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AD 821604

Phase III
Supersonic Transport Development Program

BOEING MODEL 2707

FLIGHT CONTROLS AND HYDRAULICS SUBSYSTEM SPECIFICATION

D6A10120-1

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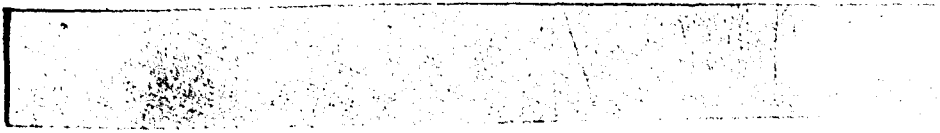
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Prepared for
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THE **BOEING** COMPANY
SUPERSONIC TRANSPORT DIVISION

Faa

ISSUE NO. **13**

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REVISION RECORD

Original release.

June 30, 1966

Document completely revised to reflect the Phase III Proposal Configuration of the 2707.

September 6, 1966

Document extensively revised to (a) reflect the 2707-100 configuration, (b) incorporate revisions requested by the FAA as documented in D6A10490-1, and (c) incorporate and identify contractual performance and compliance test requirements as documented in D6A10494-1.

December 31, 1966

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Transmittal of this document to the President of the U.S. Government must have prior approval of the Director.

*Supersonic Transportation Rev.
Federal Aviation Adm. Wash, D.C. 20590*

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1. SCOPE

This specification defines the objectives, criteria, and configuration and establishes the requirements for performance, design, test, and qualification of the flight controls and hydraulic subsystem for the prototype 2707-100 supersonic transport airplane. Differences between the prototype and production airplanes are described in Supplement I. This subsystem shall provide control of the airplane in response to pilot or guidance-navigation commands and furnish hydraulic power to the landing gear subsystem. This subsystem shall consist of primary flight controls, secondary flight controls, automatic flight controls and hydraulic power.

The statements printed in bold face type in this document are contractual requirements in accordance with D6A10494-1, Prototype Airplane Systems Performance and Compliance Requirements - Phase III Statement of Work. Changes to any of these statements will be contractual changes in accordance with Article XV of the Contract Schedule. With regard to compliance requirements shown in bold face type in Sec. 9.0, it should be noted that only that aspect or portion of the compliance requirement applicable to the specific performance requirement of D6A10494-1 is to be considered actual in the terms stated above.

2. APPLICABLE DOCUMENTS

The following documents, of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between the documents listed herein and the requirements of this specification, the requirements of this specification shall take precedence.

2.1 Specifications.

Military

MIL-E-5272C	20 Jan 60	Environmental Testing, Aeronautical and Associated Equipment, General Specification for
MIL-E-5400H	1 June 65	Electronic Equipment, Aircraft, General Specification for
MIL-F-9490C	9 Mar 66	Flight Control Systems, Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-H-8890	1 Nov 61	Hydraulic Components, Type III (-65° to +450° F), General Specification for

MIL-H-8891	1 Nov 61	Hydraulic Systems, Manned Flight Vehicles, Type III, Design, Installation, and Data Requirements for
MIL-R-5757D	18 Mar 65	Relays, Electrical (Excluding Thermo) for Electronic and Communication-Type Equipment, General Specification for
MIL-R-6106E	15 Oct 64	Relays, Electrical, Aircraft, General Specification for
MIL-R-8931	30 June 64	Reservoirs: Aircraft and Missile, Hydraulic, Separated Type
MIL-G-3786E	27 Jan 65	Switches, Rotary (Circuit Selector, Low Current Capacity), General Specification for
MIL-S-3950B	30 June 65	Switches, Toggle
MIL-S-6743	23 June 65	Switches Assemblies and Actuators; Push Button and Limit
MIL-S-6744	23 June 65	Switch Assemblies and Actuators; Push Button and Limit
MIL-P-19692B	3 July 63	Pumps, Hydraulic, Variable Delivery, General Specification for

Aeronautical

ARINC 404	1 May 56	Cases and Racking, Air Transport Equipment (including supplements 1 and 2)
ARINC 406	12 Oct 59	Standardized Interconnections and Index Pins, Airborne Electronic Equipment
ARINC 409A	1 Nov 62	Selection and Application of Semiconductor Devices
ARINC 558	1 July 64	Automatic Throttle System
AS 402A	1 Feb 59	Aeronautical Standard Automatic Pilot
TSO-C9c	15 Aug 60	Civil Aeronautical Administration Technical Standard Order

The Boeing Company

D6A10180-1	6 Sep 66	Ground Support Equipment Requirements Specification
D6A10089-1	31 Dec 66	Accessory Drive Subsystem Specification
D6A10090-1	31 Dec 66	Aircraft Integrated Data Systems Subsystem Specification
D6A10107-1	31 Dec 66	Airframe Subsystem Specification
D6A10108-1	31 Dec 66	Landing Gear Subsystem Specification
D6A10109-1	31 Dec 66	Flight Deck Subsystem Specification
D6A10116-1	31 Dec 66	Fuel Subsystem Specification
D6A10119-1	31 Dec 66	Electrical Power Subsystem Specification
D6A10122-1	31 Dec 66	Communication/Navigation/Radar Subsystem Specification
BMS 3-10	----	High Temperature Hydraulic Fluid
10-60962 Rev. A	8 July 66	Hydraulic Pump, Engine Driven

2.2 Standards.

Federal

FAR 25 (Changes 1-7)	1 Feb 65	Airworthiness Standards: Transport Category Airplanes
FAR 25 (Tentative Supplement)	1 Nov 65	Tentative Airworthiness Objectives and Standards for Supersonic Transports
A.C. No. 120-20	6 June 66	FAA Advisory Circular - Criteria for Approval of Category II Landing Weather Minimums

Military

MIL-STD-838A	7 Sep 65	Lubrication of Military Equipment
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MIL-W-16878D	5 July 61	Wire, Electrical, Insulated, High Temperature
AND 10375	5 Feb 53	Colors, Fluid Line Identification
<u>The Boeing Company</u>		
BAC T11Y	21 Apr 66	Tape, Hydraulic System, Identification
BAC-C45	----	Boeing Connector Standards

2.3 Other Publications.

The Boeing Company

D6A10064-4	9 Nov 66	Reliability Analysis Document - Automatic Flight Controls
D6A10064-8	10 Aug 66	Reliability Analysis Document - Flight Controls
D6A10064-10	28 Oct 66	Reliability Analysis Document - Hydraulic Power
D10914	7 Dec 65	Finishes and Similar Requirements for Aircraft Equipment
D6-6109	16 July 63	Dielectric Requirements for Aircraft Electric Equipment
D6-16050-1	30 Dec 66	SST Electromagnetic Interference Control Requirements
D6-16328	28 Sep 65	Electrical Requirements for Items of Equipment Installed in Model 733 Airplanes
D6-17467	12 Aug 66	Vibration Test Requirements for Items of Equipment Installed in Model 733 Airplanes
D6A10441-1	4 Nov 66	2707 Flight Control Servo Simulator Detailed Test Plan

3. SUBSYSTEM OBJECTIVES

The Flight Controls and Hydraulics Subsystem design shall meet the following objectives:

- a. Provide a powered flight control system such that the required level of flight crew skills and techniques shall be essentially the same as those required for present day commercial jet transport airplanes.
- b. Provide control capabilities and aerodynamic handling qualities essentially the same as those of present day commercial jet transport airplanes.
- c. Provide systems and components which possess service life, reliability, and ease of maintenance as good as, or better than present day commercial jet transport airplanes.
- d. Provide a design based upon proven concepts and using known materials and processes, to the greatest extent possible for this particular airplane, to achieve a high degree of cost effectiveness during commercial airline operation.
- e. Provide flight deck displays which will enable the pilot to monitor the entire subsystem performance.

4. SUBSYSTEM DESIGN CRITERIA

4.1 Operability. The allocation of the reliability and maintainability defined herein has been accomplished by analysis and experience and may be revised as long as the overall airplane requirements as specified in the Airframe Subsystem Specification D6A10107-1 are satisfied. The operability requirements below are dependent, in part, upon the reliability and maintainability definitions listed in Pars. 6.1.1 and 6.1.2 of D6A10107-1. The values included for flight dispatch delays interact so closely with the unscheduled maintenance task times that a change in either will probably require a change in both.

4.1.1. Reliability. After 18 months of scheduled airline operation, flight turnbacks or deviations due to malfunction of the flight controls and hydraulics subsystem shall not average more than 1.855 per 1,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal supersonic flight of 1.75 hr duration. Dispatch delays caused by malfunction of the flight controls and hydraulics subsystem shall not average more than 0.373 per 100 scheduled departures. Normal maintenance of the system is assumed.

These delays and deviation criteria are based on the major component mean-time-between-failure (MTBF) goals shown in Table I.

Table 1 Major Component Reliability

Major Component	MTBF (Hr)
Master Servo Power Unit	5,000
Elevon Power Control Actuator	16,000
Wing-Sweep Actuators	150,000
Hydraulic Motors	18,500
Hydraulic Pumps	3,500
Electrical Command Computer (Single Axis)	4,000
SAS Computer (Single Axis)	5,375
*Autopilot Computer (Single Axis)	2,830
Rate Gyros	6,000

*Autopilot required for last four minutes of flight for Category III landings.

Individual component reliability required to meet the overall system reliability shall be specified in the individual component procurement specifications.

4.1.2 Maintainability. The flight controls and hydraulics subsystem shall be so designed that after 18 months of scheduled airlines operation, the direct maintenance manhours per 1,000 flt/hr shall not exceed a mean value of 1,440, not including servicing of consumables, and the mean unscheduled maintenance task time at a transit or turnaround service shall be 50 min. These values are based on an average flight length of 1.75 hr with scheduled maintenance accomplished as planned.

4.1.2.1 Maintenance and Repair Cycles. Time change items shall be kept to a minimum. Whenever possible, component replacement shall be on a failure or impending failure (on condition) basis, rather than on a scheduled or time-controlled basis. The schedule check intervals for the aircraft and the down times established for these checks are shown below. All scheduled maintenance, inspections, and servicing shall be fitted within one of these cycles.

<u>Scheduled Check</u>	<u>Time Interval</u>	<u>Elapsed Time</u>
Transit Service	Not applicable	30 min
Turnaround Service	Not applicable	90 min
Daily Check	50 flt-hr	1 hr
Intermediate Check	300 flt-hr	4 hr
Periodic Check	1200 flt-hr	16 hr
Basic Check	8400 flt-hr	5 days

4.1.2.2 Service and Access. The Flight Controls and Hydraulics Subsystem design shall provide the following features to facilitate ease of maintenance:

- a. Subsystem and component assemblies shall be readily accessible to the maintenance technician for the purpose of fault isolation, calibration, adjustment servicing, and replacement. Test and calibration points shall be readily accessible without removal of the component. In all cases, accessibility provisions shall allow for efficient accomplishment of maintenance under the expected climatic extremes to be encountered.
- b. Scheduled lubrication frequencies, if required, shall coincide with the accomplishment of a scheduled maintenance check.
- c. Maintenance shall be accomplished with the skill levels required for maintaining subsonic jet aircraft.
- d. Access to components shall be provided by natural openings whenever possible. Where natural access is not available, suitable access panels shall be provided.
- e. Components shall be packaged so that they may be tested as a unit outside the airplane. Disconnect points shall be located so that control system rigging is not disturbed during replacement.
- f. Special tools shall not be required for normal maintenance and LRU removal tasks.

4.1.3 Useful Life. The Flight Controls and Hydraulics Subsystem shall have a useful life commensurate with that of the airplane (50,000 hrs), assuming normal maintenance of equipment. The Time Before Overhaul (TBO) of individual components shall be specified in the component specifications when applicable.

4.1.4 Environment. The Flight Controls and Hydraulics subsystem shall perform its intended functions under the normal operating environment defined in D6A10107-1, except as follows:

- a. Vibration environment shall be per D6-17467.
- b. Mechanical and hydraulic component environmental temperature extremes resulting from radiant effects of aerodynamic heating and from internally generated heat shall be as indicated on Fig. 39.
- c. The environment for certain automatic flight control system components shall be modified per Par. 4.2.2.

4.1.5 Human Performance. Human performance requirements shall be considered in the design of the Flight Controls and Hydraulics Subsystem and related controls and displays. Specific requirements are as follows:

- a. A fail-safe design shall be provided in those areas where failure could disable the system or cause hazardous condition through damage to equipment, injury to personnel, or inadvertent operation of critical equipment.
- b. The information displayed to the crew shall be limited to that which is necessary for them to perform specific actions or make necessary decisions.
- c. Information shall be presented in a directly usable form. Requirements for decoding, transposing, computing and interpolation shall be kept to a minimum.
- d. All controls, displays, or items of equipment that must be located, identified, read, or manipulated shall be appropriately and clearly labeled to facilitate rapid and accurate human performance.
- e. Controls shall be selected so that the direction of movement of the control will be consistent with the related movement of an associated display, equipment component, or airplane.
- f. Controls shall normally be located adjacent to their associated displays and positioned so that neither the control nor the hand normally used for setting the control will obscure the display.
- g. Controls shall be so designed and located that they are not susceptible to being moved accidentally.
- h. Labels shall be located in a consistent manner throughout the equipment and system.
- i. Labels shall be as concise as possible without distorting the intended meaning or information.
- j. Equipment shall be designed to facilitate rapid and positive fault detection and easy removal, replacement, and repair.
- k. Structural members of the unit or chassis shall not prevent access to components.
- l. Access openings provided for adjusting and handling units or components shall be sufficiently large to permit the required operations and, where possible, provide an adequate view of the components being manipulated.
- m. Units shall be so located and mounted that access to them can be achieved without danger to personnel from electrical charge, heat, or toxic substances.

4.1.6 Safety. The design shall meet the requirements of FAR 25.1309 "Tentative Airworthiness Standards - SST;" and shall assure that no single failure, as determined by a failure mode, effect, and criticality analysis shall result in a hazardous condition. Systems and

components shall be designed to receive minimum damage due to impact of foreign parts from engines or other subsystem failures. In no event shall such damage result in jeopardizing the safe flight of the airplane.

4.1.6.1 Personnel Safety.

- a. All hydraulic lines in the pressurized compartments shall be shrouded and drained overboard to prevent spillage of hot fluid on occupants or cargo.
- b. Interlocks, bleed valves, and self-sealing connectors shall be provided to protect service personnel while performing maintenance on the subsystems.

4.2 Design Criteria.

4.2.1 Primary and Secondary Flight Controls. The flight controls shall be designed to meet the requirements of FAR 25 and MIL-F-9490C, or the intent of these requirements. During the prototype program, continuous liaison with the FAA shall be maintained to develop rule changes to and interpretation of FAR 25 as supplemented by the Tentative Airworthiness Standards for Supersonic Transports, dated November 1, 1965.

4.2.2 Automatic Flight Controls.

- a. The Automatic Flight Controls shall be designed in accordance with FAR 25.
- b. The auto throttle design shall utilize the concepts described in ARINC 558.
- c. A failure detection capability shall be provided for each channel and shall, in the event of failure, automatically remove the defective channel from operation.
- d. The operational status of each channel shall be displayed on the AFCS control panel with the exception of the angle-of-attack warning and control, auto trim, and direct lift control functions.
- e. A self-test capability shall be provided for each channel such that the cause of a malfunction can be isolated down to the line replaceable unit (LRU). While self-testing is being conducted, visual indication shall be provided.
- f. Manually operated engage/disengage switches shall be provided for each channel with the exception of the functions of angle-of-attack warning and control, direct lift control, and auto trim.

- g. Computers shall be designed to ARINC 404 with Camloc type or equivalent handles.
- h. Channel isolation shall be provided.
- i. Power and warmup interlocks shall be provided for AFCS and E/CS.
- j. Materials, processes, workmanship, design, and construction shall conform to Class 3 of MIL-E-5400H as applicable.
- k. Nonmagnetic materials shall be used where possible.
- l. All internal wiring shall be suitably identified and shall be in accordance with MIL-W-16878D.
- m. Card wiring shall be on one side with components on the other side to the greatest extent possible.
- n. Electronic components shall comply with MIL-E-5400H, where applicable. Transistors shall meet ARINC 409A. Components shall be derated. Heat sinks shall be employed where required.
- o. Connectors where not covered by ARINC 404 shall be in accordance with BAC C45. Standardized indexing per ARINC 406 shall be used where applicable.
- p. Where use of solid state switching devices is impractical, the switching devices shall conform with MIL-R-6106E, MIL-R-5757D, MIL-S-6743, MIL-S-6744, MIL-S-3950A, and MIL-S-3786A, as applicable.
- q. The AS402A and TSO-C9c requirements, as applicable to the SST, shall be met.
- r. The electronic computers, control wheel transducers, acceleration sensors, control panels and throttle servos shall be capable of withstanding the environment specified in Par. 4.1.4, with the following exceptions:
 - (1) Temperature
 - (a) Operating: -65° to +180°F, with continuous operation at +180°F limited to 30 min.
 - (b) Shock: -65° to +225°F.
 - (c) Storage: -65° to +160°F.
 - (2) Humidity: 0 to 100 percent relative humidity including condensation on equipment.
 - (3) Shock: Per Par. 3.2.2.1.6.1 of MIL-E-5400H.

- s. Equipment shall utilize power with characteristics as defined by D6-16328
- t. Equipment shall meet D6-6109 dielectric requirements.
- u. Lard mode of the autopilot will comply with FAA Advisory Circular - AC No. 120-20.

4.2.3 Hydraulic Power System. The hydraulic power system shall be designed in accordance with FAR 25. MIL-H-8890 and MIL-H-8891 shall be used as design guide criteria.

4.2.3.1 Component Design Strength. The system shall be designed to withstand the proof-and-burst pressure requirements of Table I, MIL-H-8891, throughout the ambient and fluid temperature range, after loss of strength of materials caused by aging at elevated temperatures for the life of the systems or component. Design factors and tubing support shall be adequate to compensate for the fatigue and dynamic effects of steady-state pressure pulsations. Parts of the system subject to atmospheric pressure or suction pressure shall be designed to withstand an external pressure of 15 psi.

4.2.3.2 Rigid Tubing. Rigid tubing shall be fabricated of high-strength, corrosion-resistant metal. Permanent type fittings shall be used wherever possible to assure minimum weight and minimum points of potential leakage and maintenance. Lines shall, where possible, be permanently attached to, and terminated in, manifolds.

4.2.3.3 Flexible Lines. Line flexibility shall be provided by tubing coils, swivel joints or hoses, in this order of preference. Swivel joints or hoses shall be used only where dictated by envelope and motion requirements.

4.2.3.4 Straight Tube Lines. The use of straight tube lines installed between two rigid connections shall be avoided wherever possible. Where such straight lines are necessary, provisions shall be made in the mounting of the units or in the rigid connections to insure that no excessive strains will be applied to the tubing and fittings. Semi-loops, offsets, and other devices shall be provided in the tubing, as necessary, to insure proper alignment on installation, and to take care of vibration and thermal expansion.

4.2.3.5 Tube Supports. Tube supports shall be designed and spaced such that they will transmit no damaging loads to the hydraulic tubing or support structure and shall not cause galling of the tube support, which would cause reduced life.

4.2.3.6 Linear Seals Continuously pressurized high-pressure actuator rod seals shall be two-stage with bleed-off between stages to return. The design goal for actuator rod seals shall be no external leakage

other than a slight wetting of the rod. Design of first-stage seals and the leakage allowance in the reservoir design shall take into consideration the possibility of failure of the second stage which would cause fluid depletion. Seal standardization shall be a design goal. Endurance requirements for the static and dynamic seals used on flight control and engine inlet control linear actuators are 12,000,000 short stroke cycles (0.10 to 0.15 in.) and 200,000 long stroke cycles (3.0 to 3.5 in.). Endurance requirements for the static and dynamic seals used on the landing gear linear actuators are 50,000 cycles.

4.2.3.7 Rotating Shaft Seals. Leakage past rotating shaft seals on pumps and motors shall be vented overboard. The maximum rate shall be 5 cc/hr per seal.

4.2.3.8 Swivel Joints and Oscillating Seals. The design goal shall be to obtain no external leakage. If this goal cannot be achieved, provisions shall be incorporated to vent this leakage overboard, and the leakage rate shall not exceed 5 cc/hr per swivel.

4.2.3.9 Component Attachment. Self-sealing connections shall be provided, where practical, to prevent fluid loss or entry of contamination during component removal for maintenance.

4.2.3.10 Static Seals. All static seals shall have an endurance life equal to, or greater than, the life of the dynamic seals in the component.

5. SUBSYSTEM DESIGN REQUIREMENTS

5.1 Primary Flight Controls Requirements. The primary flight control surfaces shall consist of elevators and elevons for longitudinal control, spoilers for direct lift control during landing, spoilers and ailerons for low-speed lateral control, elevons and spoilers for high-speed lateral control, a rudder for directional control and variable sweep wings for optimum lift/drag characteristics.

The normal mode of operations for longitudinal and lateral control shall be through an Electric Command System (E/CS) which controls the mechanical output of the master servos. The master servos outputs shall operate the control surface hydraulic actuator valves through mechanical cable systems. Mechanical cable systems between the control columns and wheels and the master servos shall provide manual means of controlling the master servos with the electrical modes disengaged. This cable system will also back drive the control columns and wheels during E/CS operation.

The normal mode of operation of the direct lift controls shall be through the E/CS, autopilot system (A/P), and stability augmentation system (SAS). Pitch axis signals from these systems shall control an hydraulic power direct lift servo in each wing. These servos shall control the inboard spoiler hydraulic actuators through mechanical cables and linkages.

The directional control system shall be manually operated by cables to the rudder hydraulic actuator valves.

All primary control surface travels, rates, and power requirements shall be as shown in Table II.

5.1.1 Longitudinal Control. The longitudinal controls, under the most adverse loading conditions, shall provide a maneuver capability of at least plus 1.0 and minus 1.0 incremental g from trimmed 1g flight.

Low-speed control power shall be adequate to perform the landing flare and to permit a takeoff rotation capability which does not restrict airplane takeoff performance.

Table II Primary Flight Controls, Travels, Rates, and Power

Wing Sweep	Flap	Longitudinal Control				Speed Brakes	Lateral Control			Directional Control
		Primary Elev.	Aux. Elev.	Elevons	Spoilers (Direct Lift)		Spoilers	Ailerons	Elevons	
20°	Landing	±30°	0-30°	±30°	0-7 1/2° (8 Inboard Panels)	0-45° (All Panels)	0-45°	±25°	0	±25°
30°	T.O.	±30°	0-30°	±30°	0	0-45° (All Panels)	0-45°	±25°	0	±25°
30°	Faired	±30°	0	±30°	0	0-45° (All Panels)	0-45°	0	±10°	±12°
42°	Faired	±30°	0	±30°	0	0-45° (All Panels)	0-45°	0	±14°	±12°
72°	Faired	±30°	0	±25°	0	0-45° (8 Inboard Panels)	0-45° (8 Inboard Panels)	0	±20°	±8°
Maximum Surface Rate Deg/Sec	1.5	2	25	24	60	60	60	50	25	25
Design Hinge Mom (In. Lb.)	X	3,000,000	514,000	2,500,000	120,000	980,000	490,000	340,000	2,500,000	1,250,000
Control Surface Actuators	Total No.	18	10	12	36	36	36	6	6	3
	Bore (In)	4.60	3.14	7	2.5	2.5	2.5	3.5	7.225	6.88
	Stroke (In)	6.25	3.1	7	5.42	5.42	5.42	6.9	18.0	11.0
Flow/Hyd Syst. GPM		121.5	17.6	105	110.4	110.4	55.2	27.0	115.2	50.5

- 1 Sculling hinge moment with one hydraulic system
 2 Total for inboard and outboard auxiliary elevators
 3 Total for 12 spoilers
 4 Total for 6 spoilers
 5 Each actuator has 2 cylinders and pistons
 6 Total for 8 inboard spoilers
 7 6 toggle actuators, bore 4.68, stroke 5.0
 8 6 crank actuators, bore 6.12, stroke 13.3
 A Transient Flow for .658 Second max

5.1.2 Lateral Control. Lateral controls shall provide a minimum steady-roll rate of at least 25 deg/sec and a bank angle of 7 deg in 1 sec for normal takeoff and landing configurations. For operation other than takeoff and landing, roll rates of at least 15 deg/sec and a bank angle of 3 deg in 1 sec shall be attainable. Sufficient lateral control shall be provided to assure safe completion of wings-aft landings.

5.1.3 Electric Command. The Electric Command System (E/CS) shall provide feel characteristics for the pitch and roll axes to minimize the adverse effects of control input friction, back lash and resultant hysteresis. The E/CS shall be operable throughout the full range of pilot authority.

The pitch and roll axes control forces shall be capable of being preset within the ranges shown in Table III during prototype airplane development.

Table III E/CS Control Forces

E/CS Control Force		
Control Axis	Breakout	Maximum
Longitudinal	1 to 6 lb	30 to 80 lb/g
Lateral	0.5 to 3 lb	15 to 25 lb

The transients associated with engagement and disengagement of the E/CS shall not exceed 0.04 g in pitch and a roll rate of 2 deg/sec in roll.

The E/CS shall automatically engage after warmup, when power is on, and when two or more channels are active. The operational status of each channel of each axis shall be displayed on the flight deck.

5.1.4 Directional Control. The directional controls shall provide sufficient control capability: (1) for safe landings in 90 deg crosswinds up to 30 kn, (2) to maintain airplane control with failure of two engines on one side in the enroute configuration, and (3) to maintain airplane control with failure of the most critical single engine at a minimum control speed (V_{mc}) consistent with the V_1 speed required for takeoff performance. Directional control authority shall be limited, dependent on wing sweep and flap positions to provide structural protection.

5.1.5 Wing Sweep. The wing-sweep controls shall vary the sweep of both wings symmetrically. The wing-sweep controls shall move the wings from the takeoff position to the 72 deg full sweep angle in 60 ± 5 sec, from the 72 deg to the takeoff position in 120 ± 10 sec, and from the takeoff to landing position in 30 ± 3 sec. Wing-sweep actuation shall be sequenced with high lift control actuation. Automatic interlocking shall prevent asymmetric operation and prevent operation of the wings and high lift devices to incompatible positions.

5.1.6 Control Friction and Breakout Force. Longitudinal, lateral and directional controls shall exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the degree of centering shall be such that the combined effects of centering, breakout force, stability, and force gradient do not produce objectionable flight characteristics, nor permit large departures from trim conditions with controls free. The control forces for the longitudinal and lateral systems, during normal operation through E/CS, are given in Table III. In the event of complete electrical failure of E/CS the control forces through the mechanical system shall not exceed the values given in Table IV. Since the directional system does not use the E/CS, the control forces shall not exceed the values given.

Table IV Mechanical Control Mode Forces

Control Forces		
Control Axis	Breakout	Maximum
Longitudinal	12 lb	75 lb
Lateral	6 lb	50 lb
Directional	15 lb	100 lb

5.1.7 Rate of Control Displacement. The ability of the aircraft to perform maneuvers shall not be limited by the rates of control surface deflection, nor shall the rates of operation of either primary controls or auxiliary devices result in objectionable flight characteristics.

5.1.8 Control System Characteristics. Control response to cockpit control deflection or control force shall be smooth and approximately linear for all amplitudes of control input. The phase lag between the cockpit control deflection or force and the control surface deflection shall be small at all amplitudes and frequencies of control motion normally used by the pilot in controlling the airplane.

5.1.9 Ground Gust Protection. Ground gust protection shall be provided for all surfaces, for gust loads up to 70 mph, by restricting fluid flow through surface actuators.

5.1.10 Control Input Loads. The individual control modes shall be designed to withstand the loads specified in FAR 25, Pars. 25.395b, 25.397, 25.399, and 25.405.

5.1.11 Direct Lift Control. The direct lift control system shall provide precise flight path adjustment during landing approach. An approximate 0.1g incremental maneuver capability shall be provided by this system.

5.2 Secondary Flight Controls Requirements. The Secondary Flight Controls shall consist of wing trailing edge flaps and leading edge slats, the use of primary control surfaces for longitudinal, lateral and directional trim, and the use of wing spoilers as speed brakes.

5.2.1 Longitudinal Trim. Longitudinal trim shall be accomplished through electrical and mechanical control which recenters the artificial feel spring. The longitudinal trim shall have a travel authority less than that of the longitudinal control authority. In the event of an undesirable electrical trim input, the mechanical trim control shall be capable of overriding the electrical input. Longitudinal trim signals shall also move the neutral position of the control column.

5.2.2 Directional Trim. Directional trim shall be accomplished through electrical control which recenters the artificial feel spring of the rudder control. Directional trim shall have a maximum authority of ± 12 deg of rudder travel. Directional trim signals shall also move the neutral position of the control pedals.

5.2.3 Lateral Trim. Lateral trim shall be accomplished through electrical control which recenters the artificial feel spring. Lateral trim shall have a travel authority less than that of lateral control. Lateral trim signals shall also move the neutral position of the control wheel.

5.2.4 Trim Rate. Trim devices shall operate rapidly enough to maintain trim, under changing flight conditions and configurations encountered in operational usage, but shall not be excessively sensitive.

5.2.5 High Lift. The high-lift controls shall control operation of wing trailing-edge flaps and leading-edge slats. Flaps and slats shall operate at approximately 1-1/2 deg/sec. Positive and automatic interlocking shall prevent operation of slats and flaps to positions incompatible with each other and with wing sweep. The system shall prevent asymmetrical positioning in the event of any single failure.

5.2.6 Speed Brakes. Speed brakes shall be provided for operation in the air and also during landing rollout. Speed brake control shall be accomplished manually.

5.3 Automatic Flight Control System (AFCS) Requirements. The AFCS shall consist of the autopilot, auto throttle, stability augmentation system and angle of attack warning and control system.

5.3.1 Autopilot.

5.3.1.1 Engagement/Disengagement. It shall be possible to engage the autopilot, hereafter known as the A/P, at any altitude and speed within the normal flight envelope of the airplane subject to the limits delineated in this specification.

5.3.1.2 Pitch Operating Modes. The A/P shall provide the following operating pitch modes:

- a. Vertical Speed
- b. Altitude Hold
- c. Control Wheel Steering
- d. Altitude Capture
- e. Airspeed Hold
- f. Mach-Altitude
- g. Glide Slope
- h. Land
- i. Go-Around/Takeoff

5.3.1.2.1 Vertical Speed Mode. In the steady state condition, the selected vertical speed shall be held to within ± 120 ft/min. The selected vertical speed shall be established with less than a 20 percent overshoot, and shall be adjustable over a range of $\pm 8,000$ ft/min.

A change in selected vertical speed shall cause the normal acceleration to change less than ± 0.15 g incremental.

5.3.1.2.2 Altitude Hold Mode. This mode shall hold the airplane on the altitude existing at mode engagement. When this mode is engaged during a climb or descent at an altitude rate of up to 2,000 ft/min, the transient error shall not exceed 200 feet of barometric altitude. Altitude hold error shall be as shown in Table V.

Table V Autopilot Altitude Hold Tolerances

Altitude (x 1,000 feet)	Altitude Hold Tolerance		
	Bank Angle 0 Deg	Bank Angle 0 to ± 30 Deg	Bank Angle ± 30 to ± 40 Deg
0 to 30	± 45 ft	± 100 ft	± 120 ft
30 to 55	$\pm 0.15\%$	$\pm 0.4\%$	-
55 to 80	$\pm 0.2\%$	$\pm 0.4\%$	-

5.3.1.2.3 Control Wheel Steering Mode. This mode shall maintain the airplane pitch attitude to ± 0.5 deg. Control column force gradient shall be as stated in Table III.

5.3.1.2.4 Altitude Capture Mode. This mode shall control the airplane to capture a selected altitude and shall provide automatic engagement of altitude hold mode. The selected altitude range shall be from 1,000 to 80,000 ft. The capture maneuver shall not cause more than 0.15 g incremental load factor for altitude capture at an initial altitude rate of less than 2,000 ft/min. Steady-state tolerance shall be the same as for the altitude hold mode.

5.3.1.2.5 Airspeed Hold Mode This mode shall hold the airplane to the calibrated airspeed existing at mode engagement. The steady-state error shall be less than ± 3 kn, or ± 1 percent, whichever is greater.

The airspeed shall be held within ± 7.0 kn, or ± 2 percent, whichever is greater, during maneuver, gusty flight conditions, or changes in throttle setting.

5.3.1.2.6 Mach-Altitude Mode. This mode shall control the airplane to a computed mach-altitude schedule to provide a selected sonic boom overpressure control. The overpressure range shall be adjustable from a ΔP of 1.3 psf to 3.5 psf. When the airplane is subjected to a standard atmosphere and with the maximum forward acceleration, the altitude deviation from the selected trajectory shall be equal to or less than the values specified for the altitude hold mode. A change in the Mach-altitude trajectory while the mode is engaged shall not cause incremental load factor values to exceed $\pm 0.15g$. The new schedule shall be established without overshoot.

5.3.1.2.7 Glide Slope Mode. Operation shall provide for automatic capture from above or below the glide slope beam. At a beam intercept altitude between 1,000 and 3,000 ft and at a ground speed of 160 ± 5 knots, beam capture shall occur within 20 sec with less than 25 μ a overshoot. With a calibrated airspeed of 1.3 to 1.4 stall speed (V_S) and head wind shear not to exceed 4 knots per 100 ft, beam track error from beam capture down to an altitude of 75 ft shall not exceed $\pm 35 \mu$ a or ± 12 ft, whichever is larger.

5.3.1.2.8 Land Mode. This mode shall be automatically engaged at a predetermined radio altitude when the A/P engage switch is in the land position. Touchdown shall be at 1750 ± 500 feet from the approach end of the runway and touchdown vertical velocity shall be less than 3 ± 1 ft/sec. Limitations required for meeting the above requirements are head wind shear of less than 4 knots per 100 ft, runway threshold speed less than 10 kn above runway threshold reference speed and vertical gusts not to exceed 2 kn for a period of 4 sec.

5.3.1.2.9 Go-Around/Takeoff Mode. The airplane shall be controlled to the computed angle of attack reference within ± 1 deg. A go-around

maneuver shall be accomplished with an altitude loss not to exceed 45 ft with an initial flight path angle of -2.75 ± 0.5 deg, speed within ± 15 knots of the land threshold reference, and with immediate increase of engine thrust to maximum continuous. Go-around/Takeoff information shall be displayed on the flight director regardless of A/P engagement.

5.3.1.3 Roll Operating Modes.

The A/P shall provide the following operating roll modes:

- a. Roll Attitude/Heading Hold
- b. Heading Select
- c. Control Wheel Steering
- d. Localizer
- e. VOR
- f. Inertial Navigation
- g. Go-Around/Takeoff

5.3.1.3.1 Roll Attitude/Heading Hold Mode. Bank angles of ± 33 deg shall be permitted with turn knob commands. Maximum bank angle tolerance shall be ± 3 deg. When the turn knob is centered to a wings-level command and the airplane attitude is less than ± 2 deg, the heading hold mode shall be engaged. Indicated airplane heading shall be controlled to ± 1 deg.

With a step input command, the new roll attitude shall be established with an overshoot of less than 10 percent of the step input. Following an impulse type of disturbance or sideslip, the heading shall return to the reference heading with an overshoot not to exceed 20 percent of the maximum transient disturbance.

5.3.1.3.2 Heading Select Mode. Changes in airplane heading of ± 180 deg shall be selectable before or after mode engagement. The selected heading shall be captured with less than 3 deg overshoot and shall be tracked with an accuracy of ± 1 deg. In response to a step change in desired heading of 90 deg, the bank angle limit shall be reached in 8 ± 1 sec.

5.3.1.3.3 Control Wheel Steering Mode. Control wheel steering mode shall maintain the roll attitude of ± 0.5 deg. Maximum control wheel force shall be adjustable as stated in Table III.

5.3.1.3.4 Localizer Mode. Automatic capture of a localizer beam shall be permitted in either Heading Hold Mode or Heading Select Mode. At a beam intercept range of 12 ± 2 miles from the localizer transmitter, localizer beam capture shall occur with an overshoot less than 50 μ a for intercept angles up to 45 deg. Beam track error shall not exceed ± 30 μ a from the outer marker to the middle marker and shall not exceed ± 20 μ a from the middle marker to 1 mile from the localizer transmitter. Localizer mode performance shall be based on the following limitations.

- a. Runway length: 11,000 ft
- b. Transmitter location: 1,000 ft beyond far end of the runway
- c. Approach airspeed: 1.3 to 1.4 V_s
- d. Crosswind shear: not to exceed 4 kn per 100 ft

5.3.1.3.5 VOR Mode. Automatic capture of a VOR radial shall be permitted in heading hold mode or heading select mode. At intercept range of 100 $\pm$ 25 miles VOR capture shall occur with less than 1 deg overshoot for intercept angles up to 45 deg. Beam track error shall not exceed $\pm$ 15 μ a. Over-the-station passage shall be provided with roll attitude not to exceed $\pm$ 5 deg. VOR course adjustment of $\pm$ 30 deg shall be permitted while over the station.

5.3.1.3.6 Inertial Navigation Mode. Automatic capture of an inertial cross track shall be permitted in heading hold mode or heading select mode. At intercept angles up to 45 deg, cross track capture shall occur with an overshoot of less than 2 miles. During inertial cross track control, cross track error shall not exceed $\pm$ 1 mile of the indicated inertial cross track.

5.3.1.3. Go-Around/Take-Off Mode. This mode shall command a wings-level attitude when engaged. After establishment of a wings-level attitude, attitude error shall not exceed $\pm$ 1 deg. Upon engagement of this mode, the wings-level attitude shall be attained with no overshoot.

5.3.1.4 A/P Attitude Command Limits.

- a. Pitch attitude commands shall not exceed $\pm$ 30 deg.
- b. Roll attitude commands shall not exceed $\pm$ 30 deg.

5.3.1.5 A/P Authority Limits. When the A/P is engaged in single channel operation, authority limiting shall be provided to limit the incremental load factor to $\pm$ 1.0 g and the roll attitude from exceeding $\pm$ 60 deg in 3 sec.

5.3.1.6 Flight Director Display Commands. The A/P computer shall provide pitch and roll command signals to the flight director displays. During automatic navigation with the A/P, the flight director system shall display the A/P command signals.

5.3.1.7 Automatic Pitch Trim. When the A/P is engaged, automatic pitch trim shall be provided to reduce the steady state A/P command signal. Pitch transient during an A/P disengagement shall not exceed an incremental load factor of $\pm$ 0.15 g.

5.3.1.8 All Weather Landing. The A/P system shall provide for automatic control of approach in weather minimums down to 700 ft RVR.

5.3.1.9 Operational Status. The operational status of each channel of each axis of the A/P shall be displayed in the flight crew compartment. Provisions for manually operating the A/P in single channel operation shall also be provided in the flight crew compartment.

5.3.2 Auto Throttle (A/T). The A/T shall automatically vary thrust as required to hold a selected speed.

5.3.2.1 Engagement/Disengagement. The A/T operation shall be independent of E/CS, and SAS and shall be interlocked with the A/P to prevent engagement of incompatible modes. When engaged during landing it shall provide automatic throttle operation during flare.

5.3.2.2 Operating Modes. The A/T shall provide the following operating modes: (1) Air-speed select mode (including the automatic programming during flare), and (2) mach hold control mode (including Mach adjust).

a. Air-Speed Select Mode. The air-speed select mode shall capture and hold the selected air speed. The desired speed shall be selectable prior to or after mode engagement. This air speed shall be held within prescribed tolerances of commanded air speed during any condition or configuration normally experienced in flight.

(1) Static Accuracy. Static speed accuracy shall be ± 3.0 kn or ± 1 percent of command value, whichever is greater.

(2) Transient Limits. Throttle activity shall be limited by limiting rate of change of air speed to 5 kn/sec. The air speed shall be held to within ± 7.0 knots, or ± 2 percent, whichever is greater, during maneuver, gusty flight conditions, or changes in throttle setting. When stabilized on the guide slope with head wind shear of 4 kn per 100 ft of altitude, the speed error shall not exceed 3 kn.

(3) Control Inputs. The reference air speed shall be selected on the mode selector control panel.

b. Mach Hold Control Mode. The Mach hold control mode shall hold the airplane to the Mach number existing at the time of engagement. Provisions shall be incorporated to enable the reference Mach number to be changed over a range of ± 0.4 Mach.

(1) Static Accuracy. Static control accuracy shall be ± 0.01 Mach, or ± 1 percent, whichever is greater.

(2) Transient Limits. In response to a Mach adjust input, the Mach number shall be obtained without overshoot.

- (3) Control Inputs. The mode selector control panel shall provide for Mach adjustment of ± 0.4 Mach with tolerance of ± 0.1 Mach.
- (4) Drive Capability. The throttle drive servo shall be capable of moving the throttle from idle to 90 percent of full augmented (after-burning) thrust. The throttles shall move at the rate of 10 deg/sec. Under manual control mode the transition from normal thrust to after burning is indicated to the pilot by a detent. Under A/T control mode the detent is removed when the A/T is engaged. This feature permits the A/T to operate smoothly when the control is moved into or out of after-burning. Limit stops shall be provided. Upon reaching the forward position limit, the system shall prevent further increase in thrust. Any aft command signals shall cause normal aft motion. Upon reaching the aft position limit, the system shall prevent further aft motion of the levers. Any forward motion command signals shall cause normal forward motion.

5.3.2.3 No-Back Clutches. The A/T shall drive the throttle through "no-back" clutches to allow individual throttle operation without back driving A/T servos.

5.3.2.4 External Disturbances. The A/T shall be mechanized so that its disengagement is not caused by rough air or by extension of flaps, gear, air brakes or wing sweep.

5.3.2.5 Interlocks. Interlocks shall prevent engagement of the A/T when the A/P air-speed mode is engaged.

5.3.2.6 Disconnects. A/T disconnect switches shall be provided on the throttles.

5.3.2.7 Disconnect Indicator. A/T disconnect shall be displayed on pilot's panel.

5.3.3. Stability Augmentation System (SAS).

5.3.3.1 General. The SAS shall provide satisfactory handling qualities in all conditions. The performance requirements shall be to:

- a. Improve the damping of the Dutch roll, short period and phugoid modes.
- b. Improve the maneuver stability.
- c. Operate with the E/CS such that acceptable stick force/g characteristics are obtained with a simple E/CS feel system.

- d. Shape and coordinate the response to pilot control inputs so that the augmented airplane shall have desirable response characteristics which are similar in all flight conditions.
- e. Operate satisfactorily in the event that the E/CS is disengaged.
- f. Provide positive speed stability for the landing approach and low speed conditions.
- g. Provide structural loads alleviation and improve passenger and crew comfort by damping the rigid airplane modes.
- h. Provide automatic trim capability.
- i. Provide inner-loop damping and trim coordination for the autopilot and be compatible with all A/P functions.
- j. Provide pitch-up protection in the event of failure of the stick pusher system.
- k. Alleviate the transient motion following propulsion system failures, and make it easier for the pilot to maintain path control.

5.3.3.2 Engagement/Disengagement. The SAS shall be automatically engaged, after warmup, when power is on the SAS busses, and a minimum of two channels per axis are operational. A master SAS disengagement switch for disconnecting all channels of each SAS axis is provided.

5.3.3.3 Performance Characteristics. Performance characteristics of the SAS specified below shall apply for all operational flight conditions. Normal operations of the SAS shall introduce no objectionable flight or ground handling characteristics, and in no instance shall it cause airplane performance to be degraded below that for the unaugmented configuration. The SAS performance shall be compatible with the operation of the E/CS, A/P, A/T, primary cable system and engagement of an axis, or axes, of the SAS. Individual-axis or two-axis SAS operation shall be equivalent to the operation with all axes engaged.

- a. Amplitude Frequency Response. It shall be a performance goal that the amplitude frequency response of the transfer functions indicated in Figs. 1 through 3 lie within the specified bands at all frequencies from $\omega = 0.02$ rad/sec to $\omega = 10$ rad/sec.
- b. Minimum Phase. Within the frequency range specified, the pitch rate, roll rate, and yaw rate transfer functions shall have a phase-frequency response which is within 10 deg at any frequency of the minimum phase physically realizable for the corresponding amplitude-frequency response.

- c. Gusts. Any air frame oscillations due to gust increments in angle of attack or sideslip shall damp to 0.25 of the initial amplitude within one cycle.
- d. Envelope Application. The frequency response requirements specified in Par. 5.3.3.3a shall apply to all level flight conditions and steady maneuver conditions within the operational flight envelope.
- e. Residual Oscillation. When the SAS is operating, the residual oscillations measured on the flight deck during steady flight shall not produce normal acceleration (A_n), lateral accelerations (A_y), or attitude amplitudes pitch (θ), yaw (ψ) and roll (ϕ) greater than the following:

A_n : ± 0.02 g	θ : ± 0.1 deg	ϕ : ± 0.1 deg
A_y : ± 0.01 g	ψ : ± 0.15 deg	

Elimination of residual oscillations shall be a performance goal.

- f. SAS Authority. Total SAS authority in the pitch axis shall be limited to ± 1.0 g incremental after application of the maximum pitch SAS control input. Total SAS authority in the roll axis shall be limited to ± 15 deg/sec roll rate after application of the maximum roll SAS control input. Total SAS authority in the yaw axis shall be limited to ± 0.1 g of lateral acceleration after application of the maximum yaw SAS control input.

5.3.3.4 Design Characteristics.

- a. Master override switch shall be provided for disconnect of all channels of each axis.
- b. The pitch and yaw axes of SAS shall be fail-operational after two failures and fail-passive upon the third failure.
- c. The roll axis of SAS shall be fail-operational after one failure and fail-passive upon the second failure.
- d. The malfunction detection circuitry shall be capable of detecting servo actuator control loop and SAS electronic failures.

5.3.4 Angle-of-Attack Warning and Control System.

- 5.3.4.1 General. The angle-of-attack warning control system shall provide indication that the airplane is approaching a condition of excessive angle of attack by shaking the control column. The stick

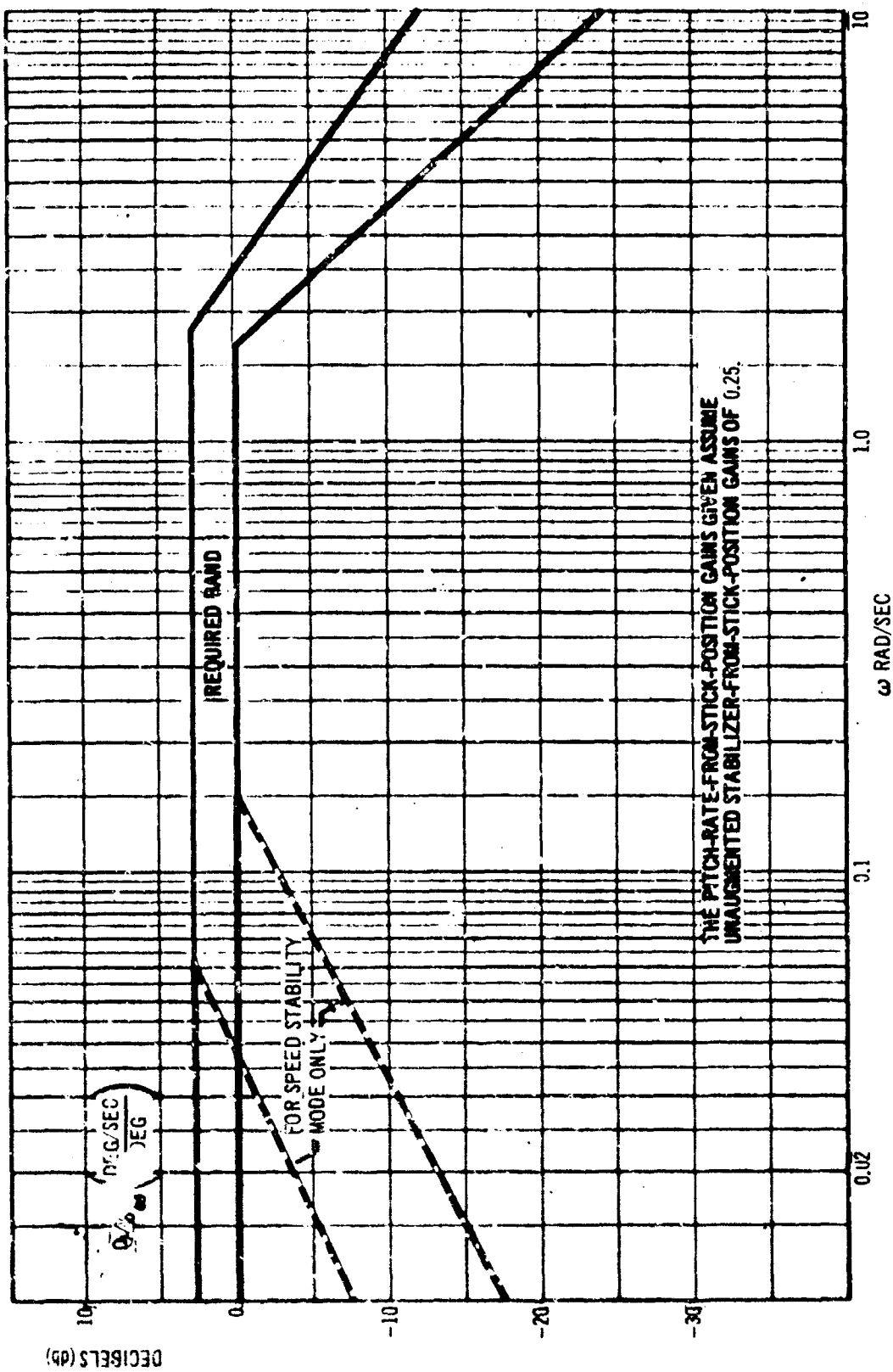


Figure 1. Frequency Response Band for Pitch Rate From Stick Position

GRAPH 20-1

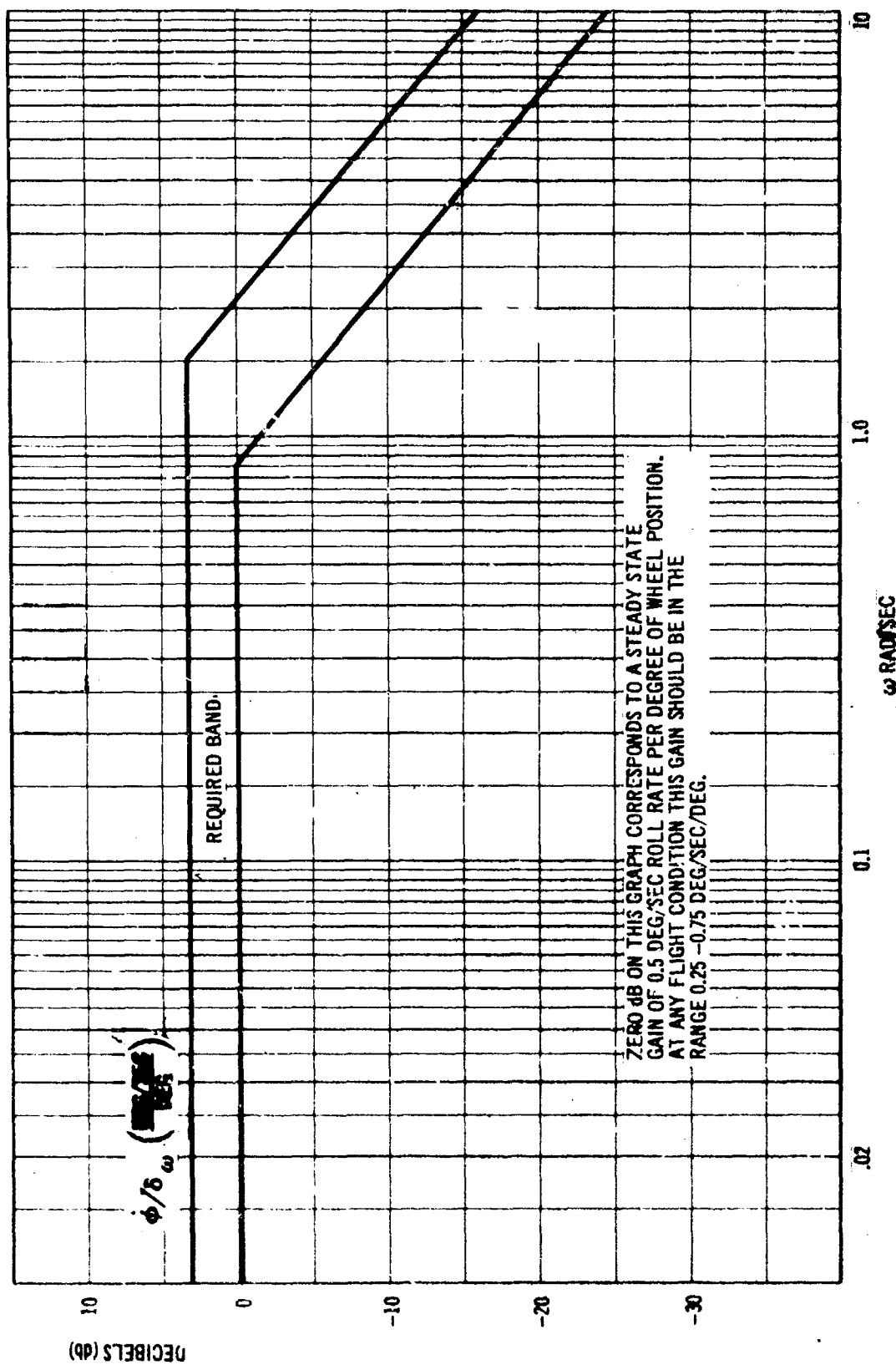


Figure 2. Frequency Response Band for Roll Rate From Pilot Wheel Position

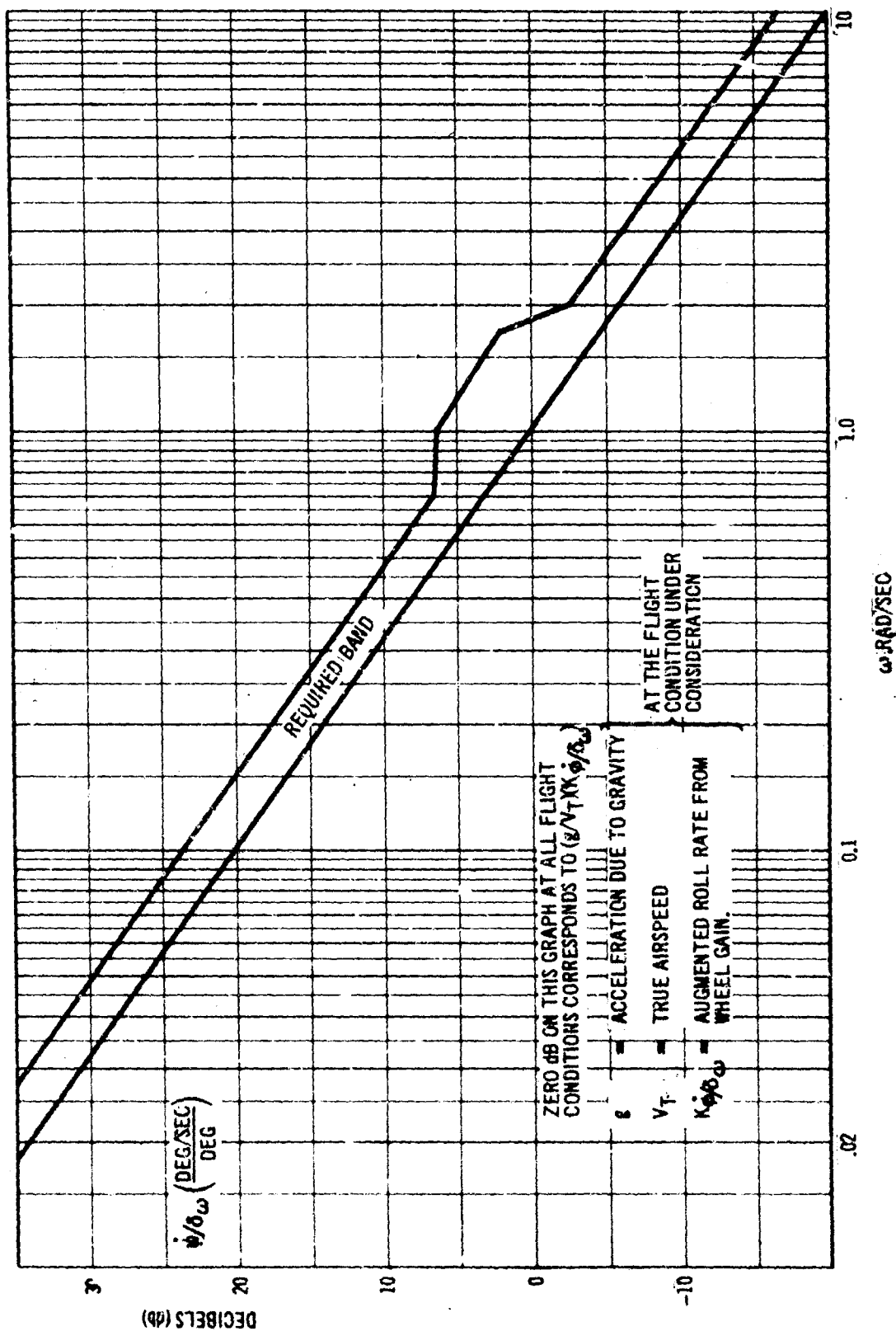


Figure 3. Frequency Response Band for Yaw Rate From Pilot Wheel Position

shaker system shall have two independent, isolated, single channels. Further protection is provided by the automatic introduction of a pitch-down signal to the E/CS if pilot response to stick shaker does not arrest the excessive angle of attack. the automatic pitch down system shall be dual channel, fail-passive.

5.3.4.2 Engagement/Disengagement. The stick shaker and automatic pitch-down system shall be engaged automatically, after warm-up, if failures do not exist. The stick shakers are disengaged by circuit breakers. The pitch-down system is disengaged by a switch.

5.4 Hydraulic Power Requirements. Three basic hydraulic power generation and distribution systems and one standby system shall be installed in the airplane. The system capabilities shall be based upon the power demands of the flight controls such that loss of one power system shall not reduce the airplane controllability. The subsequent loss of a second system shall allow safe operation throughout the flight envelope. Hydraulic pump arrangement on the accessory drive system gear boxes shall be such that the loss of any one engine, or two engines on one side, shall not cause the loss of any hydraulic system, and such that the loss of any two engines shall not cause the loss of more than one hydraulic system. Hydraulic power shall be sufficient to provide the performance characteristics required by the hydraulic using equipment, i.e., primary flight controls, secondary flight controls, and automatic flight controls, and landing gear (including extension and retraction, brakes, and steering), as defined in this specification and in Landing Gear Specification D6A10108-1.

The standby system shall provide secondary power to operate the brake system, power to operate the standby landing gear extension system, power to operate the brake system when the aircraft engines are not operating, and power to operate the aft main gear steering system both with and without the aircraft engines operating.

5.4.1 Pressure. Nominal system pressure shall be 3,000 psi.

- a. Pressure Relief. Pressure relief shall be provided, as necessary, to all parts of the system where trapped fluid may generate pressure as a result of temperature changes. The main system relief valves shall have a full flow setting of 3,850 psi maximum. Cracking pressure shall be 3,500 psi $\pm$ 50 psi and the reseal pressure shall be 3,100 psi minimum.
- b. Dynamic Pressure Effects. Pressure drop through control valves and pressure drop in the lines shall be combined to obtain required subsystem performance with minimum weight. Proper functioning of any unit shall not be adversely affected by the back pressure in the system. Malfunctioning of any subsystem shall not cause other subsystems or alternate systems to operate inadvertently or to become inoperative.

Pressure transients (such as may be caused by flow changes or pump input speed changes) in the distribution system shall not exceed 135 percent of the nominal system pressure and shall return to the nominal pressure within two seconds.

There shall be no periodic pressure pulsations which would indicate that the system is marginally stable. Periodic pressure pulsations caused by pump piston action shall be limited to ± 250 psi upstream of the system pressure filter and to ± 100 psi downstream of the pressure filter.

5.4.2 Flow Requirements. The three systems, A, B, and C, shall be capable of providing the total maximum flow demands during the typical flight profile as specified in the curves shown in Figs. 4, 5, and 6 respectively; and, they shall provide the flow for actuation, quiescent internal leakage, and control dither for each function as specified in Table VI.

5.4.3 Fluid Temperature. The normal fluid operating maximum temperature limits shall be as follows:

- | | |
|-------------------------------------|-------|
| a. Pump inlet | 320°F |
| b. Bulk Oil at heat exchanger inlet | 350°F |
| c. Maximum hot spot | 400°F |

The hydraulic systems shall be capable of starting and operating at a temperature of -50°F without causing subsequent degradation of performance and life requirements. The maximum time required to reach full performance requirements of the system from a -50°F cold soak shall not be greater than 15 min and means shall be provided to the flight crew to determine that full performance capability is available.

Each system shall be capable of full performance with the fluid at any bulk temperature between +60°F and 350°F and shall be capable of operating with a pump inlet temperature of +400°F for one hr. The systems shall have an adequate stiffness margin to operate at this temperature condition. The only degradation in maximum performance shall be due to the slight increase in internal leakage.

All parts of the system containing hydraulic fluid shall be able to function when subjected to a sudden fluid temperature change which could occur due to operation of a previously dormant component.

5.4.4 Heat Rejection. The heat removal rate, which is necessary to maintain the fluid temperature specified in Par. 5.4.3 shall be no greater than 5,500 BTU/min for the basic system. Heat shall be rejected via fluid-to-fuel heat exchangers, to the aircraft fuel or via fluid-to-air heat exchangers during subsonic operation. Sufficient cooling capability for the standby system shall be provided by the installed system line runs in the landing gear area to limit the system temperature to 350°F under all normal operating conditions. Performance parameters for these conditions are contained in specification D6A10108-1.

5.4.5 Fluid Flow Effects. Reduced flow, such as that created by single pump operation of a multi-pump system, shall not cause malfunctioning or damage to any component or subsystem. Increased flow, such as that caused by the effect of air loads, shall not adversely affect the proper functioning of any component or subsystem.

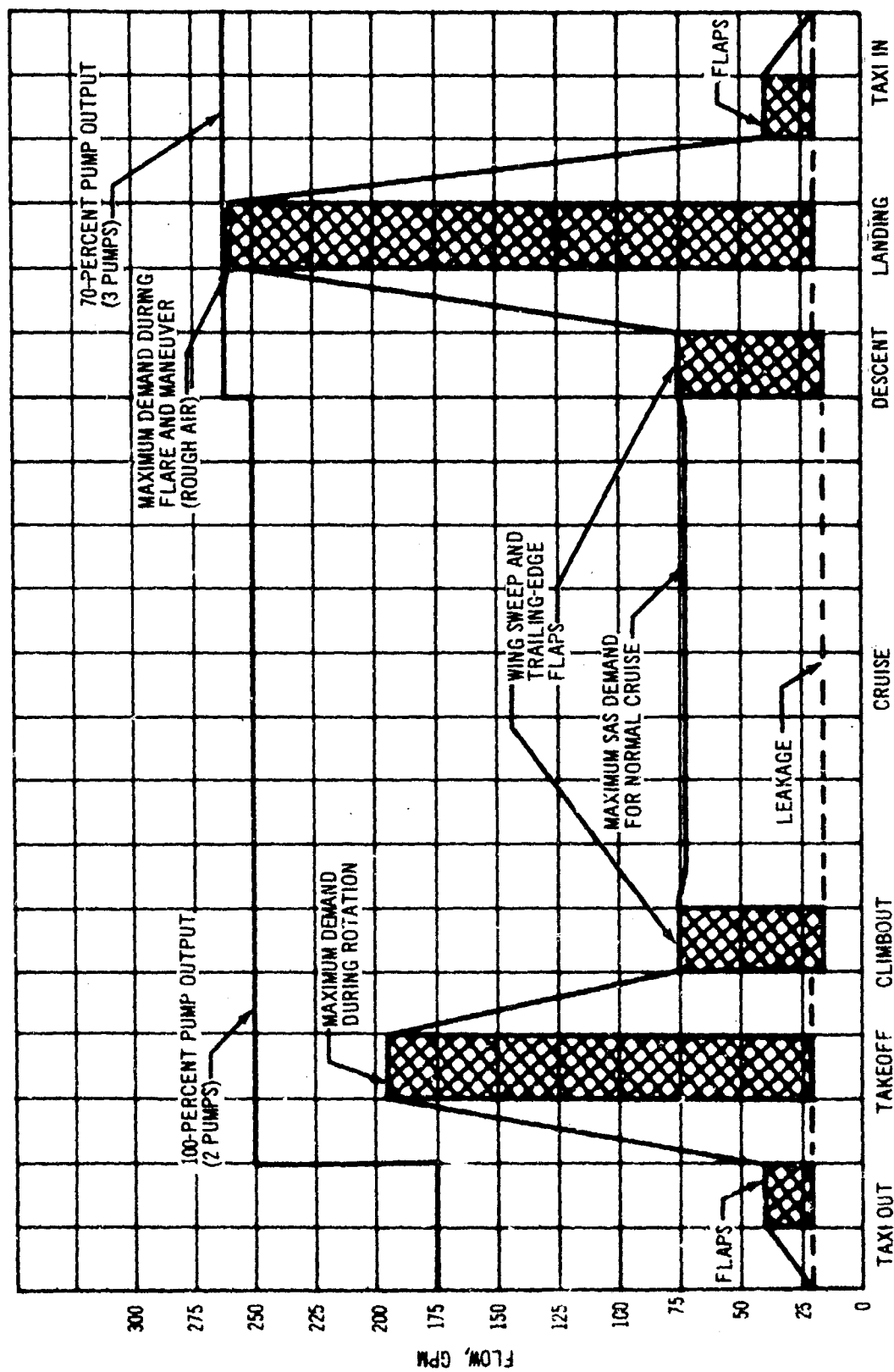


Figure 4. Hydraulic System A Hydraulic Load, Flight Profile

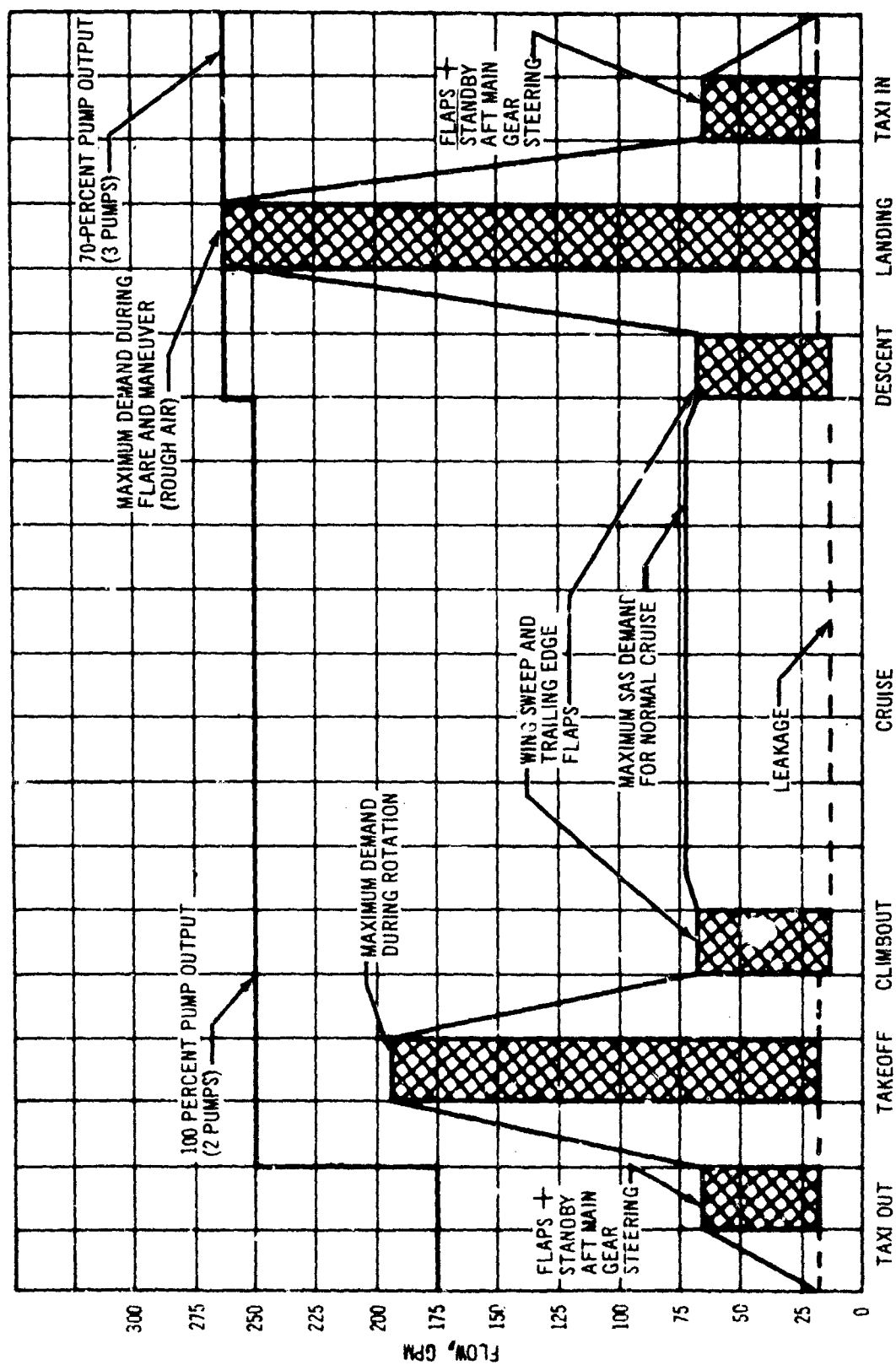


Figure 5. Hydraulic System B Hydraulic Load, Flight Profile

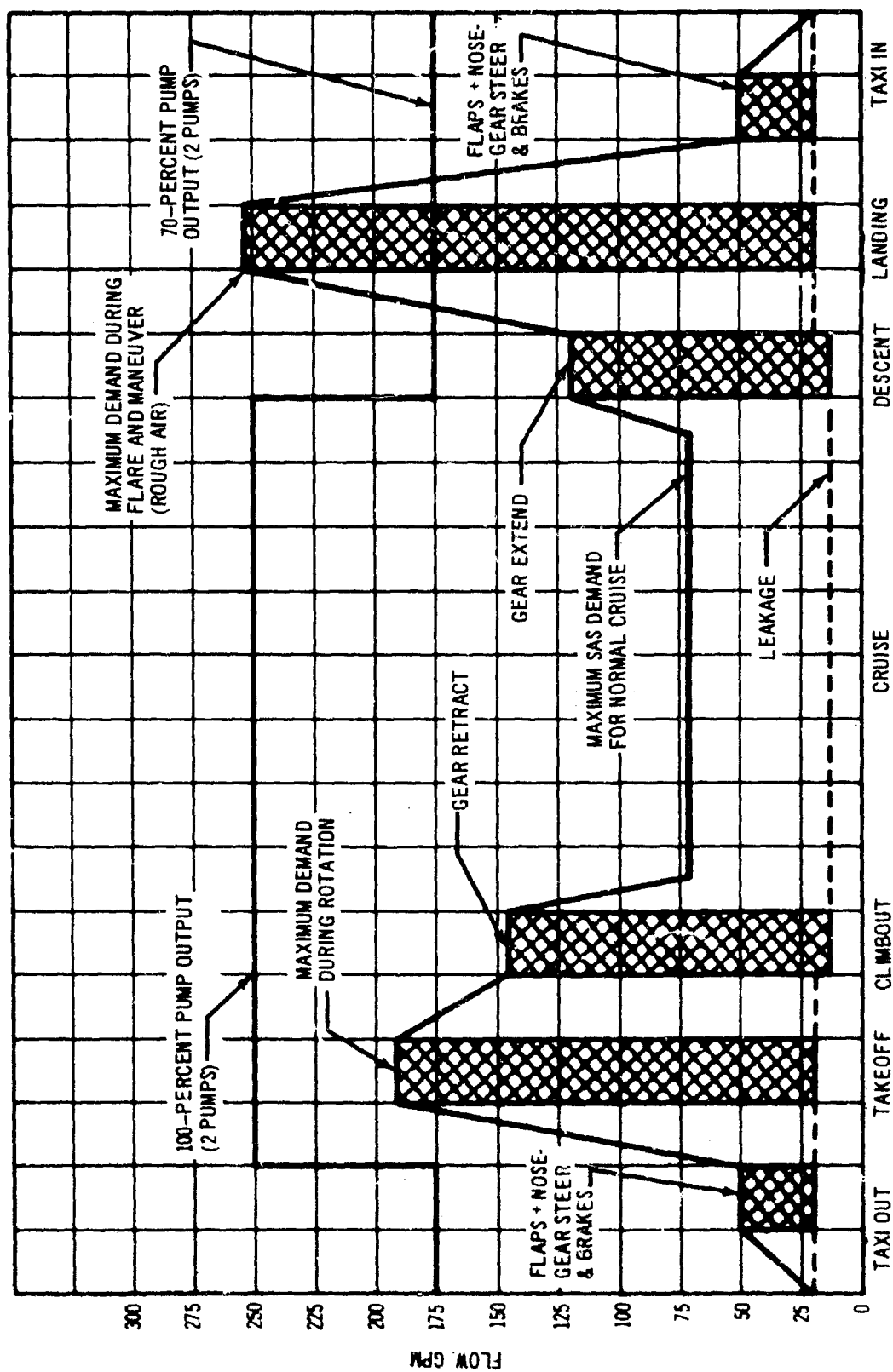


Figure 6. Hydraulic System C Hydraulic Load, Flight Profile

Table VI Hydraulic Flow Requirements

Flow Per System in GPM						
Function or Subsystem	Leakage Flow	Dither Flow	Max Demand Flow	Total Steady State Flow	Total Max Flow	System or Systems
Long. axis	9.24	5.24	244.1	9.24	253.34*	A, B, C
Lateral axis	2.82	0.82	213.8	2.82	216.62	A, B, C
Directional axis	2.03	1.03	50.5	2.03	52.53	A, B, C
Wing sweep and high lift	4.98	--	69.7	4.98	74.68	A, B, C
Landing gear retraction	--	--	126.7	--	126.7	C
Landing gear extension	--	--	100.6	--	100.6	C
Brake and nose gear steering	3.6	8.0	--	11.6	11.6	C
Aft main gear steering	0.8	--	23	0.8	24	Stby, B
System totals						
A	19.07	7.09			253.34*	
B	17.14	7.09			253.34*	
C	21.88	14.84			244.33	
Stdby	0.8				24	

* No surface rate reduction occurs because leakage flow decreases at maximum surface control rate demands.

5.4.6 Aiding Load Pressure. Portions of the hydraulic system subjected to externally applied loads shall withstand a pressure equal to the maximum loads, maximum operating pressures, and a safety factor based on the temperature effects on material strength.

5.4.7 Indication and Monitoring. Flight station indication and monitoring provisions for hydraulic power performance are contained in specification D6A10109-1. Hydraulic components shall have integral sensors (or provisions) for subsystem instrumentation and AIDS inputs as required. Reservoir fluid indication will be temperature compensated.

5.4.8 Fluid Sampling. Fluid sampling connections shall be installed in the hydraulic equipment bay.

5.4.9 Hydraulic System Components.

- a. Main System. Systems A and B shall each have three 125 GPM variable displacement pumps equipped with electrical depressurization devices, input shaft shear sections, and quick attach-detach mountings. System C shall have two pumps of the same type.
- b. Standby System Pumps. The standby system shall be primarily powered by a 24 gpm hydraulic motor-driven pump. A 6 gpm electric motor pump shall also be installed in the system.
- c. Reservoirs. The reservoir in each system shall maintain the required pressure at the inlet ports of the system pumps over a fluid temperature range of +60 to +400°F during maximum subsystem flow demand. Each reservoir shall be designed in accordance with the requirements of MIL-R-8931, and Par. 4.2.3.6. Each reservoir shall have a drain.
- d. Filters and Filtration. All system filters shall be equipped with temperature-compensated, clogged element, warning indicators. Pressure and case drain filters shall be non-bypass type and shall be located as close as practical to the pumps in order to minimize the amount of tubing and system components which must be flushed following a pump failure. The return filters shall be the two-stage type; where the normal system flow shall pass through both stages in series and higher flows shall bypass the first stage and pass through the second stage only. The filter elements shall be designed to withstand the maximum applied differential pressure without resultant failure which would release contaminants into the fluid stream. Pressure and case drain filters shall be rated at 25 microns absolute, the return filter first stage rated at 3 microns absolute and the return filter second stage 15 microns absolute. System filling shall be through installed filters. All primary flight control hydraulic actuators shall have 100 micron absolute filters installed in the supply port of each unit. Master servos, direct lift servos, and SAS servos shall have 25 micron absolute filters in the supply port of each unit.

- e. Hydraulic Fluid. The fluid for the hydraulic systems described in this specification shall meet the requirements of Boeing Material Specification 3-10.

5.5 Flight Control and Hydraulics Safety Requirements.

5.5.1 Primary and Secondary Flight Controls and Hydraulics.

5.5.1.1 Hydraulic System Safety.

5.5.1.1.1 Hydraulic System Failure. The airplane shall be capable of operating at all speeds with a loss of one hydraulic system. With two systems lost, the airplane shall be capable of operation at subsonic speeds. Operating rate of the wing-sweep mechanism, with one hydraulic system, shall be at least 30 percent of the normal rate. Each of the trim systems shall be irreversible. All surface actuators shall be irreversible.

In the event of any single disconnect or jam in the primary or the secondary flight control linkage and cable system the airplane shall be capable of being safely controlled without requiring exceptional skill or strength on the part of the flight crew.

5.5.1.2 Engine Burst. The hydraulic systems shall be so located in the aircraft that not more than two systems can be decommissioned by failure of a rotating component of any engine.

5.5.1.3 System Integrity. Interconnection of hydraulic power systems shall not be possible either in flight or on the ground.

5.5.1.4 Fire Safety. Hydraulic fluids shall have a minimum autogenous ignition temperature (AIT) of 800°F for new fluid or used fluid. The hydraulic system shall be designed to comply with the fire protection requirements specified in FAR 25.1435.

5.5.1.5 All-Engine-Out Condition Hydraulic power shall be provided as necessary to control the airplane if all engines fail.

5.5.2 Automatic Flight Controls and Electric Command. The provisions for flight safety incorporated within the Automatic Flight controls and Electric Command are categorized herein as those general requirements which apply equally to the Electric Command, Autopilot, Auto Throttle, SAS and Angle of Attack Warning and Control System, and additional requirements which are peculiar to the individual systems.

5.5.2.1 General Safety Requirements. Loss of a single electric power source shall not affect the performance of the E/CS, A/P, A/T, SAS or Angle-of-Attack Warning and Control System. Loss of a second power source shall not cause failure to more than one channel in each system.

When two or more channels of a single axis, excluding the F/CS, are engaged, transients resulting from any failure shall cause less than 0.02 g incremental load in pitch and a maximum of 3 deg roll angle.

A failure or apparent malfunction within a control axis or channel, including associated malfunction detection circuitry, shall not affect other operating axes or good channels.

Physical separation of multiple channels shall be employed.

5.5.2.2 Requirements of Electric Command. In each control axis the system shall be fail-operational on the first failure and fail-passive on the second.

The detection system shall disengage the faulty component upon first failure and activate warning signals. A second failure shall cause automatic disconnect of the remaining channels.

It shall be possible to manually select a single channel of an axis. When manual selection is made, authority may be limited as required to provide protection against a hardover failure of the F/CS.

5.5.2.3 Requirements of Autopilot and Auto Throttle. In each control axis the autopilot system shall be fail-operational on the first failure and fail-passive on the second failure. The auto throttle system shall be fail passive on the first failure.

It shall be possible to manually select a single channel of an axis. When manual selection is made, authority may be limited as required to provide protection against a hardover failure of the A/P.

Inflight testing capability of the flare mode and monitor shall be provided, which shall give a positive indication of the results to the pilot. System degradation as a result of failures shall not preclude the capability for Category II and/or Category I instrument approach, when adequate functional components exist to perform these types of instrument approaches.

Loss of commands to or from the flight directors shall not require disengaging the autopilot.

5.5.2.4 Specific Features of Stability Augmentation System.

- a. The pitch axis of the SAS shall be fail-operational after two failures and fail-passive upon the third failure.
- b. The yaw axis of the SAS shall be fail-operational after two failures and fail-passive upon the third failure.
- c. The roll axis of the SAS shall be fail-operational after one failure and fail-passive upon the second failure.
- d. The malfunction detection circuitry shall be capable of detecting failures of the servo actuator control loops as well as failures within the SAS electronic circuits.
- e. While an axis is in a fail-operational status the malfunction detection circuitry shall, upon a failure, disengage the faulty

section. When an axis enters the fail-passive status (i.e., after two failures in pitch and yaw, and one failure in roll), the malfunction detection circuitry shall activate warning signals and, upon a failure, automatically disconnect the remaining channels.

- f. It shall be possible to manually select a single channel of an axis. When manual selection is made, authority may be limited as required to provide protection against a hardover failure of the SAS.

5.5.2.5 Angle of Attack Warning and Control System. The stick shaker shall have two isolated channels, one for each stick.

The automatic pitch-down portion of the system shall have two channels which provide fail-passive operation. The pilot shall be given indication in the event of failure.

5.6 Selection of Specifications and Standards. Selection of Specifications and Standards shall be in accordance with Par. 3.3.2 of D6A10107-1.

5.7 Materials, Parts, and Processes. Selection of Materials, Parts, and Processes shall be in accordance with Par. 3.3.3 of D6A10107-1.

5.8 Standard and Commercial Parts. Standard and Commercial Parts shall be in accordance with Par. 3.3.4 of D6A10107-1.

5.9 Moisture and Fungus Resistance. Moisture and Fungus Resistance shall be in accordance with Par. 3.3.5 of D6A10107-1.

5.10 Corrosion of Metal Parts. Corrosion prevention of Metal Parts shall be in accordance with Par. 3.3.6 of D6A10107-1.

5.11 Interchangeability and Replacement. Interchangeability and Replacement shall be in accordance with Par. 3.3.7 of D6A10107-1.

5.12 Workmanship. Workmanship shall be in accordance with Par. 3.3.8 of D6A10107-1.

5.13 Electromagnetic Interference. Electromagnetic Interference shall be in accordance with Par. 3.3.9 of D6A10107-1.

5.14 Identification and Marking. Identification and Marking shall be in accordance with Par. 3.3.10 of D6A10107-1. Hydraulic lines shall be marked in accordance with AND 10375 and BAC-T11Y in a manner suitable for the environmental conditions.

5.15 Storage. Storage shall be in accordance with Par. 3.3.11 of D6A10107-1.

5.16 Subsystem Weight.

5.16.1 The 2707-100 with GE Engines. The weight of the flight controls hydraulics subsystem shall not exceed 14,170 lb. This weight is an allocation of the overall airplane weight based on analysis and design experience and may be revised as long as the overall airplane weight defined in D6A10107-1 is not exceeded.

5.16.2 The 2707-100 with P&W Engines. The weight of the flight controls and hydraulics subsystem shall not exceed 13,990 lb. This weight is an allocation of the overall airplane weight based on analysis and design experience and may be revised as long as the overall airplane weight defined in D6A10107-1 is not exceeded.

5.17 Endurance. The equivalent of 75 hours of simulated SST flights shall be performed on the flight controls servo simulator. This shall be equivalent to 75 SST mission cycles (flights). At least 25 hours of this shall be accomplished prior to the first flight of the prototype. Performance shall be within the applicable requirements as defined herein.

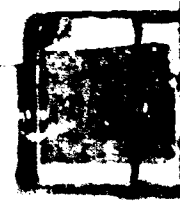
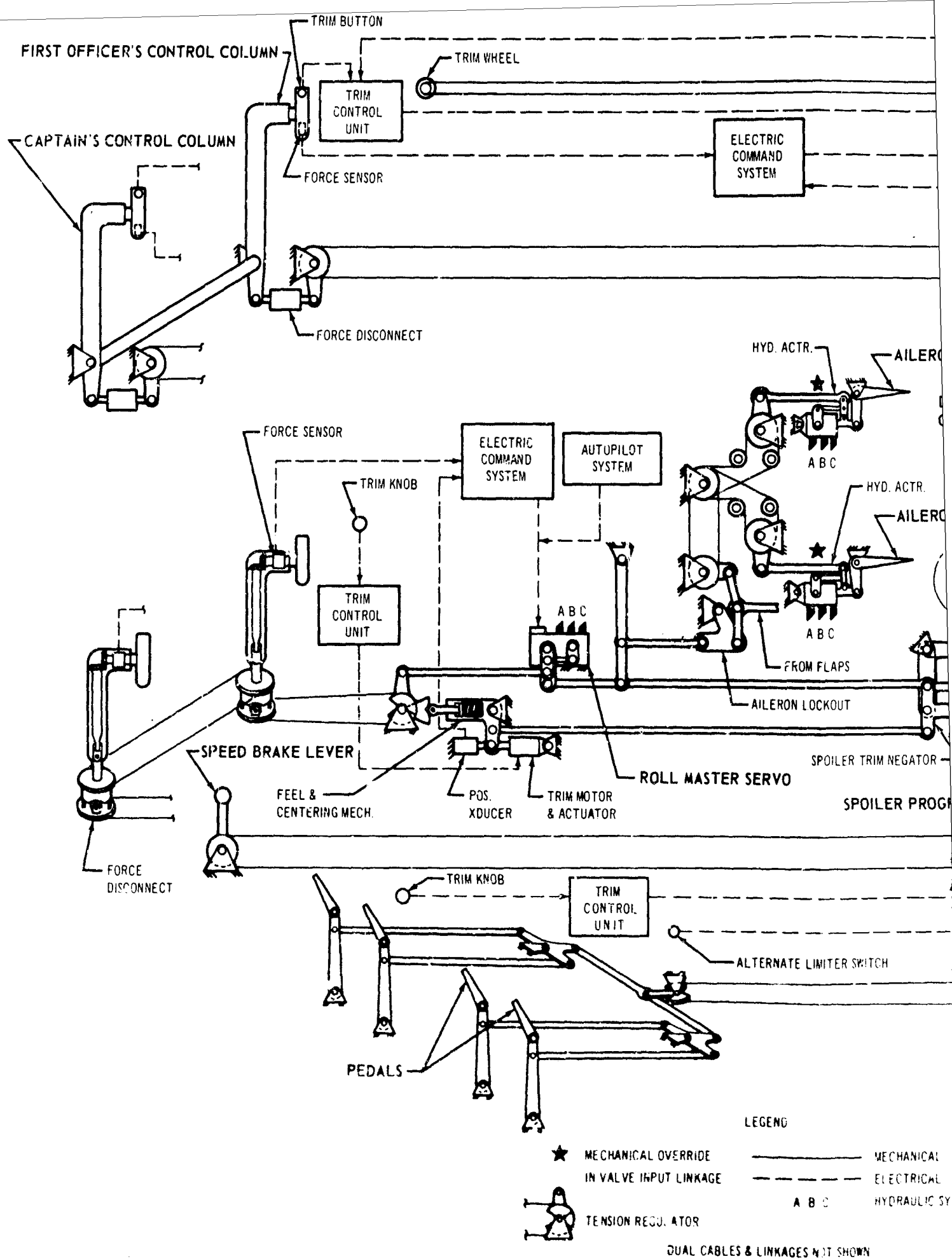
6. SUBSYSTEM DESCRIPTION

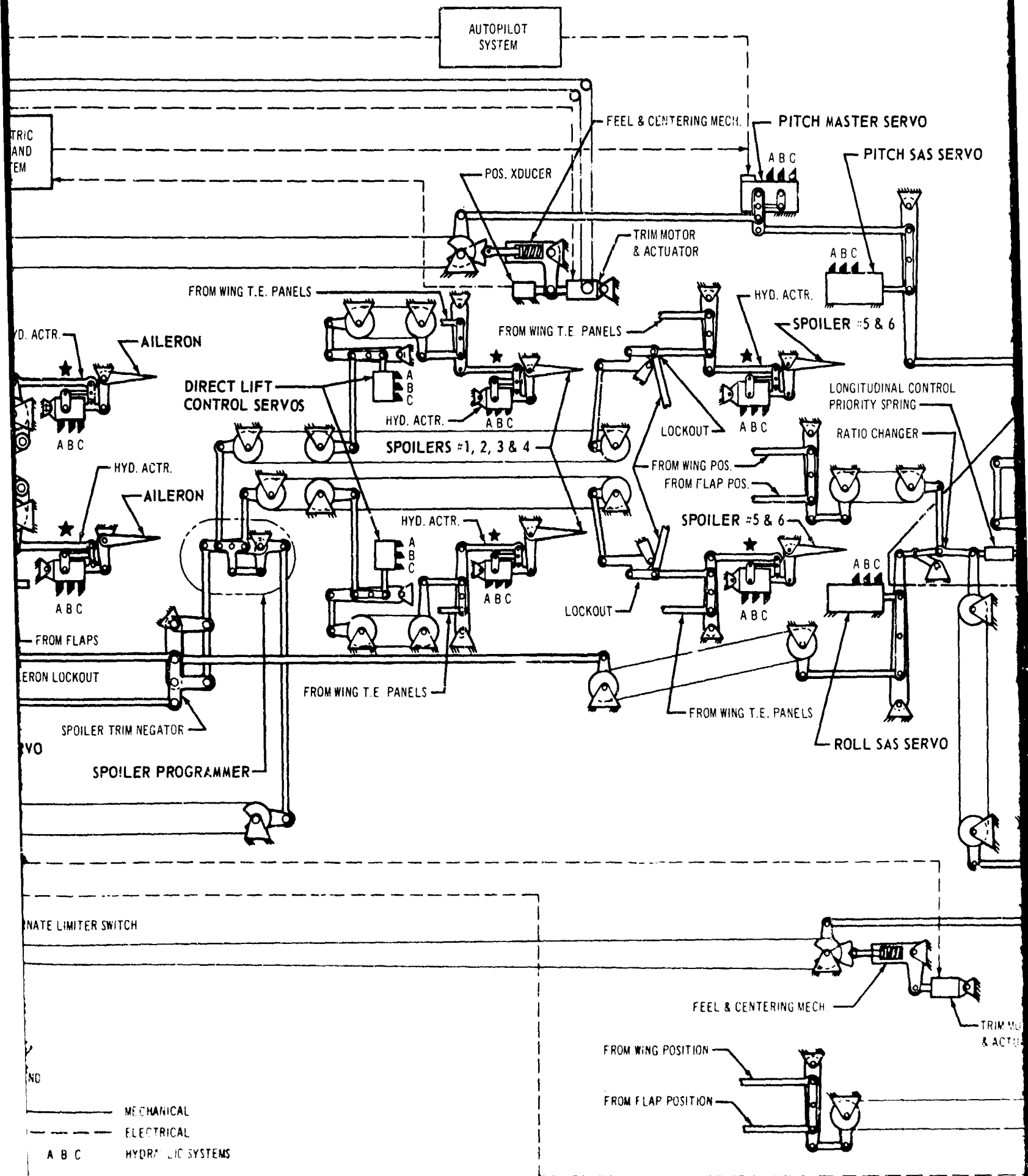
6.1 Primary Flight Controls System. The primary flight control system consists of the longitudinal control, the direct lift control, the lateral control, the directional control, and the wing sweep systems. The longitudinal, direct lift, lateral, and directional control systems are shown schematically in Fig. 7.

6.1.1 Longitudinal Control System. The longitudinal control system positions the elevons, primary elevators, and auxiliary elevators in response to pilot and automatic flight control system (AFCS) pitch commands.

The longitudinal control system consists of the following elements:

- a. The pilot's and copilot's control column.
- b. The pitch electric command system (E/CS) more fully described in Par. 6.1.1.1.
- c. A mechanical cable system from each control column to the pitch master servo.
- d. A pitch master servo. This is a triple hydro-mechanical control unit that receives E/CS or autopilot electric signals, or direct cable force inputs from the control columns. It contains three electro-hydraulic servo valves and three dual-piston push-push hydraulic actuators whose output is combined in a common output crank.
- e. The pitch stability augmentation system (SAS) servo. This is a triple hydro-mechanical control unit that receives SAS electric signals. The unit contains three electro-hydraulic servo valves and three dual piston push-push hydraulic actuators, whose output is combined in a common output crank.





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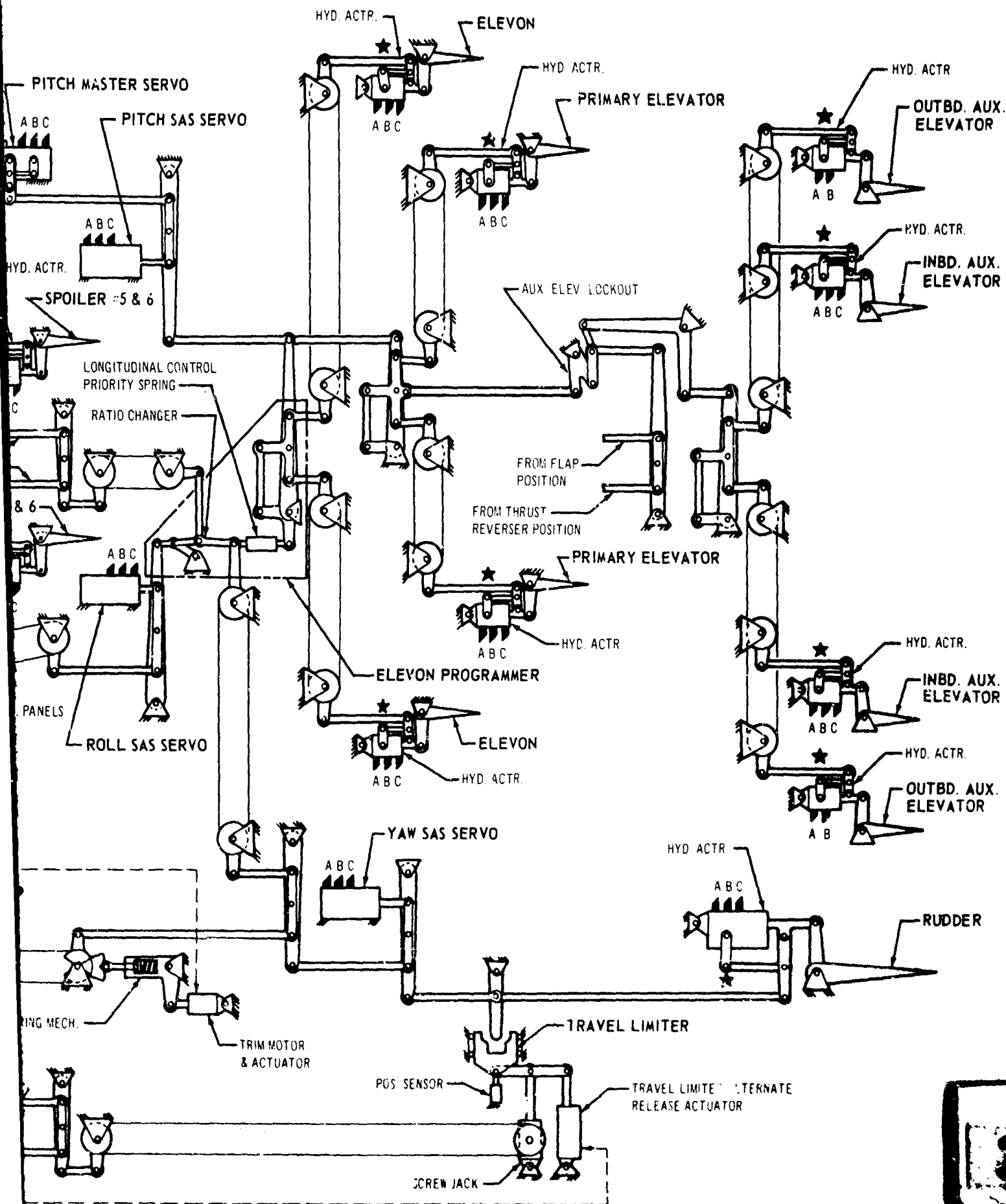


Figure 7. Longitudinal, Lateral, and Directional Control Systems Schematic

- f. A mechanical linkage which sums the outputs of the master servo and SAS servo.
- g. The elevon mechanical programmer. This unit combines the summed outputs of the pitch master servo, pitch SAS servo, and the summed outputs of the lateral master servo and lateral SAS servo, with inputs from wing position and wing flap position. One output is connected through a mechanical cable system to the control valves of the hydraulic actuators at the elevon surfaces. A second output is connected through a cable system to the directional control system as described in Par. 6.1.3d.
- h. A mechanical cable system that connects the summed outputs of the master servo and SAS servo with the control valves of the hydraulic actuators at the primary elevator surfaces.
- i. A mechanical linkage that connects the summed outputs of the master servo and SAS servo with the auxiliary elevator lockout.
- j. An auxiliary elevator lockout that combines the summed outputs of master servo and SAS servo with inputs from flap position and thrust-reverser position. Its output is connected through a mechanical cable system to the control valves of the hydraulic actuators at the auxiliary elevator surfaces.
- k. The six pairs of hydraulic actuators and two triple mechanical input servo valve assemblies, powered by the three main hydraulic systems A, B, and C, that position the two elevon surfaces.
- l. The 18 double-barreled primary elevator hydraulic actuators and six mechanical input servo valve assemblies. Nine actuators and three valve assemblies position each primary elevator. Each group of three actuator and valve assemblies for each elevator is powered independently by one of the three main hydraulic systems.
- m. The six double-barreled inboard auxiliary elevator actuator and mechanical input servo valve assemblies, three of which position each inboard auxiliary elevator. Each of the three actuator and valve assemblies on each side is powered independently by one of the three main hydraulic systems.
- n. The four double-barreled outboard auxiliary elevator actuator and mechanical input servo valve assemblies, two of which position each outboard auxiliary elevator. The two actuator and valve assemblies on each side are powered independently by hydraulic systems A and B respectively.
- o. The hydraulic pressure shutoff control is more fully described in Par. 6.1.1.2.

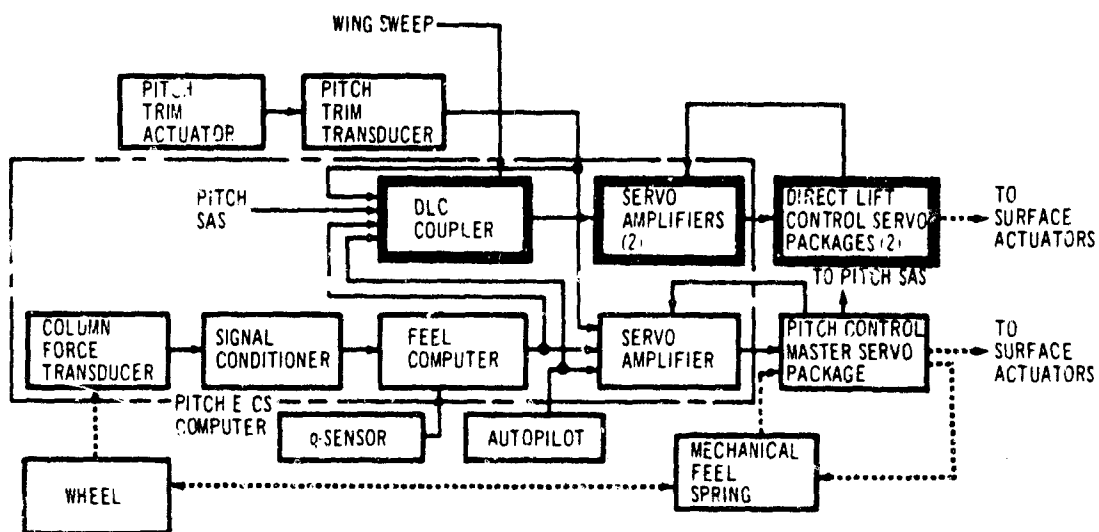
6.1.1.1 Pitch Axis Electric Command System (E/CS). In the longitudinal control system, force sensors on the pilot and co-pilot columns detect applied force and produce electrical signals for the triple-channel E/CS. In Fig. 8 is shown a functional block diagram for a single channel of the E/CS. The summed output of the pilot and co-pilot column force sensors is filtered in the signal conditioner to remove or smooth any noise component in the signal. Force sensor and dynamic pressure signals are fed to the electronic feel computer. The feel computer gain schedules column force as a function of dynamic pressure. A stick force detent is also provided electronically in the feel computer. The output of the feel computer is amplified in the servo amplifiers and fed to the pitch control master servo. When the wings are in the 20-deg sweep position, the signals are also amplified in the direct lift servo amplifiers and fed to the direct lift servos. The servo amplifier input is monitored by a failure detector which detects and removes a defective channel to provide a fail-operational system.

6.1.1.2 Longitudinal Flight Controls Pressure Shutoff Control. Each control surface actuator, the master servo, the SAS servo, and each direct lift control servo is equipped with a shutoff valve in each of the three (A, B, and C) hydraulic supply ports and an electric switch in each of its three circuits that detects failure or disagreement of any circuit. For the surface actuators and master servos, the switch actuates a light on the flight control pressure shutoff panel shown in Fig. 9. The flight crew can close the identified shutoff valve by means of the switches on this panel. For the SAS and direct lift servos, the switch actuates a light on the AFCS Control Panel shown in Fig. 18. The flight crew has the capability of selecting single channel SAS. The crew also has the capability to shut off direct lift control.

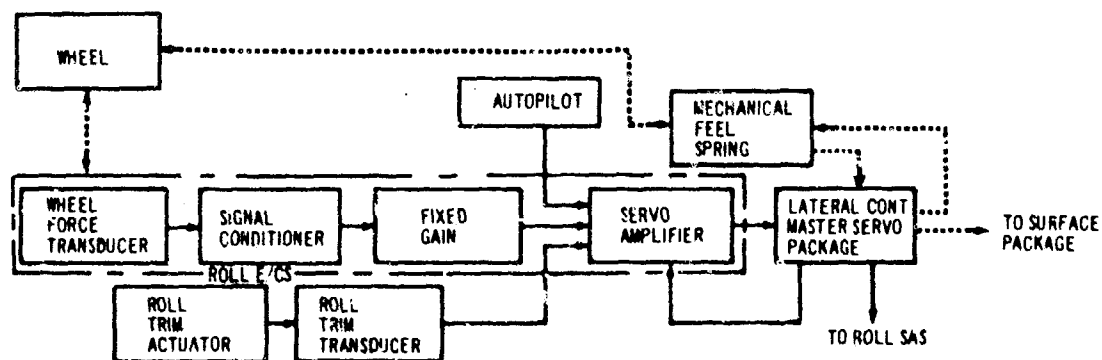
6.1.2 Lateral Control System. The lateral control system provides scheduled control of the ailerons, elevons, spoilers and rudder in response to pilot and automatic flight control system (AFCS) roll commands.

The lateral control system consists of the following elements:

- a. The pilot's and copilot's control wheel.
- b. The roll electric command system which is similar to the pitch E/CS, except that the dynamic pressure input to the electronic feel computer is not required.
- c. A mechanical cable system from each control wheel to the roll master servo.
- d. The roll master servo which is identical to the pitch master servo and which receives E/CS or autopilot electric signals, or direct cable force inputs from the control wheels.



Pitch Axis



Roll Axis

..... MECHANICAL PATH

~~XXXXXXXXXXXX~~ ELECTRICAL PATH

Figure 8. Pitch Command System Diagram, Pitch and Roll

FLIGHT CONTROLS HYDRAULIC PRESSURE SHUTOFF															
LEFT AILERON				LATERAL MASTER				LONGITUDINAL MASTER				RIGHT AILERON			
ON	A	B	C	ON	A	B	C	ON	A	B	C	ON	A	B	C
ON	A	B	C	ON	A	B	C	ON	A	B	C	ON	A	B	C
LEFT SPOILER NO 1				LEFT SPOILER NO 2				RIGHT SPOILER NO 1				RIGHT SPOILER NO 2			
ON				ON				ON				ON			
A				B				A				B			
B				C				B				C			
C				C				C				C			
LEFT SPOILER NO 3				LEFT SPOILER NO 4				RIGHT SPOILER NO 3				RIGHT SPOILER NO 4			
ON				ON				ON				ON			
A				B				A				B			
B				C				B				C			
C				C				C				C			
LEFT SPOILER NO 5				LEFT SPOILER NO 6				RIGHT SPOILER NO 5				RIGHT SPOILER NO 6			
ON				ON				ON				ON			
A				B				A				B			
B				C				B				C			
C				C				C				C			
LEFT ELEVON				LEFT MAIN ELEVATOR				RIGHT MAIN ELEVATOR				RIGHT ELEVON			
ON				ON				ON				ON			
A				B				A				B			
B				C				B				C			
C				C				C				C			
LEFT OUTBD AUX ELEVATOR				LEFT INBD AUX ELEVATOR				RIGHT INBD AUX ELEVATOR				RIGHT OUTBD AUX ELEVATOR			
ON				ON				ON				ON			
A				B				A				B			
B				C				B				C			
C				C				C				C			
RUDDER															
ON															
A															
B															
C															

Figure 9. Flight Controls Hydraulic Pressure Shutoff

- e. The mechanical linkage that combines the output of the master servo with an input from the outboard wing flaps to provide aileron control through a mechanical cable system that is connected to the control valves of the aileron surface actuators.
- f. The two triple-hydraulic aileron actuator and mechanical input servo valve assemblies that position the ailerons.
- g. The trim negator that combines inputs from the master servo and lateral trim actuator to prevent trim signals from entering the spoiler programmer.
- h. The mechanical spoiler programmer that combines inputs from the spoiler trim negator and speed brake lever and that schedules the operation of the spoiler surfaces.
- i. The mechanical cable system from the spoiler programmer to linkages that sum the programmer outputs with inputs from the upper wing trailing edge panels and operate the control valves of the inboard spoiler surface actuators. This cable system is also connected to the outboard spoiler lockouts.
- j. The outboard spoiler lockout that combines the spoiler programmer output with an input from wing position.
- k. The mechanical linkages that combine the output of the spoiler lockout with an input from the upper wing trailing edge panels and operate the control valves of the outboard spoiler surface actuators.
- l. The twelve triple-hydraulic spoiler actuator and mechanical input servo valve assemblies, one of which positions each of the six spoiler panels on each wing. Each actuator is powered by the three hydraulic systems.
- m. The roll SAS servo which is identical to the pitch SAS servo and receives electric SAS signals.
- n. The mechanical cable system from the roll master servo to a linkage which sums the master servo output with the SAS servo output to provide roll inputs to the elevon programmer.
- o. The hydraulic pressure shutoff control more fully described in Par. 6.1.1.2.

6.1.3 Directional Control System. The directional control system positions the rudder in response to pilot and AFCS commands in addition to the turn coordination commands from the lateral control system.

The directional control system consists of the following elements:

- a. The pilot's and copilot's rudder pedals.
- b. Dual mechanical cables from the rudder pedals to the feel and centering mechanism.
- c. The feel and centering mechanism that provides system feel and centering forces and receives inputs from the directional trim actuator.
- d. The mechanical linkage that sums the output of the feel and centering mechanism with the input from the elevon programmer.
- e. The mechanical linkage that sums the output from the item d. linkage with the output of the yaw SAS servo and provides an input to the rudder travel limiter.
- f. The yaw SAS servo which is identical to the pitch SAS servo and receives electric SAS signals.
- g. The rudder travel limiter. This is a mechanism that receives mechanical inputs of wing flap and wing position and limits the input to the rudder surface actuator. It also contains an alternate release actuator to increase the rudder travel limit during a wings aft landing.
- h. The mechanical linkage that connects the output of the travel limiter to the control valve of the rudder surface actuators.
- i. The triple-hydraulic rudder actuator and mechanical servo valve assembly, powered by the three main hydraulic systems, that positions the rudder.
- j. The hydraulic pressure shutoff control is more fully described in Par. 5.1.1.2.

6.1.4 Wing Sweep System. The wing sweep system positions the outboard variable sweep wings in response to pilot command for optimum lift and drag characteristics. The wing sweep control system is shown in Figs. 10 and 11.

The wing sweep system includes the following elements:

- a. The wing sweep control lever.
- b. The dual-mechanical cable system that connects the control lever to the wing-sweep/flap programmer.
- c. The wing-sweep/flap mechanical programmer. This unit schedules operation of the outboard wing flaps, leading edge slats and wings. The output is mechanically connected to the control valves of the power drive units of the flap, slats, and wing-sweep systems.

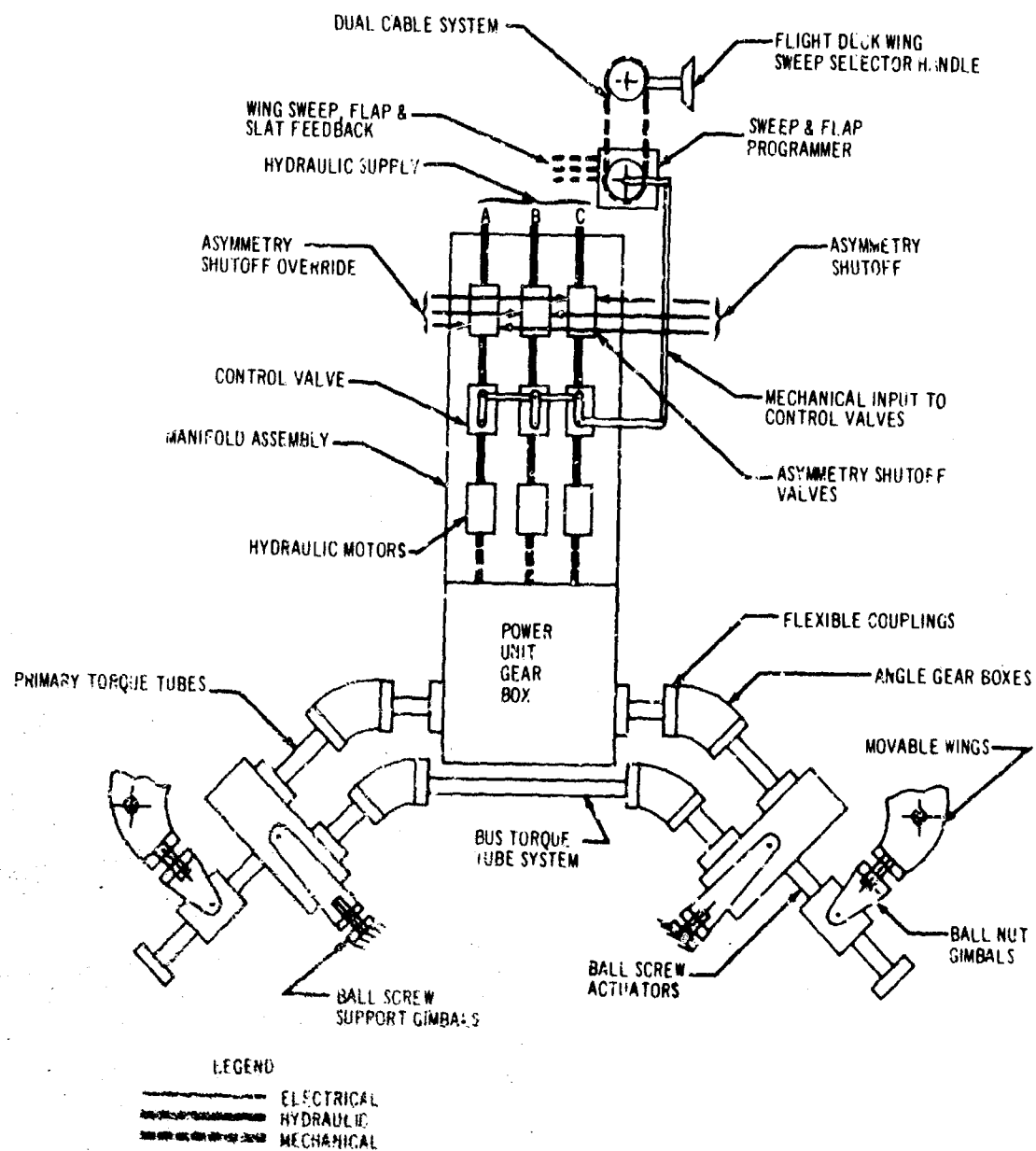


Figure 10. Wing-Sweep System Schematic

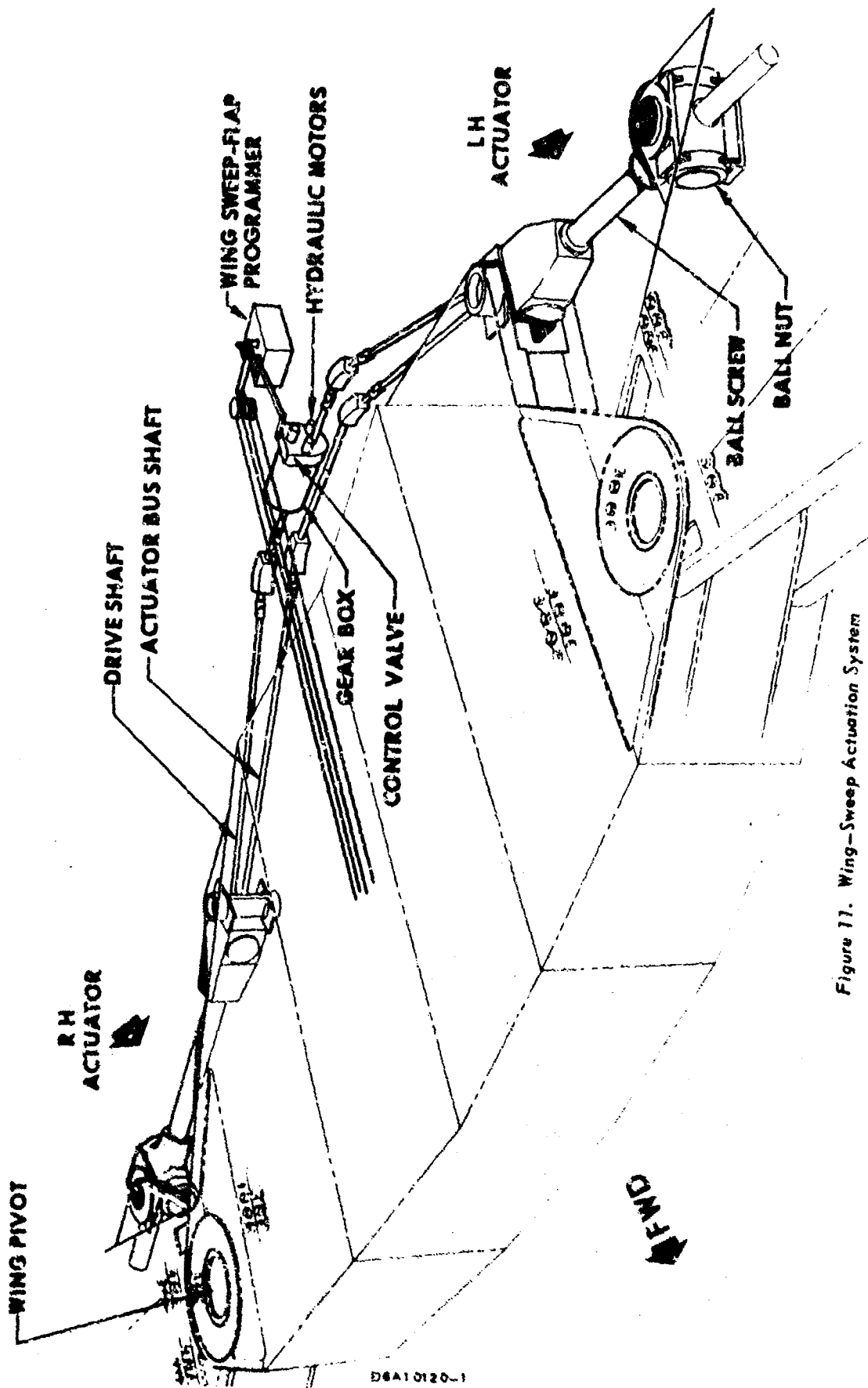


Figure 11. Wing-Sweep Actuation System

- d. The wing-sweep power drive. This unit consists of a triple differential gear box driven by three separate hydraulic motors. Each is controlled by a separate mechanical-input servo valve independently pressurized by one of the three hydraulic systems.
- e. The torque tube system which connects the output of the power drive unit to the wing-sweep actuators.
- f. The dual load path synchronizing torque tube which connects the wing-sweep actuators to ensure symmetrical positioning of the wings at all times.
- g. The wing-sweep actuators, which consists of dual-drive load-path gear box and ball bearing jackscrews. The outboard wing is attached to the nut of the jackscrew.
- h. An electric asymmetry detection and control system containing two independent electric sensor elements on each outboard wing and a comparator located in the environmentally controlled electronic bay. Hydraulic asymmetry shutoff valves are provided for each hydraulic pressure port of the power drive unit.

6.1.5 Direct Lift Control System. The direct lift control system provides programmed movement of the inboard spoilers in response to pilot and automatic flight control (AFCS) pitch commands.

The direct lift control system consists of the following elements:

- a. The two direct lift servos. Each is a triple hydromechanical control unit that receives electric command system (E/CS) or autopilot (A/P), and stability augmentation system (SAS) signals. Each unit contains three electrohydraulic servo valves and three dual piston push-push hydraulic actuators, whose output is combined in a common output crank.
- b. A mechanical linkage in each wing, which sums the output of the direct lift servo and spoiler programmer (Par. 6.1.2 item h) and introduces the sum into the mechanical cable system controlling the inboard spoiler actuator valves (Par. 6.1.2, item i).
- c. The hydraulic pressure shutoff control is more fully described in 6.1.1.2.

6.2 Secondary Flight Control Systems. The secondary flight control systems are longitudinal trim, lateral trim, directional trim, trailing edge flap, leading edge flap, and speed brake control systems.

6.2.1 Longitudinal Trim System. The longitudinal trim system is shown schematically on Fig. 12 and consists of the following elements:

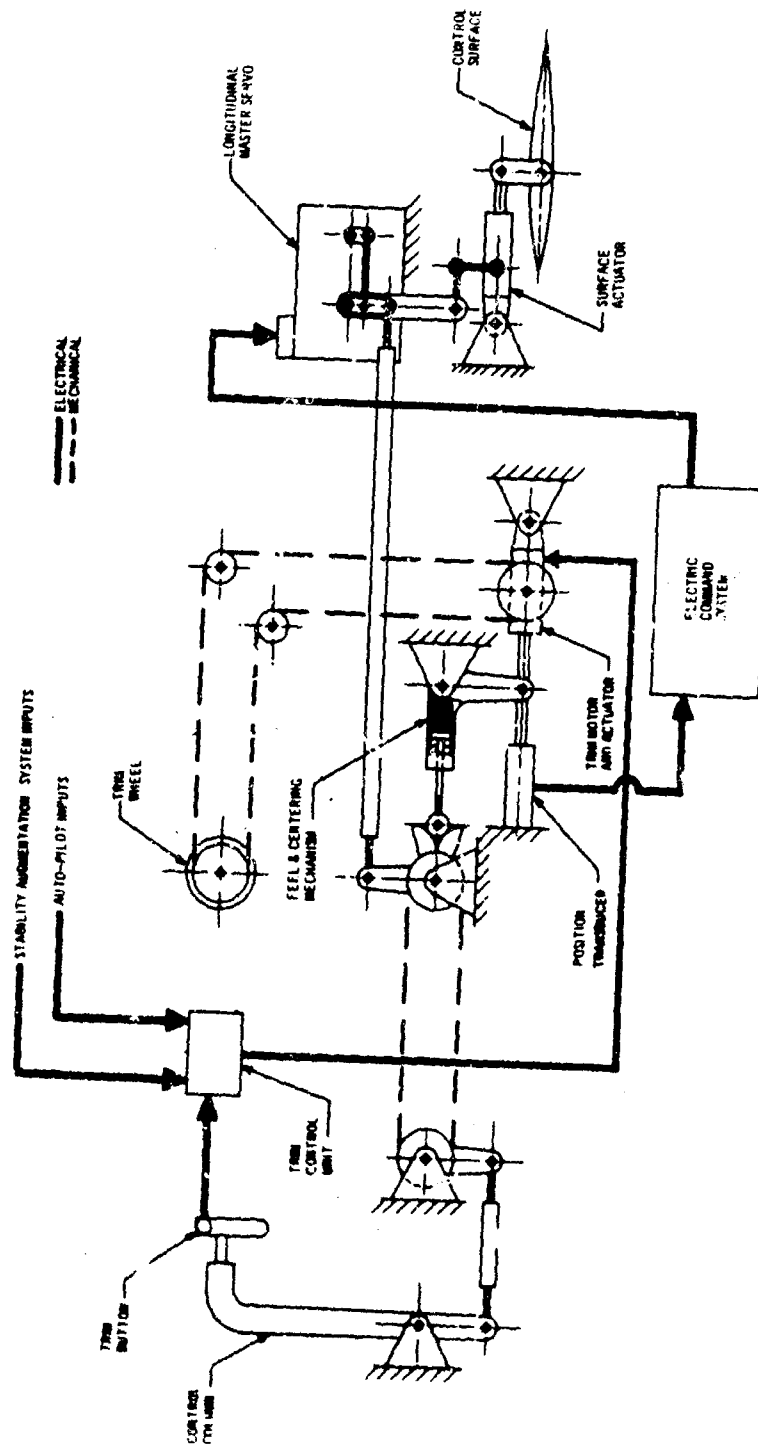


Figure 12. Longitudinal Trim System

- a. The trim button on each pilot's control wheel, the triple-channel trim control unit, the triple-winding stepping trim motor connected to the mechanical pitch trim actuator, and connecting wiring for electric pitch trim control.
- b. The trim wheel on each side of the aisle stand and mechanical cables to the mechanical trim actuator for manual pitch trim.
- c. The mechanical pitch trim actuator and three-position transducers that provide electrical trim signals through the pitch E/CS to the pitch master servo.
- d. The feel and centering mechanism and linkage connected mechanically to the pitch master servo to provide mechanical trim signals in event of pitch E/CS failure.

6.2.2 Lateral Trim System. The lateral trim system is composed of the following elements:

- a. The roll trim knob on the aisle stand, the dual channel trim control unit, the dual-winding stepping trim motor connected to the mechanical roll trim actuator, and connecting wiring for electric roll trim control.
- b. The mechanical roll trim actuators and three-position transducers that provide electric trim signals through the roll E/CS to the roll master servo.
- c. The feel and centering mechanism that serves to reposition the control wheel but which can be overridden by the wheel.
- d. A trim indicator incorporated into the trim knob.

6.2.3 Directional Trim The directional trim system is composed of the following elements:

- a. The yaw trim knob on the aisle stand, the dual-channel trim control unit, the dual-winding stepping trim motor connected to the mechanical yaw trim actuator, three-position transducers, and connecting wiring for electrical yaw trim control.
- b. The mechanical yaw trim actuator connected to the feel and centering mechanism.

6.2.4 Trailing Edge Flap System. The trailing edge flap system is shown schematically in Fig. 13 and consists of three separate flap drive systems: one for the flaps on the inboard wing, one for the inboard flaps on the outboard wing, and one for the outboard flaps on the outboard wing.

The flap system is controlled by the wing flap lever mounted on the pilot's aisle stand, mechanical cables to the wing-sweep/flap mechanical programmer, and linkage to three mechanical input hydraulic servo valves.

The inboard wing flaps are actuated by a power-drive gear box driven by two hydraulic motors pressurized by hydraulic systems A and C. The output shaft of the gear box is connected to the flap surfaces through a torque tube drive and ball bearing jackscrews.

The inboard flaps of the outboard wing are actuated by a power-drive gear box driven by three hydraulic motors pressurized separately by systems A, B, and C. The output shaft of the gear box is connected to the flap surfaces through torque tube drives and ball bearing jackscrews. The torque tube drives are routed around the wing pivot and contain linear ball bearing splines and swivel bevel gear boxes for torque tube extension and self alignment between the inboard and outboard wings.

Bevel gear boxes, designed so that one-half of each gear box is fixed and the other half rotatable, are provided at each end of the sliding splines. The gear ratios were selected so that wing-sweep motion imparts only a negligible input to the flap torque tube.

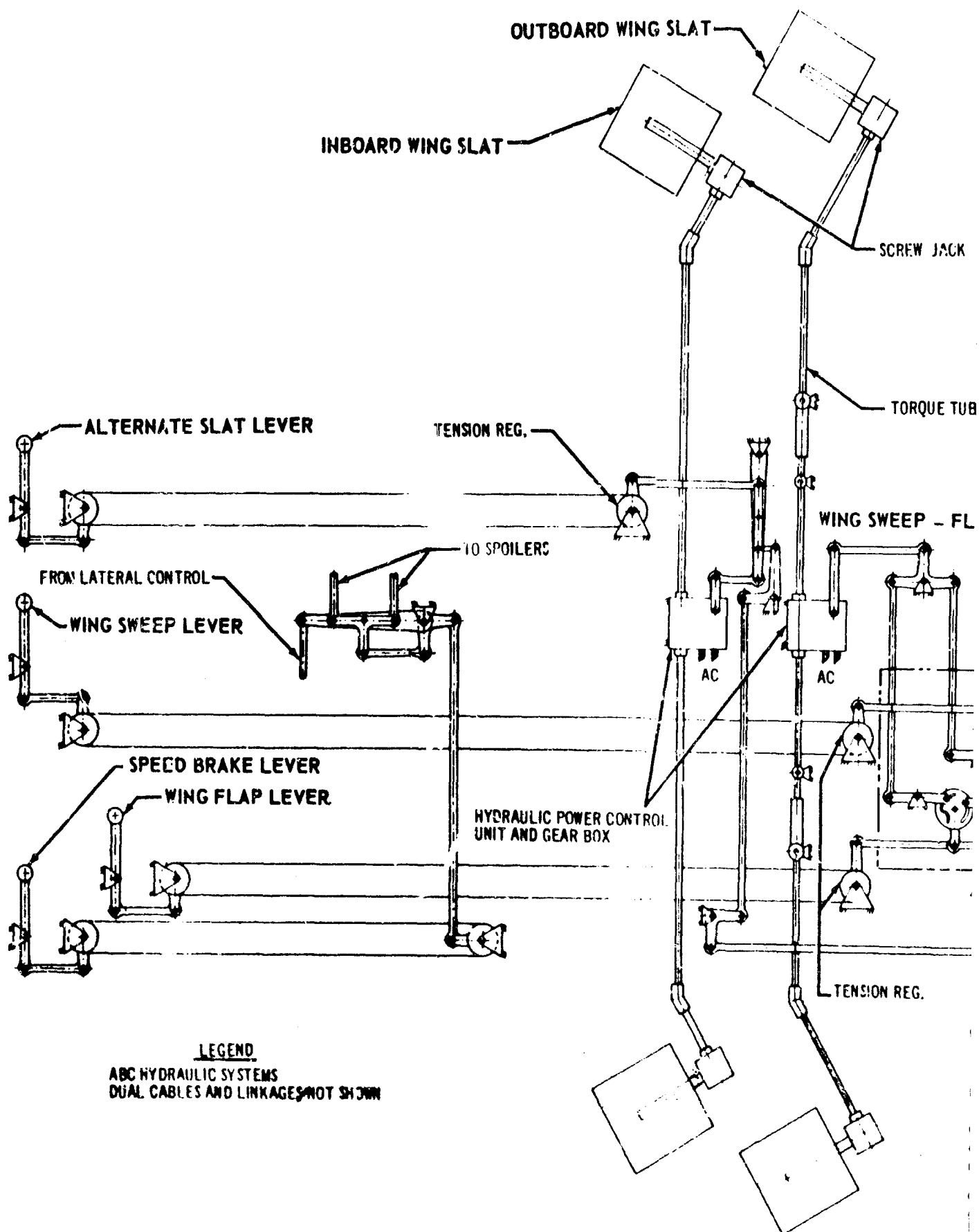
The actuation system for the outboard flaps of the outboard wing is nearly identical to the foregoing system and is also driven by three hydraulic motors pressurized by the three hydraulic systems.

An electric asymmetry detection and control system containing two independent electric sensor elements at each extreme end of the torque tube drives, and a comparator located in the environmentally controlled electronic bay are provided for each of the three flap drive systems. Hydraulic asymmetry shutoff valves are provided for each hydraulic pressure port of the flap drive control valves.

6.2.5 Leading Edge Slat System The leading edge slat system, shown in Fig. 13, consists of two separate drive systems: one for inboard wing slats, and one for outboard wing slats.

The slat system is normally controlled by the wing sweep and flap mechanical programmer, mechanical cables, and linkage to the two mechanical-input hydraulic servo valves. The alternate slat lever, mounted on the pilots' aisle stand and connected to the inboard wing slat control valve through mechanical cables and linkage, provides a means for extending inboard wing slats in the event of a wings-aft landing.

The inboard wing slats are actuated by a power-drive gearbox driven by two hydraulic motors pressurized by hydraulic systems A and C. The output shaft of the gearbox is connected to the inboard slat surfaces through torque tube drives and ball bearing jack screws.



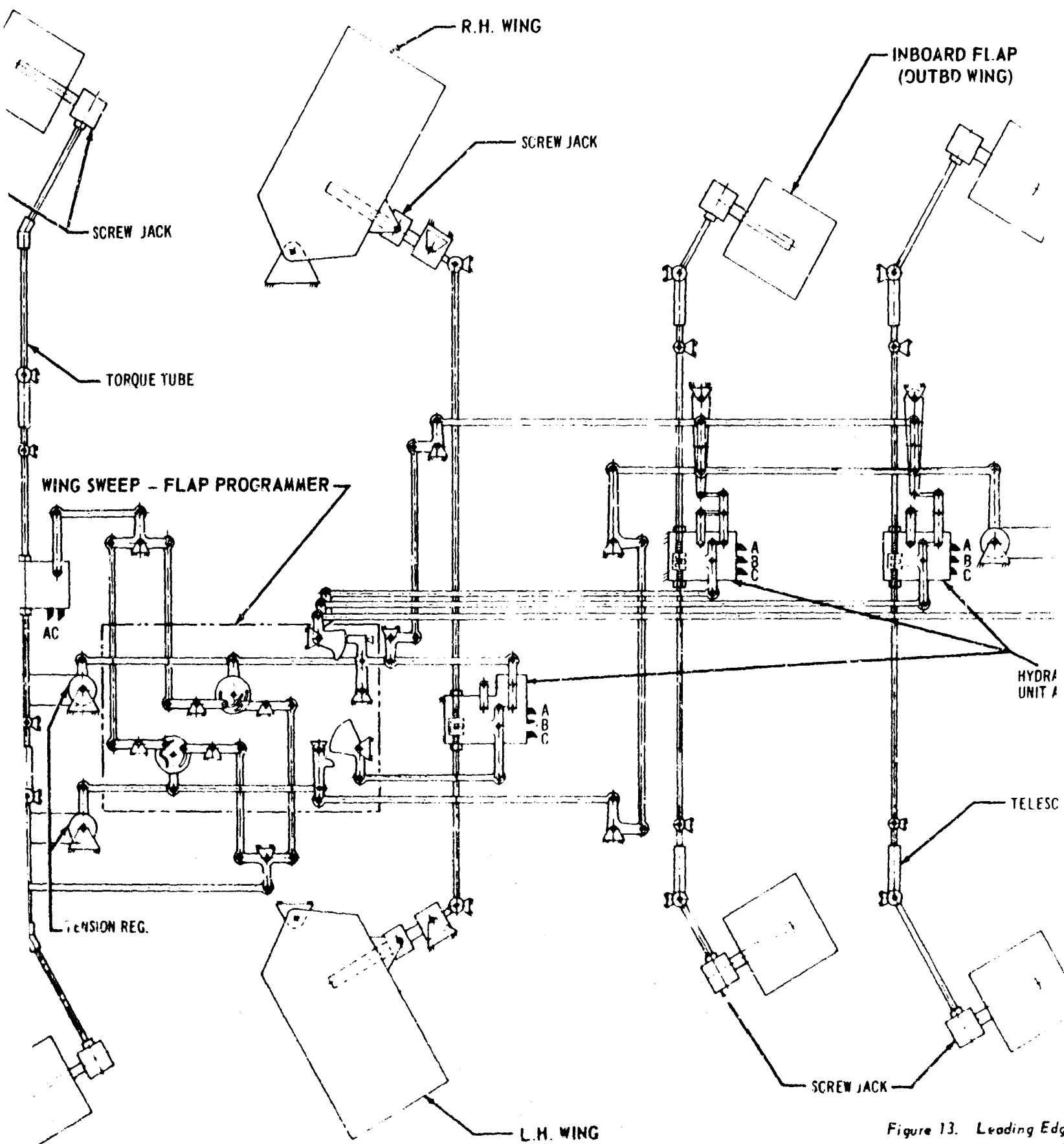


Figure 13. Leading Edge and Speed-B

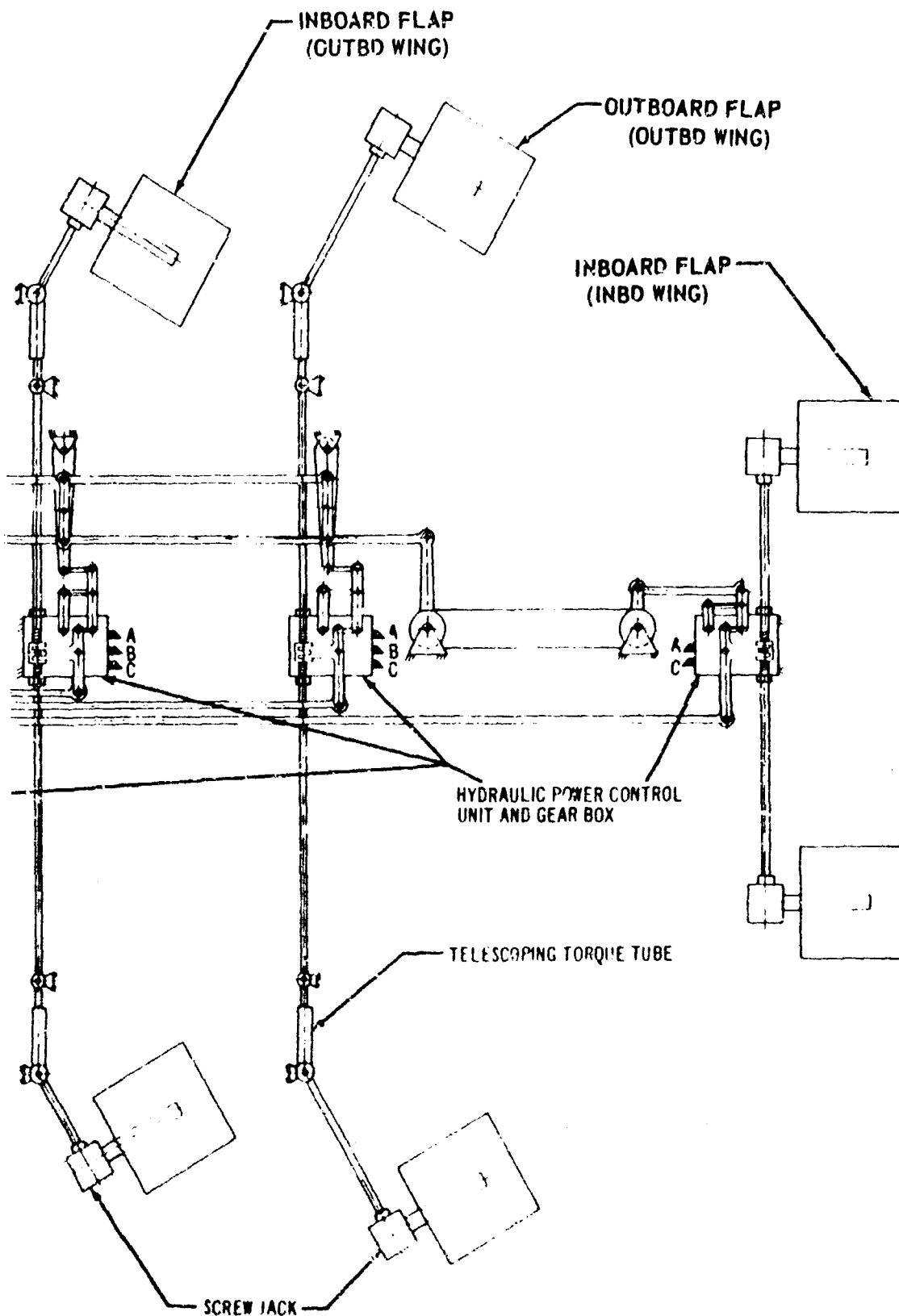


Figure 13. Leading Edge Slats, Wing-Sweep, Trailing Edge Flaps, and Speed-Brake Control Systems Schematic

The outboard wing slats are also actuated by a power-drive gearbox driven by two hydraulic motors pressurized by hydraulic systems A and C. The output shaft is connected to the outboard slat surface through torque tube drives routed around the wing pivot through linear ball bearing splines and swivel bevel gearboxes similar to the outboard wing flap system. Torque tube drive forces are transmitted to the slat surfaces through bevel gears and ball bearing jack screws similar to the other slat and flap components.

An electric asymmetry detection and control system, similar to those provided for each flap drive system, is provided for each slat drive system.

6.2.6 Speed Brake Control System. The speed brake control system, shown in Fig. 13, consists of the speed brake lever mounted on the pilots' aisle stand and mechanical cables connected to the spoiler programmer unit which allows differential motion commands for lateral control to be superimposed on speed brake commands to the spoilers.

6.3 Automatic Flight Controls System.

6.3.1 Autopilot. The A/P system consists of three A/P-F/D pitch-roll computers, the A/P flight director (F/D) mode selector control panel, and the manual control panel. Each computer contains the electronics to provide modes of operation identified in Figs. 14 and 15 which also define the input information required for each mode. The system interface data are as described in Par. 6.5. The mode selector control panel shown in Fig. 16 provides the means for engaging the A/P in the pitch and roll axes and for selecting the modes of operation. The manual control panel shown in Fig. 17 provides the means for manually controlling the airplane through the A/P. The use of these panels for control of A/P operation is explained in Par. 8.3.1. The panel shown in Fig. 18 provides the required engage/disengage switches and monitoring indicators for autopilot and autothrottle systems.

6.3.2 Autothrottle The autothrottle (A/T) system consists of a computer, control panel and servo. The computer contains electronics to process command inputs and feeds signals to the A/T servo in Mach Hold and Air Speed Hold as shown in Fig. 19. The A/T control panel provides the capability to select the mode of operation. The servo provides mechanical operation of the throttle system. The A/T is engaged by controls on the AFCS control panel shown in Fig. 18.

The system interface information identified in Fig. 19 is described in detail in Par. 6.5.

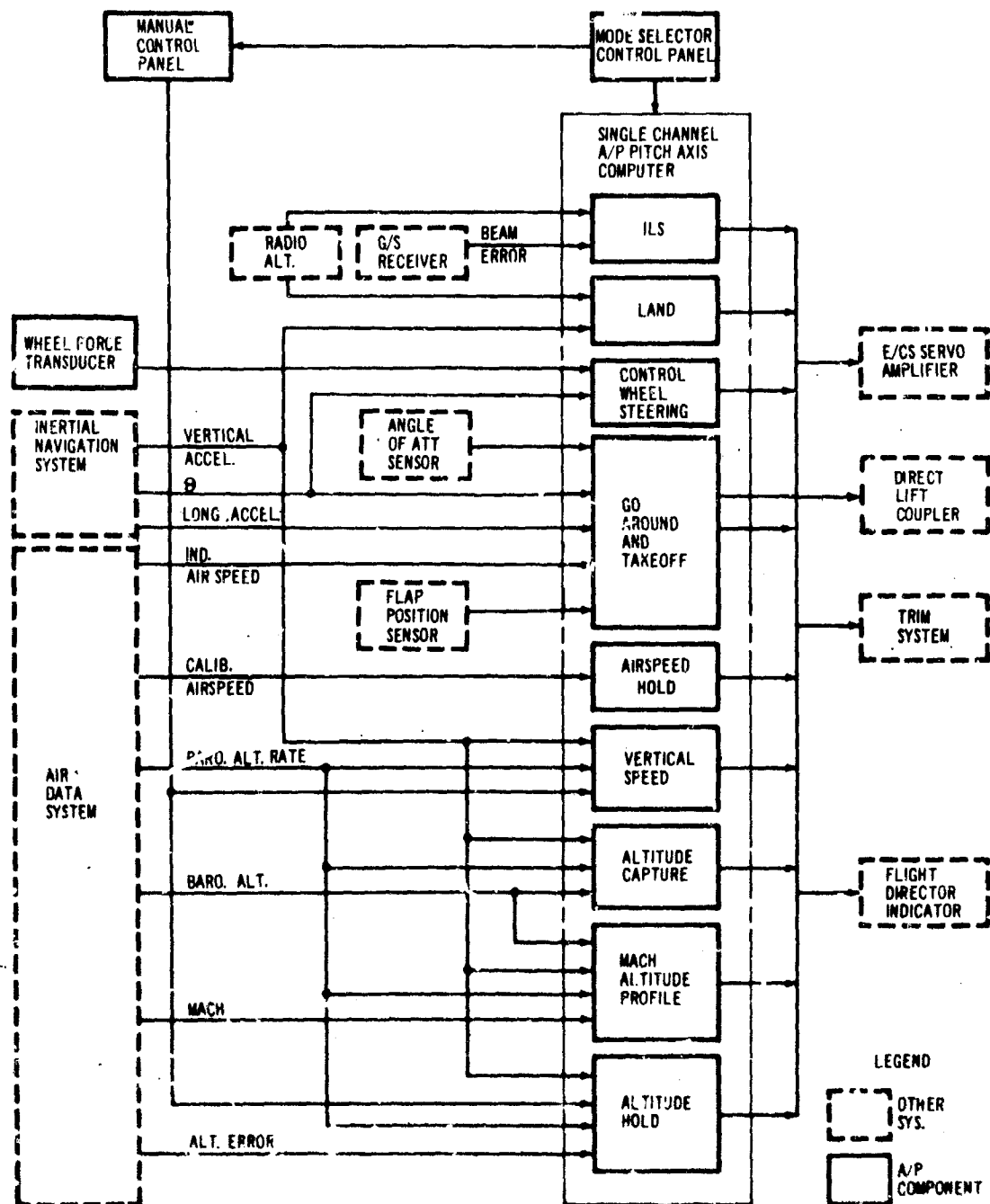


Figure 14. Autopilot Pitch Axis Diagram

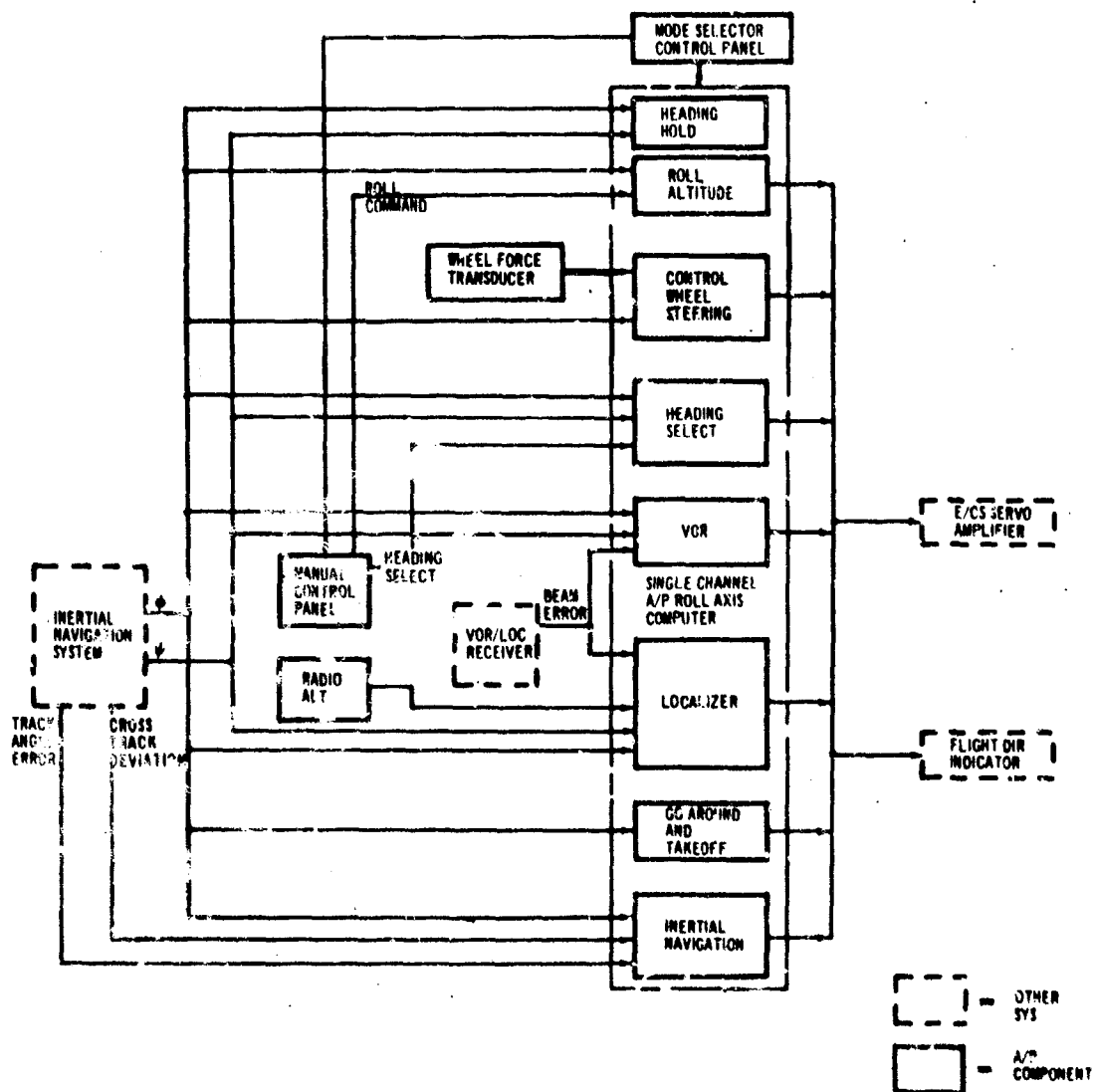


Figure 15. Autopilot Roll Axis Diagram

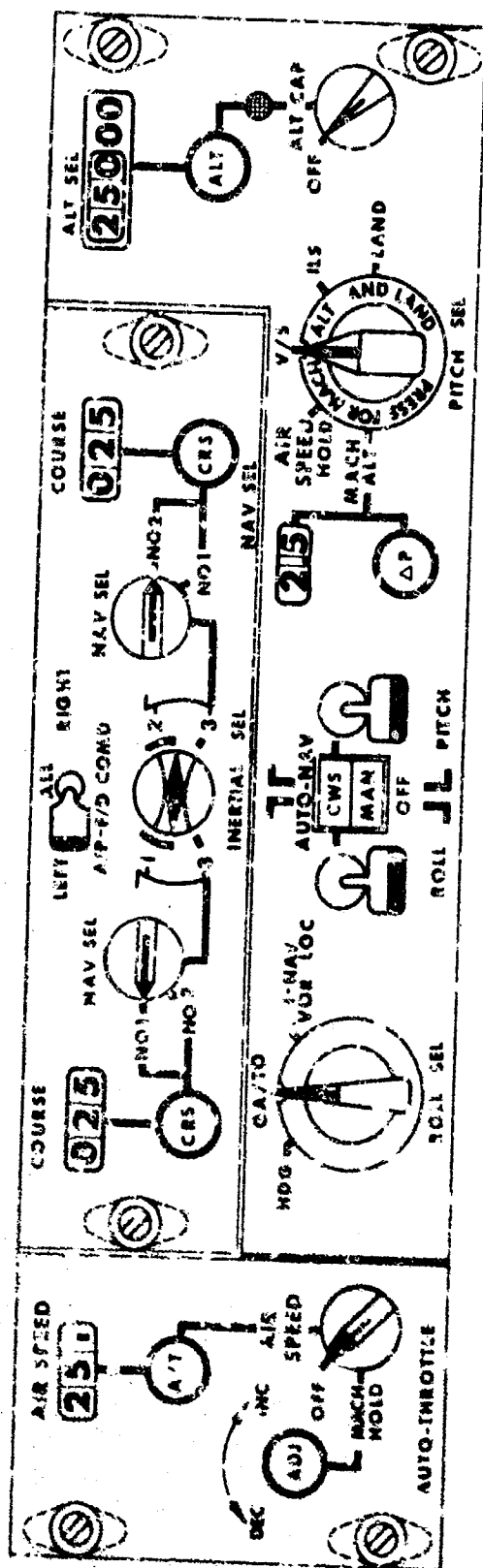


Figure 16. A/P-F/D Mode Selector Control Panel

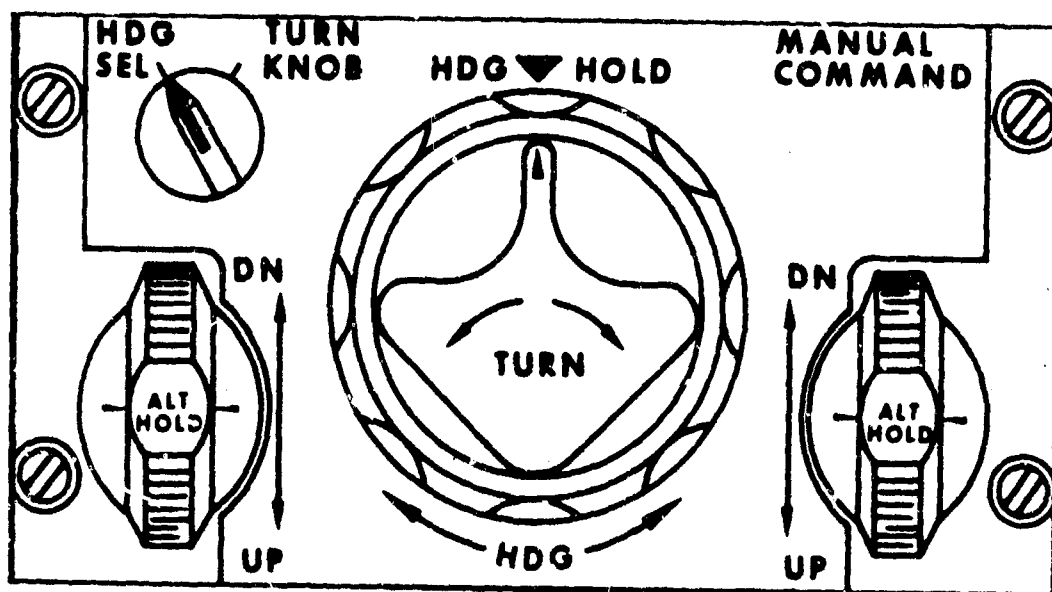


Figure 17. Manual Control Panel

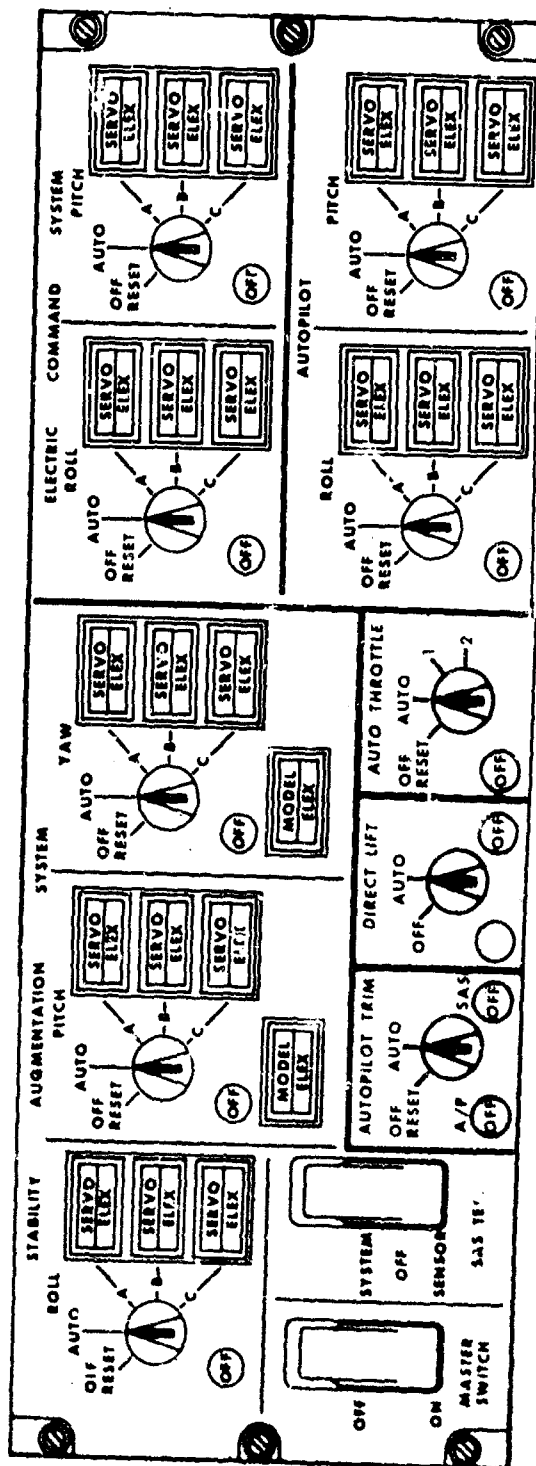


Figure 18. AFCS Control Panel

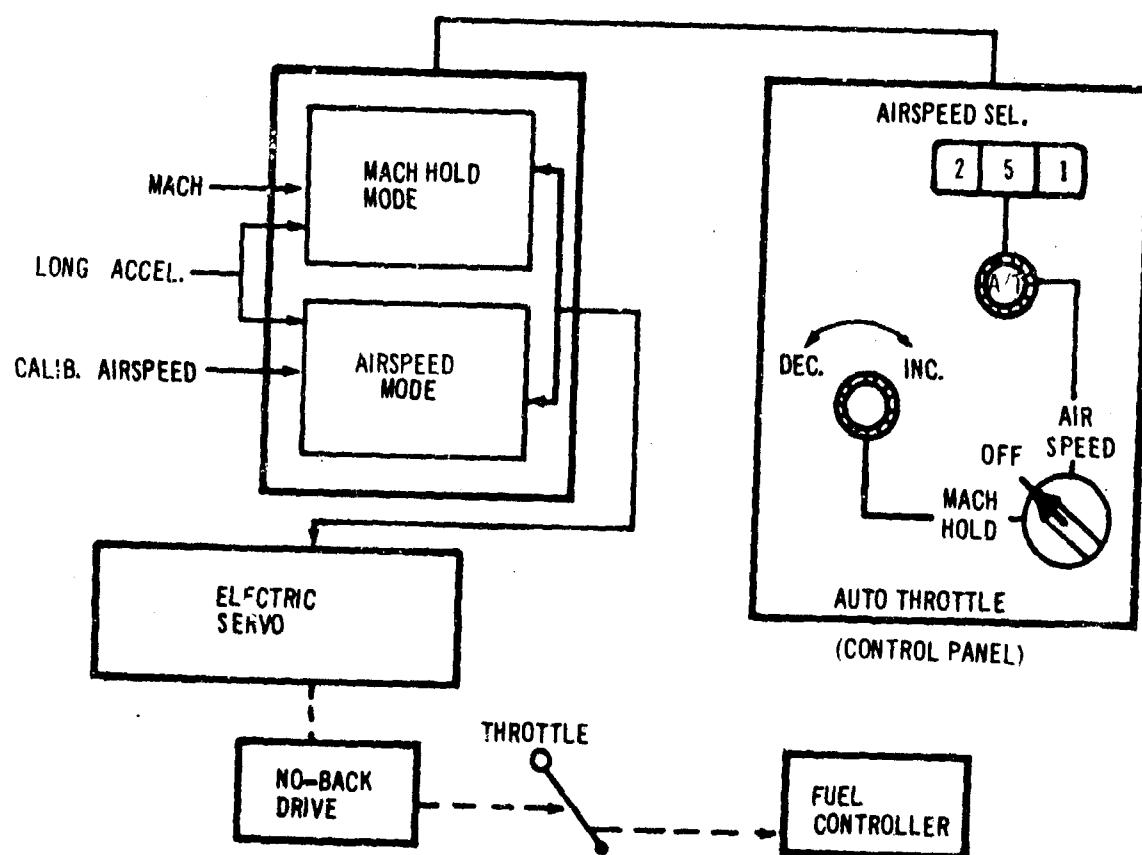


Figure 19. Autothrottle Functional Block Diagram

6.3.3 Stability Augmentation System. A full-time stability augmentation system with separate pitch, roll and yaw axes is provided. The system consists of computers for each axis, feedback sensors, a lateral accelerometer package, and servos as shown in Figs. 20, 21, and 22. The computers contain electronics to process command inputs and feed signals to the SAS servos and, with wing-sweep position at 20 deg, to the direct lift control coupler and thence to the direct lift control servos. The computers use dc mechanization techniques to provide the accuracy and capability required for filter networks. The use of multiple channels of electronics, sensors and triple force-summed SAS servos provides the required fail-operational characteristics. The system interface data are described in Par. 6.5.

The lateral accelerometer package, used for the yaw axis augmentation, is located near the CG. The rate gyros are located within the computers. The AFCS control panel shown in Fig. 18 provides the required engage/disengage switches and monitoring indicators.

6.3.4 Angle-of-Attack Warning and Control System. The angle-of-attack warning and control system consists of a stick shaker and automatic pitch-down system as shown in Figs. 23 and 24, respectively. Two stick shaker systems are provided. The electronics for the systems are located in the E/CS computer. The control panel, shown in Fig. 25, is located adjacent to the AFCS control panel and contains the engaging switches and monitor indicators. The system interface data are as described in Par. 6.5.

6.4 Hydraulic Power System. The utility and flight controls power system consists of three basic hydraulic systems ("A", "B", and "C"), one standby hydraulic system, and one control panel. Figs. 26, 27, 28, and 29 show the hydraulic power and flight control systems arrangement. See specification D6A10108-1 for diagrams and descriptions of the landing gear retraction, steering, and brake control systems. The three hydraulic systems are completely independent with no hydraulic interconnection between systems.

All hydraulic system components except pumps and heat exchangers and their associated equipment are located in the hydraulic equipment bay. The distribution system branches out from this central area. Hydraulic lines on the movable wing are routed through individual system swivels at the wing pivot. Line routing is designed to avoid engine burst pattern areas.

6.4.1 Basic Hydraulic System Description. Each hydraulic system is composed of the following elements as shown in Fig. 26:

- a. Positive displacement, variable volume pumps mounted on accessory drive gearboxes (ADS). Systems "A" and "B" each have three pumps and system "C" has two pumps. Each pump incorporates an electrically operated depressurization valve. The pump installation has self-sealing disconnects installed in the hoses leading from each port to the tubing mounted on

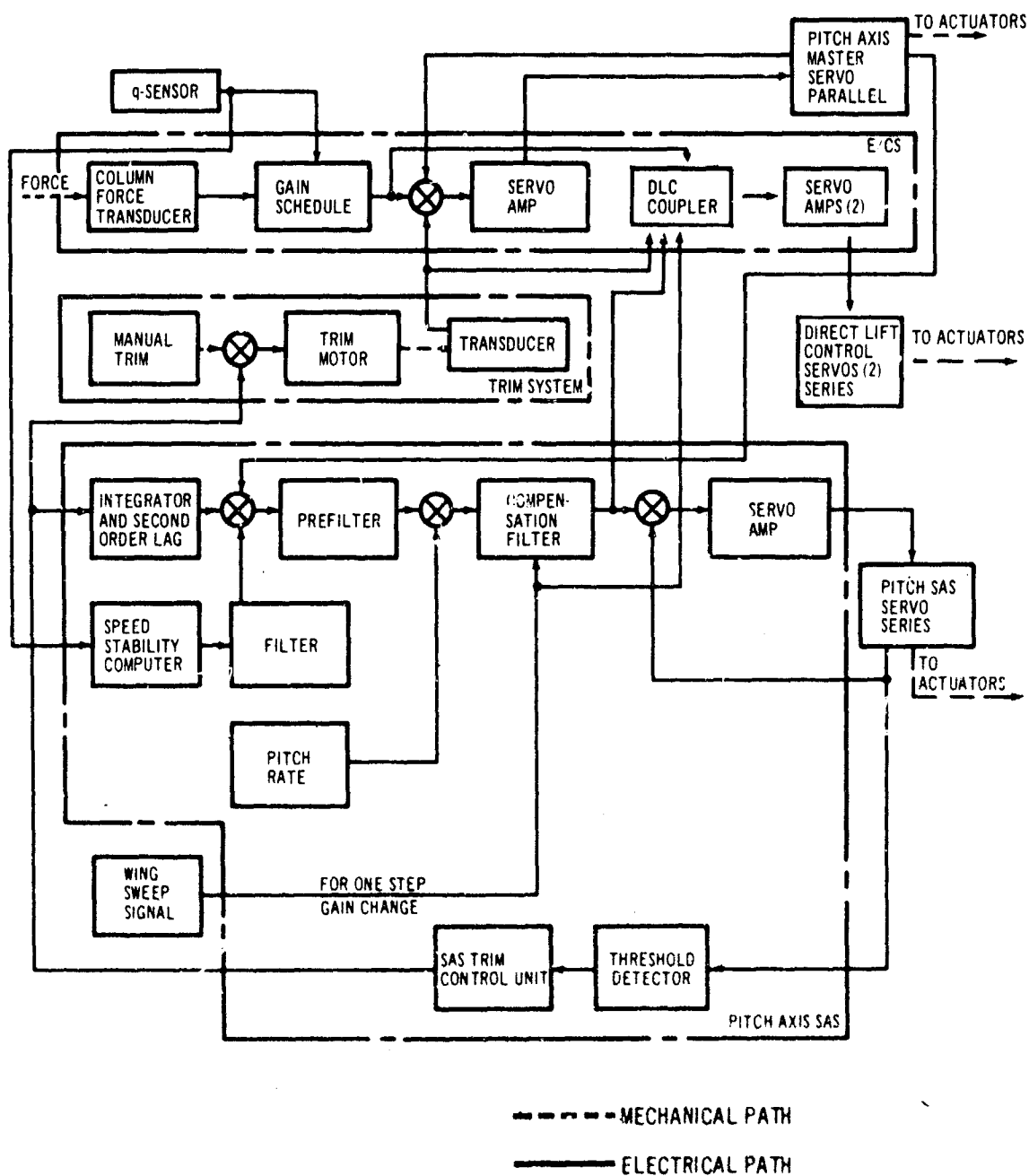


Figure 20. Pitch Axis Stability Augmentation System With E/CS and Trim

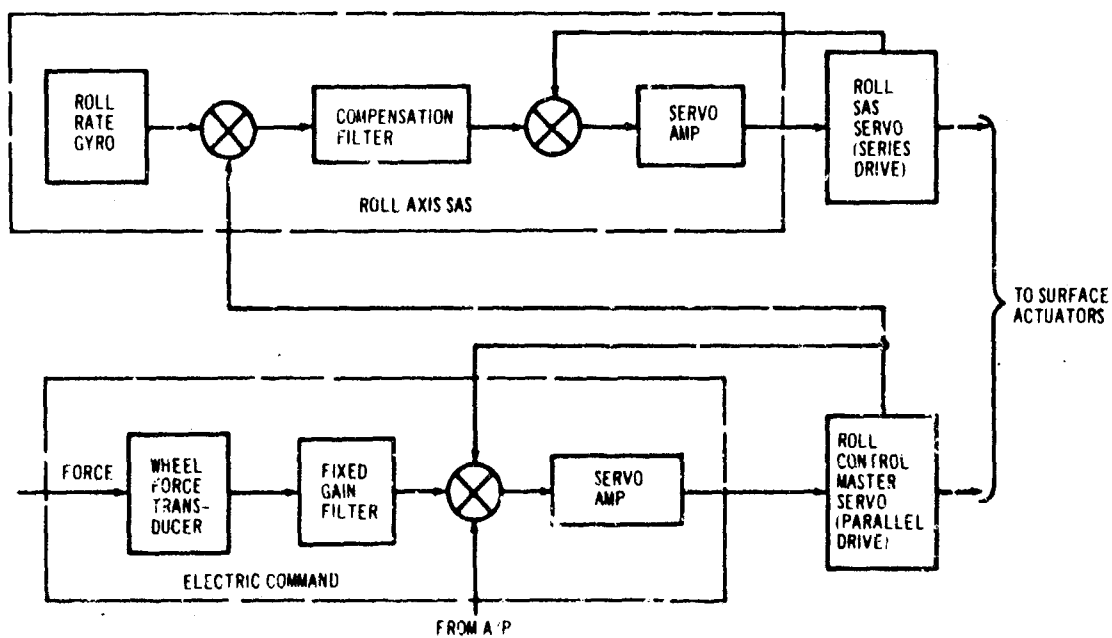


Figure 21. Roll Axis Stability Augmentation System With E/Cs

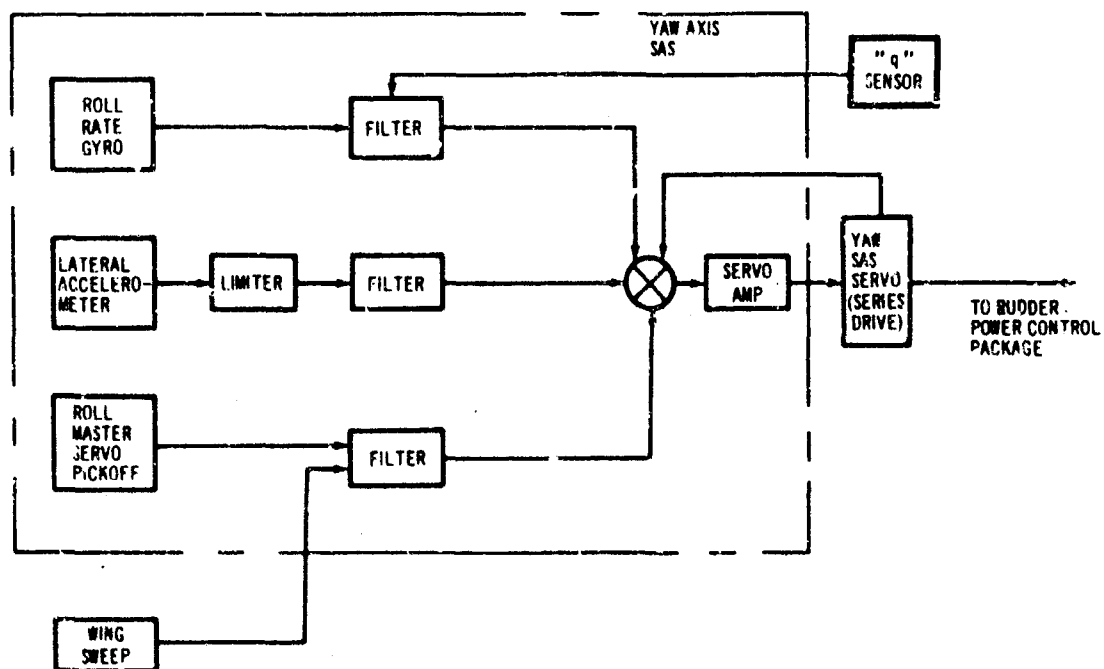


Figure 22. Yaw Axis Stability Augmentation System

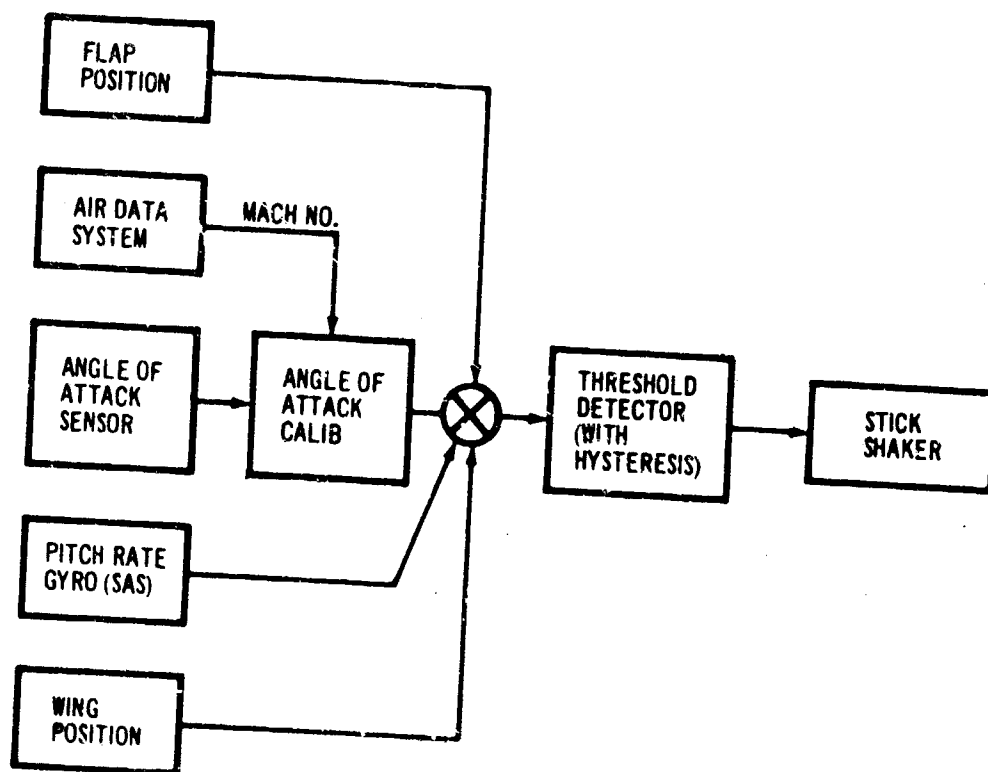


Figure 23. Stick Shaker System

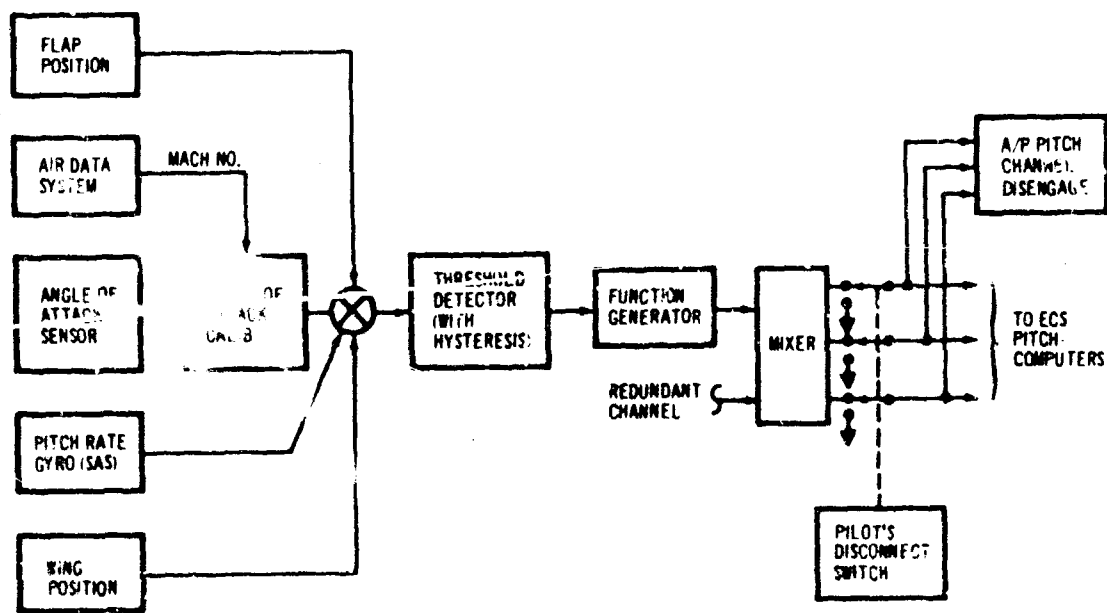


Figure 24. Automatic Pitch-Down System

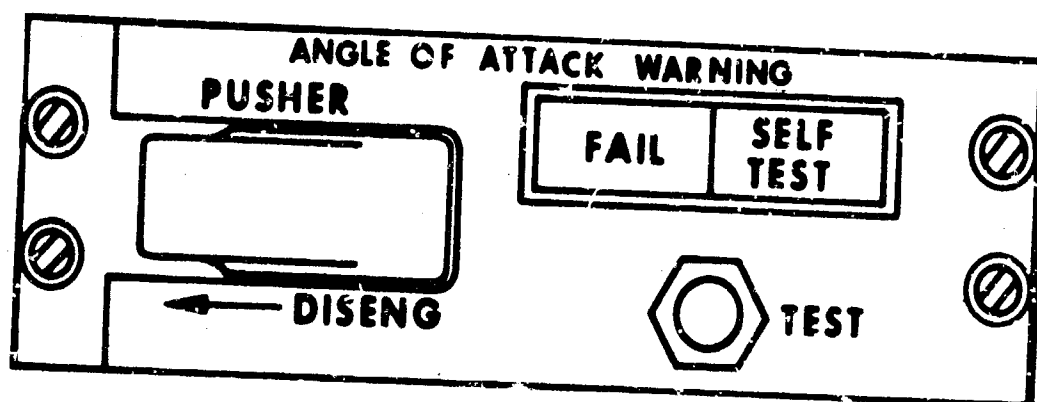
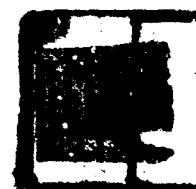
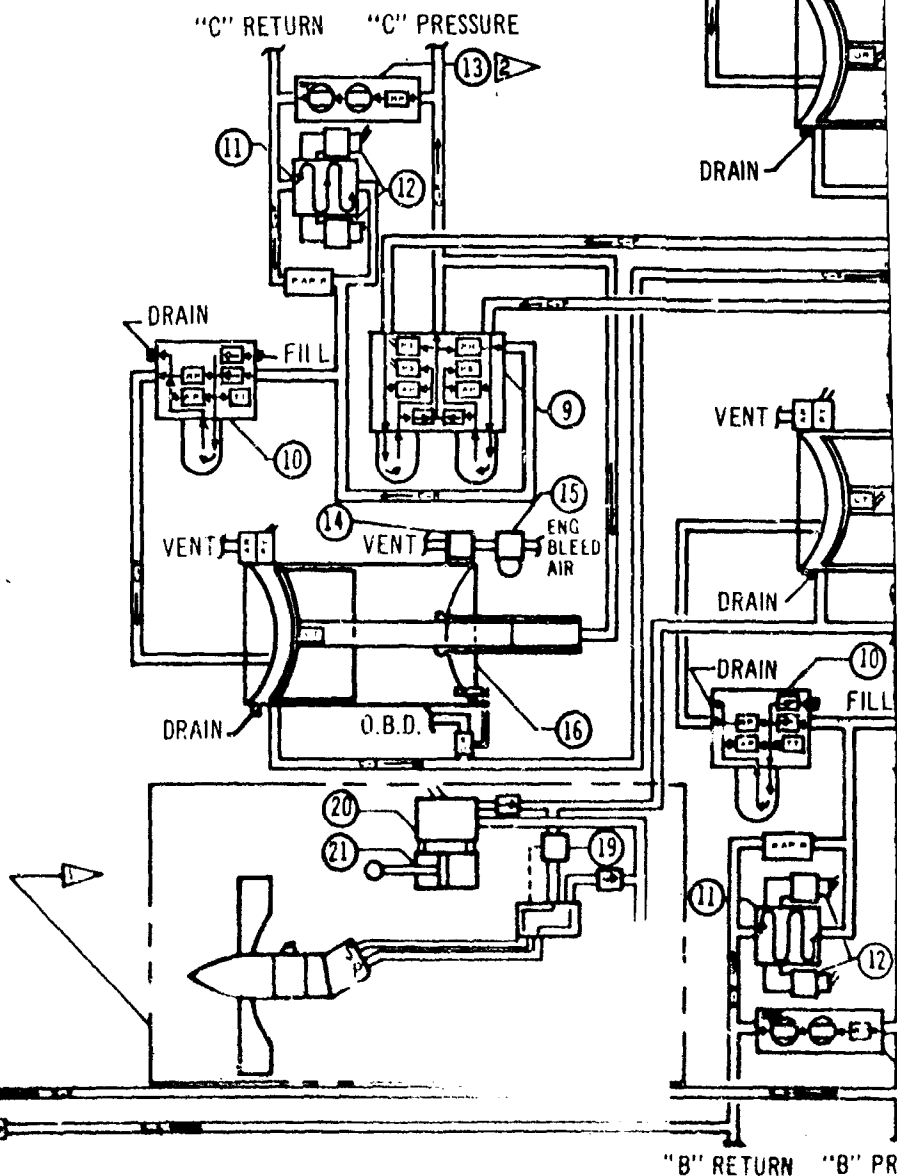
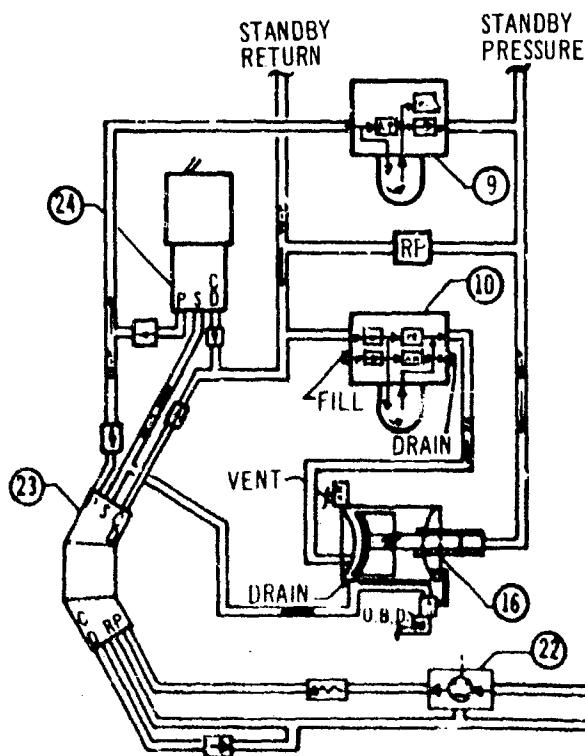


Figure 25. Angle-of-Attack Warning and Control System Panel

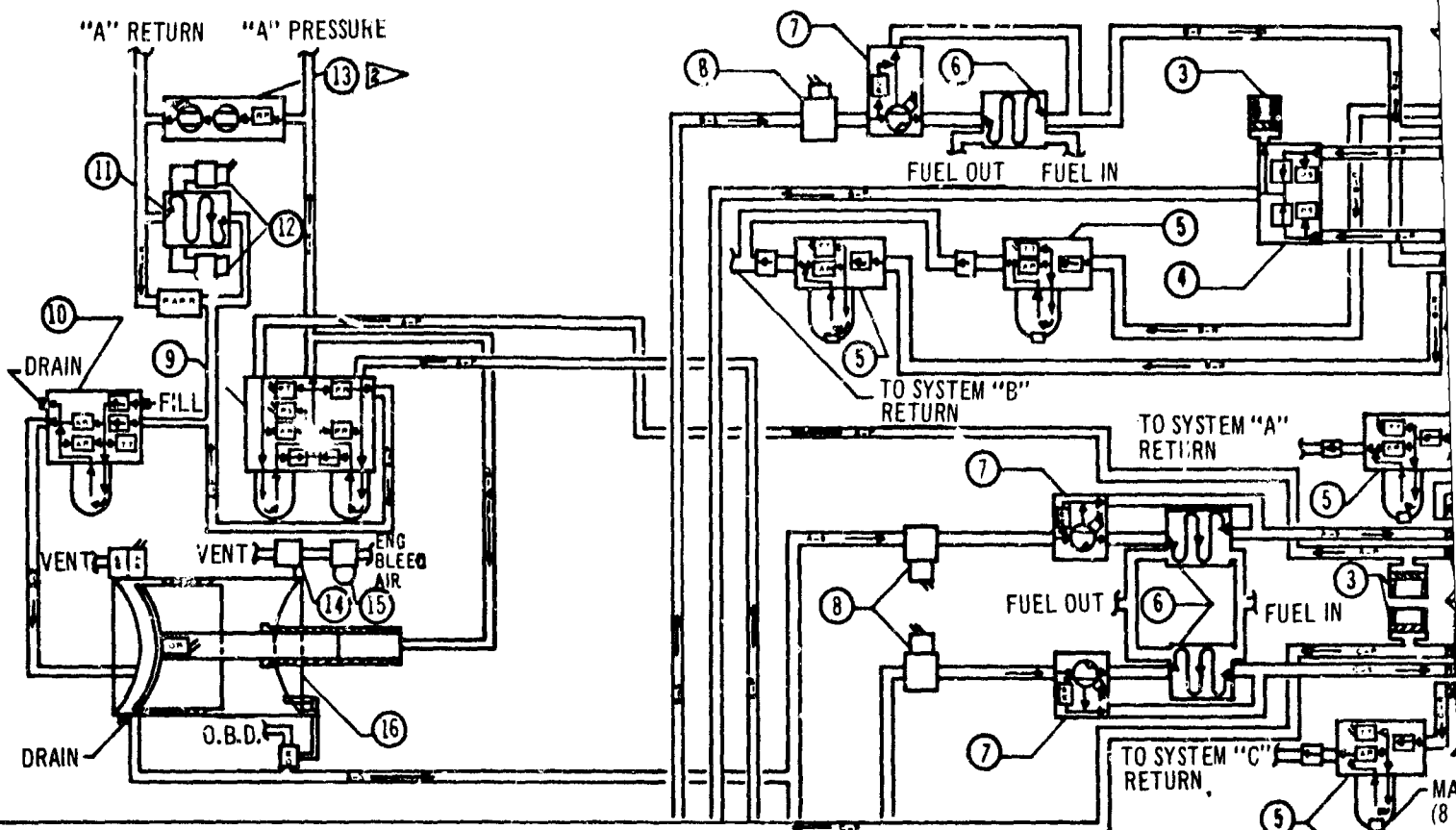
ITEM	PART NAME
1	HYDRAULIC PUMP - ENGINE DRIVEN
2	DISCONNECT - SELF SEALING
3	DAMPER - PULSATION
4	PRESSURE MODULE - PUMP
5	FILTER MODULE - CASE DRAIN
6	HEAT EXCHANGER - FUEL / OIL
7	CONTROL MODULE - HEAT EXCHANGER
8	VALVE - SUPPLY SHUT - OFF
9	FILTER MODULE - PRESSURE
10	FILTER MODULE - RETURN
11	HEAT EXCHANGER - AIR / OIL
12	AIR SCOOP - ELEC MOTOR DRIVEN
13	WARM-UP MODULE (SEE POSITIONS) 
14	PRESSURE REGULATOR
15	AIR FILTER
16	RESERVOIR - BOOT STRAP
17	HYDRAULIC PUMP - RAM AIR TURB DRIVEN 
18	SWIVEL 
19	VALVE - SHUT - OFF, DOOR GPRD 
20	VALVE - R. A. T. CONTROL 
21	INTERNAL LOCKING ACTR, R. A. T. 
22	VALVE - CONTROL, GEAR OPERATED
23	HYDRAULIC PUMP - HYDRAULIC MOTOR DRIVEN
24	HYDRAULIC PUMP - ELECTRIC MOTOR DRIVEN

SYMBOLS	
	PRESSURE
	RETURN
	SUPPLY
	HOSE
	MECHANICAL
	ELEC CONNECTION
	OVERBOARD DRAIN
	VALVE - CHECK
	VALVE - FLOW LIMITER
	VALVE - PRESSURE RELIEF
	VALVE - AIR RELEASE
	VALVE - DIFFERENTIAL PRESSURE RELIEF
	VALVE - QUANTITY RELIEF PISTON ACTUATED
	VALVE - THERMOSTAT CONTROLLED
	VALVE - ELEC OPERATED
	INDICATOR - DIFFERENTIAL PRESSURE
	TRANSMITTER - TEMP
	TRANSMITTER - LEVEL
	TRANSMITTER - PRESSURE
	SWITCH - PRESSURE
	AIRPLANE WITH GENERAL ELECTRIC ENGINES ONLY

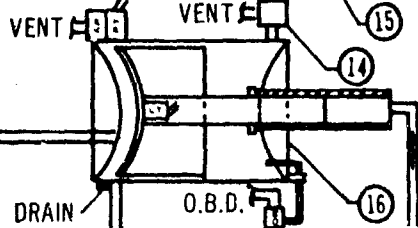


PPLY

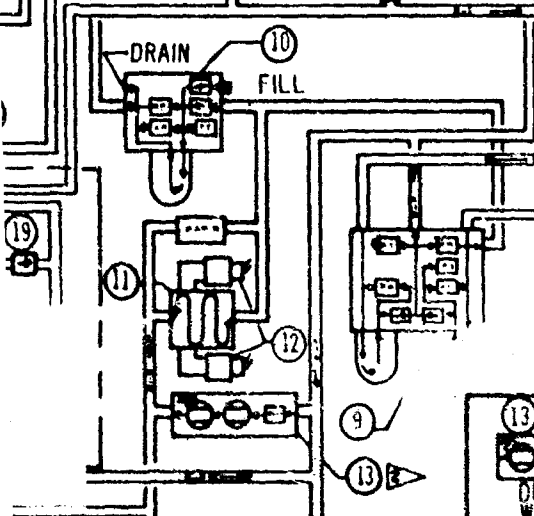
"A" RETURN "A" PRESSURE



ENG BLEED AIR



DRAIN FILL



"B" RETURN "B" PRESSURE

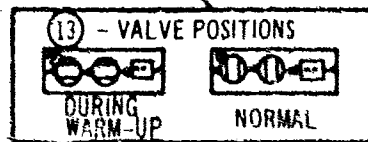


Figure 26. Hydraulic Diagram, Power System



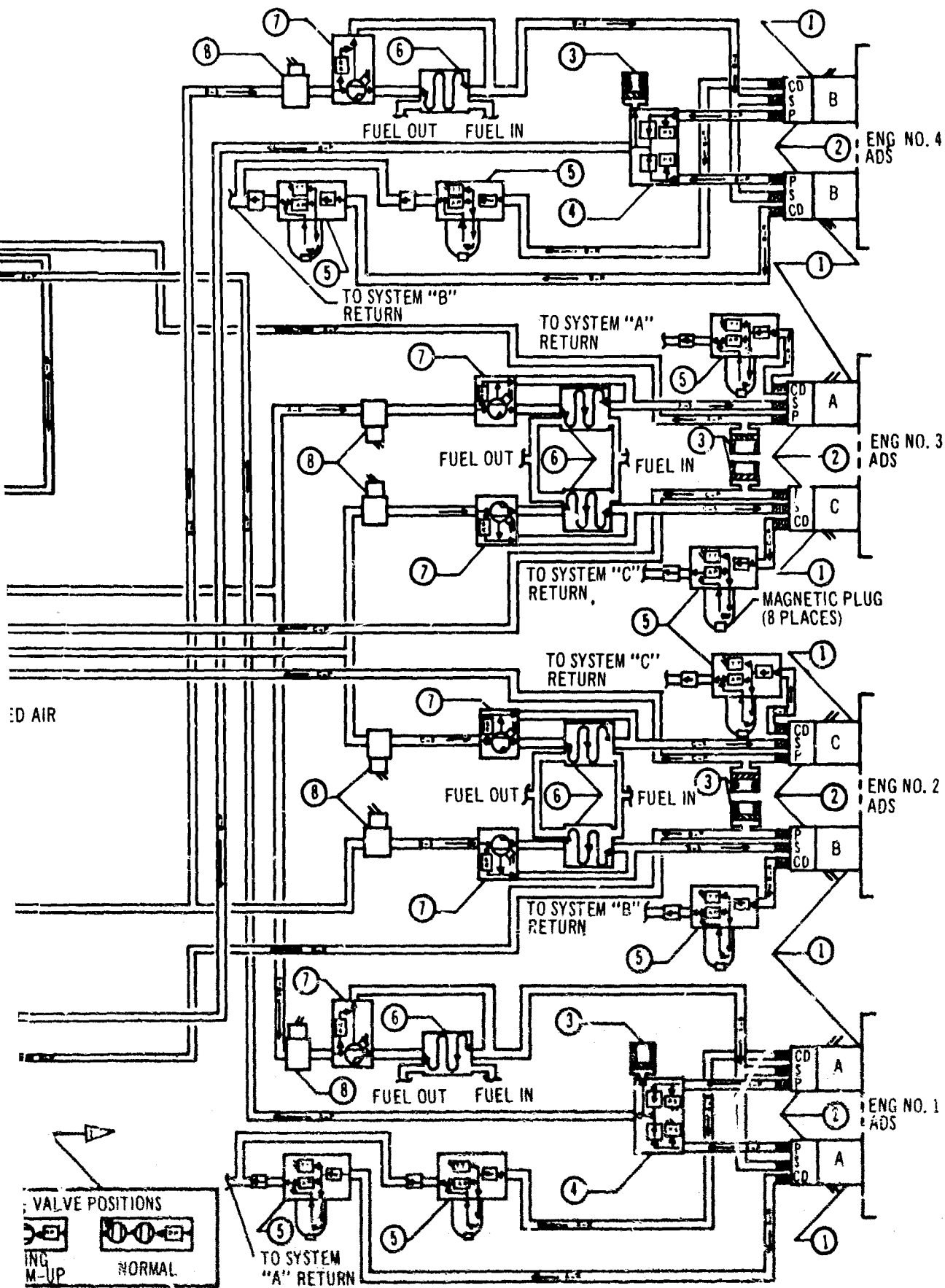
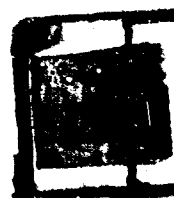
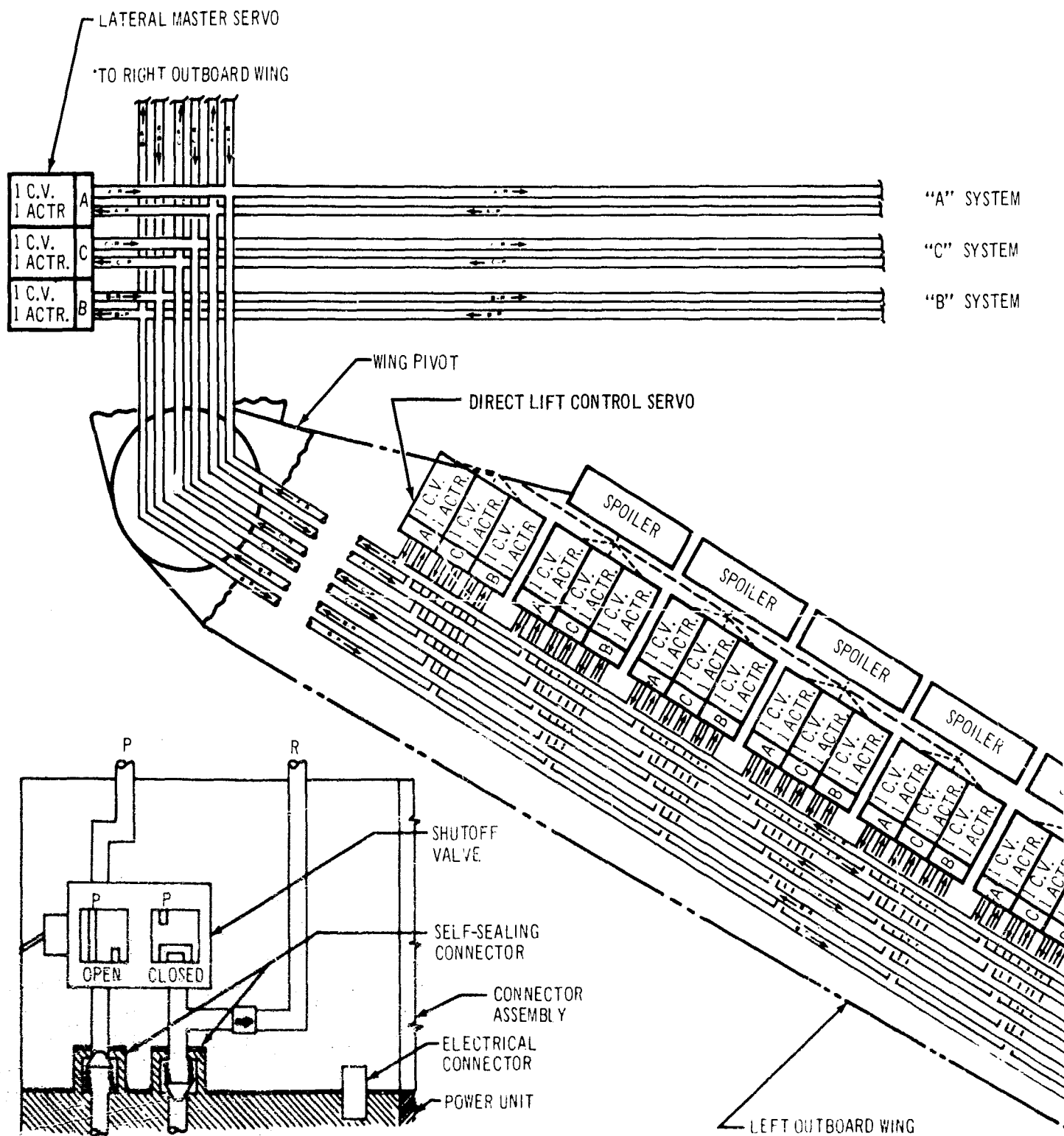


Figure 26. Hydraulic Diagram, Power Systems



TO RIGHT HORIZONTAL STABILIZER

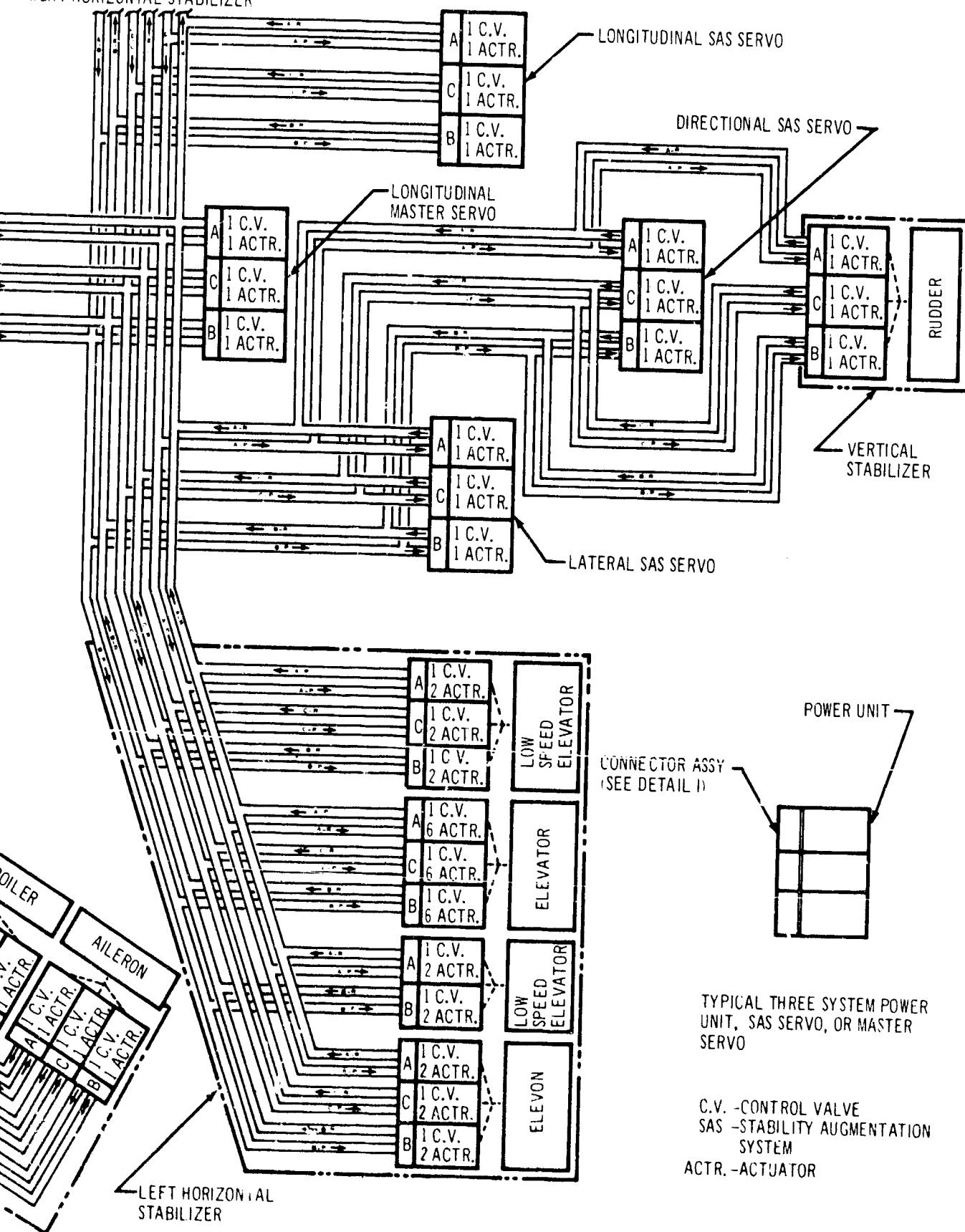
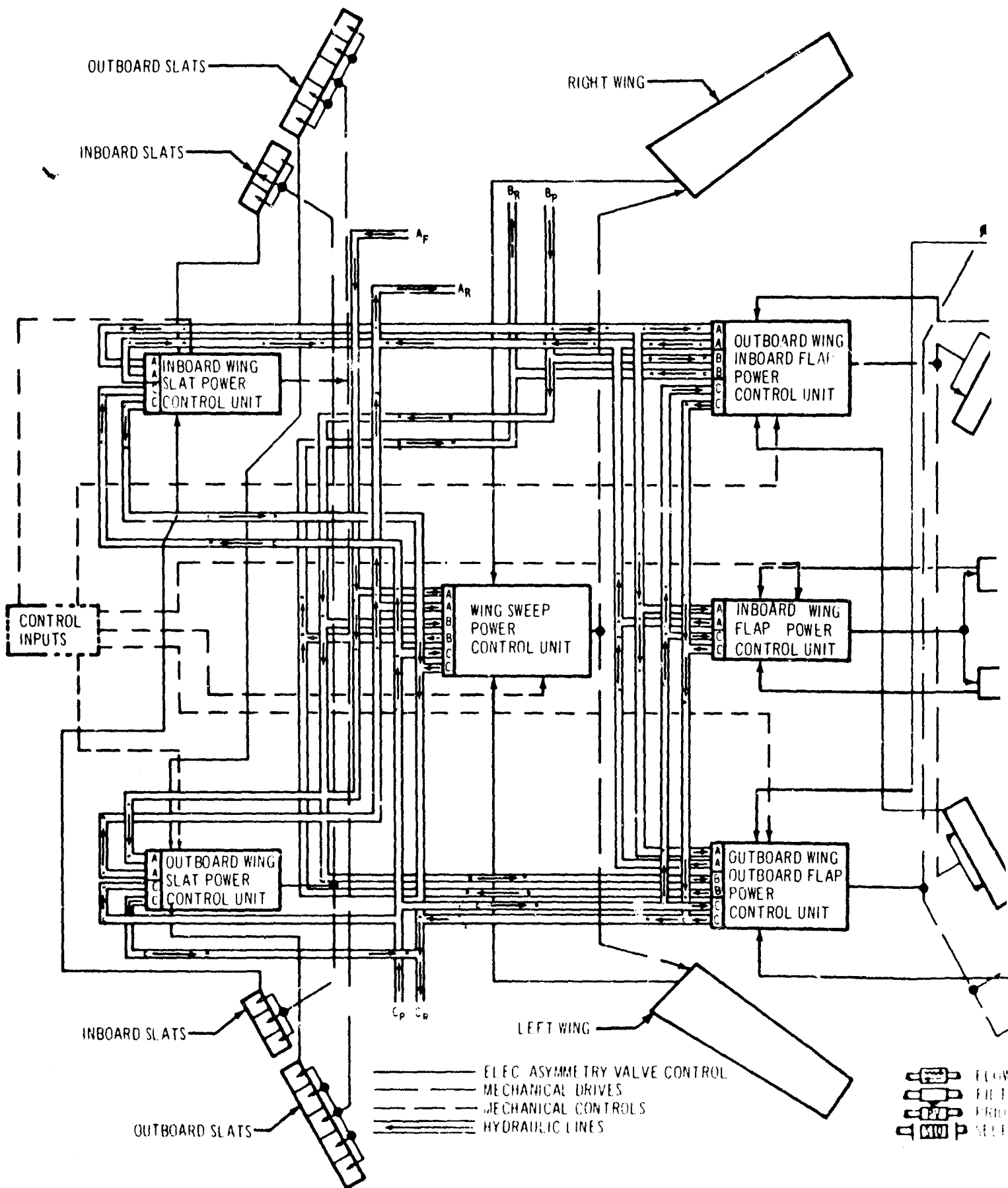
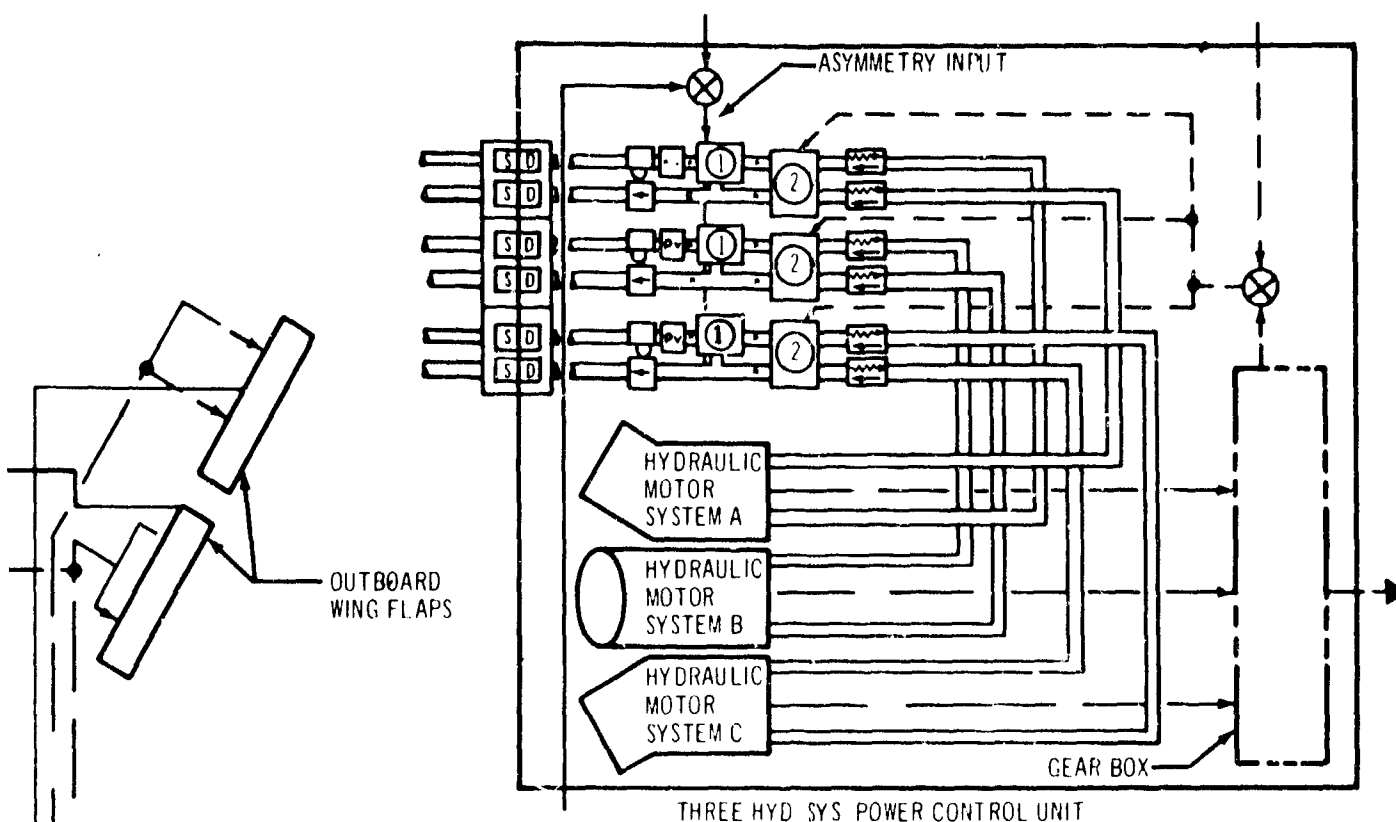


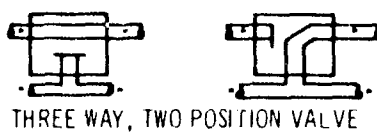
Figure 27. Hydraulic Diagram, Primary Flight Controls

D6A10120-1

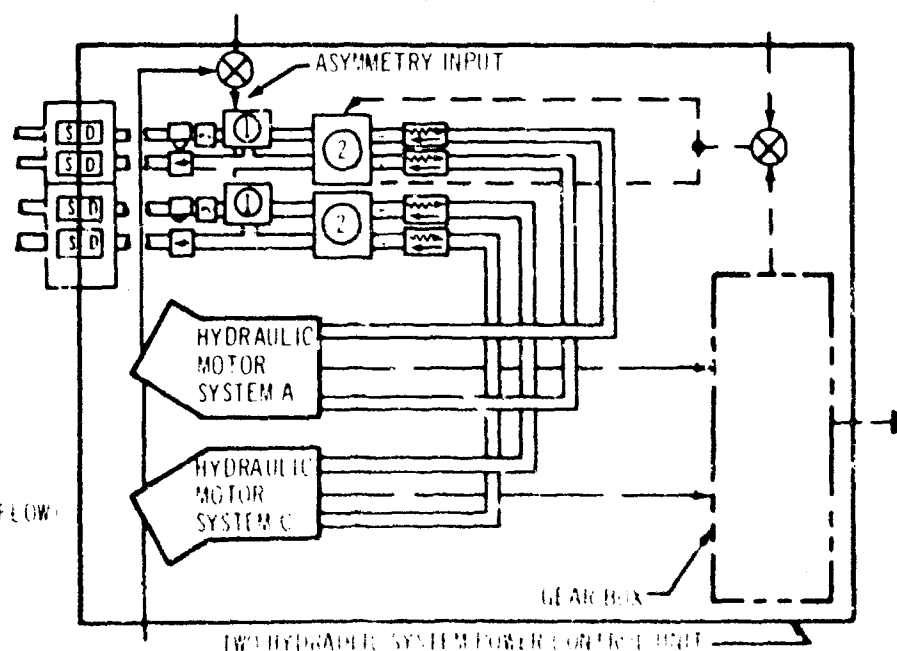




① ASYMMETRY VALVE



② CONTROL VALVE



FLOW LIMITER / FREE RETURN FLOW
FILTER
PRIORITY VALVE
SELF SEALING DISCONNECT

Figure 28. Secondary Flight Control and Wing-Sweep Hydraulic System Schematic

06410120-1

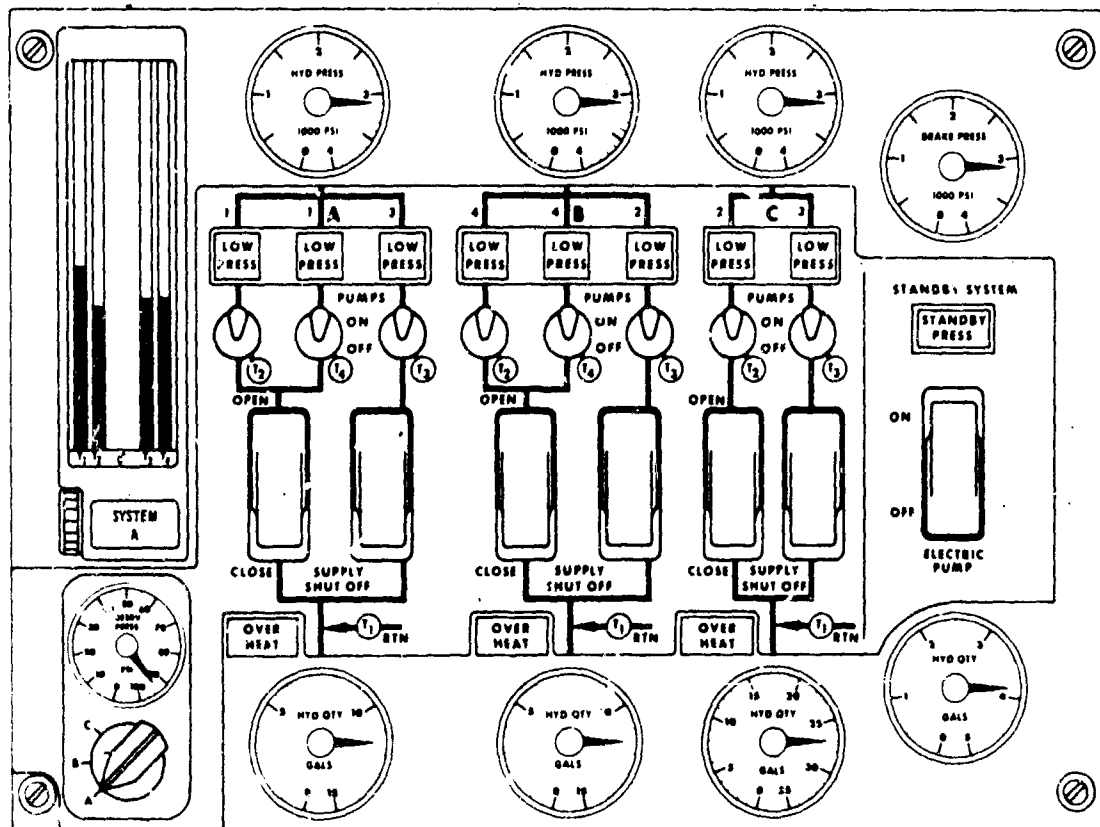


Figure 29. Flight Engineer's Hydraulic Control Panel

the nearest structure. This installation reduces to a minimum, possible loading on the pump and lines due to position tolerances and relative motion between the pump and hard lines. The self-sealing disconnects allow pump replacement without loss of fluid.

- b. One separate type reservoir with hydraulic "boot strap" and engine bleed air pressurization.
- c. One return filter module incorporating one two-stage bypass type filter, a clogged-element warning device, a temperature transmitter, system fill and drain connections, and two check valves.
- d. One pressure filter module with two non-bypass type filters, two clogged-element warning devices, the system relief valve, a pressure transmitter, two pressure switches, and two check valves.
- e. One case drain filter module for each pump incorporating one non-bypass type filter, a magnetic chip collector plug, a clogged-element warning device, a temperature transmitter, and a check valve.
- f. Two electrically operated supply shutoff valves.
- g. Two fluid-to-fuel heat exchangers.
- h. One fluid-to-air heat exchanger with electric motor driven retractable air scoops.
- i. Two heat exchanger control modules, incorporating one electrically operated flow-control valve and one pressure operated bypass valve.
- j. Two pump pulsation dampers.
- k. One system warmup module, incorporating a pressure relief valve, a thermostatically controlled valve and an electrically operated shutoff valve.

In addition, systems "A" and "B" each incorporate one pump pressure module, containing two pressure switches and two check valves. Also, system "B" contains a fixed displacement hydraulic motor and a control valve for powering the standby system.

System "A" is connected to all the flight control power units. System "B" is connected to all the flight control power units except the inboard wing slat power unit, the outboard wing slat power unit, and the inboard wing flap power unit. System "C" is connected to the landing gear retraction and extension system, the brake control and antiskid system, the nose gear steering system, and to all the flight control power units except the outboard auxiliary elevator power units.

6.4.2 Standby Hydraulic System Description. The standby hydraulic system is shown on Fig. 25. The system is composed of the following elements:

- a. One fixed displacement pump driven by a hydraulic motor.
- b. One ac electric motor-driven pump, of the variable-volume type.
- c. One separated, "boot strap" type reservoir.
- d. One return filter module, incorporating one two-stage bypass type filter, a clogged-element warning device, system fill and drain connections, and two check valves.
- e. One pressure filter module with one non-bypass filter, a clogged-element warning device, a pressure switch, and a check valve.

The standby system obtains its power from a positive displacement fixed volume pump driven by a hydraulic motor in the B system. This hydraulic motor is operated only when the landing gear handle is down and the B system is pressurized. A second standby power source, an AC electric motor driven pump, is available for ground operation and as a second means of supplying power to the standby system should B and C systems fail. The elements in the standby system are similar to those in the basic system.

The standby hydraulic system is connected to the standby landing gear extension system, the standby brake control and antiskid system, and the aft main gear steering system.

6.4.3 All-Engine-Cut Hydraulic Power Supply Description.

6.4.3.1 The 2707-100 (GE). The airplane incorporates an independent all-engine-out power supply for the "B" hydraulic system, shown schematically on Fig. 26.

The all-engine-out power supply consists of the following elements:

- a. One ram-air turbine-driven constant horsepower positive-displacement variable-volume hydraulic pump.
- b. One turbine extension and retraction hydraulic actuator with integral mechanical lock.
- c. One turbine door control valve, electrically operated.
- d. One turbine door operated shutoff valve.

The ram-air turbine and associated components are located in an unpressurized portion of the airplane fuselage.

6.4.3.2 The 2707 (P&W). The airplane utilizes engine-windmilling power in conjunction with pressure shutoff valves in the flight controls system for redistribution of the hydraulic load to make optimum use of available hydraulic power.

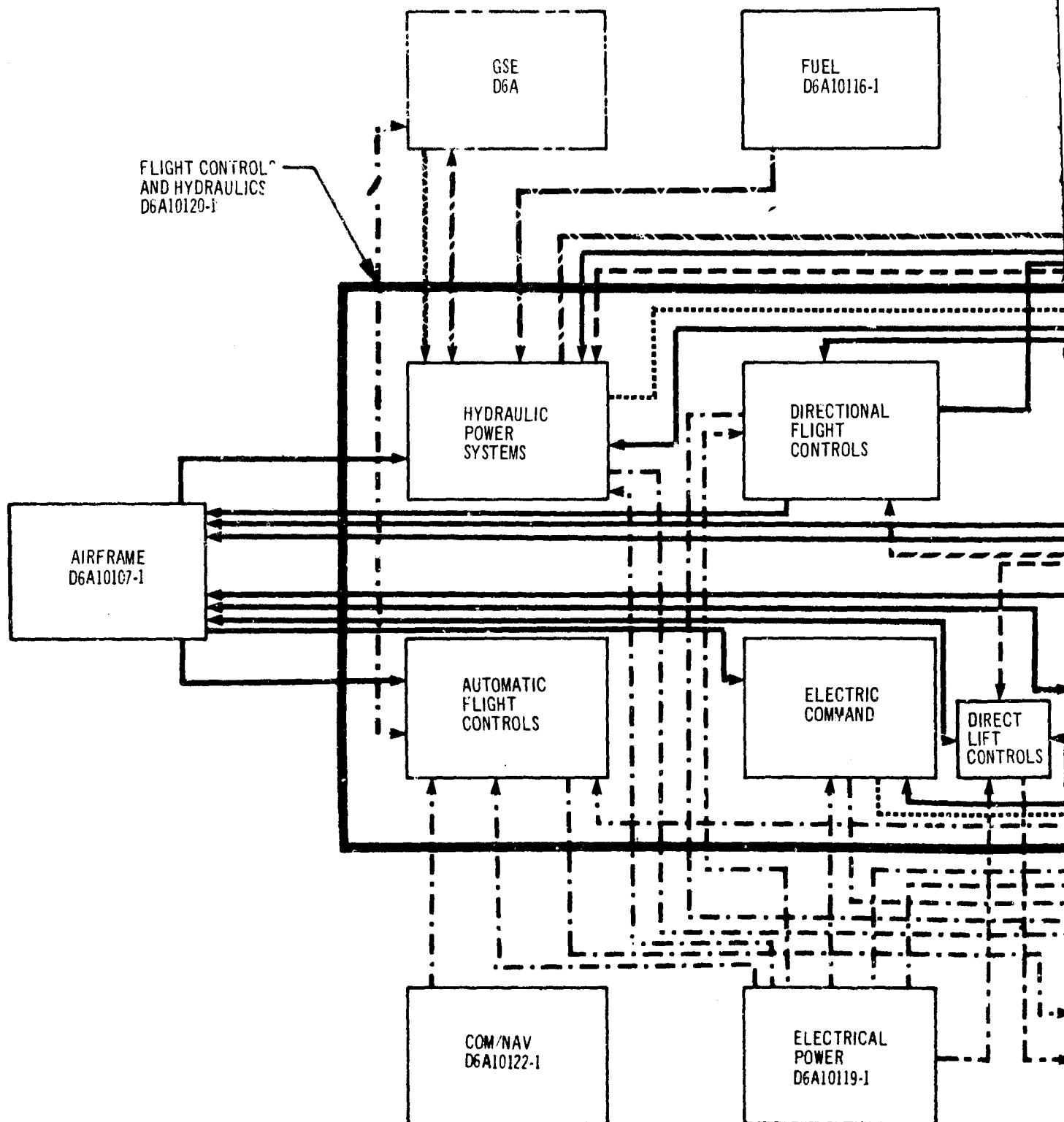
6.5 Subsystem Definition.

6.5.1 Interface Requirements. The flight controls and hydraulic subsystem shall interface with, and be compatible with, the following subsystems:

Airframe	D6A10107-1
Landing gear	D6A10108-1
Flight deck	D6A10109-1
Fuel	D6A10116-1
Accessory drive	D6A10089-1
Electrical power	D6A10119-1
AIDS	D6A10090-1
Communication/navigation	D6A10122-1
Ground support equipment	D6A10180-1

6.5.1.1 Schematic Arrangement. The flight controls and hydraulics subsystem shall interface with the subsystems listed in Par. 6.5.1 in the manner shown in Fig. 30.

6.5.1.2 Detailed Interface Definition. Detailed interface requirements are shown in Table VII.



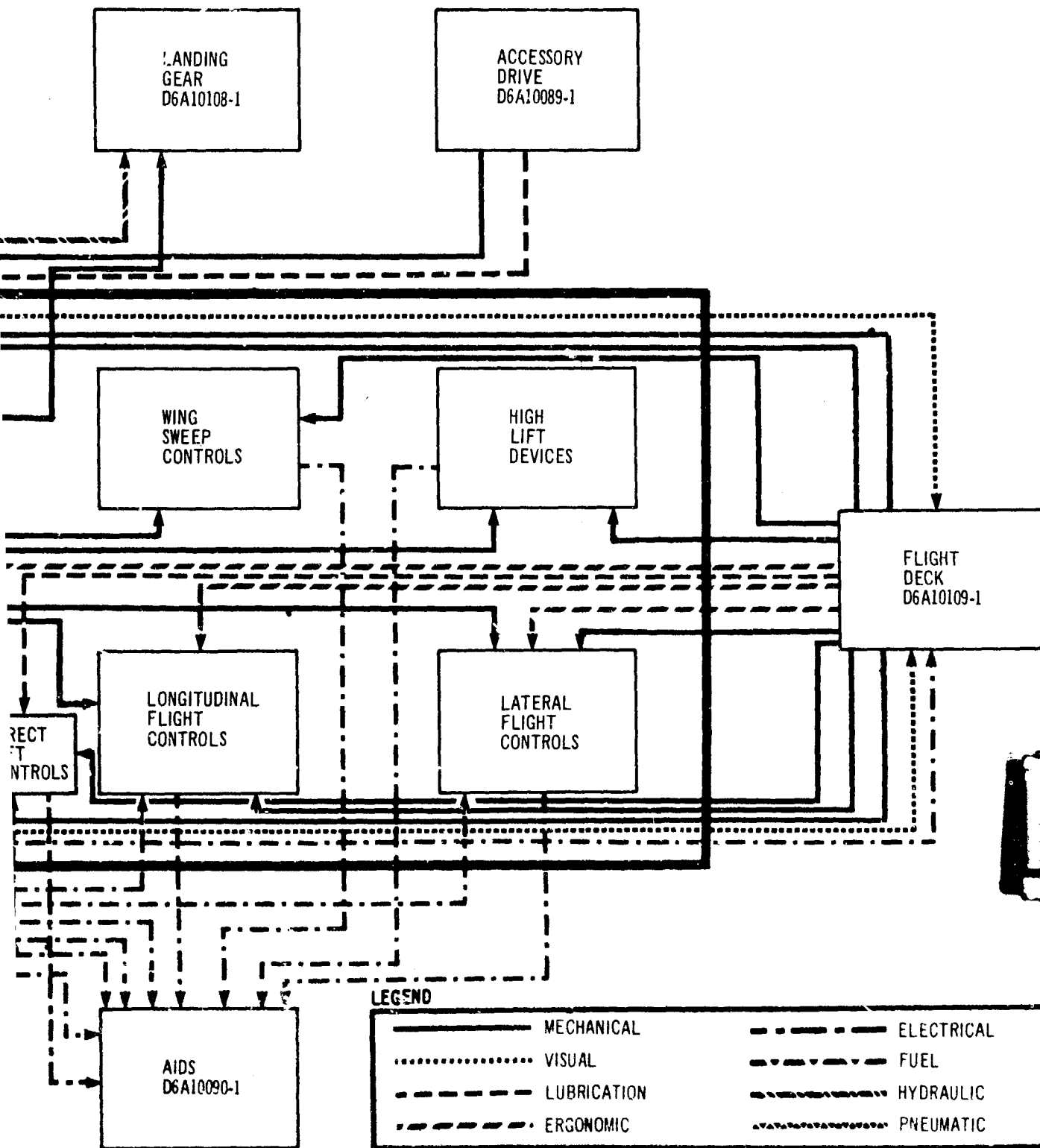


Figure 30. Schematic Interface Diagram

Table VII Detailed Interface Requirements

Subsystem	Subsystem Interface		Relationship		
	Type of interface	Mechanical	Functional	Quantitative	Tolerances
Airframe (D6A10107-1)	Input Mechanical	Brackets, mounting pads and fasteners	Mounting provisions for installation of all subsystem components		
	Output Mechanical	Hydraulic actuators	Provide means to operate: Rudder Spoilers Elevons Ailerons Elevators		
		Ball screws	Provide means to operate: LE slats TE flaps Wings		
Landing Gear (D6A10108-1)	Output Hydraulic	Tubing runs	Provide hydraulic power to: Retract and extend landing gears and doors Operate wheel brakes Steer nose and aft main gear	0 to 3,025 psi 0 to 130 gpm	Not applicable
	Mechanical	Control rod	Provide rudder pedal steering control	See D6A10108-1	Not applicable

Table VII (Continued)

Subsystem	Subsystem Interface		Mechanical	Relationship		Tolerances
	Type of interface			Functional	Quantitative	
Flight deck (D6A10109-1)	Input Mechanical		Switches	Hydraulic pump depressurization	8 on-off switches See D6A10109-1	Not applicable
				Supply shutoff valve	6 on-off switches See D6A10109-1	
				Electrical standby hydraulic pump	1 on-off switch See D6A10109-1	
				Hydraulic warmup package and heat exchanger bypass control	Three 3-position switches See D6A10109-1	
				Longitudinal trim	See D6A10109-1	
				Lateral trim	See D6A10109-1	
				Directional trim	See D6A10109-1	
				Wing sweep/flaps	See D6A10109-1	
				A/P mode selection	See D6A10109-1	
				Steering and attitude commands	See D6A10109-1	
				Channel and axis selection for A/P, A/T, SAS, E/CS and DLC	See D6A10109-1	

Table VII (Continued)

Subsystem Interface		Relationship		
Subsystem	Type of interface	Mechanical	Functional	Quantitative
Flight deck (Continued) (D6A10109-1)	Output (Continued) Visual (Continued)	Warning lights	Hydraulic pressure	9 lights
			Fluid overheat	8 lights
			E/CS, AFCS, and direct lift control system status	See D6A10109-1
	Electrical (Flight director display)		Speed command flag	28 vdc (on-off type)
			Attitude command flag	28 vdc (on-off type)
			Roll attitude command	0 $\pm$ 5000 mv
			Pitch attitude command	0 $\pm$ 100 mv
			Speed command	0 $\pm$ 2.0 v
Fuel (D6A10116-1)	Input Fuel	Tubing connections to heat exchangers	Provide fuel for cooling hydraulic fluid	6 heat exchangers 0 to 60,000 lb/hr each 0 to 120 psi
Accessory Drive (D6A10089-1)	Input Mechanical	Rotary shafts	Drive hydraulic pumps	8 shafts 0 to 4000 rpm 0 to 250 hp each

Table VII (Continued)

Subsystem Interface		Relationship		
Subsystem	Type of interface	Mechanical	Functional	Quantitative
Flight deck (Continued) (D6A10109-1)	Input (Continued) Electrical engineer's panel		Gross weight	
		Control column	Pilot input to longitudinal control and direct lift control	See D6A10109-1
		Control wheel	Pilot input to lateral control	See D6A10109-1
		Rudder pedals	Pilot input to directional control	See D6A10109-1
	Output Visual	Gages	Hydraulic system pressure	3 gages 0-4000 psi
			Hydraulic fluid temperature	4 gages -50°F to 500°F
			Reservoir quantity	2 gages 0 - 15 gal. 1 gage 0 - 35 gal. 1 gage 0 - 5 gal.
				±50 psi
				±5°F
				±1 gal.
				±1.5 gal.
				±.5 gal.

Table VII (Continued)

Subsystem Interface		Relationship		
Subsystem	Type of interface	Mechanical	Functional	Quantitative
Accessory drive (Continued) (D6A10089-1)	Input (Continued) Mechanical (Continued) Lubrication	Mounting provisions Integral	Mount hydraulic pumps Lubricate splines	8 flanges per D6A10089-1 8 splines per D6A10089-1
Electrical power (D6A10119-1)	Input Electrical	Wiring and circuit breakers	Provide power for: hydraulic instrumentation	28 vdc
			Hydraulic pump depressurization control	28 vdc 13 to 32 watts
			Hydraulic system warmup control	28 vdc
			Hydraulic system supply S.O. valve	28 vdc
			Standby hydraulic pump E/CS, AFCS, and direct lift control system Longitudinal trim actuator Lateral trim actuator	115 vac, 400 cps 3 phase, 12 kva
				-6, +1.5 vdc ±5 vdc ±5 vdc ±5 vdc System standard

Table VII (Continued)

Subsystem	Interface		Relationship		Tolerances
	Type of interface	Mechanical	Functional	Quantitative	
Electrical power (Continued) (D6A10119-1)	Input (Continued) Electrical (Continued)		Directional trim actuator Automatic flight controls		
AIDS (D6A10090-1)	Output Electrical	Wiring	Provide data on condition of subsystem components		
Communications/Navigation (D6A10122-1)	Input Electrical (air data)		Mach no.	0.2-3.0 Mach	±.005 Mach @ s.l. ±.01 Mach @ 60,000 ft ±.02 Mach @ 80,000 ft
			Mach error Calibrated airspeed	60-800 kn	±5 kn @ 50 kn ±2 kn @ 100 kn ±2 kn @ 200 kn ±5 kn @ 800 kn
			Altitude	-100 to 80,000 ft	+20 ft @ -1,000 to 1,000 ft +25 ft @ 6,000 ft +25 ft @ 12,500 ft +140 ft @ 70,000 ft +300 ft @ 80,000 ft

Table VII (Continued).

Subsystem Interface		Relationship		
Subsystem	Type of interface	Mechanical	Functional	Quantitative
Communications/Navigation (Continued) (D6A10122-1)	Input (Continued) Electrical (air data) (Continued)		Altitude	-100 to 80,000 ft
			Altitude rate	0-20,000 fpm
	Electrical (angle of attack sensor)		Angle of attack	± 25 deg
	Electrical (VOR/ LOC rcvr.)	See ARINC Characteristic #547		
	Electrical (glide slope rcvr.)	See ARINC Characteristic #551		
				Electrical correspondence to vane position ± 0.075 deg

Table VII (Continued)

Subsystem Interface		Relationship			
Subsystem	Type of interface	Mechanical	Functional	Quantitative	Tolerances
Communications/Navigation (Continued) (D6A10122-1)	Input (Continued) Electrical (inertial navigation)		Pitch attitude	0 to ± 10 deg $+10$ to ± 30 deg $- \pm 30$ to 180 deg	± 0.20 deg ± 0.50 deg ± 1.0 deg
			Roll attitude	0 to ± 10 deg $+10$ to ± 30 deg $- \pm 30$ to 180 deg	± 0.20 deg ± 0.50 deg ± 1.0 deg
			True heading	0 to 360 deg	± 0.40 deg
			Magnetic heading	0 to 360 deg	± 0.50 deg
			Vertical acc.	0 to ± 1.5 g	± 0.02 g ± 0.01
			Longitudinal acc.	0 to ± 0.5 g	± 0.01 g ± 0.01
			Cross-track	0 ± 50 nmi	$\pm (0.1$ nmi $\pm 5\%$ of deviation)
			Track-angle error	0 ± 180 deg	± 0.5 deg
			Destination warning	On-off type signal	Not applicable
			Failure warning	On-off type signal	Not applicable
Ground Operations (GSE) (D6A10180-1)	Input Hydraulic	Special coupling for servicing	Provide fluid filling and draining	See D6A10180-1	Not applicable

Table VII (Concluded)

Subsystem Interface		Relationship		
Subsystem	Type of interface	Mechanical	Functional	Quantitative
Ground Operations (GSE) (Continued) (D6A10180-1)	Input (Continued) Pneumatic	MS 28889 air valve	Charge accumulators and/or reservoirs	0 to 2,200 psi air or nitrogen
	Output Hydraulic	Sampling valves	Provide means to sample hydraulic fluid for contamination and degradation	Not applicable

7. SUBSYSTEM PERFORMANCE (Deleted)

DBA10120-

8. SUBSYSTEM OPERATION

8.1 Primary Flight Control System

8.1.1 Longitudinal Control. The pilots, through the control columns, directly control the pitch master servo. The normal mode of control utilizes the electric command system (E/CS). The dual mechanical cables from each control column provide an alternate mode by which the pilots control the master servo and have force authority over the normal (E/CS) mode. The master servo, through dual mechanical cables, controls the elevator and elevon hydraulic servoactuators and back-drives the control columns.

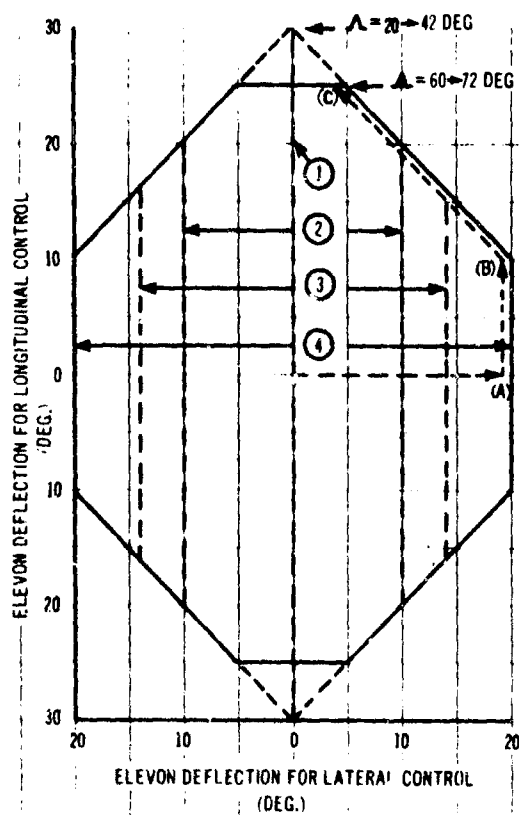
The primary elevators are controlled by the summed output of the master servo and the pitch SAS servo. The auxiliary elevators are controlled through the mechanical programmer which combines inputs from the master servo, wing flap position, and thrust reverser position. The elevons are controlled by the elevon programmer which combines inputs from the pitch master servo, pitch SAS servo, lateral master servo, and lateral SAS servo, and schedules their use for longitudinal and lateral control as a function of wing flap position and wing sweep position per the schedule shown in Fig. 31.

8.1.2 Lateral Control. The pilots, through the control wheels, directly control the lateral control master servo. The primary mode of control utilizes the E/CS. The dual mechanical cables from each control wheel provide an alternate mode by which the pilots control the master servo and have force authority over the normal E/CS mode. The master servo, through dual mechanical cables, controls the spoiler, aileron, elevon, and rudder servoactuators and back-drives the control wheels. The scheduling of these surfaces for lateral control is shown in Figs. 32 and 33.

The aileron control mechanical signals are transmitted from the lateral master servo through the aileron lockout mechanism to the aileron servoactuators. The lateral master servo controls the elevons through the elevon programmer as described in Par. 8.1.1. The spoilers are controlled through the spoiler programmer which receives inputs from the master servo and sequences spoiler motion with aileron or elevon motion as shown in Fig. 34. The programmer also combines speed brake signals with spoiler signals. The spoiler lockout mechanism cancels spoiler control to the two outboard spoiler panels on each wing when the wings are swept aft.

When the spoilers are retracted, they are faired with the upper wing trailing edge panels. As these panels move down, during wing flap extension, control linkages connecting the panels to the spoiler actuators reposition the spoiler retracted position downward to remain faired.

The rudder is controlled through the elevon programmer when it is used for turn coordination and lateral control.



COND.	WING SWEEP (DEG)	FLAP POSN	MAX. LATERAL CONTROL				MAX LONG CONTROL (DEG)
			AIL (DEG)	ELEVON (DEG)	SPOILERS (DEG)		
					INBD	OUTBD	
①	20	LAND	± 25	0	45	45	± 30
	30	T.O.	± 25	0	45	45	± 30
②	30	FAIRED	0	±10	45	45	±30
③	42	FAIRED	0	±14	45	45	±30
④	60	FAIRED	0	±20	45	0	±25

LONG. COMMANDS HAVE PRIORITY

EXAMPLE:

- 1) ASSUME CONDITION ① EXISTS
- 2) COMMAND MAXIMUM LAT. CONTROL 20 DEG. AT (A)
- 3) ADD 10 DEG. LONG. CONTROL (B)
- 4) COMMAND MAX. LONG. CONTROL = 25 DEG. AT (C) MAX. LATERAL ELEVON NOW = 5 DEG.

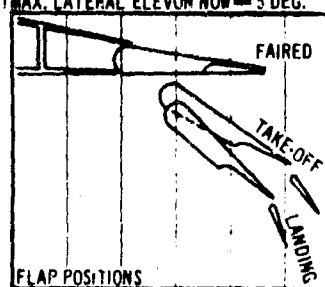


Figure 31. Elevon Mix, Longitudinal and Lateral Control

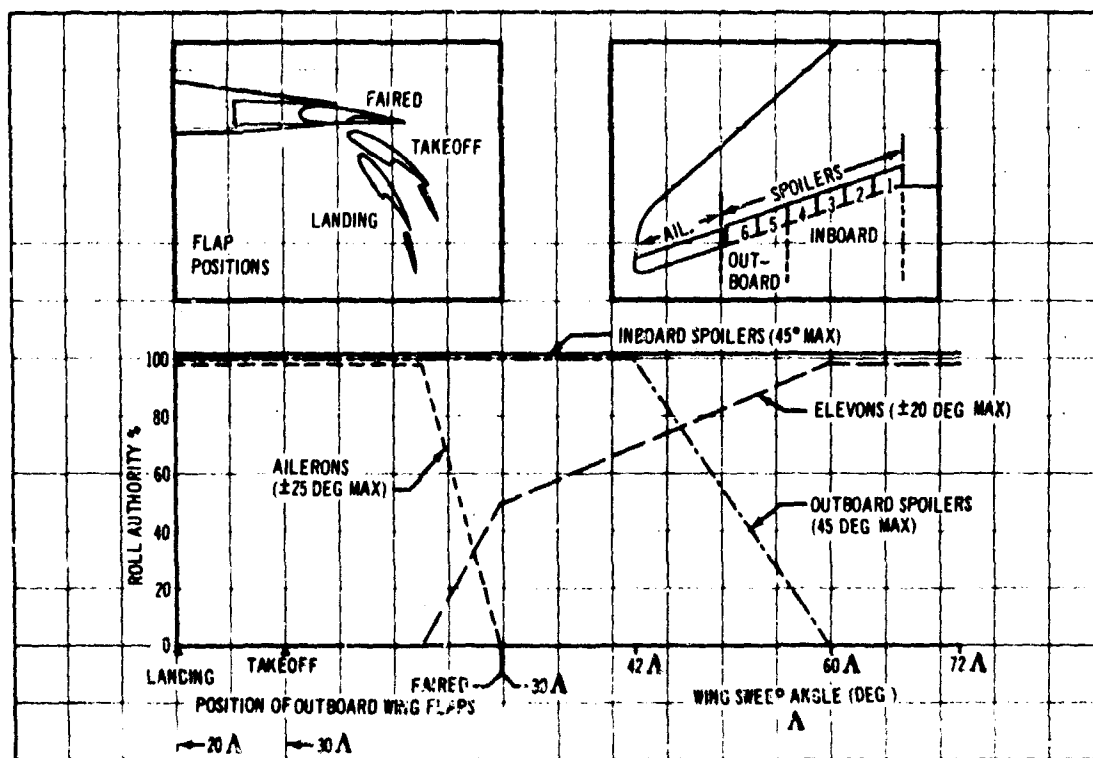


Figure 32. Lateral Control Programming

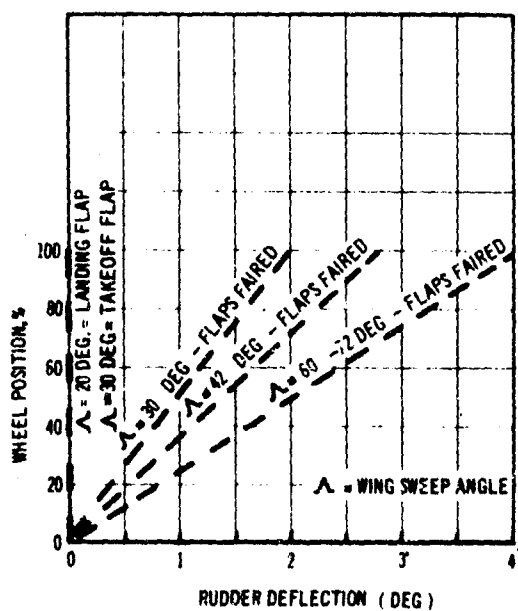


Figure 33. Rudder Versus Wheel Position

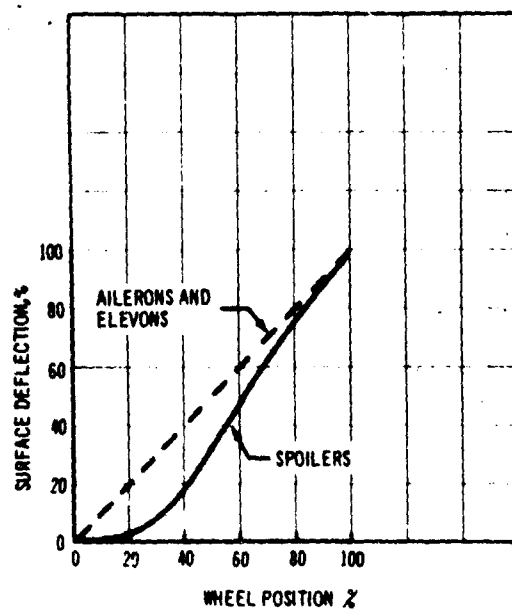


Figure 34. Spoiler Program for Lateral Control

8.1.3 Directional Control. The pilots control the rudder through a dual mechanical cable system operated by the rudder pedals. The pilot's input is summed with inputs from the yaw SAS servo and from the elevon programmer as discussed in Par. 8.1.2, when the elevons are used for lateral control. The summed input passes through the rudder travel limiter which limits the control input to the rudder servo actuator control valves as a function of the flap position and wing position as shown in Fig. 35.

8.1.4 Wing Sweep The pilot and copilot control wing sweep position through the wing sweep control lever which is connected through a dual cable system to the flap and wing sweep programmer. The programmer controls the wing sweep power drive unit and schedules fore and aft motion of the outboard wing trailing edge flaps and partial extension of the leading edge slats on the inboard and outboard wings. The programmer is so arranged that, if any flap or slat system fails to operate, the wings can be operated toward the forward direction but are inoperative toward the aft direction. The wing sweep asymmetry control system automatically shuts off hydraulic pressure to the power drive unit hydraulic motors in the event the left and right wings become asymmetric. It also includes an override feature which allows the pilots to sweep the wings if a shutdown should occur from a failure within the asymmetry control system itself.

8.1.5 Direct Lift Control. Operation of direct lift control is accomplished automatically in conjunction with operation of longitudinal control during landing. Signals from wing-sweep position sensors (Par. 6.1.4 h) activate the direct lift control at the 20-deg sweep position, causing the inboard spoilers to raise symmetrically 3 deg above fair. Pitch signals generated by pilot column forces, or auto-pilot commands and stability augmentation commands, are shaped and amplified by servo amplifiers and transmitted to the direct lift servos. (Par. 6.1.5 and Figs. 8, 14, and 20). Mechanical output of the direct lift servos control the hydraulic valves of the inboard spoiler actuators (Par. 5.1 and Fig. 7) moving the spoilers symmetrically, down to increase lift, or up to decrease lift, in parallel with movement of the elevator surfaces. With a steady state pitch command, the spoilers return to the 3-deg position with a programmed washout.

8.2 Secondary Flight Control System

8.2.1 Longitudinal Trim. The longitudinal trim actuator is controlled by an electric motor and a cable system. The pilots operate the motor by a trim button mounted on each pilot's control wheel. The cable system is operated by trim control wheels, one mounted on each side of the aisle stand.

8.2.2 Lateral Trim. The lateral trim actuator is controlled by an electric motor. The pilots operate the motor by the roll trim control knob mounted on the aisle stand.

8.2.3 Directional Trim. The directional trim actuator is controlled by an electric motor. The pilots operate the motor by the yaw trim knob mounted on the aisle stand.

8.2.4 Trailing Edge Flaps. Each flap drive system is controlled through the flap and wing sweep programmer. The fore and aft motion of the outboard wing flaps occurs with wing sweep, through pilot operation of the wing sweep lever, as described in Par. 8.1.4. The rotation of the outboard wing flaps, the operation of the inboard wing flaps and the last portion of the slat extension occurs when the pilot moves the flap lever. The flap lever is connected through a cable system to the programmer.

The programmer combines the inputs from the wing sweep lever and the flap lever and schedules the operation of the flap power drive units as shown in Fig. 36. Inputs from each flap drive system are combined within the programmer to prevent operation of wings, flaps and slats to incompatible positions.

8.2.5 Leading Edge Slats. Each slat drive system is controlled through the flap and wing sweep programmer. The first portion of extension of inboard and outboard wing slats occurs with wing sweep, through pilot operation of the wing sweep lever, as shown in Fig. 36. The last portion of slat extension occurs with flap extension, through pilot operation of the flap lever, as shown in Fig. 36. An alternate slat lever, which is connected to the programmer through a cable system, controls the extension of the inboard wing leading edge slats in the event of a wings-aft landing. Inputs from each slat drive system are combined within the programmer to prevent operation of wings, flaps and slats to incompatible positions.

8.2.6 Speed Brakes. The pilot and copilot control the spoilers in unison as speed brakes through a lever connected through a cable system to the spoiler programmer. The programmer allows differential motion of the spoilers for lateral control to be superimposed when they are raised as speed brakes.

8.3 Automatic Flight Controls.

8.3.1 Autopilot. An integrated autopilot flight director (A/P-F/D) system provides an operational capability permitting manual (OFF), manual - automatic (CWS/MAN) and automatic control (AUTO NAV). These controls are located on the lower center of the mode selector control panel, shown in Fig. 16. During manual operation the pilot controls the airplane through the control column. The manual - automatic mode allows the pilot to maneuver the airplane by control wheel steering (CWS) or with the manual controller (MAN). In the CWS portion of the manual - automatic control, commands are introduced through the control column and wheel. In the MAN portion, commands are inserted through the vertical speed wheel for the pitch axis and the turn knob or heading select ring for

the roll axis. (See Fig. 17.) In automatic control modes, the airplane is controlled to the selected flight-path reference and requires no pilot participation.

The flight director (F/D) displays are controlled by the roll and pitch selectors on the mode selector panel. These displays can be selected to show both automatic and manual commands.

8.3.1.1 Pitch Axis Manual Control Modes The following pitch axis modes are operable with the PITCH switch in the center position. The switch indicators marked CWS and MAN, located between the PITCH and ROLL switches on the mode selector control panel, must be depressed in MAN for the vertical speed mode and for the attitude hold mode, and in CWS for the control wheel steering mode.

- a. The vertical speed (V/S) is the basic pitch axis control mode and is synchronized whenever the mode is not in use. The vertical speed reference is adjusted by varying the vertical speed command wheel located on the manual control panel. This also moves the V/S command bug on the V/S indicator. Damping is provided by a vertical acceleration signal. Displacement plus integral control of vertical speed error is provided to prevent stand-off.
- b. Altitude hold is selected by rotating the vertical speed wheel to the zero rate-of-climb detent position (ALT HOLD). This feature allows gradual reduction of altitude change rate to permit capture of a desired altitude. Damping is provided by barometric altitude change rate and vertical acceleration signals. Displacement plus integral control is provided to prevent altitude stand-off.
- c. In the control wheel steering mode, the airplane is stabilized to the pitch attitude existing at the time of engagement. When a pilot pitch force command is introduced at the control column, a pitch rate command signal is generated which causes the attitude to change accordingly until the column force is returned to zero. When the column is returned to detent, the new commanded attitude is then held. This mode utilizes the E/CS pitch axis and improves upon it in that the steady-state attitude is held constant in the presence of external disturbances.

8.3.1.2 Pitch Axis Automatic Navigation Control Modes. The following pitch axis modes are operable with the PITCH switch in the AUTO NAV position:

- a. Vertical speed mode. See Par. 8.3.1.1.a.
- b. Altitude hold mode. See Par. 8.3.1.1.b.

- c. Airspeed hold mode. This mode is selected by placing the PITCH SEL switch in the AIRSPEED HOLD position.

Damping is provided by a horizontal acceleration signal. Displacement plus integral control is provided to prevent stand-off. When this mode is engaged, an interlock prevents engagement of the auto throttle.

- d. Mach altitude mode. Mach altitude control mode is selected by placing the PITCH SEL switch in the MACH ALT position.

During ascent, the mach altitude control mode commands the airplane from a point following takeoff to a preselected altitude and, during climb, maintains a desired sonic-boom overpressure as selected on the mode selector control panel.

The mach altitude flight consists of three segments; airspeed control (I), sonic boom pressure control (ΔP), and airspeed control (II).

The pilot selects a terminal altitude and a sonic boom pressure on the mode selector control panel before selecting the mach altitude mode.

After takeoff, as the airplane passes through the 380 kn reference point, the airspeed control I segment is engaged automatically. The airplane continues on airspeed control I until it reaches 0.9 Mach, at which point the sonic boom overpressure computer takes command. The longitudinal axis is controlled by the difference between the airplane altitude and the desired altitude, which is obtained by solving the sonic boom overpressure equation.

When the second segment airspeed is reached, switching occurs and control is now under the command of airspeed control II segment until the terminal altitude is reached.

Descent is accomplished by controlling a Mach altitude profile in a manner similar to that used for ascent. The pilot selects the sonic boom overpressure, arms the Mach altitude mode, and sets the engines to reduced power. The airspeed existing at the time the mode is selected is held until the sonic boom overpressure selected is approached and the ΔP control again takes command. Below Mach 0.9, the airspeed control I takes control and maintains a constant airspeed until the pilot selects another mode or disengages and manually controls the airplane.

- e. Altitude capture. This mode is armed by moving the altitude capture switch on the mode selector control panel to ALT CAP position. The desired altitude is set on the mode selector

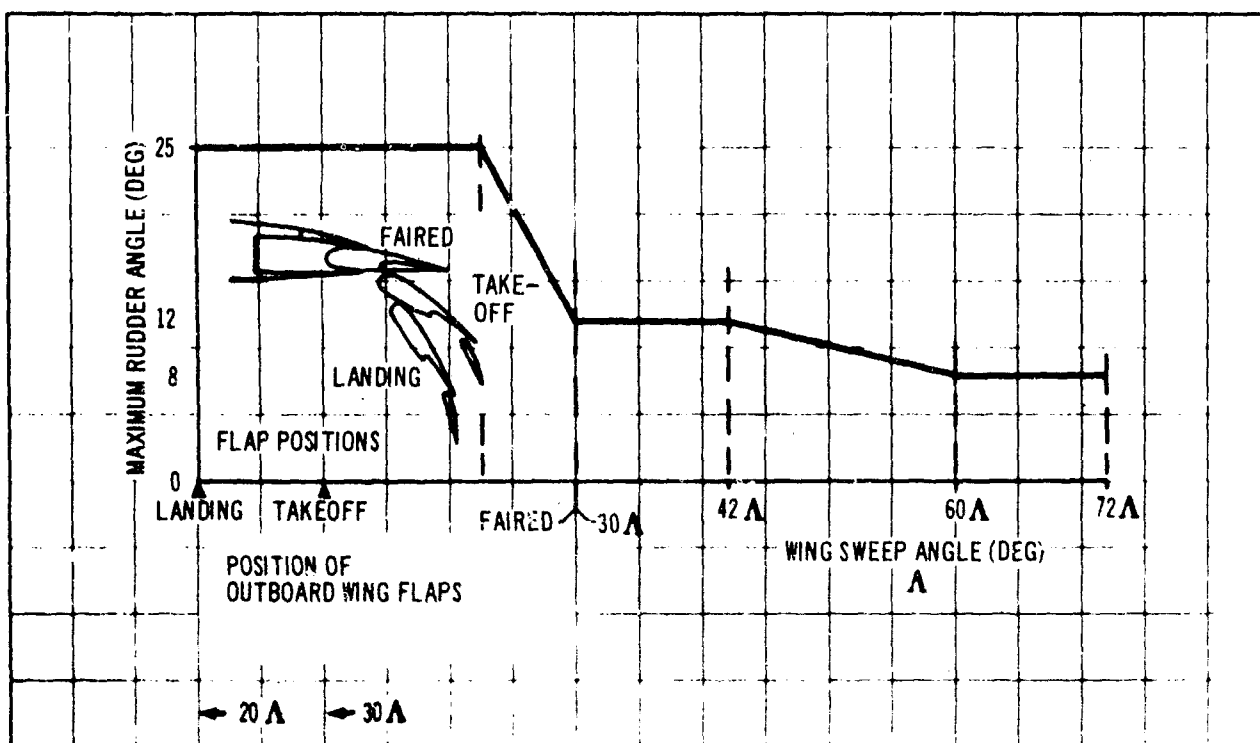


Figure 35. Rudder Travel Limiting

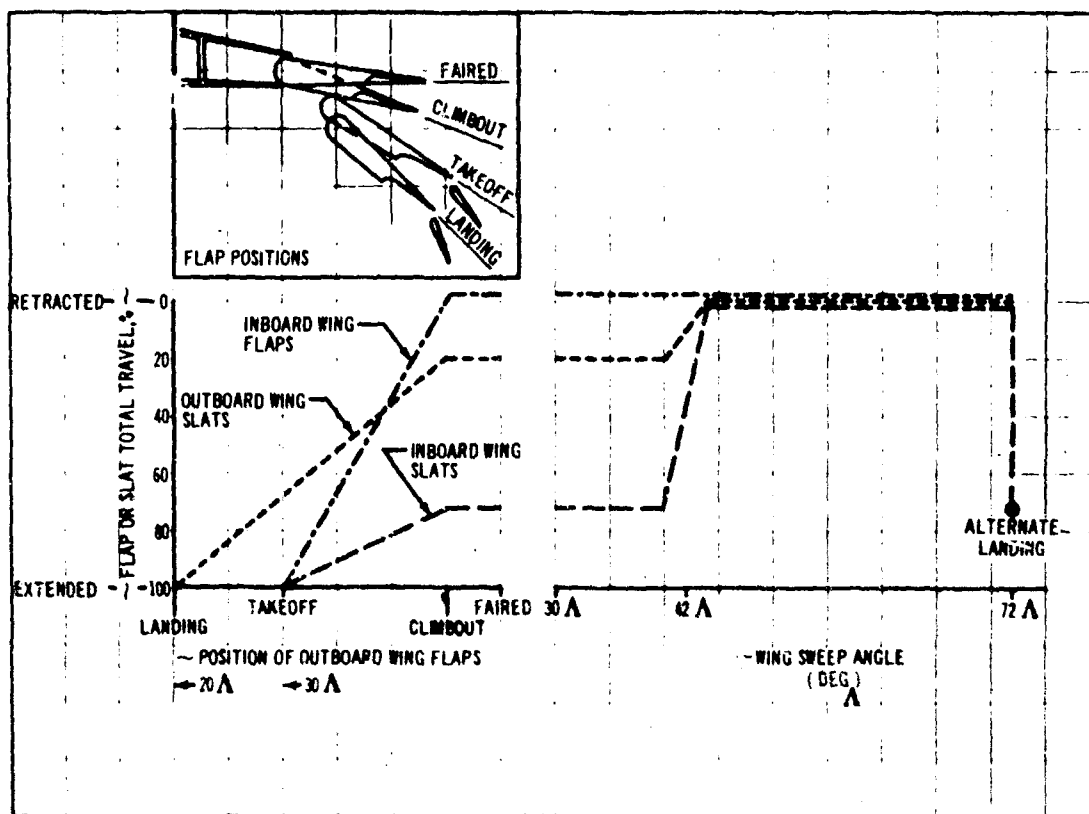


Figure 36. Slat and Flap Programming With Wing-Sweep Position

control panel and provides a reference signal for automatic capture of the selected altitude. Maneuvering of the airplane to the desired altitude is accomplished in any one of three pitch axis modes, vertical speed (V/S), airspeed hold or Mach altitude (MACH ALT). At the selected altitude, logic circuits switch out the existing pitch axis mode and switch in the altitude capture mode. The PITCH SEL switch is automatically positioned at V/S and the ALT CAP light indicates that the altitude capture mode is engaged.

The altitude capture error signal commands an airplane vertical speed to enable a smooth capture of the selected altitude. When altitude capture error is less than a predetermined value, altitude capture control is terminated and altitude hold is engaged. This is indicated by the altitude capture switch automatically moving to OFF. The altitude capture mode indicator light is extinguished and the vertical speed command wheel automatically moves to the ALT HOLD position. A vertical acceleration signal from the INS is provided for damping.

- f. Glide slope control. The glide slope control mode is selected by moving the PITCH SEL switch to the ILS position. This mode provides for automatic capture and tracking of a glide slope beam. The airplane continues to maintain either altitude hold or vertical speed control until a predetermined glide slope signal level is reached. At this point, a beam sensor is triggered to initiate glide slope capture control. During glide slope capture control, a rate of descent is commanded. This value is equivalent to the average rate of descent when the aircraft is tracking the glide slope. Glide slope track initiation is sequenced at a specific time interval after glide slope capture control has been initiated. Beam displacement plus integral control is used until flare initiation in land control mode. Radio altimeter controlled gain scheduling is provided to achieve desirable glide slope tracking. Integrated vertical acceleration is used for damping while glide slope control is in effect. This glide slope control technique is presently used on 707, 727, and 737 type airplanes.
- g. Land control. Land control is armed when the PITCH SEL switch is rotated to the LAND position. This mode provides for glide slope and automatic flare. This mode may be armed prior to glide path capture, permitting the selection of complete automatic control to touchdown prior to glide slope capture. Flare is triggered by a radio altimeter signal. The airplane vertical velocity is controlled as a function of radio altimeter altitude. A vertical acceleration signal provides damping. An asymmetrical limit blocks any nose-down command which could command an airplane attitude less than the attitude of the aircraft at initiation of land control.

- h. Go-around and take-off control mode. This mode is engaged by a manually operated switch on the aisle stand. This mode controls the airplane angle of attack. The angle of attack reference is modified by computed flight path angle to prevent excessive pitch attitude and by flap position to obtain maximum lift. When the go-around and take-off mode is selected, the PITCH SEL switch reverts to the V/S position and the ROLL SEL switch reverts to the GA/TO position. For damping, a composite forward acceleration signal is derived from a combination of horizontal acceleration and airspeed signals. During take-offs, the go-around and take-off control mode is designed for F/D commands only. It is not considered a usable A/P control mode during take-off.

8.3.1.3 Roll Axis Manual Control Modes. The following roll axis modes are operable with the ROLL switch in the center position. The switch indicators marked CWS and MAN, located between the PITCH and ROLL switches on the mode selector control panel, must be depressed in MAN for the heading select, heading hold, and roll attitude hold modes, and in CWS for the control wheel steering mode.

- a. Heading select control mode. This mode is engaged by placing the roll mode selector switch on the manual control panel in the HDG SEL position. The selected airplane heading is displayed on the horizontal situation indicator (HSI) when this mode is engaged. The selected heading is adjusted by the heading select ring on the manual control panel. This adjustment is indicated by corresponding movement of the bug on the HSI. Bank angle limits are scheduled as a function of Mach number for optimum performance throughout the flight envelope.
- b. Heading hold control mode. This mode is engaged when the turn knob is in the centered detent position, and the mode selector switch on the manual control panel is in the TURN KNOB position. A heading or signal is supplied by the inertial navigation system when this mode is in effect.
- c. Roll attitude control mode. This mode is engaged when the turn knob is not centered. The mode selector switch on the manual control panel reverts to the TURN KNOB position whenever the turn knob supplies a signal for roll attitude command. Bank angle commands of a predetermined level are permitted in this control mode.
- d. Control wheel steering mode. In this mode, the airplane is stabilized to the roll attitude existing at the time of engagement. If this attitude is less than two deg, aircraft heading is maintained. When a pilot roll force command is introduced at the control wheel, a roll rate command signal is generated which causes the attitude to change until the wheel force is returned to zero. When the wheel is returned to detent, the

new commanded attitude is then held. This mode utilizes the E/CS roll axis and improves upon it in that the steady-state attitude or heading is held constant in the presence of external disturbances.

8.3.1.4 Roll Axis Automatic Navigation Control Modes. The following roll axis control modes are operable with the ROLL switch in the AUTO-NAV position:

- a. Go-around and takeoff control mode. This mode is engaged remotely by a switch located on the aisle stand or directly by placing the ROLL SEL switch in the GA/TO position. The switch, located on the aisle stand, also places the pitch axis in go-around and takeoff control mode. This basic roll axis control mode maintains a wings-level attitude.
- b. Heading select control mode. When the ROLL SEL switch shown in Fig. 16 is in the HDG position, the heading select control mode is engaged. This mode operates identical to the heading select control mode discussed in Par. 8.3.1.3.a.
- c. Localizer control mode. The localizer control mode is selected by placing the ROLL SEL switch in the I-NAV, VOR, LOC position. The ROLL SEL switch is also placed in the I-NAV, VOR, LOC position remotely by placing the PITCH SEL switch in the ILS or LAND position. This feature permits arming or engaging an automatic approach with one selector. In order for the localizer control mode to operate properly, setting of the NAV SEL switch and the A/P - F/D CMD switch is required. One NAV SEL switch must be set to No. 1 and the other to No. 2. Both digital course indicators located on the upper portion of the mode selector panel must be set to the runway heading. The A/P - F/D CMD switch must be set to the ALL position.

Prior to capture of the localizer beam, manual commands from the manual control panel are in effect. This permits capture of the localizer beam to be accomplished in the heading hold or heading select mode. Interlocks prevent localizer capture from occurring until the turn knob is centered in the HDG HOLD position. When a predetermined beam error signal is reached, a beam sensor initiates localizer capture control.

During localizer capture control, runway course and beam error are summed to achieve a smooth capture of the localizer beam. Beam tracking control is initiated when roll attitude, beam rate, and beam error signals are less than a predetermined value.

During localizer track control, beam displacement, beam rate, beam integral, and lagged roll signals are summed to provide optimum beam tracking. Gain scheduling is controlled by the

radio altimeter with the net effective gain increasing approximately 30 percent at an altitude of 60 ft. This increase in gain ensures tight beam tracking at the runway threshold. This localizer control technique is presently incorporated in 707 and 727 automatic landing systems.

- d. VOR control mode. VOR control is selected by placing the ROLL selector in the I-NAV, VOR, LOC position. The NAV SEL switches on the mode selector control panel must indicate No. 1 or No. 2. The desired VOR radial is set on the digital course indicators on the mode selector control panel. The A/P - F/D CMD switch is set to the left or right position depending on the radio receiver that is selected. This control mode consists of automatic capture and tracking of a VOR radial. Prior to capture of a VOR radial, manual commands from the manual control panel are in effect. A beam sensor initiates VOR capture control. During VOR capture control, VOR course and beam error are summed to provide a smooth capture of a VOR radial. Beam track control is initiated when beam error and roll attitude signals are less than a predetermined value. Beam error and beam integral signals are used for beam track control. An over-the-station sensor decouples the A/P from the VOR radial when over the station. Course error signal is in effect while over the station. Selection of a new VOR radial may be made while over the VOR station. When the airplane leaves the zone of confusion, the new VOR radial is tracked.
- e. Inertial navigation control mode. I-NAV mode is selected by placing the ROLL SEL switch in the I-NAV, LOC position.

The NAV SEL switches are set to the INERTIAL SEL position. The inertial NAV system is selected on the INERTIAL SEL switch also located on the mode selector control panel. The inertial navigation control mode consists of capture and tracking of an inertial track. Prior to capture of an inertial track, manual commands from the manual control panel are in effect. When a predetermined cross-track deviation and cross-track angle error signal is reached, cross-track capture is initiated. Cross-track deviation and track angle error signals are summed to provide a smooth capture and continuous track of the selected inertial cross-track.

8.3.2 Auto Throttle. The auto throttle system (A/T) provides an operational capability permitting airspeed select, capture and hold and Mach hold. The controls are located on the mode selector control panel. When not engaged in either mode, the pilot controls the airplane through the throttle controls. The pilot can disengage auto throttle by the disconnect switches on the throttles.

The airspeed mode has a capture and hold capability. The airspeed command is selected by positioning the CAS SEL knob on the auto throttle control panel. To change airspeed the pilot dials in the desired air-

speed on the auto throttle control panel. the auto throttle automatically reduces or increases thrust, as required. The design includes proportional plus integral control to obtain the desired steady-state accuracy. A longitudinal acceleration signal is used to provide damping. Filtering minimizes the effects of wind gusts. The integrator is disabled during airspeed capture to prevent integrator runaway due to a large proportional signal input.

The airspeed reference is reduced as a function of altitude during flare to reduce the forward velocity at touchdown. This type of throttle control during flare maintains thrust control to compensate for shear winds.

Mach hold is provided through the throttle system. Short term stability is provided by airspeed with the long term reference being Mach number. This technique reduces throttle activity caused by temperature gradients. A Mach adjustment is provided to allow the pilot to make small changes in Mach as required to maintain the airplane and engines within their course operating limits. The A/T system drives the throttle through no-back clutches. These clutches allow the pilot to trim individual throttles without back-driving the A/T system.

A dual auto throttle system is used to provide fail-passive operation. The most critical failure of a thrust control system is the passive undetected fault. If the pilot is warned, he has no trouble taking control of the throttles manually. Such warning is provided by indicators which are activated by the malfunction detection capability associated with the dual redundancy. Interlocks prevent the engagement of this mode when the airspeed hold mode of the A/P is engaged.

8.3.3 SAS. A full-time, stability augmentation system (SAS) with separate pitch, roll and yaw axes is provided.

Handling quality requirements have been met by providing augmented response characteristics which allow the pilot to have precise control over the airplane pitch and roll attitudes. Path control, i.e., control of vertical speed and heading rate, is easily accomplished using attitude control as the inner loop. Turn coordination, provided by the wheel-into-rudder interconnect, is augmented by the SAS so that only the column and wheel will be used for coordinated maneuvers.

The feedback loops required to provide the good response characteristics also provide damping and frequency augmentation of the oscillatory modes, ensuring good dynamic stability.

The rate feedback loops and the electric command augmentation in the SAS have been made independent of air data. Wide frequency band filters provide the required shaping characteristics over the range of flight conditions, and configuration dependent switching (using wing sweep and flap position) changes filter characteristics at a few set points.

The use of multiple channels of electronics and sensors and triple force-summed SAS servos provides the required fail-operational characteristics. In addition, the monitoring system eliminates tolerances between channels, thereby minimizing system nuisance disengagements due to channel mistracking. Where needed, channel equalization is provided by driving each channel towards its own voted output and the use of cross-ties between channels for cross equalization is avoided.

8.3.3.1 Pitch Axis Figure 20 is a functional block diagram of the pitch axis of the SAS which also shows the close relationship with the electric command, direct lift control, and trim systems.

The unaugmented airplane has neutral or positive static stability in pitch at any trimmed straight flight condition. Short period damping, although poor at transonic and supersonic speeds, is easily corrected with SAS. At low speed, the short period damping is good but the period is long and not far removed from the phugoid which has low or negative damping. Based upon these characteristics, it is reasoned that the pitch axis of the SAS is required to be operational after two failures. After the first failure, no change in flight plan is necessary. After a second failure, the flight can continue but with a restricted flight envelope.

The pitch axis of the SAS is essentially a pitch rate feedback system with command augmentation. The E/C and SAS work together such that, in response to an applied force, the airplane quickly establishes a steady pitch rate. The response has minimum overshoot and well-damped oscillatory characteristics allowing the pilot to have precise control of pitch attitude. When the applied force is reduced to zero, the pitch rate command is zero and the airplane remains at the new attitude. This characteristic is referred to as platform stability.

Pilot command inputs to the SAS are measured at the output position of the pitch master servo and are shaped by the prefilter to give a pitch rate command signal. Body-mounted rate gyros measure the airplane pitch rate and the pitch rate error signal is then shaped by the compensation filter to give a SAS pitch axis servo position command. This servo provides a series input which is added mechanically to the pitch master servo position. The resulting linkage movement then positions the pitch control surfaces by means of the surface actuators.

The steady pitch rate developed for a given change in pitch master servo position is fixed for all flight conditions. This simplifies the gain scheduling in the electric command system required to give constant stick force/g characteristics. Scheduling for cg shift and variation in control effectiveness is therefore not required.

Because the augmented airplane has platform stability, the pitch axis of the SAS produces a control deflection to balance moment changes due to flap deflection or wing sweep. To minimize the SAS servo authority required, automatic trimming is provided when the SAS servo deflection

exceeds a threshold value. No pitch attitude error integration is provided within the SAS and therefore a slow drift in attitude occurs if the SAS is providing a balancing moment. Automatic trimming minimizes this drift.

Manual trim commands move the feel center and the pitch master servo is positioned accordingly. This master servo position change is sensed by the SAS pitch axis command augmentation and interpreted as a pitch rate command. Whether the column is controlled by the force sensors or by the trim, the airplane responds in the same manner.

To prevent the automatic trim from being interpreted as manual trim command, an electronic counterpart is generated within the SAS computer to balance the change in the master servo position signal. The components of the automatic trim function are a threshold detector, a pulse generator to command the trim stepping motor, and an electronic model of the trim motor and master servo. The same pulses are integrated by the trim stepping motor and the pulse counting circuitry of the electronic model so that the two systems track identically. This integrator is also used for pre-engage synchronization.

The SAS pitch axis basic platform stability mode exhibits neutral speed stability, i.e., changes in speed do not cause pitch attitude changes and do not require the pilot to hold a steady force proportional to the speed change. For many modes of flight, neutral speed stability simplifies the control problem; however, for landing approach and low speed operation, it may be desirable to have positive speed stability. A separate speed stability mode is provided within the pitch SAS which may be selected for these cases. When this mode is engaged, an air-speed error signal with an airspeed rate damping signal is introduced which oppose the command augmentation signals and provide ideal speed stability characteristics.

The air data required for this mode is computed within the SAS using the same dynamic pressure signals as used for electric command gain scheduling. This part of the system will be fail-operational for the first failure, and fail-passive for the second failure. Three channels are provided for the speed stability function as determined by a failure mode and effects analysis. Load alleviation is provided by the SAS pitch axis through damping of the short period. By eliminating the large pitch rate over-shoot, which is characteristic of the unaugmented airplane response to a step command, a slow well-damped normal acceleration response is obtained at high speed. This reduces the maneuver loads induced by the pilot when he is making small path corrections.

Electronic filtering of the SAS feedback loops is relied upon to prevent excitation of the flexible body modes. In the filter design, attenuation is increased at frequencies above the rigid-body modes; this ensures adequate stability margins at the first flexible mode. An initial attempt to provide one compensation filter suitable for all flight conditions resulted in too much gain at high frequencies. With

two compensation filters, one for the wings-forward flight conditions and another for the wings aft conditions, the desired high frequency cutoff is attained.

8.3.3.2 Roll Axis Figure 21 is a functional block diagram of the roll axis of the SAS.

At supersonic speeds, the roll time constant will exceed the maximum desirable. It is estimated that the sluggish roll response of the unaugmented airplane and the requirement for relief of the roll transient due to engine inlet unstarts is severe enough to require a system which is fail-operational on the first failure.

Roll rate feedback is used to provide roll damping improvement. A compensation filter and command augmentation provide essentially invariant well-damped roll response throughout the flight regime. All roll SAS commands are routed as a series input to the elevons. No gain change is required in SAS because of the mechanical elevon programming system which uses wing sweep and flap position as the controlling function.

8.3.3.3 Yaw Axis. Figure 22 is a functional block diagram of the yaw axis of the SAS.

At supersonic and high subsonic speeds the Dutch roll damping of the unaugmented airplane is very low, and turn coordination improvement is required at all speeds. These handling quality requirements, and the need to reduce the lateral acceleration resulting from engine inlet unstart, dictate the requirement for a yaw axis SAS which is required to be fail-operational after the first and second failures. The design is aimed primarily at damping the Dutch roll and ensuring good turn coordination.

When allowances were made for the very high speed and the relatively high trimmed angles of attack, it was found that body axis roll rate is a better Dutch roll damping feedback than the conventional body axis yaw rate. A suitably filtered signal from the body-mounted roll rate gyro achieves good Dutch roll damping and also decreases the roll time constant.

A limited signal from a body-mounted lateral accelerometer provides directional stiffening and reduces the sideslip angle during maneuvers. The mechanical crossfeed from the wheel into the rudder is augmented in the SAS by a shaped electrical crossfeed. This signal, like the mechanical crossfeed, is gain scheduled by wing sweep.

The roll-rate signal is gain scheduled as a function of compressible dynamic pressure. Since three sources of dynamic pressure must provide signals for four channels of the SAS yaw axis, the three dynamic pressure signals are fed through four mid-value logic circuits before performing the gain-scheduling function. Upon a failure of any dynamic pressure source, the gain is made to revert to an intermediate value which is an acceptable compromise for all flight conditions.

8.3.4 Angle-of-Attack Warning and Control System Description. The angle of attack warning and control system warns the pilot that the airplane is approaching a condition of excessive angle of attack by shaking the pilots' control columns. Further protection is provided by automatic introduction of a pitch-down signal to the electric command system if the pilots' response to the "stick shaker" signal does not arrest the approach to the excessive angle of attack.

Single channel block diagrams of the stick shaker and automatic pitch-down portions of the system are shown in Figs. 23 and 24 respectively.

The angle-of-attack warning and control system is based on monitoring angle of attack and pitch rate, in addition to wing sweep angle and flap position. When the sum of angle of attack plus pitch rate exceeds a predetermined threshold for a particular wing sweep and flap position, the stick shakers are activated. If the sum of angle of attack plus pitch rate continues to increase, a second threshold is reached which results in a pitch-down command being provided to the E/CS pitch computers and the A/P pitch axis being disengaged (if engaged). As the sum of angle of attack and pitch rate diminishes, the pitch-down signal to the E/CS is removed and the stick shaker is deactivated. To provide positive recovery from the approach to an excessive angle of attack, the levels at which signals are removed is set lower than the levels for activation.

The angle-of-attack warning and control panel, shown in Fig. 25, is provided in the pilots' compartment. Included on the panel is a disconnect switch which makes it possible for the pilot to manually disconnect the automatic pitch-down output to the E/CS. The stick shakers can be manually deactivated by turning off the circuit-breakers to the stick shaker motors.

The stick shaker portion of the angle of attack warning and control system consists of two isolated channels. The output of one channel shakes the pilots' control column and the output of the other channel shakes the copilots' control column. A failure in one channel will not affect the operation of the second stick shaker.

The automatic pitch-down portion of the system consists of two channels which provide fail-passive operation. The two pitch-down channel outputs are mixed (Fig. 24) to provide three equal outputs to the pitch channels of the E/CS computers. A continuous monitor is provided which automatically disengages the pitch-down outputs and illuminates a warning light on the pilots' panel when a failure is detected.

The pilots' panel includes provisions to initiate self test of the system. A successful self test causes the pilot's control column to shake and illuminates a light on the pilot's panel. Failure of the system to pass self check shall be indicated by the failure monitor lights.

8.3.5 AFCS Control Panels. The AFCS control panels provide the following operational functions:

- a. Pilot and copilot capabilities to engage or disengage any axis of the E/CS and A/P.
- b. Capability to select individual channels of any axis of SAS.
- c. Individual channel selection of auto throttle system.
- d. Capability to check individual rate gyros for aliveness.
- e. Capability for an end-to-end test of SAS electronics and gyros in all axes.
- f. Display of complete operational status of the AFCS, E/CS, and direct lift control system.
- g. Capability to disengage direct lift control.

8.3.5.1 Introduction. The AFCS control panels (shown in Fig. 18) display channel operational status. System operational status is displayed on the indicators shown in Fig. 37. The operation of the switches and indicators for the E/CS, the A/P, and the SAS, is similar. Therefore in the discussion which follows, the operation of the E/CS pitch axis switch and indicators will be explained and then differences in operation of the remaining axes will be noted. The SAS test, auto throttle and auto pilot trim capabilities will then be explained.

8.3.5.2 Operation

- a. E/CS pitch axis. The E/CS is designed for automatic engagement after warmup. The channel switches are normally in the AUTO position. A triple-redundant, fail-operational provision for first failure and a fail-passive provision for second failure capability is provided. For example, should a first failure occur, the dual-filament status light, located adjacent to the axis switch, will indicate the failure of the defective channel electronics or servo. No change in performance, however, will be detected by the pilot. In addition the pilot will be informed of the failure by the E/CS system warning light and the flashing of the master warning indicator on the System Warning Annunciator Panel, shown on Fig. 37.

The master system warning indicator and the associated E/CS warning light are extinguished by depressing the annunciator PUSH-TO-RESET indicator; this arms the annunciator panel for display of the next malfunction which might occur. Failures detected are differentiated into two types, servo and electronics. If a second failure occurs which is of the same

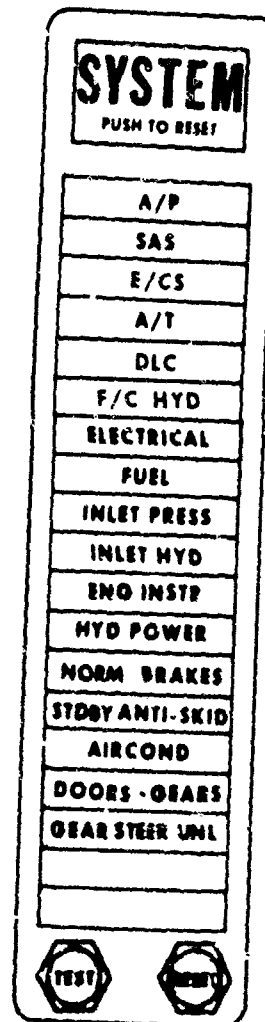
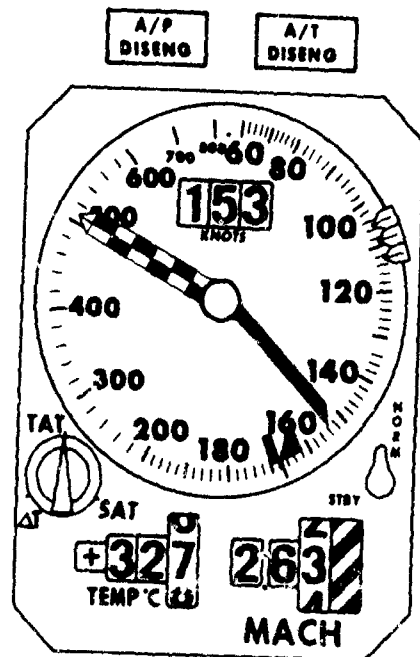


Figure 37. AFCS Warning Indicators

type as the first, the malfunction detection system automatically disconnects the axis, and the OFF light is illuminated. The disengagement transient does not cause out-of-tolerance performance. If a second failure occurs which is of the alternate type to the first failure, the malfunction detection system inherently provides for continued operation. In either case, the pilot is again apprised of the malfunction by means of the annunciator lights.

Subsequent to the automatic disengagement of an axis and the indication of this by the OFF indicator, the pilot is provided with the option of engaging a single channel. Single-channel operation automatically disables the malfunction detection system and requires "hand on" control by the pilot. The pilot may also elect to place the switch to the OFF position (which resets the malfunction detection circuitry) and then back to the AUTO position. If the second failure had been of a transient nature, automatic fail-passive operation would resume; if it was an authentic failure, the malfunction detection system would again automatically disconnect the axis.

- b. E/CS roll axis. Operation for the roll axis is identical to that described above for the pitch axis.
- c. A/P pitch and roll axes. Operation for the A/P pitch and roll axes is similar to that described for the E/CS pitch axis, with the following significant differences:

The A/P switches on the AFCS control panel are normally at AUTO. Autopilot engagement and mode selection is accomplished by the pilot at the A/P-F/D mode selector control panel (Fig. 16).

Upon automatic disengagement of an axis after two failures, the A/P switch on the mode selector panel corresponding to the disengaged axis will move automatically to the OFF position.

Since various modes of A/P operation are available, the pilot may select another mode. Automatic operation is then resumed by placing the channel switch to off (thus resetting the malfunction detection circuitry) then back to AUTO.

- d. SAS roll axis. Operation for the SAS roll axis is similar to that described for the E/CS pitch axis.
- e. SAS pitch and yaw axes. Operation is similar to that described for the E/CS pitch axis with the following significant exceptions: These axes of SAS contain four channels of electronics and an electrical model of the SAS servo actuator. This pro-

vides the capability for sustaining three similar type failures before automatic disengagement occurs.

- f. SAS test. The SAS test switch provides the pilot with gyro testing in the sensor position. This self test, when complete, returns the switch to OFF if no failure exists. If a failure of a gyro is present, the failures are indicated by the indicator lights in the appropriate axis and by the absence of switch movement to OFF. The pilot confidence test is accomplished by moving the switch to SYSTEM position. In this position the gyros are torqued and all channels, including servos, exercised. Failures are indicated in the usual fashion. The switch returns to OFF at completion of test.
- g. Auto throttle. The auto throttle contains two channels; therefore, operation is fail-passive in that automatic disconnect occurs upon the first failure. When single-channel operation is selected, monitoring provides for automatic disconnect in the event of a hard-over or electrical power failure.
- h. Auto pilot trim. Auto pilot trim is normally on whenever the A/P is engaged. Status is indicated by the switch and status light on the AFCS panel. The switch provides for manual disconnect in the event of malfunction.
- i. Direct lift control is normally on when the wings are in the 20-deg sweep angle position. Status is indicated by the switch and status lights on the AFCS panel. The amber light indicates one failed channel, and the direct lift control continues normal operation on its two remaining channels. A second failure disengages the function. The switch provides the flight crew with manual disengagement capability.

8.4 Hydraulic Power System.

8.4.1 Basic Hydraulic System. The operation of the basic hydraulic system is as follows: (See Figs. 26, 27, 28, and 29.)

Flow from the pump discharge ports is routed to the pressure filter modules. Pump pressure ripple is reduced by one pulsation damper for each pump, except for the pumps in systems A and B that are paired on ADS Nos. 1 and 4. These paired pumps are connected to pressure modules that receive the flows from two pumps through isolation check valves and route the resultant flow to one pulsation damper.

In each system's pressure filter module, the fluid is filtered, ported through isolation check valves and combined, at which point system pressure is sensed and transmitted to an indicator on the flight engineer's panel (See Fig. 29). A system relief valve between pressure and return protects the system against overpressure. A pressure

switch in each pump's discharge line actuates its individual low pressure warning light on the flight engineer's panel. The flow then continues into the hydraulic distribution network.

The system return flow is directed through the fluid-to-air heat exchanger, the return filter module, and into the reservoir. From the reservoir, the fluid goes through the supply shutoff valves, the heat exchanger control modules, the fluid-to-fuel heat exchangers and into the pumps.

The air supply to the air-fluid heat exchanger is controlled by an input from the air data system (total temperature) and engine power control lever position. Air flows through the exchanger when ram air temperature is less than 250°F and engine power control lever position is below 12,000 lb/hr fuel flow to engine. Hydraulic flow to the fuel-fluid heat exchanger is cut off by the fluid bypass shutoff valve which is controlled by the engine power control lever. Fluid flows through the exchanger when engine power control lever position is above 12,000 lb/hr. Hydraulic fluid and fuel flow through the heat exchangers is regulated by pressure drop, that is, at high flows partial bypass occurs.

The return filter is a two-stage unit arranged so that normal flow passes through both stages and high flow bypasses the first stage. Both stages are bypassed if the pressure drop becomes excessive. A hydraulic temperature sensor in the return filter module transmits the return temperature electrically to the return temperature indicator on the flight engineer's panel.

The hydraulic reservoir is pressurized by a small piston connected to the system pressure (boot strap) and by engine bleed air pressure. At system start, the air pressure provides immediate pump pressurization. The reservoir fluid quantity transmitter and fluid pressure transmitter are connected to corresponding indicators on the flight engineer's panel. A quantity relief valve, actuated by the reservoir piston, protects the reservoir against overpressure by dumping fluid overboard if the reservoir piston should reach the fully extended position. The reservoir air release valve removes air expelled from the fluid.

The electrically operated supply shutoff valves and pump depressurization valves are controlled by switches on the flight engineer's panel.

Case drain flow from each pump is directed through a separate case drain module, incorporating a nonbypass filter, and a temperature transmitter for individual pump temperature read-outs on the flight engineer's panel. The case drain return and pressure filters are equipped with differential pressure detectors for clogged filter detection.

The warm-up module, with a thermostatically controlled valve, a pressure relief valve, and an electrically operated shutoff valve connected in series, is used during low temperature system start up. Low hydraulic fluid temperatures open the thermostatically controlled valve and energy is added to the system by the pressure drop across the relief

valve. Warm fluid is circulated in the hydraulic distribution network by operation of the control surfaces. Fluid to the landing gears is warmed by fluid circulation through the landing gear subsystem. (See specification D6A10108-1.) The flight crew is provided with a check on the hydraulic system performance during system warmup by observing the hydraulic return temperatures and by comparing the movement of the control surface position indicators with the manual control inputs. The electrically operated valve in the warmup module closes whenever the landing gear oleo is extended, protecting the hydraulic system against overheating if the thermostatically controlled valve should fail.

Each system return filter module contains system fill and drain connections. When the system is filled, the incoming fluid is filtered by the two-stage return filter.

8.4.2 Standby Hydraulic System Operation. The standby hydraulic system operates as follows: (See Fig. 26.)

When the landing gear selector handle is moved to the down or to the standby extend position, the standby control valve in the B system ports fluid through a flow limiter to the standby hydraulic motor, driving the fixed displacement pump in the standby system.

When the guarded standby electric pump switch on the flight engineer's panel is closed, the standby ac electric motor driven pump is started. This pump is normally used for ground operations only.

The flow from the hydraulic motor driven pump or/and from the electric motor driven pump goes through pump isolation check valves and is filtered by one nonbypass filter in the pressure filter module. The pressure filter module contains a pressure switch which actuates the STANDBY PRESSURE light on the flight engineer's panel. A relief valve between pressure and return protects the standby system against overpressure.

Standby system return flow is combined with the case drain flows from the hydraulic motor driven pump and the electric motor driven pump. The combined flow is then directed through the return filter module, incorporating one two-stage bypass type filter.

From the return filter module, the flow goes through the reservoir to the pump suction ports. The reservoir is pressurized by the boot strap method. A transmitter, operated by the reservoir piston, actuates the standby fluid quantity indicator on the flight engineer's panel. The reservoir air release valve removes air expelled from the fluid. A quantity relief valve, actuated by the reservoir piston, protects the standby reservoir against overpressure by dumping liquid overboard if the reservoir piston should reach the fully extended position.

The standby pressure filter and return filter are provided with differential pressure indicators for clogged filter detection. The return filter module has fill and drain connections. When the standby system is filled, the incoming fluid is filtered by the return filter.

8.4.3 Operation - All Engines Out Hydraulic Power. The ALL ENGINES OUT HYDRAULIC POWER switch, located on the pilots' control stand, operates the all engines out power supply.

- a. The 2707-100 (GE) airplane. (See Fig. 26.) Placing the ALL ENGINES OUT HYDRAULIC POWER switch in the ON position operates the ram air turbine (RAT) door control valve which ports pressure to the RAT actuator to extend the RAT into the air stream. Placing the ALL ENGINES OUT HYDRAULIC POWER switch in the OFF position causes the RAT to be retracted back into the airplane fuselage.
- b. The 2707-100 (P&W) airplane. Placing the ALL ENGINES OUT HYDRAULIC POWER switch in the ON position redistributes the hydraulic loads among the basic hydraulic systems by selectively closing the electrically operated flight controls pressure shutoff valves. The all engine out hydraulic distribution is shown in Table VIII. Returning the ALL ENGINES OUT HYDRAULIC POWER switch to the OFF position restores the hydraulic load distribution to normal.

Table VIII All Engines Inoperative Hydraulic Load Distribution

		A System	B System	C System
Pitch axis	Master servo	Off	Off	On
	SAS servo	Off	Off	On
	Direct lift servos	Off	Off	On
	Main elevator	On	Off	Off
	Inbd aux. elev.	Off	Off	On
	Outbd. aux. elev.	On	Off	
	Tip elevons	Off	On	Off
Roll axis	Master servo	Off	Off	On
	SAS servo	Off	Off	On
	Spoilers	On	On	Off
	Ailerons	Off	Off	On
Yaw axis	SAS servo	Off	Off	On
	Rudder	Off	Off	On

9. SUBSYSTEM TEST PLAN

9.1 Scope.

9.1.1 Types of Testing.

9.1.1.1 Developmental. Developmental Testing establishes design criteria, component and subsystem design concepts, and design details in order to attain specification requirements.

9.1.1.2 Verification. Verification Testing proves compliance of the final design with specification requirements.

9.1.1.3 Quality Assurance. Quality Assurance Testing assures compliance with drawings, specifications, and procedures in the manufacture, assembly, or installation of parts, components, or subsystems. Quality assurance provisions exist at all levels in the form of inspection procedures, procurement specification provisions, drawings, and process specifications as a normal part of the manufacturing process.

9.1.1.4 Reliability. The reliability criteria of Par. 4.1.1 applies specifically to production systems operated in scheduled airline service. Prototype subsystem compliance with this criteria shall be accomplished as follows:

9.1.1.4.1 Reliability Tests. Tests specifically designed to verify the reliability of the prototype subsystem, shall not be conducted. Data obtained from tests conducted under Sec. 9 shall be applied to the reliability analysis specified in Par. 9.1.1.4.2 and extrapolated to anticipated airlines operational conditions.

9.1.1.4.2 Reliability Analysis. A reliability analysis shall be performed to demonstrate that the requirements of Par. 4.1.1 can be achieved. This shall be accomplished as follows:

- a. A reliability growth forecast curve shall be established based on historical experience.
- b. A Phase III target reliability level shall be established to measure achievement toward the production system reliability criteria.
- c. Design data and test results shall be applied to a reliability analysis model incorporating:
 - (1) Block diagrams summarizing the logical relationships between components success-malfunction criteria and system success-malfunction criteria.

- (2) Mathematical reliability models derived from (1) and incorporating minimum equipment requirements for continued flight.
- (3) Mathematical reliability models simulating typical airline operations and routes.
- d. Comparison shall be provided with the Phase III targets and the results extrapolated to determine expectation of achieving the requirements of Par. 4.1.1 in airline operation.

9.1.2 Levels of Testing. Testing shall be conducted at the following levels:

9.1.2.1 Materials. Laboratory testing of metals, fluids, and non-metallic materials used in component and subsystem design.

9.1.2.2 Components. Laboratory testing of representative and actual prototype components and assemblies and standard components such as fasteners, bushings, and bearings.

9.1.2.3 Subsystem Laboratory Tests. Laboratory testing of the complete subsystem or major portions thereof, such as primary flight controls, secondary flight controls, automatic flight controls, and hydraulic power systems.

9.1.2.4 Airplane Ground Tests. Ground testing of the subsystem as installed in the prototype airplane(s).

9.1.2.5 Flight Tests. Testing of the subsystem during flight of the prototype airplane(s).

9.2 Materials Testing.

9.2.1 Developmental Testing.

9.2.1.1 Bearing Materials and Lubricants. Laboratory testing of combinations of materials and lubricants shall be conducted to develop bearing design criteria for such bearing applications as hydraulic actuators, pulleys, torque tubes, and control rods. These tests shall determine load-life wear data and operating characteristics under applicable conditions of temperature, loads, direction of applied loads, vibration, angle and speed of rotation, and contamination.

9.2.1.2 Hydraulic Fluid Laboratory Evaluation. Hydraulic fluid laboratory evaluation shall be conducted on a continuing basis to insure that advantage may be taken of any improvement in fluid characteristics. Physical and chemical property testing shall be done on a variety of fluids.

9.2.2 Verification Testing.

9.2.2.1 Bearings. Laboratory testing shall be conducted to verify the use of the materials selected during the testing described in Par.

9.2.1.1.

9.2.2.2 Hydraulic Fluid. Functional testing in pump loop test set-ups shall be used to verify the performance of the selected fluid and/or any other fluids that show positive promise of improvement over the selected fluid.

9.3 Component Testing.

9.3.1 Developmental Testing.

9.3.1.1 Primary and Secondary Flight Control Components. The Developmental Servo Simulator (DSS) shall be utilized to test a conceptual servo actuator assembly consisting of a representative master servo, an SAS servo, a trim servo, and triple hydraulic actuators. A separate conceptual master servo of a different design shall also be tested on the DSS. The results of these tests shall be used to establish design criteria and requirements for the prototype airplane flight servos. These tests shall include static response, frequency response, transient response, flutter, dynamic load pressure feedback, structural compliance feedback, temperature environment, thermal shock, overheat temperature operation, and failure mode evaluation tests. A more detailed explanation of the DSS testing may be found in D6A10441-1.

All major components procured from suppliers shall be subjected to development tests by them, as required, to develop a design that meets the procurement specification requirements.

9.3.1.2 Flight Control Electronic Components. Various development tests at the component level shall be conducted on the Automatic Flight Control System (AFCS), Electric Command System (E/CS), and the Direct Lift Control System (DLC). These tests break down into two general categories: (a) tests conducted by the equipment supplier and (b) tests conducted by Boeing. The following are descriptions of the tests to be performed in each area:

9.3.1.2.1 Supplier Development Tests.

- a. Circuit Concept Tests. For any given application, several different types of circuitry approaches are usually considered. These approaches shall be breadboarded and, as a minimum, room temperature tests conducted. From these test results, as well as other considerations (cost, reliability, etc.), the best circuit for the application shall be selected.

- b. Breadboard Circuit Environment Tests. The particular circuit selected at the conclusion of tests conducted in item (a) above shall be subjected normally to high and low temperature tests. These tests verify that the circuit meets its application requirements over the temperature range for such things as stability, linearity, drift, etc.
- c. Prototype Environment Tests. The circuit that was optimized in tests conducted in item (b) above is now packaged in an arrangement similar to that used in the final design. The circuit shall then be tested to show that the final arrangement has not changed circuit characteristics. These tests shall include checks at room ambient conditions, low temperature, high temperature, vibration, and, in some instances, altitude.
- d. Prototype Integration Environment Tests. The prototype circuits shall be integrated. Tests shall then be conducted to show that the physical arrangement and integration of the various circuits has not changed the intended functional characteristics of the various circuits. These tests shall be conducted under the environments of room temperature, low temperature, high temperature, and vibration.

9.3.1.2.2 Boeing Development Tests. Additional component development tests shall be conducted at Boeing to optimize the gains, switching levels of monitor circuits, self-test signal levels, channel tracking errors, and circuit stability. These tests shall be conducted as individual bench tests and shall also utilize the Development Servo Simulator.

The design of flight control electronic components shall be further optimized during subsystem laboratory testing as defined in Par. 9.4.

9.3.1.3 Hydraulic Power Components.

- a. Pumps. Preproduction testing on two prototype pumps (one from each of two suppliers) shall be conducted. The testing shall consist of a 2,500-hr endurance test that includes flow and thermal cycling, cavitation tests, pressure tests, thermal shock tests, and operational tests. MIL-P-19692B shall be used as a guide. Testing shall be done by the supplier in accordance with the requirements of 10-60962, Hydraulic Pump, Engine Driven. Additional developmental testing shall be done at both Boeing and the supplier's facilities.
- b. Reservoir. Development testing of the reservoirs shall include endurance cycle testing and temperature cycling. Testing of reservoir assemblies or reservoir components shall be done by suppliers in accordance with Boeing purchase specification requirements. Hydraulic power system laboratory tests and flight control servo simulator tests shall be utilized in the reservoir development process.

- c. Tubing & Fittings. Development testing of both permanent and reconnectable fittings and tube-to-fitting connections shall be conducted. Impulse and flexure tests shall be conducted on tubing and on permanent and reconnectable fittings as well as repeated use tests on reconnectable fittings. Boeing shall conduct the tubing and fitting development test program to assure uniform comparative data.
- d. Seals and Swivels. Design development testing on applicable types of seals shall be conducted. The seal development program shall include fluid and material compatibility tests and configuration evaluation of static, linear, and rotary dynamic seals. The life and compatibility tests shall be done in conjunction with the development testing of the hydraulic swivels. All development testing shall be done at Boeing.
- e. Hoses. Impulse and flexure testing on hose sizes that must be developed shall be conducted. Tests will be per requirements of Boeing procurement specifications and shall be conducted by the supplier(s).
- f. Other Hydraulic Components. All other hydraulic power system components shall be developed and tested in accordance with the Boeing procurement specifications. Final development testing shall be done in the Boeing laboratory and on the Flight Control Servo Simulator.

9.3.2 Verification Testing.

9.3.2.1 Primary and Secondary Flight Control Components. Verification testing of purchased components shall be conducted by the supplier to prove compliance with the design requirements of the individual procurement specifications.

The Flight Control Servo Simulator (FCSS) shall be used for verification testing of the components as installed within the subsystem. This verification testing shall consist of functional checkout tests and performance tests. The functional checkout tests shall ensure that each component is operable throughout the required working ranges without binding or interference and that operation occurs in the proper direction and sequence. The performance tests shall ensure that each component meets the required performance criteria of the applicable installation drawing, procurement specification, and this specification and shall include static response, frequency response, and transient response tests. A more detailed explanation of the testing accomplished on the FCSS may be found in D6A10441-1.

The following components of the primary and secondary flight control systems shall be tested: master servos, SAS servos, direct lift servos, feel and centering mechanisms, trim motors and actuators, torque tube drives, cable runs and cable tension regulators, all surface actuators,

lockouts and programmers, rudder travel limiter, and power control units for wing sweep, flaps, and slats.

9.3.2.2 Flight Control Electronic Components. Verification tests to prove compliance with specification requirements shall be conducted by the component suppliers and by Boeing. These tests shall be as follows:

- a. Supplier Verification Tests. Prototype components, representing the final design, shall be subjected to performance verification tests before, during, and after exposure to all specified environments. Reports of test results shall be submitted to Boeing for review, comment, and approval.
- b. Boeing Verification Tests. Prototype components shall also be subjected to verification tests at Boeing. These tests will be conducted as bench tests and on the Development Servo Simulator. They shall include some limited environmental tests and shall be aimed at verifying supplier results in such areas as frequency response, phasing, transducer thresholds, transducer linearity, monitor circuit threshold levels, and circuit tracking capability. The satisfactory completion of subsystem laboratory tests outlined in Par. 9.4 shall also be utilized to verify component design compliance.

9.3.2.3 Hydraulic Power.

- a. Pumps. The hydraulic pumps shall be endurance tested to a minimum of 300 hr before use in the prototype airplane. Performance of the hydraulic pumps shall be verified by the supplier in accordance with the procurement specification and by their performance during flight control servo simulator and laboratory hydraulic power system testing.
- b. Reservoir. The performance and endurance data obtained from use of the reservoirs in the servo simulator and in hydraulic power system laboratory tests, including cold temperature, shall be used as verification of reservoir performance within the system.
- c. Tubing & Fittings. Tubing and fittings in all sizes and types to be used in the prototype airplane shall be subjected to impulse cycle, proof pressure, and burst pressure testing at Boeing as verification that these components will satisfy applicable strength requirements.
- d. Seals. Verification testing of the suitability of seals shall occur as a result of the operation and testing of the various units in the flight control servo simulator, hydraulic power system laboratory testing, and component verification testing.
- e. Hoses. All sizes of hoses to be used in the prototype airplane shall be impulse cycle, proof, and burst tested at Boeing in

addition to that testing required of the supplier by procurement specifications.

- f. Verification of the performance of other hydraulic power system components shall be as a result of the flight control servo simulator testing and hydraulic power system laboratory testing in addition to the testing required of the supplier by the procurement specifications.

9.4 Subsystem Laboratory Testing.

9.4.1 Developmental Testing.

9.4.1.1 Flight Control Servo Simulator (FCSS). The primary task of the FCSS shall be to evaluate the dynamic nature of the flight control servo systems and the flight controls and hydraulic subsystem of which the servos are a part. The FCSS shall be arranged so that the entire subsystem may be joined hydraulically, mechanically, and electrically to simulate the complete integrated airplane subsystem. In addition, the arrangement shall permit major portions of the subsystem to be isolated and tested separately with controlled inputs. (Refer to Fig. 38.)

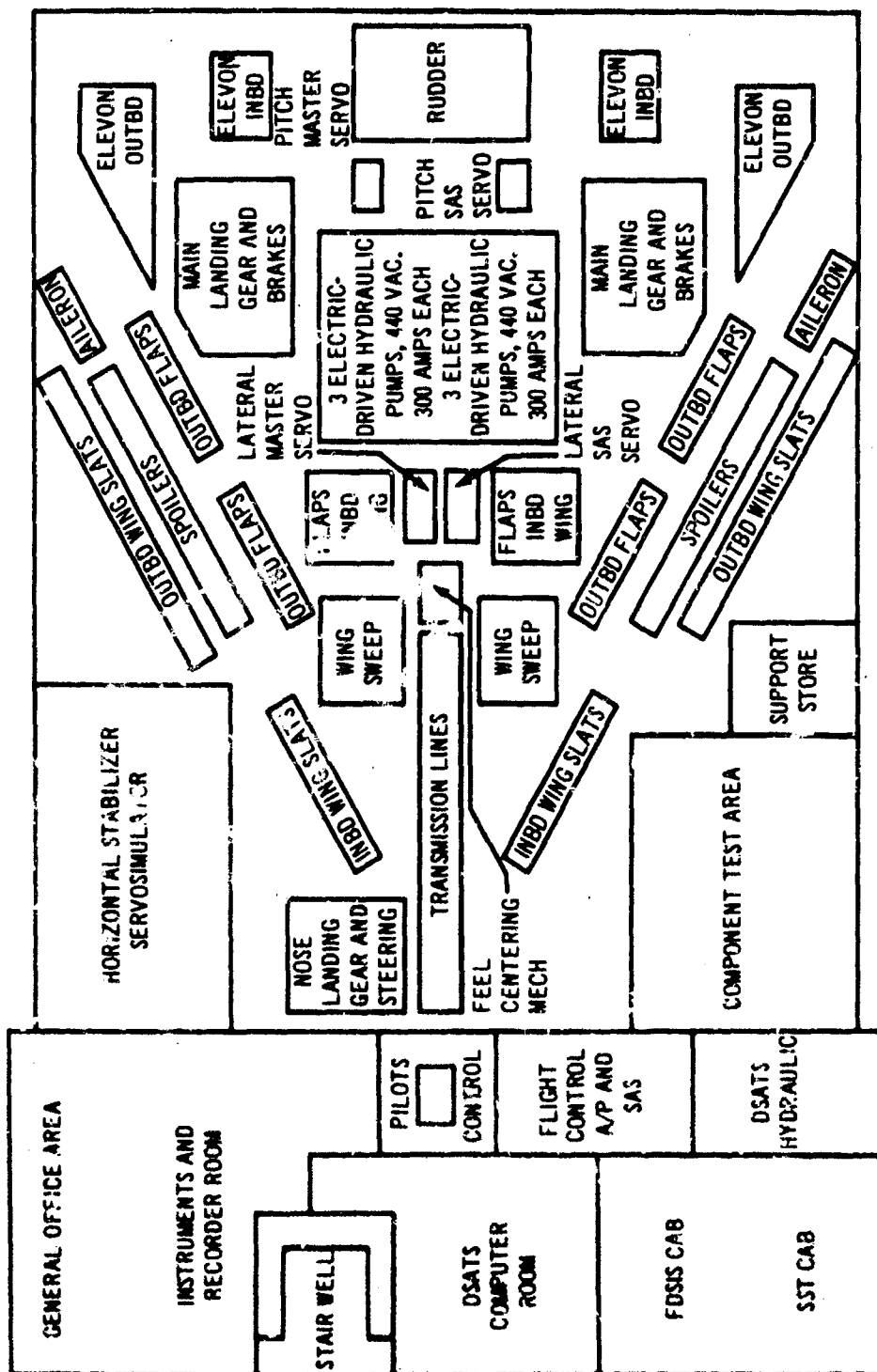
The testing accomplished on the FCSS shall determine the stability, under simulated operational conditions (refer to Fig. 39) of each of the primary flight control servo actuators and each of the high lift control systems and shall determine the friction and springiness of the mechanical control systems and the interaction between the mechanical and electrical command systems. The FCSS shall be used in determining the accuracy of the phasing of ailerons, spoilers, and elevons in roll control and elevators and elevons in pitch control. The action of slats and flaps shall be correlated with wing position.

The effect of loads and structural springiness on the stability of the primary and automatic control systems shall be established. The flow phenomena in the hydraulic systems under all flight conditions shall be determined. The FCSS shall be used to ensure that each component operates within the required working range without binding or interference when installed within the complete subsystem. The failures of elements within the subsystem shall be simulated under various flight conditions to determine automatic and manual recovery capabilities and techniques.

The results of these tests shall be used to further develop the design of individual components, as required, to ensure that the overall requirements of the subsystem are met.

The FCSS shall determine that the airplane handling characteristics are satisfactory under at least the following conditions:

Ground Roll:	Wings at 30 deg. Flaps, slats and landing gear each operated simultaneously with flight controls.
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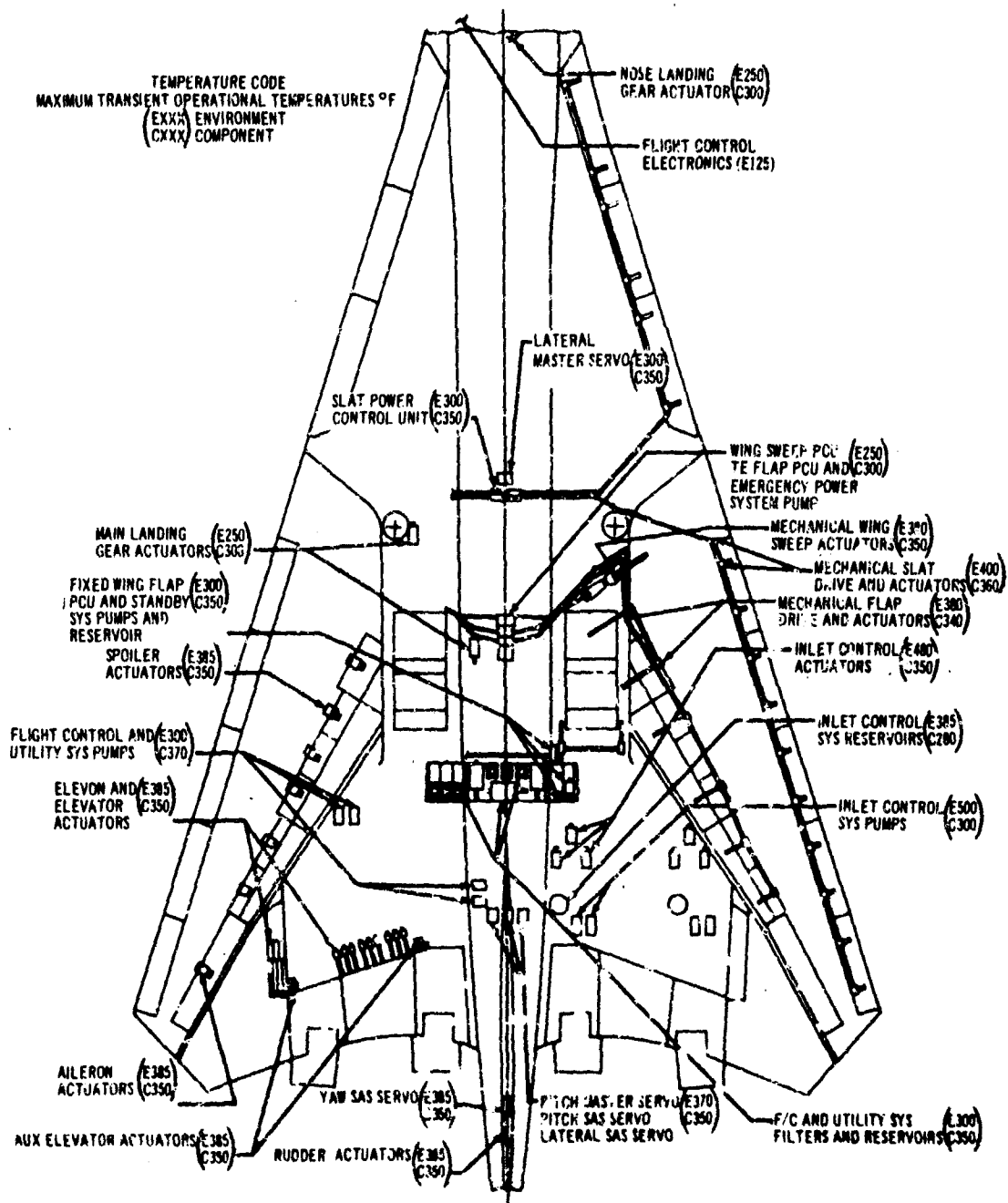


Figure 39. Hydraulic and Flight Control Component and Environmental Temperatures

Takeoff: Wings at 30 deg. Flaps and slats extended and landing gear retracted.

Subsonic Climb: Wings at 42 deg. Flaps and slats retracted.

Subsonic Cruise: Wings at 42 deg. Air loading to simulate altitudes from 1,000 to 40,000 ft. Mach number simulated to 0.9.

Supersonic Cruise: Wings at 72 deg. Air loading to simulate altitudes to 80,000 ft and full Mach capability, system operating temperatures, and simulated ambient temperatures at selected locations.

Subsonic Letdown: As in subsonic climb but with high system temperatures.

Approach and Landing: Wings at 20 deg. Flaps, slats, and landing gear each operated simultaneously with flight controls.

9.4.1.2 Primary and Secondary Flight Controls in FCSS. The following testing shall be accomplished on the FCSS in order to meet the intent of the developmental testing described in Par. 9.4.1.1.

9.4.1.2.1 Longitudinal, Lateral, and Directional Systems.

- a. Each system shall be tested to ensure that it is not possible to manually induce oscillations.
- b. The spring rates of the appropriate components and structural mountings of each system shall be determined.
- c. The resolution between pilot inputs and master servos and direct lift servos, between master servos and direct lift servos and control surfaces, between pilot inputs and control surfaces, between SAS inputs and control surfaces, and between trim inputs and control surfaces shall be determined.
- d. Each system shall be tested for manual transient and frequency responses under all load conditions.
- e. Each system shall be tested for electrical transient and frequency responses under all load conditions.
- f. Control surface movements due to engagement or disengagement of hydraulic systems, servos, or surface actuators shall be evaluated.

- g. The rates of master servo, SAS servo, and direct lift servo outputs and control surfaces shall be determined under all load conditions.
- h. The operation of each system shall be tested in the mechanical control mode, E/CS control mode, A/P mode, SAS mode, trim control mode, and appropriate combinations of these modes.
- i. The tests described in a, c, d, e, g, and h shall be performed with one, two, and three hydraulic systems operable and with hydraulic power simulating an all-engine-out condition.
- j. The breakout forces and force gradients of the longitudinal and lateral axes on the electric command mode shall be tested and values established for subsequent flight testing. This mode shall also be checked for engagement and disengagement transients.
- k. The longitudinal, lateral, and directional axes centering shall be tested in combination with breakout force, stability and force gradient, to assure that no objectionable flight characteristics are apparent. The mechanical system centering breakout forces and force gradients shall be tested with the electric command system disengaged.

9.4.1.2.2 Wing Sweep and High Lift Systems.

- a. Each system shall be tested to ensure that it is not possible to manually induce oscillations.
- b. The spring rates of the appropriate components and structural mountings of each system shall be determined.
- c. The operation of power drive units, torque tube systems, and surface actuators shall be tested to determine the accuracy and repeatability of commanded position and to ensure that rates meet specification requirements.
- d. The amount of asymmetry that exists during normal system operation and the degree of asymmetry required to shut off or h of the systems shall be determined.
- e. The tests described in a. and c. shall be performed with one, two, and three hydraulic systems operable and with hydraulic power simulating an all-engine-out condition.

9.4.1.2.3 Combined Systems Operational Tests.

- a. Spoiler deflection as a function of aileron or elevon position, both with and without speed brake and direct lift servo inputs.

- b. Outboard spoiler panel lockout as a function of wing and TE flap position.
- c. Spoiler faired position as a function of wing TE panel position.
- d. Aileron operation and lockout as a function of TE flap position.
- e. The operation of the elevon programmer and elevon surfaces, for longitudinal and lateral control, as a function of wing and TE flap position.
- f. Auxiliary elevator operation and lockout as a function of TE flap and thrust reverser position.
- g. Operation of rudder, with combined inputs from the lateral and directional systems, as a function of wing and TE flap position.
- h. The tests described in a. through g. shall be performed with appropriate inputs from pilots, SAS, autopilot, and trim systems.
- i. Operation of wing sweep/flap programmer and resultant scheduled operation of wings, TE flaps and LE slats, including operation of alternate slot system.
- j. Operation of the wing sweep system following actuation within the asymmetry detection systems of wing sweep, TE flaps, and LE slats.

9.4.1.2.4 Failure Tests.

- a. The operation of each system shall be determined with simulation of typical failures described in D6A10064-8, Reliability Analysis Document - Flight Controls.
- b. The operation of each system shall be determined with simulation of typical failures of the automatic flight controls and hydraulics systems as described in D6A10064-4 and D6A10064-10.

9.4.1.3 Flight Control Electronics in FCSS. The AFCS, E/CS, and DLC systems shall be optimized to meet desired performance requirements of this specification by refining gains, shaping, and gradients. This optimization shall be primarily accomplished by changing fixed resistors, variable resistors, and capacitors. All the functions and modes of the system shall receive this type of testing and are listed below:

- a. Electric Command System
- b. Stability Augmentation System

- c. Auto Throttle System
- d. Angle-of-Attack Warning and Control System
- e. Direct Line Control System
- f. Autopilot Pitch and Roll Modes
 - (1) Vertical Speed
 - (2) Altitude Hold
 - (3) Control Wheel Steering
 - (4) Altitude Capture
 - (5) Airspeed Hold
 - (6) Mach-Altitude
 - (7) Glide Slope
 - (8) Land
 - (9) Go-Around/Takeoff
 - (10) Roll Altitude/Heading Hold
 - (11) Heading Select
 - (12) Inertial Navigation
 - (13) Localizer
 - (14) VOR

9.4.1.4 Hydraulic Power System in FCSS. The complete airplane hydraulic system and its loads shall be simulated. The simulator system shall incorporate airplane components, the lengths and sizes, types of joints, and mountings. Variable-speed drives shall provide simulated airplane engine speeds and accelerations, providing them a capability of complete dynamic simulation. The fluid temperature variation capability shall vary from room temperature to +430°F. System dynamics at various temperatures, load conditions, and condition of failure shall be demonstrated. Pressure surges, transients, compliance, and stability shall be thoroughly investigated.

The hydraulic power system's capability to operate the flight controls and utility functions at their specified performance requirements shall be demonstrated by use of the flight control serve simulator

Heat rejection requirements, temperature data, pressure, pulsation and transient pressure characteristics, flow requirements, and leakage characteristics shall be specifically monitored.

Controllability of the airplane shall be demonstrated under the following abnormal conditions of available hydraulic power:

- a. Two hydraulic systems operative
- b. One hydraulic system operative
- c. Hydraulic Power available with all engines out. (See Par. 9.7.)

9.4.1.5 Other Laboratory Developmental Testing.

9.4.1.5.1 Mockups. Full-scale operating mockups of the critical areas shall be constructed to check clearance of components and routing of control cables and hydraulic and electrical lines.

9.4.1.5.2 Flight Control Electronic Systems.

- a. Supplier Bench Testing. The components comprising the system shall be connected together and bench tested. The interfaces with other systems, such as air data, servos, various receivers, etc., shall be simulated as applicable loads, input signals, etc. The tests shall be conducted primarily in a room ambient environment and shall optimize system performance by pointing out areas needing gain or shaping changes, as well as verifying phasing relationships, and allowing necessary component modifications required to satisfy electromagnetic interference requirements.
- b. Boeing Bench Testing. Boeing shall conduct a series of subsystem tests to optimize various parameters. These tests differ from those conducted by the supplier in that actual interface hardware shall be used in place of simulated inputs and outputs.

9.4.1.5.3 Hydraulic Power System Tests. Tests shall be conducted to develop and evaluate the system and its components by operation of the major portions of a typical hydraulic power system under various load and flow conditions. The test set-up shall consist of essential elements such as the pumps with variable-speed drives, heat exchangers, filters, pulsation dampers, modular packages, etc. Additional cold temperature testing of this portion of a typical hydraulic power system shall be accomplished by use of a cold box. Operation and warm-up capability tests shall be conducted at temperatures of from -50° to +60°F. Response, warm-up, and extreme temperature performance of the hydraulic power system shall be demonstrated with the hydraulic power laboratories test set-up.

9.4.2 Verification Testing.

9.4.2.1 Flight Control Servo Simulator (FCSS). The testing described in Pars. 9.4.1.1, 9.4.1.2, 9.4.1.3, and 9.4.1.4 shall be conducted as verification testing when the developmental testing has resulted in an acceptable design configuration.

The subsystem shall be endurance tested during simulated flights in which the FCSS is programed to subject each of the systems to a schedule of operation and loading that is representative of the various phases of a typical airplane flight as stated in Par. 9.4.1.1. The subsystem shall be endurance tested to an equivalent of 75 hr of simulated SST flights. This shall be equivalent to 75 SST mission cycles (flights). At least 25 hr of this testing shall be accomplished prior to first flight of the prototype airplane.

9.4.2.2 Other Laboratory Verification Testing.

9.4.2.2.1 Flight Control Electronic Systems. Subsystem verification testing shall be accomplished in two phases: a. at the supplier and b. at Boeing. The following is a description of this testing and its intended purpose:

- a. Supplier Tests. The supplier shall tie together electrically, and where applicable physically, the entire subsystem. He shall conduct, in a room ambient environment, tests to verify that the subsystem meets the specified requirements for electromagnetic interference (generation and susceptibility) and voltage stability.
- b. Boeing Tests. Boeing shall conduct tests for electromagnetic interference and voltage stability similar to the tests described in item a above, except that the Flight Control Electronic Systems shall be subjected to these tests when they are operated in conjunction with other subsystems (Air Data, Inertial Navigation, etc.).

9.4.2.2.2 Hydraulic Power. Subsystem performance under cold start and warm-up conditions shall be verified by the testing described in Par. 9.4.1.5.3.

9.5 Airplane Ground Tests.

9.5.1 Proof Tests.

- a. Prior to first flight, the flight controls of the airplane shall be proof-loaded to 100 percent limit load. The system shall be loaded from the pilots' input to the control surface attachment. The load shall be applied in the direction, and with the surface in the position determined by analysis to be critical. The tests shall be conducted on the following controls: elevons, elevators,

rudder, ailerons, spoilers, leading edge slats, and trailing edge flaps.

- b. Prior to taxi testing and first flight, the entire hydraulic system, as installed in the airplane, shall be subjected to the proof pressure specified in Par. 4.2.3.1.

9.5.2 Functional Tests. Functional tests shall be conducted to determine that the operation of the subsystem as installed in the airplane meets the operating criteria developed during subsystem testing on the FCSS. The following tests shall be included:

- a. Control surface deflection relative to control column and pedal force and position inputs.
- b. Mechanical and electric command modes centering, breakout force, and force gradients.
- c. Electric command mode and AFCS engagement and disengagement transients.
- d. Subsystem operation with one and two hydraulic systems in-operative.
- e. Wing sweep positions and rates and wing sweep-high lift system interlocks and asymmetry detection systems.
- f. The scheduled operation of elevons, auxiliary elevators, spoilers, and direct lift servos as a function of wing and flap positions.
- g. The calibration of surface position indicators.
- h. The hydraulic system pressure transients and temperature rise versus time characteristics during operation of flight controls and during operation of the landing gear while the airplane is jacked.
- i. Stability Augmentation System. Torque each rate gyro and note proper channel operation and surface motion.
- j. Angle-of-Attack Warning and Control System. From a static pressure source, place a predetermined pressure on the Air Data Computer, then set the Angle-of-Attack Vane at a predetermined angle of attack, note control column shaker operation. Repeat above test at another predetermined pressure and angle of attack, note control column pusher operation.
- k. Electric Command System. Put pitch commands into the control column and note surface motion for proper directions. Repeat the test for roll commands. With wings in the 20-degree sweep

position, the control column pitch inputs shall also actuate direct lift control spoilers.

- l. Auto Throttle System. From a static pressure source, place a predetermined cyclic pressure into the Air Data Computer and throttle handle motion should occur.
- m. Autopilot System. Predetermined signals shall be inserted into the autopilot at the Interface Box Test Connectors. The control surfaces and/or flight director shall be observed to determine proper operation. The modes of operation that shall be tested are as follows:

- (1) Vertical Speed
- (2) Altitude Hold
- (3) Altitude Capture
- (4) Airspeed Hold
- (5) Mach - Altitude
- (6) Glide Slope
- (7) Land
- (8) Go-Around/Takeoff
- (9) Roll Attitude /Heading Hold
- (10) Heading Select
- (11) Localizer
- (12) VOR
- (13) Inertial Navigation
- (14) Control Wheel Steering. This test will use inputs at the wheel rather than at the Interface Box Test Connectors.

In addition to tests l. through m., tests will be run using the autopilot built-in test equipment to further verify proper autopilot axis and channel operation.

9.5.3 Electromagnetic Interference Test. This subsystem shall be tested as a part of the airplane level electromagnetic interference test. A successful airplane level test, as defined in D6A10107-1, shall constitute verification of the requirements of this subsystem.

9.5.4 Maintainability Tests. The suitability of service and access provisions shall be determined by the observation of technicians performing maintenance and servicing tasks on the subsystem. All activities involving scheduled checks, repairs, and servicing of line replaceable units (LRU) shall be observed and data recorded.

9.6 Flight Testing. The flight test program shall cover operation of actual test aircraft under conditions outlined in Par. 9.4.1.1 as a minimum. Temperatures of system components as well as their environment shall be monitored. Within the 100 hours of flight testing, the specific allocation of flight test time to the Flight Controls and Hydraulics Subsystem shall be 33 hr for Stability and Control Tests.

Flight tests shall include evaluation of control effectiveness, handling qualities, flutter characteristics, buffet, and vibration. Inflight evaluation of all modes of operation of primary and secondary flight controls shall be made. Hydraulic systems temperatures, pressures, and pressure transients shall be thoroughly monitored and evaluated. The effects upon the hydraulic system of flight control and landing gear loads under all flight conditions shall be evaluated. Simulated failure conditions shall be imposed and subsystem performance evaluated under these conditions.

The following tests shall be included among those required to meet the objectives of the flight test program:

- a. The mechanical and electric command modes centering, breakout forces, and force gradients shall be tested to verify that they meet specification requirements and that they do not produce objectional flight characteristics.
- b. The electric command mode shall be tested to verify that the engagement and disengagement transients do not exceed specification requirements.
- c. The longitudinal control system shall be tested to verify that, under the most adverse airplane loading conditions, it provides a maneuver capability of at least plus 1.0 and minus 1.0 incremental g from trimmed 1.0 g flight, during the following flight conditions:

Takeoff

Subsonic climb

Supersonic cruise

Subsonic cruise

Subsonic letdown

Approach and landing

- d. The lateral control system shall be tested to verify: that it provides a minimum steady-roll rate of at least 25 deg/sec and a bank angle of 7 deg in one sec for normal takeoff and landing configurations; that it provides, for operation other than takeoff and landing, roll rates of at least 15 deg/sec and a bank angle of 3 deg in 1 sec. This maneuver capability shall be verified during takeoff, subsonic climb, subsonic cruise, supersonic cruise, subsonic letdown, approach, and landing flight conditions.
- e. The Flight Control Electronic Systems shall be tested to verify compliance with specification requirements. These tests shall include, as a minimum, a test of the following systems and modes of operation:
- (1) Stability Augmentation System
 - (2) Electric Command System
 - (3) Auto Throttle System
 - (4) Angle-of-Attack Warning and Control System
 - (5) Auto Trim System
 - (6) Autopilot System Modes:
 - (a) Vertical Speed
 - (b) Altitude Hold
 - (c) Control Wheel Steering
 - (d) Altitude Capture
 - (e) Airspeed Hold
 - (f) Mach-Altitude
 - (g) Glide Slope
 - (h) Land
 - (i) Go-Around/Takeoff
 - (j) Roll Attitude/Heading Hold
 - (k) Heading Select
 - (l) Localizer
 - (m) VOR
 - (n) Inertial Navigation

- f. The wing sweep and high life systems shall be tested to verify proper position response, rate, interlocks, and asymmetry detection function.

9.7 Analyses

- a. Loads. An analysis shall be conducted to verify that the system power requirements are compatible with the aerodynamic surface loads. This analysis shall be updated utilizing the results of wind tunnel and airplane tests.
- b. Stress. Stress analysis shall be conducted to verify the structural design of all components and mountings. This analysis shall be updated utilizing the results of component and airplane tests.
- c. Dynamics. A dynamic analysis shall be conducted to evaluate control response and performance in all flight modes. This analysis shall be updated utilizing the results of FCSS and flight tests.
- d. Fatigue. A fatigue analysis shall be conducted to verify that the components of the subsystem will have the structurally useful life specified. This analysis shall be updated utilizing the results of component development tests, FCSS tests, and flight tests.
- e. Useful Life. An analytical review of applicable design, tests, and service data shall be provided to justify useful life requirements of Par. 4.1.3.
- f. Safety. The safety requirements identified in Par. 4.1.6 shall be verified analytically by the identification of compensating provisions for each failure mode defined in the failure mode effect and criticality analysis.
- g. Electromagnetic Compatibility. An analysis shall be conducted to verify compliance with the electromagnetic compatibility (EMC) requirements of Par. 5.13. Data for this analysis shall be obtained from component tests and the results of ground and flight tests.
- h. All-Engine-Out Hydraulic Power. Data on the windmilling power characteristics of the engine-inlet combination shall be obtained from the engine manufacturer. This data shall be used, together with data on other emergency hydraulic power sources, if any, to determine by analysis the hydraulic power available to control the airplane if all engines fail.

10. PREPARATION FOR DELIVERY

The subsystem shall be delivered as part of the prototype airplane delivered as defined by specification D6A10107-1. Delivery requirements for the subsystem or components shall be defined by the contract.

11. NOTES

11.1 Definitions. The definitions shown in Sec. 6 of specification D6A10107-1 shall apply.

11.2 Supporting Data.

11.2.1 Maintenance Design Guide. The Maintenance Design Guide, D6-9458, Commercial Supersonic Transport, may be used for subsystem maintainability design guidance.

11.2.2 Reliability Analysis. Reliability analyses to show compliance with the requirements of 4.1.1 and to assess the verification data required in 9.1.1.4.2 will be included in documents D6A10064-4, D6A10064-8, and D6A10064-10, respectively, for AFCS, Flight Controls, and Hydraulic Power.

SUPPLEMENT I

Production Airplane Differences from the Prototype Airplane.

The paragraphs below, applicable to the production 2707-100 airplane, are numbered to correspond to the paragraphs that give the respective requirements on the prototype airplane.

5.16 Subsystem Weight.

5.16.1 2707-100 with GE Engines. The weight of the flight controls and hydraulics subsystem shall not exceed 14,450 lb. This weight is an allocation of the overall airplane weight based on analysis and design experience and may be revised as long as the overall airplane weight defined in D6A10107-1 is not exceeded.

5.16.2 2707-100 with P&W Engines. The weight of the flight controls and hydraulics subsystem shall not exceed 14,270 lb. This weight is an allocation of the overall airplane weight based on analysis and design experience and may be revised as long as the overall airplane weight defined in D6A10107-1 is not exceeded.