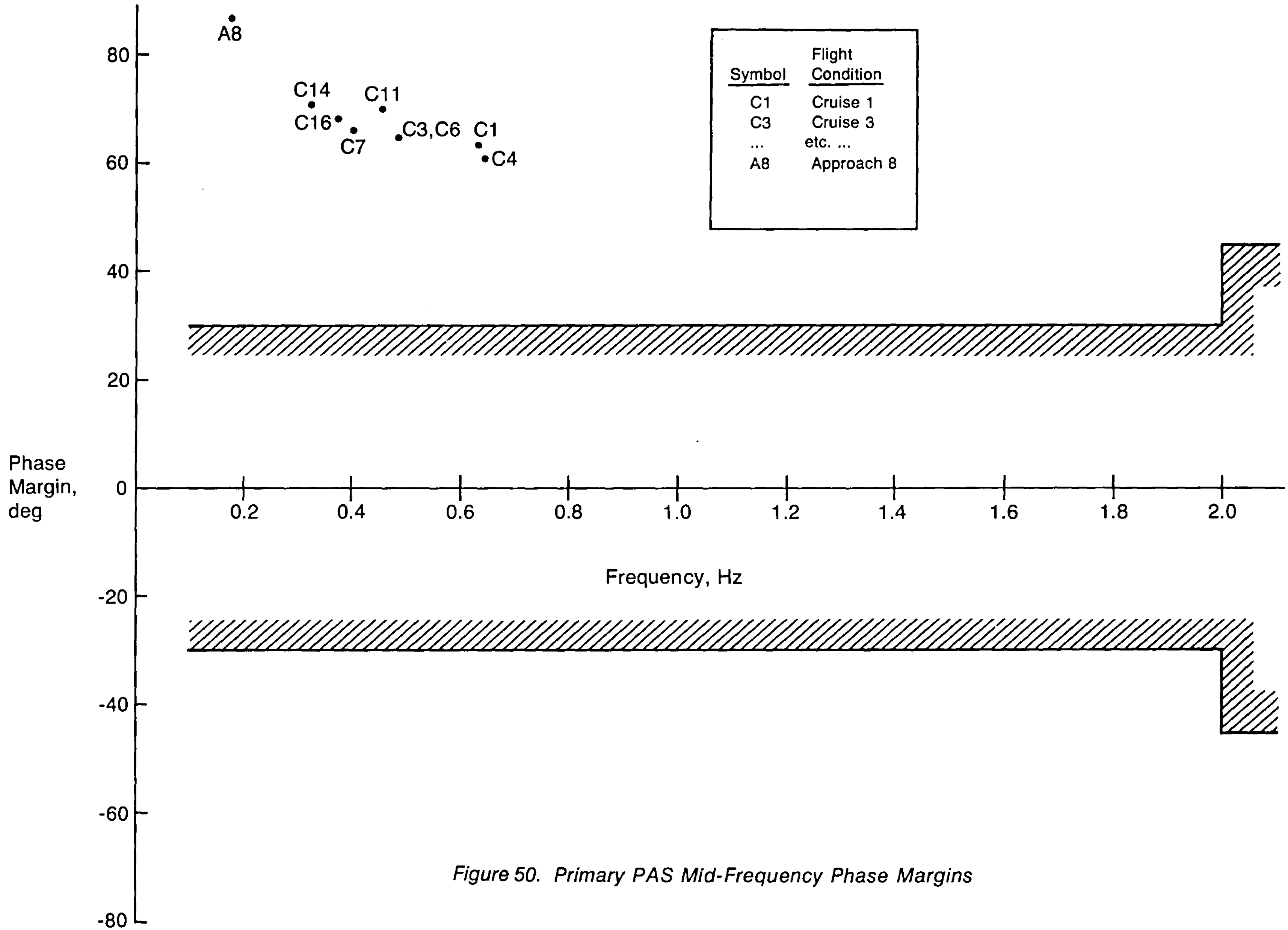


Figure 50. Primary PAS Mid-Frequency Phase Margins

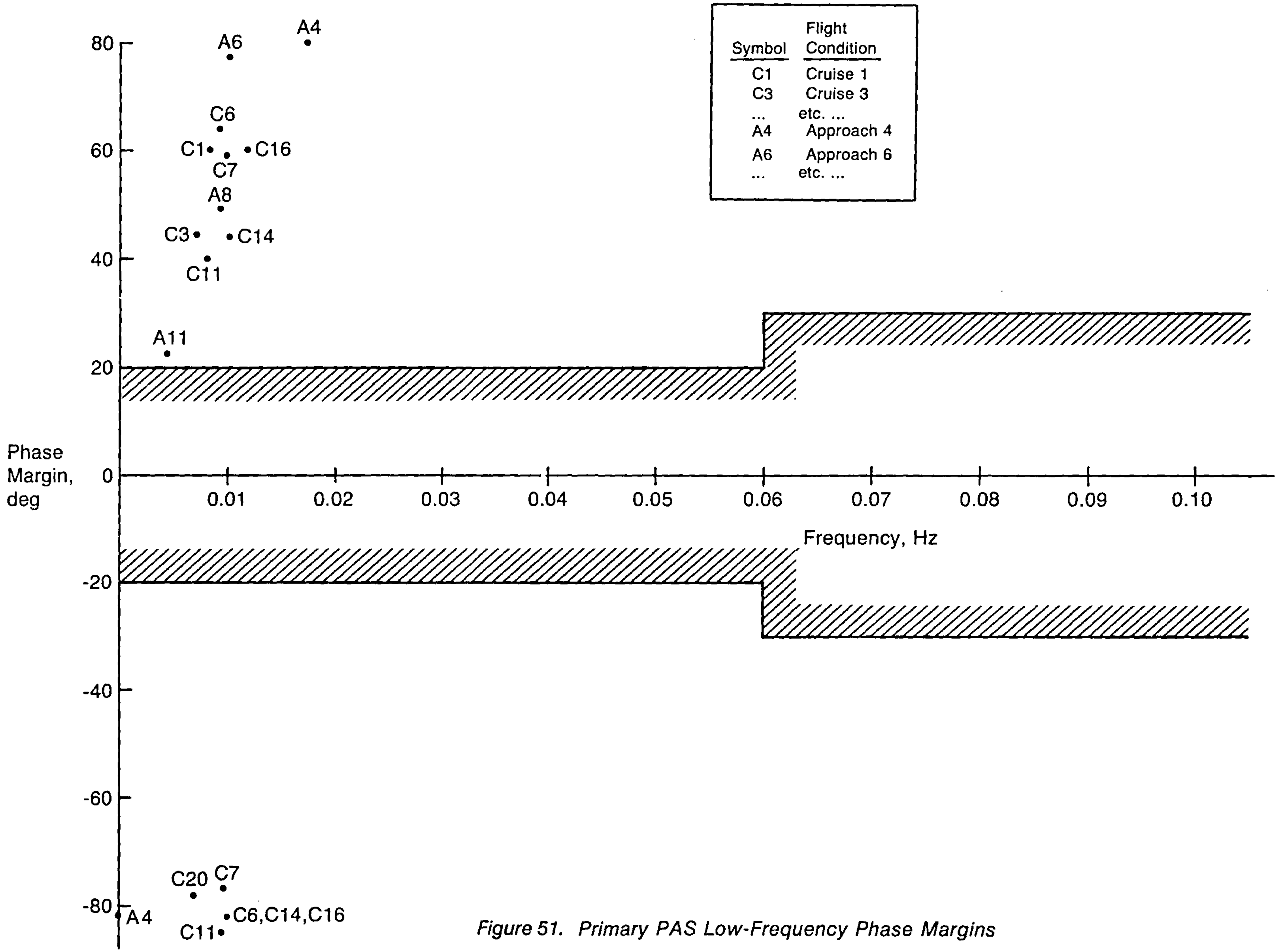


Figure 51. Primary PAS Low-Frequency Phase Margins

Primary PAS. Essential PAS responses were also within acceptable limits for all cases in which a "steady-state" pitch-rate response to a step column command could be estimated.

Once a column command is removed (at t = 5 sec in fig. 52), the airplane, augmented with the Primary PAS, tends to return to the original trim condition required in FAR, Part 25 (ref. 11), but with considerable overshoot in both attitude and speed. An additional pitch-attitude-error feedback loop was therefore included to improve the damping in the absence of column inputs. The prevailing pitch attitude when the column returns to the detent position was selected as the attitude error reference. In order that the airplane might return to the trim conditions existing prior to the column input, the loop included a washout circuit with a 5-sec time constant.

The column forces required to produce an incremental load factor of "1g" are listed in Table 5 as a function of flight condition and cg location. Because no FAR requirement exists for this parameter, the production 757 airplane requirement was used as an objective for the Primary PAS. It states that "maneuvering stick force levels shall be between 133 and 356N (30 and 80 lb) at 1-g incremental load factor." A less restrictive objective of 67 to 490 N/g (15 to 110 lb/g) was assumed for the Essential PAS. As shown in the table, these objectives were substantially met. Only two conditions were significantly out of range, and both of them related to the Primary PAS: CRU 7 at 102 N/g (23 lb/g) and CRU 20 at 111 N/g (25 lb/g). Both of these conditions are at extremes of the probable test envelope. Although gain adjustments to bring these conditions within range can easily be made, this will tend to increase the maneuvering forces at more likely flight conditions. Although still within range, the more probable operating conditions would then approach 356 N/g (80 lb/g). Because piloted simulation results have indicated a preference for light column forces, the gain revision decision will be deferred to the flight test phase. For the unstable Essential PAS conditions (i.e., no steady-state column-force-per-g exists), the value of incremental load factor 3 sec after application of column force was used for the computation.

Table 5 also lists the airspeed change per unit of column force for the Primary PAS. Federal Aviation Regulations, Section 25.173c (ref. 11), states that "the average gradient of the stable slope of the stick force versus speed curve may not be less than 1 pound for each 6 knots." This is equivalent to requiring $\Delta V/F_{col} < 0.7 \ \frac{m/sec}{N}$ (6 kn/lb) in the table. As shown, the requirement was essentially met, with the only significant deviation

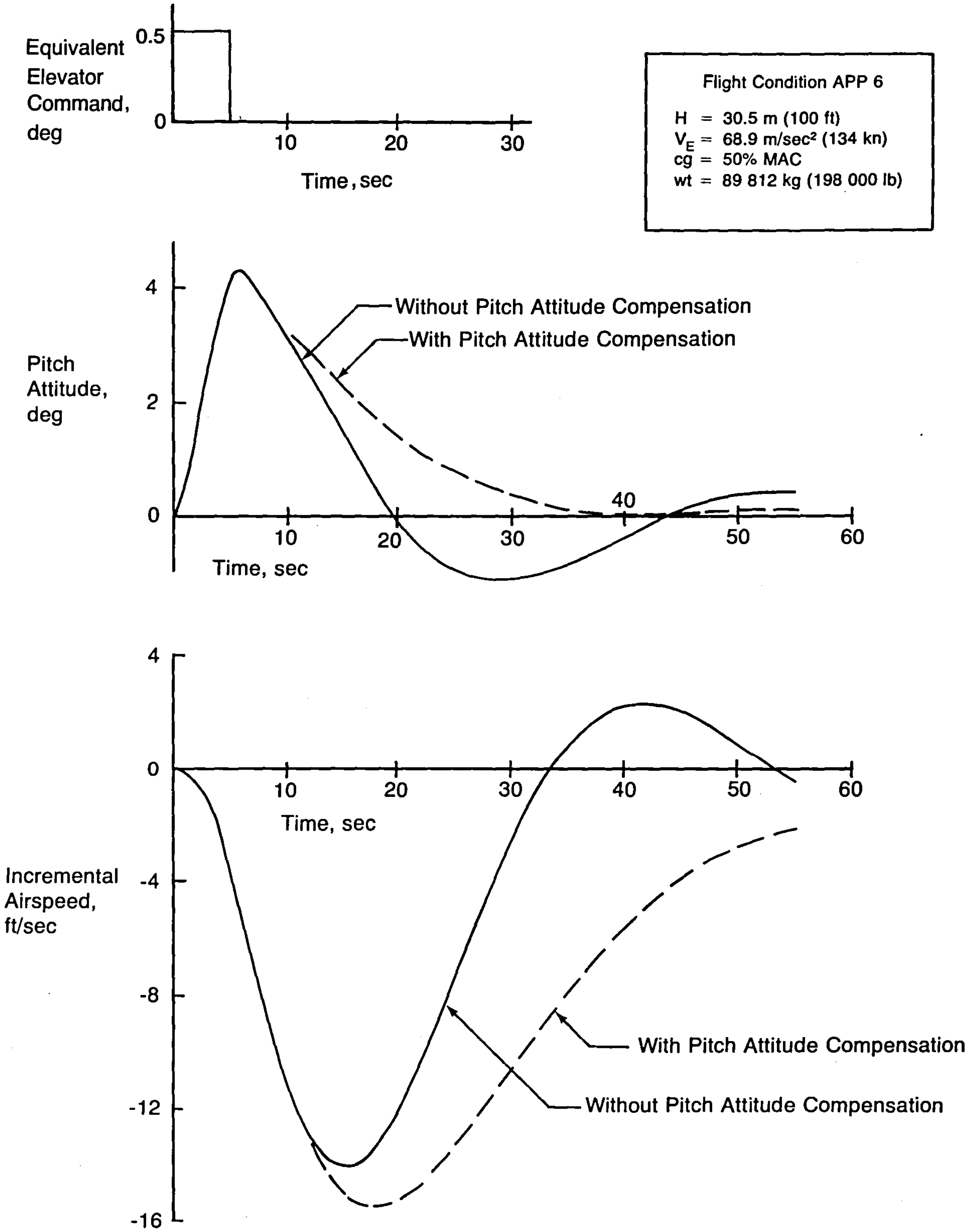


Figure 52. Response to a 5-sec Column Pulse

Table 5. Column-Force-per-g and Column-Force-per-knot Data

Flight Condition	cg Location (% MAC)	F_{col}/g N/g (lb/g) Essential PAS	Primary PAS	V/F_{col} $\frac{m/sec}{N}$ (kn/lb) Primary PAS
CRU 1	20	205 (46)	178 (40)	0.567 (4.9)
CRU 3	50	102 (23)	133 (30)	0.717 (6.2)
CRU 4	20	196 (44)	173 (39)	0.428 (3.7)
CRU 6	50	80.1 (18)	129 (29)	0.405 (3.5)
CRU 7	55	62.3 (14)	102 (23)	0.463 (4.0)
CRU 11	50	182 (41)	249 (56)	0.567 (4.9)
CRU 14	50	227 (51)	191 (43)	0.705 (6.1)
CRU 16	50	107 (24)	169 (38)	0.382 (3.3)
CRU 20	50	80.1 (18)	111 (25)	0.278 (2.4)
APP 4	50	334 (75)	285 (64)	0.439 (3.8)
APP 5	20	454 (102)	298 (67)	0.254 (2.2)
APP 6	50	280 (63)	258 (58)	0.879 (7.6)
APP 8	50	236 (53)	205 (46)	N/A
APP 11	50	254 (57)	182 (41)	N/A

at flight condition APP 6 where the value was 0.88 $\frac{m/sec}{N}$(7.6 kn/lb). It was not corrected because of the belief that it would result in excessive stability at other conditions—a potential problem not addressed in the FAR. Simulator results suggest that if $\Delta V/F_{col}$ is less than 0.23 $\frac{m/sec}{N}$(2 kn/lb), the speed feedback opposes pitch-rate commands from the column such that the airplane is difficult to maneuver.

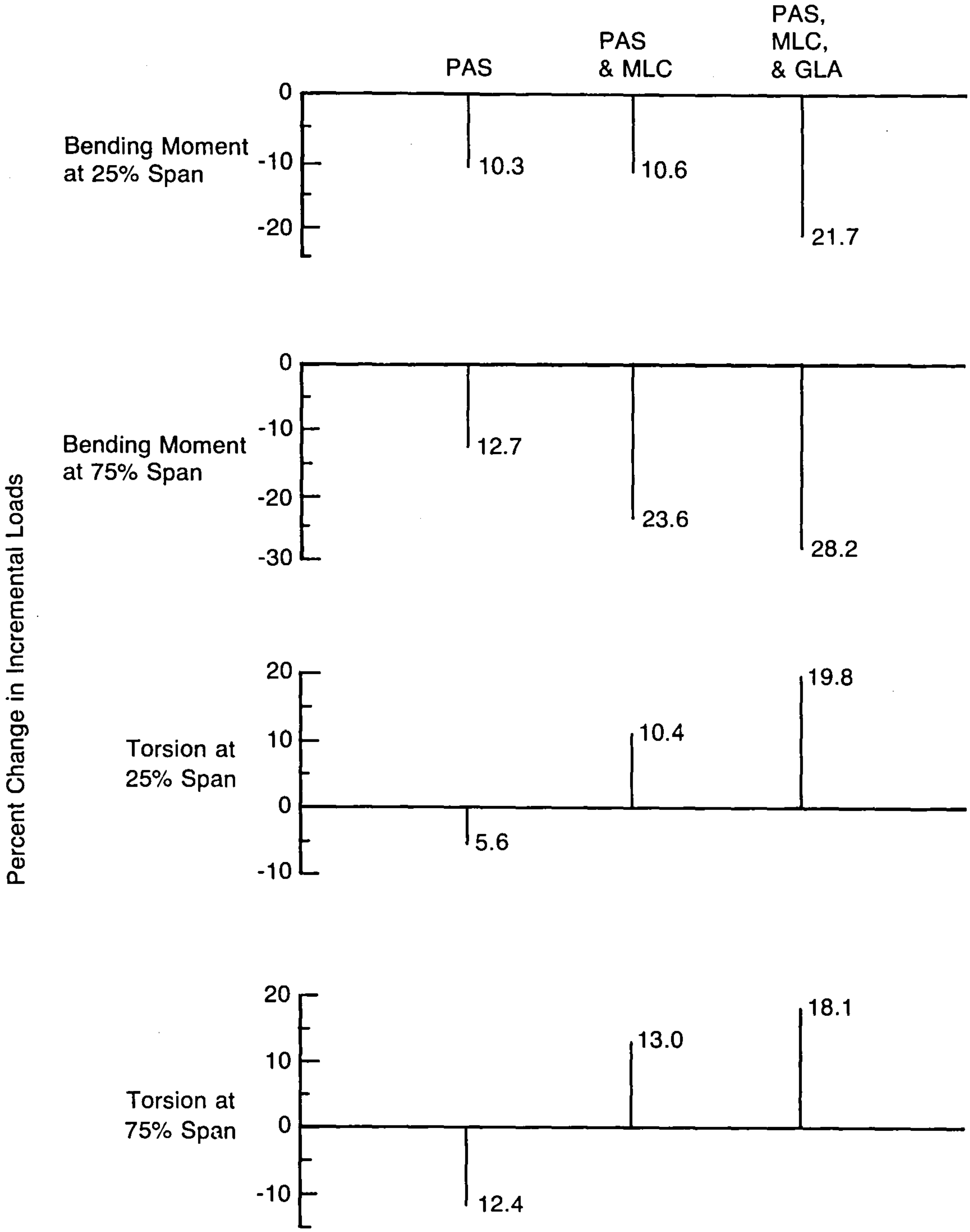
10.2.3.3 Wing-Load Alleviation Performance

The performance of the WLA was evaluated using Von Karman turbulence with an rms isotropic gust level (σ_{GUST}) of 7.3 m/sec (24 ft/sec). A summary of the system effectiveness is shown in Figures 53 and 54. Percent changes in the rms levels of incremental wing bending and torsion loads, with successive closure of the PAS, MLC, and GLA control loops, are shown. The changes are referenced to the rms loads of the unaugmented airplane. Also shown is the increase in the peak incremental value of elevator and aileron displacement and rate.

The data show that PAS alone yields a 12.7% decrease in the incremental bending moment near the wing tip (75% of wing span) compared to that for the unaugmented airplane. The peak incremental demands on the 757 elevator are 2.2 deg in displacement and 9.8 deg/sec in rate. Both figures are well within the capabilities of the elevator drive servos.

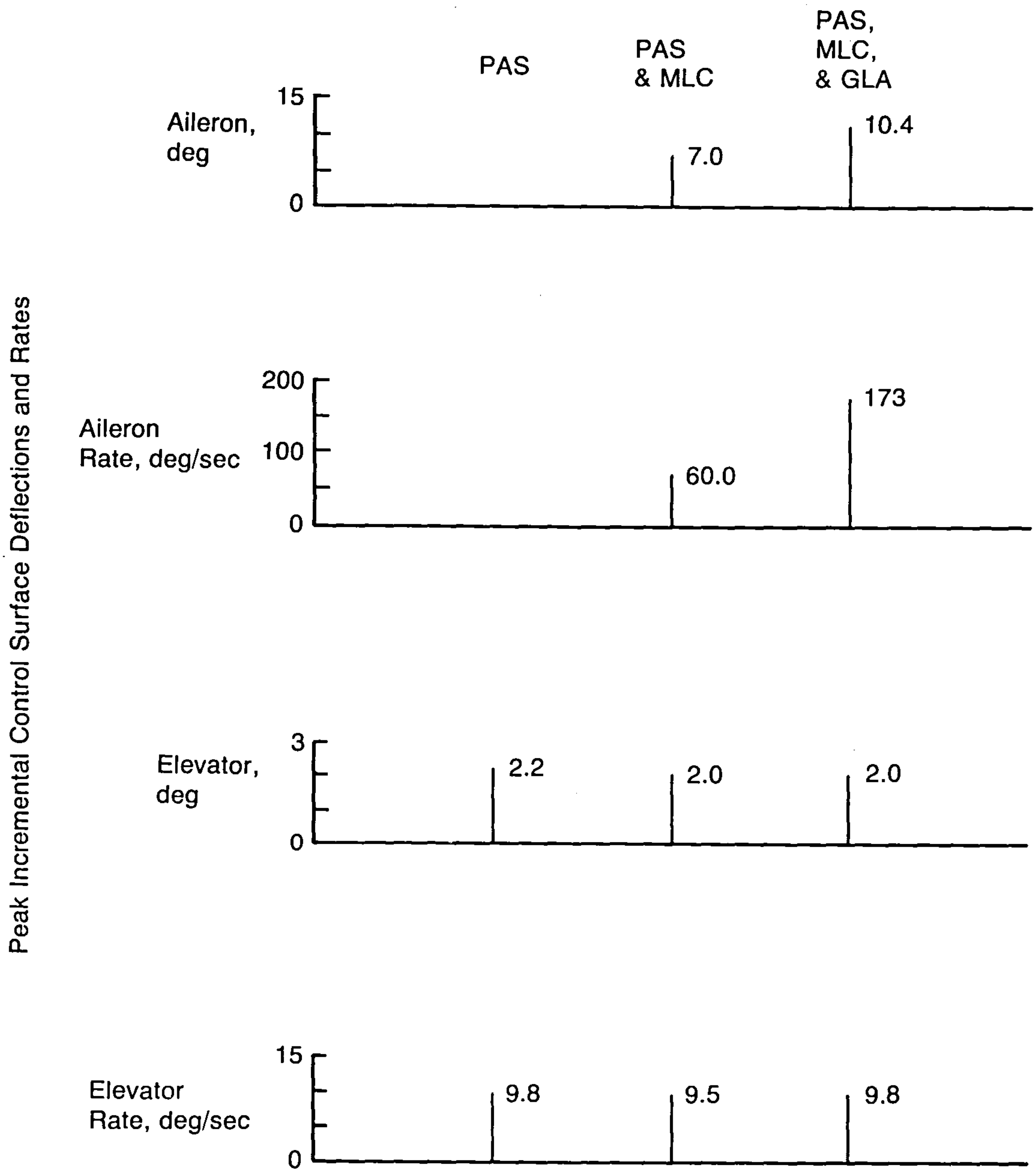
Adding MLC to the PAS yields a 23.6% decrease in the outboard incremental bending moment, again compared to the value for the unaugmented airplane. This additional reduction in moment is obtained by symmetric deflection of the ailerons and has a minimal effect on elevator servo activity. The peak demand on the aileron is 7.0 deg, which is 58% of the trailing-edge-down authority limit at V_{MO}. The rate demand is 60 deg/sec, which exceeds the current 55-deg/sec no-load rate limit of the 757 aileron drive system. This level of exceedance is not significant due to the extremely low probability of occurrence of the selected gust level.

As shown in Figure 53, the GLA system reduced outboard bending moments by approximately 5% but at the expense of a 10% increase in inboard torsion—an acceptable trade. However, the peak aileron rate rose from 60 to 173 deg/sec. It is therefore questionable whether or not this particular GLA design could be justified for a Production ACT System. It should nevertheless be adequate to meet the limited objectives of the WLA flight test program, which is concerned more with verifying system architecture than with validating WLA control laws.



Note: σ_{GUST} = 7.3 m/sec (24 ft/sec), Flight Condition CRU 4
Data relative to the unaugmented airplane response

Figure 53. Wing Loads in Turbulence Due to PAS and WLA Systems



Note: σ_{GUST} = 7.3 m/sec (24 ft/sec), Flight Condition CRU 4

Figure 54. Control Surface Activity in Turbulence Due to PAS and WLA Systems

10.3 RELIABILITY

As stated in Section 10.1, the reliability requirements, system and airplane schedule reliability, apply only to the Production ACT System. The Test ACT System is not configured to achieve such reliability nor can the projected flight test program be expected to provide assurance of Production ACT reliability. This must therefore be accomplished by the analytical methods of this section.

The summary of Production ACT reliability requirements and predictions is as follows:

	Requirement	Prediction
Probability of loss of Essential PAS/FBW in a 1-hr flight	1×10^{-9}	0.69×10^{-9}
Probability of loss of Primary PAS in a 1-hr flight	1×10^{-5}	0.16×10^{-5}
Probability of loss of WLA in a 1-hr flight	1×10^{-5}	0.17×10^{-5}
Number of interruptions per 100 000 departures	65	53.7

10.3.1 ESSENTIAL SYSTEM

Basic Requirement—The Essential system is required to provide sufficient pitch control and augmented stability to permit continued safe flight and landing without exceptional pilot skill or strength after any failure condition that has not been shown to be extremely improbable (probability is less than 10^{-9} for a 1-hr flight) (sec. 4.1.4). The performance requirement of the Essential PAS is therefore to provide at least Level 3 handling qualities (as defined in fig. 4) over the normal operating envelope. At least Level 2 handling qualities are to be provided over a restricted flight envelope that is sufficient for continuation to the scheduled destination.

Design Consequences—In order to meet the reliability requirement using state-of-the-art electronic components, the Essential system was configured to be quadruply redundant and "brick walled" (i.e., each channel is completely independent of the others, from the sensors to the output of the secondary actuators).

Essential System Reliability—The calculated probability of total Essential system failure in a 1-hr flight is 0.69×10^{-9}, as shown in Figure 55.

The reliability analysis for the Essential system is based on the following assumptions:

- Worst case ambient temperature of the equipment racks is $65^{o}C$.
- One Primary computer has been failed for 25 flight hours prior to dispatch.
- Dispatch is not permitted with a known failure in the Essential system.
- Failure rates of components are as shown in Table 6.
- Generic faults and false condemnations are considered negligible.

Figure 55 shows four sources of failure for the Essential System. The major contributors to its failure (passive fault sequences leading to a "two-two split") refer to those failures that may not be detectable by the detent monitor during flight (table 7). The "two-two split" in a voted quadruple redundant control function is a possible failure pattern that defeats the normal failure detection and reconfiguration plan. Normally the first failure is identified by a three-versus-one vote and the discrepant channel is disconnected. A second failure is then detected by a two-versus-one vote. However, if two of four operating channels fail similarly and are detected simultaneously, the voter configuration is two versus two with no sure way of identifying the failed pair. This is the "two-two split." Such failures are detectable during preflight test. The rate of inhibiting failures of inline monitors is small—on the order of 10^{-7} or 10^{-8} per hour. Consequently, even if the inline monitors are not retested for 30 000 hr (assumed airplane life), passive fault sequences involving inhibited inline monitors do not contribute materially to the probability of a two-two split. The 7.1×10^{-6} failure probability shown in Figure 55 is derived for passive faults for which there is no inline monitoring. The computation is conservative since not all combinations of undetected passive faults cause two-two splits. The conclusion that inline monitors are effective, despite 30 000-hr exposure times, is based on the assumption that the monitoring components are periodically tested. A preflight test of the spin monitor has been developed and has been assumed to be available for the Production ACT System.

Another failure mode results where one or more channels are disabled by a soft detent and not detected because of a failed Primary computer. (The secondary actuators have a connect-disconnect feature that enables them to be selectively engaged to the summing link. Engagement results from applying hydraulic pressure to two pistons within the

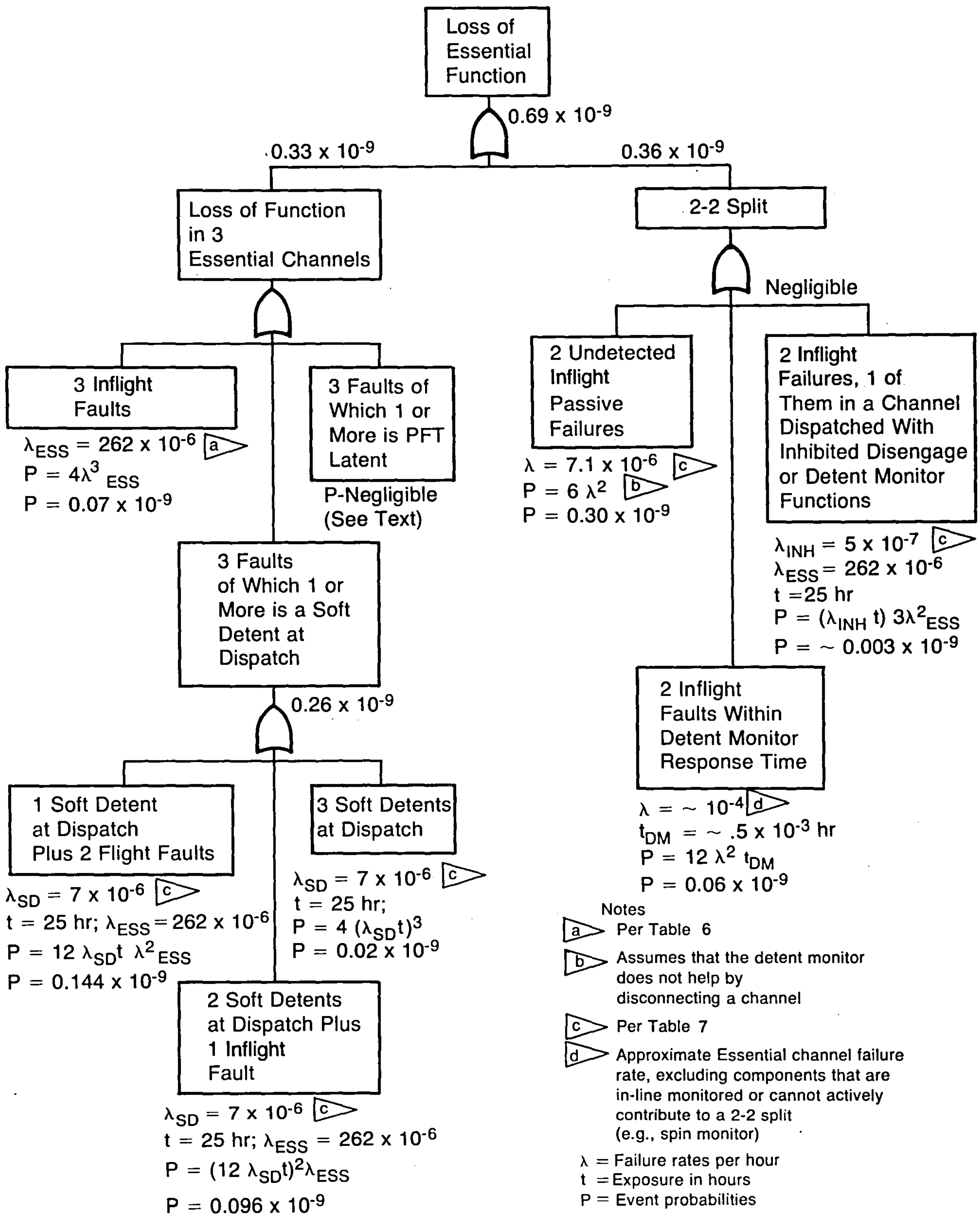


Figure 55. Essential Function Fault Tree

Table 6. Essential Channel and Essential Computer Failure Rates

Essential Channel

Component or Function	Failures per 10^6 flight hour	Source
Loss of electrical power	0^a	NASA CR-3519
Loss of hydraulic power channel	24	757 estimate
Servo (secondary actuator)	32	NASA CR-3519
LVDTs (2)[c]	14	NASA CR-3519
Engage logic (A9 card)[b]	18	Analysis
Essential computer (A6 and A8 cards)[b]	33	Analysis
Secondary power loss (PS total)[c]	62	Analysis
Force transducers (2)	22	NASA CR-3519
Gyro[c]	50	Gyro spec
Gyro monitor/torquer	7	Analysis
Total λ_{ESS}	262	
Soft detent	7	d
Inhibiting failure of inline monitors	0.02 to 0.2	Analysis

[a]Negligible.
[b]Total component failure rate at 65°C ambient.
[c]Components having inline monitors.
[d]Factored from a Boeing piece-part analysis.

Essential Computer

Component	Failures per 10^6 flight hour, 50°C	Failures per 10^6 flight hour, 65°C
Engage logic (A9 card)	11	18
Computer (A6, A8 cards)	14	33
Power supply	42	62
Total	67	113

Table 7. Rate of Passive Failures Causing 2-2 Splits

Component or Function	Classification	Failures per 10^6 flight hour
Hydraulic power	A	00
Servo	B	1.1*
LVDTs	C	00
Engage logic (A9)	A	00
Power supply	C	00
Force transducers	C	00
Gyro (except pickoff)	C	00
Gyro pickoff	B	1.0*
Gyro monitor torquer	A	00
	Subtotal	2.1
	Estimate for Essential computations	5.0
	Total:	7.1

Classification key:

- A A 2-2 split requires a 3-fault sequence.
- B Unmonitored and can lead to a 2-2 split.
- C Component has high coverage inline monitor.

*Vendor FMEA for similar equipment.

power piston, causing them to clamp down on a roller on the output link, thus making the output link follow the power piston. Any fault that prevents the pistons from clamping down on the roller prevents engagement. This is called a "soft detent.") This failure combination contributes 0.26×10^{-9} to the probability of Essential system failure, under the assumption that a failed Primary computer at dispatch precludes soft detent detection and that the detent validation is waived for a maximum of 25 flight hours. In this case, a soft detent failure is not detected by the preflight check nor is it apt to be detected in flight. The figure of 25 flight hours is based on 757 and 767 minimum equipment list

(MEL) requirements for the Yaw Damper System, which allows dispatch with one channel inoperative for a maximum of 25 flight hours.

The system, as a result of failure of three channels in flight, has a failure rate of 0.07×10^{-9} per flight hour.

Negligible failure probability has been assigned to a three-fault sequence involving preflight latent faults. It is assumed that any identified failure sequence of substance will require modification of the preflight test procedure so that the condition would be detectable.

10.3.2 PRIMARY SYSTEM

Basic Requirements–The Primary system, operating in conjunction with the Essential, is required to provide Level 1 handling qualities. Loss of the Primary system does not require diversion, although dispatch would not be permitted. Some change of flight parameters, such as speed or altitude, may be required. To improve the schedule reliability of this system without excessive redundancy, dispatch is permitted with any one of the following items inoperative:

- One Digital Air Data System
- One Inertial Reference System
- One Primary Active Controls Computer

The reliability requirement of the Primary system is such that the probability of complete failure of either the Primary PAS or the WLA shall be less than 10^{-5} in a 1-hr flight. Because loss of the Primary PAS or WLA affects only flight envelope modification, the failure probabilities are averaged over all dispatchable flights.

Design Consequences–In order that the inflight requirements be met (with the predicted component failure rate), the Primary PAS must be capable of operating with only a single IRS or DADC. As discussed in Section 10.1, it is expected that this can be accomplished with inline monitors and authority limiters. It is believed that inline monitors can provide 95% coverage, where coverage represents the percentage of component faults that are correctly identified by the monitors and correctly reconfigured.

Calculated Reliability of Primary PAS/FBW—The failure rate of a Primary computer, excluding components dedicated to WLA functions, is estimated to be 22 x 10^{-5} per flight hour. This is based on the failure rates listed in Table 8. Applicable failure rate estimates from the 757 Autopilot/Flight Director System (AFDS) Program have been used, and the remainder have been obtained by analysis.

Table 8. Primary Computer Failure Rate Estimate

Computer Card Number - Function	Primary PAS/FBW Failure Rate Per 10^6 Flight Hour[c,d]	Primary Computer Failure Rate Per 10^6 Flight Hour[c]
A2 - I/O[b]	8.9	8.9
A3 - I/O[b]	9.9	9.9
A4 - Analog in	13.4	13.4
A5 - Analog output	–	14.2
A5 -- PAS/FBW only	1.8	–
A7 - Monitors	8.9	8.9
A10 - X-channel receiver	11.4	11.4
A11 - RAM	30.7	30.7
A12 - Logic	–	17.5
A12 -- PAS/FBW only	1.4	–
A13 - I/O control[b]	10.8	10.8
A14 - I/O data[b]	11.6	11.6
A15 - CPU control	12.9	12.9
A16 - CPU data	8.6	8.6
Program memory (30 000)	38.8	38.8
Power supply[a,b]	–	42.1
Power supply without 400-Hz inverter	37.0	–
Interconnect	18.3	18.3
Total	224.4	258.0

[a]Power supply is counted against Essential.
[b]Components in series with sensor.
[c]At 50°C average equipment ambient.
[d]Excludes WLA-unique components and 400-Hz power supply for Essential computers.

Certain Primary computer failures can intercept the flow of data from local channel sensors to the other Primary computers. The involved components in Table 8 are computer cards A2, A3, A13, A14, and the power supply. To account for such failures, the approximate failure rate of these components (8×10^{-5} per flight hour) is added to the failure rate of each sensor (table 9). Failure rates so adjusted are indicated by an asterisk (e.g., λ*IRS).

The probability of total loss of Primary PAS/FBW functions is calculated to be 0.16×10^{-5} for a 1-hr flight (fig. 56). Failures of the Primary computer do not contribute significantly to this value.

Calculated Reliability of WLA—In Figure 57, the probability of total loss of WLA is calculated to be 0.17×10^{-5} for a 1-hr flight. Again, neither the Primary computers nor the dedicated sensors contribute materially to the result. The most significant contributor is the Inertial Reference System.

Table 9. Airplane Sensor and Sensor System Failure Rates

Component	Failures Per 10^6 Flight Hour
Inertial Reference System	600
Digital Air Data Computer	290
Inertial Reference System (IRS*)	680
Digital Air Data Computer (DADC*)	370

*The failure rates for IRS and DADC include the failure rate of input-output processing of their signals by the Primary computers.

Notes:

a. It is assumed that dispatch may take place with one Primary PAS computer or one Inertial Reference System or one Digital Air Data Computer inoperative. The probability that no additional module is inoperable is one minus the probability that any additional module is inoperable.

b. All failure rates (λ's) are expressed per flight hour.

Glossary:

λ_{PPC} Failure rate of primary PAS computer

λ^{*}_{IRS}, λ^{*}_{DADC} Failure rates of Inertial Reference System (IRS) and Digital Air Data Computer (DADC) supplemented to include parts of the computer that provide input-output functions

P Probability of an event occurring

t Exposure time

C Coverage; fraction of faults correctly attributed to the malfunctioning component and correctly managed by configuration control.

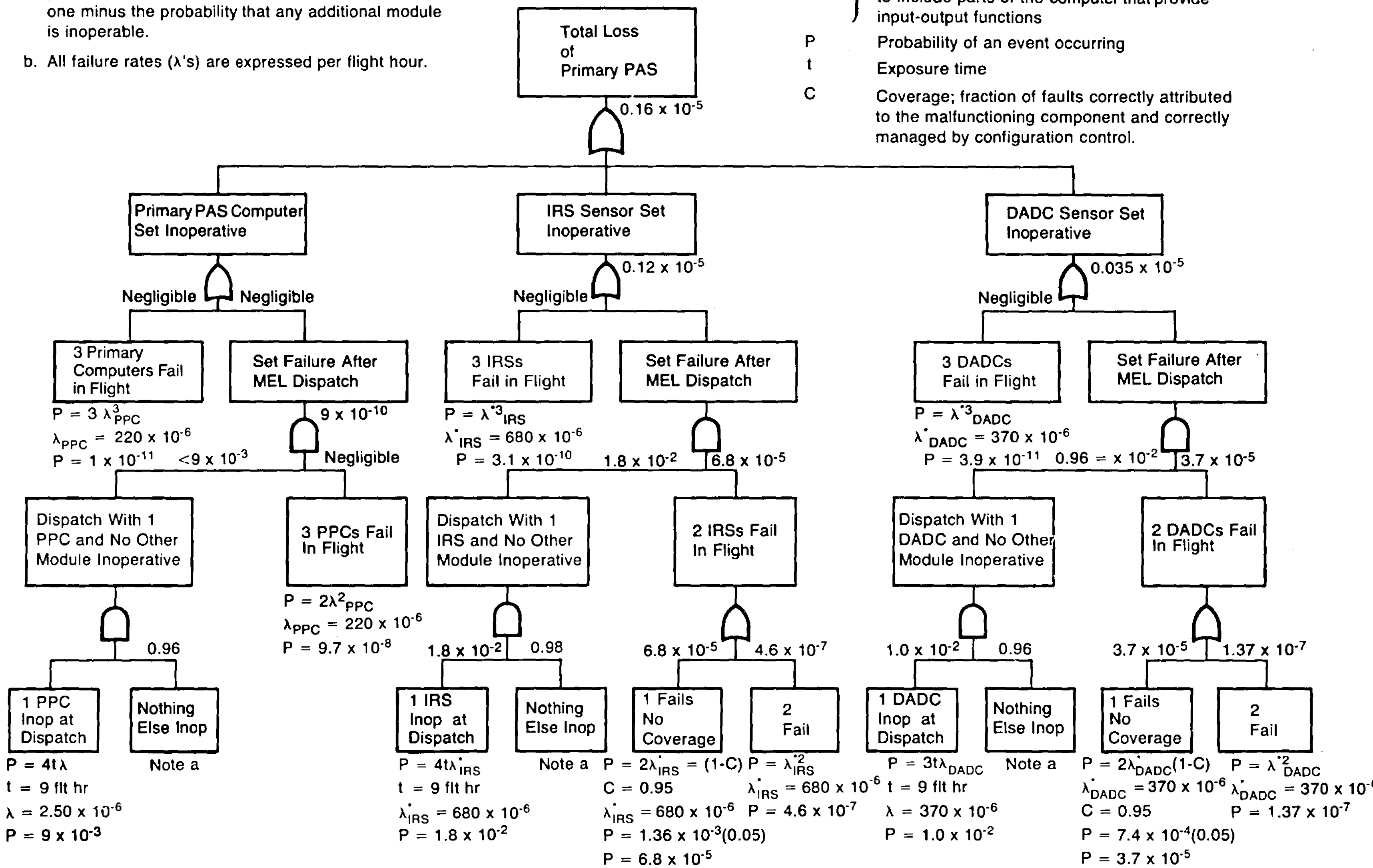
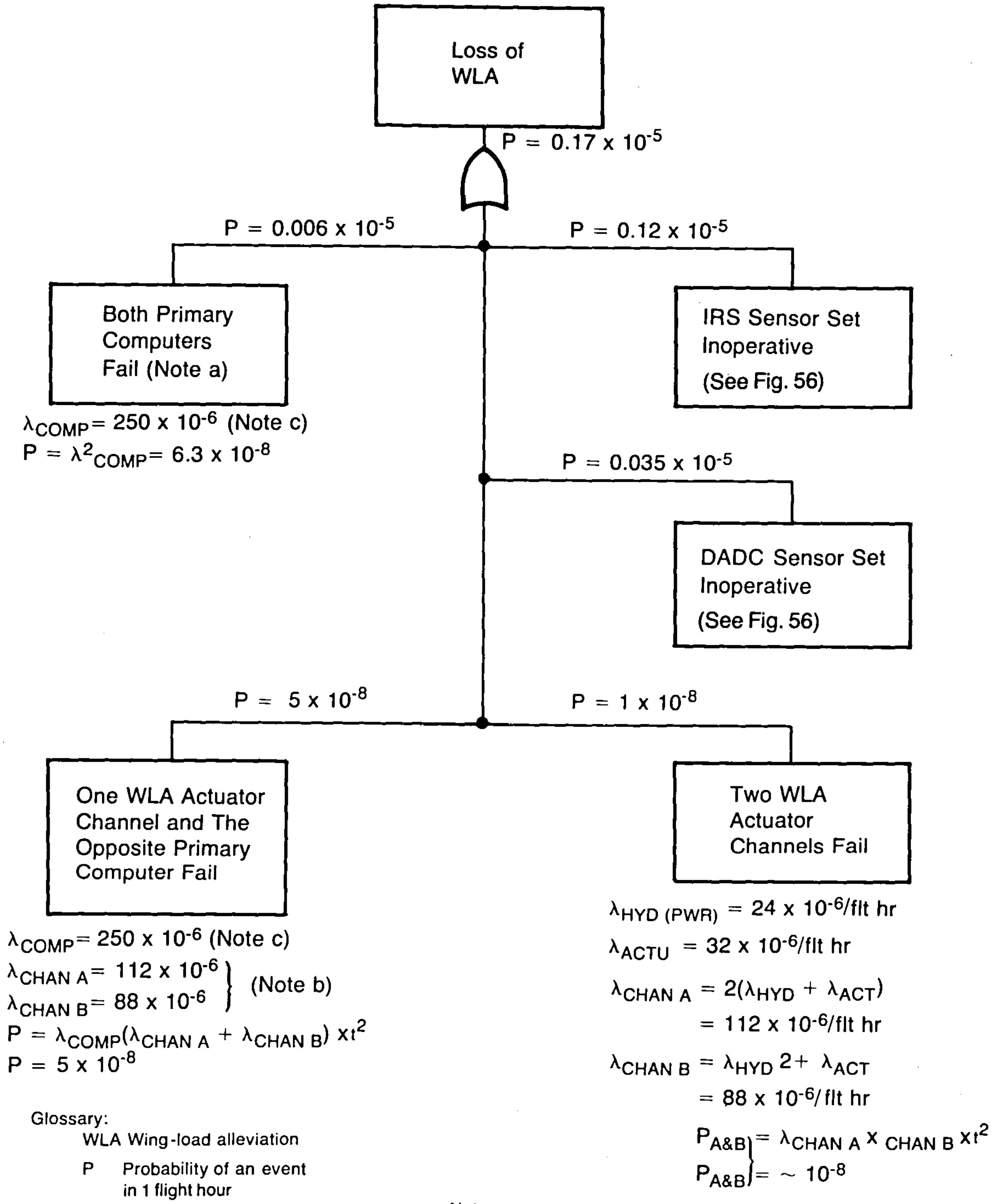


Figure 56. Fault Tree for Total Loss of Primary PAS



Glossary:

WLA Wing-load alleviation

P Probability of an event in 1 flight hour

λ_{HYD} Mean failure rate for hydraulic supplies

λ_{ACT} Actuator failure rate

$\lambda_{CHAN\ A}$, $\lambda_{CHAN\ B}$ Failure rates of actuator channels A & B

t Exposure time, 1 flight hour

Notes:

a. Wing-load alleviation is connected to only two Primary computers at a time

b. Channel failure rates computed elsewhere on this page

c. The WLA function uses computer components not used by Primary PAS, thus the computer λ is different

Figure 57. Fault Tree for Total Loss of WLA

10.3.3 SCHEDULE RELIABILITY

Schedule reliability per the definition in Section 4.1.5 can be predicted in either of two ways:

- By an analysis based on the probability of (1) a required ACT function being made inoperable due to failure of a component and (2) an interruption of service due to such failure

- By comparison to interruption rates experienced in commercial service resulting from failure of components that are similar to the ACT components

Although it is possible to compute probability, the variables such as (1) the time a malfunction occurs and (2) the available ground time (through-stop, turnaround, or overnight) to trouble shoot, repair, or replace—coupled with whether or not the flight crew will accept an airplane that is legally dispatchable with faults—make it a very cumbersome task with questionable results.

The second method is based on actual airline service experience for components of similar function and deferrability. Deferrability refers to the use of the minimum equipment list, which identifies those components that can be inoperable without precluding dispatch. Previous experience has shown that the second method provides good predictions, and it is therefore used here.

- Basic Requirement—The Production ACT System shall add no more than 65 interruptions per 100 000 departures (sec. 4.1.5).

- Calculated Schedule Reliability—The Production ACT System interruption rate is predicted to be 53.7 per 100 000 departures (table 10).

Table 10 is a tabulation of the Production ACT components to be analyzed, the components currently in airline service chosen to approximate the ACT components, and the airplane types from which the experience interruption rates were obtained. Each ACT system component interruption rate is developed by factoring the data base component by the following factors:

Table 10. Schedule Reliability Calculation

Name of Part Added to the Baseline Airplane	Comparison Airplane	Comparison Part, Name	Number per Airplane Factor	Flight Length Factor	Failure Rate Factor	Deferrability Factor	Interruptions Per 100 000 Departures: Comparison Part	Interruptions Per 100 000 Departures: ACT Part	Remarks
			A	B	C	D	E	F=A•B•C•D•E	
Essential Computer and Dedicated Pitch-Rate Gyro	DC-10	Digital Air Data Computer	4/2	0.625	67/98	1	11.6	9.9	No MEL Dispatch for Rate Gyro or DADC
Elevator and Aileron Secondary Actuators (Note 2)	747	Elevator PCU	8/4	0.510	32/17	1	2.77	5.3	No MEL Dispatch for Secondary Actuators or PCUs
Flap Position Sensor	747	Flap Position Sensor	4/2	0.510	5/5	2 (Note 1)	2.77	5.6	ACT Requires All Flap Position Sensors; 747 Allows One Inoperative Under MEL
Column Force Sensors (LVDTs)	DC-10	Digital Air Data Computer	8/2	0.625	7/98	1	11.6	2.1	No MEL Dispatch for Column Force Sensors or DADCs
Preflight Test Panel	DC-10	Digital Air Data Computer	1/2	0.625	33/98	1	11.6	1.2	No MEL Dispatch for Test Panel or DC-10 DADC
Control Panel	DC-10	Digital Air Data Computer	1/2	0.625	16.6/98	1	11.6	0.6	No MEL Dispatch for Control Panel or DC-10 DADC
Primary ACT Computer	DC-10	Digital Air Data Computer	4/2	0.625	224/98	0.5 (Note 1)	11.6	16.6	All DADCs Required for Dispatch; One Primary Computer May Be Inoperative Under MEL
						Component Total		41.3	
						System (1.3 x Component Total)		53.7	

Note 1 A deferrability factor other than one is provided to account for those instances where the comparison part has a different MEL requirement than does the production ACT component.

Note 2 Wing-mounted accelerometers will have sufficient dispatch flexibility to be considered negligible contributors to schedule unreliability.

- Number per Airplane Factor—The ratio of the number of components in the ACT airplane to the number of similar components in the experience data base.

- Flight Length Factor—Determined for the 757 and 767 Programs, it assigns a ratio of time versus cycle-related failures for each two-digit ATA classification and provides a formula for adjusting data base flight lengths to the flight length of the airplane under study. The flight length of the airplane under study is assumed to be 1.25 hr and is consistent with the expected average flight of the 767 and 757 fleets.

- Failure Rate Factor—The ratio of the anticipated failure rate of an ACT component to the experienced failure rate.

- Deferrability Factor—A correction to allow for the fact that the ACT airplane component and the reference airplane component are not treated alike by the MEL.

The sum of the interruption rates for each of the listed components represents those interruptions that are traceable to the particular ACT components. Experience has shown that all schedule interruptions for an automatic flight control system are approximately equal to 1.3 times the sum of all interruptions traceable to particular components. To account for this, the ACT System totals are computed at 1.3 times the sum of the components.

11.0 TEST PLANNING

Page

11.0 TEST PLANNING

Testing of the Test ACT System will be performed in two phases. The first will take place in an avionics laboratory and a flight control test rig and will determine readiness for flight tests. The second phase will be a flight test that will demonstrate the performance and safety of the Test ACT System in a relaxed static stability-configured airplane.

11.1 LABORATORY TEST PLAN

The primary objective of the Test ACT System laboratory tests will be to verify and validate the system hardware and software in time to support the flight test currently scheduled for 1985. The majority of the laboratory testing will be performed in the Boeing Digital Avionics Flight Controls Laboratory and a Boeing flight control test rig. Other testing was performed by Collins in Cedar Rapids. The testing at Collins verified that the system, as designed, meets the requirements; the testing at Boeing will validate most of these requirements and prove the system is safe for flight.

The activities covered by the Laboratory Test Plan include tests of system hardware and software, as well as system integration. The testing begins with the delivery of the Test ACT Console (TAC) and system hardware and progresses through troubleshooting and updating of hardware and software, as required to show full readiness for the flight tests.

11.1.1 LABORATORY TEST DEVELOPMENT AND SCHEDULE

11.1.1.1 Buildup in Test Complexity

The laboratory test features a programmed buildup in the completeness and complexity of the test setup. Starting with single-channel testing of an analog Essential computer, the testing will build in complexity by slowly adding multiple channels, the digital Primary computers, an airplane simulation, and finally a pilot with a simulated flight deck, moving-base cockpit, and computer-generated imagery.

The earlier, simpler tests will be performed using general-purpose test equipment under the control of an Eclipse S250 computer. In this case the Eclipse simply automates the testing. As the tests grow in complexity, the Eclipse will be used to simulate airplane sensors and then the airplane dynamics themselves. At the highest levels of complexity (i.e., those involving a pilot), the tests will switch to a Harris Series 800 computer that will provide a higher fidelity model of the airplane aerodynamics, as well as provisions for interfacing with the pilot and a flight deck simulator.

11.1.1.2 Test Design and Development

Review and analysis of system requirements documentation and the supplier's design and verification documentation will be used to develop a test matrix. This matrix will be used in turn to design the test cases (Plans of Test) for which procedures will be developed. The test cases will be chosen in a manner that covers the requirements in a balanced fashion, with a minimum of duplication of previous verification testing. Plans of Test, as well as Detailed Test Procedures (DTP), will be developed prior to the beginning of trial test runs and finalized prior to the formal test runs.

The design and development process for step-by-step procedures, test drivers, comparison checkers, and analytical programs will continue through the test period. The DTPs will become more specific as familiarity with the laboratory facilities and operational procedures increases; then the Plans of Test will be updated.

11.1.1.3 Analysis and Review

The analysis and review step of the test design process is critical to the selection and preparation of detailed tests that will adequately examine the features of the Test ACT System within the constraints of time and other fixed resources.

The test analysis task begins with the development of laboratory test categories and the determination of potential system error classifications. These will serve to direct and order the test design effort. The next step is the preparation of a test matrix, relating documented requirements to planned tests. This effort is intended to ensure that no stated requirement goes untested. Further, it will aid in avoiding unnecessary duplication of tests. The test matrix is also expected to provide insight into retest requirements of new versions of the system, following revision or problem correction.

11.1.1.4 Laboratory Test Categories

The design of the test cases and the overall planning of the testing is aided by initially dividing the requirements that must be tested into logical categories. These categories will provide the first-level organization for the development of the test cases (Plans of Test).

A set of categories that meets the needs of the Test ACT System laboratory test effort follows:

- Open-loop hardware tests
- Open-loop software tests
- Failure detection tests
- System integration tests
- Closed-loop system performance tests
- Closed-loop failure response tests

Table 11 illustrates the test categories and their contents, as envisioned for the Test ACT System laboratory tests.

11.1.1.5 Test Matrix

The test matrix for each category will be developed by examining the requirements defined by the Test ACT Specification Control Drawing. In addition, supplier design and implementation documents will be examined for requirements generated by the system design process. The specification paragraph number will be used to cross-reference the various documents with the test matrix.

11.1.1.6 Development of Test Procedures

The general test philosophy will be first to verify that the system meets all requirements given in the Specification Control Drawing and then to determine that they result in a safe system that meets the performance requirements. To ensure this, the matrix described in Section 11.1.1.5 will be used. It will indicate how each requirement will be met (inspection, analysis, or test) and who (the vendor or Boeing) will perform the test.

Table 11. Laboratory Test Plan Categories

OPEN-LOOP HARDWARE TESTS

- EMI, power transient, and quality susceptibility
- Control panel operation
- Input/output interface
- Primary system hardware monitors
- Essential control laws
- Essential PAS/FBW monitors
- Primary system output voting

OPEN-LOOP SOFTWARE TESTS

- Flight deck interface
- Pitch control laws
- Wing-load alleviation
- Signal selection and fault detection
- Output management
- Program control
- Fault reporting and recording

FAILURE DETECTION TESTS

- Power-up
- Preflight
- Periodic tests/monitors

SYSTEM INTEGRATION TESTS

- Test ACT/DADC integration
- Test ACT/IRS integration
- Test ACT/CSEU/FSEU integration
- Test ACT/secondary actuator integration

CLOSED-LOOP PERFORMANCE TESTS

- Closed-loop stability and response (unpiloted)
- Closed-loop stability and response (piloted)

CLOSED-LOOP FAILURE RESPONSE TESTS

- Airplane response to failures (unpiloted)

Each requirement to be tested by Boeing will be covered by at least one DTP. In general, each of them will verify several requirements.

The DTP will be developed in three parts or steps. Step one, as explained previously, is to identify the requirements to be verified. Step two will be to write a short description of a test that will verify those requirements. Step three, which will be performed in the laboratory, will be to determine the procedures required to perform a repeatable test (a primary requirement on DTPs is that they contain enough information to repeat the procedure years later). It is in this period that the computer programs used to automate the test will be developed for the Eclipse.

11.1.1.7 Test Schedule

Figure 58 gives the schedule for the initial phase of laboratory testing. Laboratory development and test planning will proceed concurrently. Both are scheduled to be complete on July 6, when the Test ACT hardware and software arrive from Collins. At that time laboratory testing will begin. This phase of laboratory testing will be completed by mid-December 1983. The need for system changes will be determined at that time.

Figure 59 is the full-term schedule. Test planning for 1984 is only preliminary, as the data for software update and actuator availability are not firm. The major test events for 1984 are a test of the four actuator interconnectors and validation of the final Test ACT Configuration.

11.1.2 LABORATORY FACILITY DESCRIPTION

11.1.2.1 Digital Avionics Flight Controls Laboratory

All tests will be conducted at the Digital Avionics Flight Controls Laboratory (DAFCL). This laboratory is located adjacent to the Renton Flight Simulation Center (RFSC). The primary function of the DAFCL is 757/767 flight control system validation, although support is also provided to advanced technology programs, such as Test ACT, and to software development. The laboratory layout is shown in Figure 60.

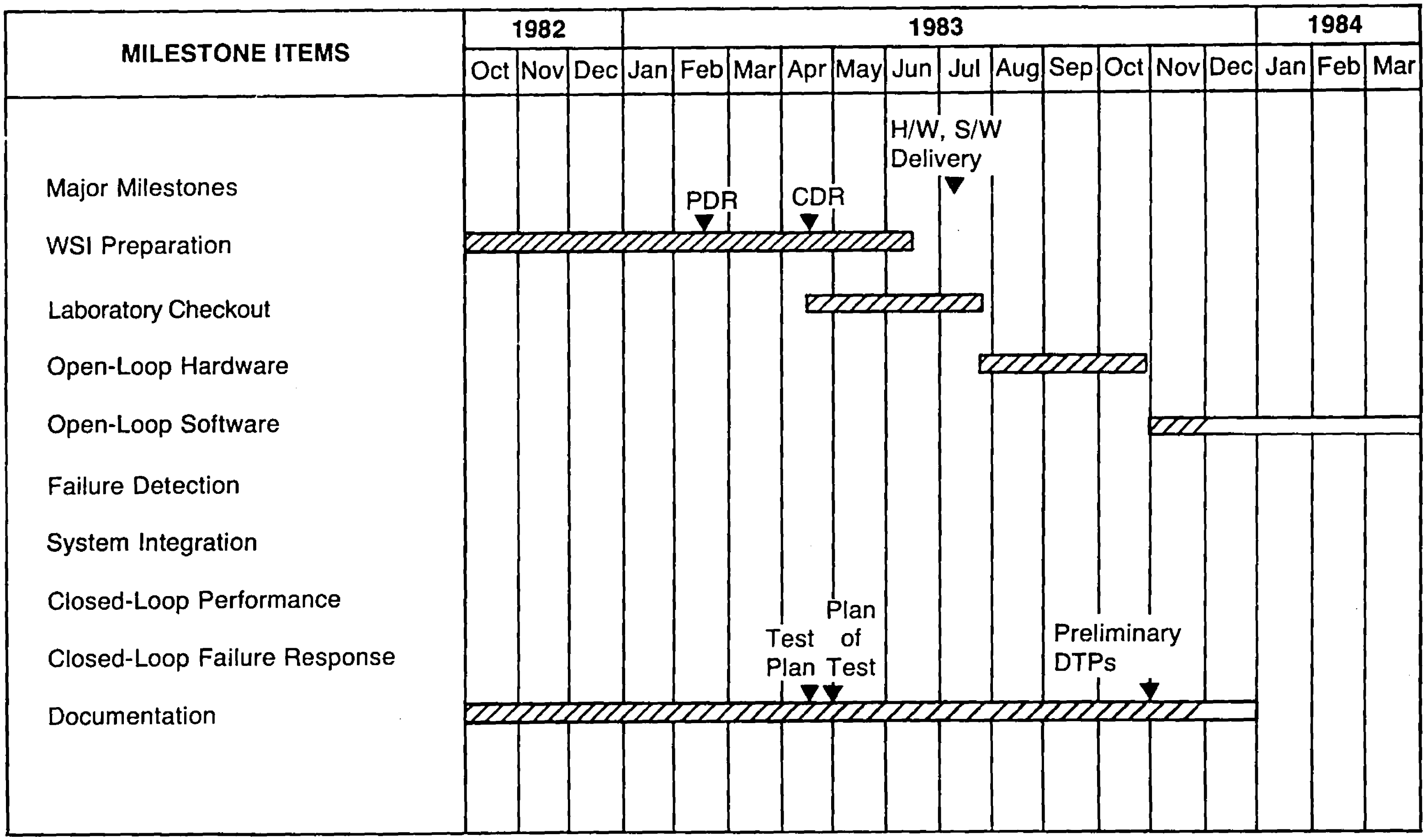


Figure 58. Test ACT System Laboratory Test Schedule

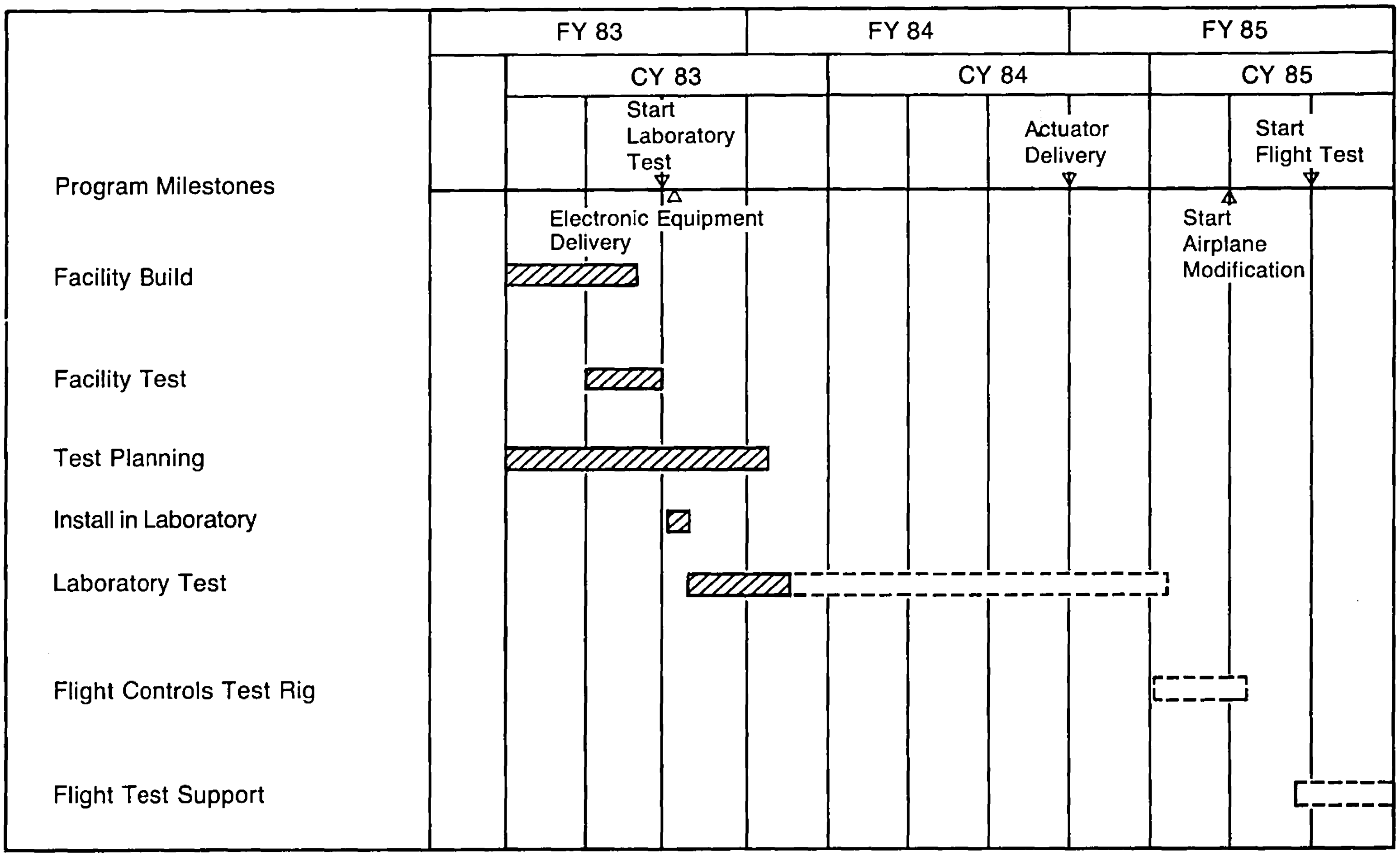
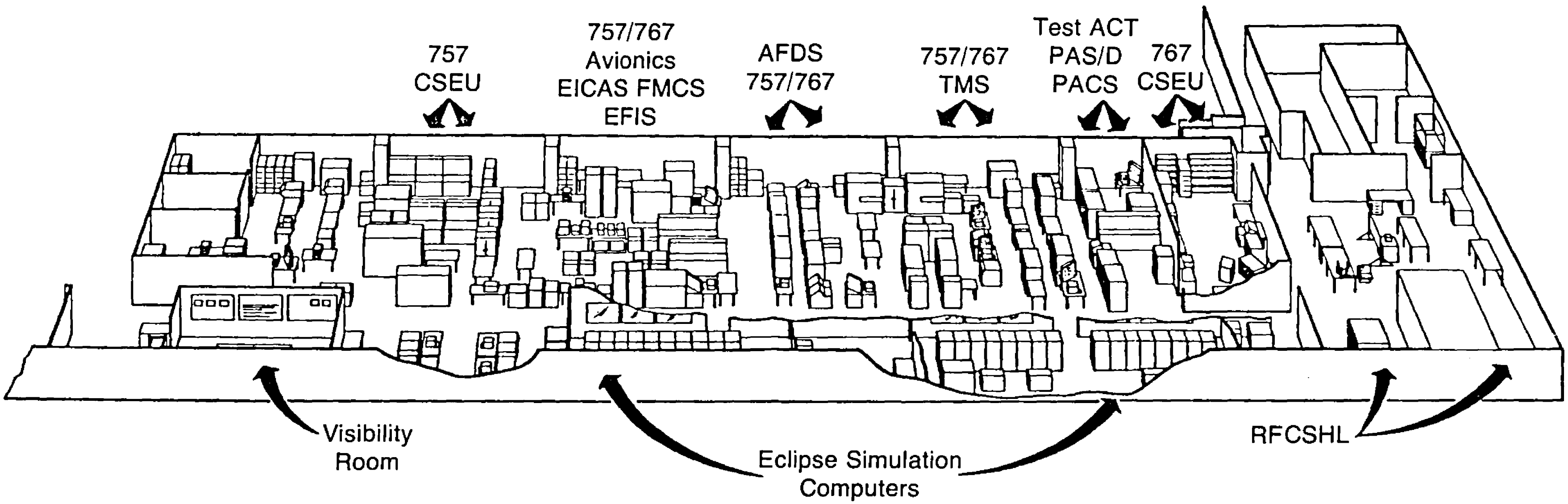


Figure 59. Overall Test Schedule

Digital Avionics Flight Controls Laboratory



TMS: Thrust Management System
PAS/D: Pitch Augmented Stability Demonstrator

Figure 60. DAFCL Laboratory Layout

The DAFCL contains work areas with digital simulation hardware, test consoles, analog computers, interface equipment, and other support hardware. The ability is provided to conduct real-time simulations with simulated control inputs. The complex can be interconnected with the RFSC for testing with crew cabs and maximum fidelity aerodynamic models.

11.1.2.2 Digital Avionics Flight Controls Laboratory Simulations

The DAFCL Data General Eclipse S250 computers host simulations of the airplane equations of motion, sensors, servos, airplane environment, and engine dynamics. The sensor and servo simulation will include proper timing, scaling, and significant nonlinearities.

The Eclipse aerodynamic model is derived from the maximum fidelity aerodynamic models used in the RFSC. This process provides good configuration control of airplane data. The Eclipse computers will be used in the early phases of testing to (1) provide an automated test sequence, (2) record test results, (3) simulate sensors and actuators, and (4) provide input and output simulation and failure models.

11.1.2.3 Renton Flight Simulation Center

The Renton Flight Simulation Center contains all components necessary to conduct man-in-the-loop, real-time aircraft simulations. The components include (1) digital and hybrid-analog computers, (2) cockpits with instrument displays, (3) computer-generated cathode ray tube (CRT) displays, (4) visual displays, (5) a cockpit motion system, (6) an extensive computer software library, and (7) an experienced simulation staff. The floor plan and location of major elements of the RFSC are shown in Figure 61.

The Multipurpose (M2) Cab is equipped with a three-degree-of-freedom motion base and an electrohydraulic force feel system. The M2 cab can be configured to represent any Boeing commercial airplane. The cab has a complete captain and first officer instrumentation set, modular aisle stands and dash panels, test conductor's station, and out-the-window visual displays. The Guidance and Flight Display Cab and the Systems and Work Load Cab are commonly called Configuration (C) Cabs. The C cabs support the 757 and 767 Programs. Included in the cabs is a complete digital interface system to the RFSC laboratory computers.

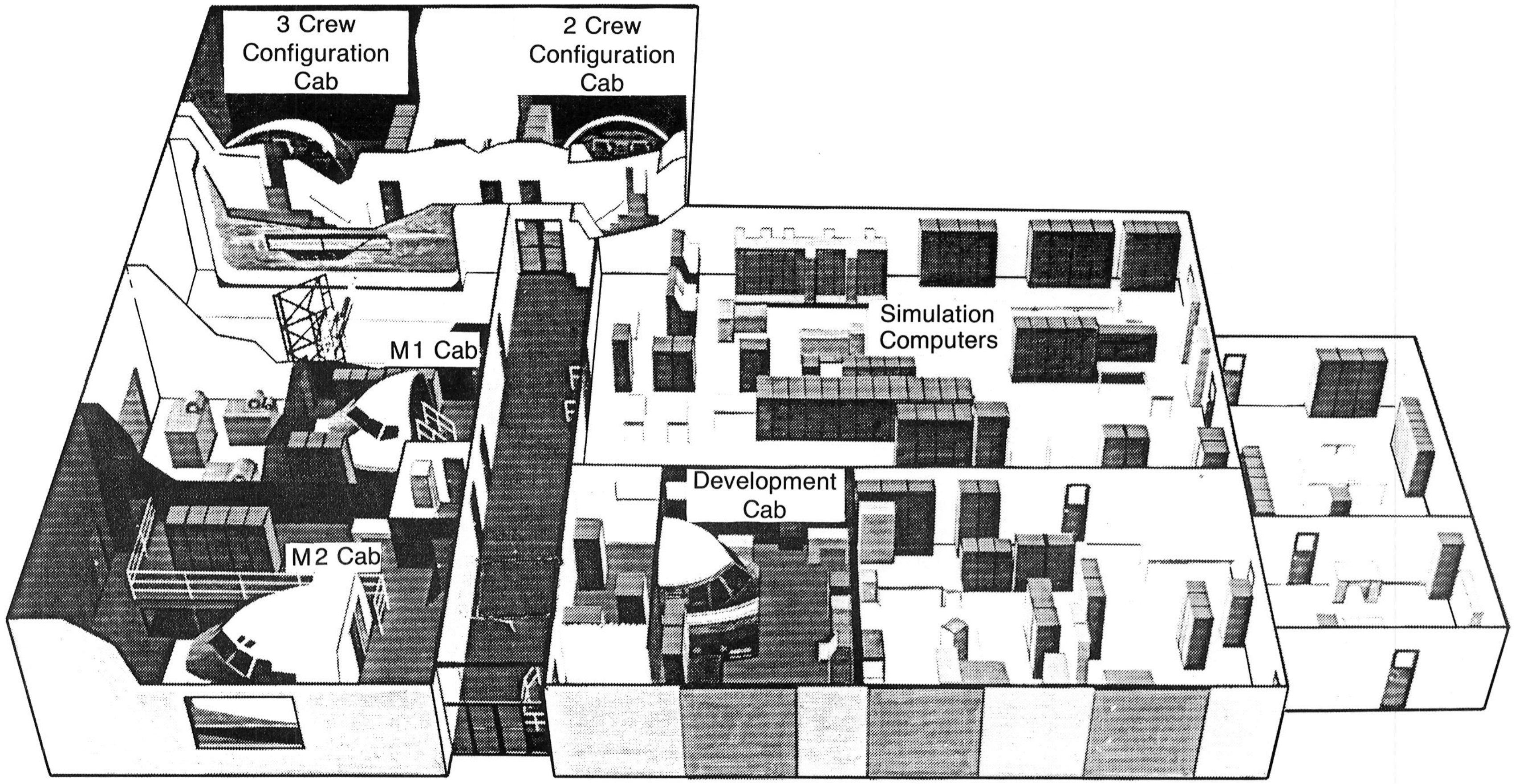


Figure 61. Boeing-Renton Flight Simulation Center

A high-fidelity, fully nonlinear 757 simulation model, resident in Harris Series 800 computers, is provided in the RFSC. This facility will be used for system integration and piloted simulation work in support of this test plan. The simulation cab facility is used for tests that require pilot-in-the-loop control and display interaction. Most piloted simulations will utilize the fixed-base C cab for system integration testing since that cab configuration closely matches the cockpit layout of the 757 flight test airplane. The moving-base M2 cab will also be used to a lesser extent, where motor cues are important.

11.1.2.4 Renton Flight Control Systems Hydromechanical Laboratory

The Renton Flight Control Systems Hydromechanical Laboratory (RFCSHL), located adjacent to the DAFCL (fig. 60), is used for development, verification, and validation testing of aircraft actuators and mechanical fixtures. The laboratory contains complete electrical and hydraulic fluid distribution systems with convenient connect points. Large environmental test and fluid flow benches are available, as is a protected operator control and data gathering room.

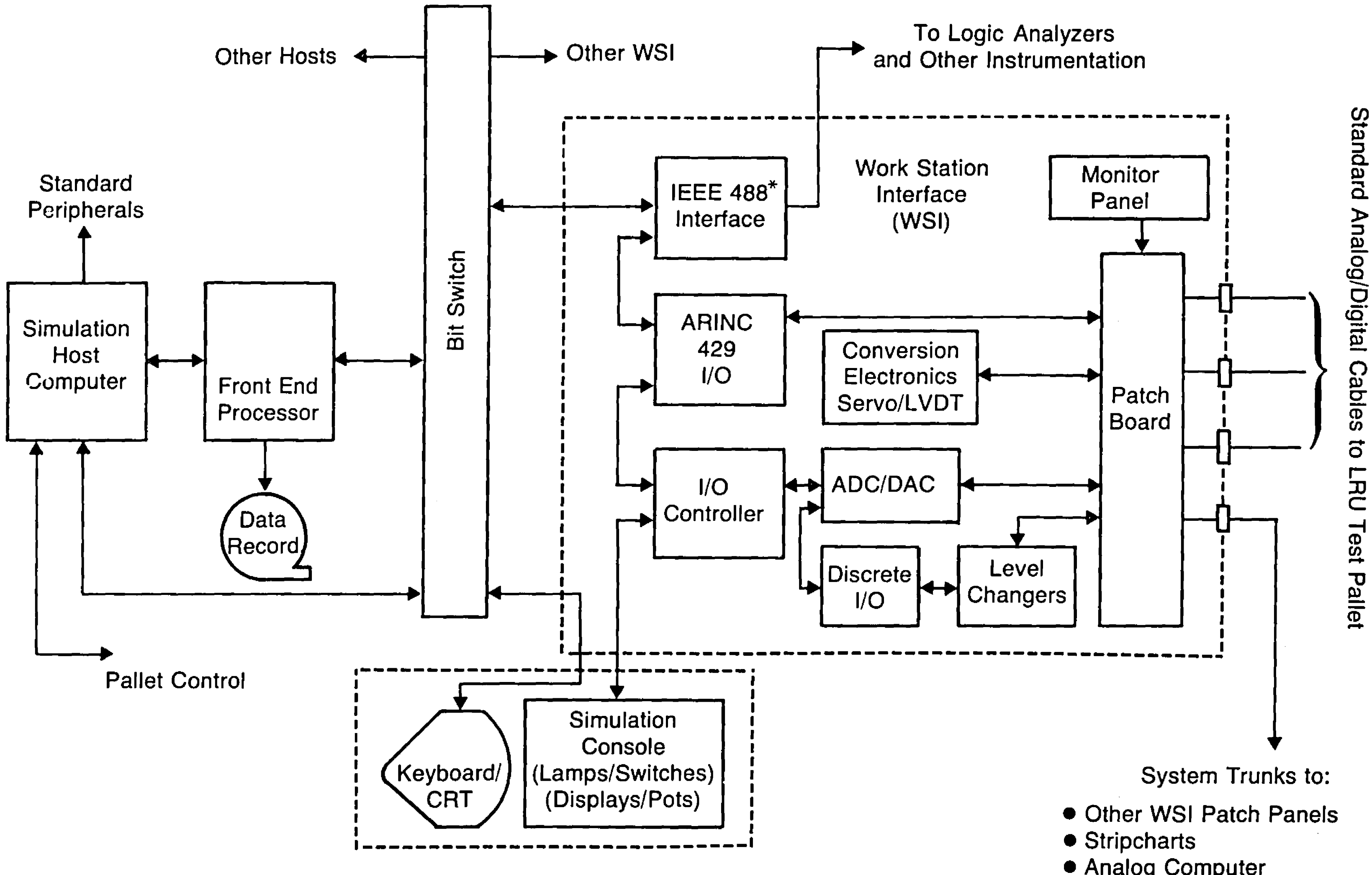
Future plans call for building a special test fixture (mini-rig) in the RFCSHL to test the servoactuators in a four-channel closed-loop configuration. This mini-rig will simulate the mechanical interconnections between the actuators that will exist on the test 757 airplane. These actuators will be controlled, through the work station interface (WSI), by the Active Controls Computers (ACC) located in the adjacent DAFCL.

11.1.3 TEST SUPPORT HARDWARE AND SOFTWARE

11.1.3.1 Digital Avionics Flight Controls Laboratory Test Support Equipment

In addition to the simulation host computers, the other equipment required for DAFCL testing is shown in Figure 62. The major elements are described as follows:

- A front-end processor (Data General Eclipse S230) is provided with each simulation host. It has the capability to format the floating-point simulation data to or from the equivalent fixed-point format that is compatible with the line replaceable unit (LRU) interface equipment.



*IEEE 488: Institute of Electrical and Electronics Engineers' data transmission standard

Figure 62. Typical DAFCL Test Configuration

- A WSI is provided for interfacing the Primary and Essential computers and TAC with the rest of the laboratory. This device controls the buffering, conversion, signal conditioning, and other required LRU interfacing. A patch panel is provided for flexible interconnection to the TAC.

- A Simulation Control Console is provided for the test conductor interface. This console includes a simulation control CRT and a panel of programmable controls and displays (discrete switches, lamps, rotary input encoders, and numerical displays).

Figure 63 illustrates the interfacing required for the Test ACT work station.

11.1.3.2 Digital Avionics Flight Controls Laboratory Support Software

The DAFCL support software will include a real-time simulation and test executive, plotting routines, test driver software (e.g., frequency response test), interface software and data bases, diagnostic software, and analysis programs. The real-time simulation and test executive is a higher order language within which simulation models and open-loop test drivers are combined to form a simulation package. It also provides a standard user interface for simulation initialization, control, and data acquisition.

Supporting aids, such as LRU and Aeronautical Radic Incorporated (ARINC) data bases, high-fidelity LRU models, comparison checkers, problem tracking, and report generation capabilities, are also provided.

11.1.3.3 Work Station Interface Patch Board Preparation and Checkout

The test area of the DAFCL includes a WSI patch board development and checkout bench, consisting of a work area and a microcomputer-based automatic patch board checkout station. A patch board configuration control program is provided for interactive maintenance and control of the patch configurations and for downloading the patch data base for the checkout station.

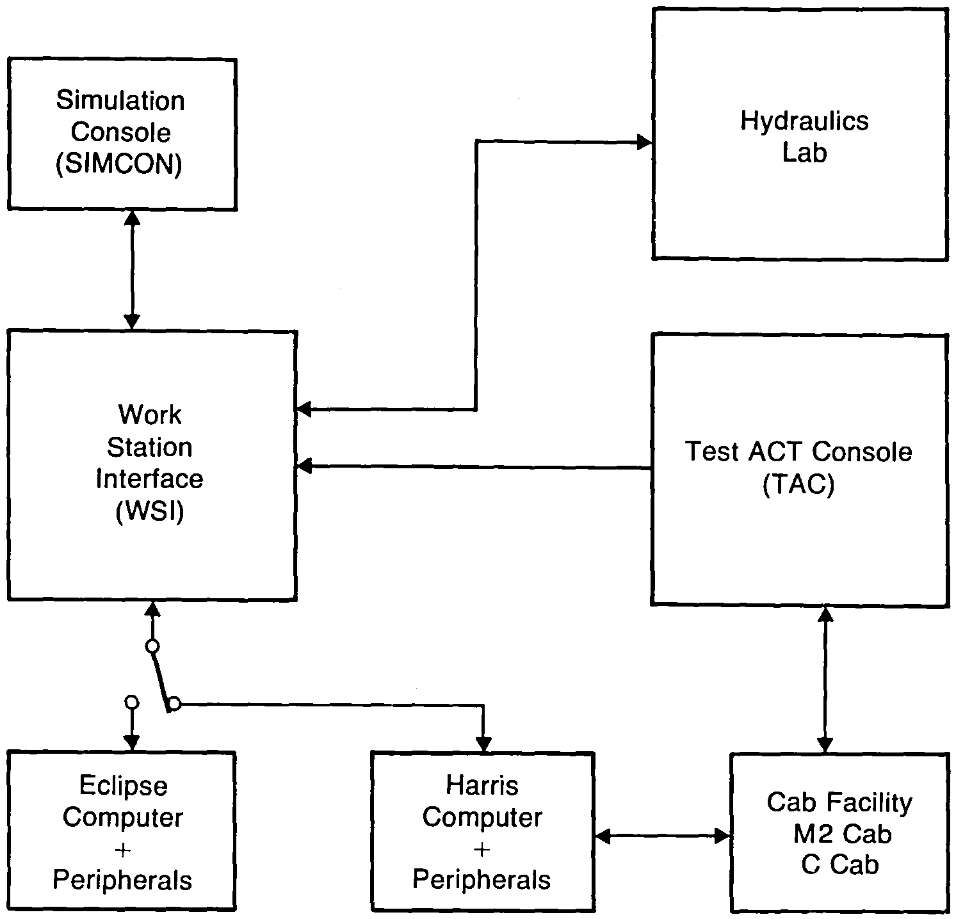


Figure 63. Test ACT Work Station

11.2 FLIGHT CONTROL TEST RIG TEST PLAN

Test ACT System and proof testing will be performed on the 757-200 Flight Control Test Rig (FCTR) prior to installing the system in the airplane.

11.2.1 FLIGHT CONTROL SYSTEM TEST

All mechanical and actuation components of the Test ACT System will be installed in the FCTR and integrated with the normal airplane systems. The function of the ACCs will be simulated by laboratory test drivers.

The objectives of the test are to:

- Verify elevator rate and deflection capability
- Determine system resolution in terms of deadband and hysteresis
- Determine frequency response characteristics
- Determine effects of asynchronous computer operation
- Determine system engage and disengage characteristics
- Determine system transient characteristics in response to failures

11.2.2 PROOF AND OPERATIONS TEST

A one-time limited mechanical and hydraulic system proof and operations test will be conducted. This will be used to verify proper operation and adequate structural capacity of Test ACT elements for operating modes that are beyond the scope of the system functional test. This will subject Test ACT components, supports, and plumbing to limit or near-limit load conditions. Component operating clearance under extra-normal conditions, such as combined mechanical and electrical commands, will be verified.

The above testing will be conducted on the FCTR wherever practicable, with the remainder accomplished upon the 757 airplane prior to first flight.

11.3 PRELIMINARY FLIGHT TEST PLAN

This section presents the Preliminary Plan for the testing of the Test ACT System after installation in the test airplane. The plan includes engineering ground tests, as well as the actual flight test.

11.3.1 TEST OBJECTIVES

The objectives of the engineering ground tests are to verify:

- Correct operation of the installed Test ACT System
- Correct operation of the flight test instrumentation system
- Flight test operational procedures and overall flight readiness before proceeding to the flight test phase

The objectives of the flight tests are to validate:

- The performance and safety of the Test ACT pitch fly-by-wire (FBW) and stability augmentation system on a test airplane configured for relaxed static stability
- The open-loop performance (sensor-to-control surface command signal) of the Test ACT wing-load alleviation (WLA) system

11.3.2 TEST AIRPLANE DESCRIPTION

11.3.2.1 General Description

The test airplane will be the Boeing-owned 757 (NA001). It is a new-generation, subsonic, commercial transport aircraft, as shown in Figure 64, that will be powered by two Pratt & Whitney 2037 high-bypass-ratio turbofan engines. Its longitudinal control surfaces include full-span elevators and trimmable horizontal stabilizer. The principal characteristics of the aircraft are summarized in Table 12.

Passengers (Mixed Class)	178
Engine Thrust (each)	156 000N (35 000 lb)

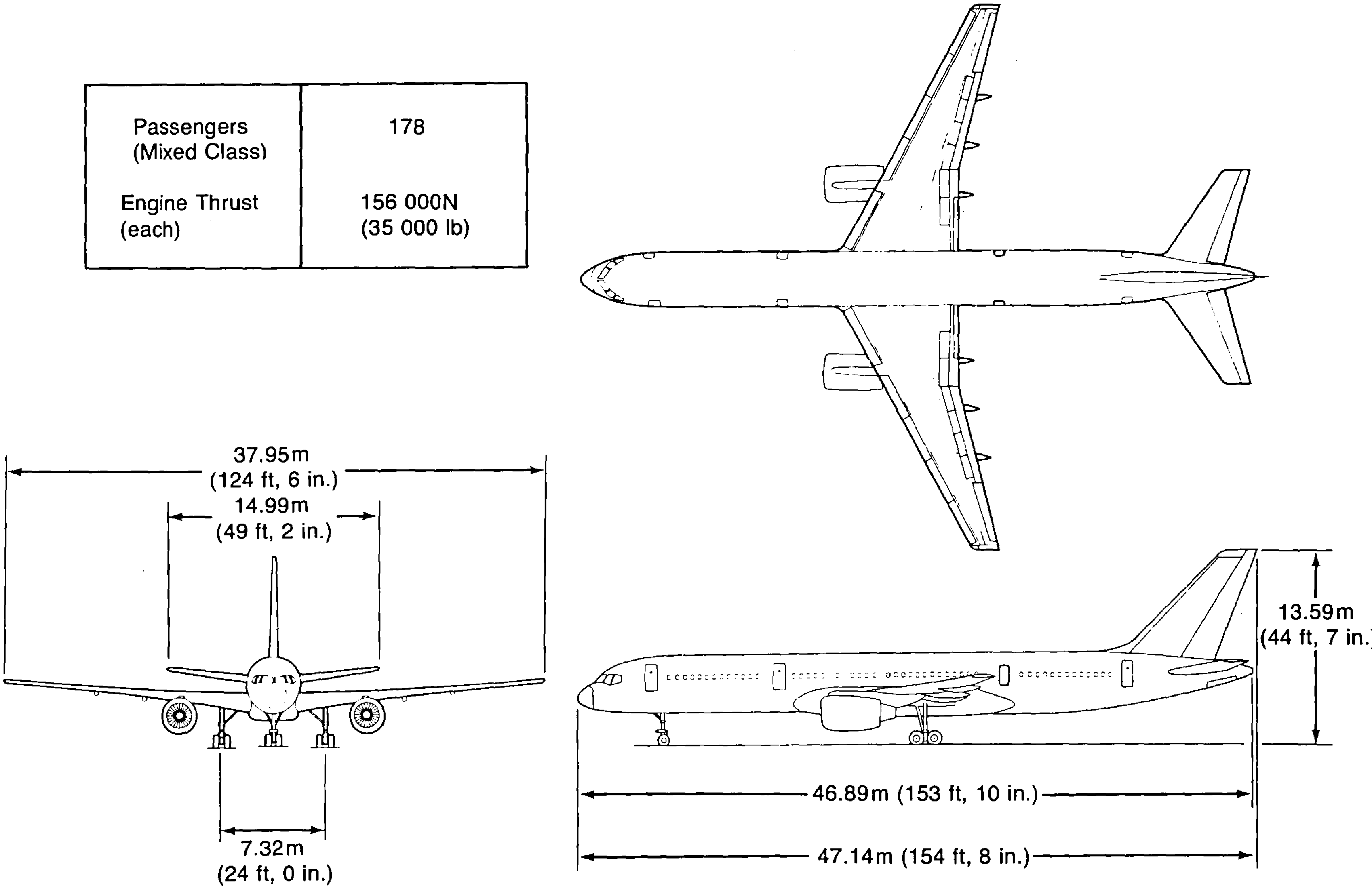


Figure 64. General Arrangement of Test Airplane

Table 12. Principal Characteristics of the Test Airplane

	Metric Units	English Units
Maximum taxi weight	100 236 kg	221 000 lb
Maximum takeoff weight	99 792 kg	220 000 lb
Maximum landing weight	89 813 kg	198 000 lb
Maximum zero fuel weight	83 462 kg	184 000 lb
Engine thrust		
P&W 2037	156 kN approx.	35 000 lb approx.
Passenger capacity–typical		
Mixed class	178	178
All tourist	196	196
Fuel capacity	41 180 liters	10 880 U.S. gal
Cargo capacity		
All bulk	52.11 m^3	1 840 ft^3
Maximum operating speed		
CAS	180 m/sec	350 kn
Mach number	0.86	0.86

11.3.2.2 Ballast System

The test airplane has a system of movable ballast enabling the center of gravity (cg) to be placed anywhere between 7% and 39% mean aerodynamic chord (MAC), the normal airplane cg range. Fixed ballast will be added to shift this range aft in order to attain 55% MAC. A description of the planned ballast system is presented in Appendix A.

11.3.2.3 Flight Test Instrumentation

The flight test instrumentation system consists of both airplane- and ground-station-based equipment. The airplane equipment provides a data acquisition function by recording selected test data from various airplane subsystems and formatting the data as desired for real-time monitoring. A telemetry link with the ground station allows real-time monitoring of system performance during minimum crew flights. Recorded data include subsystem performance parameters and pertinent aircraft states that are available for engineering evaluation after postflight processing.

Test ACT System parameters available within the ACCs will be transmitted to flight test instrumentation via ARINC 429 digital data buses. Real-time monitoring capability of Test ACT performance will be provided via CRT displays, hard copy printers, x-y plotters, strip chart recorders, and frequency spectrum analyses.

11.3.3 GROUND TEST PLAN

Engineering ground tests will be performed to verify that the new and modified systems associated with the Test ACT System function as intended. Functional and vibration tests will be performed. Additionally, operational procedures relative to control system reconfiguration and possible emergency scenarios will be verified.

11.3.3.1 Functional Tests

Functional tests wll be performed to verify correct operation of the modified primary longitudinal flight control system, the FBW and pitch augmentation control system, the WLA system, the cg management system, and the flight test instrumentation and data acquisition system.

11.3.3.2 Ground Vibration Tests of Test Airplane

Ground vibration testing will be performed, if necessary, to verify that the Test ACT-configured airplane with the most aft cg does not have any unexpected or undesirable characteristics compared with the basic airplane.

11.3.3.3 Operational Procedures

Data Acquisition and Display—Procedures for acquiring and displaying system and airplane performance data from the test instrumentation system will be verified.

Flight Program Changes—Procedures for reconfiguring the Test ACT control laws via the flight test programmer will be verified.

Cg Management—Procedures for managing the location of the airplane cg by means of the ballast system will be verified under both normal and emergency operating conditions.

Control System Reconfiguration—Procedures for shifting from the normal primary flight control system operation to Test ACT System operation, and vice versa, will be verified.

11.3.4 FLIGHT TEST PLAN

The objective of the Test ACT System flight test is to validate the performance and safety of the Test ACT pitch FBW and stability augmentation system and the open-loop performance of the WLA system.

11.3.4.1 Test Scope

The scope of the flight test will include the following:

- Evaluation of relaxed stability airplane longitudinal handling qualities with the Test ACT System not operating
- Development of the active control system, using the provisions for inflight software program changes

- Evaluation of the active control system stability and performance

- Evaluation of relaxed stability airplane longitudinal handling qualities with the Test ACT System operating

- Validation of the Test ACT System fault tolerance and safety

11.3.4.2 Test Ground Rules and Assumptions

Prior to flight test, the following will have been completed:

- All flight test conditions and tasks will have been simulated in the 757 engineering flight simulator. A summary of predicted flight test performance will have been prepared for use during real-time evaluation of flight test results.

- Engineering ground test of the Test ACT and associated systems will have been completed.

The flight test plan assumes the following:

- Dedicated flutter testing will not be required. The airplane will be operated well within flutter placards. Flutter engineers will monitor closed-loop testing for absence of system coupling with airplane elastic modes.

- The WLA subsystem will have sensors and computers operational, but no actuators will be installed. The operation of the sensors and control law performance will be monitored, but no dedicated flight tests will be conducted for WLA.

- Initial flight tests will be performed with a minimum crew, necessitating telemetering of flight test data to a ground station for real-time monitoring and evaluation by the engineering team. Subsequent tests will have Test ACT and Flight Test Analysis engineers on board.

- Access to the ACC software programs via the TAC keyboard will be prohibited in flight.

- Loading of alternate, previously verified, control law software modules will be allowed during flight.

- System fault tolerance will be verified by inserting simulated faults at the TAC interface. However, actual initiation of the fault will be controlled by the pilot.

- The greater part of the testing will be performed at aft cg locations that are controllable via the mechanical reversion system, in the unlikely event that the augmentation function is lost. The final phase of testing will evaluate "flight crucial" conditions after the reliability and fault tolerance of the system have been demonstrated.

11.3.4.3 Detailed Flight Test Phases

The flight test will consist of four phases:

1. Determination of handling qualities of the unaugmented and augmented airplane at a benign flight condition; system development to correct any shortcomings discovered during flight test

2. Continuation of phase 1-type testing but with the flight envelope expanded to include most critical flight conditions up to V_{MO}/M_{MO} and maximum aft cg locations that allow safe control unaugmented; testing of takeoff, landing, and cruise configurations

3. Demonstration of Test ACT System fault tolerance

4. Demonstration of handling qualities with the cg location moved farther aft such that the Test ACT System is performing a flight-crucial function; i.e., one required for safe flight

11.3.4.4 Flight Test Results

Flight test results will be appropriately documented to describe the flight performance of the Test ACT System.

12.0 CONCLUDING REMARKS

Page

12.0 CONCLUDING REMARKS

The objectives of the Integrated Application of Active Controls (IAAC) Project were threefold: (1) to establish a credible assessment of the potential benefit of Active Controls Technology (ACT) applied to a commercial transport, (2) if there were a positive potential benefit of ACT, to identify the risks that preclude its use today, and (3) to initiate test and development activities to reduce the risk of such applications in a commercial transport. The first two objectives have been accomplished; the third is well under way with the completion and delivery of the Test ACT System described herein. The test and development work necessary to prepare the ACT technology (represented by this experimental system) for commercial application is just beginning.

12.1 TEST ACT SYSTEM STATUS

The Test ACT System, consisting of flightworthy hardware and software, was designed to represent a potentially certifiable system. It implements the following active control functions:

- Pitch-augmented stability
- Wing-load alleviation, including maneuver-load control and gust-load alleviation

Since the system is necessary for continued safe flight, a natural adjunct to the ACT system is the use of fly by wire for control. Therefore the Test ACT System implements both control and augmentation through the use of fly by wire.

Collins Air Transport Division of Rockwell International was selected to fabricate the Test ACT System equipment, including all system sensors. The following equipment was built, verified, and delivered:

- Four Active Controls Computers
- One Test ACT Control Panel
- One Preflight Test Panel
- One Flight Test Programmer
- Four pitch-rate gyros
- Two column force transducers with quadruple pickoffs
- Eight linear accelerometers
- One Test ACT Console

12.2 FUTURE WORK

The test phases began with installation of the electronics in the Boeing Digital Avionics Flight Controls Laboratory and will proceed through addition of the servos, installation of the servos and associated mechanisms in a Flight Control Test Rig ("Iron Bird"), and installation and flight test in a Boeing-owned test airplane. This extensive laboratory and flight test of the Test ACT System will identify solutions for the technical risk areas as a step toward validating and establishing confidence that such a crucial system can be developed for a commercial application.

13.0 REFERENCES

Page

13.0 REFERENCES

13.0 REFERENCES

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APPENDIX A: 757 NA001 CENTER-OF-GRAVITY MANAGEMENT FOR TEST ACT

Page

APPENDIX A
757 NA001 CENTER-OF-GRAVITY MANAGEMENT FOR TEST ACT

A.1 GENERAL

The Test ACT System will be flown at center-of-gravity (cg) positions considerably aft of the certificated limits for the 757 aircraft. In order to achieve the desired aft cg limits for flight and remain within the certificated limits for takeoff and landing, a system of fixed and movable ballast is incorporated. This system provides a cg forward of 39% mean aerodynamic chord (MAC) for taxi, takeoff, and landing and enables a rearward shift to 55% in flight.

A.2 TEST ACT BALLAST SYSTEM

The ballast system used is shown schematically in Figure A-1. The 757 Test Configuration is achieved by removing passenger seats, aft lavatories, and passenger-related items. All galleys except for the forward one, the class divider, and miscellaneous storage provisions are removed. Flight test instrumentation racks plus 20 seats for test personnel are installed. Approximately 6800 kg (15 000 lb) of fixed ballast, consisting of 23-kg (50-lb) sand bags, are placed on the floor of the main deck and in the lower cargo compartments. Center-of-gravity control during flight is accomplished by means of fore and aft water barrels on the main and cargo decks; pumps and interconnecting plumbing are provided for fore and aft wake transfer.

A.3 CENTER-OF-GRAVITY CONTROL

Figure A-2 shows the gross weight—cg diagram for the Test ACT 757 Configuration. The cg for the operating empty weight is seen to be well forward of the certificated aft limit, but loading the fixed ballast moves it to approximately that limit. The water ballast system then allows the cg to be varied from within the certified limits for takeoff and landing to 55% for flight testing.

Fuel only slightly affects the airplane cg, which ensures reasonable test duration for each selected cg location.

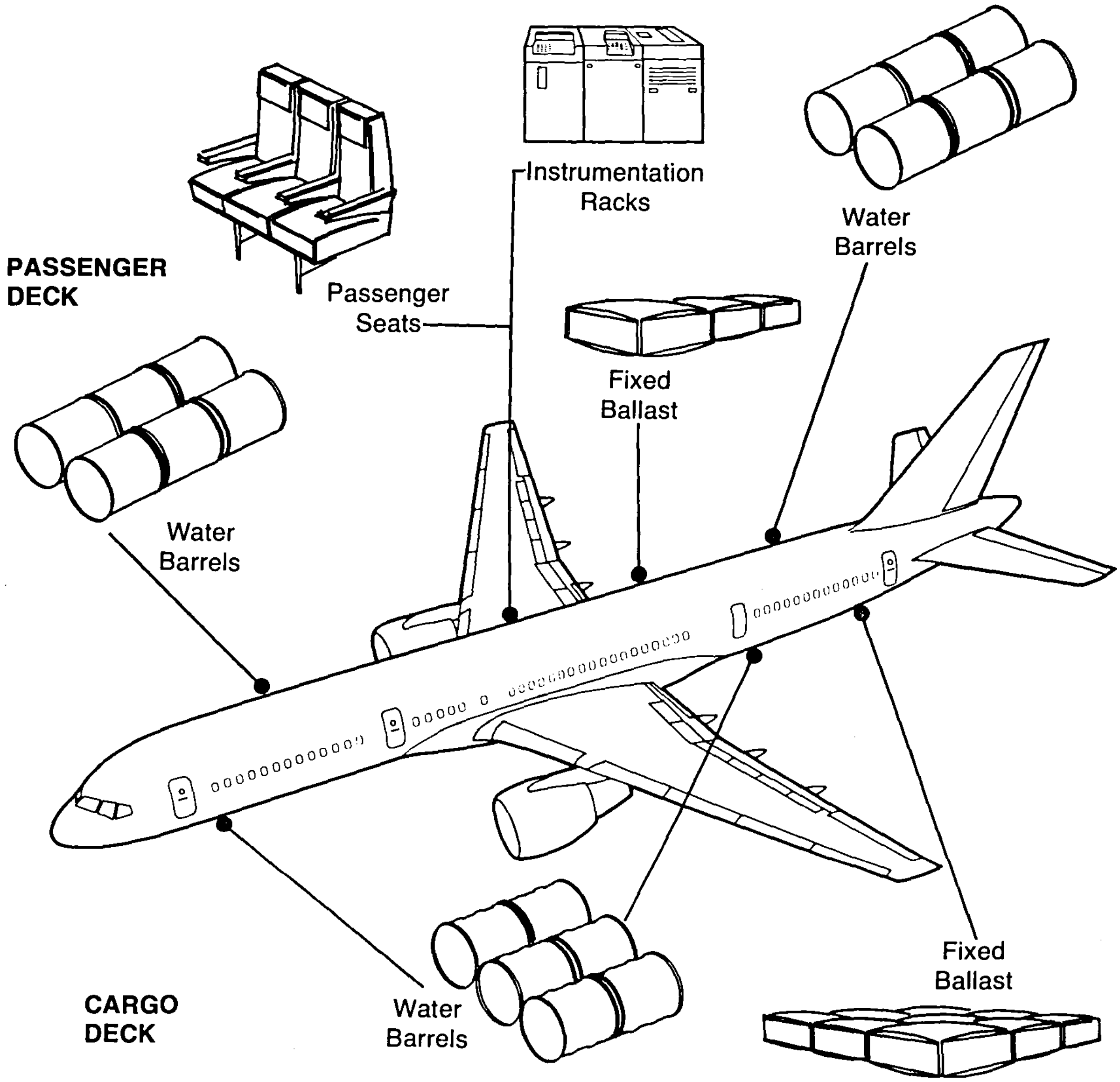


Figure A-1. Test Airplane Center-of-Gravity Control

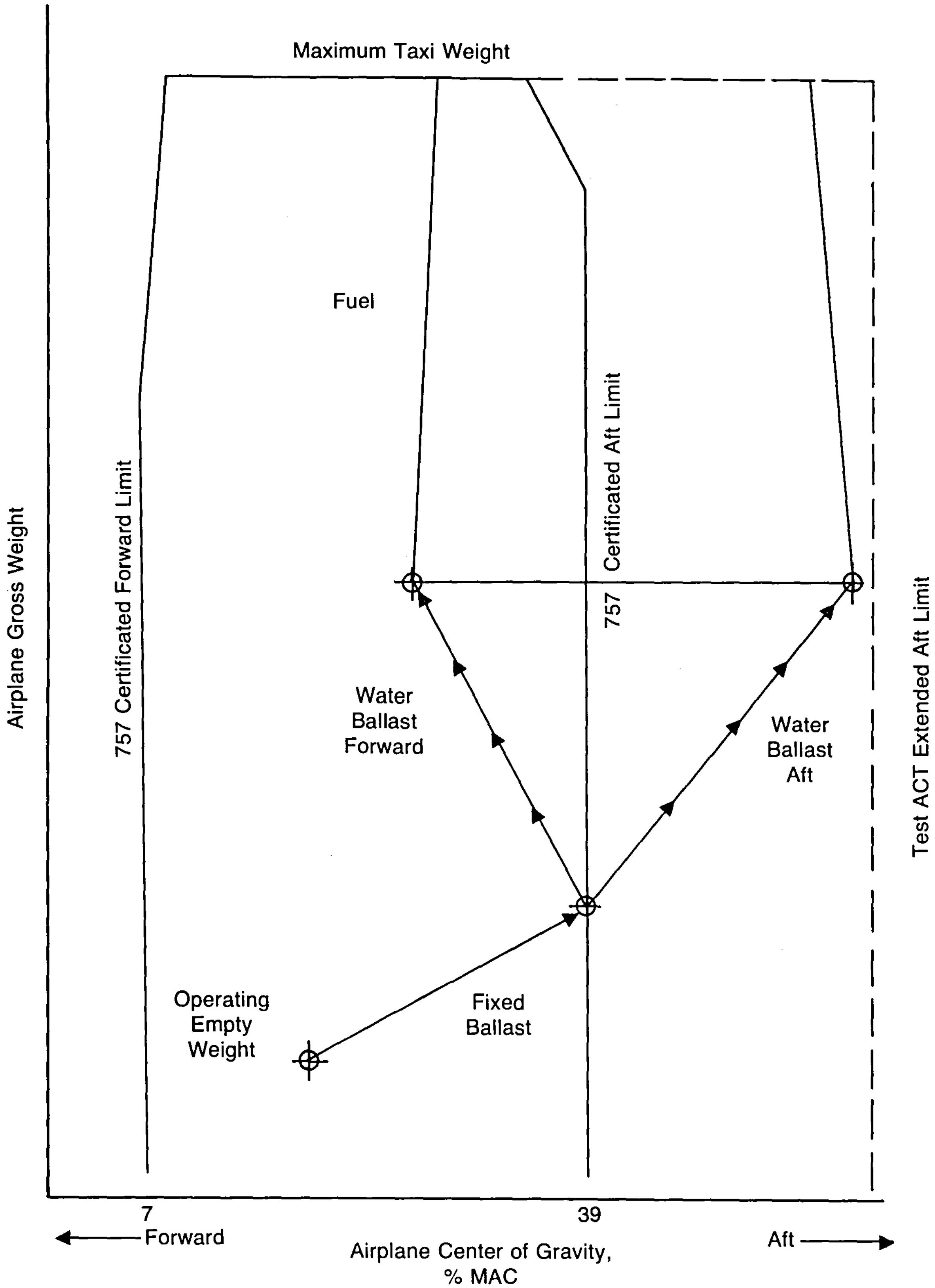


Figure A-2. Test Airplane Gross Weight–Center-of-Gravity Diagram

APPENDIX B: ELEVATOR CONTROL LINKAGE HYSTERESIS TESTS

APPENDIX B: ELEVATOR CONTROL
LINKAGE HYSTERESIS TESTS

APPENDIX B
ELEVATOR CONTROL LINKAGE HYSTERESIS TESTS

This appendix reports on tests run in the 757 Flight Control Test Rig (FCTR, "Iron Bird") facility. The tests investigate the hysteresis of the left elevator control linkage between the aft quadrant and the middle power control unit (PCU) to evaluate its effect on operation of the Active Controls Technology (ACT) fly-by-wire system proposed for testing on NA001.

B.1 TEST ASSEMBLY

The system tested was the complete elevator control system. The following special equipment was installed:

- Elevator surface protractor
- Feel computer pneumatic system
- Rig pins

These two procedures were executed per 757 specifications before the hysteresis runs described in the following:

- Elevator system rigging
- Elevator system functional test

B.2 TEST PROCEDURES

Refer to Figure B-1 for the following test procedures used during the 757 Iron Bird hysteresis investigations.

Test runs 1 through 5 were made to investigate the effects of proper tightness of fasteners in the control system linkages under test. Runs 1 and 2 checked total hysteresis (pitch autopilot input voltage versus elevator trailing-edge angle) and control linkage hysteresis (movements of aft quadrant versus middle PCU input lever) before tightening the fasteners in the linkage to the proper torque values. Runs 3 through 5 repeated the test after adjusting the linkage fasteners. All of these tests are grouped together and labeled "TEST A" in Table B-1.

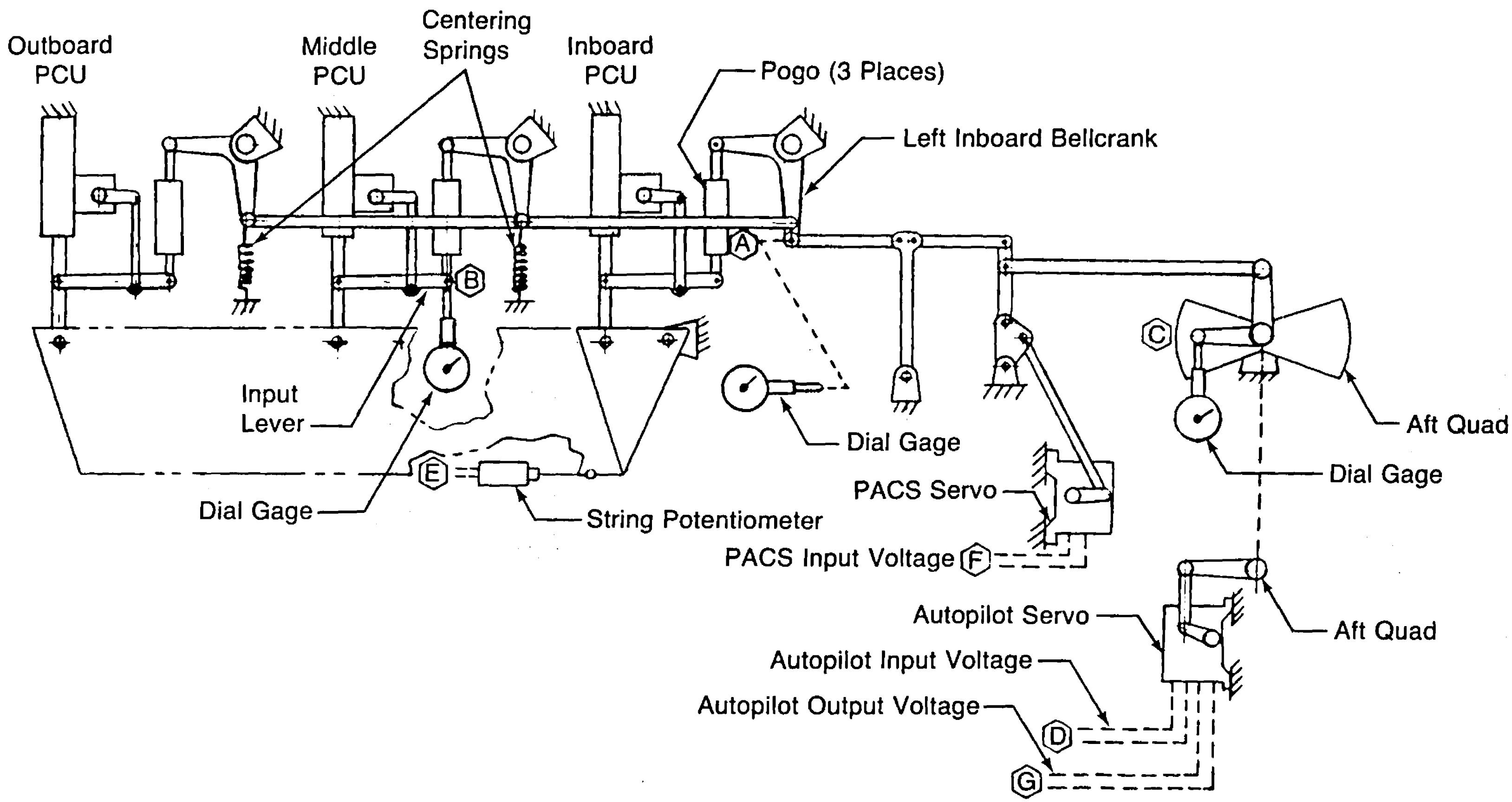


Figure B-1. Test Configuration, FCTR Control Linkage Hysteresis Measurements, Elevator System–Left Side

Table B-1. Hysteresis in 757 Elevator Control System

RUN NO.	δ_E (deg)	PERIOD (sec/cycle)	AMPLITUDE (deg δ_E)	HYSTERESIS AVERAGE OF RUNS (deg δ_E)	HYSTERESIS MEASURED BETWEEN[d]	SYSTEM CONFIGURATION	COMMENTS
		TEST "A"	**RETORQUING LINKAGE JOINT BOLTS**				
1&2	+1	100	±0.25	0.0508	C&B, D&E*	• A/P driving	Runs 1&2 take before retorquing
3-5	+1	100	±0.25	0.0274	C&B, D&E*	• PACS servo grounded[b]	Runs 3-5 taken after retorquing Reduction in hysteresis $= \frac{(0.0508-0.0274)\text{x}100}{0.0508} = 46.06\%$
		TEST "B"	**TOTAL LINKAGE HYSTERESIS**				
3-5	+1	100	±0.25[a]	0.0274	C&B, D&E*	• A/P driving	Average of total hysteresis $= \frac{0.0274+0.0326+0.0398+0.0339}{4}$ $= 0.0334\text{-deg } \delta_E$
19&20	+5	105	±0.25	0.0326	C&B, D&E*	• PACS servo grounded[b]	
15-18	+10	93	±0.25	.0398	C&B D&E*		Feel unit disconnected to prevent A/P disengaging
16&17	+13	94	±0.25	.0339	C&B D&E*	• Feel & cent. unit disconnected	
		TEST "C"	**TEST ACT CONFIGURATION (PACS SERVO DRIVING)**				
28-30	+1.25	100	±0.25[a]	0.0680	F&E*	• PACS servo driving • A/P grounded	Represents Test ACT configuration
		TEST "D"	**EFFECT OF CENTERING SPRING REMOVAL**				
6&7	+5	100	±0.25	0.0410	C&B, D&E*	• A/P driving • Feel unit disconnected	Runs 6&7 taken before spring removal
8&9	+5	100	±0.25	0.0759	C&B, D&E*	• PACS servo grounded[b]	Runs 8&9 taken after spring removal Increase in hysteresis due to spring removal = 85%
		TEST "E"	**EFFECT OF LOAD REVERSAL ON POGO SUPPORT BEARINGS**				
19&20	+5	105	± 0.25	0.0326	C&B D&E*	• A/P driving	Contribution to linkage hysteresis of PCU bellcrank, and pogo support bearings downstream of centering springs.
19&20	+5	105	± 0.25	0.0207	C&A D&E*	• Feel unit disconnected	δ_E = +5 deg contribution $= 100 \text{ x}\left[\frac{.0326-.0207}{.0326}\right] = 36.50\%$
15&18	+10	93	± 0.25	0.0398	C&B D&E*	• PACS grounded[b]	
15&18	+10	93	±0.25	0.0193	C&A D&E*		δ_E = +10 deg contribution $= 100 \text{ x}\left[\frac{.0398-.0194}{.0398}\right] = 51.51\%$
16&17	+13	94	±0.25	0.0339	C&B D&E*		δ_E = +13 deg contribution $= 100 \text{ x}\left[\frac{.0339-.0194}{.0339}\right] = 42.77\%$
16&17	+13	94	±0.25	0.0194	C&A D&E*		

a Feel unit connected
b PACS system grounded by existing springs
c Comparison of results of test "B"
d See Figure B-1
* Overall hysteresis measured on x-y plotter

Table B-1. Hysteresis in 757 Elevator Control System (Continued)

RUN NO.	δ_E (deg)	PERIOD (sec/cycle)	AMPLITUDE (deg δ_E)	HYSTERESIS AVERAGE OF RUNS (deg δ_E)	HYSTERESIS MEASURED BETWEEN [d]	SYSTEM CONFIGURATION	COMMENTS
		TEST "F"	**TOTAL HYSTERESIS (A/P HYSTERESIS NOT INCLUDED)**				
22&23	+1	100	± 0.25	0.0945	G&E*	• A/P driving • Feel unit disconnected	Runs taken to isolate surface PCU hysteresis from total At δ_E = +1 deg : .0945 deg -.0274 deg = .0671 deg δ_E PCU hysteresis
24&25	+5	100	± 0.25	0.0810	G&E*	• PACS grounded[b]	At δ_E = +5 deg : .0810 deg -.0326 deg c = .048 deg δ_E PCU hysteresis
		TEST "G"	**OVERALL HYSTERESIS (A/P DRIVING)**				
19&20	+5	105	± 0.25	0.0775	D&E*	• A/P driving • Feel unit disconnected • PACS grounded[b]	
		TEST "H"	**EFFECT OF PCU BIAS SPRINGS- PACS DRIVING**				
31-34	1.25	97	0.25	0.0448	F & E*	• Elev PCU bias springs installed • PACS driving • Right A/P engaged • Feel unit connected • All hyd on	Same as Test "C" with the addition of bias springs. Reduction in hysteresis $100 \times \left[\frac{.0680 - .0448}{.0680}\right] = 34.1\%$
		TEST "I"	**EFFECT OF PCU BIAS SPRINGS – A/P DRIVING**				
35-37	5	103	±0.25	0.0912	D & E*	• Elev PCU bias springs installed • Center A/P driving	Same as Test "G" with the addition of bias springs. Test "I" increased hysteresis, under investigation.
38-39	1	37	±0.25	0.1033	D & E*	• Feel unit disconnected • All hyd on	

a Feel unit connected
b PACS system grounded by existing springs
c Comparison of results of Test "B"
d See Figure B-1
* Overall hysteresis measured on x-y plotter

Test runs 6 through 9 were made to investigate the effects of removing the control system linkage centering springs while cycling the elevator ± 0.25 deg from 5 deg down. Runs 6 and 7 recorded hysteresis data before spring removal, and runs 8 and 9 recorded data after spring removal. They are grouped as "TEST D" in Table B-1.

In test runs 10 through 14, mechanical dial gages were used to measure the movement of the intermediate linkages from the aft quadrant to the inboard bellcrank. This would allow the hysteresis contribution of the elevator PCUs to be determined from the overall hysteresis measurements. Runs 10 and 11 recorded data cycling the elevator at ± 0.25 deg from 5 deg down, while runs 12 through 14 recorded data at ± 0.25 deg from 1-deg down elevator.

Test runs 15 through 20 were made to measure the effects of increasing linkage load on hysteresis and were accomplished by disconnecting the feel and centering unit from the aft quadrant and cycling the linkage ± 0.25 deg at increasing elevator down angles. Runs 15 and 18 were taken at 5-deg down elevator, runs 16 and 17 at 10-deg down elevator, and runs 19 and 20 at 13-deg down elevator (the pitch autopilot authority limit). These runs also utilized dial gages at the middle PCU input lever and the inboard bellcrank to supplement the data of runs 10 through 14. These tests are grouped together and labeled as "TEST E" and are then combined with runs 3 through 5 as "TEST B" in Table B-1. Run 21 was intentionally deleted.

Runs 22 through 25 were conducted to determine the hysteresis contribution of the driving pitch autopilot servo. This was accomplished by plotting elevator trailing-edge angle against autopilot servo output voltage instead of input voltage previously monitored. Comparison of data from these runs with data from runs 12 through 14 at 1-deg down elevator and runs 15 and 18 at 5-deg down elevator resulted in the autopilot servo internal hysteresis. These tests are grouped and labeled as "TEST F" in Table B-1. Runs 26 and 27 were intentionally deleted.

Runs 28 through 30 were made to gather hysteresis data while driving the control system linkage at a point nearest that of the Test ACT System. By driving the linkage with the Pitch Augmentation Control System (PACS) servoactuator, a portion of the linkage not common to the Test ACT System could be eliminated. Since the PACS servo travel is limited by mechanical stops internally, the cycling had to be performed at ± 0.25 deg from

1.25-deg down elevator. The difference between this point and the 1-deg down point of previous tests is negligible. These tests are grouped and labeled as "TEST B" in Table B-1.

Test runs 31 through 34 were made to measure the effect of a PCU bias spring on the elevator hysteresis. The runs were made with the PACS driving the system ± 0.25 deg from 1.25-deg down elevator. These runs are grouped and labeled as "TEST H."

Test runs 35 through 39 were also made to measure the effect of a PCU bias spring on elevator hysteresis. The runs were made with the autopilot servo driving the system. Runs 35 through 37 drove the elevator ± 0.25 deg at 5 deg down. Runs 38 through 39 drove the elevator ± 0.25 deg at 1 deg down. These runs are grouped and labeled as "TEST I."

B.3 SUMMARY OF RESULTS

Satisfactory performance of the fly-by-wire system projected for the 757 elevator control (Test ACT Program) depends largely on the level of hysteresis in the mechanical path between the electric command servos and the input to the surface PCUs.

An analysis that considered the contributions of the Test ACT servos, the surface PCUs, and the intermediate linkage resulted in a calculated hysteresis of 0.15-deg surface angle (δ_E). In order to verify this value experimentally, measurements of hysteresis in the control path between the pitch autopilot servos and the elevator surface were taken from the 757 FCTR elevator control system. In addition, measurements were taken between the PACS, an experimental series pitch augmentation servo, and the elevator surface. It should be noted that the analysis included the effects of low temperature on bearing friction, as well as the additional linkages of the Test ACT System, while the values of hysteresis measured on the FCTR were exclusive of these elements.

The following measurements were recorded at approximately 0.01-deg δ_E per second:

- Overall (PACS input voltage versus surface rotation) hysteresis measured was 0.07-deg δ_E.

- Overall (PACS input voltage versus surface rotation) hysteresis measured, with PCU bias springs, was 0.045-deg δ_E.

- Total intermediate linkage hysteresis measured was 0.03-deg δ_E.

- Excluding local PCU input links, intermediate linkage hysteresis measured was 0.02-deg δ_E.

- Across PCUs, hysteresis measured was 0.04-deg δ_E.

- Retorquing of linkage bolts reduced hysteresis from 0.0508- to 0.0274-deg δ_E (46%).

B.4 DISCUSSION AND CONCLUSIONS

Two methods were used to measure hysteresis in the elevator control system. One plotted overall hysteresis loops on a plotter using a string potentiometer to measure surface position versus autopilot or PACS servo input voltage. The other method took dial indicator measurements of travel at points along the linkage (fig. B-1). Measurements using both methods were taken simultaneously. To assess the effects of load increase in the linkage on hysteresis, plots were taken at about four different elevator trim angles (1, 5, 10, and 13 deg) while cycling the surface at $\pm$0.25-deg δ_E at a period of 100 sec/cycle—the maximum rate that allowed reading of the dial gages. Changes in elevator angles were accomplished by applying sinusoidal and offset voltages to the center hydraulic system autopilot servo. The results of these runs, shown in Figure B-2, illustrate that the linkage hysteresis is not extremely sensitive to load increases.

To determine the effect on hysteresis of insufficient tightness of linkage joint bolts, two runs were made—one before and one after tightening all bolts in the linkage. A reduction in hysteresis of 46% was measured on the run subsequent to tightening (table B-1, runs 1 through 5, "TEST A"). The PCU pogo rod end bearings, because they are located downstream of the centering springs, will be subjected to load reversals and will therefore contribute a significant amount of hysteresis and backlash to the linkage system. Runs 15 through 20, "TEST E," showed a contribution of 36.5% and 51.5% to the total linkage hysteresis.

Most tests, as noted herein, were run using the parallel autopilot servo to cycle the elevator. To better represent the Test ACT Configuration, hysteresis loops were also plotted using the series PACS servo input. Loops were plotted using inputs from the

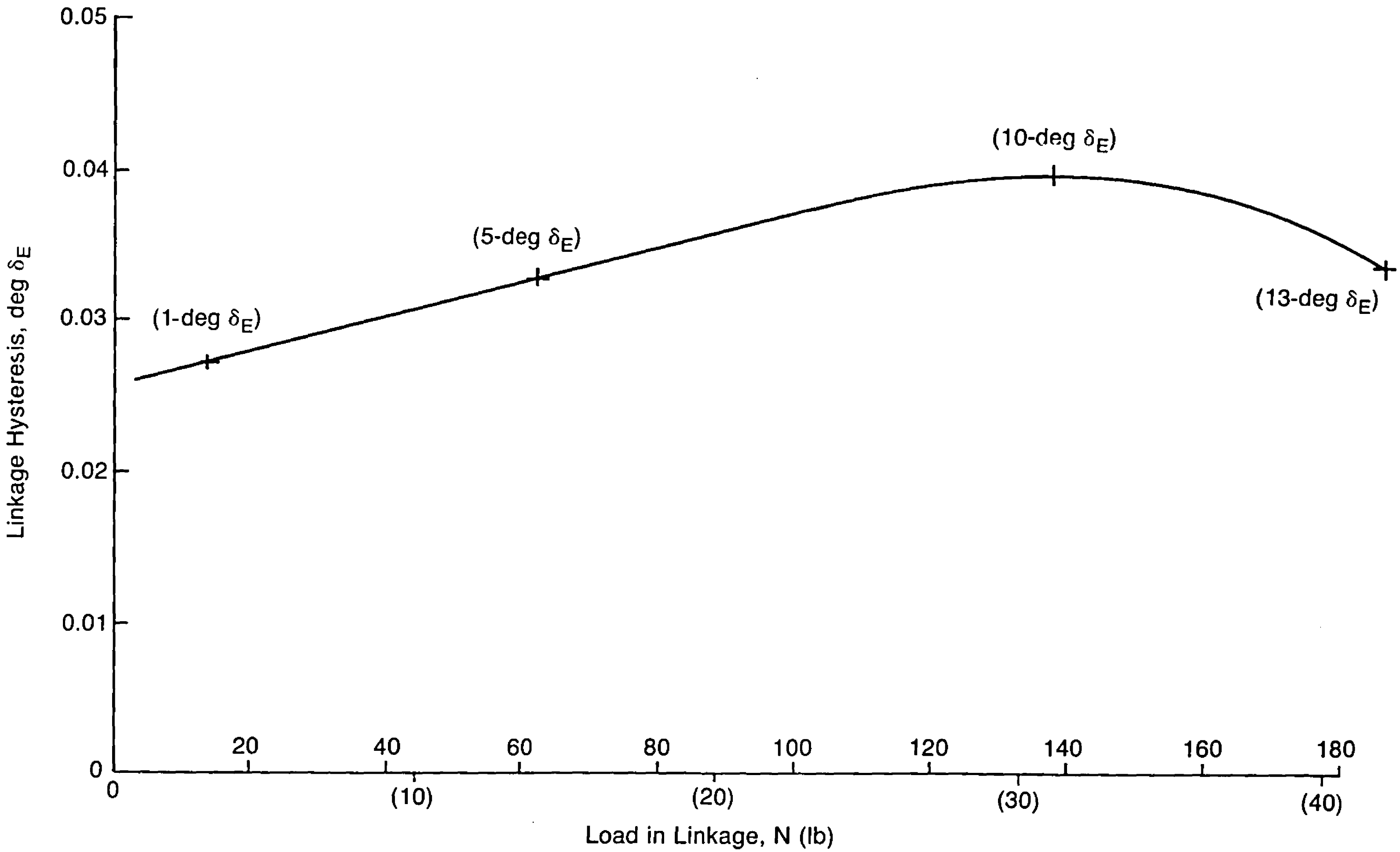


Figure B-2. Test ACT Program Hysteresis Study–757

PACS servo input voltage versus elevator trailing-edge string potentiometer output voltage, "TEST C." These measurements were taken with one autopilot engaged in order to increase the stiffness-to-ground of the aft quadrant. The hysteresis measured in this case was 0.07-deg δ_E and is shown in Figure B-3.

The elevator PCUs were modified by installing a 2.2-N (0.5-lb) servovalve bias spring. The conditions of "TEST C" were repeated with the modified PCUs. This was labeled as "TEST H." Under these conditions the average hysteresis was reduced to 0.045-deg δ_E and is shown in Figure B-4.

Runs 6 through 9 were made to determine the effect of removing the centering springs on the linkage hysteresis and backlash (fig. B-1). Linkage measurements were taken at the aft quadrant and the input point on the middle surface PCU. Removal of the springs caused an increase of 85% in hysteresis and backlash in the linkage, "TEST D."

To obtain a hysteresis value for the surface PCU and its support bearings, additional runs were made in which autopilot hysteresis was not included. Autopilot output linear variable differential transformer (LVDT) voltage was used to plot hysteresis loops rather than autopilot input voltage. PCU hysteresis was obtained by subtracting the total linkage hysteresis. This was found to be between 0.048- and 0.067-deg δ_E, "TEST F."

These measurements may be compared with those of a similar test performed on the 767 FCTR elevator in November 1982. Those results showed an overall hysteresis (from PACS input to elevator output angle) of about 0.05-deg δ_E, with the linkage contributing 0.03-deg δ_E and the PCUs showing 0.02-deg δ_E. The results from this system, which include PCU valve bias springs, compare closely with those of the 757 system with bias springs added, "TEST H."

B.5 RECOMMENDATIONS

The foregoing results indicate that the existing surface PCUs should be fitted with servovalve bias springs to reduce the freeplay existing between the servovalve and linkage downstream of the centering springs when used in the Test ACT airplane. The elevator linkage joints must be maintained to proper torques throughout the flight test program.

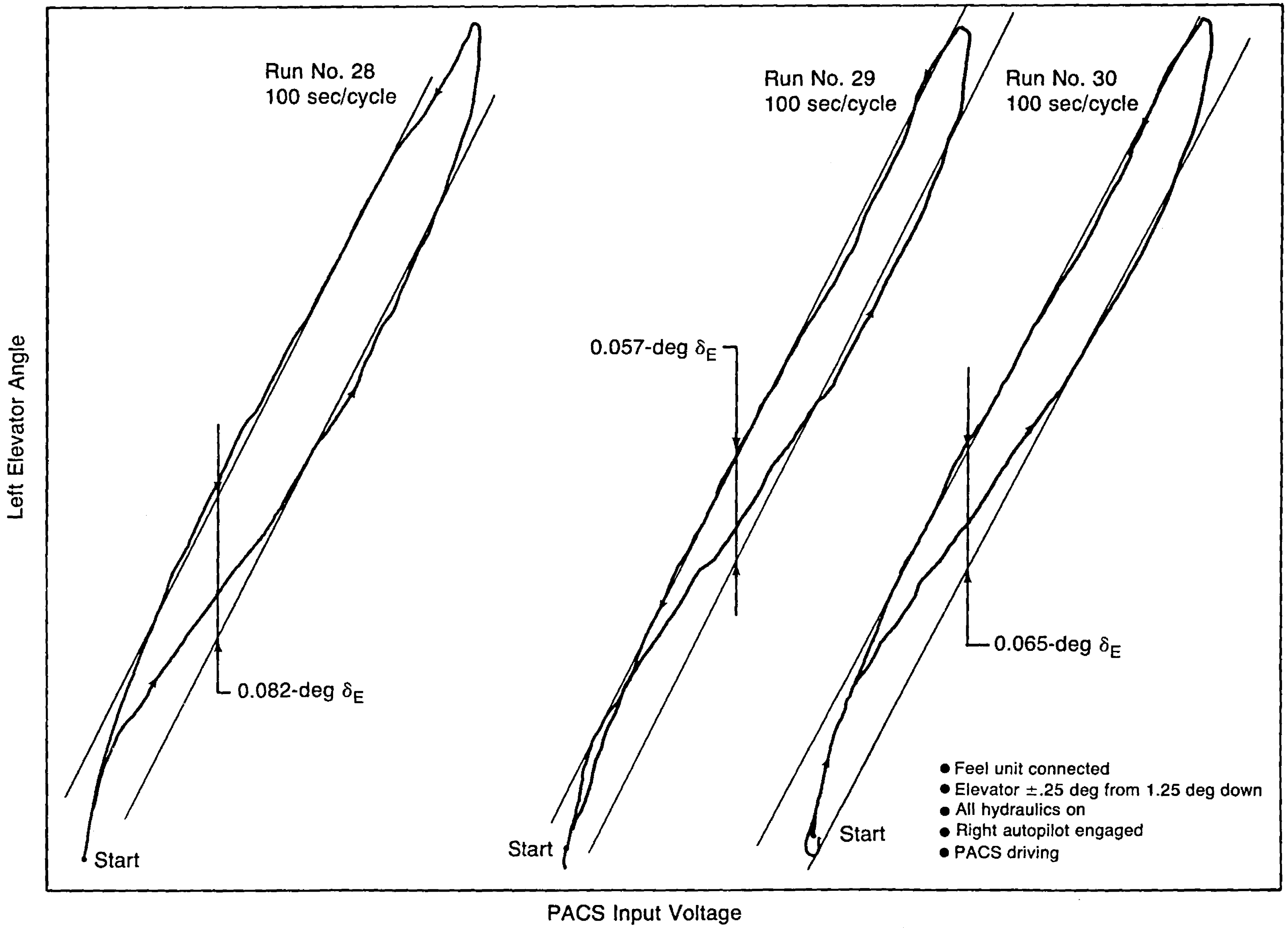


Figure B-3. Hysteresis–PACS to Elevator Surface

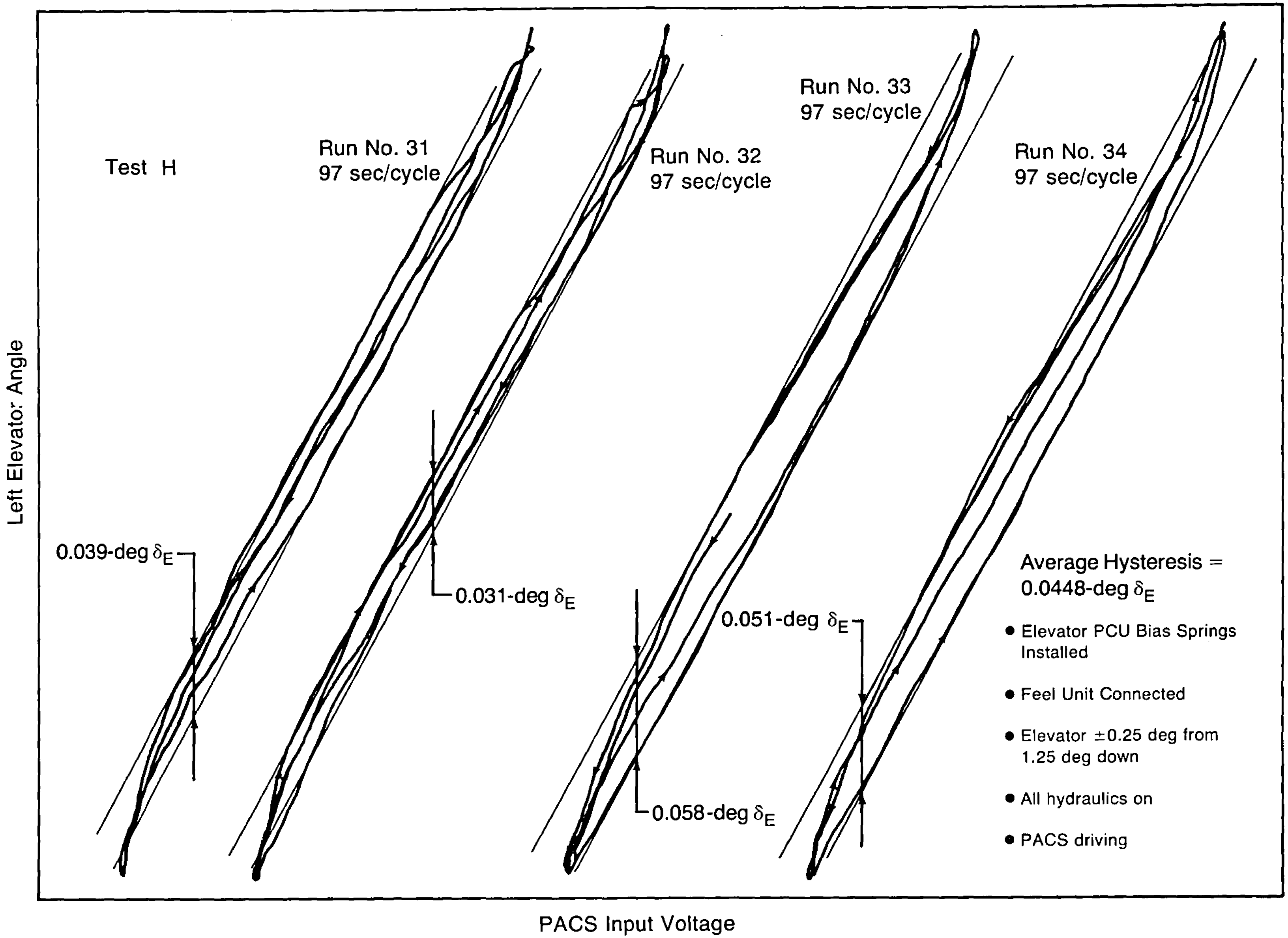


Figure B-4. Hysteresis–PACS With PCU Bias Springs to Elevator Surface

APPENDIX C: STRUCTURAL
CONSIDERATIONS

Page

APPENDIX C
STRUCTURAL CONSIDERATIONS

C.1 GENERAL

Preliminary structural design loads and flutter analyses have been conducted for the 757 Test ACT aircraft to assess the airframe structural capability during flight testing. These analyses have considered the proposed Test ACT flight test envelope, as well as variations in gross weight and center of gravity (cg) provided by the ballast system. The results of these studies consist of preliminary recommended operating limitations and placards, ensuring that the structural capability of the 757 airframe will not be exceeded.

C.2 STRUCTURAL OPERATING LIMITATIONS

Preliminary structural operating limitations are presented in the following:

Takeoff and Landing

- Maintain center of gravity within certificated limits
- Limit sink rate at touchdown to 1.8 m/sec (6 ft/sec)

Flight–Flaps Up

- Limit maneuver load factors to:
  - +2.0g
  - 0g
- Do not operate in greater than moderate turbulence

Flight–Flaps Down

- Limit maneuver load factors to:
  - +1.5g
  - 0g

C.3 STRUCTURAL LOADS ANALYSES

To accomplish the objectives of the flight demonstration program, it will be necessary to conduct flying quality tests at centers of gravity aft of certificated structural design limits. An aft cg envelope that can be achieved in flight has been established using the planned 757 flight test water ballast system, supplemented with 23-kg (50-lb) sand bags piled in the aisles and around seats in the aft end of the airplane and the aft lower lobe. For taxi, takeoff, and landing, the cg is maintained within design limits by storing all water ballast in water barrels on the forebody main deck and in the forward lower lobe. After takeoff, the desired aft cg is obtained by pumping water aft into barrels on the aft-body main deck and in the aft lower lobe.

This arrangement requires appropriate flight placards to ensure that flight test loads remain within existing structural design loads for the horizontal tail, aft-body monocoque, and aft-body floor beams. Landing sink speed limitation may be required if design load limits are not to be exceeded during landing operations.

Time history simulations were performed on the 757 engineering flight simulator to substantiate the static maneuver envelope for aft cg positive check maneuvers, which are the horizontal tail maximum up-load cases. The pitch-augmented stability (PAS) control laws were included to provide airplane and tail load dynamic response due to combined pilot and PAS commands. Figures C-1 and C-2 show selected time history results for one of the critical positive check maneuvers. Results are shown for both the Primary PAS control law and the Essential PAS control law for the same column command. The action of PAS on elevator deflection and airplane response is evident in these figures.

Figures C-3 and C-4 show body load envelopes for symmetric high-speed maneuvers and indicate the effect of the recommended flight limitations.

These analyses of structural design loads will be updated prior to flight test in order to reflect the final Test ACT System Configuration and test requirements. Included will be consideration of balanced flight maneuver, pitch and yaw maneuver, gust, landing, and assumed control system dynamic failure conditions for the final flight test envelope. Critical combinations of speed, altitude, and loading configurations will be analyzed.

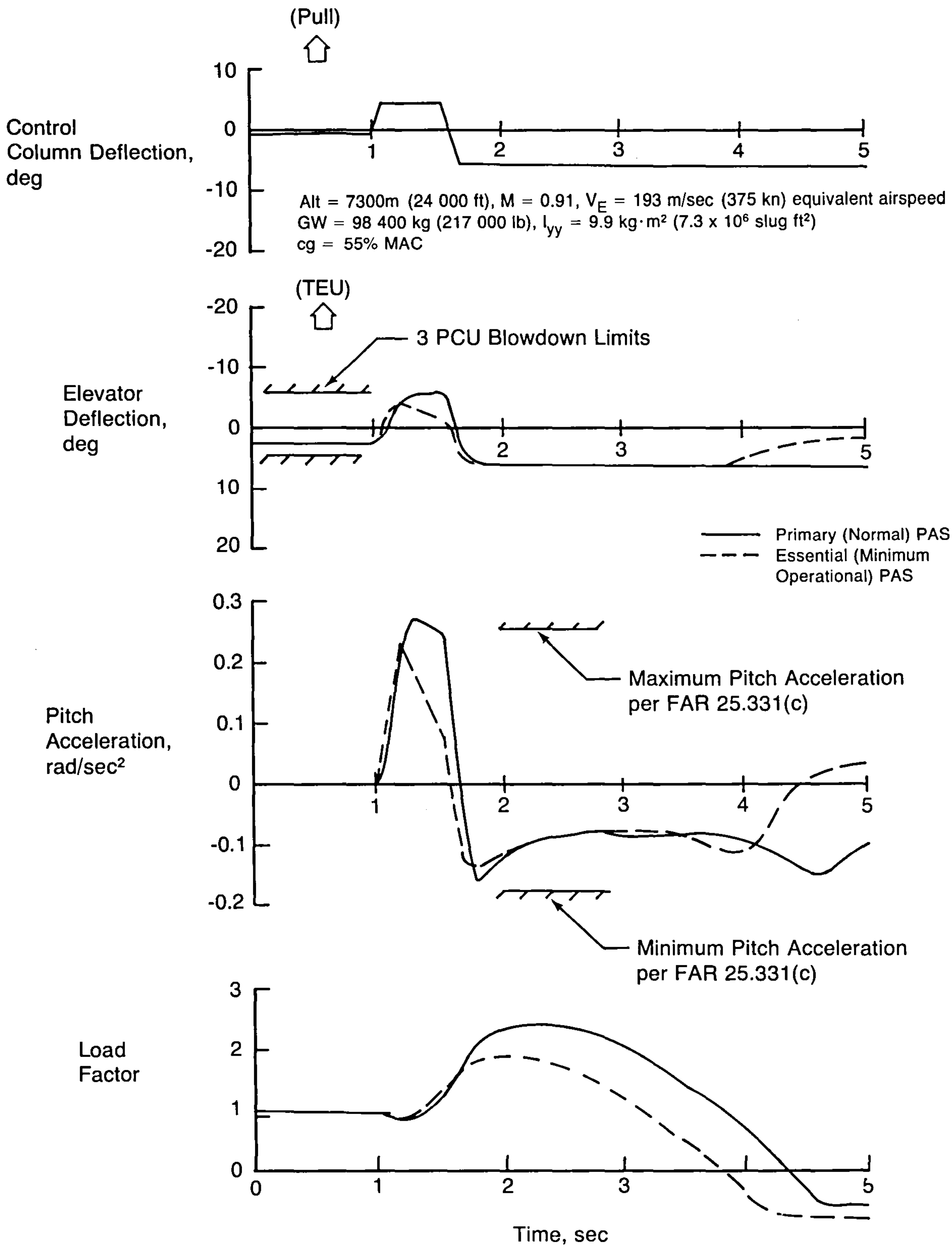


Figure C-1. Positive Check Maneuver at Dive Speed

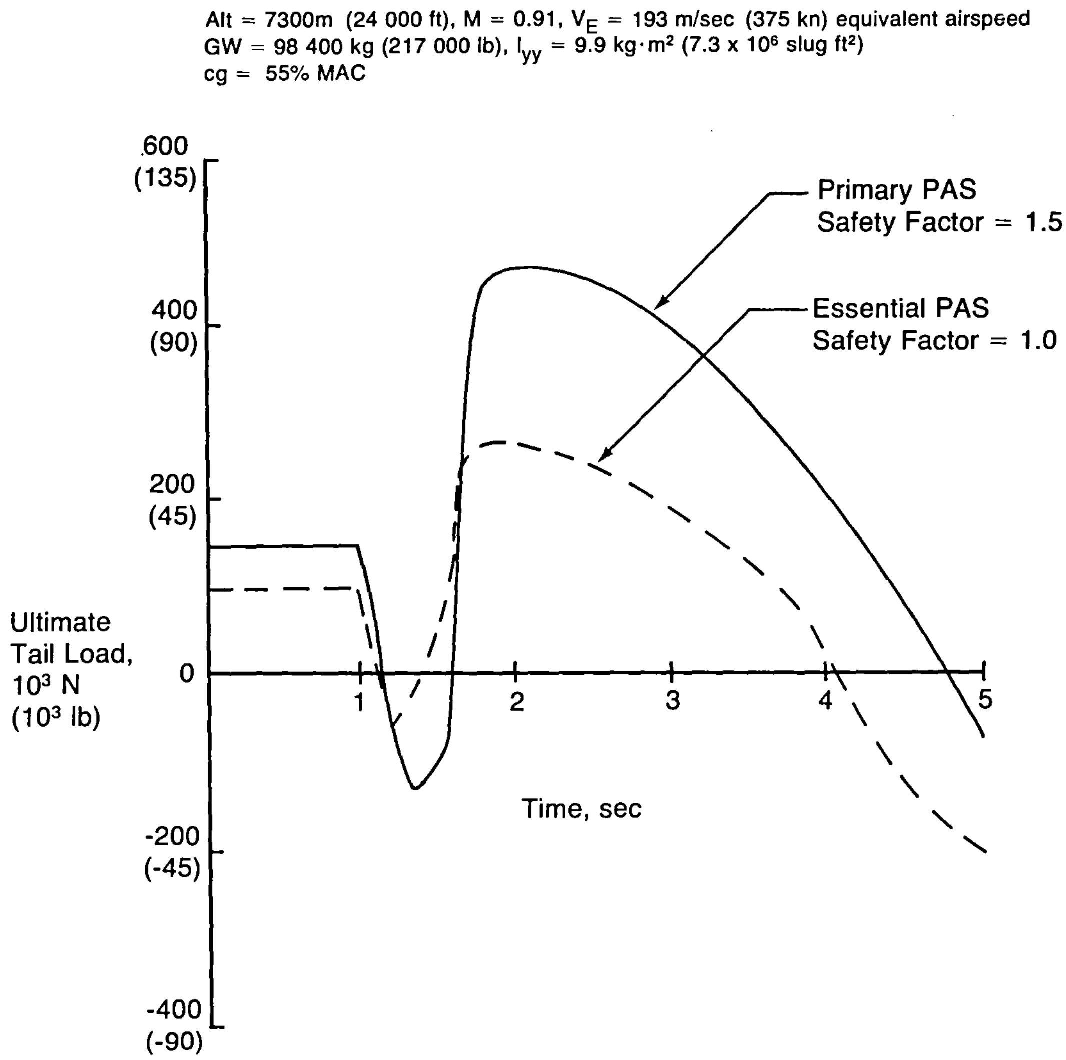
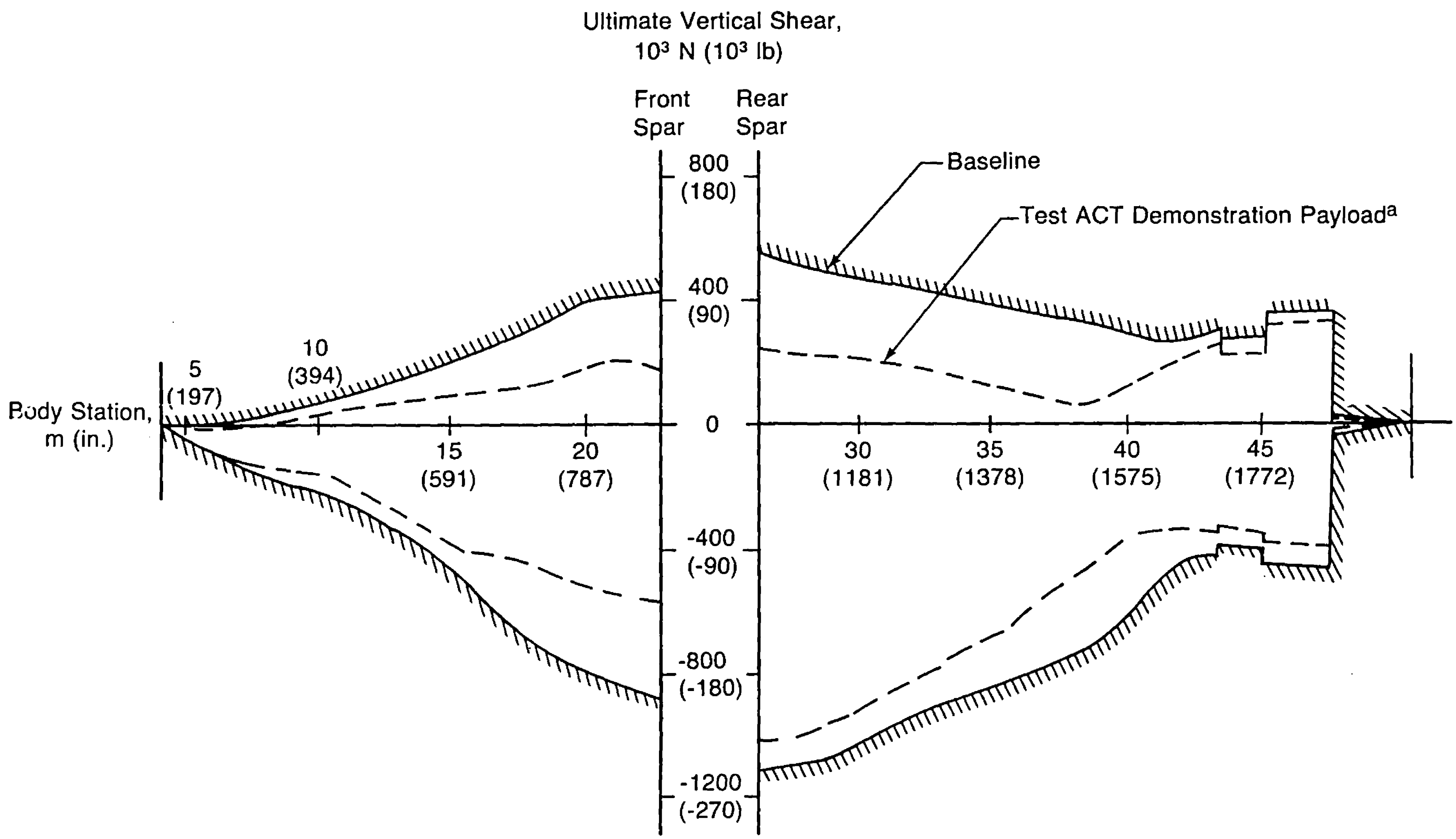


Figure C-2. Tail Load Time Histories for Positive Check Maneuver at Dive Speed

Balanced and pitch maneuvers and gust conditions will include the effects of the Primary plus the Essential system, as well as the Essential system alone, for the range of flight conditions and cg locations that will be flight tested. Loads analyses of critical oscillatory and hardover Test ACT System failure conditions will use beam theory to represent the structural flexibility of the test airplane. Lifting line aerodynamic theory, with empirical corrections derived from wind tunnel and flight tests, will be utilized.

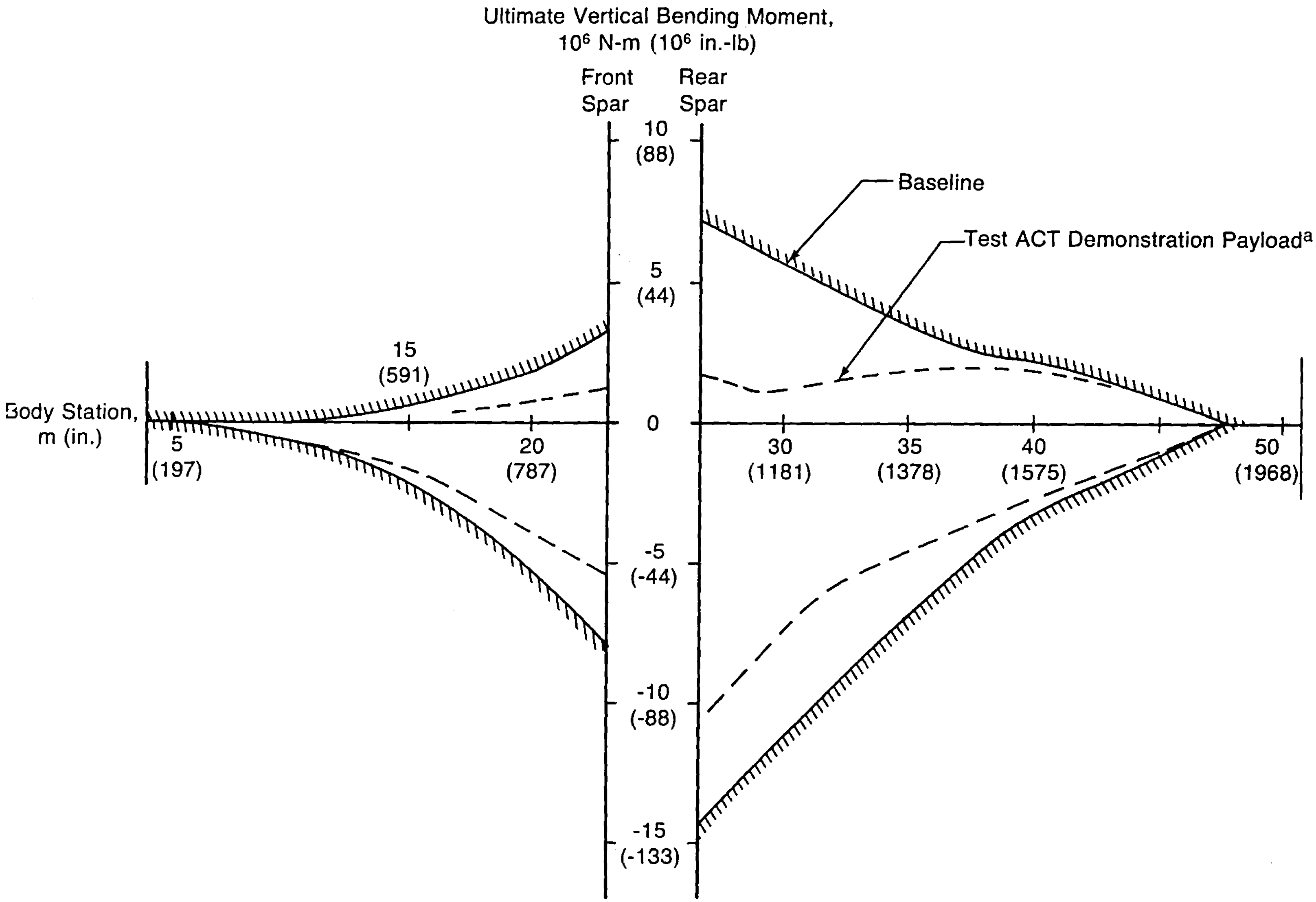
Final test limitations and placards based upon these loads analyses will ensure the structural integrity of the test airplane while minimizing structural changes.



Safety Factor = 1.5 for FAR conditions

[a]Vertical maneuver load factor limited between 0 and 2 on Test ACT demonstration

Figure C-3. Fuselage Vertical Shear Envelope for Flight Conditions



Safety Factor = 1.5 for FAR conditions

[a]Vertical maneuver load factor limited between 0 and 2 on Test ACT demonstration

Figure C-4. Fuselage Vertical Bending Moment Envelope for Flight Conditions

APPENDIX D: MECHANICAL CONTROL SYSTEM MODIFICATIONS

APPENDIX D
MECHANICAL CONTROL SYSTEM MODIFICATIONS

This appendix presents an expansion of the material in Section 8.1.3. It is a more detailed description of the mechanical changes to the flight deck under-floor and empennage areas of the test airplane.

D.1 FLIGHT DECK INSTALLATIONS

The current 757 control column and tension regulator installation is shown in Figure D.1-1. A cam and follower breakout unit limits torque that can be transmitted between columns in the column bus tube. Separated bearings in the bus tube overlap allow independent rotation and provide bending continuity across the joint. The tension regulator quadrants are driven by motion imparted to symmetrical cranks by rods from each control column; rotations of the two quadrants are in opposite directions.

Test ACT flight deck installations are shown in Figure D-2. Removal of the follower and springs from the bus tube override allows the two columns to rotate independently on the tube splice bearings, described previously. The first officer's column is disconnected from the right tension regulator by removing the input rod, and the right and left tension regulators are connected with a diagonal bus to preserve the stiffness of two cable loops for elevator inputs from the captain's column (fig. D-3, 3 sheets).

The Test ACT pilot input mechanism, supported on truss-like brackets attached to flight deck beams aft of the first officer's column, consists of a torque tube, mounted parallel to the column bus tube, to which the feel cam and input force transducers are attached. The column is connected to the unit through a rod fastened between the existing column crank clevis and a crank pivoted on the input mechanism torque tube. The load path from this crank to the feel cam is divided to pass through two four-channel force transducers such that the sum of the signals in a single channel from each transducer is approximately proportional to the column load.

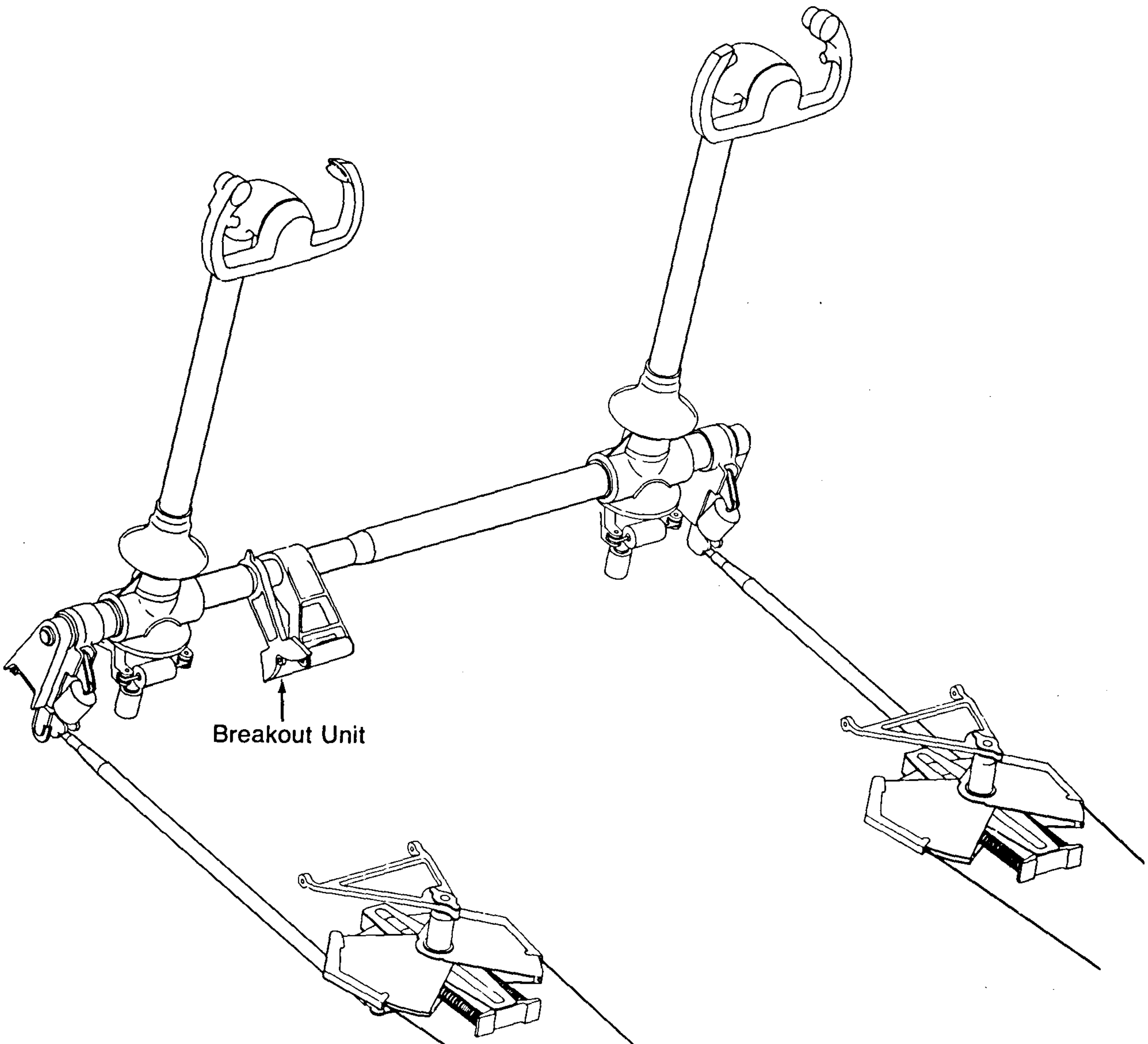


Figure D-1. 757 Column and Tension Regulator Installation

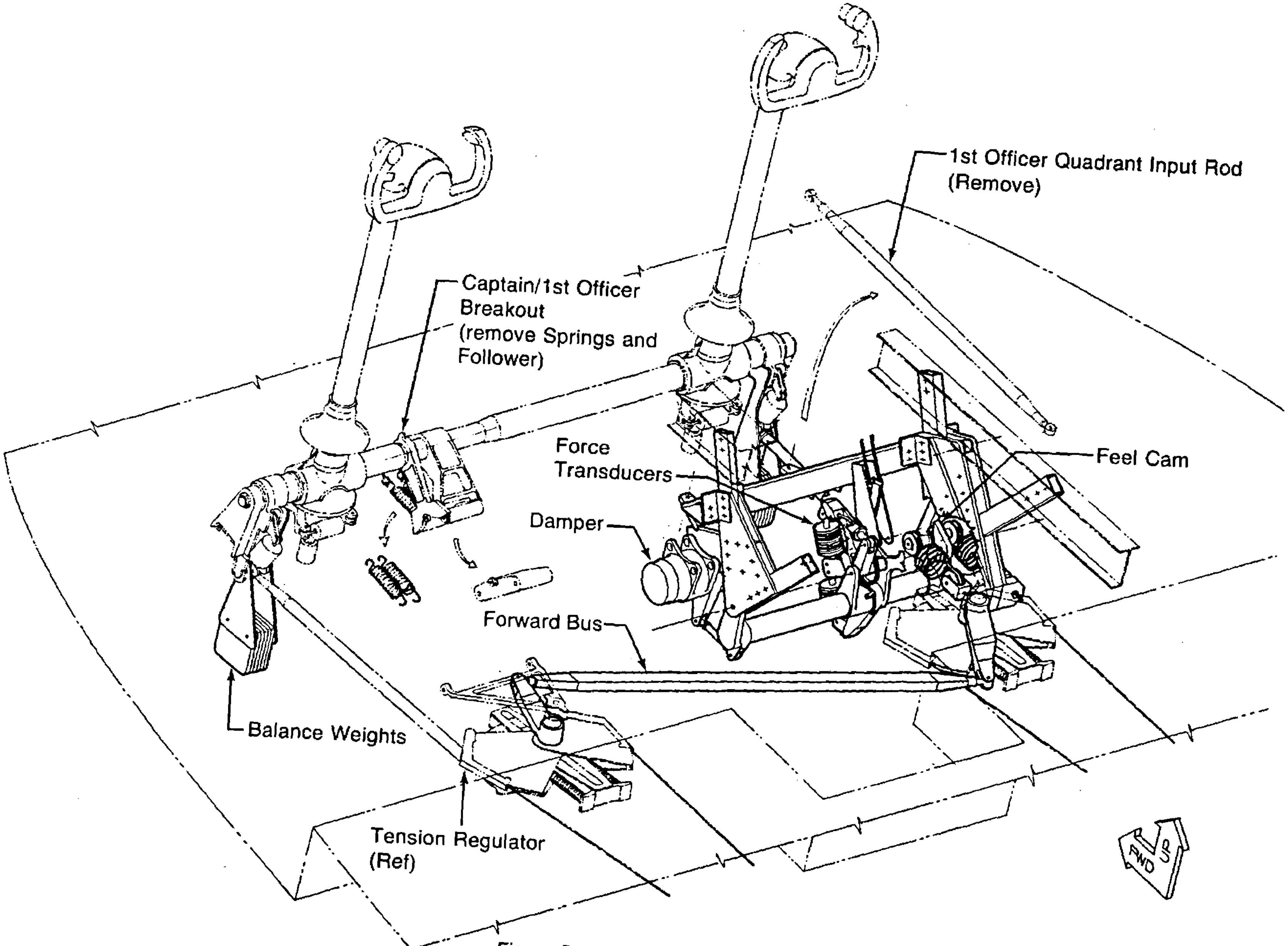
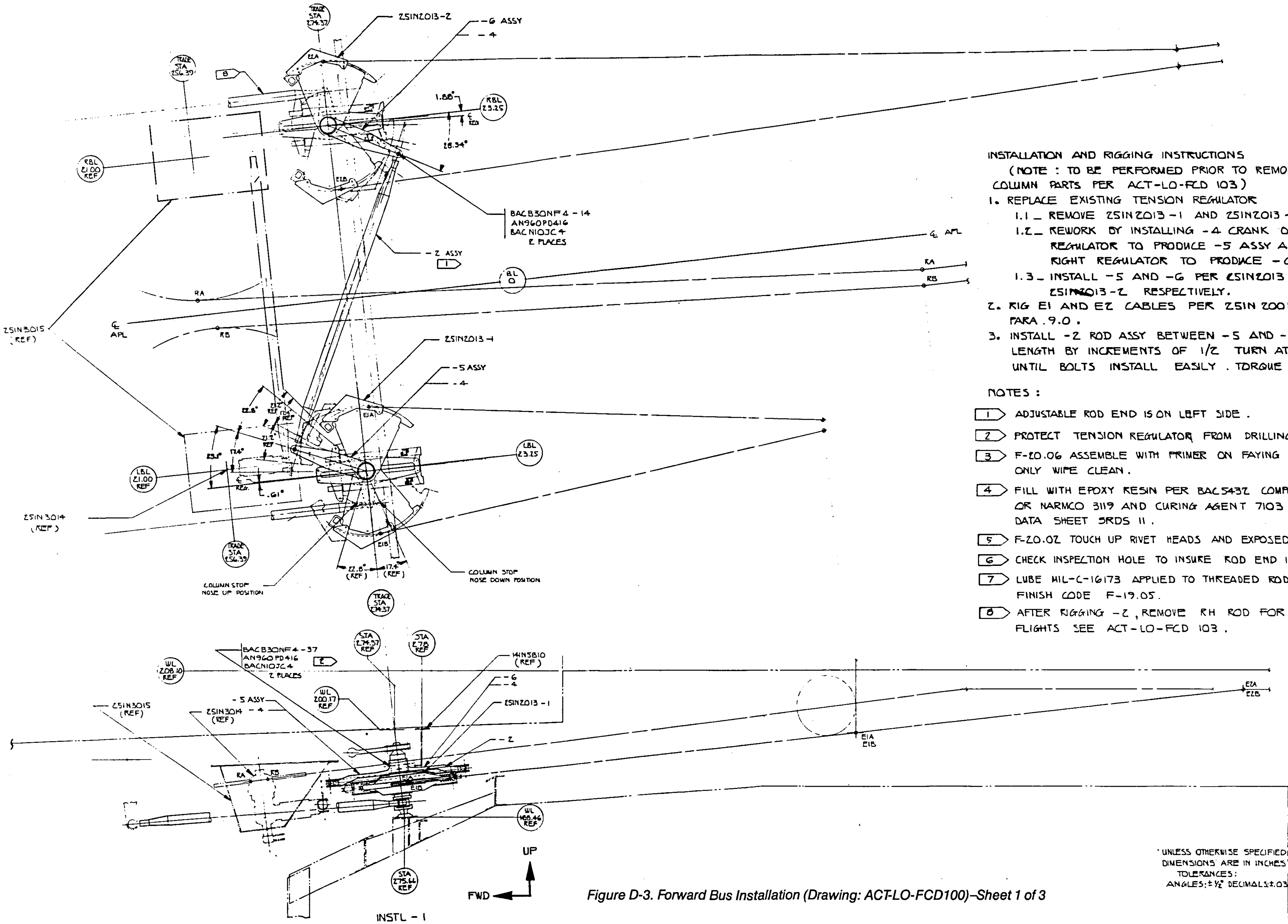


Figure D-2. Test ACT Flight Deck Mechanical Installation



INSTALLATION AND RIGGING INSTRUCTIONS
(NOTE : TO BE PERFORMED PRIOR TO REMOVAL OF COLUMN PARTS PER ACT-LO-FCD 103)

1. REPLACE EXISTING TENSION REGULATOR
   1.1 _ REMOVE 25IN2013-1 AND 25IN2013-2
   1.2 _ REWORK BY INSTALLING -4 CRANK ON LEFT REGULATOR TO PRODUCE -5 ASSY AND ON RIGHT REGULATOR TO PRODUCE -6 ASSY
   1.3 _ INSTALL -5 AND -6 PER 25IN2013-1 AND 25IN2013-2 RESPECTIVELY.
2. RIG E1 AND E2 CABLES PER 25IN 2001, SHT 2, PARA. 9.0.
3. INSTALL -2 ROD ASSY BETWEEN -5 AND -6. ADJUST LENGTH BY INCREMENTS OF 1/2 TURN AT ROD END UNTIL BOLTS INSTALL EASILY. TORQUE LOCK NUT.

NOTES :

1. ADJUSTABLE ROD END IS ON LEFT SIDE.
2. PROTECT TENSION REGULATOR FROM DRILLING CHIPS.
3. F-20.06 ASSEMBLE WITH PRIMER ON FAYING SURFACES ONLY WIPE CLEAN.
4. FILL WITH EPOXY RESIN PER BAC5432 COMPOUND NO.8 OR NARMCO 3119 AND CURING AGENT 7103 PER NARMCO DATA SHEET SRDS 11.
5. F-20.02 TOUCH UP RIVET HEADS AND EXPOSED SURFACE METAL.
6. CHECK INSPECTION HOLE TO INSURE ROD END IS IN SAFETY.
7. LUBE MIL-C-16173 APPLIED TO THREADED ROD ENDS PER FINISH CODE F-19.05.
8. AFTER RIGGING -2, REMOVE RH ROD FOR TEST ACT FLIGHTS SEE ACT-LO-FCD 103.

Figure D-3. Forward Bus Installation (Drawing: ACT-LO-FCD100)–Sheet 1 of 3

Figure D-3. Forward Bus Installation (Drawing: ACT-LO-FCD100)–
Sheet 1 of 3

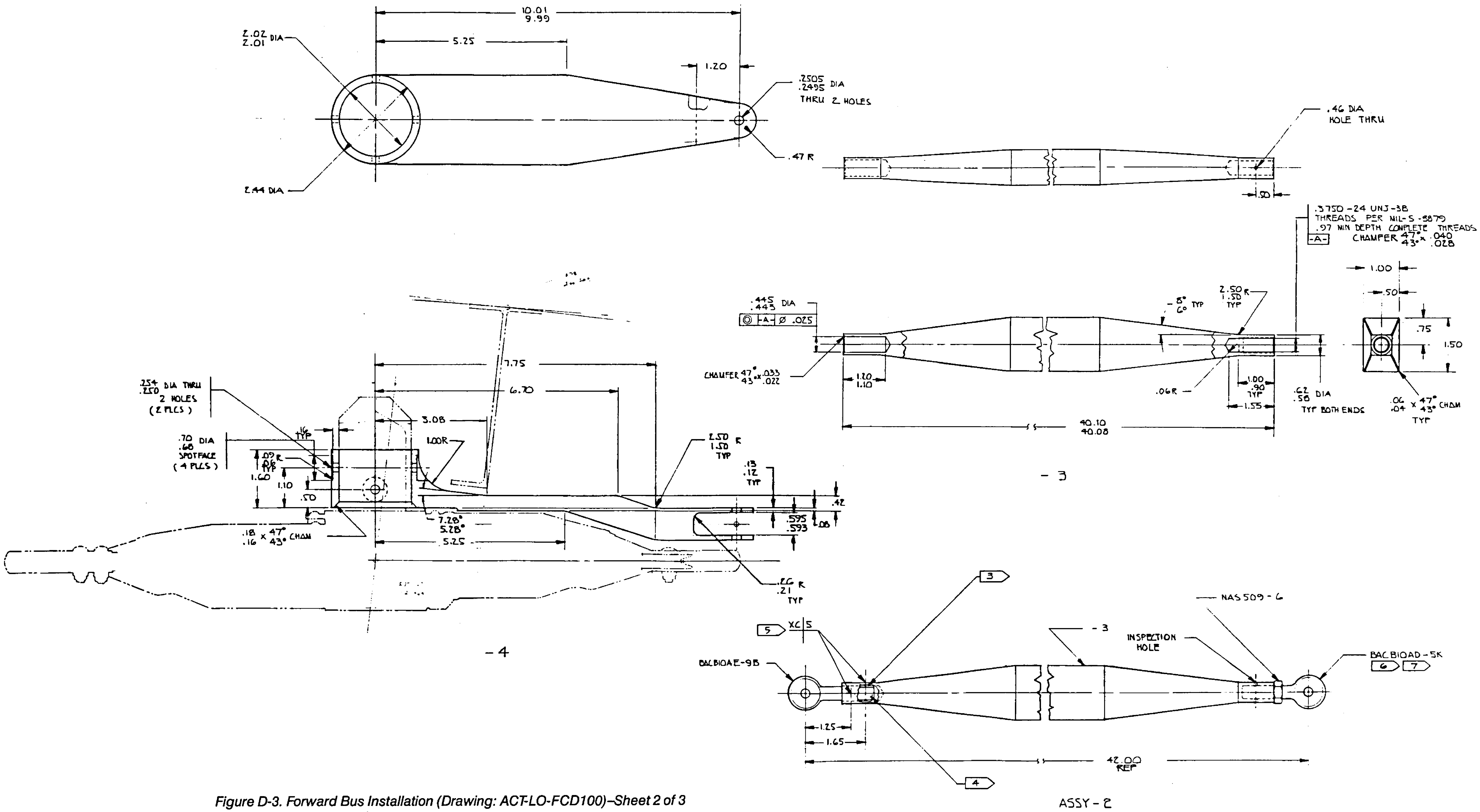


Figure D-3. Forward Bus Installation (Drawing: ACT-LO-FCD100)–Sheet 2 of 3

Figure D-3. Forward Bus Installation (Drawing: ACT-LO-FCD100)–
Sheet 2 of 3

ACT-LO-FCD100 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-6	-5	-2	-1				
					ACT-LO-FCD 100-1	BUS INSTL - TENSION REGULATOR TEST ACT		
			-	1	-2	ROD ASSY		P2, P3
			1		-3	ROD	1.00 X 1.50 X 41.00	M1, F1, P1, P4, P5
	1	1			-4	CRANK	3.00 X 3.00 X 11.00	M1, F1, P1, P4, P5
		-		1	-5	CRANK ASSY		
	-			1	-6	CRANK ASSY		
		1			251N2013-1	TENSION REGULATOR INSTL.		
	1				251N2013-2	TENSION REGULATOR INSTL.		
			1		BACB10AE-9B	ROD END		
			1		BACB10AD-5K	ROD END		
	2	2			BACB30NF-37	BOLT		
				2	BACB30NF-14	BOLT		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		-6	-5	-1				
		2	2	2	BACN10JC4			
		2	2	2	AN960PD416			

M1: 15-5PH STAINLESS STEEL BAR PER AMS5659. HEAT TREAT TO 180 - 200 KSI PER BAC-5619.

P1: MAGNETIC PARTICLE INSPECT PER BAC5424.

P2: COAT FAYING SURFACES WITH WET BMS 5-95, SEALANT PER BAC-5000.

P3: INSTALL FASTENERS WITH F-20.06.

P4: MARK PARTS PER BAC5307.

P5: PENETRANT INSPECT PER BAC5423.

F1: F-16.01.

Figure D-3. Forward Bus Installation (Drawing: ACT-LO-FCD100)–Sheet 3 of 3

The force-displacement program of the feel cam, follower, and springs arrangement was designed to meet the requirements of Figure D-4. The degree of proportionality of the transducer input force is shown in Figure D-5. The signal is virtually linear for about two-thirds of the column travel and falls off at the travel limits due to linkage geometry effects. System gains are such that all elevator requirements for Test ACT, including full travel ground demonstrations, are achievable within the linear range of transducer output signal.

An adjustable hydraulic rotary rate damper is attached to the torque tube system through a crank and link. The damping force is adjustable in the range of 113 to 226 N-m per rad/sec (1000 to 2000 in.-lb per rad/sec). This amounts to 1.8 to 2.6 N/cm (1 to 2 lb/in./sec) at the wheel.

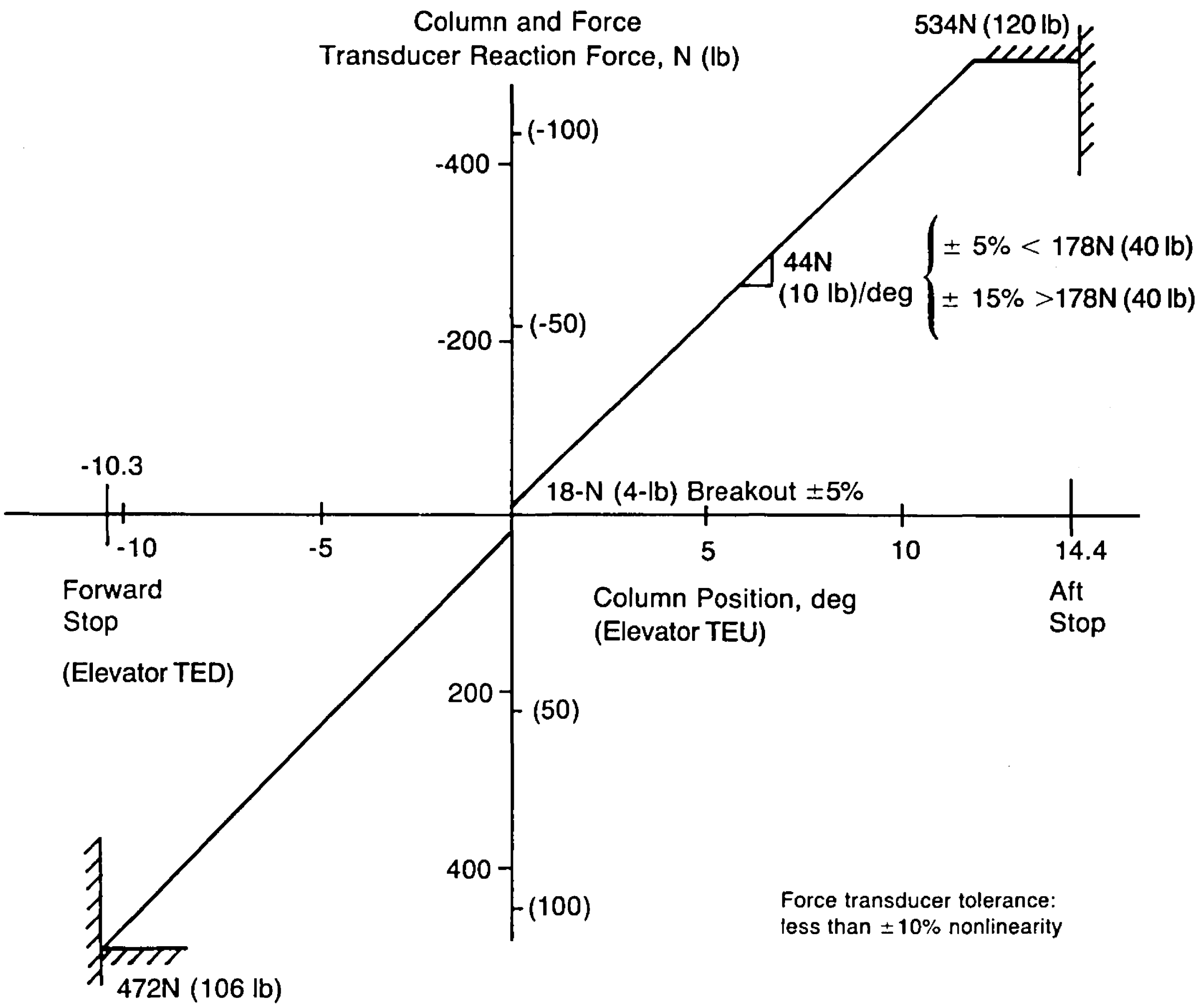


Figure D-4. Control Column Feel System Requirements

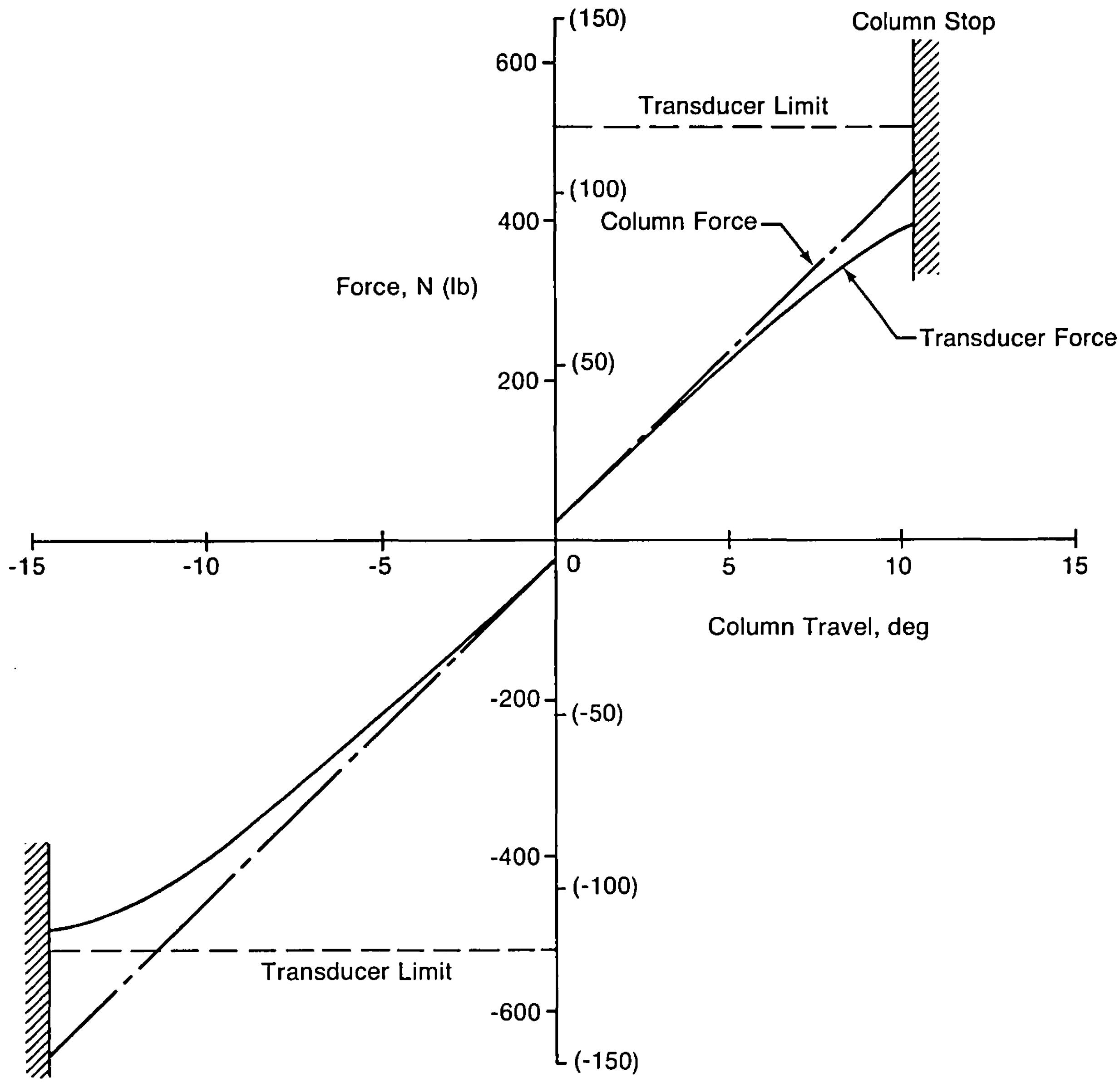
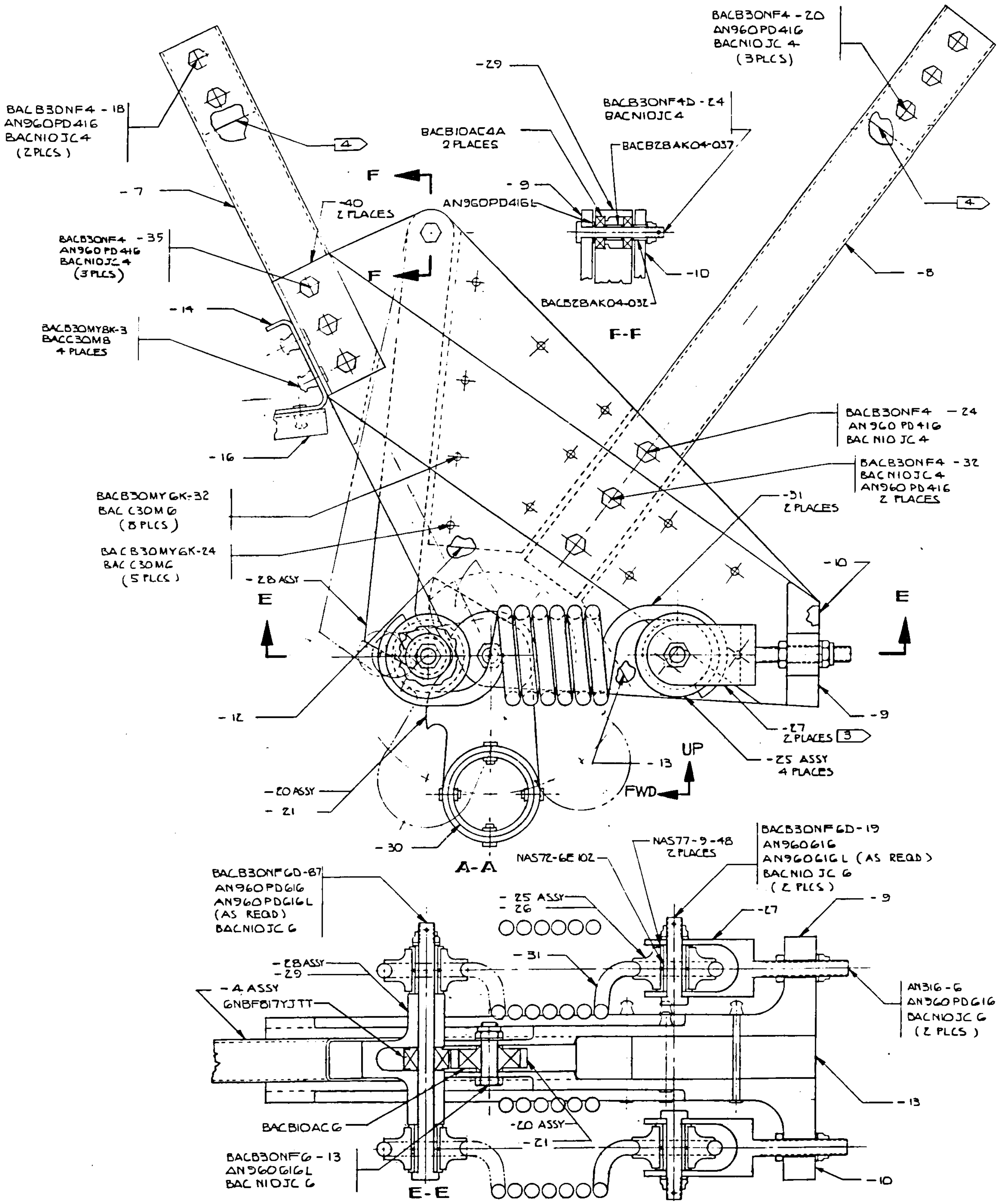


Figure D-5. Force Transducer Characteristics

Column inertial forces pass through the force transducers so column unbalance is a concern. Each column weighs approximately 14 kg (30 lb), with a cg of about 0.25m (10 in.) above the pivot. Balance weight of about 8.6 kg (19 lb) was added on a special support assembly, available from 757 flight testing, to bring the first officer's column unbalance to approximately zero. The left column was also balanced in this way to preclude any inputs or reduction of summing lever detents due to attitudes or accelerations during fly-by-wire (FBW) flight modes. Flight deck installation details are shown in Figure D-6 (9 sheets).



1 FOR TEST ACT FLIGHTS :
REMOVE SPRINGS 25IN1083 -1 AND FOLLOWER ARM 25IN1078-1
FROM COLUMN TORQUE TUBE , SEE DETAIL I .
REMOVE AND STORE 250N2004-104 . RH ROD ASSY ONLY .
INSTALL -35 ROD ASSY .
FOR CONVENTIONAL FLIGHT CONFIGURATIONS :
REMOVE -35 ROD ASSY , INSTALL MECHANISM STOWAGE BOLT
SEE DETAIL I AND REINSTALL ROD ASSY 250N2004-104 (RH.)
REINSTALL SPRINGS 25IN1083 -1 AND FOLLOWER ARM 25IN1078 -1
ON COLUMN TORQUE TUBE , DETAIL I .

2 ADJUST -35 ROD ASSY AS FOLLOWS :
(1) INSTALL RIG PIN IN CAPTAINS CONTROL COLUMN .
(2) CLAMP A STRAIGHT EDGE ACROSS CAPTAINS AND FIRST OFFICERS CONTROL WHEELS . THIS TOOL SHOULD HOLD THE TWO COLUMNS IN LINE .
(3) ADJUST -27 EYEBOLTS TO ENSURE A SMALL PRELOAD ON -20 CAM DETENT (BE SURE STOWAGE BOLT , DETAIL I , IS REMOVED)
(4) ADJUST -35 TO FIT BETWEEN COLUMN CRANK AND -18 CLEVIS .
(5) REMOVE CONTROL COLUMN WHEEL CLAMPS .

3 ADJUST -27 EYE BOLT UNTIL THE FORCE ON THE FIRST OFFICERS CONTROL WHEEL FINGER REF POINT REQUIRED TO BREAK OUT IN THE AFT DIRECTION IS : 4±.25 LB .

4 FILL WITH EPOXY RESIN PER BAC - 5432 COMPOUND NO. 8 OR NARMCO 3119 AND CURING AGENT 7103 PER NARMCO DATA SHEET SRDS 11 .

5 FEMALE SPLINE TO MATCH P/NA-16920-1 .

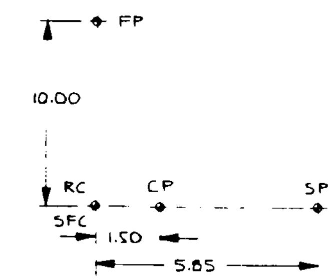
6 INSTALL THIS TRANSDUCER FIRST.

7 INSTALL THIS TRANSDUCER SECOND , ADJUST AS NECESSARY TO INSTALL ROD END BOLTS .

8 REMOVE EXISTING BOLT BACB30NF4 - 3 , REPLACE BACB30NF4-4

9 INSTALL WITH F20.06

BASIC GEOMETRY FOR COMPUTER CAM PROGRAM 'FEEL 7'



SPRING DATA MATL · 17-7PH STEEL

P1 LOAD	: 32 LB
P1 LENGTH	: 5.85 IN
P2 LOAD	: 534.21 LB
P2 LENGTH	: 6.960 IN
TOTAL COILS	: 5.2
OD	: 2.35 IN
WIRE DIA	: .343 IN
STRESS AT P2	: 85,225 PSI
Sb HOOKS	: 163,208 PSI
St HOOKS	: 81,604 PSI
RATE (REF)	: 452.44 LB/IN
HOOK ID	: 1.664 IN

FP - CP	: 10.111
FP - SP	: 11.341
FP - SFC	: 10.000
FP - RC	: 10.000
CP - SP	: 4.35
RC - CP	: 1.50
CP - SFC	: 1.50
ROLLER DIA	: 1.0625
SPRING PRELOAD	64 LB

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 1 of 9

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–
Sheet 1 of 9

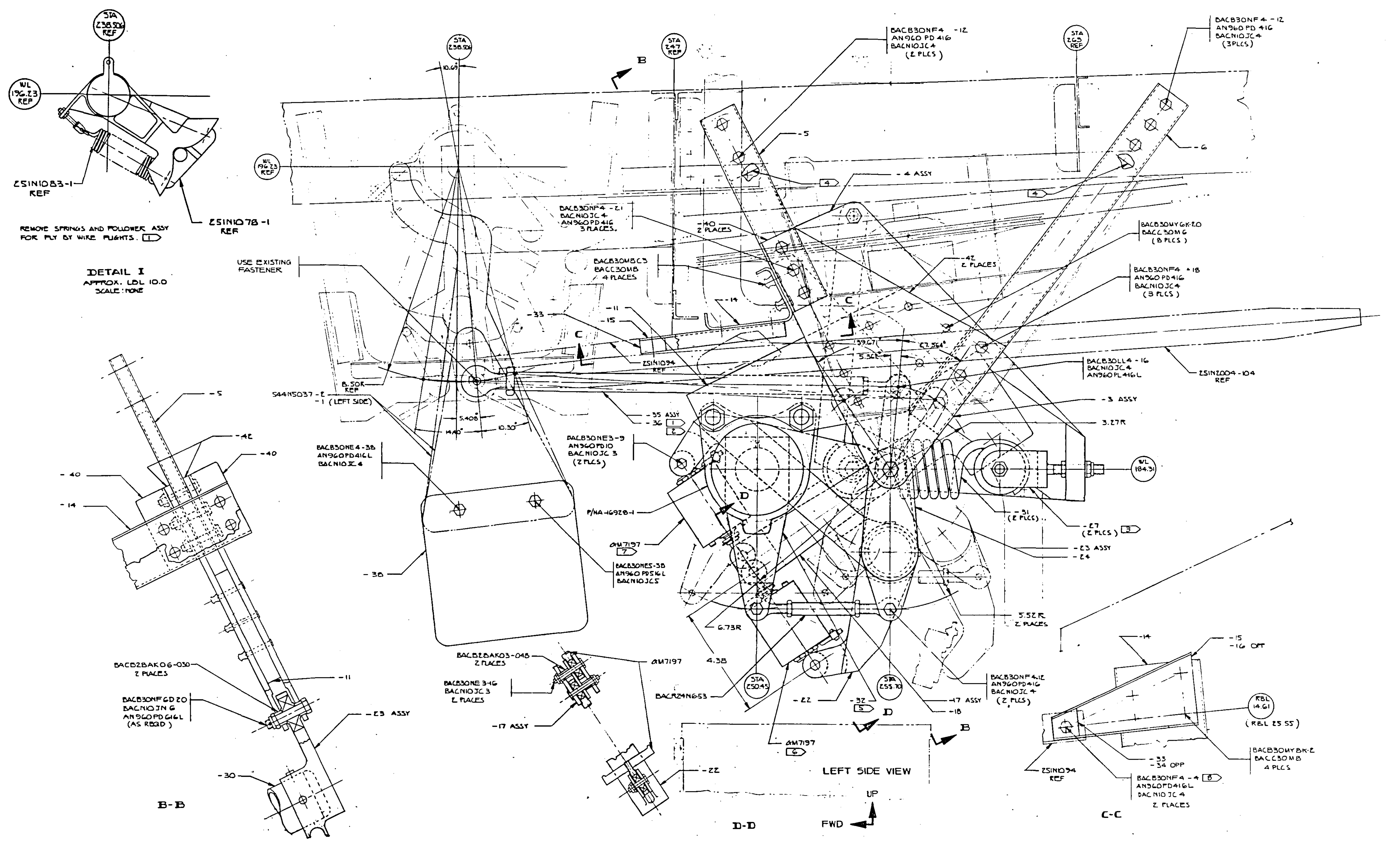


Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 2 of 9

*Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–
Sheet 2 of 9*

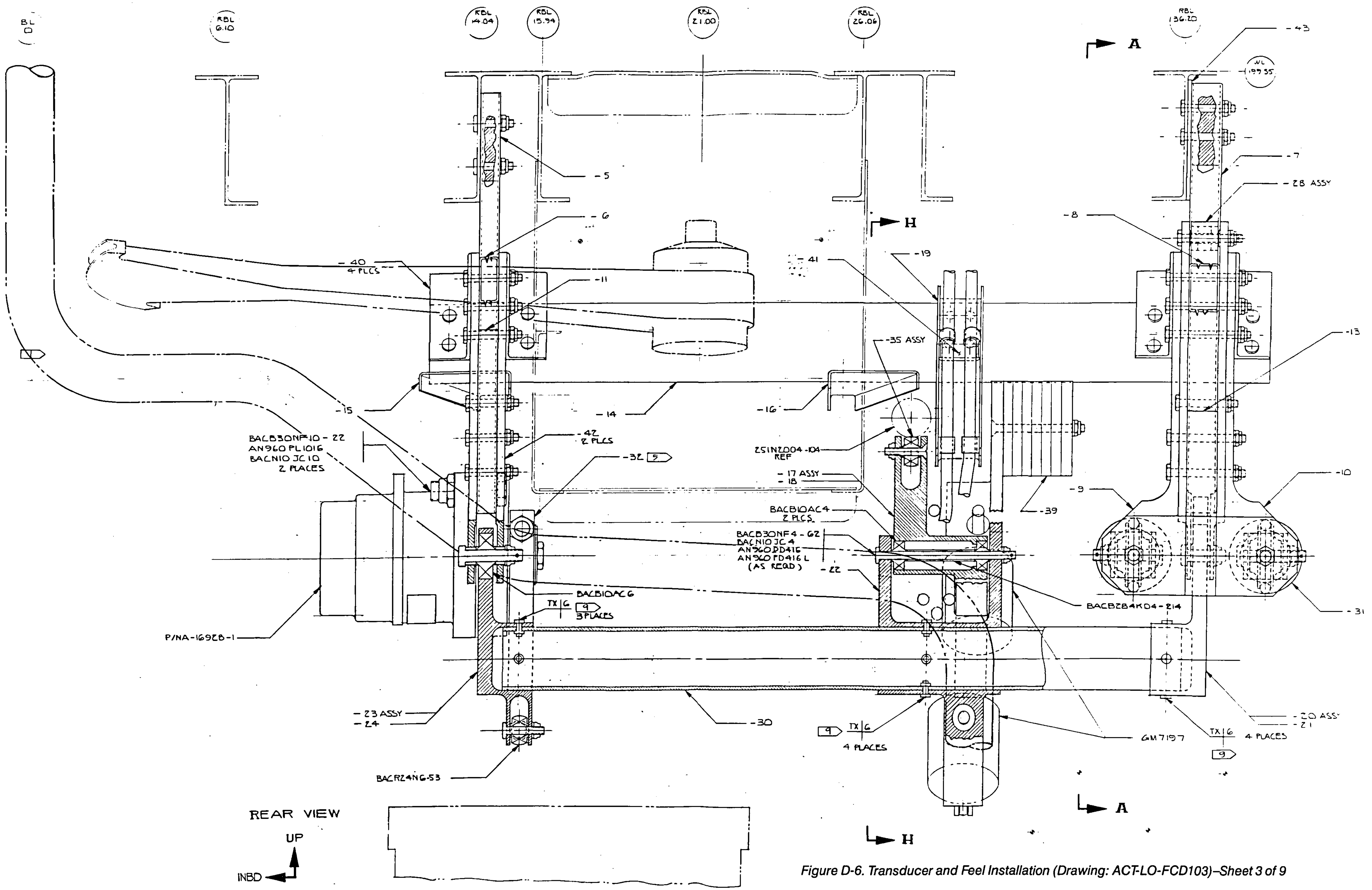


Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 3 of 9

*Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–
Sheet 3 of 9*

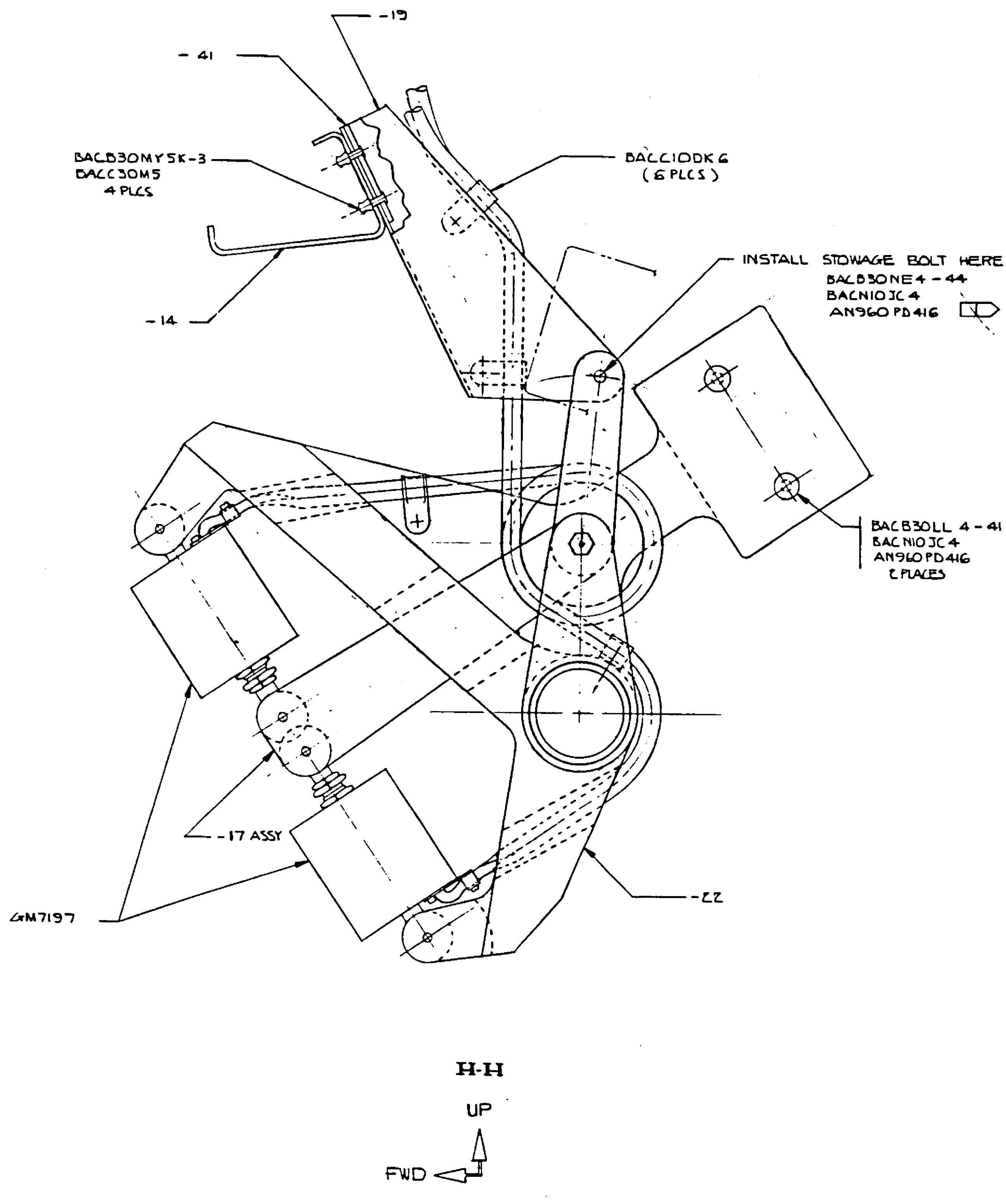


Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 4 of 9

ACT-LO-FCD103 Parts List

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			-4	-3	-2	-1				
						-	ACT-LO-FCD 103-1	ACT TRANSDUCER & FEEL INSTL		
					-	1	-2	ACT TRANSDUCER & FEEL ASSY		
				-	1		-3	TRUSS ASSY - INBD		
			-		1		-4	TRUSS ASSY - OUTBD		
				1			-5	RECT. TUBE	BAC1501 -1145 .625 X 1.5 X 10.0	F2, M1
				1			-6	RECT. TUBE	BAC1501 -1145 .625 X 1.5 X 17.0	F2, M1
			1				-7	RECT. TUBE	BAC1501 -100033 1.00 X 1.50 X 12.00	F2, M1
			1				-8	RECT. TUBE	BAC1501 -100033 1.00 X 1.50 X 17.00	F2, M1
			1				-9	PLATE	10.00 X 16.00 X 3.00	F1, M8, P1

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		-20	-17	-4	-3	-2				
				1			-10	PLATE	10.00 X 16.00 X 3.00	F1, M8, P1
					1		-11	SPACER PLATE	6.00 X 12.00 X .625	F1, M8, P1
				1			-12	SPACER PLATE	6.00 X 9.00 X 1.00	F1, M8, P1
				1			-13	SPACER PLATE	6.00 X 7.00 X 1.00	F1, M8, P1
						1	-14	SUPPORT	8.00 X 28.00 X .071	F1, M2
						1	-15	SUPPORT	6.00 X 7.00 X .071	F1, M3
						1	-16	SUPPORT	6.00 X 7.00 X .071	F1, M3
			-				-17	INPUT CRANK ASSY		
			1				-18	INPUT CRANK	5.00 X 12.00 X 4.00	F1, M6, P1
						1	-19	WIRE SUPPORT BRACKET	8.00 X 9.00 X .071	F1, M3
		-				1	-20	FEEL CAM ASSY		
		1					-21	FEEL CAM	3.00 X 4.00 X 8.00	F7, M9, P2

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 5 of 9

ACT-LO-FCD103 Parts List

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-4	-28	-25	-23	-2	-1				
					1		-22	LOAD CRANK	14.00 X 14.00 X 5.00	F1, M6
				-	1		-23	CRANK ASSY		
				1			-24	CRANK	3.00 X 8.00 X 2.75	F1, M6
			-		4		-25	HOOK SUPPORT ASSY		
			1				-26	HOOK SUPPORT	2.50 DIA. X 1.50	F1, M7
					2		-27	EYE BOLT	1.75 X 1.75 X 6.00	F5, M9, P2
	1	-					-28	FOLLOWER ARM ASSY		
		1					-29	FOLLOWER ARM	3.00 X 12.00 X 4.00	F7, M9, P2
					1		-30	SHAFT	2.000 D X .08 WALL X 22.00	F2, M1
					2		-31	SPRING		F6, M10
						1	-32	CRANK	3.00 X 8.00 X 1.00	F3, M6, P1
						1	-33	FILLER	1.50 X 1.50 X .625	F1, M8

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-3	-37	-35	-28	-2	-1				
							-34	FILLER	1.5 X 1.5 X .625	F1, M8
			-			1	-35	ROD ASSY		
			1				-36	ROD	.75 DIA. X 16.00	F1, M4
		-				2	-37	COLUMN BALANCE ASSY		
		10					-38	COL. BALANCE WT.	6.0 X 6.0 X .30	F5, M9
					7		-39	XDCR BALANCE WT.	3.0 X 4.0 X .30	F5, M9
					4		-40	ANGLE	AND10134-1406 1.25 X 1.50 X 3.50	F1, M5
					1		-41	FILLER	BAC1511-3701 1.28 X .125 X 2.00	F1, M5, P1
	2						-42	PLATE	9.0 X 13.0 X .25	F1, M8
						1	-43	SPACER PLATE	3.0 X 6.0 X .125	F1, M2

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 6 of 9

ACT-LO-FCD103 Parts List

QTY -28	-25	-23	-20	-17	-2	-1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
						1	BACR24N653	ROD ASSY		
						1	544N5037-1	COL. BALANCE INSTL		
						1	-2	COL. BALANCE INSTL		
					2		GM7197	TRANSDUCER		VC1
						1	P/NA-16928-1	HYD. DAMPER		VC2
1							6NBF817YJTT	TRACK ROLLER		VC3
				2			BACB10AC4	BEARING		P3
2							BACB20AC4A	BEARING		P3
		1	1				BACB10AC6	BEARING		P3
	1						NAS72-6E102	BUSHING		P4
	2						NAS77-9-48	BUSHING		P4
				1			BACB28AK03-048	BUSHING		P4

QTY -35	-28	-17	-4	-3	-2	-1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			1				BACB28AK04-032	BUSHING		P4
	1						BACB28AK04-037	BUSHING		P4
				2			BACB28AK06-030	BUSHING		P4
		1					BACB28AK04-214	BUSHING		P4
					2		BACN10JC3	NUT		
						12	BACN10JC4	NUT		
	1				4		BACN10JC6	NUT		
					2		BACN10JC10	NUT		
1							BACB10AD-8	ROD END		P7
1							BACB10AD-5	ROD END		P7
1							AN316-10	NUT		
1					2		AN316-6	NUT		

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 7 of 9

ACT-LO-FCD103 Parts List

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			-4	-3	-2	-1				
					2		BACB30LL4-41	BOLT		
						1	BACB30LL4-16	BOLT		
						1	BACB30NE4-44	BOLT		
			5				BACB30MY6K-24	BOLT		
			5				BACB30MY6K-32	BOLT		
					8		BACB30MY8K-2	BOLT		
					8		BACB30MY8K-3	BOLT		
					4		BACB30MY5K-2	BOLT		
				8			BACB30MY6K-20	BOLT		
			3				BACB30NF4-24	BOLT		
				3		2	BACB30NF4-18	BOLT		
				3			BACB30NF4-21	BOLT		

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-28	-17	-4	-3	-2	-1				
					2		BACB30NE3-9	BOLT		
					2		BACB30NE3-16	BOLT		
						5	BACB30NF4-12	BOLT		
						3	BACB30NF4-20	BOLT		
			1				BACB30NF4D-24	BOLT		
			2				BACB30NF4-32	BOLT		
			3				BACB30NF4-35	BOLT		
					1		BACB30NF4D-62	BOLT		
					1		BACB30NF6-13	BOLT		
					1		BACB30NF6D-20	BOLT		
					2		BACB30NF6D-19	BOLT		
	1						BACB30NF6D-87	BOLT		

Figure D-6. Transducer and Feel Installation (Drawing: ACT-LO-FCD103)–Sheet 8 of 9

ACT-LO-FCD103 Part List and Notes

QTY							PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			-4	-3	-2	-1				
					2		BACB30NF4-4	BOLT		
						1	BACB30LL4-16	BOLT		
							AN960PD416	WASHER		
							AN960PD416L	WASHER		
							AN960PD616	WASHER		
							AN960PD616L	WASHER		
							AN960PD1016	WASHER		
							BACC30M5	COLLAR		
							BACC30M6	COLLAR		
							BACC30M8	COLLAR		

M1: 2024-T3 AL. ALLOY TUBE PER BMS 7-196

M2: 2024-T3 AL. ALLOY SHEET PER QQ-A-250/4

M3: 2024-0 AL. ALLOY SHEET PER QQ-A-250/4. HEAT TREAT TO T42 PER BAC5602.

M4: 2024-T42 AL. ALLOY ROD PER QQ-A-225/6

M5: 2024-T3511 AL. ALLOY EXTRUSION PER QQ-A-200/3.

M6: 7075-T7351 AL. ALLOY PLATE PER QQ-A-250/12. ULTRASONIC INSPECT PER BAC5439, CLASS B.

M7: 7075-T7351 AL. ALLOY ROD PER QQ-A-225/9.

M8: 2024-T351 AL. ALLOY PLATE PER QQ-A-250/4. ULTRASONIC INSPECT PER BAC5439, CLASS B.

M9: 15-5 PH, BAR PER BMS7-240 TYP I, HEAT TREAT TO 180-200 KSI PER BAC5619.

M10: 17-7 PH PER MIL-S-25043 HEAT TREAT TO 180-200 KSI.

P1: PENETRANT INSPECT PER BAC-5423, CLASS B.

P2: MAGNETIC PARTICLE INSPECT PER BAC5424.

P3: INSTALL BEARING PER BAC-5435.

P4: INSTALL BUSHING PER BAC-5435.

P5: ROLLER SWAGE PER BAC-5435.

P6: FILL WITH EPOXY RESIN PER BAC-5432 COMPOUND NO.8 OR NARMCO 3119 AND CURING AGENT 7103 PER NARMCO DATA SHEET SRDS II.

P7: COAT FAYING SURFACES WITH WET BMS 5-95 SEALANT PER BAC-5000.

P8: INSTALL FASTENERS WITH F-20.06.

F1: F-17.04 + F-20.02

F2: F-18.07

F3: F-17.04 + F-20.03

F4: F-16.03

F5: F-20.02

F6: F-16.03

F7: F-16.01

VC1: KAVLICO CORPORATION
CHATSWORTH, CA. 91311

VC2: HYDRAULICS HOUDAILLE INC. (SEE ATTACHMENT)
BUFFALO, NY. 14211

VC3: TORRINGTON
TORRINGTON, CONN.

Figure D-6. Transducer and Feei Installation (Drawing: ACT-LO-FCD103)–Sheet 9 of 9

D.2 EMPENNAGE INSTALLATION

Figure D-7 shows the unmodified 757 test airplane elevator linkage arrangement. Dual cable quadrants are linked to a dual feel unit, which in turn is grounded through an override and cam to the stabilizer. Stabilizer motions provide a programmed elevator input to augment nose-down trim effectiveness. Triple autopilot servos also attach to the cable quadrants, and right and left output linkages transmit quadrant motions through a summing mechanism near the side of the body to elevator surface actuator linkages on the stabilizer spar. Triple-parallel hydraulic surface actuators are positioned by conventional mechanical closed-loop controls. These actuators are the balanced, moving piston type. A body-mounted valve is positioned by a summing lever attached to the piston for feedback and, through a collapsible link (pogo), to a spar-mounted bellcrank for input. The three bellcranks are linked together in parallel and driven by a pushrod system from the aft quadrant.

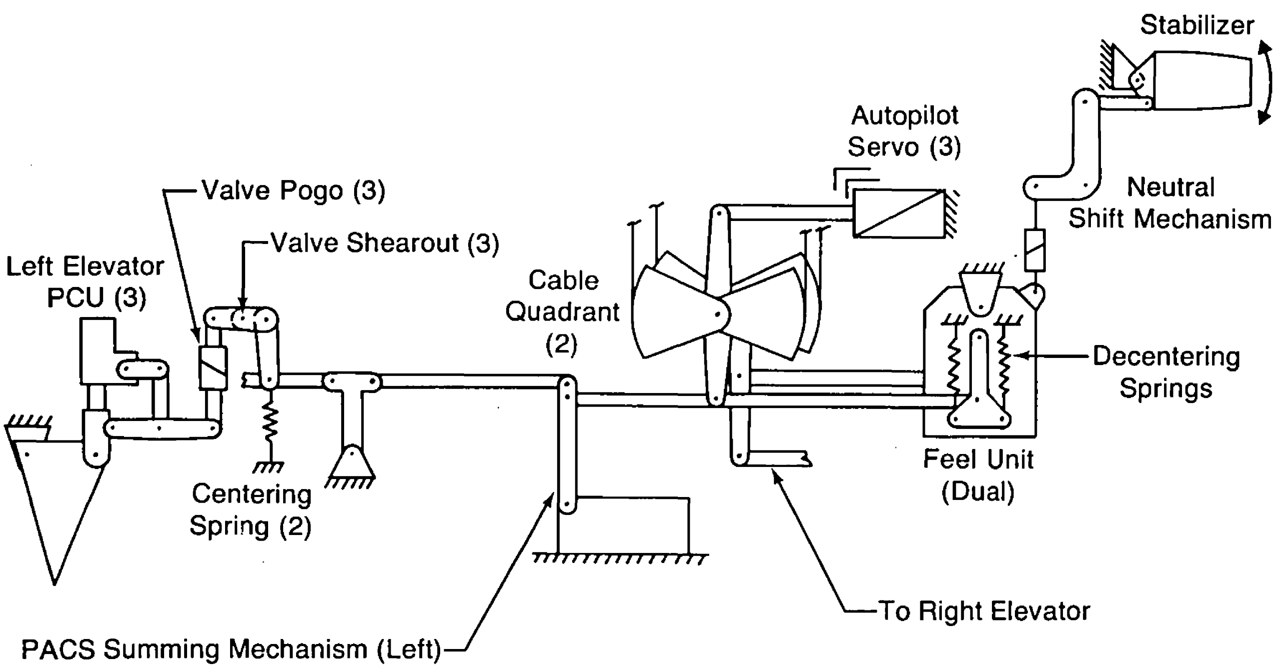


Figure D-7. 757 Empennage Installations–Linkage Schematic

Centering springs are fitted between the bellcrank and ground at the two outboard actuator positions. These springs maintain the elevator in a near-faired position in the event of a disconnect of an inboard component.

The input pogo links are provided to protect against a jammed valve. In such a case, the pogo associated with the offending valve collapses with an additional column load of about 53N (12 lb). This allows the two healthy actuators to bring full load to bear on the immobilized one, which bypasses at its relief valve setting of about 24.1×10^3 Pa (3500 lb/in.^{2}). Normal single system pressure is 19.3×10^3 to 20.7×10^3 Pa (2800 to 3000 lb/in.^{2}). Thus, continued surface control is available with reduced hinge moment capability. In the event of a frozen or corroded pogo, the pilot can bypass a jammed valve by shearing out the special fused joint in the associated bellcrank with a one-time load of about 178N (40 lb) at the column.

The Pitch Augmentation Control System (PACS) summing mechanisms near the side of body were established early in the 757 design to meet an anticipated need for augmentation during stall approaches. Flight testing proved the augmentation unnecessary, but the mechanisms remain deactivated in early airplanes, including the test airplane. These sites are used to mount a similar summing mechanism for introduction of the Test ACT FBW servo mechanical commands into the elevator actuator input linkage.

The pitch feel unit features decentering springs that neutralize the effect on pilot feel schedules of the two centering springs in each elevator surface linkage. This arrangement is retained in the Test ACT Configuration to ensure the same feel characteristics during mechanical-mode flight.

The Test ACT empennage linkage arrangement is shown in Figure D-8. Changes to the 757 cable system input and surface actuator installations are:

- Reduction from three to one modified autopilot servo
- Grounding of the neutral shift input
- Addition of a solid bus link between right and left quadrants
- Addition of small bias springs (not shown) in the surface actuator valves

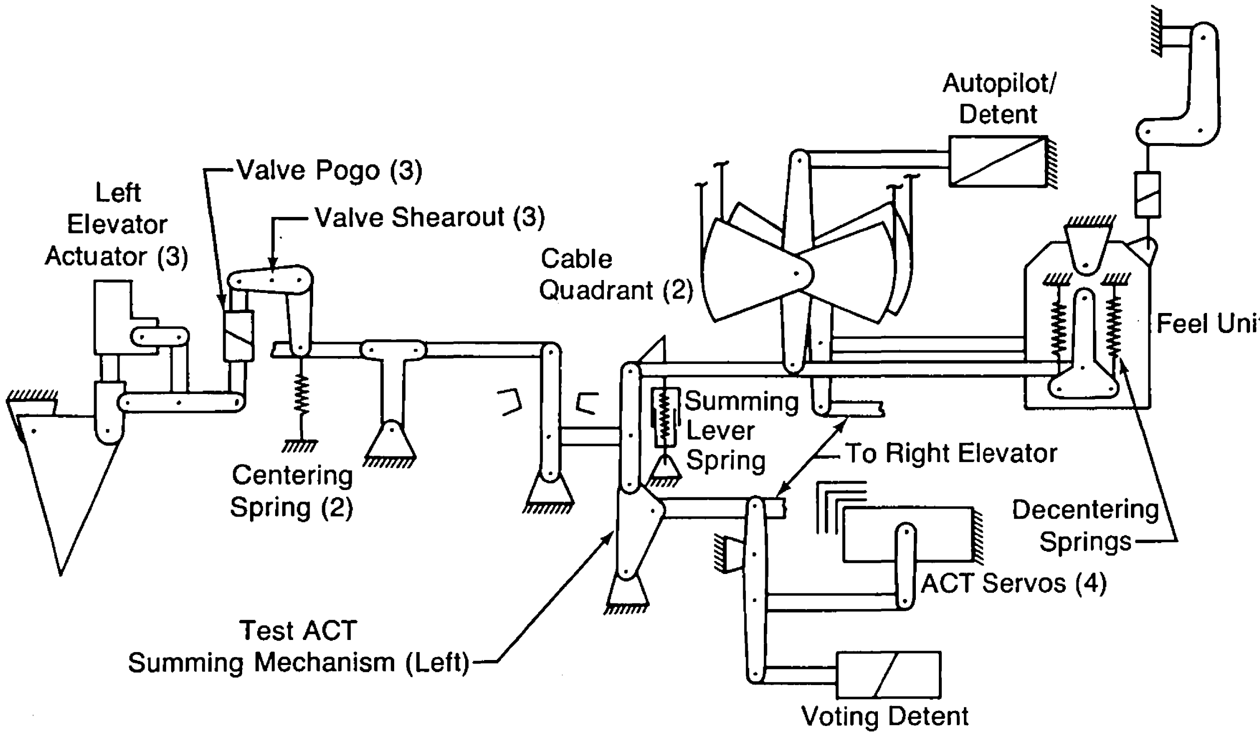


Figure D-8. Test ACT Empennage Installations–Linkage Schematic

Fly-by-wire and augmentation inputs are provided by four added Test ACT servos, force summed on a single shaft to which a centering mechanism is attached in order to provide a fifth force vote. Servo shaft motions are delivered by pushrods to a right and left Test ACT summing mechanism where electrical and mechanical commands to the elevator actuators are combined. The summing link ratio has been chosen to preserve the full elevator authority of the mechanical control system while allowing nearly full authority in the Test ACT flight mode. Figure D-9 illustrates the electric control input-output characteristics of the summing mechanism for two ratios. The chosen ratio provides a reasonably linear and symmetrical elevator output close to the desired travels of +20 deg, -30 deg.

In the Test ACT summing mechanism the summing lever, combined with a parallel motion idler crank, has the same length and airplane location as the equivalent member on the current PACS summing mechanism. Thus, with the Test ACT servos in the detent (Test ACT "off") position, the summing mechanism reproduces current 757 gearing between the aft quadrant and the elevator actuator input linkage.

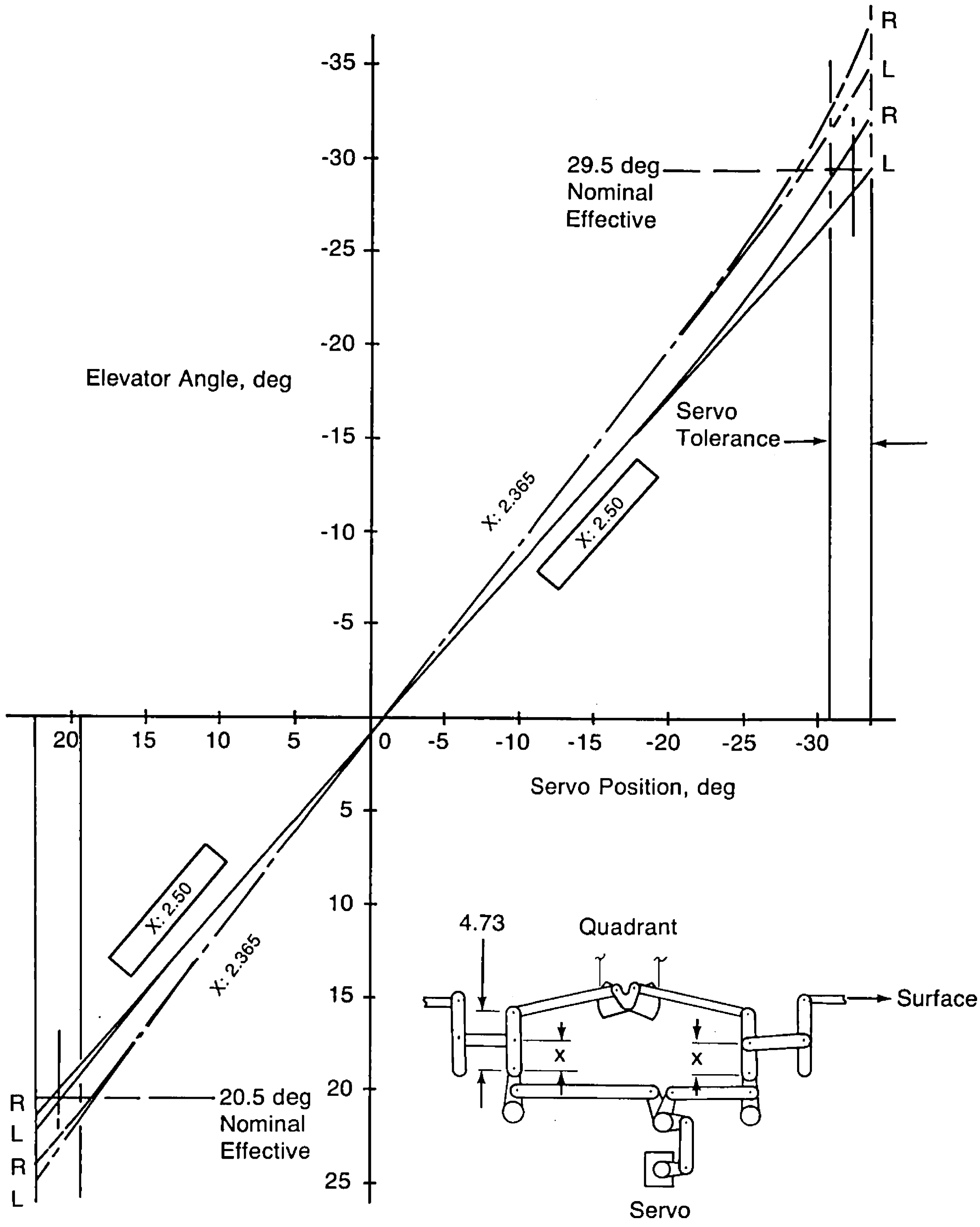


Figure D-9. Empennage Installations–Summing Ratio

A major requirement in the design of the Test ACT servo linkage was the assured availability of adequate elevator control in either input mode and under both normal and failure conditions. The important parameter in this regard is the detent level of the reactive end of the summing lever. This defines what load can be transmitted to the elevator surface linkage by one control mode before the summing lever begins to "back out" against the detent provided by the other control system. Also of interest is the authority remaining when the surface load exceeds the detent setting and when a complete disconnect of the summing lever has taken place.

After consideration of several multichannel active locking designs, a passive summing lever spring arrangement was chosen that effectively neutralizes the normal operating surface input load encountered during the Test ACT inputs. When the mechanical mode is in use, the spring imparts no load since its line of action is maintained along the summing lever centerline. A functional model of the mechanism is shown in Figure D-10. Figure D-11 shows the reactive load for zero backout required by electric-mode control for various operational conditions, including a jammed surface valve in which the valve input pogo is compressed. The loads are shown in terms of an equivalent control column reaction force. Also illustrated is the neutralizing effect of the summing lever springs.

The detent levels available for reaction of Test ACT operational loads are shown in Figure D-12 superimposed on the load curves of Figure D-11. The total reaction is made up of two load components: that from the feel unit and system friction, totaling about 22N (5 lb), and that from the specially modified autopilot servo, about 58N (13 lb), expressed as equivalent column force. The combined 80-N (18-lb) detent will cover normal and valve-jammed Test ACT control requirements, as shown.

Autopilot servo modifications that provide a detent function are described later in this section. The function is activated through switching procedures during initiation of the Test ACT flight mode.

Since the autopilot detent control is single channel, the feel unit, which is a dual hydromechanical device, must provide a reaction level sufficient for normal Test ACT operation during the period of time following an autopilot servo disconnect. As shown in Figure D-13, a small backout occurs in the nose-down direction with autopilot servo loss; however, elevator travels are sufficient for any requirement encountered in Test ACT flight.

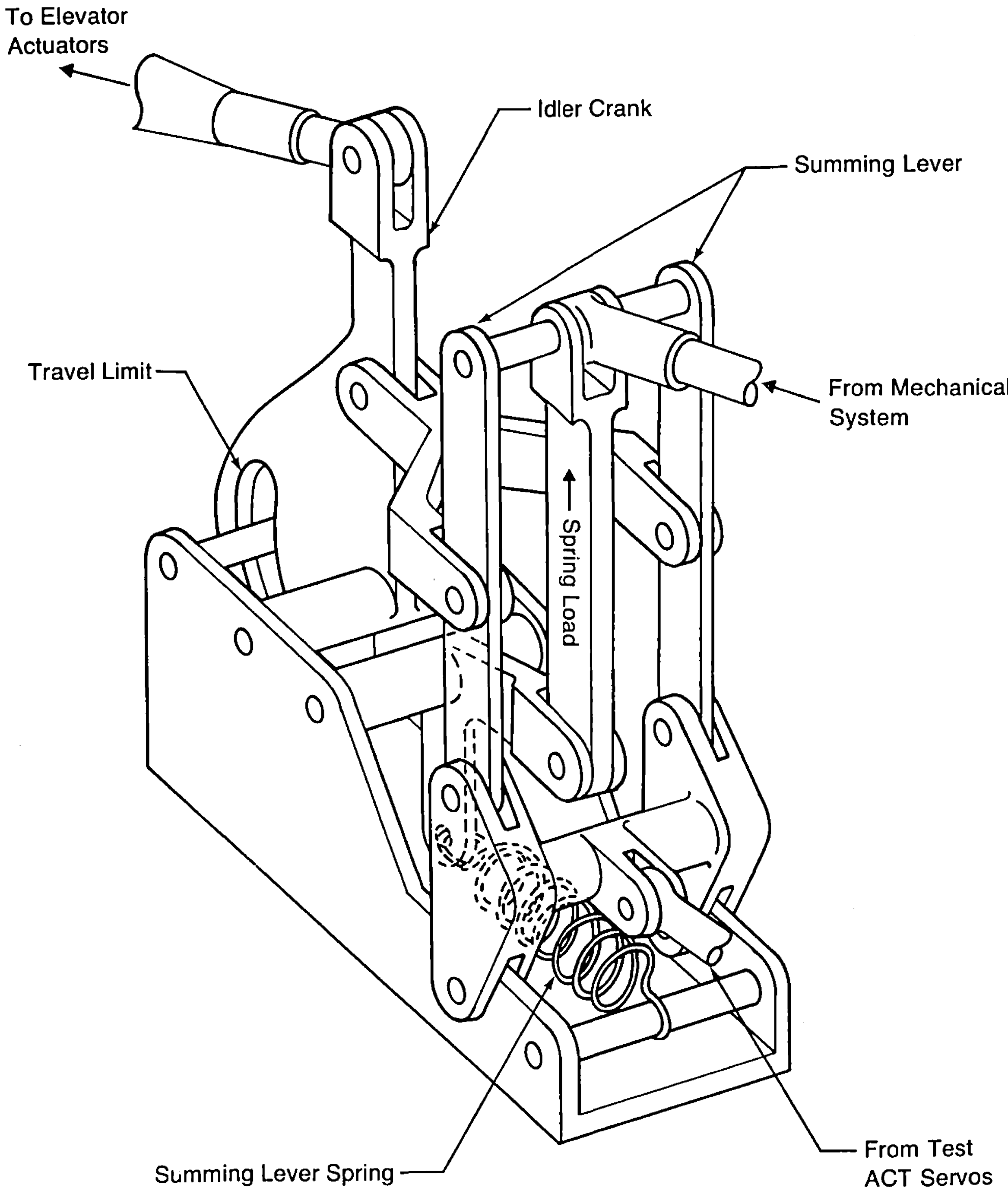


Figure D-10. Test ACT Summing Mechanism–Functional Model

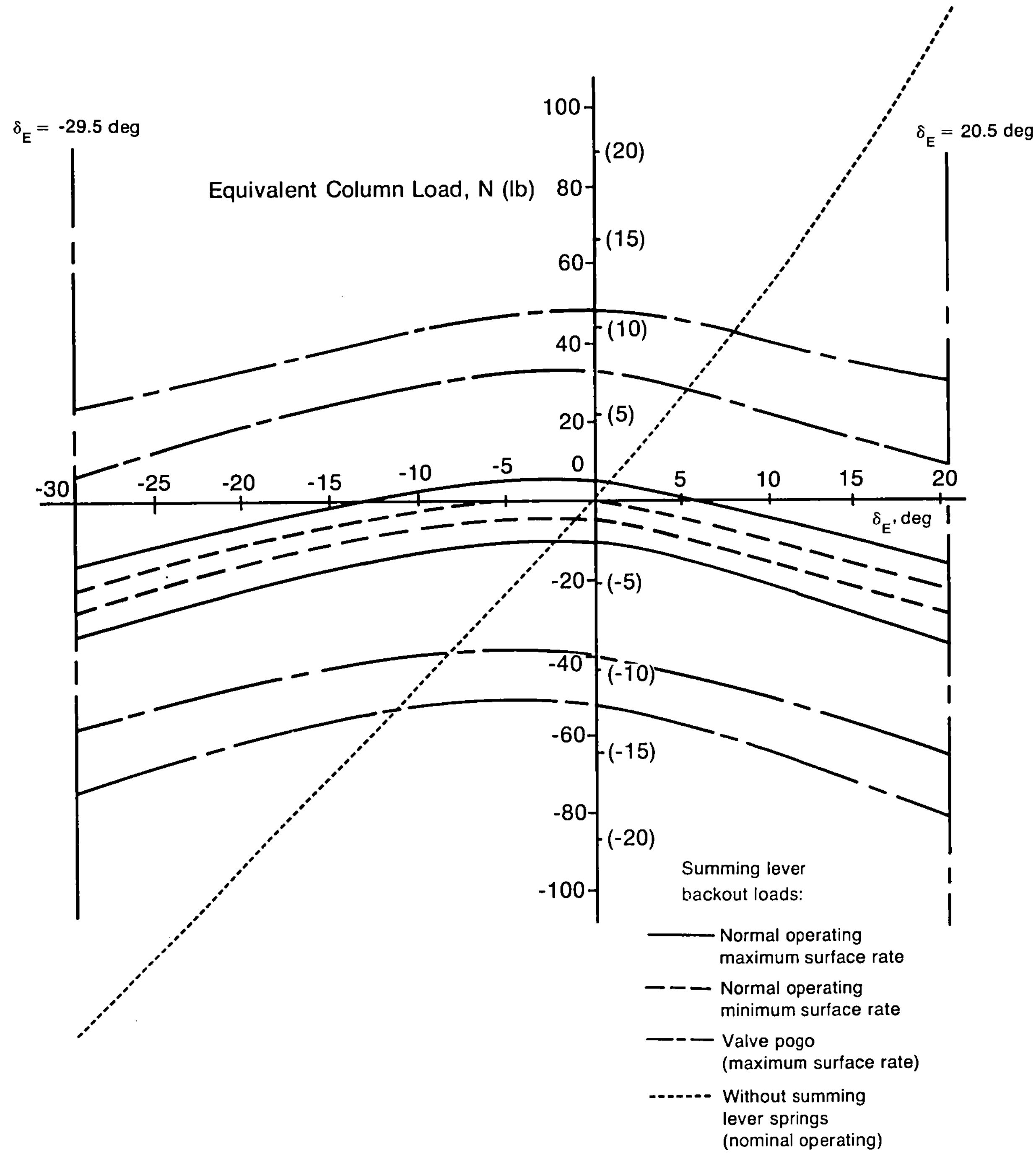


Figure D-11. Summing Lever Loads–Electric Control

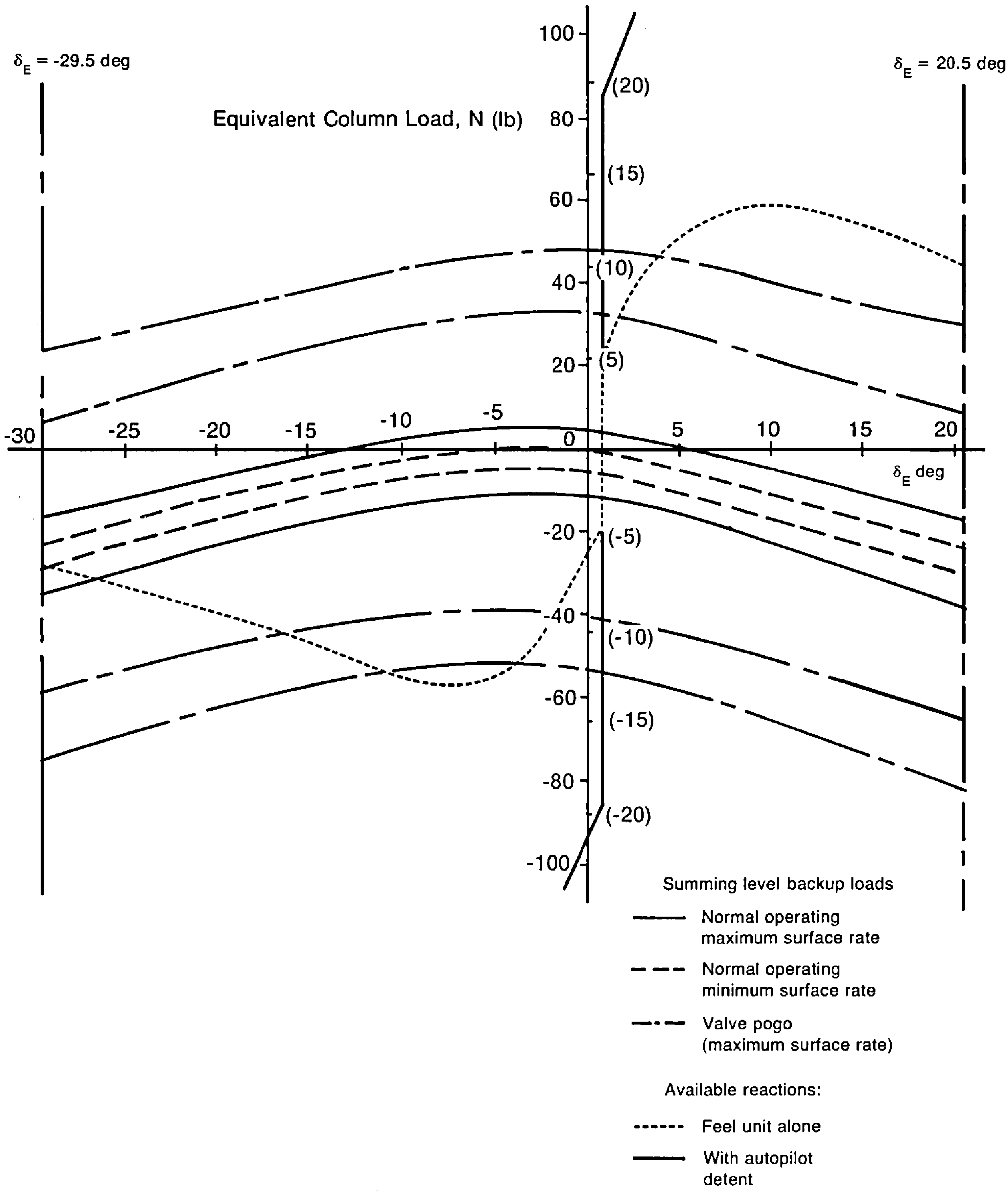


Figure D-12. Summing Lever Reaction–Electric Control

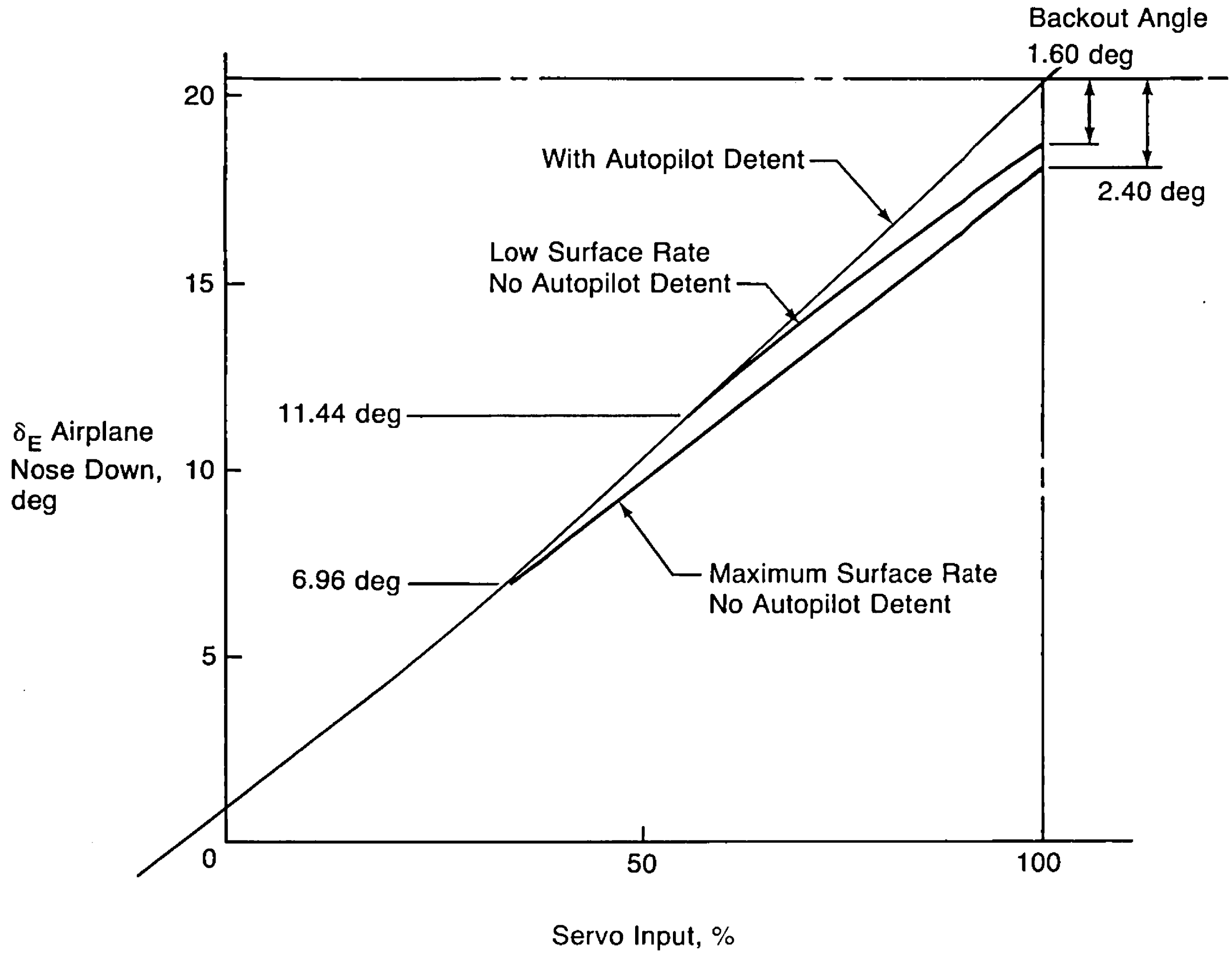


Figure D-13. Electric Control Authority–Effect of Autopilot Detent

Because of the neutralizing effect of the summing lever spring, very little elevator authority is lost following a complete disconnect of one summing mechanism from the cable quadrant. As shown in Figure D-14, the affected elevator is shifted a small amount in the nose-up direction. The null transient would be equivalent to a step input of about 0.5 deg of elevator deflection (δ_E). Of course, mechanical-mode control authority would be reduced by half after such a break.

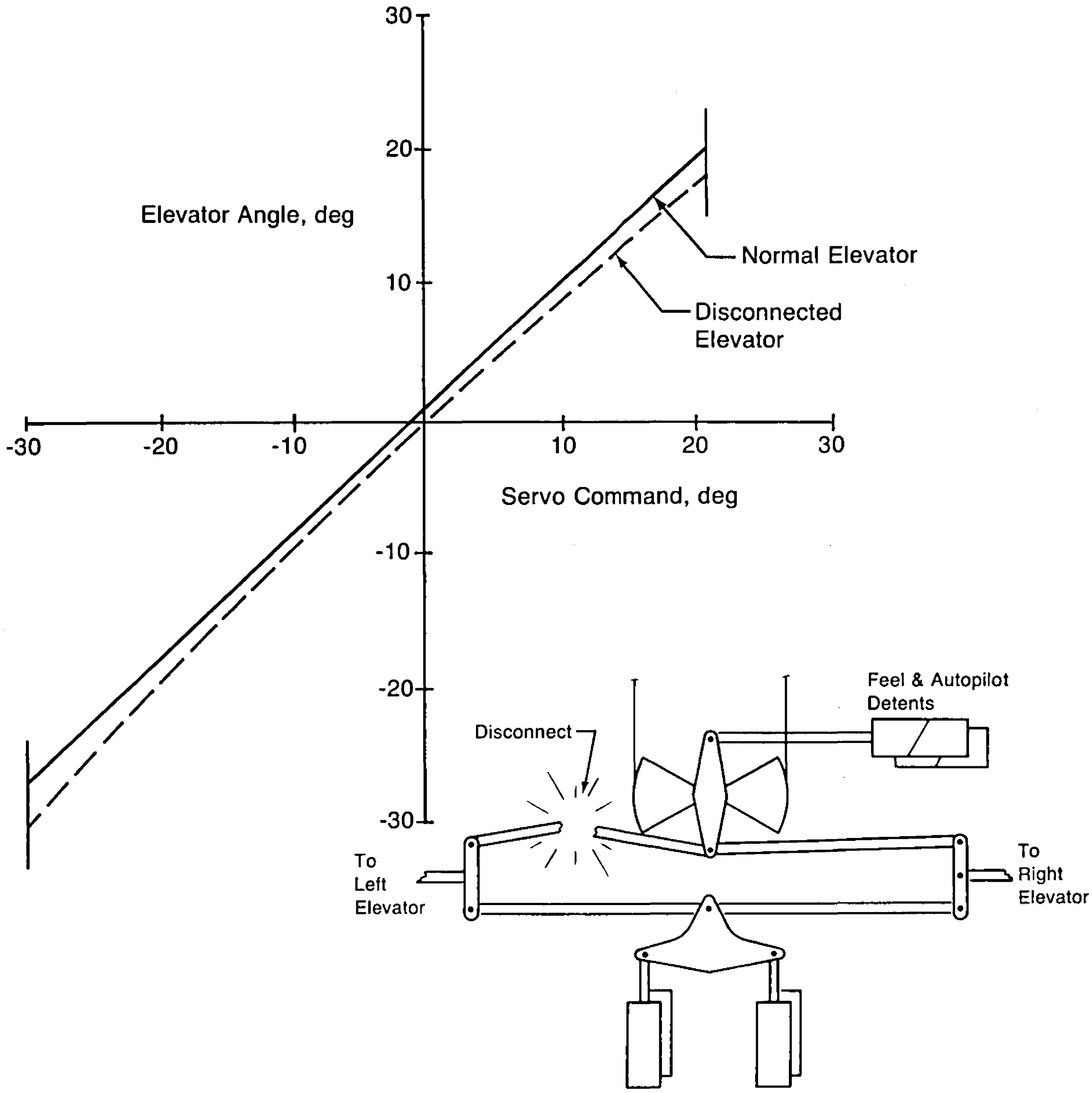


Figure D-14. Electric Control Authority–Effect of Quadrant Rod Disconnect

Figure D-15 shows the operational load reactions when pitch control is in the mechanical (Test ACT "off") mode. In addition to normal operating and valve pogo loads, the requirement for a bellcrank shearout is shown. Because of its low probability of occurrence, shearout capability is a strict requirement only in the mechanical control mode. At all except very low speeds, the electric control mode is capable of effecting a shearout.

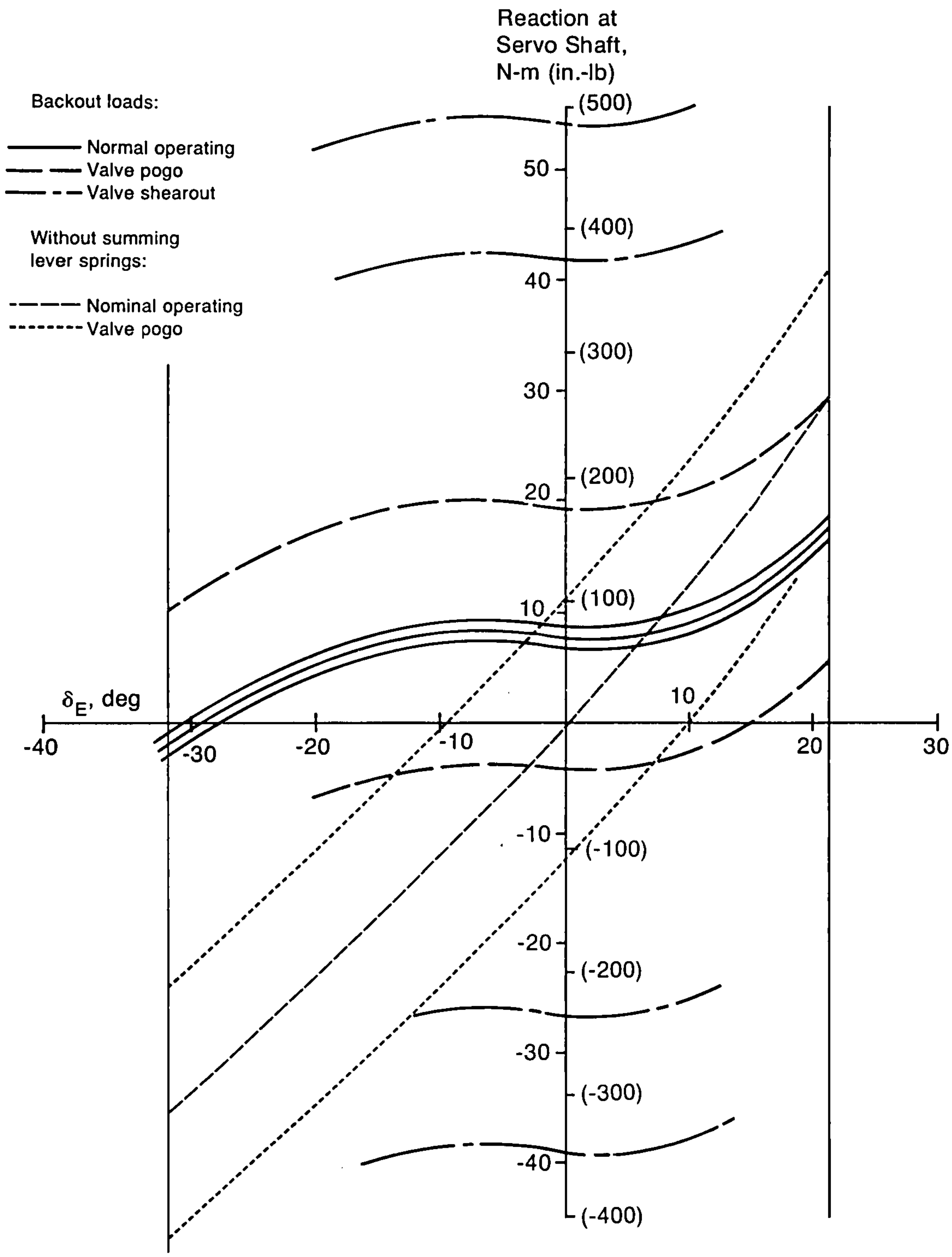


Figure D-15. Summing Lever Loads–Mechanical Control

The reaction requirements shown in Figure D-15 are given in terms of equivalent torques at the Test ACT servo shaft. Again, as in the electric control mode, the summing lever springs produce a significant reduction in reaction loads, illustrated by the curves of loads with springs omitted.

Loads for normal operation and valve pogo compression during mechanical control are reacted by the servo centering unit detent (also called the "voting detent"). To meet the force voting requirement, this detent should be about 17 N-m (150 in.-lb), nose up and nose down. This value is satisfactory for reaction of nose-up loads; however, nose-down load reaction must be about 28 N-m (250 in.-lb). Accordingly, an asymmetrical detent has been incorporated in the servo centering unit torque program (fig. D-16).

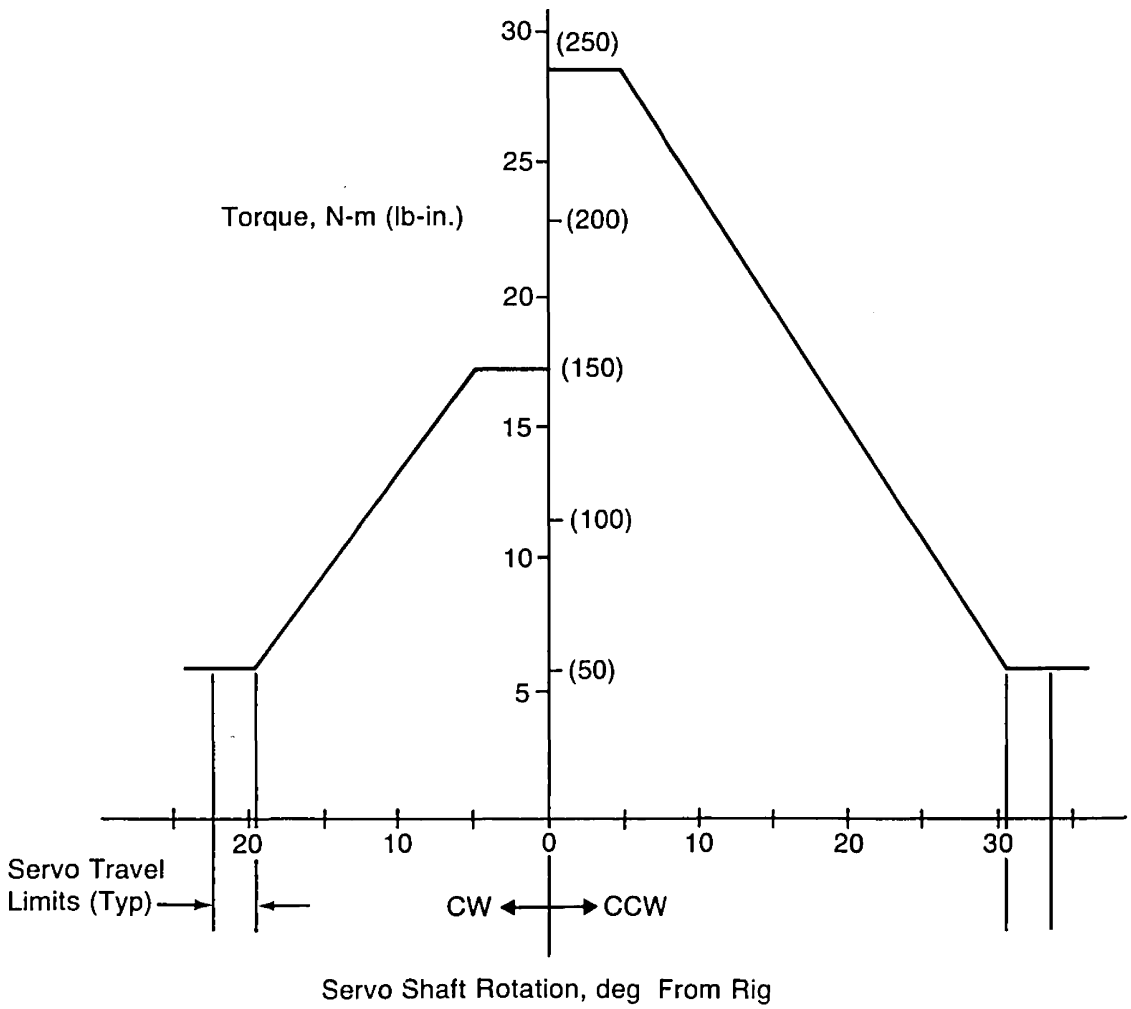


Figure D-16. Servo Centering Unit Torque

As shown in Figure D-17, the voting detent covers all normal and valve pogo loads with zero backout. The out-of-detent gradient, provided by the summing lever spring and servo centering unit torques, reacts shearout loads with a backout of less than 50% servo travel. Once a shearout is effected, the system returns to center, and the control loads return to the normal range.

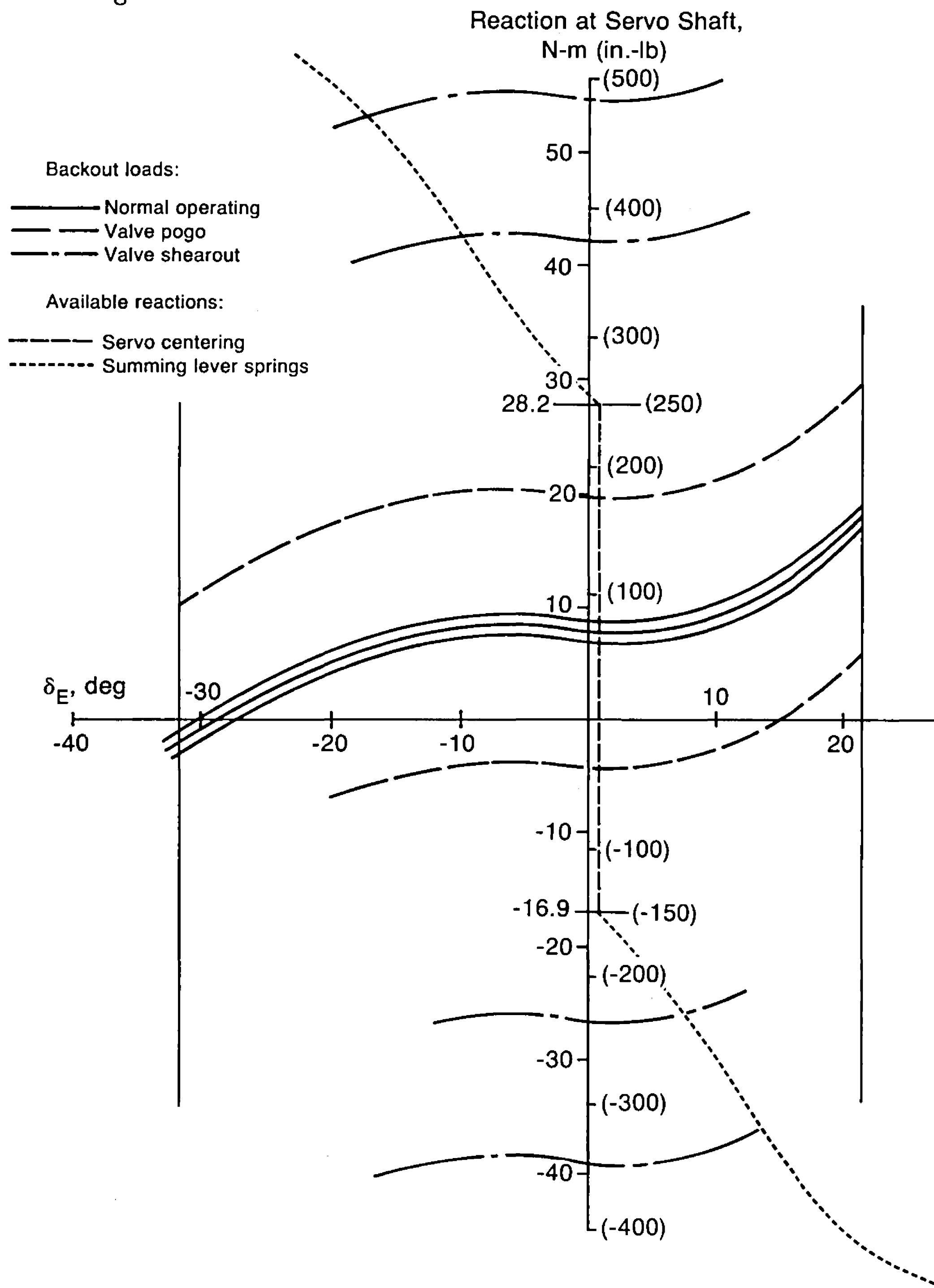


Figure D-17. Summing Lever Reactions–Mechanical Control

Figure D-18 shows the effect of a servo rod disconnect on mechanical control. The result is a net elevator authority of approximately 80%, following a null shift equivalent of about 1.0-deg δ_E.

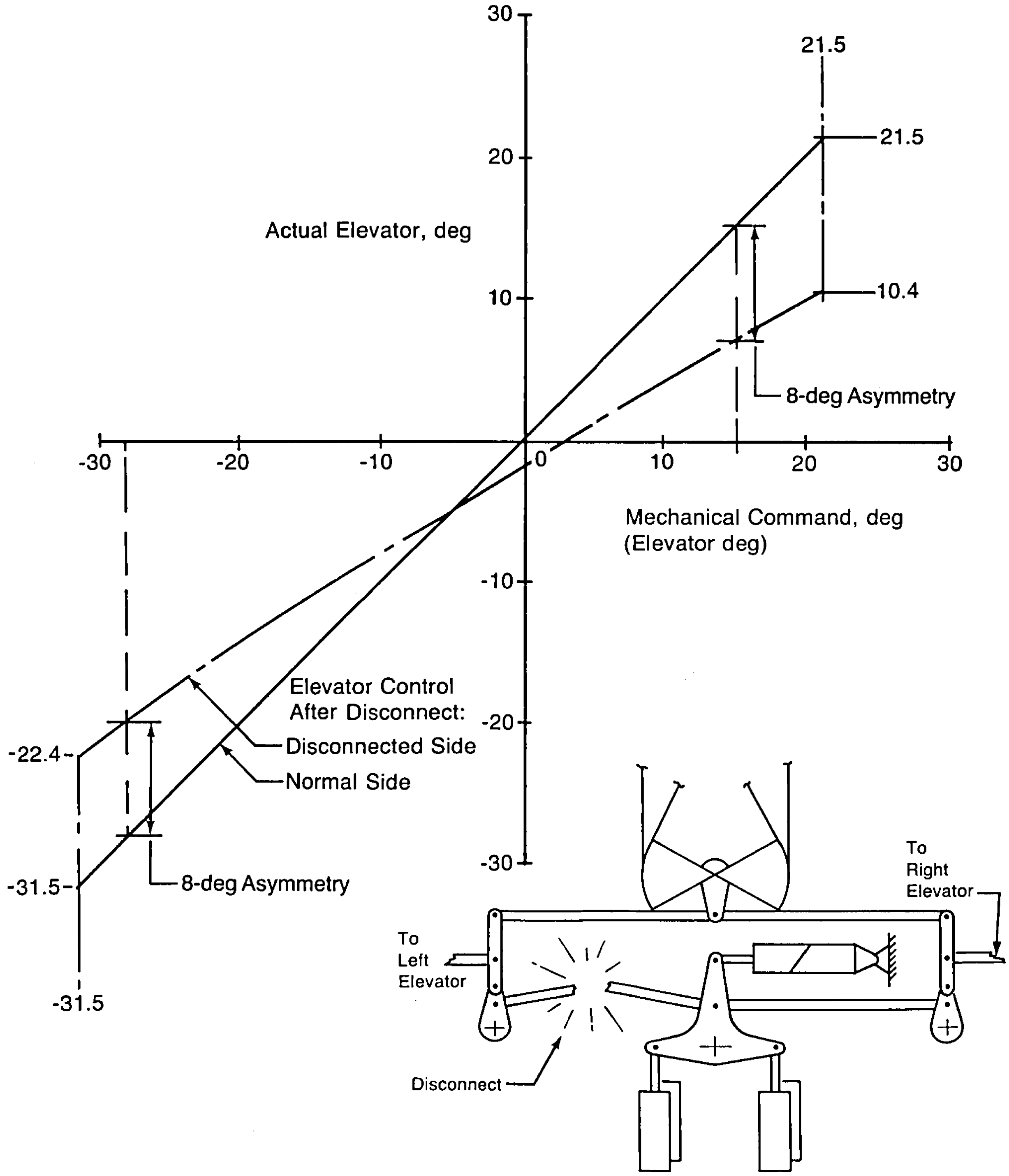


Figure D-18. Mechanical Control Authority

Figure D-19 shows Test ACT servo torque requirements for normal control. All elevator travel requirements, including ground demonstration of full travel, are within the combined minimum torque capability of two servos. Single servos would cam out at somewhat greater than half travel. However, operation is applicable only to single-servo ground tests, which will require travels of less than ± 5-deg δ_E. See Figures D-20 (10 sheets) and D-21 (16 sheets) for empennage installation details.

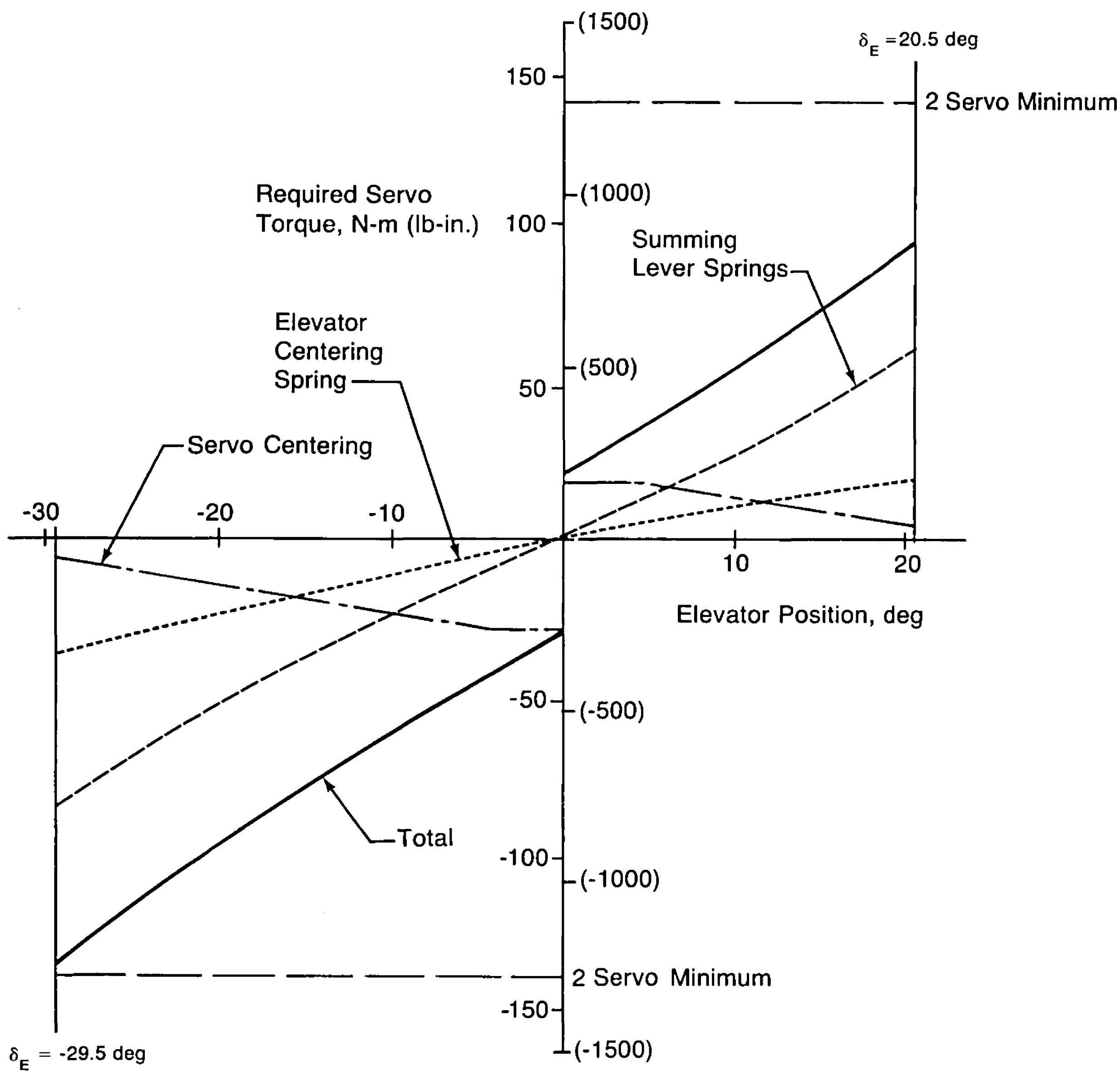


Figure D-19. Test ACT Servo Torque Requirements

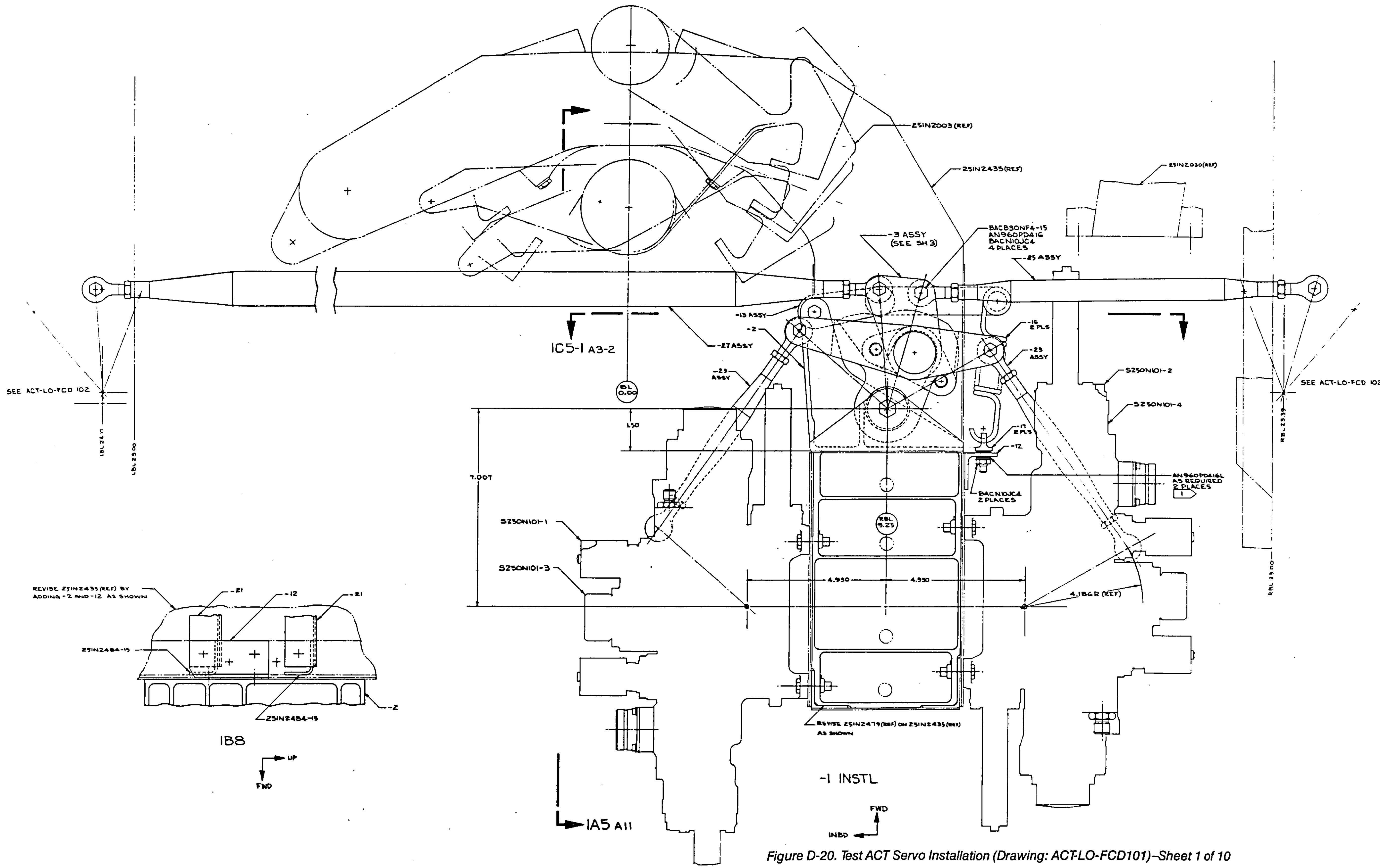


Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 1 of 10

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–
Sheet 1 of 10

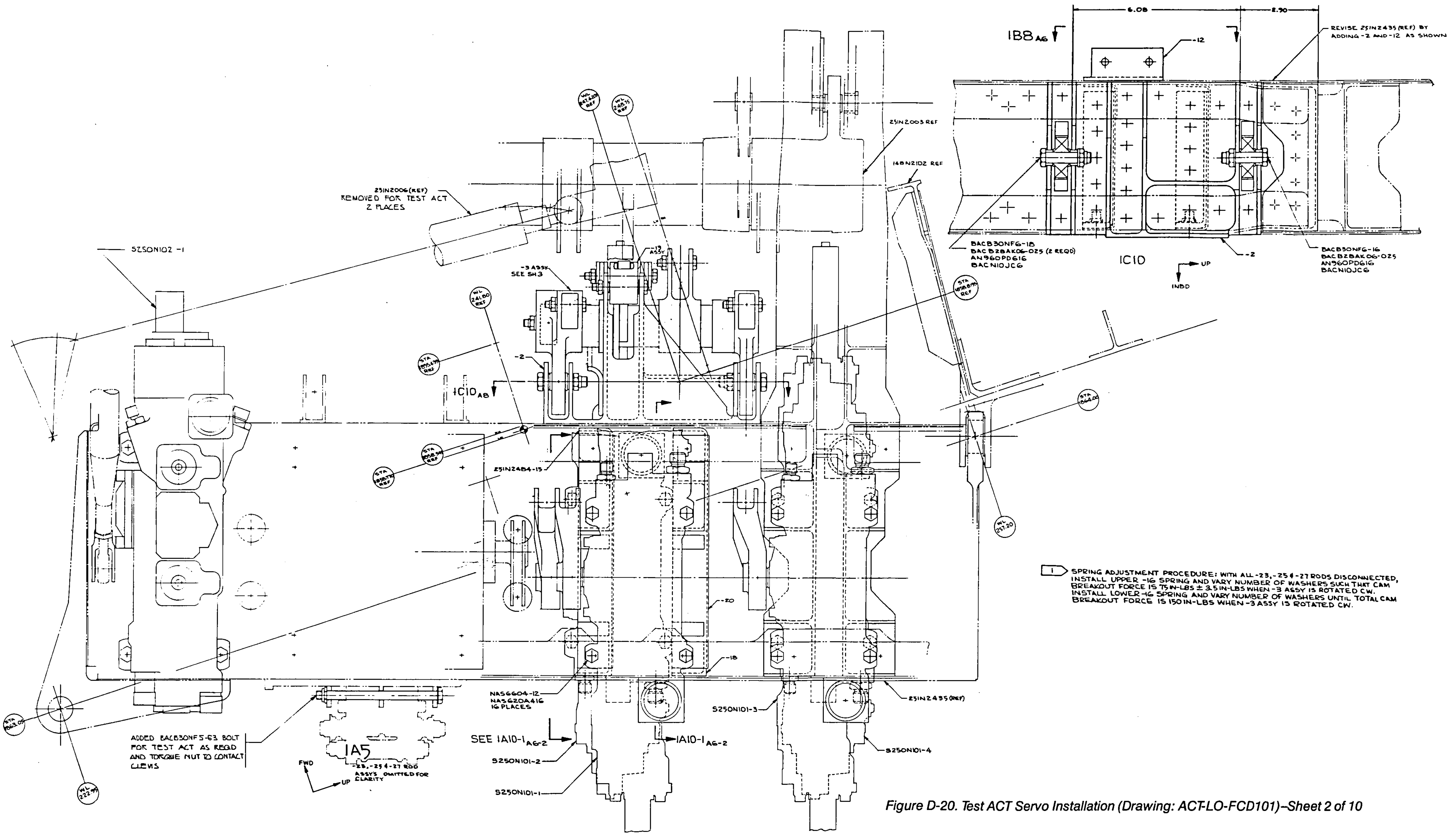


Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 2 of 10

*Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–
Sheet 2 of 10*

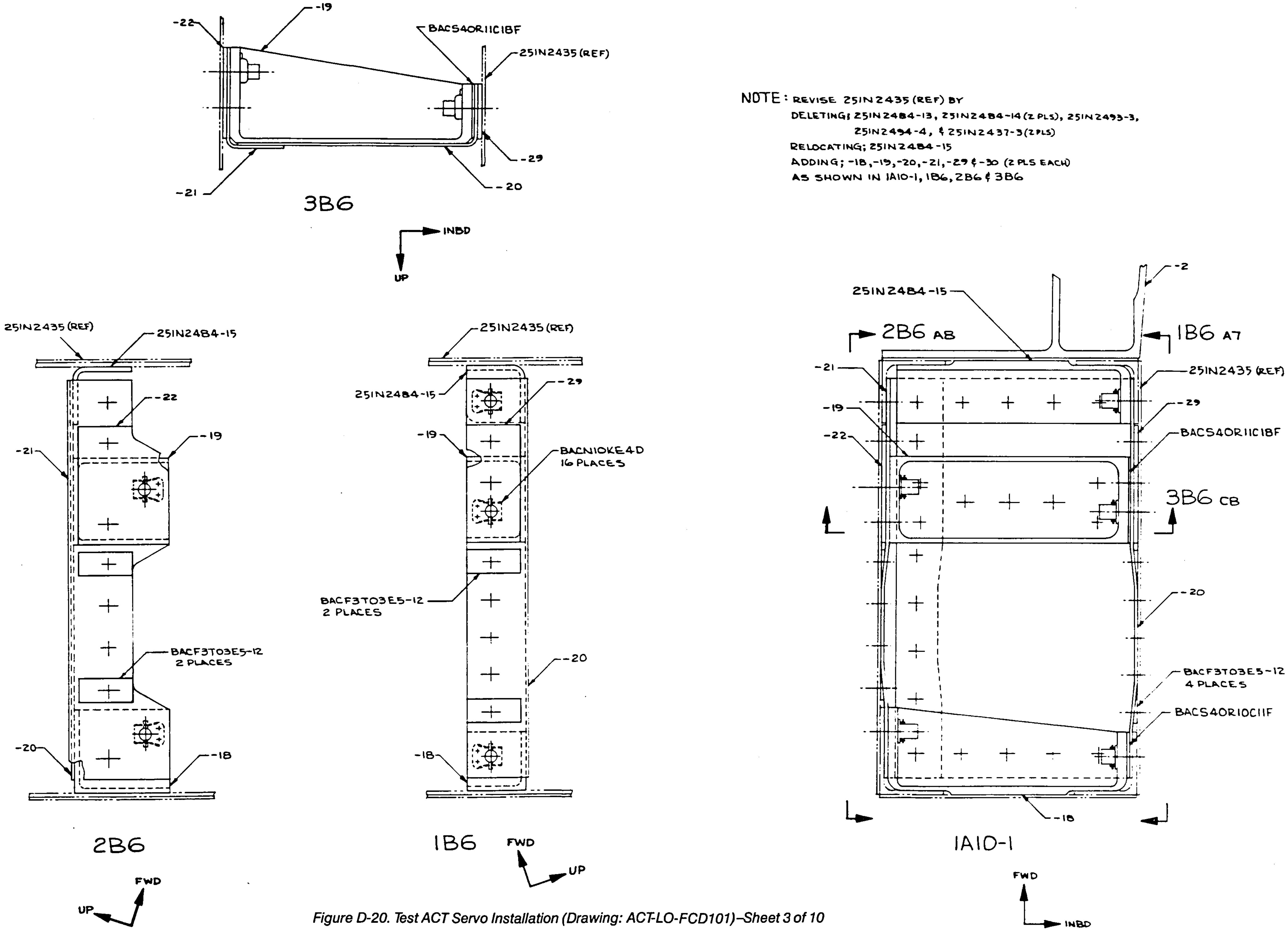
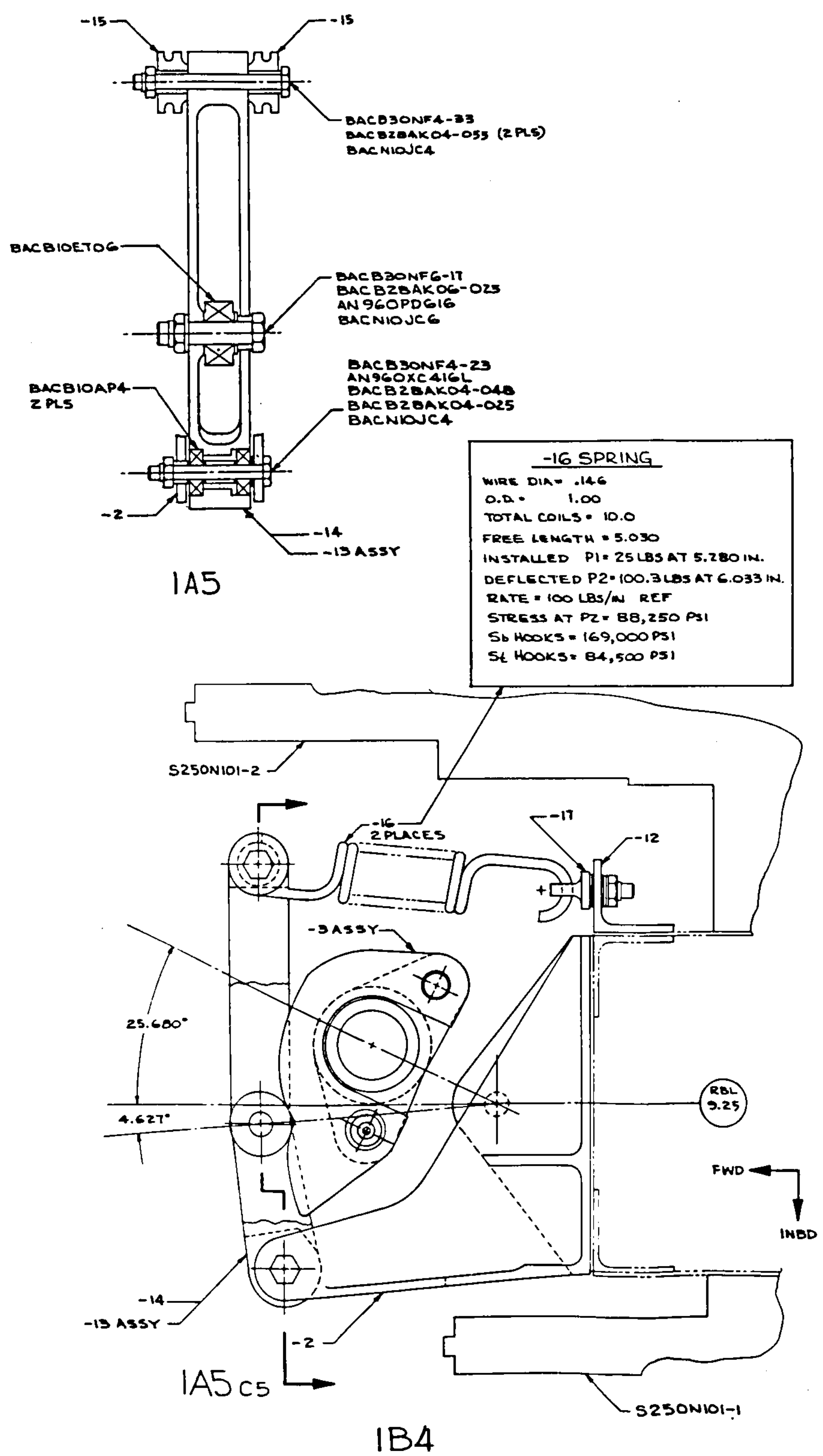


Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 3 of 10

*Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–
Sheet 3 of 10*



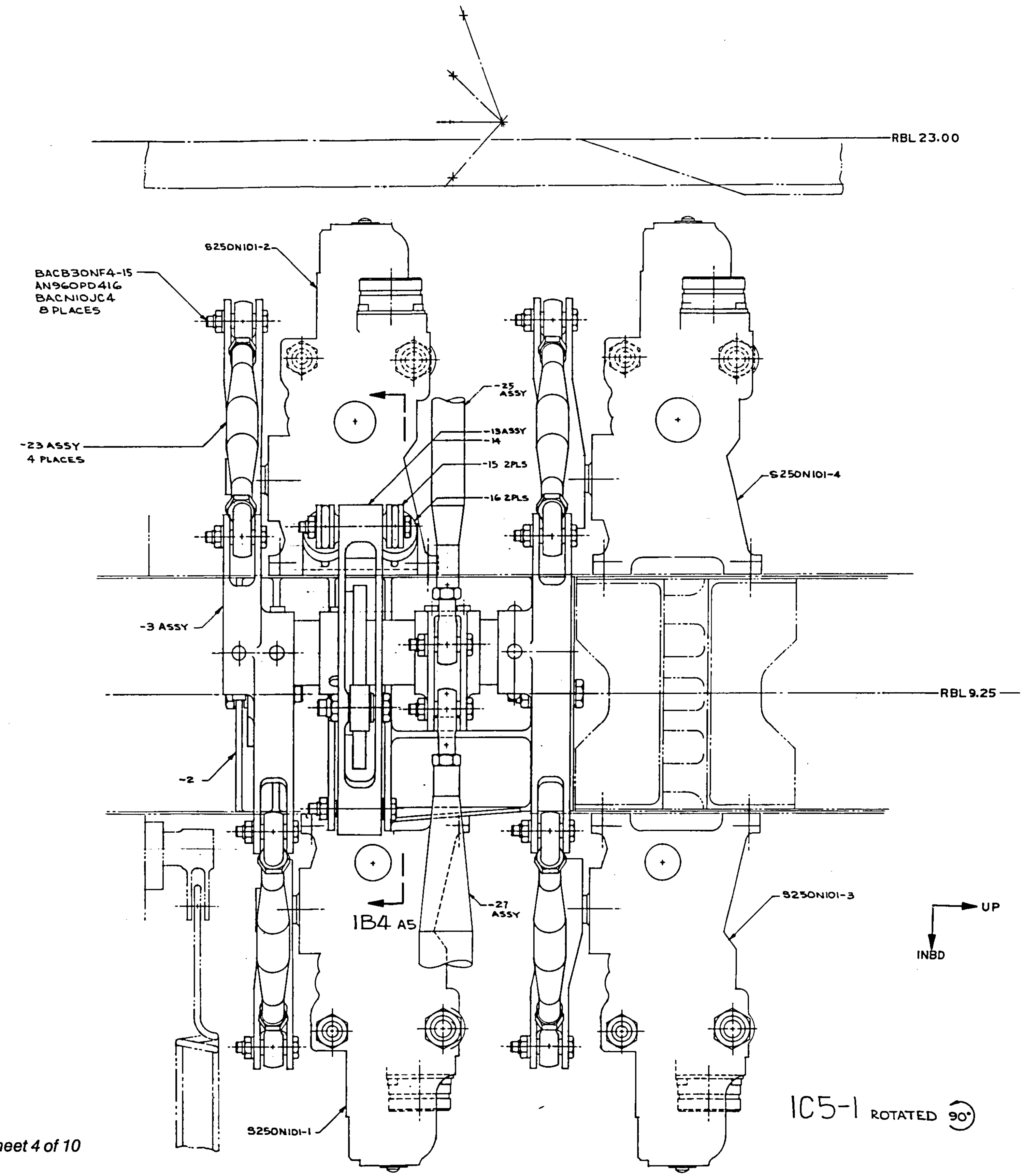
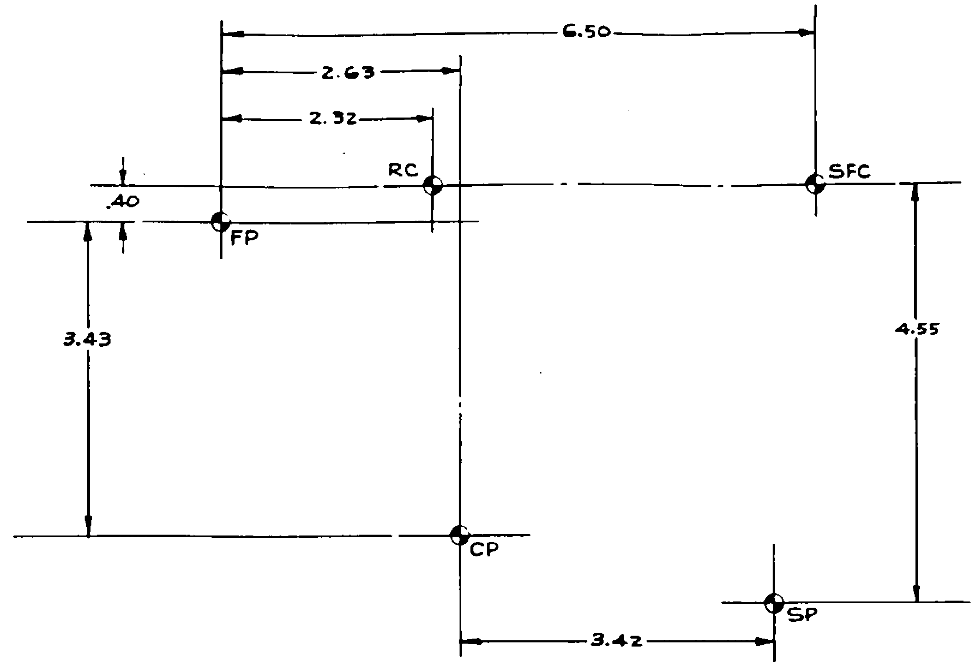
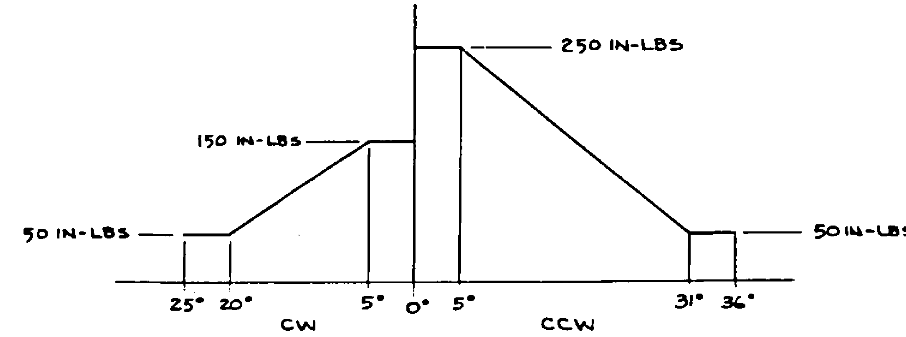


Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 4 of 10

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)– Sheet 4 of 10

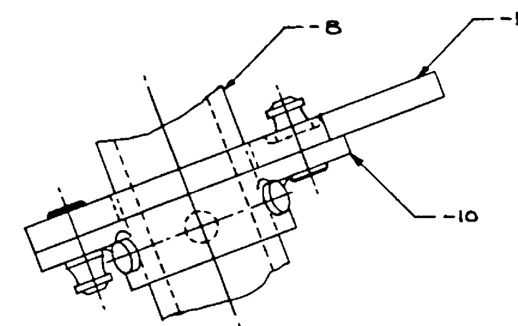




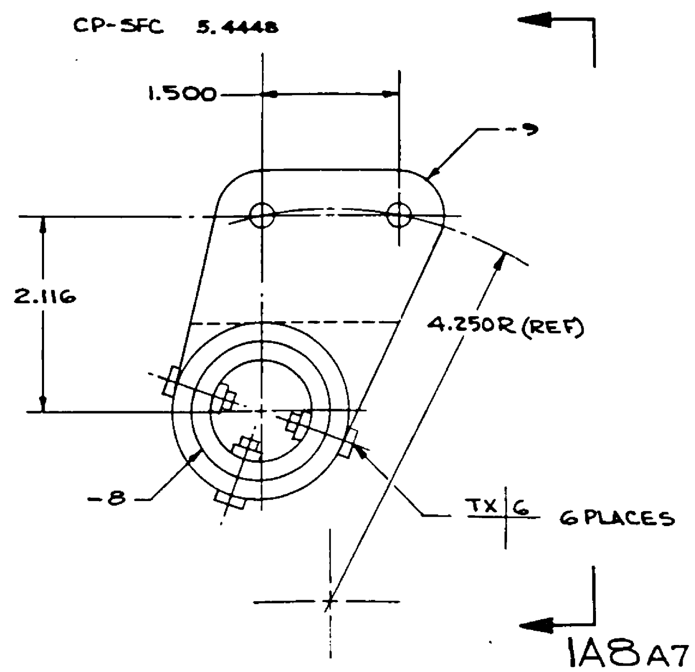
CAM TORQUE CURVE

BASIC GEOMETRY FOR COMPUTER CAM PROGRAM "FEEL 7"

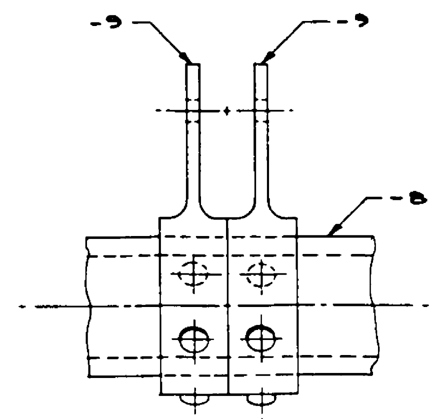
FP-CP	4.3222	ROLLER DIA	1.0625 DIA
FP-SP	7.3366	SPRING PRELOAD	50 LBS
FP-SFC	6.5123	SPRING RATE	200 LBS/IN
FP-RC	2.3542		
CP-SP	3.4950		
RC-CP	3.8425		
CP-SFC	5.4448		



2B6



3A3



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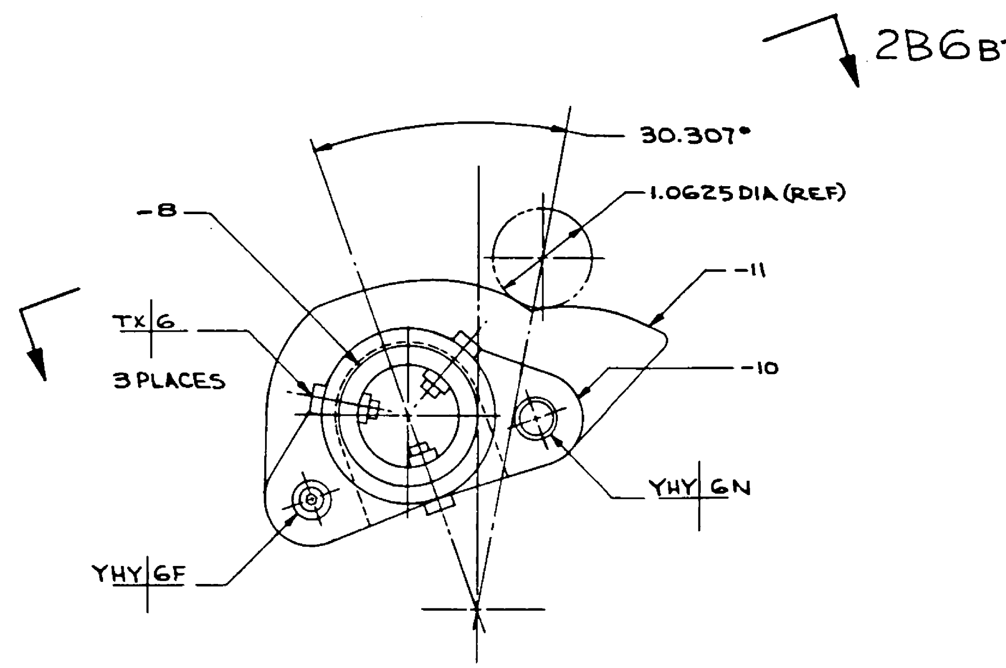


2A3

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 5 of 10

*Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–
Sheet 5 of 10*

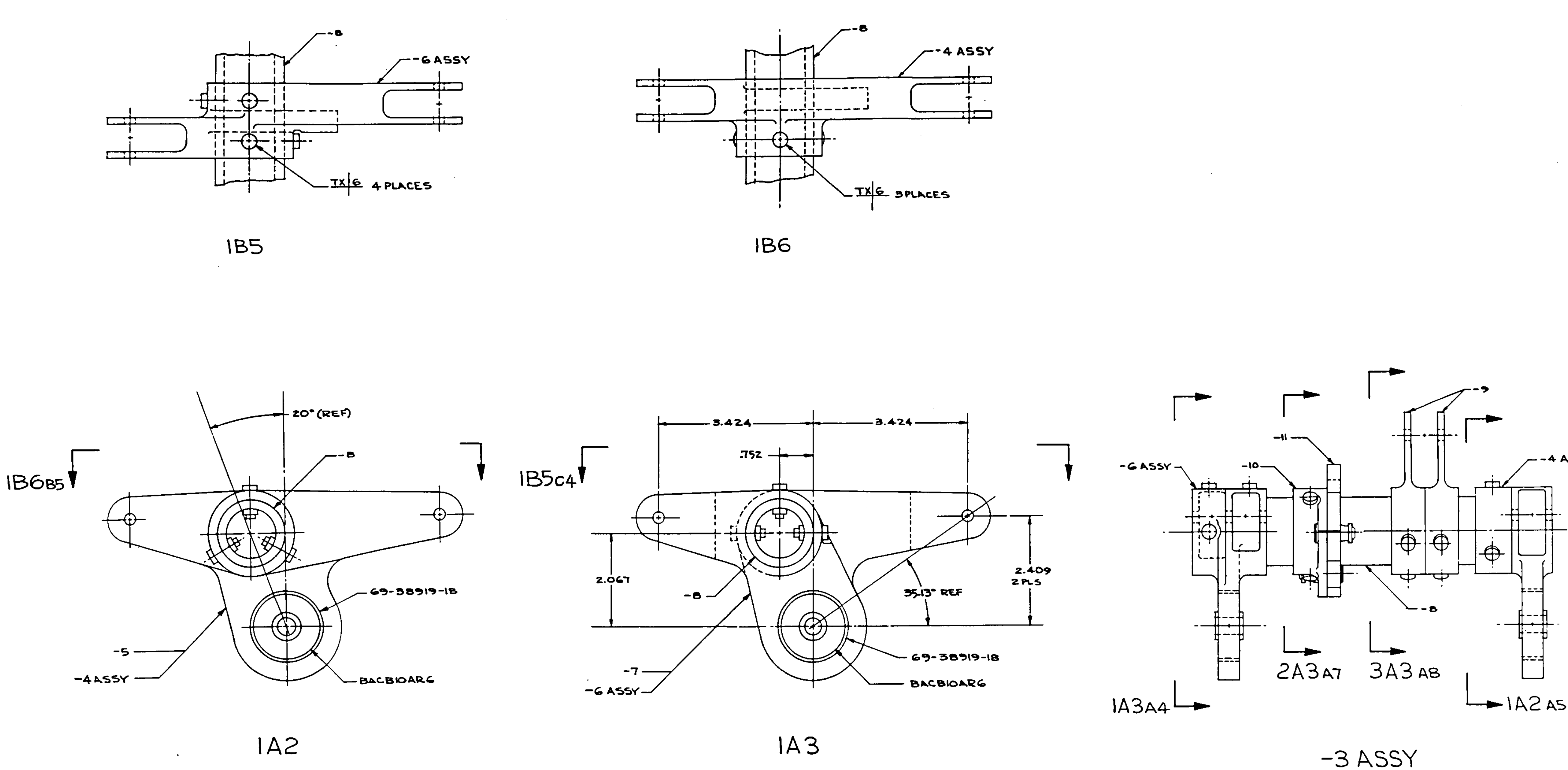


Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 6 of 10

*Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–
Sheet 6 of 10*

ACT-LO-FCD101 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-6	-4	-3	-1				
				-	-1	QUAD SERVO INSTALLATION		
				1	-2	BRACKET	5.5 X 6.0 X 8.0	M1,P1,F1
				1	-3	OUTPUT SHAFT ASSY		
			1		-4	UPPER CRANK ASSY		
		1			-5	UPPER CRANK	1.8 X 4.3 X 8.0	M1,P1,F1
			1		-6	LOWER CRANK ASSY		
	1				-7	LOWER CRANK	1.8 X 4.3 X 8.0	M1,P1,F1
	1	1			BACB10AR6	BEARING		P3
	1	1			69-38919-18	SLEEVE		P4,P5
			1		-8	SHAFT	1.6 DIA X 8.2 LONG	M1,P1,F1
			2		-9	OUTPUT CRANK HALF	1.0 X 3.0 X 3.7	M1,P1,F1
			1		-10	CAM SUPPORT	1.0 X 2.0 X 3.7	M1,P1,F1

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		-13	-3	-1				
			1		-11	CAM	.3 X 2.5 X 5.0	M2,P2,F2
				1	-12	ANGLE	MAKE FROM AND10134-1407	M3,P1,F1
				1	-13	FOLLOWER ARM ASSY		
		1			-14	FOLLOWER ARM	1.0 X 1.5 X 7.8	M1,P1,F1
		2			-15	SPRING GUIDE	1.0 DIA X .6 LONG	M1,P1,F1
		1			BACB10FT06	BEARING		
		1		1	BACB10AP4	BEARING		P3,P4
		2			BACB28AK04-055	BUSHING		
		1		3	BACB28AK06-025	BUSHING		
		1			BACB30NF4-33	BOLT		
		1			BACB30NF6-17	BOLT		
		1		14	BACN10JC4	NUT		

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 7 of 10

ACT-LO-FCD101 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		-23	-13	-1				
			1	3	BACN10JC6	NUT		
			1	1	AN960PD616	WASHER		
				2	-16	SPRING	.146 DIA X 32 LONG	M4,P2,F2
				2	-17	SPRING ANCHOR	.6 DIA X 1.7 LONG	M2,P2,F5
				2	-18	END FITTING	1.9 X 2.0 X 5.1	M1,P1,F1
				2	-19	CENTER FITTING	1.9 X 2.0 X 5.1	M1,P1,F1
				2	-20	WEB	.063 X 6.5 X 8.5	M5,F1
				2	-21	ANGLE	.063 X 3.5 X 8.5	M5,F1
				1	-22	SPACER	.090 X 2.0 X 2.5	M7,F4
				4	-23	ROD ASSY-SERVO		
		1			-24	TUBE	.75 DIA X .035 WALL X 6.2 LONG	M6,F3
				1	-25	ROD ASSY-R.H.		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-27	-25	-23	-1				
		1			-26	TUBE	.75 DIA X .035 WALL X 11.5 LONG	M6,F3
				1	-27	ROD ASSY-L.H.		
	1				-28	TUBE	1.25 DIA X .035 WALL X 30.7 LONG	M6,F3
	1	1	1		BACB10AD12	ROD END		
	1	1	1		BACB10AD13	ROD END		
	1	1	1		NAS509-6	NUT		
	1	1	1		NAS509-7	NUT		
				1	-29	SPACER	.090 X 1.2 X 2.5	M7,F4
				1	S250N101-1	SERVO		
				1	S250N101-2	SERVO		
				1	S250N101-3	SERVO		
				1	S250N101-4	SERVO		

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 8 of 10

ACT-LO-FCD101 Parts List

QTY				-1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
				2	251N2484-15	FITTING		
				1	BACB28AK04-025	BUSHING		
				1	BACB28AK04-048	BUSHING		
				12	BACB30NF4-15	BOLT		
				1	BACB30NF4-23	BOLT		
				1	BACB30NF6-16	BOLT		
				1	BACB30NF6-18	BOLT		
				16	NAS6604-12	BOLT		
				16	NAS620A416	WASHER		
				12	AN960PD416	WASHER		
				1	AN960XC416L	WASHER		
				8	BACF3T03E5-12	TAPERED FILLER		

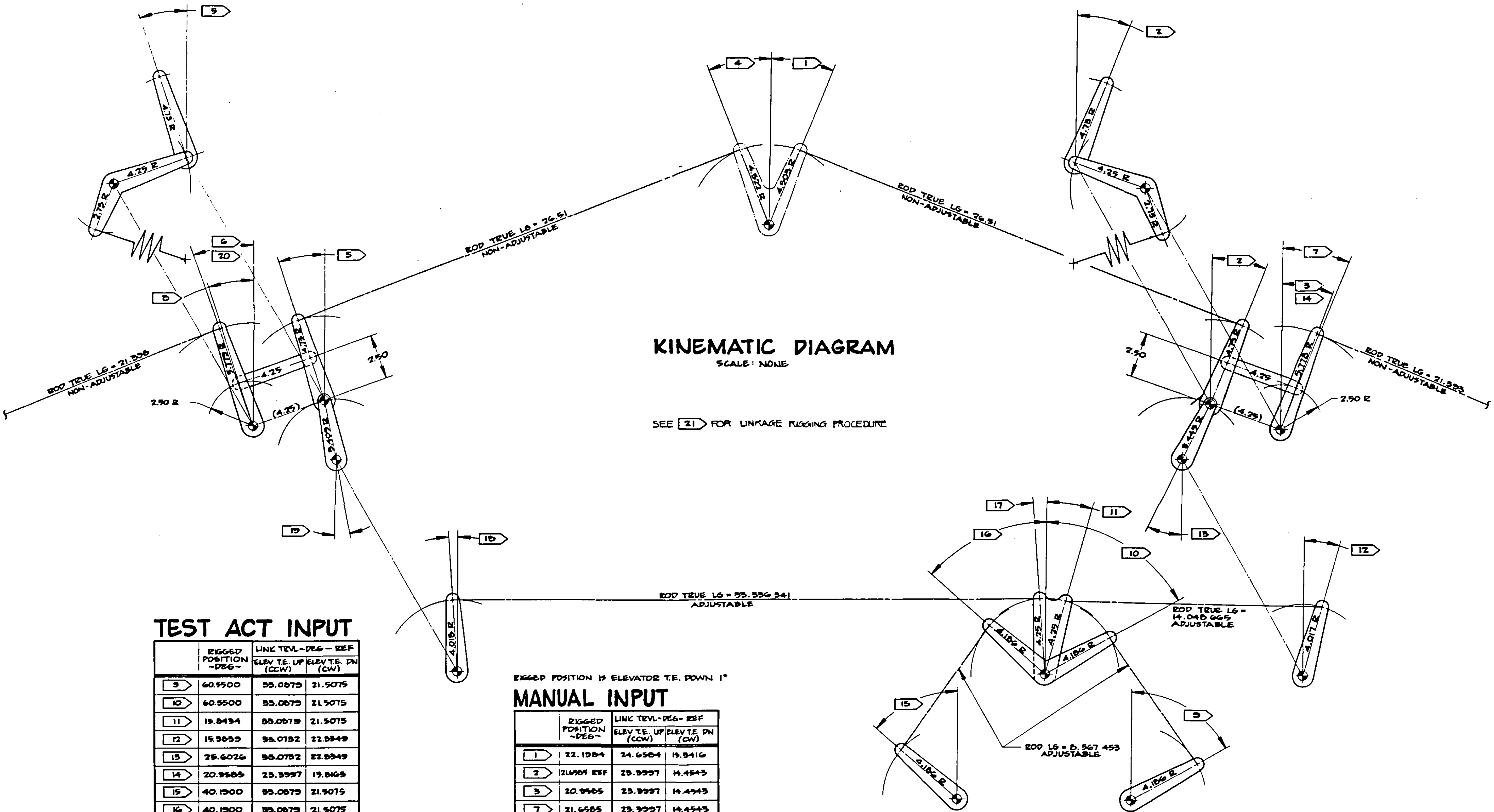
QTY				-1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
				2	BACS40R10C11F	LAMINATED SHIM		
				2	BACS40R11C18F	LAMINATED SHIM		
				AR	AN960PD416L	WASHER		
				16	BACN10KE4D	NUT PLATE		
				1	S250N102-1	SERVO		
				1	BACB30NF5-63	BOLT		

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 9 of 10

Notes for ACT-LO-FCD 101 Parts List

M1	7075-T7351 BAR PER QQ-A-225/9. ULTRASONIC INSPECT PER BAC5439 CLASS B.
M2	15-5 PH BAR PER AMS5659. HT TR 180-200 KSI PER BAC5619.
M3	2024-T42 EXTRUSION PER QQ-A-200/3
M4	17-7 PH WIRE PER AMS5678. HT TR TO CONDITION CH900 PER BAC5619.
M5	7075-0 BARE SHT PER QQ-A-250/12. HT TR TO T6 PER BAC5602.
M6	2024-0 TUBE PER WW-T-700/7. HT TR TO T42 PER BAC5602.
M7	2024-T42 CLAD SHT PER QQ-A-250/5.
P1	PENETRANT INSPECT PER BAC5423 AFTER MACHINING.
P2	MAGNETIC PARTICLE INSPECT PER BAC5424.
P3	INSTALL PER BAC5435.
P4	ROLLER SWAGE PER BAC5435.
P5	FILL GAP IN SLEEVE WITH DOW-CORNING Q3-0121 SEALANT PER BAC5000 AFTER SWAGING.
F1	F-18.13
F2	F-17.09
F3	F-18.07
F4	F-18.06
F5	F-16.01

Figure D-20. Test ACT Servo Installation (Drawing: ACT-LO-FCD101)–Sheet 10 of 10



TEST ACT INPUT

	RIGGED POSITION ~DEG~	LINK TRVL~DEG~REF	
		ELEV T.E. UP (CCW)	ELEV T.E. DN (CW)
9	60.5500	33.0879	21.5075
10	60.5500	33.0879	21.5075
11	15.8434	33.0879	21.5075
12	15.3839	33.0732	22.8349
13	25.6026	33.0732	22.8349
14	20.9585	23.3997	13.8165
15	40.1900	33.0879	21.5075
16	40.1900	33.0879	21.5075
17	4.4886	33.0879	21.5075
18	8.5383	33.0790	22.8419
19	12.0131	33.0790	22.8419
20	19.5932	21.3802	14.6442

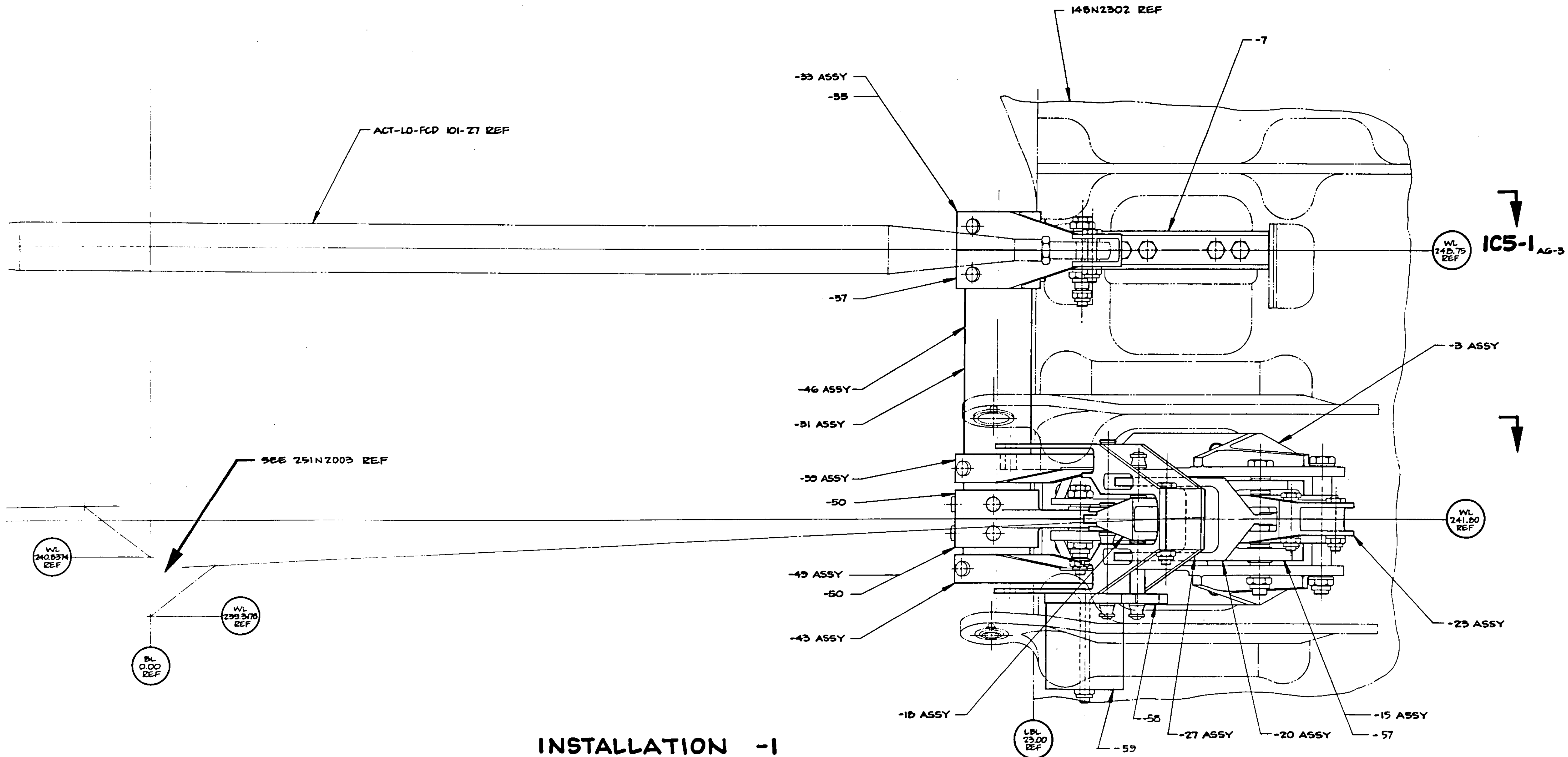
RIGGED POSITION IS ELEVATOR T.E. DOWN 1°

MANUAL INPUT

	RIGGED POSITION ~DEG~	LINK TRVL~DEG~REF	
		ELEV T.E. UP (CCW)	ELEV T.E. DN (CW)
1	22.1984	24.6504	15.3416
2	21.6585 REF	23.3997	14.4543
3	20.9585	23.3997	14.4543
7	21.6585	23.3997	14.4543
4	21.8016	24.6504	15.3416
5	18.8835 REF	22.7800	14.6442
6	19.5932	22.7800	14.6442
8	18.8835	22.7800	14.6442

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 1 of 16

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)– Sheet 1 of 16



INSTALLATION -1

CONTROL RODS -52, -53, 251N2662-1 AND 251N2662-3 OMITTED FOR CLARITY

FRONT VIEW

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 2 of 16

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)– Sheet 2 of 16

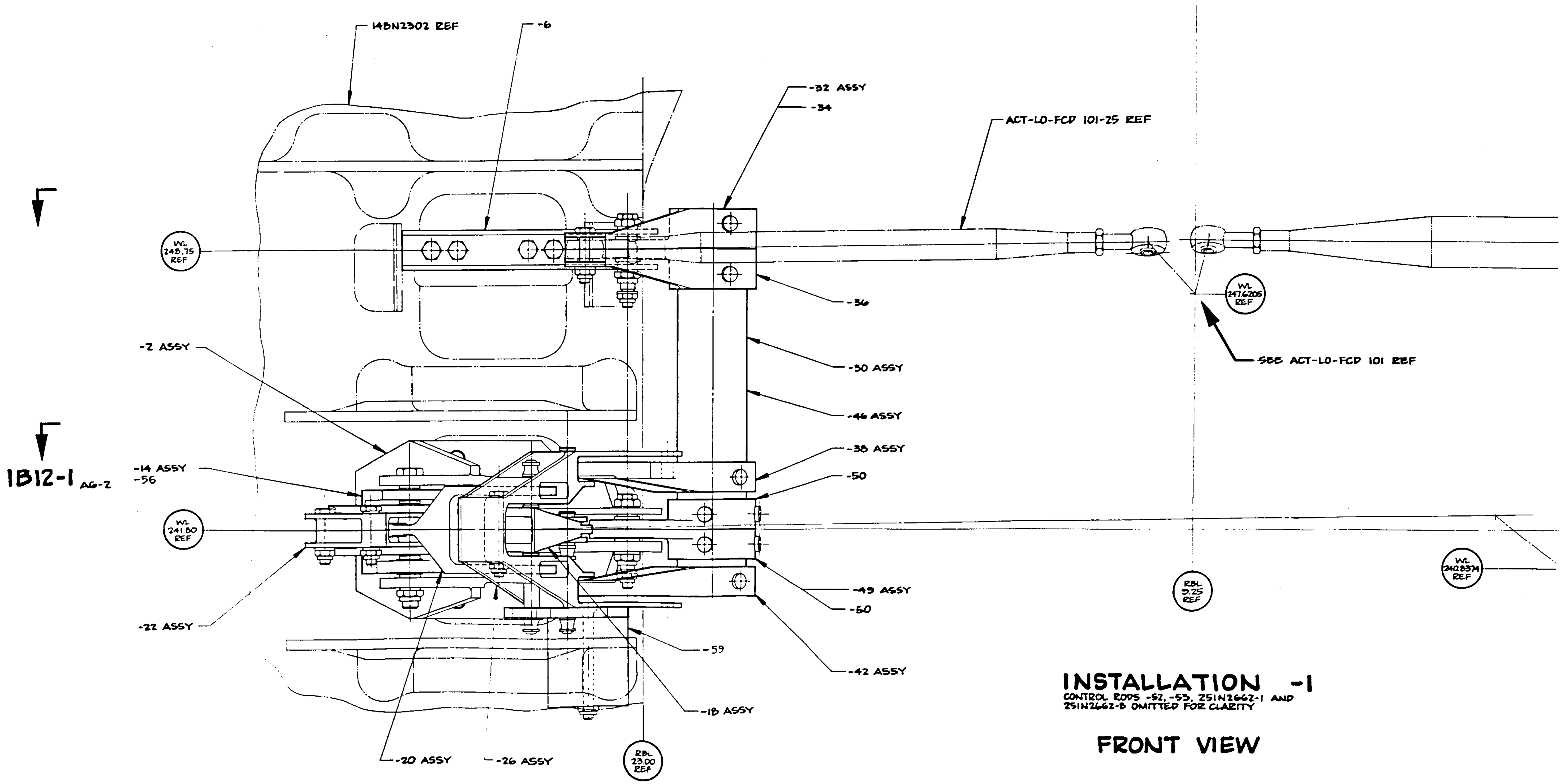


Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 3 of 16

*Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–
Sheet 3 of 16*

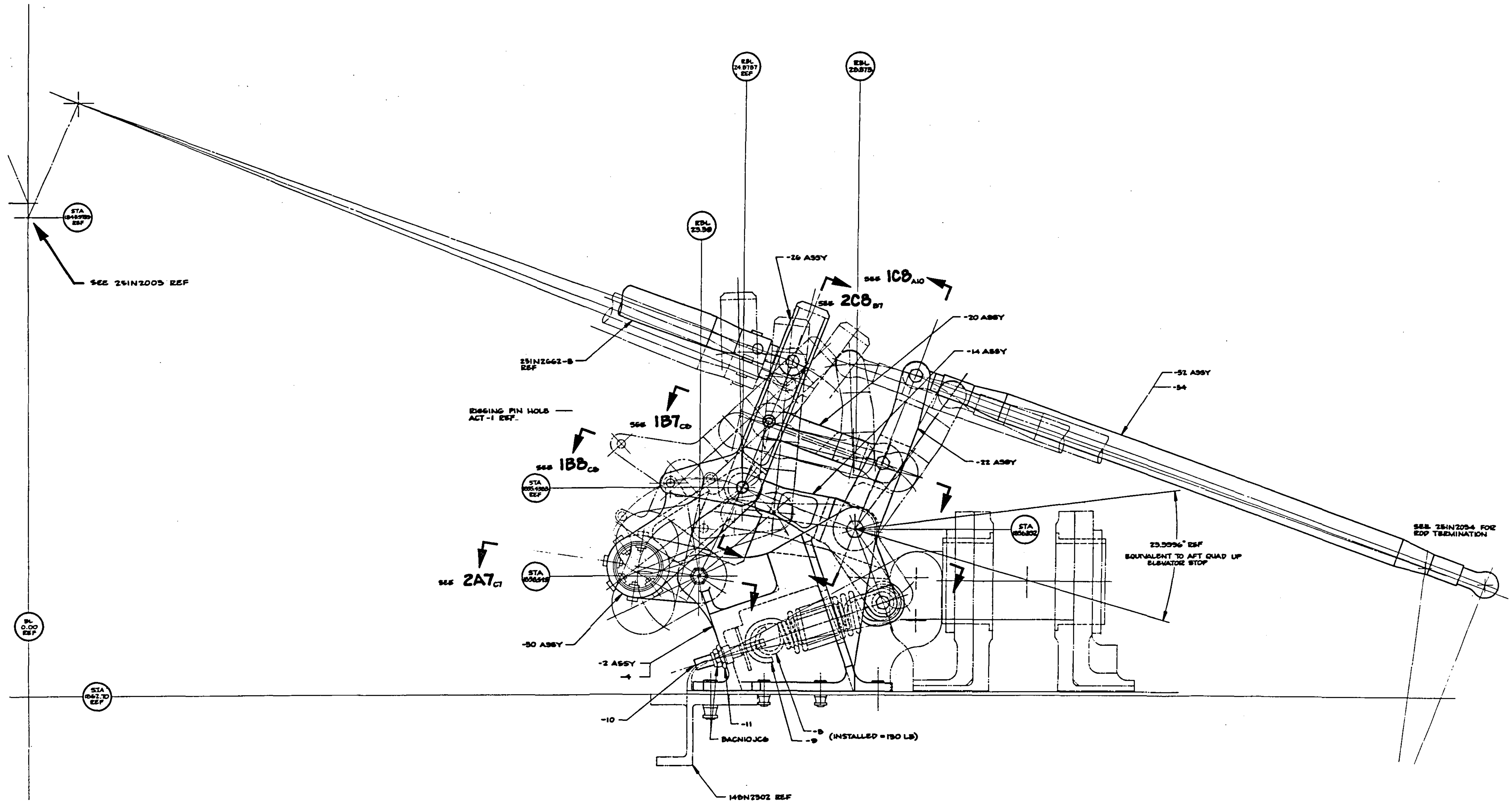
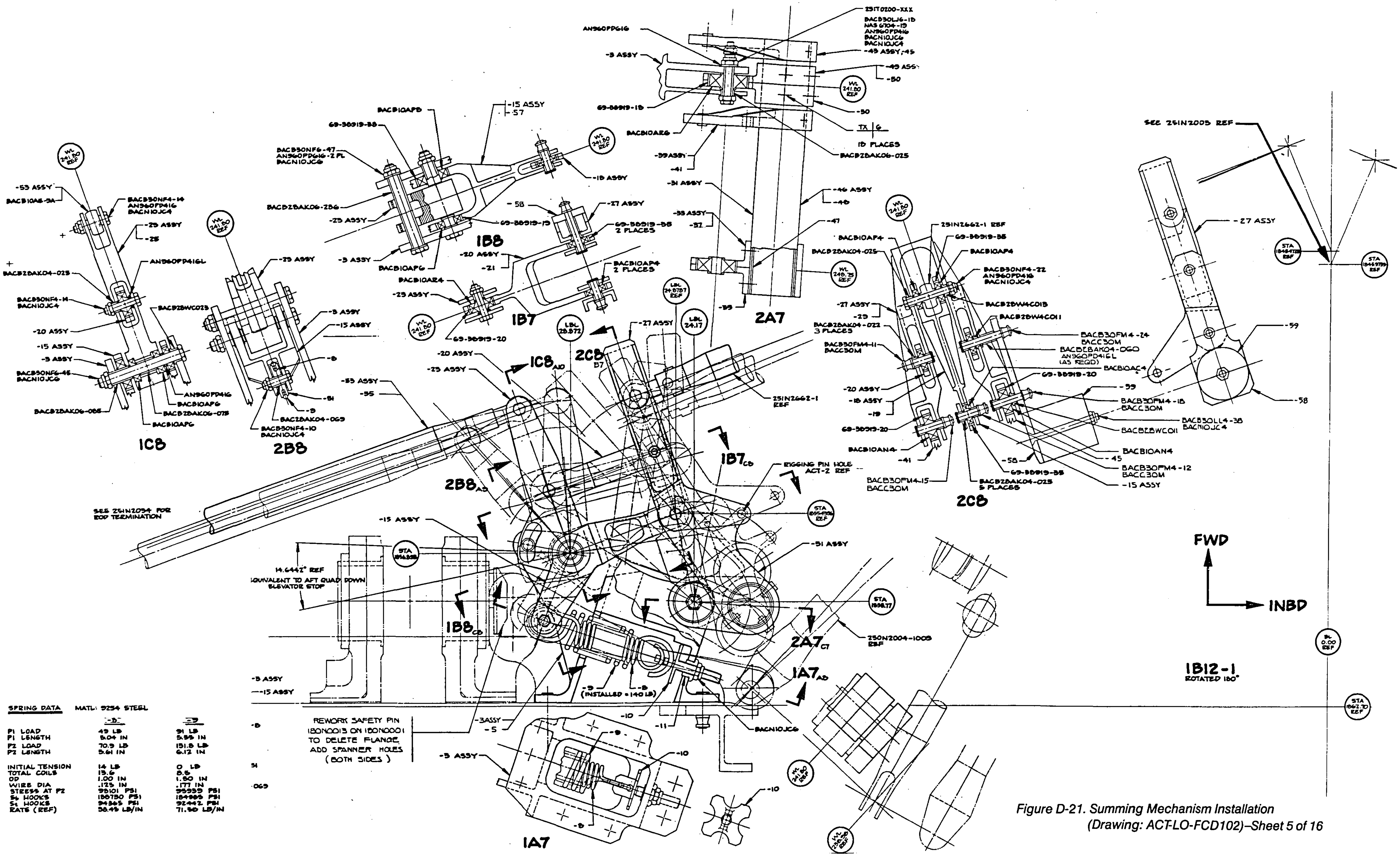


Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 4 of 16

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–
Sheet 4 of 16



	-8	-9
P1 LOAD	49 LB	91 LB
P1 LENGTH	5.04 IN	5.55 IN
P2 LOAD	70.9 LB	151.8 LB
P2 LENGTH	5.61 IN	6.12 IN
INITIAL TENSION	14 LB	0 LB
TOTAL COILS	13.6	8.5
OD	1.00 IN	1.80 IN
WIRE DIA	.125 IN	.177 IN
STRESS AT P2	90101 PSI	95939 PSI
Sb HOOKS	180730 PSI	184985 PSI
St HOOKS	94365 PSI	92442 PSI
RATE (REF)	38.45 LB/IN	71.50 LB/IN

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 5 of 16

*Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–
Sheet 5 of 16*

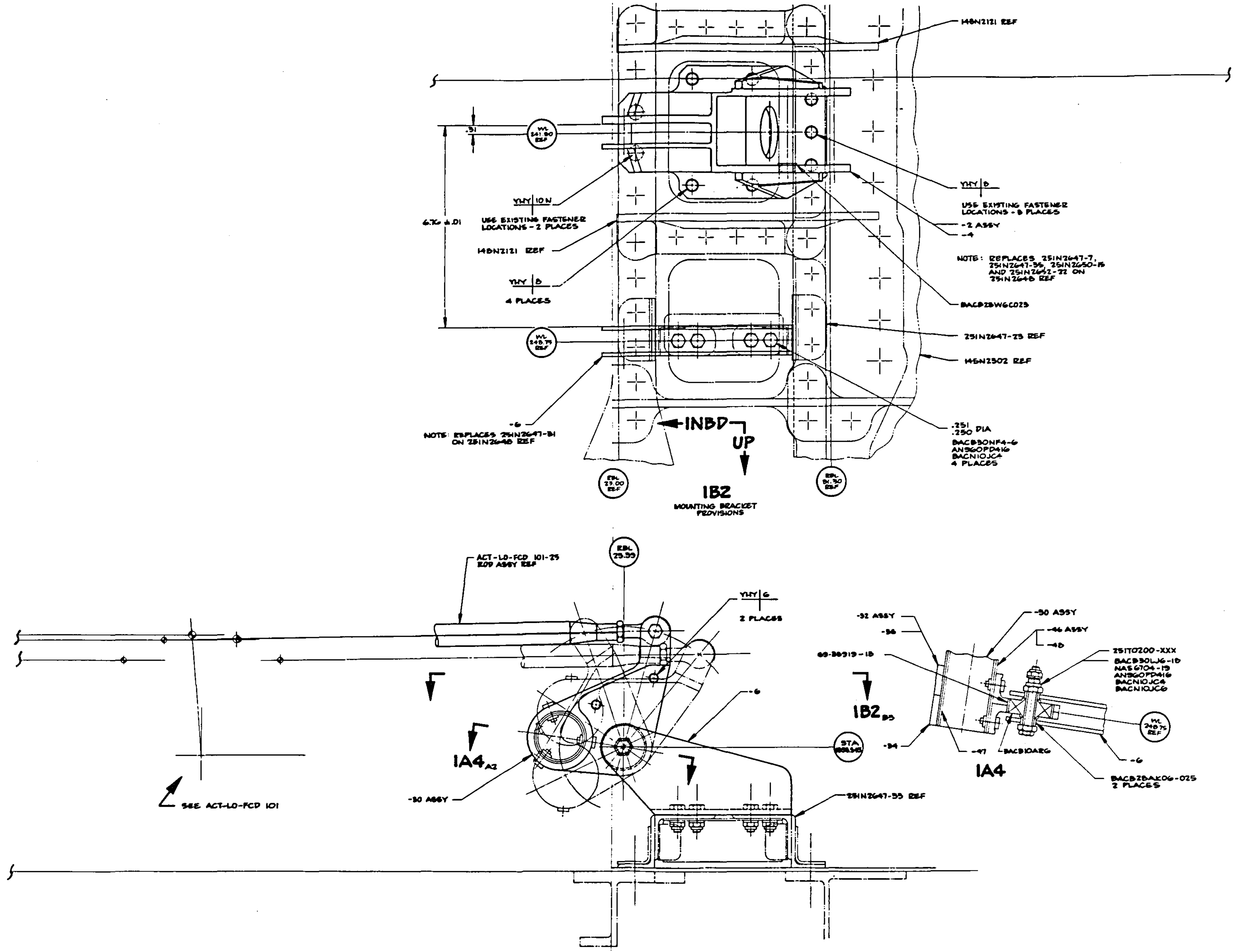


Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 6 of 16

Figure D-21 Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–
Sheet 6 of 16

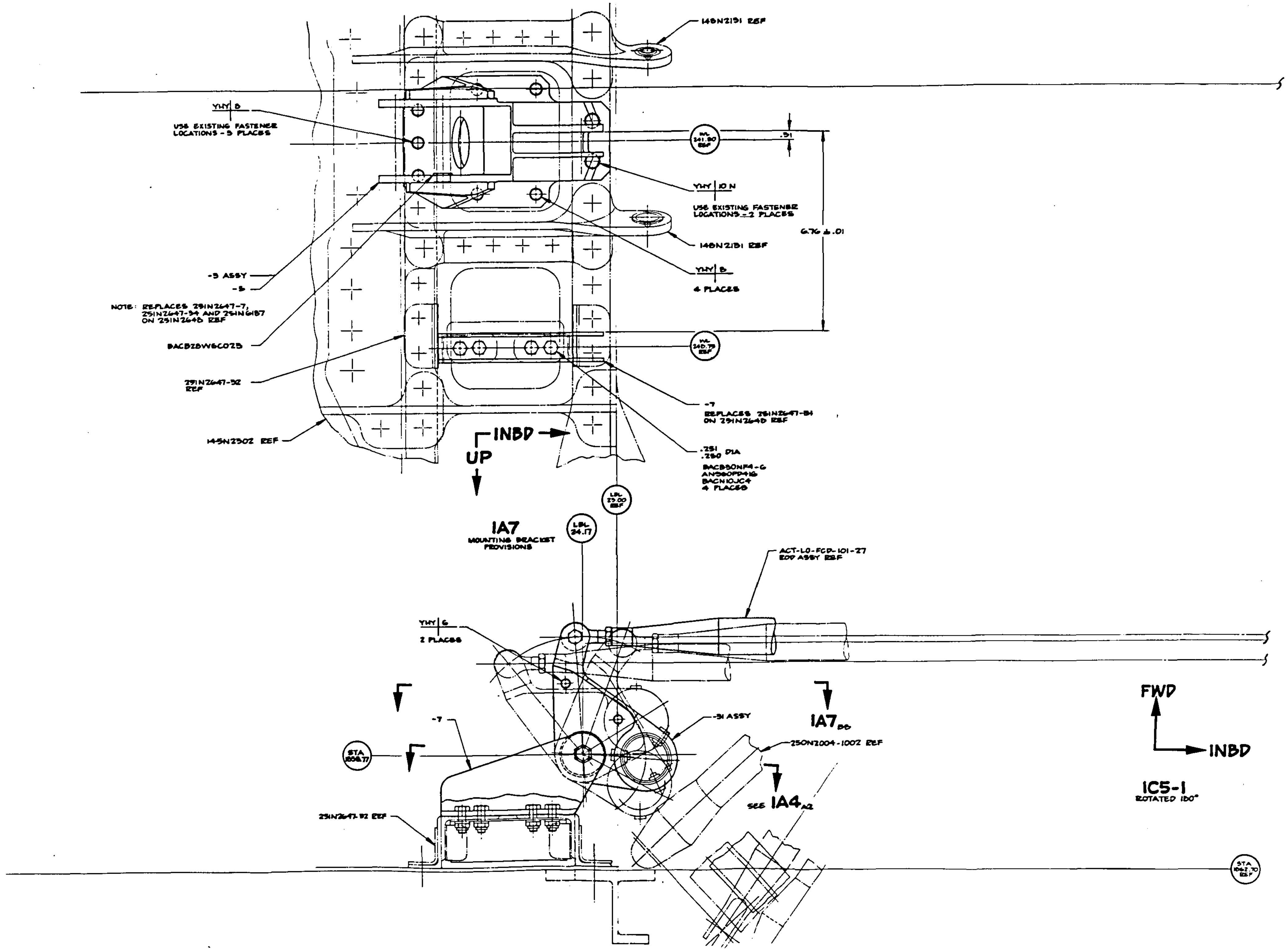


Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 7 of 16

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–
Sheet 7 of 16

ACT-LO-FCD102 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		-3	-2	-1				
				-	-1	SUMMING MECHANISM INSTALLATION		21 (See Sheet 15 of 16)
			-	1	-2	SUPPORT ASSY-LOWER		P9
			1		-4	SUPPORT	4.5 X 6.8 X 8.5	M1, F1
			1		BACB28W6C023	BUSHING-FLANGED		
		-		1	-3	SUPPORT ASSY-LOWER		P9
		1			-5	SUPPORT	4.5 X 6.6 X 8.0	M1, F1
		1			BACB28W6C023	BUSHING-FLANGED		
				1	-6	SUPPORT-UPPER	1.1 X 3.1 X 6.5	M1, F1, F3
				1	-7	SUPPORT-UPPER	1.1 X 2.9 X 5.6	M1, F1, F3
				2	-8	TENSION SPRING-INNER	.125 DIA X 52 LG	M3, F5
				2	-9	TENSION SPRING-OUTER	.177 DIA X 50 LG	M3, F5
				2	-10	ADJUSTER-SPRING	1.7 X 1.7 X 2.8	M5, P2, F4

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			-14	-1				
				2	-11	NUT-ADJUSTER	.75 HEX X .5 LG	M5, P2, F4
			-	1	-14	SPRING LINK ASSY		P3, P4, P5 P6, P8, F3
			1		-56	LINK		OPPOSITE -57 M1, P1, F1
			1		BACB10AP6	BEARING		
			1		69-38919-19	SLEEVE-BRG RETENTION		
				2	-58	COUNTER BALANCE PLATE	.25 X 2.50 X 7.0	M9, P2, F5
			1		BACB10AP8	BEARING		
			1		69-38919-38	SLEEVE-BRG RETENTION		

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 8 of 16

ACT-LO-FCD102 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
-18		-15		-1				
		-		1	-15	SPRING LINK ASSY		P3, P4, P5 P6, P8, F3
		1			-57	LINK	2.5 X 4.0 X 7.5	M1, P1, F1
		1			BACB10AP6	BEARING		
		1			69-38919-19	SLEEVE- BRG RETENTION		
				2	-59	MASS BALANCE	2.0 DIA X 2.50 LG	M9, P2, F5
		1			BACB10AP8	BEARING		
		1			69-38919-38	SLEEVE BRG RETENTION		
-				2	-18	ERECTION LINK ASSY		P3, P4, P5
1					-19	LINK	1.2 X 1.3 X 6.1	M1, P1, F1
1					BACB10AP4	BEARING		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-22	-20	-18	-1				
			2		69-38919-35	SLEEVE- BRG RETENTION		
			1		BACB10AC4	BEARING		
		-		2	-20	DRAG LINK ASSY		P3, P4, P5
		1			-21	LINK	1.3 X 2.3 X 5.7	M1, P1, F1
		2			BACB10AP4	BEARING		
		2			69-38919-35	SLEEVE- BRG RETENTION		
		1			BACB10AR4	BEARING		
		1			69-38919-20	SLEEVE- BRG RETENTION		
	-			1	-22	LINK ASSY- OUTPUT		P3
	1				-24	LINK	1.4 X 3.0 X 7.2	M1, P1, F1
	1				BACB10AP6	BEARING		

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 9 of 16

ACT-LO-FCD102 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
	-27	-26	-23	-1				
			-	1	-23	LINK ASSY- OUTPUT		P3
			1		-25	LINK	1.4 X 2.9 X 7.2	M1, P1, F1
			1		BACB10AP6	BEARING		
		-		1	-26	SUMMING LINK ASSY		OPPOSITE -27 P9
		1			-28	SUMMING LINK	3.0 X 4.0 X 6.9	OPPOSITE -29 M1, P1, F1
		1			BACB28W4C013	BUSHING- FLANGED		
		2			BACB28W4C011	BUSHING- FLANGED		
	-			1	-27	SUMMING LINK ASSY		P9
	1				-29	SUMMING LINK	3.0 X 4.0 X 6.9	M1, P1, F1
	1				BACB28W4C013	BUSHING- FLANGED		
	2				BACB28W4C011	BUSHING- FLANGED		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
-49	-32	-46	-30	-1				
			-	1	-30	TORQUE TUBE ASSY		F3, P6, P8
		-	1		-46	TUBE ASSY		P7
		1			-47	TUBE-INNER	1.625 OD X .065 WALL X 9.8 LG	M4, P1
		1			-48	TUBE-OUTER	1.625 ID X .065 WALL X 9.8 LG	M4, P1
	-		1		-32	LUG ASSY- UPPER PIVOT		F3, P3, P4 P5, P6, P8
	1				-34	LUG-UPPER	1.1 X 4.0 X 6.5	M1, P1, F1
	1				-36	LUG-LOWER	1.1 X 4.0 X 6.5	OPPOSITE -34 M1, P1, F1
	1				BACB10AR6	BEARING		
	1				69-38919-18	SLEEVE- BRG RETENTION		
-			1		-49	LUG ASSY- LOWER PIVOT		F3, P3, P4 P5, P6, P8
2					-50	LUG	.9 X 2.3 X 4.4	M1, P1, F1
1					BACB10AR6	BEARING		

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 10 of 16

ACT-LO-FCD102 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
-42	-38	-49	-30	-1				
		1			69-38919-18	SLEEVE- BRG RETENTION		
	-		1		-38	LUG ASSY UPPER SUMMING		P3, P4, P5
	1				-40	LUG	.9 X 3.0 X 6.5	M1, P1, F1
	1				BACB10AN4	BEARING		
	1				69-38919-20	SLEEVE- BRG RETENTION		
-			1		-42	LUG ASSY- LOWER SUMMING		P3, P4, P5
1					-44	LUG	.9 X 2.3 X 6.5	M1, P1, F1
1					BACB10AN4	BEARING		
1					69-38919-20	SLEEVE- BRG RETENTION		
			18		MS 90354-6	RIVET		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
-49	-33	-46	-31	-1				
			-	1	-31	TORQUE TUBE ASSY		F3, P6, P8
		-	1		-46	TUBE ASSY		P7
		1			-47	TUBE-INNER	1.625 OD X .065 WALL X 9.8 LG	M4, P1
		1			-48	TUBE-OUTER	1.625 ID X .065 WALL X 9.8 LG	M4, P1
	-		1		-33	LUG ASSY- UPPER PIVOT		F3, P3, P4 P5, P6, P8
	1				-35	LUG-UPPER	1.1 X 4.4 X 6.3	M1, P1, F1
	1				-37	LUG-LOWER	1.1 X 4.4 X 6.3	OPPOSITE-35 M1, F1, P1
	1				BACB10AR6	BEARING		
	1				69-38919-18	SLEEVE- BRG RETENTION		
-			1		-49	LUG ASSY- LOWER PIVOT		F3, P3, P4 P5, P6, P8
2					-50	LUG	.9 X 2.3 X 4.4	M1, P1, F1
1					BACB10AR6	BEARING		

Figure D-21. Summing Mechanism installation (Drawing: ACT-LO-FCD102)–Sheet 11 of 16

ACT-LO-FCD102 Parts List

QTY -43	QTY -39	QTY -49	QTY -31	QTY -1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
		1			69-38919-18	SLEEVE BRG RETENTION		
	-		1		-39	LUG ASSY- UPPER SUMMING		P3,P4,P5
	1				-41	LUG	.9 X 2.8 X 6.4	M1,P1,F1
	1				BACB10AN4	BEARING		
	1				69-38919-20	SLEEVE- BRG RETENTION		
-			1		-43	LUG ASSY- LOWER SUMMING		P3,P4,P5
1					-45	LUG	.9 X 2.3 X 6.4	M1,P1,F1
1					BACB10AN4	BEARING		
1					69-38919-20	SLEEVE- BRG RETENTION		
			18		MS90354-6	RIVET		

QTY	QTY	QTY -53	QTY -52	QTY -1	PART NO.	DESCRIPTION	STOCK SIZE	NOTES
				2	-51	SPRING GUIDE	1.38 DIA X .4 LG	M5,P2,F4
			-	1	-52	CONTROL ROD ASSY		P8,F3
			1		-54	TUBE	1.00 DIA X .035 WALL X21.6 LG	M2,F2
			2		BACB10AE-9A	BEARING- ROD END		
		-		1	-53	CONTROL ROD ASSY		P8,F3
		1			-55	TUBE	1.00 DIA X .035 WALL X 21.6 LG	M2,F2
		2			BACB10AE-9A	BEARING ROD END		
				8	BACB30NF4-6	BOLT		
				2	BACB30NF4-10	BOLT		
				4	BACB30NF4-14	BOLT		
				2	BACB30NF4-22	BOLT		

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 12 of 16

ACT-LO-FCD102 Parts List

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
			-X	-1				
			-	4	251T0200-X	DUAL BOLT ASSY		
			1		BACB30LJ6-18	BOLT-OUTER		
			1		NAS6704-19	BOLT-INNER		
			1		AN960PD416	WASHER		
			1		BACN10JC6	NUT		
			1		BACN10JC4	NUT		
				2	BACB30NF6-45	BOLT		
				2	BACB30NF6-47	BOLT		
				16	BACN10JC4	NUT		
				6	BACN10JC6	NUT		

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
				-1				
				12	AN960PD416	WASHER		
				8	AN960PD616	WASHER		
				4	BACB30FM4-11	BOLT		
				2	BACB30FM4-12	BOLT		
				4	BACB30FM4-15	BOLT		
				10	BACC30M	COLLAR		
				2	BACB10AP6	BEARING		P3
				2	BACB10AP4	BEARING		P3
				AR	AN960PD416L	WASHER		
				6	BACB28AK04-022	BUSHING		
				14	BACB28AK04-025	BUSHING		
				2	BACB28AK04-069	BUSHING		

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 13 of 16

ACT-LO-FCD102 Parts List and Notes

QTY					PART NO.	DESCRIPTION	STOCK SIZE	NOTES
				-1				
				2	BACB28AK06-075	BUSHING		
				2	BACB28AK06-085	BUSHING		
				2	BACB28AK06-286	BUSHING		
				6	BACB28AK06-025	BUSHING		
				2	BACB28AK04-060	BUSHING		
				2	BACB30FM4-24	BOLT		
				2	BACB30FM4-18	BOLT		
				2	BACB30LL4-38	BOLT		

M1 ALUMINUM ALLOY 7075-T7351 BAR PER QQ-A-225/9. ULTRASONIC INSPECT PER BAC5439, CLASS B.

M2 2024-0 ALUMINUM ALLOY TUBE PER WW-T-700/3. HEAT TREAT TO T42 PER BAC5602.

M3 9254 STEEL WIRE PER ASTM A401.

M4 2024-T3 ALUMINUM ALLOY SEAMLESS TUBING PER WW-T-700/3 TYPE 1.

M5 15-5 PH BAR PER AMS 5659. HEAT TREAT TO 150-170 KSI PER BAC5619.

P1 PENETRANT INSPECT PER BAC5423.

P2 MAGNETIC PARTICLE INSPECT PER BAC5424.

P3 INSTALL BEARING PER BAC5435.

P4 ROLLER SWAGE SLEEVE PER BAC5435.

P5 FILL END GAP IN 69-38919 SLEEVE WITH DOW-CORNING Q3-0121 SEALANT PER BAC5000 AFTER SWAGING.

P6 FAYING SURFACE SEAL WITH BMS 5-95 SEALANT PER BAC5000. INSTALL RIVETS/FASTENERS WITHIN CURING TIME OF SEALANT.

P7 CLEAN WITH M.E.K. PER BAC5750 AND BOND OVER CURED BMS10-11 PRIMER PER BAC5010 TYPE 70. SWAGE OR DRAW OUTER TUBE OVER INNER TUBE (AFTER APPLICATION OF PRIMER AND ADHESIVE) TO MEET REQUIRED OUTSIDE DIA. OPTIONAL FINISH MACHINE AFTER SWAGING.

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 14 of 16

Notes for ACT-LO-FCD102 Parts List (Continued)

P8	INSTALL FASTENERS WITH F-20.06
P9	INSTALL BUSHING PER BAC5435. FAY FILLET SEAL WITH BMS 5-95 SEALANT PER BAC5000.
F1	F-18.13. OMIT PRIMER ON FINSHED HOLE DIAMETERS.
F2	F-18.07
F3	AFTER MACHINING F-17.10.
F4	F-16.01
F5	F-20.03

21 **ACT SUMMING UNIT RIGGING PROCEDURE**

This procedure should be used in place of 251N2001 Sht. 2 Para. 15.0 P. C. U. input rod adjustment. (757 Elevator Control Rigging Instructions)

A. Summing Lever Spring Adjustment

Depressurize all Hydarulic Systems

1. Remove all tension from the R. H. summing lever spring. (ACT-LO-FCD 102)
2. Insert rig pin #7 (251N2001) at aft quadrant.
3. Disconnect the control rod between the R. H. summing mechanism and the R. H. idler lever.
4. Remove the control rod between the L. H. summing mechanism and the quad shaft. (ACT-LO-FCD 101)
5. Remove all spring tension from the voting cam follower arm. (ACT-LO-FCD 101)
6. Rotate the quadshaft clockwise by applying a force to the L. H. quadshaft output arm. Force should be increased to a level at which the R. H. ACT rig Pin can be inserted between the summing lever and sumshaft. Force applied at the quadshaft input arm to allow insertion of the rig pin should be 7 lb. $\pm$ 2 lbs.

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 15 of 16

Notes for ACT-LO-FCD102 Parts List (Concluded)

7. To obtain the correct force at the quadshaft input arm, adjust the summing lever spring tension by rotating the nut provided at the spring anchor.

8. Remove Rig Pin #7 and R. H. ACT Rig Pin.

9. To adjust the L. H. summing lever spring, repeat steps 1 thru 8 substituting "L. H." for "R. H." and "R. H." for "L. H." in all cases.

10. Replace voting cam follower arm spring.

11. Replace all control rods.

B. Summing Mechanism Rigging

Depressurize all Hydraulic Systems

1. Complete R. H. Summing lever spring adjustment.

2. Insert Rig Pins # 7 & # 9. (251N2001)

3. Remove the new control rod connecting the R. H. summing mechanism to the quadshaft. (ACT-LO-FCD 101)

4. Adjust the existing control rod between the inb'd R. H. P.C.U. bellcrank and the R. H. idler lever until the R. H. ACT Rig Pin can be freely inserted.

5. With the R. H. ACT Rig Pin inserted, install the control rod between the R. H. summing mechanism and the quad shaft and adjust the rod length until the rod end bolts can be freely inserted.

6. Remove Rig Pins 7, 9, and R. H. ACT Rig Pin.

7. To rig L. H. summing mechanism repeat steps 1 thru 6 substituting "L. H." for "R. H." in all cases.

Figure D-21. Summing Mechanism Installation (Drawing: ACT-LO-FCD102)–Sheet 16 of 16

D.3 SECONDARY ACTUATORS (S250N101)

A survey was conducted to determine if a servoactuator in current use on a Boeing airplane could meet the restrictions of electrical compatibility and physical envelope while providing adequate output torque and angular authority. The unit selected for use as the Test ACT elevator secondary actuator is the 757 rollout guidance servoactuator, part number S251N312-4, primarily because of its output torque, authority limits, and physical envelope.

To create the Test ACT secondary actuator from the 757 rollout guidance servo, the following modifications must be made:

- Replace the existing electrohydraulic servovalve (EHSV, 10-60813-1) with one that will provide sufficient flow to operate the actuator at a rate of 55 deg/sec. The replacement EHSV must have mounting and hydraulic provisions compatible with the existing actuator manifold. Electrical characteristics (excitation voltage, resistance, inductance, connections, etc.) must be the same as those of the existing valve.

- To accommodate the increase in flow rate through the unit, some drill passages in the actuator manifold need to be enlarged and the flow restrictors omitted.

- Replace the existing output shaft with a solid shaft of larger diameter at both the outer seal and outer bearing locations. These changes increase the stiffness of the driving elements, as a means of reducing the compliance and overall hysteresis of the actuator, to meet the requirements of Figure D-4.

- Increase the size of the outer bearing to further reduce actuator compliance.

- Increase the size of the outer shaft seal to accommodate the previously mentioned shaft diameter change.

- Because of 757 space constraints, the actuator installation necessitates a different output lever for each of the four units. The material of the levers will be changed from aluminum to 15-5 PH steel to achieve additional stiffness.

- The effect of the changes to the output shaft and levers will increase the actual stiffness values of the output drive train from 698 kN/m (3990 lbf/in.) to 1136 kN/m (6500 lbf/in.).

- A fifth actuator will be procured for use as a spare unit to support the Test ACT flight test program. Since all four actuators are unique assemblies by virtue of their output lever differences, the existing permanent fasteners used to secure the levers to the output shafts will be replaced with bolts and locking nuts to facilitate adaptation of the spare unit to any installation position.

D.4 AUTOPILOT SERVOACTUATOR (S250N102)

The requirement for a single autopilot channel during mechanical flight will be met by retaining one of the 757 pitch autopilot servos. The following modifications will be made to satisfy additional requirements levied against the mechanical control system:

- The addition of a centering valve to the pitch autopilot servo, sandwiched between the EHSV and the actuator manifold, provides a solenoid-controlled hydraulic bypass of the mod-piston chambers. Figure D-22 shows the autopilot hydraulic schematic and the functional relationship of the centering valve. The valve is in the deenergized or bypass position during fly-by-wire flight, preventing transmission of autopilot position signals, while the actuator is serving as an added detent. During mechanical flight, the valve is in the energized or transmitting position, allowing use of the actuator for conventional autopilot functions.

- To provide a detent force sufficiently high to react the FBW-mode linkage loads, the existing centering springs in the autopilot servoactuator will be replaced with springs and a retainer that result in a nominal breakout force of 452.9N (102 lb) at the mod-piston centerline. This enhances the existing feel unit centering force by the equivalent of 57.7N (13 lb) at the control column.

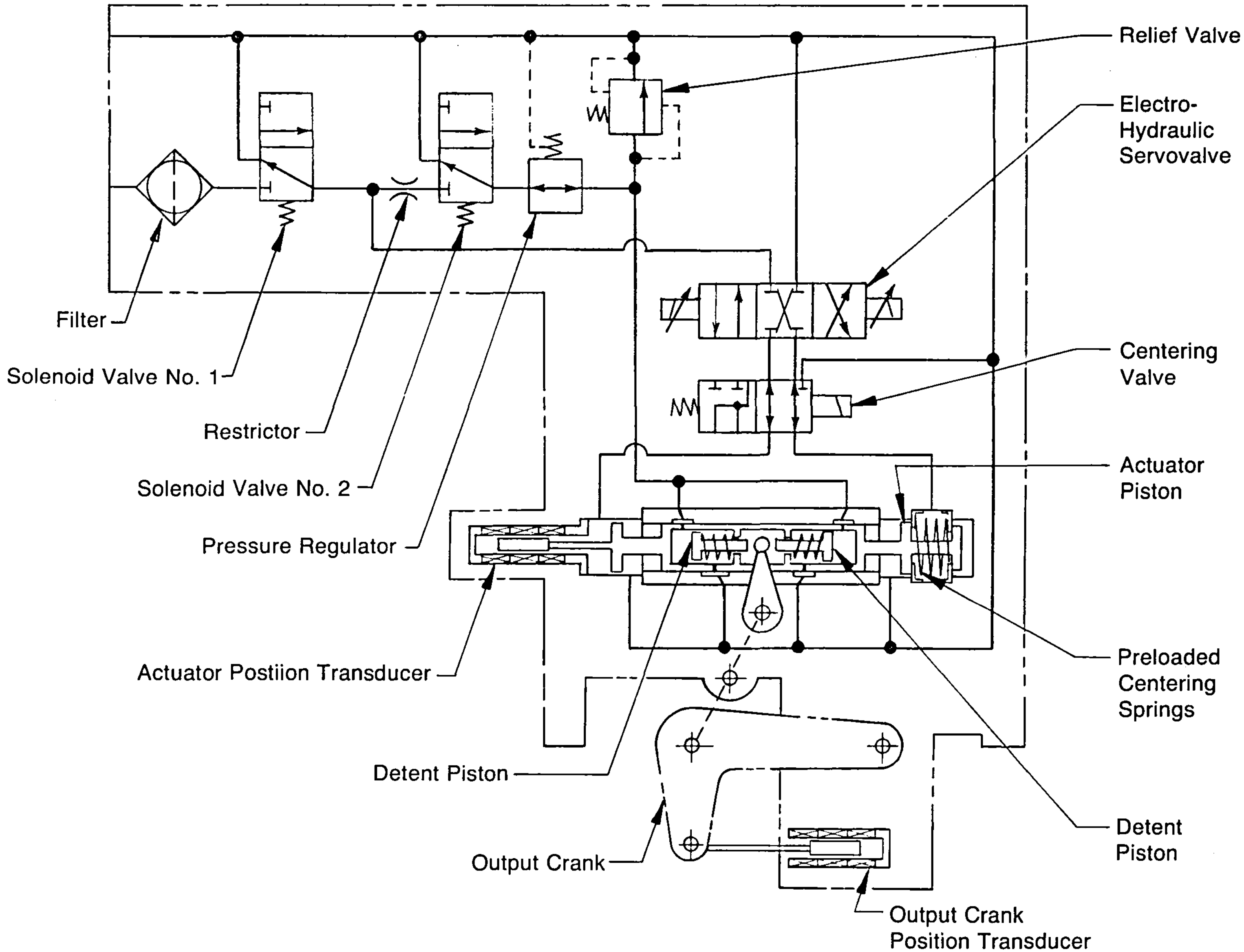


Figure D-22. Autopilot Servo Functional Schematic

D.5 PRIMARY ACTUATORS

Measurements on the 757 Flight Control Test Rig (FCTR) indicate the hysteresis of the elevator mechanical control system may be too high to satisfy the Test ACT requirements during the aft cg flight conditions. These and similar 767 FCTR tests indicate the surface power control unit (PCU) contribution to overall hysteresis is significant and can be reduced by the addition of a bias spring to the PCU control valve spool, as shown in Figure D-23.

To create the Test ACT primary actuators from the existing 757 elevator PCUs, the following modifications are required:

- Add a 2.2-N (0.5-lb) bias compression spring behind the control valve spool
- Add spring retainers at both ends of the bias spring to ensure positive application of load and to prevent generation of contaminants in the valve cavity

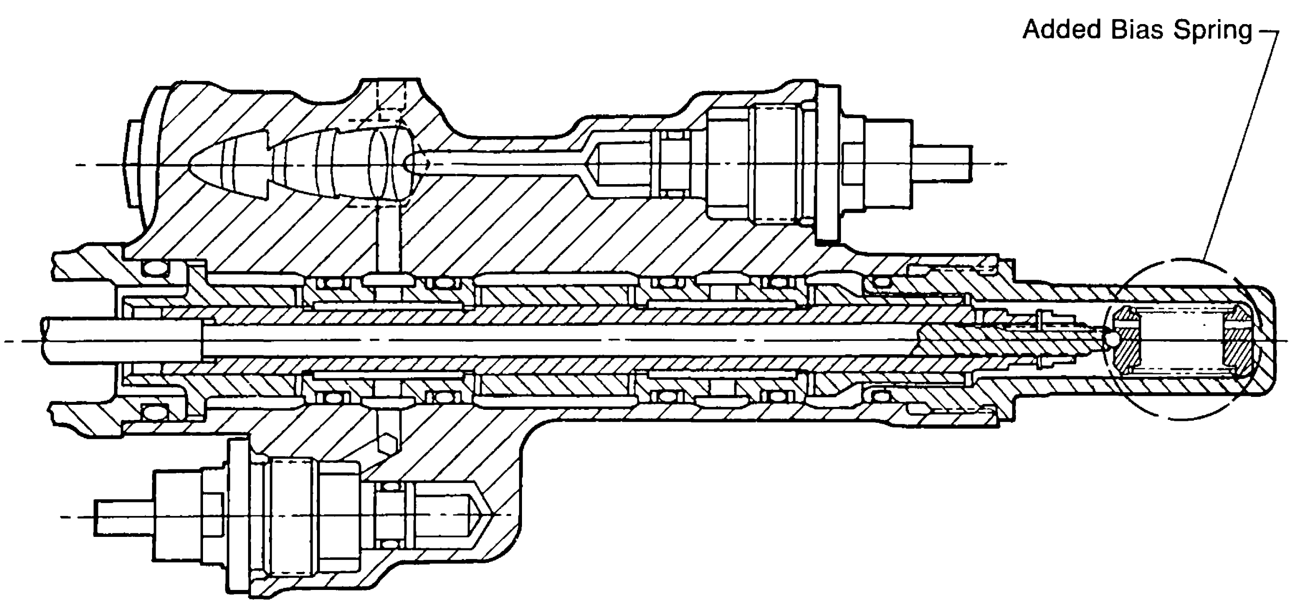


Figure D-23. PCU Bias Spring Installation

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16. Abstract

This report documents the engineering and fabrication of the Test ACT System, produced in the third program element of the IAAC Project. The system incorporates pitch-augmented stability and wing-load alleviation, plus full authority fly-by-wire control of the elevators. The pitch-augmented stability is designed to have reliability sufficient to allow flight with neutral or negative inherent longitudinal stability. With reduction of the risk involved in the design of an active controls airplane as a goal, the system will be extensively tested in the Boeing Digital Avionics Flight Controls Laboratory and then test flown in a Boeing 757-200 airplane.

17. Key Words (Suggested by Author(s))
Energy Efficient Transport, Active Controls Technology, Redundant Automatic Flight Controls, Augmented Stability, Fly by Wire, Wing Load Alleviation

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