A. File the attached Temporary Revision/Alerts in the Manual Supplement in ATA Chapter/Section/Subject/Page sequence

B. File this Control Page in front of the Chapter TRs/Alerts.

C. The following list shows active TRs/Alerts together with TRs/Alerts added by this control page.

Chapter Section Subject	Page	TR/Alert No.
22-11-02	201	22-523
22-11-03	403	* 22-529
22-14-01	401	22-521
22-21-01	401	22-522
22-31-02	401	22-520

D. Remove and Destroy the following TRs/Alerts:

* Indicates TRs/Alerts issued with this control page

MAINTENANCE MANUAL

TEMPORARY REVISION No. 22-523

THIS TEMPORARY REVISION IS ISSUED BY BRITISH AIRWAYS ENGINEERING (TECHNICAL INFORMATION SERVICES, G2, TBA, S401, P. O. BOX 10, HEATHROW AIRPORT, HOUNSLOW, MIDDLESEX TW6 2JA).
CAA DESIGN APPROVAL No. DAI/8566/78.

Manual Reference 22-11-02 Page 201 para 2 and 3

(Tasks 22-11-02-022-001 and 22-11-02-402-007)

REASON FOR REVISION

To carry out the requirements of the Safety and Technical Strategy Board Cross-Connection Project G/38/98.

ACTION

After the existing paragraph 2.D.(3) and before 2.D.(4) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

Before the existing paragraph 3.C.(5)and before the existing paragraph 3.C.(6) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

Originator: Gary Kerr
Reference: 757-W-MCR-22-GK-99-587
Workbook: JS 22-009

MAINTENANCE MANUAL

TEMPORARY REVISION No. 22-529

THIS TEMPORARY REVISION IS ISSUED BY BRITISH AIRWAYS ENGINEERING (TECHNICAL INFORMATION SERVICES, G2, TBA, S401, P. O. BOX 10, HEATHROW AIRPORT, HOUNSLOW, MIDDLESEX TW6 2JA).
CAA DESIGN APPROVAL No. DAI/8566/78.

Manual Reference 22-11-03 Page 403 sub task 034-006

REASON FOR REVISION

An alternate part number Autopilot disconnect switch is now in use which has different wire colour codes.

ACTION

Read the following for sub task 034-006

S 034-006

(5) Use the extraction tool to disconnect these eight wires from the switch side of the quick-disconnect terminals:

WIRE LABEL AND COLOR CODE TABLE			
TERMINAL NUMBER	P/N 721101A1 SWITCH SIDE WIRE LABEL	AIRFRAME SIDE WIRE COLOR CODE	ALTERNATE SWITCH P/N SW43896 WIRE COLOR CODE
1	BKB	WHITE/BLACK	BLACK
2	BRN	WHITE/BLUE	BROWN
4	GRN	WHITE/GREEN	GREEN
5	YEL	WHITE/YELLOW	YELLOW
7	BKG	RED/GREEN	GREY
8	BKY	WHITE/BROWN	ORANGE
10	BKA	WHITE/PINK	WHITE
12	BLU	BLACK/YELLOW	BLUE

CAUTION: Two types of disengage switch are available. Attention is drawn to the wiring differences shown in the table above.

Originator: T. D'Arcy
Reference: 6143
Workbook: JS 22-013

MAINTENANCE MANUAL

TEMPORARY REVISION No. 22-521

THIS TEMPORARY REVISION IS ISSUED BY BRITISH AIRWAYS ENGINEERING (TECHNICAL INFORMATION SERVICES, G2, TBA, S401, P. O. BOX 10, HEATHROW AIRPORT, HOUNSLOW, MIDDLESEX TW6 2JA).
CAA DESIGN APPROVAL No. DAI/8566/78.

Manual Reference 22-14-01 Page 401 para 2.C and 3.C.

(Tasks 22-14-01-004-001 and 22-14-01-404-008)

REASON FOR REVISION

To carry out the requirements of the Safety and Technical Strategy Board Cross-Connection Project G/38/98.

ACTION

After the existing paragraph 2.C.(2) and before 2.C.(3) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

After the existing paragraph 3.C.(1) and before 3.C.(2) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

Originator: Garry Kerr
Reference: 757-MCR-22-GK-99-430
Workbook: JS 22-007

MAINTENANCE MANUAL

TEMPORARY REVISION No. 22-522

THIS TEMPORARY REVISION IS ISSUED BY BRITISH AIRWAYS ENGINEERING (TECHNICAL INFORMATION SERVICES, G2, TBA, S401, P. O. BOX 10, HEATHROW AIRPORT, HOUNSLOW, MIDDLESEX TW6 2JA).
CAA DESIGN APPROVAL No. DAI/8566/78.

Manual Reference 22-21-01 Page 401 para 2 and 3

(Tasks 22-21-01-024-001 and 22-21-01-424-007)

REASON FOR REVISION

To carry out the requirements of the Safety and Technical Strategy Board Cross-Connection Project G/38/98.

ACTION

After the existing paragraph 2.D.(2) and before 2.D.(3) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

After the existing paragraph 3.C. and before the existing paragraph 3.C.(1) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

After the existing paragraph 3.E.(3) add the following:

(4) Push the left yaw damper engage switch
  (a) Verify that the left yaw damper ON light extinguishes.
(5) Push the left yaw damper engage switch
  (a) Verify that the left yaw damper ON light illuminates.
(6) Push the right yaw damper engage switch
  (a) Verify that the right yaw damper ON light extinguishes.
(7) Push the right yaw damper engage switch
  (a) Verify that the right yaw damper ON light illuminates.

Originator: Gary Kerr
Reference: 757-W-MCR-22-GK-99-446
Workbook: JS 22-008

MAINTENANCE MANUAL

TEMPORARY REVISION No. 22-520

THIS TEMPORARY REVISION IS ISSUED BY BRITISH AIRWAYS ENGINEERING (TECHNICAL INFORMATION SERVICES, G2, TBA, S401, P. O. BOX 10, HEATHROW AIRPORT, HOUNSLOW, MIDDLESEX TW6 2JA).
CAA DESIGN APPROVAL No. DAI/8566/78.

Manual Reference 22-31-02 Page 401 para 2 and 3

(Tasks 22-31-02-004-002 and 22-31-02-404-005)

REASON FOR REVISION

To carry out the requirements of the Safety and Technical Strategy Board Cross-Connection Project G/38/98.

ACTION

After the existing paragraph 2.B.(3) and before 2.B.(4) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

Before the existing paragraph 3.C.(1) add the following:

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

Originator: Gary Kerr

Reference: 757-W-MCR-22-GK-99-432
Workbook: JS 22-006

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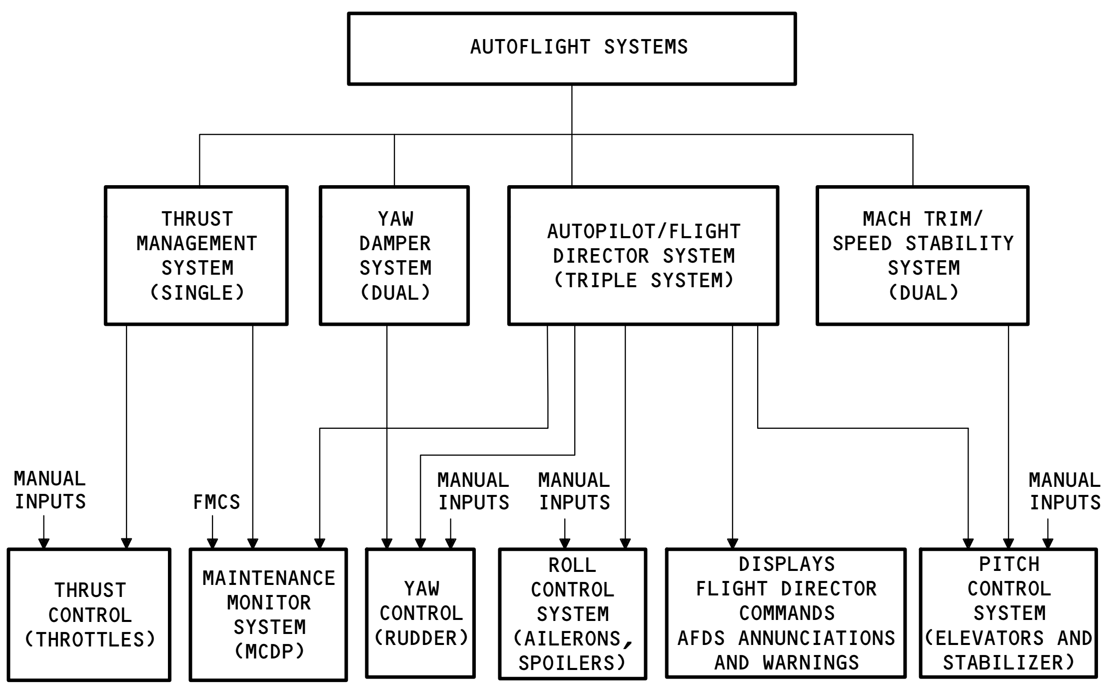
AUTOFLIGHT - GENERAL - DESCRIPTION AND OPERATION

1. General
   A. Autoflight Systems (Fig. 1)
      (1) The Autopilot Flight Director System provides automatic control of the ailerons, elevator, stabilizer, and rudder control systems for flying selected flight modes. It provides pitch and roll flight director commands, system warnings, and mode annunciations. Yaw control is provided during the landing rollout.
      (2) The Yaw Damper System controls the rudder to minimize oscillations due to Dutch roll and gust induced side slip. It also provides turn coordination.
      (3) The Mach Trim/Speed Stability System controls horizontal stabilizer position as a function of airspeed, Mach, and elevator loading. Mach trim is disabled when flaps are extended.
      (4) The Thrust Management System provides automatic thrust control of the engines based on selected modes, current conditions, and engine limits.



Autoflight Systems
Figure 1

(5) The Maintenance Monitoring System provides centralized flight fault recording of three systems. It also allows maintenance personnel to perform programmed tests.

(6) The ILS Deviation Monitor indicates excessive deviations from ILS centerlines during an approach to landing.

B. Autopilot/Flight Director System (AFDS) (Fig. 2)

(1) The Autopilot (Flight Control) (AMM 22-10-00) consists of the following five autopilot/flight director subsystems.

(2) Autopilot/Flight Director Power (AMM 22-11-00)

(a) This section covers the general description of ac and dc power distribution to components common to the autopilot/flight director system. It also covers the component details and operation of the three flight control computers, Autoflight Control System (AFCS) mode control panel, autopilot disengage switches, and AFCS go-around switches.

(3) Autopilot/Flight Director Pitch Channel (AMM 22-12-00)

(a) This section covers the general description of the autopilot/flight director pitch channel. It gives the component details and operation of the Autopilot Pitch Control Servos (APCSs); APCS electrohydraulic servovalve and solenoid valves.

(4) Autopilot/Flight Director Roll and Yaw Channel (AMM 22-13-00)

(a) This section covers the general description of the autopilot/flight director roll and yaw channel. It gives the component details and operation of the Autopilot Rollout Guidance Servos (ARGSs), Autopilot Lateral Control Servos (ALCSs), ARGS and ALCS electrohydraulic servovalve and solenoid valves.

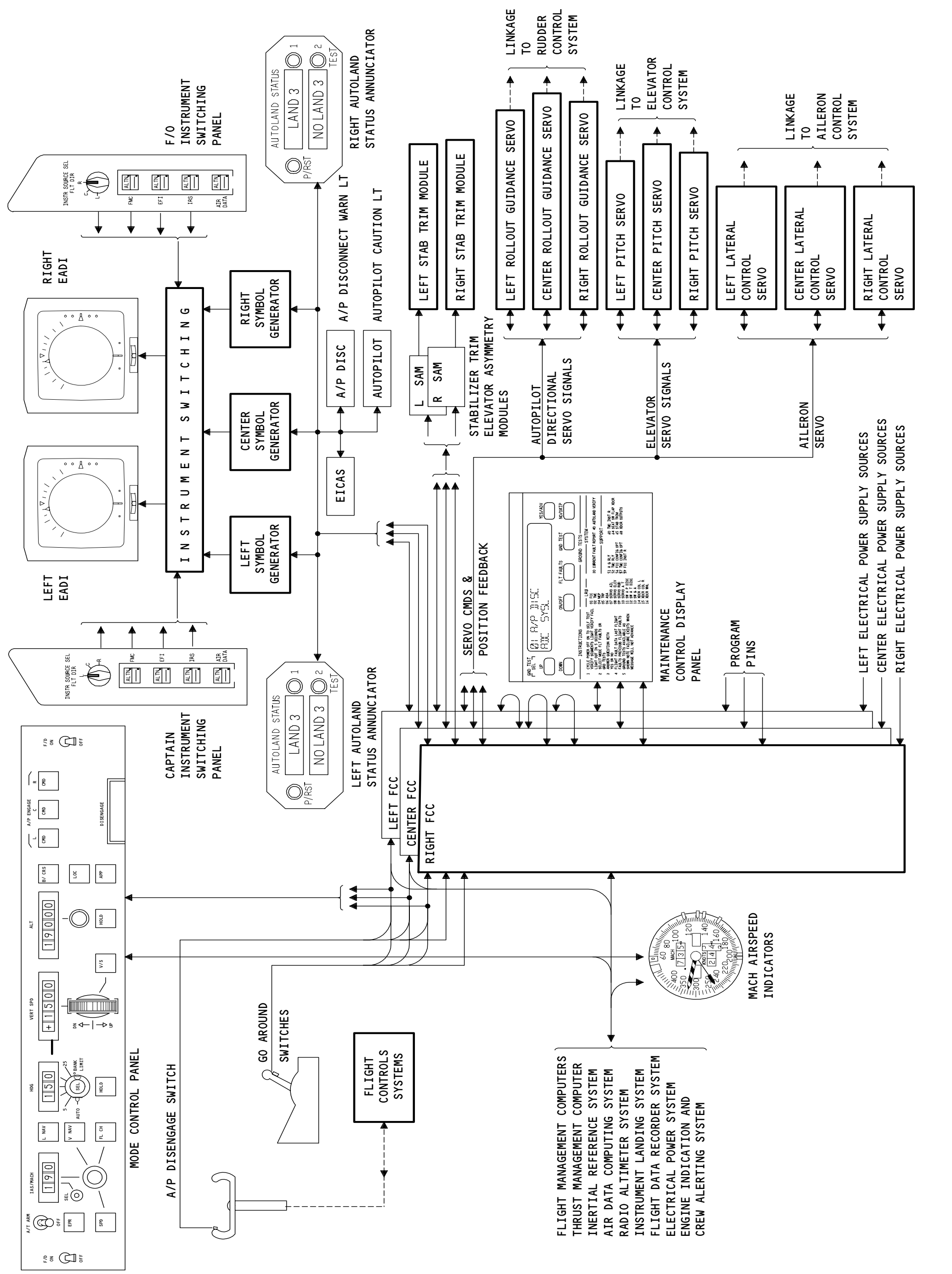
(5) Autopilot/Flight Director Warning and Annunciation (AMM 22-14-00)

(a) This section covers the description of the autopilot/flight director warning and annunciation. It covers the component details and operation of the autoland status annunciators, autopilot caution and disconnect lights, and Engine Indication and Crew Alerting System (EICAS) autopilot displays.

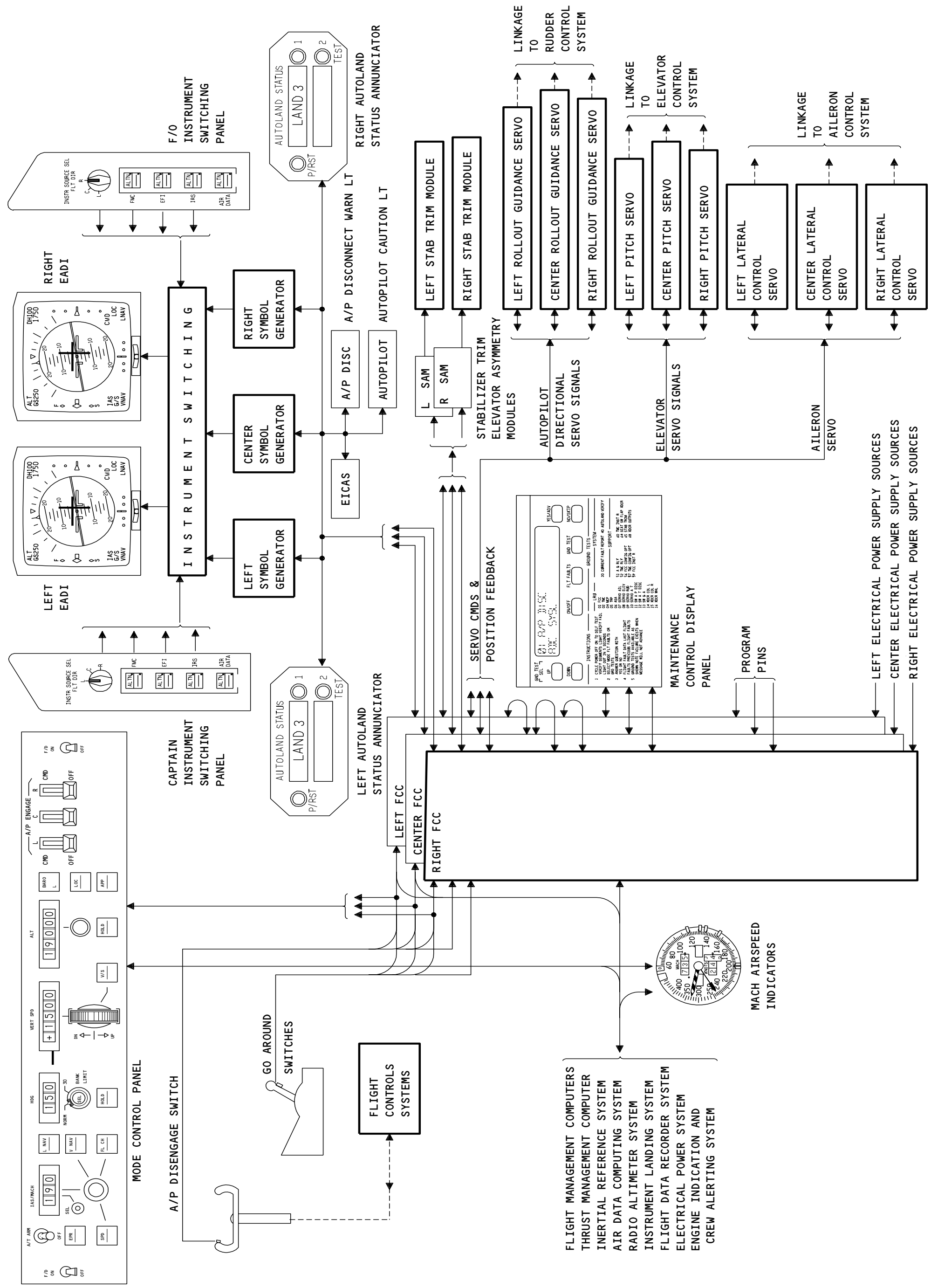
(6) Autopilot/Flight Director Interchannel Data (AMM 22-15-00)

(a) This section covers the general description and operation of cross channel data buses. Digital data is transmitted between the three flight control computers.

C. Yaw Damper System (AMM 22-21-00) (Fig. 3)



Autopilot Flight Director System
Figure 2 (Sheet 1)



Autopilot Flight Director System
Figure 2 (Sheet 2)

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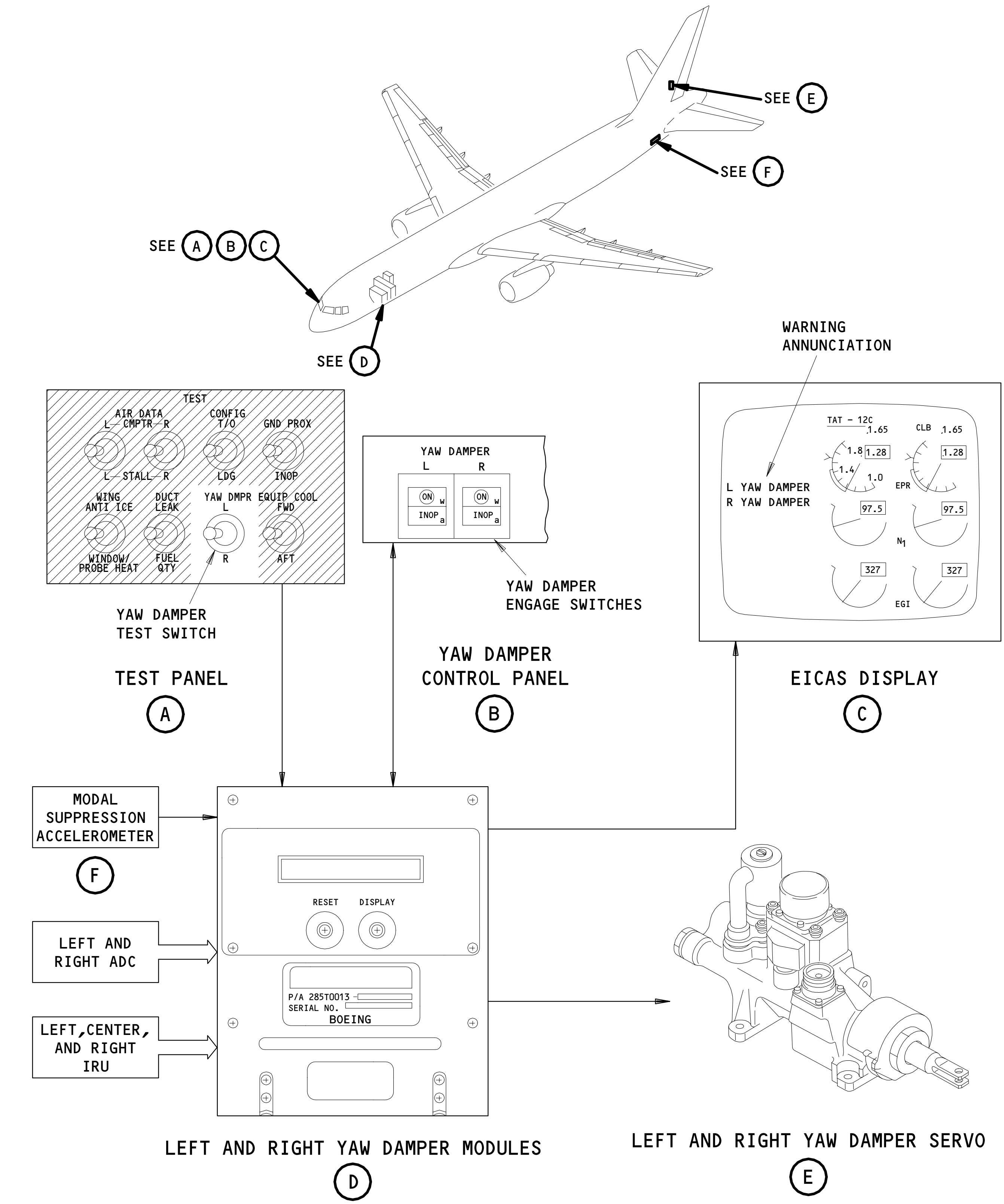


Yaw Damper System
Figure 3

(1) The yaw damper system provides commands to the rudder that dampen undesirable side slip and provide coordination for normal turns. The system consists of a control panel with status indication, two yaw damper servos, and two yaw damper modules. The yaw damper modules receive data from the Air Data Computers (ADCs) and Inertial Reference Units (IRUs). The modules also receive on-off commands from the control panel. The modules send status information to the control panel, and electrical control commands to the two yaw damper servos. The servos translate the electrical commands to mechanical outputs which drive the rudder system.

(2) This section covers the general description of the yaw damper system. It gives the component details and operation of the yaw damper panel, Yaw Damper Modules, Yaw Damper Servos (YDSs), and YDS electrohydraulic servovalve and solenoid valves.

D. Automatic Stabilizer Trim and Mach Trim/Speed Stability (Fig. 4)

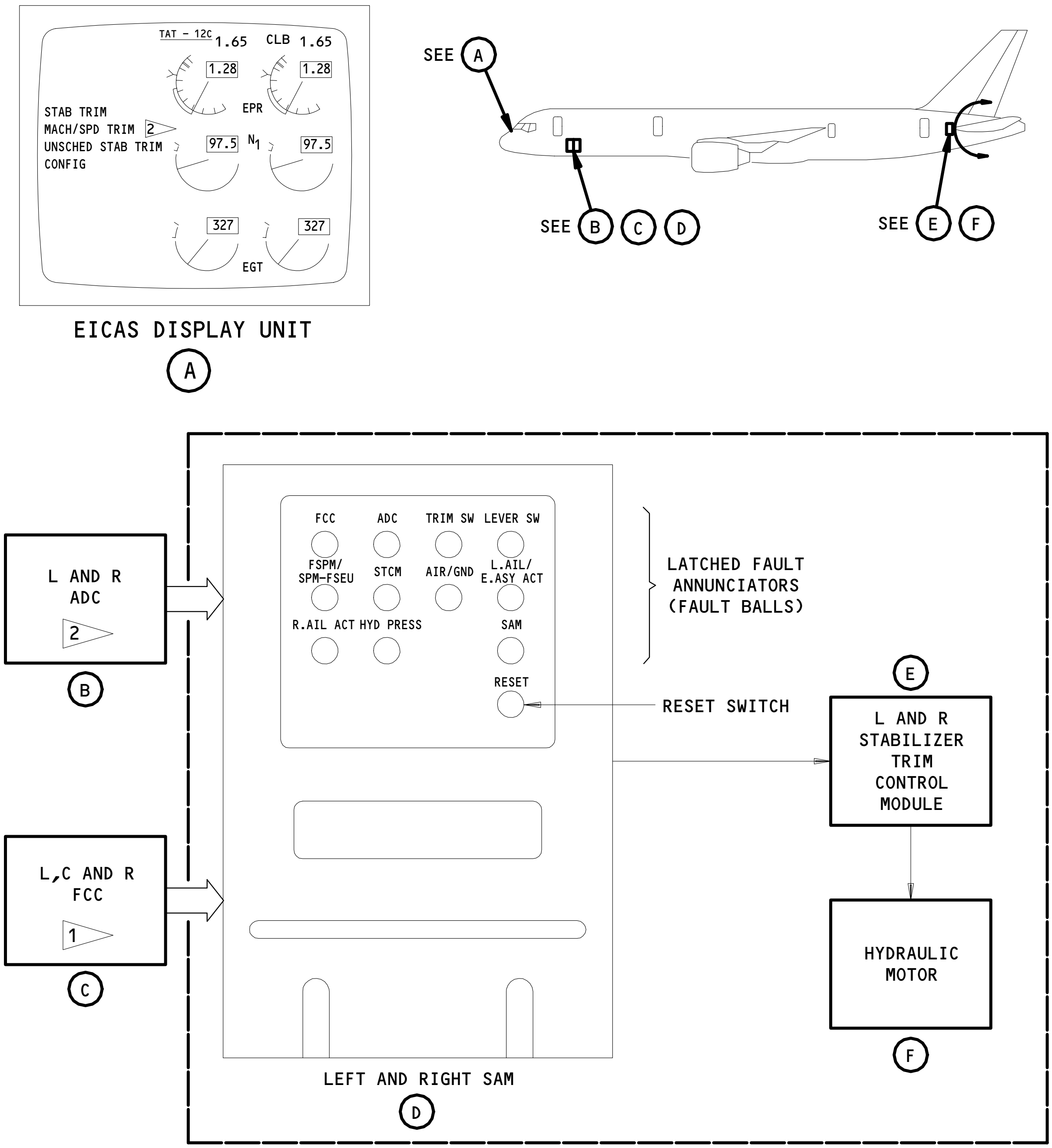
(1) The automatic stabilizer trim system (AMM 22-22-00) uses commands from the FCCs to trim the stabilizer. The stabilizer is trimmed to reduce elevator loads. The automatic trim mode is only active when the autopilot is engaged.

(2) The Mach/speed trim mode (AMM 22-24-00) receives its commands from the Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) and is active whenever the autopilot or manual trim is not engaged. The Mach/speed mode trims the stabilizer opposite to any Mach or speed changes.

(3) These sections cover the general description and operation of the automatic stabilizer trim system (AMM 22-22-00) and the Mach trim/speed stability system (AMM 22-24-00).

E. Thrust Management System (Fig. 5)

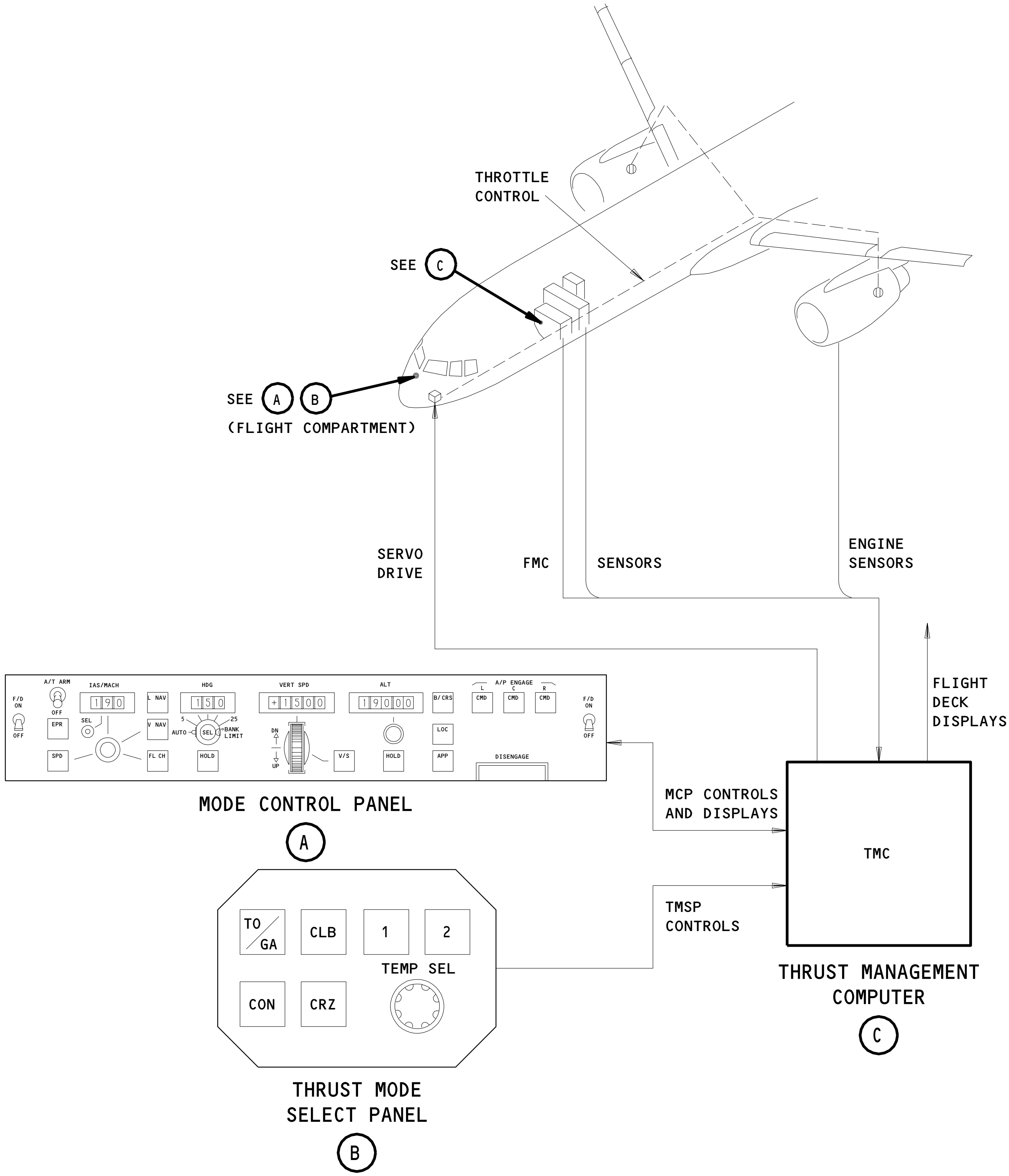
(1) The Thrust Management System (TMS) controls the engine thrust settings during takeoff, cruise and landing (full flight regime). The TMS functions include thrust limit computation, autothrottle control, and information display. Engine thrust limits are computed by the Thrust Management Computer (TMC) based on the selected mode on the Thrust Mode Select Panel (TMSP). The TMS can be operated independent of, or coordinated with the AFDS and FMCS.



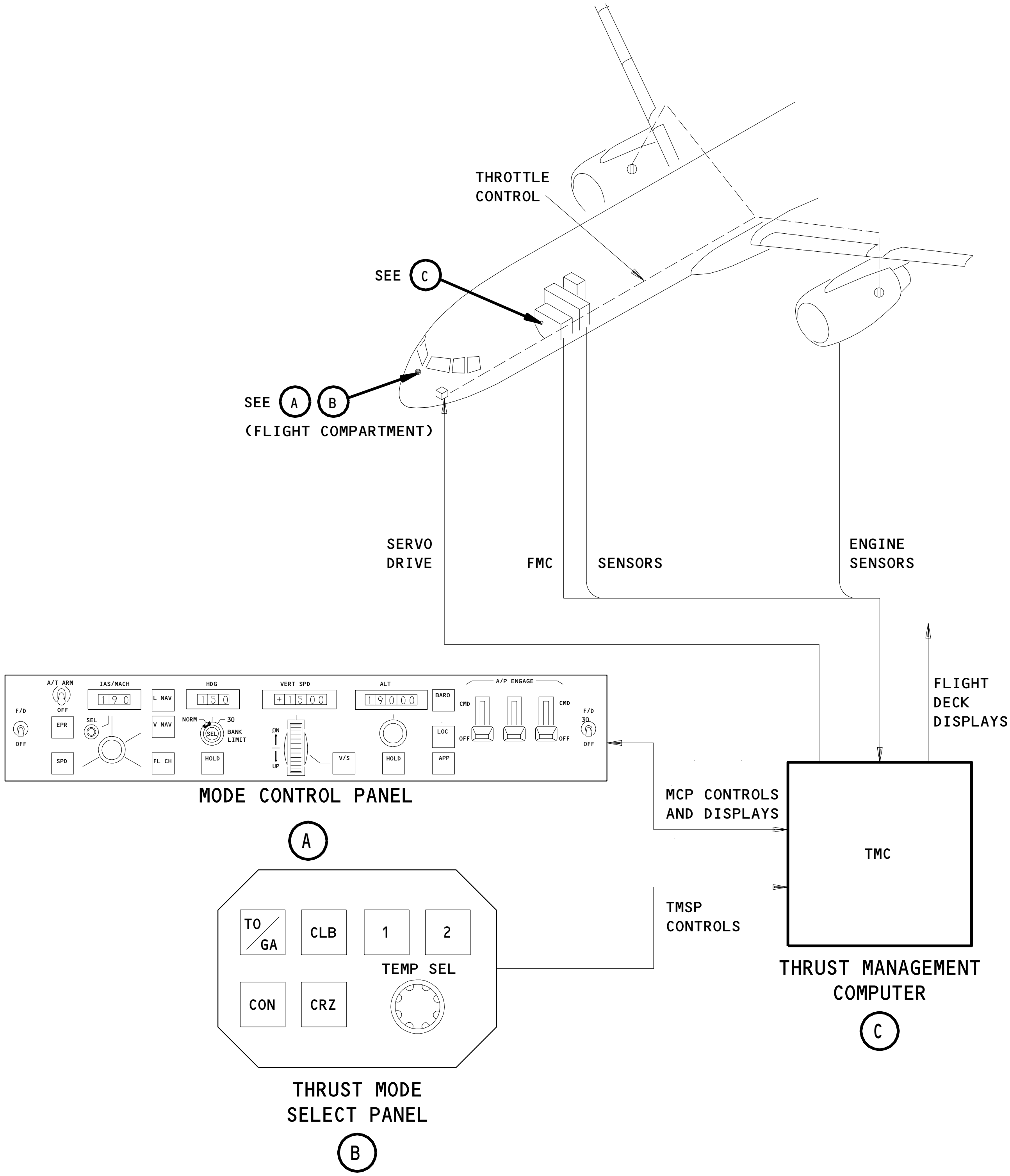
1 AUTOMATIC STABILIZER TRIM SYSTEM ONLY

2 MACH SPEED STABILITY SYSTEM ONLY

Automatic Stabilizer Trim and Mach Trim/Speed Stability
Figure 4



Thrust Management System
Figure 5 (Sheet 1)



Thrust Management System
Figure 5 (Sheet 2)

(2) The thrust management system consists of the following four subsystems:

(3) Thrust Management Power (Ref 22-31-00)

(a) This section covers the general description of the ac and dc power distribution and interface components common to the TMS. It covers the component detail and operation of the TMC and TMSP.

(4) Thrust Management System (Ref 22-32-00)

(a) This section covers the general description of the TMS. It covers the component details and operation of the autothrottle servomotor generator, disengage switches, leading edge slat switch, microswitch pack with thrust reverse switches, and clutch pack assembly.

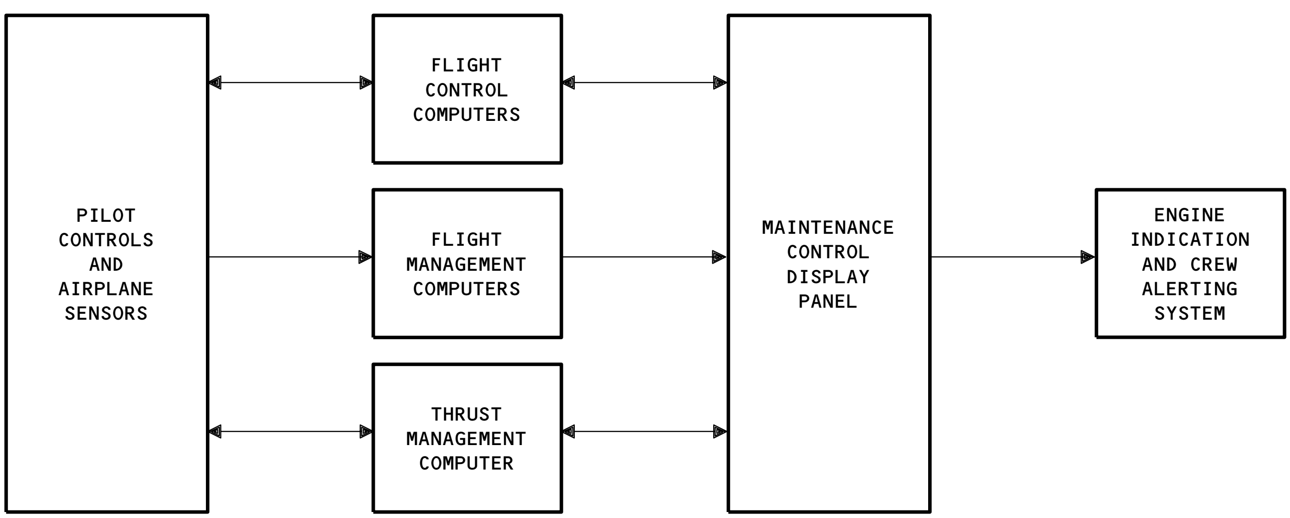
(5) Thrust Management Engine (Ref 22-33-00)

(a) This section covers the component detail and operation of the power lever angle transducers on the engines.

(6) Thrust Management Warning and Annunciation (AMM 22-34-00)

(a) This section covers the general description and operation of the thrust management system warning and annunciation.

F. Maintenance Monitor (AMM 22-41-00) (Fig. 6)



Maintenance Monitor System
Figure 6

(1) The maintenance monitor system consists of a Maintenance Control Display Panel (MCDP). It provides a centralized record of flight faults from the flight control computers, thrust management computer, flight management computers, and their related sensors. It allows maintenance crews to perform ground tests on the AFDS, TMS, and their related sensors without additional ground test equipment.

(2) This section covers the general description, component details, and operation of the MCDP.

G. ILS Deviation Monitor (AMM 22-42-00).

(1) The ILS Deviation Monitor indicates excessive deviations from ILS centerlines (glideslope or localizer) during an approach to landing. This section covers the general description and operation of the ILS Deviation Monitor.

2. Abbreviations

A. The following list contains abbreviations commonly used in this chapter.

ACARS	ARINC Communications Addressing and Reporting System
A/D	Analog to Digital
ADC	Air Data Computer
ADCS	Air Data Computing System
ADF	Automatic Directing Finder
ADIRS	Air Data Inertial Reference System
ADIRU	Air Data Inertial Reference Unit
AFCS	Autoflight Control System
AFDS	Autopilot/Flight Director System
AIDS	Airborne Integrated Data System
AIL	Aileron
ALCS	Autopilot Lateral Control Servo
ALT	Altitude or Altimeter
AOA	Angle of Attack
A/P	Autopilot
APCS	Autopilot Pitch Control Servo
APP	Approach
APU	Auxiliary Power Unit
ARINC	Aeronautical Radio Inc.
ARR	Arrival
ARGS	Autopilot Rollout Guidance Servo
ASA	Autoland Status Annunciator
A/T	Autothrottle

ATC	Air Traffic Control
B/A	Bank Angle
BAT	Battery
BCD	Binary Coded Decimal
B/CRS	Back Course
BIT	Built-In-Test
BITE	Built-In-Test-Equipment
BNR	Binary Numerical Representation
CAA	Civil Aviation Agency
CAS	Computed Airspeed
CDU	Control Display Unit
CFT	Control Force Transducer
CLB	Climb
CMD	Command
CON	Continuous
CPU	Central Processing Unit
CRS	Course
CRT	Cathode Ray Tube
CRZ	Cruise
CSEU	Control System Electronics Unit
CWS	Control Wheel Steering
D/A	Digital to Analog
DADC	Digital Air Data Computer
DDM	Difference in Depth of Modulation
DEP	Departure
DES	Descent
DFDAU	Digital Flight Data Acquisition Unit
DH	Decision Height
DITS	Digital Information Transfer System
DME	Distance Measuring Equipment
D-TO	Derated Takeoff
EADI	Electronic Attitude Director Indicator
EAROM	Electrically Alterable ROM
ECS	Environmental Control System
EDHP	Engine Driven Hydraulic Pump
E/E	Electrical/Electronics
EFCU	Elevator Feel and Centering Unit
EFIS	Electronic Flight Instrument System
EGT	Exhaust Gas Temperature

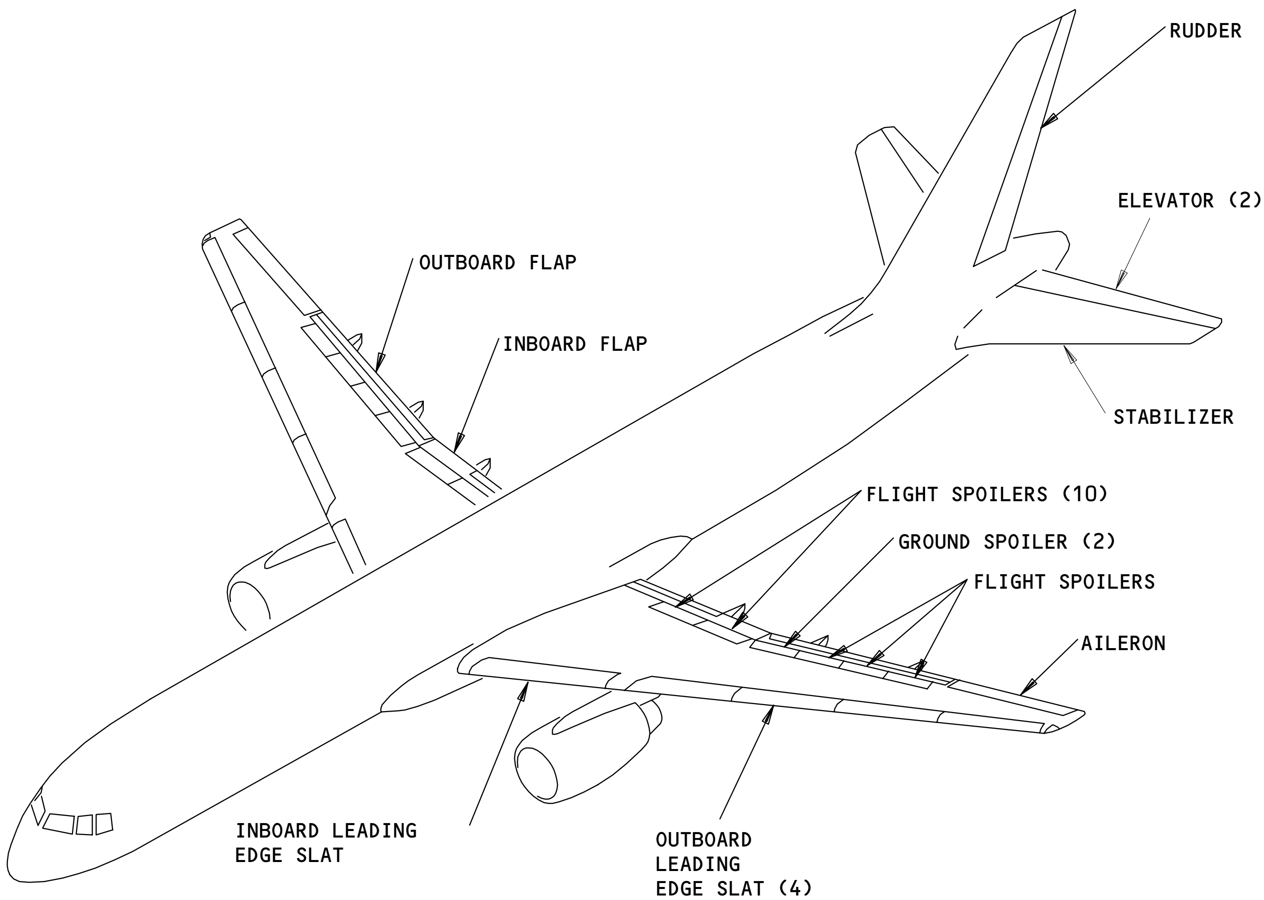
EHSI	Electronic Horizontal Situation Indicator
EHSV	Electrohydraulic Servovalve
EICAS	Engine Indication and Crew Alerting System
EMHP	Electric Motor Hydraulic Pump
EMI	Electromagnetic Interference
ENG or ENGA	Engage
EPR	Engine Pressure Ratio
FAA	Federal Aviation Administration
FCC	Flight Control Computer
F/D	Flight Director
FLCH	Flight Level Change
FMA	Flight Mode Annunicator
FMC	Flight Management Computer
FMCS	Flight Management Computer System
FMS	Flight Management System
F/O	First Officer
FPM	Feet Per Minute
FSAM	Flap/Slat Accessory Module
FSEU	Flap/Slat Electronics Unit
GA or G/A	Go Around
G/S	Glideslope
G.S.	Gain Schedule
HDG	Heading
HLD	Hold
HR	Height, Radio
HYD	Hydraulic
IAS	Indicated Airspeed
ILS	Instrument Landing System
INFC	Interface
INTC	Intercept
I/O	Input/Output
IRU	Inertial Reference Unit
IRS	Inertial Reference System
LNAV	Lateral Navigation
LOC	Localizer
LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transducer
M	Mach
M/ASI	Mach/Airspeed Indicator
MB	Millibars

MCDP	Maintenance Control Display Panel
MCHENG	Multi-Channel Engage
MCP	Mode Control Panel
MCU	Modular Concept Unit
MMO	Maximum Operating Mach
MMR	Multi-Mode Receiver
MSL	Mean Sea Level
NAV	Navigation
NCD	No Computed Data
PCA	Power Control Actuator
PCU	Power Control Unit
PDU	Power Drive Unit
PLA	Power Lever Angle
PROG	Program
PROM	Programmable Read Only Memory
PTU	Power Transfer Unit
RA	Radio Altitude or Radio Altimeter
RAT	Ram Air Turbine
RCVR	Receiver
RDMI	Radio Distance Magnetic Indicator
REF	Reference
RF	Radio Frequency
RMI	Radio Magnetic Indicator
RNAV	Area Navigation
ROM	Read Only Memory
RTE	Route
RTG	Rating
RVDT	Rotary Variable Differential Transducer
RVR	Runway Visual Range
SAM	Stabilizer Trim and Elevator Asymmetry Limit Module
SAT	Static Air Temperature
SCM	Spoiler Control Module
SG	Symbol Generator
SID	Standard Instrument Departure
SOV	Shut-Off Valve
SPD	Speed
SPM	Stabilizer Position Module
SSM	Sign Status Matrix

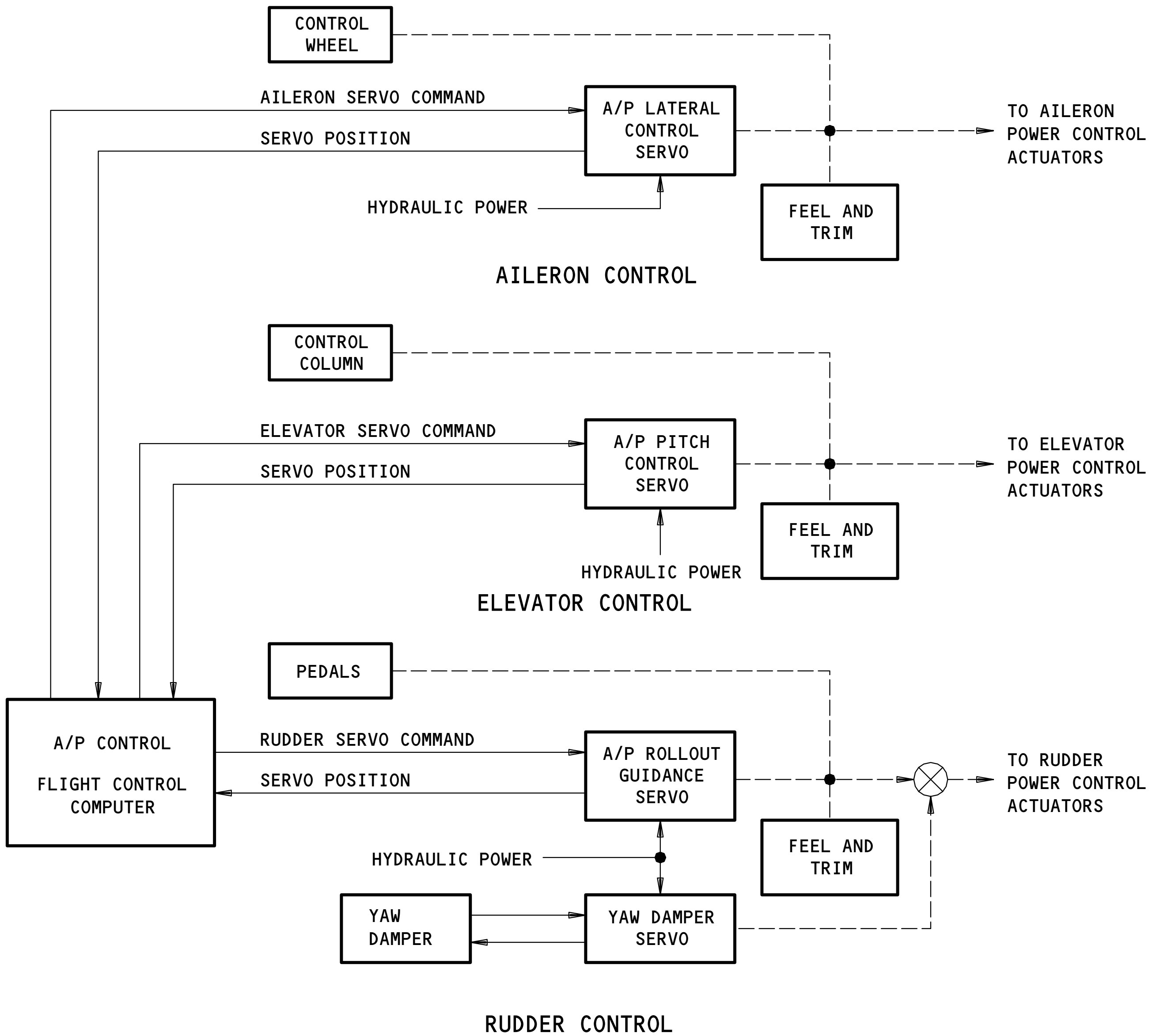
STAB	Stabilizer
STAR	Standard Terminal Arrival Route
STCM	Stabilizer Trim Control Module
STLM	Stabilizer Trim Limit Switch and Position Transmitter Module
TAS	True Airspeed
TAT	Total Air Temperature
TCA	Terminal Control Area
TGT	Target
TLA	Throttle Lever Angle
TMC	Thrust Management Computer
TMS	Thrust Management System
TMSP	Thrust Mode Select Panel
TO or T/O	Takeoff
TOC	Top of Climb
TOD	Top of Decent
TRK	Track
TRU	Transformer/Rectifier Unit
VFR	Visual Flight Rules
VHF	Very High Frequency
VMO (Vmo)	Maximum Operating Airspeed (Knots)
VNAV	Vertical Navigation
VOR	VHF Omnidirectional Range
V/S	Vertical Speed
WEU	Warning Electronics Unit
XCVR	Transceiver
XDCR	Transducer
XFMR	Transformer
XMTR	Transmitter
YDM	Yaw Damper Module
YDS	Yaw Damper Servo
YSM	Yaw Damper/Stabilizer Trim Module

HYDRAULIC AND FLIGHT CONTROLS (AUTOFLIGHT) - DESCRIPTION AND OPERATION

1. General
   A. Autoflight systems drive hydraulically powered servo actuators which in turn drive hydraulic Power Control Actuators (PCAs) connected to the ailerons, elevators, and rudder. The PCAs are driven either manually or from autoflight system servos. Secondary flight controls such as spoilers, flaps, slats, and stabilizer are also discussed. The following pages provide a general discussion of airplane Hydraulic Power (Chapter 29) and Flight Controls (Chapter 27).
   B. Flight Controls - Introduction (Fig. 1)
      (1) The primary flight controls consist of five movable surfaces for the three control axis:
         (a) Roll control: Ailerons, one surface each wing (2)
         (b) Pitch control: Elevators, one surface each side (2)
         (c) Directional control (yaw): Rudder, one surface
      (2) Secondary flight controls consist of 28 surfaces for four systems.
         (a) Spoilers: provide lift and drag, augment roll control, six surfaces each wing (12)
         (b) Leading edge slats: provide high lift, five surfaces each wing (10)
         (c) Trailing edge flaps: provide high lift, two surfaces each wing (4)
         (d) Stabilizer: augments pitch control, one surface each side (2)
      (3) Actuation of all control surfaces is by hydraulic servos or actuators. Primary flight control actuators are controlled mechanically by both the pilots and the electrohydraulic autoflight servos. Spoilers are controlled only with electrohydraulic actuators. Flaps and slats are driven with hydraulic motors; they also are driven by a backup system using electric motors.
   C. Flight Control Inputs (Fig. 2)
      (1) All primary flight control surfaces are physically moved by hydraulic power control actuators (PCAs). Mechanical and hydraulic devices are used to provide normal control system feel. The PCAs are controlled by movement of mechanical linkage. The mechanical linkage is moved as a result of:
         (a) Conventional manual flight controls consisting of control columns, cables and quadrants, etc.
         (b) Servos responding to autopilot command signals.
      (2) Two or more actuators drive each primary flight control surface. Each PCA is powered by only one hydraulic system. This provides dual or triple path control for each primary control surface ensuring control of the airplane in the event of a hydraulic system or actuator failure. The servos and actuators of controlling systems also use hydraulic power.



Flight Controls - Introduction
Figure 1



Flight Controls Inputs
Figure 2

(3) Operation of the autopilot servos is similar except that when electrically engaged the input command is electrical rather than mechanical. The servos electrical feedback (position) neutralizes the control input (command) from the system computer when the new position is reached.

D. Power Control Actuator Reaction (Fig. 3)

(1) The mechanical inputs to the PCAs provide the commands for movement of the control surfaces. The inputs are neutralized by the reaction of the PCA. For the aileron and elevator systems, the action of the summing linkage maintains the PCA control lever in the neutral position. The linkage sums the motion of the control input and the output piston, which move in opposite directions. The resultant establishes the position of the control lever. For the rudder system, the PCA is basically floating except it is referenced to structure to prevent lateral motion. Longitudinal motion occurs when the rudder is moved in response to a command. That motion, caused by movement of the reaction link, neutralizes the position of the control lever.

(2) The PCAs continue to move in the commanded direction as long as the control lever is displaced from neutral. When more than one PCA is used on a single surface, movement is synchronized by adjustment of the control linkage. All control surfaces are moved by hydraulic power input to the PCAs. There is no reversion to manual movement.

E. Hydraulic Power Systems - Introduction (Fig. 4)

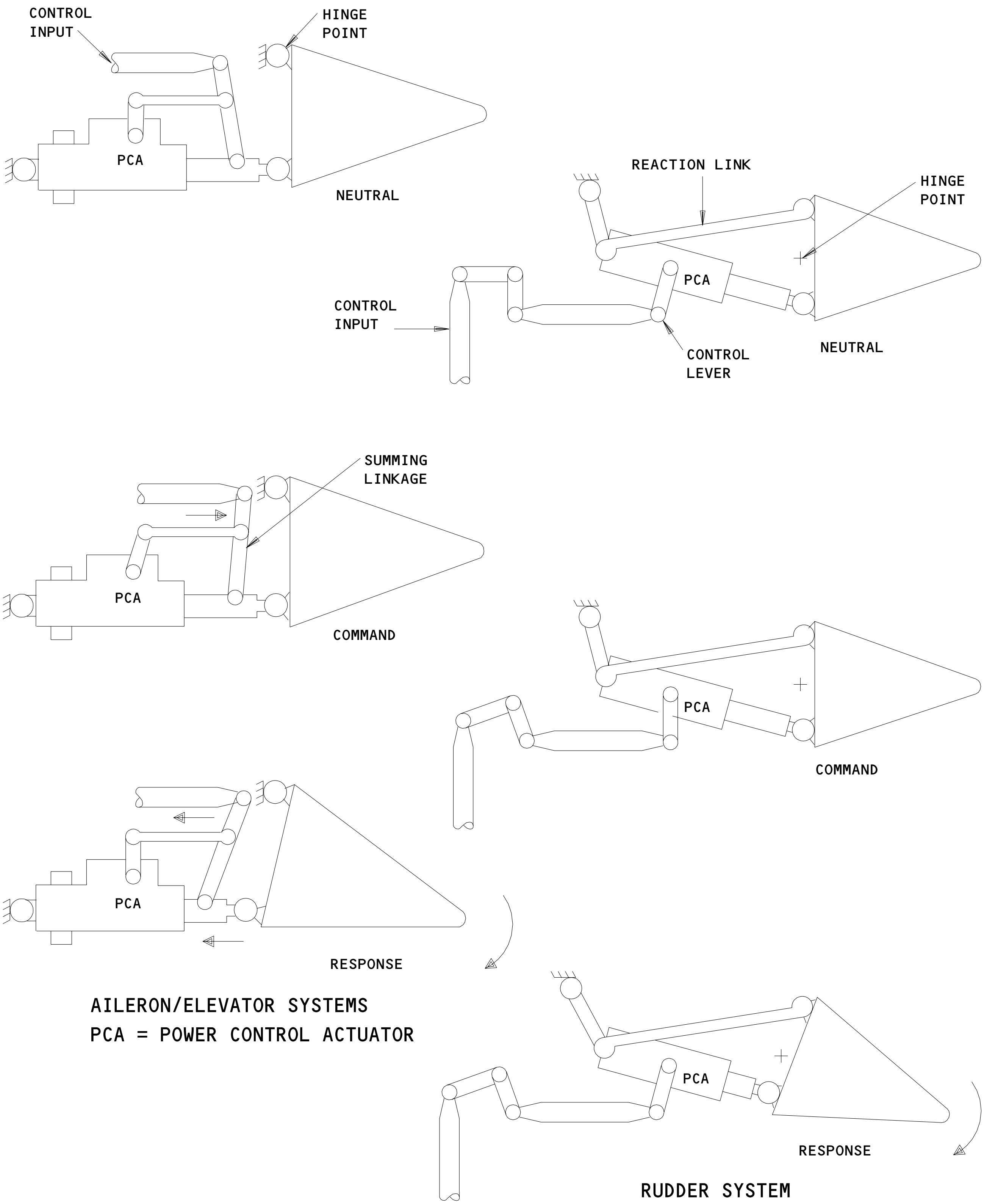
(1) Three independent hydraulic systems are installed. System tubing is color coded for easy identification. The left is red, the center is blue, and the right is green. Each hydraulic system provides power for all essential flight control systems.

(2) Three types of hydraulic pumps are used: Engine Driven Pumps (EDPs), Alternating Current Motor Pumps (ACMPs), and Ram Air Turbine (RAT). The EDPs are in the left and right systems. The ACMPs are in all three systems. The RAT is in the center system.

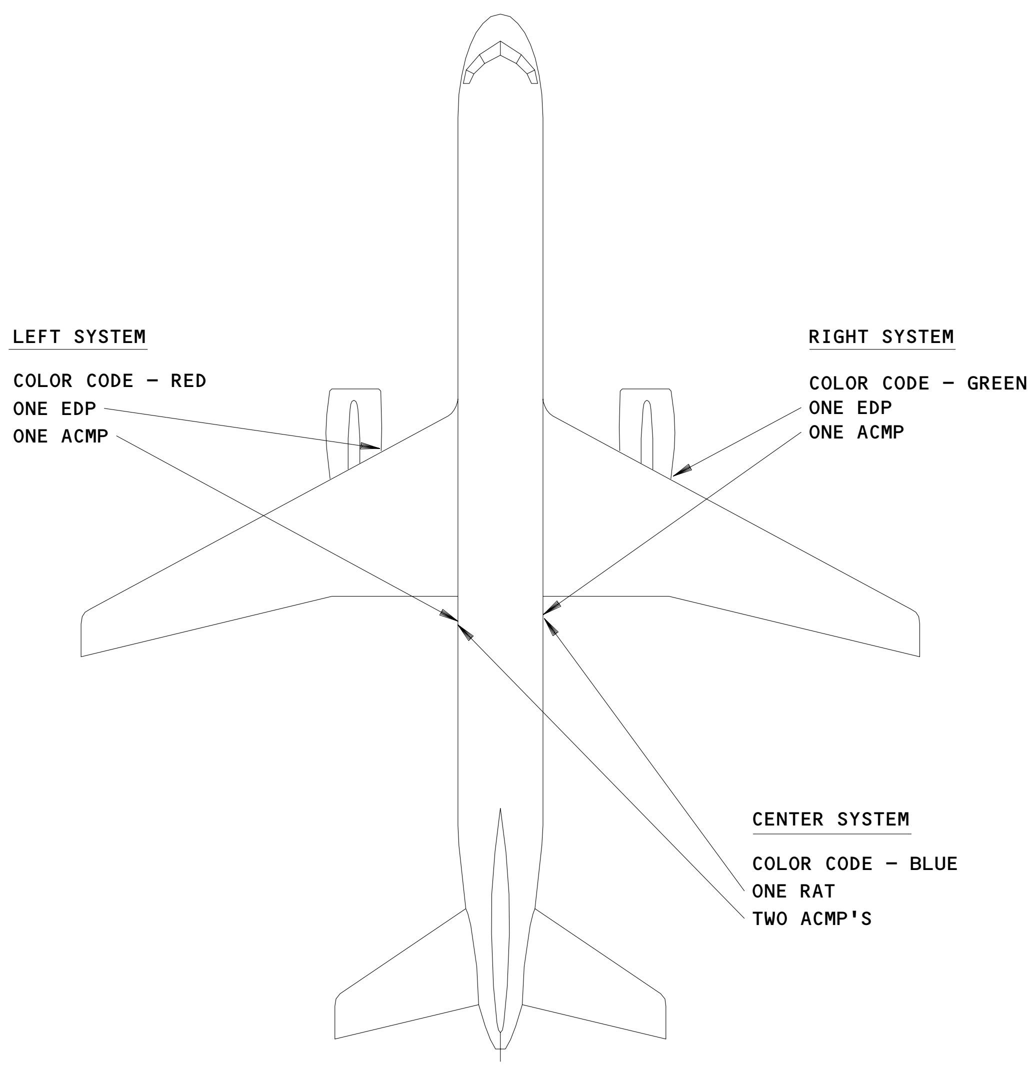
(3) Other equipment and components for each system includes independent pressurized reservoirs, filters for each pump, and associated valves, gages, and sensors. Rated pressure of each system is 3000 psi. Fluid used is fire resistant BMS 3-11 Type IV, which is purple in color.

(4) Components of the left system are in the hydraulic bay of the left engine strut. Components of the center system are in the right wheel well. Components of the right system are in the hydraulic bay of the right engine strut.

(5) The system also uses a Power Transfer Unit (PTU) to drive the landing gear, flaps, and slats, which are normally driven by the left system. The PTU consists of a hydraulic motor, powered by the right system, which drives a hydraulic pump connected to the left system.



Power Control Actuator Reaction
Figure 3



LEGEND:

ENGINE DRIVEN HYDRAULIC PUMP - EDP
ALTERNATING CURRENT MOTOR PUMP - ACMP
RAM AIR TURBINE - RAT

Hydraulic Power Systems - Introduction
Figure 4

F. Hydraulic System - Simplified (Fig. 5)

(1) The left, center, and right hydraulic systems provide power for primary flight control systems: ailerons, rudder, elevators, flight spoilers, and autopilot. The right and center hydraulic systems provide power for the stabilizer trim system, and elevator feel (computer and actuator). The left and right hydraulic systems provide power for the thrust reversers, brakes, high lift devices, landing gear actuation, and nose gear steering. The right hydraulic system provides power for the ground spoilers.

(a) Each system uses primary and demand pumps. Primary pumps run continuously. Demand pumps run when required.

(b) The left and right systems use EDPs as the primary pumps. The center system uses two ACMPs as primary pumps.

(c) The left and right systems use ACMPs as demand pumps which start when system pressure is low.

(d) The center system has a RAT for emergency use if both engines fail in flight. The RAT provides hydraulic power for stabilizer trim, autopilot, ailerons, rudder, elevators, elevator feel, and spoilers.

(e) All hydraulic systems have system pressure and temperature sensors. Each pump's pressure is monitored by an independent sensor. Check valves on each pumps output and return lines allow each system to maintain pressure when some of the pumps in the system are not operating. The left and right EDPs have fluid supply shut-off valves.

(f) Reservoirs can be pressure filled or manually filled. A hand pump and reservoir fill valve (selector) allow filling of the selected reservoir.

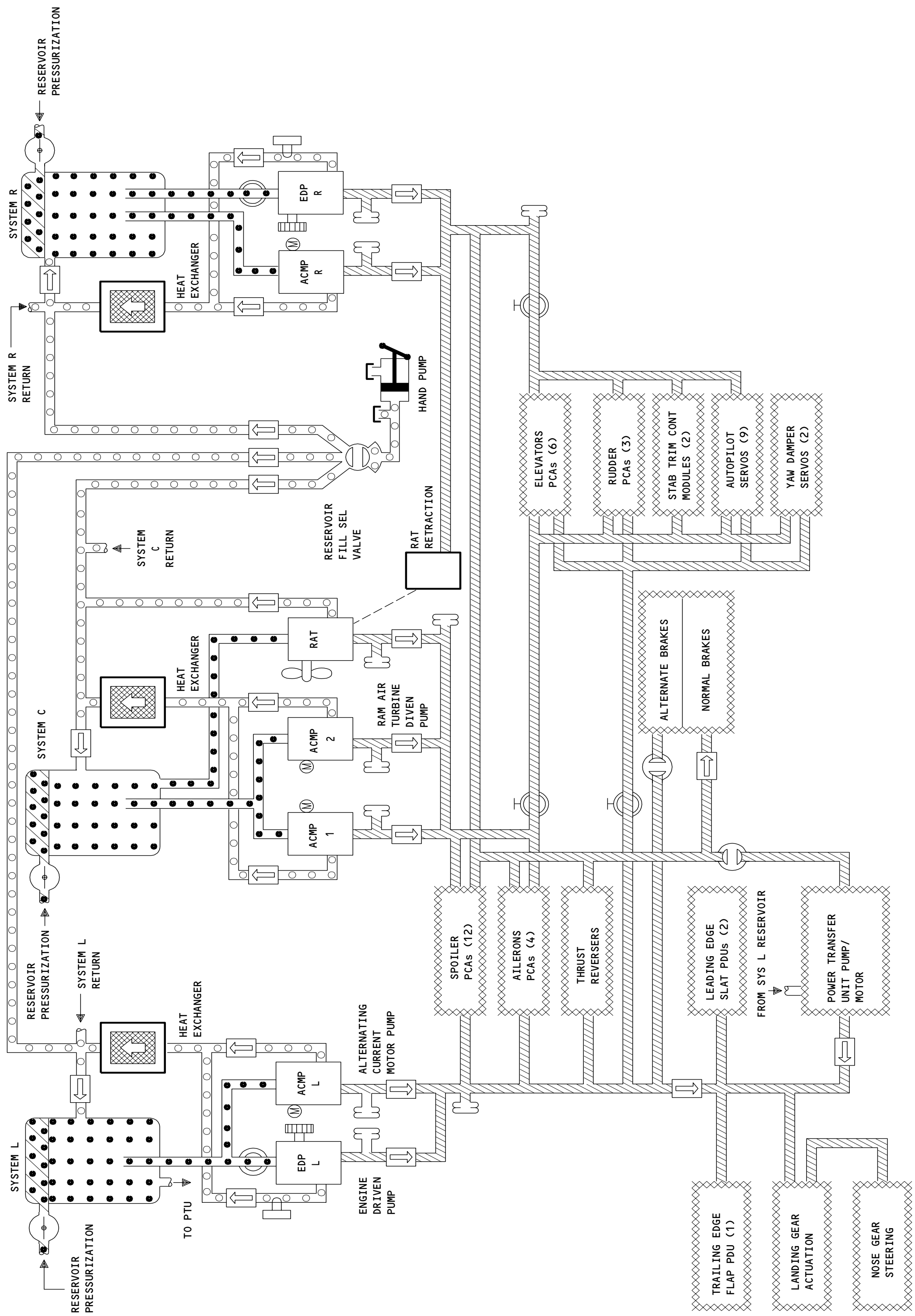
G. Left and Right Hydraulic System Components (Fig. 6)

(1) The left and right hydraulic systems each have one engine driven pump (EDP) and one alternating current motor pump (ACMP). The EDP is attached to the engine gearbox on the under side of the engine. The pump is a line replaceable unit (LRU) with a rated output of 37 gpm.

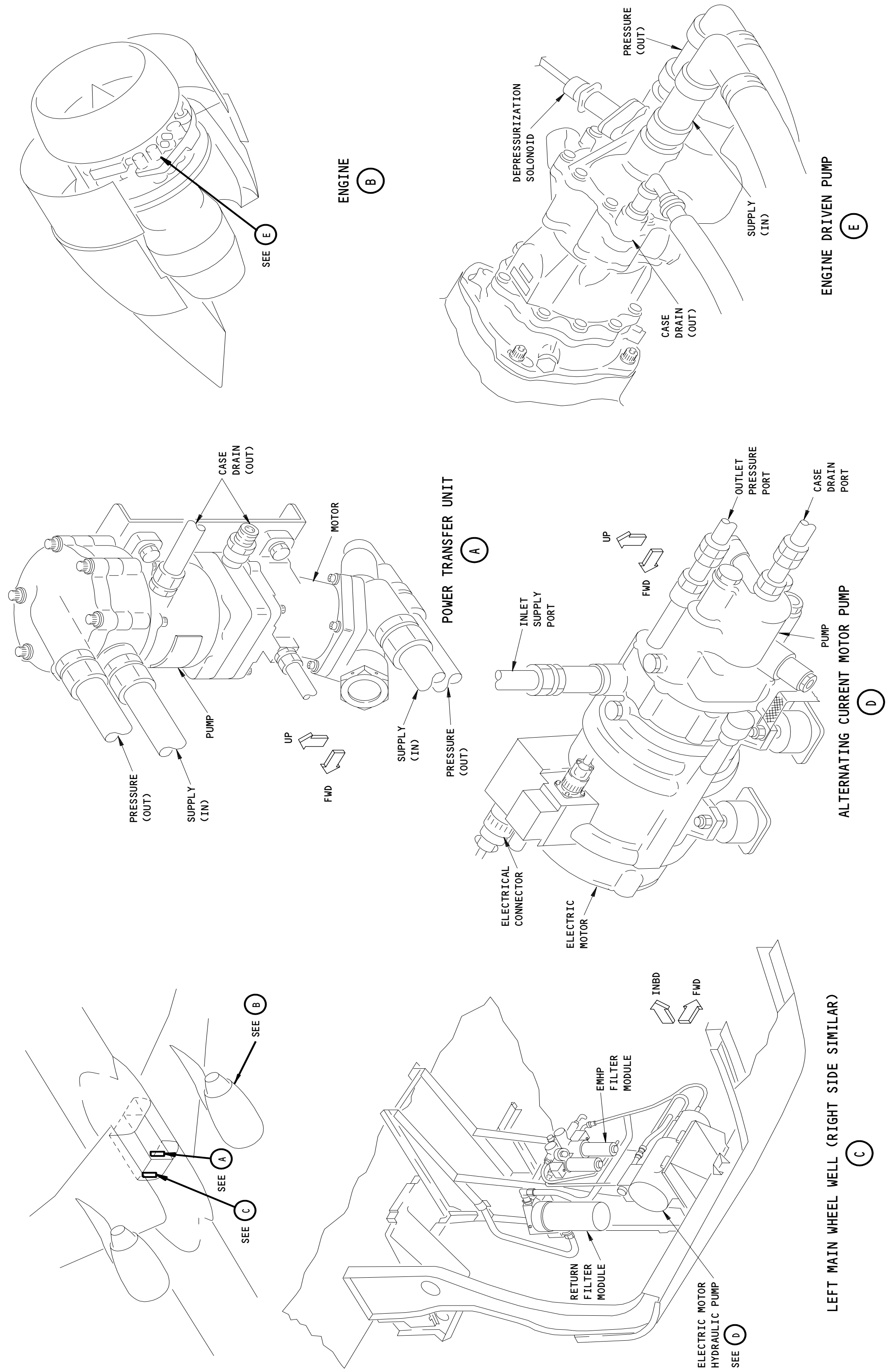
(2) The ACMPs are in the left and right mainwheel forward fairings. The associated filters, valves, and reservoir are also located in that area. The electric motor is powered by 115 vac using 45 amps at rated output. The pump is a LRU with a rated hydraulic output of 7 gpm.

H. Center Hydraulic Systems Components (Fig. 7)

(1) The two ACMP primary sources are on the left side of the airplane in the fairing aft of the main wheel well. They are mounted on shelves on the keel structure. The associated filters, valves, and reservoir are also in that area. The center ACMPs are identical to the left and right ACMPs.



Hydraulic System - Simplified
Figure 5



Left and Right Hydraulic System Components
Figure 6

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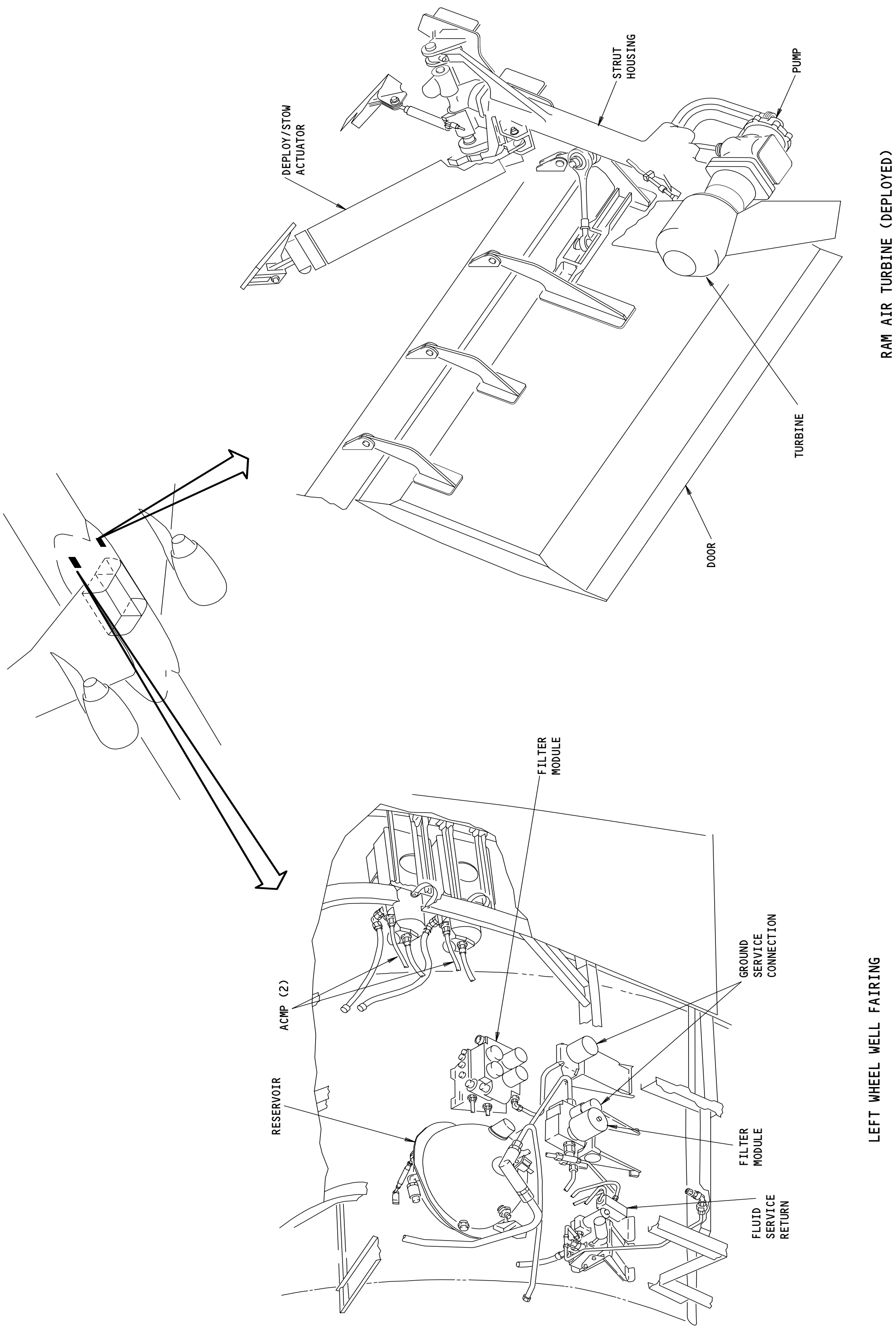


Center Hydraulic System Components
Figure 7

28064

(2) The Ram Air Turbine (RAT) is in the fairing aft of the right wheel well, from which it deploys downward into the slipstream. The RAT is extended and retracted by an electric motor. The swing arm, turbine, and pump are all one LRU.

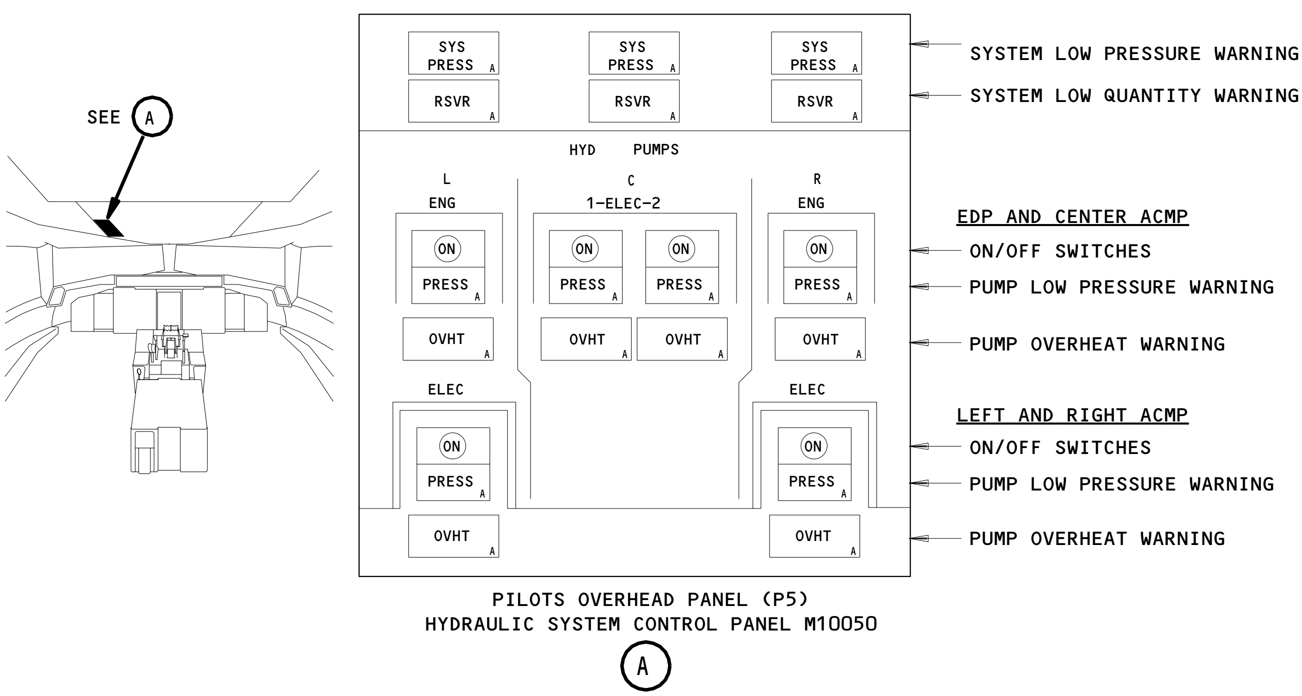
(3) The hydraulic service panel is aft of the right wheel well. It provides a central capability to service all three system reservoirs.

I. Hydraulic System Control Panel (Fig. 8)

(1) The hydraulic system control panel is on the pilot's overhead panel P5. The panel incorporates the switches and indicators required for normal control and monitoring of the three hydraulic systems.

(2) Each system has the following switches and indications:

(a) System pressure, EICAS indicator (AMM 31-41-00)
(b) System low pressure warning light (amber)
(c) Primary pump overheat warning light (1 left, 2 center, 1 right) (amber)
(d) Primary pump ON/OFF switch (1 left, 2 center, 1 right)
(e) Primary pump low pressure warning light (1 left, 2 center, 1 right) (amber)
(f) Demand pump low pressure warning light (amber)



Hydraulic System Control Panel
Figure 8

(g) Demand pump control switch
(h) System quantity low warning light (amber)
(i) System reservoir quantity, EICAS indicator (AMM 31-41-00)
(j) System reservoir temperature, EICAS indicator (AMM 31-41-00)

(3) The primary pump switch/lights are push-on/push-off type. When pushed on, the ON legend is illuminated white. The left and right switches control EDPs. The center switches control ACMPs.

(4) The demand pump switches are also push-on/push-off switches. The left and right switches control ACMPs.

J. Hydraulic System Controls (Fig. 9)

(1) Hydraulic Flight Control Panel

(a) The hydraulic flight control panel on P61 contains three push-on/push-off switch/lights. Each switch controls a shutoff valve that selectively interrupts hydraulic power to the flight controls. The three switches isolate hydraulic power to the left, center, and right hydraulic systems.

(b) The OFF position of the switch/light illuminates amber when pressed. Isolating hydraulic pressure to the selected areas aids in trouble shooting and allows maintenance without removing all hydraulic power.

(2) Engine Ignition and Start Control Panel

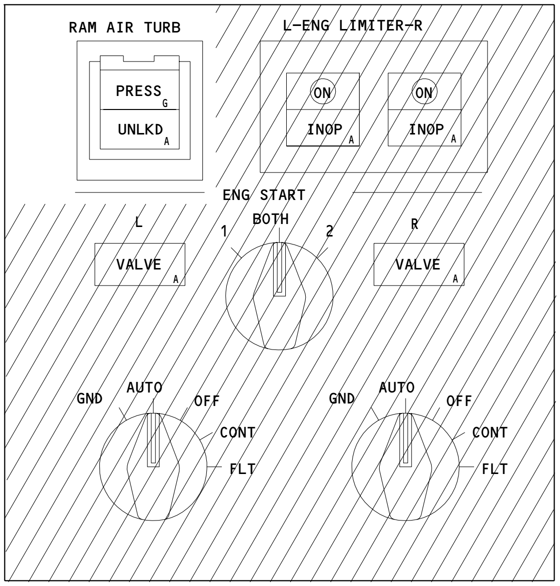
(a) The engine ignition and start control panel incorporates the ram air turbine (RAT) switch/light. Pressing the switch starts an electric motor that extends the RAT. The lower part of the switch/light marked UNLKD (unlocked) illuminates amber. The upper part marked PRESS illuminates green when the RAT is generating at least 1000 psi.

(b) Retraction of the RAT is a ground operation. The turbine blades must be centered before the RAT can be retracted. Retraction is done electrically using a control panel in the right wing aft fairing.

(3) Stabilizer Trim Hydraulic Power Cutout Switches

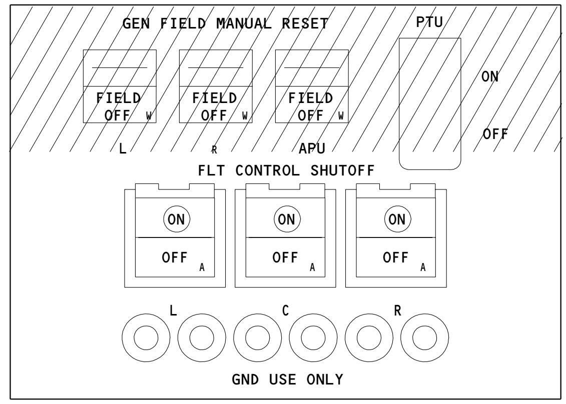
(a) Two stabilizer trim cutout switches (center and right) interrupt hydraulic power to stabilizer trim components in the tail. The switches are guarded to the NORM position. Positioning them to CUTOUT closes the cutout valves, which are part of the stabilizer trim control module. This enables trouble shooting and performance of ground maintenance on the stabilizer trim drive hydraulic motors without removing hydraulic system power.

K. Flight Control - Hydraulic Distribution (Fig. 10)



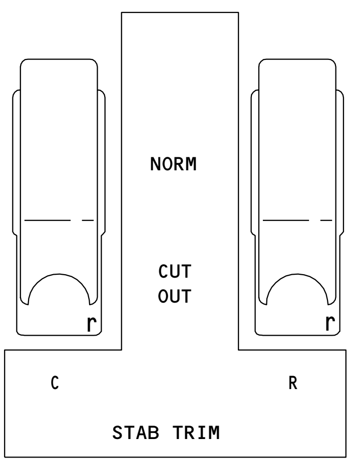
PILOTS OVERHEAD PANEL, P5
- ENGINE IGNITION AND START CONTROL PANEL (EXAMPLE)

A



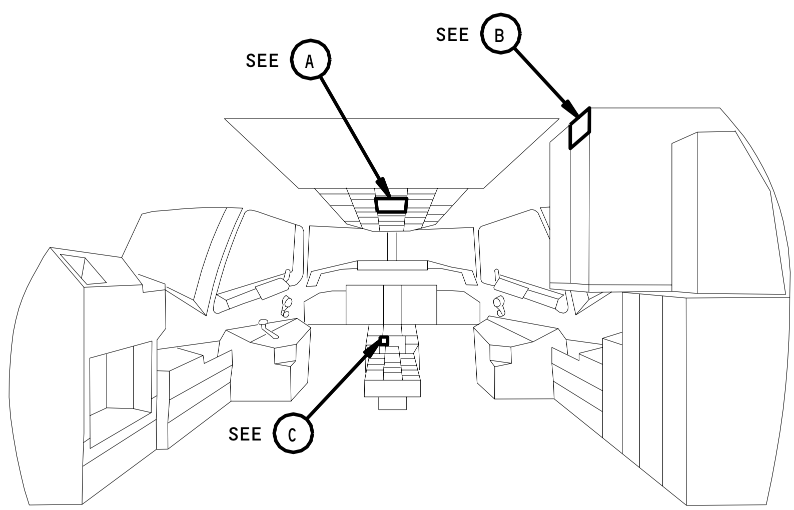
RIGHT SIDE PANEL, P61
- HYDRAULIC FLIGHT CONTROL SHUTOFF SWITCHES (EXAMPLE)

B



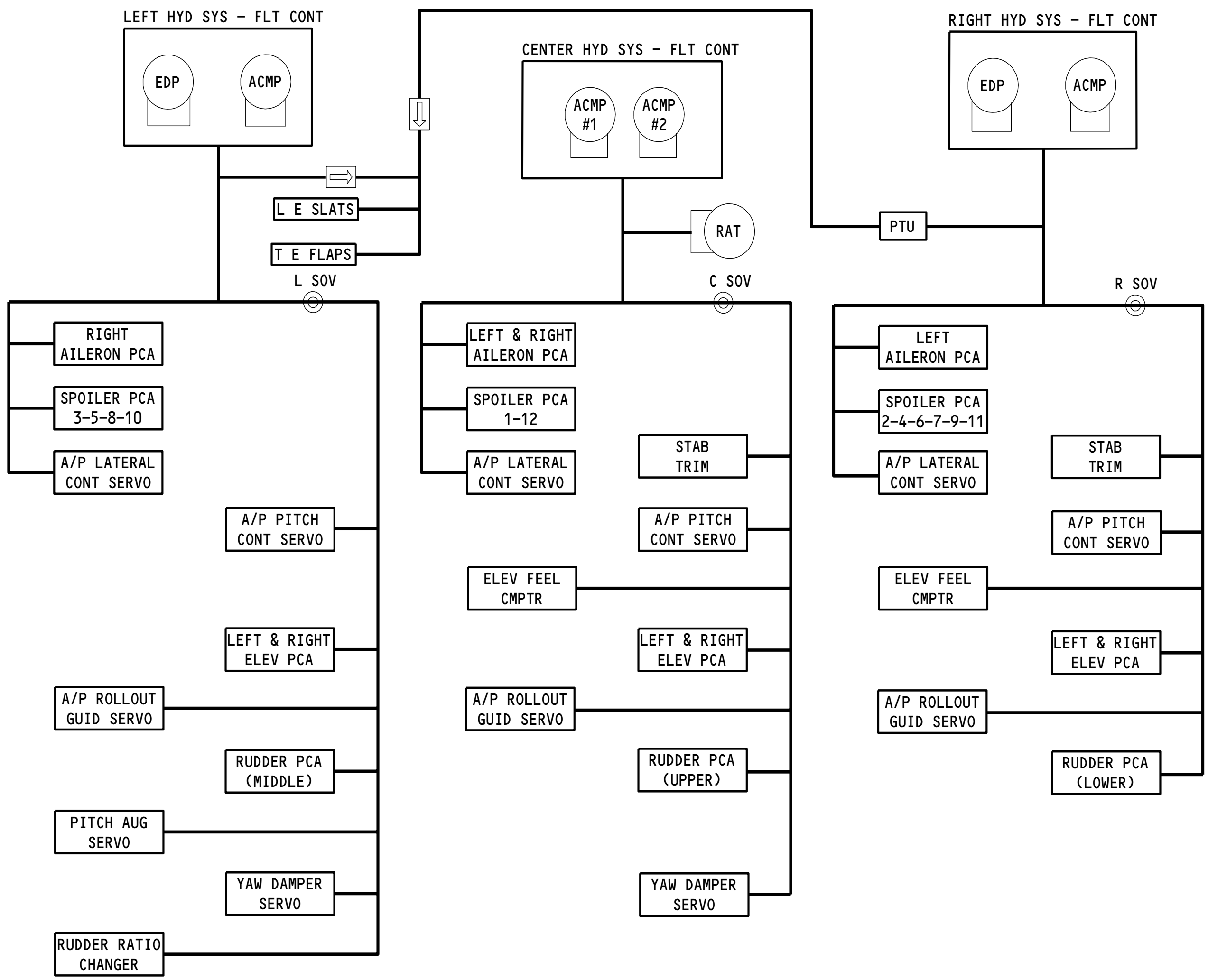
QUADRANT STAND, P10
- STABILIZER TRIM HYDRAULIC POWER CUTOUT SWITCHES

C



FLIGHT COMPARTMENT

Hydraulic System Controls
Figure 9



Flight Control - Hydraulic Distribution
Figure 10

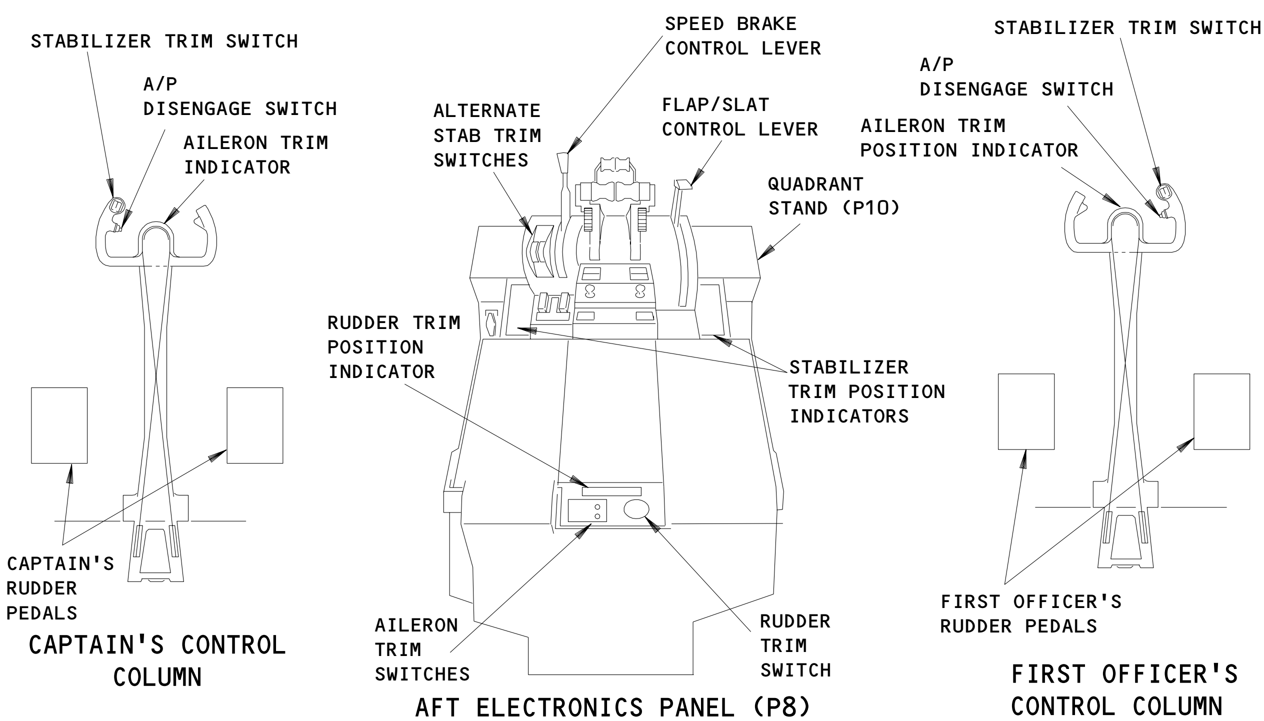
(1) The left, center and right hydraulic systems provide hydraulic power to the respective left, center and right flight control systems actuators and servos. Hydraulic power required for all flight control systems except flap/slats and stabilizer trim, is routed through shut-off valves (SOV). The stabilizer trim system has two individual hydraulic power cutout valves, one in the right and one in the center systems.

(2) All three main hydraulic systems provide power for the following components:
- (a) Ailerons - Autopilot Lateral Control Servos (ALCSs) and Power Control Actuators (PCAs)
- (b) Spoilers - PCAs
- (c) Elevators - PCAs and Autopilot Pitch Control Servos (APCSs)
- (d) Rudder - PCAs and Autopilot Rollout Guidance Servos (ARGSs)

(3) The Elevator Feel Computers (EFCs) hydraulic power is from the right and center systems. The Yaw Damper Servos (YDSs) hydraulic power is from the left and center systems. The left hydraulic system provides power for the Rudder Ratio Changer (RRC).

L. Flight Control - Flight Deck Inputs (Fig. 11)

(1) Primary Flight Control Inputs
- (a) Primary flight control inputs are made using the control column/wheel, rudder pedals, and trim switches.

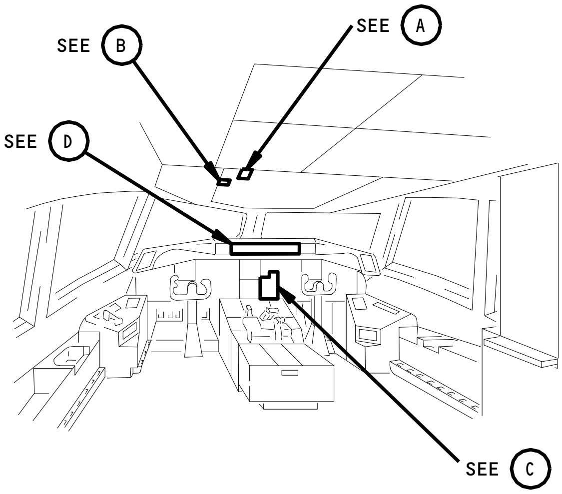


Flight Control - Flight Deck Inputs
Figure 11

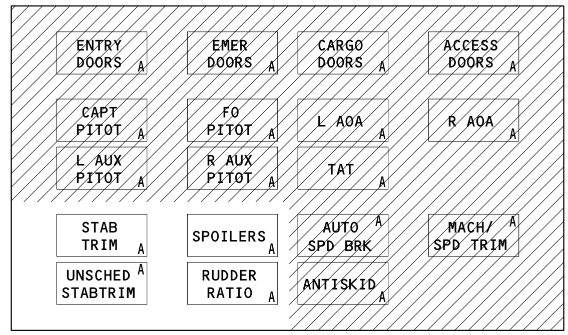
(b) The captain's and first officer's control column/wheels provide the manual control cable input for pitch and roll control. The spoiler system transducers assist in roll control of the airplane.
(c) The outboard horn of each control wheel has an autopilot disengage switch, stabilizer trim switch, and switch for communication systems. An aileron trim indicator is on the top of each control wheel hub.
(d) The captain's and first officer's rudder pedals provide the manual cable input for yaw control. The autopilot and yaw damper systems also provide yaw control inputs.
(e) The aft section of the control stand (P8) contains aileron and rudder trim switches, and a rudder trim position indicator.

(2) Secondary Flight Control Inputs
(a) The stabilizer trim switches (thumb operated) on each control wheel provide an electrical input to drive the stabilizer up and down.
(b) The center section of the control stand (quadrant stand - P10) incorporates the following items:
1) Alternate stab trim switches on the left side for manual trimming of the stabilizer.
2) Stabilizer trim position indicators on each side.
3) Stabilizer trim hydraulic power CUTOUT switches (right and center hydraulic systems) on the left side.
4) Speedbrake control lever on the left side for manually extending the speedbrakes.
5) Flap/slat control lever on the right side for manually extending the flaps and slats.

M. Flight Deck Control Panels (Fig. 12)
(1) Annunciator Lights Panel
(a) Six amber annunciators indicate faulty operation of various flight control systems. The annunciator lights panel is on P5.
(2) Yaw Damper Panel
(a) The two independent yaw damper systems are controlled by individual push-on/push-off switch/lights. The yaw damper is engaged when the switch translucent ON legend illuminates white. The lower part of the switch/light illuminates amber INOP for a yaw damper failure. A YAW DMPR test switch on the test panel on P61 enables ground testing for the yaw damper system.

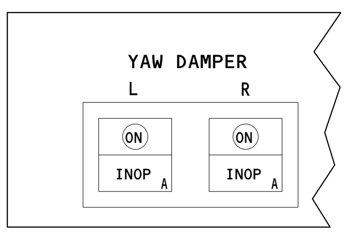


FLIGHT COMPARTMENT



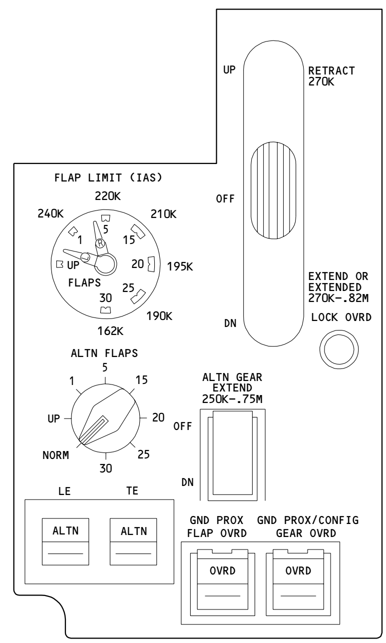
ANNUNCIATOR PANEL, M10394 (P5)

A



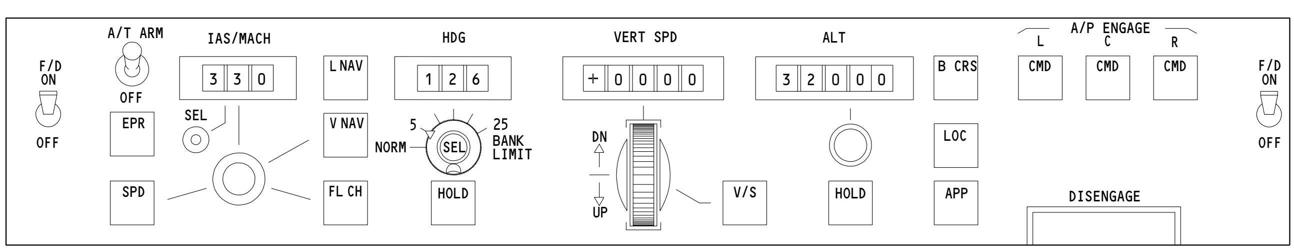
YAW DAMPER PANEL, M10250 (P5)

B



ALTERNATE FLAP CONTROL (P3-1)

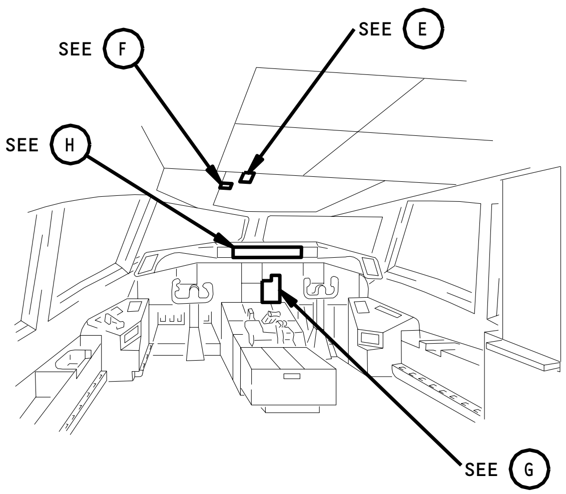
C



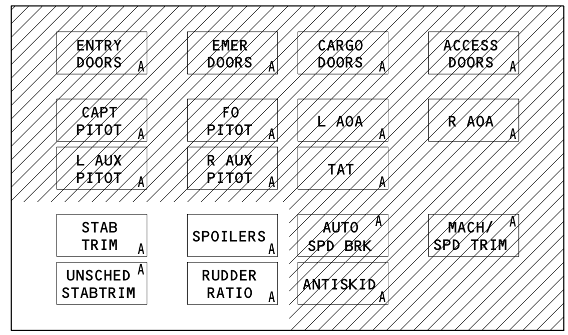
AFDS MODE CONTROL PANEL, M90 (P55)

D

Flight Deck Control Panels
Figure 12 (Sheet 1)

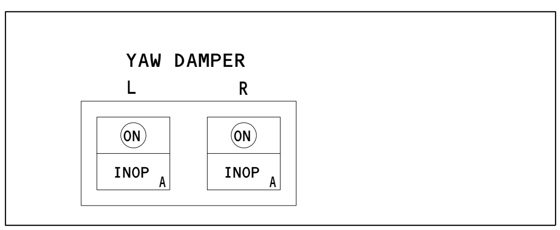


FLIGHT COMPARTMENT



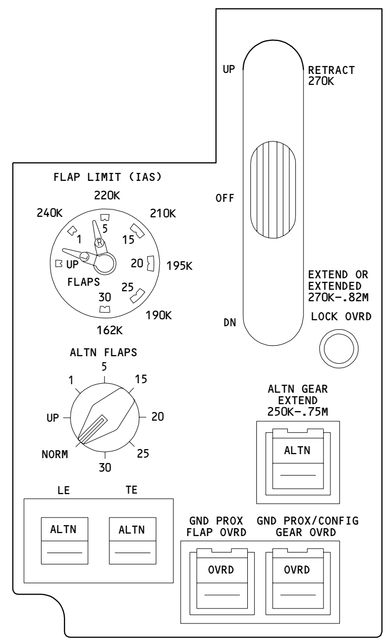
ANNUNCIATOR PANEL, M10394 (P5)

E



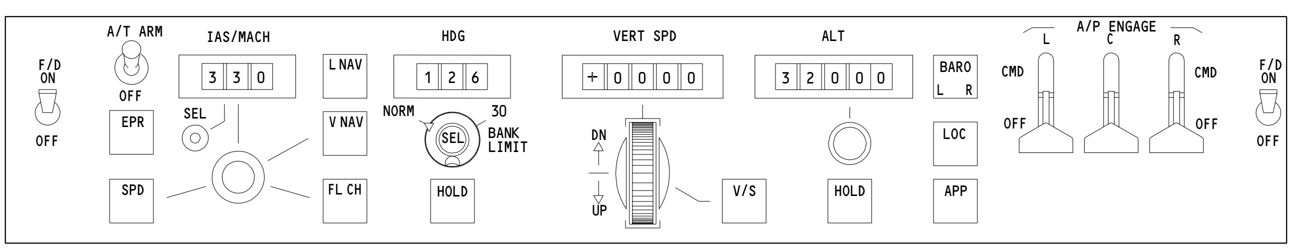
YAW DAMPER PANEL, M10250 (P5)

F



ALTERNATE FLAP CONTROL (P3-1)

G



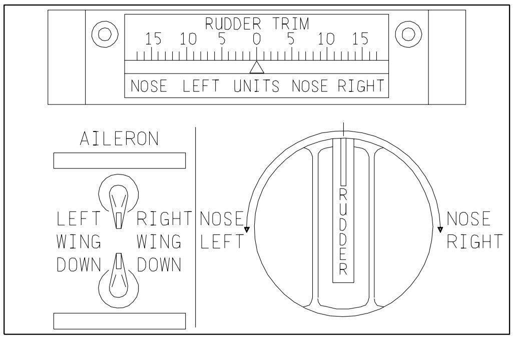
AFDS MODE CONTROL PANEL, M90 (P55)

H

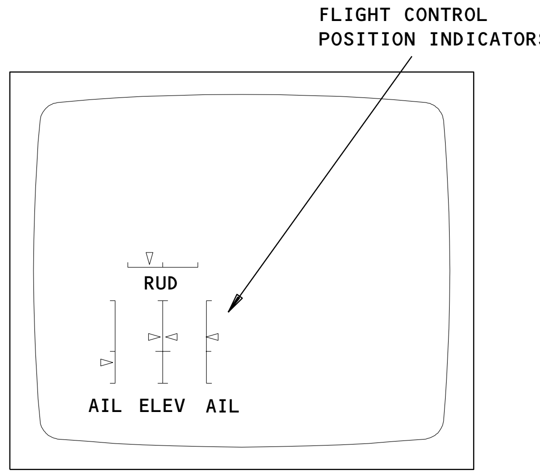
Flight Deck Control Panels
Figure 12 (Sheet 2)

(3) Alternate Flaps Control Panel
   (a) Two push-on/push-off switch/lights arm individual electrical control of leading edge slats and trailing edge flaps. The degree of extension of the flaps/slats is controlled with an 8-position rotary switch. The switch is left in NORM for normal hydraulic control of the flaps and slats.
(4) Autopilot Mode Control Panel
   (a) The autopilot mode control panel enables autopilot control of the primary flight control surfaces in various selected modes. Thrust management system arming and mode selection switches are on the left side of the panel. Captain's and first officer's flight director on-off switches at each end of the panel cause flight director commands to be displayed on the electronic attitude director indicators (EADIs).

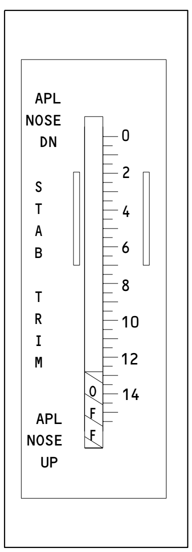
N. Flight Deck Flight Control Indicators (Fig. 13)
(1) Trim Indicators
   (a) Two stabilizer trim indicators are installed, one on each side of the quadrant section of the control stand (P10). The indicators have a green band and units of stabilizer position. The green band indicates the safe takeoff stabilizer position range. The units indicate relative stabilizer position for airplane nose-up and nose-down. The indicators are driven by position transmitters at the stabilizer.
   (b) Rudder trim position is on the aft electronic control panel P8. The indicator is electrically driven by position transmitters at the rudder.
   (c) Aileron trim is indicated on top of the hub of each control wheel. Trim is indicated by the physical offset of the control wheel.
(2) Aileron and Rudder Trim Switches
   (a) Aileron and rudder trim switches are on the aft section of the control stand (P8). The aileron trim switches (2) are spring loaded to the center position. When both are held toward LEFT WING DOWN or RIGHT WING DOWN, they activate arm and control signals that cause the ailerons to be trimmed.



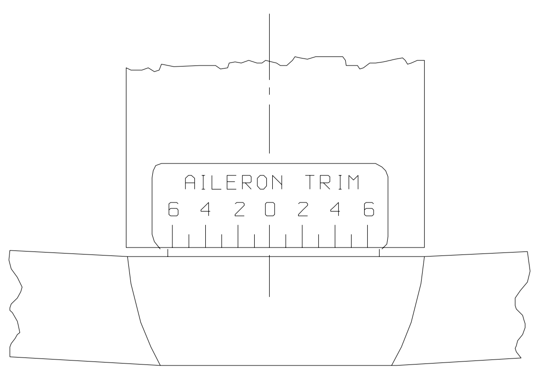
RUDDER TRIM POSITION INDICATOR
RUDDER TRIM SWITCH
AILERON TRIM SWITCHES
(P8)



LOWER EICAS DISPLAY (P2)



STABILIZER TRIM
POSITION INDICATORS
(2) (P10)



AILERON TRIM
POSITION INDICATOR
(CONTROL WHEEL)

Flight Deck Flight Control Indicators
Figure 13

(b) The rudder trim switch is a single rotary switch spring loaded to the center position. When held toward NOSE LEFT or NOSE RIGHT, it activates signals that cause the rudder to be trimmed.

(c) The switches cause trim motors to move cam/roller mechanisms. These change the center position of the control system.

(3) Flight Control Position Indicators

(a) The flight control position indicators are on the lower EICAS display unit when the status page is selected (Ref 31-41-00). The pointers indicate the position of each aileron (2), elevator (2), and rudder (1). A position transmitter at each of the following control surfaces drives the corresponding pointer on the indicators:

1) Left aileron
2) Right aileron
3) Left elevator
4) Right elevator
5) Rudder

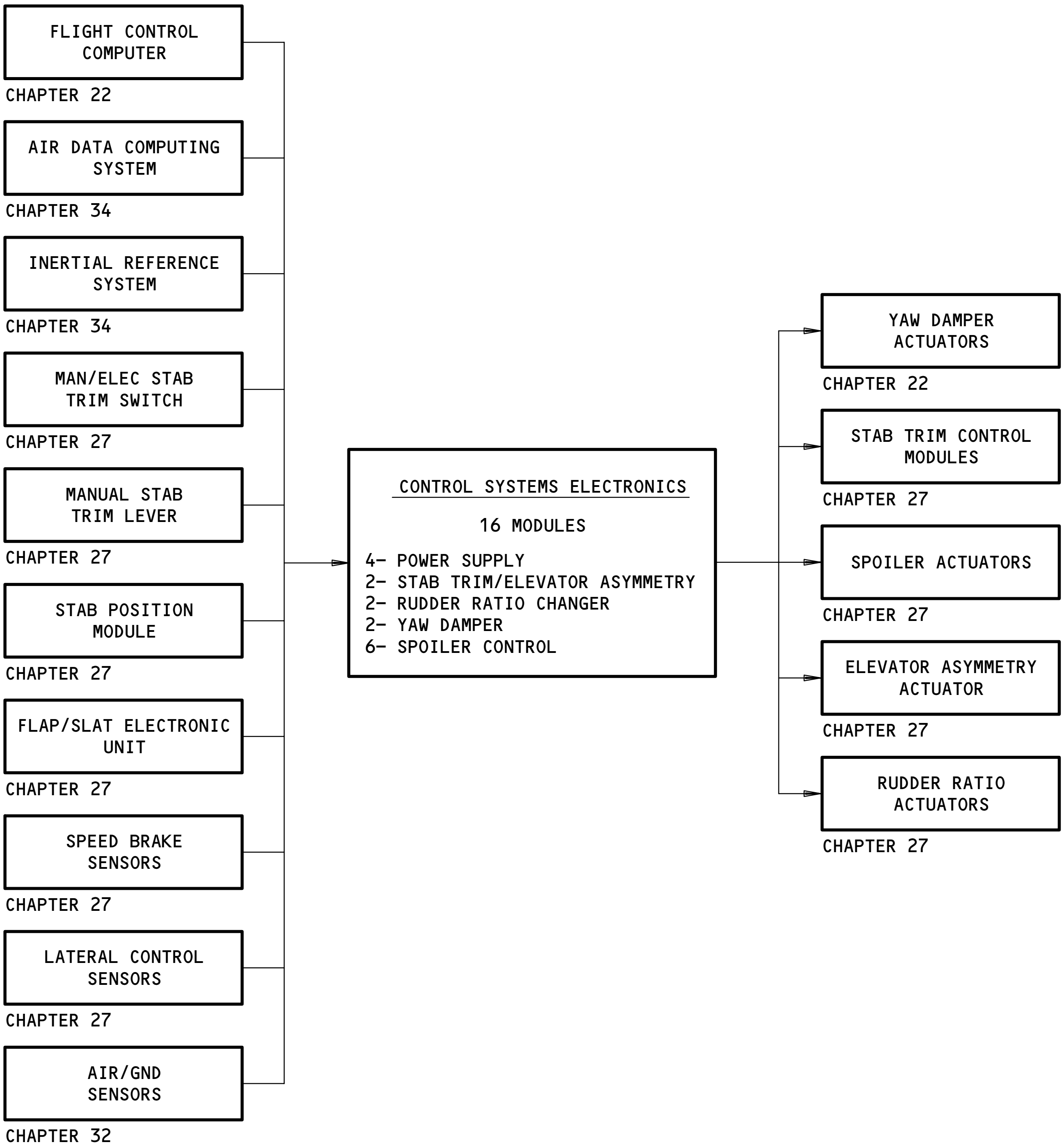
O. Flight Control System Electronics (Fig. 14)

(1) The two Control System Electronic Units (CSEU) consist of 8 separate modules in the left system and 8 separate modules in the right system (Ref Chapter 27). The modules are on the electronic equipment racks in the main electrical/electronics equipment compartment.

(2) The modules drive flight control actuators and stabilizer trim control modules. Sensor inputs are from electrical, mechanical, and information (as shown) systems. The CSEU modules act as buffers for inputs, define required control movements, and then modify and program the inputs to provide the output command signal.

(3) The two power supplies provide power monitoring and necessary voltages for each module.

(4) The stabilizer trim and elevator asymmetry limit module (SAM) processes autopilot automatic trim signals, manual electric trim signals, and drives the stabilizer to relieve elevator loads and maintain pitch authority.



Flight Control System Electronics
Figure 14

(5) The rudder ratio changer module increases rudder travel limits as airspeed is reduced. This maintains rudder authority at low airspeeds.

(6) The yaw damper module controls the rudder to maintain directional control during gusting and dutch roll frequencies. It also provides turn coordination when making turns at low airspeeds.

(7) The three spoiler control modules drive spoiler power control actuators which augment roll authority and act as speed brakes.

P. Hydraulic Power Control Actuators and Servos (Fig. 15)

(1) Hydraulic Actuators for Manual Control

(a) Elevators, ailerons, spoilers, and rudder are driven by Power Control Actuators (PCAs). Each PCA uses hydraulic power from one of the three main hydraulic systems.

1) Each elevator (2) uses three PCAs.
2) Each aileron (2) uses two PCAs.
3) Each spoiler (12) uses one PCA.
4) The rudder uses three PCAs.

(2) Hydraulic Actuators For Automatic Stability Control

(a) Two Yaw Damper System (YDS) servos provide stability control through the rudder. The YDS servos are in the vertical fin below the rudder PCAs.

(3) Hydraulic Actuators for Autopilot Control

(a) Autopilot control of the ailerons is through the Autopilot Lateral Control Servos (ALCSs). The ALCSs use autopilot electrical inputs to drive the aileron PCAs.

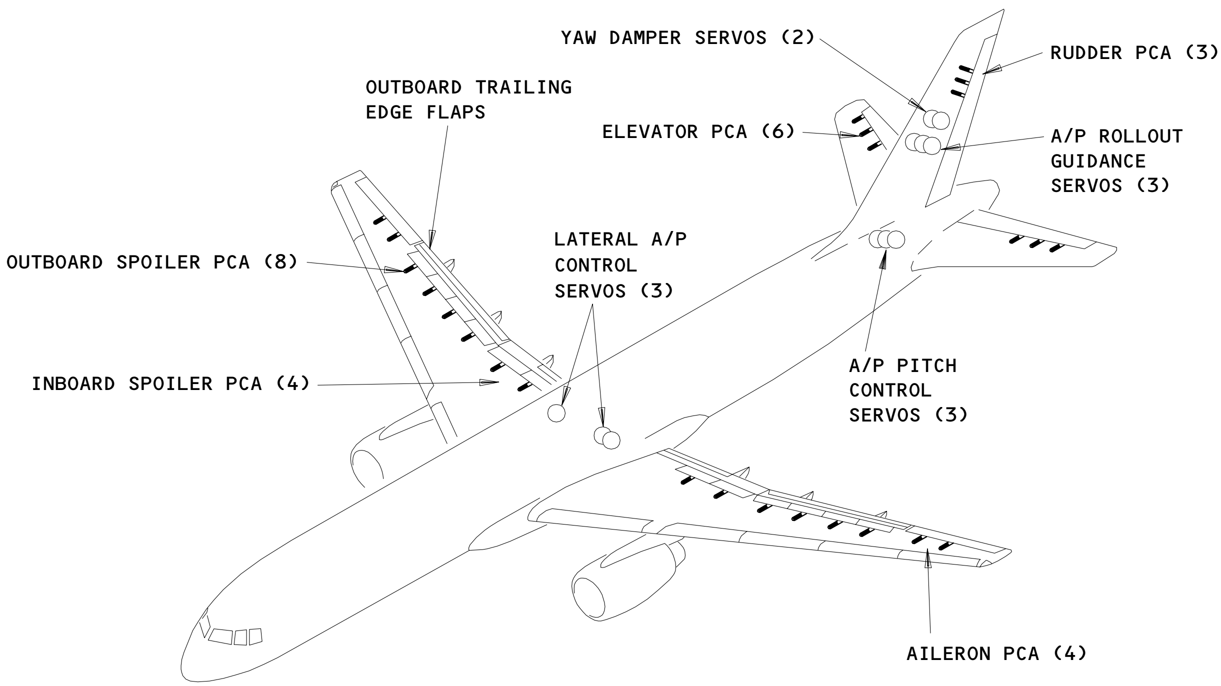
(b) Three Autopilot Pitch Control Servos (APCSs) drive the elevator PCAs. The APCSs are aft of the stabilizer.

(c) Three Autopilot Rollout Guidance Servos (ARGSs) drive the rudder PCAs. The ARGSs are in the vertical fin.

(d) The PCAs are the same ones used with manual control of the airplane.

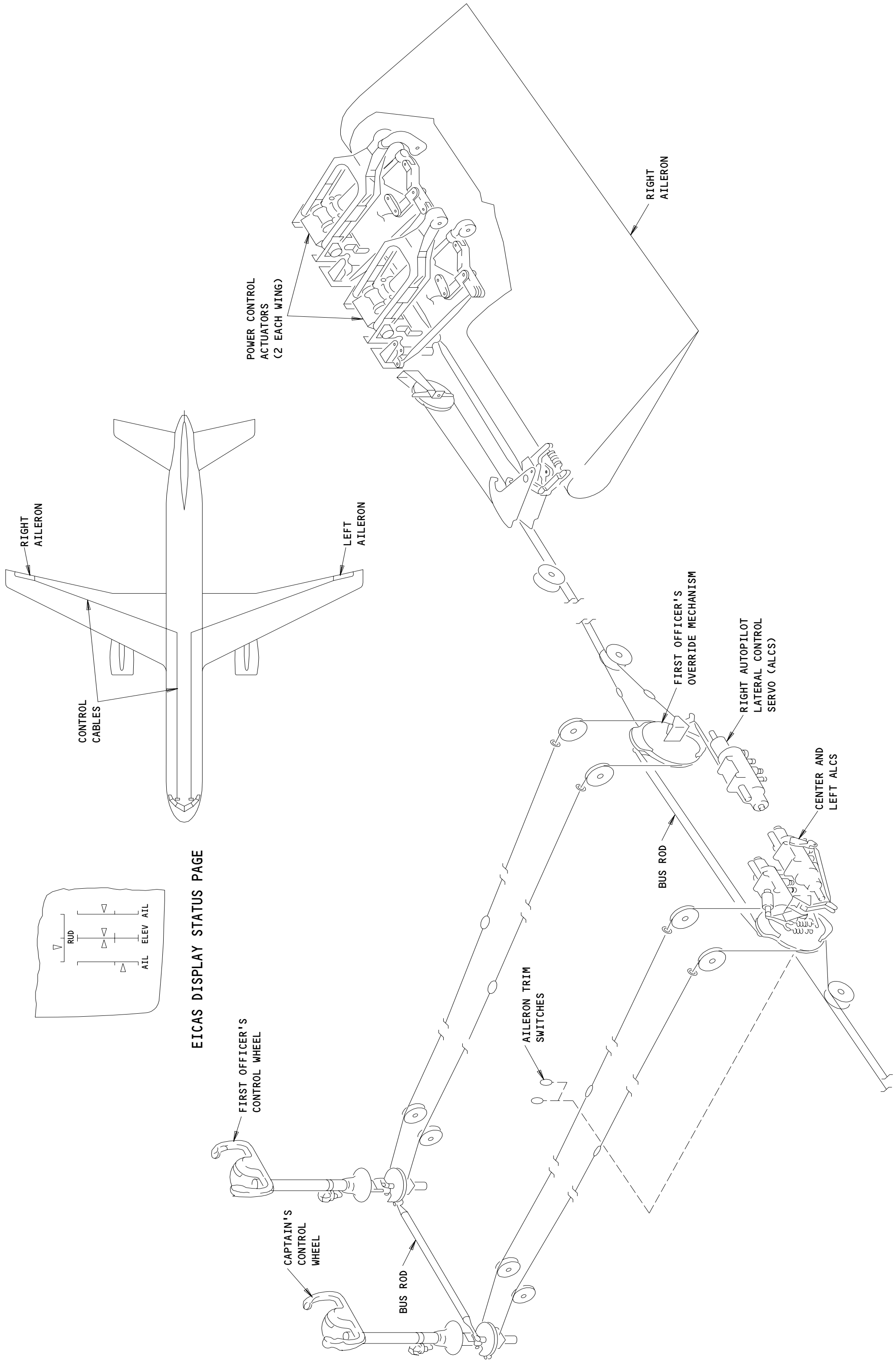
Q. Aileron Control System (Fig. 16)

(1) Primary lateral control (roll) is provided by one aileron on each wing. The ailerons are controlled by inputs from either pilots' control wheel. The inputs are transmitted to the left and right aileron actuators via mechanical control paths. Each aileron is positioned by its two Power Control Actuators (PCAs). The captains and first officers control paths are interconnected but are separated via override devices in case of a jam.



PCA = POWER CONTROL ACTUATOR

Hydraulic Power Control Actuators and Servos
Figure 15



Aileron Control System
Figure 16

28072

(2) Control wheel inputs are conditioned by the lateral feel, trim, and center mechanism. The assembly provides a sense of feel and centering for the pilots' control wheels.

(3) The flight control computers provide electrical inputs to the Autopilot Lateral Control Servos (ALCSs). The ALCSs then drive the PCAs for roll control and back-drive the control cables to turn the control wheels.

R. Lateral Feel, Trim and Centering

(1) The lateral feel, trim and centering mechanism is part of the left aft quadrant. It is mounted on the lower part of the left wheel well forward bulkhead. The assembly provides a sense of feel and centering for the pilots control wheels. Artificial feel is necessary because aerodynamic pressure is interrupted by the hydraulic power control units.

(2) Feel is provided by a moving cam with a spring loaded cam follower. The cam is on the aft quadrant shaft, which is rotated by cables to the control wheels. The follower is a roller referenced to structure through the trim actuator. Movement of the aft quadrant shaft and cam is resisted by the spring loaded cam follower. The cam causes the follower to stretch the spring, and provide a centering force.

(3) The trim actuator changes the relative position of the cam follower, thereby changing the center position of the cam. The actuator is an electric motor with a jackscrew drive shaft. Trim centering occurs when the actuator moves the spring loaded cam follower causing the roller to ride up the sides of the cam. This increases tension on the spring and the cam. The quadrant and control wheels relieve this spring tension by centering the cam on the roller.

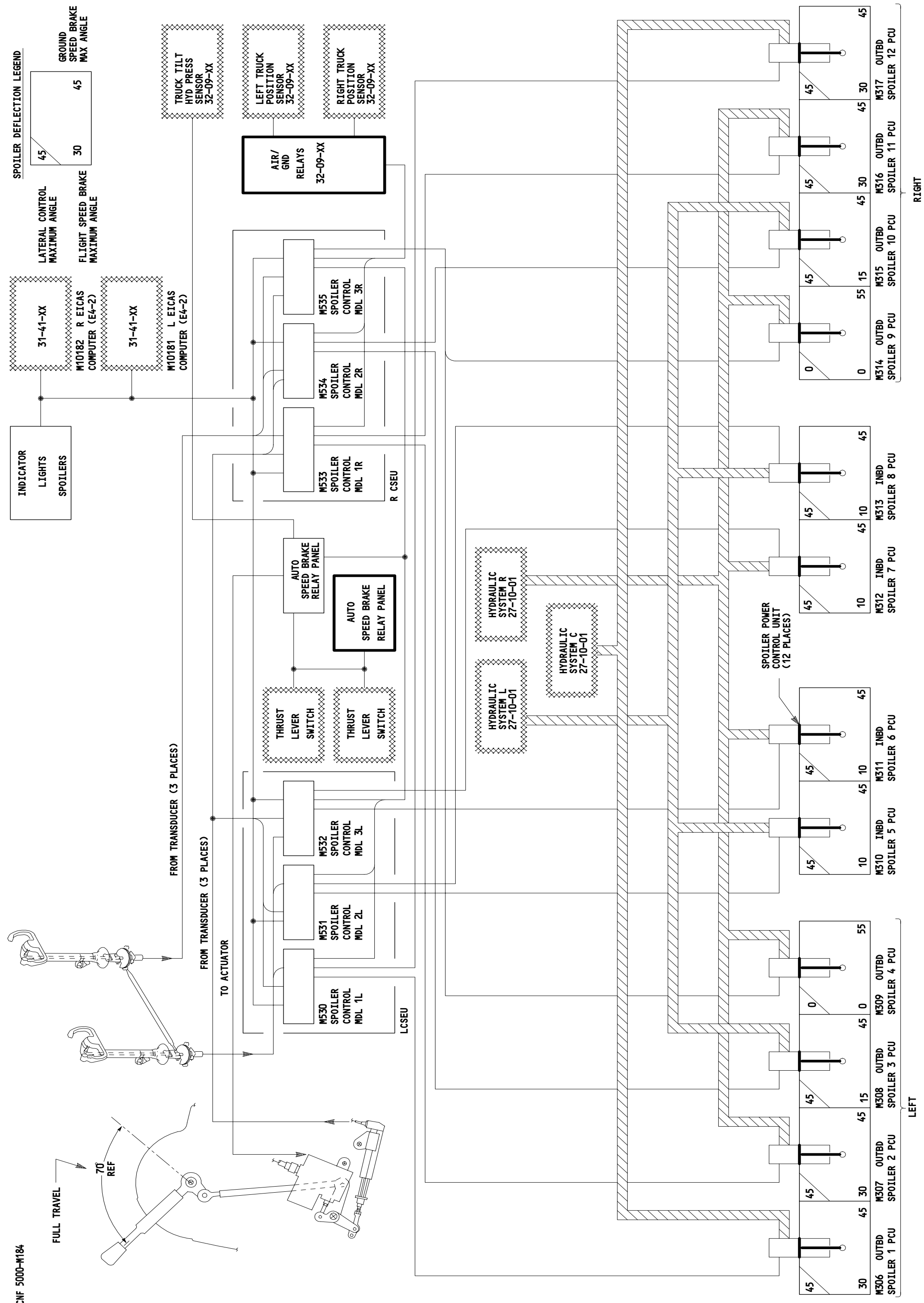
S. Aileron Control System Operation

(1) The aileron control system consists of interconnecting cables and pushrods. Control wheel movement is either by manual force, or is back-driven by an ALCS. Any one ALCS can drive the primary system through interconnects in the wheel well. Aerodynamic feel is artificially applied by the feel, trim, and centering mechanism, since hydraulic force overcomes aerodynamic forces.

(2) Each aileron is powered by two PCAs. The PCAs are controlled by manual inputs or autopilot inputs from the FCCs through the ALCSs. Each PCA and ALCS uses power from one of three main hydraulic systems. The aileron position is detected by position transmitters on each aileron and is indicated on the lower EICAS display unit status page (Ref 31-41-00).

(3) Three Rotary Variable Differential Transducers (RVDTs) are on each forward quadrant. Each RVDT senses control wheel movement and provides independent electrical signals to Spoiler Control Modules (SCMs). The signals are processed by the SCMs to command spoiler movement that augments roll control. The spoiler RVDT primary winding uses 26 vac from the CSEU power supplies.

T. Spoiler Control System Operation (Fig. 17)



Spoiler Control System
Figure 17

(1) Twelve spoilers are on the airplane wings. Spoilers one through six are on the left. Spoilers seven through twelve are on the right. Spoiler pick-up is at 10 degrees of control wheel rotation. Inputs from the RVDTs on the forward quadrants are proportional to control wheel movement. The inputs are processed by six spoiler control modules (SCMs).

(2) Each SCM drives two PCAs, one per spoiler. Feedback from each PCA RVDT provides panel position to the SCMs. The SCMs command spoiler travel proportional to control wheel rotation. Spoiler travel is limited to a 45-degree maximum, which is equal to 85 degrees control wheel rotation.

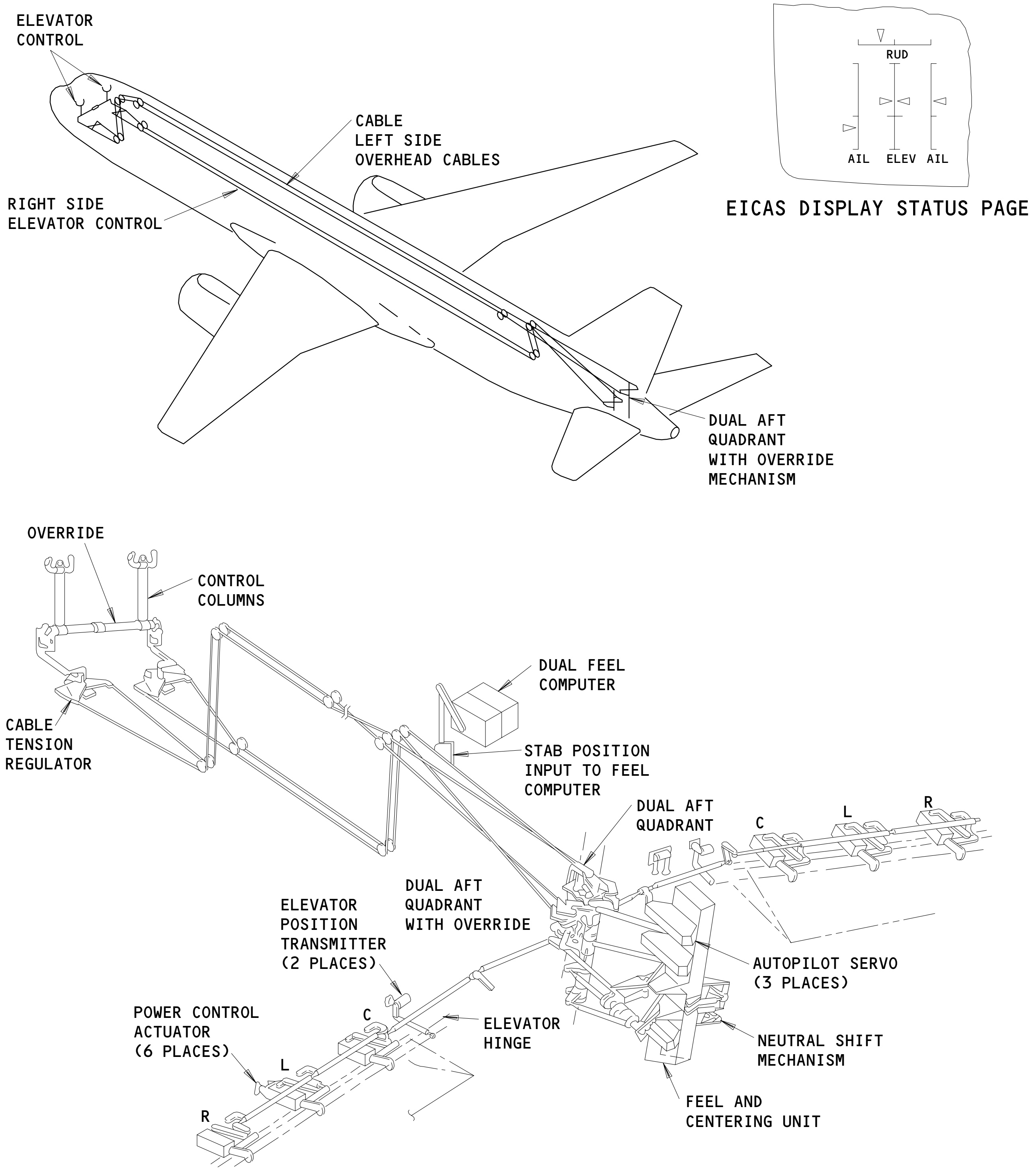
(3) All the spoilers are used for lateral control except panels four and nine. These are used as ground speedbrakes only and have a maximum travel of 55 degrees. The remaining spoilers can be used as inflight speedbrakes with varying maximum limits. When the airplane lands, the automatic speedbrake system deploys all panels to the maximum ground limit.

U. Elevator Control System (Fig. 18)

(1) The elevator control system is operated either manually or by the autopilot. Manual pitch control is obtained with separate cable systems. Control columns on a torque tube move tension regulators, cable, aft quadrants, linkages, PCAs, and elevators. Manual control is unassisted mechanical movement of the PCAs, which then move the elevators. The aft quadrants are aft of the stabilizer. The PCAs are on the stabilizer aft spar.

(2) Autopilot input is from three flight control computers (FCCs) to the three Autopilot Pitch Control Servos (APCSs). The servos drive the elevators and back-drive the elevator linkage to move the control columns. The servos contain LVDTs that provide servo and surface position feedback to the FCCs.

(3) Control inputs are conditioned to provide a sense of feel and centering. Feel is introduced by a feel and centering unit. Inputs are from the neutral shift and a feel computer. The feel computer uses dynamic pressure (CAS) and stabilizer position to vary the feel. The feel unit includes the centering device. Faults are indicated with an ELEV FEEL message on the upper EICAS display unit (Ref 31-41-00).



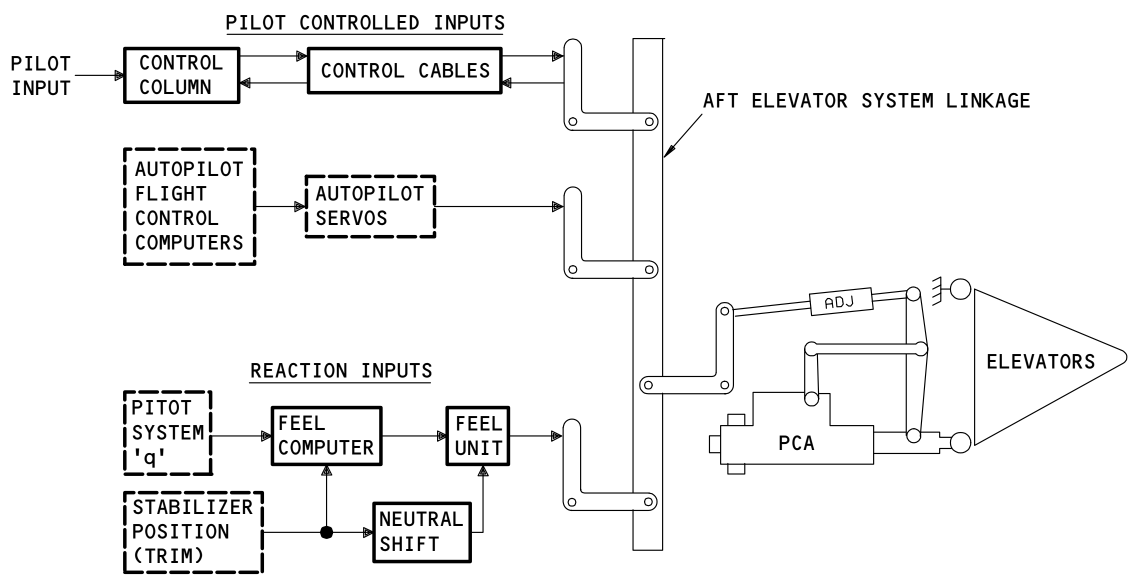
Elevator Control System
Figure 18

(4) Elevator control movements are routed to both left and right power control actuators by mechanical linkage. There is an interconnection between elevator PCA linkages by slave cable to prevent major asymetry. Position transmitters are located at each elevator to provide control surface position on the lower EICAS display unit status page.

V. Elevator Control Modes (Fig. 19)

(1) All elevator control is accomplished by movement of a common set of linkages. Both aft quadrant torque tubes operate together. The mechanical linkage translates torque tube movement into PCA control lever movement. The hydraulically powered PCAs move the elevators. Linkage between the PCA output shaft and input linkage centers the control lever when the elevator position corresponds with the input linkage position.

(2) Pitch control is accomplished by pilot controlled inputs through the control columns and autopilot. Manual input, through the control columns, mechanically moves the PCAs. If the autopilot is engaged outputs are processed by an FCC to control an autopilot pitch control servo. The servo then drives the linkage to the PCAs and backdrives the control cables to move the control columns.



Elevator Control Modes
Figure 19

(3) Pilot controlled inputs are conditioned by reaction inputs. The feel computer reacts to stabilizer position and dynamic pressure, and then provides appropriate inputs to the feel unit. Neutral shift reacts to stabilizer position and maintains the relative stabilizer-to-elevator position by mechanically moving the feel unit.

W. Elevator Control Conditioning Devices

(1) Elevator Neutral Shift

(a) The elevator neutral shift assembly recenters the elevators in proportion to stabilizer position to maintain elevator authority. The effective range is between 8 and 12 1/2 degrees of stabilizer leading edge down (airplane nose-up). The assembly moves the feel and centering unit through an override cam and follower. The feel and centering then moves linkages to the PCAs, which reposition the elevators. Elevators move from a normal position of 3 degrees down to a maximum of 1 degree up.

(2) Elevator Feel and Centering Unit

(a) Mechanical feel and centering is provided by a cam and follower inside the feel unit. They are between a feel actuator and the feel unit control arms. Control arms are connected to the elevator aft quadrant torque tubes. The feel computer varies hydraulic pressure to the feel actuator in proportion to airspeed and stabilizer position. This provides the appropriate pressures for the sense of feel and centering required by the pilots.

X. Elevator Feel Unit

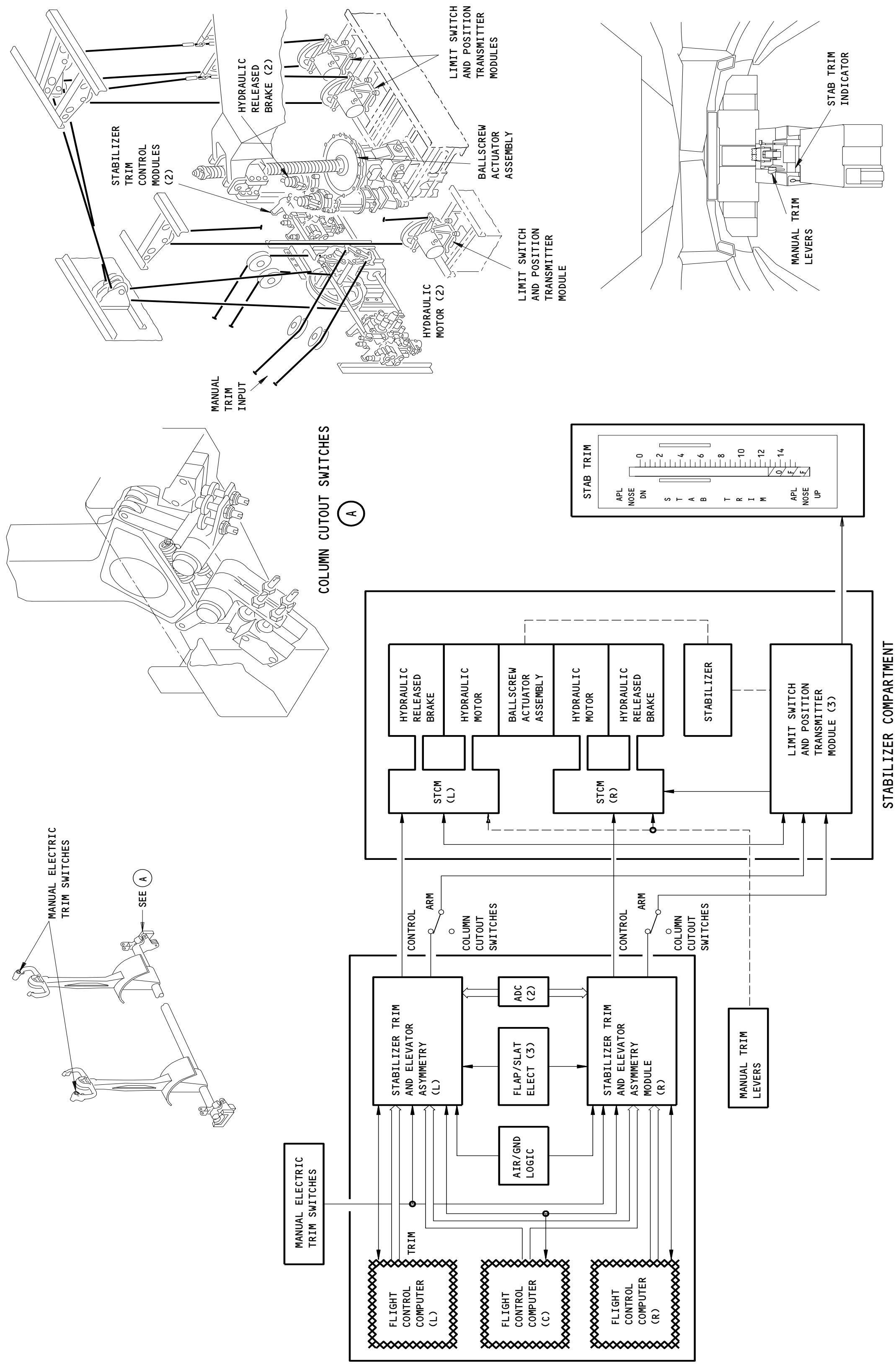
(1) The elevator feel unit generates increasing force with increased control system movement. The scheduling of force application is determined by the feel computer, which uses dynamic pressure and stabilizer position to meter hydraulic pressure to the feel actuator.

(2) The feel actuator resists motion of both left and right elevator control cranks on the feel unit. Resistance varies with hydraulic pressure from the feel computer. Crank motion in either direction pulls the outer levers together, which extends the feel actuator piston rods.

(3) Mechanical feel is then accomplished with a cam and spring-loaded cam follower on the first officer's control crank working against the feel actuator. A cam detent also creates a centering force to keep elevators in the neutral (faired) position.

(4) Total elevator feel force is the combination of mechanical and hydraulic force from the feel unit, plus system friction. Feel force returns the controls to neutral and resists their displacement from neutral.

Y. Stabilizer Trim System (Fig. 20)



Stabilizer Trim System Components
Figure 20

(1) The stabilizer consists of two outboard control surfaces joined together with a center section to form one assembly. The stabilizer pivots about a point at the aft end of the center section. The leading edge of the center section is moved by a ballscrew actuator assembly. The actuator is controlled by two stabilizer trim control modules.

(2) The stabilizer trim control modules respond to either mechanical or electrical inputs. Manual stabilizer trim levers drive cables which move ARM and CONTROL hydraulic valves on both stabilizer trim control modules. The left cables provide ARM inputs. The right cables provide CONTROL inputs. Electrical inputs from the SAM control solenoid operate ARM and CONTROL valves on each module.

(3) Three limit switch and position transmitter modules are mounted on fixed structure below the center section leading edge. Switches and transmitters in the modules are positioned by control cables driven by the stabilizer. Each module includes a Rotary Linear Variable Transformer (RVDT), a synchro, and limit switches. The RVDT provides stabilizer position inputs to the Stabilizer Position Module (SPM). The synchro provides position signals for the stabilizer trim position indicator on the control stand.

(4) Stabilizer trim faults are indicated by an amber STAB TRIM light on the P5 panel.

Z. Stabilizer Trim Control and Operation

(1) Stab Trim Control

(a) Four priorities of trim are available.

1) First priority trim control is manual via the trim levers and cables. The manual ARM input to the Stabilizer Trim Control Modules (STCM) inhibits all electrical inputs (Ref 27-41-00, Horizontal Stabilizer Trim Control System).

2) Second priority is pilots' electric trim (using the thumb switches) if only one autopilot FCC is engaged. Thumb switch electric trim is third prioirty if more than one FCC is engaged.

3) Third priority is autopilot automatic stabilizer trim control if one FCC is engaged. Automatic stabilizer trim is second priority if more than one FCC is engaged (Ref 22-22-00, Automatic Stabilizer Trim).

4) Fourth priority is Mach/speed trim (Ref 22-24-00, Mach Trim/Speed Stability).

(b) The two hydraulic stabilizer trim CUTOUT switches on the control stand operate shutoff valves on the stabilizer trim control modules.

(2) Stab Trim Operation

(a) The STCMs are operated from two inputs, mechanical and electrical. Mechanical input is with the manual trim levers, which move control arms on both STCMs. All electrical ARM and CONTROL input signals come from the stabilizer trim and elevator asymmetry limit modules (SAM). These signals are interrupted by column cutout and limit switches between each SAM and STCM.

(b) The SAM output signals are initiated by operating manual electric trim, automatic stabilizer trim, or within the SAM (Mach/speed trim). Inputs which initiate SAM outputs are:
1) Pilots' thumb operated switches - analog discrete 28 vdc for manual electric trim.
2) FCC - Analog discrete ARM and digital CONTROL signals for automatic stabilizer trim.
3) SPM and ADC - analog stabilizer position (SPM) and digital airspeed (ADC) for Mach/speed trim.

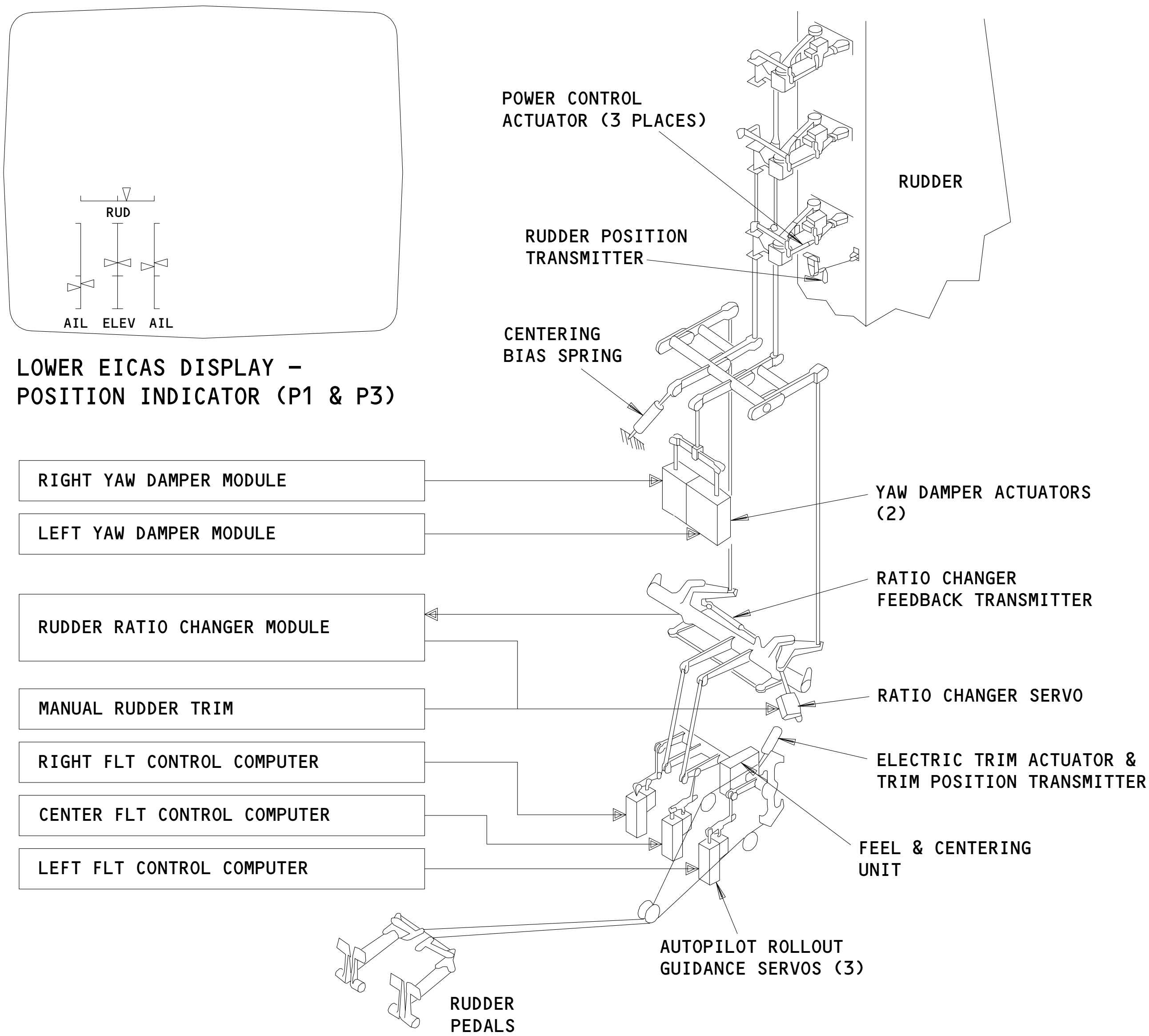
(c) The STCMs provide the hydraulic pressure to the hydraulic motors on the ballscrew actuator assembly. Both motors operate with manual lever and thumb switch inputs. The maximum rate of stabilizer movement is 0.5 degrees per second.

(d) The automatic stabilizer trim and Mach/speed trim control use only one hydraulic motor. Only one SAM is in command at any time as determined by the FCC engaged. The rate of stabilizer movement is 1/2 of that used for manual or manual electric control. Maximum rate is 0.25 degrees per second.

(e) A hydraulic pressure signal from the elevator feel computer modulates the flow rate of fluid to the hydraulic motors via a rate valve in the STCM. This signal causes the stabilizer trim rate to vary as a function of airspeed. The trim rate produced by one STCM is between 0.1 and 0.25 degreees per second. Two STCMs produce a trim rate of 0.2 to 0.5 degrees per second.

AA. Rudder Control System (Fig. 21)

(1) Rudder system control components are primarily in the vertical stabilizer. They consist of the following items:
(a) Power Control Actuators (PCAs) (3)
(b) Position Transmitter
(c) Yaw Damper Servo (2) - Actuator
(d) Rudder Ratio Changer - Servo and Assembly
(e) Electric Trim Actuator and Trim Position Transmitter
(f) Feel and Centering Device
(g) Aft Quadrant
(h) Autopilot Rollout Guidance Servos (3)



Rudder Control System
Figure 21

(i) Interconnecting Linkages with dual (primary and secondary) output rods.

(2) Manual control is accomplished by a cable system from the pilots' rudder pedals to the aft quadrant. Autopilot rudder control is only used for directional guidance during landing rollout. Autopilot control signals from the FCCs operate the autopilot rollout guidance servos, which provide an input to the aft quadrant.

(3) Automatic stability control is by the CSEU yaw damper modules, which provide yaw damping and turn coordination through the yaw damper servos.

(4) Control inputs are conditioned by feel and centering, and by the trim and ratio changer. Feel, centering, and trim devices are on the aft quadrant. The ratio changer servo is controlled by the CSEU rudder ratio changer modules. All inputs are translated to move each PCA control valve. The control valve causes the PCA to extend or retract, which moves the rudder. Each PCA is powered by a separate hydraulic system.

(5) Rudder surface position is indicated on the control surface position indicator on the lower EICAS display unit status page (Ref 31-41-00). The rudder position transmitter on the rudder provides an input to the indicator.

(6) An amber RUDDER RATIO light is on the panel P5. Faults in the rudder ratio changer turn on the light.

(7) A rudder trim control switch is on the aft end of the control stand panel P8. The trim control drives the rudder trim actuator, which recenters the rudder.

AB. Rudder Control Conditioning Devices

(1) Rudder Feel, Centering, and Trim

(a) The rudder feel, centering, and trim device is beside the aft quadrant in the vertical stabilizer. Feel and centering is accomplished by a cam and spring loaded cam follower. Feel force increases as the cam follower is forced up the cam's slope by rotation of the aft control quadrant. Centering force is a result of the cam detent being forced to align with the cam follower after input forces to the aft quadrant are removed.

(b) Rudder trim is accomplished when the trim actuator rotates the feel cam to establish a new center position. As the feel cam detent shifts, the cam follower moves and provides an input to the PCAs, which then repositions the rudder. The rudder trim switch on the control stand provides electrical inputs to the trim actuator.

(2) Rudder Ratio Changer

(a) The rudder ratio changer is above the aft quadrant in the vertical stabilizer. The mechanism changes the output control linkage movement in proportion to aft quadrant and CSEU rudder ratio changer module inputs. Rudder displacement is increased at low airspeeds to maintain rudder authority.

(b) A hydraulic actuator responds to inputs from the CSEU rudder ratio changer modules. The rudder ratio changer modules provide an output in proportion to airspeed. The actuator repositions output lever arms to vary the ratio of control rod input to output. The yaw damper system servos also connect to the rudder ratio changer output rods that drive the rudder PCAs.

AC. Yaw Damper and Rudder PCAs (Fig. 22)

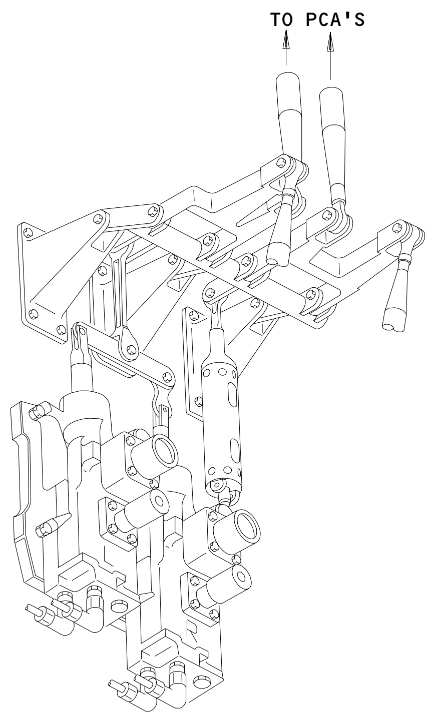
(1) Yaw Damper Assembly

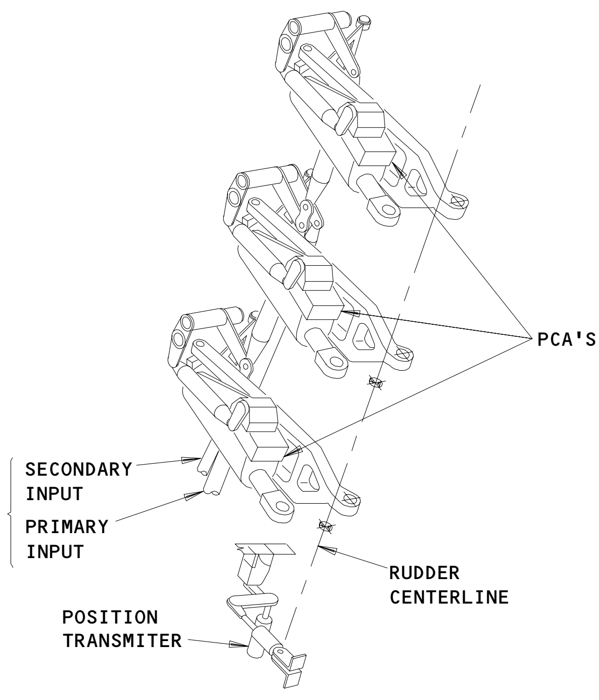
(a) The yaw damper assembly is above the rudder ratio changer assembly. Two yaw damper servos respond to commands from CSEU yaw damper modules to counteract undesired yaw motion. The yaw damper modules provide an output in response to yaw rate and turn coordination (roll) inputs.

(b) Summing linkages combine the output of the two yaw damper servos with the output of the rudder ratio changer. The final control input to rudder PCAs is with two rods, a primary input rod and a secondary input rod.

(2) Rudder PCAs

(a) Three PCAs are above the yaw damper assembly. The PCAs translate control lever inputs to rudder surface movement. Each PCA uses a separate hydraulic system.





Yaw Damper and Rudder PCAs
Figure 22

(b) All PCA control valves are driven by the PCA control levers, while all are interconnected. The primary input rod from the yaw damper assembly and ratio changer is connected to the center PCA. The secondary input rod is connected to the lower PCA. The PCAs cause movement of the rudder about the hinge line in opposition to the reaction link. The rudder position transmitter provides an output to the control surface position indicator on the lower EICAS display unit status page.

AD. High Lift System - Introduction (Fig. 23)

(1) Leading Edge Slats

(a) Five leading edge slat segments are on each wing. Control is by movement of the flap lever on the center section of the control stand (P10). Control cables from the flap lever operate the slat and flap Power Drive Units (PDUs), which use hydraulic or electrical power. Primary control is hydraulic.

(b) The flap lever also enables the Flap/Slat Electronic Unit (FSEU), which controls the hydraulic shutoff valve for each PDU. The PDUs drive a gearbox and rotary actuator at each slat segment. Slat position is transmitted to the FSEU for fault detection and annunciation.

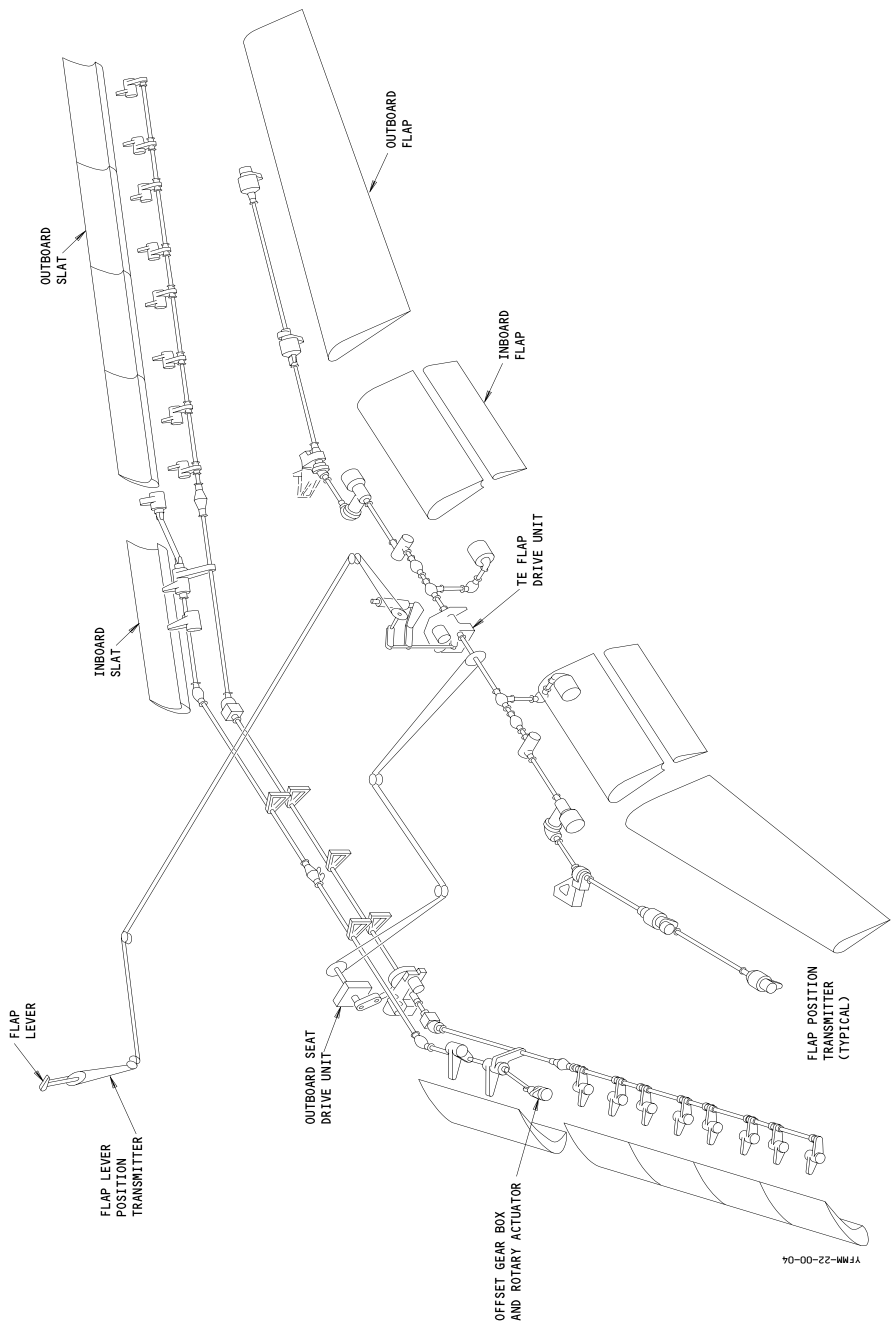
(c) Alternate control is via the alternate flaps control panel on the pilots center instrument panel (P3-1). This panel controls the FSEU which provides electrical output signals that cause the PDUs to extend or retract slats (or flaps) to the selected position.

(2) Trailing Edge Flaps

(a) Two trailing edge flap segments are on each wing. The outboard flap is composed of two flap surfaces.

(b) The flap lever controls flap PDUs and the FSEUs as previously described. Hydraulic power for the flap PDUs is enabled by the FSEU. Flap position is sent to FSEU for fault detection and annunciation. It is also sent to the FSEU for interfacing with other autoflight systems.

(c) Alternate control is via the alternate flaps control panel, as previously described.



High Lift System
Figure 23

(d) The primary drive is powered from the left hydraulic system. The alternate drive uses 115 vac.

AE. Flap Lever/Flap-Slats Positions (Fig. 24)

(1) The flap lever can be positioned between 0 and 30 units. The units do not necessarily relate to degrees of flap extension. Detents are at 0, 1, 5, 15, 20, 25 and 30 units. Trailing edge flap and inboard and outboard leading edge slat relative positions are indicated.

(a) Position 0 fully retracts the flaps and slats.

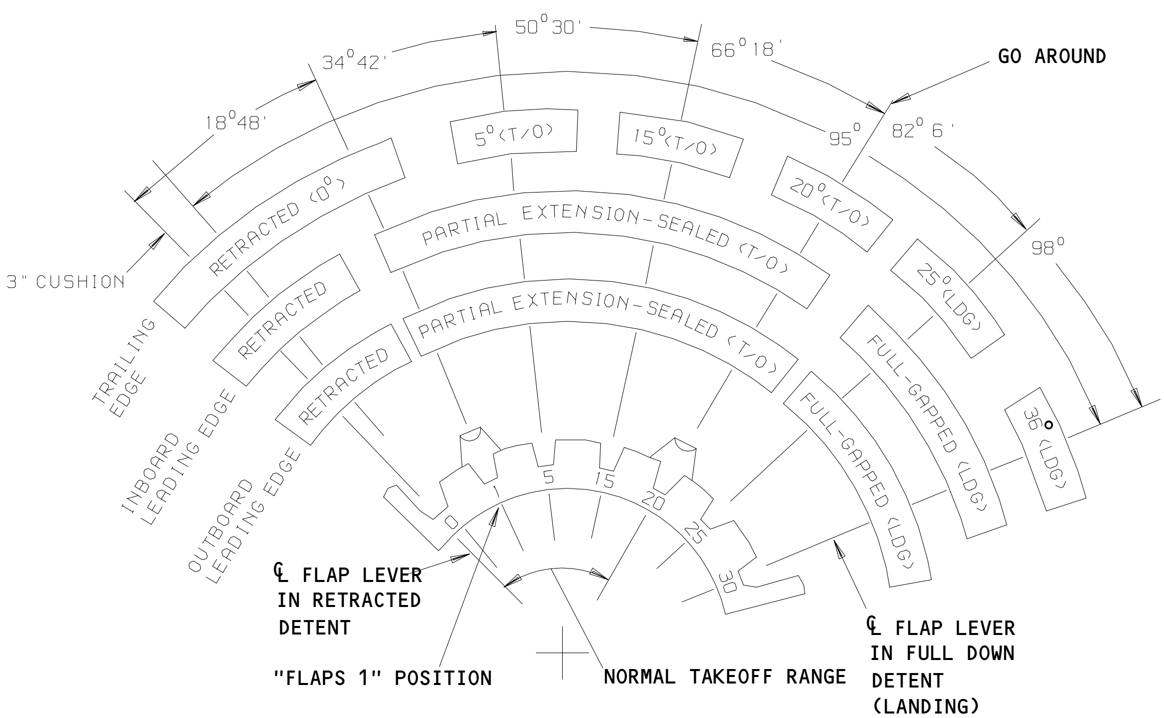
(b) Position 1 results in partial slat extension with the slats still sealed (no gap). The flaps remain fully retracted.

(c) Position 5 results in the flaps extending to 5 degrees after the slats are partially extended (no gap).

(d) Position 15 and 20 extend the flaps to 15 and 20 degrees with the slats still partially extended (no gap).

(e) Position 25 and 30 extends the flaps to 25 and 36 degrees, and fully extends (full-gapped) the slats.

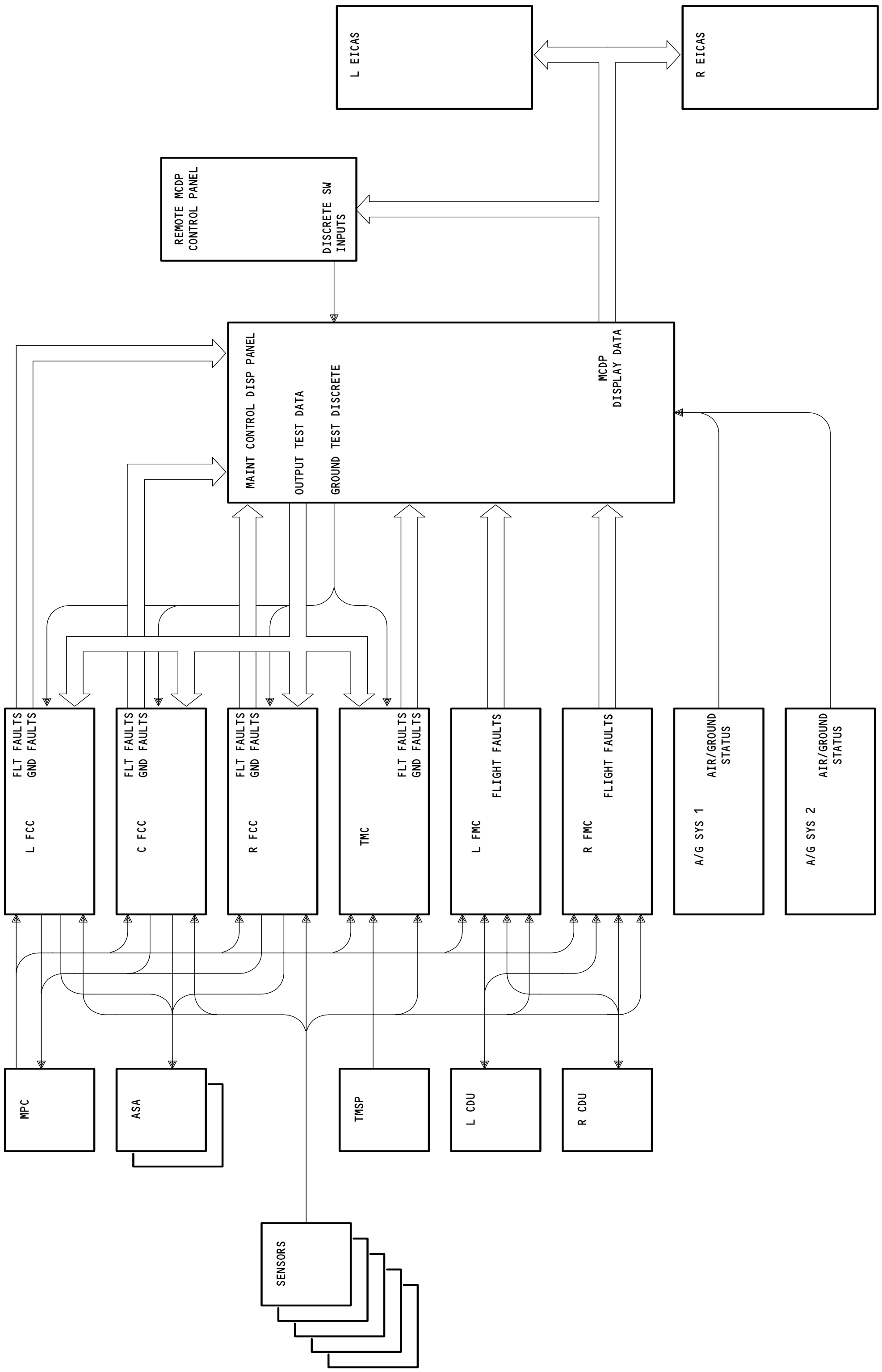
(f) Detent positions 5, 15, and 20 are normally used for takeoff. Detent positions 25 and 30 are normally used for landing.



Flap Lever/Flap Slat Positions
Figure 24

AUTOFLIGHT BITE - DESCRIPTION AND OPERATION

1. General
   A. Autoflight BITE provides an automated, rapid means of testing and verifying the operational status of line replacable units associated with autoflight. BITE testing can be performed with a minimum of effort using only equipment installed on the airplane. Functions contributing to BITE include:
      (1) Built-In-Test-Equipment in individual LRUs.
      (2) Built-In-Tests in individual LRU software.
      (3) Monitor and reporting functions in individual LRUs.
      (4) Control and display functions provided by the Maintenance Control Display Panel (MCDP).
   B. Autoflight BITE includes monitoring and reporting in-flight faults and conducting ground tests.
      (1) Flight fault data is available in two categories.
         (a) Last flight faults contains all fault data detected during the last flight leg, and represents current fault data.
         (b) Previous flight faults contains historical fault data from previous flights.
      (2) Ground tests provide a means of exercising and testing hardware on the ground. Three general categories of tests are available.
         (a) LRU tests provide testing of specific LRUs in the autoflight systems. There are 15 LRU tests.
         (b) System tests exercise the autoflight systems, including LRUs, interfaces, and supporting systems. There are 2 system tests.
         (c) Support tests exercise and monitor supporting functions and components. There are 12 support tests.
   C. BITE messages are provided in four categories.
      (1) Fault messages identifying specific problems.
      (2) Diagnostic codes providing amplifying data on selected faults.
      (3) Status information related to selected functions or components.
      (4) Instruction/questions related to BITE operation and interactive testing requiring operator actions.
2. Component Details
   A. Autoflight BITE is supported by a hardware network and series of software controlled tests. Hardware components and interfaces are shown in Fig. 1.
      (1) Most autoflight BITE functions are associated with the Maintenance Control Display Panel (MCDP). Three major systems interface with the MCDP, providing fault data for system components and interfacing LRUs. These systems are: Autopilot/Flight Director System
      Thrust Mangement System
      Flight Management Computer System
      (2) The Yaw Damper System and associated LRUs provide a BITE independent of the MCDP.



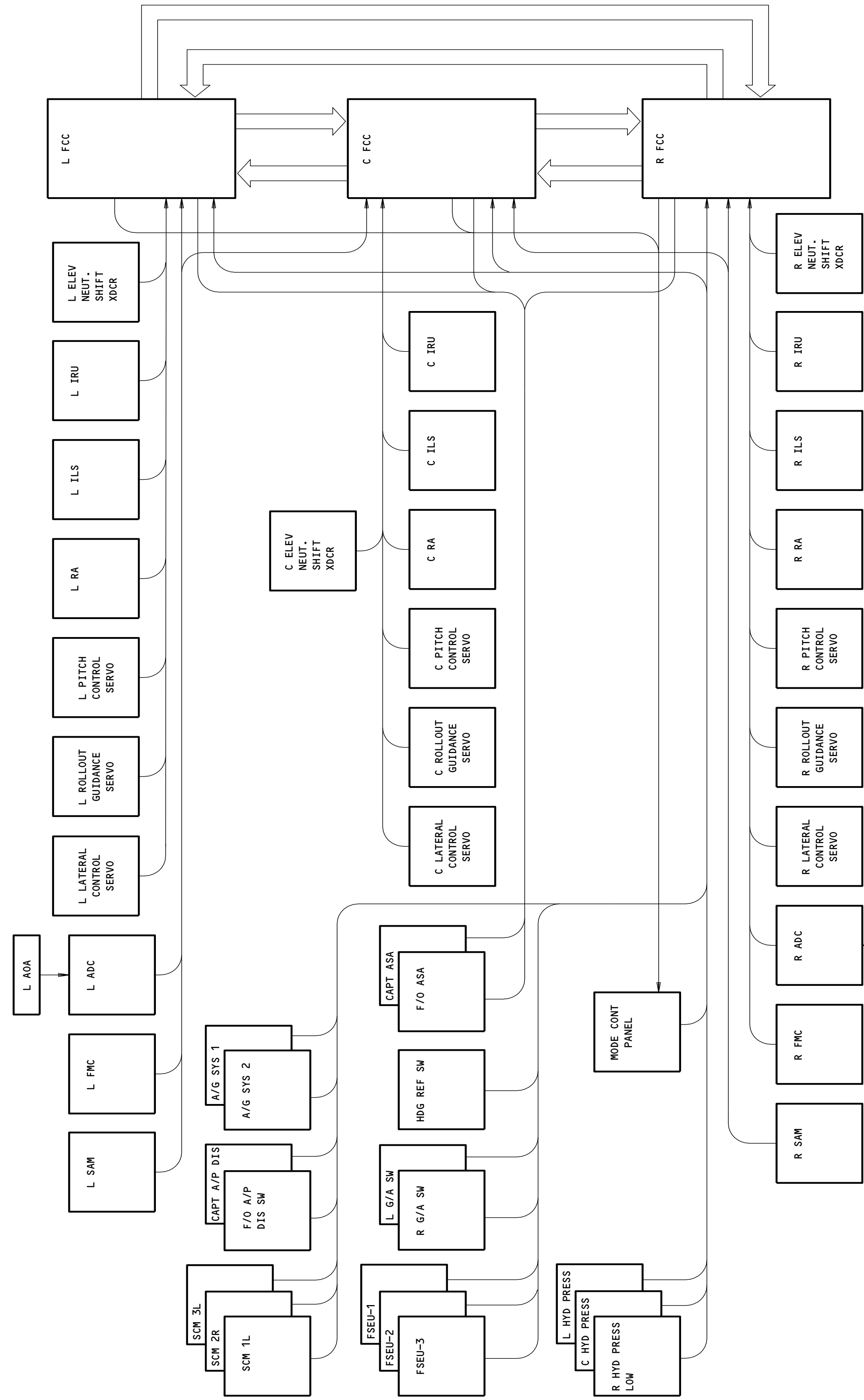
Autoflight Bite Components
Figure 1

148852

B. Autopilot/Flight Director System (Fig. 2)
   (1) The autopilot/flight director system consists of three FCCs and all cross channel data buses, one mode control panel (MCP), and two autoland status annunciators (ASAs).
   (2) The FCCs receive and report fault data from the following LRUs.
      (a) Inertial Reference Unit (IRU) L,C,R
      (b) Instrument Landing System (ILS) Receiver L,C,R
      (c) Radio Altimeter (RA) L,C,R
      (d) Autopilot Pitch Control Servo (APCS) L,C,R
      (e) Autopilot Rollout Guidance Servo (ARGS) L,C,R
      (f) Autopilot Lateral Control Servo (ALCS) L,C,R
      (g) Air Data Computer (ADC) L,R
      (h) Flight Management Computer (FMC) L,R
      (i) Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) L,R
      (j) Low Hydraulic Pressure Light L,C,R
      (k) Flap/Slat Electronic Unit (FSEU -1, -2, -3)
      (l) Go-around switch L,R
      (m) Heading reference switch
      (n) Spoiler Control Module (SCM 3L, 2R, 1L)
      (o) F/O and Capt. A/P Disconnect Switch
      (p) Elevator Neutral Shift Transducer L,C,R

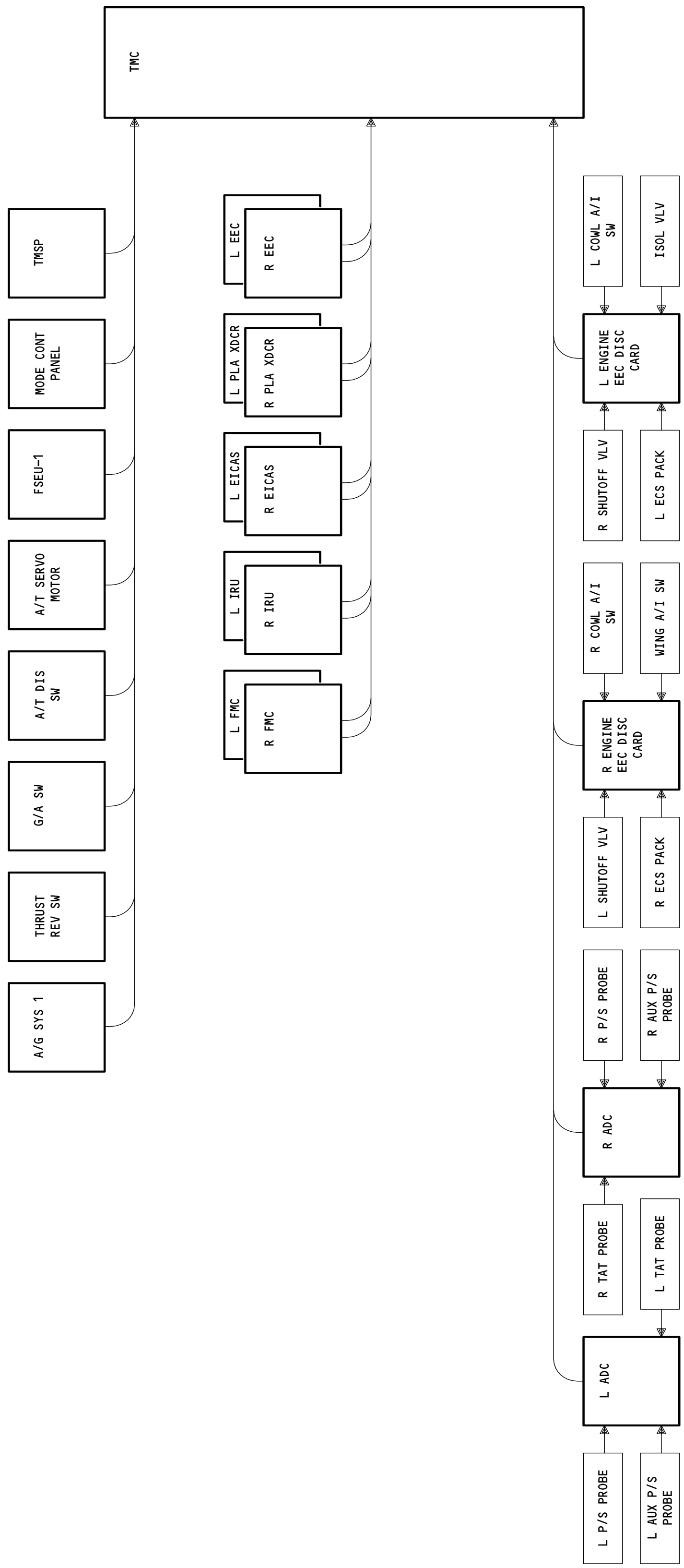
C. Thrust Management System (Fig. 3)
   (1) The thrust management system consists of one TMC, one MCP, and one thrust mode select panel (TMSP).
   (2) The TMC receives and reports fault data from the following LRUs.
      (a) Electronic Engine Control (EEC) L,R
      (b) Engine Indicating and Crew Alerting System (EICAS) L,R
      (c) Inertial Reference Unit (IRU) L,R
      (d) Flight Management Computer (FMC) L,R
      (e) Flap/Slat Electronic Unit (FSEU -1)
      (f) A/T Servo Motor
      (g) A/T Disconnect SW
      (h) Go-around Switch
      (i) Thrust Reverser Switch
      (j) Air/Ground System 1
      (k) Air Data Computers (ADC) L,R with associated probes
      (l) Engine EEC Discrete cards L,R with associated valves, switches, etc.
      (m) Power Lever Angle (PLA) Transducers L,R

D. Flight Management Computer System (Fig. 4)
   (1) The flight management computer system consists of two FMCs and cross channel data buses, one MCP, and two FMC control display units (CDUs).



Autopilot Flight Director System Bite Interfaces and Sensors
Figure 2

152684



Thrust Management System Bite Interfaces and Sensors
Figure 3

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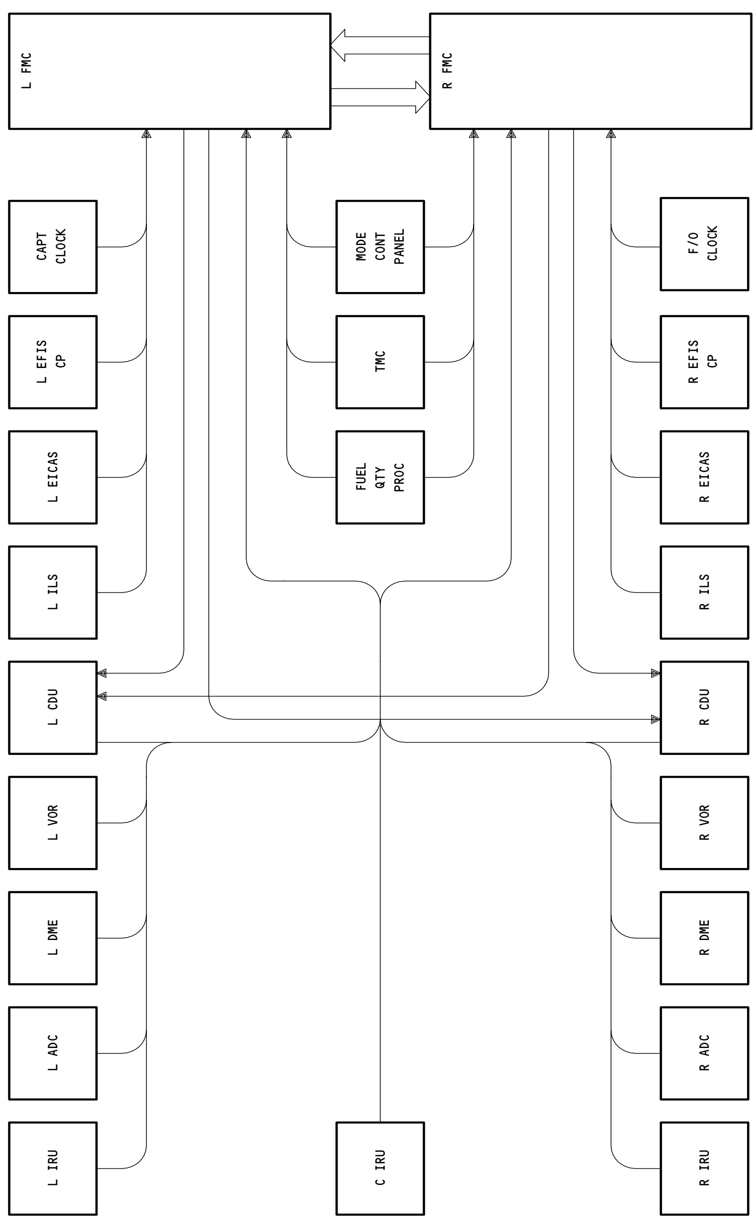


Flight Management Computer System Bite Interfaces and Sensors
Figure 4

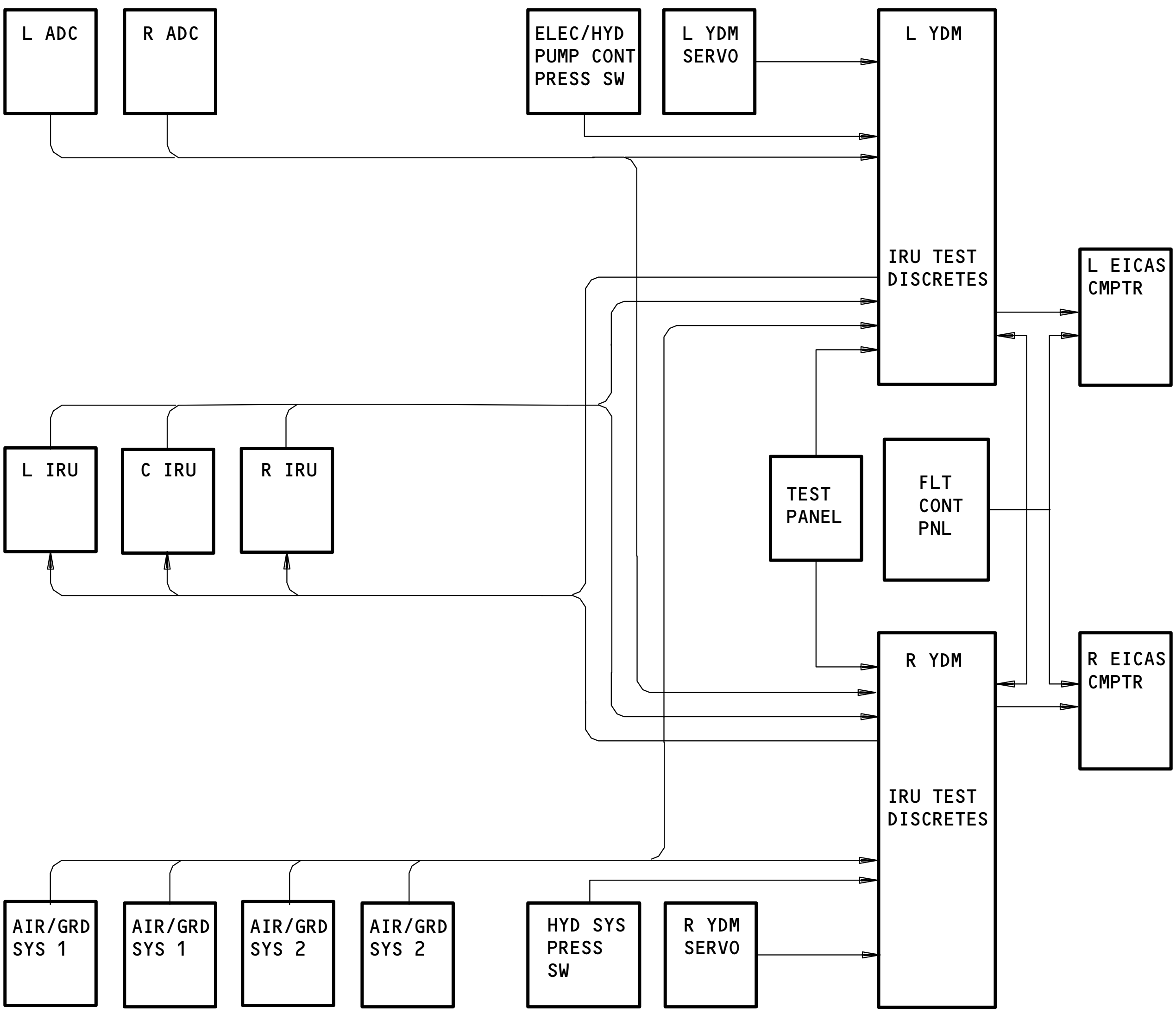
148849

(2) The FMCs receive and report fault data from the following LRUs.
  (a) Captains and first officers clock
  (b) EFIS control panel L,R
  (c) EICAS computer L,R
  (d) ILS receiver L,R
  (e) VOR receiver L,R
  (f) DME L,R
  (g) ADC L,R
  (h) IRU L,C,R
  (i) Fuel Quantity Processor
  (j) TMC

E. Yaw Damper System (Fig. 5)
(1) The yaw damper system consists of two yaw damper modules (YDM), a control panel, and a test switch.
(2) The YDMs monitor the following LRUs.
  (a) YDM Servo L,R
  (b) Elect/Hydraulic Pump and Hydraulic System Pressure
  (c) IRU L,C,R
  (d) ADC L,R
  (e) Air/Ground System 1, 2

3. Operation

A. Flight Faults
(1) At takeoff the FCCs, FMCs, and TMC will purge previously recorded fault data from memory. Throughout the flight each computer will monitor interfaces, interfacing LRUs, and its own circuitry for faulty operation. Any detected faults will be stored in memory for later use.
  (a) Faults detected in-flight can be stored in non-volatile memory and stored in volatile memory. Faults permanently stored in non-volatile memory are for shop use and can only be cleared in the shop. Faults temporarily stored in volatile memory are reported to the MCDP and cleared from memory. Data stored in the two types of memory generally will not be identical.
  (b) Faults reported to the MCDP are those that have a flight deck effect.
(2) Transfer of fault data from the FCCs, FMCs, and TMC takes place at touchdown.
  (a) At touchdown, when the air/ground system transitions from air to ground and ground speed drops below 40 knots, the MCDP will automatically power up and undergo self test.
  (b) Upon successful completion of power up and self test the MCDP will interrogate the FCCs, FMCs, and TMC for stored fault data, initiating automatic transfer.



Yaw Damper Bite Interface and Sensors
Figure 5

(3) Data transferred to the MCDP at touchdown is stored in non-volatile memory. Two classifications of flight faults are last faults and previous faults.
  (a) Last flight faults are those occuring during the last flight leg and represent faults that currently exist.
  (b) Previous flight faults occured prior to the last flight leg. The MCDP can store up to 99 past flights if no more than 10 past flights have reported faults. This data is for historical use, checking reoccurring faults, or holding data for later use.
  (c) Flight fault data stored in the MCDP can only be erased in the shop.

(4) Fault messages can be displayed on the MCDP or EICAS if a remote panel is used. The message is displayed in two lines.
  (a) Top line of the flight faults messages displays the flight deck effect and historical flight tag. The historical flight tag identifies the flight leg in which the fault occurred. 00 is present fault and 01 thru 99 occurred 1 to 99 flights ago.
  (b) Bottom line of the flight faults messages displays the faulty LRU.

(5) Diagnostic codes related to flight faults can be displayed by pressing the flight fault mode switch/light causing the flight deck effect displayed on the message top line to be replaced by the diagnostic.

B. Ground Tests
  (1) Ground tests provide a means of verifying and amplifying on flight faults to aid in fault isolation. Ground test also provide fault indications when no flight faults were recorded.
    (a) Tests 01 thru 16 are LRU tests oriented towards specific components.
    (b) Tests 30 and 40 are system tests involving the entire autoflight system.
    (c) Tests 51 thru 69 are supporting tests oriented towards specific function involving several components.
  (2) 01 FCC (Left, Right, and Center)
    (a) Displays any FCC self-test faults and FCC interface faults from the associated system interfaces.
  (3) 02 TMC
    (a) Displays any TMC self-test faults and TMC interface faults from associated system interfaces.

(4) 04 MCP
  (a) Displays any FCC and TMC self-test faults. Displays FCC and TMC interface faults related to MCP.

NOTE: In later MCP test steps failure of FCC L and FCC C, or FCC R will cause half lighting of lower half of MCP switch/light indicators. Failure of TMC will prevent EPR/SPD/THR switch/lights from lighting.

  (b) Displays the interaction message for checking the MCP lightswitches.
  (c) Displays interaction messages for checking the on and off position of the captain's flight director switch.
  (d) Displays interaction messages for checking the arm and off position of the autothrottle ARM switch.
  (e) Displays the interaction message for checking the command function of the indicated airspeed/mach select switch.
  (f) Displays the interaction message for checking the command function of heading select switch.
  (g) Displays the interaction message for checking the ON and OFF positions of the first officer's flight director switch.
  (h) GUI 115;
  displays interaction messages for checking the engaged and disengaged position of the bat handle switches.
  (i) GUI 001-114, 116-999;
  displays interaction messages for checking the down and up position of the disengage bar.
  (j) Displays interaction messages for checking the command function for each position of the bank limit switch.
  (k) Displays interaction messages for checking the control function of the indicated airpseed/mach display rotary control.
  (l) Displays interaction messages for checking the control function of heading display rotary control.
  (m) Displays interaction messages for checking the control function of the vertical speed display rotary control.
  (n) Displays interaction messages for checking the control functions of the altitude select display rotary control.

(5) 05 TMSP
  (a) Displays TMC self-test faults.
  (b) Displays TMSP-to-TMC interface faults
  (c) Does a TMSP SW/MODE DISPLY test routine which checks the mode display and command function of continuous, climb, and cruise mode switching. Displays an interactive message at end of the test.
  (d) Does a TMSP TEMP SELECT test with interaction messages to check control function of temperature select control display.

(6) 06 ASA
  (a) Displays any FCC self-test faults.
  (b) Displays FCC interface faults related to ASA.
  (c) Checks all ASA displays and their reset function.

(7) 07 SERVO AIL
  (a) Displays any FCC self-test faults and FCC interface faults related to aileron servos.
  (b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
  (c) Provides interaction message to verify aileron trim set to zero.
  (d) GUI 115;
      provide interaction message to position MCP CMD bat handle switches up to CMD. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
  (e) GUI 001-114, 116-999;
      provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
  (f) Wheel centers, then moves clockwise and then counterclockwise and back to center. Inboard ailerons are driven to a plus and minus 10 degree position. FCC's check that each position is within limits.

(8) 08 SERVO ELEV
  (a) Displays any FCC self-test faults and FCC interface faults related to elevator servo.
  (b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
  (c) Provides interaction message to verify elevator trim set to zero.
  (d) Displays elevator feel position pressure for operator to verify pitot pressure is zero.
  (e) GUI 115;
      provide interaction message to position MCP CMD bat handle switches up to CMD. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
  (f) Provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.

(g) GUI 001-114, 116-999;
provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.

(h) Column centers, then moves aft (trailing edge up), then forward (trailing edge down) and back to center. Elevators are driven to a plus 20 degrees (trailing edge up) and a minus 9 degrees (trailing edge down) and the FCCs check that each position of the elevator servos and neutral shift tranducers (FEEL POS) are within limits.

(9) 09 SERVO RUD

(a) Displays any FCC self-test faults and FCC interface faults related to rudder servo.

(b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).

(c) Provides interaction message to verify rudder trim set to zero.

(d) Rudder centers, then moves rudder trailing edge right, then left and back to center. Rudder driven to a plus and minus 20 degrees and FCCs check that each position is within limits.

NOTE: Airplanes with -133 FCCs and option group 1 with a value of 6; If the RUD SERVO X FAIL (X = L, C, R) message shows intermittenly when you do this ground test, ignore this message unless the RUD SERVO X FAIL (X = L, C, R) message is shown in the last flight faults or previous flight faults with the diagnostic code 206.

(10) 10 SERVO A/T

(a) Displays TMC self-test faults.

(b) Throttle is automatically driven aft for 8 seconds at 10 degrees/second and checked for a Power Lever Angle (PLA) of 44 to 60 degrees. The test fails if the PLA test value is not sensed within six seconds. The autothrottle is then driven forward for eight seconds at 10 degrees/second and checked for a PLA of 120 to 136 degrees, and a tachometer signal feedback of 7 to 12 degrees/second. The autothrottle is driven aft for eight seconds at 10 degrees/second, and checked for a PLA of 44 to 60 degrees, and a tachometer feedback signal of 7 to 12 degrees/second.

(11) 11 SW A/P DISC

(a) Displays any FCC self-test faults, and FCC interface faults related to A/P Disc switch.

(b) Instructs the operator to close the captain's and first officer's autopilot disconnect switches. The switch output is monitored for autopilot disconnect command response.

(12) 12 SW A/T DISC

(a) Displays any TMC self-test faults and TMC interface faults related to A/T disconnect switch.

(b) Instructs operator to close the left and right autothrottle disconnect switches. Switch output is monitored for autothrottle disconnect command response.

(13) 13 SW G/A

(a) Displays any FCC and TMC self-test faults FCC and TMC interface faults related to G/A switch.

(b) Instructs the operator to close the captain's and first officer's Go-Around (G/A) switches. The G/A switch command response is monitored at each FCC and TMC.

(14) 14 XDCR COL L

(a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 14 NO TEST THIS A/C CONFIG.

(15) 15 XDCR COL R

(a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 15 NO TEST THIS A/C CONFIG.

(16) 16 XDCR WHL

(a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 16 NO TEST THIS A/C CONFIG.

(17) 17 PVDC

(a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 17 NO TEST THIS A/C CONFIG.

(18) 30 CURRENT FAULT REPORT

(a) Displays all LRU and interface faults detected and stored in the FCCs and TMC while the MCDP operates in the ground test mode. Ground fault messages and diagnostic codes are the same as flight fault messages except the flight deck effect and intermittent bit status are not displayed. Diagnostic codes are displayed by pressing the GRD TEST mode switch.

(b) Provides interaction message to allow operator to skip Interface Fault messages.

(c) Provides interaction message to allow operator to select current Ground Fault display.

(19) 40 AUTOLAND

(a) Displays the current ground faults and interface faults related to the autoland system. The autoland related interfaces are automatically monitored. It automatically performs LRU and support tests related to the autoland system that do not require interaction with the operator.

(b) Automatically performs LRU and support tests related to the autoland system that require no manual preconditioning except for applying electrical power.

(c) Automatically performs LRU and support tests related to the autoland system that require manual preconditions but no hydraulic power.

(d) Performs manual interface checks related to the autoland system.

(e) Performs manual LRU and support tests related to the autoland system.

(f) Performs A/P servo and stab trim tests.

(g) Performs Autothrottle tests.

(20) 51 AIR/GRD RLY
- (a) Displays any FCC and TMC self-test faults.
- (b) Interaction test of the landing gear system 1 air/ground relay and its interface to the FCCs and TMC.
- (c) Interaction test of the landing gear system 2 air/ground relay and its interface to the FCCs.

(21) 52 TMC RLY/SW
- (a) Displays any TMC self-test faults and TMC interface faults related to Environmental Control System (ECS) cards, pneumatic system, anti-ice, and thrust reverser lever position.
- (b) Tests for discrete output signals from the left and right Electronic Engine Controls (EEC) discrete cards, from the autothrottle microswitch pack assembly, pneumatic shutoff and isolation valves, and anti-ice valves.

(22) 56 FCC CONFIG/OPT
- (a) Displays any FCC self-test faults and FCC interface faults related to parity.
- (b) Checks the status of the FCC program pin wiring for airplane and hardware configurations and customer selected options.

(23) 57 TMC CONFIG/OPT
- (a) Displays any TMC self-test faults and TMC interface faults related to parity.
- (b) Checks status of TMC program strapping for engine and airplane configuration, autothrottle, and thrust limit customer selected options.

(24) 59 FCC INSTR
- (a) Displays any FCC self-test faults.
- (b) Sends test routine command and parameters to FCCs, and displays operator interaction message.
- (c) Tests operation of captain's and first officer's MCP flight director switches and the ability of FCCs to move the pitch command bar 10 degrees. Checks each valid FCC separately.

(25) 60 TMC INSTR
- (a) Displays any TMC self-test faults.
- (b) Sends test routine number and parameters to TMC and displays operator action message.
- (c) Tests the ability of the TMC to control the captain's and first officer's EPR instruments target index. EPR target index driven to a test value of 1.50

(26) 64 SPD BK/FLAP XDCR
- (a) Displays any TMC and FCC self-test faults and FLAP XDCR related interface faults.
- (b) Displays a VFY HYD ON message.

(c) Provides interaction messages to check the FCCs sensing of the speedbrakes down, armed, and up.
(d) Displays action messages to set flap lever to 25, 15, 1, and 0.
(e) Tests operation of flap position sensing transducer, and the slat switch position for the FCC and TMC interfaces.

(27) 65 STAB TRIM
(a) Displays any FCC self-test faults and FCC interface faults related to stab trim.
(b) Provides action message VFY HYD ON.
(c) Tests the ability of the FCC to command the Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) to drive the horizontal stabilizer UP and DOWN. It checks the FCC-to-SAM interface.

(28) 66 XDCR OUTPUTS
(a) Displays any FCC and TMC self-test faults, and transducer related interface faults.
(b) Tests and displays the force of horizontal stabilizer, aileron servo, aileron surface, elevator servo, elevator surface, rudder servo, rudder surface, and flaps to the FCC and TMC. It tests and displays the left and right power lever angle position sent to the TMC.

(29) 67 AIL SURF LIMIT
(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.
(b) Provides action messages VFY HYD ON, SET AIL TRIM = 0-ADV, and A/P ENG LCR TO CMD (for valid FCCs only) (LCR = Left, Center, Right).
(c) Drives aileron to its positive limit if operator responds YES/ADV to action message AIL POS LIM?
(d) Drives aileron to its negative limit if operator responds YES/ADV to action message AIL NEG LIM?
(e) Displays aileron surface position in degress.

    AIL SURF DEG
±XX.X ±XX.XX ±XX.XX   where XX.X = decimal number

(30) 68 ELEV SURF LIM
(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.
(b) Provides action messages VFY HYD ON, SET STAB TRIM = 6-ADV, and A/P ENG LCR TO CMD (for valid FCCs only) (LCR = Left, Center, Right).
(c) Drives elevator to its positive limit if operator responds YES/ADV to action message ELEV TO POS LIM?
(d) Displays elevator surface position in degress.

    ELEV SURF DEG
±XX.X ±XX.X ±XX.X   where XX.X = decimal number

(e) Drives elevator to its negative limit if operator responds YES/ADV to action message ELEV TO NEG LIM?

(f) Displays elevator surface position in degress.

ELEV SURF DEG
±XX.X ±XX.X ±XX.X

(31) 69 RUD SURF LIM

(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.

(b) Provides action messages VFY HYD ON, SET RUD TRIM = 0-ADV, and A/P ENG LCR TO CMD (for valid FCCs only) (LCR = Left, Center, Right).

(c) Drives rudder to its positive limit if operator responds YES/ADV to action message RUD TO POS LIM?

(d) Displays rudder surface position in degress.

RUD SURF DEG
±XX.X ±XX.X ±XX.X

(e) Drives rudder to its negative limit if operator responds YES/ADV to action message RUD TO NEG LIM?

(f) Displays rudder surface position in degress.

RUD SURF DEG
±XX.X ±XX.X ±XX.X

C. Fault Message Interpretation

(1) Fault messages displayed by the MCDP are intended to reflect flight deck effects, and identify as near as possible the cause of a fualt. Hardware monitors and software monitors detect unit malfunctions and generate fault data for storage and reporting.

(a) The common mode monitors, used to make sure of the LVDT operation, and the hardware detent comparators are the hardware minotors in the system. All the other monitors are software monitors.

(b) Software monitors test hardware functioning, interfaces, and data reasonableness. The various types of software monitors are:

1) Activity monitors.
2) Wrap-around test monitors.
3) Arinc 429 sign status matrix monitors (SSM).
4) Command/response monitors.

5) Discrete I/O monitors.
6) Detent comparators.
7) Operational monitors.
8) Signal selection fault detection voting circuits (SSFD).

(2) Interface Messages
(a) Interfaces are tested at power up and during ground tests. Messages "NO INFC LRU/LRU" will not be reported as flight faults. Flight faults will report group faults which can be related to a ground test NO INFC message.
(b) If the MCDP generates a NO INFC message for FCC, TMC, or FMC on power up, the MCDP will not interrogate that LRU for flight faults and will not exercise ground tests for that LRU.
(c) The FCC and TMC will report NO INFC messages if an activity monitor fails to detect any change in data received over the interface for specified period of time.

(3) Mode Errors
(a) Mode errors are reported for flight deck effects caused by crew procedures. For most mode errors there is no fault and no need to replace hardware.
1) Typical crew caused mode errors include:
a) Failure to select TRUE heading when flying near the north pole.
b) Attempting to change ILS frequency after capturing glideslope.
c) Failure to disconnect a single channel engaged autopilot prior to flight and touchdown.
d) Pilot input via wheel/column that overrides a single channel engaged autopilot.
(b) Mode errors with diagnostic codes 247 or 143 may be hardware problems and require corrective action.

(4) Sign Status Matrix (SSM) Faults
(a) LRUs transmitting ARINC 429 data provide SSM information in data. The SSM provides status of normal operation, functional test, no computed data, or failure warning.
(b) FCCs, FMCs, and TMC will normally store any SSM other than normal operation in non-volatile memory as fault data. This data can be read in the shop.

(c) SSM data forwarded to the MCDP will normally consist of only failure warn. Three situations can occur that will result in SSM of no computed data to be forwarded to the MCDP.
  1) Loss of data from IRU can result in diagnostic codes 57, 62, 75, 76, and 247 associated with NCD from IRU.
  2) Loss of ILS data at or after capture for glide slope or localize can result in diagnostic codes 199, 200, 202 and 203 associated with NCD from ILS.
  3) Loss of FMC data caused by NCD from IRU can result in diagnostic codes 120 and 121 associated with NCD from FMC.

(5) Signal Selection Fault Detection (SSFD) Faults
  (a) Signal selection fault detection is a process used to identify and eliminate a suspected source of faulty data. SSFD is used when dual or triple data sources are available. Any one data source can be voted out as faulty or if that sources data compares unfavorably with a computed median data value.
  (b) SSFC monitors have a process they go through to not let the autopilot and flight director use the bad data from a defective sensor to calculate the data they will use. SSFD monitors come on when the airspeed is more than 60 knots and an autopilot is in use, or if one or two of the flight directors are on. The monitors continue to compare data values independently of how many autopilot channels are in use. The diagnostic code intermittent bit indicates which of the three FCCs has bad data. This FCC, because of the bit, will not be permitted to have a say as to what data is good or bad. Without this bit, this FCC will possibly have that say and permit a data output from an FCC that has bad data.
  (c) SSFD faults are inhibited on the ground and cannot be duplicated during ground test.
  (d) SSFD faults can be generated for the following conditions.
    1) ILS data fail during ILS capture; diagnostic codes 218, 219.
    2) Localizer or glideslope deviation disagrees with data receiver on cross channel; diagnostic codes 236, 237.
    3) IRU data faulty or missing; diagnostic codes 114 thru 118, 122 thru 125, 132, 215, 216, 230, 231, 238, 240, 241, and 249.
    4) LRRA data faulty; diagnostic codes 176, 221.
    5) Elevator surface position data fault; diagnostic codes 142, 143.
    6) Aileron surface position data fault; diagnostic codes 211, 214.
    7) Rudder surface position data fault; diagnostic codes 212, 217.
    8) Feel computer data fault; diagnostic codes 222, 223.
    9) Stabilizer position data error; diagnostic codes 213, 229.
    10) Flap position data error; diagnostic codes 210, 224.

D. Yaw Damper Module BITE

(1) The yaw damper module BITE consists of a number of monitors that check critical data, and of test routines designed into the module.

(a) Power-up testing occurs when power is applied to the module. Testing consists of fault disengage, servo loop monitor, command coincidence monitor, and memory read/write tests.

(b) Preflight testing occurs when the YDM test switch on the P61 panel is set to L or R. Testing causes the inertial reference system to self test, moves the rudder and checks rudder position, and check IRU inputs.

(c) Continuous inflight testing of data inputs and internal operation is performed by monitor circuits.

(2) Faults detected by the yaw damper monitor circuits are shown on the yaw damper display. The faults are divided into FAULTS NOW and LAST LEG FAULTS. FAULTS NOW are faults that occurred during the last air/ground transition. LAST LEG FAULTS are the faults that occurred before the last air/ground transition. The messages that follow can be shown by the yaw damper module.

(a) NO FAULTS - There are no FAULTS NOW and no LAST LEG FAULTS.

(b) NO FAULT NOW - There are no FAULTS NOW but there are LAST LEG FAULTS.

(c) NO LEG FAULT - There are FAULTS NOW but there are no LAST LEG FAULTS.

(d) FAULTS NOW - There are one or more FAULTS NOW faults, and they are displayed by pressing the DISPLAY button on the faceplate of the yaw damper module.

(e) LAST LEG - There are one or more LAST LEG FAULTS, and they are the DISPLAY button on the faceplate of the yaw damper module.

(f) TEST RUNNING - Power Up Test (PUT) or Pre-Flight Test (PFT) is running on the yaw damper module.

(g) ACCEL - The yaw damper module has determined that there is a fault with the modal suppression accelerometer.

(h) L ADC, R ADC - The yaw damper module has determined that the data from the indicated air data computer is invalid.

(i) ADC DIFF - The yaw damper module has detected an invalid input from an air data computer but is unable to determine which one is at fault.

(j) AIRGND 1, AIRGND 1/2, AIRGND 2 - The yaw damper module has determined that the indicated air/ground discrete is invalid.
(k) AIR DIFF - The yaw damper module has detected an invalid air/ground discrete but is unable to determine which air/ground relay is at fault.
(l) 26 VAC - The yaw damper module has detected the loss of or the invalid level of the 26V RMS reference voltage.
(m) HYD SWITCH - The hydraulic pressure high signal agreed with the hydraulic pressure low signal for more than 2 seconds.
(n) L IRU, C IRU, R IRU - The yaw damper module has determined that the data from the indicated inertial reference unit is invalid.
(o) IRU OFF - The IRUs are not on or no data is transmitted.
(p) IRU NCD - IRUs are on but no data or bad data is transmitted.
(q) IRU DIFF - The yaw damper module has detected an invalid input from an inertial reference unit but is unable to determine which one is at fault.
(r) LVDT - The yaw damper module has detected that the LVDT common mode voltage is low. During the preflight test, an LVDT fault is recorded if the common mode voltage is low and a position test fault is detected (valid high hydraulic pressure must be present).
(s) YD ACT - During normal operation, a yaw damper actuator fault occurs when the yaw damper module detects a servo loop fault. During the preflight test, a yaw damper actuator fault is recorded if a position test fault occurs. To record a YD ACT fault, valid high hydraulic pressure must be present and the LVDT common mode voltage must be normal.
(t) YD TEST - Failure of PFT (Pre-Flight Test).
(u) YDM - A yaw damper module fault is recorded and latched with the following faults: power up test fault, preflight test fault, A/C type code input fault, control or monitor channel command coincidence fault, or a computing channel fault. A YDM fault is also recorded when either control or monitor channel detected auto-disengage relay faults.
    1) If no YDM fault exists, the YDM fault latch resets for the following conditions:
        a) Successful completion of power up test or maintenance reset on ground.
        b) Power up reset, manual engage or YDM autoreset in air or on ground.

E. Troubleshooting Techniques

(1) Autoflight BITE can reduce troubleshooting time and eliminate unnecessary replacement of LRUs if used properly.

(a) Always check flight faults, record all faults and related diagnostic.

(b) If there are no current flight faults, perform ground test 30 and record all faults and associated diagnostic codes.

(c) If no flight fault exists for a particular flight squawk, perform an associated ground test.

(2) Interface faults should be corrected first, and associated ground tests rerun to ensure a faulty LRU is not being masked by another fault that renders the interface inoperative.

(a) If an interface fault cannot be isolated to a specific LRU or wire, redundant LRUs can be swapped and the fault tracked to relate it to an LRU.

(b) The YDM display and other fault indicators not related to autoflight BITE can be checked and individual LRU self-tests run.

(3) LRU faults detected by MCDP ground tests will remain in the MCDP even after the LRU is replaced. It is necessary to exit ground test to clear the faults from the MCDP.

(4) Faults that occur prior to take-off or after landing will not be reported as flight faults. These faults will only be detected and reported by ground tests.

AUTOFLIGHT BITE - MAINTENANCE PRACTICES

1. General
   A. This maintenance practice tells how to use the Maintenance Control Display Panel (MCDP) to review stored flight faults, test, and trouble-shoot.
   B. Ground tests can be done by the MCDP for the individual computers, the line replaceable units (LRUs) and interfaces. The tests are shown on the MCDP front panel. This procedure gives the operation of each test.
   C. Procedures are given to energize the MCDP and do the self-test, to do a check for flight faults, and to do the ground tests.

TASK 22-00-02-712-001

2. Energize the MCDP and Do the Self Test
   A. General
      (1) The MCDP is energized manually on the ground to read flight faults or do ground tests. The MCDP is energized when the MCDP ON/OFF switch is pushed.
      (2) When the MCDP is started, it does a sequence of self-tests. The FAIL light in the ON/OFF switch/light will come on when the MCDP is energized and remain on for five seconds. Also, the FLT FAULTS mode light and the GRD TEST mode light will come on. All segments of the the display will come on. When the FAIL light goes off, the MCDP is prepared for use. The FAIL light will stay on if there is a failure during self-test.
      (3) Fault indications related to self-test failures show after the self-test is completed. These messages are not kept in non-volatile memory.
      (4) The MCDP will go off if power is off for more than 200 milliseconds. The MCDP will stay on if power is off for less than 200 msec. Messages related to not having power are these:

MCDP FAIL RESTART	The MCDP had an error when the power came back on. Open and then close the MCDP circuit breakers to energize the MCDP.
PWR FAIL RST IN PROGRESS	This indication will show if the power goes off and the MCDP starts again during the ground test mode. The indication comes on for three seconds, then the ground test can be done.

   B. Reference
      (1) 24-22-00/201, Electrical Power - Control

C. Access
   (1) Location Zones
       119   Main Equipment Center (Left)

   (2) Access Panels
       119BL   Main Equipment Center

D. Energize the MCDP and Do the Self Test

   S 862-002
   (1) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:

      (a) 11C30, LANDING GEAR POS SYS 1
      (b) GUI 001-114, 116-999;
          11S3, MAINT CONT DSPL
      (c) GUI 115;
          11S6, MAINT CONT DSPL
      (d) 11S15, AIR/GND SYS 1
      (e) 11S19, AIR/GND SYS 2
      (f) 11S23, POS SYS 2

   S 862-003
   (2) Supply electrical power (Ref 24-22-00).

   S 862-004
   (3) Put the MCDP ON/OFF switch to the ON position.
      (a) Make sure the FAIL light, the FLT FAULTS light, and the GRD TEST light come on and then go off.

   S 712-005
   (4) Do the Flight Faults and Ground Test procedures as necessary.

TASK 22-00-02-742-006

3. Flight Faults

A. General
   (1) The Flight Faults Mode gives faults that occurred during a flight. Only the faults that cause a flight deck effect are shown as flight faults. The faults can be read after each flight when the MCDP is manually energized. After the MCDP has been energized and the self-tests done, the MCDP will automatically go into the flight faults mode.

B. Reference
   (1) 24-22-00/201, Electrical Power - Control

C. Access
   (1) Location Zones
       119   Main Equipment Center (Left)
       211/212   Flight Compartment

(2) Access Panel
119BL Main Equipment Center

D. Show the Last Flight Faults

S 712-007
(1) Do the "Energize the MCDP and Do the Self-Test" procedure.

NOTE: Last Flight Faults shows the faults kept by the MCDP the last time the MCDP was automatically turned on at touchdown. If the Flight Faults Mode is engaged on the ground, the MCDP shows "LAST FLT FAULT?". Push the "YES/ADV" button to show a fault that was found during the last flight. Each time the "YES/ADV" switch is pushed, a different fault will show until all of the faults have been shown. The display will then show "ALL LAST FLT FAULTS DSPLY".

S 982-008
(2) Push the NO/SKIP switch to go out of Last Flight Faults.

NOTE: If no faults are found, the message that shows after the YES/ADV switch is pushed will be "NO LAST FLT FAULT".

E. Show the Previous Flight Faults

S 712-009
(1) The Previous Flight Faults shows the flight faults kept from flights before the the last flight. When the MCDP shows the message "ALL LAST FLT FAULT DSPLY" or "NO LAST FLT FAULT", push either YES/ADV or NO/SKIP to view the Previous Flight Faults.

NOTE: When you go into Previous Flight Faults, the message "PREV FLT FAULT?" shows. The MCDP will show faults from the flight before the last the same as with Last Flight Faults messages. The same messages for all previous faults and no previous faults before are shown as applicable. The Previous Flight Faults gives historical flight data. Faults from up to 99 flights are kept in the MCDP unless more than 10 flights have had faults. Faults for a specified flight are kept together and given a number (02 = two flights before, 05 = 5 flights before; etc.). Flight numbers are not used if the flight had no faults.

S 982-010
(2) To show subsequent Previous Flight Faults, push the YES/ADV switch.

TASK 22-00-02-742-011

4. Ground Test

A. General

(1) The Ground Test Mode gives a sequence of different tests to make sure specified parts of the Autopilot/Flight Director System (AFDS) and Thrust Management System (TMS) are satisfactory. Some of the tests are part of the AFDS Adjustment/Test (MM 22-10-00) and the TMS Adjustment/Test (MM 22-32-00). Others tests are specified by different Removal/Installation procedures and Maintenance Practices. The ground tests can also be used to make sure AFDS components are OK before takeoff. After the MCDP has been energized and the self-test done, the MCDP will automatically go into the flight faults mode. Push the GRD TEST switch to go into the Ground Test Mode.

(2) The tests are given by number sequence. The tests can be done by themselves or in a different sequence if the applicable steps to prepare for the test have been done.

(3) The MCDP can be put into GRD TEST MODE after the MCDP has been manually energized. The GRD TEST MODE program tells the FCC's and the TMC to do self-tests. The MCDP tells each computer to to make sure its interfaces are OK. The results of the self-tests and interface tests are kept by the MCDP for future reference. The MCDP makes sure the self-tests and interface test results are OK every time a ground test is done. A computer that is not OK and is necessary to do the specified test is shown. The test can still be done, but not for the channel of the bad computer. The MCDP then shows interface faults kept by the MCDP that effect the specified test. Leave the GRD TEST MODE and then go into GRD TEST MODE again to bring the data up to date. When a GND TEST fault is corrected, you must go to the FLT FAULTS MODE and then go back into the GRD TEST MODE to remove that fault from the MCDP.

(4) For some steps it is necessary for a person to do an operation or reply. This may be to answer a question (YES or NO) or to do or to not do the next step (ADV or SKIP). The step can possibly tell the person to operate a switch or circuit breaker, supply or remove hydraulic power, or monitor a display or effect.

(5) Use the YES/ADV switch to start the test that shows, give a reply to MCDP messages or instructions, to show all of the failures for the test being done, and to continue with a test when a failure stops the test.

(6) The NO/SKIP switch usually causes the MCDP to stop the GRD TEST that is in operation and go to the subsequent GND TEST. The MCDP will not go to the subsequent GND TEST when the messages that follow show on the MCDP.

(a) 30 INFC FAULT? - the MCDP will go to 30 CURRENT OP FAULT? when the NO/SKIP switch is pushed.

(b) 30 VFY THRST LVR SYST CLR-ADV - the MCDP will go to 30 CURRENT OP FAULT? when the NO/SKIP switch is pushed.

(c) 40 A/L BUS ISLN TEST? - the MCDP will go to to 40 VFY ASA BLANK - RST ASA when the NO/SKIP switch is pushed.

(d) 40 A/L SW LIGHTS TEST? - the MCDP will go to 40 A/L SRVO TEST? when the NO/SKIP switch is pushed.
(e) 40 A/L SRVO TEST? - the MCDP will go to advance to 40 A/L A/T TEST? when the NO/SKIP switch is pushed.
(7) When a ground test is complete, the MCDP display will show TEST COMPLETE with the test number.

B. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
(2) Remote MCDP, A22001-22

C. References
(1) 09-11-00/201, Towing
(2) 22-41-01/401, Maintenance Monitor
(3) 24-22-00/201, Electrical Power - Control
(4) 27-21-00/501, Rudder and Rudder Trim Control System
(5) 27-61-00/201, Spoiler/Speedbrake Control System
(6) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(7) 31-41-00/501, Engine Indication and Crew Alerting System (EICAS)
(8) 34-11-00/201, Pitot-Static System
(9) 34-21-00/501, Inertial Reference System
(10) 36-00-00/201, Pneumatic General (Provide/Remove Pneumatic Power)
(11) 76-11-00/501, Engine Control System
(12) 78-31-00/201, Thrust Reverser System (Deactivation/Activation)

D. Access
(1) Location Zones

119	Main Equipment Center (Left)
211/212	Flight Compartment

(2) Access Panel

119BL	Main Equipment Center

E. Prepare to Test (Fig. 201)

S 862-012
(1) Make sure the circuit breakers necessary for the MCDP test to be done are closed before the Ground Test Mode is entered.

S 862-013
(2) Open these P11 panel circuit breakers to prevent incorrect aural warnings:
(a) 11B16, AURAL WARN SPEAKER LEFT
(b) 11H35, AURAL WARN SPEAKER RIGHT

S 862-014
(3) Supply electrical power (Ref 24-22-00).

S 862-015
(4) Do the "Energize the MCDP and Do the Self Test" procedure.

CIRCUIT BREAKER			MCDP GROUND TESTS																												
LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16	30	40	51	52	56	57	59	60	64	65	66	67	68	69
11A2	C595	VOR MKR LEFT	X												NT	NT	NT	X													
11A3	C606	ILS CENTER	X															X	X												
11A7	C622	EFIS DSPL SW LEFT	X															X						X							
11A10	C625	AIR DATA CMPTR LEFT	X	X														X	X												
11A11	C1	AIR DATA AOA SENSOR LEFT	X	X														X	X												
11A12	C2	AIR DATA BARO CORRECT LEFT	X	X														X	X												
11A17	C521	AUTOFLIGHT WARN	X				X											X	X												
11A33	C1201	INDICATOR LIGHTS 2						X	X	X																X	X				
11B2	C1338	ISOL VALVE CONT		X														X			X										
11B3	C1337	ISOL VALVE PWR		X														X			X										
11B17	C573	CLOCK IND LEFT	X															X													
11B18	C566	WARN ELEX B	X															X	X												
11C6	C1538	FLT CONT ELEC 1L AC	X															X	X							X	X	X			
11C7	C1534	FLT CONT ELEC 1L DC	X															X	X							X	X	X			
11C8	C1537	FLT CONT ELEC 2L AC	X															X	X							X	X	X			
11C9	C1533	FLT CONT ELEC 2L DC	X															X	X							X	X	X			
11C14	C1521	FLAP/SLAT ELEC UNIT 2 POWER	X															X								X		X			
11C15	C1541	FLAP/SLAT ELEC UNIT 2 CONT	X															X								X		X			
11C16	C1524	FLAP/SLAT ELEC UNIT 2 SENSOR	X															X								X		X			
11C30	C1175	LANDING GEAR POS SYS 1	X	X	X	X	X	X	X	X	X	X	X	X				X	X	X	X	X	X	X	X	X	X	X	X	X	X
11C34	C1048	FUEL QTY 1																X													
11D6	C826	CAT III BUS ISOL BAT	X															X	X												
11D21	C4113	ENGINE HP BLD VLV LEFT		X														X													
11D22	C4114	ENGINE HP BLD VLV RIGHT		X														X													
11E1	C580	IAS MACH LEFT		X														X													
11E3	C593	ADI LEFT	X	X														X						X							
11E4	C633	EFIS CONT PNL LEFT	X															X						X							
11E6	C588	HSI LEFT	X															X													
11E8	C597	FMCS CDU LEFT	X															X													
11E9	C609	FMCS CMPTR LEFT	X	X														X													
11E10	C603	ILS L	X															X													
11E11	C582	DME LEFT	X															X													
11E16	C516	MODE CONT PNL LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E17	C513	FLT CONT COMPUTER POWER LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E18	C522	FLT CONT COMPUTER SERVO LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E20	C515	FLIGHT CONT COMPTR PWR CENTER	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E21	C524	FLIGHT CONT COMPTR SERVO CENTER	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E22	C581	IAS MACH RIGHT		X														X													
11E24	C594	ADI RIGHT	X															X						X							
11E25	C634	EFIS CONT PNL RIGHT	X															X						X							
11E29	C598	FMCS CDU RIGHT	X															X													
11E30	C610	FMCS CMPTR RIGHT	X	X														X													
11E31	C605	ILS RIGHT	X															X													
11E32	C583	DME RIGHT	X															X													
11E33	C596	VOR RIGHT	X															X													
11E34	C517	MODE CONT PNL RIGHT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E35	C514	FLT CONT CMPTR PWR RIGHT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E36	C523	FLT CONT CMPTR SERVO RIGHT	X	X	X		X	X	X	X		X		X	NT	NT	NT	X	X	X		X		X		X	X	X	X	X	X

MCDP Ground Tests Circuit Breaker Requirements
Figure 201 (Sheet 1)

290504

CIRCUIT BREAKER			MCDP GROUND TESTS																												
LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16	30	40	51	52	56	57	59	60	64	65	66	67	68	69
11F1	C611	IRS LEFT	X	X											NT	NT	NT	X	X												
11F5	C600	RAD ALTM LEFT	X															X	X												
11F8	C637	EFIS SYM GEN LEFT	X	X														X						X							
11F9	C639	EFIS SYM GEN CENTER	X															X						X							
11F14	C501	TMC AC	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F15	C525	TMC DC	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F16	C512	TMC SERVO	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F19	C1525	STAB POS MOD CENTER	X															X	X								X	X			
11F20	C602	RADIO ALTM CENTER	X															X	X												
11F21	C613	IRS CENTER	X															X	X												
11F22	C612	IRS RIGHT	X	X														X	X												
11F24	C623	EFIS DSPL SW RIGHT	X															X						X							
11F26	C601	RADIO ALTM RIGHT	X															X	X												
11F29	C638	EFIS SYM GEN RIGHT	X															X						X							
11F30	C626	AIR DATA CMPTR RIGHT	X	X														X	X												
11F31	C3	AIR DATA AOA SENSOR RIGHT	X															X	X												
11F32	C4	AIR DATA BARO CORRECT RIGHT	X															X	X												
11G10	C1031	RUD RATIO																										RUD			
11G12	C1025	FLAP/SLAT ELEC UNIT 1 POWER		X														X	X							X		X			
11G13	C1539	FLAP/SLAT ELEC UNIT 1 CONT		X														X	X							X		X			
11G14	C1037	FLAP/SLAT ELEC UNIT 1 SENSOR		X														X	X							X		X			
11G15	C1523	STAB POS MOD LEFT	X															X	X								X	X			
11G17	C1536	FLT CONT ELEC 1R AC	X															X	X							X	X	X			
11G18	C1531	FLT CONT ELEC 1R DC	X															X	X							X	X	X			
11G21	C4210	FLAP/SLAT ELEC UNIT 3 POWER	X															X	X							X		X			
11G22	C1540	FLAP/SLAT ELEC UNIT 3 CONT	X															X	X							X		X			
11G23	C1038	FLAP/SLAT ELEC UNIT 3 SENSOR	X															X	X							X		X			
11G24	C1526	STAB POS MOD RIGHT	X															X	X								X	X			
11G27	C1535	FLT CONT ELEC 2R AC	X															X	X							X	X	X			
11G28	C1532	FLT CONT ELEC 2R DC	X															X	X							X	X	X			
11H10	C1002	STAB TRIM LEFT POS IND																									X	X			
11H11	C1017	STAB TRIM LEFT CONT																									X	X			
11H12	C1008	FLAP POS IND LEFT																								X	X	X			
11H13	C1522	FLAP POS IND RIGHT																								X	X	X			
11H19	C1009	STAB TRIM POS IND RIGHT																									X	X			
11H20	C1018	STAB TRIM CONT RIGHT																									X	X			
11J2	C4078	EICAS CMPTR LEFT	X	X	R	X	R	R	R	R	R	R	R	R				X	X	R	R	R	R	R	R	R	R	R	R	R	R
11J3	C4081	EICAS UPPER DISPLAY	X			X												X	X												
11J13	C4101	ELEVATOR POS LEFT																										ELE			
11J14	C4099	AILERON POS LEFT																										AIL			
11J15	C1035	AILERON TRIM													NT	NT	NT											AIL			
11J16	C1005	RUDDER POS																										RUD			

MCDP Ground Tests Circuit Breaker Requirements
Figure 201 (Sheet 2)

CIRCUIT BREAKER			MCDP GROUND TESTS																												
LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16	30	40	51	52	56	57	59	60	64	65	66	67	68	69
11J17	C1034	RUDDER TRIM POS													NT	NT	NT											RUD			
11J22	C4102	ELEV POS RIGHT																										ELE			
11J23	C4100	AILERON POS RIGHT																										AIL			
11J29	C4079	EICAS CMPTR RIGHT	X	X	R	X	R	R	R	R	R	R	R	R				X	X	R	R	R	R	R	R	R	R	R	R	R	R
11J30	C4082	EICAS LOWER DISPLAY	X	R	R	R	R	R	R	R	R	R	R	R				X	X	R	R	R	R	R	R	R	R	R	R	R	R
11J31	C4189	EICAS DISPLAY SW	X	R	R	R	R	R	R	R	R	R	R	R				X	R	R	R	R	R	R	R	R	R	R	R	R	R
11J32	C4094	EICAS DISPLAY SELECT	X	R	R	X	R	R	R	R	R	R	R	R				X	R	R	R	R	R	R	R	R	R	R	R	R	R
11J33	C565	WARN ELEX A	X															X	X												
11J35	C574	CLOCK IND RIGHT	X															X													
11L4	C4119	LEFT ENGINE ELECTRONIC ENGINE																													
		CONTROL LIMITER		X														X													
11L5	C4129	LEFT ENGINE ELECTRONIC ENGINE		X														X													
		CONTROL SUPV																X													
11L9	C4103	LEFT ENGINE EPR XMTR		X														X													
11L19	C1053	FUEL QTY 2	X																												
11L31	C4127	RIGHT ENGINE ELECTRONIC ENGINE																													
		CONTROL LIMITER		X														X													
11L32	C4130	RIGHT ENGINE ELECTRONIC ENGINE																													
		CONTROL SUPV		X														X													
11L36	C4104	RIGHT ENGINE EPR XMTR		X																											
11P2	C1298	L IND LTS 2			X																										
11P29	C1284	R IND LTS 2			X																										
11Q10	C1339	ENG BLEED L		X														X	X		X										
11Q19	C1340	R ENG BLEED		X														X	X		X										
11R3	C824	CAT III BUS ISOL LEFT	X															X	X												
11R30	C825	CAT III BUS ISOL RIGHT	X															X	X												
11S3	C520	MAINT CONT DSPL	X	X	X	X	X	X	X	X	X	X	X	X				X	X	X	X	X	X	X	X	X	X	X	X	X	X
11S15	C1182	AIR/GND SYS 1	X	X	X	X	X	X	X	X	X	X	X	X				X	X	X	X	X	X	X	X	X	X	X	X	X	X
11S19	C1170	AIR/GND SYS 2	X	X	X	X	X	X	X	X	X	X	X	X				X	X	X	X	X	X	X	X	X	X	X	X	X	X
11S23	C4279	POS SYS 2	X		X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
6A3	C811	DC STBY BUS OFF	X															X													
6D3	C614	IRS L	X	X														X	X												
6D4	C621	IRS C	X															X	X												
6D5	C620	IRS R	X	X														X	X												
6G2	C563	CLOCK TIME BASE L	X															X													
6G3	C576	CLOCK TIME BASE R	X															X													
6L11	C813	INV PWR BAT	X															X	X												
6M13	C892	AC STBY BUS OFF	X												NT	NT	NT	X	X												

AIL = NECESSARY TO TEST THE AILERON POSITION
ELE = NECESSARY TO TEST THE ELEVATOR POSITION
PLA = NECESSARY TO TEST THE PLA POSITION
R = NECESSARY WHEN THE REMOTE MCDP CONTROL PANEL IS USED
RUD = NECESSARY TO TEST THE RUDDER POSITION
NT = NO TEST FOR THIS AIRPLANE CONFIGURATION

MCDP Ground Tests Circuit Breaker Requirements
Figure 201 (Sheet 3)

A74429

LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16	30	40	51	52	56	57	59	60	64	65	66	67	68	69
11A2	C595	VOR MKR LEFT	X												NT	NT	NT	X													
11A3	C606	ILS CENTER	X															X	X												
11A10	C625	AIR DATA CMPTR LEFT	X	X														X	X												
11A11	C1	AIR DATA AOA SENSOR LEFT	X	X														X	X												
11A12	C2	AIR DATA BARO CORRECT LEFT	X	X														X	X												
11A17	C521	AUTOFLIGHT WARN	X				X											X	X												
11A33	C1201	INDICATOR LIGHTS 2						X	X	X																X	X				
11B2	C1338	ISOL VALVE CONT		X														X			X										
11B3	C1337	ISOL VALVE PWR		X														X			X										
11B17	C573	CLOCK IND LEFT	X															X													
11B18	C566	WARN ELEX B	X															X	X												
11C4	C622	EFIS DSPL SW LEFT	X															X						X							
11C6	C1538	CSEU 1L AC	X															X	X							X	X	X			
11C7	C1534	CSEU 1L DC	X															X	X							X	X	X			
11C8	C1537	CSEU 2L AC	X															X	X							X	X	X			
11C9	C1533	CSEU 2L DC	X															X	X							X	X	X			
11C14	C1521	FSEU 2 PWR	X															X								X		X			
11C15	C1541	FSEU 2 CONT	X															X								X		X			
11C16	C1524	FSEU 2 SENSOR	X															X								X		X			
11C30	C1175	LANDING GEAR POS SYS 1	X	X	X	X	X	X	X	X	X	X	X	X				X	X	X	X	X	X	X	X	X	X	X	X	X	X
11C34	C1048	FUEL QTY L																X													
11D6	C826	CAT III BUS ISOL	X															X	X												
11D21	C4113	ENGINE HP BLD VLV LEFT		X														X													
11D22	C4114	ENGINE HP BLD VLV RIGHT		X														X													
11E1	C580	IAS MACH LEFT		X														X													
11E3	C593	ADI LEFT	X	X														X						X							
11E4	C633	EFIS CONT PNL LEFT	X															X						X							
11E6	C588	HSI LEFT	X															X													
11E8	C597	FMCS CDU LEFT	X															X													
11E9	C609	FMCS CMPTR LEFT	X	X														X													
11E10	C603	ILS L	X															X													
11E11	C582	DME LEFT	X															X													
11E16	C516	MODE CONT PNL LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E17	C513	FLT CONT COMPUTER POWER LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E18	C522	FLT CONT COMPUTER SERVO LEFT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E20	C515	FLT CONT CMPTR PWR CENTER	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E21	C524	FLT CONT CMPTR SERVO CENTER	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E22	C581	IAS MACH RIGHT		X														X													
11E24	C594	ADI RIGHT	X															X						X							
11E25	C634	EFIS CONT PNL RIGHT	X															X						X							
11E29	C598	FMCS CDU RIGHT	X															X													
11E30	C610	FMCS CMPTR RIGHT	X	X														X													
11E31	C605	ILS RIGHT	X															X													
11E32	C583	DME RIGHT	X															X													
11E33	C596	VOR RIGHT	X															X													
11E34	C517	MODE CONT PNL RIGHT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E35	C514	FLT CONT CMPTR PWR RIGHT	X	X	X		X	X	X	X		X		X				X	X	X		X		X		X	X	X	X	X	X
11E36	C523	FLT CONT CMPTR SERVO RIGHT	X	X	X		X	X	X	X		X		X	NT	NT	NT	X	X	X		X		X		X	X	X	X	X	X

MCDP Ground Tests Circuit Breaker Requirements
Figure 201A (Sheet 1)

A74435

CIRCUIT BREAKER			MCDP GROUND TESTS																												
LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16	30	40	51	52	56	57	59	60	64	65	66	67	68	69
11F1	C611	IRS LEFT	X	X											NT	NT	NT	X	X												
11F5	C600	RADIO ALTM LEFT	X															X	X												
11F8	C637	EFIS SYM GEN LEFT	X	X														X						X							
11F9	C639	EFIS SYM GEN CENTER	X															X						X							
11F10	C4263	EFIS INSTR COMPARATOR	X															X						X							
11F14	C501	TMC AC	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F15	C525	TMC DC	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F16	C512	TMC SERVO	X	X	X	X					X		X	X				X	X		X		X		X			PLA			
11F19	C1525	STAB POS MOD CENTER	X															X	X								X	X			
11F20	C602	RAD ALTM CENTER	X															X	X												
11F21	C613	IRS CENTER	X															X	X												
11F22	C612	IRS RIGHT	X	X														X	X												
11F24	C623	EFIS DSPL SW RIGHT	X															X						X							
11F26	C601	RADIO ALTM RIGHT	X															X	X												
11F29	C638	EFIS SYM GEN RIGHT	X															X						X							
11F30	C626	AIR DATA CMPTR RIGHT	X	X														X	X												
11F31	C3	AIR DATA AOA SENSOR RIGHT	X															X	X												
11F32	C4	AIR DATA BARO CORRECT RIGHT	X															X	X												
11G10	C1031	RUD RATIO																										RUD			
11G12	C1025	FSEU 1 PWR		X														X	X							X		X			
11G13	C1539	FSEU 1 CONT		X														X	X							X		X			
11G14	C1037	FSEU 1 SENSOR		X														X	X							X		X			
11G15	C1523	STAB POS MOD LEFT	X															X	X								X	X			
11G17	C1536	CSEU 1R AC	X															X	X							X	X	X			
11G18	C1531	CSEU 1R DC	X															X	X							X	X	X			
11G21	C4210	FSEU 3 PWR	X															X	X							X		X			
11G22	C1540	FSEU 3 CONT	X															X	X							X		X			
11G23	C1038	FSEU 3 SENSOR	X															X	X							X		X			
11G24	C1526	STAB POS MOD RIGHT	X															X	X								X	X			
11G27	C1535	CSEU 2R AC	X															X	X							X	X	X			
11G28	C1532	CSEU 2R DC	X															X	X							X	X	X			
11H10	C1002	STAB TRIM POS IND L																									X	X			
11H11	C1017	STAB TRIM CONT L																									X	X			
11H12	C1008	FLAP POS IND LEFT																								X	X	X			
11H13	C1522	FLAP POS IND RIGHT																								X	X	X			
11H19	C1009	STAB TRIM POS IND RIGHT																									X	X			
11H20	C1018	STAB TRIM CONT RIGHT																									X	X			
11J2	C4078	EICAS CMPTR LEFT	X	X	R	X	R	R	R	R	R	R	R	R				X	X	R	R	R	R	R	R	R	R	R	R	R	R
11J3	C4081	EICAS UPPER IND	X			X												X	X												
11J13	C4101	ELEVATOR POS LEFT																										ELE			
11J14	C4099	AILERON POS LEFT																										AIL			
11J15	C1035	AILERON TRIM																										AIL			
11J16	C1005	RUDDER POS													NT	NT	NT											RUD			

MCDP Ground Tests Circuit Breaker Requirements
Figure 201A (Sheet 2)

A74456

CIRCUIT BREAKER			MCDP GROUND TESTS																																			
LOC	CB	NOMENCLATURE	01	02	04	05	06	07	08	09	10	11	12	13	14	15	16			30	40			51	52	56	57	59	60	64	65	66	67	68	69			
11J17	C1034	RUDDER TRIM POS													NT	NT	NT															RUD						
11J22	C4102	ELEV POS RIGHT																														ELE						
11J23	C4100	AILERON POS RIGHT																														AIL						
11J29	C4079	EICAS CMPTR RIGHT	X	X	R	X	R	R	R	R	R	R	R	R						X	X			R	R	R	R	R	R	R	R	R	R	R	R			
11J30	C4082	EICAS LOWER IND	X	R	R	R	R	R	R	R	R	R	R	R						X	X			R	R	R	R	R	R	R	R	R	R	R	R			
11J31	C4189	EICAS DSPL SW	X	R	R	R	R	R	R	R	R	R	R	R						X	R			R	R	R	R	R	R	R	R	R	R	R	R			
11J32	C4094	EICAS PILOTS DSP	X	R	R	X	R	R	R	R	R	R	R	R						X	R			R	R	R	R	R	R	R	R	R	R	R	R			
11J33	C565	WARN ELEX A	X																	X	X																	
11J35	C574	CLOCK IND RIGHT	X																	X																		
11L4	C4119	LEFT ENGINE ELECTRONIC ENGINE CONTROL LIMITER		X																X																		
11L5	C4129	LEFT ENGINE ELECTRONIC ENGINE CONTROL SUPV		X																X																		
11L9	C4103	LEFT ENGINE EPR XMTR		X																X																		
11L19	C1053	FUEL QTY R	X																	X																		
11L31	C4127	RIGHT ENGINE ELECTRONIC ENGINE CONTROL LIMITER		X																X																		
11L32	C4130	RIGHT ENGINE ELECTRONIC ENGINE CONTROL SUPV		X																X																		
11L36	C4104	RIGHT ENGINE EPR XMTR		X																																		
11P2	C1298	L IND LTS 2			X																																	
11P29	C1284	R IND LTS 2			X																																	
11Q10	C1339	ENG BLEED L		X																X					X													
11Q19	C1340	R ENG BLEED CONT		X																X					X													
11R3	C824	CAT III BUS ISOL LEFT	X																	X	X																	
11R30	C825	CAT III BUS ISOL RIGHT	X																	X	X																	
11S6	C520	MAINT CONT DSPL	X	X	X	X	X	X	X	X	X	X	X	X						X	X			X	X	X	X	X	X	X	X	X	X	X	X			
11S15	C1182	AIR/GND SYS 1	X	X	X	X	X	X	X	X	X	X	X	X						X	X			X	X	X	X	X	X	X	X	X	X	X	X			
11S19	C1170	AIR/GND SYS 2	X	X	X	X	X	X	X	X	X	X	X	X						X	X			X	X	X	X	X	X	X	X	X	X	X	X			
11S23	C4279	POS SYS 2	X		X		X	X	X	X		X		X						X	X			X		X		X		X	X	X	X	X	X			
6A3	C811	DC STBY BUS OFF	X																	X	X																	
6D3	C614	IRS L	X	X																X	X																	
6D4	C621	IRS C	X																	X	X																	
6D5	C620	IRS R	X	X																X	X																	
6G2	C563	CLOCK TIME BASE L	X																	X																		
6G3	C576	CLOCK TIME BASE R	X																	X																		
6L11	C813	INV PWR BAT	X																	X	X																	
6M13	C892	AC STBY BUS OFF	X												NT	NT	NT			X	X																	

AIL = NECESSARY TO TEST THE AILERON POSITION
ELE = NECESSARY TO TEST THE ELEVATOR POSITION
PLA = NECESSARY TO TEST THE PLA POSITION
R = NECESSARY WHEN THE REMOTE MCDP CONTROL PANEL IS USED
RUD = NECESSARY TO TEST THE RUDDER POSITION
NT = NO TEST FOR THIS AIRPLANE CONFIGURATION

MCDP Ground Tests Circuit Breaker Requirements
Figure 201A (Sheet 3)

A74458

S 482-016

(5) If the remote MCDP is used, make sure the MCDP self-test (above) is done on the main MCDP and then do these steps.

(a) Install the remote MCDP panel.

(b) Close the EICAS circuit breakers (6 each) on the P11 panel (Ref 31-41-00).

(c) On the EICAS maintenance panel, P61, push the CONF/MCDP switch to see the MCDP display on the EICAS bottom display unit.

NOTE: The messages that show on the MCDP in the main equipment center will also show on the EICAS bottom display unit. The indication MCDP FLT FAULTS or MCDP GRD TEST shows to identify which mode the MCDP is in.

(d) Make sure the EICAS bottom display unit shows MCDP OFF.

(e) Put the MCDP remote panel ON/OFF switch in the ON position.

(f) Push the GRD TEST switch.

F. Supply Hydraulic Power (Necessary for Ground Tests 07, 08, 09, 30, 40, 64, 65, 66, 67, 68, 69)

NOTE: Hydraulic power is necessary to test the engage and disengage function of the Autopilot/Flight Director System during MCDP Ground Test 30.

S 862-017

(1) Supply power by either the airplane electrically driven pumps or the ground hydraulic carts.

S 202-018

(2) Do not activate hydraulic equipment other than those necessary by these tests. This makes sure there is sufficient power when the airplane electrically driven hydraulic pumps are used.

S 862-019

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Supply power to the left, the right, and the center hydraulic systems (Ref 29-11-00) when the MCDP indication VFY HYD ON shows, unless told differently by the procedure. Push the YES/ADV switch to show the hydraulics are on.

G. 01 FCC

S 862-020

(1) Do the "Prepare to Test" steps.

S 862-021

(2) If the test is to be done after an LRU replacement or a fault correction, make sure the MCDP is put out of the GRD TEST MODE. Go into the FLT FAULTS MODE or set the MCDP to OFF and then go into the GRD TEST MODE again.

NOTE: Before MCDP Test 01-FCC is done, put the MCDP out of the GRD TEST MODE. This will remove faults that are corrected from the MCDP memory.

S 742-022

(3) Do MCDP test 01.

NOTE: The MCDP shows FCC # FAIL if the FCC had a self-test failure when it went into the GRD TEST MODE. The MCDP shows FCC # interface faults if FCC the passed the self-test. The MCDP does not test LRUs or input signals. Each FCC is tested independently as set by the MCDP operator.

S 862-023

(4) Put the airplane back to its usual condition if no other ground tests are necessary.

H. 02 TMC

S 862-024

WARNING: DO SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (MM 27-61-00). IT IS NECESSARY TO MOVE THE THRUST LEVERS DURING THIS TEST WHICH CAN CAUSE SPEEDBRAKE MOVEMENT IF HYDRAULIC POWER IS ON. THIS CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT.

(1) Do Spoiler/Speedbrake Deactivation procedure (Ref 27-61-00).

S 862-183

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(2) Make sure the engines are not in operation.

S 862-141

(3) Do the "Prepare to Test" steps.

S 862-025

(4) If the test is to be done after an LRU replacement or a fault correction, make sure the MCDP is put out of GRD TEST MODE. Go into the FLT FAULTS MODE or set the MCDP to OFF and then go into the GRD TEST MODE again.

NOTE: Before MCDP Test 02-TMC is done, put the MCDP out of the GRD TEST MODE. This will remove faults that are corrected from the MCDP memory.

S 862-029

(5) On the P5 panel, set the L and R ENG LIMITER switches on the ENGINE START/RAM AIR TURB control panel to ON.

S 862-040

(6) On the P5 panel, set the L and R ELEC ENG CONTROL switches to ON.

S 742-035

(7) Set the left and right EEC GND TEST switches to the up (ON) position. This will supply power to the EEC. The EEC GND TEST switch is below each EEC unit in the main equipment center.

NOTE: If you do not supply power to the EEC before you set the MCDP ground test mode, these messages will show during the test:

NO INFC TMC EEC L BUS IN
NO INFC TMC EEC R BUS IN

S 742-036
(8) Do MCDP Test 02

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into the GRD TEST MODE. If no TMC message shows, the MCDP shows messages to clear the thrust lever system and set the A/T switch to ARM. The MCDP then shows TMC interface faults related to the autothrottle servo. The TMC test does not do a check of the LRUs or the input signals.

S 862-290
(9) Put the airplane back to its usual condition if no other ground tests are necessary.

I. 04 MCP

S 862-038
(1) Do the "Prepare to Test" steps.

S 862-278
(2) On the MCP, set switches as follows:
(a) Set the A/T ARM switch to OFF.
(b) GUI 115;
Set the A/P ENGAGE switches to OFF.
(c) GUI 001-114, 116-999;
Set the DISENGAGE BAR to the UP position.
(d) Set the two F/D switches to OFF.

S 862-187
(3) Do the MCDP test 04.

NOTE: The MCDP shows FCC # and TMC FAIL for the computers that had a self-test fialure when it went into the GRD TEST MODE. If no TMC or FCC messages show, the MCDP shows FCC and TMC interface faults related to the MCP for the FCC's and the TMC that passed the self-test. A manual test is then done to make sure switches, switch/lights, controls, and displays on the MCP are OK. It is necessary for a person to operate controls during this test.

S 862-201
(4) Put the airplane back to its usual condition if no other ground tests are necessary.

J. 05 TMSP

S 802-042
(1) Do the "Prepare to Test" steps.

S 742-043
(2) Do MCDP test 05.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC message shows, the MCDP shows TMC interface faults related to the TMSP. A test is done manually to make sure the TMSP switches and the temperature select control are OK. The switch indications are shown on the top EICAS display during the 05 test.

S 862-044
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

K. 06 ASA

S 862-045
(1) Do the "Prepare to Test" steps.

S 742-046
(2) Do MCDP test 06.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows stored FCC interface faults related to the ASA. A person operates controls to do a check of the ASA reset function.

S 862-291
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

L. 07 SERVO AIL

S 862-279
(1) Do the "Prepare to Test" steps.

S 862-049

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply Hydraulic Power" steps.

S 862-289
(3) Set the Flap handle to the zero detent position.

S 742-051
(4) Do MCDP test 07.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the aileron servos. The MCDP then tells the person to make sure hydraulic power is supplied. Other messages tell the person to set the aileron trim to zero, then engage the A/P in the command mode. After the servos engage, the MCDP tells the FCCs to move the servos to adjust the control wheel to center, then turn it clockwise, then counter-clockwise, and then back to center.

S 862-052
(5) Put the airplane back to its usual condition if no other ground tests are necessary.

M. 08 SERVO ELEV

S 862-184
(1) Do the "Prepare to Test" steps.

S 862-185

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply Hydraulic Power" steps.

S 862-053
(3) Set the stabilizer to 6 units of trim.

S 742-054
(4) Do MCDP test 08.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the elevator servos. The MCDP then shows a message to make sure hydraulic power is supplied. Other messages tell the person to set the STAB TRIM to 6 units, and set the ELEV FEEL pitot-static pressure to zero. The MCDP then tells the person to engage the A/P in command. After the servos engage, the MCDP tells the FCCs to move the servos to adjust the control column to center and then move it forward, aft, and back to center.

S 862-292
(5) Put the airplane back to its usual condition if no other ground tests are necessary.

N. 09 SERVO RUD

S 862-280
(1) Do the "Prepare to Test" steps.

S 982-057

WARNING: LOCK THE NOSE GEAR STEERING WHEN RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Move the towing lever on the nose gear metering valve module to the TOW POSITION.

S 432-186
(3) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 862-281
(4) Do the "Supply Hydraulic Power" steps.

S 742-059
(5) Do MCDP test 09.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the rudder servos. The MCDP then shows a message to make sure the hydraulic power is supplied. Other messages tell the person to set the rudder trim to zero. The MCDP tells the FCCs to move the servos to adjust the rudder to center and then move it side to side. A person will monitor the movement of the rudder pedals.

S 862-293
(6) Put the airplane back to its usual condition if no other ground tests are necessary.

O. 10 SERVO A/T

S 862-200

WARNING: DO SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (MM 27-61-00). IT IS NECESSARY TO MOVE THE THRUST LEVERS DURING THIS TEST WHICH CAN CAUSE SPEEDBRAKE MOVEMENT IF HYDRAULIC POWER IS ON. THIS CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT.

(1) Do Spoiler/Speedbrake Deactivation procedure (Ref 27-61-00).

S 752-188

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(2) Make sure the engines are not in operation.

S 862-188
(3) Do the "Prepare to Test" steps.

S 862-068
(4) Set the A/T switch on the MCP to OFF.

S 742-069
(5) Do MCDP test 10.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. The MCDP then shows messages to make sure the thrust levers and cables are clear, and then to set the A/T switch on the MCP to ARM. Data sent to the TMC by the MCDP initializes the throttles to the aft position. Then they move forward and then back to the aft position.

S 862-294
(6) Put the airplane back to its usual condition if no other ground tests are necessary.

P. 11 SW A/P DISC

S 862-282
(1) Do the "Prepare to Test" steps.

S 742-073
(2) Do MCDP test 11.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the autopilot disengage switch. The MCDP then tells the person to push and hold, then release, each (Capt, F/O) A/P disengage switch. The switches are monitored for the A/P disengage command operation.

S 862-295
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

Q. 12 SW A/T DISC

S 862-075

(1) Do the "Prepare to Test" steps.

S 742-076

(2) Do MCDP test 12.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC FAIL message shows, the MDCP shows TMC interface faults related to the autothrottle disengage switch. The MCDP tells the person to push and hold, then release the two (L, R) autothrottle disengage switches. The switches are monitored for autothrottle disengage command operation.

S 862-077

(3) Put the airplane back to its usual condition if no other ground tests are necessary.

R. 13 SW G/A

S 862-078

(1) Do the "Prepare to Test" steps.

S 742-079

(2) Do MCDP test 13.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC and TMC interface faults related to the go-around switch. The MCDP then tells the person to push and hold, then release, the two (Capt, F/O) go-around switches. The FCCs and the TMC monitor the G/A switch command operation.

S 862-080

(3) Put the airplane back to its usual condition if no other ground tests are necessary.

S. 14 XDCR COL L

S 742-242

(1) This test is not used.

T. 15 XDCR COL R

S 742-244

(1) This test is not used.

U. 16 XDCR WHL

S 742-248

(1) This test is not used.

V. 17 PVD/PVDC

S 742-252

(1) This test is not used.

W. 30 CURRENT FAULT REPORT

S 862-094

(1) Align the Left, the Center, and the Right Inertial Reference Systems in the NAV MODE (Ref 34-21-00).

NOTE: It takes 10 minutes to initialize the IRS. The MCDP self-test can be done during the IRS initialization.

S 862-051

WARNING: DO SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (MM 27-61-00). IT IS NECESSARY TO MOVE THE THRUST LEVERS DURING THIS TEST WHICH CAN CAUSE SPEEDBRAKE MOVEMENT IF HYDRAULIC POWER IS ON. THIS CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT .

(2) Do Spoiler/Speedbrake Deactivation procedure (Ref 27-61-00).

S 862-314

WARNING: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(3) Make sure the engines are not in operation.

S 862-287

(4) Do the "Prepare to Test" steps.

S 862-095

(5) Set the A/T switch on MCP to OFF.

S 862-097
(6) On the P5 panel, set the L and the R ENG LIMITER switches on the ENGINE START/RAM AIR TURB control panel to ON.

S 862-320
(7) On the P5 panel, set the L and R ELEC ENG CONTROL switches to ON.

S 862-101
(8) Set the left and right EEC GND TEST switches to the up (ON) position. This will supply power to the EEC. The EEC GND TEST switch is below each EEC unit in the main equipment center.

NOTE: If you do not supply power to the EEC before you set the MCDP ground test mode, these messages will show during the test:

NO INFC TMC EEC L BUS IN
NO INFC TMC EEC R BUS IN

NOTE: Because the engines are not in operation, these messages can show:

CURRENT FAULT EEC L
CURRENT FAULT EEC R

S 742-104
(9) Do MCDP test 30.
(a) The MCDP shows FCC (L, C, R) FAIL and TMC FAIL for a FCC and TMC that had a self-test failure when it went into GRD TEST MODE.
(b) The MCDP shows "30 INFC FAULT?".
(c) Push YES/ADV to see interface faults or push NO/SKIP to go to "30 CURRENT OP FAULT?".
(d) The MCDP shows "30 CURRENT OP FAULT?".
(e) If a check of autopilot engage problems is to be done, do these steps before YES/ADV is pushed:
1) Do the "Supply Hydraulic Power" steps.
2) Engage the applicable autopilot channel.

3) Wait 20-30 seconds for the autopilot to automatically disengages or manually disengage the autopilot.
4) Push the YES/ADV switch.

NOTE: The autopilot must be disengaged before YES/ADV is pushed to go to "30 CURRENT OP FAULT?". If the MCDP goes to CURRENT OP FAULT with the autopilot engaged, the MCDP will show SRVO ON - NO GRD TEST FCC (L, C, R).

If SRVO ON - NO GRD TEST is shown, set FLT FAULTS MODE, disengage the autopilot, and then set GRD TEST MODE.

(f) When the MCDP goes to CURRENT OP FAULT, it will show all current op faults. Push the GRD TEST MODE switch to show the diagnostic codes.

S 862-105

(10) Put the airplane back to its usual condition if no other ground tests are necessary.

NOTE: Put the MCDP out of ground test mode before MCDP ground test 02 is done or MCDP ground test 30 is done again. If you do not do this, many "NO INFC TMC" messages can show. Set POWER switch on EEC Maintenance Panle (P-61) to NORMAL, and safety wire switch.

X. 40 AUTOLAND

S 862-181

WARNING: DO SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (MM 27-61-00). IT IS NECESSARY TO MOVE THE THRUST LEVERS DURING THIS TEST WHICH CAN CAUSE SPEEDBRAKE MOVEMENT IF HYDRAULIC POWER IS ON. THIS CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT.

(1) Do Spoiler/Speedbrake Deactivation procedure (Ref 27-61-00).

S 862-076

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(2) Make sure the engines are not in operation.

S 862-189

(3) Do the "Prepare to Test" steps.

S 862-107

WARNING: LOCK THE NOSE GEAR STEERING WHEN RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Move the towing lever on the nose gear metering valve module to the TOW POSITION.

S 862-108
(5) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 862-106

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) Do the "Supply Hydraulic Power" steps.

NOTE: Do not supply hydraulic power until the MCDP message 40 VFY HYD-ON-ADV shows.

S 862-202
(7) On the P10 panel, set the C and R STAB TRIM CUTOUT switches to NORM.

S 862-182
(8) On the MCP, set switches as follows:
(a) Set the A/T ARM Switch to ARM.
(b) GUI 115;
Set the A/P ENGAGE Switches to OFF.
(c) GUI 001-114, 116-999;
Set the DISENGAGE BAR to the UP position.

S 742-110

(9) Do MCDP test 40.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to autoland. The ILS Tune and RA Test inhibit functions are tested for a "not inhibited" condition and for an "inhibited" condition when inhibit is asked for. Fault messages for either failed condition will show. If the FCC R/MCDP interface is inactive, the MCDP shows that the bus isolation test cannot be done. The MCDP then tells the person to make sure the standby power is in the AUTO position, the battery switch is on, and the left bus tie is in the AUTO position. An advisory message shows that the EICAS display will not be on during the bus isolation test and the person pushes the YES/ADV switch. The remaining tests use parts of other ground tests to do a check of the autoland related functions. Autoland autothrottle test is done after the autoland servo test. After the start of the autoland autothrottle test, the MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC FAIL message shows, the MCDP shows all TMC interface faults. The MCDP then does the autothrottle servo test and the go-around switch test.

S 862-111

(10) Do these steps to clear the EICAS maintenance message(s) "STAB TRIM" and/or YAW DAMPER" that can show after the Autoland Bus Isolation Test is done:

WARNING: BE CAREFUL WHEN YOU OPEN THESE CIRCUIT BREAKERS. ACCIDENTAL SPOILER MOVEMENT CAN OCCUR WHEN THESE CIRCUIT BREAKERS ARE OPENED. INJURY TO PERSONS CAN OCCUR.

(a) Open these circuit breakers on the P11 panel and then close them in less than 25 seconds:
1) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
2) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
3) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
4) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

(b) If a faultball stays set to yellow on a Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) and/or a Yaw Damper Module (YDM) after the above circuit breakers have been opened and closed, push the reset switch on the front of the SAM.

S 862-112

(11) Put the airplane back to its usual condition if no other ground tests are necessary.

Y. 51 AIR/GRD RLY

S 862-113

(1) Do the "Prepare to Test" steps.

S 862-114

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS AND DAMAGE TO EQUIPMENT

(2) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoilers.

S 742-115

(3) Do MCDP test 51.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP then tells a person to to open and close the air/ground circuit breakers to test the FCC and the TMC reply to air/ground relay operation. MCDP messages C/B LAND GEAR SYST 1 and C/B LAND GEAR SYST 2 refer to 11S15, AIR/GND SYS 1, and 11S19, AIR/GND SYS 2, 11C19, LANDING GEAR POS SYS 2 ALTN, C/Bs on panel P11.

S 862-115

(4) Put the airplane back to its usual condition if no other ground tests are necessary.

Z. 52 TMC RLY/SW

S 842-178

(1) Test 52 can be done by one of these two procedures.

(a) Procedure 1 uses the TEST HI and TEST LO switches on the left (M1033) and right M10312 ECS cards in the P50 panel to supply the high and low inputs. This procedure does a check of the ECS cards and their outputs to the TMC without pneumatic power.

(b) Procedure 2 uses the control switches on the overhead P5 panel to supply the high and low inputs to the ECS cards. This procedure does a check of the complete ECS system and needs pneumatic power.

S 862-116
(2) Open these circuit breakers on the P11 panel:
    (a) 11G11, AUTO SPEED BRAKE
    (b) 11S21, AUTO BRAKE

S 862-117
(3) Do the "Prepare to Test" steps.

S 862-118
(4) Set these switches on the P5 panel:
    (a) Set the L and R PACK switch/lights to OFF.
    (b) Set the WING and L and R ENGINE switch/lights on the ANTI-ICE panel to OFF.
    (c) Set the BLEED AIR panel ISOLATION switch/light to the open position (the switch is pushed in).
    (d) Set the BLEED AIR panel L ENG and R ENG switch/lights to OFF (the switch is pushed in).

S 862-119
(5) Make sure the reverse thrust levers are set at idle and in the forward thrust position.

S 862-120

WARNING: DO THE THRUST REVERSER ISOLATION VALVE DEACTIVATION PROCEDURE (AMM 78-31-00). FAILURE TO DEACTIVATE THE THRUST REVERSER ISOLATION VALVE CAN CAUSE ACCIDENTAL THRUST REVERSER OPERATION. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Deactivate the thrust reverser isolation valve for ground maintenance (AMM 78-31-00).

S 862-123
(7) Supply pneumatic power (AMM 36-00-00/201) when Procedure 2 is used.

S 742-124
(8) Do MCDP test 52.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GND TEST MODE. If no TMC FAIL message shows, MCDP shows TMC interface faults related to the ECS relay. These test steps make sure all TMC analog input discrete signal paths are OK.

TEST 52 - TMC RLY/SW - PROCEDURE 1		
TEST STEP	MCDP INTERACTION MESSAGE	OPERATION ACTION NOTE: PUSH AND HOLD THE TEST SWITCH UNTIL MCDP MOVES TO THE NEXT TEST SWITCH. IF THE MCDP TEST DOES NOT MOVE REFER TO 22-00-04, FIG. 102
1	52 SET L PACK SW ON	PUSH THE TEST HIGH SWITCH ON THE L ECS CARD M10313.
2	52 SET L PACK SW OFF	RELEASE THE TEST HIGH SWITCH AND PUSH THE TEST LOW SWITCH ON THE L ECS CARD M10313.
3	52 SET R PACK SW ON	RELEASE THE L ECS CARD M10313 TEST LOW SWITCH AND PUSH THE TEST HIGH SWITCH ON THE R ECS CARD M10312.
4	52 SET R PACK SW OFF	RELEASE THE TEST HIGH SWITCH AND PUSH THE TEST LOW SWITCH ON R ECS CARD M10312.
5	52 ECS PACK L/R LO LO	RELEASE THE R ECS CARD M10312 TEST LOW SWITCH AND PUSH THE TEST HIGH SWITCHES ON THE L AND THE R ECS CARDS (M10313, M10312).
6	52 ECS PACK L/R HI HI	RELEASE THE TEST HIGH SWITCHES AND PUSH THE TEST LOW SWITCH ON THE L AND THE R ECS CARD (M10313, M10312).
7	52 ECS PACK L/R LO LO	RELEASE THE TEST SWITCHES ON THE L AND R ECS CARDS (M10313, M10312) AND PUSH THE MCDP YES/ADV SWITCH
8	52 PUSH WG AI ON	PUSH THE TEST HIGH SWITCH ON THE L ECS CARD M10313.
9	52 PUSH ALL AI SW OFF	RELEASE THE TEST HIGH SWITCH AND PUSH THE TEST LOW SWITCH ON THE L ECS CARD M10313.
10	52 COWL AI L/R OFF OFF	RELEASE THE TEST LOW SWITCH ON THE L ECS CARD M10313 AND PUSH THE TEST HIGH SWITCH ON THE L AND R ECS CARD (M10313, M10312)
11	52 COWL AI L/R ON ON	RELEASE THE TEST HIGH SWITCHES AND PUSH THE TEST LOW SWITCHES ON THE L AND R ECS CARDS (M10313, M10312).

TEST 52 - TMC RLY/SW - PROCEDURE 1		
TEST STEP	MCDP INTERACTION MESSAGE	OPERATION ACTION NOTE: PUSH AND HOLD THE TEST SWITCH UNTIL MCDP MOVES TO THE NEXT TEST SWITCH. IF THE MCDP TEST DOES NOT MOVE REFER TO 22-00-04, FIG. 102
12	52 COWL AI L/R OFF OFF	RELEASE THE TEST LOW SWITCHES ON THE L AND R ECS CARDS (M10313, M10312). PUSH THE MCDP YES/ADV SWITCH.
13	52 PUSH SOV SW L CLOSE	PUSH THE TEST LOW SWITCH ON THE L ECS CARD M10313.
14	52 PUSH SOV SW L OPEN	RELEASE THE TEST LOW SWITCH AND THE PUSH TEST HIGH SWITCH ON THE L ECS CARD M10313.
15	52 PUSH SOV SW R CLOSE	RELEASE THE TEST HIGH SWITCH ON THE L ECS CARD M10313. PUSH THE TEST LOW SWITCH ON THE R ECS CARD M10312.
16	52 PUSH SOV SW R OPEN	RELEASE THE TEST LOW SWITCH AND PUSH THE TEST HIGH SWITCH ON THE R ECS CARD M10312.
17	52 PUSH ISLN VLV SW L CLOSE	RELEASE THE TEST HIGH SWITCH ON THE R ECS CARD M10312. PUSH THE TEST LOW SWITCH ON THE L ECS CARD M10313.
18	52 PUSH ISLN VLV SW L OPEN	RELEASE THE TEST LOW SWITCH AND PUSH THE TEST HIGH SWITCH ON THE L ECS CARD M10313.
19	52 VFY THRST LVR SYST CLR - ADV	RELEASE THE TEST HIGH SWITCH ON THE L ECS CARD M10313. MAKE SURE THE THRUST LEVER SYSTEM IS CLEAR FOR OPERATION AND THE THRUST REVERSERS ARE DEACTIVATED (AMM 78-31-00). PUSH THE MCDP YES/ADV SWITCH WHEN THE THRUST REVERSER SYSTEM IS CLEAR FOR OPERATION.
		NOTE: IN THESE STEPS THE THRUST REVERSER LEVERS ARE USED TO ACTIVATE THE THRUST REVERSER SWITCHES IN THE AUTOTHROTTLE MICROSWITCH PACK ASSEMBLY (M966).

TEST 52 - TMC RLY/SW - PROCEDURE 1		
TEST STEP	MCDP INTERACTION MESSAGE	OPERATION ACTION NOTE: PUSH AND HOLD THE TEST SWITCH UNTIL MCDP MOVES TO THE NEXT TEST SWITCH. IF THE MCDP TEST DOES NOT MOVE REFER TO 22-00-04, FIG. 102
20	52 SET L REV THRST LVR - ON	1. PUT THE THROTTLES AGAINST THE AFT MECHANICAL STOPS. 2. MOVE THE LEFT REVERSER THRUST LEVER UP AND AFT TO THE REVERSE THRUST INTERLOCK POSITION.
21	52 SET L REV THRST LVR - OFF	MOVE THE LEFT THRUST REVERSE LEVER TO THE FORWARD THRUST POSITION (REVERSERS STOWED).
22	52 SET R REV THRST LVR - ON	1. PUT THE THROTTLES AGAINST THE AFT MECHANICAL STOPS. 2. MOVE THE RIGHT REVERSER THRUST LEVER UP AFT TO THE REVERSE THRUST INTERLOCK POSITION.
23	52 SET L REV THRST LVR - OFF	MOVE THE RIGHT THRUST REVERSER LEVER TO THE FORWARD THRUST POSITION (REVERSER STOWED).
	52 TEST COMPLETE	PUSH THE MCDP YES/ADV OR THE NO/SKIP SWITCH TO GET OUT OF THE TEST AND ARM TEST 56.

TEST 52 - TMC RLY/SW - PROCEDURE 2		
TEST STEP	MCDP INTERACTION INDICATION	OPERATION ACTION NOTE: IF THE MCDP TEST DOES NOT MOVE FORWARD REFER TO 22-00-04, FIG. 102
1	52 SET L PACK SW ON	TURN THE L PACK SWITCH ON THE AIR CONDITIONING CONTROL PANEL M10193 ON THE OVERHEAD P5 PANEL TO THE AUTO OR STBY POSITION. NOTE: THIS TEST AUTOMATICALLY MOVES TO THE NEXT STEP IF THE L PACK SWITCH IS IN AUTO OR STBY POSITION.
2 2	52 SET L PACK 52 SET L PACK SW OFF	TURN THE L PACK SWITCH ON THE AIR CONDITIONING CONTROL PANEL M10193 TO THE OFF POSITION.
3	52 SET R PACK SW ON	TURN THE R PACK SWITCH ON THE AIR CONDITIONING CONTROL PANEL M10193 ON ON OVERHEAD P5 PANEL TO THE AUTO OR STBY POSITION. NOTE: THE TEST AUTOMATICALLY MOVES TO THE NEXT STEP IF THE R PACK SWITCH IS IN THE AUTO OR STBY POSITION.
4	52 SET R PACK SW OFF SW OFF	TURN THE R PACK SWITCH ON THE AIR CONDITIONING PANEL M10193 TO THE OFF POSITION.
5	52 ECS PACK L/R LO LO	WITH BOTH THE L AND THE R PACK SWITCHES IN THE OFF POSITION, THE MCDP BOTTOM LINE MESSAGE SHOWS LO LO. PUSH THE TEST HIGH SWITCH ON THE LAND THE R ECS CARD (M10313, M10312). THE MCDP BOTTOM LINE MESSAGE WILL SHOW HI HI.
6	52 ECS PACK L/R HI HI	RELEASE THE TEST HIGH SWITCHES.
7	52 ECS PACK L/R LO LO	PUSH THE MCDP YES/ADV SWITCH.

TEST 52 - TMC RLY/SW - PROCEDURE 2		
TEST STEP	MCDP INTERACTION INDICATION	OPERATION ACTION NOTE: IF THE MCDP TEST DOES NOT MOVE FORWARD REFER TO 22-00-04, FIG. 102
8	52 PUSH WG AI ON	PUSH THE WING ANTI-ICE SWITCH TO THE ON POSITION.
9	52 PUSH ALL AI SW OFF	PUSH ALL (WING AND ENGINES) ANTI-ICE SWITCHES TO THE OFF POSITION (THE ON INDICATION DOES NOT SHOW).
10	52 COWL AI L/R OFF OFF	PUSH THE L ENGINE ANTI-ICE SWITCH TO THE ON POSITION (ON INDICATION SHOWS)
11	52 COWL AI L/R ON OFF	PUSH THE R ENGINE ANTI-ICE SWITCH TO THE ON POSITION (ON INDICATION SHOWS)
12	52 COWL AI L/R ON ON	PUSH THE L AND THE R ENGINE ANTI-ICE SWITCHES TO THE OFF POSITION (ON INDICATION DOES NOT SHOW).
13	52 COWL AI L/R OFF OFF	PUSH THE MCDP YES/ADV SWITCH.
14	52 PUSH SOV SW L OPEN	1. PUSH THE L ENG SWITCH ON THE BLEED AIR CONTROL PANEL M10259 TO THE OPEN POSITION (VERTICAL LINE SHOWS). 2. MANUALLY OPEN THE L AIR SUPPLY PRESSURE AND THE SHUTOFF VALVE (V10116) (MM 36-11-09).
15	52 PUSH SOV SW R OPEN	1. PUSH THE L ENG SWITCH ON THE BLEED AIR CONTROL PANEL TO THE CLOSE POSITION (VERTICAL LINE DOES NOT SHOW). 2. PUSH THE R ENG SWITCH ON THE BLEED AIR CONTROL PANEL M10259 TO THE OPEN POSITION (VERTICAL LINE SHOWS). 3. MANUALLY OPEN THE R AIR SUPPLY PRESSURE AND THE SHUTOFF VALVE (V43) (MM 36-11-09).

TEST 52 - TMC RLY/SW - PROCEDURE 2		
TEST STEP	MCDP INTERACTION INDICATION	OPERATION ACTION NOTE: IF THE MCDP TEST DOES NOT MOVE FORWARD REFER TO 22-00-04, FIG. 102
16	52 PUSH ISLN VLV SW L OPEN	1. PUSH THE R ENG SWITCH ON THE BLEED AIR CONTROL PANEL TO THE CLOSE POSITION (VERTICAL LINE DOES NOT SHOW). 2. PUSH THE ISOLATION VALVE SWITCH ON THE BLEED AIR CONTROL PANEL M10259 TO THE OPEN POSITION (HORIZONTAL LINE SHOWS).
		NOTE: THESE STEPS INCLUDE OPERATION OF THE THRUST REVERSER LEVERS TO OPERATE THE THRUST REVERSER SWITCHES IN THE AUTOTHROTTLE MICROSWITCH PACK ASSEMBLY (M966).
17	52 VFY THRST LVR SYS CLR - ADV	1. PUSH THE L ENG AND THE R ENG SWITCHES ON THE BLEED AIR CONTROL PANEL M10259 TO THE CLOSE POSITION (VERTICAL LINE DOES NOT SHOW). PUSH THE ISOLATION VALVE SWITCH ON THE BLEED AIR CONTROL PANEL M10259 TO THE CLOSE POSITION (HORIZONTAL LINE DOES NOT SHOW). 2. MAKE SURE THE THRUST LEVER SYSTEM IS CLEAR FOR OPERATION. PUSH THE MCDP YES/ADV SWITCH WHEN THE THRUST REVERSER SYSTEM IS CLEAR FOR OPERATION.
18	52 SET L REV THRST LVR - ON	1. PUSH THE THROTTLE LEVERS AGAINST THE AFT MECHANICAL STOPS. 2. MOVE THE LEFT REVERSER THRUST LEVER UP AND AFT TO THE REVERSE THRUST INTERLOCK POSITION.
19	52 SET L REV THRST LVR - OFF	MOVE THE LEFT THRUST REVERSE LEVER TO THE FORWARD THRUST POSITION (REVERSERS STOWED).

TEST 52 - TMC RLY/SW - PROCEDURE 2		
TEST STEP	MCDP INTERACTION INDICATION	OPERATION ACTION NOTE: IF THE MCDP TEST DOES NOT MOVE FORWARD REFER TO 22-00-04, FIG. 102
20	52 SET R REV THRST LVR - ON	1. PUSH THE THROTTLE LEVERS AGAINST THE AFT MECHANICAL STOPS. 2. MOVE THE RIGHT REVERSER THRUST LEVER UP AND AFT TO THE REVERSE THRUST INTERLOCK POSITION.
21	52 SET L REV THRST LVR - OFF	MOVE THE RIGHT THRUST REVERSER LEVER TO THE FORWARD THRUST POSITION (REVERSER STOWED).
	52 TEST COMPLETE	PUSH THE MCDP YES/ADV OR THE NO/SKIP SWITCH TO GET OUT OF THE TEST AND ARM TEST 56.

S 862-126
(9) Operate the thrust reverser isolation valve (Ref 78-31-00).

S 862-127
(10) Remove pneumatic power if it is not necessary.

S 862-128
(11) Put the airplane back to its usual condition if no other ground tests are necessary.

AA. 56 FCC CONFIG/OPT

S 862-129
(1) Do the "Prepare to Test" steps.

S 742-190
(2) Do MCDP test 56.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to parity. A check is done to make sure the FCC program wiring has the correct configuration and customer options. The pins are read as a binary count and the decimal equivalent shows on the MCDP. All three channels show at the same time.

S 862-131
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

AB. 57 TMC CONFIG/OPT

S 862-132
(1) Do the "Prepare to Test" steps.

S 742-133
(2) Do MCDP TEST 57

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC FAIL message shows, the MCDP shows TMC interface faults related to parity. A check is done to make sure the TMC program pin wiring has the correct configuration and customer options. The pins are read as a binary count and the decimal equivalent shows on the MCDP.

S 862-134
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

AC. 59 FCC INSTR

S 862-135
(1) Do the "Prepare to Test" steps.

S 742-136
(2) Do MCDP test 59.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. Tests of the operation of the captain's and first officer's flight director switches, and the instrument source select switches are done.

S 862-137
(3) Put the airplane back to its usual condition if no other ground tests are necessary.

AD. 60 TMC INSTR

S 862-138
(1) Do the "Prepare to Test" steps.

S 212-139
(2) Make sure the THRUST REF SET knob on the EICAS DISPLAY SELECT PANEL is pushed in.

S 742-140
(3) Do MCDP test 60.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. The MCDP sends test parameters to the TMC to control the captain's and first officer's EPR instruments target index.

S 862-191
(4) Put the airplane back to its usual condition if no other ground tests are necessary.

AE. 64 SPD BK/FLAP XDCR

S 862-142
(1) Do the "Prepare to Test" steps.

S 862-326

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, ADN STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply Hydraulic Power" steps.

S 742-144

WARNING: MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS MOVE TO THEIR COMMANDED POSITION IN LESS THAN ONE SECOND AND CAUSE INJURY TO PERSONS AND DAMAGE TO EQUIPMENT.

(3) Move all persons and equipment away from the spoilers.

S 742-192
(4) Do MCDP test 64.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC and TMC interface faults related to the slat switch and flap transducer. The MCDP shows a message to make sure hydraulic power is on. To do tests of the speedbrake and flap lever, the MCDP tells a person to set the handle or lever to a specified position. The MCDP then does a check for the correct position.

S 862-145
(5) Put the airplane back to its usual condition if no other ground tests are necessary.

AF. 65 STAB TRIM

S 862-146
(1) Do the "Prepare to Test" steps.

S 862-327

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply of Hydraulic Power" steps.

S 742-193
(3) Do MCDP test 65.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to stab trim. The MCDP shows messages to make sure the SAM and hydraulic power is ON and then to make sure the A/P is engaged in command. The MCDP then tells the FCCs to drive the Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) to move the stabilizer up and down. A test for each FCC is done separately.

S 862-296
(4) Put the airplane back to its usual condition if no other ground tests are necessary.

AG. 66 XDCR OUTPUTS

S 862-297
(1) Do the steps shown to prepare to test each of these components:
(a) Aileron Autopilot Servos
1) Set the aileron trim to 0 units of trim.
2) Set the flaps to 0.
(b) Elevator Autopilot Servo
1) Set the stabilizer to 6 units of trim.
2) Pressurize the elevator feel pitot system to 350 knots (AMM 34-11-00).
(c) Neutral Shift Transducers
1) Set the stabilizer to 2 units of trim.
2) Pressurize the elevator feel pitot system to 350 knots (AMM 34-11-00).

(d) Rudder Autopilot Servo
1) Make sure the test switches on the L and R Rudder Ratio Modules in the Main E/E Compt are set to NORM (Ref 27-21-00).
2) Set the rudder trim to 0 units of trim.
3) Make sure the rudder is at the neutral position (Ref 27-21-00).

(e) Stabilizer Position
1) Set the stabilizer trim to six units of trim.

(f) Power Lever Angle Transducers (PLA)
1) Set the thrust levers against the aft (idle) steps.

S 862-283

(2) Do the "Prepare to Test" steps.

S 862-288

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Move the towing lever on the nose gear metering valve module to the TOW POSITION.

S 862-130

(4) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 862-152

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(5) Move the speed brake lever to the ARMED position.

S 862-194

(6) Do the "Supply Hydraulic Power" steps.

S 742-286

(7) Do MCDP test 66.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC and TMC interface faults related to the transducers. The MCDP does tests and shows the position of the horizontal stabilizer, the aileron servo, the aileron surface, the elevator servo, the elevator surface, the elevator feel position, the rudder servo, the rudder surface, the flaps, and the power lever angle. Test 66 also does a test and shows the percentage of the speedbrake handle. Refer to Fig. 202 for the rigging parameters.

S 862-158

(8) Move the speed brake lever to the down-and-locked detent position.

S 862-195

(9) Put the airplane back to its usual condition if no other ground tests are necessary.

AH. 67 AIL SURF LIM

S 862-159

(1) Do the "Prepare to Test" steps.

S 862-328

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, ADN STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply Hydraulic Power" steps.

MCDP DISPLAY	RIG VALUE	MAX CH TRACKING DIFFERENCE	
		NULL	MAX INPUT
66 STAB DEG/LCR	-2 ±2°	0.3°	0.6°
66 AIL SRVO DEG	1		
66 AIL SURF DEG	0 ±0.5°	0.4°	0.8°
66 ELEV SRVO DEG	1		
66 ELEV SURF DEG 3	1.1 ±0.2° 2	0.4°	0.8°
66 FEEL POS DEG 4	0 ±0.1° 2	0.2°	0.4°
66 RUD SRVO DEG	1		
66 RUD SURF DEG	0 ±0.6°	1.2°	2.4°
66 SPD BK PCT	15.4 ±5%	N/A	N/A
66 FLAP DEG/FCC	0 ±2°	0.6°	1.2°
66 FLAP DEG/TMC	0 ±2°	0.6°	1.2°
66 PLA DEG/L/R	50 ±1.25°	2.5°	N/A

1 THE SERVO POSITIONS WERE SET BY THE CENTERING SPRINGS THE LAST TIME HYDRAULIC POWER WAS SET TO OFF. NO TOLERANCES ARE SPECIFIED. ONLY USE THE DATA TO MAKE SURE THE SERVO IS ALMOST IN THE CENTER AND IS NOT AT THE POSITIVE OR NEGATIVE LIMIT.

2 IF THE NULL (RIG) VALUE IS NOT IN TOLERANCE, PRESSURIZE THE ELEVATOR FEEL PITOT SYSTEM TO 350 KNOTS (REF 34-11-00).

3 SET THE STABILIZER TRIM TO 6 UNITS

4 SET THE STABILIZER TRIM TO 2 UNITS

MCDP DISPLAY	POSITIVE LIMIT	NEGATIVE LIMIT
67 AIL SURF DEG (SINGLE CHANNEL)	+10 ±3°	-10 ±3°
67 AIL SURF DEG (TRIPLE CHANNEL)	19° MINIMUM	19° MINIMUM
68 ELEV SURF DEG (SINGLE CHANNEL)	6.2° MINIMUM	5.5° MINIMUM
68 ELEV SURF DEG (TRIPLE CHANNEL)	+15 ±3°	-25 ±3°
69 RUD SURF DEG	22.8° MINIMUM	22.8° MINIMUM

Rigging Tolerances and Position Limits - Tests 66,67,68,69
Figure 202

S 742-284
(3) Do MCDP test 67.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the aileron autopilot servos. The MCDP shows messages to make sure the hydraulic power is on, the aileron trim is set to zero, and that all three A/P channels are engaged in command mode. The MCDP then moves the aileron servo to the positive limit and shows the aileron surface position in degrees for each channel. The same is done for the negative limit. This does a test of the autopilot aileron servo limits. Refer to Fig. 202 for the limit parameters.

S 862-163
(4) Put the airplane back to its usual condition if no other ground tests are necessary.

AI. 68 ELEV SURF LIM

S 862-285
(1) Do the "Prepare to Test" steps.

S 862-329

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, ADN STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the "Supply Hydraulic Power" steps.

S 742-165
(3) Do MCDP test 68.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to elevator autopilot servos. The MCDP shows messages to make sure the hydraulic power is on, the stab aileron trim is set to 6 units, and that all three A/P channels are engaged in command. The MCDP then moves the elevator servo to the positive limit and shows the elevator surface position in degrees for each channel. The same is done for the negative limit. This does a test of the elevator autopilot servo limits. Refer to Fig. 202 for the limit parameters.

S 862-298

(4) Put the airplane back to its usual condition if no other ground tests are necessary.

AJ. 69 RUD SURF LIM

NOTE: The Rudder Ratio Changer must be in the Low Speed position and the rudder in neutral for this test. The Rudder is at neutral when the two sides of the rudder trailing edge are aligned within the rudder index plate groove.

S 862-167

(1) Do the "Prepare to Test" steps.

S 982-168

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Move the towing lever on the nose gear metering valve module to the TOW POSITION.

S 432-169

(3) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 862-196

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, ADN STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(4) Do the "Supply Hydraulic Power" steps.

S 742-170

(5) Do MCDP test 69.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to the rudder autopilot servos. The MCDP shows messages to make sure the hydraulic power is on, the rudder trim is set to zero, and that all three A/P channels are engaged in command. The MCDP then moves the rudder servo to the positive limit and shows the rudder surface position in degrees for each channel. The same is done for the negative limit. Refer to Fig. 202 for the limit parameters.

S 862-299

(6) Put the airplane back to its usual condition if no other ground tests are necessary.

AK. Put the Airplane Back to Its Usual Condition

S 032-172

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN THE LOCKPIN IS REMOVED THE NOSE WHEEL CAN MOVE QUICKLY TO THE CENTER POSITION. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Make sure the nose gear wheels are centered and remove the nose gear steering valve lockpin.

S 862-197

(2) Make sure the towing lever goes to the NORMAL POSITION.

S 982-173

(3) Set the MCDP to OFF.

S 082-174

(4) Remove the remote MCDP control unit if it is installed.

S 862-305

(5) On the P5 panel, set the L and R ENG LIMITER switches on the ENGINE START/RAM AIR TURB control panel to OFF.

S 862-186

(6) On the P5 panel, set the L and R ELEC ENG CONTROL switches to OFF.

S 862-198

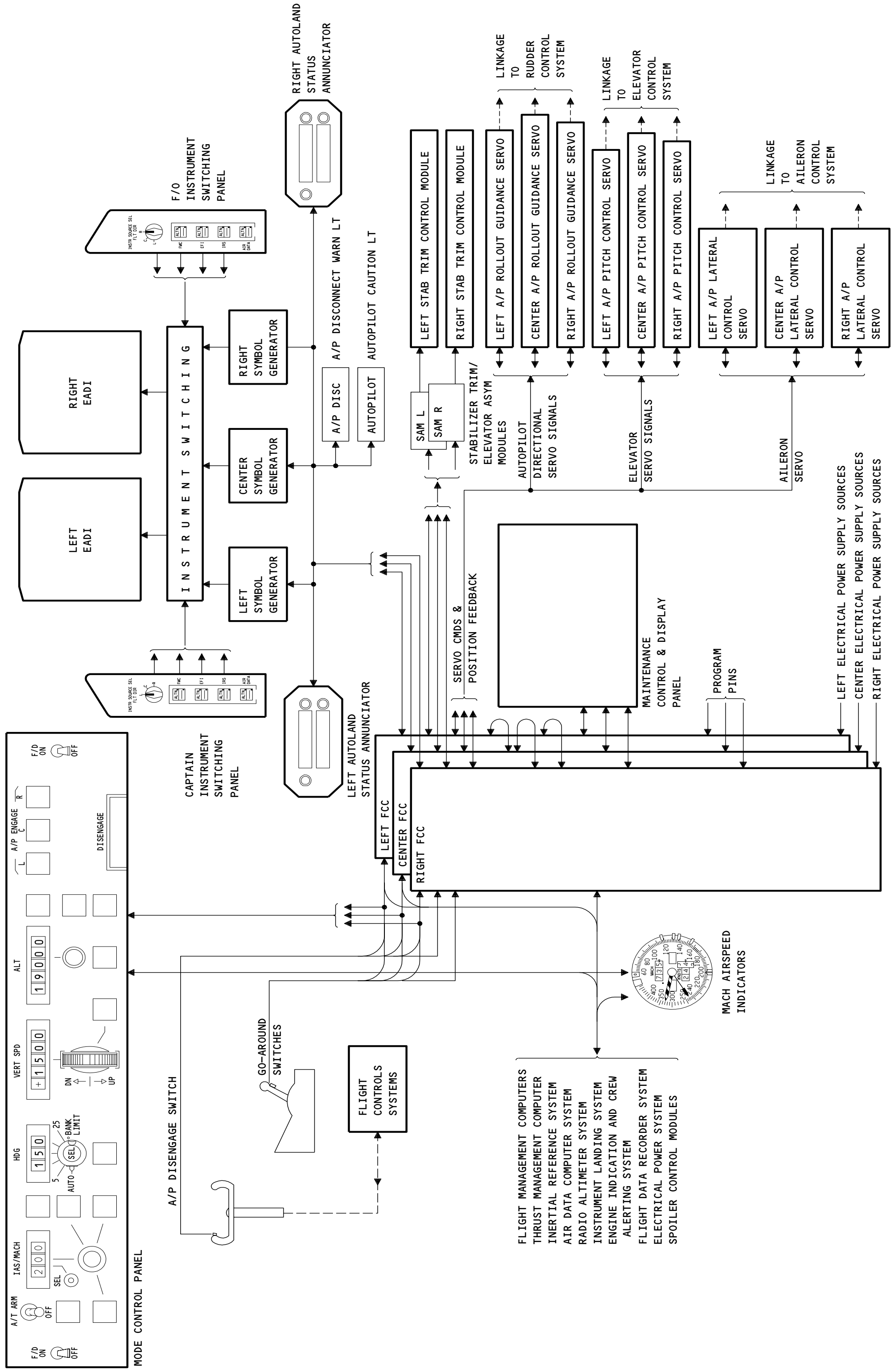
(7) Remove hydraulic power if is not necessary (AMM 29-11-00/201).

S 862-199

(8) Remove electrical power if it is not necessary (AMM 24-22-00/201).

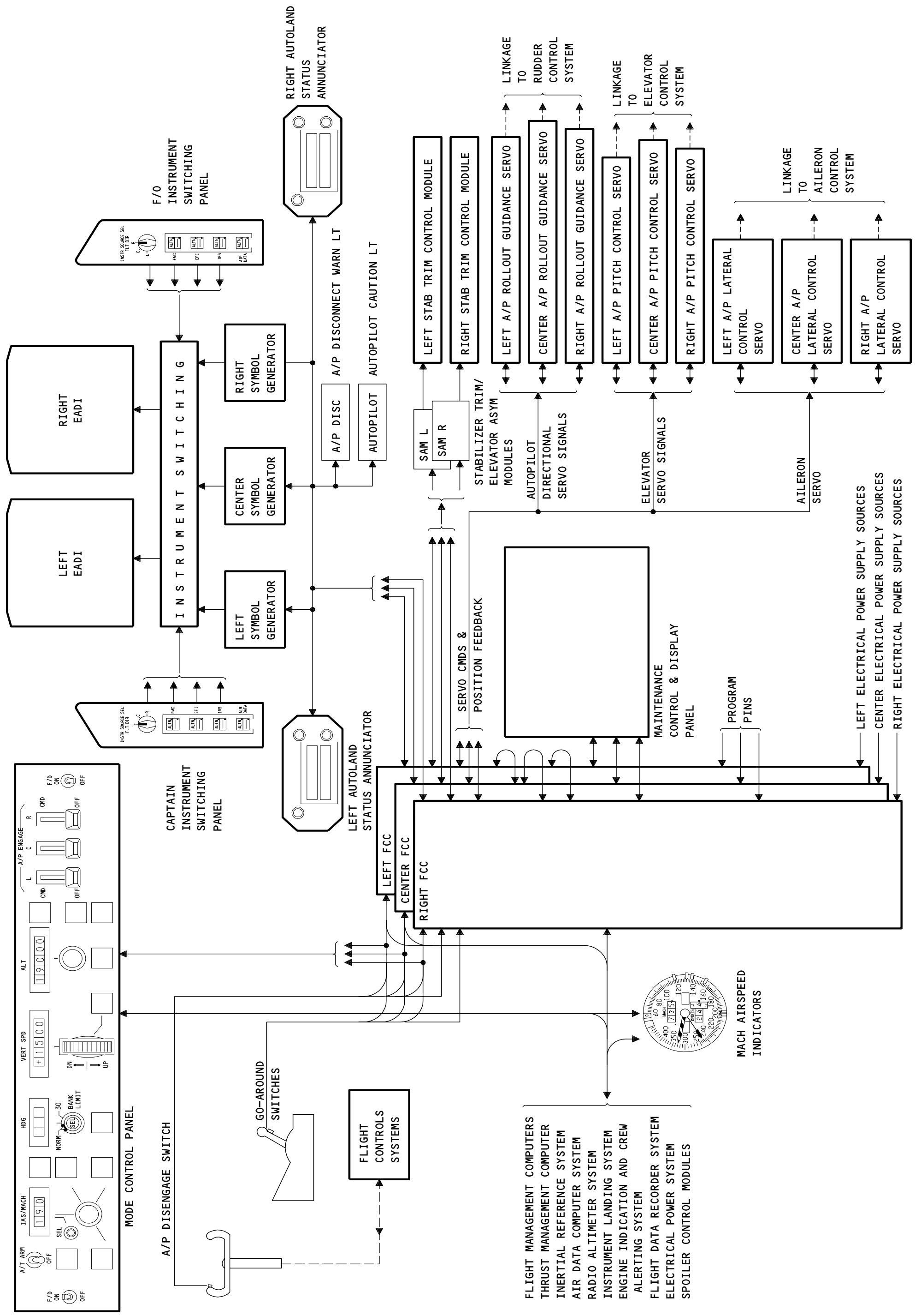
AUTOPILOT (FLIGHT CONTROL) - DESCRIPTION AND OPERATION

1. General
   A. The Autopilot/Flight Control System (AFCS) provides automatic control of the roll, pitch, and yaw axes through all phases of flight except takeoff (T/O). The Flight Director (F/D) provides guidance commands for all phases of flight including takeoff using manual control. The AFDS supplies display data to the Electronic Horizontal Situation Indicators (EHSI) and Electronic Attitude Director Indicators (EADI) of the Flight Instrument System (Ref 34-22-00).
   B. Components (Fig. 1)
      (1) The AFDS has three digital Flight Control Computers (FCCs) and one Autoflight Control Systems (AFCS) Mode Control Panel (MCP). The system is capable of providing autopilot/flight director guidance and control for Autoland Category III operations.
      (2) The left, center, and right (L, C, and R) FCCs contain all logic and signal handling circuitry for pitch, roll, and yaw axis control. Yaw control is used only during an automatic landing and rollout.
      (3) Each FCC provides Autopilot (A/P) command output signals which drive hydraulic servos that deflect control surfaces. Each FCC also provides Flight Director (F/D) output signals for steering commands and mode annunciator's displayed on the Electronic Attitude Director Indicators (EADIs).
      (4) The primary digital interface between the AFDS and associated sensor systems uses the ARINC 429 Digital Information Transfer System (DITS).
      (5) The AFDS flight deck components (Fig. 2) are located as follows:
         (a) Mode Control Panel - center glare shield, P55 panel
         (b) Autoland Status Annunciator - Captain's P1-3 panel and First Officer's (F/O) P3-1 panel.
         (c) A/P DISC red warning light - P1-3.
         (d) AUTOPILOT amber caution light - P1-3
         (e) A/P disengage switches - Inboard side of outboard horn on each control wheel
         (f) AFDS go-around switches - One for each FCC and TMC on each thrust lever

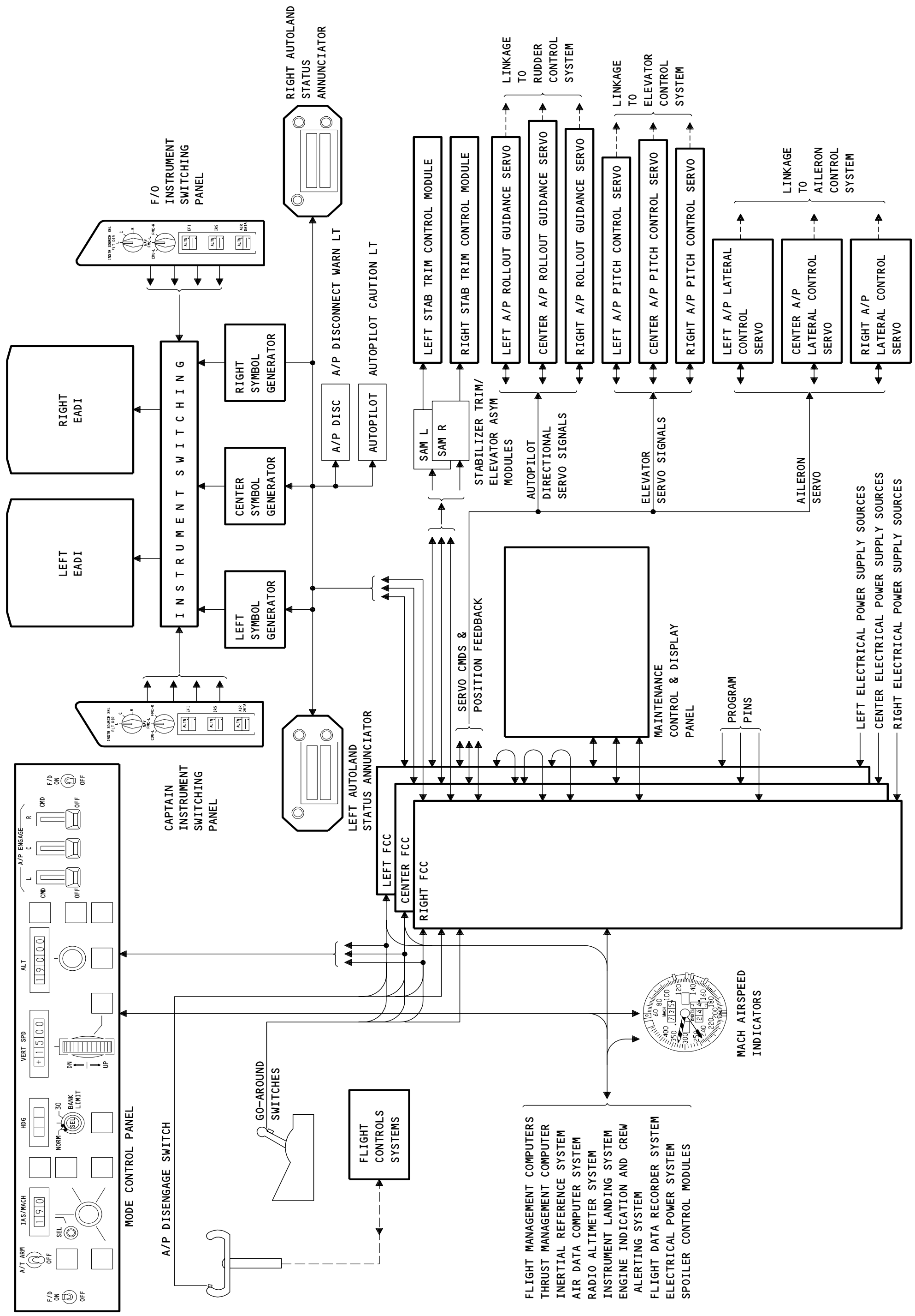


Autopilot Flight Director System
Figure 1 (Sheet 1)

A71195

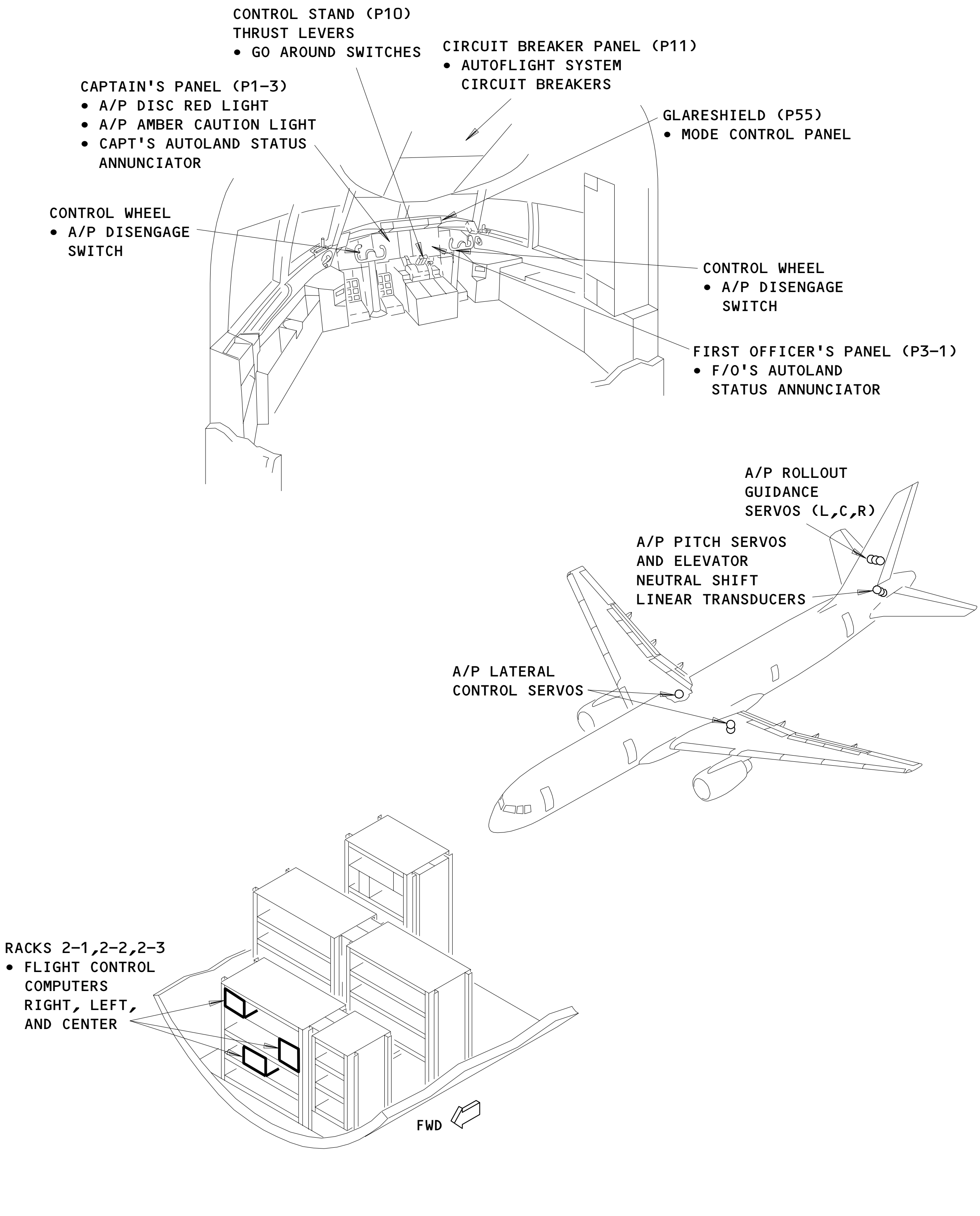


Autopilot Flight Director System
Figure 1 (Sheet 2)



Autopilot Flight Director System
Figure 1 (Sheet 3)

1576822



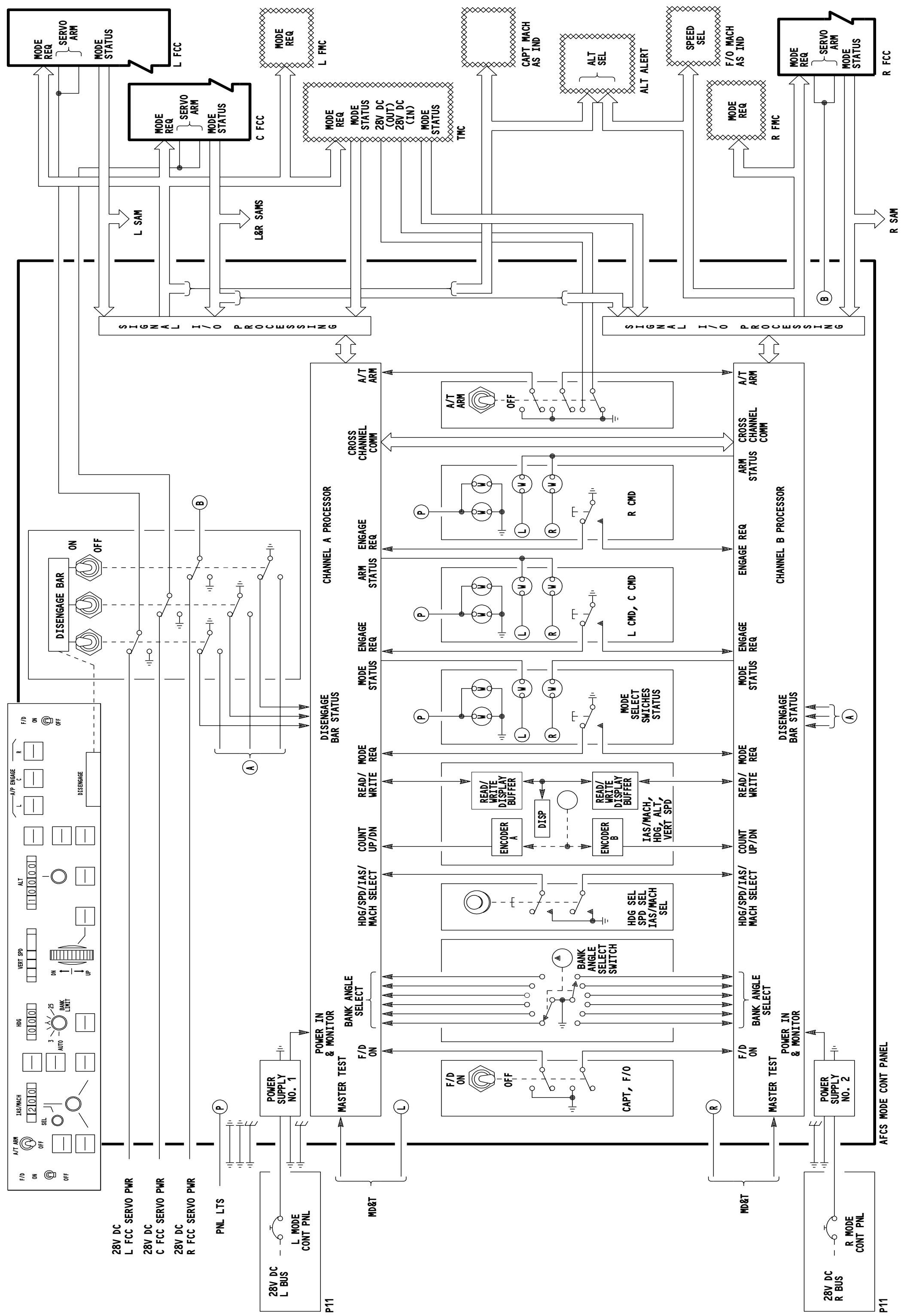
Autopilot Flight Director System - Component Locations
Figure 2

(6) The below-deck components of the AFDS are located as follows:
  (a) Flight Control Computers (FCCs-3) - on main Electrical/Electronics (E/E) racks (E3-1, E3-2, and E3-3)
(7) The Autopilot flight control servos are located as follows:
  (a) Autopilot Lateral Control Servos - Left (L) and Center (C) in the left wing root, Right (R) in the right wing root.
  (b) Autopilot Pitch Control Servos - L, C, and R aft of the horizontal stabilizer
  (c) Autopilot Rollout Guidance Servos - L, C, and R in the lower area of the vertical stabilizer.

C. Interfacing Components (Fig. 1)
(1) Each FCC provides an output to a Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) for automatic trim of the stabilizer when an autopilot is engaged. The SAMs (2) are on main E/E racks E3-1 and E4-1 (Ref 22-22-00, Automatic Stabilizer Trim System).
(2) Each FCC also provides fault data to the Maintenance Control Display Panel(MCDP) and performs ground tests when requested by the MCDP. The MCDP is on main E/E rack E4-2 (Ref 22-41-00, Maintenance Monitor).
(3) The F/D source select switches on P1-1 and P3-3 allows each pilot to select which FCC to use for Flight Director Commands on his EADI (Ref 34-22-00, Flight Instrument System).

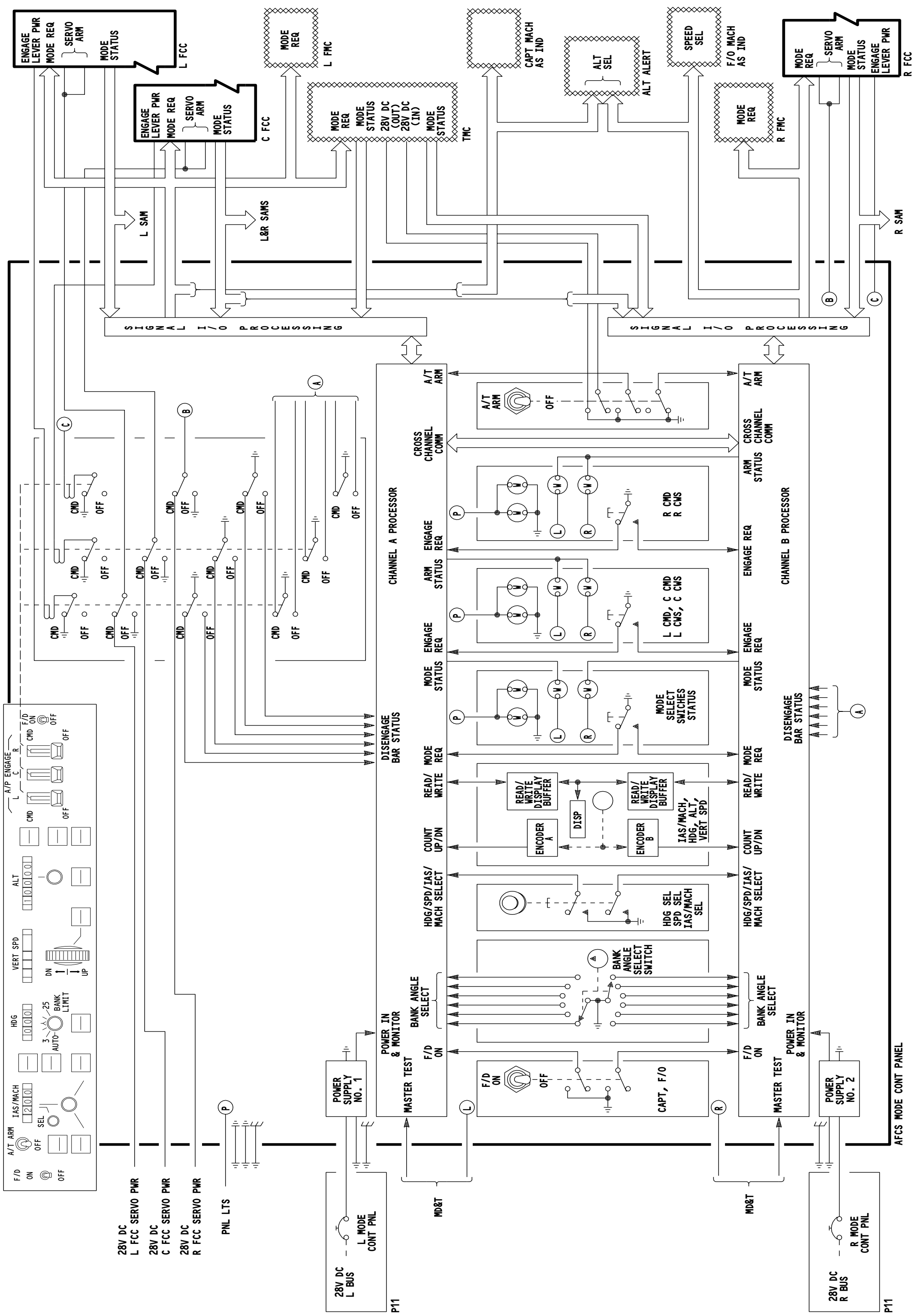
D. Indications
(1) The AFDS condition and status is indicated by various lights and displays as follows:
  (a) The mode select switch/lights on the MCP have identifying legends in the upper half of the switch cover. The legends light when panel lights power is applied to the panel. A dot-bar matrix on the lower half lights when the FCC has accepted the associated mode request. Digital readouts on the MCP indicate selected airspeed, selected vertical speed, selected heading, and selected altitude. Panel legends identify the status of the F/D, A/T (auto throttle), and bank angle (B/A) limit switches.
  (b) The Electronic Attitude Director Indicator (EADI) provides Flight Director displays and Flight Mode Annunciation (FMA) armed or engaged roll and pitch modes.
  (c) A red autopilot disconnect (A/P DISC) light and an amber AUTOPILOT caution light are on the captain's instrument panel (P1-3).
  (d) A caution and warning system consists of a warning horn, master WARN lights, and an A/P DISC (Red) message on the Engine Indicating and Crew Alerting System (EICAS) (Ref 31-51-00, Warning System).

E. Autopilot/Flight Director System Mode Control Panel Interfaces Schematic (Fig. 3)



Autopilot/Flight Director System Mode Control Panel Interfaces Schematic
Figure 3 (Sheet 1)

A71200



Autopilot/Flight Director System Mode Control Panel Interfaces Schematic
Figure 3 (Sheet 2)

A71202

(1) Autopilot (A/P) and flight director (F/D) engage requests are generated by the Mode Control Panel (MCP) and are sent to the Flight Control Computers (FCCs). Once accepted, the requests can result in the arming and engaging of the A/P servos.

(2) Mode requests for both A/P and F/D are generated at the MCP and transmitted on an ARINC 429 digital data bus to the FCCs.

(3) GUI 001-114, 116-999;
the A/P disengage bar logic tells the FCC if the bar is up or down. This allows the FCC to enable or disable the A/P servos.

(4) GUI 115;
the FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the A/P mode select switch/lights. Mode status is also indicated on the EADIs (Ref 22-14-00).

(5) GUI 001-114, 116-999;
the FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the A/P engage and mode select switch/lights. Mode status is also indicated on the EADIs (Ref 22-14-00).

F. FCC Input Discretes From Remote Switches

(1) Two A/P disengage switches (one on each control wheel) provide discrete signals to the FCCs which disengage the autopilot. The go-around switches (one for each FCC and TMC on each thrust lever) provide discretes to the FCCs to start the Go-Around (GA) mode.

G. AFDS Input Power

(1) The FCCs receive 28v dc and 115v ac, 400 Hz power for internal use. The 115v ac is also reduced to 26v ac in the FCC and supplied to the autopilot servos position LVDT's and elevator neutral shift transducers. The MCP power supply receives 28v dc. Both the MCP and the FCC use 28v dc for the actuator and servo arm, engage, and disengage functions.

(2) Autopilot/Flight Director System Mode Control Panel Operation

(a) MCP Functional Partitioning

1) The MCP contains channels A and B which use a common display element. This prevents a single failure from causing a total loss of the MCP function. Each channel receives inputs from panel switches and encoders and processes the data independently.

2) The A channel receives ARINC 429 data from the L and C FCCs, L FMC, and the TMC. Channel A transmits 429 data to the L and C FCCs, L FMC, TMC, and pilot's instruments.

3) The B channel receives 429 data from the R and C FCC, R FMC, and TMC. Channel B transmits 429 data to the R FCC, R FMC, and pilot's instruments.

(b) MCP Autopilot Engage Switch/lights
   1) GUI 115;
      placing any one of the MCP engage switches in CMD (L, C, or R) generates an engage request signal. The L and C switch requests are processed by channel A; the R request by channel B. The switch hold solenoid is energized when the A/P solenoids are armed. The hold solenoid keeps the A/P engage switch in the CMD position.
   2) GUI 001-114, 116-999;
      pressing any one of the CMD switch/lights (L, C, or R) generates an engage request signal. The L and C switch/light requests are processed by channel A; the R requests by channel B. Illumination of the dot-bar matrices on the switch/lights is controlled by status logic generated in the appropriate FCC. The matrix lights when the request is accepted.
   3) The dot-bar matrix lights on the mode select switch/lights are checked by the Master Dim and Test System (Ref (33-16-00).

(c) MCP Autopilot Mode Select Switch/lights
   1) The mode select switch/lights provide mode request signals to both A and B channels of the MCP. The request also sets a bit in a digital word which is transmitted to the FCC or TMC. The illumination of the dot-bar matrices of these switches is controlled by mode status logic generated in the engaged FCC(s) or TMC. No illumination is provided for the IAS/MACH SEL switch or the HDG SEL switch.

(d) MCP Autopilot Reference Data Controls and Displays
   1) Reference values for altitude, heading, vertical speed, and airspeed or mach number are entered using the related select knob or thumbwheel on the MCP. These reference values are optically encoded and supplied to both the A and B channels and routed to the appropriate MCP window for display. The reference values are also sent to the FCC, FMC, and TMC.

(e) Vertical Speed Mode
   1) When the vertical speed (V/S) mode select switch/light is pressed, a V/S mode request discrete is initiated. This discrete generates a manual ON signal which arms the vertical speed mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. The vertical speed mode is automatically initiated by an engaged FCC (CMD or F/D) if no other pitch modes are selected or engaged (Ref 22-12-00).

2) The dot-bar matrix annunciator in the V/S switch/light illuminates when the MCP receives V/S OPER from the FCC's. V/S OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once V/S mode is engaged, it can be disabled if CMD is disabled and both the F/Ds are turned off. The V/S mode will also be disabled by the selection of the following pitch modes: ALT HLD select, FLCH select, VNAV select, APP select, or GA select.
3) The V/S pushbutton can also be illuminated using the TEST switch on the master dim and test panel. This test input checks MCP circuits (data management) as well as the dot-bar matrices.

(f) Heading Hold Mode
1) Pressing the HDG HOLD mode select switch/light initiates a heading hold mode request discrete. This discrete generates a manual ON signal which arms the heading hold mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. The heading hold mode is automatically initiated by an engaged FCC (CMD or F/D) if no other roll modes are selected or engaged (Ref 22-13-00).
2) The dot-bar matrix annunciator in the HDG HOLD switch/light lights when the MCP receives HDG HOLD OPER from the FCC's. HDG HDL OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once HDG HOLD mode is engaged, it can be disabled if CMD is disabled and both the F/Ds are turned off. Heading hold is also disabled by selection of the following modes: HDG select, LNAV select, LOC override, or GA select.

(g) Altitude Hold Mode
1) Pressing the ALT HOLD mode select switch/light initiates an altitude hold mode request discrete. This discrete generates a manual ON signal which arms the altitude hold mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. Altitude hold can also be automatically initiated by an engaged FCC (CMD or F/D) during altitude capture (Ref 22-12-00).
2) The dot-bar matrix in the ALT HOLD switch/light lights when the MCP receives ALTITUDE MODE OPER from the FCC(s). ALT HLD OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once altitude hold mode is engaged, it can be disabled if CMD is disabled and both the F/Ds are turned off. Altitude hold will also be disabled by selection of the following modes V/S select, FLCH select, VNAV select, APP select, or GA select.

(h) Thrust Mode
1) Pressing the EPR mode select switch/light initiates a thrust mode request discrete. This discrete results in a thrust mode select signal if the autothrottle (A/T) is armed and operating with no faults while the airplane is either on the ground, in the air, or in the takeoff mode.

2) The dot-bar matrix in the thrust mode select switch/light lights when the MCP receives THRUST MODE OPER from the TMC. Once the thrust mode is engaged, it can be disabled if the A/T is disengaged or another mode is selected (SPD select, FLCH select, or VNAV select).

(i) Speed Mode

1) Pressing SPD switch/light initiates a speed mode request discrete. This discrete generates a speed mode select signal if the A/T is enabled and the airplane is airborne. The speed mode can also be generated by FLCH mode or VNAV mode disengagement.

2) The dot-bar matrix in the SPD switch/light lights when the MCP receives SPEED MODE OPER from the TMC. Once SPEED MODE is initiated, it can be disabled if the A/T is disabled, A/T retard engaged, or GA mode selected.

(j) Flight Level Change Mode

1) Pressing the Flight Level Change (FLCH) mode switch/light initiates a flight level change discrete. This discrete generates a flight level change select signal if the FCC is engaged in CMD or the F/D is on. The FLCH select signal arms the flight level change mode.

2) The dot-bar matrix in the FLCH switch/light lights when the MCP receives FLIGHT LEVEL CHANGE OPER from the FCCs. FLIGHT LEVEL CHANGE OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once flight level change is engaged it can be disabled if the following modes are selected: ALT HOLD select, VS select, VNAV select, or GA select.

(k) Lateral Navigation Mode

1) Pressing the Lateral Navigation (LNAV) mode select switch/light initiates an LNAV mode request discrete. This discrete generates an LNAV SELECT signal if the FCC is engaged in CMD or the F/D is on. The LNAV SELECT signal arms the LNAV mode. The LNAV mode can be manually disarmed by pressing the LNAV switch/light a second time.

2) The dot-bar matrix in the LNAV switch/light lights when the MCP receives LNAV OPER or LNAV ARM from the FCCs. The LNAV OPER signal is also sent to the EFIS symbol generators for mode display on the EADIs. Once LNAV mode is engaged, it can be disabled if the following modes are selected: HDG HLD select, HDG select, or GA select.

3) The same signal that lights the LNAV switch/light is provided, by the MCP, to the Flight Management System (FMS) to enable LNAV steering commands from the FMC (Ref 34-61-00).

(l) Vertical Navigation Mode
1) Pressing the Vertical Navigation (VNAV) mode select switch/light initiates a VNAV mode request discrete. This discrete generates a VNAV SELECT signal if the FCC is engaged in CMD or the F/D is on. The VNAV SELECT signal arms the vertical navigation mode. The VNAV mode can be manually disarmed by pressing the VNAV switch/light a second time.
2) The dot-bar matrix in the VNAV switch/light lights when the MCP receives VNAV OPER from the FCCs. The VNAV OPER signal is also sent to the EFIS symbol generators for mode display on the EADIs. Once VNAV MODE is engaged it will be disabled if the following modes are selected: ALT HLD select, V/S select, or FLCH select.
3) The same signal that lights the VNAV switch/light is provided, by the MCP, to the FMS to enable VNAV steering commands from the FMC (Ref 34-61-00).

(m) Localizer Mode
1) Pressing the localizer (LOC) mode select switch/light initiates a localizer mode request discrete. This discrete generates a LOC MODE request signal if the FCC is engaged in CMD or the F/D is on. The LOC MODE signal arms the localizer mode. This signal can be manually disarmed by pressing the LOC mode switch/light a second time.
2) The dot-bar matrix in the LOC switch/light lights when the MCP receives LOC MODE ARM/OPER. LOC MODE OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once the localizer mode is engaged it can be disabled if the following modes are selected: HDG HLD select, HDG select, or LNAV select.

(n) Approach Mode
1) Pressing the Approach (APP) mode select switch/light initiates an approach mode request discrete. This discrete generates an APP MODE request signal if the FCC is engaged in CMD or the F/D is on. The APPROACH MODE request signal arms the approach mode. This signal can be manually disarmed by pressing the APP switch/light a second time.
2) The dot-bar matrix in the APP switch/light lights when the MCP receives APPROACH MODE ARM/OPER from the FCCs. Once the approach mode is engaged, it can be disabled when the following modes are selected: HDG HLD Select, HDG select, or LNAV select.

(o) GUI 001-114, 116-999;
Backcourse Mode
1) Pressing the Backcourse (B/CRS) mode select switch/light initiates the backcourse mode request discrete. This discrete generates a BACKCOURSE SELECT signal if the FCC is engaged in CMD or the F/D is on. If BCRS ARM VALID is true, BCRS SELECT generates BCRS MODE ARM/OPERATE. This signal can be manually disarmed by pressing the B/CRS switch/light a second time.

2) The dot-bar matrix in the B/CRS pushbutton lights when the MCP receives BCRS MODE ARM/OPER. Once BCRS MODE ARM/OPERATE has been initiated, it can be disabled if BCRS ARM VALID is lost or the following modes are selected: HDG HLD select, HDG select, or LNAV select.

(3) ALT, VERT SPD, & HDG Reference Displays/Controls

(a) Encoders and Displays

1) Each numerical magnetic wheel display has an optical encoder. This is controlled by a knob or thumbwheel which is used for modifying the display. Each encoder has dual redundant circuits on a common shaft for interfacing with channels A and B of the MCP.

2) Each display has initial, maximum, and minimum values and an incremental value as previously defined in the component details section.

3) The V/S and HDG displays can be initialized/synchronized by the FCCs or manually set by the crew.

(4) IAS/MACH Reference Displays/Controls

(a) IAS/MACH Display States and Transitions

1) The IAS/MACH display has four valid states.

a) OFF

b) Airspeed being displayed: Airspeed (IAS) display is entered automatically when MCP power is turned on or when the SPD knob is pressed. Airspeed is also displayed if the SEL switch is pressed when MACH is being displayed, or if the display is blank when SPD or FLCH modes are initiated.

c) MACH being displayed: this state is entered if the SEL switch is pressed when airspeed is being displayed and the airplane's speed is between 0.40 and 0.95 mach.

d) Speed display blank: this state is entered from either IAS or MACH display states if the VNAV mode is initiated or the SPD knob is pressed with the AFDS not in the FLCH, SPD, T/O, or G/A modes. (These are airspeed-being-displayed only modes.)

(b) Encoders and Displays

1) The numerical magnetic wheel display is modified by an optical encoder which is controlled by the rotary knob. The encoder has dual redundant circuits on a common shaft for interfacing with channels A and B of the MCP. The display shows airspeed from 100 to 399 knots in increments of one knot, and Mach number from 0.40 to 0.95 in increments of 0.01 Mach. If the reference airspeed from the FCC is invalid, the display shows 200 knots.

2) The IAS/MACH display can be initialized/synchronized by the FCCs or manually set by the crew.

(5) GUI 001-114; 116-999;
AFDS MCP Disengage Bar - Operation

(a) The DISENGAGE bar provides total disengagement of all three FCCs. The switch assembly consists of three DPDT toggle switches (one for each FCC) tied together with a bar across the bat handles. When the bar is up, the servos may be engaged. When down, 28 Vdc is removed from the servos, preventing engagement. When the bar is down, a dayglow orange strip is visable, annunciating the disengaged position of the switch.

H. Autopilot/Flight Director System Interfaces (Fig. 4)

(1) The integrated Flight Management Systems (FMS) consists of the L and R FMCs, the L, R, and C AFDSs, the Thrust Management System (TMS), and the Maintenance Control Display Panel (MCDP).

(a) The AFDS can function with the A/P off, and the F/D on; or with single channel A/P engaged in CMD with the F/D on. The configuration for autoland operation is normally three channels but can be two autopilot channels in CMD and F/D off or on.

(b) In CMD ENG, the AFDS provides flight path guidance and control using raw data input from interfacing sensors. In an FMS coupled mode, it receives computed VNAV and LNAV pitch and roll commands from the FMC. The AFDS also provides the commands for automatic stabilizer trim.

(c) Full system monitoring and Built-In-Test Equipment (BITE) detect system faults in flight. This data is stored for recall by the MCDP after landing.

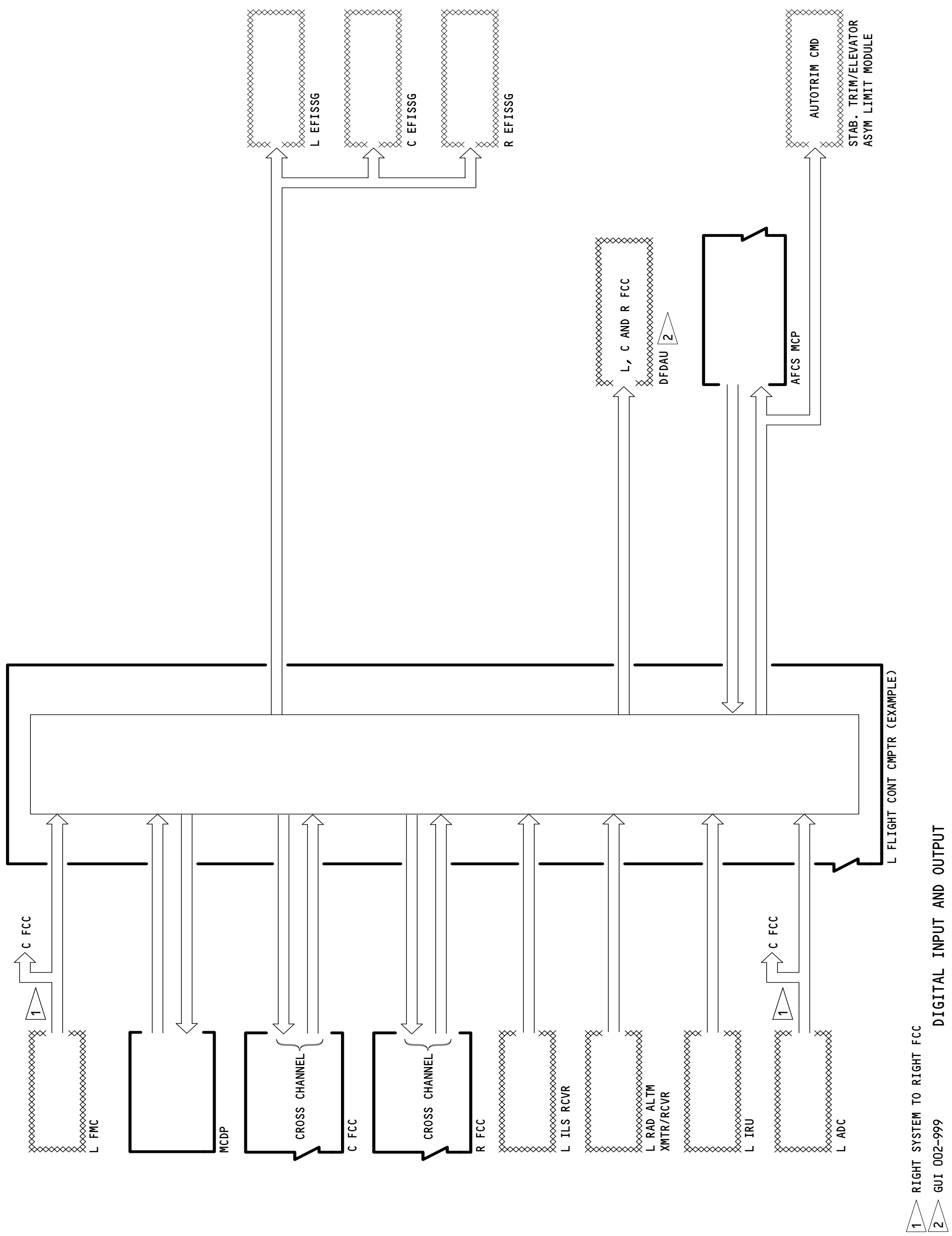
(d) Digital interfaces between the AFDS and associated systems use ARINC 429 DITS buses.

(2) AFDS Interfaces

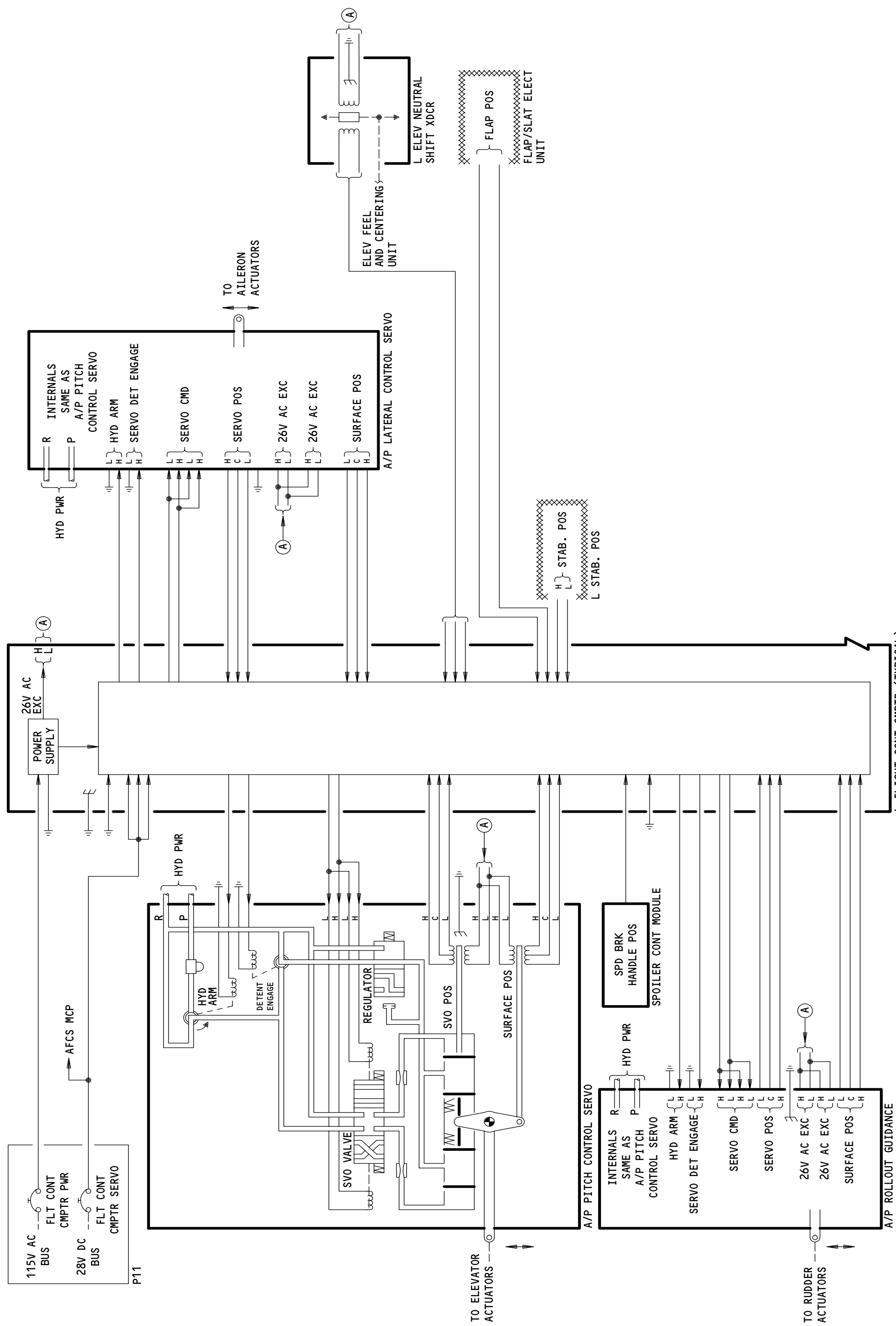
(a) Each FCC has two-way interfaces with all A/P servos, the MCP, the SAMs, the MCDP, and the other two FCCs.

(b) Each FCC receives data from critical sensors (dedicated sources for AUTOLAND signals) and non-critical sensors (non-dedicated sources and switched inputs). Each FCC also uses signals from the A/P disengage switches, go-around switches, and FCC program pins.

(c) Each FCC sends data to the Autoland Status Annunciator (ASA), the Engine Indication and Crew Alerting System (EICAS) (Ref 31-41-00), and symbol generators (SGs) of the Electronic Flight Instrument System (EFIS) (Ref 34-22-00, Flight Instrument System).

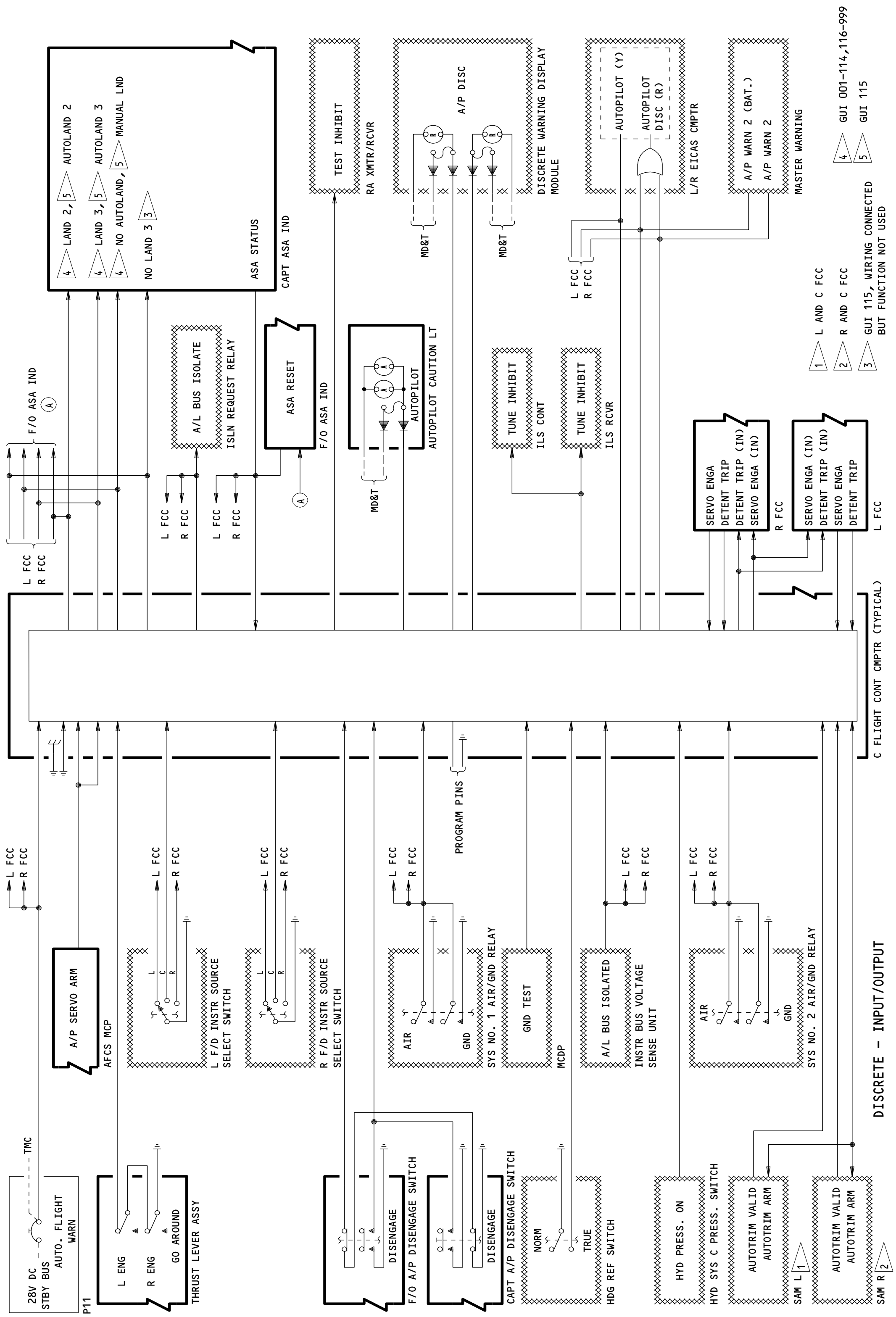


Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 1)



ANALOG INPUT AND OUTPUT

Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 2)



DISCRETE - INPUT/OUTPUT

Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 3)

(3) The MCP interfaces with both mach/airspeed indicators (Chapter 34) the Warning Electronics Unit (WEU) (Chapter 31), each FMC (Chapter 34) and the TMC (Chapter 22).
   (a) The MCP has a two-way analog and digital data link with the left and center FCCs, and a separate two-way link with the right FCC. Each FCC has a two-way link with the other FCCs and the MCDP.
   (b) The MCP interfaces with the following units via ARINC 429 data links:
      1) Captain's and F/O's mach/airspeed indicators (M/ASI) (Ref 34-13-00, Air Data Instruments)
      2) Left, center, and right FCCs (Ref 22-11-00).
      3) Left and right FMCs (Ref 34-61-00, Flight Management Computer System).
      4) Thrust Management Computer (Ref 22-31-00).
      5) Altitude alert warning module in the Warning Electronic Unit (WEU) (Ref 31-51-00, Warning System).
(4) The A/P disengage switches on the control wheels interface with all three FCCs.
(5) The go-around switches on the thrust levers interface with the TMC and the FCCs.
(6) Each (2) FLT DIR (Flight Director) select switch selects the L, C, or R FCC as the data source for F/D displays. The left switch, (P1-1 panel) controls the captain's EADI. The right switch (P3-3 panel) controls the first officer's EADI.
(7) The left EFI (Electronic Flight Instrument) source select switch (P1-1 panel) selects the left or center EFIS SG (Symbol Generator) as the display processor for the left Electronic Horizontal Situation Indicator (EHSI) and EADI. The right EFI source select switch (P3-3 panel) selects the right or center (ALTN) symbol generator for the right EHSI and EADI.
(8) If the EFI switch on the left and right INSTR SOURCE SEL panels are selected to the same source, single tone chime sounds and the EICAS displays a yellow INSTR SW message. Instrument switching is covered in detail in 34-22-00, Flight Instrument System.

I. Sensors
(1) Critical sensors are those which provide raw data inputs during Autoland. Each FCC receives a dedicated input from a critical sensor. Autoland critical sensors are the Instrument Landing System (ILS) receiver, Inertial Reference Unit (IRU), Radio Altimeter (RA), and the Electrical Power (Control) System (EPCS).
(2) Non-critical sensors are those which do not provide essential raw data during Autoland. The FCCs receive non-critical sensor inputs through instrument switching or common data links. Non-critical sensors are the Flight Management Computer System (FMCS), Air Data Computing System (ADCS), Proximity Switch Electronic Unit (PSEU), and the flap and slat position sensors.

J. AFDS Outputs
(1) The AFDS produces the following outputs:
   (a) Autopilot servo commands
   (b) Flight director displays

(c) Mode status and Autoland annunciation
(d) Caution and Warning outputs
(e) Flight data acquisition signals

(2) Interchannel data permits data sharing and total system verification during Autoland (Ref 22-15-00).

(3) Each FCC provides analog servo commands to its own Electrohydraulic Servo Valves. Each FCC also provides analog discretes to the other two FCCs when its pitch and roll A/P servos are engaged.

(4) Output buses one, two, and three from each FCC, supply the MCP, MCDP, and EFIS symbol generators respectively (Fig. 5). Output bus four from each FCC supplies the Digital Flight Data Acquisition Unit (DFDAU) (Ref 31-31-00).

(5) Output buses one, two, and three from each FCC, supply the MCP, MCDP, and EFIS symbol generators respectively.

(6) FCC - Analog Discrete Outputs

(a) GUI 115;
pin G11, of each FCC, supplies 28 vdc (engage hold) to its own ENGAGE lever (switch) in the MCP.

(b) Pin A6 supplies a ground to the electrical power (control) System when the AUTOLAND BUS ISOLATE REQUEST discrete is produced after triple channel APP selection (Ref 24-22-00).

(c) Pin F9 normally supplies 28 vdc, but changes to ground if the FCC detects any detent trip condition. The signal is fed to the other two FCCs.

(d) Pin G10 supplies 28 vdc to the other two FCCs when A/P servos are engaged. A ground is supplied when they are disengaged.

(e) Pin K2 supplies an A/P CAUTION - 1 ground which illuminates the AUTOPILOT amber caution light on the P1-3 panel.

(f) Pin K3 supplies an A/P CAUTION - 2 ground to the Engine Indication and Crew Alerting computer (Ref 31-41-00).

(g) Pin E12 (A/P WARN 2 - BAT) and pin E12 (A/P WARN 2) both supply grounds in the A/P WARN condition to two independent inputs of the aural warning module in the WEU (Ref 31-51-00).

(h) Pin G12 (A/P WARN 1 - BAT) and pin G12 (A/P WARN 1) both supply grounds in the A/P WARN condition to two pairs of bulbs in the red A/P DISC warning light on the P1-3 panel.

K. AFDS Inputs

(1) Each FCC has analog inputs as shown from its own autopilot lateral control servo, autopilot pitch control servo, autopilot rollout guidance servo, and from the respective (L, C, and R) flap and stabilizer position modules.

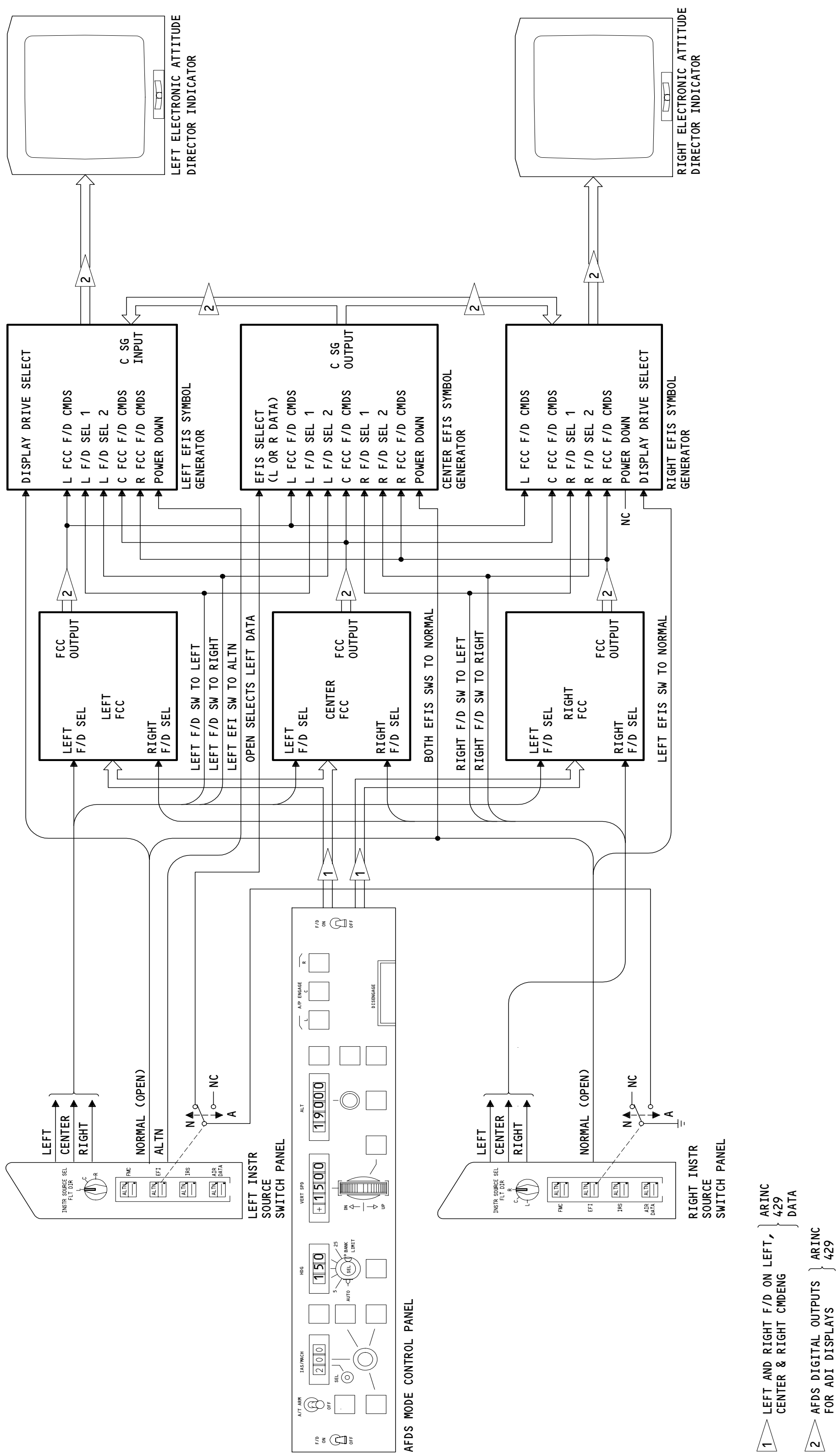
(2) FCC Digital Inputs and Cross Channel Lines.

(a) Cross-channel data is encoded in vendor unique code; all other data is transmitted in standard ARINC 429 format. Critical sensor data is transmitted on dedicated data links while non-critical data is stored or switched.

(b) The MCP has two identical micro-processors, A and B, which perform identical functions. The A processor receives ARINC 429 data from the left and center FCCs, Left FMC, and TMC. The B processor receives ARINC 429 Data from the right and center FCCs and TMC. The A processor transmits ARNIC 429 data to the left and center FCCs and TMC, and the B processor transmits ARNIC 429 data to the right FCC and FMC.
(c) Each FCC has independent data links with the MCDP.

(3) FCC Analog Discrete Inputs
(a) Pin H15 receives 28 vdc from the MCP when the A/P servos are armed. Pin C6 receives 28v dc from the hydraulic system low pressure light, on the P10 panel, when the light is off.
(b) Pin C7 changes from ground to open after the EPCS has completed Autoland power bus isolation.
(c) Pin A3 receives a ground from the MCDP when the ground test mode is selected.
(d) Pins C13, F3, or F1 receive a ground from the air/ground relay when the airplane is on the ground. Two of these three pins for each FCC is wired to the Air/Ground Systems 1 and 2.
(e) Pin H2 receives a ground or open when the MAG/TRUE switch is in the TRUE or MAG position respectively.
(f) Pin H11 receives a ground whenever the A/P DISC switch is pressed.
(g) Pin F15 (connector B) normally has a ground through both A/P DISC switches (in series) and receives an open whenever either switch is pressed.
(h) Pin K1 or K3 receives 28 volts SERVO ENG DC from another FCC (from pin G10 output).
(i) Pin E3 or G1 receives 28 volts DETENT TRIP from another FCC (from pin F9).
(j) Pins J10 and K10 receive ground whenever the RESET switch is pressed on either ASA.
(k) Pin G9 on both connectors receive 28 vdc autotrim valid from the Stabilizer Trim/Elevator Asymmetry Limit Module (SAM). The center FCC receives signals from the left and right SAMs. The right and left FCCs interface only with the applicable left or right SAM.
(l) Pin F10 or F9 receives a ground when the FCC switch on the left or right INSR SOURCE SEL panel selects the center FCC.
(m) Pin F15 (connector A) normally receives a ground when both go-around (GA) switches are relaxed and an open when either switch is pressed.
(n) Pins A13, G13, K13, J02 and C13 are program pins.

L. AFDS Switching (Fig. 5)
(1) Source Select Switches
(a) The FLT DIR switches on the left and right INSTR SOURCE SEL panels connect the selected FCC computed F/D commands and selected data to the associated EADI.
(b) The EFI switches on the left and right INSTR SOURCE SEL panels enable the NORMAL or ALTERNATE SG to process incoming AFDS and FMCS data.



AFDS Data Switching
Figure 5 (Sheet 1)

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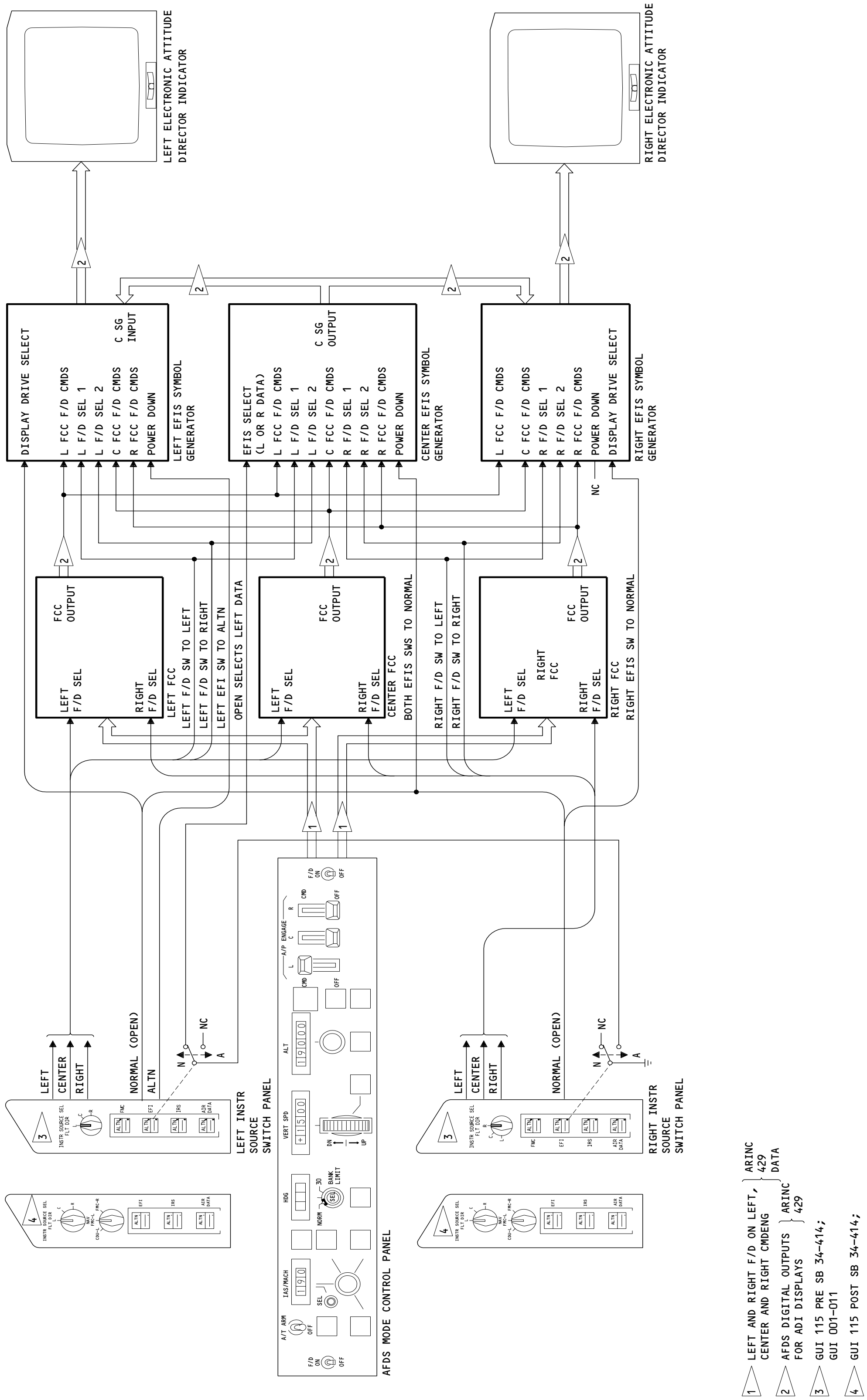


AFDS Instrument Switching Schematic
Figure 5 (Sheet 2)

(c) The switches on the left and right INSTR SOURCE SEL panels do not control IRU inputs to FCCs but they do control attitude display signals.
(d) The AIR DATA switches on the left and right INSTR SOURCE SEL panels do not change the ADC sources to the FCCs.

(2) MCP Switching
(a) Positioning the captain's or first officer's F/D switch to ON permits the FCCs to compute F/D commands and activates the EADI FMA. These commands can be generated simultaneously by an FCC engaged in CMD mode.

(3) F/D Switching
(a) The FLT DIR switch on the left (right) INSTR SOURCE SEL panel is a three-position, rotary switch which provides a ground to the left, center or right FCC. This enables that FCC to supply computed F/D signals to all SGs.
(b) The left LFT DIR switch supplies a ground to the left and center EFIS SGs when in the L or R position. These ground are identified as L F/D SEL 1 and L F/D SEL 2 respectively.
(c) The right FLT DIR switch supplies a ground to the right and center EFIS SGs when in the R or L position. These grounds are identified as R F/D SEL 2 and R F/D SEL 1 respectively.
(d) The FLT DIR switch ground logic allows the EFIS SGs to determine which F/D data input to use for the L or R F/D display (EADI). When a FLT DIR switch is in the C position, no F/D source ground is supplied.

(4) EFI Switching
(a) The EFI switches on the left and right INSTR SOURCE SEL panels are alternate action pushbutton switches. The legend ALTN (alternate) is displayed when the associated switch is pressed to select another Symbol Generator (SG). The EFI switches control the DISPLAY DRIVE SELECT inputs to the left and right EFIS SGs and the EFI select to the center EFIS SG. Each switch in the normal position supplies an open to the DISPLAY DRIVE SELECT input. This enables the left or right SG to select the source of video for data processing. If either EFI switch is in the ALTN position, the normal SG for that switch is disabled.
(b) The POWER DOWN input to the left SG is an open when the left switch is normal, but grounded when in ALTN. This ground disables the left SG. The POWER DOWN input to the center SG is grounded when the two EFI switches are normal. Thus, the center SG is disabled. If either EFI switch is in ALTN, the center SG is powered up. The right SG has a permanently open POWER DOWN input.

(c) The EFI SELECT input controls which data source is used by the center SG. This signal is routed through both EFI source select switches. When open, the center SG selects the left data source. This signal input is an open under all combinations of EFI switch positions except when the left switch is in normal and the right is in ALTN. If both switches are set at ALTN, the left has priority use of the center SG. The EICAS displays a yellow INSTR SW message, the master CAUTION light illuminates, and a single tone chime sounds.

(5) FCC Program Pin Connections (Fig. 6)

(a) The FCC program pins allow selection of different control laws and operational characteristics by either connecting an option pin to ground or leaving the pin not connected at the FCC shelf connector. The following description of the FCC program options covers the actual airplane configuration:

(b) FCC configuration - provides five pin connections at the shelf to identify the aircraft type. The FCC reads these pins as a binary code; there are 32 possible combinations. Pin 1 is grounded for all 757-200 airplanes.

(c) FCC Interlock - five pins are provided to prevent the utilization of FCC's with incompatible part numbers. Presently the interlock pins are only for the -109 and subsequent FCC's. Interlock protection to make sure other FCC dash numbers are not mixed is done in the FCC option 1.

(d) FCC Option 1

1) Customer Option Pin 1 (pin P1B-A12 grounded) - When this pin is grounded, the -103 FCC cannot be used with subsequent dash number FCCs. Also, the Customer Option Pin 1 includes these functions:

a) CHANGED BANK ANGLE LIMIT SPEED SCHEDULE: The bank angle limit speed schedule in the AUTO position is changed to give increased turn operation in the terminal area with airspeed between 200 and 250 knots.

b) CHANGED LOGIC FOR ENGINE-OUT INPUT: The operation of the engine-out logic during electrical power transfers is changed to prevent the command of nuisance V2 airspeed during F/D takeoff mode.

OPTION DESCRIPTION	BARRIER GROUND CONNECTIONS INSERT-PIN TO INSERT-PIN		1 = CONN TO GRD 0 = OPEN	MCDP TEST 56 READOUT
FCC CONFIGURATION - PIN 1	A-A12	A-A11	0	04 4
PIN 2	-B12	-B11	0	
PIN 3	-C12	-C11	1	
PIN 4	B-A11	B-G8	0	
PIN 5	-B11	-H8	0	
FCC INTERLOCK - PIN 1	B-C11	B-J8	0	00 4
PIN 2	-F11	-K9	0	
PIN 3	-D12	-D15	0	
PIN 4	-F12	-K7	0	
PIN 5	-H12	-J15	0	
FCC OPTION 1				06 4
CUSTOMER OPTION PIN 1	B-A12	B-B15	0	
CUSTOMER OPTION PIN 2	-B12	-D15	1	
CUSTOMER OPTION PIN 3	-C12	-E5	1	
CUSTOMER OPTION PIN 4	-A13	-A14	0	
CUSTOMER OPTION PIN 5	-B13	-B14	0	
FCC OPTION 2				37 4
A/P MODE ENGAGE	B-A3	B-E4	1	
G/S CAP INHIBIT	-H13	-J14	0	
FULL TIME NO LAND	-K13	-K14	1	
METHOD 2	-G13	-H14	0	
AUTOLAND STATUS ANNUNCIATOR	-C8	-E8	0	
LATERAL COMMAND ENGAGE	-C10	-C14	1	
FCC OPTION 3				31 4
CWS INHIBIT	B-J2	B-J4	1	
FULL TIME F/D	-D3	-K4	1	
F/D AUTOMATIC ON	-A4	A-D4	1	
ILS ANOM DLY	-D4	B-H4	1	
A/P ENGA TO/GA OPT	-G1	A-G2	1	
SYS ARCHITECTURE	-H1	-H2	0	
FCC OPTION 4 3				16 (MAG) 18 (TRUE) 4
GS RELAY STATUS	B-G2	A-G4	0	
MAG/TRUE IN	-H2	1	0 OR 1	
LOC RELAY STATUS	-D7	B-G4	0	
EO A/L INHIBIT	-G10	A-G8	0	
SINGLE SOURCE A/P	-J10	-E10	1	
PARITY (ODD) THE SUM OF THE GROUNDED PINS (TOGETHER WITH THE PARITY PIN) MUST BE EQUAL TO AN ODD NUMBER. YOU MUST GROUND THE PARITY PIN OR KEEP IT OPEN TO GET ODD PARITY.	B-C13	B-F4	L = 0 OR 1 C = 1 OR 0 R = 0 OR 1	2 4

FCC PIN CONFIGURATION (EXAMPLE) 4

FCC	CH IDENT 1	CH IDENT 2
LEFT	GROUND	OPEN
CENTER	GROUND	GROUND
RIGHT	OPEN	GROUND

CHANNEL IDENTIFICATION TABLE

1 CONNECTED TO MAG/TRUE HDG SWITCH ON P3-1 PANEL

2 THE PARITY SHOWS ON THE DISPLAY. IF THE PARITY IS INCORRECT, THE STATUS WILL BE SHOWN AS "ADD" OR "DEL" FOR THE ASSOCIATED CHANNEL.

3 FCC OPTION 4 GROUP NOT INCLUDED IN PARITY CHECK

4 ALL PIN CONNECTIONS AND "MCDP TEST 56 READOUT" VALUES ARE SHOWN AS AN EXAMPLE. FOR SPECIFIC VALUES FOR EACH AIRPLANE, REFER TO MCDP GROUND TEST 56 (FIM 22-00-04/101, FIG. 103).

FCC Program Pin Connections
Figure 6

c) CHANGED TAKEOFF AND GO-AROUND CONTROL LAW: The takeoff and go-around control law is changed to give better airplane guidance when there is windshear.

d) CHANGED HEADING SELECT CONTROL LAW: The heading select control law is changed to make the turn rate of the airplane better.

e) CHANGED FLIGHT LEVEL CHANGE (FLCH) CONTROL LAW: The FLCH control law is changed to make the climb performance during flap retraction better.

f) ADDED GLIDESLOPE ANOMALY ANNUNCIATION AND MAINTENANCE REPORTING: A glideslope anomaly detector and associated annunciation and maintenance reporting are added to decrease the risk of the autopilot track a disturbed glideslope beam.

g) CHANGED LNAV AND VNAV MODE ARMING LOGIC: The mode select logic is changed so that LNAV and VNAV are not armed while in the go-around mode below 400 feed radio altitude so that pitch or roll errors do not add up before LNAV or VNAV engagement.

h) CHANGED ALTITUDE CAPTURE LOGIC DURING TAKEOFF AND GO-AROUND: The altitude capture logic is changed to prevent altitude capture below 800 feet radio altitude during takeoff or go-around to prevent accidental altitude capture during windshear.

i) CHANGED NOSE LET-DOWN LOGIC AT TOUCHDOWN: The nose let-down logic is changed to keep pitch-up at touchdown to a minimum when the speed brakes are extended.

j) CHANGED FLIGHT MODE ANNUNCIATION: The flight mode annunciation is changed to remove ALPHA annunciations during FLCH and go-around modes.

k) CHANGED GO-AROUND LOGIC: The go-around logic is changed to allow a transition from takeoff mode to go-around mode.

l) ADDED F/D AUTO ON OPTION: A program pin is available which supplies flight director guidance by pilot operation of the takeoff/go-around switches if the flight director was not engaged before.

NOTE: This feature is part of the windshear protection system and the F/D AUTO ON program pin must be grounded for this feature to operate.

2) Customer Option Pin 1 (pin P1B-A12 grounded) and, Customer Option Pin 2 (pin P1B-B12 grounded) - When these pins are grounded, the -105 FCC cannot be used with subsequent dash number FCCs. The functions listed for Customer Option Pin 1 are still applicable. Also, the Customer Option Pin 2 includes these functions:
   a) CHANGED BACKCOURSE (BCRS) MODE LOGIC: Mode logic is changed to permit a change from the flight director takeoff mode directly to the flight director BCRS mode. This will make BCRS departures easier at airports where it is necessary to use these procedures.
   b) NUISANCE GLIDESLOPE ANOMALY TRIPPING DECREASED: The delay on the glideslope anomaly detector annunciation is increased from 10 seconds to 15 seconds to decrease the risk of nuisance glideslope beam error fault indications. Also, the glideslope anomaly detector is disconnected for flight director only approaches.
   c) CHANGE F/D AUTO-ON LOGIC: The F/D auto on feature is prevented below 80 knots indicated airspeed.
   d) RUDDER VALVE TEST PREVENTED FOR ENGINE-OUT: The rudder valve test is prevented if one engine has failed before approach to prevent nuisance NO LAND 3 annunciations because of rudder trim.
   e) CHANGED STABILIZER TRIM LOGIC: Nose up stabilizer mistrim at 330 feet AGL is prevented during LAND 3 operation.
   f) CHANGED HEADING SELECT LOGIC: The heading select logic is changed to prevent a wrong-way turn flight director command after a go-around.
   g) CHANGED MULTI-CHANNEL A/P ENGAGE LOGIC: Multi-channel autopilot engagement is prevented if there is an autopilot caution. This will prevent LAND 3 annunciation with the glideslope anomaly detector open.
   h) CHANGED ADC SELECT LOGIC: A latch delay is added to the Air Data Computer (ADC) select logic prevent inadvertent transfers to cross-channel ADC inputs.
3) The three remaining pins are provided for selecting customer unique configurations which are currently undefined.

(e) FCC Option 2
   1) A/P Mode Engage (Pin P1B-A3 grounded) - When the autopilot is initially engaged in command (CMD) mode, the autopilot will engage, with exception of Takeoff or Go-Around modes, into the current flight director modes. If the flight directors were initially OFF or the original flight director modes were Takeoff or Go-Around, the modes will become Vertical Speed(pitch) and Heading Hold (roll). Glideslope and localizer active modes will transition to Heading Hold and Vertical Speed modes, and the glideslope and localizer will be rearmed.

2) Autopilot Disengage Warning, Double/Single Push Reset (FCC -134 and prior)
   a) Whenever an A/P disengage warning occurs due to a control wheel disconnect switch action, the warning reset procedure depends on grounding of the option pin.
   b) Double Push Reset (P1B-H13 open) - If the autopilot warning light is on due to an autopilot disconnect from the wheel disconnect switch, the control wheel A/P disengage switch must be pressed and released a second time to reset the warning.
   c) Single Push Option (P1B-H13 grounded) - The autopilot disconnect warning is reset automatically after 2 seconds.
   d) Reset of the autopilot disconnect warning due to any other means of disconnect is achieved by pressing and releasing either control wheel A/P disengage switch. The warning continues until the switch is released.
3) Glideslope Capture Inhibit (FCC -135 and on)
   a) Baseline configuration: The Glideslope Capture is enabled prior to Localizer Capture (P1B-H13 open).
   b) Option: The Glideslope Capture is inhibited until the Localizer is captured (P1B-H13 grounded). This option is available only if FCC Interlock pins P1B-C11 and P1B-F11 are also grounded.
4) Full-Time No Land 3 Option
   a) This option controls the display capability of each ASA.
   b) Baseline configuration: (P1B-K13 open) The autoland capability is not displayed until the Approach (APP) mode is operative.
   c) Option: (P1B-K13 grounded) The autoland capability is displayed full-time.
5) Method 1/Method 2 Approach Mode Channel Selection
   a) At the selection of the APP mode, this option determines how off-line A/P channels are armed.
   b) GUI 001-114, 116-999;
      Method 1: (P1B-G13 open) All off-line channels automatically arm for multichannel operation and dot-bar matrices on the CMD ENGAGE switch/lights illuminate.
   c) GUI 115;
      Method 2: (P1B-G13 grounded) Multichannel arm requires that each engage lever be placed in CMD.
   d) GUI 001-114, 116-999;
      Method 2: (P1B-G13 grounded) Multichannel arm requires that each CMD ENGAGE switch/light be individually pressed.

6) GUI 115;
optional Autoland Status Annunciator (ASA) option (P1B-C8 grounded) permits use of an alternate type of ASA. The two types of ASAs are functionally equivalent. The alternate type uses different nomenclature on the panel face and prisms (STATUS, AUTOLAND 3, AUTOLAND 2, MANUAL LND).

7) Lateral Command Engage Option (LCEO)
   a) When the A/P is initially engaged in CMD or a F/D is turned on, the roll channel response depends upon this option.
   b) Baseline configuration: (P1B-C10 open) When the A/P is engaged in CMD, the airplane is rolled to wings level and the A/P engages into HDG HOLD mode regardless of the bank angle (B/A).
   c) Option: (P1B-C10 grounded) If the B/A is less than 5 degrees, A/P action is the same as in the baseline configuration. If the B/A is between 5 and 30 degrees at CMDENG, the A/P engages into Roll Attitude Hold. When the B/A is greater than 30 degrees at CMDENG, the A/P rolls the airplane back to 30 degrees and then engages into Roll Attitude Hold.

(f) FCC Option 3
1) CWS Inhibit Option
   a) This option controls whether control wheel steering function is available.
   b) Baseline Configuration: With the option pin not grounded, control wheel steering function is active.
   c) Option: With the option pin grounded, control wheel steering function is inhibited. The force transducers required for CWS are not installed when option pin is grounded.
2) Full Time Flight Director Option
   a) This option controls the flight director command displays of each EADI.
   b) Baseline configuration: The flight director bars associated with the engaged channel will bias out of view.
   c) Option: The flight director bars are displayed for the engaged channel.
3) Flight Director Automatic On Option
   a) Flight director engage requests will be automatically generated by the FCC's in the go-around mode. This option supports windshear capabilities.
4) Flight Director Display on Rollout Option
   a) When the A/P is in MCHENG and in the ROLLOUT mode, F/D command bar display depends upon this option.
   b) Baseline configuration: The F/D command bar display is biased from view during rollout.

(g) Two pin connections of the shelf identify the FCC as L, C, or R channel. The binary code used is shown in the table of Figure 10.

(h) The parity pin provides odd parity for the total of program pin ground connections to meet ARINC 429 standards.

2. Component Details

A. The flight control computer, AFCS mode control panel, AFDS go-around switch and A/P disconnect switch are described in Autopilot/Flight Director Power - Description and Operation (22-11-00).

B. The autopilot pitch control servos and elevator neutral shift transducers are described in Autopilot/Flight Director Pitch Channel - Description and Operation (22-12-00).

C. The autopilot rollout guidance servos and autopilot lateral control servos are described in Autopilot/Flight Director Roll and Yaw Channel - Description and Operation (22-13-00).

D. The autoland status annunciator, autopilot caution light, and A/P disconnect warning light are described in Autopilot/Flight Director Warning and Annunciator - Description and Operation (22-14-00).

E. The flight control computers digital cross channel data buses are described in Autopilot/Flight Director Interchannel Data - Description and Operation (22-15-00).

3. Operation

A. Functional Description

(1) General (Fig. 7)

ENTRY CONFIGURATION	ACTIVE CONFIGURATION	EXIT CONFIGURATION
ARM, CMD	OFF (PITCH & ROLL)	ARM
OFF	ARM (PITCH & ROLL)	OFF, CMD
ARM	CMD (PITCH & ROLL)	OFF
ARM (ROLLOUT), CMD (ROLLOUT)	OFF (ROLLOUT)	ARM (ROLLOUT)
OFF	ARM (ROLLOUT)	OFF, CMD (ROLLOUT)
ARM (ROLLOUT	CMD (ROLLOUT)	OFF

PITCH & ROLL CONFIGURATIONS ARE APPLICABLE TO SINGLE CHANNEL COMMAND ENGAGE OPERATIONS. ROLLOUT CONFIGURATIONS ARE APPLICABLE TO MULTI-CHANNEL COMMAND ENGAGE OPERATIONS, PROVIDING PITCH, ROLL & YAW CONTROL/GUIDANCE OUTPUTS FOR AUTOLAND APPROACHES.

AFDS Configurations
Figure 7

(a) These AFDS configurations define, in parentheses, the type of guidance or control provided. Transition into and out of the ACTIVE configuration is possible as defined in the table. In multichannel engage operation, each FCC may be in a different configuration before all FCCs become CMD (ROLLOUT).

(2) Single Channel Mode Functions (Fig. 8)

(a) The table shows the basic control functions for modes which can be active during single channel operation. Simultaneous pitch and roll modes are those in which both pitch and roll control functions are enabled at mode selection. Pitch and roll axes control laws are not variable by pilot action.

(b) Pitch and roll command and F/D modes may be independently selected and allow pilot control interface actions.

(3) Flight Director Only Block Diagram (Fig. 9)

(a) In the flight director only mode, either or both F/Ds are on and all A/P channels are disconnected. The F/D can use stand-alone reference inputs (uncoupled FMC) or FMC steering signals (coupled FMC) derived from a selected flight plan.

(b) The FCC generates F/D steering commands which are converted into video signals in the EFIS SG and displayed on the EADI. The F/D mode annunciation is also indicated on the EADI. The pilot controls the airplane attitude to follow F/D commands with control wheel and column movement.

(c) In the F/D only mode, surface position (aileron, elevator, rudder) feedback is used to synchronize the servo commands to null. All other signal loops are synchronized in the FCC. Hydraulic power is always required to move control surfaces. In the F/D only mode, solenoids in the actuators permit mechanical input signals only to control surface position. Autopilot servo commands are not used.

(d) The AFDS is used in this condition for takeoff. Only wings level, and rotation target angle guidance is available.

(4) Single Channel Command Engage (Fig. 10)

(a) In the single channel CMD ENG mode, the A/P receives inputs from the MCP, FMC steering commands, or pilot-selected navigational commands.

MODE CLASSIFICATION	MODES	FUNCTIONS	ANNUNCIATED ON FLIGHT MODE ANNUNCIATOR
SIMULTANEOUS PITCH AND ROLL MODES	CONTROL INACTIVE	NO SERVO OR F/D COMMANDS. ALL CHANGES SYNCHRONIZE ALL CHANNELS MONITORED FOR MODE ENGAGEMENT CAPABILITY	___
	TAKEOFF	F/D ONLY MODE, AUTOMATICALLY SELECTED ON GROUND. ROLL F/D CMDS FOR WINGS LEVEL UNTIL LIFTOFF, THEN TRACK HOLD PITCH F/D CMDS FOR ROTATION ATTITUDE, THEN SAFE CLIMBOUT SPEED	YES YES
F/D OR CMD	GO AROUND	ROLL CMDS GROUND TRACK HOLD PITCH CMDS FOR SAFE CLIMB RATE USING SPEED SCHEDULE	YES YES
	B/CRS MODE [2]	INDEPENDENT OF A/P OR F/D EXCEPT AFTER LOC CAPTURE	YES
PITCH MODES (F/D OR CMD)	VERTICAL SPEED	BASIC F/D OR A/P CMD SELECTION MODE CONTROLS TO SELECTED V/S	YES
	ALTITUDE ACTIVE	ALTITUDE HOLD & ALTITUDE SELECT	YES
	FLIGHT LEVEL CHANGE	CONTROLS TO IAS OR MACH SELECTED, DURING CLIMB OR DESCENT TO SELECTED ALTITUDE. PROVIDES FLAP PLACARD AND MINIMUM SPEED PROTECTION	YES
	VNAV MODE	CONTROLS TO VERTICAL STEERING COMMANDS FROM FMC	YES
	GLIDESLOPE ACTIVE	GLIDESLOPE CAPTURE & TRACK	YES
ROLL MODES (F/D OR CMD)	HEADING HOLD	BASIC F/D OR A/P COMMAND SELECTION MODE [1]	YES
	HEADING SELECT	CONTROLS TO SELECTED HEADING	YES
	LNAV MODE	CONTROLS TO HORIZONTAL (LATERAL) STEERING CMDS FROM FMC	YES
	LOC ACTIVE	CAPTURE & TRACK LOCALIZER CENTERLINE	YES
	B/CRS LOC ACTIVE [2]	CAPTURE & TRACK BACKCOURSE LOCALIZER CENTERLINE	YES

[1] FUNCTION MAY INCLUDE ROLL ATTITUDE HOLD DEPENDING ON LATERAL CMD ENGAGE OPTION

[2] GUI 001-114,116-999

Single Channel Mode Functions
Figure 8

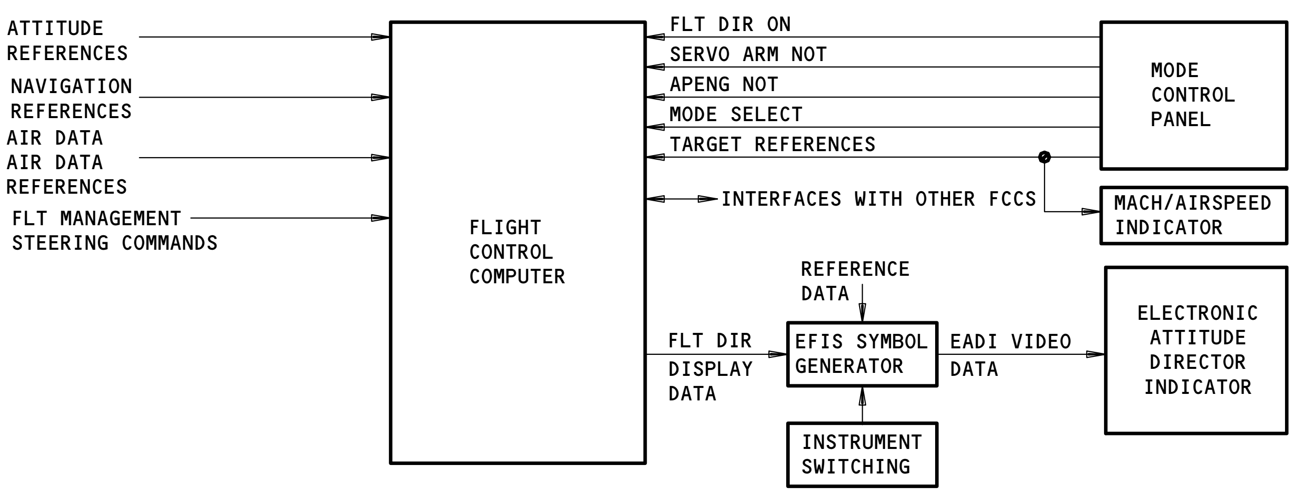
(b) Reference inputs come from the sensor system, FMC or MCP selected target references, position feedback from the A/P servos, and mechanical control feedback to the control wheel and column.

(c) GUI 010-114, 116-999;
the FCC outputs in CMD ENG include servo commands to A/P servos, stabilizer trim commands, annunciaion discretes to the EFIS, and caution and warning discretes. Flight director bar commands to the EADI are available in CMD ENG.

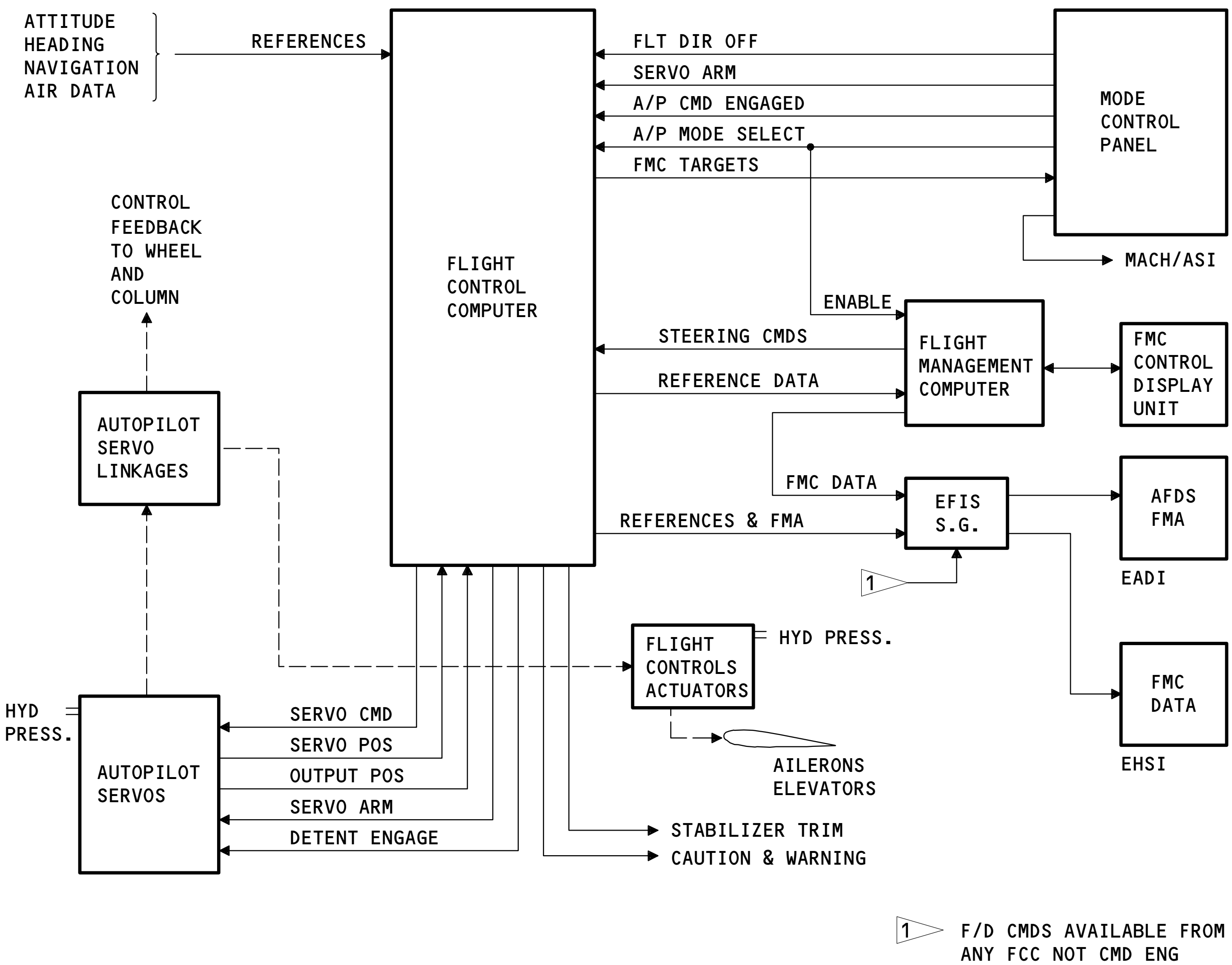
(d) GUI 001-009, 115;
the FCC outputs in CMD ENG include servo commands to A/P servos, stabilizer trim commands, annunciaion discretes to the EFIS, and caution and warning discretes. Flight director bar commands to the EADI are inhibited in CMD ENG.

(e) The CMD ENG mode is capable of all climb, cruise, and descent control. It is capable of single channel ILS approach control except for the runway alignment, flare, landing, and rollout control.

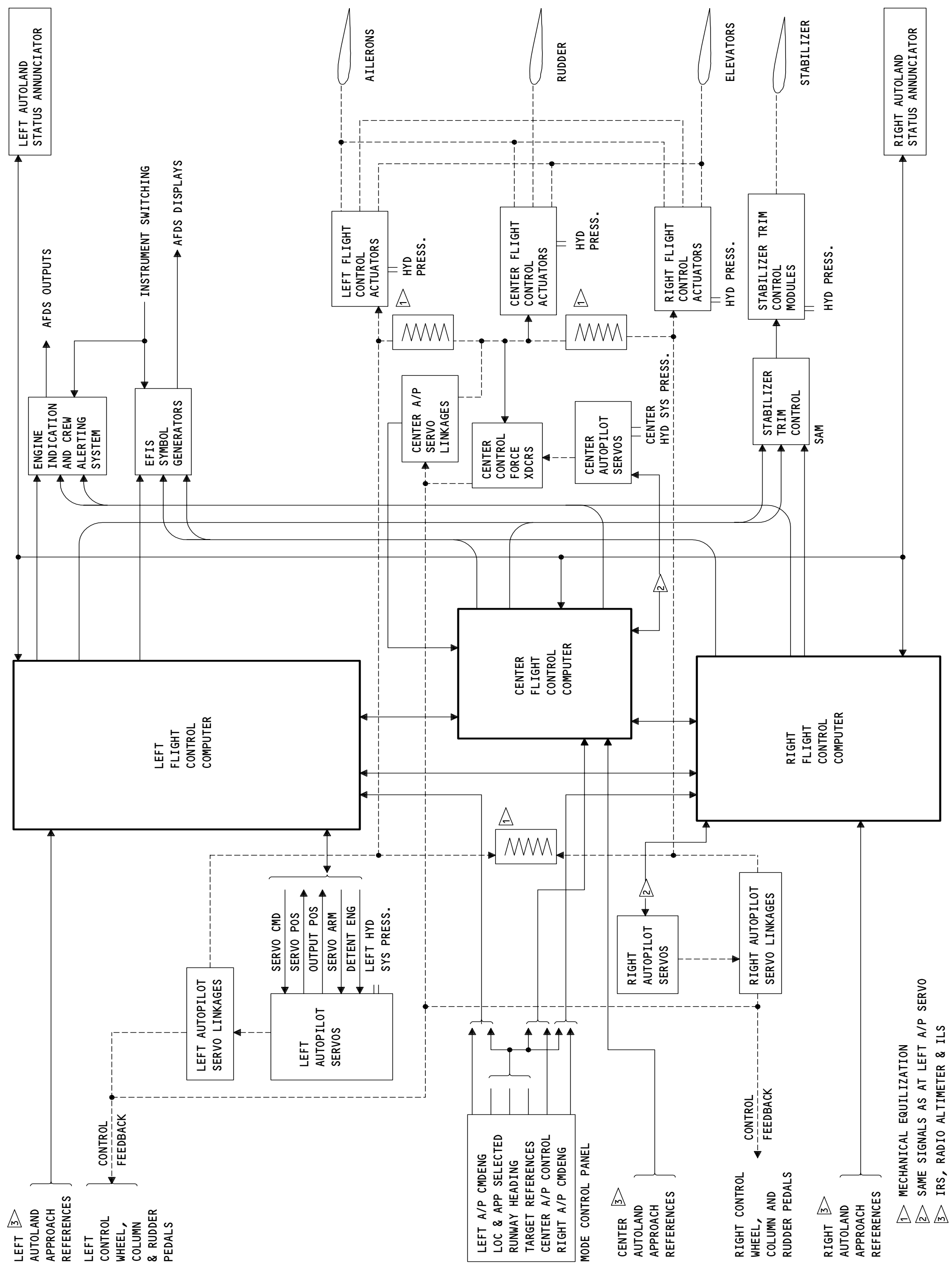
(5) Multichannel Engage (Autoland) (Fig. 11)



Flight Director Only - Block Diagram
Figure 9



Single Channel Command Engage Block Diagram
Figure 10



Multi-Channel Engage (Autoland) Block Diagram
Figure 11

(a) Each FCC has dedicated interfaces with A/P reference sensors, A/P servos (LCS, RGS, PCS), EFIS symbol generators, the caution and warning system, and on electrical power source. Each FCC receives data from the AFCS MCP and each of the other FCCs.

(b) All A/P servos of one channel are powered from that channel's hydraulic system and provide mechanical inputs to the respective flight control actuator.

(c) Each FCC supplies data to the engine indication and crew alerting system, the A/P DISC (red) warning light, the AUTOPILOT (amber) caution light and the stabilizer trim control system.

(d) GUI 001-114, 116-999;
autoland requires 3 or 2 A/Ps to be CMDENG. Autoland status is indicated on the Autoland Status Annunciator (ASA) as LAND 3 (green), LAND 2 (green), NO LAND 3 (amber), and NO AUTOLND (amber).

(e) GUI 115;
autoland requires 3 or 2 A/Ps to be CMDENG. Autoland status is indicated on the Autoland Status Annunciator (ASA) as AUTOLAND 3 (green), AUTOLAND 2 (white), and MANUAL LND (amber).

(f) The AFDS status is displayed on the EADIs. The AFDS flight mode annunciation (FMA) display indicates roll and pitch modes that are armed or engaged.

1) GUI 010-114, 116-999;
flight director commands are available from a CMDENG channel.

2) GUI 001-009, 115;
flight director commands are not available from a CMDENG channel.

(6) Autoland Categories and Configurations (Fig. 12)

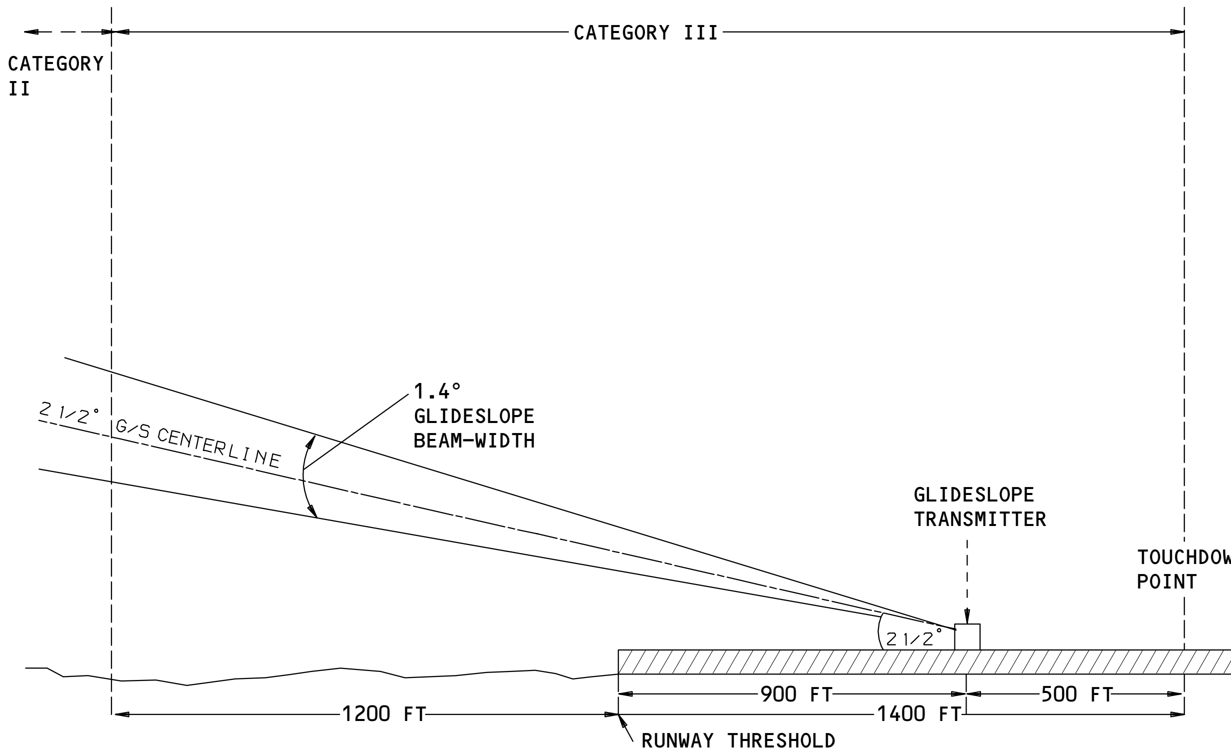
(a) The AFDS is capable of controlling the airplane in Category III approaches, depending on the ground facilities available. A typical arrangement is as follows.

1) The G/S transmitter is approximately 900 feet from the runway threshold and transmits a beam with a centerline inclined at a minimum of 2 1/2 degrees to the horizontal.

2) The AFDS controls the airplane to touchdown 500 feet beyond the G/S transmitter. Lateral and landing rollout control is guided by the localizer transmitter at the far end of the runway.

(b) A Category III approach is divided into three sub-categories (A, B, C) according to the runway visual range (RVR).

(c) GUI 001-114, 116-999;
the Fail Operational configuration has three A/P channels hydraulically engaged in roll, pitch, and yaw. All three use independent data and independent electrical and hydraulic power systems. The ASA shows LAND 3 (green).



AUTOLAND CATEGORY	DH	RVR
I	200	≥1800
II	100	≥1200
IIIA	0	≥700
IIIB	0	≥150
IIIC	0	≥0

DH= DECISION HEIGHT
RVR= RUNWAY VISUAL RANGE

Autoland Categories and Configurations
Figure 12

(d) GUI 115;
the Fail Operational configuration has three A/P channels hydraulically engaged in roll, pitch, and yaw. All three use independent data and independent electrical and hydraulic power systems. The ASA shows AUTOLAND 3 (green).

(e) GUI 001-114, 116-999;
The Fail Passive configuration has 2 or 3 channels hydraulically engaged in roll, pitch, and yaw, using two independent sources of data, and electrical and hydraulic power. The ASA shows LAND 2 (green).

(f) GUI 115;
The Fail Passive configuration has 2 or 3 channels hydraulically engaged in roll, pitch, and yaw, using two independent sources of data, and electrical and hydraulic power. The ASA shows AUTOLAND 2 (white).

(g) Autoland configurations are not active until each FCC recognizes LOC and G/S capture; the airplane is below 1500 feet radio altitude; and other autoland logic requirements are met. Autoland configurations provide control for runway alignment, flare, landing, and rollout.

(7) Autoland Functional Sequence (Fig. 13)

(a) The left (L) A/P is CMDENG (pitch and roll) on ILS approach. The LOC has been captured and the A/P is in ALT HOLD prior to glideslope intercept. The center (C) and right (R) FCCs supply captain's and F/O's F/D displays.

(b) The APP mode enables multichannel engage (MCHENG). With the method 2 option active, the pilot must place the engage switches in CMD for the C and R A/Ps. This arms all A/Ps for MCHENG and makes the L FCC the first channel in CMD. The FMA on the EADI now displays roll and pitch ARM modes.

(c) GUI 001-114, 116-999;
multichannel engage occurs when all A/Ps have detected LOC and G/S capture and 1500 feet from a valid radio altimeter (RA). The first channel in command must receive acceptable CMDENG criteria from the other two FCCs. All A/Ps engage pitch, roll, and yaw channels. The ASA displays LAND 3 (Fail Operational) or LAND 2 (Fail Passive). All phases of autoland are possible with LAND 3 or LAND 2 displayed.

(d) GUI 115;
multichannel engage occurs when all A/Ps have detected LOC and G/S capture and 1500 feet from a valid radio altimeter (RA). The first channel in command must receive acceptable CMDENG criteria from the other two FCCs. All A/Ps engage pitch, roll, and yaw channels. The ASA displays AUTOLAND 3 (Fail Operational) or AUTOLAND 2 (Fail Passive). All phases of autoland are possible with AUTOLAND 3 or AUTOLAND 2 displayed.

(e) If the current first-channel-in-command cannot provide autostabilizer trim control, automatic selection of a new first-in-command is made by a prescribed sequence (i.e., L, R, C, L...).

REQUIREMENTS/CONDITIONS FOR FAIL-OPERATIONAL AUTOLAND	LEFT AUTOPILOT			CENTER AUTOPILOT			RIGHT AUTOPILOT		
	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS
L A/P CMDENG • LOC CAPT • GS ENG.HR>1500•ALT HOLD. CAPT F/D ON USING CFCC F/O F/D ON USING RFCC (1)	CONTROLS LEFT A/P LATERAL CONTROL SERVO FOR LOCALIZER TRACKING	CONTROLS LEFT A/P PITCH SERVO FOR BARO ALT HOLD	SYNCHRONIZING TO RUDDER POSITION	A/P CH SYNCS TO AIL POSN. SUPPLIES F/D CMDS FOR LOCALIZER TRACKING	A/P CH SYNCS TO ELEVATOR POSN. SUPPLIES F/D CMDS FOR ALT HOLD	SYNCS TO RUDDER POSITION	AS FOR CENTER A/P	AS FOR CENTER A/P	AS FOR CENTER A/P
AS ABOVE, PILOT PRESSES APP SWITCH (METHOD 2 OPTION) (2)	AS ABOVE - L FCC REMAINS CMDENG BUT ALSO IS MULTI-CHANNEL ARM			AS ABOVE - PILOT PRESSES CMD SWITCH, (2) CENTER A/P BECOMES MULTI-CHANNEL ARM			AS ABOVE - PILOT PRESSES CMD SWITCH, (2) RIGHT A/P BECOMES MULTI-CHANNEL ARM		
AS ABOVE, BUT NOW GSENG AND ALT HOLD.	AS ABOVE	CONTROLS LEFT A/P PITCH SERVO FOR GLIDESLOPE CAPTURE & TRACK	AS ABOVE	AS ABOVE	A/P CH SYNCS F/D CMDS FOR G/S CAPT & TRACK	AS ABOVE	AS ABOVE	AS FOR CENTER A/P	AS ABOVE
AS ABOVE, BUT HR < 1500FT. ALL VALID A/PS ARE MCHENG. ROLL & YAW CH ENTER ROLLOUT ARM PITCH CH ENTERS FLARE ARM. ASA SHOWS LAND 3 (GREEN)	AS ABOVE, PLUS LIMITED BANK CMD FOR RUNWAY ALIGNMENT. ENTER ROLLOUT ARM MODE	AS ABOVE, ENTERS FLARE ARM MODE. ADJUSTS PITCH ATTITUDE DUR-ING APPROACH SPEED AND CONFIGURATION CHANGES. RE-MAINS IN CON-TROL OF AUTO STAB TRIM. FOLLOWS C FCC AS FIRST IN CMD	CONTROLS A/P ROLLOUT GUIDANCE SERVO DURING RUNWAY ALIGNMEMT, PROVIDING YAW IF BANK CAN-NOT MAINTAIN LOC CENTER-LINE	ENGAGES TO CONTROL CENTER A/P LATERAL CONTROL SERVO, USING CENTER A/P APPROACH REFERENCES IN ROLLOUT ARM MODE. F/D CMDS DISABLED	ENGAGES TO CONTROL CENTER A/P PITCH SERVO, USING APPROACH REF-ERENCES IN FLARE ARM MODE. F/D CMDS DIS-ABLED. FOLLOWS RIGHT FCC AS FIRST IN CMD FOR AUTO STAB TRIM CONTROL IF RIGHT A/P DISCONNECTS	ENGAGES TO CONTROL CENTER A/P ROLLOUT GUIDANCE SERVO DURING ROLLOUT ARM MODE	ENGAGES TO CONTROL RIGHT A/P LATERAL CONTROL SERVO USING RIGHT A/P APPROACH REF-ERENCES IN ROLLOUT ARM MODE. F/D CMDS DISABLED	ENGAGES TO CONTROL RIGHT PITCH SERVO, USING RIGHT A/P REFERENCES IN CMD FOR STAB TRIM CONTROL, IF LEFT A/P DISCONNECTS	ENGAGES TO CONTROL RIGHT A/P ROLLOUT GUIDANCE SERVO DURING ROLLOUT ARM MODE
AS ABOVE, BUT WHEN HR<50FT PITCH CH ENTERS FLARE ENGAGE AND GSENG.	AS ABOVE, BUT AILERON COMMANDS RE-DUCE AS ALT-ITUDE REDUCES	CONTROLS PITCH ATTITUDE TO RE-DUCE DESCENT RATE TO 2 FT/SEC AT TOUCH-DOWN. TD POINT TARGET IS 1400 FT FROM RUNWAY THRES-HOLD APPROX.	AS ABOVE		AS FOR LEFT A/P			AS FOR LEFT A/P	
TOUCHDOWN ROLL CH ENTERS ROLLOUT ENGAGE. PITCH CH DISENGAGES FLARE MODE	COMMANDS WINGS LEVEL	APPLIES NOSE DOWN PITCH CMD SO NOSEWHEEL STEERING CAN BE EFFECTIVE	MAINTAINS RUD-DER CONTROL DURING HIGH SPEED SECTOR OF LANDING		AS FOR LEFT A/P			AS FOR LEFT A/P	
LANDING ROLL - CONTROL APPLIED UNTIL AUTOPILOT DISCONNECT	AS ABOVE	AS ABOVE	PROVIDES NOSE-WHEEL STEERING CMDS TO MAIN-TAIN RUNWAY CENTERLINE		AS FOR LEFT A/P			AS FOR LEFT A/P	
GO-AROUND REMAINS MCHENG UNTIL EXIT CONFIGURATION IS COMPLETE	CONTROLS TO HOLD GROUND TRACK	CONTROLS FOR SAFE CLIMB RATE USING SPEED/ FLAPS REF	CONTROLS TO HOLD GROUND TRACK		AS FOR LEFT A/P			AS FOR LEFT A/P	

(1) L A/P ASSUMED TO BE CMDENG INITIALLY. C OR R A/P COULD ALSO BE CMD INITIALLY.

(2) IF METHOD 1 OPTION IS ACTIVE, OTHER CMDENG A/PS AUTOMATICALLY BECOME MULTICHANNEL ARM AND MCP CMD SWITCHES ILLUMINATE.

Autoland Functional Sequence
Figure 13 (Sheet 1)

A70354

REQUIREMENTS/CONDITIONS FOR FAIL-OPERATIONAL AUTOLAND	LEFT AUTOPILOT			CENTER AUTOPILOT			RIGHT AUTOPILOT		
	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS
L A/P CMDENG • LOC CAPT • GS ENG.HR>1500•ALT HOLD. CAPT F/D ON USING CFCC F/O F/D ON USING RFCC [1]	CONTROLS LEFT A/P LATERAL CONTROL SERVO FOR LOCALIZER TRACKING	CONTROLS LEFT A/P PITCH SERVO FOR BARO ALT HOLD	SYNCHRONIZING TO RUDDER POSITION	A/P CH SYNCS TO AIL POSN. SUPPLIES F/D CMDS FOR LOCALIZER TRACKING	A/P CH SYNCS TO ELEVATOR POSN. SUPPLIES F/D CMDS FOR ALT HOLD	SYNCS TO RUDDER POSITION	AS FOR CENTER A/P	AS FOR CENTER A/P	AS FOR CENTER A/P
AS ABOVE, PILOT PRESSES APP SWITCH (METHOD 2 OPTION)	AS ABOVE - L FCC REMAINS CMDENG BUT ALSO IS MULTI-CHANNEL ARM			AS ABOVE - PILOT PLACES ENGAGE SWITCH IN CMD, CENTER A/P BECOMES MULTI-CHANNEL ARM			AS ABOVE - PILOT PLACES ENGAGE SWITCH IN CMD, RIGHT A/P BECOMES MULTI-CHANNEL ARM		
AS ABOVE, BUT NOW GSENG AND ALT HOLD.	AS ABOVE	CONTROLS LEFT A/P PITCH SERVO FOR GLIDESLOPE CAPTURE & TRACK	AS ABOVE	AS ABOVE	A/P CH SYNCS F/D CMDS FOR G/S CAPT & TRACK	AS ABOVE	AS ABOVE	AS FOR CENTER A/P	AS ABOVE
AS ABOVE, BUT HR < 1500FT. ALL VALID A/PS ARE MCHENG. ROLL & YAW CH ENTER ROLLOUT ARM PITCH CH ENTERS FLARE ARM. ASA SHOWS AUTOLAND 3 (GREEN)	AS ABOVE, PLUS LIMITED BANK CMD FOR RUNWAY ALIGNMENT. ENTER ROLLOUT ARM MODE	AS ABOVE, ENTERS FLARE ARM MODE. ADJUSTS PITCH ATTITUDE DURING APPROACH SPEED AND CONFIGURATION CHANGES. REMAINS IN CONTROL OF AUTO STAB TRIM. FOLLOWS C FCC AS FIRST IN CMD	CONTROLS A/P ROLLOUT GUIDANCE SERVO DURING RUNWAY ALIGNMEMT, PROVIDING YAW IF BANK CANNOT MAINTAIN LOC CENTERLINE	ENGAGES TO CONTROL CENTER A/P LATERAL CONTROL SERVO, USING CENTER A/P APPROACH REFERENCES IN ROLLOUT ARM MODE. F/D CMDS DISABLED	ENGAGES TO CONTROL CENTER A/P PITCH SERVO, USING APPROACH REFERENCES IN FLARE ARM MODE. F/D CMDS DISABLED. FOLLOWS RIGHT FCC AS FIRST IN CMD FOR AUTO STAB TRIM CONTROL IF RIGHT A/P DISCONNECTS	ENGAGES TO CONTROL CENTER A/P ROLLOUT GUIDANCE SERVO DURING ROLLOUT ARM MODE	ENGAGES TO CONTROL RIGHT A/P LATERAL CONTROL SERVO USING RIGHT A/P APPROACH REFERENCES IN ROLLOUT ARM MODE. F/D CMDS DISABLED	ENGAGES TO CONTROL RIGHT PITCH SERVO, USING RIGHT A/P REFERENCES IN CMD FOR STAB TRIM CONTROL, IF LEFT A/P DISCONNECTS	ENGAGES TO CONTROL RIGHT A/P ROLLOUT GUIDANCE SERVO DURING ROLLOUT ARM MODE
AS ABOVE, BUT WHEN HR<50FT PITCH CH ENTERS FLARE ENGAGE AND GSENG.	AS ABOVE, BUT AILERON COMMANDS REDUCE AS ALTITUDE REDUCES	CONTROLS PITCH ATTITUDE TO REDUCE DESCENT RATE TO 2 FT/SEC AT TOUCHDOWN. TD POINT TARGET IS 1400 FT FROM RUNWAY THRESHOLD APPROX.	AS ABOVE		AS FOR LEFT A/P			AS FOR LEFT A/P	
TOUCHDOWN ROLL CH ENTERS ROLLOUT ENGAGE. PITCH CH DISENGAGES FLARE MODE	COMMANDS WINGS LEVEL	APPLIES NOSE DOWN PITCH CMD SO NOSEWHEEL STEERING CAN BE EFFECTIVE	MAINTAINS RUDDER CONTROL DURING HIGH SPEED SECTOR OF LANDING		AS FOR LEFT A/P			AS FOR LEFT A/P	
LANDING ROLL - CONTROL APPLIED UNTIL AUTOPILOT DISCONNECT	AS ABOVE	AS ABOVE	PROVIDES NOSEWHEEL STEERING CMDS TO MAINTAIN RUNWAY CENTERLINE		AS FOR LEFT A/P			AS FOR LEFT A/P	
GO AROUND REMAINS MCHENG UNTIL EXIT CONFIGURATION IS COMPLETE	CONTROLS TO HOLD GROUND TRACK	CONTROLS FOR SAFE CLIMB RATE USING SPEED/FLAPS REF	CONTROLS TO HOLD GROUND TRACK		AS FOR LEFT A/P			AS FOR LEFT A/P	

[1] L A/P ASSUMED TO BE CMDENG INITIALLY.
C OR R A/P COULD ALSO BE CMD INITIALLY.

Autoland Functional Sequence
Figure 13 (Sheet 2)

A70355

(f) GUI 001-114, 116-999;
if a critical sensor fails during a Fail-Operational approach, the ASA shows NO LAND 3, and the FMA displays a mode fault annunciation. When the pilot presses the ASA reset, the AFDS reverts to a FAIL-PASSIVE approach and the ASA displays LAND 2.

(g) GUI 115;
if a critical sensor fails during a Fail-Operational approach, the ASA shows AUTOLAND 2, and the FMA displays a mode fault annunciation. When the pilot presses the ASA reset, the AFDS reverts to a FAIL-PASSIVE approach and the ASA displays AUTOLAND 2 and the FMA removes the mode fault annunciation.

(h) The A/P channel having its own dedicated sensor invalid selects a valid equivalent sensor input from the cross-channel data bus and the A/P remains MCHENG. If this channel is FIRST IN CMD, it remains FIRST IN CMD. Loss of two similar sensor signals disables autoland capability. The autopilot disengages and the ASA goes blank. The FMA displays fault modes.

(i) Single channel A/P disconnect causes the autoland configuration to revert to FAIL-PASSIVE.

(j) In single channel ILS approach, no runway alignment, flare, or rollout control or guidance is available.

(k) The Autoland configuration is normally disconnected by pressing the A/P disconnect switch on either control wheel just before runway turnoff. It is automatically disconnected by an ENGAGE REMOVAL FAULT.

(l) When disconnect warnings occur, they must be reset. The A/P disengage warning reset depends on whether the one-push or two-push option is active. The autopilot stays in a multichannel configuration when Go-Around (GA) is selected. When exit from GA occurs, MCHENG reverts to single channel CMDENG (FIRST IN CMD).

(8) Control and Display

(a) Control

1) Automatic navigation is available from selectable reference sources in single channel CMD engage for climbout, cruise, descent, and ILS approach.

2) When two or three A/P channels are engaged in CMD, Autoland approaches are possible.

3) The autopilot is not certified for takeoff. The takeoff mode is a Flight Director only mode.

4) In A/P stand-alone (no F/D is on) operation, Lateral Navigation (LNAV) is accomplished through the following modes: heading hold, heading select, localizer only, approach guidance and rollout, and turbulance penetration.

5) When coupled with the Flight Management Computers (FMCs), the LNAV mode utilizes VOR (Very high frequency Omnidirectional Range), IRS (Inertial Reference System), and stored flight plan information.

6) In A/P stand-alone operation, Vertical Navigation (VNAV) is accomplished with the following modes:

a) Pitch altitude hold

b) Autostabilizer trim - through Stabilizer Trim Elevator Asymmetry Module (SAM)
c) Vertical speed select and hold
d) Airspeed/Mach select and hold
e) Go-around
f) Approach guidance, flare, and landing
g) GUI 001-114, 116-999;
Glideslope with backcourse

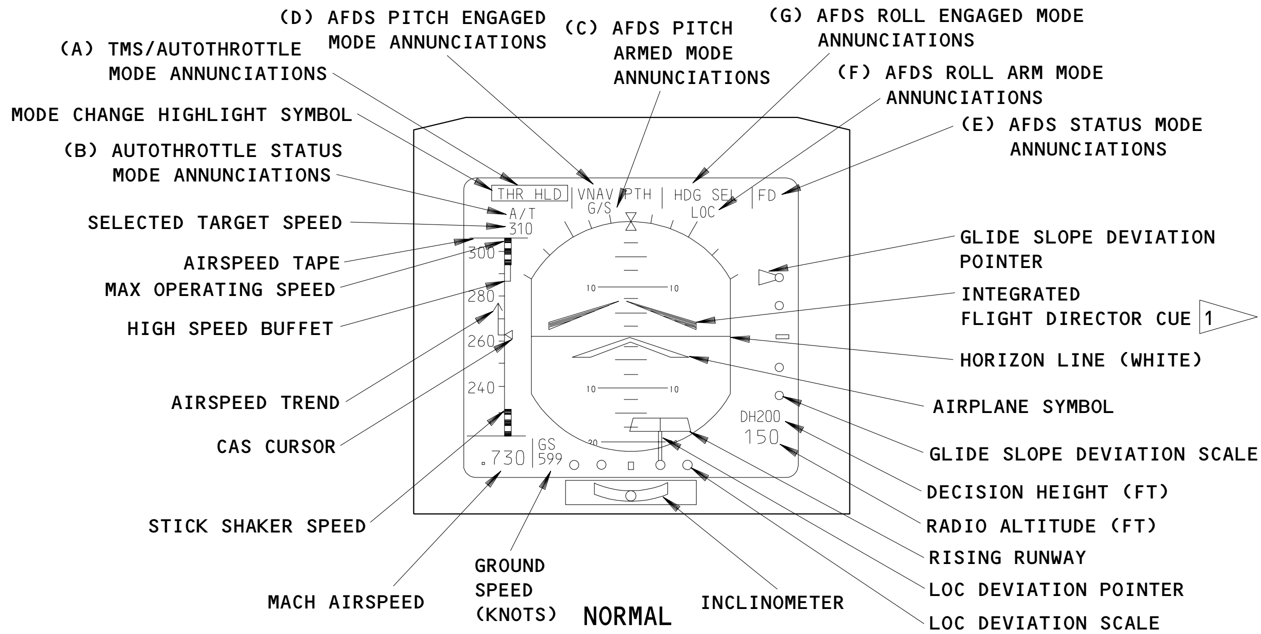
7) When coupled to the FMCS, the VNAV mode uses the following references:
a) Stored flight plan incorporating Standard Terminal Arrival Routes (STARs) and Standard Instrument Departures (SIDs).
b) Computed flight plan depending on the IAS/MACH mode selected.
c) Computed flight plan depending on the FMCs selected mode.
d) IRS.

8) The FCC roll modes do not use any functions associated with Stabilizer Trim Elevator Asymmetry Module (SAM).

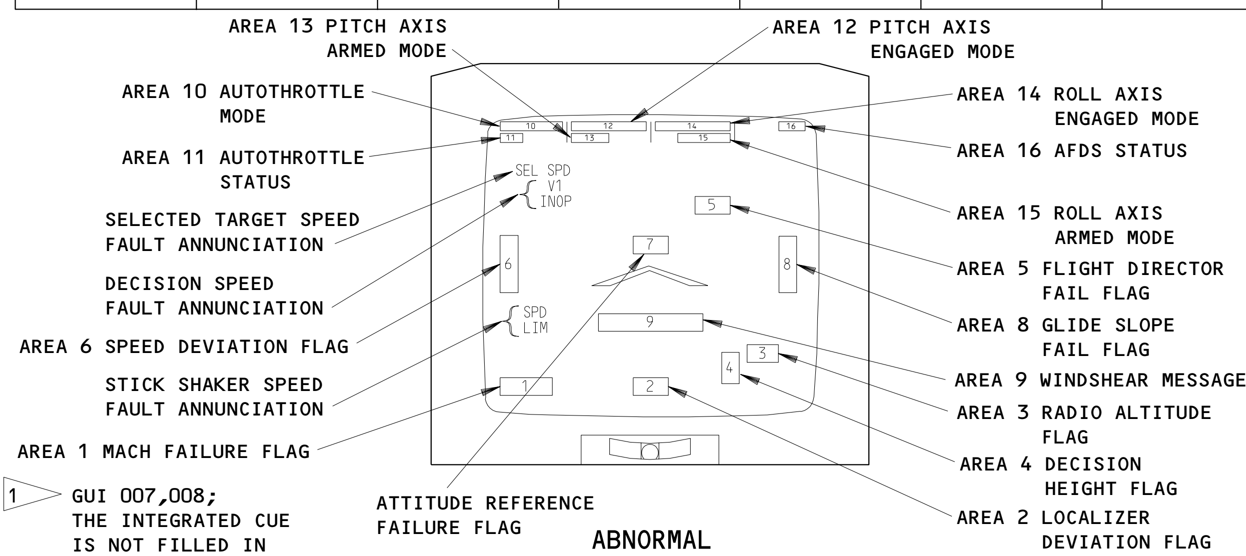
9) The FCC pitch modes use functions associated with SAMs in the following systems:
a) Horizontal Stabilizer Trim System (Ref 27-41-00).
b) Automatic Stabilizer Trim System (Ref 22-22-00)

(b) Displays

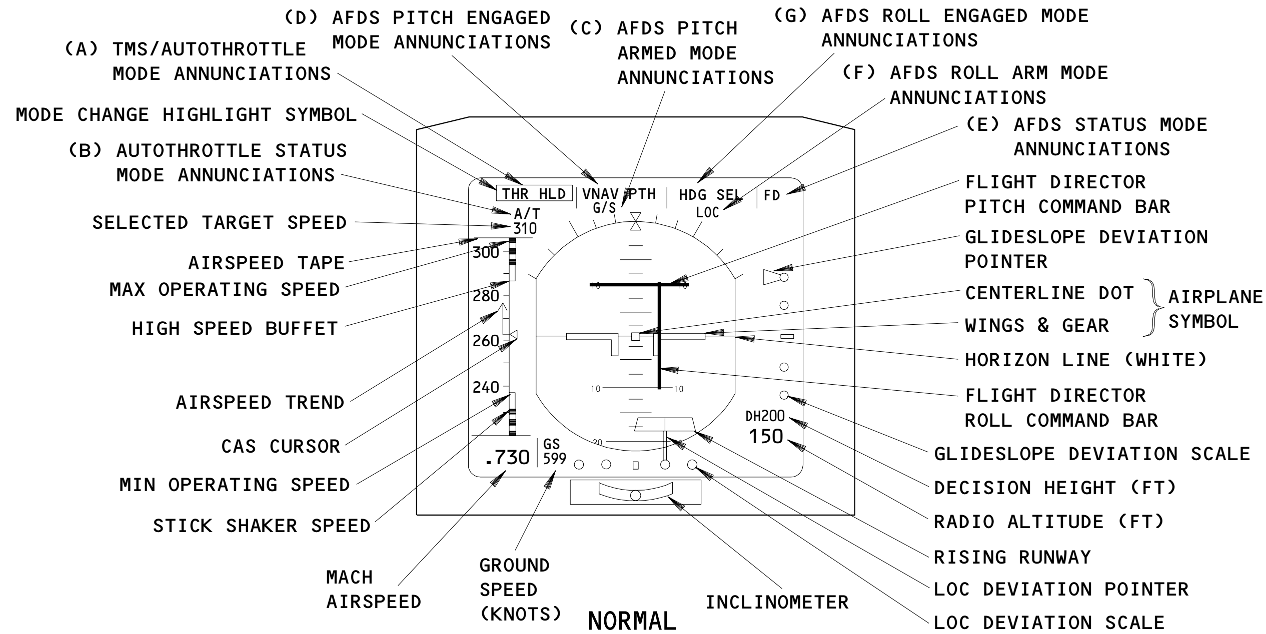
1) EADI - AFDS Displays (Normal) (Fig. 14)
a) GUI 002-006, 010-999;
the airplane is represented on the EADI as a fixed, white symbol consisting of a center dot flanked by two L-shaped bars. The vertical lines represent the airplane wheels; the horizontal lines, the wings.
b) GUI 001, 007-009;
the airplane is represented on the EADI as a fixed, white symbol consisting of an inverted V-bar.
c) GUI 002-006, 010-999;
the roll command bar is a magenta vertical line capable of lateral movement within ±0.625 inches from the center dot. The roll command scale factor is 30 degrees per inch.
d) GUI 002-006, 010-999;
the pitch command bar is a magenta horizontal line capable of vertical movement within ±0.625 inches from the center dot. The pitch command scale factor is 14.3 degrees per inch.
e) GUI 001, 007-009;
the integrated cue command bar is a magenta inverted vee capable of vertical and horizontal movement.
f) If the commanded fly-to point (CFP) is outside the normal deflection range for the command bars, both bars deflect together. They remain intersected at 0.125 inches (minimum) from the bar end to indicate the CFP.



AFDS AND TMC EADI MODE ANNUCIATION						
A	B	C	D	E	F	G
EPR SPD SPD LIM FLAP LIM THR-HOLD IDLE TEST FLCH GA ALPHA	A/T F/S	G/S FLARE VNAV	TO ALT HOLD V/S VNAV VNAV PTH VNAV SPD SPD SPD LIM FLAP LIM G/S FLARE ALT CAP GA	CMD FD TST	LOC ROLLOUT B/CRS LNAV LOC/NAV	TRK HOLD HDG HOLD HDG SEL LNAV LOC B/CRS ROLLOUT ATT TO GA

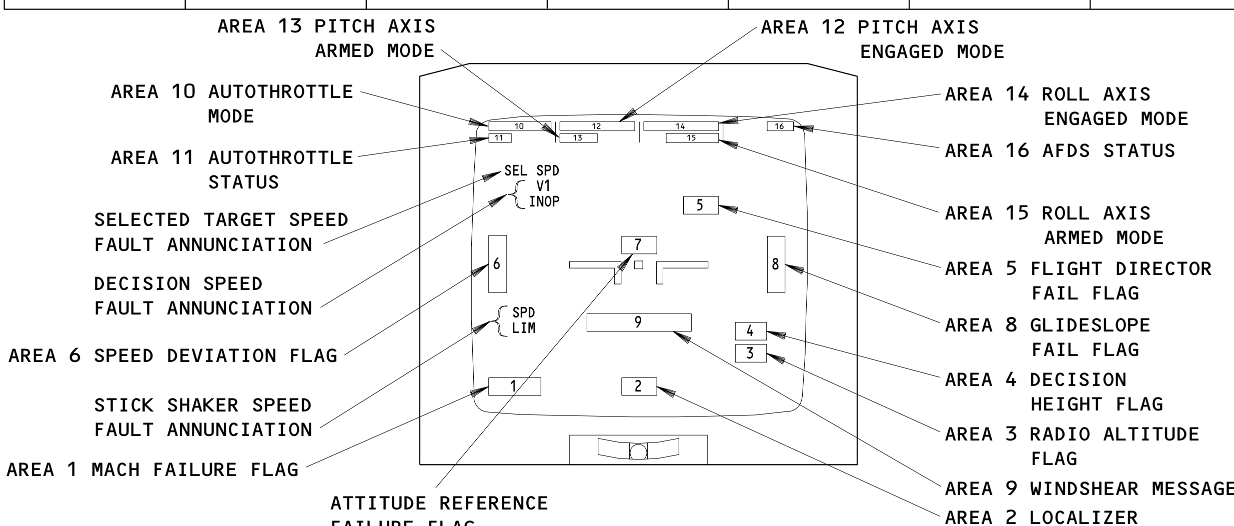


EADI - AFDS Displays
Figure 14 (Sheet 1)



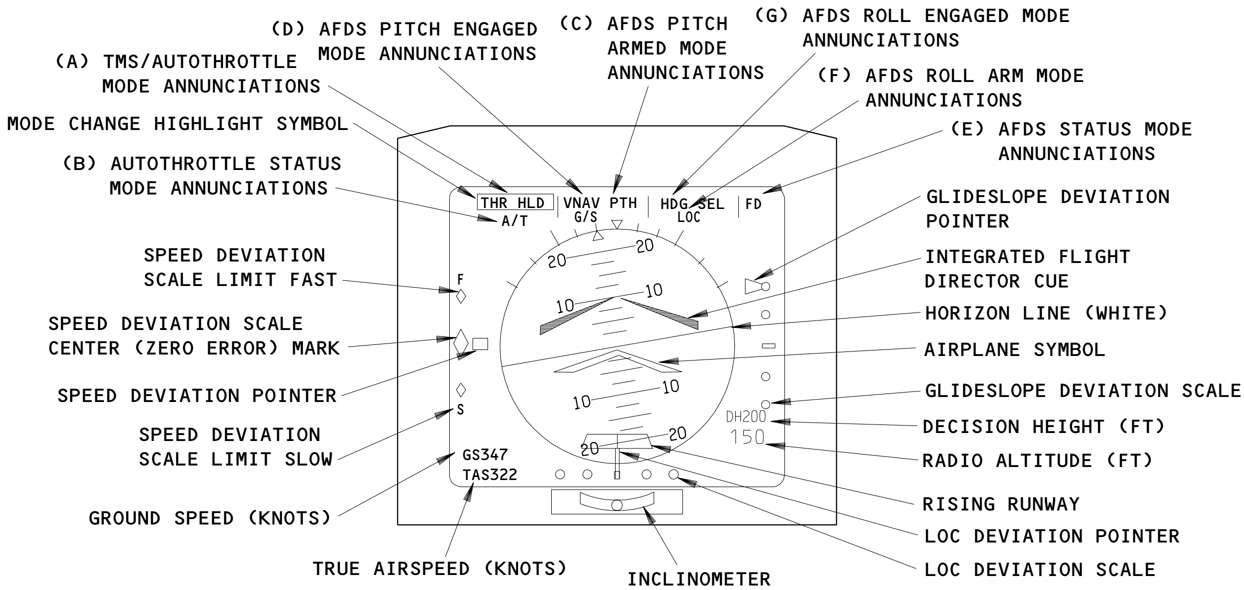
NORMAL

AFDS AND TMC EADI MODE ANNUCIATION						
A	B	C	D	E	F	G
EPR SPD SPD LIM FLAP LIM THR-HOLD IDLE TEST FLCH GA	A/T F/S	G/S FLARE VNAV	TO ALT HOLD V/S VNAV VNAV PTH VNAV SPD SPD SPD LIM FLAP LIM G/S FLARE ALT CAP GA	CMD FD TST	LOC ROLLOUT B/CRS LNAV LOC/NAV	TRK HOLD HDG HOLD HDG SEL LNAV LOC B/CRS ROLLOUT ATT TO GA



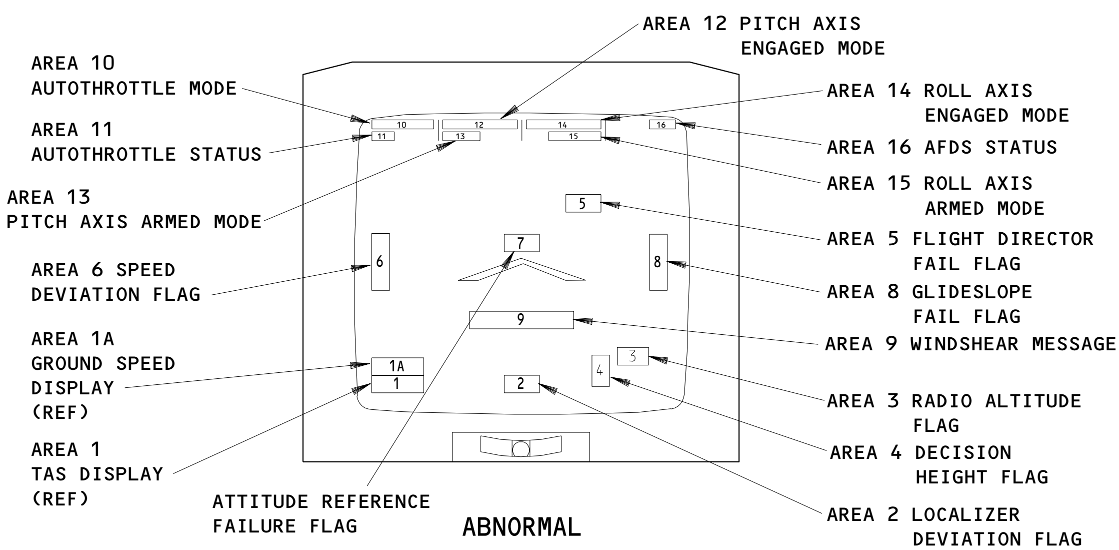
ABNORMAL

EADI - AFDS Displays
Figure 14 (Sheet 2)



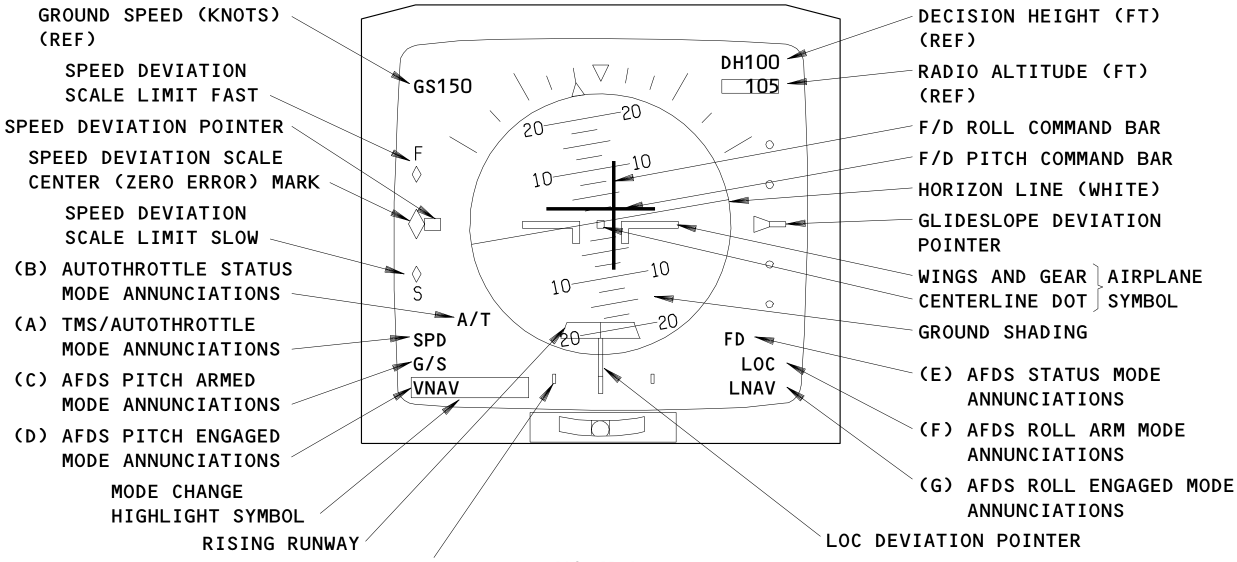
NORMAL

AFDS AND TMC EADI MODE ANNUCIATION						
A	B	C	D	E	F	G
EPR SPD SPD LIM FLAP LIM THR-HOLD IDLE TEST FLCH GA ALPHA	A/T F/S	G/S FLARE VNAV	TO ALT HOLD V/S VNAV VNAV PTH VNAV SPD SPD SPD LIM FLAP LIM G/S FLARE ALT CAP GA	CMD FD TST	LOC ROLLOUT B/CRS LNAV LOC/NAV	TRK HOLD HDG HOLD HDG SEL LNAV LOC B/CRS ROLLOUT ATT TO GA



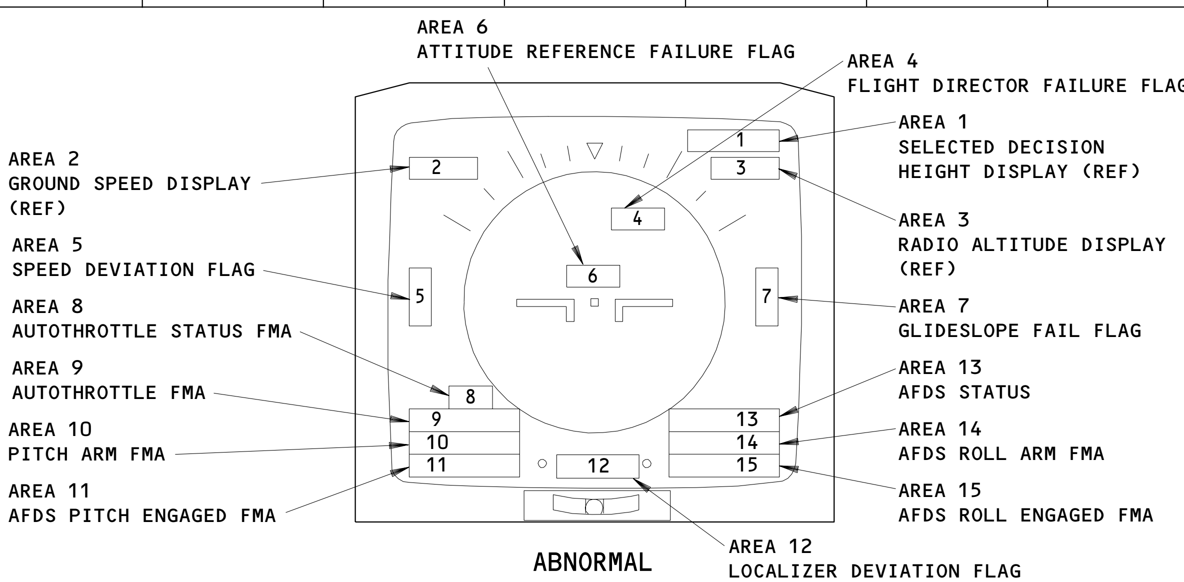
ABNORMAL

EADI - AFDS Displays
Figure 14 (Sheet 3)



NORMAL

AFDS AND TMC EADI MODE ANNUCIATION						
A	B	C	D	E	F	G
EPR	A/T	G/S	TO	CMD	LOC	TRK HOLD
SPD	F/S	FLARE	ALT HOLD	FD	ROLLOUT	HDG HOLD
SPD LIM		VNAV	V/S	TEST	LNAV	HDG SEL
FLAP LIM			VNAV		LOC/NAV	LNAV
THR-HOLD			VNAV PTH			LOC
IDLE			VNAV SPD			ROLLOUT
TEST			SPD			ATT
FLCH			SPD LIM			TO
GA			FLAP LIM			GA
ALPHA			G/S			
			FLARE			
			ALT CAP			
			GA			



ABNORMAL

EADI - AFDS Displays
Figure 14 (Sheet 4)

g) GUI 115;
up to 8 alphanumeric characters, 0.165 inches high, identify one of the following modes: CMD or FD (left aligned) or TEST (right aligned). This readout appears on the top line of the Flight Mode Annunciation (FMA) on the right side. The FMA is located in the lower, right and left corners of the ADI display.

h) GUI 001-114, 116-999;
up to 3 alphanumeric characters, 0.167 inches high, identify one of the following AFDS status annunciations: TST, CMD, or FD (right justified). These annunciations are green and appear in area E.

i) GUI 115;
up to 8 alphanumeric characters, 0.165 inches high, identify GS or FLARE. These are white and appear on the middle line, left side.

j) GUI 001-114, 116-999;
up to 5 alphanumeric characters, 0.123 inches high, identify one of the following armed pitch mode annunciations: G/S, FLARE or VNAV. These annunciations are white, appear in area C and are left justified.

k) GUI 115;
up to 8 alphanumeric characters, 0.165 inches high, identify one of the following: TO, ALT HOLD, ALT CAP, V/S, VNAV, IAS, IAS LIM, MACH, MACH LIM, G/S, FLARE, GA, ALPHA, FLAP LIM or blank. This display appears on the left side, bottom line. Colors used are: green for A/P, magnenta for F/D only operative, yellow for all LIM messages and for ALPHA.

l) GUI 001-114, 116-999;
up to 8 alphanumeric characters, 0.167 inches high, identify one of the following engaged pitch mode annunciations: TO, ALT HOLD, ALT CAP, V/S, VNAV, SPD, SPD LIM, G/S, FLARE, GA, FLAP LIM, VNAV PTH or VNAV SPD. These annunciations are green, appear in area D and are left justified.

m) GUI 115;
up to 8 alphanumeric characters, 0.165 inches high, identify one of the following: LOC, ROLLOUT, or blank. This display appears on the middle line, right side, in white.

n) GUI 001-114, 116-999;
up to 8 alphanumeric characters, 0.123 inches high, identify one of the following armed roll mode annunciations: LOC, ROLLOUT, B/CRS, LNAV or LOC/LNAV. These annunciations are white, appear in area F and are right justified.

o) GUI 115;
up to 8 alphanumeric characters, 0.165 inches high, identify one of the following: HDG HOLD, HDG SEL, TRK HOLD, LNAV, LOC, ROLLOUT, ATT, or blank. The display is on the bottom line, right side, in green.

p) GUI 001-114, 116-999;
up to 8 alphanumeric characters, 0.167 inches high, identify one of the following engaged roll mode annunciations: HDG HOLD, HDG SEL, TRK HOLD, LNAV, LOC, B/CRS, ROLLOUT, ATT, TO or GA. These annunciations are green, appear in area G and are right justified.

q) GUI 115;
up to 7 alphanumeric characters, 0.167 inches high, identify one of the following TMS autothrottle operational mode annunciations: EPR, SPD, SPD LIM, IDLE, FLP LIM, ALPHA, TEST, THR HOLD, FLCH or GA. These annunciations are green and appear on the third line from the bottom, left side (area A).

r) GUI 001-114, 116-999;
up to 7 alphanumeric characters, 0.167 inches high, identify one of the following TMS autothrottle operational mode annunciations: EPR, SPD, SPD LIM, IDLE, FLP LIM, ALPHA, TEST, THR HOLD, FLCH or GA. These annunciations are green, appear in area A and are left justified.

s) GUI 001-114, 116-999;
up to 3 alphanumeric characters, 0.167 inches high, identify one of the following TMS status mode annunciations: A/T or F/S. These annunciations are green, appear in area B and are left justified.

t) GUI 001-114, 116-999;
the mode change highlight symbol is a box that appears around the engaged pitch, roll and TMS/autothrottle mode annunciations for 10 seconds after transition into an ingaged mode.

u) GUI 009, 115;
the speed deviation scale is a fixed white symbol on the left side of the EADI. It consists of a large central diamond and two smaller diamonds (above and below) indicating the ±10 knots deviation. A corresponding F (top) and S (bottom) indicate fast or slow.

v) GUI 009, 115;
the speed deviation pointer is a magenta colored rectangle immediately to the right of the deviation scale. It is capable of vertical deflection to ±12.5 knots.

w) GUI 001-008, 010-114, 116-999;
the airspeed tape is a display that appears on the left side of the EADI. It consists of a white calibrated airspeed tape moving against a fixed white cursor with rolling digits which represents current computed airspeed (CAS). White numerics are placed against the 10 knot indices every 20 knots. Pointers, bugs, and barber poles are displayed against the scale to indicate certain special speeds, as shown below:

MINIMUM FLAP RETRACTION SPEED	GREEN
MINIMUM OPERATING SPEED	YELLOW
STICK SHAKER SPEED	RED/BLACK
HIGH SPEED BUFFET	YELLOW
MAXIMUM OPERATING SPEED	RED/BLACK
V1 SPEED	GREEN
ENGINE OUT SPEED	GREEN
SELECTED TARGET SPEED	MAGENTA

x) GUI 115;
the localizer deviation scale is at the bottom center of the EADI in white symbols. The standard scale is ±2 dots when A/P or F/D is in LOC ARM mode. The outer dot represents ±0.155 DDM (Difference in Depth of Modulation). An expanded scale of ±0.5 dot is used when the A/P or F/D is in LOC ENG. The outer dot represents 0.388 DDM.

y) GUI 001-114, 116-999;
the localizer deviation scale is at the bottom center of the EADI in white symbols. The standard scale is ±2 dots when A/P or F/D is in LOC ARM or B/CRS ARM mode. The outer dot represents ±0.155 DDM (Difference in Depth of Modulation). An expanded scale of ±0.5 dot is used when the A/P or F/D is in LOC ENG or B/CRS ENG. The outer dot represents 0.388 DDM.

z) The localizer deviation pointer is magenta and appears whenever the LOC deviation scale is in view. It is rectangular and is capable of ±0.73 inch deflection left and right of center. Maximum deviation represents ±0.17 DDM (2.2 dots) against the standard scale and ±0.0485 (0.625 dot) against the expanded scale.

aa) The G/S deviation scale is a vertical scale on the right side of the EADI. The symbols are white. The standard scale is ±2 dots when the A/P or F/D is in G/S ARM or G/S ENG. The outer dot represents ±0.175 DDM.

ab) The G/S deviation pointer is a magenta colored, truncated triangle capable of vertical movement inside the scale. Maximum deflection is ±0.193 DDM (2.2 dots).

2) EADI - AFDS Displays - Abnormal (Fig. 14)
   a) If pitch or roll data is invalid the following display elements are removed: horizon line, pitch attitude scale lines, roll pointer, sky/ground shading and pitch reference line. A yellow, rectangular flag with black letters ATT appears in area 7. A No Computed Data (NCD) condition removes the display elements, but the flag does not appear.
   b) Invalid data or the loss of the F/D-on bit, removes the command bars and shows a rectangular FD failure flag (yellow with black letters) in area 5. Command bars are removed if an NCD condition exists; no flag is displayed.

      NOTE: AIRPLANES WITH -420 AND SUBSEQUENT EFIS SYMBOL GENERATORS;
      If the airplane is on the ground and the FCC circuit breaker for the selected FCC is open, the FD failure flag will not show.
   c) GUI 115;
      invalid data removes the speed deviation scale and pointer and draws a rectangular SPD failure flag (yellow with black letters) in area 6. An NCD condition removes the pointer but the scale remains.
   d) GUI 115;
      if the TMS SPD DEV ERROR option is active, a speed error greater than ±10 knots from the commanded speed index is indicated on the EADI. The speed deviation pointer changes color to yellow and flashes at 0.9 Hz.
   e) GUI 115;
      invalid data removes LOC and/or G/S scales and pointers and draws LOC and/or G/S failure flags in area 12 and area 7 respectively. Failure flags are yellow with black letters. An NCD condition removes the pointer but not the scale.
   f) GUI 001-114, 116-999;
      invalid data removes LOC and/or G/S scales and pointers and draws LOC and/or G/S failure flags in area 2 and area 8 respectively. Failure flags are yellow with black letters. An NCD condition removes the pointer but not the scale.
   g) Provision for a ILS Deviation Monitor is available as an option. The ILS Deviation Monitor is enabled when any A/P is in APP (after LOC and G/S capture) and either the LOC or G/S exceeds the critical values below for more than one second. (The ILS DEV WARN program pin on the SGs must be grounded.) The critical value for LOC is more than 0.0155 DDM BELOW 500 FEET: FOR G/S, more than 0.0875 DDM between 500 and 1000 feet.

h) The ILS deviation monitor is indicated by a flashing (1 Hz) LOC or G/S deviation pointer and a change in scale color from white (normal) to yellow (caution).

i) Invalid and NCD data remove FMA characters of the affected display (areas 8, 9, 10, 11, 13, 14, 15). When an FMA FAULT bit is set in a specific AFDS discrete input word, a horizontal line is drawn through the center of the displayed annunciation.

j) GUI 001-008, 010-114, 116-999;
invalid data removes the speed tape and cursor and draws a rectangular SPD failure flag (yellow with black letters) in area 6. An NCD condition removes the pointer but the tape remains.

k) GUI 001-008, 010-114, 116-999;
invalid data removes selected target speed pointer and annunciates SEL SPD on the top left side of the EADI. A NCD condition removes the pointer only.

l) GUI 001-008, 010-114, 116-999;
invalid and NCD data removes V1 pointer and annunciates V1 INOP. This shows as a 2-line message immediately below SEL SPD annunciation.

m) GUI 001-008, 010-114, 116-999;
invalid data removes stick shaker speed barber pole and annunciates SPD LIM. A NCD condition on the ground or in air for less than 10 seconds will remove barber pole. A NCD condition in the air (greater than 10 seconds) will remove barber pole and annunciates SPD LIM on lower left side of EADI.

n) GUI 001-008, 010-114, 116-999;
invalid data blanks MACH readout and draws MACH flag (yellow with black letters). A NCD condition draws four dashes in place of numerics.

o) GUI 009, 115;
invalid data removes numerics and FT letters. No computed data draws five dashes in place of numerics but retains the FT legend.

B. BITE and Monitor

(1) The FCC provides in-flight detection of AFDS and related component failures. Detection includes:

(a) System fault detection and fault location.

(b) Monitoring of servos, sensors, input interfaces, and output interfaces.

(c) Self test.

(d) Storage of identified faults.

(2) Fault Detection and Location

(a) Faults are detected through self tests, system monitors within the FCC, and external fault data from interfacing LRUs. The FCC also receives hydraulic valid signals to aid in fault location.

(b) The FCC, in conjunction with the MCDP, can identify the location of an in-flight detected failure to the FCC itself, to a specific interfacing LRU, or to a functional group of LRUs.

(3) Monitoring

(a) The FCC monitors variable and discrete inputs and internally generated valid signals, including:

1) Servo loop operation.
2) Flap and stabilizer position.
3) Hydraulic power.
4) Data bus operation.
5) Status words.
6) Input signal fault.
7) Input signal tolerance.
8) Valid discretes.
9) Internal AC and DC power.

(b) The FCC monitors all incoming and outgoing interfaces, checking data for reasonableness and checking sign status matrix bits on data buses. The FCC also makes cross channel data comparison with other FCCs.

(4) In-flight Self Test

(a) The in-flight self test is performed by hardware and software independently in each of the FCCs. Safety critical functions are tested at least once a second. Non-critical functions are tested at least once every ten seconds.

(b) Faults detected by FCC self-test are stored in memory for access by shop support equipment. A maximum of five faults can be stored at one time. This fault data can only be removed from memory in the maintenance shop.

(5) Fault Data Storage

(a) The FCC provides permanent storage for up to five internal faults which result in an FCC invalid, and temporary storage for five words of fault data per flight leg.

(b) Fault data in temporary storage is formatted for transmission over a ARINC 429 data bus to the MCDP. The FCC clears temporary data at the start of each flight leg. Data provided to the MCDP includes:

1) Fault identity and location
2) Source of fault detection
3) Fault effect on flight deck
4) Fault intermittent or constant
5) Fault occured in-flight or on-ground

(6) MCDP Fault Recording and Testing

(a) The MCDP decodes and stores, in non-volatile memory, flight fault data from three flight control computers, one thrust management computer, and two flight management computers. The MCDP can store up to 350 independent flight faults. Ground faults during tests are annunciated but not stored for recall.

(b) The MCDP outputs test discrete signals to the FCCs to initiate ground testing. The test discrete signals are inhibited when the airplane is not on the ground. The MCDP provides test control and test parameters for the FCC ground test.

(c) The FCC tests the AFDS, including:
1) Central Processor Unit
2) Memories
3) Interfacing electronics
4) Power supplies

(d) FCC tests include:
1) Automatic self-test - tests FCC hardware not tested during in-flight automatic self-test.
2) Automatic interface tests.
a) ILS Tune Inhibit
b) RA Test Inhibit
c) Bus Isolate
d) Autopilot Servos
e) Automatic Stabilizer Trim
3) Manual interface tests.
a) Servo Detents
b) Transducer Output
c) MCP
d) ASA
e) FCC-EFIS Display
f) Analog Data Comparison

(e) The MCDP displays flight faults, ground faults, or initiates ground testing (Ref 22-41-00, Maintenance Monitor).

C. Control
(1) The autopilot is ON when power is applied. Proper operation requires interfacing systems be ON and operational.
(a) Ensure the following systems are operational.
1) Thrust Management Power (22-31-00, Adjustment/Test)
2) Maintenance Monitor (22-41-00, Adjustment/Test)
3) AC Generation (24-20-00, Adjustment/Test)
4) DC Generation (24-30-00, Adjustment/Test)

5) Aileron and Aileron Trim Control System (27-11-00, Adjustment/Test)
6) Rudder and Rudder Trim Control System (27-21-00, Adjustment/Test)
7) Elevator Control System (27-31-00, Adjustment/Test)
8) Horizontal Stabilizer Trim Control System (27-41-00, Adjustment/Test)
9) Leading Edge Slat System (27-81-00, Adjustment/Test)
10) Main Hydraulic Systems (29-11-00, Adjustment/Test)
11) Engine Indication and Crew Alerting System (31-41-00, Adjustment/Test)
12) Landing Gear Position Indicating and Warning System (32-61-00, Adjustment/Test)
13) Integral Panel Lights (33-13-00, Maintenance Practices)
14) Master Dim and Test (33-16-00, Adjustment/Test)
15) Air Data Computing System (34-12-00, Adjustment/Test)
16) Inertial Reference System (34-21-00, Adjustment/Test)
17) Flight Instrument System (34-22-00, Adjustment/Test)
18) ILS Navigation System (34-31-00, Adjustment/Test)
19) Radio Altimeter System (34-33-00, Adjustment/Test)
20) DME System (34-55-00, Adjustment/Test)
21) Flight Management Computer System (34-61-00, Adjustment/Test)

(b) Ensure the following circuit breakers on circuit breaker panel P11 are closed.
1) FLT CONT CMPTR SERVO L
2) MODE CONT PNL L
3) FLT CONT CMPTR PWR L
4) FLT CONT CMPTR SERVO R
5) MODE CONT PNL R
6) FLT CONT CMPTR PWR R
7) FLT CONT CMPTR SERVO C
8) FLT CONT CMPTR PWR C

(2) The F/D is operational when the FD switches are positioned to ON. The F/D command bars are then in view on the EADIs.

(3) GUI 001-114, 116-999;
the A/P is operational when a CMD switch is pressed and the lower half of the switch/lights illuminate.

(4) GUI 115;
the A/P is operational when an engage switch is placed and remains in the CMD position.

(5) Various modes can be selected on the MCP. Corresponding mode indications are annunciated on the EADIs.

(6) The autopilot is turned OFF when power is removed.

AUTOPILOT (FLIGHT CONTROL) - ADJUSTMENT/TEST

1. General
   A. This procedure contains two tasks. The first task is an operational test of the Autopilot/Flight Director System (AFDS). The second task is a system test of the AFDS.
   B. The operational test is to be used without tools or test equipment for a fast check of the operation of the system. The system test uses the MCDP to test the AFDS interfaces and autoland fuction.

TASK 22-10-00-715-002

2. AFDS Operational Test
   A. General
      (1) This task makes sure the Flight Control Computers (FCCs), Mode Control Panel (MCP), and Autoland Status Annunciators (ASAs) operate correctly. The test uses only equipment installed on the airplane.
   B. Equipment
      (1) Hand-held light source
   C. Consumable Materials
      (1) Black tape
   D. References
      (1) 24-22-00/201, Electrical Power - Control
      (2) 29-11-00/201, Hydraulic Power
      (3) 31-51-00/601, Warning System
      (4) 33-16-00/501, Master Dim and Test
      (5) 34-12-00/501, Air Data Computing System
      (6) 34-21-00/501, Inertial Reference System
      (7) 34-22-00/501, Flight Instrument System
      (8) 34-33-00/501, Radio Altimeter System
      (9) 34-61-00/501, Flight Management Computer System
   E. Access
      (1) Location Zones
          211/212 Flight Compartment

   F. Prepare to Test

      S 865-003
      (1) Make sure these circuit breakers on the overhead circuit-breaker panel, P11, are closed:
          (a) 11A2, VOR MKR LEFT

(b) 11A3, ILS CENTER
(c) GUI 001-114, 116-999;
11A7, EFIS DSPL SW LEFT
(d) 11A10, AIR DATA CMPTR LEFT
(e) 11A11, AIR DATA AOA SENSOR LEFT
(f) 11A12, AIR DATA BARO CORRECT LEFT
(g) 11A17, AUTOFLIGHT WARN
(h) 11A31, INDICATOR LIGHTS TEST
(i) 11A32, INDICATOR LIGHTS 1
(j) 11B17, CLOCK IND LEFT
(k) 11B18, WARN ELEX B
(l) GUI 115;
11C4, EFIS DSPL SW LEFT
(m) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(n) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC
(o) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(p) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC
(q) 11C14, FLAP/SLAT ELEC UNIT 2 POWER
or
FSEU 2 PWR
(r) 11C15, FLAP/SLAT ELEC UNIT 2 CONT
or
FSEU 2 CONT
(s) 11C16, FLAP/SLAT ELEC UNIT 2 SENSOR
or
FSEU 2 SENSOR
(t) 11C30, LANDING GEAR POS SYS 1

(u) 11D6, CAT III ISOL
or
CAT III BUS ISOL BAT
(v) 11E3, ADI LEFT
(w) 11E4, EFIS CONT PNL LEFT
(x) 11E6, HSI LEFT
(y) 11E8, FMCS CDU LEFT
(z) 11E9, FMCS CMPTR LEFT
(aa) 11E10, ILS L
(ab) 11E11, DME LEFT
(ac) 11E16, MODE CONT PNL LEFT
(ad) 11E17, FLT CONT COMPUTER POWER LEFT
(ae) 11E18, FLT CONT COMPUTER SERVO LEFT
(af) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(ag) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(ah) 11E24, ADI RIGHT
(ai) 11E25, EFIS CONT PNL RIGHT
(aj) 11E29, FMCS CDU RIGHT
(ak) 11E30, FMCS CMPTR RIGHT
(al) 11E31, ILS RIGHT
(am) 11E32, DME RIGHT
(an) 11E33, VOR RIGHT
(ao) 11E34, MODE CONT PNL LEFT
(ap) 11E35, FLT CONT CMPTR PWR RIGHT
(aq) 11E36, FLT CONT CMPTR SERVO RIGHT
(ar) 11F1, IRS LEFT
(as) 11F5, RADIO ALTM LEFT
(at) 11F8, EFIS SYM GEN LEFT
(au) 11F9, EFIS SYM GEN CENTER
(av) 11F14 or 11F16, TMC AC
(aw) 11F15 or 11F17, TMC DC
(ax) 11F16 or 11F18, TMC SERVO
(ay) 11F19, STAB POS MOD CENTER
(az) 11F20, RADIO ALTM CENTER
(ba) 11F21, IRS CENTER
(bb) 11F22, IRS RIGHT
(bc) 11F24, EFIS DSPL SW RIGHT
(bd) 11F26, RADIO ALTM RIGHT
(be) 11F29, EFIS SYM GEN RIGHT
(bf) 11F30, AIR DATA CMPTR RIGHT

(bg) 11F31, AIR DATA AOA SENSOR RIGHT
(bh) 11F32, AIR DATA BARO CORRECT RIGHT
(bi) 11G12, FLAP/SLAT ELEC UNIT 1 POWER
or
FSEU 1 PWR
(bj) 11G13, FLAP/SLAT ELEC UNIT 1 CONT
or
FSEU 1 CONT
(bk) 11G14, FLAP/SLAT ELEC UNIT 1 SENSOR
or
FSEU 1 SENSOR
(bl) 11G15, STAB POS MOD LEFT
(bm) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(bn) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(bo) 11G21, FLAP/SLAT ELEC UNIT 3 POWER
or
FSEU 3 PWR
(bp) 11G22, FLAP/SLAT ELEC UNIT 3 CONT
or
FSEU 3 CONT
(bq) 11G23, FLAP SLAT ELEC UNIT 3 SENSOR
or
FSEU 3 SENSOR
(br) 11G24, STAB POS MOD RIGHT
(bs) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(bt) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC
(bu) 11J2, EICAS CMPTR LEFT
(bv) 11J3, EICAS UPPER IND
or
EICAS UPPER DISPLAY
(bw) 11J29, EICAS CMPTR RIGHT
(bx) 11J30, EICAS LOWER IND
or
EICAS LOWER DISPLAY
(by) 11J31, EICAS DISPLAY SW
(bz) 11J32, EICAS PILOTS DSP
or
EICAS DISPLAY SELECT

(ca) 11J33, WARN ELEX A
(cb) 11J35, CLOCK IND RIGHT
(cc) 11P2, L IND LTS 2
(cd) 11P29, R IND LTS 2
(ce) 11R3, CAT III BUS ISOL LEFT
(cf) 11R30, CAT III BUS ISOL RIGHT
(cg) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(ch) GUI 115;
11S6, MAINT CONT DSPL
(ci) 11S15, AIR/GND SYS 1
(cj) 11S19, AIR/GND SYS 2
(ck) 11S23, POS SYS 2

S 865-004

(2) Make sure that these main power distribution panel P6 circuit breakers are closed:
(a) 6A3, DC STBY BUS OFF
(b) 6D3, IRS L
(c) 6D4, IRS C
(d) 6D5, IRS R
(e) 6L11, INV PWR BAT
(f) 6M13, AC STBY BUS OFF

S 865-005

(3) Supply electrical power (Ref 24-22-00).

S 715-006

(4) To complete this test it is necessary that these systems operate correctly. Use references only if and as necessary:
(a) Warning System (Ref 31-51-00).
(b) Master Dim and Test (Ref 33-16-00).
(c) Air Data Computer (Ref 34-12-00).
(d) Inertial Reference System (Ref 34-21-00).
(e) Flight Instrument System (Ref 34-22-00).

(f) Radio Altimeter System (Ref 34-33-00).
(g) Flight Management Computer System (Ref 34-61-00).

S 865-007

WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF CONTROL SURFACES AND CONTROL COLUMNS BEFORE HYDRAULIC SYSTEMS ARE PRESSURIZED. CONTROL SURFACES CAN MOVE WHEN AN HYDRAULIC SYSTEM IS PRESSURIZED AND CAUSE INJURY TO PERSONS.

(5) Make sure the L, C and R FLT CONTROL SHUTOFF valves are open.

S 865-009
(6) Supply hydraulic power (Ref 29-11-00).

S 865-010
(7) Make sure the MCDP is off.

S 865-011
(8) Align the left, center, and right Inertial Reference Units in NAV MODE (Ref 34-21-00).

NOTE: Approximately 10 minutes are necessary to align the IRU's. The steps before the "Test AFDS Operation" step can be done during this time.

S 865-012
(9) On the P1-1 panel, set the captain's instrument source select switches to these positions:
(a) FLT DIR switch to L.
(b) ADC, FMC, EFI, and IRS switches to normal (ALTN indication on switch does not show).

S 865-013
(10) On the P3-3 panel, set the first officer's instrument source select switches to these positions:
(a) FLT DIR switch to R.

(b) ADC, FMC, EFI, and IRS switches to normal (ALTN indication on switch does not show).

S 865-014

(11) Set the A/T switch and the two F/D switches on the MCP to OFF.

G. Test AFDS Operation

S 715-015

(1) Do these steps to do a check of the A/P DISC light, AUTOPILOT light, and MCP switch-lights:

(a) Make sure the A/P DISC and AUTOPILOT lights on the P1-3 panel are not on.

NOTE: If lights are on, push A/P disconnect switch two times to cancel the warning.

(b) On the P5 panel, set the IND LTS DIM/BRT switch on the right lighting control panel to BRT.

(c) On the P5 panel, push the TEST switch on the right lighting control panel to set Master Dim and Test to on, and hold if necessary .

(d) Make sure the A/P DISC light and AUTOPILOT light are on.

(e) Make sure the bottom half of the MCP switch-lights (dot-bar matrices) are on.

(f) AIRPLANES WITH COLLINS 822-1492-102 MCP;
Make sure the LCD windows display 8's and blink.

(g) Push the TEST switch on the right lighting control panel to set the test to off, if necessary.

S 715-016

(2) Do these steps to do a check of the MCP light-plate, switch-lights, and automatic brightness sensor:

(a) On the P5 panel, turn the PANEL/FLOOD GLARESHIELD switch on the left lighting control panel full clockwise (CW).

(b) Make sure the MCP light-plate and the top half of the switch-lights are on.

(c) Turn the PANEL/FLOOD GLARESHIELD switch counterclockwise (CCW).

(d) Make sure the MCP light-plate and switch-lights dim.

(e) AIRPLANES WITH -231 OR -232 MCP;
Do these steps to check the bezel light sensor:
1) Turn the PANEL/FLOOD GLARESHIELD switch an equal distance between fully clockwise and fully counterclockwise.
2) Put tape over the MCP light sensor.
3) Make sure the MCP display backlighting dims.
4) Remove the tape from the MCP light sensor.
5) Make sure the MCP backlighting become brighter.
6) Point a light source at the MCP light sensor.
7) Make sure the MCP backlighting becomes brighter.

S 715-017
(3) Do these steps to do a check of the MCP controls and displays:
(a) Make sure the MCP light-plate is on.
(b) Make sure the MCP digital displays show as follows :

1)	IAS/MACH	200
2)	HDG	000
3)	VERT SPD	Blank
4)	ALT	10,000

NOTE: The MCP digital display values given are only for when the airplane is initially supplied with power. The display values can be different if airplane power has been on since the last flight.

(c) Do a check of the IAS/MACH control and display as follows:
1) Turn the IAS/MACH control CW to increase the value that shows to 399 (maximum value).
2) Turn the control to set the display to 250(depress the knob if the window is blank)
3) Make sure the speed bugs on the left and right IAS/MACH Airspeed instruments are indicating 245 to 255.
4) Turn the control CCW to decrease the value that shows.
5) Turn the control to set the display back to 200.
(d) Do a check of the HDG SEL control and display as follows:
1) Turn HDG SEL control CW to increase the value that shows.
2) Make sure the MCP HDG display value and the two EHSIs selected heading indicator locations agree (±1 degree).
3) Turn the HDG SEL control to some other values and monitor the results.
4) Turn the control to set the display back to the initial heading.
(e) Do a check of the VERT SPD control and display as follows:
1) Turn VERT SPD control UP and DN.
2) Make sure the display does not change.
(f) Do a check of the ALT control and display as follows:
1) Turn the ALT control CW to increase the value that shows.
2) Turn the control CCW to decrease the value that shows.
3) Turn the control to set the display back to 10,000.

S 715-018

(4) Do these steps to do a check of the Autoland Status Annunciator (ASA):

(a) Push and hold the TEST 1 switch on the Captain's ASA .

(b) GUI 001-114, 116-999;
Make sure the ASA shows LAND 3 (green) on the top display and NO LAND 3 (yellow) on the bottom display.

(c) GUI 115;
Make sure the ASA shows AUTOLAND 3 (green) on the top display and the bottom display stays black.

(d) Release the test switch .

(e) GUI 001-114, 116-999;
Make sure the displays become black.

(f) GUI 115;
Make sure the top display becomes black.

(g) Press and hold the TEST 2 switch.

(h) GUI 001-114, 116-999;
Make sure the ASA shows LAND 2 (green) on the top display and NO AUTOLAND (yellow) on the bottom display.

(i) GUI 115;
Make sure the ASA shows AUTOLAND 2 (white) on the top display and MANUAL LND (yellow) on the bottom display.

(j) Release the test switch.

(k) Make sure the displays become black.

(l) Do the above steps again for the first officer's ASA.

S 715-019

(5) Do these steps to do a check of the Flight Director Annunciations:

(a) Set the Captain's and First Officer's F/D switch on the MCP to ON.

(b) Make sure these indications show:

1) GUI 002-006, 010-999;
The Captain's and First Officer's flight director command bars show on their EADIs.

2) GUI 001, 007-009;
The Captain's and First Officer's flight director command cue shows on their EADIs.
3) GUI 115;
FD (green letters) shows in the bottom right corner of each EADI.
4) GUI 001-114, 116-999;
FD (green letters) shows in the top right corner of each EADI.

(c) Set the Captain's INSTR SOURCE SEL FLT DIR switch to the C position, then the R position, and back to the L position. Give the switch at least 5 seconds at each position.
1) Make sure these indications show at each switch position.
a) GUI 002-006, 010-999;
The Flight Director command bars stay on the Captain's EADI.
b) GUI 001, 007-009;
The Flight Director command cue stays on the Captain's EADI.
c) GUI 115;
FD stays in the bottom right corner of each EADI.
d) GUI 001-114, 116-999;
FD stays in the top right corner of each EADI.

(d) Set the First Officer's INSTR SOURCE SEL FLT DIR switch to the C position, then the L position, and back to the R position. Give the switch at least 5 seconds at each position.
1) Make sure these indications show at each switch position:
a) GUI 002-006, 010-999;
the Flight Director Command bars stay on the First Officer's EADI.
b) GUI 001, 007-009;
The Flight Director Command cue stays on the First Officer's EADI.
c) GUI 115;
FD stays in the bottom right corner of each EADI.
d) GUI 001-114, 116-999;
FD stays in the top right corner of each EADI.

H. Test Autopilot Engage/Disengage Operation.

S 865-020
(1) Engage the left autopilot in CMD.

S 755-021
(2) Make sure that HDG HOLD and V/S appear on the two EADIs.

S 865-022
(3) Push the HDG SEL control on the MCP.

S 755-023
(4) Make sure the HDG HOLD indication changed to HDG SEL on the two EADIs.

S 975-024
(5) Write on paper the heading that shows in the HDG display of the MCP at this time. This is the initial heading.

S 865-025
(6) Turn the HDG SEL control to set a heading that is 100 degrees to the left of the initial heading.

S 755-026
(7) Make sure the control wheel moves CCW and the ailerons move in a left-wing-down direction.

S 865-027
(8) Set a heading that is 100 degrees to the right of the initial heading.

S 755-028
(9) Make sure the control wheel moves CW and the ailerons move in a right-wing-down direction.

S 865-029
(10) Set HDG display to initial heading.

S 865-030
(11) Turn VERT SPD control to set -1000 FPM in the VERT SPD display of the MCP.

S 755-031
(12) Make sure the column moves forward and the elevator moves in a down direction in less than 15 seconds.

S 865-032
(13) Set +1000 FPM in VERT SPD display.

S 755-033
(14) Make sure the column moves aft and the elevator moves in an up direction in less than 15 seconds.

S 865-034
(15) Push the captain's A/P Disengage switch to disengage the autopilot.

S 865-035
(16) Push the captain's A/P Disengage switch again to cancel the disengage warning.

S 865-154
(17) Turn the FD OFF and then ON again.

S 715-036
(18) Do the above steps again to test autopilot engage/disengage operation for the center and right autopilots.

I. Put the Airplane Back to Its Usual Condition

NOTE: Do not put the airplane back to its usual condition if AFDS System Test is to be done.

S 865-037
(1) Set IRS to off if it is not necessary (Ref 34-21-00).

S 865-038

(2) Remove the pressure from the left, center, and right hydraulic systems if it is not necessary (Ref 29-11-00).

S 865-039

(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-10-00-735-040

3. AFDS System Test

A. General

(1) This task uses the Maintenance Control Display Panel (MCDP) Ground Test procedures to test the AFDS interfaces and autoland function.

(2) The MCDP is installed in the main equipment center on the E1 rack. The MCDP control switches and display are found on the front of the MCDP. If a remote MCDP control unit is not to be used, two persons are necessary to do this system test. One person will be in the the main equipment center and one person will be in the control cabin.

(3) A connection is in the control cabin for a hand-held MCDP remote control unit. This lets MCDP Ground Tests be done from the control cabin. The MCDP data shows on the bottom EICAS display. Instructions for use of the MCDP are found in the Autoflight BITE - Maintenance Practices (Ref 22-00-02).

B. Equipment

(1) Nose Gear Steering Valve Lockpin - A09003-1

(2) Remote Control Unit, Maintenance Control Display Panel - A22001-22 (preferred), A22001-15 (optional).

C. References

(1) 09-11-00/201, Towing

(2) 22-00-02/201, Autoflight BITE

(3) 22-41-00/501, Maintenance Monitor

(4) 24-22-00/201, Electrical Power - Control

(5) 29-11-00/201, Hydraulic Power

(6) 31-41-00/501, Engine Indication and Crew Alerting System (EICAS)

(7) 34-12-00/501, Air Data Computing System
(8) 34-31-00/501, Instrument Landing System

D. Access
(1) Location Zones

119	Main Equipment Center
211/212	Flight Compartment

(2) Access Panels

119AL	Main Equipment Center Access Panel

E. Prepare to Test

S 865-041
(1) Make sure the circuit breakers shown in the Operational Test are closed before the MCDP is put into GRD TEST mode.

S 485-042
(2) If the remote MCDP is to be used, do the steps that follow:
(a) Install the MCDP remote control unit as follows:
1) Remove the dust cover from connector D41101J found behind the P6-5 panel.
2) Connect the MCDP remote control unit to connector D41101J.
(b) Make sure the EICAS circuit breakers on the P11 panel are closed.
(c) On the EICAS maintenance panel on the P61 panel, push the CONF/MCDP switch to show MCDP data on the EICAS bottom display.

NOTE: The same messages that show on the MCDP in the main equipment center will show on the EICAS bottom display. The indication MCDP FLT FAULTS or MCDP GRD TEST shows to identify which mode the MCDP is in.

(d) Make sure EICAS bottom display shows the message MCDP OFF.

S 715-064
(3) Do MCDP Power-Up and Self Test (AMM 22-00-02/201).

F. Test AFDS

S 715-043
(1) Do AFDS Operational Test

S 895-044
(2) Remove the pressure from the left, center, and right hydraulic systems (AMM 29-11-00/201).

S 715-045
(3) Do MCDP Operational Test (AMM 22-41-00/501).

S 865-000

WARNING: DO SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (MM 27-61-00). IT IS NECESSARY TO MOVE THE THRUST LEVERS DURING THIS TEST WHICH CAN CAUSE SPEEDBRAKE MOVEMENT IF HYDRAULIC POWER IS ON. THIS CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT .

(4) Do Spoiler/Speedbrake Deactivation procedure (AMM 27-61-00/201).

S 755-000

WARNING: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(5) Make sure the engines are not in operation.

S 865-067
(6) Set the A/T switch on the MCP to ARM.

S 865-070
(7) On the P5 panel, set the L and R ENG LIMITER switches on the ENG START/RAM AIR TURB control panel to ON.

S 865-125
(8) On the P5 panel, set the L and R ELEC ENG CONTROL switches to ON.

S 745-085
(9) Do MCDP Ground Test 30-CURRENT FAULT REPORT.
(a) Make sure that only these messages show during INFC FAULT ?

NO INFC TMC EEC L BUS IN
NO INFC TMC EEC R BUS IN

NOTE: The above messages will not show if you supply power to the EEC before you set the MCDP to the ground test mode.

(b) These messages can show during CURRENT OP FAULT?

CURRENT FAULT EEC L
CURRENT FAULT EEC R

NOTE: The above messages will not show the if engines are in operation.

S 865-092

WARNING: NOSE GEAR STEERING MUST BE LOCKED WHEN RUDDER MOVEMENT CAN OCCUR TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(10) Move the towing lever on the nose gear metering valve module to the TOW POSITION.

S 495-095
(11) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 865-122
(12) On the P10 panel, set the C and R STAB TRIM CUTOUT switches to the NORM position.

S 865-123
(13) On the MCP, set switches as follows:
(a) A/T ARM switch - ARM
(b) GUI 115;
A/P ENGAGE switches - OFF
(c) GUI 001-114, 116-999;
DISENGAGE bar - up position

S 755-096
(14) Make sure the ASA has no indication.

S 715-097
(15) Do the Instrument Landing System (ILS) Operational Test (AMM 34-31-00/501).
(a) Make sure no failures occur.

S 865-124
(16) It will be necessary during this test for the hydraulics to be pressured. Pressurization of hydraulics must obey the steps that follow:
(a) Hydraulic pressure can be supplied by the airplane electrically driven pumps or by ground hydraulic carts.
(b) To make sure there is sufficient power when the airplane electrically driven pumps are used, operate only the hydraulic devices necessary for this test.

WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF CONTROL SURFACES AND CONTROL COLUMNS BEFORE HYDRAULIC SYSTEMS ARE PRESSURIZED. CONTROL SURFACES CAN MOVE WHEN AN HYDRAULIC SYSTEM IS PRESSURIZED AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(c) Do not apply pressure to the left, right, and center hydraulic systems until MCDP message VFY HYD ON shows.

S 745-089

CAUTION: YOU MUST NOT ALLOW EQUIPMENT COOLING TO BE OFF FOR MORE THAN ONE MINUTE OR EQUIPMENT DAMAGE CAN OCCUR. DURING THIS TEST, RIGHT BUS POWER IS SWITCHED OFF WHICH CAUSES EQUIPMENT COOLING TO BE OFF. RIGHT BUS POWER WILL BE OFF LESS THAN 10 SECONDS FOR A SATISFACTORY TEST. IF THIS TEST DOES NOT OPERATE SATISFACTORILY, WAIT 5 MINUTES BEFORE THE TEST IS DONE AGAIN.

(17) Do MCDP Ground Test 40-AUTOLAND.

NOTE: The EICAS bottom display is supplied with power from the right AC BUS. It will be temporarily without power during operation of the R BUS tie breaker (BTB) if APU or external power is used as airplane power. The remote MCDP switches will stay in operation but the display will operate only in the main equipment center. After the EICAS bottom display starts operation again, it is necessary to press the CONF/MDCP switch again.

(a) Make sure no failure messages show.

S 865-048

(18) Do the steps that follow to clear EICAS maintenance message(s) "STAB TRIM" and/or YAW DAMPER" if they show:

WARNING: ACCIDENTAL SPOILER MOVEMENT CAN OCCUR WHEN THESE CIRCUIT BREAKERS ARE OPENED. INJURY TO PERSONS CAN OCCUR.

(a) Open these circuit breakers on the overhead circuit-breaker panel, P11, and then close them in less than 25 seconds:
1) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
2) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

3) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
4) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

(b) If a faultball stays set to yellow on a Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) and/or a Yaw Damper Module (YDM) after the above circuit breakers have been opened and closed, push the reset switch on the front of the SAM.

G. Put the Airplane Back to Its Usual Condition

S 095-101

WARNING: STAY AWAY FROM NOSE GEAR WHEELS WHEN YOU REMOVE THE LOCKPIN. THE NOSE WHEELS CAN MOVE QUICKLY TO THE CENTER POSITION AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do these steps to remove nose gear steering lockpin:
(a) Make sure the nose gear wheels are in the center position.
(b) Remove nose gear steering valve lockpin.
(c) Make sure the towing lever goes back to NORMAL POSITION.

S 085-050
(2) Do these steps to remove the MCDP remote control unit if it is installed:
(a) Disconnect the MCDP remote control panel from connector D41101J.
(b) Install the dust cover on connector D41101J.

S 865-052
(3) Set the IRS to off if it is not necessary (AMM 34-21-00/201).

S 865-055
(4) On the P5 panel, set the L and R ENG LIMITER switches on the ENG START/RAM AIR TURB control panel to OFF.

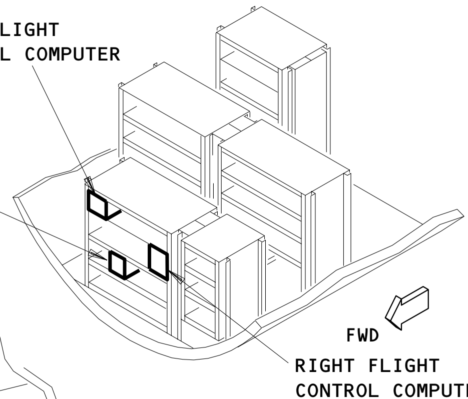
S 865-073
(5) On the P5 panel, set the L and R ELEC ENG CONTROL switches to OFF.

S 865-062
(6) Remove the pressure from the left, center, and right hydraulic systems if it is not necessary (AMM 29-11-00/201).

S 865-063
(7) Remove electrical power if it is not necessary (amm 24-22-00/201).

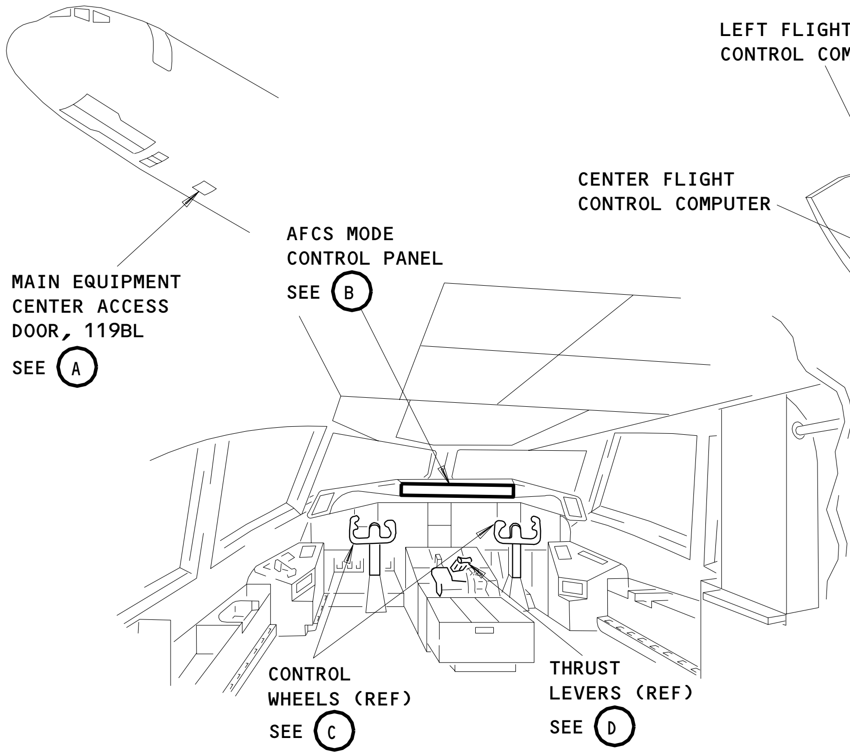
AUTOPILOT/FLIGHT DIRECTOR POWER - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. Power to the Autopilot/Flight Director System (AFDS) is provided by the right and left primary power sources (AC) and the right and left Transformer-Rectifier Units (DC). During Autoland operation, the center AFDS receives power from the standby inverter and the airplane battery.
   B. The controls and interlocks of the AFDS consist of the following units:
      (1) One Autoflight Control Systems (AFCS) Mode Control Panel (MCP), located below the glareshield (P55).
      (2) Three Flight Control Computers (FCCs) located in the forward Electrical/Electronics (E/E) compartment in racks E2-1 (L), E2-2 (C), and E2-3 (R).
      (3) One autopilot disengage switch, located on the inboard side of the outboard horn of each control wheel.
      (4) One go-around switch, located on the underside of each thrust lever.
      (5) AFDS mode annunciations displayed on the Electronic Attitude Director Indicator (EADI).
   C. AFDS Power and Engage Interlocks (Fig. 2)
      (1) Autopilot (A/P) and flight director (F/D) engage requests are generated at the Mode Control Panel (MCP) and are supplied to the Flight Control Computers (FCCs). Once accepted, the autopilot (A/P) engage requests can result in the arming and engaging of the A/P servos.
      (2) Mode requests for both A/P and F/D operation are generated at the MCP and transmitted by digital data bus to the FCCs.
      (3) GUI 001-114, 116-999;
      the A/P disengage bar logic tells the FCC if the bar is up or down. This allows the FCC to enable or disable the A/P servos.
      (4) GUI 115;
      the FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the select switch/lights. Flight director and autopilot engage status is indicated by the position of the CMD switches. Status is also indicated on the EADIs (Ref 22-14-00).

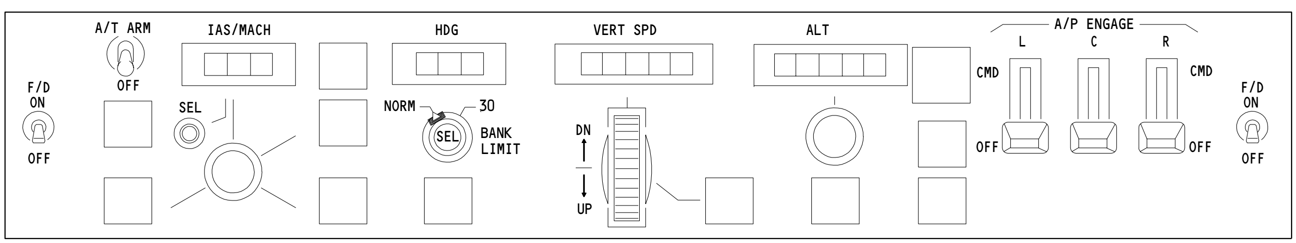


MAIN EQUIPMENT CENTER

A

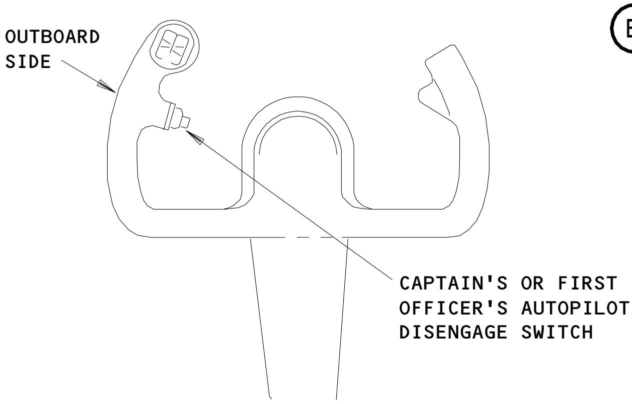


FLIGHT COMPARTMENT



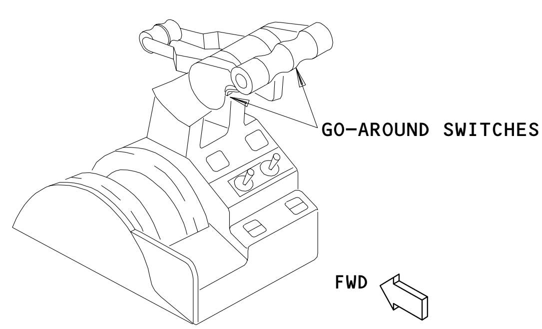
AFCS MODE CONTROL PANEL

B



CONTROL WHEELS (REF)

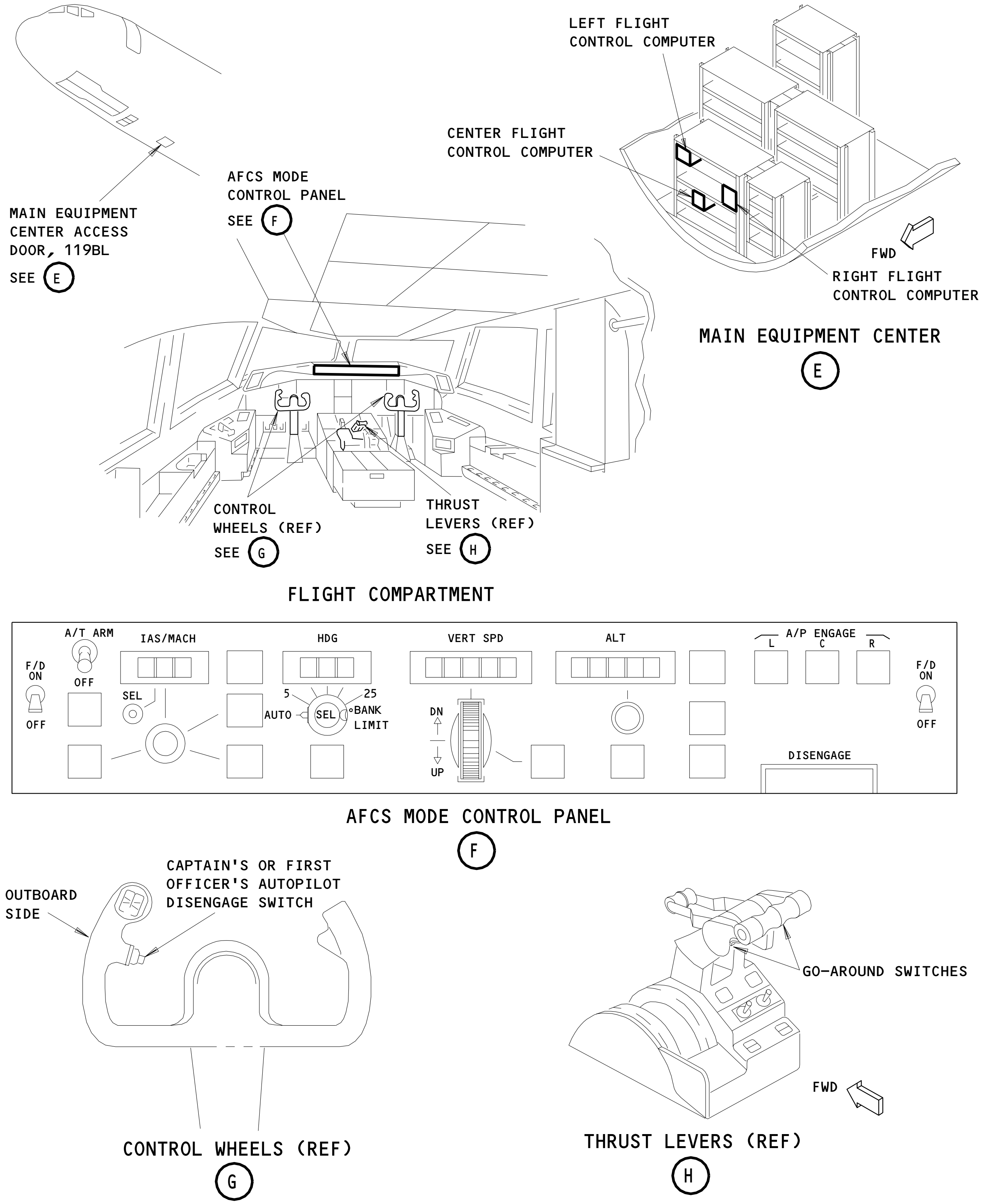
C



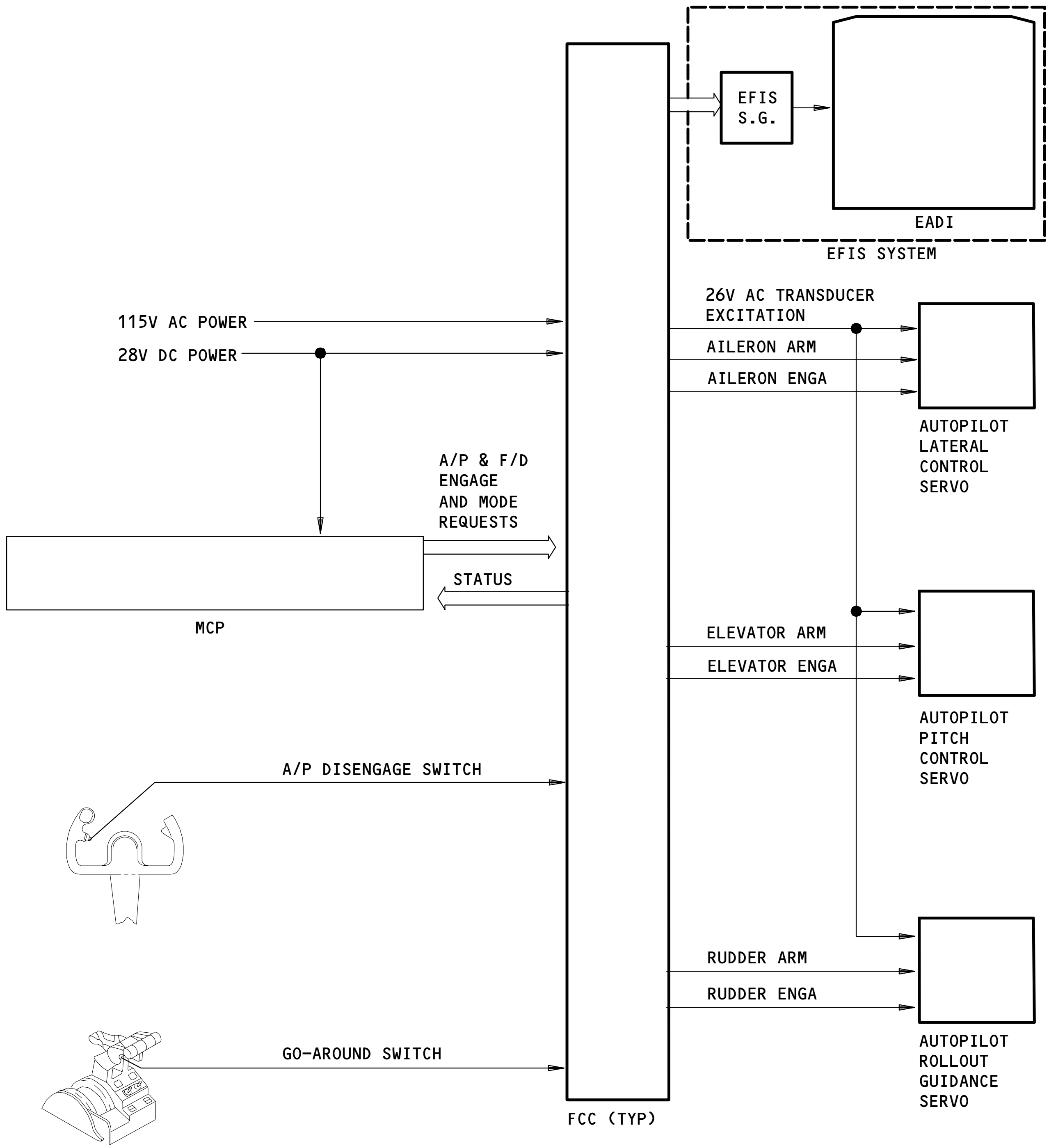
THRUST LEVERS (REF)

D

Autopilot/Flight Director System - Component Location
Figure 1 (Sheet 1)



Autopilot/Flight Director System - Component Location
Figure 1 (Sheet 2)



AFDS Power and Engage Interlocks
Figure 2

(5) GUI 001-114, 116-999;
the FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the select switch/lights. Flight director and autopilot engage status is indicated by the dot-bar matrix on the CMD switches. Status is also indicated on the EADIs (Ref 22-14-00).

D. FCC Input Discretes From Remote Switches

(1) Two A/P disengage switches (one on each control wheel) provide discrete signals to the FCCs which disengage the autopilot. The go-around switches (one on each thrust lever) provide discretes to the FCCs to start the Go-Around (GA) mode.

E. AFDS Input Power

(1) The FCCs receive 28 vdc and 115 vac, 400 Hz power for internal use. The 115 vac is reduced to 26 vac in the FCC and supplied to the A/P servos Autopilot Lateral Control Servo (ALCS), Autopilot Rollout Guidance Servo (ARGS), and Autopilot Pitch Control Servo (APCS). The MCP power supply receives 28 vdc. Both the MCP and the FCC use 28 vdc for the actuator servo arm, engage, and disengage functions.

2. Component Details

A. AFDS Mode Control Panel (Fig. 3)

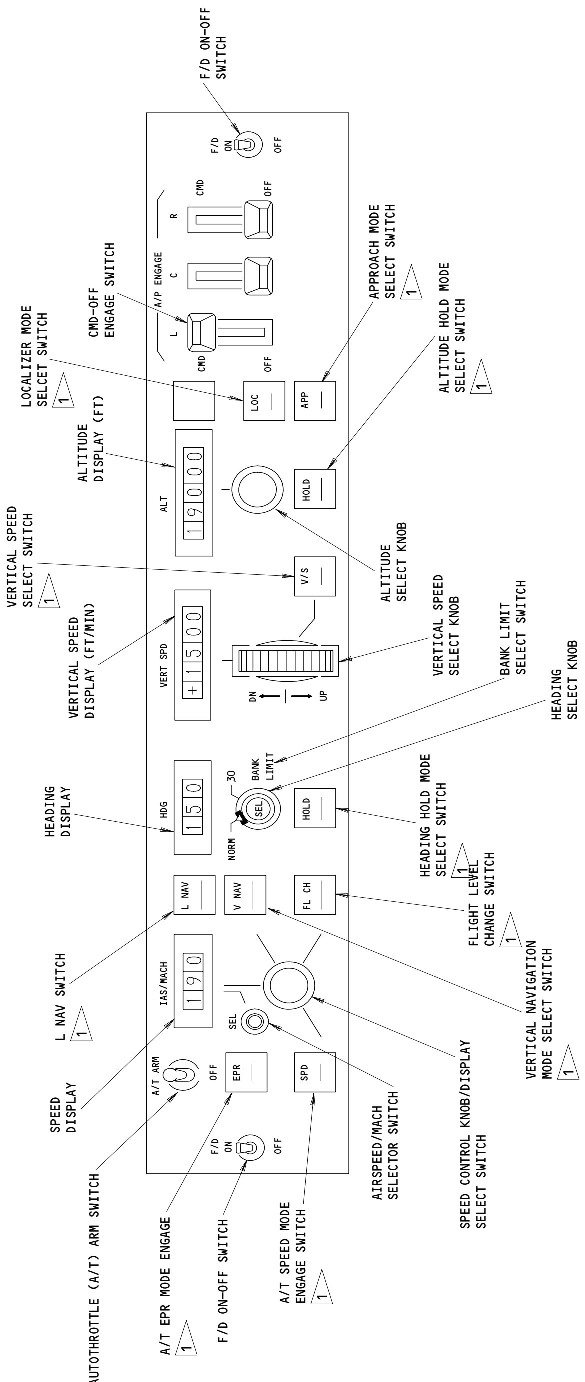
(1) The AFDS Mode Control Panel provides the primary interface between the flight crew and the autopilot/flight director and autothrottle (A/T) systems. It contains switches and logic circuits for mode selectors and control of all A/P, F/D, and A/T functions.

(2) Autopilot Engage/Disengage Switches

(a) GUI 001-114, 116-999;
the MCP has 3 pushbutton switches for manual engagement of the L, C, and R autopilot channels in the CMD mode. Pushbutton legends are lighted whenever power is applied to the MCP. A dot-bar matrix on each pushbutton is lit by FCC logic when the request has been accepted by the FCC.

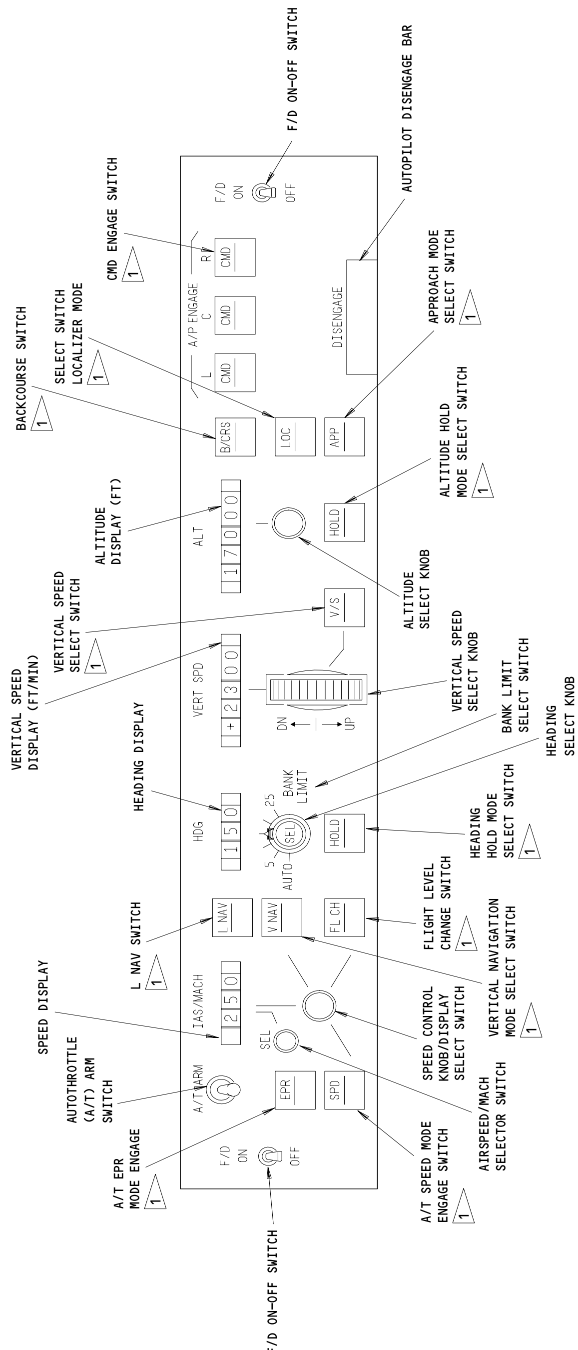
(b) GUI 115;
the MCP has 3 engage switches for manual engagement of the L, C, and R autopilot channels in the CMD mode. The switches are locked in the CMD position when the engage request has been accepted by the FCC.

(c) GUI 001-114, 116-999;
the A/P disengage bar is a rocker switch with contacts that interface with all three FCCs. It provides positive disengagement of all A/P servos. A dayglow-orange stripe on top of the bar gives a clear indication of the disengaged state.



1 DOT/BAR MATRIX ON LOWER HALF OF SWITCH COVER IS ILLIMINATED WHEN COMPUTER CONFIRMS VALID CONDITIONS AND SATISFIES REQUEST

AFDS Mode Control Panel
Figure 3 (Sheet 1)



1 DOT/BAR MATRIX ON LOWER HALF OF SWITCH COVER IS ILLUMINATED WHEN COMPUTER CONFIRMS VALID CONDITIONS AND SATISFIES REQUEST

AFCS Mode Control Panel
Figure 3 (Sheet 2)

A70271

(3) Autothrottle Arm Switch
  (a) The Autothrottle (A/T) toggle switch allows arming and disarming of the autothrottle system for engagement in the EPR or speed mode.
(4) Autothrottle Mode Engage Switches
  (a) Two pushbutton switches provide manual selection of the A/T modes. Contacts of these switches interface with the Thrust Management Computer (TMC) (AMM 22-31-00). The dot-bar matrix, controlled by TMC logic, lights when the mode is engaged.
(5) Flight Director Switches
  (a) The flight director system is turned on/off with either of two F/D toggle switches on the MCP. The left F/D switch controls the captain's EADI command bars. The right F/D switch controls the first officer's EADI command bars.
(6) Autopilot/Flight Director Mode Select Controls
  (a) The MCP has eight pushbuttons for manual selection of the following AFDS modes:
    1) Vertical speed - V/S
    2) Altitude Hold - ALT HLD
    3) Vertical Navigation - VNAV
    4) Flight Level Change - FLCH
    5) Heading Hold - HDG HLD
    6) Lateral Navigation - LNAV
    7) Localizer - LOC
    8) Approach - APP
  (b) Each pushbutton operates switch contacts that interface with all three FCCs. Dot-bar illumination is controlled by FCC logic.
  (c) A pushbutton switch labeled SEL, allows manual selection of either airspeed or mach for FLCH (flight level change) or A/T SPD (speed) modes. Switch contacts interface with the FCCs and the TMC.
  (d) GUI 001-114, 116-999;
  a pushbutton switch/light is provided for manual selection of backcourse (B/CRS). Switch contacts interface with the FCCs, and dot-bar illumination is controlled by FCC generated logic.

(e) The HDG SEL (heading select) inner knob assembly contains a push action switch for manual selection of the HDG SEL (heading select) mode. The switch contacts interface with the FCCs.

(7) Heading Reference Display and Control

(a) The heading reference portion of the MCP consists of a magnetic wheel heading display window and a rotary control with two concentric knobs. The display window is capable of showing 000 to 359 degrees in increments of one degree. The display is initialized to 000 when power is first applied to the MCP. At LOC capture, the display is initialized to the selected runway heading from the FCC.

(b) GUI 001-114, 116-999;
the outer knob of the rotary control has six positions for selection of bank angle limits (5, 10, 15, 20, 25, AUTO). The knob position is digitally transmitted to the FCCs. The inner knob increments the heading display when rotated. This heading reference is optically sensed to drive the decoders and then digitally transmitted to the FCCs.

(c) GUI 115;
the outer knob of the rotary control has two positions for selection of bank angle limits (NORM and 30). The knob position is digitally transmitted to the FCCs. The inner knob increments the heading display when rotated. This heading reference is optically sensed to drive the decoders and then digitally transmitted to the FCCs.

(8) Vertical Speed Reference Display and Control

(a) The vertical speed (V/S) portion of the MCP consists of a magnetic wheel display window and a thumbwheel control. The display window displays V/S from +6000 to -8000 ft/min in 100 ft/min increments. When V/S mode is not selected, the display shows zero. With V/S active, the display will show the vertical speed at the time the mode was selected. (If V/S is zero ft/min, A/P goes into an altitude hold control while in V/S mode). The V/S reference value is optically sensed to drive the decoders and then digitally transmitted to the FCCs.

(b) The thumbwheel is used to select a given vertical speed. Upward rotation of the thumbwheel decreases the selected value, downward rotation increases it.

(9) Altitude Reference Display and Control

(a) The altitude select portion of the MCP consists of a display window and a rotary select knob. The display can show 0 to 50,000 feet in increments of 100 feet. When power is applied to the MCP, the altitude window is set to 10,000 feet. The rotary knob increments the display when rotated (clockwise increases value). The altitude reference is digitally transmitted to the FCCs, the FMCs, and the warning electronics module.

(10) IAS/MACH Reference Display and Control

(a) The indicated airspeed (IAS) reference section of the MCP consists of a rotary knob with an integral pushbutton switch and an IAS/MACH display window. Display limits are 100 to 399 knots in increments of one knot, and 0.40 to 0.95 Mach in increments of 0.01 Mach.

(b) If the FCC airspeed is invalid, the IAS/MACH window displays 200 knots. The display is blanked when the VNAV mode becomes active and when the speed control switch in the knob is pressed while speed (or Mach) is displayed. If the switch is pushed when either A/T SPD, FLCH, or F/D takeoff mode is active, the display does not change.

(c) The airspeed and Mach number reference values are stored in the MCP and are transmitted in digital form to the TMC, the FCCs, and the mach/airspeed indicators to drive the airspeed reference cursors.

B. Flight Control Computer

(1) The FCC is an 8 MCU component with a maximum weight of 28.1 lbs. It is cooled by blow-through air and has a single shell electrical connector per ARINC 600. Two removable side covers permit access to all circuit cards and modules.

(2) Autopilot outputs of the FCC include arm, engage, and disengage logic to the local A/P servos; system engagement and autoland status annunciations; and aural and visual warnings of system disconnect.

(3) GUI 010-114, 116-999;
the FCC generates steering signals for the F/D command bars. The command bars are displayed on the EADI when the FCC is in the CMD or F/D modes. The FCC to Electronic Flight Instrument System (EFIS) symbol generator selection is accomplished externally by the instrument source select switch.

(4) GUI 001-009, 115;
the FCC generates steering signals for the F/D command bars. The command bars are displayed on the EADI when the FCC is in the F/D mode. The FCC to Electronic Flight Instrument System (EFIS) symbol generator selection is accomplished externally by the instrument source select switch.

(5) The FCC provides automatic stabilizer trim commands to the Stabilizer Trim/Elevator Asymmetry Limit Module (SAM).

C. Autopilot Disengage and Go-Around Switches

(1) Autopilot Disengage Switches

(a) The A/P disengage switches are single action, multiple pole, pushbutton switches.

(b) Pressing either of the A/P disengage switches disconnects any engaged FCC. The autopilot disengage switch must be pressed a second time to reset the autopilot disengage warning.

(2) Go-Around (GA) Switches

(a) Each palm actuated GA switch contains a set of contacts for each FCC. The switch provides discrete logic used in the GA mode.

3. <u>Operation</u>

A. Functional Description

(1) Autopilot/Flight Director System Power

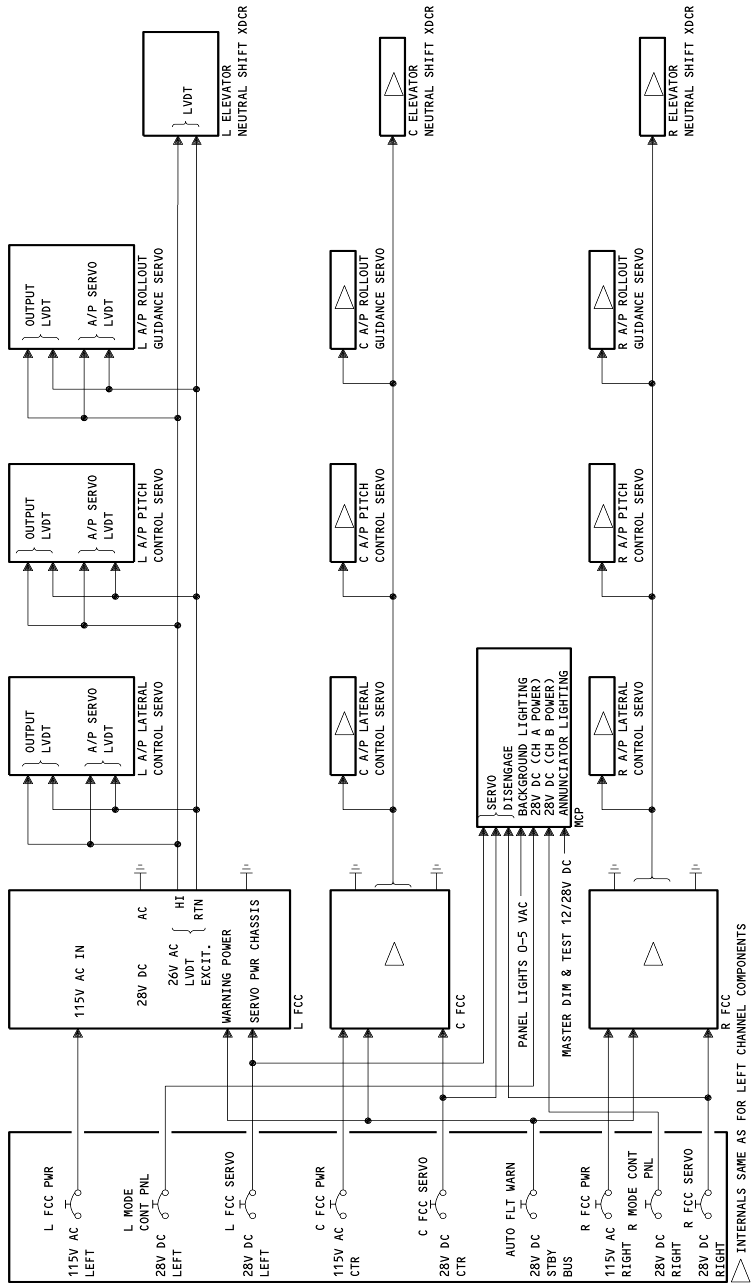
(a) General

1) Power for the Autopilot/Flight Director System is provided by the engine or APU driven generators via the left and right AC buses. The left and right transformer rectifier units (TRUs) provide DC power to the left and right DC buses respectively. During normal operation, airplane cruise or takeoff, the center AC and DC buses are powered by the left buses. During autoland operation, the center AC and DC buses are powered by the static inverter and hot battery bus. Center bus switching is provided by the FCCs internal isolation request and system relay logic. The switching provides each FCC with an independent power source during autoland operation.

(b) Autopilot/Flight Director System Power Distribution (Fig. 4)

1) AC Power

a) The 115v ac supplied to the FCCs is used by the FCC internal power supply. The FCCs output 26v ac to operate neutral shift transducers and Linear Variable Differential Transducers (LVDT)s in the flight control servos (ALCSs, APCSs, & ARGSs). The MCP receives 5v ac power for background lighting from the GLARESHIELD panel lights.



Autopilot/Flight Director System Power Distribution Schematic
Figure 4

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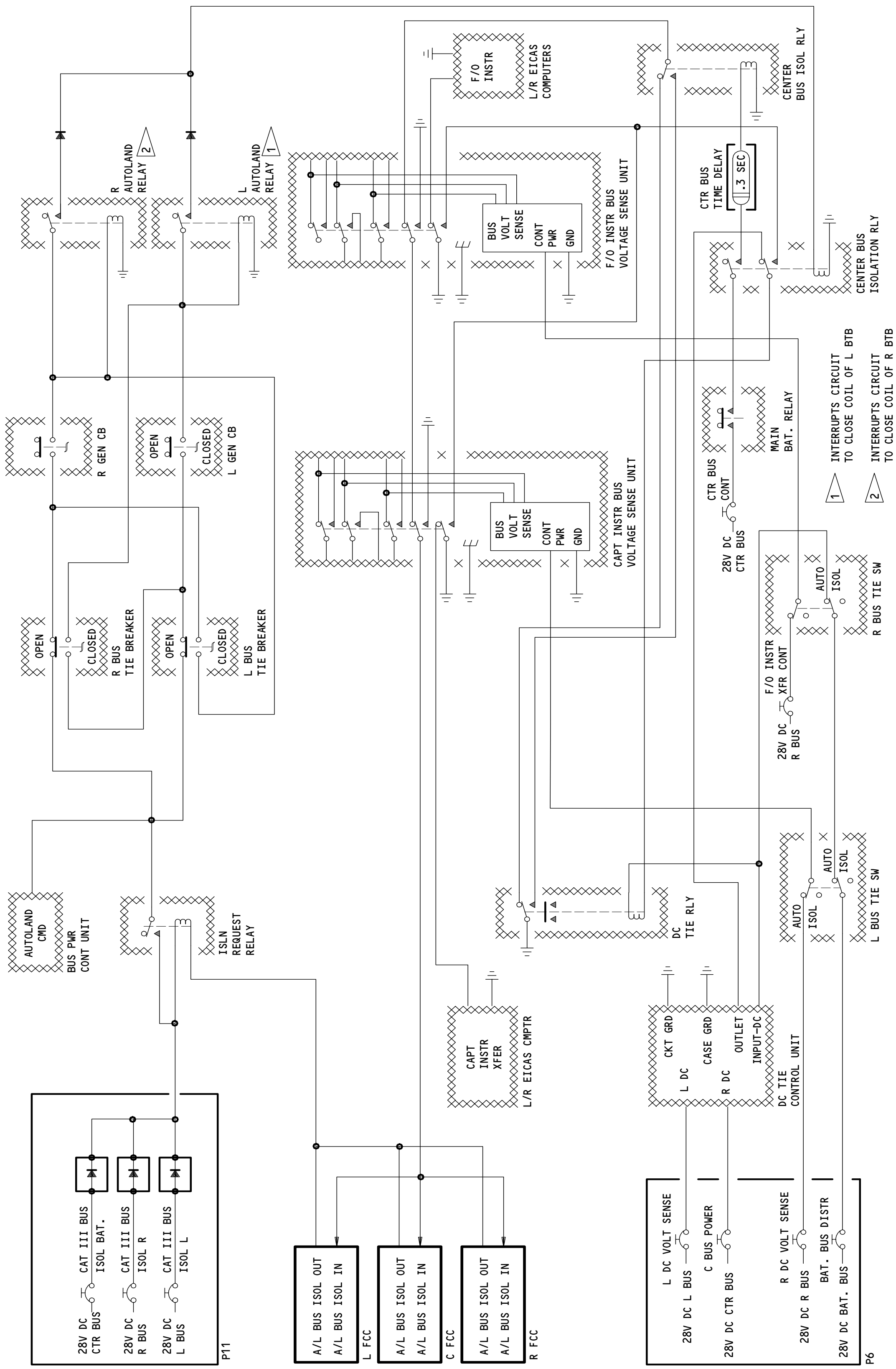
2) DC Power
   a) Each FCC receives 28v dc from the standby bus thru the AUTOFLIGHT WARN circuit breaker for caution and warning monitor circuits. The FCCs receive 28v dc servo power for servo arming and engaging. The MCP receives 28v dc thru the MODE CONT PNL circuit breakers for power. A variable 12 or 28v dc is provided to the MCP from the master dim & test for annunciator illumination.

(c) Autopilot/Flight Director System AC Power
  1) FCC AC Power
     a) The FCC power supply receives 115v ac and generates dc voltages required for internal FCC circuitry. The 115v ac is also reduced to 26v ac to provide servo LVDT excitation.

(d) Autopilot/Flight Director System DC Power
  1) FCC DC Power
     a) Standby 28v dc power is supplied to the FCC as an independent power source for A/P caution and warning monitor circuits.
     b) GUI 115;
        left FCC 28v dc servo power is provided to the left FCC. It is used to power the aileron, elevator, and rudder arm and engage solenoids. The solenoids are energized when the FCC ARM/ENGAGE logic is satisfied. Within the MCP, the servo power is used by the CMD engage switches.
     c) GUI 001-114, 116-999;
        left FCC 28v dc servo power is provided to both the FCC and the MCP. Within the FCC, this is used to power the aileron, elevator, and rudder arm and engage solenoids. The solenoids are energized when the FCC ARM/ENGAGE logic is satisfied and the MCP DISENGAGE bar is up.
  2) MCP DC Power
     a) The primary source of 28v dc for the MCP power supply is provided by Left and Right Mode Cont PNL circuit breakers.

(e) Autopilot/Flight Director System Autoland Bus Isolation (Fig. 5)
  1) Three independent AC and DC power sources are required to initiate a fail-operational Category III landing. The main engine-driven generators are the normal sources for two of the independent channels and the standby power system (battery and static inverter) provides the third. If one engine-driven generator is inoperative, the APU-driven generator may be substituted for that source.
  2) The L and C AFDS channels are supplied from the left power system and the R AFDS channel from the right power system. The bus tie breakers (BTBs) are normally open but will close automatically to maintain power to a main AC bus if the associated generator fails.



AFDS Autoland Bus Isolation Schematic
Figure 5

3) If the APU is running prior to a generator failure, the auxiliary power breaker (APB) and the associated BTB will close allowing the APU generator to power the failed generator's channel. If the APU is not running, both BTBs will close and the opposite engine-driven generator will assume the load (Ref 24-22-00).

4) The ground allows the ISLN REQUEST RELAY (K122) to energize thereby providing 28 vdc to the AUTOLAND RELAYS (K526 and K527). The 28 vdc energizes both relays which in turn provides 28 vdc to energize the CENTER BUS ISOL RELAY (K123). With K123 energized, the MAIN BAT RELAY (K104) provides power to energize the CENTER BUS XFER RELAY (K107). This removes the ground to the FCC's A/L BUS ISOL IN pin. The open signals the FCC's that the center buses are now powered from the hot battery bus (DC) and static inverter (AC). If either the left or right INSTR BUS VOLTAGE SENSE UNIT (M1079 or M1217) de-energizes, then the ground to the FCC's A/L BUS ISOL in pin is completed.

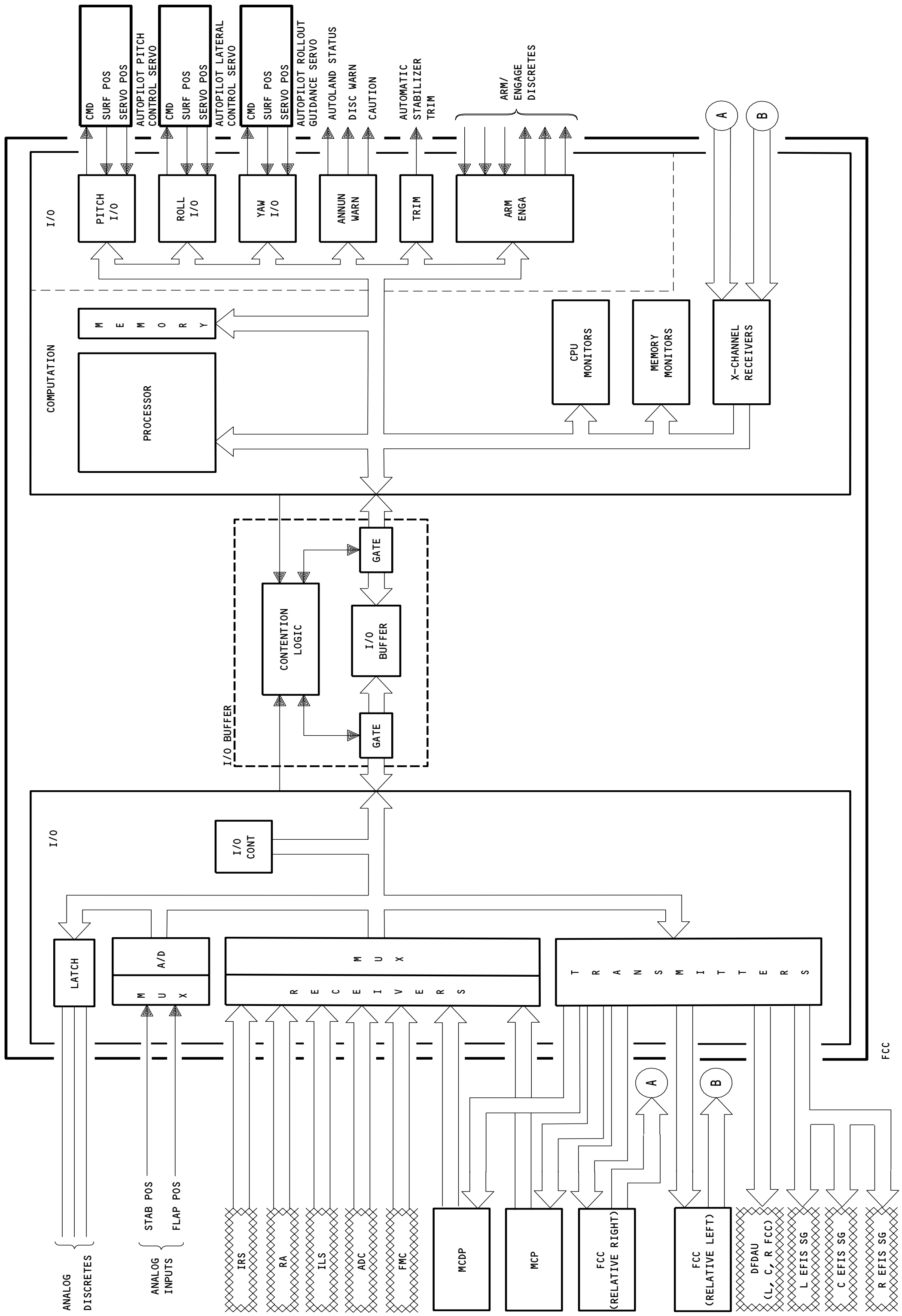
5) The autoland bus isolation logic maintains the center AFDS on battery power for the duration of the approach unless reset by one of the following conditions:
   a) A failure is detected which reduces the system redundancy level to less than the fail operational configuration.
   b) Total autopilot disconnect.
   c) Go-around has been initiated and the airplane is at an altitude of 100 feet or more.

6) Except for total autopilot disconnect, center bus isolation is latched when:
   a) The airplane is below alert height and not in the G/A mode.
   b) The airplane is below 100 feet and in G/A mode.
   c) Center bus isolation was reset previously due to reduced system redundancy.

(2) AFDS FCC Operation

(a) FCC Functional Partitioning (Fig. 6)

1) The FCC is divided into two separate sections; one section is controlled by the CAPS processor and the other section is controlled by the Input/Output (I/O) controller. The section under the I/O controller provides input buffering, signal conversion, and storage of analog, digital, and discrete signals received from external sensors and control switches. The section under the CAPS processor uses the CAPS transfer bus to provide processor related functions such as memory access, computation of analog servo commands, processing of discrete input information, monitoring functions, and processing of cross channel receiver information. The CAPS processor section also contains and implements the primary application program for the FCC.



Flight Control Computer Functional Partitioning Schematic
Figure 6

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2) The link between the I/O controller section and the CAPS processor section is provided by the I/O buffer. Data words may be stored or retrieved by either the I/O controller or the CAPS processor, with the storage location based on the source and contents of the data word. The I/O buffer also contains three internally generated status bits for each parameter. The status bits are read by the processor in order to determine the status of the data.

(b) ARINC 429 I/O

1) The ARINC 429 I/O assembly provides an interface between the internal I/O bus and the ARINC 429 input and output buses. The assembly handles seven ARINC 429 input buses from external sources, and transmits data to external units via three output buses. Three ARINC 429 input ports are also provided for FCC internal signal wraparound to facilitate built-in test requirements.

(c) Digital Discrete Input

1) The digital discrete assembly provides an interface between the internal FCC I/O bus and discrete inputs. The assembly also receives one ARINC 429 input (from Internal Reference System (IRS) and provides three ARINC 429 inputs (to the MCP, Flight Data Acquisition Unit (FDAU), and EFIS). Data transfer is directed by the I/O controller.

(d) Analog Input

1) The analog input assembly provides for the conversion of analog input signals into digital words for use in FCC mode control computations. Internal test voltages are monitored for software scaling and fault detection. Wraparound signals from pitch servo assembly and roll/yaw servo assembly are converted to digital words for use in software monitoring routines. The converted inputs are multiplexed onto the I/O data bus as two 8-bit words.

(e) Roll/Yaw Servo

1) The roll/yaw servo assembly provides an interface between the CAPS processor and the aileron and rudder hydraulic servo valves. Servo commands from the CAPS processor are converted to analog signals and then summed with the present servo position signal to provide the drive for the airplane hydraulic servo valves.

(f) Discrete/Warn I/O

1) The discrete/warn I/O assembly provides discrete-to-TTL level translation of +28 volt discrete signals for use by the FCC processors. The assembly also monitors the status of the autopilot servo engage signals, and provides warning outputs if a problem is detected.

(g) Monitor
   1) The monitor assembly provides many of the monitor and miscellaneous system functions. These functions include computation and iteration monitoring, system clocks, real-time interrupts, transfer bus termination logic, and two non-volatile fault indicators (used to provide an indication of failures in the CAPS processor or power supply).

(h) Pitch Servo
   1) The servo assembly provides an interface between the CAPS processor and the elevator servo loop. Servo commands from the CAPS processor are converted to analog signals and then summed with the present servo position signal to provide the drive for the aircraft hydraulic servo valves.

(i) Engage/Warn
   1) The engage/warn assembly provides an interface between the CAPS processor and the autopilot servo engage solenoids. The assembly also provides discrete-to-TTL level translation of elevator, aileron, and rudder engage signals, and the left and right autopilot engage signals for use by the CAPS processor. Window comparators are provided to monitor the power supply voltages.

(j) Cross Channel Receiver
   1) The cross channel receiver assembly provides the FCC with an input buffer for data from the other two FCCs in the system. The input data is received on ARINC 429 buses and converted to parallel words for use by the CAPS processor. Separate input buffers are maintained for each of the other FCCs in the system.

(k) Read/Write Memory
   1) The read/write memory assembly provides 4k words of high speed CMOS RAM for use by the CAPS processor as scratch pad memory. The assembly also provides 32 words of non-volatile memory for storage of fault information. In addition to the memory circuits, transfer bus logic decoding for up to 13 outputs is divided. These outputs are used to enable devices which require no more than four transfer bus addresses.

(l) Arm/Warn
   1) The arm/warn assembly provides an interface between the CAPS processor and the autopilot servo arm solenoids. Circuitry is also provided for half of the warn output and for half of the dual FCC power supply monitor. Additional circuits allow the CAPS processor to examine the status of interface data for verification or safety reasons.

(m) I/O Control
 1) The I/O control assembly provides the processing control functions for the various I/O interface assemblies. The assembly services the I/O assemblies by any of three methods: unconditional periodic service, conditional periodic service, or interrupt service. The controller operates independent of the CAPS processor and communicates with the CAPS through I/O data path assembly or by means of an interrupt.

(n) I/O Data Path
 1) The I/O data path assembly provides an interface between the I/O processor and the CAPS processor. The I/O data parameters are stored in the assembly buffer by the I/O processor. The buffer is also read/write accessed by the CAPS processor.

(o) CAPS Processor
 1) The CAPS processor is comprised of a CPU control assembly and a CPU data path assembly. The CPU data path assembly provides the data manipulation and processing circuitry required to execute the CAPS instruction set. The CPU control assembly contains the circuitry required for operation and control of the processor system and program memory.

(p) Program Memory
 1) The program memory assemblies contain the operating program (software) for the FCC. The software provides the commands which control the operation of the CAPS processor.

(q) Power Supply
 1) GUI 115;
 the power supply assembly provides the required operating voltages for the FCC. The input to the power supply is provided by the aircraft 115V ac, 400 Hz system. The outputs of the power supply are 26V ac, +28, -28, +15, -15, +12, -12, +5 Vdc, and a 40 milliamp current source for an engage lever holding circuit contained in the mode control panel.
 2) GUI 001-114, 116-999;
 the power supply assembly provides the required operating voltages for the FCC. The input to the power supply is provided by the aircraft 115V ac, 400 Hz system. The outputs of the power supply are 26V ac, +28, -28, +15, -15, +12, -12, and +5V dc.

(3) FCC Signal Selection
 (a) General
 1) The selection of local or voted FCC input data depends on the mode of operation and type of data used. Local data is used when two of the three sensor sources are invalid and the system operation will only have F/D guidance. Voted data is used for all of the autopilot and flight director modes if only two of three signals are valid.

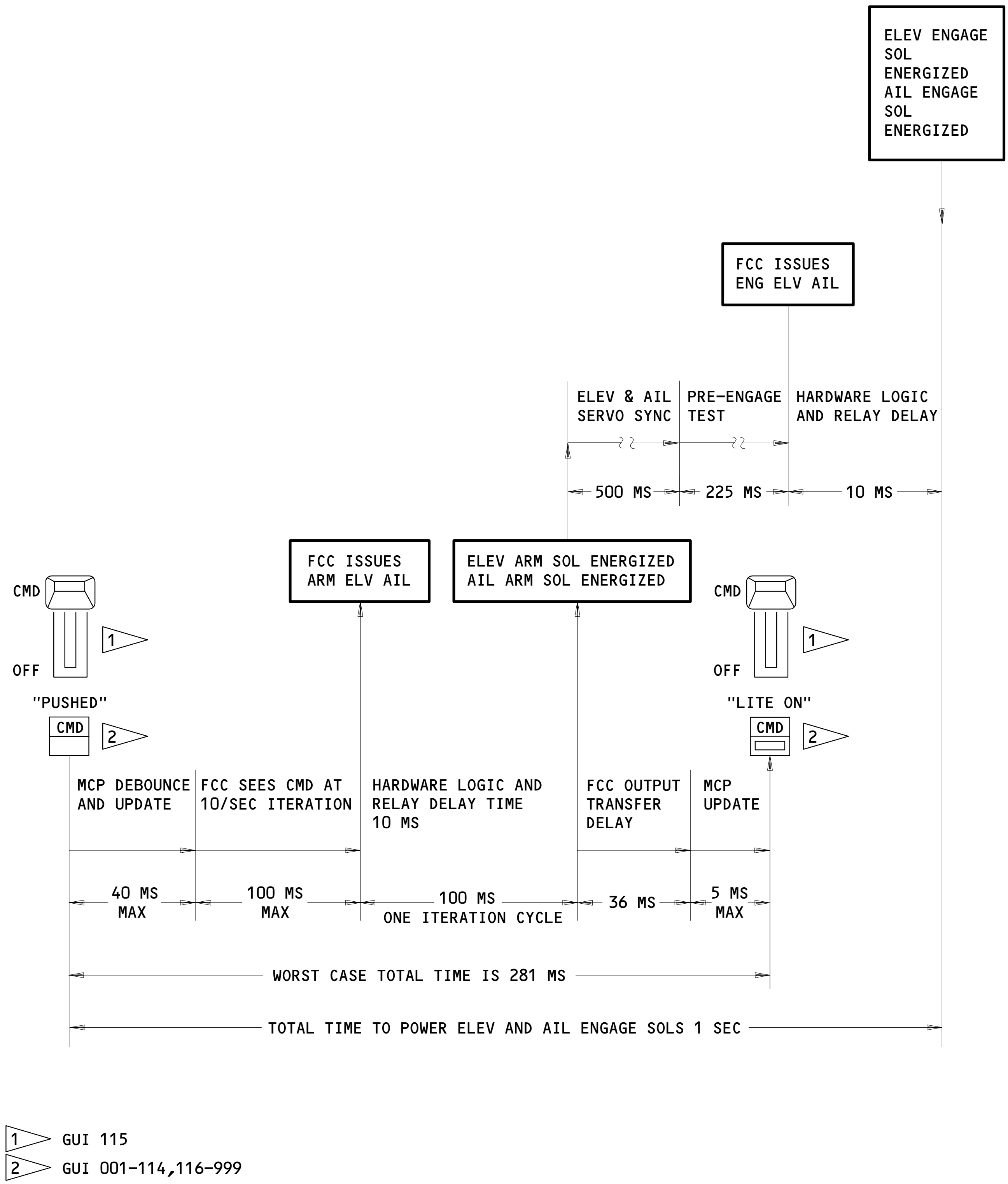
2) Inner loop data is from the Inertial Reference System (IRS) and is used for autopilot stability. These signals are:
   a) Pitch, pitch rate, roll, and roll rate during cruise.
3) Inner loop data is considered critical and always used in Signal Selection Fault Detection (SSFD) voting. The SSFD provides high-integrity input monitoring.
4) Outer loop data is from the following units:
   a) Instrument Landing System (ILS)
   b) Radio Altimeter (RA)
5) The above units provide the following signals:
   a) Angle of attack, altitude rate, altitude, airspeed, mach, and impact pressure (ADC).
   b) VNAV and LNAV control laws (FMC).
   c) Localizer and glideslope deviation and runway course (ILS).
   d) Altitude above terrain (RA).

(b) Signal Selection
1) Triple Sensors
   a) For single-channel engage or F/D modes, local outer loop data is used by the FCC. An invalid outer loop signal required for an autopilot engage mode causes the A/P to hold attitude and display mode fail and caution annunciations (Ref 22-14-00). An invalid outer loop signal does not disengage the A/P. If the invalid condition corrects before a timeout, the mode is automatically re-entered. An invalid local signal required for any F/D mode removes the F/D command bars from the EADI.
   b) During multi-channel CMD, SSFD voted ILS and RA data is used by the FCC. The SSFD voting sets invalid signals to zero or the last voted value and selects the mid value signal of the three inputs. The first invalid signal required for multi-channel operation changes the ASA display (Ref 22-14-00). A second invalid input of the same signal disengages all A/P channels, inhibits engagement in channels with local failures, changes the ASA display, and display warning annunciations (Ref 22-14-00).
   c) Signal selection while on the ground is inhibited. Local sensor data will be used for computation within each FCC.

2) Dual Sensors
   a) Signal selection for ADC data allows selection of one of the two outer loop ADC data sources. For a F/D mode, left ADC data is used by all FCC's if the captain's F/D switch is ON and right ADC data is used if only the first officer's F/D switch is ON. If the autopilot is engaged, left ADC data is used if the left or center channel is engaged and right ADC data is used if the right channel is engaged. If any of the data being used becomes invalid, the FCC will automatically switch over and use the opposite ADC data.
   b) For single-channel engage or F/D modes, failure of both ADC signals causes the autopilot to show mode fail and caution annunciations and to disengage. During multi-channel engage, failure causes no change until after touchdown and disengage.

(4) Flight Director Engage
   (a) The two F/D ON inputs to the FCCs originate at the two F/D switches at each end of the MCP. Either input will generate an F/D SELECT signal. This signal is sent to FCC mode logic and to the EFIS symbol generators for mode and F/D bar annunciation on the EADIs.
   (b) When the airplane is in the air, the F/D SELECT signal is also used to generate a MODE ON output. This is used to prepare the FCC mode logic to activate the CMD or FD modes. The MODE ON output will also result from the following conditions.
      1) Autopilot engaged with F/D OFF.
      2) Autopilot engaged with F/D ON since the A/P mode engage option is used. The FCC holds the airplane bank angle at the time of F/D or CMD engagement.

(5) Autopilot Engagement Timing (Fig. 7)
   (a) CMD Engage Sequence
      1) GUI 001-114, 116-999;
         when a CMD switch/light is pressed, MCP software debounces the switch contacts and encodes the engage request for transmission to the appropriate FCC. The FCC varifies the request and then applies power to the elevator and aileron ARM solenoids. Once armed these servos synchronize with the actual position of the flight controls.



Autopilot Engagement Timing Diagram
Figure 7

2) GUI 115;
when a MCP engage switch is placed in the CMD position, MCP software debounces the switch contacts and encodes the engage request for transmission to the appropriate FCC. The FCC verifies the request and then applies power to the elevator and aileron ARM solenoids. Once armed, these servos synchronize with the actual position of the flight controls.

3) A pre-engage test monitors the synchronization and verifies the servo's response to command. Upon completion of the test, the FCC powers the elevator and aileron engage solenoids.

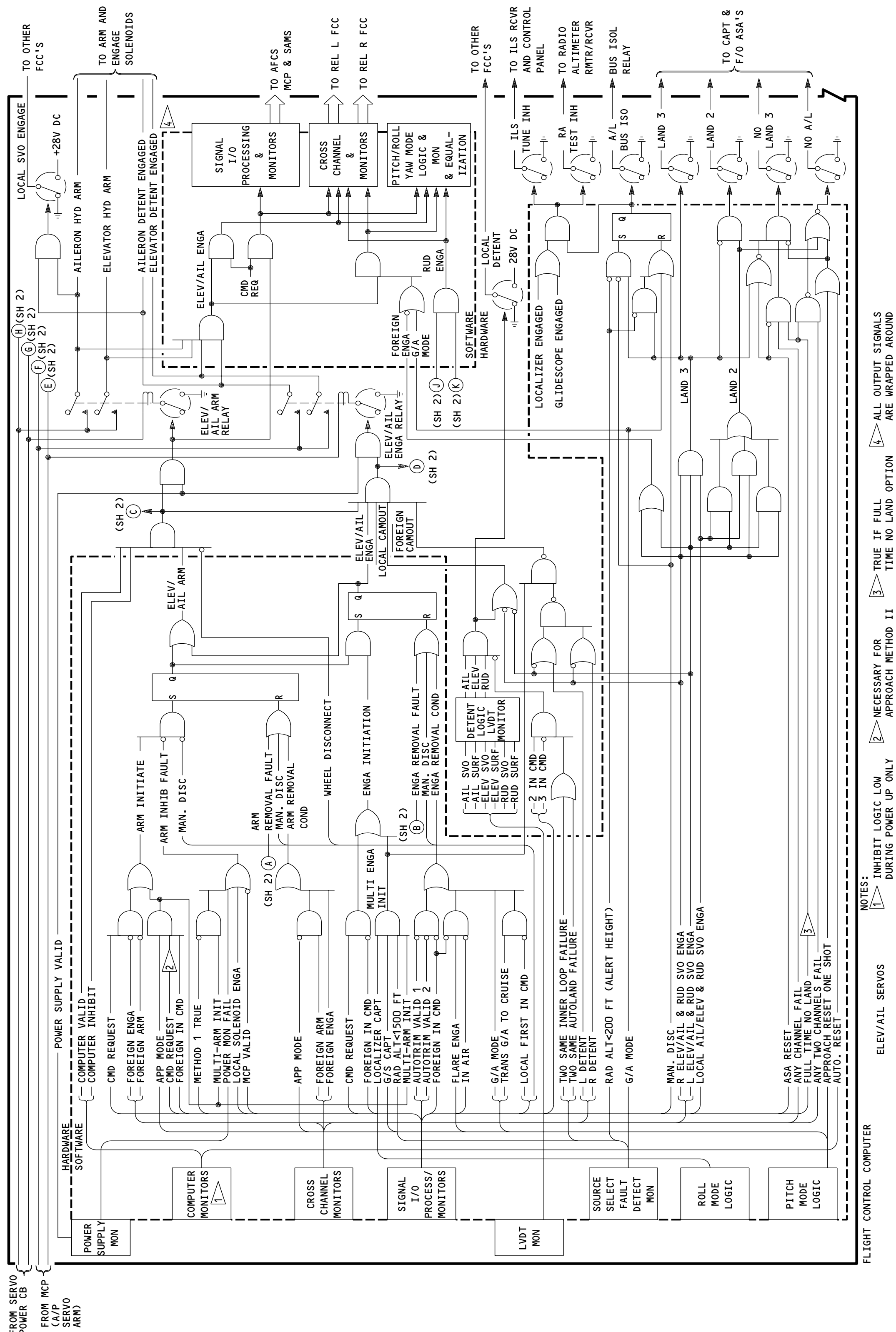
4) GUI 001-114, 116-999;
the dot-bar matrix in the CMD switch/light is lighted by an engage request accepted signal within 41 milliseconds after the ARM solenoids are energized.

5) GUI 115;
the engage switch hold solenoid is energized by a CMD HOLD signal within 41 milliseconds after the arm solenoids are energized.

(6) Autopilot/Flight Director System ARM and ENGAGE logic (Fig. 8)

(a) GUI 115;
a CMD engage request is initiated when a single MCP engage switch is placed in the CMD position. The MCP software verifies that the switch activity is valid and only then encodes the activity into an engage request digital transmission. The engage request is sent to the appropriate FCC. FCC software and hardware monitors determine if the engage request can be implemented.

(b) GUI 001-114, 116-999;
a CMD engage request is initiated when the appropriate MCP switch/light is pressed. The MCP software verifies that the switch activity is valid and only then encodes the activity into an engage request digital transmission. The engage request is sent to the appropriate FCC. FCC software and hardware monitors determine if the engage request can be implemented.



Autopilot/Flight Director System Arm and Engage Logic Schematic
Figure 8 (Sheet 1)

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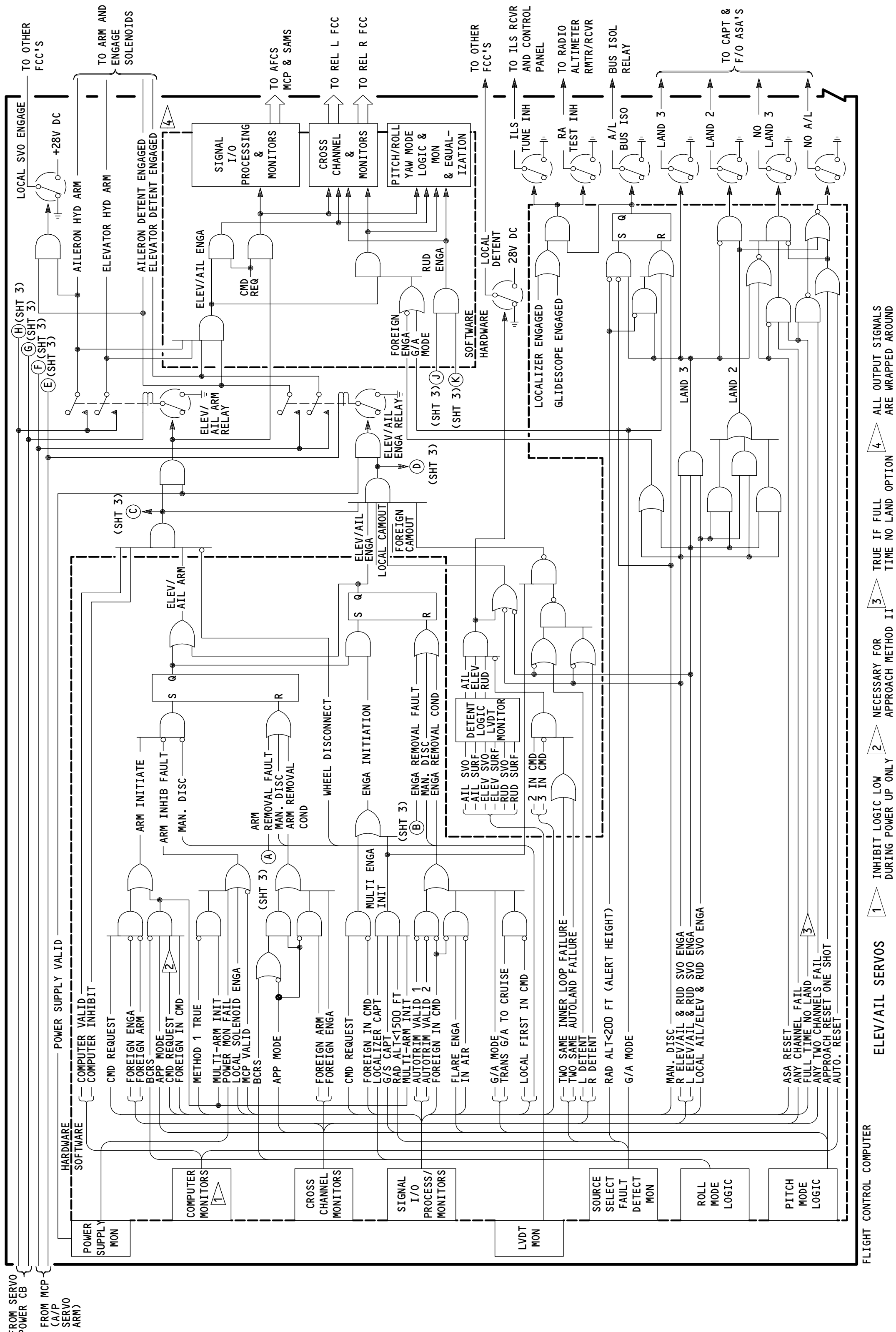
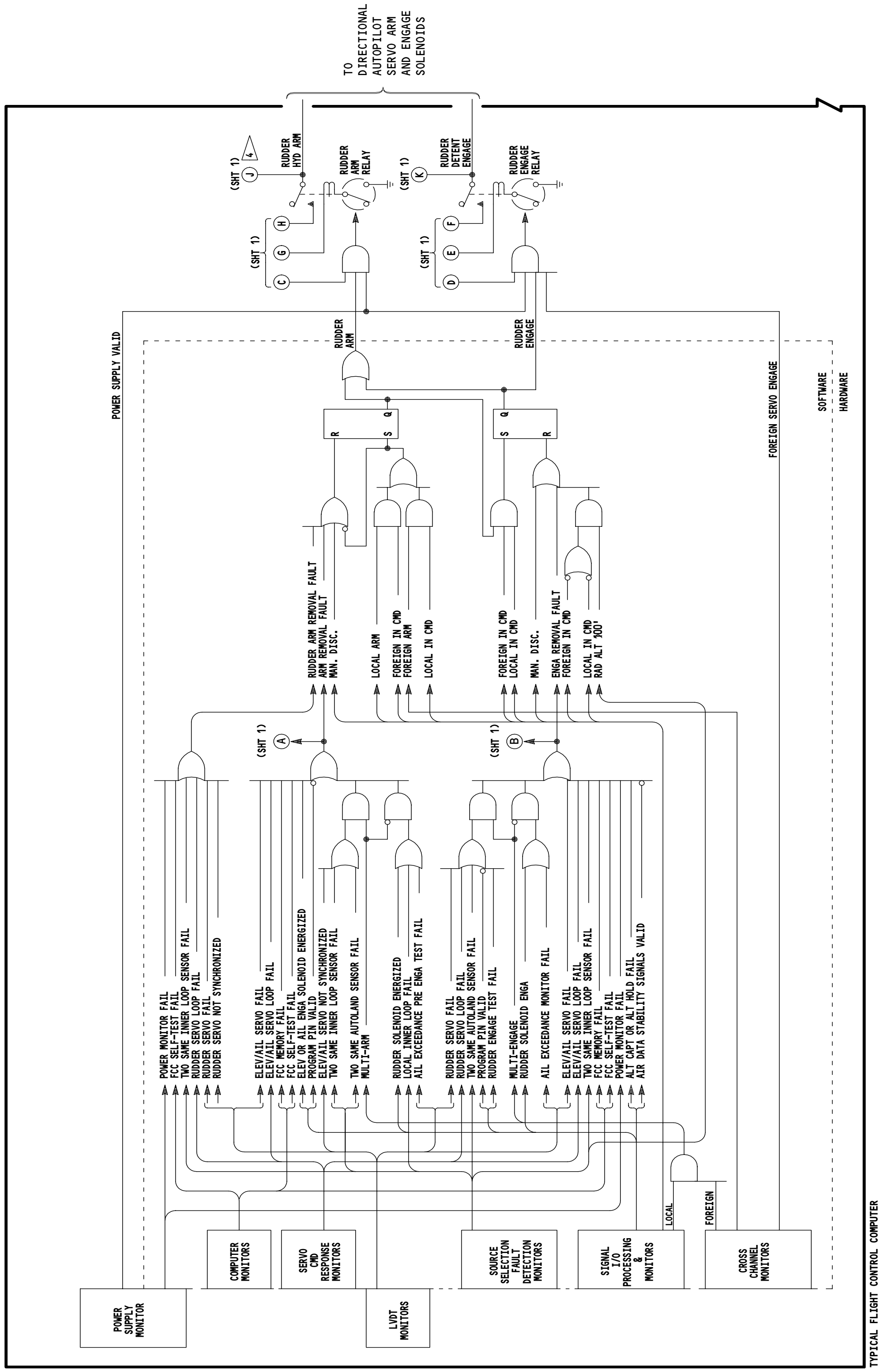


Autopilot/Flight Director System Arm and Engage Logic Schematic
Figure 8 (Sheet 2)



RUDDER SERVOS

Autopilot/Flight Director System Arm and Engage Logic Schematic
Figure 8 (Sheet 3)

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(c) Once the engage request is received, the FCC energizes the appropriate arm solenoids if the FCC is not inhibited. The FCC is inhibited if:
    1) aileron or elevator arm solenoids are already energized
    2) electrical power monitor detects a failure
    3) MCP parameters or interface is invalid

(d) Once armed, the A/P channel transitions from ARM to OFF if:
    1) aileron or elevator engage solenoids are energized
    2) autopilot disengage switch is pressed

(e) GUI 115;
after the arm solenoids are energized, a hold enable signal is sent to CMD HOLD circuitry. This circuitry applies 28 VDC to the MCP engage switch hold solenoid. The energized hold solenoid keeps the engage switch in the CMD position.

(f) GUI 001-114, 116-999;
after the arm solenoids are energized, an engage request acceptance signal is sent to the MCP. This signal causes the MCP to illuminate the appropriate CMD switch/light. The appropriate engage solenoids are also energized.

(g) The appropriate engage solenoids are then energized if the FCC is not inhibited. The FCC remains in the ARM state if:
    1) rudder solenoid is energized
    2) aileron or elevator servos fail or servo loops not verified
    3) FCC program pin status not valid

(h) Once engaged, the A/P channel transitions from CMD to OFF if:
    1) FCC memory failed or self test monitors are not verified
    2) rudder solenoid is energized
    3) aileron or elevator servo or servo loop failed
    4) any two identical inner loop sensors failed or invalid
    5) electrical power monitor detects a failure
    6) aileron exceedance detector monitor detects a failure
    7) ALT HLD or override function cannot be performed
    8) autopilot disengage switch pressed
    9) Manual electric stabilizer trim requested by pitot.

(i) An engage request accepted signal is sent to the EFIS symbol generators for mode annunciation on the EADI. The cross-channel data buses allow each FCC to provide its engage status to the other two.

(7) Multichannel Autopilot Engage
   (a) GUI 115;
       multichannel CMD engage requests are initiated when the approach mode is selected and two or three MCP engage switches are placed in the CMD position. MCP software activity and engage request transmission is the same as single channel engagement. Each engage request is sent to its appropriate FCC. FCC software and hardware monitors determine if the engage requests can be implimented.
   (b) GUI 115;
       once the engage requests are received, each FCC energizes its appropriate arm solenoids if the FCC is not inhibited. The FCC is inhibited if:
       1) FCC program pin status invalid
       2) conditions under single channel arm inhibit occur
   (c) GUI 001-114, 116-999;
       multichannel CMD engage requests are initiated when the APP mode is selected and one FCC is engaged in the CMD mode. Each unengaged FCC automatically energizes its appropriate arm solenoids if the FCC is not inhibited. The FCC is inhibited if:
       1) FCC program pin status invalid
       2) conditions under single channel arm inhibit occur
   (d) Once armed, each A/P channel transitions from ARM to OFF if:
       1) aileron, elevator, or rudder servos fail or servo loops not verified
       2) aileron or elevator servos not synchronized to surface position
       3) yaw servo not synchronized to surface position within 3 seconds
       4) any two identical aileron or elevator inner loop or autoland sensor parameters failed or invalid
       5) any two identical rudder inner loop sensor parameters or interfaces invalid
       6) electrical power monitor detects a failure
       7) conditions under single channel ARM to OFF transition occur
   (e) GUI 115;
       after the arm solenoids are energized, each engage switch is held in the CMD position.

(f) GUI 001-114, 116-999;
after the arm solenoids are energized, each CMD switch/light is lighted.

(g) The appropriate engage solenoids are then energized if each FCC is not inhibited. The FCC remains in the arm state if:
1) aileron, elevator, or rudder inner loop data not valid
2) FCC program pin status not valid

(h) Once engaged, each A/P channel will transition from CMD to OFF if:
1) FCC memory failed or self test monitors are not verified
2) aileron, elevator, or rudder servo or servo loop fails
3) any two identical inner loop sensors failed or invalid
4) electrical power monitor detects a failure
5) any two identical autoland sensors failed or invalid
6) FCC program pin status invalid
7) ALT HOLD or override function cannot be performed
8) autopilot disengage switch pressed

(i) The cross-channel data buses allow each FCC to provide its engage status to the other two. If the first-in-command channel (first FCC engaged in CMD during multichannel operation) is disengaged, the first-in-command status is automatically transferred to the next channel as follows:. . . LEFT, RIGHT, CENTER, LEFT, etc..

B. BITE

(1) The FCC provides in-flight detection of AFDS and related component failures. Detection includes:
(a) System fault detection and fault location.
(b) Monitoring of servos, sensors, input interfaces, and output interfaces.
(c) Self Test
(d) Storage of internal fault identification.

(2) Fault Detection and Location.
(a) The FCC detects flight faults in the following AFDS System Components:
1) Autopilot Servos

2) Transducer Inputs and Outputs
3) MCP - FCC Interfaces
4) FCC

(b) The FCC, in conjunction with the MCDP, can identify the location of the flight faults listed above. The MCDP ground tests verify the proper operation of the MCP front panel switch/lights, switches, and displays. The ground tests also verify proper operation of the A/P disengage and go-around switches.

(3) FCC Interface Monitors

(a) The A/P servo control functions are checked by command/response monitors which verify the function of the servo loop, and by current drive wraparounds, which check the circuitry to the servo drivers. The command/response monitors verify that the actuator moves when commanded. The current drive wraparounds verify that the servo is functioning correctly by comparing actual servo movement with the movement commanded by the FCC software.

Detent comparators verify that the aircraft control surfaces are following the commanded signals. Dual comparators, one in hardware and one in software, monitor the servo position and the aircraft control surface position. An incorrect comparison generates a fault signal for use in software.

(b) Linear variable differential transducer (LVDT) signal outputs are checked by hardware monitors. The hardware monitors detect faults caused by shorted wires, broken wires, or loss of excitation, and generate a fault signal for use in software. These monitors are used for the servo position sensor inputs. Signal inputs from sensors other than LVDTs are monitored by software. Signal selection failure detection routines are used to determine sensor correctness and to provide voter outputs for multichannel operation. Incorrect sensor signals are voted out of the final results.

(c) Discrete input and output interfaces are monitored by software which detects shorted wires. These monitors are used for the caution/warning inputs and outputs, engage and arm solenoid power, trim outputs, engage and arm commands and inputs, and autoland status annunciator inputs and outputs. The interface monitor circuits are tested, as appropriate, at power up or prior to engagement of autopilot/autoland.

The ARINC 429 inputs from system LRUs are checked in two ways. First, the FCC checks the sign status matrix to verify the validity of the incoming data. The buffer memory locations are also periodically tested to determine if the data words have been updated. If the words have not been updated, then a fault is assumed and the data words at the tested memory locations are not used.

The ARINC 429 test words are generated by the FCC. The test words are used to verify the correct operation of the circuitry involved in communication with the mode control panel and the other FCCs in the system. The test words are sent to the receiving system unit, operated on, and then transmitted back to the originating FCC. The results are compared with the expected answers to determine if faults exist.

(4) FCC Self Test

(a) The FCC provides a number of self-tests. The tests are divided into two basic categories: power-up/pre-engage tests and operation verification tests. The power-up/pre-engage category includes tests which would cause adverse system reactions if performed during flight. The operation verification category provides critical function monitoring such as the CAPS processor memories, interfacing electronics, and power supplies. These types of functions are monitored on a continuous or recurring basis.

(b) CAPS Processor

1) The CAPS processor self-test is implemented using several techniques, and provides checks of processor components and data, address, and control paths. During execution of the self-test program, the program monitor counts the number of microcycles in each iteration of the self-test program and also monitors the execution time of the program. A second hardware-implemented check is provided by the processor parity monitor. The parity monitor checks the instruction mapping into the microprogram memory. Both self-test program monitors are checked at power up.

2) The computation rate of the processor is continuously checked during the normal operation of the FCC. During each 25 millisecond computation period, a software generated signal is sent to the program monitor. If the signal is more than ±5 milliseconds out of tolerance, a fault signal is generated. The circuit does not reset until two consecutive within tolerance signals are received. The monitor circuit is checked at power up for correct fault response.

(c) Memories

1) The program memory is checked by a software routine. The checksum of the ROM contents is compared with the expected sum (stored in memory), and a fault signal is generated if the comparison fails.

2) The RAM is checked by three monitors in the FCC. At power up, a write-then-read test is performed on each memory location. During flight, the RAM is in use, so two RAM locations, with complementary addresses, are checked with a write-then-read test. The in-flight test verifies the condition of the address and data lines. During normal operation, parity bits are added to words written into memory and checked when words are read from memory.

(d) Interfacing Electronics

1) Internal I/O monitoring is performed primarily in software. If the I/O logic fails, the CAPS processor indicates a fault. The I/O processor address and data lines are checked by the CAPS processor with a write-then-read test. A failure is indicated if the CAPS processor does not receive the proper word. Wraparound monitoring is used to check, in part, the ARINC 429 I/O and analog I/O circuits. Test words are wrapped around to test the 429 UART and 429 multiplexer circuits. The MCP and MCDP outputs provide the low-speed 429 wraparounds and the FCC left cross channel output provides the high-speed 429 wraparound. The analog wraparound signals are generated and fed back into the FCC analog inputs. The analog signals wrapped around are servo position, force link position, servo drive, and commanded output.

2) The inertial reference system (IRS) input is brought in, through two separate paths. The IRS information is stored in separate memory locations, thus allowing data comparisons by the CAPS processor. If a fault exists in either path, the comparison will indicate an inequality.
3) The ARINC 429 outputs which are not wrapped around are checked by hardware monitors. The monitors detect failures of one side of each ARINC 429 driver output and send a fail signal to the software if a fault is detected.
4) The A/D and D/A converter is used to check the reference voltage used in the conversion logic. The reference voltage is scaled and inputted. The digital result is compared in software, with the expected result, and a fault is indicated if the voltage is out of tolerance. The analog wraparounds cannot be used to find reference voltage faults because both the D/A and A/D converters use the reference voltage generated by the A/D converter. A separate reference is, therefore, required.

(e) Power Supplies
1) The power supply voltages are continuously monitored for overvoltage, undervoltage, and overcurrent conditions. The +5, +15, and -28 Vdc outputs are monitored by additional overvoltage monitors which latch the entire power supply in a shut-down state. The 115 Vac input power must be interrupted to reset these monitors. The voltage and frequency of the 26 Vac supply output are checked by software.

(5) Internal Fault Storage
(a) The FCC provides permanent storage for up to five internal faults which result in an FCC invalid, and temporary storage for five words of fault data per flight leg.
(b) Faults detected by FCC self-test are stored in memory for access by shop support equipment. A maximum of five faults can be stored at one time. This fault data can only be removed from memory in the maintenance shop.
(c) Fault data in temporary storage is formatted for transmission over a ARINC 429 data bus to the MCDP. The FCC clears temporary data at the start of each flight leg. Data provided to the MCDP includes:
1) Fault identity and location
2) Source of fault detection
3) Fault effect on flight deck
4) Fault intermittent or constant
5) Fault occured in-flight or on-ground.

AUTOPILOT/FLIGHT DIRECTOR POWER

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
ANNUNCIATOR - (FIM 22-14-00/101)				
AUTOLAND STATUS, CAPT, N70				
AUTOLAND STATUS, F/O, N71				
CIRCUIT BREAKER -	--		FLT COMPT, P11	
AUTOFLIGHT WARN, C521		1	11A17	*
FLIGHT CONT CMPTR PWR CENTER, C515		1	11E19 OR 11E20	*
FLT CONT COMPUTER POWER LEFT, C513		1	11E17	*
FLT CONT CMPTR PWR RIGHT, C514		1	11E35	*
FLIGHT CONT CMPTR SERVO CENTER, C524		1	11E20 OR 11E21	*
FLT CONT COMPUTER SERVO LEFT, C522		1	11E18	*
FLT CONT CMPTR SERVO RIGHT, C523		1	11E36	*
MAINT CONT DSPL, C520		1	11S6	*
MODE CONT PNL LEFT, C516		1	11E16	*
MODE CONT PNL RIGHT, C517		1	11E34	*
COMPUTER - C FLIGHT CONTROL, M140	--	1	119BL, MAIN EQUIP CTR, E2-3	22-11-01
COMPUTER - L FLIGHT CONTROL, M139	--	1	119BL, MAIN EQUIP CTR, E2-1	22-11-01
COMPUTER - R FLIGHT CONTROL, M141	--	1	119BL, MAIN EQUIP CTR, E2-2	22-11-01
COMPUTER - (FIM 22-31-00/101)				
THRUST MANAGEMENT, M183				
COMPUTER - (FIM 31-41-00/101)				
L EICAS, M10181				
R EICAS, M10182				
COMPUTER - (FIM 34-12-00/101)				
L AIR DATA, M100				
R AIR DATA, M101				
COMPUTER - (FIM 34-61-00/101)				
L FLIGHT MANAGEMENT, M134				
R FLIGHT MANAGEMENT, M135				
INDICATOR - (FIM 29-31-00/101)				
HYDRAULIC SYSTEM CONTROL PANEL, M10				
INDICATOR - (FIM 34-13-00/101)				
MACH AIRSPEED, CAPT, N1				
MACH AIRSPEED, F/O, N41				
LIGHT - (FIM 22-14-00/101)				
A/P DISC				
AUTOPILOT CAUTION, L269				
MODULE - (FIM 27-09-00/101)				
1L SPOILER CONTROL, M530				
2R SPOILER CONTROL, M534				
3L SPOILER CONTROL, M532				
MODULE - (FIM 27-48-00/101)				
C STABILIZER POSITION, M10409				
L STABILIZER POSITION, M10408				
R STABILIZER POSITION, M10410				
MODULE - (FIM 34-16-00/101)				
ALTITUDE ALERT, M617				
PANEL - AFCS MODE CONTROL, M90	--	1	FLT COMPT, P55	22-11-02
PANEL - (FIM 22-41-00/101)				
MAINTENANCE CONTROL DISPLAY, M168				
RECEIVER - (FIM 34-31-00/101)				
C ILS, M157				
L ILS, M156				
R ILS, M158				

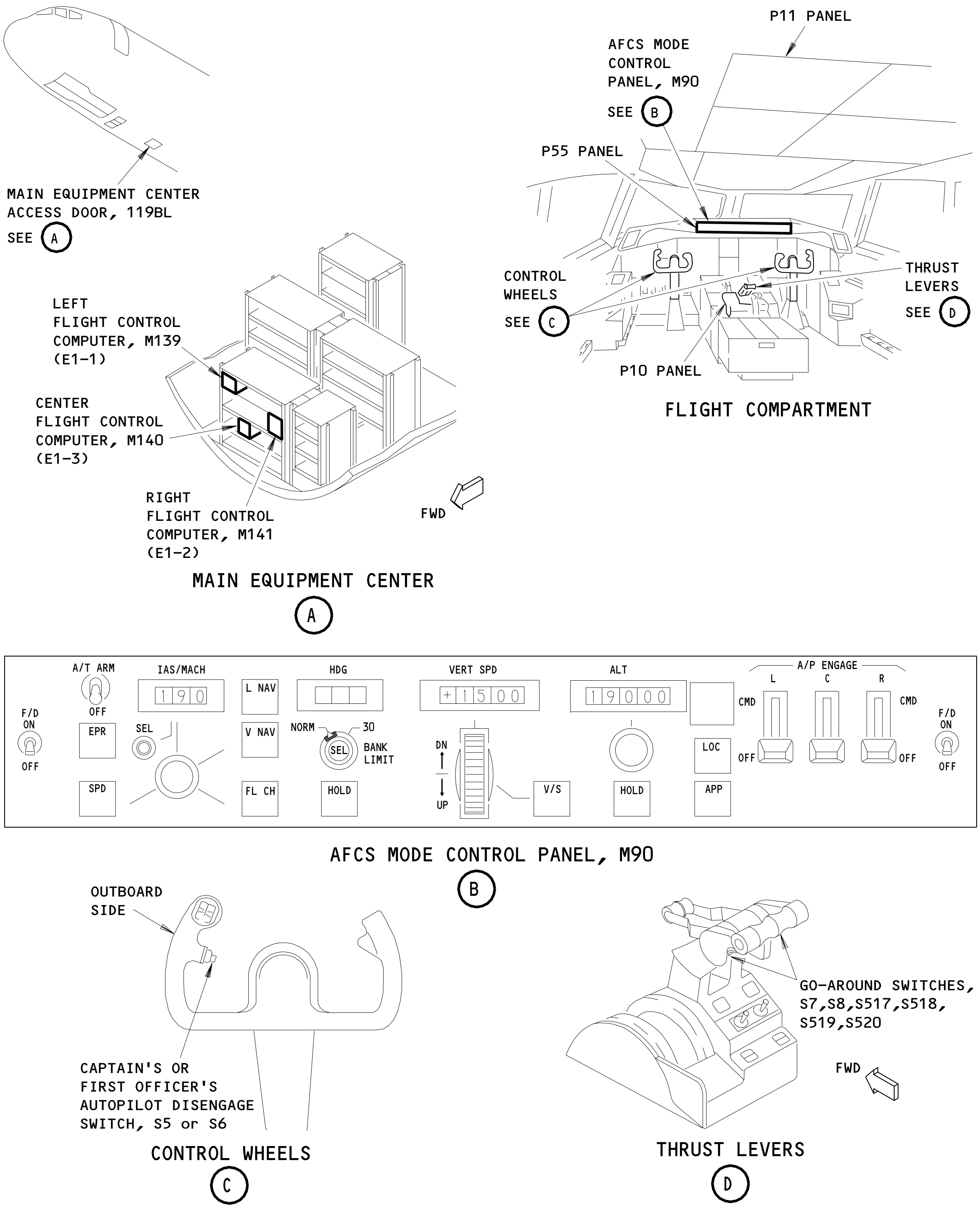
* SEE THE WDM EQUIPMENT LIST

1 GUI 001-114, 116-999   2 GUI 115

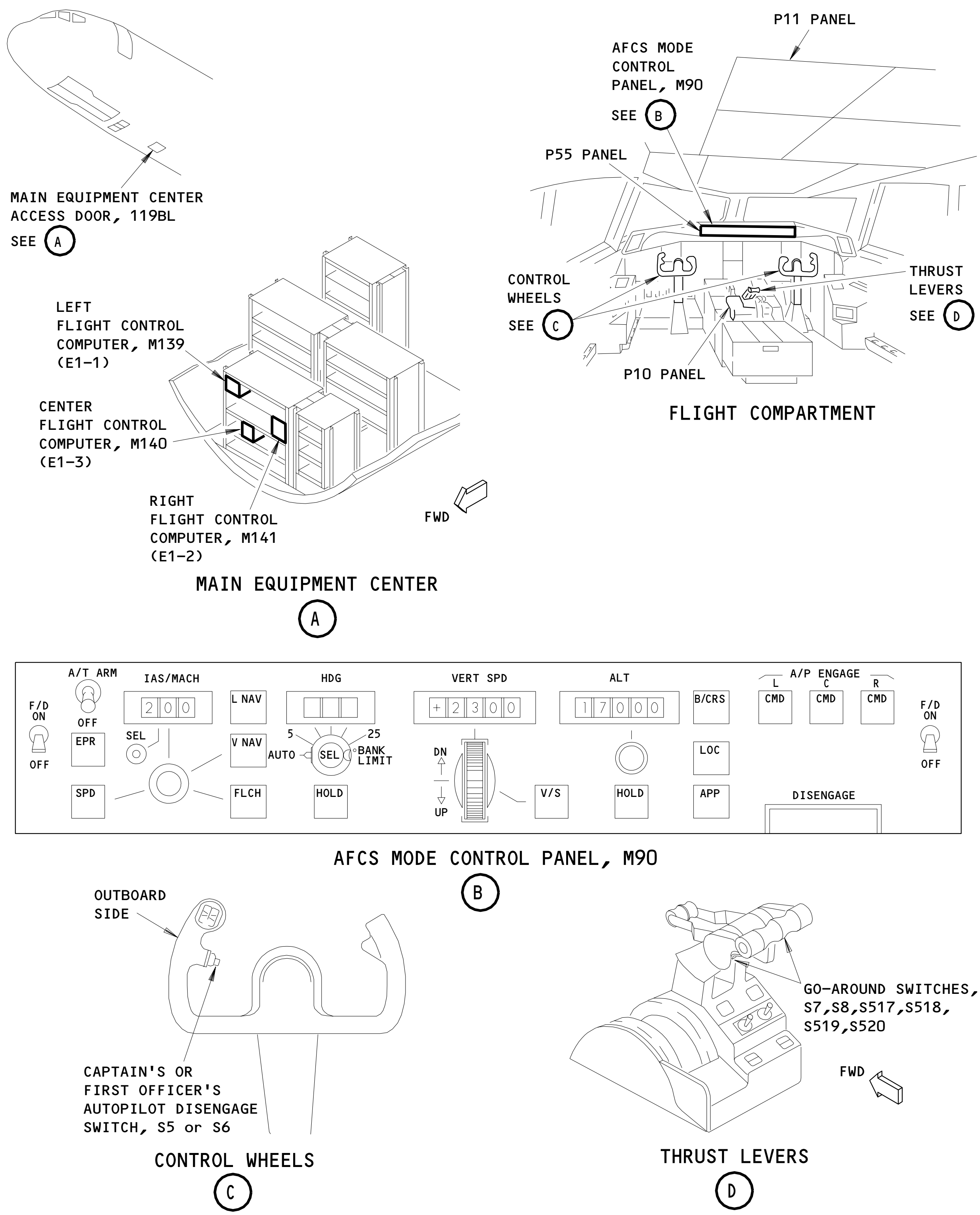
Autopilot/Flight Director Power - Component Index
Figure 101 (Sheet 1)

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
RELAY - (FIM 31-01-06/101)				
CENTER BUS ISOLATION, K122				
RELAY - (FIM 31-01-36/101)				
AIR/GROUND, SYS NO. 1, K142				
RELAY - (FIM 31-01-37/101)				
AIR/GROUND, SYS NO. 2, K202				
SERVO - (FIM 22-12-00/101)				
C AUTOPILOT PITCH CONTROL, M272				
L AUTOPILOT PITCH CONTROL, M271				
R AUTOPILOT PITCH CONTROL, M273				
SERVO - (FIM 22-13-00/101)				
C AUTOPILOT LATERAL CONTROL, M10041				
C AUTOPILOT ROLLOUT GUIDANCE, M278				
L AUTOPILOT LATERAL CONTROL, M10040				
L AUTOPILOT ROLLOUT GUIDANCE, M277				
R AUTOPILOT LATERAL CONTROL, M10042				
R AUTOPILOT ROLLOUT GUIDANCE, M279				
SWITCH - (FIM 34-22-00/101)				
INSTRUMENT SOURCE SELECT, CAPT, S1				
INSTRUMENT SOURCE SELECT, F/O, S9				
SWITCH - AUTOPILOT DISENGAGE, CAPT, S5	--	1	FLT COMPT CONTROL WHEEL, CAPT	22-11-03
SWITCH - AUTOPILOT DISENGAGE, F/O, S6	--	1	FLT COMPT CONTROL WHEEL, F/O	22-11-03
SWITCH - GO-AROUND, S7,S8,S517,S518,S519,S520	--	6	FLT COMPT, P10, THRUST LEVERS	22-11-04
SYMBOL GENERATOR - (FIM 34-22-00/101)				
C EFIS, M149				
L EFIS, M148				
R EFIS, M150				
TRANSDUCER - (FIM 22-12-00/101)				
C ELEVATOR NEUTRAL SHIFT, TS5153				
L ELEVATOR NEUTRAL SHIFT, TS5151				
R ELEVATOR NEUTRAL SHIFT, TS5152				
TRANSMITTER/RECEIVER - (FIM 34-33-00/101)				
C RADIO ALTIMETER, M204				
L RADIO ALTIMETER, M202				
R RADIO ALTIMETER, M203				
UNIT - (FIM 27-51-00/101)				
FLAP/SLAT ELECTRONICS 1, M10331				
FLAP/SLAT ELECTRONICS 2, M10332				
FLAP/SLAT ELECTRONICS 3, M10333				
UNIT - (FIM 31-01-06/101)				
INSTRUMENT BUS SENSING, CAPT, M10374				
INSTRUMENT BUS VOLTAGE SENSE, M1079				
UNIT - (FIM 31-51-00/101)				
WARNING ELECTRONICS, P51				
UNIT - (FIM 34-21-00/101)				
C INERTIAL REFERENCE, M160				
L INERTIAL REFERENCE, M159				
R INERTIAL REFERENCE, M161				

Autopilot/Flight Director Power - Component Index
Figure 101 (Sheet 2)



Autopilot/Flight Director Power - Component Location
Figure 102 (Sheet 1)



Autopilot/Flight Director Power - Component Location
Figure 102 (Sheet 2)

FLIGHT CONTROL COMPUTER - REMOVAL/INSTALLATION

1. General
   A. Three Flight Control Computers (FCC) are installed in the main equipment center. The left FCC is on shelf E2-1, the center FCC is on shelf E2-3, and the right FCC is on shelf E2-2. The computers have hold-down hooks on the front to attach them to the rack. One multi-pin connector divided into three areas is found at the rear of each unit.

TASK 22-11-01-004-001

2. Remove the Flight Control Computer
   A. References
      (1) 20-10-01/401, E/E Rack Mounted Components
      (2) 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zones
          119/120   Main Equipment Center

      (2) Access Panels
          119BL   Main Equipment Center

   C. Prepare For Removal

      S 864-006
      (1) Set the two F/D switches on the MCP to OFF.

      S 864-007
      (2) For left FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E17, FLT CONT COMPUTER PWR LEFT

      S 864-008
      (3) For center FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E19, or 11E20, FLIGHT CONT CMPTR PWR CENTER

      S 864-009
      (4) For right FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E35, FLT CONT CMPTR PWR RIGHT
   D. Remove the Flight Control Computer

S 914-002

CAUTION: DO NOT TOUCH THE FLIGHT CONTROL COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE FLIGHT CONTROL COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

S 024-003
(2) Remove the FCC (Ref 20-10-01).

S 434-010
(3) Install a protective cover on the connector at the rear of the FCC.

TASK 22-11-01-404-011
3. Install the Flight Control Computer
A. References
(1) 20-10-01/401, E/E Rack Mounted Components
(2) 20-41-01/201, Electrostatic Discharge Sensitive Devices
(3) 22-00-02/201, Autoflight BITE
(4) 24-22-00/201, Electrical Power - Control
B. Access
(1) Location Zones
119/120 Main Equipment Center

(2) Access Panels
119BL Main Equipment Center

C. Install the Flight Control Computer

S 914-004

CAUTION: DO NOT TOUCH THE FLIGHT CONTROL COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE FLIGHT CONTROL COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

S 034-012
(2) Remove the protective cover from the connector at rear of the FCC.

S 424-013

(3) Install the FCC (Ref 20-10-01).

D. Test the Flight Control Computer.

S 864-014

(1) Supply electrical power (Ref 24-22-00).

S 864-015

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11A17, AUTOFLIGHT WARN
   (b) 11E17, FLT CONT COMPUTER POWER LEFT
   (c) 11E18, FLT CONT COMPUTER SERVO LEFT
   (d) 11E19, OR 11E20, FLIGHT CONT CMPTR PWR CENTER
   (e) 11E20, or 11E21, FLIGHT CONT CMPTR SERVO CENTER
   (f) 11E35, FLT CONT CMPTR PWR RIGHT
   (g) 11E36, FLT CONT CMPTR SERVO RIGHT

S 864-016

(3) Make sure MCDP is out of Ground Test Mode. Set MCDP to OFF or go into Flight Faults Mode.

NOTE: Before MCDP Ground Test 01-FCC is done, the MCDP must be put out of Ground Test Mode. This removes faults that are corrected from the MCDP memory.

S 864-017

(4) Put the MCDP into Ground Test Mode (Ref 22-00-02).

S 724-005

(5) Do MCDP Ground Test 01-FCC (Ref 22-00-02).
   (a) Make sure no failures occur during the test.

E. Put the Airplane Back to Its Usual Condition.

S 864-018

(1) Set the MCDP to off.

S 864-019

(2) Remove electrical power if it is not necessary (Ref 24-22-00).

AFCS MODE CONTROL PANEL - MAINTENANCE PRACTICES

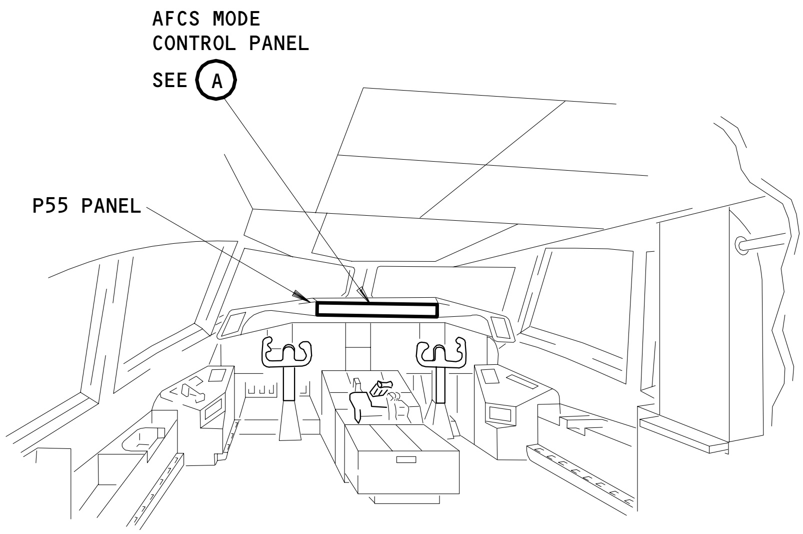
1. General
   A. This procedure has these tasks:
      (1) The removal of the AFCS Mode Control Panel (MCP).
      (2) The installation of the MCP.
      (3) The removal of a mode select switch and/or a lamp for the mode select switch.
      (4) The installation of a mode select switch and/or a lamp for the mode select switch.
   B. The AFCS Mode Control Panel (MCP) is in the center of the glareshield, P55, in the flight compartment. The MCP is installed on supports in the glareshield structure. Four captive screws in the bottom of the glareshield attach the MCP to the glareshield. Three electrical connectors are at the rear of the MCP. An opening at the rear of the MCP automatically connects with the equipment cooling system when the panel is installed.

TASK 22-11-02-022-001

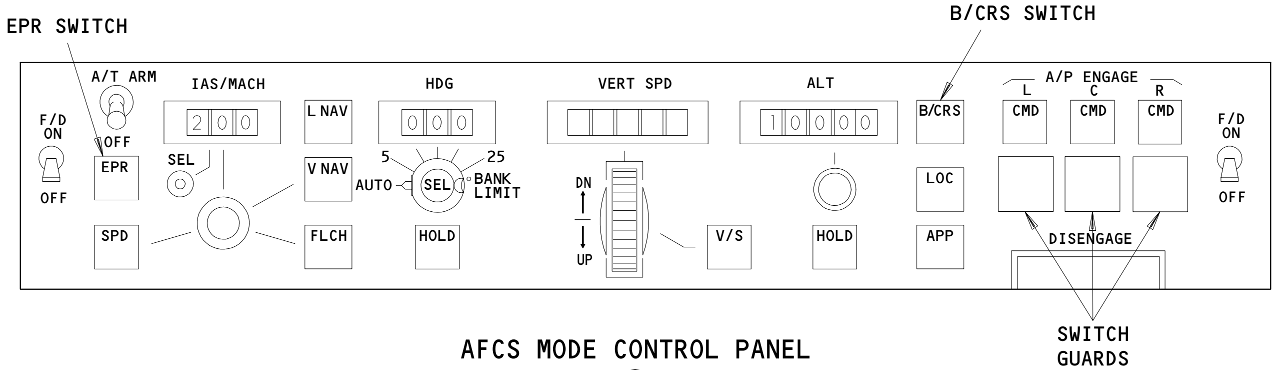
2. Mode Control Panel Removal (Fig. 201)
   A. Reference
      (1) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zone
          211/212 Flight Compartment

   C. Prepare For Removal

      S 862-002
      (1) Open these circuit breakers and attach DO-NOT-CLOSE tags:
          (a) P11 Overhead Circuit Breaker Panel:
              1) 11E16, MODE CONT PNL LEFT
              2) 11E34, MODE CONT PNL RIGHT
              3) 11N30, INSTRUMENT & PANEL GLARE SHIELD
              4) 11P1, IND LTS L 1
              5) 11P2, IND LTS L 2
              6) 11P28, IND LTS R 1
              7) 11P29, IND LTS R 2
   D. Remove the Mode Control Panel

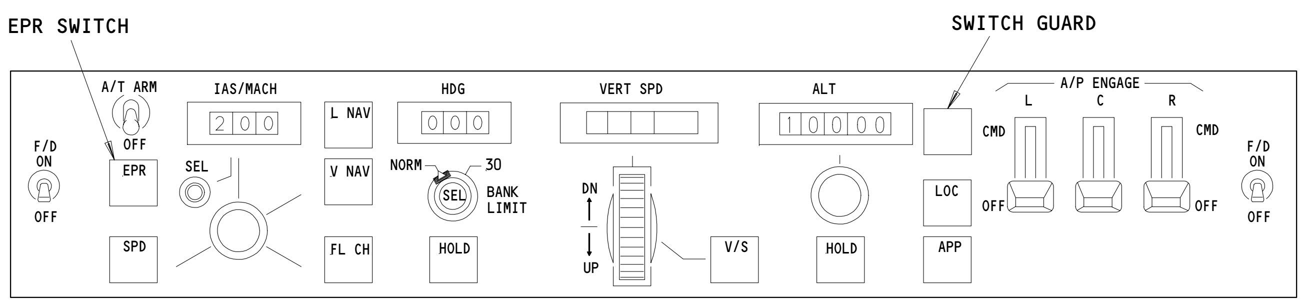


FLIGHT COMPARTMENT



AFCS MODE CONTROL PANEL

A 1



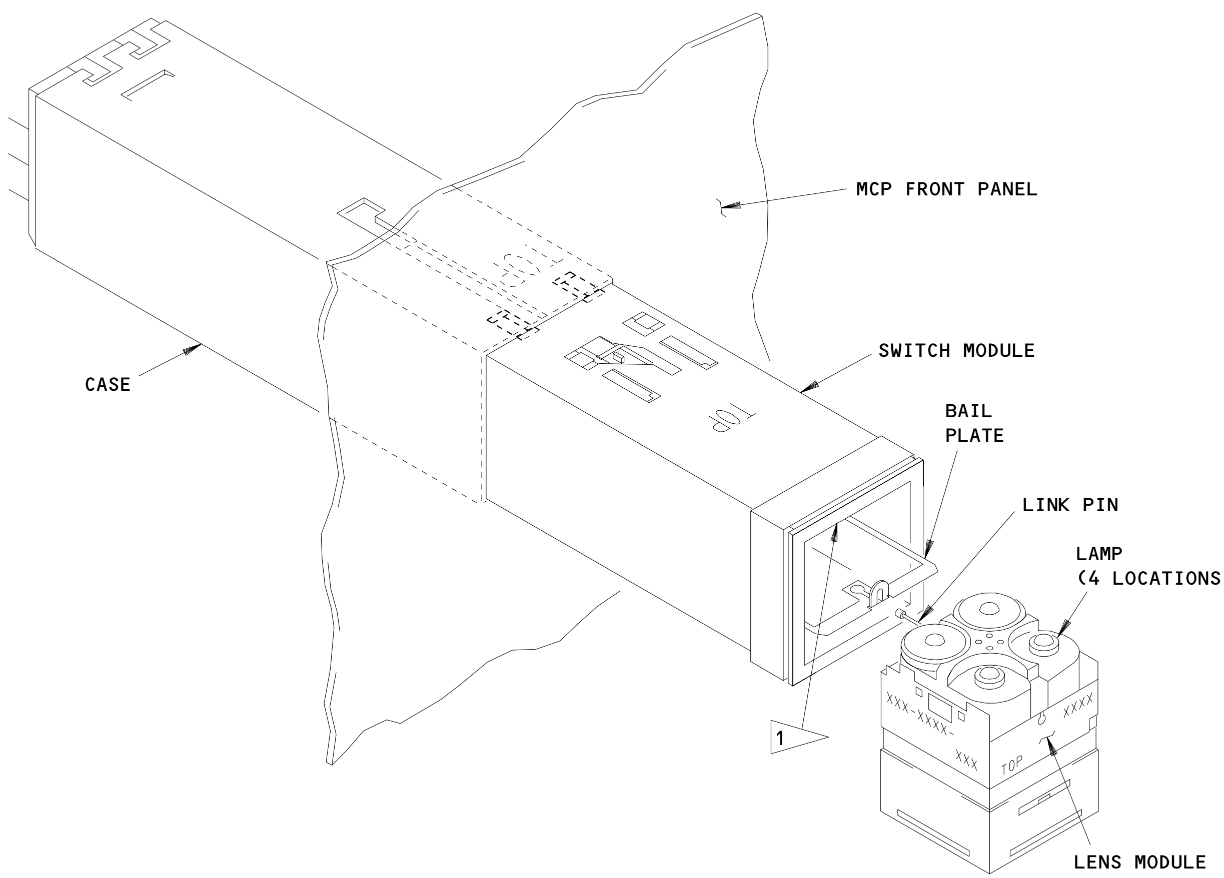
AFCS MODE CONTROL PANEL

A 2

1 GUI 001-114,116-999

2 GUI 115

AFCS Mode Control Panel Installation
Figure 201



MODE SELECT SWITCH

1 THE JACKSCREW IS AT THE TOP IN THE SWITCH MODULE, AND BEHIND THE MATING PLATE FOR THE LENS MODULE.

Mode Select Switch Removal/Installation
Figure 202

S 912-003

CAUTION: DO NOT TOUCH THE MCP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCP.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 342-004

(2) Loosen the four screws on the bottom of the MCP supports in the glareshield that attach to the MCP.

S 012-005

CAUTION: CAREFULLY PULL THE MCP OUT FROM ITS SUPPORTS IN THE GLARESHIELD TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO THE ELECTRICAL CABLES COULD OCCUR.

(3) Pull the MCP out from its supports in the glareshield until you can get access to the electrical connectors at the rear of the MCP.

S 862-043

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

(4) Disconnect the electrical connectors.
(a) Apply protective covers to the electrical connectors.

S 022-038
(5) Remove the MCP from its supports in the glareshield.

TASK 22-11-02-402-007
3. Mode Control Panel Installation (Fig. 201)
A. References
(1) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Control
B. Access
(1) Location Zone
211/212 Flight Compartment

C. Install the Mode Control Panel

S 912-008

CAUTION: DO NOT TOUCH THE MCP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCP.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 342-009
(2) Remove the protective covers from the electrical connectors at the rear of the MCP.

S 342-010

CAUTION: MAKE SURE THE COVER ON THE AIR INLET OF THE MCP IS REMOVED. DAMAGE TO EQUIPMENT COULD OCCUR IF THIS OPENING IS BLOCKED.

(3) Make sure the cover on the air inlet of the MCP is removed.

S 212-011

CAUTION: MAKE SURE THE AIR OUTLET IN THE GLARESHIELD THAT CONNECTS WITH THE REAR OF THE MCP IS CLEAR OF BLOCKAGE. EQUIPMENT DAMAGE COULD OCCUR IF THIS OPENING IS BLOCKED.

(4) Make sure the air outlet in the glareshield that connects with the rear of the MCP is clear of blockage.

S 412-012

CAUTION: CAREFULLY PUT THE MCP INTO ITS SUPPORTS IN THE GLARESHIELD TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO THE ELECTRICAL CABLES COULD OCCUR.

(5) Move the MCP into its supports in the glareshield until the electrical cables can be connected to the rear of the MCP.

S 862-044

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

(6) Connect the electrical connectors.

S 422-014

(7) Move the MCP into its installed position.

S 432-015

(8) Tighten the four screws on the bottom of the MCP supports in the glareshield.

S 752-016

(9) Make sure the new MCP has the same switch configuration as the MCP that was removed (Fig. 201).
   (a) Make sure the new MCP has an EPR lens installed. If the new MCP has an N1 lens installed, replace the N1 lens with the EPR lens from the removed MCP.
   (b) GUI 115;
   make sure the new MCP has the BARO L/R lens installed above the LOC switch. If the new MCP has a switch guard installed, replace the switch guard with the BARO L/R lens from the removed MCP.
   (c) GUI 001-114, 116-999;
   make sure the new MCP has a B/CRS lens installed. If the new MCP has a switch guard installed above the LOC switch, replace the switch guard with the B/CRS lens from the removed MCP.
   (d) GUI 001-114, 116-999;
   make sure the new MCP has switch guards installed below the CMD switches. If the new MCP has CWS lenses installed, replace the CWS lenses with the switch guards from the removed MCP.

D. Test the Mode Control Panel

S 862-017

(1) Supply electrical power (AMM 24-22-00/201).

S 862-018

(2) Set the two F/D switches on the MCP to OFF.

S 862-068
(3) Set the MCP A/T ARM Switch to ARM

S 862-069
(4) Set the OVERHEAD MASTER DIM/TEST IND LIGHTS to BRIGHT.

S 862-019
(5) Remove DO-NOT-CLOSE tags and close these circuit breakers:
   (a) P11 Overhead Circuit Breaker Panel:
      1) 11E16, MODE CONT PNL LEFT
      2) 11E34, MODE CONT PNL RIGHT
      3) 11N30, INSTRUMENT & PANEL GLARE SHIELD
      4) 11P1, IND LTS L 1
      5) 11P2, IND LTS L 2
      6) 11P28, IND LTS R 1
      7) 11P29, IND LTS R 2

S 722-020
(6) Make sure the MCP lightplate comes on.

S 712-021
(7) Do MCDP Ground Test 04-MCP (AMM 22-00-02/201).
   (a) Make sure no failures occur during the test.

S 862-063
(8) Open the following circuit breaker:
   (a) 11E16 MODE CONT PNL L

S 712-064
(9) Select Flight Faults, then re-select Ground Faults.

S 712-065
(10) Do MCDP Ground Test 04-MCP (AMM 22-00-02/201).
   (a) Advance through 04 MCP MAN TEST to obtain the following message:
      1) 04 PUSH IAS/MACH SEL SW
   (b) Follow MCDP instructions to the end of the test.

S 862-066
(11) Close the following circuit breaker:
   (a) 11E16 MODE CONT PNL L

S 862-022
(12) Set the MCDP to off.

S 862-023
(13) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-11-02-022-024

4. Mode Select Switch and Lamp Removal (Fig. 202)

A. General

(1) This task has steps to remove a mode select switch and to remove a lamp for the mode select switch.

B. Equipment

(1) 50445 - Extractor Tool, Jay-E1 Product Inc., 23301 S. Wilmington, Carson, CA 90745

C. Access

(1) Location Zones

211/212 Flight Compartment

D. Procedure

S 862-025

(1) Open these circuit breakers and attach the DO-NOT-CLOSE tags:

(a) P11 Overhead Circuit Breaker Panel

1) 11E16, MODE CONT PNL LEFT
2) 11E34, MODE CONT PNL RIGHT
3) 11N30, INSTRUMENT & PANEL GLARESHIELD
4) 11P1, IND LTS L 1
5) 11P2, IND LTS L 2
6) 11P28, IND LTS R 1
7) 11P29, IND LTS R 2

S 032-026

(2) Use your fingernails (or the extractor tool) in the grooves at the sides of the lens module to pull the lens module out.

NOTE: The lens module is now held by a link pin and a bail plate.

(a) Push the link pin to the rear of the groove on the bail plate to remove the lens module if it is necessary.

S 022-027

(3) To remove a lamp, pull the lamp out of the rear of the lens module.

S 022-028

(4) To remove the switch, do these steps:

(a) Turn the jackscrew counterclockwise until it stops.

NOTE: The jackscrew is at the top in the switch module, and behind the mating plate for the lens module.

(b) Pull the switch module out from the case.

(c) NOTE: If removing lens module with P/N 837-0034-XXX, rotate the lens module 90 degrees. At 90 degrees the bail feature will unlock from the lens module.

TASK 22-11-02-422-029

5. Mode Select Switch and Lamp Installation (Fig. 201)

A. General
   (1) This task has steps to install a mode select switch and to install a mode select switch lamp.

B. References
   (1) AMM 22-10-00/501, Autopilot/Flight Director System
   (2) AMM 24-22-00/201, Manual Control

C. Access
   (1) Location Zones
       211/212 Flight Compartment

D. Procedure

S 862-039

(1) Make sure that these circuit breakers are open and the DO-NOT-CLOSE tags are attached:
   (a) P11 Overhead Circuit Breaker Panel
      1) 11E16, MODE CONT PNL LEFT
      2) 11E34, MODE CONT PNL RIGHT
      3) 11N30, INSTRUMENT & PANEL GLARESHIELD
      4) 11P1, IND LTS L 1
      5) 11P2, IND LTS L 2
      6) 11P28, IND LTS R 1
      7) 11P29, IND LTS R 2

S 422-030

(2) To install the switch, do these steps:
   (a) Turn the jackscrew counterclockwise until the jackscrew stops before you install the switch module.
   (b) Push the switch module into the case.

   NOTE: Put the side that has the word TOP in the up position.

CAUTION: DO NOT USE THE JACKSCREW TO PULL THE SWITCH MODULE INTO ITS POSITION. THIS CAN CAUSE DAMAGE TO THE SWITCH MODULE AND CASE.

   (c) Push the switch module with your finger while you lightly turn the jackscrew clockwise.
      1) Continue to push the switch module and turn the jackscrew clockwise until the jackscrew is tight.

S 862-031

CAUTION: DO NOT USE A DIFFERENT LAMP PART NUMBER. THE INCORRECT LAMP CAN CAUSE DAMAGE TO THE MCP.

(3) To install a new lamp, put the lamp into the rear of the lens module.

S 432-032

(4) To install the lens module, do these steps:
(a) Put the link pin into the rear of the groove on the bail plate and pull the link pin to the other end of the groove.
(b) NOTE: If installing lens module with P/N 837-0034-XXX, rotate the lens module 90 degrees after placing the bail feature into the grove of the lens module.

(c) Push the lens module into the switch module.

NOTE: Put the side that has the word TOP in the up position.

E. Test the mode select switch and lamp.

S 862-033

(1) Supply the electrical power (AMM 24-22-00/201).

S 862-034

(2) Set the two F/D switches on the MCP to OFF.

S 862-035

(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers:
(a) P11 Overhead Circuit Breaker Panel
1) 11E16, MODE CONT PNL LEFT
2) 11E34, MODE CONT PNL RIGHT
3) 11N30, INSTRUMENT & PANEL GLARESHIELD
4) 11P1, IND LTS L 1
5) 11P2, IND LTS L 2
6) 11P28, IND LTS R 1
7) 11P29, IND LTS R 2

S 712-036

(4) Do MCDP ground test 04-MCP (AMM 22-00-02/201).
(a) Make sure no failures occur during test.

S 862-037

(5) Remove the electrical power if it is not necessary (AMM 24-22-00/201).

AUTOPILOT DISENGAGE SWITCHES - REMOVAL/INSTALLATION

1. General
   A. One autopilot disengage switch is found on each control wheel. The removal and installation tasks for each switch are the same.

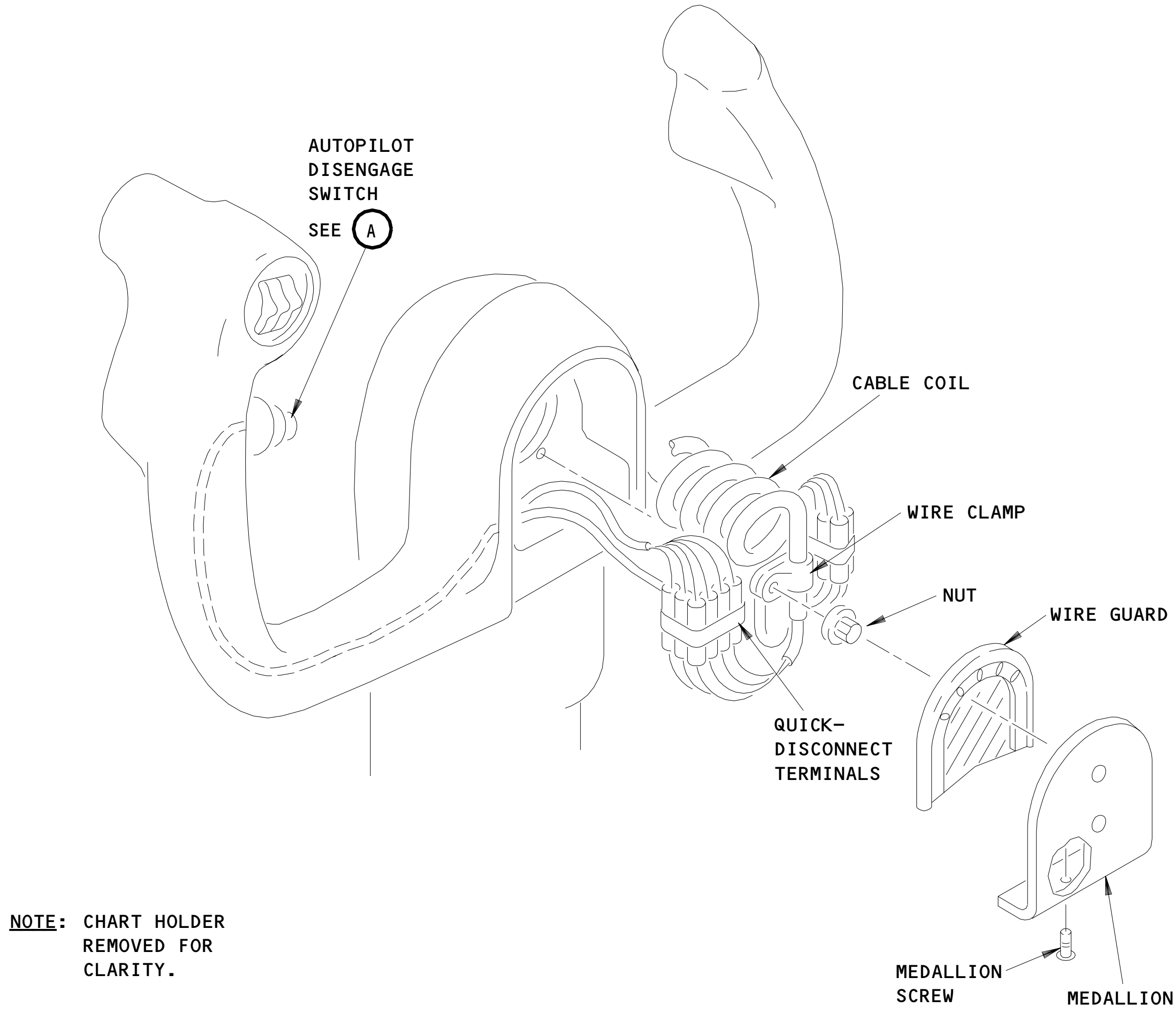
TASK 22-11-03-004-001

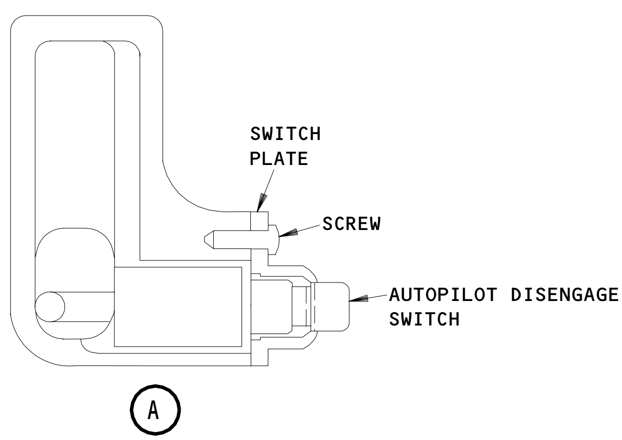
2. Remove the Autopilot Disengage Switch (Fig. 401)
   A. Equipment
      (1) Insertion and Extraction tool (Duetch No. M15570-20)
   B. Consumable Materials
      (1) Threading wire, a 30-inch length of standard wire (No. 18 to 22 gage), without insulation if possible (used to pull the wire bundle through the control wheel)
   C. Access
      (1) Location Zone
          211/212 Flight Compartment

   D. Prepare for Removal

      S 864-002
      (1) Set the two F/D switches on the MCP to OFF.

      S 864-003
      (2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11E16, MODE CONT PNL LEFT
         (b) 11E17, FLT CONT COMPUTER POWER LEFT
         (c) 11E18, FLT CONT COMPUTER SERVO LEFT
         (d) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
         (e) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
         (f) 11E34, MODE CONT PNL RIGHT
         (g) 11E35, FLT CONT CMPTR PWR RIGHT
         (h) 11E36, FLT CONT CMPTR SERVO RIGHT





Autopilot Disengage Switch Installation
Figure 401

E. Remove the Autopilot Disengage Switch

S 034-007
(1) Remove the screw that attaches the medallion to the control wheel.

S 034-004
(2) Remove the medallion with chart holder.

NOTE: It is not necessary to remove the chart holder from the medallion.

S 034-005
(3) Remove the nuts to disconnect the wire clamps.

S 034-008
(4) Pull the quick-disconnect terminals away from the control wheel.

S 034-006
(5) Use the extraction tool to disconnect these eight wires from the switch side of the quick-disconnect terminals:

WIRE LABEL AND COLOR CODE TABLE FOR SWITCH MFG P/N SW43896	
SWITCH SIDE WIRE COLOR	AIRFRAME SIDE WIRE COLOR CODE
BROWN	WHITE/BLUE
YELLOW	WHITE/YELLOW
GREEN	WHITE/GREEN
BLACK	WHITE/BLACK
ORANGE	WHITE/BROWN
WHITE	WHITE/PINK
BLUE	BLACK/YELLOW
GRAY	RED/GREEN

WIRE LABEL AND COLOR CODE TABLE FOR SWITCH MFG P/N 721101-A1	
SWITCH SIDE WIRE LABEL	AIRFRAME SIDE WIRE COLOR CODE
BRN	WHITE/BLUE
YEL	WHITE/YELLOW
GRN	WHITE/GREEN
BKB	WHITE/BLACK
BKY	WHITE/BROWN
BKA	WHITE/PINK
BLU	BLACK/YELLOW
BKG	RED/GREEN

S 434-010
(6) Attach one end of the threading wire around the wire bundle (eight wires) that is connected to the switch.

S 034-009
(7) Remove the screw that holds the switchplate to the control wheel.

S 024-011
(8) Remove the switchplate assembly and pull the wires through the control wheel.

NOTE: Permit approximately 3 or 4 inches of the threading wire to come out of the control wheel.

S 034-012
(9) Remove the threading wire from the switch wire bundle.

S 034-013
(10) Remove the switchplate from the switch.

TASK 22-11-03-404-014
3. Install the Autopilot Disengage Switch (Fig. 401)
A. Equipment
(1) Insertion and Extraction tool (Duetch No. M15570-20)
B. Consumable Materials
(1) Threading wire, a 30-inch length of standard wire (No. 18 to 22 gage), without insulation if possible (used to pull the wire bundle through the control wheel)
(2) C00259 Primer, BMS 10-11, type 1

C. References
(1) 20-30-03/201, Finishing Materials
(2) 22-00-02/201, Autoflight BITE
(3) 24-22-00/201, Electrical Power - Control

D. Access
(1) Location Zone
211/212 Flight Compartment

E. Install the Autopilot Disengage Switch

S 644-024
(1) Put primer on the threads of the switchplate and switch (Ref 20-30-03).

S 434-015
(2) Install the switchplate on the switch.

S 434-016
(3) Attach the end of the threading wire, that comes out of the switch hole on the control wheel, around the switch wire bundle.

S 424-025
(4) Pull the switch wire bundle down through the control wheel with the other end of the threading wire.

S 034-017
(5) Remove the threading wire from the switch wire bundle.

S 434-018
(6) Use the insertion tool to connect the switch wires to the quick-disconnect terminals. Refer to the Wire Label and Color Code Table.

S 434-019

CAUTION: MAKE SURE THE WIRE CLAMPS ARE INSTALLED IN THE POSITION SHOWN IN FIG. 401. PUT THE CABLE COIL IN THE CENTER OF THE CONTROL WHEEL. FRICTION WILL RESULT IF THE WIRE CLAMPS ARE NOT INSTALLED CORRECTLY AND THE CABLE COIL IS NOT PUT IN THE CENTER. DAMAGE TO EQUIPMENT CAN OCCUR.

(7) Install the wire clamps into the control wheel with the nuts.

NOTE: Install the wire clamp to put the cable coil in the center of the control wheel.

(a) Tighten the nuts to 20 pound-inches.
(b) Put the quick-disconnect terminals into the control wheel.

S 434-030

CAUTION: ENSURE THAT WIRE GUARD ASSEMBLY SLIDES SMOOTHLY IN GROOVE OF WHEEL ASSEMBLY DURING INSTALLATION. KEEP ALL WIRES AND CONNECTORS CLEAR OF PLANE OF GROOVE TO PREVENT POSSIBLE WIRE INSULATION DAMAGE DURING ASSEMBLY.

(8) Install wire guard assembly. Insert lower end of wire guard assembly inside base of wheel. Firmly press upper edge of seal until guard is below medallion groove.

S 434-020

(9) Install the medallion and chart holder with the screw.

S 434-021

(10) Install the switchplate assembly with the screw.

F. Test Autopilot Disengage Switch

S 864-022

(1) Supply electrical power (Ref 24-22-00).

S 864-023

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11E16, MODE CONT PNL LEFT
   (b) 11E17, FLT CONT COMPUTER POWER LEFT
   (c) 11E18, FLT CONT COMPUTER SERVO LEFT
   (d) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
   (e) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
   (f) 11E34, MODE CONT PNL LEFT
   (g) 11E35, FLT CONT CMPTR PWR RIGHT
   (h) 11E36, FLT CONT CMPTR SERVO RIGHT

S 724-026

(3) Do MCDP Ground Test 11-SW A/P DISC (Ref 22-00-02).
   (a) Make sure no failure messages show.

S 864-027

(4) Set the MCDP to off.

S 864-028

(5) Remove electrical power if not necessary (Ref 24-22-00).

AFCS GO-AROUND SWITCHES - REMOVAL/INSTALLATION

1. General
   A. Six AFCS go-around switches are found in the engine thrust levers. Three switches are installed in the side of each thrust lever. The Removal/Installation procedure is the same for each set of switches.

TASK 22-11-04-004-001

2. Remove the AFCS Go-Around Switches (Fig. 401).
   A. Access
      (1) Location Zones
         211/212 Flight Compartment

   B. Remove the AFCS Go-Around Switches

      S 864-002
      (1) Set the two F/D switches on the MCP to OFF.

      S 864-003
      (2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11E16, MODE CONT PNL LEFT
         (b) 11E17, FLT CONT COMPUTER POWER LEFT
         (c) 11E18, FLT CONT COMPUTER SERVO LEFT
         (d) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
         (e) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
         (f) 11E34, MODE CONT PNL RIGHT
         (g) 11E35, FLT CONT CMPTR PWR RIGHT
         (h) 11E36, FLT CONT CMPTR SERVO RIGHT
         (i) 11F14 or 11F16, TMC AC
         (j) 11F15 or 11F17, TMC DC
         (k) 11F16 or 11F18, TMC SERVO

      S 034-004
      (3) Remove the screws and the cover from the thrust lever.

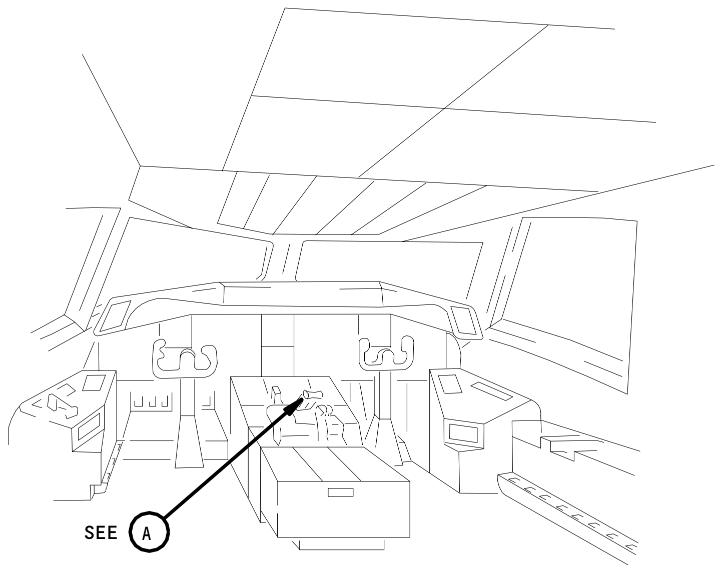
      S 024-005
      (4) Remove the screws from the applicable switch and actuator assembly.

      S 034-006
      (5) Remove the heat shrink tubing from the wiring terminals.

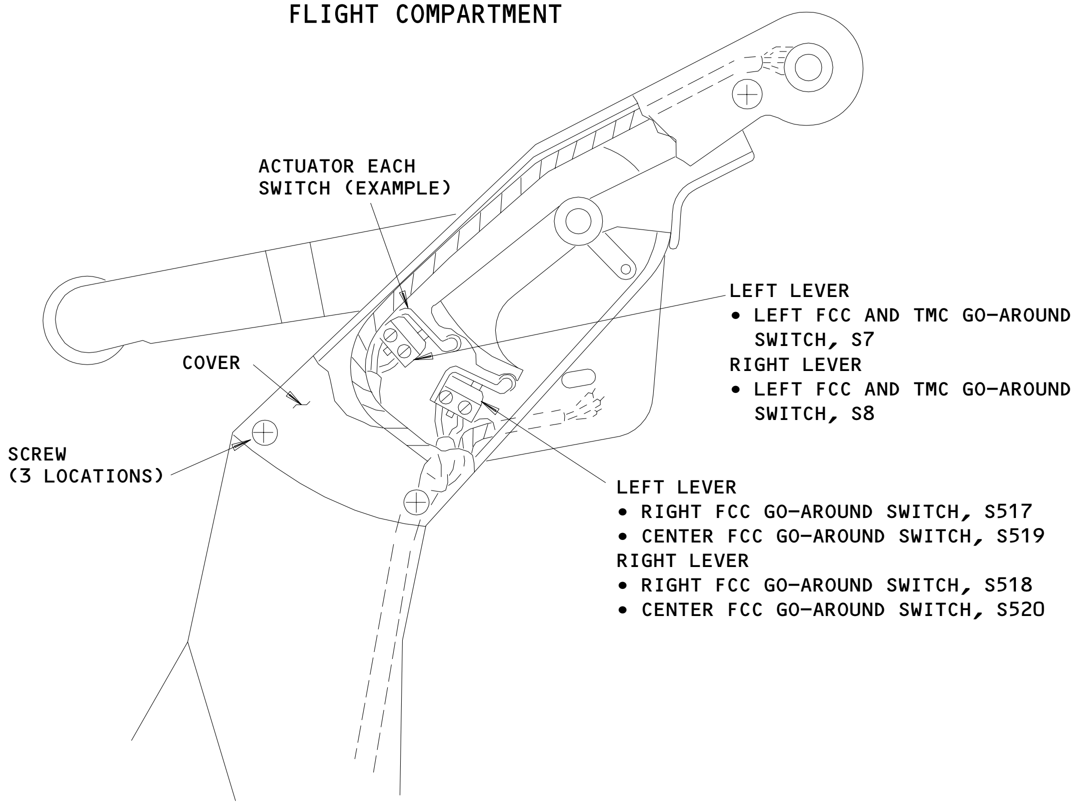
      S 024-007
      (6) Unsolder the wires from the switch.

TASK 22-11-04-404-008

3. Install the AFCS Go-Around Switches (Fig. 401)
   A. Equipment
      (1) Heat gun - commercially available



FLIGHT COMPARTMENT



AFCS GO-AROUND SWITCH

A

NOTE: SPRING REMOVED FOR CLARITY.

AFCS Go-Around Switches Installation
Figure 401

B. Consumable Materials
   (1) G02104 Heat Shrink Tubing, R876, color yellow, Raychem Corp, 300 Constitution Dr., Menlo Park, CA 94025
C. References
   (1) 22-00-02/201, Autoflight BITE
   (2) 24-22-00/201, Electrical Power - Control
D. Access
   (1) Location Zones
       211/212 Flight Compartment

E. Install the AFCS Go-Around Switches

S 434-009
   (1) Apply the heat shrink tubing. Do not shrink the tubing until after the wires are soldered to the switch.

S 424-010
   (2) Solder the wires to the new switch.

S 434-011
   (3) Shrink the heat shrink tubing over the wire terminals.

S 424-012
   (4) Install the switch and actuator assembly with the screws.

S 434-013
   (5) Install the cover on the thrust lever with the screws.

F. Test the Go-Around switches

S 864-014
   (1) Supply electrical power (Ref 24-22-00).

S 864-015
   (2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
       (a) 11E16, MODE CONT PNL LEFT

(b) 11E17, FLT CONT COMPUTER POWER LEFT
(c) 11E18, FLT CONT COMPUTER SERVO LEFT
(d) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(e) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(f) 11E34, MODE CONT PNL RIGHT
(g) 11E35, FLT CONT CMPTR PWR RIGHT
(h) 11E36, FLT CONTR CMPTR SERVO RIGHT
(i) 11F14 or 11F16, TMC AC
(j) 11F15 or 11F17, TMC DC
(k) 11F16 or 11F18, TMC SERVO

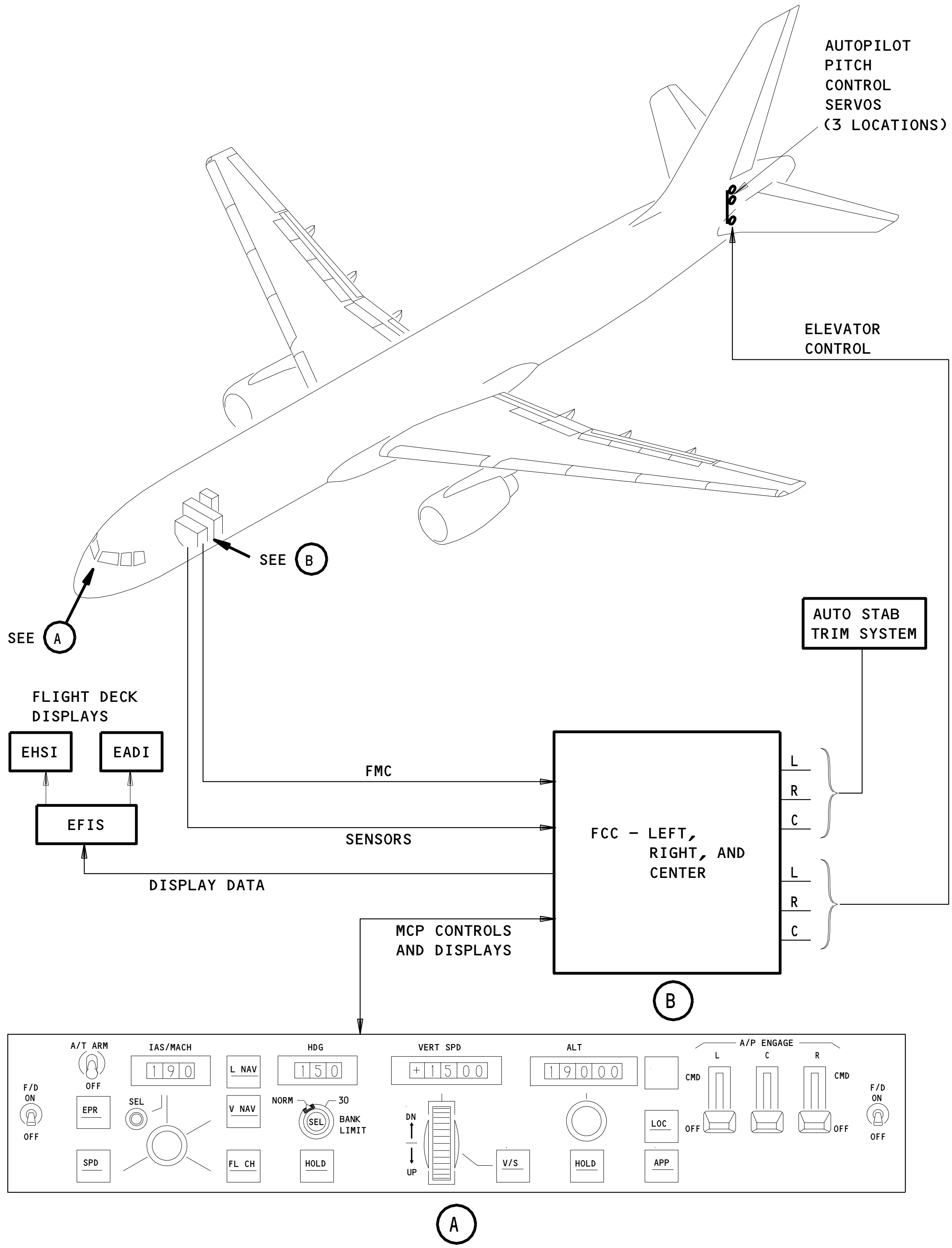
S 724-016
(3) Do MCDP Ground Test 13-SW G/A (Ref 22-00-02).
(a) Make sure no failure messages show during the test.

S 864-017
(4) Set the MCDP to off.

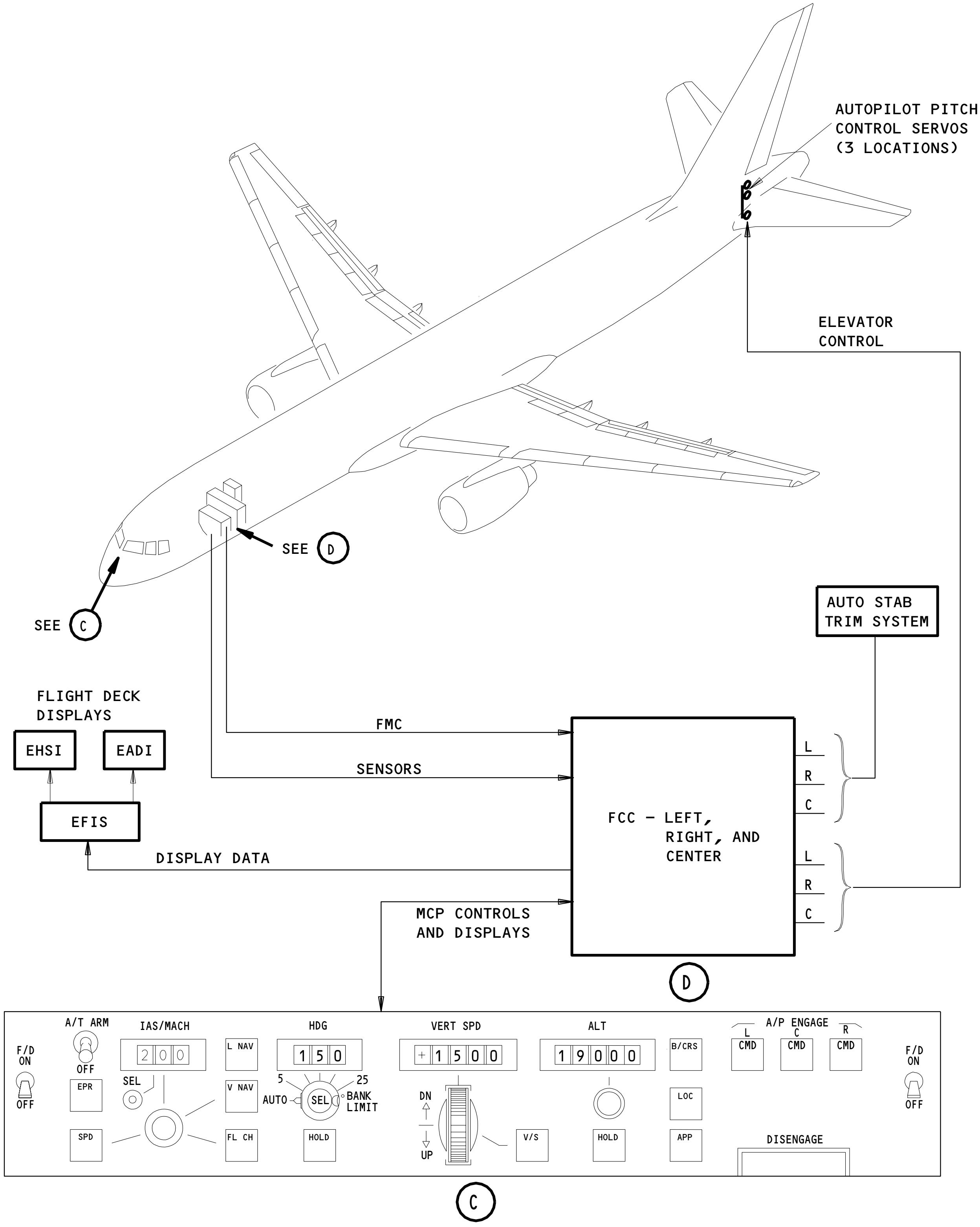
S 864-018
(5) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR PITCH CHANNEL - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The pitch channel consists of pitch mode switches on the AFCS mode control panel, pitch modes of the three Flight Control Computers (FCCs) three Autopilot Pitch Control Servos (APCS) and three elevator neutral shift transducers. Each servo and transducer is dedicated to a single flight control computer. Primary AFDS control modes are selected on the AFCS Mode Control Panel (MCP). Flight director and mode displays are on the Electronic Attitude Director Indicators (EADIs).
   B. GUI 010-114, 116-999;
   each FCC generates commands for the Flight Director (F/D) and Autopilot (A/P). The FCC selected provides commands for either the A/P or F/D, or both at the same time. The A/P and F/D modes are selected on the MCP. F/D displays are turned on by the F/D switches on the MCP. The FCC providing F/D displays is selected by the pilots FLT DIR source select switch.
   C. GUI 001-009, 115;
   each FCC generates commands for the Flight Director (F/D) and Autopilot (A/P). The FCC selected provides commands for either the A/P or F/D, but not both at the same time. The A/P and F/D modes are selected on the MCP. F/D displays are turned on by the F/D switches on the MCP. The FCC providing F/D displays is selected by the pilots FLT DIR source select switch.
      (1) Pitch axis control functions of the FCC provide pitch axis stability, manual pitch attitude control and programmed pitch attitude control of the airplane. The functions control the autopilot or provide flight director steering commands. The autopilot pitch control servos and elevator control system provide for pitch axis maneuvering. The automatic stabilizer trim system provides for the transfer of sustained elevator loads to the horizontal stabilizer. A Stabilizer trim and elevator Asymmetry limit Module (SAM) provides the interface between the FCC and the stabilizer trim system.
      (2) Pitch Axis Stability
         (a) The attitude hold function of the FCC maintains airplane pitch attitude during turbulence with the autopilot engaged or provides F/D commands.
      (3) Manual Pitch Attitude Control
         (a) The basic mode of the pitch channel is vertical speed (V/S). Altitude hold is V/S equal to zero. Both are active whenever a flight director is on or an autopilot is engaged in CMD. Altitude hold is manually selected by pressing the HOLD button on the MCP. Vertical speed is manually selected by pressing the V/S button on the MCP or is automatically selected when CMD is engaged without selecting a mode. Altitude hold is automatic whenever the vertical speed is less than 300 feet per minute.



Pitch Axis Control
Figure 1 (Sheet 1)



Pitch Axis Control
Figure 1 (Sheet 2)

(b) Altitude select control is available whenever a flight director is on or an autopilot is engaged in CMD. A vertical speed may be selected and V/S pushbutton pressed. The airplane then climbs or descends to the altitude selected on the MCP. If in a climb or descent at the time the F/D is on or the autopilot is engaged,the airplane continues at the existing vertical speed until the selected altitude is reached. The mode then automatically transfers to altitude hold. The V/S or HOLD pushbuttons do not have to be pressed if the vertical speed and altitude differences are greater than altitude hold capture limits.

(4) Programmed Pitch Attitude Control

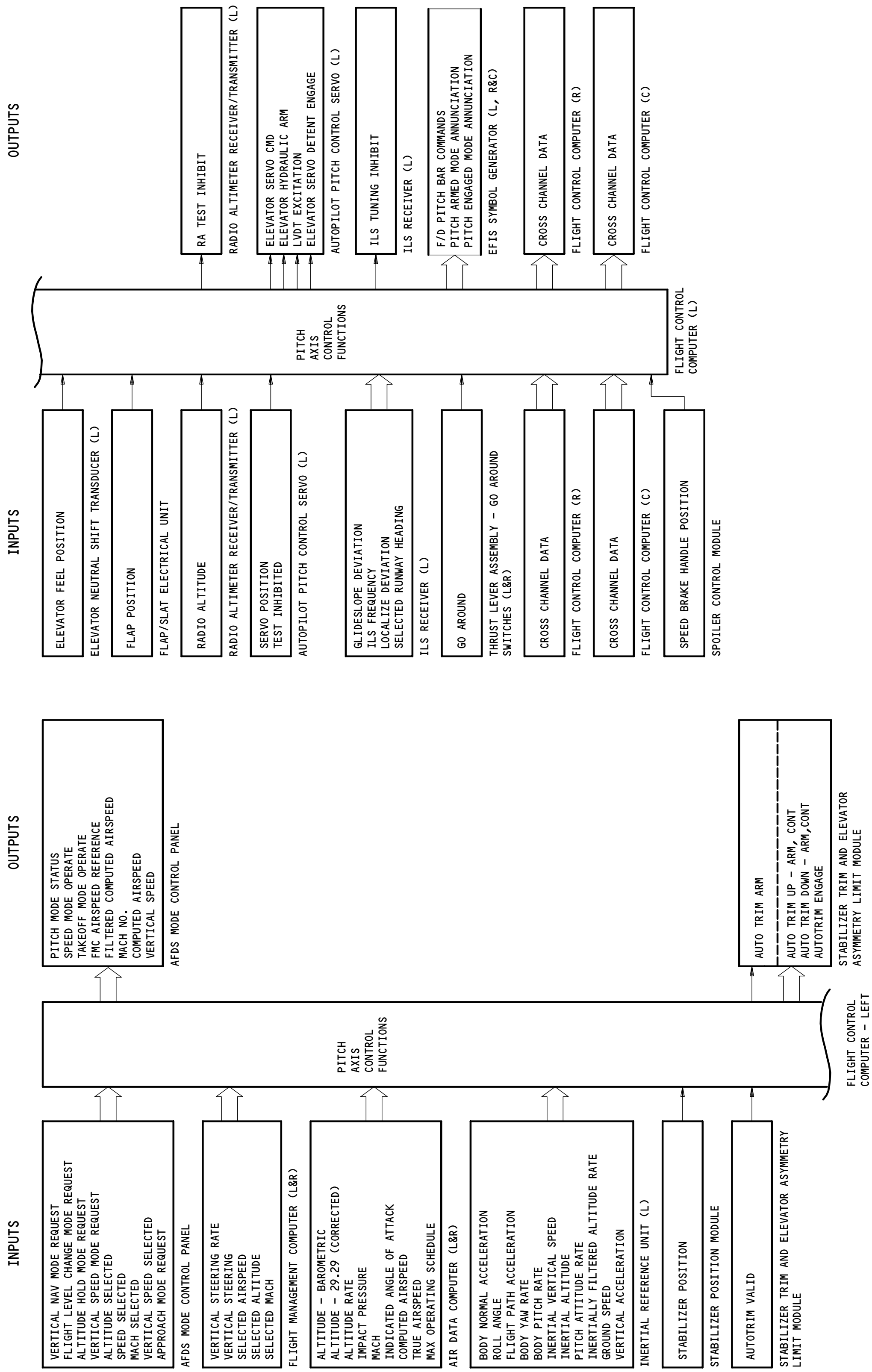
(a) Any of the programmed pitch attitude control functions can be used with a flight director engaged or an autopilot engaged in CMD. During sustained elevator deflections, the stabilizer may be retrimmed. The FCC generates auto stabilizer trim commands for the SAM. The SAM drives the stabilizer trim system until FCC pitch attitude circuits return elevators to neutral and remove auto stab trim commands (Ref 22-22-00).

(b) The flight management computer system provides pitch steering commands when VNAV is selected. The airplane is then guided to the vertical flight profile programmed in the FMCS.

(c) Programmed pitch attitude control functions provide autopilot or flight director guidance to capture and track the glideslope centerline during an ILS approach. Additional programmed functions provide automatic compensation for windshear and perform the flare maneuver prior to touchdown. Vertical position correction provides a consistant touchdown point on the runway. A controlled pitchdown attitude maneuver lowers and holds the nose wheel down after the main gear touchdown. If a go-around is initiated, flight director or autopilot guidance establishes a pitch-up attitude and positive rate of climb. Go-around mode disengages at 500 feet, or when another mode is selected. A selected vertical mode is then used for airplane guidance.

D. Flight Control Computer Pitch Axis Control Inputs and Outputs (Fig. 2)

(1) Inputs include mode requests from the MCP, servo and surface position from the APCS, flight reference data from IRUs and ADCs, and navigation data from ILS and RA systems. Vertical steering, rate target altitude, and airspeed/mach commands are supplied by the FMC. Auto stabilizer trim data is received from the stabilizer position module, stabilizer trim and elevator asymmetry limit module, and elevator neutral shift transducer.



Flight Control Computer Pitch Axis Control Inputs and Outputs
Figure 2

(2) The FCC outputs include mode status discretes, air speed/mach number, and vertical speed data to the MCP. Autopilot pitch commands are sent to the autopilot pitch control servos and stabilizer trim and elevator asymmetry limit module. Flight director pitch bar command, pitch armed and engaged mode annunciation data is sent to the EFIS symbol generators (Ref 34-22-00).

2. <u>Component Details</u>

A. Autopilot Pitch Control Servo Locations and Linkages (Fig. 3)

(1) The three Autopilot Pitch Control Servos (APCS) are at the aft end of the stabilizer center section on fuselage Section 48. The right APCS is the upper unit, center APCS the center unit and left APCS the bottom unit.

(2) Each APCS is mounted with four mounting bolts on the torque box aft of the horizontal stabilizer. Output rods connect the APCS crank to the torque tube. Two hydraulic lines and an electrical connector complete the installation.

(3) Each APCS output rod is connected to the aft torque tube. The captains elevator control column cables are connected to the quadrant on the aft torque tube. The first officer's elevator control cables are connected to the forward torque tube quadrant. The two torque tubes are linked together so either pilot can control the elevators.

B. Autopilot Pitch Control Servo (Fig. 4)

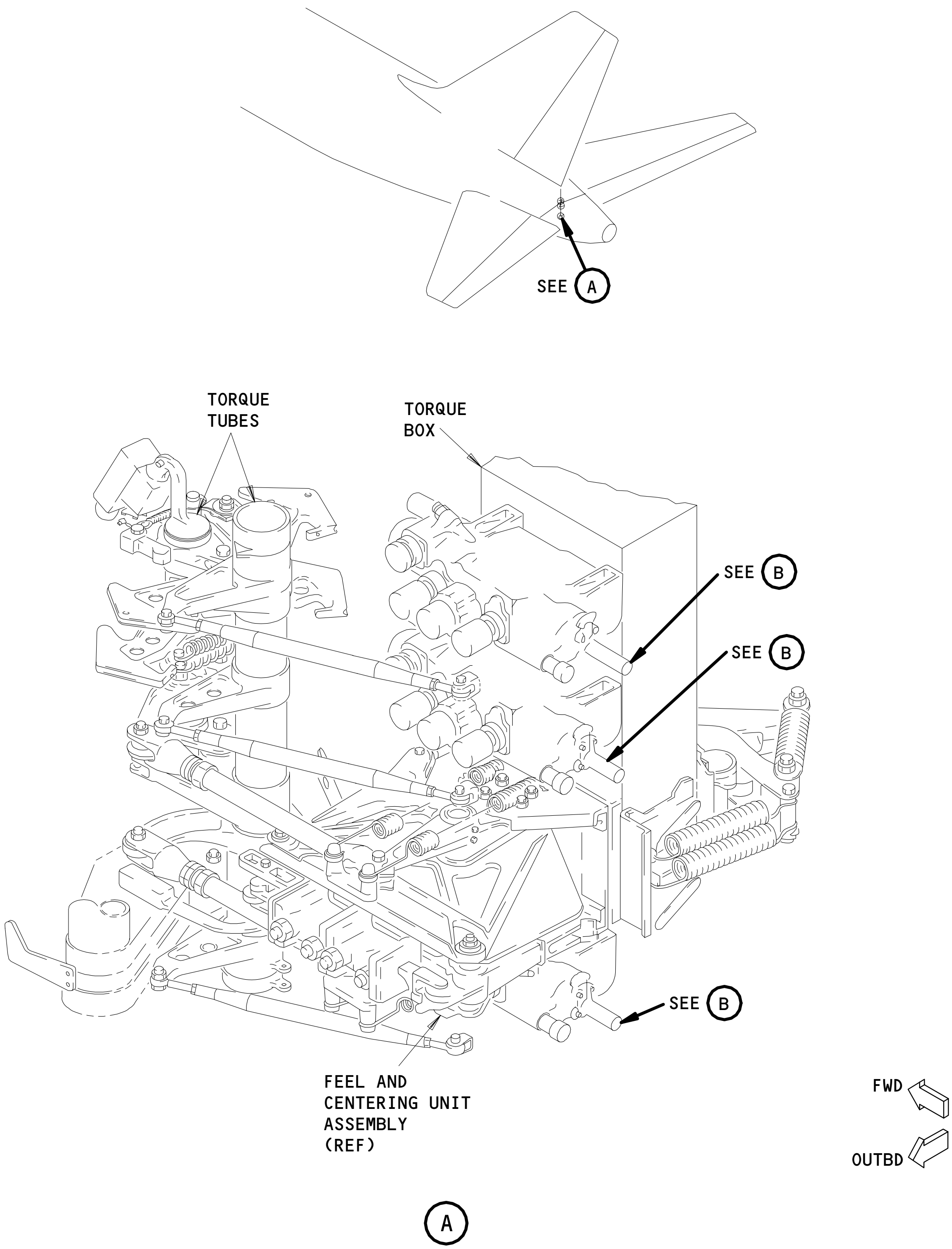
(1) The autopilot pitch control servo has 2 solenoid valves and an electrohydraulic servovalve. Two linear variable differential transducers (LVDTs) provide position feedback signals representing the position of an actuator piston and an output crank (lever) on the APCS. The APCS has a combined pressure regulator-relief valve, and an inline filter at the hydraulic pressure connection.

(2) Electrohydraulic Control Valves

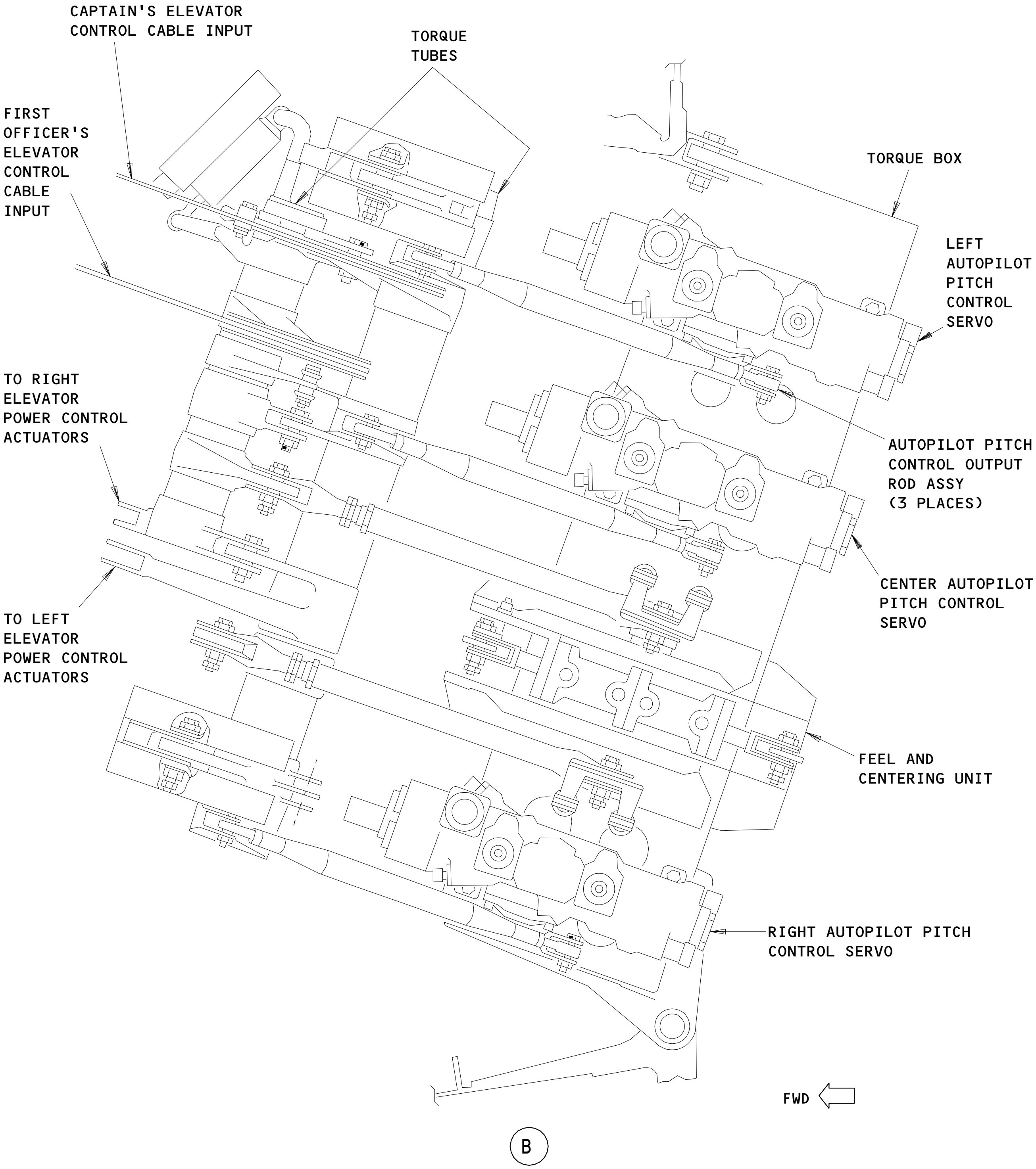
(a) Solenoid Valves

1) Two solenoid valves (SV1 and SV2) are installed on the APCS. Each solenoid valve is an electrically operated open-close valve which completes hydraulic pressure through the APCS when autopilot arm and engage logic circuits are completed. Each valve is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mated when the solenoid is bolted in position.

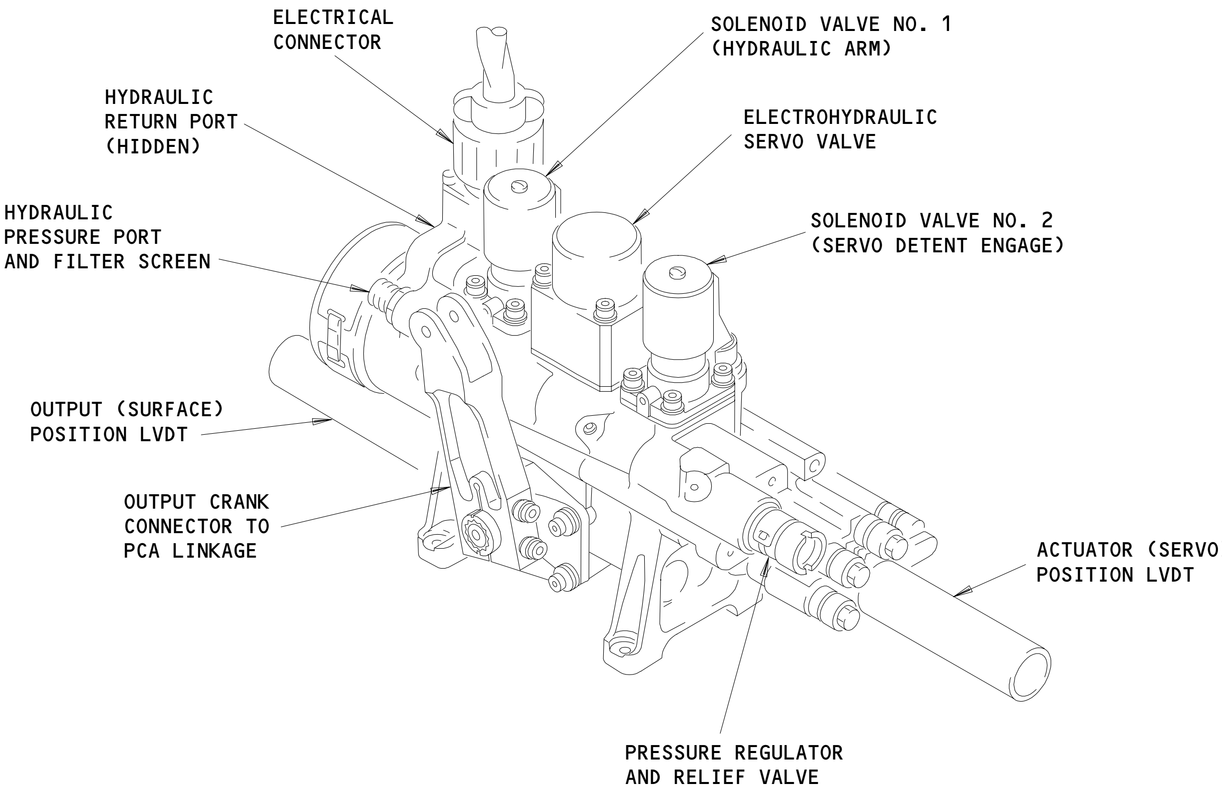
2) Solenoid valve number 1 (SV1) opens when ELEV HYD ARM DC is applied (autopilot is armed). It ports hydraulic pressure to solenoid valve number 2 (SV2) and the electrohydraulic servovalve (EHSV). Solenoid valve number 2 opens when ELEV DETENT ENG DC is applied (autopilot is engaged in CMD) and ports hydraulic pressure to the detent pistons. The detent pistons clamp the output linkage crank and transmit actuator piston position to the output crank.

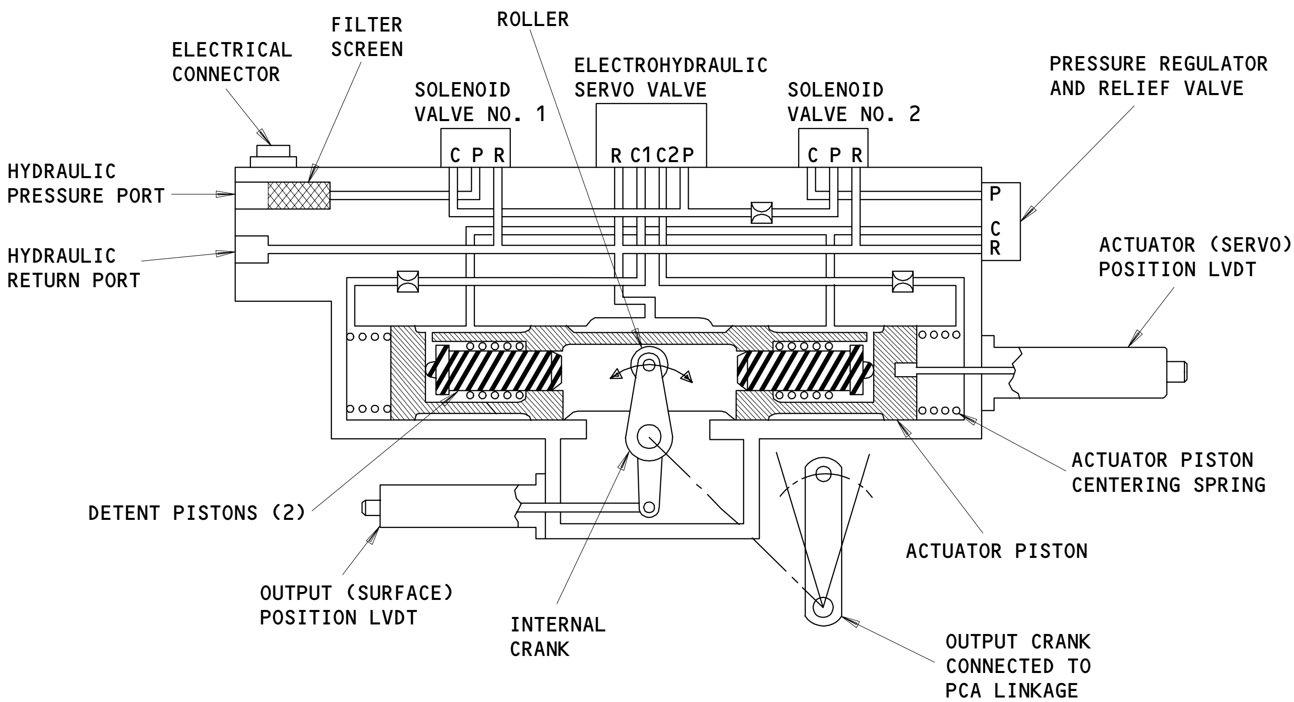


Autopilot Pitch Control Servo Locations and Linkages
Figure 3 (Sheet 1)



Autopilot Pitch Control Servo Locations and Linkages
Figure 3 (Sheet 2)





Autopilot Pitch Control Servo
Figure 4

(b) Electrohydraulic Servovalve (EHSV)

1) One electrohydraulic servovalve is installed on each APCS. The EHSV contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydraulic pressure through the valve can be applied to either of two output ports. When no error signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor and out the projector jet. From the jet, equal pressure is applied to opposite ends of the piston holding it at center and closing both output ports.

2) When an error signal is applied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston than the other causing it to move and opens the corresponding output port to complete hydraulic fluid flow through the APCS. When the error signal is nulled, the motor armature and jet return to center. This equalizes the pressures on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.

3) The EHSV is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mated when the EHSV is bolted in position.

4) The EHSV is controlled by the elevator error signal (ELEV SERVO CMD) from the FCC. In response to this command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the actuator piston LVDT nulls the error signal at the FCC and piston movement stops.

(3) Linear Variable Differential Transducers (LVDT)

(a) The APCS has two LVDTs. The actuator piston (servo position) LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The output position (surface position) is operated by the internal crank which is connected to the output crank and elevator control linkage. Both LVDTs are variable reluctance transformers with an output that varies directly with linear motion. The LVDT uses 26 vac excitation from the associated FCC. The LVDTs are not considered line replaceable units since they require nulling adjustments to be completed on the bench to match actuator piston with crank position.

(4) Pressure Regulator and Relief Valve

(a) The pressure regulator-relief valve regulates and limits hydraulic pressure applied to the actuator piston and detent piston. The pressure relief function allows manual inputs from the control linkage to override autopilot control.

(5) Actuator Piston Assembly

(a) The actuator piston translates autopilot input command through the EHSV into mechanical positioning of the elevators. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV when the autopilot is armed or engaged. The springs center the actuator piston when pressure is released.

(b) Two detent pistons inside the actuator piston are normally retracted by disengage springs. When the autopilot is engaged, SV2 opens. Hydraulic pressure through SV2 overrides spring tension and locks the detent pistons against the roller of the internal crank. Detent pistons may be forced back if sufficient force to overcome hydraulic pressure is applied manually through the control linkage (camout).

(c) The actuator piston LVDT provides an electrical signal proportional to actuator piston position. This signal nulls the autopilot elevator position error signal from the FCC to stop movement of the actuator piston.

(6) Cranks

(a) The internal crank roller is clamped by the detent pistons when the autopilot is engaged. The crank moves with the actuator piston. Motion of the internal crank moves the output position LVDT. One end of the output crank is directly connected to the internal crank at a common pivot point. The other end of the output crank is connected to the elevator linkage.

(7) Mechanical Control Sequence

(a) Initially the actuator piston is fixed by the centering springs. With the autopilot not engaged (armed), SV1 is open, SV2 is closed, detent pistons are disengaged and the internal crank is free to move within the piston cavity. The output position LVDT provides internal crank position to the FCC for autopilot synchronization to elevator position. The FCC commands through the EHSV cause the actuator piston to follow the internal crank so that the internal crank remains centered within the piston cavity. The actuator piston position LVDT nulls the command signal to stop the actuator piston.

(8) Autopilot Control Sequence

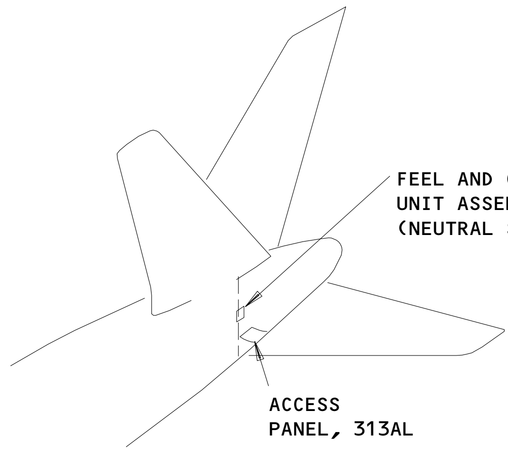
(a) With the autopilot engaged, SV1 and SV2 are open, the detent pistons are pressurized and the internal crank is clamped on the center of the actuator piston. When the EHSV receives a command from the FCC, hydraulic pressure is ported to one end of the actuator piston. The detent pistons carry the internal crank with the actuator piston to its commanded position. The output crank moves the linkage to the elevator power control actuators while the output position LVDT sends position information back to the FCC to null the command signal and stop the elevator.

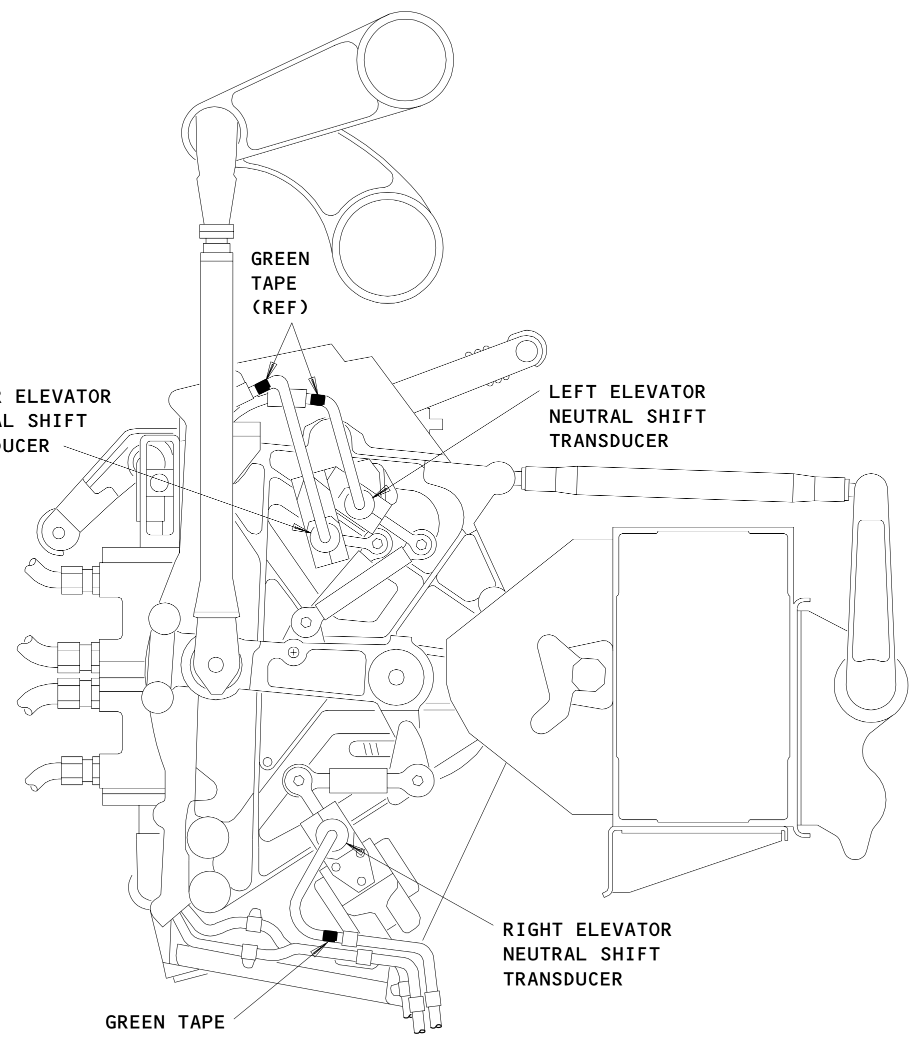
(9) Camout
(a) Camout occurs when the output crank position (surface position) does not correspond to the actuator piston position (servo position) as sensed by differing LVDT outputs. Causes of camout are mechanical jamming of output linkage, pilot override of an engage autopilot channel, or mechanical override of one autopilot channel by the remaining two during multi-channel operation.

C. Feel and Centering Unit Assembly (Fig. 5)
(1) The neutral shift transducers provide an analog input to each FCC. The input is proportional to the elevator position which varies with stabilizer position. The three transducers are installed on the feel centering assembly aft of the horizontal stabilizer in fuselage Section 48.
(2) The stud on each transducer housing is mounted to a bearing in the centering unit housing assembly with a nut and washer. The adjustable rod end of each transducer is attached to the centering unit crank assembly. The left and right transducer rod ends are each attached to the crank assembly with a bolt and nut. The center transducer attaches to the bottom side of the crank assembly with a countersunk screw. The rod ends are adjusted during transducer installation to set the null voltage of each unit. Each transducers electrical cable is clamped to the centering unit housing assembly.
(3) The primary winding of each neutral shift transducer uses 400 Hz 26 vac from its associated FCC. The secondary winding output to the FCC is in phase with the primary winding when the transducer is in the retracted position. The secondary output phase is reverse of the primary when the transducer is in the extended position. The output is nulled with the centering unit assembly crank is in a neutral position.

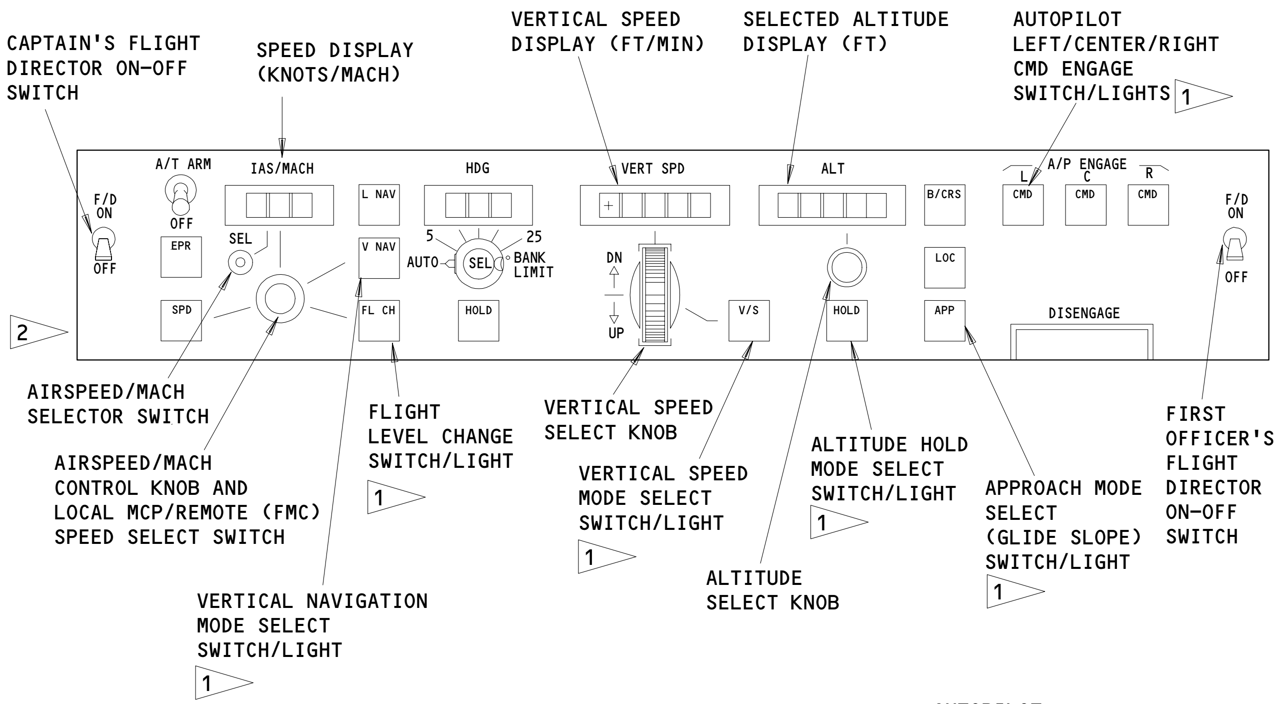
3. Operation

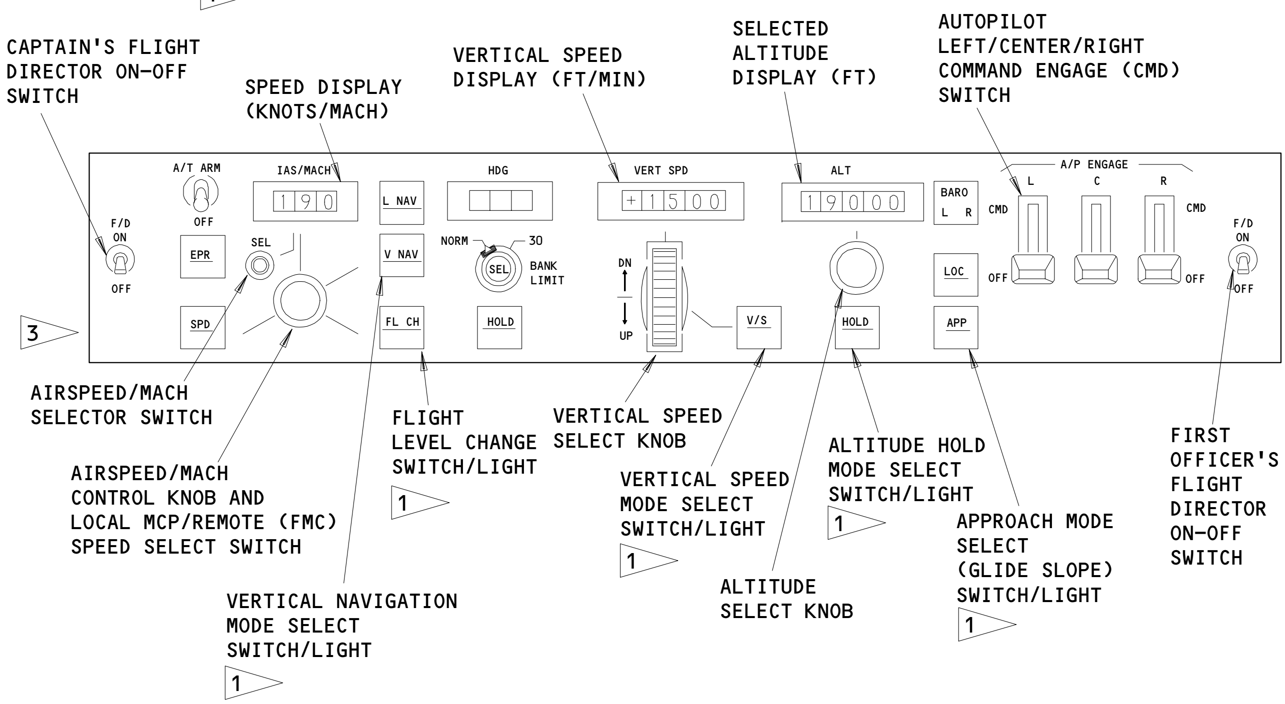
A. Functional Description
(1) Mode Control Panel - Pitch Controls (Fig. 6)
(a) Flight Director ON/OFF switches
1) Two F/D toggle switches control the left and right flight director dislays.
(b) Command Engage switches
1) GUI 115;
three CMD ENGAGE paddle switches (L, C, R) provide inputs for command engage logic to dedicated FCCs.





Feel and Centering Unit Assembly
Figure 5





1. DOT/BAR MATRIX ON LOWER HALF OF SWITCH COVER IS ILLUMINATED WHEN FCC CONFIRMS REQUEST IS VALID.
2. GUI 001-114,116-999
3. GUI 115

Mode Control Panel - Pitch Controls
Figure 6

2) GUI 001-114, 116-999;
three CMD ENGAGE switch/lights (L, C, R) provide inputs for command engage logic to dedicated FCCs.

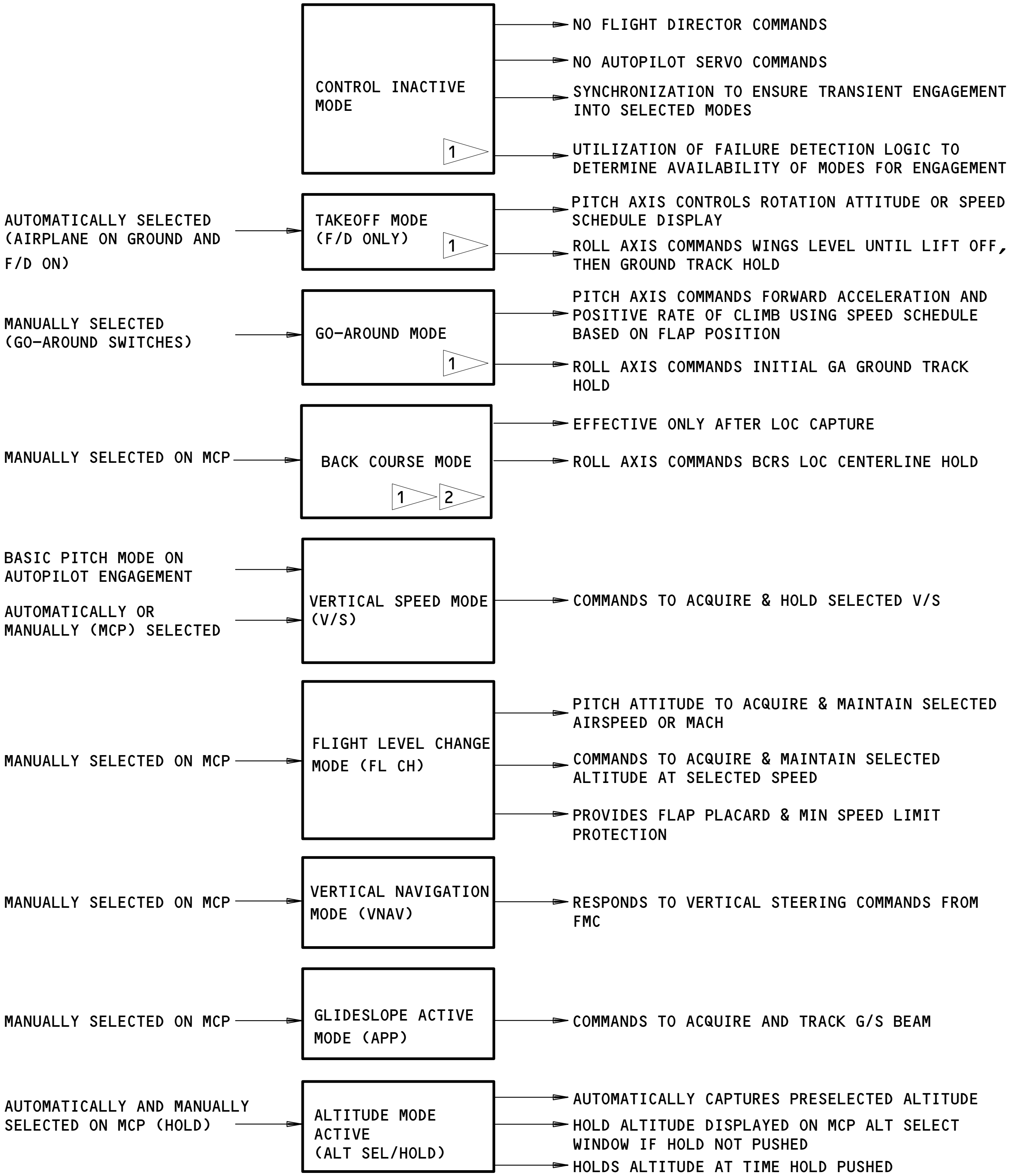
(c) The following MCP controls are associated with the pitch function:
1) VNAV switch/light
2) FLCH switch/light (Flight Level Change)
3) Vert SPD select thumbwheel
4) V/S SPD switch/light
5) ALT Select Knob
6) ALT HOLD switch/light
7) APP switch/light
8) IAS/MACH Control Knob and local MCP/remote FMC speed select switches
9) IAS/MACH select switch

(2) Pitch Mode Functions (Fig. 7)

(a) The control inactive mode is the dormant condition of the AFDS when electrical power is on and no A/P or F/D commands are generated. Control loops are synchronized to control surface position for transient-free engagement. Failure detection logic determines the availability of each mode.

(b) Takeoff Mode (F/D Mode Only)
1) The takeoff mode consists of two submodes: takeoff on-ground, and takeoff in-air. The takeoff on-ground submode provides the F/D with 8 degrees pitch up and wings level roll commands during the ground roll and until 2 seconds after radio altitude of 5 feet. The takeoff in-air submode begins 2 seconds after a radio altitude of 5 feet. The F/D pitch up command is washed out over a 3-second time period and replaced with pitch attitude commands referenced to scheduled airspeed. Airspeed commands are scheduled according to flap position and engine failure. This ensures safe F/D pitch attitude commands until the selected airspeed on the MCP is obtained. The F/D roll command is changed to provide ground track hold.

(c) Go-Around Mode
1) Go-around mode is selected by pressing the go-around switches on the thrust levers. The go-around mode provides A/P and F/D pitch commands for a safe positive rate of climb using speed scheduling based on flap position. The A/P and F/D roll commands provide ground track hold based on runway heading with crosswind compensation.



1 SIMULTANEOUS PITCH AND ROLL MODES
2 GUI 001-114,116-999

Pitch Mode Functions
Figure 7

(d) GUI 001-114, 116-999;
Backcourse (B/CRS) Mode
1) The backcourse mode is selected with the B/CRS switch/light. If not within localizer capture range, the B/CRS switch reverses only the localizer displays on the ADIs. The standby attitude/ILS indicator also has a B/CRS selector which is used during a back course approach, however, it has no interconnect with the AFDS.
2) After localizer capture, the A/P or F/D provides commands to track the airplane to the back course centerline. The pitch channel continues to provide guidance from the selected pitch mode.

(e) Vertical Speed
1) Vertical speed is the basic pitch axis mode. When the V/S switch/light is pressed with the A/P in CMD or the F/D on, the vertical speed command acquires the selected vertical speed. The existing V/S mode is automatically acquired when the F/D is initially turned on or the F/D is overridden by engaging the A/P in CMD.

(f) Flight Level Change
1) The flight level change mode is manually selected by pressing the FLCH switch/light on the MCP. The mode uses software programmed elevator control laws to generate pitch attitude A/P or F/D commands. The pitch attitude commands control the airplane to obtain and maintain an MCP selected altitude using the MCP selected airspeed or Mach. The FLCH mode provides flap placard and minimum speed limit protection. When limits are encountered, corresponding annunciation is provided on the EADI.

(g) Vertical Navigation Mode
1) The mode is manually selected by pressing the VNAV switch/light on the MCP. The A/P or F/D responds to pitch steering commands from the FMC. Pitch steering commands are generated in the FMC as necessary to acquire and track a vertical path along a preplanned flight route.

(h) Glideslope Active Mode (Approach)
1) The mode is manually selected by pressing the APP switch/light on the MCP. Pitch commands are generated for A/P or F/D control to acquire and maintain capture of the glideslope. The A/P channels can be used for single channel or multi-channel ILS approaches.

(i) Autoland Mode
1) The autoland mode is only active during multi-channel operation. Pitch commands provide flare and nose gear letdown during rollout.

(3) Output Signal Synchronization (Fig. 8)

(a) Elevator Servo Command

1) Servo command signals are synchronized to the position of the servo to prevent transients upon engagement. Elevator surface position is summed with the elevator command output which is initially at zero. Assuming the elevator is displaced, the increase in elevator position LVDT output is integrated and limited to +10 and -28 degrees of elevator displacement. The resulting output is summed at the integrator until LVDT input is nulled.

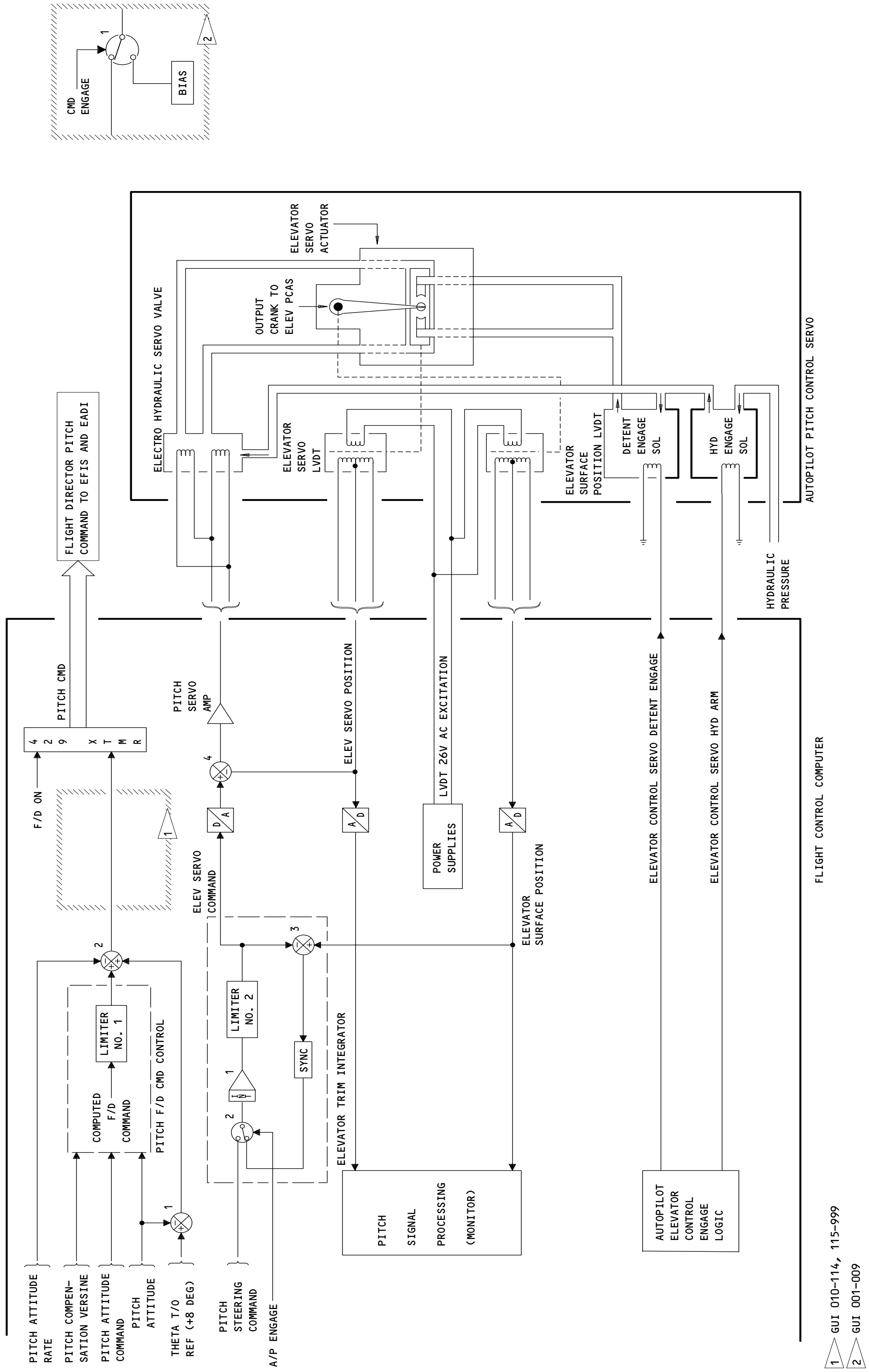
2) The resulting elevator command signal is applied to the pitch servo amplifier to drive the electrohydraulic servovalve. With elevator servo hydraulic arm logic applied, the electrohydraulic servovalve moves to center the actuator piston (servo actuator) about the output crank linkage. The servo position LVDT provides feedback to null the elevator command signal and stop the servo actuator. At A/P engage, elevator servo detent engage logic is applied to the servo. Servo detent pistons then clamp the actuator piston to the output crank linkage without generating a transient. The A/P engage logic transfers pitch steering commands to the elevator trim integrator for A/P engaged operation.

(b) Flight Director Pitch Command

1) Flight director pitch commands are computed continuously whether or not the A/P is engaged or the F/D is on.

2) The F/D pitch commands are computed from selected pitch attitude commands, present pitch attitude, pitch attitude rate, and versine. Computed F/D pitch commands are summed with pitch rate. If in takeoff (T/O) mode, a reference +8 degrees (theta) pitch signal is summed with pitch attitude and then summed with the F/D pitch commands. Computed F/D pitch commands are applied to the signal generators and EADIs of the EFIS.

3) GUI 001-009, 115;
a F/D ON discrete causes the F/D command bars to be in view and the F/D pitch command displayed. With the F/D switches off, the F/D ON discrete is not provided and the EFIS inhibits the F/D displays. When the airplane radio altitude is greater than 5 feet and the A/P is CMD engaged, a fixed bias is sent to the EFIS which biases the F/D bars out of view. The airplane is then under A/P CMD ENGAGE control to selected modes.



Output Signal Synchronization
Figure 8

C31305

4) GUI 010-114, 116-999;
a F/D ON discrete causes the F/D command bars to be in view and the F/D pitch command displayed. With the F/D switches off, the F/D ON discrete is not provided and the EFIS inhibits the F/D displays. When the airplane radio altitude is greater than 5 feet and the A/P is CMD engaged, the airplane is under A/P CMD ENGAGE control to selected modes.

(4) Flight Director - Pitch Channel (Fig. 9)

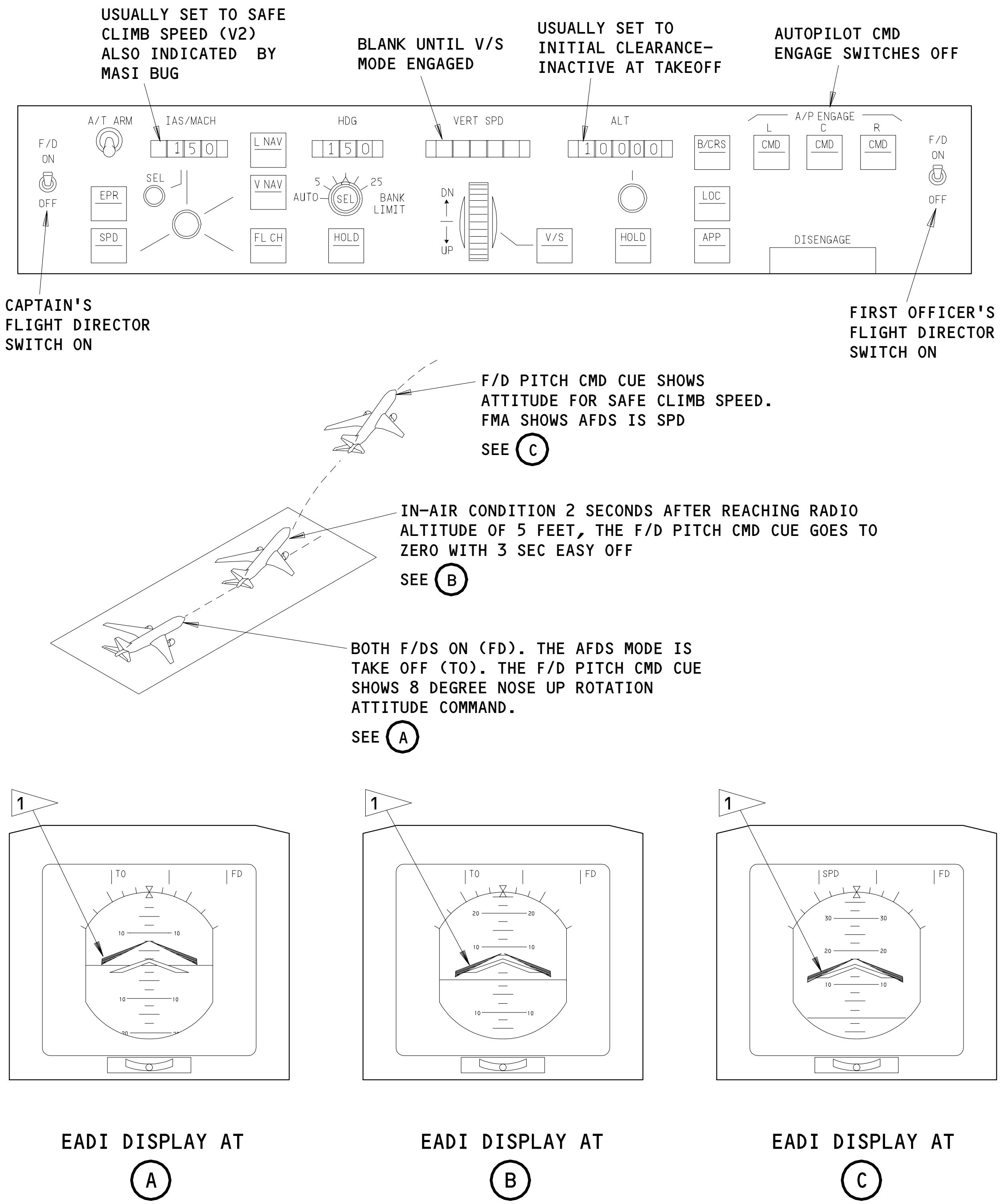
(a) Flight director pitch commands are computed continuously regardless of whether or not the A/P is engaged or the F/D is on. Positioning the F/D switches ON enables flight director displays for all available modes.

(b) The VNAV, V/S, ALT SEL, ALT HOLD, FLCH and APP modes are available as F/D modes. The IAS/MACH speed selection is available for the FLCH mode. The pilot controls the airplane in response to F/D commands on the EADI directly with the control column/wheel.

(c) The F/D pitch commands are computed from selected pitch attitude commands, present pitch attitude, pitch attitude rate, and versine. Computed F/D pitch commands are summed with pitch rate. If in takeoff (T/O) mode, a reference +8 degrees (theta) pitch signal is summed with pitch attitude and then summed with the F/D pitch commands. Computed F/D pitch commands are applied to the symbol generators and EADIs of the EFIS.

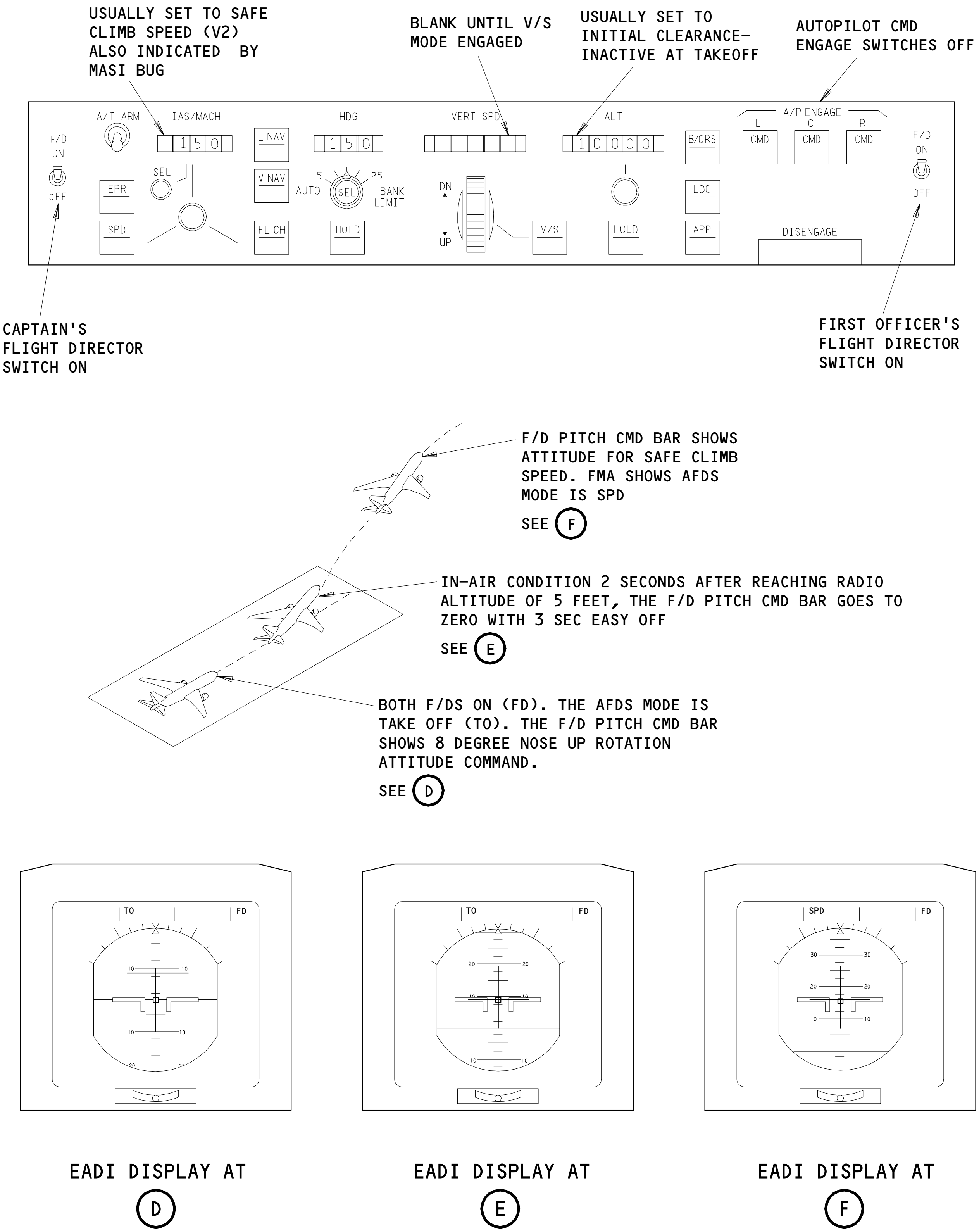
(d) GUI 001-009, 115;
a F/D on discrete causes the F/D command bars to be in view and the F/D pitch command displayed. With the F/D switches off, the F/D ON discrete is not provided and the EFIS inhibits the F/D displays. When the airplane radio altitude is greater than 5 feet and the A/P is CMD engaged, a fixed bias is sent to the EFIS which biases the F/D bars out of view. The airplane is then under A/P CMD ENGAGE control to selected modes.

(e) GUI 010-114, 116-999;
a F/D on discrete causes the F/D command bars to be in view and the F/D pitch command displayed. With the F/D switches off, the F/D ON discrete is not provided and the EFIS inhibits the F/D displays. When the airplane radio altitude is greater than 5 feet and the A/P is CMD engaged, the airplane is under A/P CMD ENGAGE control to selected modes.

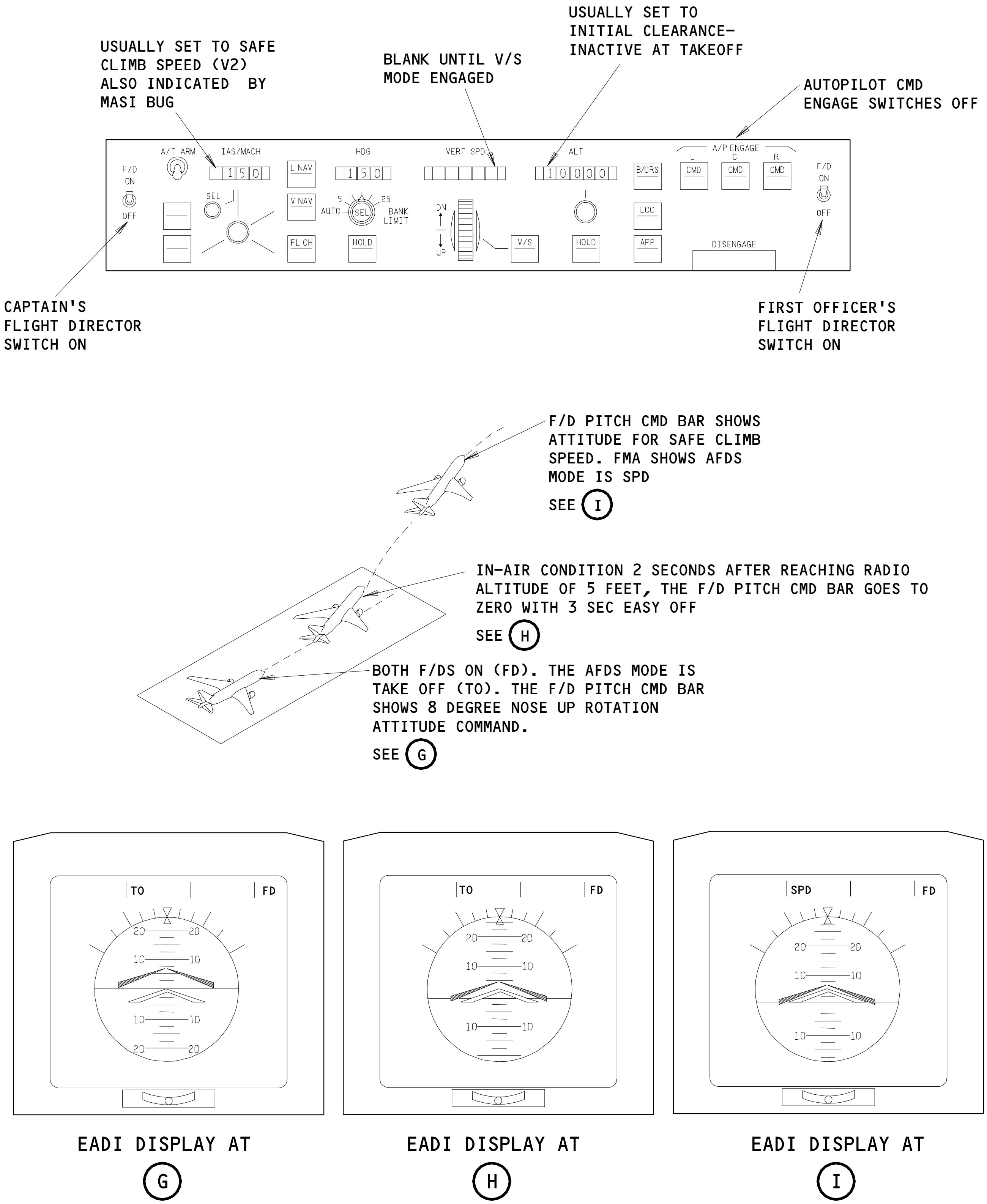


1 ON GUI 007,008 THE FLIGHT DIRECTOR CUE IS NOT FILLED

Flight Director - Pitch Channel
Figure 9 (Sheet 1)



Flight Director - Pitch Channel
Figure 9 (Sheet 2)



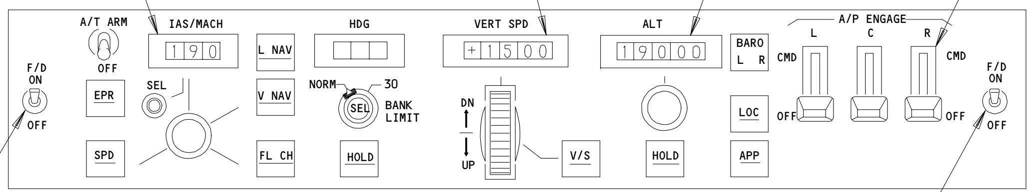
Flight Director - Pitch Channel
Figure 9 (Sheet 3)

USUALLY SET TO SAFE CLIMB SPEED (V2) ALSO INDICATED BY MASI BUG

BLANK UNTILL V/S MODE ENGAGED

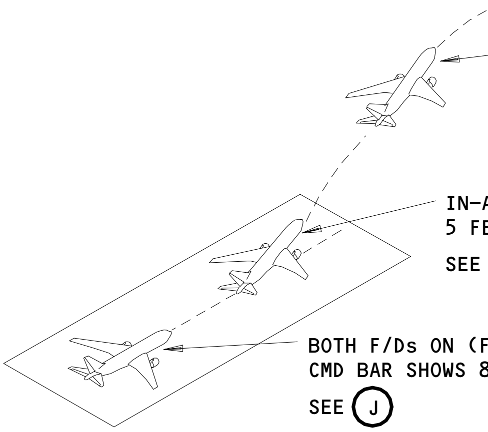
USUALLY SET TO INITIAL CLEARANCE- INACTIVE AT TAKEOFF

AUTOPILOT CMD ENGAGE SWITCHES OFF



CAPTAIN'S FLIGHT DIRECTOR SWITCH ON

FIRST OFFICER'S FLIGHT DIRECTOR SWITCH ON



F/D PITCH CMD BAR SHOWS ATTITUDE FOR SAFE CLIMB SPEED. FMA SHOWS AFDS MODE IS SPD.

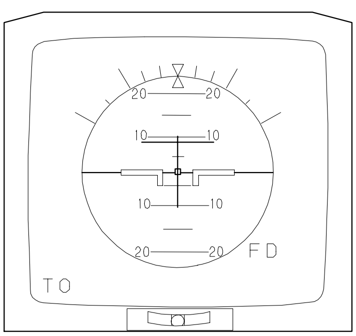
SEE (L)

IN-AIR CONDITION 2 SECONDS AFTER REACHING RADIO ALTITUDE OF 5 FEET, THE F/D PITCH CMD BAR GOES TO ZERO WITH 3 SEC EASY OFF.

SEE (K)

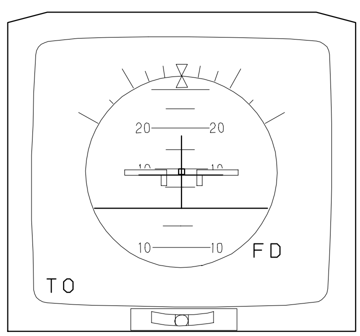
BOTH F/Ds ON (FD). THE AFDS MODE IS TAKE OFF (TO). THE F/D PITCH CMD BAR SHOWS 8 DEGREE NOSE UP ROTATION ATTITUDE COMMAND.

SEE (J)



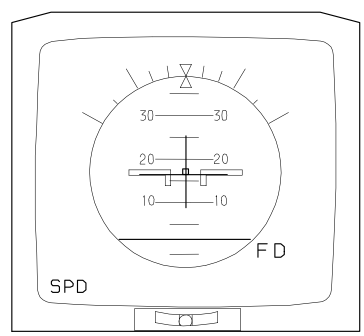
EADI DISPLAY AT

J



EADI DISPLAY AT

K



EADI DISPLAY AT

L

Flight Director - Pitch Channel
Figure 9 (Sheet 4)

(f) The flight director switches on the MCP cause the flight director bars to appear on the EADI for that side selected. The flight director bars provide the pilots information for manually flying a preselected flight profile. If the F/D is switched ON while the aircraft is on the ground, takeoff mode will automatically engage. If the F/D is switched ON in flight and no other mode is selected, the vertical speed mode automatically engages.

(g) GUI 001-009, 115;
the flight director EADI indications are controlled by the FCC selected with the FLT DIR instrument select switch. The FCC controls the flight director or the autopilot in CMD, but not both at the same time. If the flight directors source FCC is also engaged in CMD, the flight director bars are biased out of view.

(h) GUI 010-114, 116-999;
the flight director EADI indications are controlled by the FCC selected with the FLT DIR instrument select switch. The FCC controls the flight director or the autopilot in CMD. If the flight directors source FCC is also engaged in CMD, the flight director bars remain displayed on EADI.

(5) Pitch Channel - Takeoff Mode

(a) Takeoff mode is automatically engaged when either or both flight director switches are switched ON while the airplane is on the ground. Takeoff mode displays appropriate flight director bar commands for proper rotation and climbout. The flight director remains in takeoff mode until another valid mode is engaged.

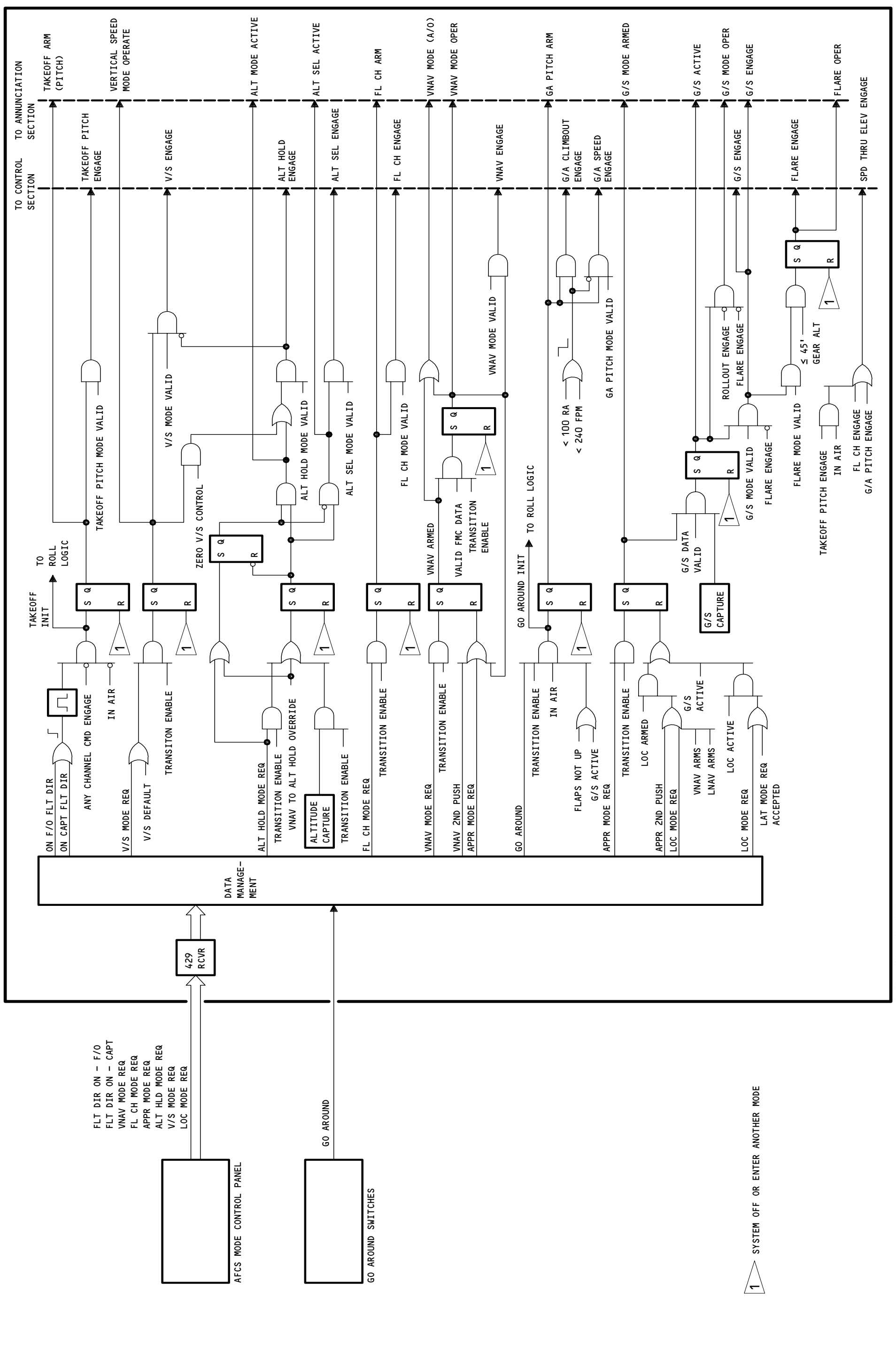
(b) Before takeoff, the crew must set a climbout speed on the MCP. The flight director will show a pitch-up command of 8.0 degrees. When the radio altimeter is more than 5.0 feet for 2.0 seconds, an INAIR signal adjusts the 8.0 degree command to the altitude command necessary to keep the takeoff speed.

(c) The EADI flight mode annunciator displays TO when takeoff mode is engaged. It continues to display TO until takeoff mode is disengaged by selecting any other valid mode or by switching both flight directors OFF. The following are valid modes that disengage the takeoff mode:

1) Altitude mode active - select altitude hold or selected altitude captured.

2) Flight level change - select FLCH mode.
3) Vertical navigation - VNAV captured
   a) The VNAV ARM is a submode. When VNAV is pressed, the mode request is accepted and the dot-bar matrix is illuminated, VNAV is annunciated on the MCP and in the pitch mode armed section of the EADI. When the VNAV path is captured, VNAV mode is entered and VNAV is annunciated in the pitch mode engaged section of the EADIs. Pressing the VNAV switch/light a second time deactivates the VNAV modes.
4) If a valid mode is not selected and CMD is engaged, vertical speed mode automatically engages.

(6) Autopilot/Flight Director System Pitch Mode Logic and Control Signal Flow (Fig. 10)
   (a) Takeoff Mode Control Logic
      1) Takeoff mode logic causes Takeoff Attitude engage (TAKEOFF ATT ENG) to be a logic one when the airplane is not in the air, CMD switches are off, either or both F/D switches are ON, and TAKEOFF SPD ENG is logic zero. When TAKEOFF ATT ENG is logic one, the 8 degree pitch up bar command is enabled.
      2) At a radio altitude of 5 feet a logic zero is sent to the AND and OR gates in the IN-AIR logic circuit. The logic zero is put through a 2 second delay. The NAND gate outputs a logic one (IN-AIR) 2 seconds after the airplane passed the 5 foot point. The IN-AIR logic one enables the SPD ENG and disables the 8 degree pitch-up bar command (TAKEOFF ATT ENG). This adjusts the 8 degree pitch-up command to the set pitch altitude in 2 seconds to keep the takeoff speed.
   (b) Takeoff Mode Control Signal Flow and Monitoring
      1) The TAKE OFF SPD ENG provides flight director pitch commands using speed schedules for a safe climbout speed. The roll channel provides a versine compensation gain schedule to provide pitch corrections for an airplane bank angle greater than ±1 degree.
         a) The versine signal is modified by the versine compensation gain schedule as a function of flap position and impact pressure (IAS).



Autopilot/Flight Director System Pitch Modes Logic Schematic
Figure 10 (Sheet 1)

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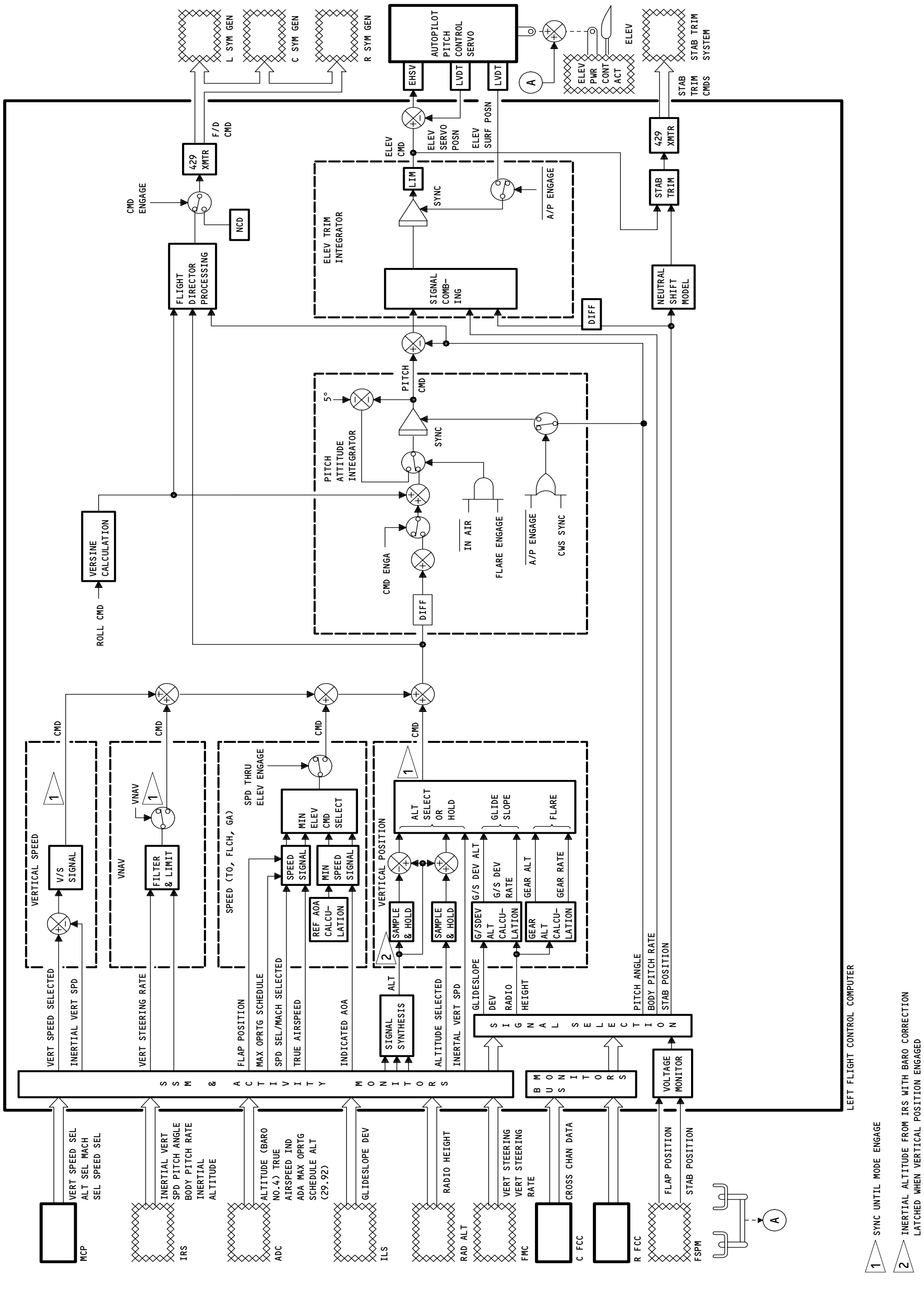

Autopilot/Flight Director System Pitch Control Signal Flow Schematic
Figure 10 (Sheet 2)

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b) Increased flap position is directly proportional and increases the signal. An increase in impact pressure is inversely proportional and decreases the signal.

(c) Speed Through Elevator

1) Speed Through Elevator Command Processing

a) The Speed Through Elevator (STE) function computes elevator commands to capture and hold the selected speed. It is normally used with autothrottle to achieve airspeed and rate of climb control. The alpha floor calculation computes elevator commands to capture and hold minimum speed (130% of stall speed) for angle of attack control. The A/T mode predicts the elevator response necessary to compensate for throttle change and adds this to the other two signals.

b) Command selection normally uses STE signals but automatically selects alpha floor signals when necessary to prevent flying below minimum speed. These signals produce the FLCH proportional command in degrees of pitch attitude.

c) Three flight modes use Speed Through Elevator. The takeoff mode, Flight Level Change (FLCH) mode, and go-around mode use the speed through elevator function. The FLCH mode combines with throttles to reach a new flight altitude. In the go-around mode, throttles are used to gain altitude and maintain speed.

2) Takeoff Mode (Speed Engage)

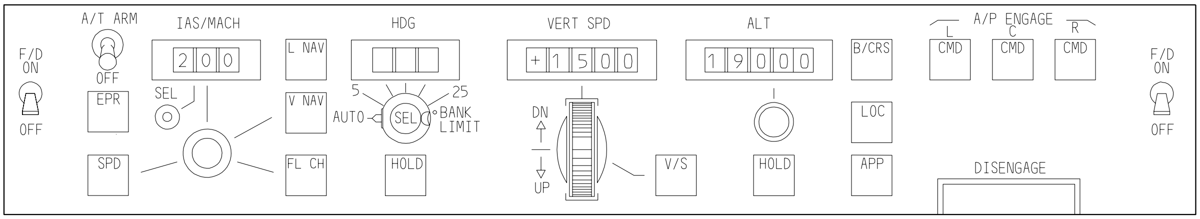
a) The takeoff mode is a F/D only mode. The speed through elevator function provides a smooth transition from rotation bar bias to speed control when in the air.

3) Flight Level Change Mode

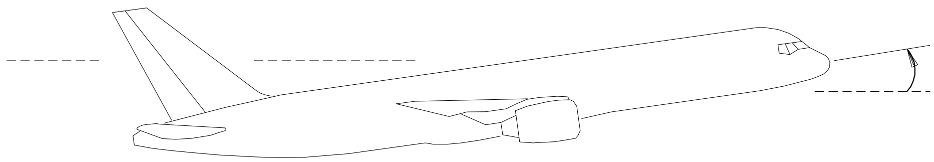
a) In FLCH, the pitch channel controls the elevators to capture and maintain the selected IAS/MACH. The throttles are used to control rate of climb or descent. With the A/T armed, throttles control vertical speed based on the test altitude change requested. With the A/T off, the pilots control the throttles as required. Without any change of thrust, capturing a new speed results in vertical speed change.

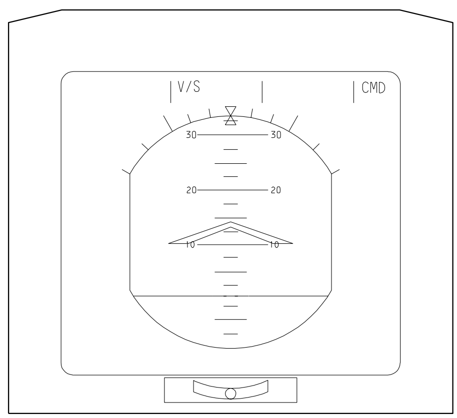
4) Go-Around Mode
   a) The go-around (G/A) mode is ARMED when the glideslope is captured or flaps are extended. The pitch channel controls speed and the throttles control rate of climb (either automatic or manual throttle). The airplane must achieve 400 ft radio altitude in order to exit the G/A mode.

(7) Pitch Channel - Vertical Speed Mode (Fig. 11)
   (a) V/S entry and exit is possible provided transition logic and acceptance criteria are satisfied. Initial entry requires that an A/P or F/D be engaged. Entry from other modes requires that the V/S switch/light be pressed, the request accepted and no incompatable modes be in operation.
   (b) Exit from V/S to control inactive requires switching A/P and F/D to off. Exit to FLCH, ALT HOLD and G/A modes requires that the mode switch/light be pressed and the request accepted. Vertical speed is active during VNAV ARM and G/S ARM conditions. Exit to VNAV capture requires that VNAV arm is established which includes VNAV switch/light pressed and accepted, no ILS modes are established, and the airplane is within the capture zone. Exit to glideslope capture requires that the APP switch/light is pressed and accepted and that the airplane is within the glidelsope capture zone.
   (c) Two submodes are available: VNAV ARM and G/S ARM. The airplane is controlled with V/S until reaching the capture zone of VNAV or G/S. After capture, V/S control is stopped and VNAV or G/S signals provide pitch commands. VNAV and G/S armed are annunciated on the EADI.
   (d) A valid V/S mode switch request is required for acceptance into V/S mode. However, entry into V/S is automatic without pressing a V/S switch/light when an F/D or A/P is engaged from control inactive, and an autothrottle SPD or EPR mode is engaged.
   (e) With V/S engaged, the airplane holds the existing V/S if within ±30 degrees pitch angle. If greater than 30 degrees, the airplane attitude is reduced to 30 degrees and the corresponding vertical speed is used as the reference. The EADI flight mode annunciator displays V/S when the V/S pitch mode is engaged.



AFDS MCP





EADI

- COMMAND ENGAGE (CMD ENG)
  - AIRPLANE
    - VERTICAL SPEED (V/S) MODE
    - CLIMBOUT AT PRESENT V/S DISPLAYED
    - V/S MAY BE CHANGED WITH V/S KNOB
  - MCP
    - A/P CMD SWITCH ENGAGED
    - ALT DISPLAY SET AT ATC CLEARED ALTITUDE
    - VERTICAL SPEED DISPLAY SYNC TO V/S AT ENGAGEMENT, THEN CHANGED AS DESIRED
    - V/S SWITCH DOT-BAR MATRIX ILLUMINATED
  - EADI
    - F/D BARS INDICATE AIRPLANE AT COMMANDED PITCH ATTITUDE IF F/D SWITCH ON.
    - FMA INDICATES:
      - STATUS-CMD
      - PITCH MODE ENGAGED-V/S

Pitch Channel - Vertical Speed Mode
Figure 11 (Sheet 1)

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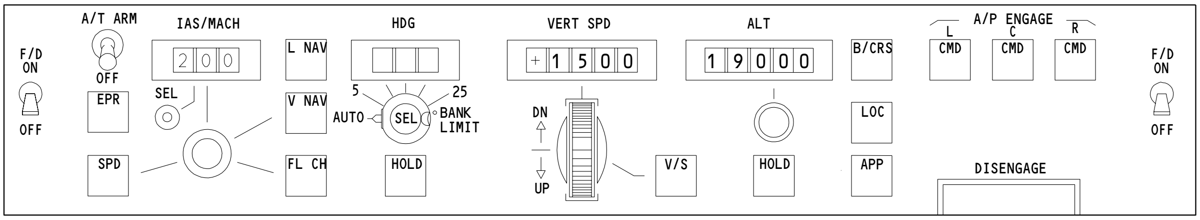

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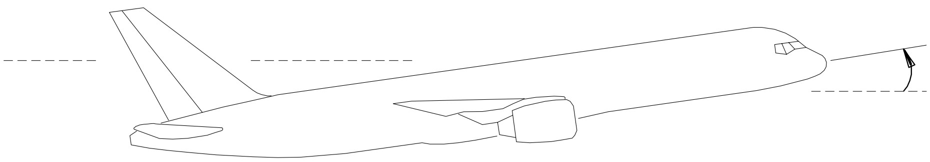

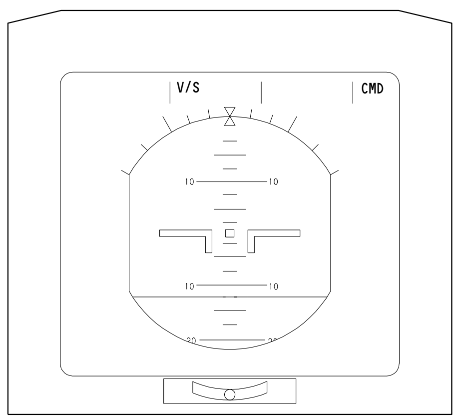

EADI

- COMMAND ENGAGE (CMD ENG)
  - AIRPLANE
    - VERTICAL SPEED (V/S) MODE
    - CLIMBOUT AT PRESENT V/S DISPLAYED
    - V/S MAY BE CHANGED WITH V/S KNOB
  - MCP
    - A/P CMD SWITCH ENGAGED
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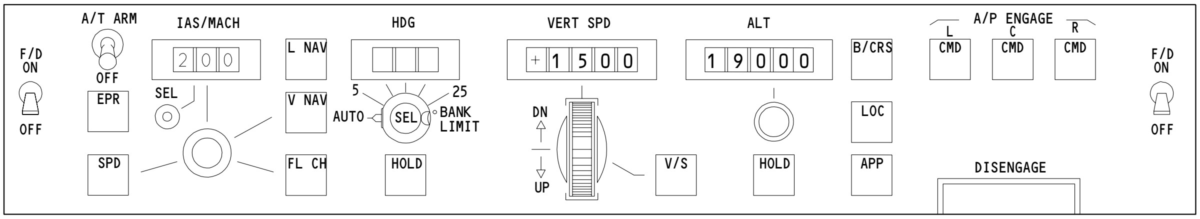
    - F/D BARS INDICATE AIRPLANE AT COMMANDED PITCH ATTITUDE IF F/D SWITCH ON.
    - FMA INDICATES:
      - STATUS-CMD
      - PITCH MODE ENGAGED-V/S

Pitch Channel - Vertical Speed Mode
Figure 11 (Sheet 2)



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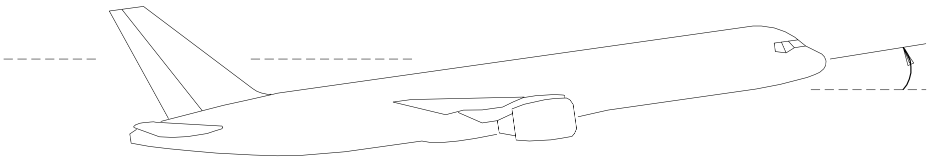

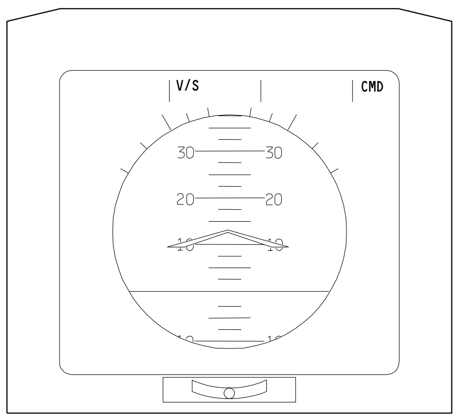

EADI

- COMMAND ENGAGE (CMD ENG)
  - AIRPLANE
    - VERTICAL SPEED (V/S) MODE
    - CLIMBOUT AT PRESENT V/S DISPLAYED
    - V/S MAY BE CHANGED WITH V/S KNOB
  - MCP
    - A/P CMD SWITCH ENGAGED
    - ALT DISPLAY SET AT ATC CLEARED ALTITUDE
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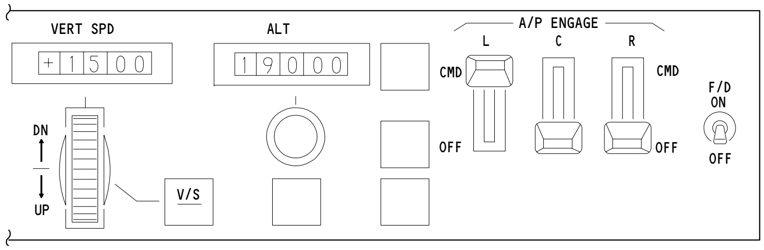
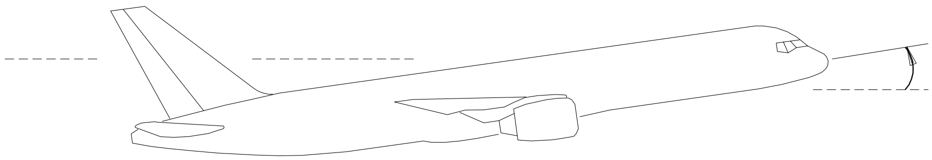
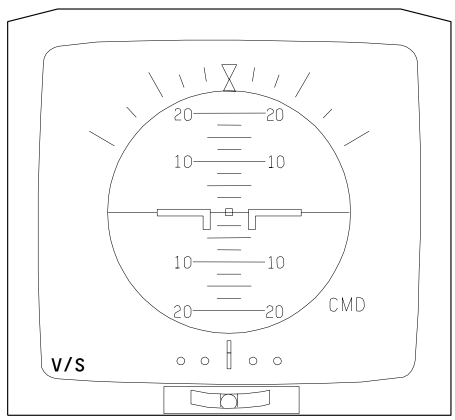
  - EADI
    - F/D BARS INDICATE AIRPLANE AT COMMANDED PITCH ATTITUDE IF F/D SWITCH ON.
    - FMA INDICATES:
      - STATUS-CMD
      - PITCH MODE ENGAGED-V/S

Pitch Channel - Vertical Speed Mode
Figure 11 (Sheet 3)



AFDS MCP





EADI

- COMMAND ENGAGE (CMD ENG)
  - AIRPLANE
    - VERTICAL SPEED (V/S) MODE
    - CLIMBOUT AT PRESENT V/S DISPLAYED
    - V/S MAY BE CHANGED WITH V/S KNOB
  - MCP
    - F/D SWITCH OFF
    - A/P CMD SWITCH ENGAGED
    - ALT DISPLAY SET AT ATC CLEARED ALTITUDE
    - VERTICAL SPEED DISPLAY SYNC TO V/S AT ENGAGEMENT, THEN CHANGED AS DESIRED
    - V/S SWITCH DOT-BAR MATRIX ILLUMINATED
  - EADI
    - F/D BARS INDICATE AIRPLANE AT COMMANDED PITCH ATTITUDE
    - FMA INDICATES:
      - STATUS-CMD
      - PITCH MODE ENGAGED-V/S

Pitch Channel - Vertical Speed Mode
Figure 11 (Sheet 4)

(f) Vertical Speed Command Processing
1) Vertical speed command processing computes the pitch path (THETA proportional) command, the pitch rate integral command in degrees of pitch attitude per second, and versine in degrees of pitch attitude.
2) The pitch reference signals appropriate to the V/S mode are selected. Pitch rate limits, gain schedules, and filters are applied as necessary. Signals are synchronized during the non-pitch active modes.

(g) Vertical Speed Processing Logic
1) Vertical speed processing logic generates vertical speed engage and vertical path control logic for the vertical speed command processing circuits.

(h) Vertical Speed Signal Processing
1) Vertical speed signal processing provides the following gain scheduling signals:
a) Versine compensation
b) True airspeed inverse
c) Angle of Attack/true airspeed

(i) Roll Channel Signal Processing
1) Roll channel signal processing generates the cosine of the roll attitude command which is used for the versine computation.

(j) Pitch Flight Director Signal Processing
1) Pitch flight director signal processing generates the pitch F/D bar command from the pitch path command input.

(8) Vertical Speed Control Logic
(a) Vertical Speed Mode Inactive
1) The logic zero condition for vertical speed engage shown in Fig. 10 is generated with the following 3 conditions:
a) F/D ON with the CMD switch selected to CMD. This generates a F/D DISENG momentary high pulse.
b) Altitude hold mode selected. This causes the input to vertical speed engage gate to go high.
c) Vertical speed entry conditions not satisfied. This causes the input to vertical speed engage gate to go low.

2) Any of the above conditions generate a vertical speed engage logic zero which allows synchronization of the existing V/S command. The vertical speed engage valid computation is normally set to one as a configuration management interlock.
3) The logic zero condition for vertical path control is generated when:
   a) Vertical speed engage is zero (disengaged)
   b) Glideslope engage or altitude hold engage and a one-shot pulse for a vertical position mode change is generated.
4) Both of the above conditions generate a vertical path control logic zero. This opens the path for pitch rate compensation which is composed of wind acceleration and AOA rate filters.

(b) Vertical speed Mode Engaged
1) Vertical speed engage logic one is generated to provide vertical speed command control with the following conditions:
   a) The F/D switch ON and a REQUEST and CMD VALID pulse gone low provide a zero state for F/D Disengage.
   b) Altitude hold is not engaged.
   c) Manual or automatic V/S mode entry conditions are satisfied for V/S mode transition generates a high.
   d) V/S engage valid equals one.
2) When vertical speed engage equals one, the pitch rate compensation path to the pitch attitude integrator is closed. This logic one condition for vertical speed control is generated when:
   a) Vertical speed engage is equal to one or
   b) Glideslope engage, flare engage, or altitude hold engage is high and the one-shot input to G11 is low due to no mode changes.

(9) Vertical Speed Control Signal Flow and Monitoring
(a) Gain Schedule Signal Processing
1) The cosine of the PHI CMD gain schedule shown in Fig. 11 is developed from a cosine computation of the roll attitude command.

2) The angle of attack (AOA) and true airspeed (TAS) gain schedule computation is computed from impact pressure (IAS) and TAS which have been modified by flap position.
3) The TAS inverse gain schedule computation is computed from TAS modified by flap position.
4) The versine compensation gain schedule computation is computed from impact pressure and flap position.

(10) Vertical Speed Command Processing

(a) Pitch Attitude Integrator Functions

1) The pitch attitude integrator computes four functions: pitch compensation - Versine, pitch rate integral command, pitch path command, and pitch F/D bar command. The integrator synchronizes pitch computations when the A/P is not in the V/S mode. It also selects the correct pitch reference for the V/S mode and ensures the pitch command produced is in the correct direction. Gain programs are applied as necessary.

(b) Gain Schedules

1) Four gain schedules are used. Gain schedule 1 controls versine as an inverse function of impact pressure (IAS) and as a direct function of flap position. Gain schedule 2 controls flight path acceleration as an inverse function of IAS and TAS. Gain schedule 3 controls pitch rate as a function of the cosine of roll attitude commands. The pitch rate is the summation of wind acceleration and AOA rate components. Gain schedule 4 controls V/S error as a function of inverse TAS. The V/S error is the difference between selected V/S and existing V/S.

(c) Synchronization

1) During synchronization, the V/S command from the vertical speed low pass filter goes to zero when vertical speed engage is a logic zero. The logic zero is generated as a pulse when CMD is engaged and when switching from one mode to another. During synchronization the pitch attitude integrator maintains a constant output for elevator stability.

2) The V/S error is synchronized to zero by V/S command lag filter when vertical speed engage is a logic zero. The output from V/S low pass filter is zero. With vertical patch control a logic zero, the pitch rate path compensation signal is removed from the pitch rate integral command signal. With inputs zero, the pitch path integral command is zero. The pitch F/D bar command and pitch path command are zero.

(d) V/S Mode Logic/Switch Signal Status

1) After synchronization, V/S engage and vertical path control are both logic 1. Vertical speed command lag filter and vertical speed low pass filter are activated and S1 is closed. The FLCH and vertical position commands are zero. The VNAV, FLCH and vertical position integral commands are zero.

(e) V/S CMD Signal Processor Outputs

1) With V/S ENG mode, pitch attitude integrator output is proportional to filtered V/S error path command and V/S error integral command and pitch rate path compensation. The V/S error path command is provided as the pitch path command input to the pitch attitude integrator. The V/S error integral command and pitch attitude integrator are summed and provided as the pitch rate integral command input to the pitch attitude integrator.

(f) V/S Mode Reference

1) The V/S reference from V/S command lag filter is developed from the MCP selected vertical speed (altitude rate) display. The display is automatically set to the existing vertical speed at V/S ENG. The display is manually set by rotating the VERT SPD wheel.

2) Prior to V/S engage, V/S mode reference is synchronized to the IRS altitude rate value. The output is zero when V/S engage is zero.

3) After V/S engage, the V/S mode reference signal in V/S command lag filter is time delayed and integrated to dampen any rapid VERT SPD wheel changes. Rate changes are limited to prevent exceeding 0.lg during V/S capture. The output from V/S command lag filter is summed with IRS altitude rate value to provide the V/S error.

(g) V/S Error Signal Processing
1) The V/S error signal is modified as a function of inverse TAS and applied to a V/S low pass filter. The low pass filter processes the gain scheduled V/S error through an integrator with a time constant of 1.0 seconds. This provides the pitch path CMD.
(h) Pitch Path Command
1) The pitch path command provides a signal, proportional to V/S error, to change elevator deflections for short turn (immediate response) due to V/S errors from the selected or present value. It diminishes as the airplane attitude changes and so decreases the V/S error magnitude. The V/S error will not go to zero with pitch path command only, due to changed aerodynamic effects.
(i) Pitch Rate Integral Command
1) The pitch rate integral command is composed of V/S error which is reduced by a factor of 0.3 and pitch rate path compensation, which is composed of short duration wind accelerations and AOA rate components. These summed signals provide the long term integral command for elevator trim to reduce the V/S error to zero (selected V/S = actual V/S).
(j) Versine - Pitch Compensation
1) Versine pitch compensation is computed from the cosine of roll attitude and is modified by versine compensation. It is an inverse function of impact pressure and a direct function of flap position. The pitch compensation versine is applied to the pitch attitude integrator to provide a proportional pitch up signal component during a bank.
(k) Vertical Speed Flight Director Command
1) The pitch path command provides the input for developing the pitch F/D bar command.
(l) Pitch Rate Path Compensation
1) The sum of flight path acceleration and AOA rate filtered make up pitch rate path compensation. The flight path acceleration is modified as an inverse function of impact pressure and TAS. The AOA rate filtered is limited to a 1.0 deg/sec change to prevent rapid rate changes. The resultant signal is modified as a function of the cosine of roll attitude command. The pitch rate path compensation attenuates vertical accelerations and elevator activity during turbulance.

(m) Flight Path Acceleration
   1) Inertial flight path acceleration from the IRS and the flight path acceleration wind signal from the wind acceleration filter are summed to make up flight path acceleration. This is then modified as an inverse function of impact pressure and TAS. The modified flight path acceleration provides a desired pitch rate representing actual flight path acceleration and a small component of wind acceleration used in computing an integral command compensation signal to the elevator. This reduces vertical acceleration in the V/S mode.

(n) AOA - Rate Filtered
   1) The AOA rate filtered is computed from the computed AOA rate from the IRS. This is integrated and summed with corrected AOA (vane angle from the ADC and IRS) and applied through a low pass filter. The filter limits the signal to ±1.0 deg/sec for damping of elevator movement. The AOA rate filtered represents a component of pitch rate path compensation to the elevator to allow for vertical wind gusts on the AOA vane.

(o) Pitch Attitude Integrator Processing
   1) The pitch attitude integrator establishes a pitch attitude reference. It integrates pitch commands from MCP manually selected modes, automatically selected modes, and navigation sources. It also generates pitch attitude error which is processed to provide the elevator servo command. Pitch signal summation to the pitch attitude integrator is provided from the pitch rate integral command, the pitch path command, and the versine command.
   2) Vertical Speed Signal Logic
      a) Switches 1, 4 and 5, remain as drawn since their control logic, TCLIM IN AIR NOT and TSYNC, are all zero. Switches 2, 3, and 6 transfer (IN AIR, TPATH, and PCE are logic one).
   3) Vertical speed Signal Processing
      a) The pitch rate integral command is summed with pitch attitude error to provide a long standing error washout at the rate of 0.1 deg/sec/deg.

b) Pitch path (proportional) command provides the immediate response elevator command. This is differentiated and summed with pitch rate integral command. V/S ENG allowing the composite pitch command signal to be summed with any possible signal.

c) The versine command is controlled as a direct function of IAS and as an inverse function of flap position. This is differentiated and summed to form composite pitch rate command.

d) The composite pitch command is integrated and its output is limited to ±30 degrees of pitch angle. This is summed with the reference pitch attitude to provide a pitch attitude error command for processing.

(p) Pitch Altitude Hold Mode

1) The pitch altitude hold mode is manually requested by pressing the HOLD switch/light on the MCP. A valid request is indicated by the lighting of the dot-bar matrix on the lower half of the switch cover. The altitude hold mode is automatically selected when in the vertical speed mode and V/S is zero. (The V/S light will remain lighted.)

2) The altitude hold mode is also selected automatically when in the VNAV mode and the FMC has commanded a target altitude in a direction away from the MCP selected altitude. Under these conditions, VNAV disengages and the VNAV switch/light extinguishes. Pitch altitude hold is active in CMD ENG or F/D on.

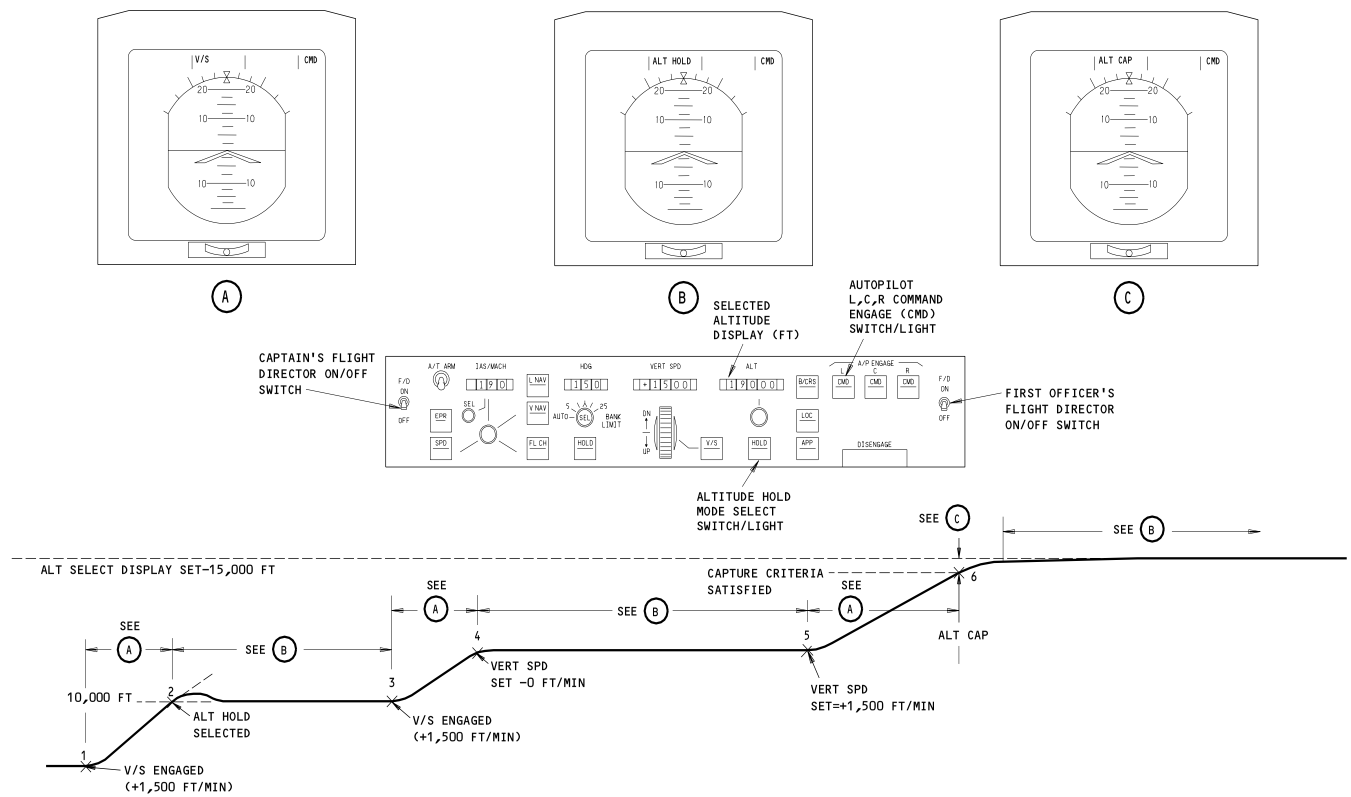
(q) Pitch Altitude Select Mode

1) Altitude select arm and altitude select engage are not manually selectable on the MCP. Altitude select occurs at altitude capture, which is determined by capture criteria, and may be entered from any of the following modes: V/S, FLCH, VNAV, T/O (in F/D only), and GA. Altitude select may not be entered from the G/S mode.

(11) Altitude Select/Hold Control Logic (Fig. 10)

(a) Flare Engage Logic

1) Flare engage logic is generated by computation of the appropriate data from the data management bus.



1 • ALTITUDE SELECT DISPLAY MANUALLY SET FOR ATC CLEARANCE OF 15,000 FT
• V/S SWITCH PUSHED FOR V/S ENG
  - V/S DOT-BAR ILLUMINATES
• VERT SPD DISPLAY SET FOR +1,500 FT/MIN
  - A/P CLIMBS TO POINT 2 IN V/S MODE

2 • ALT HOLD SWITCH PUSHED (AT 10,000 FT)
  - V/S DOT-BAR EXTINGUISHES
  - V/S DISPLAY BLANKS
  - ALT HOLD DOT-BAR ILLUMINATES
  - A/P LEVELS TO BAROMETRIC ALTITUDE AT TIME OF ALT HOLD SELECTED
  - AFDS UNDER ALT HOLD LOGIC CONTROL
  - FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES ALT HOLD

3 • V/S SWITCH PUSHED FOR V/S ENG
  - ALT HOLD DOT-BAR EXTINGUISHES
  - V/S DOT-BAR ILLUMINATES
  - VERT SPD DISPLAY INDICATES 0
• VERT SPD DISPLAY FOR +1500 FT/MIN
  - A/P CLIMBS TO POINT 4 IN V/S MODE

4 • VERT SPD DISPLAY SET TO 0 AT SELECTED POINT
  - A/P LEVELS & HOLDS BAROMETRIC ALTITUDE
  - UNDER ALT HOLD LOGIC CONTROL
  - V/S MODE ARMED
  - FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES V/S

5 • VERT SPD DISPLAY SET FOR +1500 FT/MIN
  - V/S MODE ENGAGED
  - V/S DOT-BAR ILLUMINATES
  - A/P CLIMBS TO POINT 6 IN V/S MODE

6 • CRITERIA SATISFIED FOR ALT CAPTURE
  - V/S DOT-BAR EXTINGUISHES
  - HOLD DOT-BAR ILLUMINATES
  - FMA ON EADI INDICATES ALT CAP UNTIL <10 FT FROM SELECTED ALT, THEN INDICATES ALT HOLD
  - A/P LEVELS OFF TO 15,000 FT
  - AFDS UNDER ALT SELECT LOGIC CONTROL

Pitch Channel - Altitude Select/Hold Mode
Figure 12 (Sheet 1)

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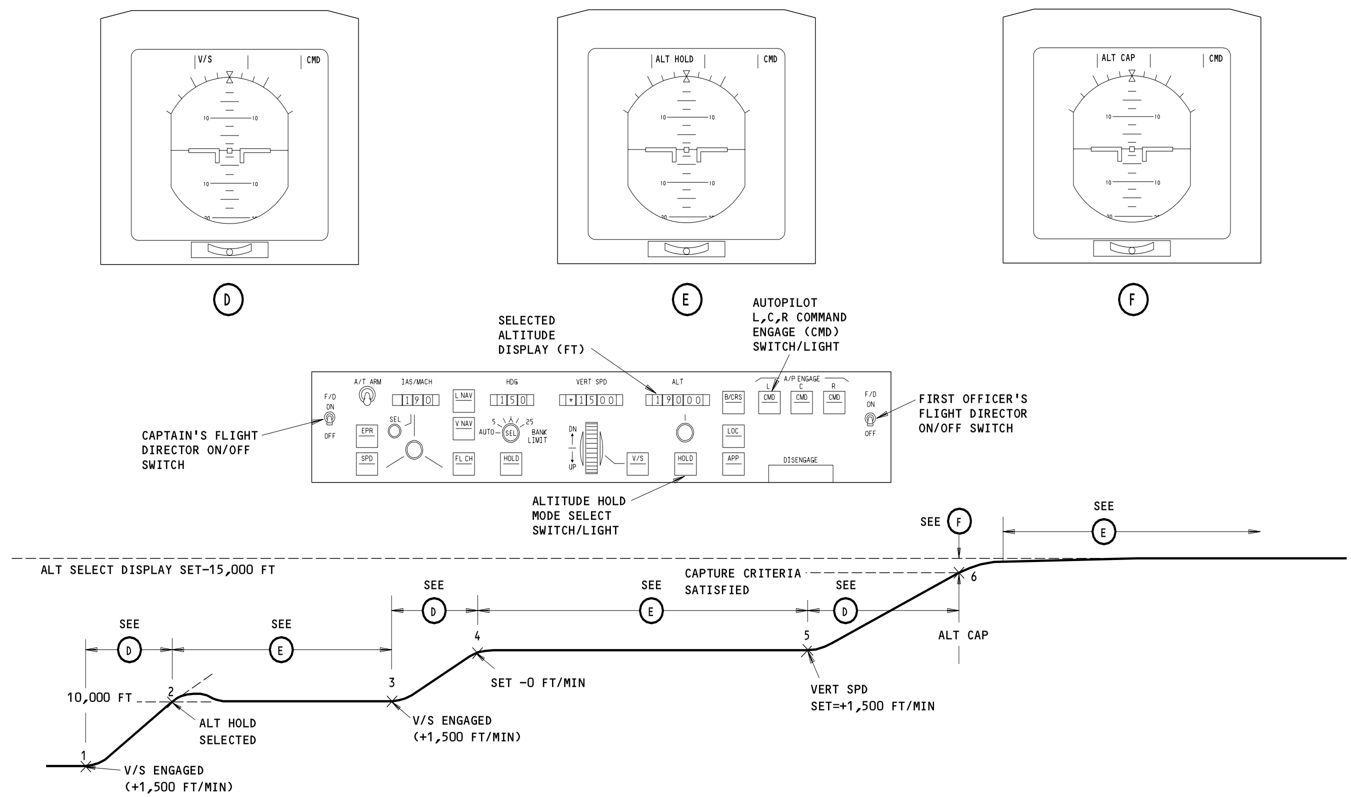

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2 • ALT HOLD SWITCH PUSHED (AT 10,000 FT)
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– V/S DISPLAY BLANKS
– ALT HOLD DOT-BAR ILLUMINATES
– A/P LEVELS TO BAROMETRIC ALTITUDE AT TIME OF ALT HOLD SELECTED
– AFDS UNDER ALT HOLD LOGIC CONTROL
– FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES ALT HOLD

3 • V/S SWITCH PUSHED FOR V/S ENG
– ALT HOLD DOT-BAR EXTINGUISHES
– V/S DOT-BAR ILLUMINATES
– VERT SPD DISPLAY INDICATES 0
• VERT SPD DISPLAY FOR +1500 FT/MIN
– A/P CLIMBS TO POINT 4 IN V/S MODE

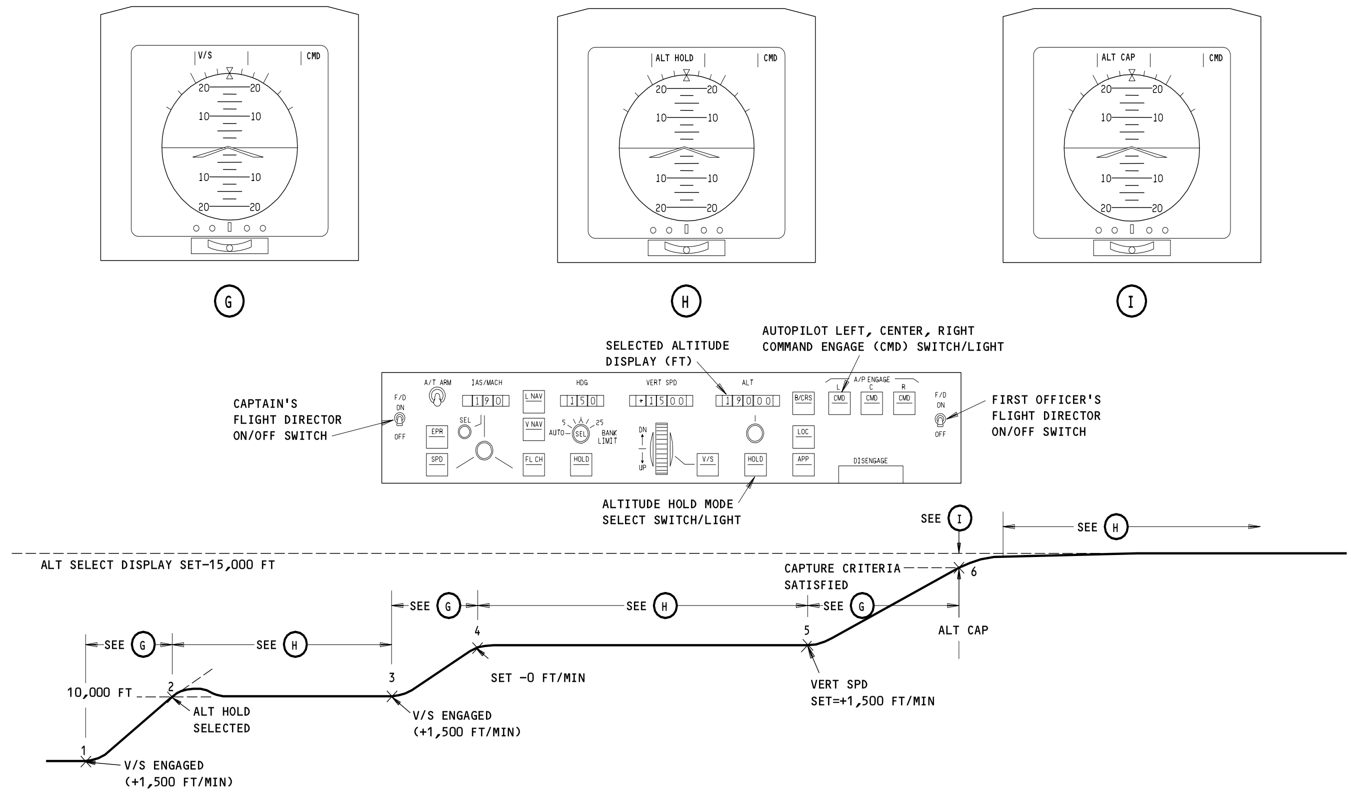
4 • VERT SPD DISPLAY SET TO 0 AT SELECTED POINT
– A/P LEVELS & HOLDS BAROMETRIC ALTITUDE
– UNDER ALT HOLD LOGIC CONTROL
– V/S MODE ARMED
– FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES V/S

5 • VERT SPD DISPLAY SET FOR +1500 FT/MIN
– V/S MODE ENGAGED
– V/S DOT-BAR ILLUMINATES
– A/P CLIMBS TO POINT 6 IN V/S MODE

6 • CRITERIA SATISFIED FOR ALT CAPTURE
– V/S DOT-BAR EXTINGUISHES
– HOLD DOT-BAR ILLUMINATES
– FMA ON EADI INDICATES ALT CAP UNTIL <10 FT FROM SELECTED ALT, THEN INDICATES ALT HOLD
– A/P LEVELS OFF TO 15,000 FT
– AFDS UNDER ALT SELECT LOGIC CONTROL

Pitch Channel – Altitude Select/Hold Mode
Figure 12 (Sheet 2)

A44542



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  - A/P CLIMBS TO POINT 2 IN V/S MODE

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  - V/S DISPLAY BLANKS
  - ALT HOLD DOT-BAR ILLUMINATES
  - A/P LEVELS TO BAROMETRIC ALTITUDE AT TIME OF ALT HOLD SELECTED
  - AFDS UNDER ALT HOLD LOGIC CONTROL
  - FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES ALT HOLD

3 • V/S SWITCH PUSHED FOR V/S ENG
  - ALT HOLD DOT-BAR EXTINGUISHES
  - V/S DOT-BAR ILLUMINATES
  - VERT SPD DISPLAY INDICATES 0
• VERT SPD DISPLAY FOR +1500 FT/MIN
  - A/P CLIMBS TO POINT 4 IN V/S MODE

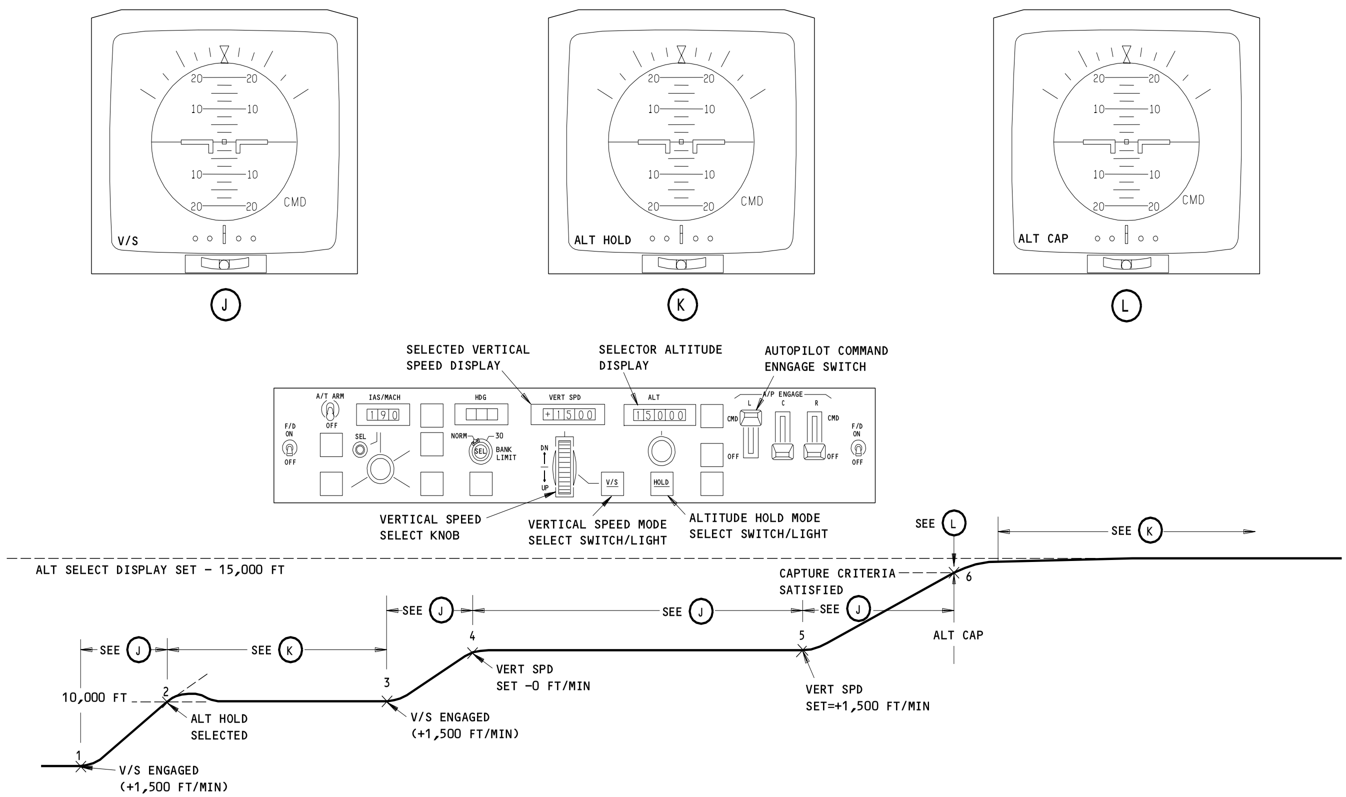
4 • VERT SPD DISPLAY SET TO 0 AT SELECTED POINT
  - A/P LEVELS & HOLDS BAROMETRIC ALTITUDE
  - UNDER ALT HOLD LOGIC CONTROL
  - V/S MODE ARMED
  - FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES V/S

5 • VERT SPD DISPLAY SET FOR +1500 FT/MIN
  - V/S MODE ENGAGED
  - V/S DOT-BAR ILLUMINATES
  - A/P CLIMBS TO POINT 6 IN V/S MODE

6 • CRITERIA SATISFIED FOR ALT CAPTURE
  - V/S DOT-BAR EXTINGUISHES
  - HOLD DOT-BAR ILLUMINATES
  - FMA ON EADI INDICATES ALT CAP UNTIL <10 FT FROM SELECTED ALT, THEN INDICATES ALT HOLD
  - A/P LEVELS OFF TO 15,000 FT
  - AFDS UNDER ALT SELECT LOGIC CONTROL

Pitch Channel - Altitude Select/Hold Mode
Figure 12 (Sheet 3)

C32051



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- V/S DOT-BAR ILLUMINATES
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2 • ALT HOLD SWITCH PUSHED (AT 10,000 FT)
- V/S DOT-BAR EXTINGUISHES
- V/S DISPLAY BLANKS
- ALT HOLD DOT-BAR ILLUMINATES
- A/P LEVELS TO BAROMETRIC ALTITUDE AT TIME OF ALT HOLD SELECTED
- AFDS UNDER ALT HOLD LOGIC CONTROL
- FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES ALT HOLD

3 • V/S SWITCH PUSHED FOR V/S ENG
- ALT HOLD DOT-BAR EXTINGUISHES
- V/S DOT-BAR ILLUMINATES
- VERT SPD DISPLAY INDICATES 0
• VERT SPD DISPLAY SET FOR +1500 FT/MIN
- A/P CLIMBS TO POINT 4 IN V/S MODE

4 • VERT SPD DISPLAY SET TO 0 AT SELECTED POINT
- A/P LEVELS & HOLDS BAROMETRIC ALTITUDE
- UNDER ALT HOLD LOGIC CONTROL
- V/S MODE ARMED
- FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES V/S

5 • VERT SPD DISPLAY SET FOR +1500 FT/MIN
- V/S MODE ENGAGED
- V/S DOT-BAR ILLUMINATES
- A/P CLIMBS TO POINT 6 IN V/S MODE

6 • CRITERIA SATISFIED FOR ALT CAPTURE
- V/S DOT-BAR EXTINGUISHES
- HOLD DOT-BAR ILLUMINATES
- FMA ON EADI INDICATES ALT CAP UNTIL <10 FT FROM SELECTED ALT, THEN INDICATES ALT HOLD
- A/P LEVELS OFF TO 15,000 FT
- AFDS UNDER ALT SELECT LOGIC CONTROL

Pitch Channel - Altitude Select/Hold Mode
Figure 12 (Sheet 4)

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(b) Vertical Position Approach Logic
  1) The vertical position approach logic is generated from flare engage or glideslope engage. This logic is required for switching between flare engage/glideslope engage circuitry and ALT HOLD/SELECT circuitry.

(c) Vertical Position Engage Logic
  1) Vertical position engage logic is generated by either flare engage, glideslope engage, altitude select engage and altitude hold engage and generates vertical position engage. Whenever a vertical position mode change occurs, vertical position engage is momentarily interrupted. The vertical position engage logic is required for switching from a synchronization mode to active flare engage, glideslope engage, altitude select engage, and altitude hold engage mode in the vertical position command processor.

(d) Altitude Select Engage
  1) The altitude select engage logic is generated and is momentarily zero at F/D disengage. The input requires altitude select engage valid and altitude select arm to be logic one. The altitude select arm logic requires that the ALT HOLD switch not be pressed and that the FMC does not command a target altitude in a direction away from the MCP altitude when in the VNAV mode. the ALT SELECT/HOLD mode must be active for altitude select arm to be logic one.
  2) The altitude select engage logic is required to establish the altitude select mode in the ALT SEL/HOLD signal development.

(e) Altitude Hold Engage
  1) The altitude hold engage logic is generated and is momentarily zero at F/D disengage. The altitude input can be generated three ways (provided altitude hold/engage valid is logic one).
    a) If the ALT HOLD switch/light dot bar matrix on the MCP is lighted and ALT SEL/HOLD mode is active.

b) When the FMC (in VNAV mode and not in ALT SEL ENG) commands the airplane to fly to a target altitude in a direction away from the MCP set altitude.

c) In the V/S mode, the V/S display set to zero will also generate altitude hold engage.

(12) Altitude Select/Hold Control Signal Flow and Monitoring (Fig. 10)

(a) Vertical Position Time Constant (Natural Frequency)

1) The vertical position time constant is generated for the three regions in which altitude capture can occur, to provide a safe and smooth transition path to the selected altitude. The three regions of concern are overshoot (region I), acceleration limit (region II) and no acceleration limit (region III). Additional requirements for the time constant calculation are vertical acceleration command limit, delta altitude command and delta altitude rate command. The delta altitude command provides the initial path command and the delta altitude rate command provides the initial path rate command at ALT CAP or when the airplane achieves the proper delta altitude and rate to smoothly achieve the selected altitude.

2) The three regions in which ALT CAP occurs defines the logic which activates time constant calculation for one iteration period. The time constant value is limited between 0.01 and 0.50 radian/second for a smooth transition path generation.

(b) Vertical Position Limit

1) The vertical position limit is generated by a logic 1 input from ALT CAP in either overshoot limit region or acceleration limit region. The vertical position limit is required for switching logic in the vertical command processor to input vertical acceleration command limit when the altitude is captured in the overshoot or acceleration limit regions.

(c) Vertical Acceleration Command Limit

1) The vertical acceleration command limit is a fixed value of vertical acceleration from the IRS at the time of ALT CAP, or it is 3.2 ft/sec (0.1g) prior to ALT CAP. Either of these fixed values will control the airplane until the altitude rate is zero or the absolute value of the rate of approach to selected altitude is reduced to the normal capture limit.

2) The sign of HDDCC is opposite that of delta altitude rate command. This provides an acceleration that will bring the airplane closer to the desired altitude. The altitude capture (ALT CAP) logic is required for vertical acceleration command limit control.

(13) Pitch Channel - Flight Level Change (Fig. 13)

(a) The flight level change mode is entered when the FLCH switch/light is pressed with the autopilot system in any mode except control inactive. Flight level change mode can exit into any autopilot mode except takeoff mode by pressing the appropriate mode button.

(b) If a different autothrottle mode is entered, the AFDS exits FLCH and enters the V/S mode. Two submodes, VNAV ARM and G/S ARM, are possible in FLCH mode. Automatic transition to G/S occurs if G/S is ARMED when G/S is captured. The FMA on the EADI will display G/S in the pitch mode armed annunciation section.

(c) In the FLCH mode, the AFDS commands the elevators to capture and hold the selected airspeed/mode, climb/decent rate controlled by the throttle. This occurs whether in autothrottle or manual operation and is automatically limited to maximum speed limits. The FMA shows SPD as the pitch mode engaged if no limits exist. If a maximum limit exists, the FMA shows FLAP LIM or SPD LIM.

(14) Pitch Channel - Vertical Navigation (VNAV) Mode (Fig. 14)

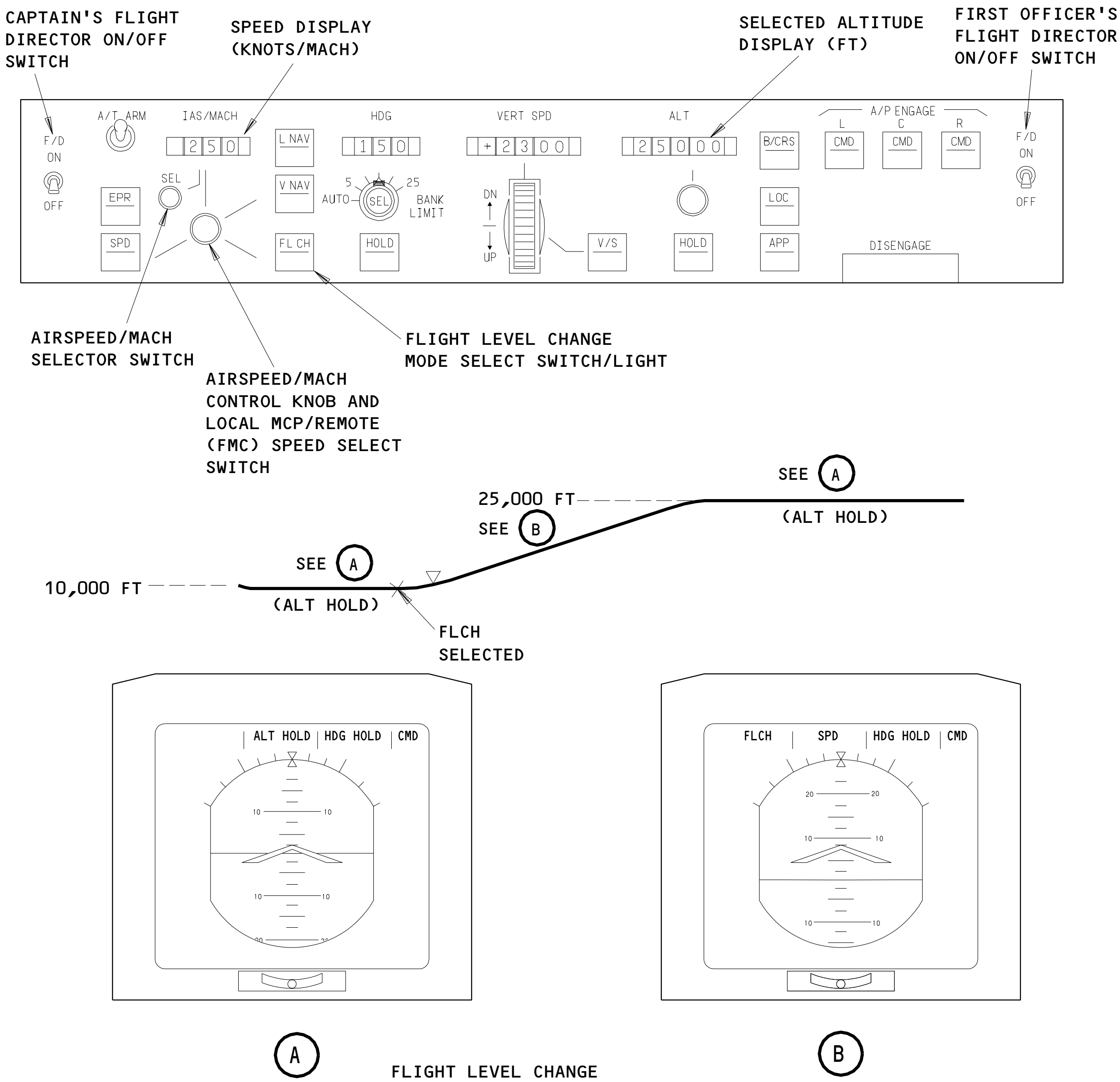
(a) Mode Control Panel

1) The VNAV mode is active in CMD ENG or F/D ON status. It is manually requested by pressing the VNAV switch/light on the MCP. A valid VNAV request is indicated by the illumination of the dot-bar matrix on the lower half of the switch/light. The flight mode annunciator on the EADI indicates VNAV as the pitch mode engaged. The IAS/MACH speed display and the VERT SPD display are both blank when VNAV is engaged. The control source for the VNAV mode is the Flight Management Computer (FMC).

2) VNAV cannot be entered directly from glideslope arm mode. The glideslope mode is a valid submode of VNAV. The flight mode annunciator FMA on the EADI shows G/S in the pitch mode armed section.

3) A valid VNAV mode switch request is required to engage the VNAV mode (except from control inactive and G/S active).

4) The VNAV mode is armed by pressing the VNAV switch/light on the MCP. The mode is engaged for control of the vertical axis. The current vertical axis mode is disengaged when the FMC outputs valid vertical steering commands. If VNAV steering signals become NCD (no computed data) after engagement, VNAV will disengage but remain ARMED. The VNAV mode is disengaged if the A/P is disconnected (disengaged) and both F/Ds are OFF, or if any other vertical axis mode is engaged.



FLIGHT LEVEL CHANGE

- USED TO CLIMB OR DESCEND TO NEW FLIGHT LEVEL
- COMBINED A/T AND FCC MODES
- THROTTLES ADVANCE/RETARD TO ACHIEVE CLIMB/DESCENT RATE BASED ON ALTITUDE CHANGE REQUESTED
- IF CREW OVERRIDES THROTTLES, A/T GOES TO THROTTLE HOLD
- SPEED THROUGH ELEVATOR CONTROLS SPEED TO SELECTED IAS/MACH
- AUTOMATIC TRANSITION TO VERTICAL POSITION AND A/T SPEED WHEN NEW ALTITUDE APPROACHED

Pitch Channel - Flight Level Change
Figure 13 (Sheet 1)

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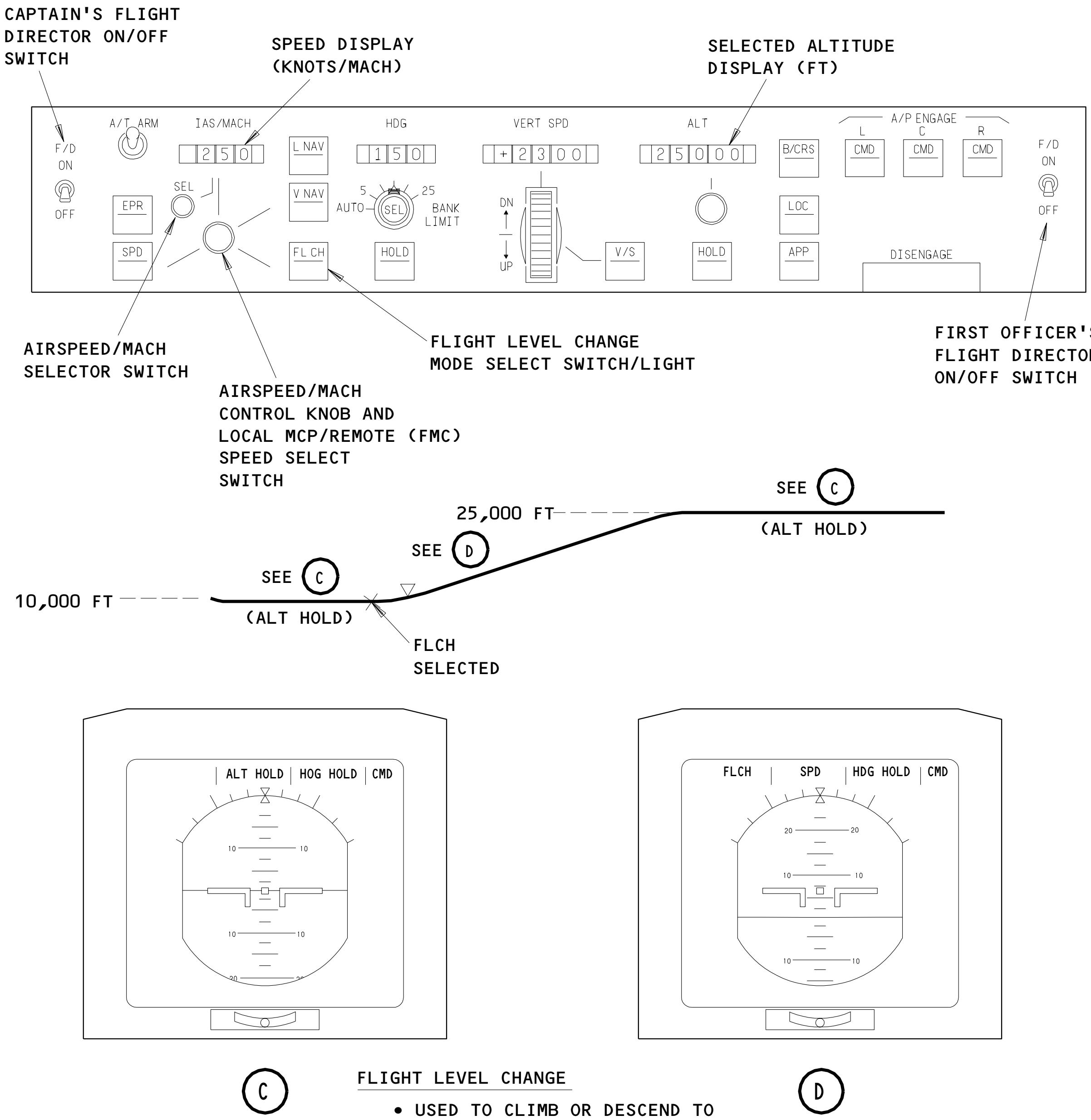

FLIGHT LEVEL CHANGE

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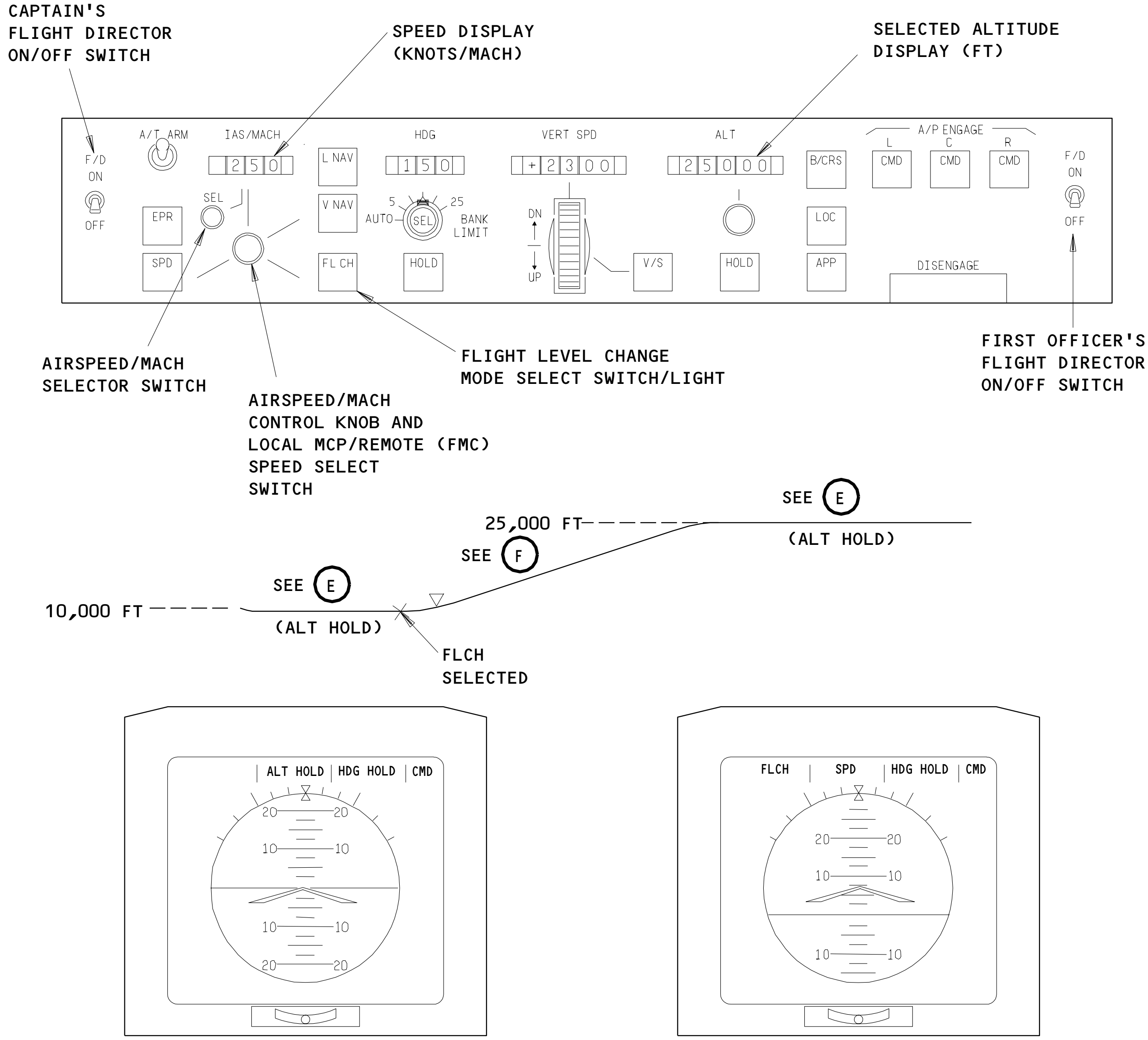
- COMBINED A/T AND FCC MODES
- THROTTLES ADVANCE/RETARD TO ACHIEVE CLIMB/DESCENT RATE BASED ON ALTITUDE CHANGE REQUESTED
- IF CREW OVERRIDES THROTTLES, A/T GOES TO THROTTLE HOLD
- SPEED THROUGH ELEVATOR CONTROLS SPEED TO SELECTED IAS/MACH
- AUTOMATIC TRANSITION TO VERTICAL POSITION AND A/T SPEED WHEN NEW ALTITUDE APPROACHED

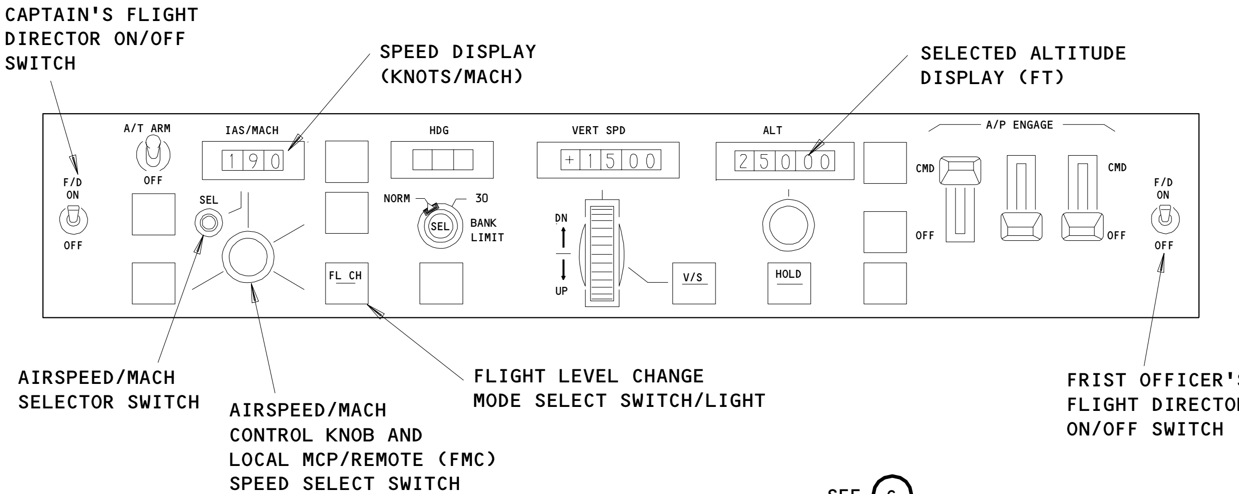
Pitch Channel - Flight Level Change
Figure 13 (Sheet 2)

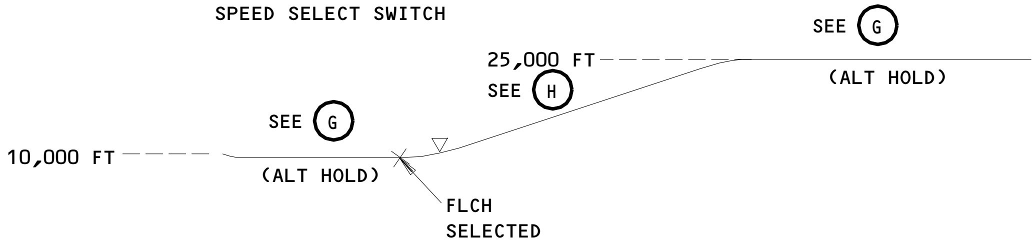


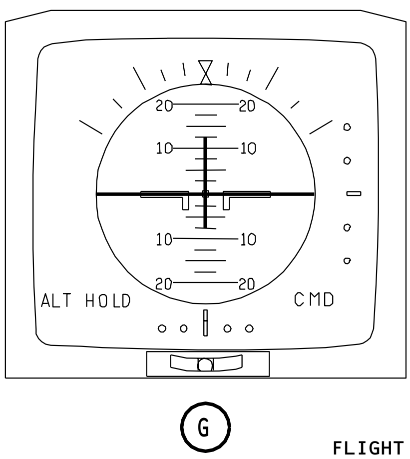
FLIGHT LEVEL CHANGE

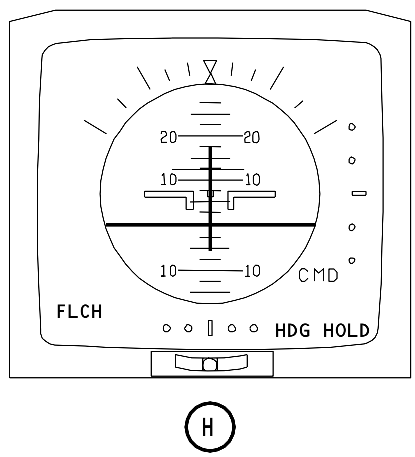
- USED TO CLIMB OR DESCEND TO NEW FLIGHT LEVEL
- COMBINED A/T AND FCC MODES
- THROTTLES ADVANCE/RETARD TO ACHIEVE CLIMB/DESCENT RATE BASED ON ALTITUDE CHANGE REQUESTED
- IF CREW OVERRIDES THROTTLES, A/T GOES TO THROTTLE HOLD
- SPEED THROUGH ELEVATOR CONTROLS SPEED TO SELECTED IAS/MACH
- AUTOMATIC TRANSITION TO VERTICAL POSITION AND A/T SPEED WHEN NEW ALTITUDE APPROACHED

Pitch Channel - Flight Level Change
Figure 13 (Sheet 3)





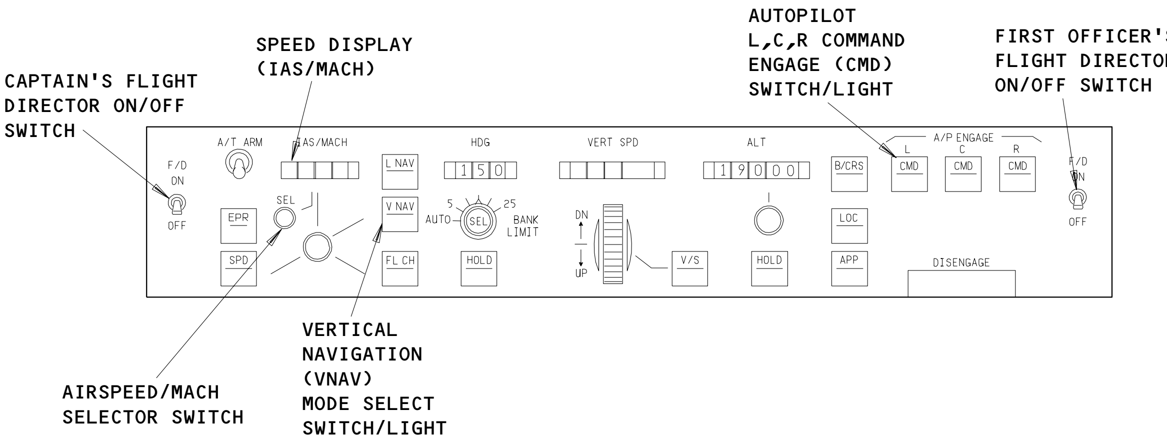


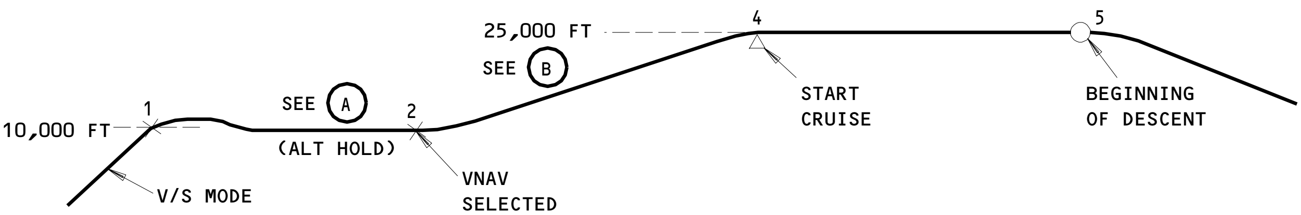


FLIGHT LEVEL CHANGE

- USED TO CLIMB OR DESCEND TO NEW FLIGHT LEVEL
- COMBINED A/T AND FCC MODES
- THROTTLES ADVANCE/RETARD TO ACHIEVE CLIMB/DESCENT RATE BASED ON ALTITUDE CHANGE REQUESTED
- IF CREW OVERRIDES THROTTLES, A/T GOES TO THROTTLE HOLD
- SPEED THROUGH ELEVATOR CONTROLS SPEED TO SELECTED IAS/MACH
- AUTOMATIC TRANSITION TO VERTICAL POSITION AND A/T SPEED WHEN NEW ALTITUDE APPROACHED

Pitch Channel - Flight Level Change
Figure 13 (Sheet 4)





FMA ON ADI INDICATES VNAV

- ALT SELECT DISPLAY MANUALLY SET FOR ALT OF 30,000 FT (HIGHER THAN FMC ROUTE)

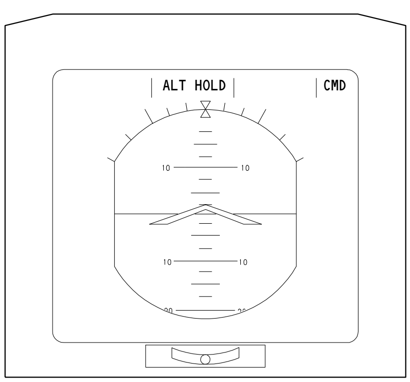
1 • ALT HOLD SW PUSHED AT 10,000 FT
  - V/S SWITCH DOT-BAR EXTINGUISHES
  - V/S DISPLAY BLANKS
  - HOLD SWITCH DOT-BAR ILLUMINATES
  - A/P LEVELS TO BARO-ALT AT TIME OF ALT HOLD
  - FMA ON EADI IN PITCH MODE ENGAGES SECTION INDICATES ALT HOLD

2 • VNAV SW PUSHED
  - HOLD SWITCH DOT-BAR EXTINGUISHES
  - VNAV SWITCH DOT-BAR ILLUMINATES
  - IAS/MACH DISPLAY BLANKS
  - FMA ON ADI INDICATES VNAV
  - AFDS UNDER FMC CONTROL

3 • FMC COMMANDS ELEVATORS FOR CLIMB TO 25,000 FT AT PRESCRIBED RATE

4 • FMC COMMANDS ELEVATORS FOR LEVEL CRUISE

5 • FMC COMMANDS ELEVATORS FOR DESCENT



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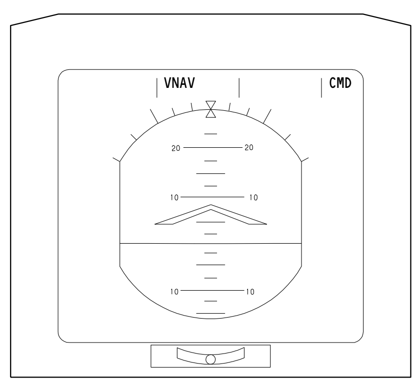

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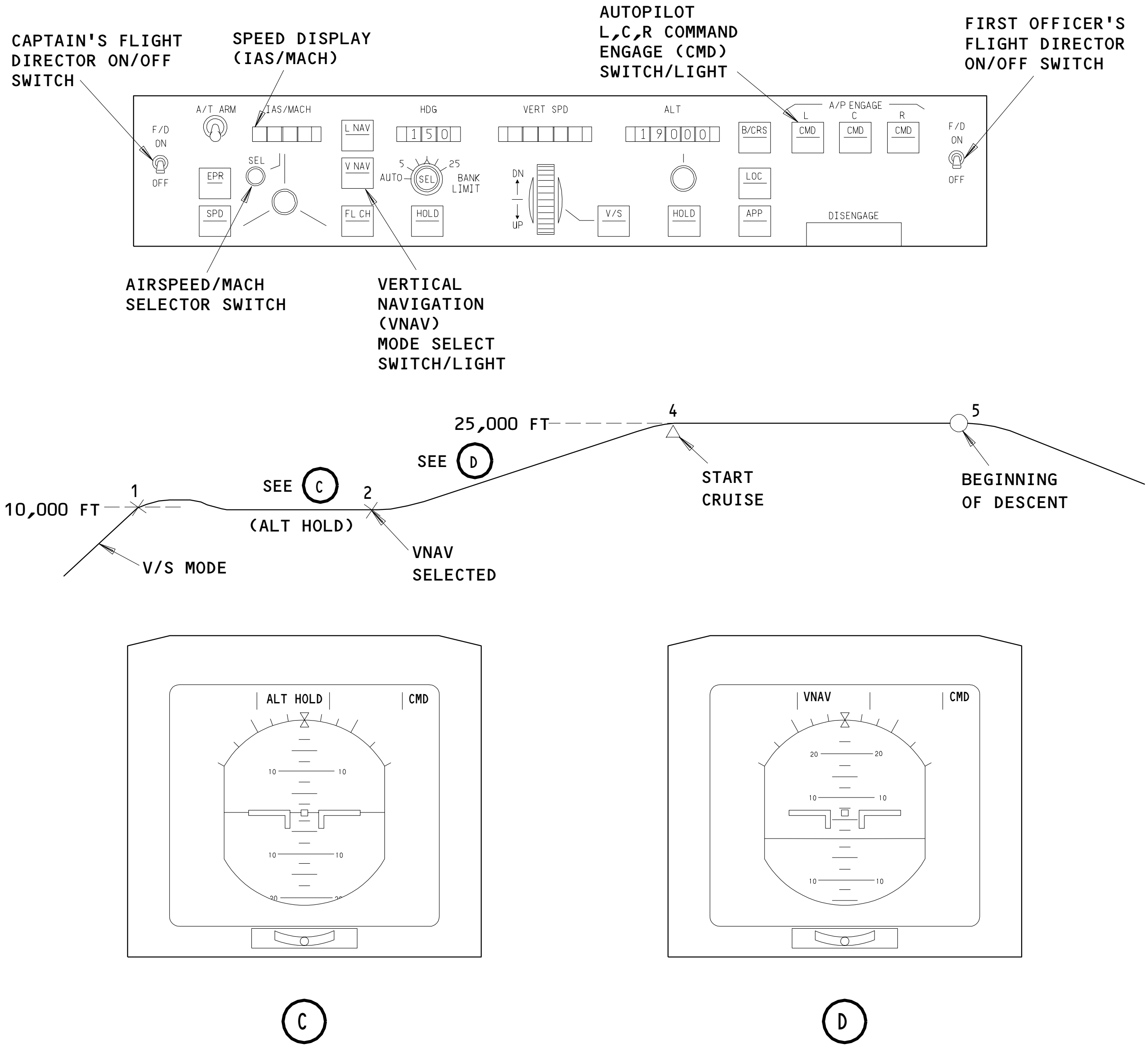
Pitch Channel - Vertical Navigation (VNAV) Mode
Figure 14 (Sheet 1)



- ALT SELECT DISPLAY MANUALLY SET FOR ALT OF 30,000 FT (HIGHER THAN FMC ROUTE)

1 • ALT HOLD SW PUSHED AT 10,000 FT
  - V/S SWITCH DOT-BAR EXTINGUISHES
  - V/S DISPLAY BLANKS
  - HOLD SWITCH DOT-BAR ILLUMINATES
  - A/P LEVELS TO BARO-ALT AT TIME OF ALT HOLD
  - FMA ON EADI IN PITCH MODE ENGAGES SECTION INDICATES ALT HOLD

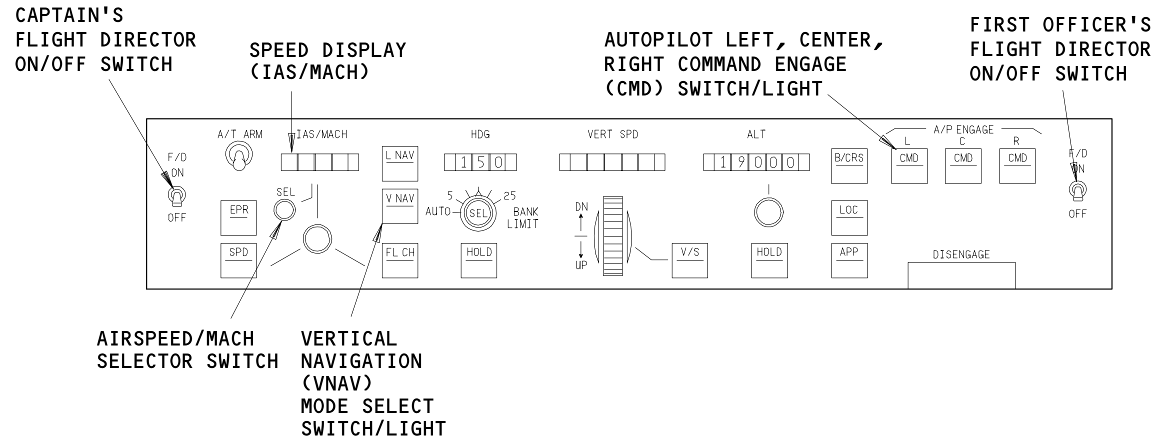
2 • VNAV SW PUSHED
  - HOLD SWITCH DOT-BAR EXTINGUISHES
  - VNAV SWITCH DOT-BAR ILLUMINATES
  - IAS/MACH DISPLAY BLANKS
  - FMA ON ADI INDICATES VNAV
  - AFDS UNDER FMC CONTROL

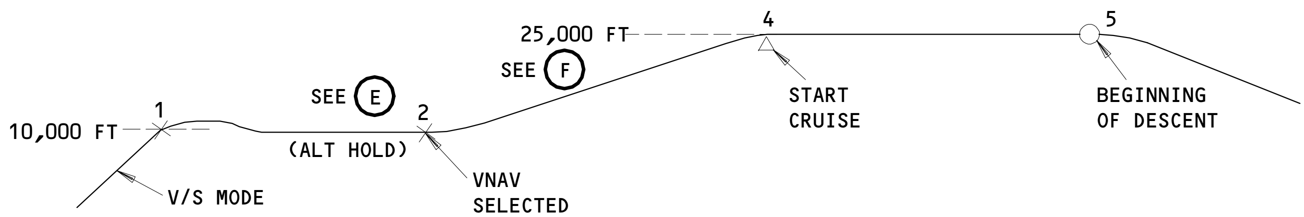
3 • FMC COMMANDS ELEVATORS FOR CLIMB TO 25,000 FT AT PRESCRIBED RATE

4 • FMC COMMANDS ELEVATORS FOR LEVEL CRUISE

5 • FMC COMMANDS ELEVATORS FOR DESCENT

Pitch Channel - Vertical Navigation (VNAV) Mode
Figure 14 (Sheet 2)





- ALT SELECT DISPLAY MANUALY SET FOR ALT OF 30,000 FT (HIGHER THAN FMC ROUTE)

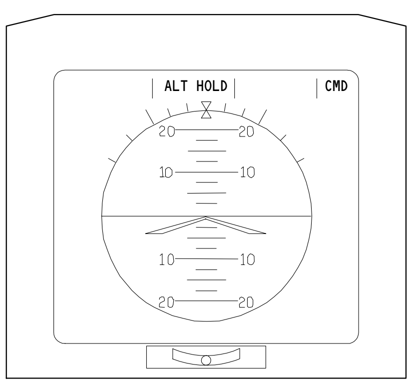
1 • ALT HOLD SW PUSHED AT 10,000 FT
- V/S SWITCH DOT-BAR EXTINGUISHES
- V/S DISPLAY BLANKS
- HOLD SWITCH DOT-BAR ILLUMINATES
- A/P LEVELS TO BARO-ALT AT TIME OF ALT HOLD
- FMA ON EADI IN PITCH MODE ENGAGES SECTION INDICATES ALT HOLD

2 • VNAV SW PUSHED
- HOLD SWITCH DOT-BAR EXTINGUISHES
- VNAV SWITCH DOT-BAR ILLUMINATES
- IAS/MACH DISPLAY BLANKS
- FMA ON ADI INDICATES VNAV
- AFDS UNDER FMC CONTROL

3 • FMC COMMANDS ELEVATORS FOR CLIMB TO 25,000 FT AT PRESCRIBED RATE

4 • FMC COMMANDS ELEVATORS FOR LEVEL CRUISE

5 • FMC COMANDS ELEVATORS FOR DESCENT



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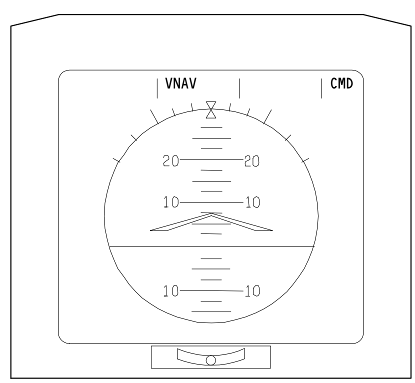

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Pitch Channel - Vertical Navigation (VNAV) Mode
Figure 14 (Sheet 3)

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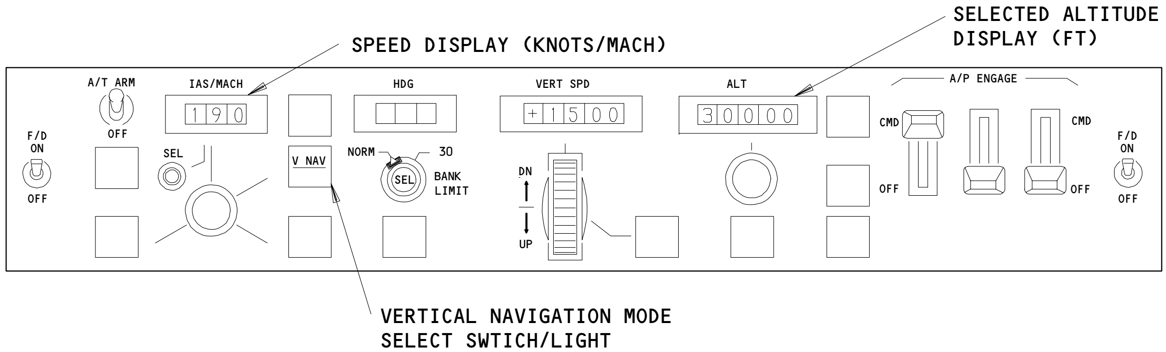
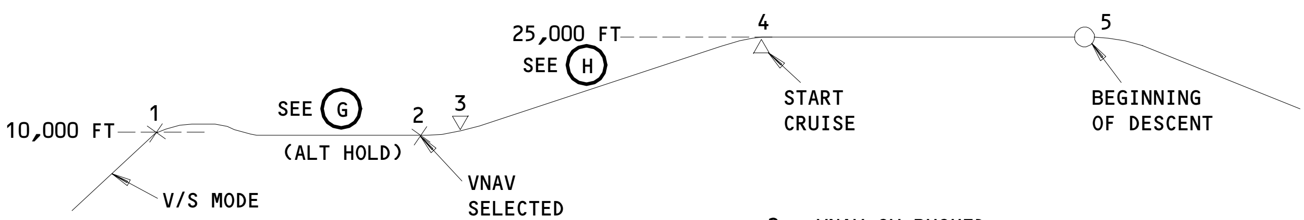




- ALT SELECT DISPLAY MANUALLY SET FOR ALT OF 30,000 FT (HIGHER THAN FMC ROUTE)

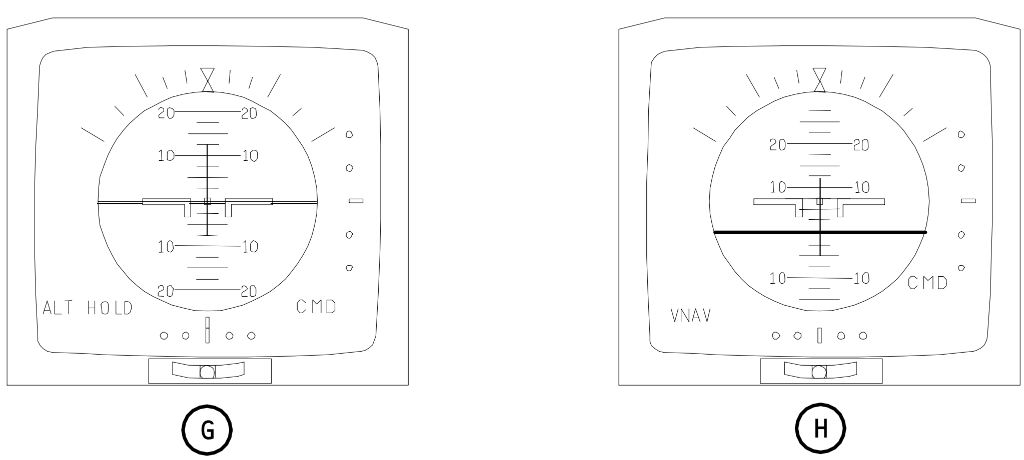
1 • ALT HOLD SW PUSHED AT 10,000 FT
  - V/S SWITCH DOT-BAR EXTINGUISHES
  - V/S DISPLAY BLANKS
  - HOLD SWITCH DOT-BAR ILLUMINATES
  - A/P LEVELS TO BARO-ALT AT TIME OF ALT HOLD
  - FMA ON EADI IN PITCH MODE ENGAGES SECTION INDICATES ALT HOLD

2 • VNAV SW PUSHED
  - HOLD SWITCH DOT-BAR EXTINGUISHES
  - VNAV SWITCH DOT-BAR ILLUMINATES
  - IAS/MACH DISPLAY BLANKS
  - FMA ON EADI INDICATES VNAV
  - AFDS UNDER FMC CONTROL

3 • FMC COMMANDS ELEVATORS FOR CLIMB TO 25,000 FT AT PRESCRIBED RATE

4 • FMC COMMANDS ELEVATORS FOR LEVEL CRUISE

5 • FMC COMANDS ELEVATORS FOR DESCENT



Pitch Channel - Vertical Navigation (VNAV) Mode
Figure 14 (Sheet 4)

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(15) VNAV Control Logic (Fig. 10)
  (a) Vertical Navigation Mode Engage
    1) The VNAV mode engage logic is generated by three input conditions: F/D not disengaged, VNAV mode active, and VNAV engaged valid. The VNAV mode active means that the VNAV switch/light dot-bar matrix is lighted and the FMC is sending valid data.
  (b) TAS Inverse Gain Schedule
    1) The TAS inverse gain schedule control amplitude is inversely proportional to TAS and is used for gain schedule control.
  (c) Transition Control Logic From VNAV to ALT HOLD
    1) The altitude hold auto engage logic is produced, for one iteration when it is enabled by VNAV mode engage and the following three conditions exist:
      a) The airplane is above or below the MCP selected altitude.
      b) The FMC is in the process of having the target altitude changed.
      c) The FMC is notifying the AFDS that the next target altitude is farther away from the MCP selected altitude.
  (d) Transition Control Logic From VNAV to ALT SELECT
    1) The altitude select auto engage logic is produced for one sampling when it is enabled by VNAV mode engage and the following four conditions exist:
      a) The airplane is above or below the MCP selected altitude.
      b) Altitude capture criteria are met.
      c) The FMC notifies the AFDS that the next target altitude is such that the airplane will pass through the MCP selected altitude.
      d) The FMC target altitude is not equal to the MCP altitude.
  (e) VNAV Override
    1) If altitude hold auto engage and altitude select auto engage are generated, VNAV override is generated which automatically disengages the VNAV mode and engages ALT HOLD or ALT SEL.

(16) VNAV Control Signal Flow and Monitoring (Fig. 10)

(a) Output Requirements to the Pitch Attitude Integrator

1) Inputs to the VNAV processor are VNAV command and VNAV proportional limit. The VNAV command VNAVS is the VNAV steering command to be processed under VNSLIM limitation to provide TCMD. The VNSLIM is developed from a 4.83 ft/sec^2 (0.15g) constant acceleration command and controlled by inverse TAS.

(17) Pitch Channel - Approach (Glideslope) (Fig. 15)

(a) The approach mode is armed by pressing the APP switch/light. This also arms the glideslope. Glideslope will engage when the glideslope capture criteria are satisfied. Command engage configuration can be single channel or multi-channel. glideslope is a valid submode in single or multi-channel engage. Flare is a valid submode in multi-channel engage only.

(b) Manual entry from any permissible mode is possible when the APP switch/light dot bar matrix is lighted. If the switch/light is pressed again, prior to LOC or G/S capture, the APP mode is disarmed.

(c) Automatic exit from the APP mode can occur three ways: selecting a different roll mode after LOC capture but prior to glideslope capture; selecting a different pitch mode after glideslope capture but before LOC capture; and when go-around is selected. There is a one-sampling time interruption of the APP mode if only the F/D is engaged and one channel is initially engaged in CMD.

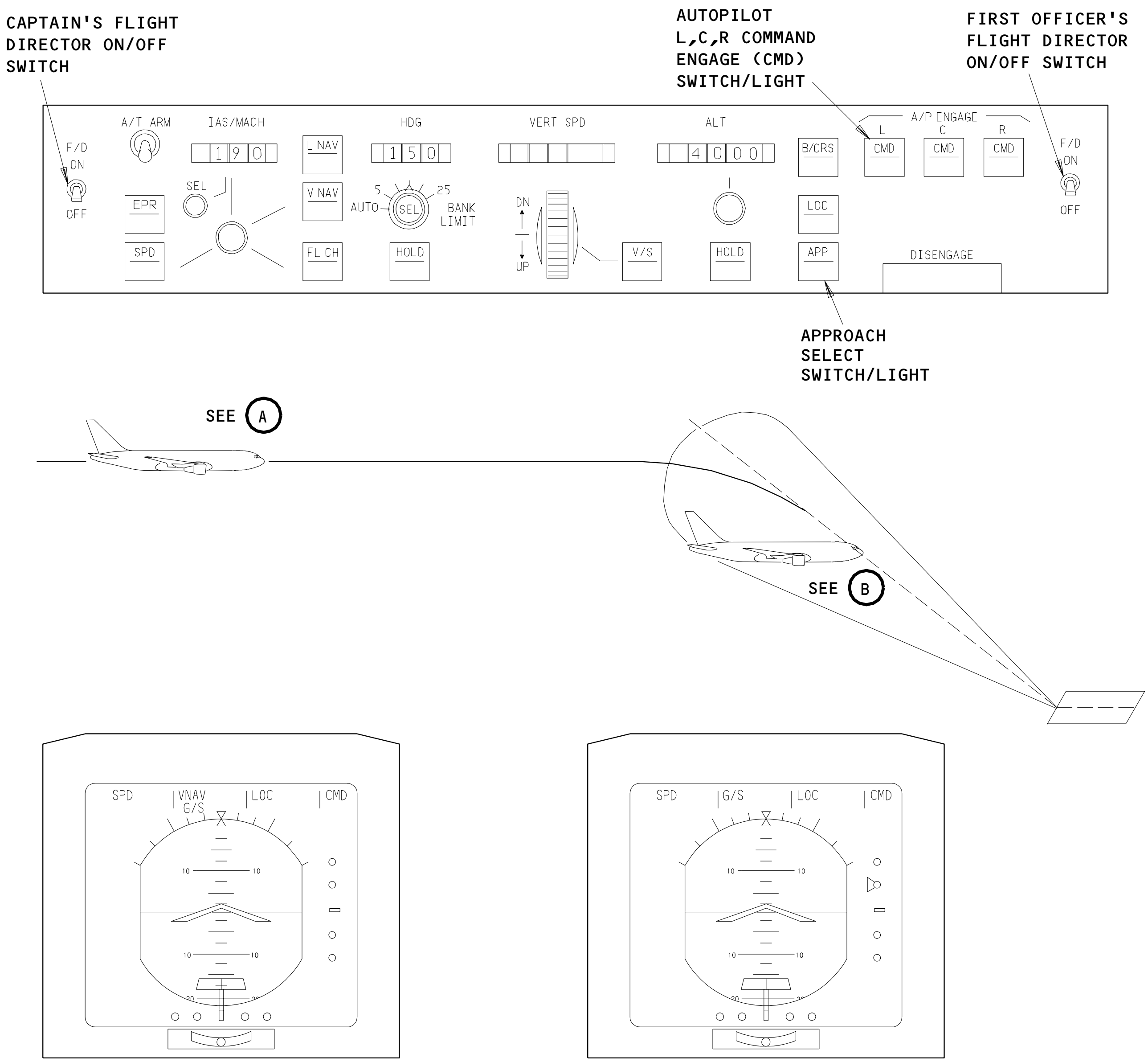
(d) Two submodes are permissible in APP ARM: glideslope ARM and glideslope engaged. The flight mode annunciator on the EADI shows G/S (glideslope) as the pitch mode armed and pitch mode engaged respectively.

(18) Glideslope Command Logic

(a) Glideslope Capture

1) Glideslope capture is armed by APP mode ARM. A bias is added to gear deviation altitude to reference the deviation back to the glide slope antenna. When deviation and rate meet the capture criteria and the airplane is approaching the beam, a capture signal is generated.

2) If deviation and rate reach minimum values without capture criteria being met, capture is forced by a second capture signal. The glideslope engage logic is latched until APP ARM is removed, or Flare is engaged, or until the airplane is on the ground.



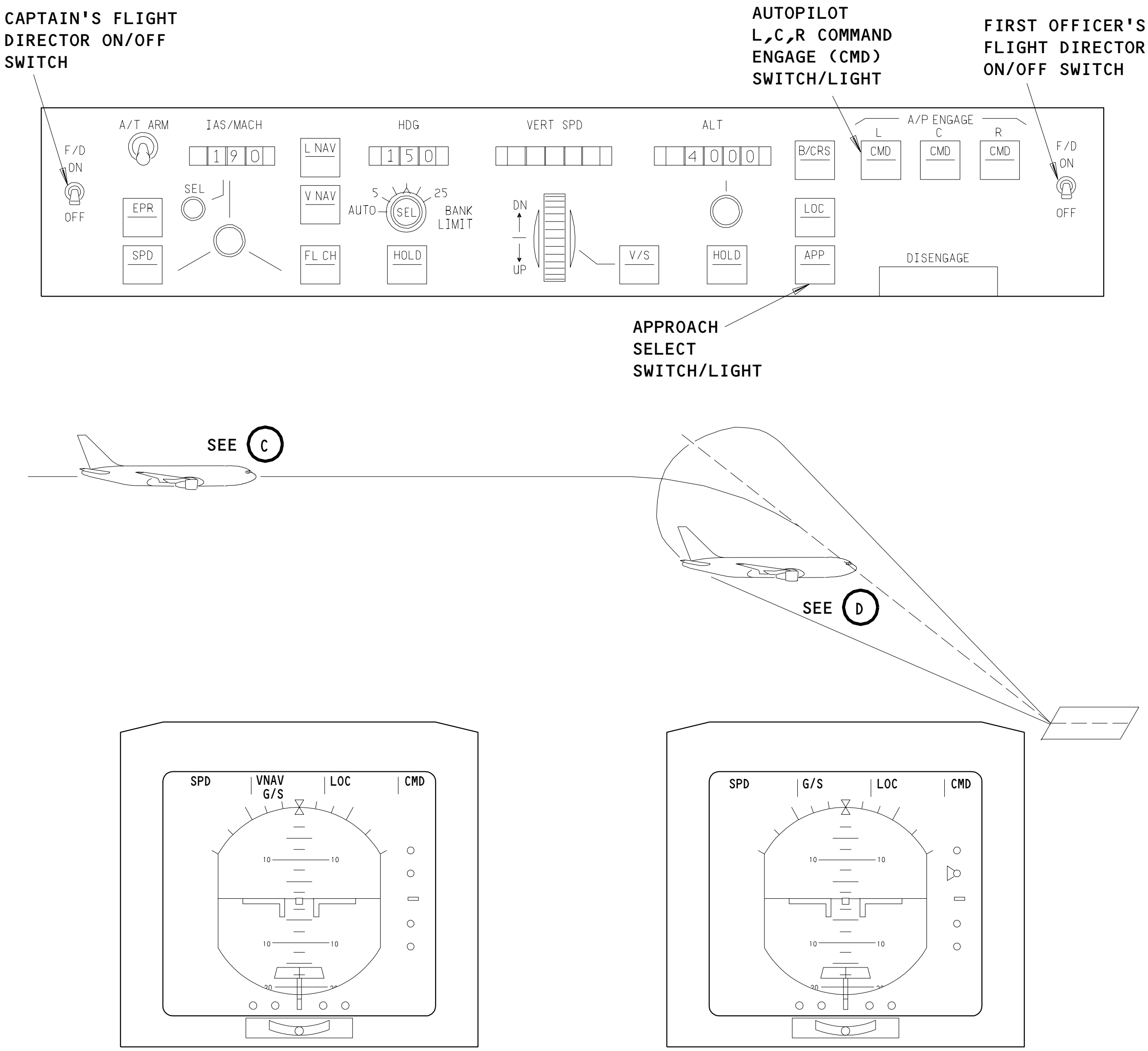
- GLIDE SLOPE ARMED
- APP SWITCH PRESSED
  - APP SWITCH DOT-BAR ILLUMINATED
  - FMA ON EADI INDICATES G/A ARMED
- POSSIBLE ACTIVE MODES
  - VNAV
  - V/S
  - FLCH
  - ALT HOLD

A

- G/S CAPTURED
- CAP CRITERIA BASED ON
  - DISTANCE FROM GLIDE PATH
  - RATE OF APPROACH TO GLIDE PATH
- FMA ON EADI INDICATES G/S IN PITCH MODE ENGAGED SECTION

B

Pitch Channel - Approach Mode (Glide slope)
Figure 15 (Sheet 1)



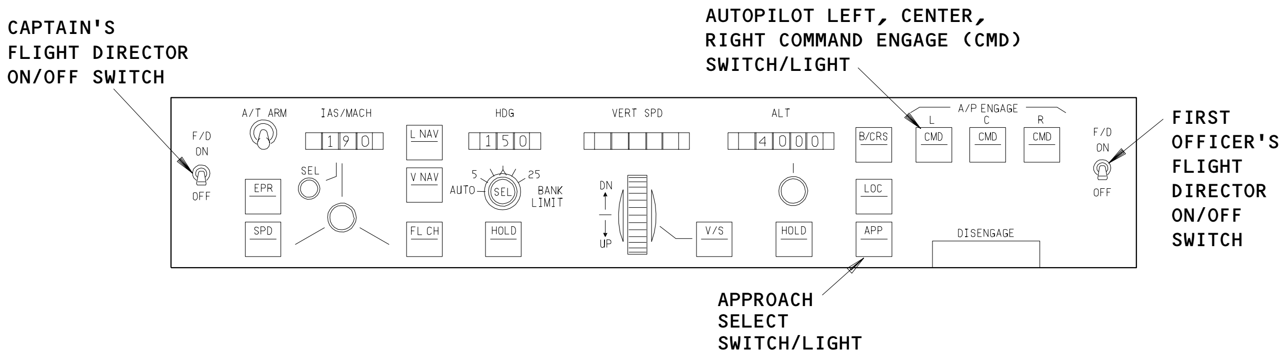
- GLIDE SLOPE ARMED
- APP SWITCH PRESSED
  - APP SWITCH DOT-BAR ILLUMINATED
  - FMA ON EADI INDICATES G/A ARMED
- POSSIBLE ACTIVE MODES
  - VNAV
  - V/S
  - FLCH
  - ALT HOLD

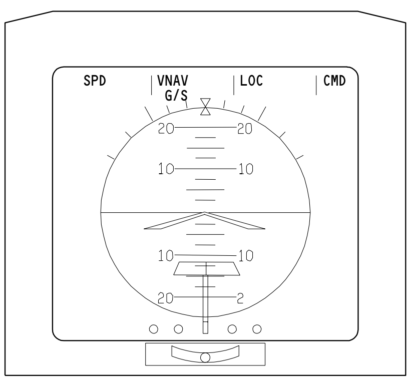
C

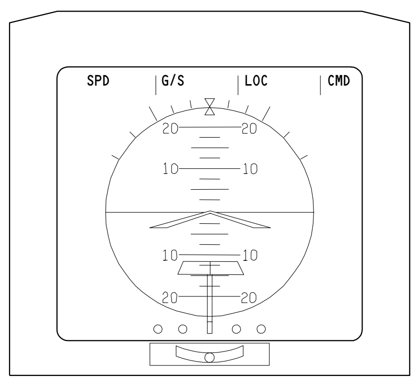
- G/S CAPTURED
- CAP CRITERIA BASED ON
  - DISTANCE FROM GLIDE PATH
  - RATE OF APPROACH TO GLIDE PATH
- FMA ON EADI INDICATES G/S IN PITCH MODE ENGAGED SECTION

D

Pitch Channel - Approach Mode (Glide slope)
Figure 15 (Sheet 2)







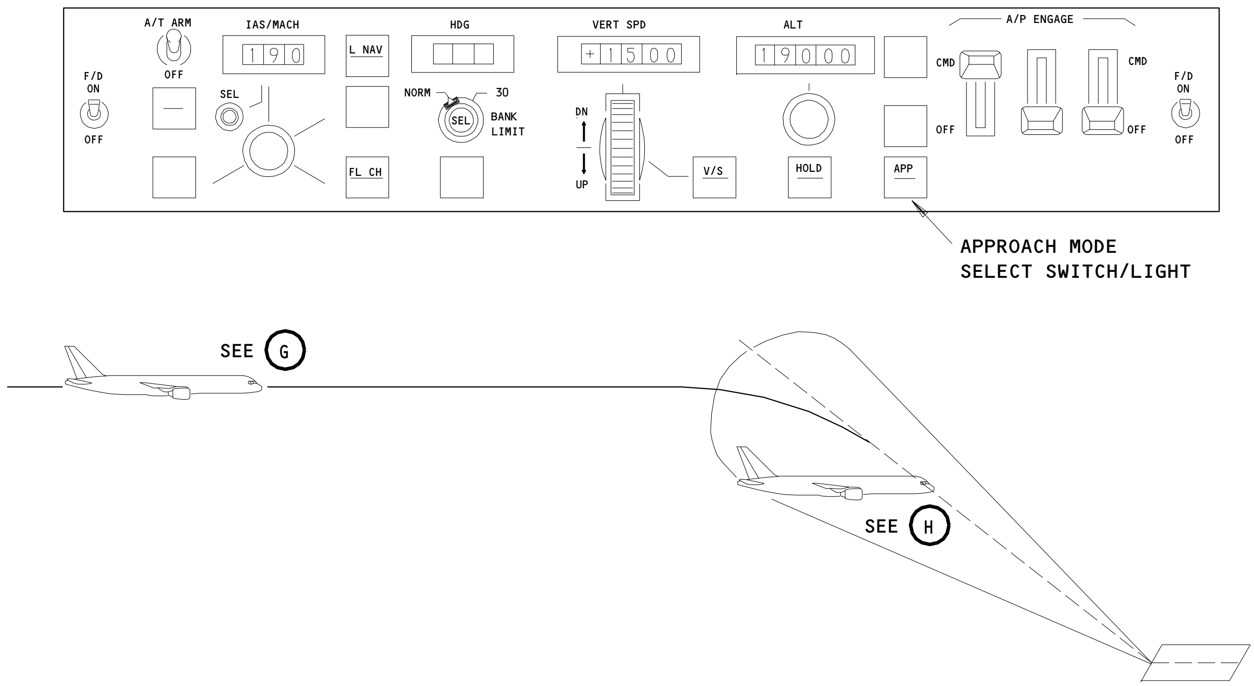
- GLIDE SLOPE ARMED
- APP SWITCH PRESSED
  - APP SWITCH DOT-BAR ILLUMINATED
  - FMA ON EADI INDICATES G/A ARMED
- POSSIBLE ACTIVE MODES
  - VNAV
  - V/S
  - FLCH
  - ALT HOLD

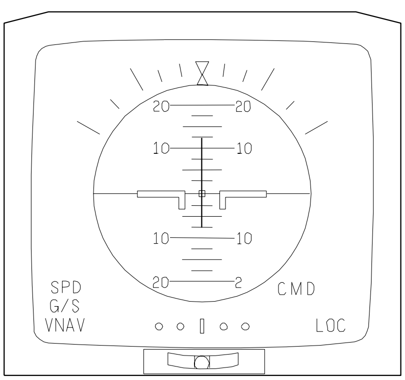
E

- G/S CAPTURED
- CAP CRITERIA BASED ON
  - DISTANCE FROM GLIDE PATH
  - RATE OF APPROACH TO GLIDE PATH
- FMA ON EADI INDICATES G/S IN PITCH MODE ENGAGED SECTION

F

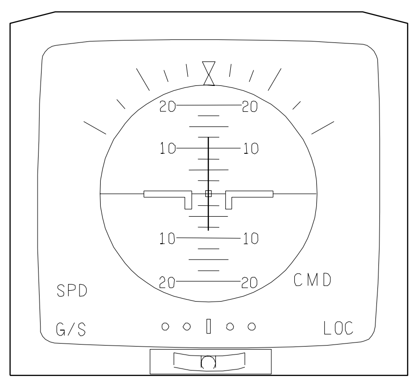
Pitch Channel - Approach Mode (Glide slope)
Figure 15 (Sheet 3)





- GLIDE SLOPE ARMED
- APP SWITCH PRESSED
  - APP SWITCH DOT-BAR ILLUMINATED
  - FMA ON EADI INDICATES G/S ARMED
- POSSIBLE ACTIVE MODES
  - VNAV
  - V/S
  - FLCH
  - ALT HOLD

G



- G/S CAPTURED
- CAP CRITERIA BASED ON
  - DISTANCE FROM GLIDE PATH
  - RATE OF APPROACH TO GLIDE PATH
- FMA ON EADI INDICATES G/S IN PITCH MODE ENGAGED SECTION

H

Pitch Channel - Approach Mode (Glide Slope)
Figure 15 (Sheet 4)

(b) Vertical Position Logic
1) The vertical position engage logic goes invalid for one iteration after a vertical position mode change. This allows synchronization to new values. Vertical position engage transition logic goes valid for one frame during the next iteration time. This logic is used by acceleration forecast circuits for acceleration limit determination. One time frame later, vertical position engage delayed logic becomes valid and enables the acceleration limit reset.

(19) Approach Control Signal Flow and Monitoring
(a) Signal Selection
1) The vertical position approach logic selects filtered glideslope deviation altitude and filtered glideslope deviation rate inputs to command processing. Since the deviation altitude is the distance from the main landing gear to the glideslope beam centerline, a bias of 19 feet is added to reference it to the glideslope antenna.
2) The vertical position approach logic selects the gains for glideslope operation. The position and rate errors are combined to form the integral and proportional commands.

(b) Acceleration Forecast
1) At glideslope capture, the capture region is determined by the relationship between deviation altitude and deviation rate. If no acceleration limit is required, the output is a no acceleration limit region computation pulse. If some acceleration limit is required, acceleration limit region computation latch is set. This is reset when exceeding the acceleration limit is no longer forecast. The overshoot limit region computation latch is set if overshoot is forecast. When the airplane is parallel to the glideslope beam centerline, overshoot limit region computation is reset.

(c) Vertical Acceleration Command Limit
1) GUI 115;

A natural frequency (time constant) is calculated for the command processor at capture if no limit is imposed or when an imposed limit is removed. It is held, limited and sent to command processing.

2) When a limit is imposed, the acceleration commanded by the command processor will be set at this value. For glideslope it is a fixed value of 2.5 ft/sec/sec.
3) If no acceleration limits are imposed, the natural frequency sets the gains in the second order filter. The commanded deviation and rate decay from the initial values at a time constant set by the natural frequency.
4) If acceleration limits are imposed, vertical position limit logic operates and the commanded rate is reduced at the fixed 2.5 ft/sec/sec acceleration limit. When the limit is removed, the commanded position and rate decay to zero with the calculated time constant.
5) Vertical position error and vertical position rate error are the differences between commanded and actual position and rate respectively.

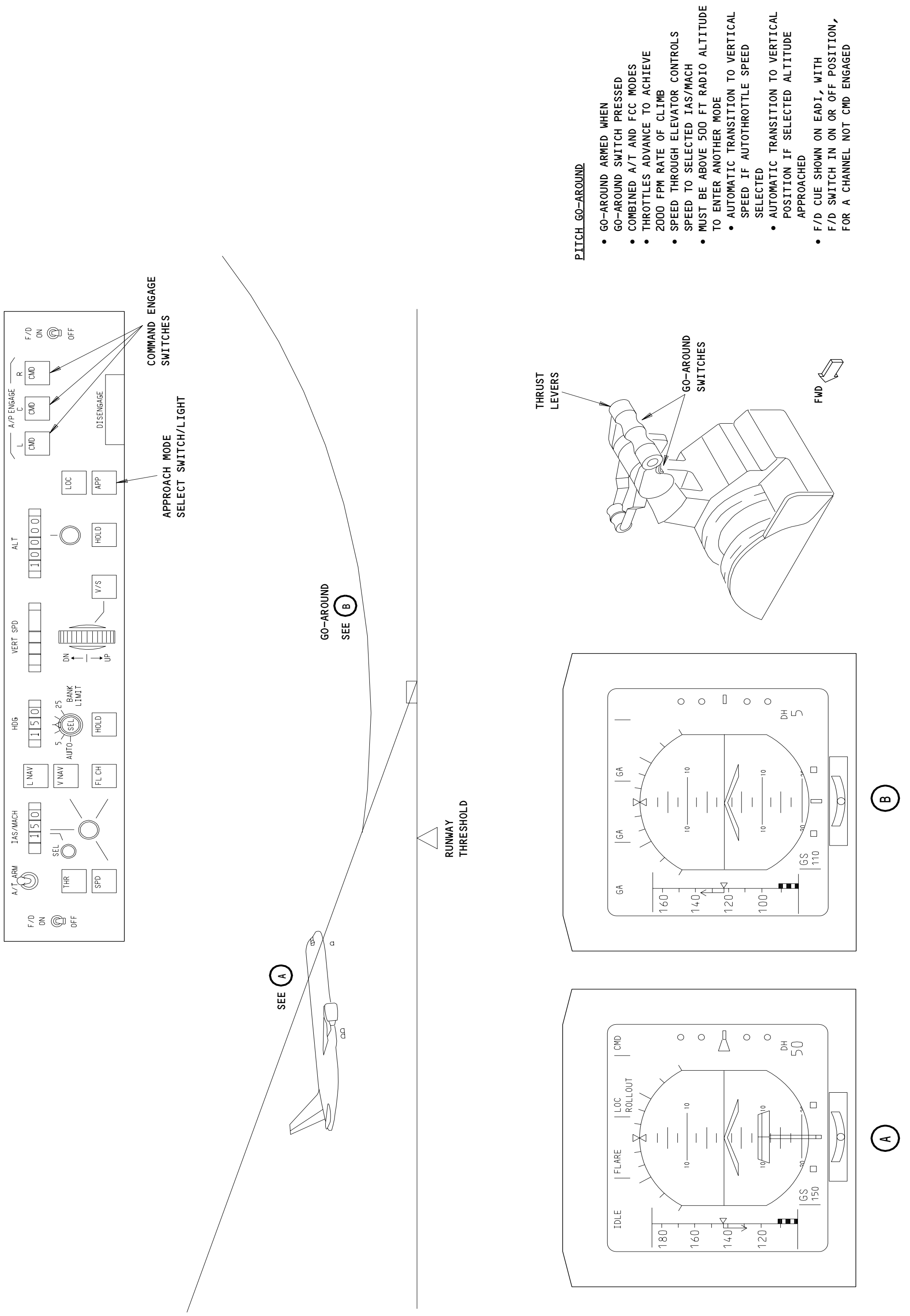
(20) Pitch Channel - Go-Around Mode (Fig. 16)

(a) Transition to Go-Around mode is possible from all modes except control inactive and takeoff. Go-around mode can exit into all modes except takeoff and glideslope. The VNAV ARM submode is permissible for go-around. A valid go-around switch request is required to enter go-around mode.

(b) On entry into go-around mode the elevators are commanded to capture and hold selected airspeed and the autothrottle (if engaged) commands a 2000 FPM climb rate. A pitch up signal to the elevators is provided to compensate for the increased thrust. The flight mode annunciator on the EADI displays GA as the go-around mode is engaged.

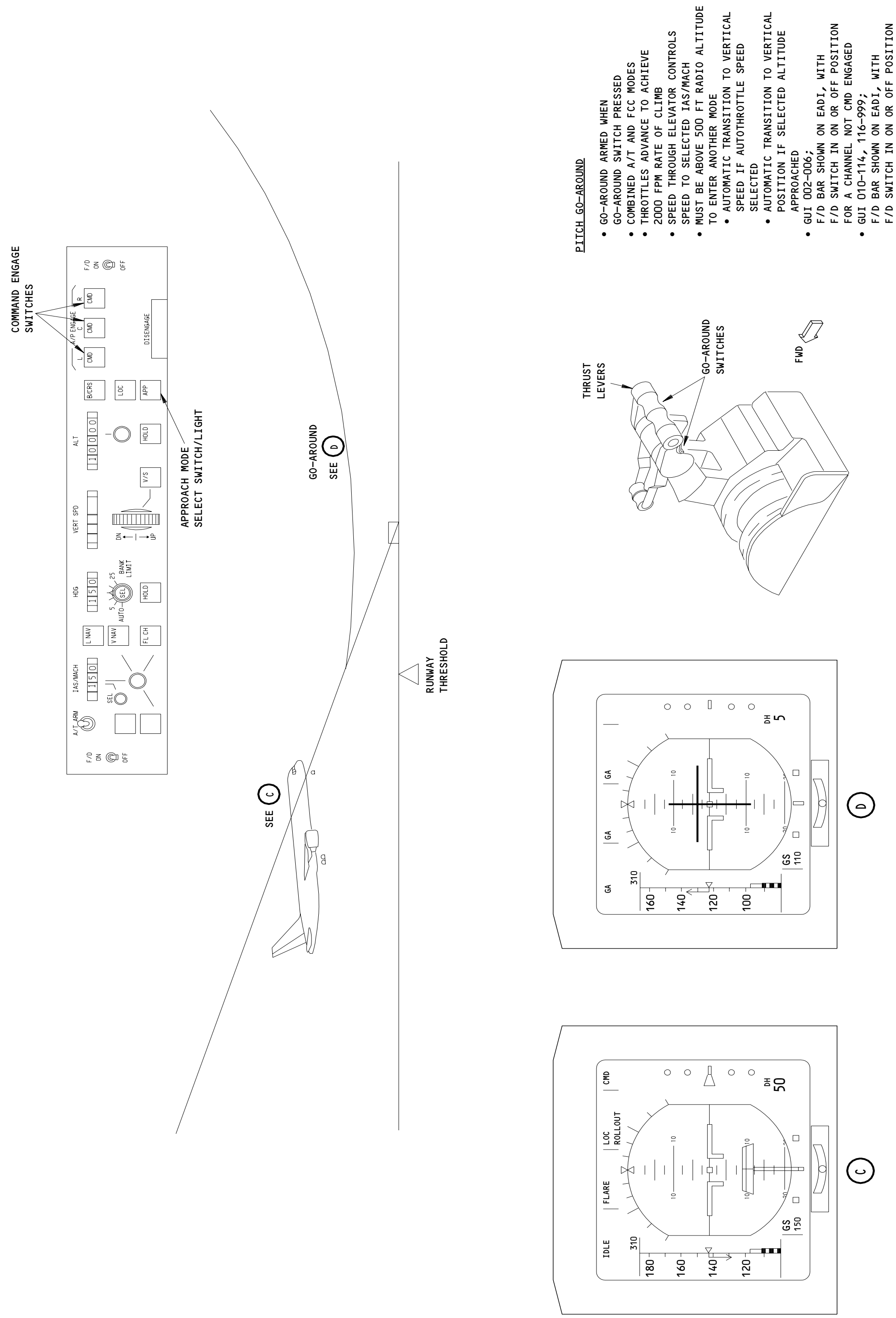
(21) Pitch Channel - Autoland (Fig. 17)

(a) Multichannel engage is initiated when one channel is engaged in CMD and one or more A/P channels are manually placed in CMD with APP selected. The switches remain in CMD only if APP is selected. The Autoland Status Annunciator (ASA) shows autoland capability (Ref 22-14-00).

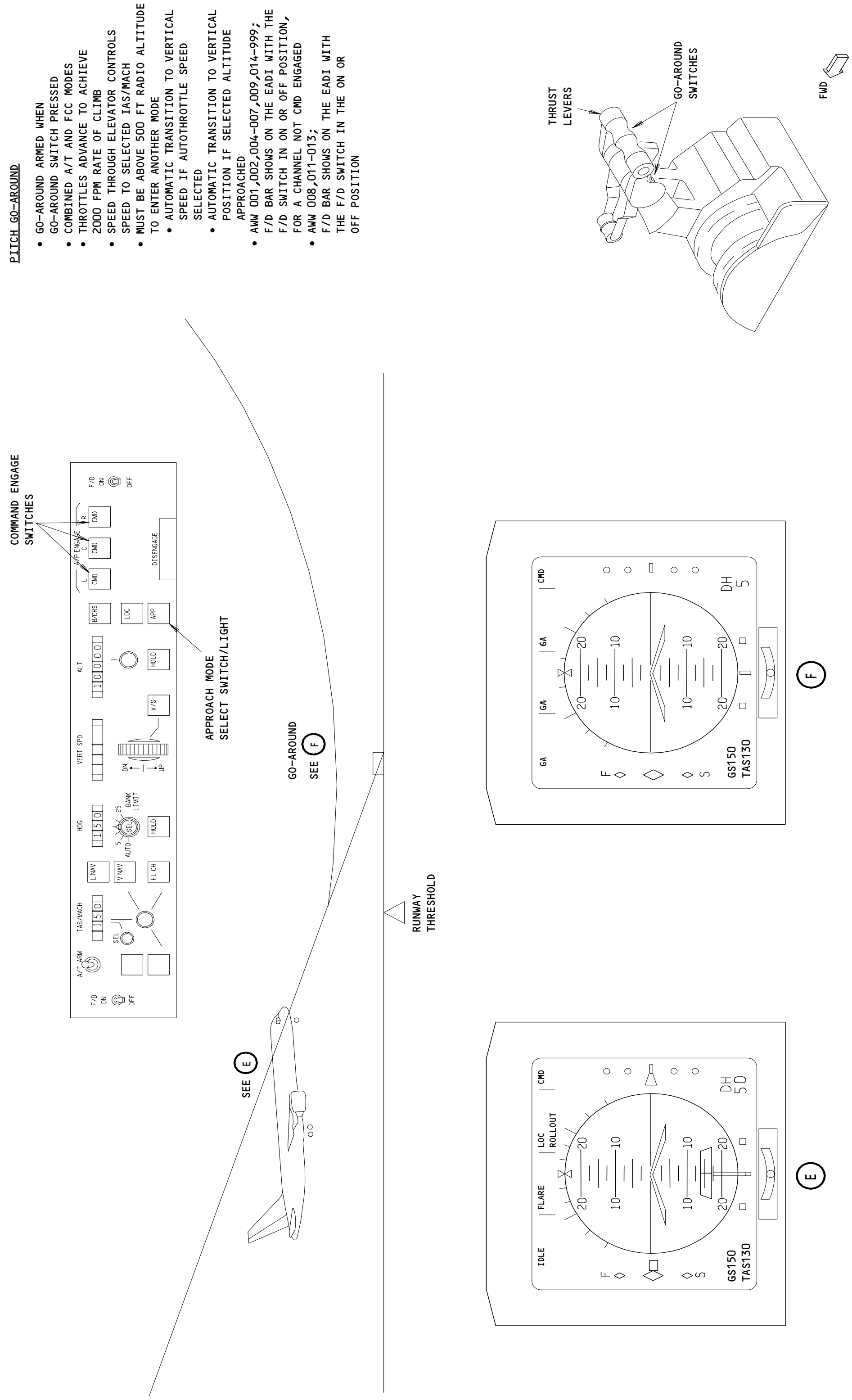


Pitch Channel - Go-Around Mode
Figure 16 (Sheet 1)

835329

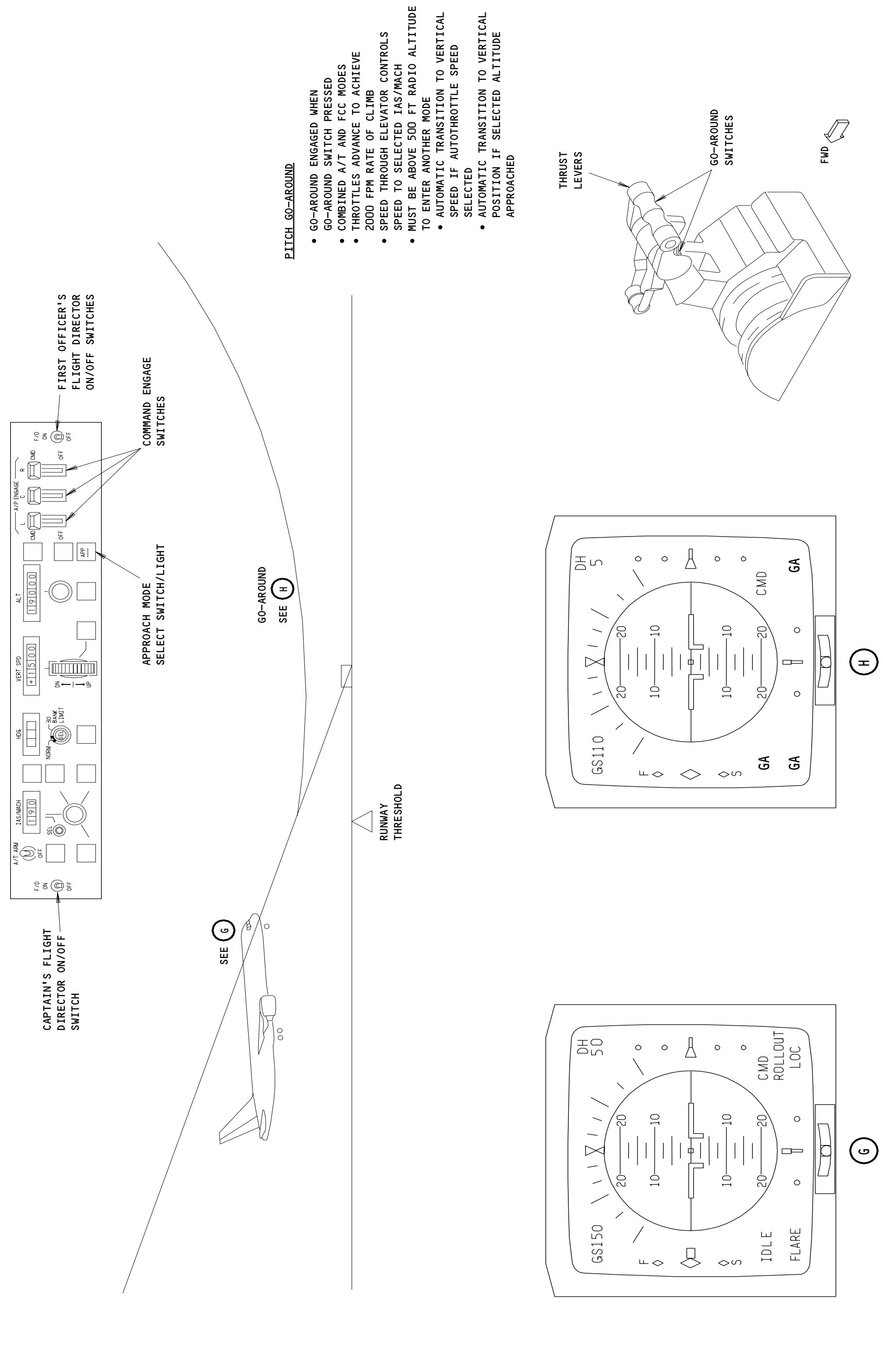


Pitch Channel - Go-Around Mode
Figure 16 (Sheet 2)



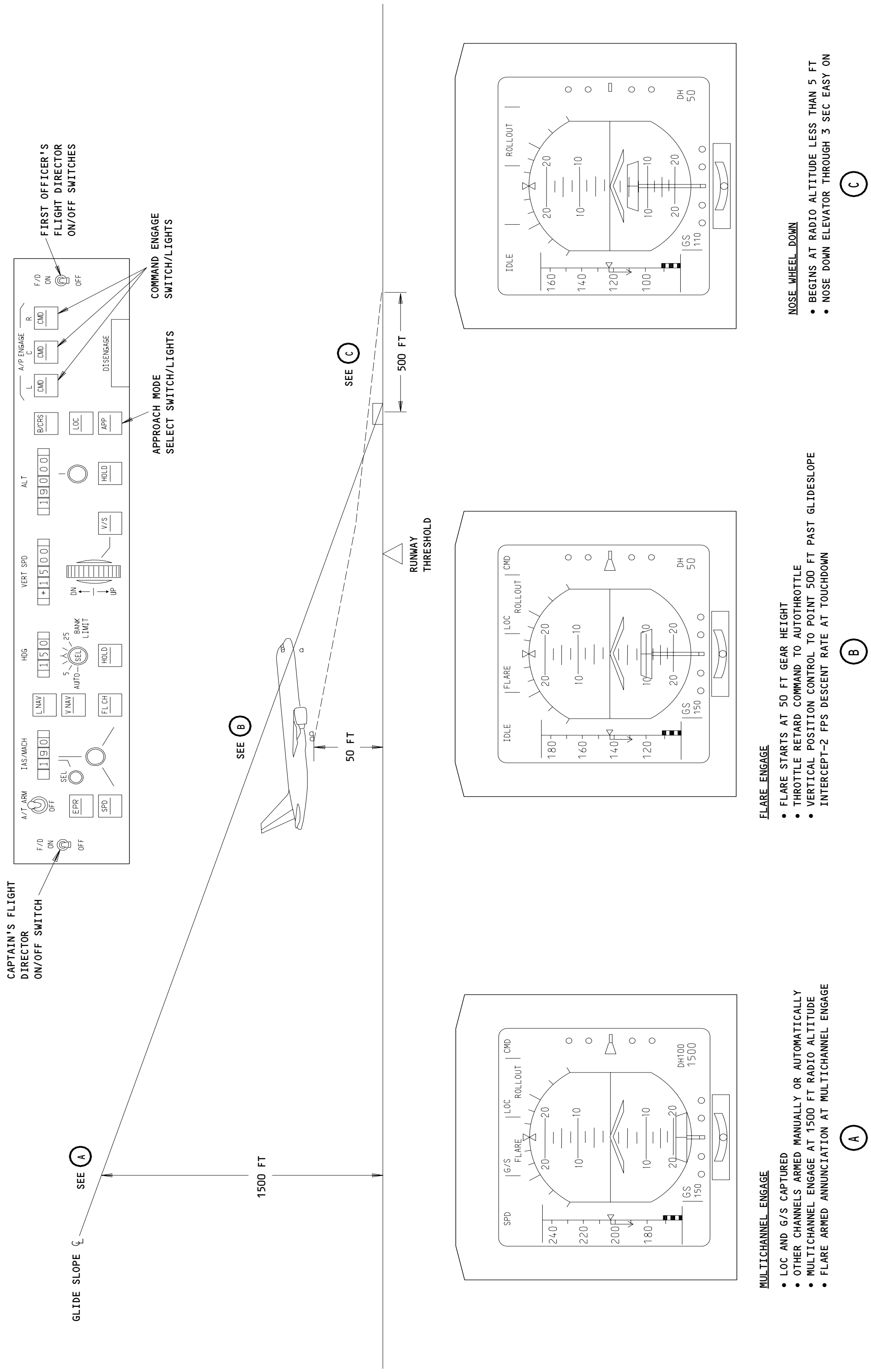
Pitch Channel - Go-Around Mode
Figure 16 (Sheet 3)

C33489



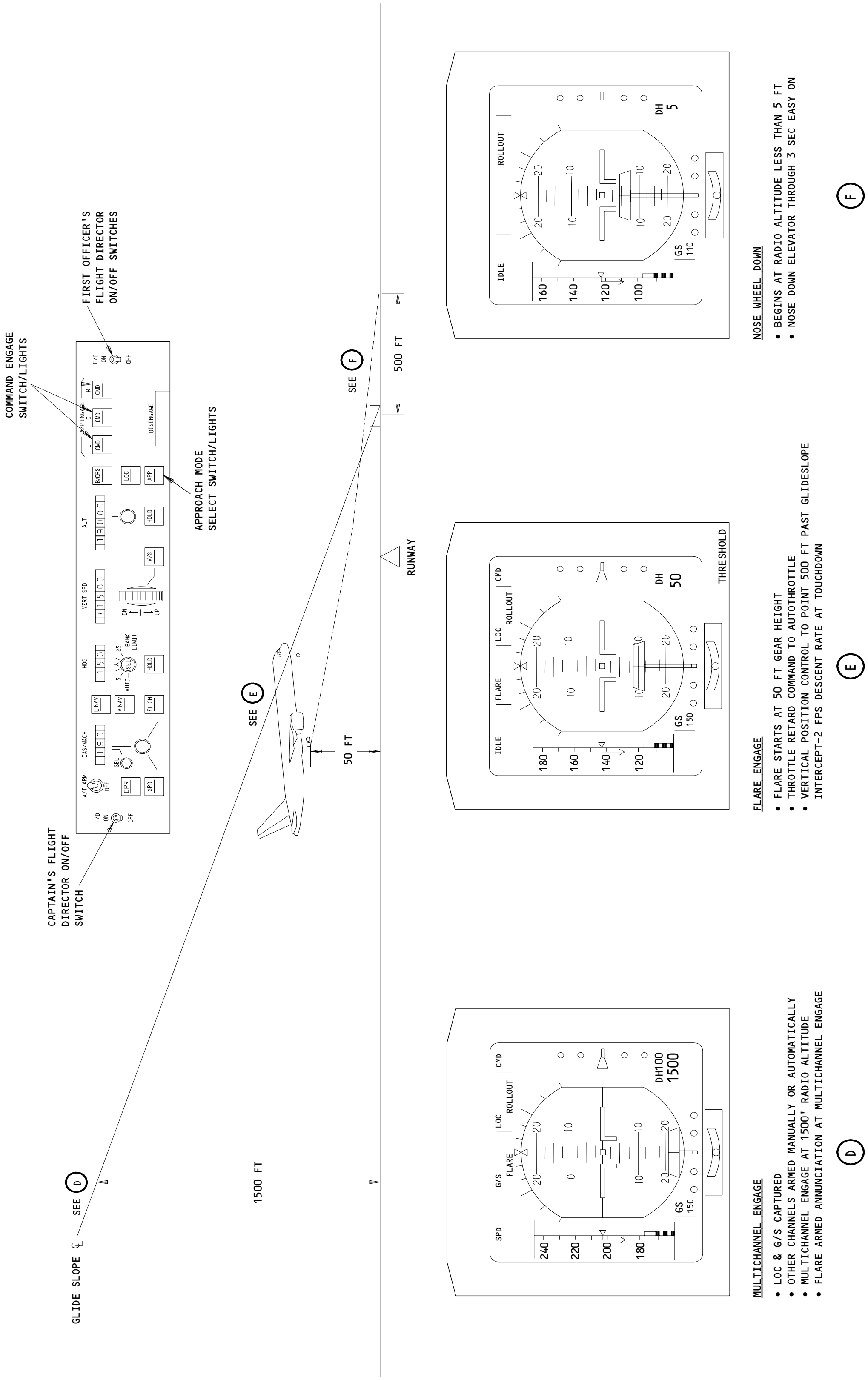
Pitch Channel - Go-Around Mode
Figure 16 (Sheet 4)

A70668



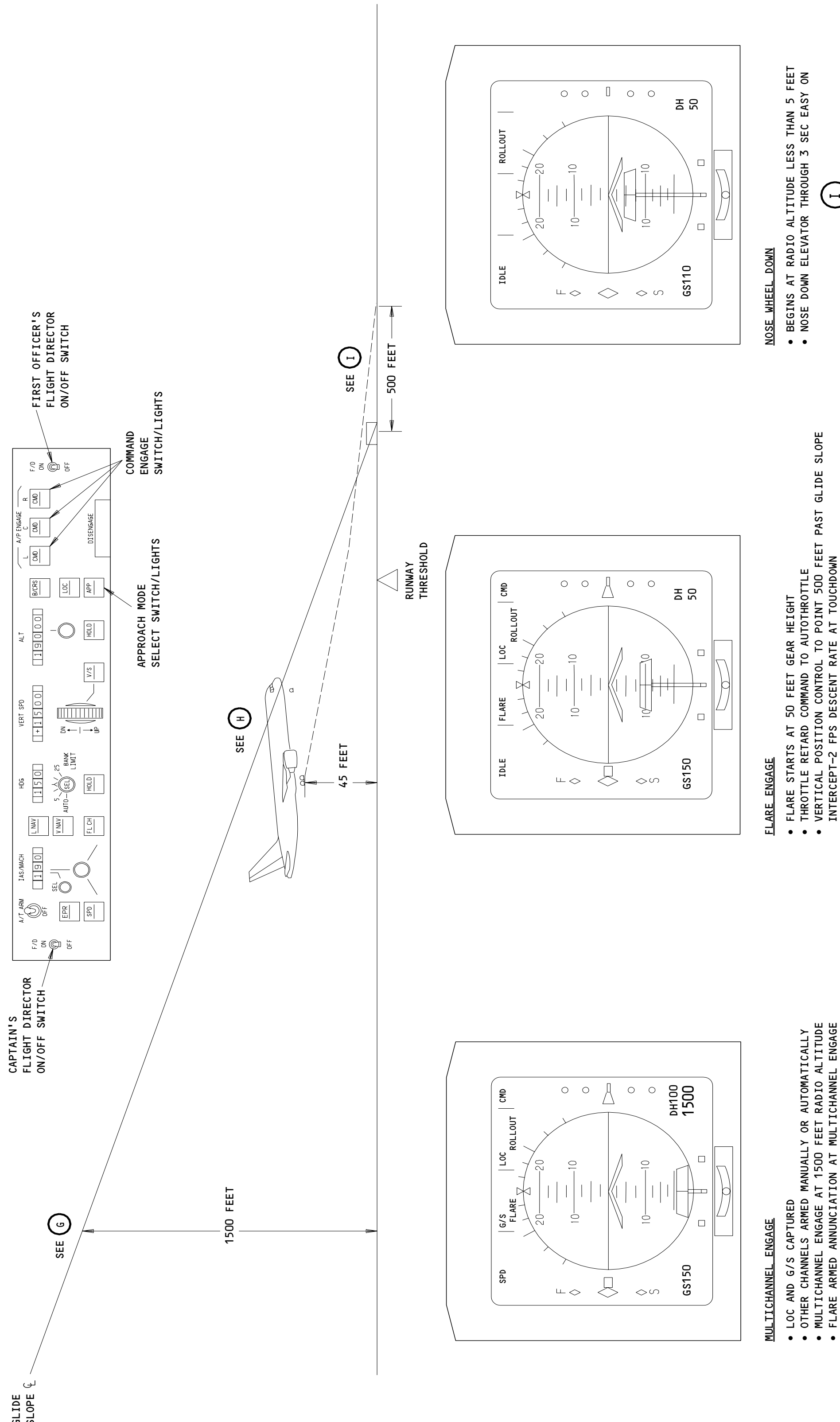
Pitch Channel - AutoLand
Figure 17 (Sheet 1)

984572



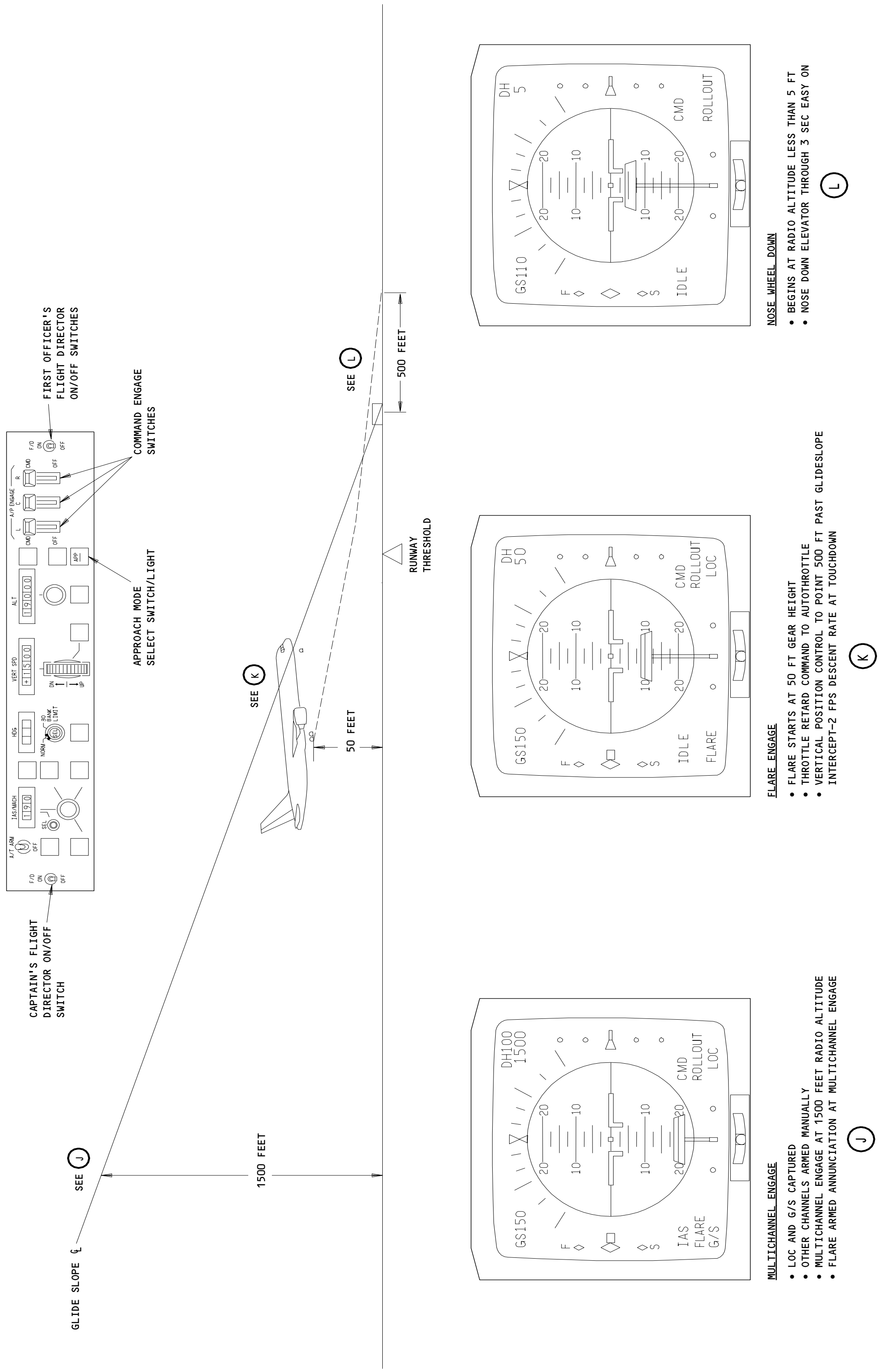
Pitch Channel - Autoland
Figure 17 (Sheet 2)

A44677



Pitch Channel - Autoland
Figure 17 (Sheet 3)

C33591



Pitch Channel - Autoland
Figure 17 (Sheet 4)

A70699

(b) Pitch Multi-Channel Engage
1) Pitch multi-channel engage occurs at 1500 feet radio altitude if G/S and LOC are captured and two or more channels are armed. The flight mode annunciator on the EADI shows FLARE as the pitch armed mode.
a) Flare is engaged when the main gear is 50 feet above the ground. A throttle retard command is sent to the autothrottle to control vertical position to a point 500 feet past the glideslope intercept point. The descent rate at touchdown is 2 feet per second.
b) At radio altitude of less than 5 feet, a nose-down command is applied to the elevators to ensure nose wheel steering is affective.

(22) Windshear
(a) The autoflight windshear system causes the airplane to climb out of a windshear when in the TO or GA modes, or when both flight director switches are off and autopilot is not in CMD. The system is started when one of the go-around switches is pushed. If the usual AFDS pitch commands do not cause a sufficient rate of climb (at least 1200 fpm), the AFDS smoothly changes to a 15 degree pitch attitude or a little less than the pitch limit indicator, whichever is less.
(b) The autothrottle sets the go-around thrust unless it is during takeoff when in the THR HOLD mode. When the go-around switch is pushed while in the TO mode, all the performance derates are cancelled and the EPR pointers move to maximum takeoff thrust. When THR HOLD is on, the throttles must be operated manually.
(c) When the windshear conditions go away, the autoflight system smoothly changes back to the usual takeoff or go-around mode.
(d) If the windshear system turns on with the flight directors off, the flight director command bars come on automatically and give the applicable commands, and the ADIs show GA. When the windshear condition is gone, the flight directors can be reset (command bars off) by selecting an AFDS mode, or by setting the flight director switches off, then on.

(23) Autoland Flight Path Control Logic (Fig. 10)

(a) Upon receipt of the APP mode ARMED or OPERATIVE discrete, the APP switch dot-bar matrix is illuminated to indicate request accepted. Each CMD ENGAGE switch initiates a CMDENG request discrete which goes to its corresponding FCC. A valid request generates a REQUEST ACCEPTED discrete which latches the CMD switch in the up position. Each F/D switch generates an ON/OFF discrete, but there is no visual feedback to the switches.

(b) Command Engage Sequence Detection Logic

1) Each FCC receives CMDENG REQUEST ACCEPTANCE logic from the other two FCCs via the cross-channel bus and records the order in which each become CMDENG. One of the discretes (L FIRST, C FIRST, R FIRST) is supplied to the first-in-command detection logic. This logic determines which FCC will supply FMA signals to the EFIS. It identifies LOCAL FIRST IN COMMAND and FOREIGN channels during multi-channel engage. On detection of a failure in the first-in-command channel, the fail sequence logic selects a different channel to be first-in-command.

(c) Autoland Processing Logic

1) The APP mode request generates the APPROACH ARM logic when the F/D disengage logic is zero. This is applied to the localizer logic as an alternate to the LOC mode arm logic and is also used in autoland logic.

(d) Transition to Rollout Arm

1) When autoland status is ROLLOUT ARM and the left FCC is LOCAL FIRST IN CMD, and the yaw channel remains in sync.

2) Each of the remaining FCCs (R and C) detects that the L FCC is CMDENG and each must receive individual CMDENG requests. The LOCAL FIRST IN CMD logic for the C and R FCCs is zero. The MULTIARM 1/2 logic is generated in the C FCC and R FCC. Both C and R multichannel engage (MCHENG) logic outputs are zero until MCHENG occurs. The R and C FCCs remain ARMED for CMDENG and are also ROLLOUT ARMED.

3) Multi-channel engage requires G/S and LOC capture and an indication of less than 1500 feet from a valid radio altimeter. Engage logic in the L FCC generates MCHENG (multichannel engage). In the C and R FCCs, engage logic generates MCHENG and C or R CMDENG logic. At MCHENG, the L FCC causes the yaw channel to be in CMD. The C FCC and R FCC cause roll, pitch and yaw to be in CMD. All three FCCs are now in ROLLOUT ENGAGE.

(e) Rollout Engage Default to OFF

1) Rollout engage defaults to OFF occurs when an engage removal fault is detected from the engage interlocks (Ref 22-11-00). If the L FCC defaults to OFF, for example, L CMDENG is logic zero to all FCCs. Sequence detection logic then selects a new LOCAL FIRST IN CMD from a prescribed sequence. This channel controls the ASA display.

(24) Autoland Flight Path Control Signal Flow and Monitoring (Fig. 10)

(a) Autoland pitch signals provide control during the following positions of the flight profile:

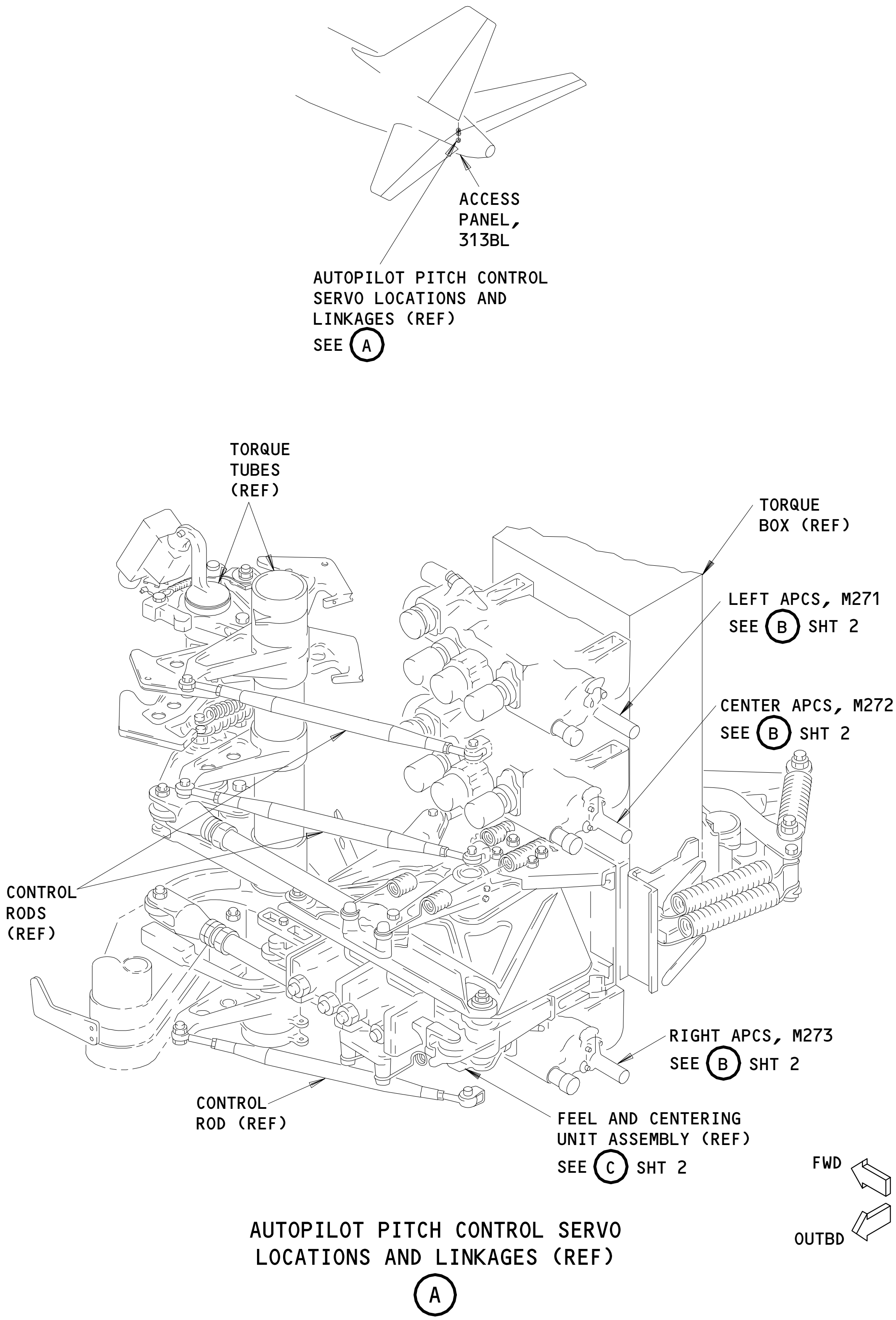
1) Glideslope (less than 1500 feet)
2) Flare (from gear altitude at 50 feet to touchdown)
3) Nose wheel let down
4) Elevator Equalization

(b) At multi-channel engage, elevator equalization logic becomes valid. The elevator commands are adjusted to remove any small errors between surface position feedback and servo position feedback.

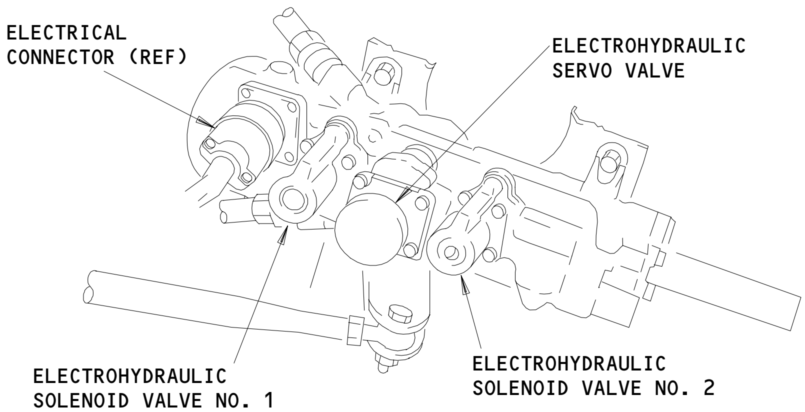
AUTOPILOT/FLIGHT DIRECTOR PITCH CHANNEL

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
SERVO - C AUTOPILOT PITCH CONTROL, M272	1	1	313BL, SECT 48	22-12-01
SERVO - L AUTOPILOT PITCH CONTROL, M271	1	1	313BL, SECT 48	22-12-01
SERVO - R AUTOPILOT PITCH CONTROL, M273	1	1	313BL, SECT 48	22-12-01
TRANSDUCER - C ELEVATOR NEUTRAL SHIFT, TS5153	2	1	313BL, SECT 48	22-12-04
TRANSDUCER - L ELEVATOR NEUTRAL SHIFT, TS5151	2	1	313BL, SECT 48	22-12-04
TRANSDUCER - R ELEVATOR NEUTRAL SHIFT, TS5152	2	1	313BL, SECT 48	22-12-04
VALVE - ELECTROHYDRAULIC SERVO	2	1	313BL, SECT 48, EA AUTOPILOT PITCH CONTROL SERVO	22-12-02
VALVE - ELECTROHYDRAULIC SOLENOID	2	2	313BL, SECT 48, EA AUTOPILOT PITCH CONTROL SERVO	22-12-02

Autopilot/Flight Director Pitch Channel - Component Index
Figure 101

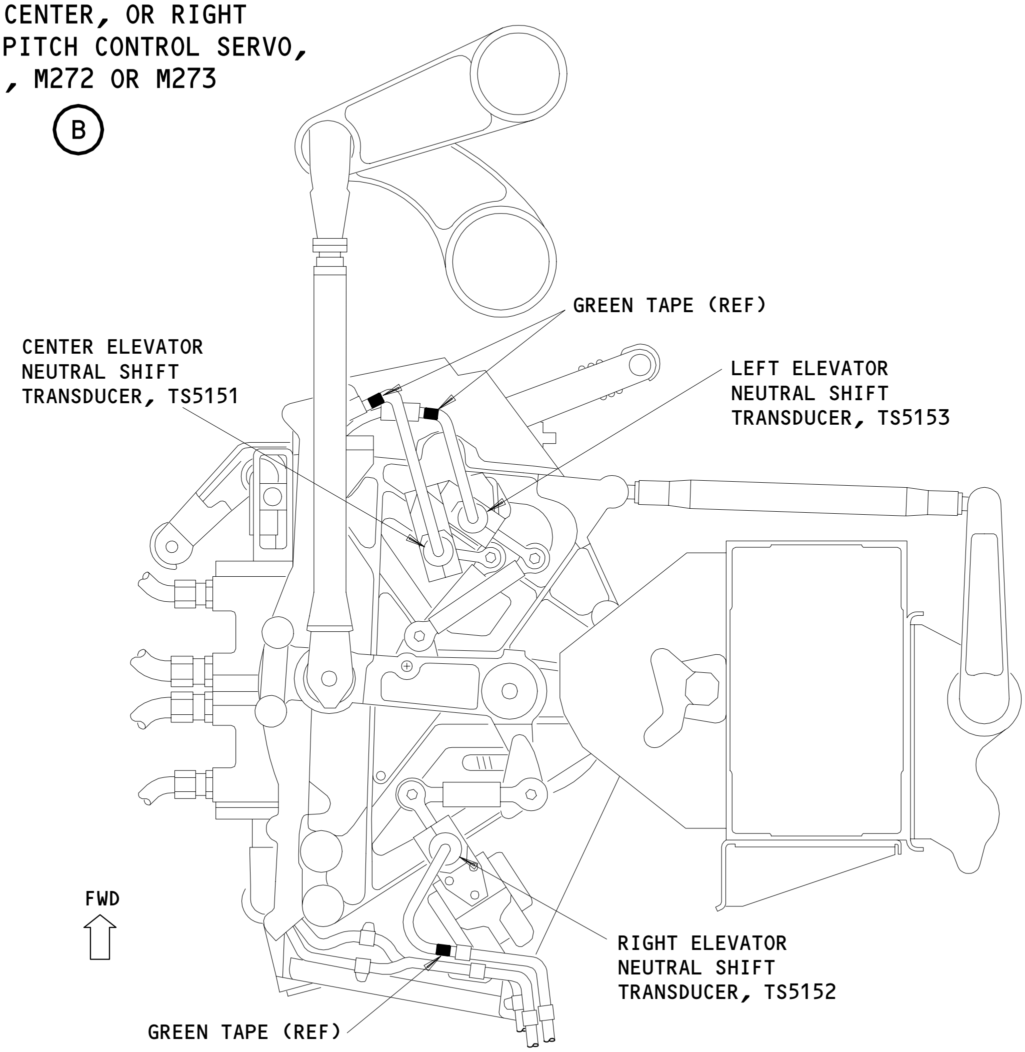


Autopilot/Flight Director Pitch Channel - Component Location
Figure 102 (Sheet 1)



LEFT, CENTER, OR RIGHT
AUTOPILOT PITCH CONTROL SERVO,
M271, M272 OR M273

B



FEEL AND CENTERING UNIT ASSEMBLY (REF)

C

Component Location (Details from Sht 1)
Figure 102 (Sheet 2)

AUTOPILOT PITCH CONTROL SERVO - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. There are three autopilot pitch servos. They are attached to the torque box aft of the horizontal stabilizer at section 48 of the airplane.
   B. This procedure includes tasks to remove, install, adjust, and test the Autopilot Pitch Control Servo. A task is also included to replace the control rod.
   C. There are two tasks to adjust the Autopilot Pitch Control Servo:
      (1) Adjustment with the Maintenance Control Display Panel (MCDP)
      (2) Adjustment with the Phase Synchronous Voltmeter Model 101A

TASK 22-12-01-004-065

2. Remove the Autopilot Pitch Control Servo
   A. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   B. Access
      (1) Location Zone
          313 Stabilizer Center Section Compartment (Left)

      (2) Access Panel
          313AL Control Bay Access Door

   C. Prepare for Removal

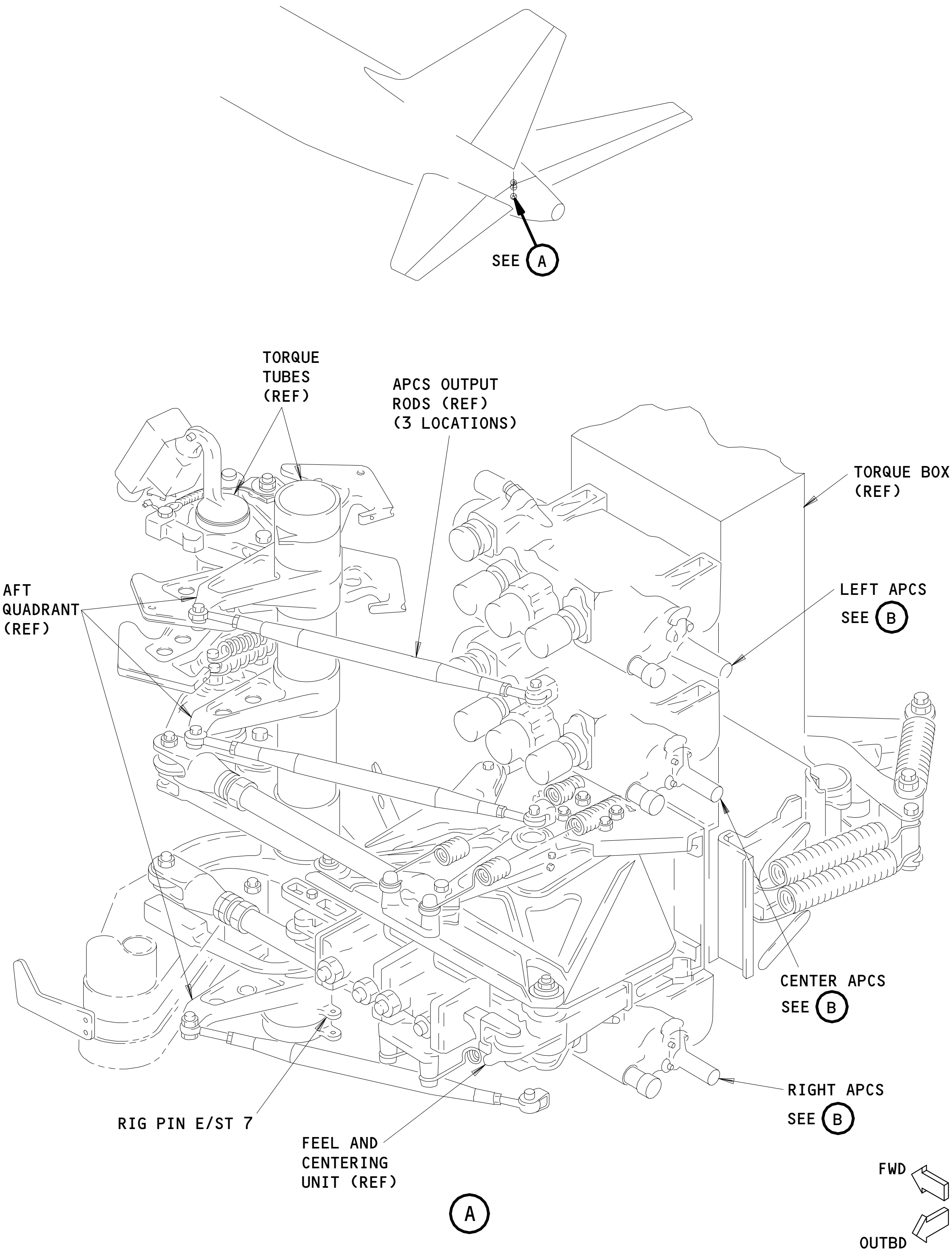
      S 864-001
      (1) Supply electrical power (Ref 24-22-00).

      S 864-002
      (2) Remove the pressure from the left, center and right hydraulic systems (Ref 29-11-00).

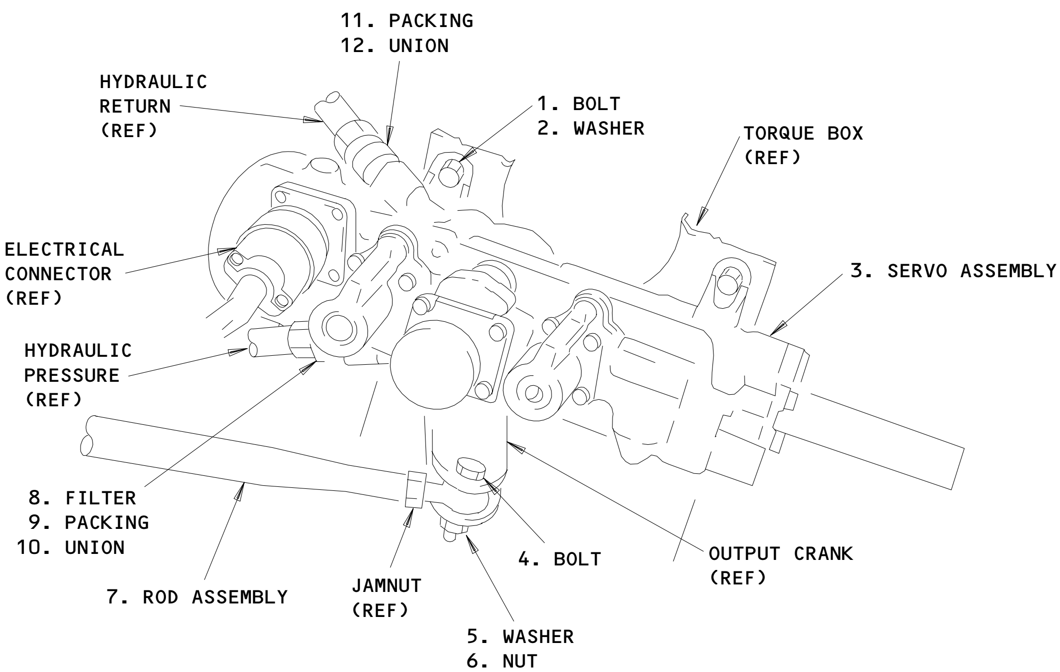
      S 864-003
      (3) On right side panel, P61, set the L, C and R FLT CONTROL SHUTOFF switches in the OFF position and attach DO-NOT-OPERATE tags.

      S 864-004
      (4) Set the two F/D switches on the MCP in the OFF position.

      S 864-005
      (5) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11H17, FLT CONT SHUTOFF LEFT
          (b) 11H18, FLT CONT SHUTOFF CENTER
          (c) 11H28, FLT CONT SHUTOFF RIGHT
          (d) For the left servo:
              1) 11E17, FLT CONT COMPUTER POWER LEFT
              2) 11E18, FLT CONT COMPUTER SERVO LEFT



Autopilot Pitch Control Servo
Figure 401 (Sheet 1)



AUTOPILOT PITCH CONTROL SERVO
(TYPICAL)
B

Autopilot Pitch Control Servo
Figure 401 (Sheet 2)

(e) For the center servo:
1) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
2) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER

(f) For the right servo:
1) 11E35, FLT CONT CMPTR PWR RIGHT
2) 11E36, FLT CONT CMPTR SERVO RIGHT

S 014-006

WARNING: STAY OFF THE CONTROLS BAY ACCESS DOOR, 313AL. THE WEIGHT OF PERSONS ON THIS DOOR COULD CAUSE ITS SPRING-LOADED LATCHES TO RELEASE. PERSONS COULD FALL THROUGH THE OPENING AND BE INJURED.

WARNING: STAY OFF THE SERVICE ACCESS DOOR, 311AL, IF IT IS OPENED (NOT REQUIRED FOR THIS PROCEDURE). THE WEIGHT OF PERSONS ON THIS DOOR COULD CAUSE ITS SPRING-LOADED LATCHES TO RELEASE. PERSONS COULD FALL THROUGH THE OPENING AND BE INJURED.

(6) Open access panel 313AL (Ref 06-42-00).

D. Remove Autopilot Pitch Control Servo

S 034-007

(1) Disconnect the electrical connector from servo.

S 034-008

(2) Remove the bolt (4), the nut (6) and the washer (5) from the rod assembly (7).

S 034-066

(3) Disconnect the rod assembly from the servo output crank.

S 034-009

(4) Disconnect the hydraulic lines.

S 434-010

(5) Seal the hydraulic lines and servo hydraulic ports with caps.

S 024-011

(6) Hold the servo and remove the bolts (1), the washers (2) and the servo (3).

NOTE: The servo assembly weighs 17 pounds.

TASK 22-12-01-424-067

3. Install the Autopilot Pitch Control Servo

A. Equipment

(1) Rig Pin E/ST 7 - P/N B20003-21 from set B20003-XX (Ref 20-10-24)

B. Parts

MM		NOMENCLATURE	IPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	3 3	Servo Assembly Servo Assembly	27-31-53	01	105 107

C. Reference
(1) 20-10-24/201, Rig Pins
D. Access
(1) Location Zone
313 Stabilizer Center Section Compartment (Left)

(2) Access Panel
313AL Control Bay Access Door

E. Install the Autopilot Pitch Control Servo

S 834-012
(1) Install the rig pin E/ST 7 (Ref 20-10-24).

S 034-013
(2) Remove the caps from the hydraulic lines and servo hydraulic ports.

S 434-014
(3) Connect the hydraulic lines to the servo and tighten them by hand.

S 434-015
(4) Use the bolts (1) and washers (2) to install the servo (3).

S 434-016
(5) Tighten the hydraulic lines.

S 434-017
(6) Use the bolt (4), washer (5), and nut (6) to connect the rod assembly (7) to the servo output crank.

S 434-018
(7) Connect the electrical connector.

S 094-047
(8) Remove the rig pin E/ST 7 if it is not necessary to replace the control rod.

S 824-068
(9) Adjust the servo with the Maintenance Control Display Panel (MCDP) or Model 101A Phase Synchronous Voltmeter if it is not necessary to replace the control rod.

TASK 22-12-01-904-019

4. Replace the Control Rod

A. Equipment
(1) Rig Pin E/ST 7 - P/N B20003-21 from set B20003-XX (Ref 20-10-24)

B. Parts

MM		NOMENCLATURE	IPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	3 3	Servo Assembly Servo Assembly	27-31-53	01	105 107

C. Reference
(1) 20-10-24/201, Rig Pins

D. Access

(1) Location Zone

313 Stabilizer Center Section Compartment (Left)

(2) Access Panel

313AL Control Bay Access Door

E. Replace the Control Rod

S 834-020

(1) Install the rig pin E/ST 7.

S 034-021

(2) Remove the bolt (4), washer (5) and nut (6) to disconnect the rod assembly (7) from the servo (3).

S 034-022

(3) Remove the bolt, washer and nut to disconnect the rod assembly (7) from the aft quadrant.

S 214-023

(4) Make sure the correct length of the control rod assembly (7) is 15.30 inches from the bolt center to bolt center.

NOTE: The correct length is used to decrease the amount of adjustment.

S 434-024

(5) Use the bolt, the washer and the nut to connect the end of the rod assembly (7) to the aft quadrant.

S 094-048

(6) Remove the rig pin E/ST 7.

S 824-025

(7) Adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

TASK 22-12-01-824-026

5. Adjust the Autopilot Pitch Control Servo with the Maintenance Control Display Panel (MCDP)

A. Parts

B. References
   (1) 22-00-02/201, Autoflight BITE
   (2) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   (3) 34-11-00/201, Pitot-Static System (Pressurization)

C. Access
   (1) Location Zone
       313 Stabilizer Center Section Compartment (Left)

   (2) Access Panel
       313AL Control Bay Access Door

D. Prepare to Adjust

   S 864-069
   (1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
       (a) 11E17, FLT CONT COMPUTER POWER LEFT
       (b) 11E18, FLT CONT COMPUTER SERVO LEFT
       (c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
       (d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
       (e) 11E35, FLT CONT CMPTR PWR RIGHT
       (f) 11E36, FLT CONT CMPTR SERVO RIGHT
       (g) 11H17, FLT CONT SHUTOFF TAIL LEFT
       (h) 11H18, FLT CONT SHUTOFF TAIL CENTER
       (i) 11H28, FLT CONT SHUTOFF TAIL RIGHT

   S 864-075
   (2) On panel P61, remove the DO-NOT-OPERATE tags and set the L, C and R FLT CONTROL SHUTOFF switches in the ON position.

S 864-070

WARNING: STAY OFF ALL OF THE CONTROL SURFACES. THE CONTROL SURFACES USE HYDRAULIC POWER AND MAY MOVE WHEN ANY CONTROLS ARE MOVED OR WHEN THE HYDRAULIC SYSTEMS ARE PRESSURIZED. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Supply hydraulic pressure (Ref 29-11-00).

S 864-076
(4) Manually operate the elevator system three times.

S 864-071
(5) Make sure these circuit breakers on the P11 panel are closed:

(a) 11C30, LANDING GEAR POS SYS 1
(b) 11E16, MODE CONT PNL LEFT
(c) 11E34, MODE CONT PNL RIGHT
(d) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(e) GUI 115;
11S6, MAINT CONT DSPL
(f) 11S15, AIR/GND SYS 1
(g) 11S19, AIR/GND SYS 2
(h) 11S23, POS SYS 2

S 864-072
(6) Pressurize the elevator feel pitot system to 350 knots (Ref 34-11-00).

S 864-073
(7) Set the stabilizer trim to six units of trim.

S 834-039
(8) Install the rig pin E/ST 7 (Ref 20-10-24).

S 864-074
(9) Set the MCDP to Ground Test 66-XDCR OUTPUTS (Ref 22-00-02).

S 864-077
(10) Use the YES/ADV switch to move the test forward until the top line of the MCDP shows 66-ELEV SURF DEG.

E. Adjust the Autopilot Pitch Control Servo

S 824-035
(1) With the control rod connected to the servo output crank, adjust the servo output control rod until the value on the bottom line of the MCDP display is 1.1 ±0.2 degrees.

S 434-036
(2) Use the bolt (4), the washer (5), and the nut (6) to connect the servo control rod assembly (7).

S 434-037
(3) Tighten the control rod (7) jamnut.

S 084-038
(4) Remove the rig pin E/ST 7.

S 714-039
(5) Do the "Test the Autopilot Pitch Control Servo Installation" steps.

TASK 22-12-01-824-040
6. Adjust the Autopilot Pitch Control Servo with the Model 101A Phase Synchronous Voltmeter

A. Equipment
(1) Rig Pin E/ST 7 - P/N B20003-21 from set B20003-XX (Ref 20-10-24)
(2) Voltmeter, Phase Synchronous Model 101A
Kit - 101AC5,
Electronic Aviation systems
300 Airport Way
Renton, WA 98005
(a) Voltmeter, Phase Synchronous
(b) Adapter Cable Assembly 1014002 -5

B. Parts

C. Reference
(1) 20-10-24/201, Rig Pins
(2) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(3) 34-11-00/201, Pitot-Static System (Pressurization)

D. Access
(1) Location Zone
313 Stabilizer Center Section Compartment (Left)

(2) Access Panel
313AL Control Bay Access Door

E. Prepare to Adjust the Autopilot Pitch Control Servo

S 864-040

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11H17, FLT CONT SHUTOFF TAIL LEFT
(b) 11H18, FLT CONT SHUTOFF TAIL CENTER
(c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-041

(2) On panel P61, remove the DO-NOT-OPERATE tags and set the L, C and R FLT CONTROL SHUTOFF switches in the ON position.

S 864-042

<u>WARNING</u>: STAY OFF ALL OF THE CONTROL SURFACES. THE CONTROL SURFACES USE HYDRAULIC POWER AND MAY MOVE WHEN ANY CONTROLS ARE MOVED OR WHEN THE HYDRAULIC SYSTEMS ARE PRESSURIZED. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Supply hydraulic pressure (Ref 29-11-00).

S 864-043

(4) Manually operate the elevator system three times.

S 864-044

(5) Pressurize the elevator feel pitot system to 350 knots (Ref 34-11-00).

S 864-045

(6) Set the stabilizer trim to six units of trim.

S 834-038

(7) Install the rig pin E/ST 7 (Ref 20-10-24).

S 484-041

(8) Connect the Phase Synchronous Voltmeter to the power source given on the voltmeter placard.

S 864-066
(9) Set the ON/OFF switch in the ON position and let the voltmeter warm up for 10 minutes.

S 864-067
(10) Set the CAL/RUN switch in the CAL CHK position.
(a) Make sure in phase volts are 9.990 to 10.010.

S 864-068
(11) Set the CAL/RUN switch in the RUN position.

S 034-045
(12) Disconnect the electrical connector from the Autopilot Pitch Control Servo.

S 484-046
(13) Connect the -5 cable assembly to the Autopilot Pitch Control Servo and Phase Synchronous Voltmeter.

F. Adjust the Autopilot Pitch Control Servo

S 824-047
(1) Adjust the Autopilot Servo control rod (7) until the Phase Synchronous voltmeter shows +3.72 ±.03 with the control rod (7) connected to the servo.

S 434-048
(2) Use the bolt (4), the washers (5), and the nut (6) to connect the servo control rod assembly (7).

S 434-049
(3) Tighten the control rod (7) jamnut.

S 864-050
(4) Disconnect the Phase Synchronous Voltmeter
(a) Set the ON/OFF switch in the OFF position.
(b) Disconnect the -5 cable assembly from the Autopilot Pitch Control Servo.
(c) Disconnect the voltmeter from the power source.

S 434-051
(5) Connect the airplane electrical connector to the autopilot pitch control servo.

S 084-052
(6) Remove the rig pin E/ST 7.

S 734-053
(7) Do the "Test the Autopilot Pitch Control Servo Installation" steps.

TASK 22-12-01-734-054

7. Test the Autopilot Pitch Control Servo Installation

A. References
   (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
   (2) 22-00-02/201, Autoflight BITE
   (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

B. Access
   (1) Location Zone
       313 Stabilizer Center Section Compartment (Left)

   (2) Access Panel
       313AL Control Bay Access Door

C. Prepare to Test

S 864-078

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11E17, FLT CONT COMPUTER POWER LEFT
   (b) 11E18, FLT CONT COMPUTER SERVO LEFT
   (c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
   (d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
   (e) 11E35, FLT CONT CMPTR PWR RIGHT
   (f) 11E36, FLT CONT CMPTR SERVO RIGHT
   (g) 11H17, FLT CONT SHUTOFF TAIL LEFT
   (h) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (i) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-079

(2) On the P61 panel, remove the DO-NOT-OPERATE tags and set the L, C and R FLT CONTROL SHUTOFF switches on panel P61 in the ON position.

S 864-080

WARNING: STAY OFF ALL OF THE CONTROL SURFACES. THE CONTROL SURFACES USE HYDRAULIC POWER AND MAY MOVE WHEN ANY CONTROLS ARE MOVED OR WHEN THE HYDRAULIC SYSTEMS ARE PRESSURIZED. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Supply hydraulic pressure (Ref 29-11-00).

S 864-081

(4) Manually operate the elevator system three times.

S 864-082

(5) Make sure these circuit breakers on the P11 panel are closed:

(a) 11C30, LANDING GEAR POS SYS 1
(b) 11E16, MODE CONT PNL LEFT
(c) 11E34, MODE CONT PNL RIGHT
(d) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(e) GUI 115;
11S6, MAINT CONT DSPL
(f) 11S15, AIR/GND SYS 1
(g) 11S19, AIR/GND SYS 2
(h) 11S23, POS SYS 2

S 864-083
(6) Pressurize the elevator feel pitot system to 350 knots (Ref 34-11-00).

S 864-084
(7) Set the stabilizer trim to six units of trim.

S 864-085
(8) Set the MCDP to Ground Test 66-XDCR OUTPUTS (Ref 22-00-02).

S 864-086
(9) Use the YES/ADV switch to move the test forward until the top line of the MCDP shows 66-ELEV SURF DEG.

D. Test the Autopilot Pitch Control Servo Installation

S 214-056
(1) With the control column released, make sure the MCDP shows 1.1 ±0.2 degrees on the bottom line of the display.

S 984-057
(2) Move the control column to the full forward position.
(a) Make sure the value on the bottom line of the MCDP display is more than +5 degrees for the left, center and right servos.

S 984-037
(3) Move the control column to the full aft position.
(a) Make sure the value on the bottom line of the MCDP display is more than -5 degrees (-6 or subsequent) for the left, center and right servos.

S 744-059
(4) Do the MCDP Ground Test 08-SERVO ELEV (Ref 22-00-02).
(a) Make sure no MCDP fault messages show during the test.

S 214-036
(5) Examine the Autopilot Pitch Control Servo for leaks and fix them as necessary.

E. Put the Airplane Back to its Usual Condition

S 414-061
(1) Close the service access panel 313AL.

S 044-062
(2) Set the MCDP to off.

S 864-046
(3) Slowly release the pressure from the elevator feel pitot system (Ref 34-11-00).

S 864-063
(4) Remove the hydraulic power if it is not necessary (Ref 29-11-00).

S 864-064
(5) Remove the electrical power if it is not necessary (Ref 24-22-00).

APCS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - MAINTENANCE PRACTICES

1. General (Fig. 201)
   A. There are three autopilot pitch control servos. They are attached to the torque box that is aft of the horizontal stabilizer at section 48 of the airplane. Each servo has two LRU solenoid valves and one LRU electrohydraulic servovalve.
   B. This procedure includes tasks to remove, install, and test the servovalve and solenoid valves. The tasks are the same for the two types of valves.

TASK 22-12-02-002-001

2. Remove the APCS Servovalve or Solenoid Valve
   A. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   B. Access
      (1) Location Zone
          313 Stabilizer Center Section Compartment (Left)

      (2) Access Panels
          313AL Controls Bay Access Door

   C. Prepare to Remove the Valve.

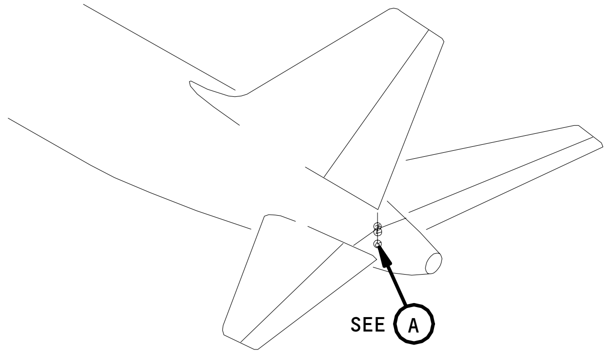
      S 862-002
      (1) Supply electrical power (Ref 24-22-00).

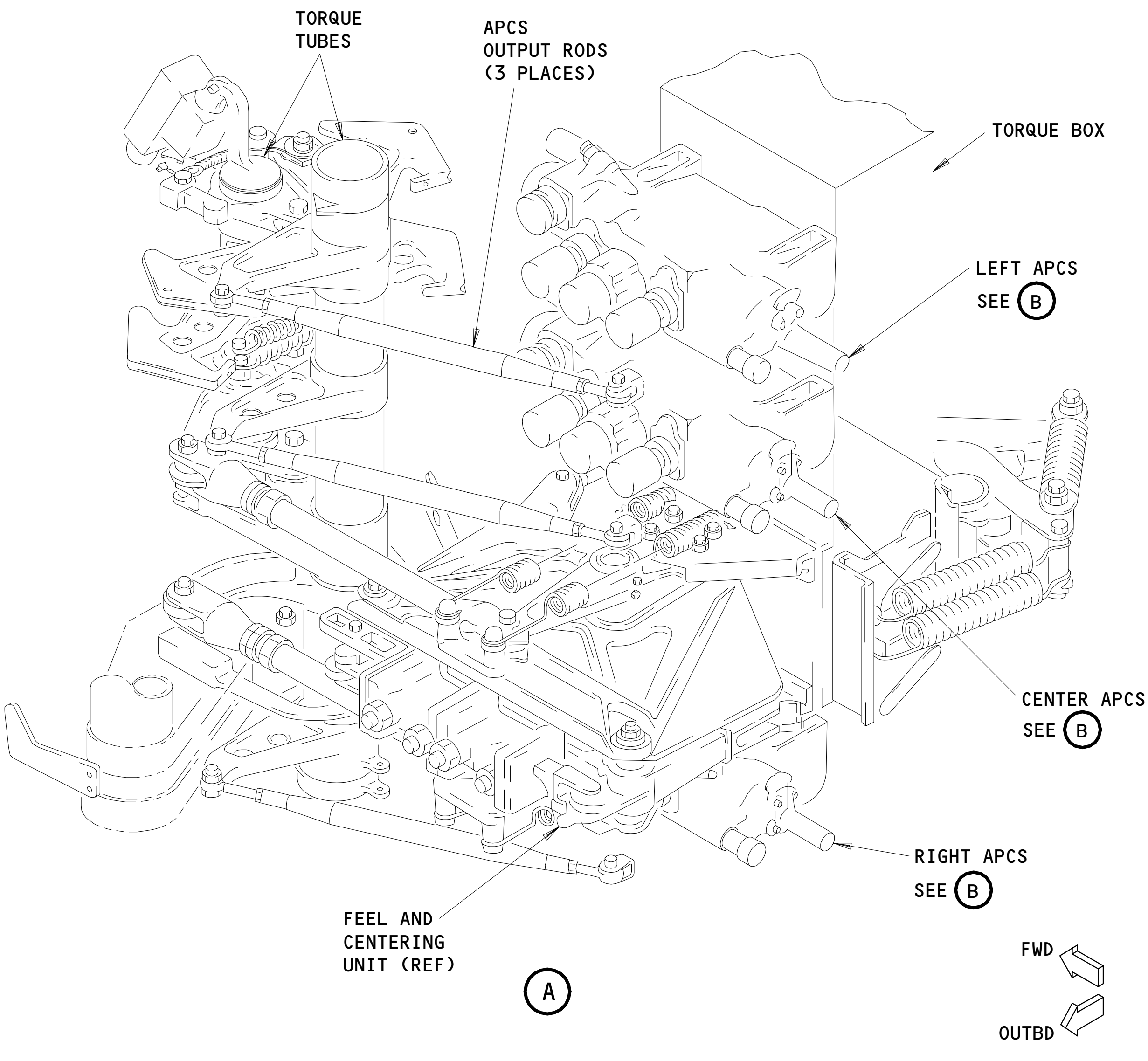
      S 862-003
      (2) Remove the pressure from the left, the center and the right hydraulic systems (Ref 29-11-00).

      S 862-004
      (3) On right side panel, P61, set the L, C, and R FLT CONTROL SHUTOFF switches in the OFF position and attach DO-NOT-OPERATE tags.

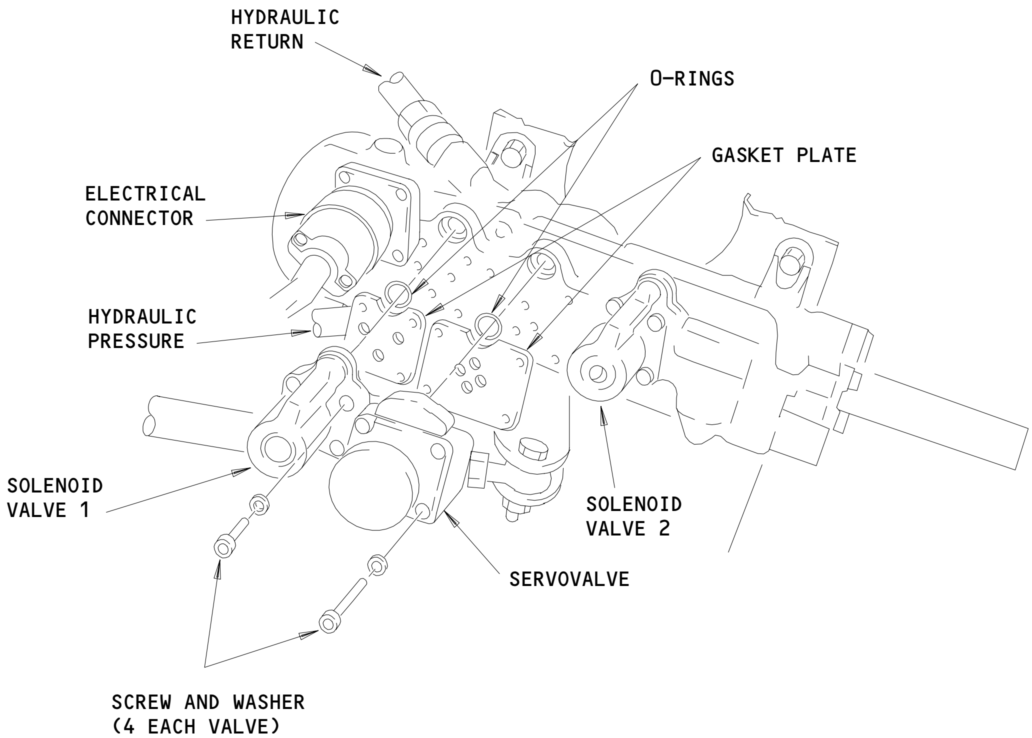
      S 862-005
      (4) Set the two F/D switches on the MCP in the OFF position.

      S 862-006
      (5) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11H17, FLT CONT SHUTOFF LEFT
          (b) 11H18, FLT CONT SHUTOFF CENTER
          (c) 11H28, FLT CONT SHUTOFF RIGHT
          (d) For the left servo:
              1) 11E17, FLT CONT COMPUTER POWER LEFT
              2) 11E18, FLT CONT COMPUTER SERVO LEFT
          (e) For the center servo:
              1) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
              2) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER





APCS Electrohydraulic Servo Valve and Solenoid Valves
Figure 201 (Sheet 1)



TYPICAL APCS ELECTROHYDRAULIC SERVO VALVE
AND SOLENOID VALVES

B

APCS Electrohydraulic Servo Valve and Solenoid Valves
Figure 201 (Sheet 2)

(f) For the right servo:
1) 11E35, FLT CONT CMPTR PWR RIGHT
2) 11E36, FLT CONT CMPTR SERVO RIGHT

S 012-007

WARNING: STAY OFF THE CONTROLS BAY ACCESS DOOR, 313AL. THE WEIGHT OF PERSONS ON THIS DOOR COULD CAUSE ITS SPRING-LOADED LATCHES TO RELEASE. PERSONS COULD FALL THROUGH THE OPENING AND BE INJURED.

WARNING: STAY OFF THE SERVICE ACCESS DOOR, 311AL, IF IT IS OPENED (NOT) NECESSARY FOR THIS PROCEDURE). THE WEIGHT OF PERSONS ON THIS DOOR COULD CAUSE ITS SPRING-LOADED LATCHES TO RELEASE. PERSONS COULD FALL THROUGH THE OPENING AND BE INJURED.

(6) Open access panel 313AL (Ref 06-42-00).

D. Remove the Valves.

S 102-008

CAUTION: DO NOT LET ANY CONTAMINATION GET INTO THE ELEVATOR AUTOPILOT SERVO AND VALVES. CONTAMINATION MAY CAUSE EQUIPMENT DAMAGE AND/OR FAILURE.

(1) Fully clean the external areas of the valve and around the surface of the valve where it attaches to the servo.

S 032-009
(2) Remove the screws and washers from the valve assembly.

S 032-010
(3) Remove the valve assembly.

S 032-011
(4) Remove the O-ring.

S 032-012
(5) Remove the gasket plate.

S 432-013
(6) Put covers on all openings.

TASK 22-12-02-422-019

3. Install the APCS Servovalve or Solenoid Valve

A. Consumable Materials

(1) D00153 Fluid - Hydraulic, Erosion Arresting, Fire Resistant BMS 3-11

(2) D00633 Grease - BMS 3-33 (Preferred)
(3) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)

B. Access
(1) Location Zone
313 Stabilizer Center Section Compartment (Left)

(2) Access Panels
313AL Controls Bay Access Door

C. Install the Valve.

S 032-014
(1) Remove the covers from all openings.

S 112-015
(2) Clean the surfaces of valve that will attach to the servo the new seal plate, the O-ring, and the servo assembly with clean hydraulic fluid. Do not allow the hydraulic fluid to go into the electrical connector.

S 432-016
(3) Install the O-ring.

S 432-019
(4) Install the gasket plate.

S 422-018
(5) Put the valve assembly in its position on the servo.

S 642-017
(6) Apply a layer of grease to the screws.

S 432-014
(7) Install the screws and the washers to attach the valve to the servo assembly.
(a) Tighten screws to 50-70 inch-pounds.

TASK 22-12-02-702-020
4. <u>Test the APCS Electrohydraulic Servovalve and Solenoid Valves</u>

A. References
(1) 22-00-02/201, Autoflight BITE
(2) 24-22-00/201, Electrical Power - Control
(3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

B. Access
  (1) Location Zone
      313 Stabilizer Center Section Compartment (Left)

  (2) Access Panels
      313AL Controls Bay Access Door

C. Prepare to Test the Valves

  S 862-020
  (1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
    (a) 11E17, FLT CONT COMPUTER POWER LEFT
    (b) 11E18, FLT CONT COMPUTER SERVO LEFT
    (c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
    (d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
    (e) 11E35, FLT CONT CMPTR PWR RIGHT
    (f) 11E36, FLT CONT CMPTR SERVO RIGHT
    (g) 11H17, FLT CONT SHUTOFF TAIL LEFT
    (h) 11H18, FLT CONT SHUTOFF TAIL CENTER
    (i) 11H28, FLT CONT SHUTOFF TAIL RIGHT

  S 012-021
  (2) Remove the DO-NOT-OPERATE tags from the left, center, and right FLT CONT SHUTOFF switches on panel P61 and set them in the ON position.

  S 782-022

  WARNING: STAY OFF ALL OF THE CONTROL SURFACES. THE CONTROL SURFACES CAN MOVE WHEN CONTROLS ARE MOVED OR WHEN THE HYDRAULIC SYSTEMS ARE PRESSURIZED. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

  (3) Pressurize the left, center and right hydraulic systems.

  S 982-023
  (4) Manually operate the elevator system three times.

D. Test the Valves

  S 862-024
  (1) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:

    (a) 11C30, LANDING GEAR POS SYS 1
    (b) 11E16, MODE CONT PNL LEFT
    (c) 11E34, MODE CONT PNL RIGHT

(d) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(e) GUI 115;
11S6, MAINT CONT DSPL
(f) 11S15, AIR/GND SYS 1
(g) 11S19, AIR/GND SYS 2
(h) 11S23, POS SYS 2

S 742-028
(2) Do MCDP Ground Test 08-SERVO ELEV (Ref 22-00-02).
(a) Make sure MCDP failure messages do not show during test 08 SERVO ELEV.

S 212-026
(3) Examine the system for leaks and fix them as necessary.

E. Put the Airplane Back to Its Usual Condition.

S 412-025
(1) Close the access panel 313AL.

S 862-031
(2) Set the MCDP to off.

S 862-030
(3) Remove the pressure from the left, center and right hydraulic systems if it is not necessary (Ref 29-11-00).

S 862-029
(4) Remove electrical power if it is not necessary (Ref 24-22-00).

ELEVATOR NEUTRAL SHIFT TRANSDUCER - REMOVAL/INSTALLATION

1. General
   A. The three elevator neutral shift transducers are on the feel centering assembly aft of the horizontal stabilizer in fuselage section 48. The transducers are accessible through access panel 313AL, aft of the stabilizer.

TASK 22-12-04-004-001

2. Remove the Elevator Neutral Shift Transducer
   A. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Hydraulic System
   B. Access
      (1) Location Zone
          313 Stabilizer Center Section Compartment

      (2) Access Panel
          313AL APU Intake Port and Door Installation

   C. Prepare to Remove the Elevator Neutral Shift Transducer

      S 864-002
      (1) Supply electrical power (Ref 24-22-00).

      S 864-003

      WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF THE CONTROL SURFACES BEFORE THE HYDRAULIC SYSTEMS ARE PRESSURIZED. THE CONTROL SURFACES CAN MOVE WHEN THE HYDRAULIC SYSTEMS ARE PRESSURIZED OR IF ANY OF THE CONTROLS ARE MOVED. THIS CAN CAUSE INJURY TO PERSONS AND DAMAGE TO EQUIPMENT.

      (2) Supply hydraulic pressure (Ref 29-11-00).

      S 864-004
      (3) Put the stabilizer in leading-edge down position (full nose up) for easy access.

S 864-005
(4) Set the two STAB TRIM switches on the P10 panel to the CUT OUT position.

S 864-006
(5) Remove the pressure from the left, center and right hydraulic systems (Ref 29-11-00).

S 864-007
(6) Set the L, C, and R FLT CONTROL SHUTOFF switches on the right side panel P61 to the OFF position and attach DO-NOT-OPERATE tags.

S 864-008
(7) Set the two F/D switches on the MCP to OFF.

S 864-009
(8) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
- (a) 11C12, STAB TRIM SHUTOFF LEFT
- (b) 11C13, STAB TRIM SHUTOFF RIGHT
- (c) 11H17, FLT CONT SHUTOFF LEFT
- (d) 11H18, FLT CONT SHUTOFF CENTER
- (e) 11H28, FLT CONT SHUTOFF RIGHT
- (f) For the left transducer:
  - 1) 11E17, FLT CONT COMPUTER POWER LEFT
  - 2) 11E18, FLT CONT COMPUTER SERVO LEFT
- (g) For the center transducer:
  - 1) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
  - 2) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER
- (h) For the right transducer:
  - 1) 11E35, FLT CONT CMPTR PWR RIGHT
  - 2) 11E36, FLT CONT CMPTR SERVO RIGHT

S 864-010

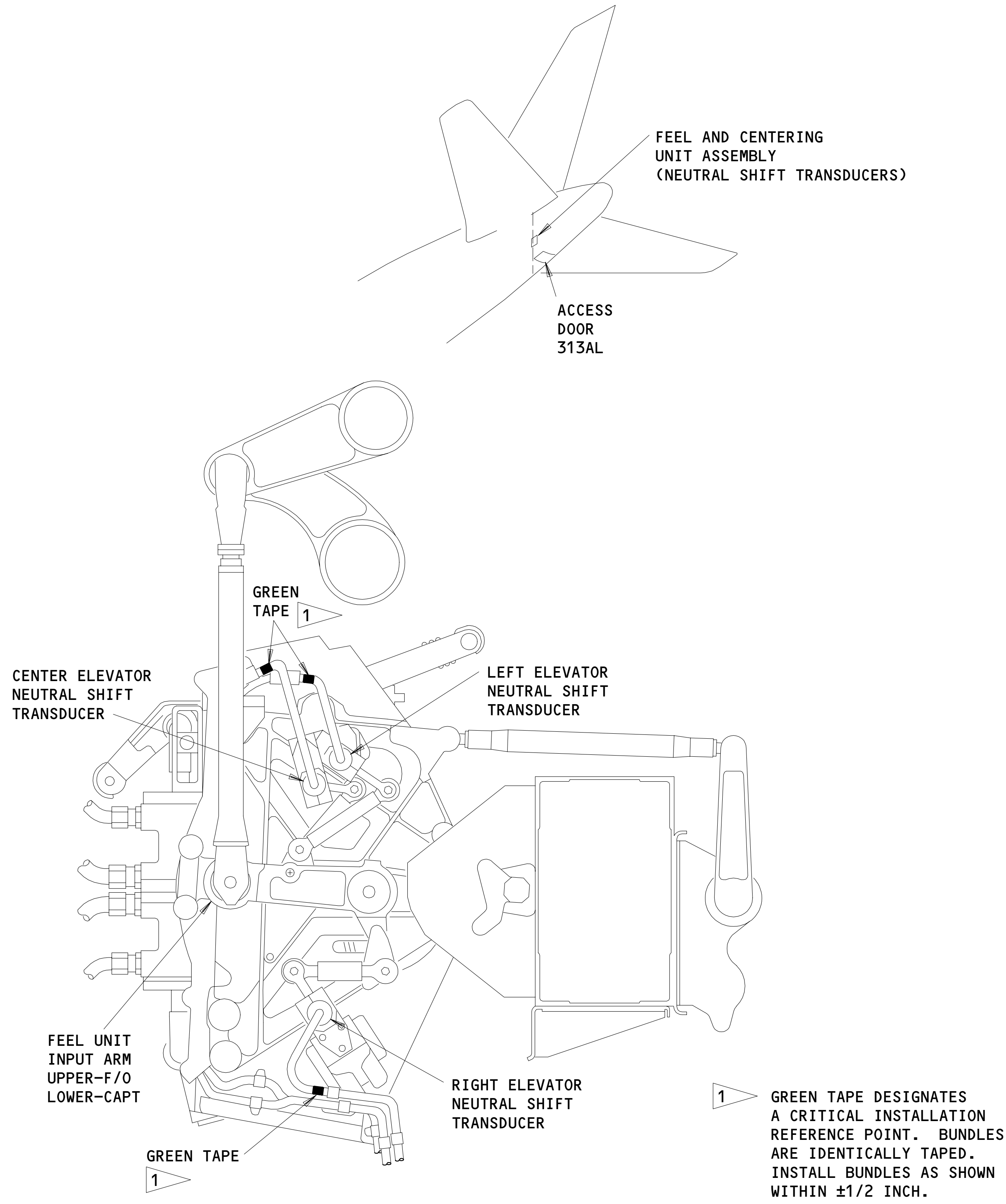
WARNING: STAY OFF THE SERVICE ACCESS DOOR 311AL AND THE ACCESS DOOR FOR THE CONTROLS BAY, 313AL. YOUR WEIGHT CAN CAUSE THE SPRING-LOADED LATCHES TO RELEASE. IF YOU FALL THROUGH THE DOOR, INJURY CAN OCCUR.

(9) Open the access panel 313AL (Ref 06-42-00).

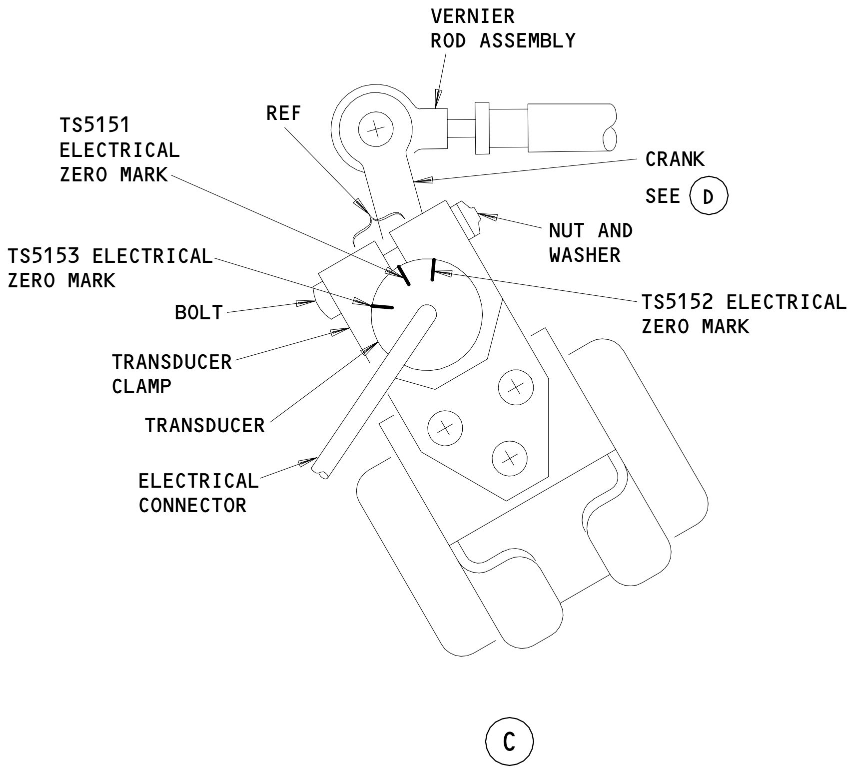
D. Remove the Elevator Neutral Shift Rotary Transducer (Fig. 401)

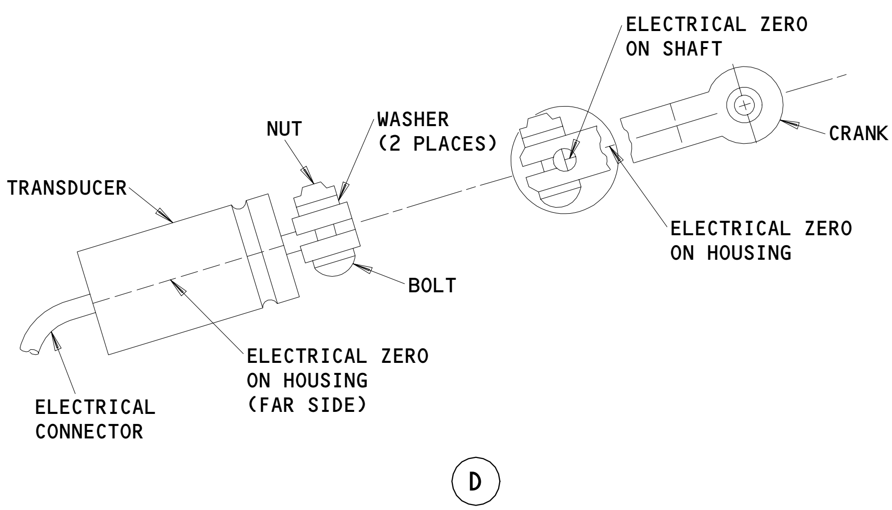
S 034-011
(1) Disconnect the electrical connector.

S 034-012
(2) Remove the bolts, the washers, and the nuts from the transducer electrical cable clamps.



Elevator Neutral Shift Rotary Transducers
Figure 401 (Sheet 1)





Elevator Neutral Shift Rotary Transducer
Figure 401 (Sheet 2)

S 034-013
(3) Remove the nut, the washer, and the bolt from the crank and vernier rod assembly.

S 034-014
(4) Remove the screws from the transducer clamp.

S 014-015
(5) Remove the transducer and the transducer clamp as an assembly.

S 034-016
(6) Remove the crank from the transducer.

S 034-017
(7) Remove the nut, the washer, and the bolt from the transducer clamp.

S 034-018
(8) Remove the transducer assembly from the transducer clamp.

TASK 22-12-04-404-019

3. Install the Elevator Neutral Shift Rotary Transducer

A. General

(1) Two procedures are given to adjust the transducer after installation. One uses a phase angle voltmeter for the transducer adjustment and the other uses the MCDP.

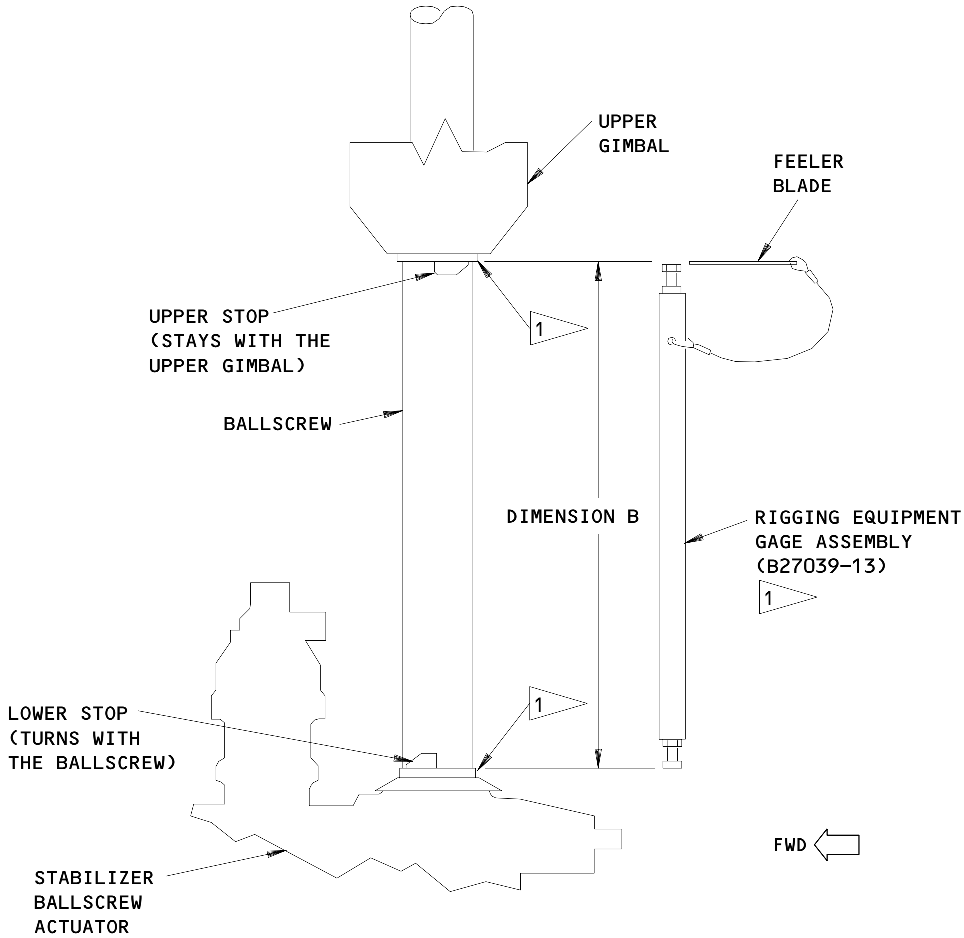
B. Equipment

NOTE: Necessary when the MCDP is not used for the adjustment.

(1) Breakout Box - Position Sensors, Flight Controls Rigging - A27063-88
 (a) Breakout Box - 2
 (b) Cable Assemblies -26, -27
(2) Phase angle voltmeter with a 10 mv accuracy at 5.0 volts in the phase sensitive mode

C. References

(1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
(2) 22-00-02/201, Autoflight BITE



1 TO USE THE GAGE ASSEMBLY TO SET DIMENSION B TO 22.24 ±0.05 INCHES (564.90 ±0.25 mm), DO THE STEPS THAT FOLLOW:

1. PUT THE GAGE ASSEMBLY AGAINST THE BALLSCREW SUCH THAT THE BOLT HEADS ARE ON THE SHOULDERS OF THE UPPER AND LOWER STOPS
2. MAKE SURE THAT THE FEELER BLADE CAN BE PUT BETWEEN THE BOLT HEAD OF THE GAGE ASSEMBLY AND ON THE SHOULDER OF THE UPPER STOP (THIS WILL PREVENT PRELOADING)

Horizontal Stabilizer Position
Figure 402

(3) 24-22-00/201, Electrical Power - Control
(4) 29-11-00/201, Pressurize/Depressurize Hydraulic System
(5) 34-11-00/201, Pitot-Static System

D. Access
(1) Location Zone
313 Stabilizer Center Section Compartment (Left Side)

(2) Access Panel
313AL APU Intake Port and Door Installation

E. Install the Component (Fig 401)

S 434-077
(1) Install the transducer in the transducer clamp. View the transducer from the top and align the electrical zero mark on the transducer as follows:
(a) TS 5151 - align the electrical zero mark with the center of the transducer clamp reference in view C.
(b) TS 5152 - set the electrical zero mark approximately 16 degrees right of the reference in view C.
(c) TS 5153 - set the electrical zero mark approximately 63 degrees left of the reference in view C.

S 434-020
(2) Install the transducer clamp bolt, washer, and nut.

S 434-021
(3) Install the crank on the transducer with the centerline on the crank aligned with the electrical zero mark on the shaft.

S 824-022
(4) Set the vernier rod length to 3.68 inches.

S 434-023
(5) Connect the crank and the vernier rod and install the bolt, the washer, and the nut.

S 434-024
(6) Install the transducer electrical cable with the clamps.

S 214-025
(7) Make sure the transducer electrical cable is correctly installed at the critical installation reference points. The critical installation reference points are identified by green tape.

S 434-078
(8) Tighten the bolts, the washers, and the nuts on the clamps.

S 824-079
(9) Adjust the transducer with the Maintenance Control Display Panel (MCDP) or the Phase Angle Voltmeter.

F. Adjust the Elevator Neutral Shift Rotary Transducer With the MCDP

S 864-026
(1) Set the stabilizer trim to 2 units so dimension B=22.24 ±0.05 inches (Fig. 402).

S 864-027
(2) Pressurize the auxiliary pitots 1 and 2 to 350 knots (4.0 psig) (Ref 34-11-00).

S 434-028
(3) Connect the transducer electrical connectors.

S 214-029
(4) Shake the lower (Captain's) feel unit input arm to make sure the arm is in the center.

S 434-082
(5) Using a suitable block/clamp combination, restrain the upper (First Officer's) feel unit input arm.

S 034-031
(6) Loosen the transducer clamp to let the transducer body turn.

S 864-032
(7) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E17, FLT CONT COMPUTER POWER LEFT
(b) 11E18, FLT CONT COMPUTER SERVO LEFT
(c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(e) 11E35, FLT CONT CMPTR PWR RIGHT
(f) 11E36, FLT CONT CMPTR SERVO RIGHT

S 714-033
(8) Start MCDP Ground Test 66-XDCR OUTPUTS (AMM 22-00-02).

S 714-080
(9) Push the YES/ADV switch until the MCDP top line message shows 66 FEEL POS DEG.

S 214-034
(10) Turn the transducer body until the MCDP bottom line shows less than +1 degree.
(a) Make sure the electrical zero mark on the transducer body is less than 30 degrees from the electrical zero mark on the shaft.

S 214-035
(11) Tighten the clamping nut.
(a) Make sure the MCDP bottom line value stays less than +1.0 degree.

S 034-036
(12) Remove the safety wire from the vernier rod.

S 214-037
(13) Turn the knurled ring on the vernier rod until the MCDP bottom line shows 0.00 ±0.1 degrees with the control column released.

S 434-038
(14) Lock the knurled ring to the vernier rod with wire.

S 034-039
(15) Remove the block and clamp from the upper feel unit input arm.

S 714-040
(16) Do the Test the Transducer Installation steps.

G. Adjust Elevator Neutral Shift Rotary Transducer with the Phase Angle Voltmeter

S 864-041
(1) Set the stabilizer trim to 2 units so dimension B=22.24 ±0.05 inches (Fig. 402).

S 864-042
(2) Pressurize the auxiliary pitots 1 and 2 to 350 knots (4.0 psig) (Ref 34-11-00).

S 864-043
(3) Shake the lower (Captain's) feel unit input arm to make sure the arm is in the center.

S 434-044
(4) Using a suitable block/clamp combination, restrain the upper (First Officer's) feel unit input arm.

S 034-045
(5) Loosen the transducer clamp to let the transducer body turn.

S 434-046
(6) Install the breakout box between the transducer and the airplane wiring.

S 864-047
(7) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E17, FLT CONT COMPUTER POWER LEFT
(b) 11E18, FLT CONT COMPUTER SERVO LEFT
(c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CTR
(d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CTR
(e) 11E35, FLT CONT CMPTR PWR RIGHT
(f) 11E36, FLT CONT CMPTR SERVO RIGHT

S 214-048
(8) Measure the output of the transducer on pins 3 and 5 with a Phase Angle Voltmeter.

S 214-049
(9) Turn the transducer body until you get an in-phase (absolute) output voltage of less than 1.25v ac. Make sure the electrical zero mark on the transducer body is less than 30 degrees from the electrical zero mark on the shaft.

S 214-050
(10) Tighten the clamping nut. Make sure the in-phase output voltage is less than 1.25v ac.

S 034-051
(11) Remove the lockwire from the vernier rod.

S 214-052
(12) Turn the knurled ring on the vernier rod to get the smallest in-phase voltage. Make sure the voltage is less than 0.040v ac.

S 434-053
(13) Lock the knurled ring to the vernier rod with wire.

S 034-054
(14) Remove the breakout box and the phase angle voltmeter.

S 434-055
(15) Connect the transducer electrical connector.

S 034-056
(16) Remove the block and the clamp from the upper feel unit input arm.

S 714-057
(17) Do the Test the Transducer Installation steps.

H. Test the Transducer Installation

S 864-058
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT
(c) 11E17, FLT CONT COMPUTER POWER LEFT
(d) 11E18, FLT CONT COMPUTER SERVO LEFT
(e) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(f) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(g) 11E35, FLT CONT CMPTR PWR RIGHT
(h) 11E36, FLT CONT CMPTR SERVO RIGHT
(i) 11H17, FLT CONT SHUTOFF LEFT
(j) 11H18, FLT CONT SHUTOFF CENTER
(k) 11H28, FLT CONT SHUTOFF RIGHT

S 864-059
(2) Remove the DO-NOT-OPERATE tags and set the L, C, and R FLT CONTROL SHUTOFF switches on the panel P61 to the ON position.

S 864-060
(3) Remove the DO-NOT-OPERATE tags and set the two STAB TRIM switches on the P10 panel to the NORM position.

S 864-061
(4) Set the stabilizer at two units of trim so dimension B=22.24 ±0.01 inches (Fig. 402).

S 864-062
(5) Pressurize the elevator feel pitot system to 350 knots (Ref 34-11-00).

S 864-063
(6) Pressurize the L, C, and R main hydraulic systems (Ref 29-11-00).

S 714-064
(7) Start MCDP Ground Test 66-XDCR OUTPUTS (Ref 22-00-02).

S 984-065
(8) Push the YES/ADV switch until the message 66 FEEL POS DEG ±X.XX ±X.XX ±X.XX shows.

S 214-066
(9) Make sure the three values that show are 0.00 ±0.1 degree with the control column released. Make sure the maximum difference between the channels is 0.1 degree.

S 214-067
(10) Write the values that show.

S 864-068
(11) Set the stabilizer to six units of trim as measured on the indicator.

S 214-069
(12) Make sure the three values that show are still 0.00 ±0.1 degree with the control column released.

S 214-070
(13) Make sure the difference between the FEEL POS DEG values at 2 units of trim and at six units of trim is 0.05 degree or less for each channel. Adjust the elevator feel unit upper arm (MM 27-31-00) if the difference is more than 0.05 degree.

S 864-071
(14) Remove pressure from the elevator feel pitot system.

S 714-072
(15) Do MCDP Ground Test 08-SERVO ELEV (Ref 22-00-02).
(a) Make sure no fault messages show during the test.

I. Put the Airplane Back to Its Usual Condition

S 864-073
(1) Set the MCDP to off.

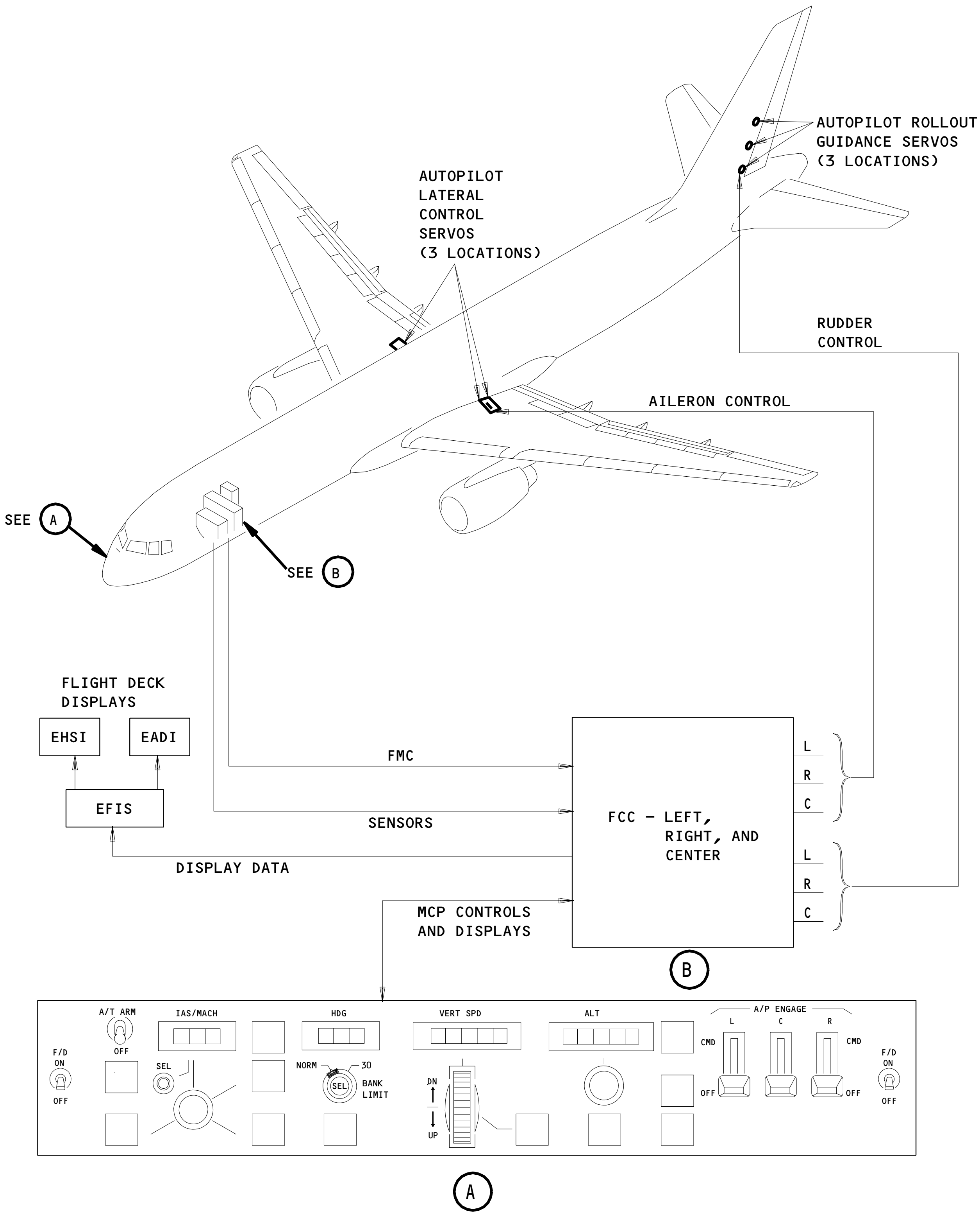
S 864-074
(2) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 864-075
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

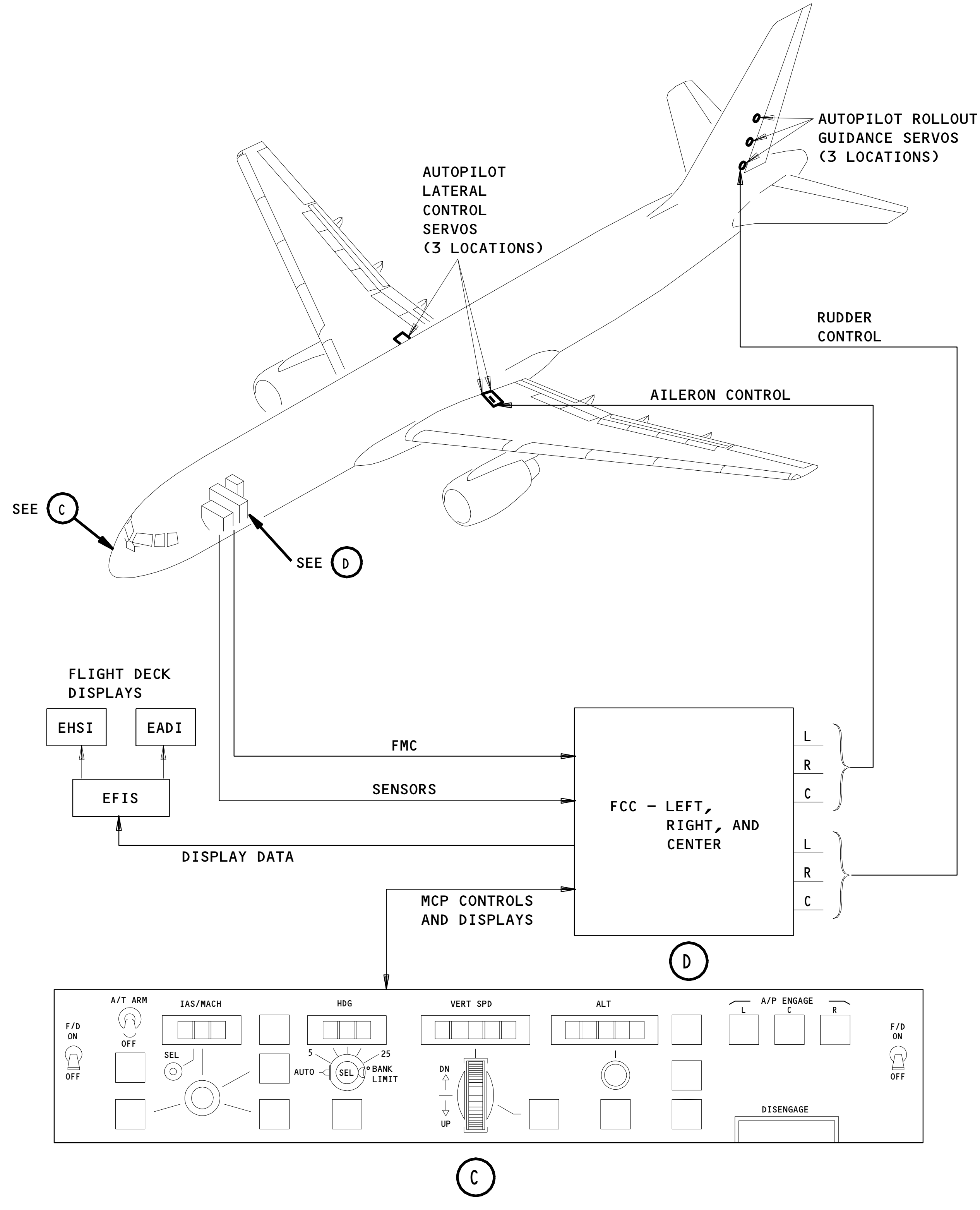
S 864-076
(4) Close the service access panel 313AL.

AUTOPILOT/FLIGHT DIRECTOR ROLL AND YAW CHANNEL - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The roll and yaw channels consist of three flight control computers (FCCs), three autopilot lateral control servos (ALCSs) and three autopilot rollout guidance servos (ARGSs). Each servo is dedicated to a single FCC. Primary control of the A/P is through selections made on the AFCS Mode Control Panel (MCP). Mode and Flight Director (F/D) displays are presented on the Electronic Attitude Director Indicators (EADIs) of the Electronic Flight Instrument System (EFIS).
   B. Each FCC generates F/D and A/P commands. The F/D commands are supplied to the EFIS symbol generators for display on the EADI. The F/D displays are enabled by the F/D switches on the MCP.
   C. With the A/P not engaged, control wheel rotation provides input to the aileron power control units; rudder pedals and rudder trim control provide inputs to the rudder power control actuators. The output Linear Variable Differential Tranducers (LVDTs) of the ALCS and ARGS sense control surface movement. They generate aileron and rudder servo feedback commands for synchronizing the FCC to the control surface positions.
   D. With the A/P engaged in single channel CMD, sensor signals are combined in the FCC and autopilot roll command signals drive the dedicated ALCS. Off-line FCCs synchronize to existing conditions. The F/D command bars are biased out of view for the FCC that has the autopilot engaged in CMD.
   E. Two or three FCCs in CMDENG constitute the multi-channel engage (MCHENG) configuration for Autoland. In MCHENG, each engaged FCC receives dedicated sensor data and drives the dedicated ALCS and ARGS with roll and yaw servo commands.
   F. Roll Axis Control Requirements
      (1) Roll axis control functions of the FCC provide roll attitude hold and programmed roll attitude control of the airplane. The ALCSs, provide for roll axis maneuvering, which drives the ailerons and spoilers through control wheel movement.
      (2) If a flight director is on, roll attitude inputs from the IRU provides F/D command to maintain the roll attitude of the airplane during aerodynamic disturbances.
      (3) Programmed attitude control is provided automatically:
         (a) From the Flight Management Computer System (FMCS) during lateral steering (LNAV mode).
         (b) During localizer capture and centerline tracking (LOC and APP modes).
         (c) During automatic compensation for wind drift (track conditions).
         (d) During runway alignment maneuvering prior to touchdown (APP mode).



Roll and Yaw Axis Control
Figure 1 (Sheet 1)



Roll and Yaw Axis Control
Figure 1 (Sheet 2)

G. Yaw Axis Control

(1) Yaw axis control functions provide directional control of the airplane during an autopilot dual or triple channel engaged controlled touchdown and rollout. During an ILS approach with two or three autopilots engaged, an FCC directional control module aligns the airplane on the centerline prior to touchdown and maintains the airplane on runway centerline after touchdown. During high speed rollout, the rudder is used for steering. Through an interconnect with the rudder pedals, nose wheel steering is used during low speed rollout.

H. Appropriate rate limiting and compensation signals are provided to the roll and yaw channels in each A/P mode. The validity of input data used in each mode is monitored and appropriate actions taken in the event a no computed data (NCD) bit is detected.

I. FCC Roll and Yaw Axes - Inputs and Outputs (Fig. 2)

(1) Figure 2 shows the FCC inputs and outputs related to the roll and yaw axes. The inputs include mode requests (MCP), servo and surface position (ALCS & ARGS), flight reference data (IRU and ADC), and navigation data (ILS and RA). Runway length and horizontal steering commands are received from the FMC. The go-around switches on the thrust levers provide an input for engagement of G/A mode.

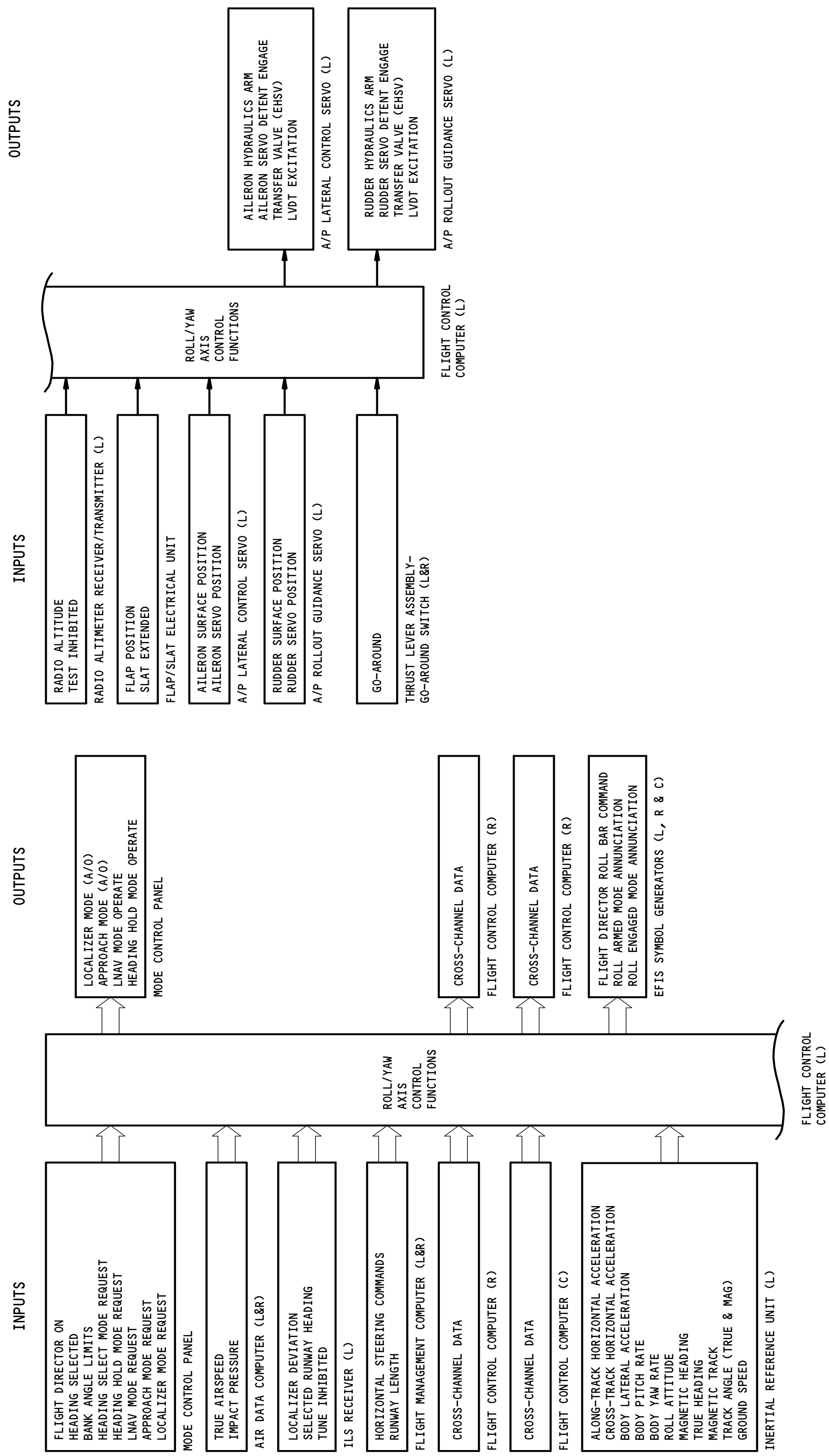
(2) Outputs are servo arm and engage commands (ALCS and ARGS), mode status (MCP), F/D roll commands (EADI), and Flight Mode Annunciations (EADI FMA). Cross-channel data is provided to and from each of the other FCCs.

2. Component Details

A. A/P Lateral Control Servos - Location and Linkages (Fig. 3)

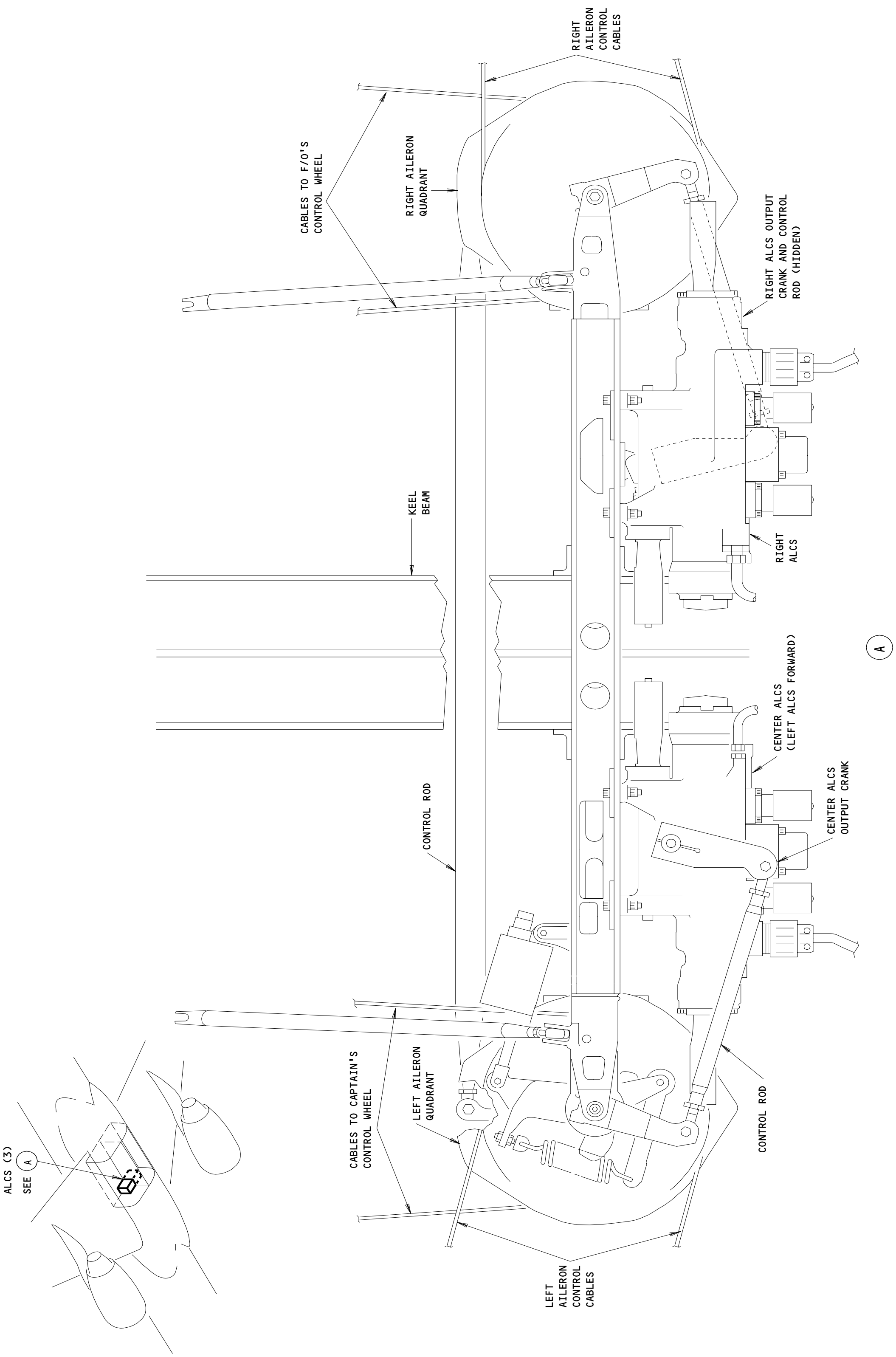
(1) The left and center ALCSs are located in the main landing gear wheel well, left of the keel beam (left ALCS forward; center ALCS aft). The right ALCS is located to the right of the keel beam. They are installed with four mounting bolts. Two hydraulic lines and an electrical connector complete the installation. Left, center, and right ALCSs are controlled by L, C, and R FCCs and powered by the L, C, and R hydraulic systems respectively.

(2) Outputs from the left and center servos are mechanically linked to the feel, centering, and trim assembly. The right servos output is linked directly to the override mechanism and, indirectly to the feel, centering and trim assembly through a bus cable, quadrant, and linkage. Servo output as a result of autopilot command is fed back to the control wheels through the feel, centering, and trim assembly.



FCC Roll and Yaw Axes - Control Inputs and Outputs
Figure 2

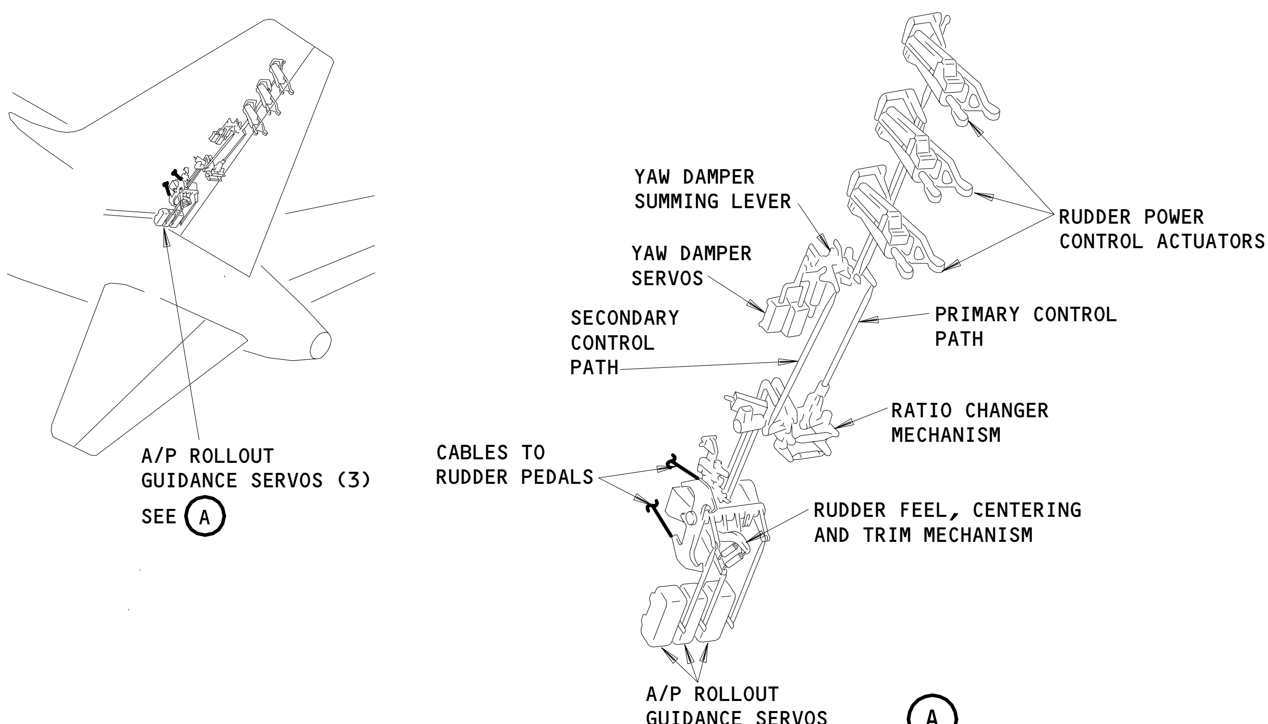
30330



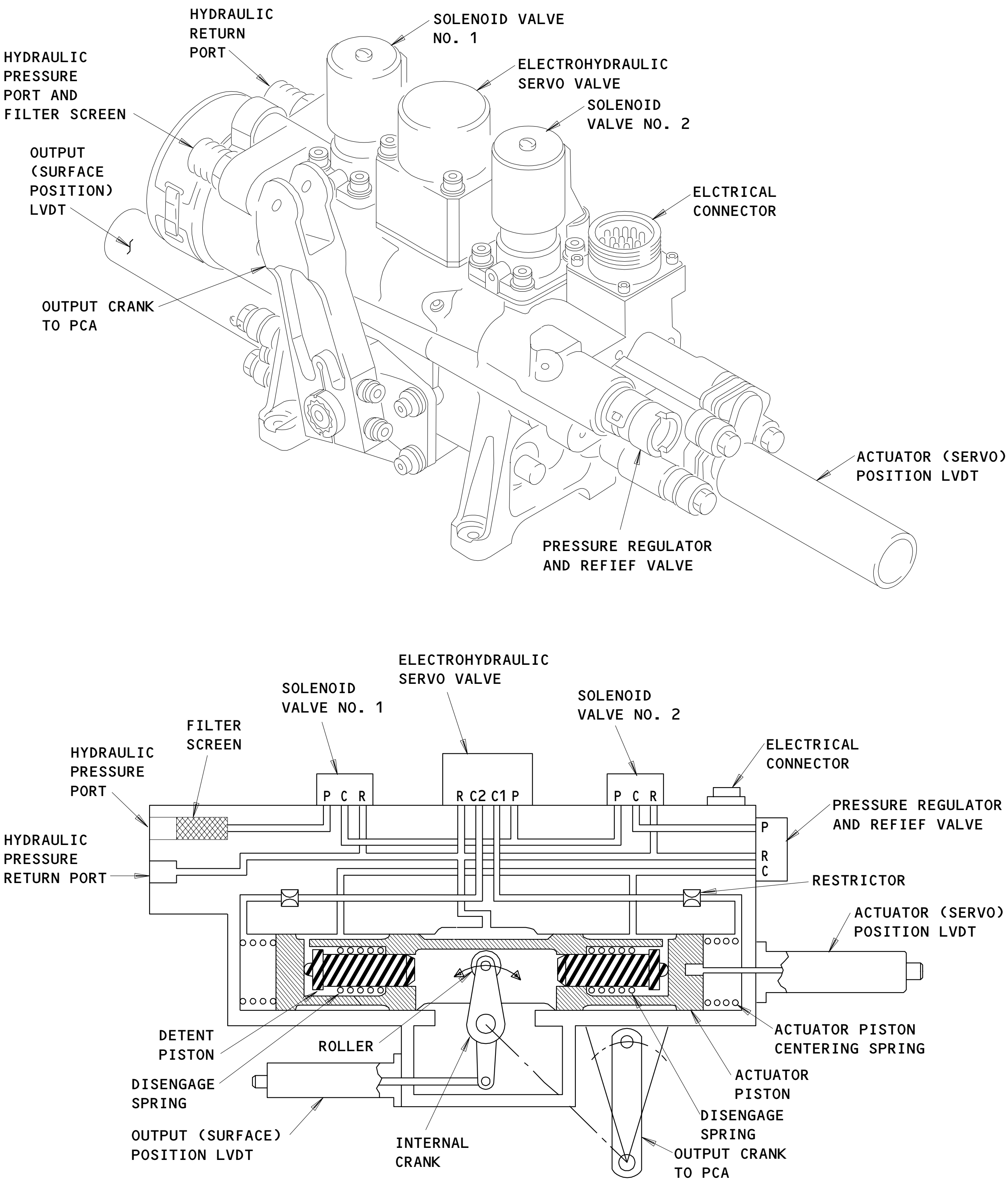
ALCS Location and Linkage
Figure 3

B. A/P Rollout Guidance Servos - Location and Linkage (Fig. 4)
   (1) Three autopilot rollout guidance servos (ARGSs) are side by side at the base of the vertical stabilizer immediately forward of the rudder hinge line. Each servo has an output rod connected to the rudder feel, centering and trim mechanism on the aft quadrant. Access to the servos is through a removable panel on the lower left side of the vertical stabilizer.
   (2) Each ARGS has its own drive linkage to the aft quadrant which provides feedback to the rudder pedal. The rudder control system has dual mechanical drive links to the ratio changer mechanism with primary and backup linkages to the yaw damper summing lever. There is no feedback from the yaw damper to the rudder pedals. Three power control actuators (PCAs) use a common mechanical input to deflect the rudder surface.

C. Autopilot Servo Components (Fig. 5)
   (1) Each A/P servo (ALCS and ARGS) contains two solenoid valves, an electrohydraulic servo valve (EHSV), two linear variable differential transducers (LVDTs), and a pressure regulator/relief valve. Hydraulic power is applied through pressure and return ports at one end of the servo. A 25-micron filter is in the hydraulic input port of unit.



ARGS Location and Linkage
Figure 4



Autopilot Servo Components (ALCS and ARGS)
Figure 5

(2) Solenoid Valves

(a) Each solenoid valve (SV1 and SV2) is an electrically operated open-close valve which completes hydraulic pressure through the servo when autopilot arm and engage logic circuits are completed. The valve is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mated when the solenoid valve is bolted in position.

(b) Solenoid valve number 1 (SV1) opens when the autopilot is armed. It ports hydraulic pressure to solenoid valve number 2 (SV2) and the electrohydraulic servo valve (EHSV). Solenoid valve number 2 opens when the autopilot is engaged, and ports hydraulic pressure to the detent pistons. The detent pistons clamp the output linkage crank and transmit actuator piston position to the output linkage.

(3) Electrohydraulic Servovalve (EHSV)

(a) The EHSV is controlled by the output command signal from the FCC. The EHSV contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydrualic pressure through the valve can be applied to either of two output ports. When no command signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor armature and out of the projector jet. From the jet, equal pressure is applied to opposite ends of the piston holding it at center.

(b) When a command signal is applied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston than the other causing it to move and open the corresponding output port to complete hydraulic pressure through the servo. When the command signal is nulled, the motor armature and jet return to center. This equalizes the pressure on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.

(c) In response to the FCC output command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the actuator piston LVDT (linear variable differential transducer) nulls the command signal at the FCC and piston movement stops.

(d) The EHSV is installed with four bolts and sealed with a gasket plate. Electrical pins are mated when the EHSV is bolted in position.

(4) Linear Variable Differential Transducer (LVDT)

(a) Each servo has two identical LVDTs. The actuator piston (servo position) LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The output position (surface position) LVDT is operated by the intermediate crank, which is connected to the surface control linkage. Both LVDTs are variable reluctance transformers with an output that varies directly with linear motion. The LVDT uses 26vac excitation from the associated FCC. The LVDTs are not considered line replaceable units. They require nulling adjustments to be completed which match actuator piston with piston position when the servo is on the bench.

(5) Pressure Regulator and Relief Valves

(a) The pressure regulator and relief valve regulates and limits hydraulic pressure applied to the actuator piston and detent pistons. The pressure relief function allows manual inputs from the control linkage to override autopilot control.

(6) Actuator Piston Assembly

(a) The actuator piston translates autopilot input commands through the EHSV into mechanical positioning of the control surface. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV when the autopilot is armed or engaged. The springs center the actuator piston when pressurization is released.

(b) Two detent pistons inside the actuator piston are normally retracted by disengage springs. When the autopilot is engaged, SV2 opens. Hydraulic pressure through SV2 overrides spring tension and locks the detent pistons against the roller of the internal crank. Detent pistons may be forced back if sufficient force to overcome hydraulic pressure is applied manually through the control linkage (camout).

(c) The actuator piston LVDT provides an electrical signal proportional to actuator piston position. This signal nulls the autopilot command signal from the FCC to stop movement of the actuator piston.

(7) Cranks

(a) The internal crank roller is clamped by the detent pistons when the autopilot is engaged. The crank moves with the actuator piston. Motion of the internal crank moves the output position LVDT. One end of the output crank is directly connected to the internal crank at a common pivot point. The other end of the output crank is connected to the control surface linkage.

(8) Mechanical Control Sequence - Autopilot Disengaged

(a) Initially, the actuator piston is fixed by the centering springs. With the autopilot not engaged (armed), SV1 is open, SV2 is closed; detent pistons are disengaged, and the internal crank is free to move within the piston cavity. The output position LVDT provides internal crank position to the FCC for autopilot synchronization to surface position. The FCC commands through the EHSV cause the actuator piston to follow the internal crank so that the crank remains centered within the cavity. The actuator piston position LVDT nulls the command signal to stop the actuator piston.

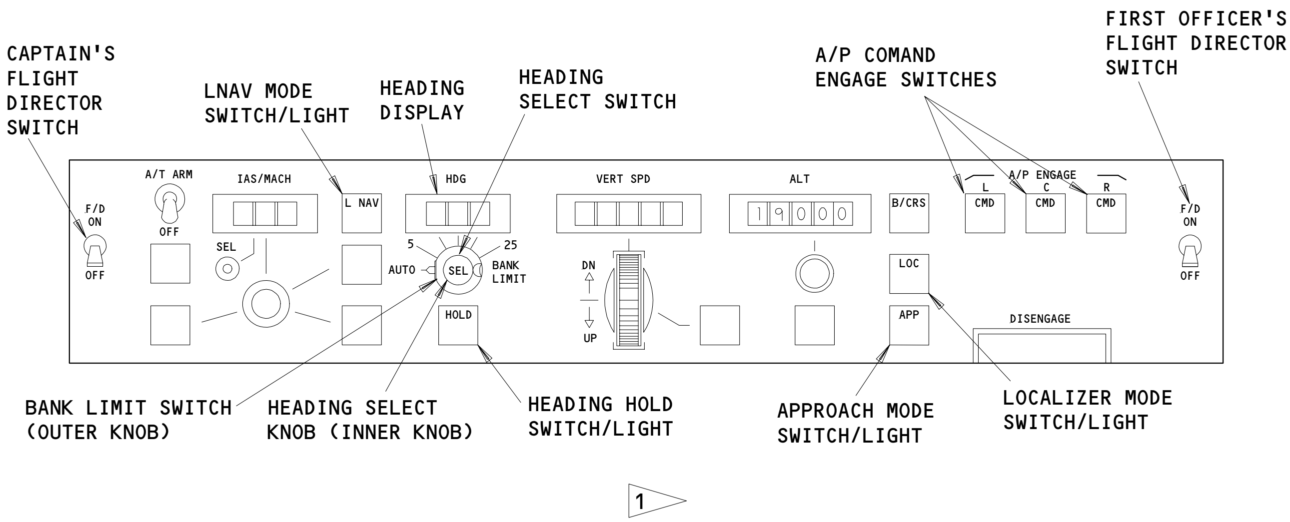
(9) Autopilot Control Sequence - Autopilot Engaged

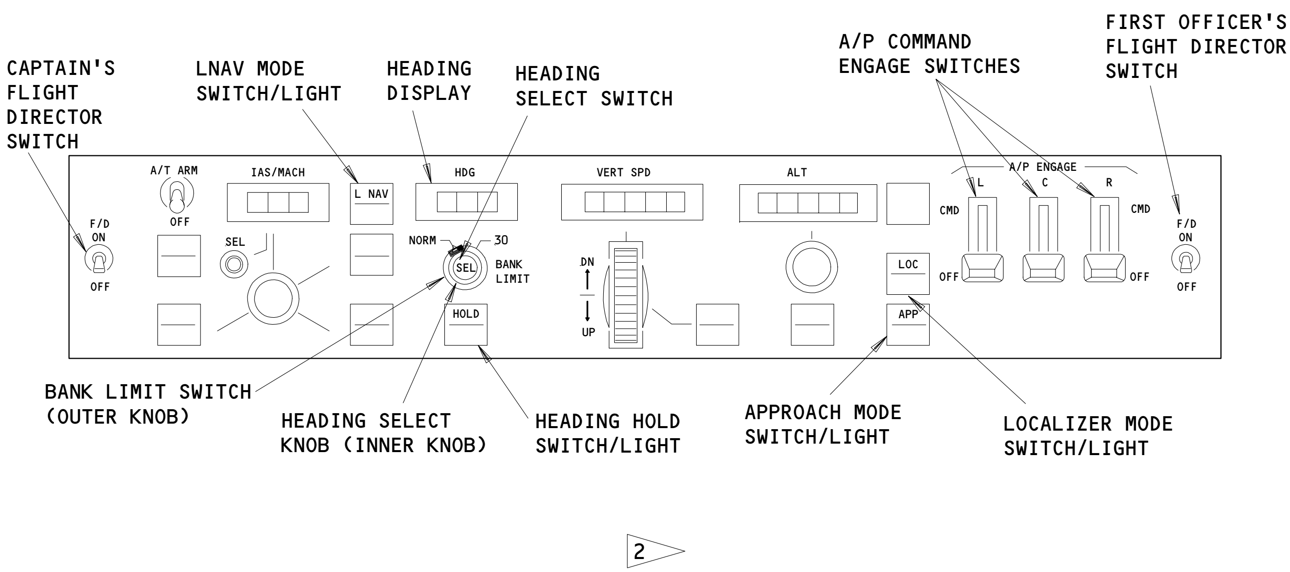
(a) With the autopilot engaged, SV1 and SV2 are open, the detent pistons are pressurized and the internal crank is clamped in the center of the actuator piston. When the EHSV receives a command from the FCC, hydraulic pressure is ported to one end of the actuator piston. The detent pistons carry the internal crank with the actuator piston to its commanded position. The output crank moves the linkage to the surface power control actuators and the output position LVDT sends position information back to the FCC to null the command signal and stop surface movement.

(10) Camout

(a) Camout occurs when the output crank position (surface position) does not correspond to the actuator piston position (servo postion) as sensed by differing LVDT outputs. Causes of camout are mechanical jamming of output linkage, pilot override of an engaged autopilot channel, or mechanical override of one autopilot channel by the remaining channels during multi-channel operation.

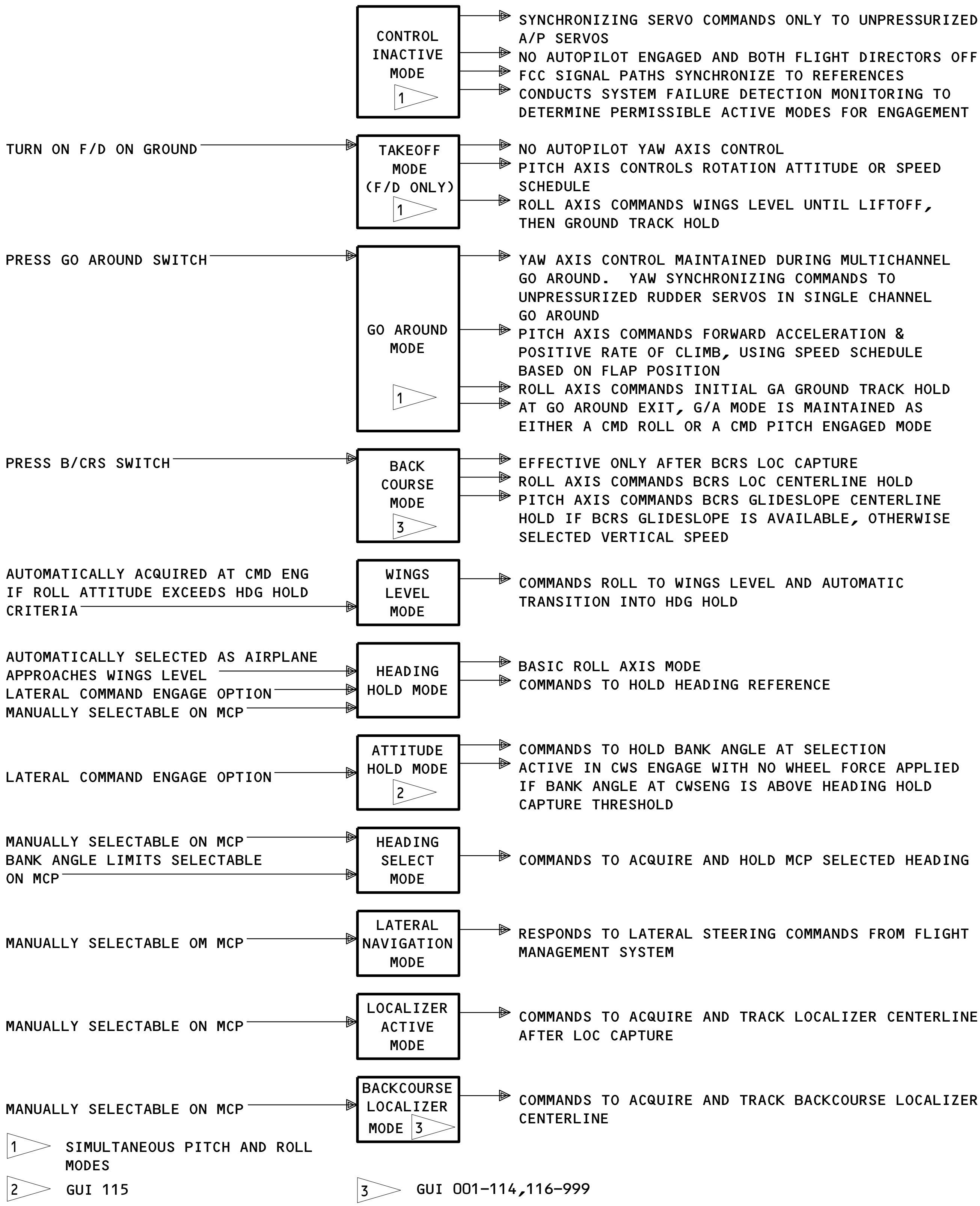
3. Operation
   A. Functional Description
      (1) Mode Control Panel - Roll and Yaw Controls (Fig. 6)
         (a) Flight Director ON/OFF Switches
            1) Two F/D toggle switches control the left and right flight director displays.
         (b) Command Engage Switches
            1) Three CMD ENGAGE switches (L, C, R) provide inputs for command engage logic to dedicated FCCs.
         (c) Other MCP controls associated with the roll and yaw modes.
            1) LNAV switch/light
            2) HDG HOLD switch/light
            3) HDG SEL knob
            4) HDG SEL switch (within HDG SEL knob)
            5) Selected HDG display
            6) LOC switch/light
            7) APP switch/light
            8) GUI 001-114, 116-999;
               B/CRS switch/light
      (2) Command Roll Mode Functions (Fig. 7)
         (a) Simultaneous Modes
            1) Simultaneous modes are those in which both pitch and roll control laws engage at the same time. These modes are Takeoff (T/O), and Go-Around (G/A).
         (b) Control Inactive
            1) Control Inactive is the configuration of the AFDS when no autopilot or flight director is engaged. During control inactive, the FCC processes roll angle commands from Runway Alignment, Heading/Track, and Localizer Processors to produce the Roll Attitude Command. This maintains a roll attitude reference when the A/P is not engaged and generates a roll to wings level command at CMDENG. The aileron servo command synchronizes to aileron position when the A/P is not engaged. This permits a smooth transition to A/P control at CMDENG. Processors not active in command computations synchronize to maintain a zero output.





1 GUI 001-114,116-999
2 GUI 115

Mode Control Panel - Roll and Yaw Controls
Figure 6



Command Roll Modes
Figure 7

(c) Takeoff Mode (F/D only)
  1) The takeoff mode has two submodes: on-ground and in-air. In the on-ground submode, the F/D provides a 8-degree nose-up command for initial climbout. The F/D also provides a wings level roll command.
  2) In the in-air submode, the F/D pitch command is referenced to the speed schedule with flap position and engine failure corrections. The F/D roll commands provide ground track hold after lift-off. There is no F/D yaw axis control.

(d) Go-Around
  1) The AFDS provides commands for a positive rate of climb using a speed reference schedule based on flap position. Roll channel commands initiate go-around (G/A) ground track hold. This mode is manually selected by pressing the go-around switch on the throttles.

(e) GUI 115;
Lateral Command Engage Option (LCEO)
  1) The lateral command engage option (LCEO) is selected by grounding a dedicated pin on the FCC. The baseline configuration is LCEO inoperative (pin ungrounded). In this configuration, the A/P engages into HDG HLD mode at CMDENG regardless of the bank angle (B/A). With LCEO operative (pin grounded) the A/P engages into HDG HOLD if the B/A is equal to or less than 5 degrees, and into Roll Attitude Hold if the B/A is greater than 5 degrees.

(f) Autopilot Mode Engage Option (APMEO)
  1) The APMEO is enabled by grounding a dedicated pin on the FCC. The baseline configuration is APMEO enabled. This option causes the A/P to couple smoothly into the path command modes being used by the F/D when the A/P engages in CMD (except during takeoff or go-around). The G/S ACTIVE, LOC ACTIVE, and G/A modes transition to V/S and HDG HLD modes and rearm the G/S and LOC if they were armed or active before A/P CMD ENGAGE.

(g) Heading Hold
  1) Heading hold is the basic roll axis mode. It is automatically selected when the F/D only mode is initially chosen or when any A/P channel is engaged in single-channel CMD (both F/D off). Heading hold can also be manually selected by pressing the HOLD switch/light on the MCP.

(h) GUI 115;
Attitude Hold
  1) Attitude hold is in operation when the AFDS is initially engaged, the LCEO is operative and the B/A is greater than 5 degrees. If the autopilot is engaged, or the flight director is on, with the B/A greater than 30 degrees, the A/P rolls the airplane back to 30 degrees and holds that attitude.

(i) Heading Select Mode
  1) GUI 001-114, 116-999;

heading select mode acquires and maintains the heading selected on the MCP. It has 6 selectable bank angle limits: AUTO and 5°-25° in 5 degree steps. The AUTO B/A limit varies from 15 to 25 degrees as a function of true airspeed and heading error. The heading select mode can be manually called by pushing the HDG SEL switch on the MCP.

2) GUI 115;
heading select mode acquires and maintains the heading selected on the MCP. It has two selectable bank angle limits: NORM and 30°. The NORM position selects a B/A based on true airspeed and heading error with a maximum angle of 30 degrees. The heading select mode can be manually called by pushing the HDG SEL switch on the MCP.

(j) Lateral Navigation Mode (LNAV)

1) The LNAV mode responds to FMC lateral steering commands to acquire and track selected flight plan paths. It is manually selected by pressing the LNAV switch/light on the MCP.

(k) Localizer Active Mode

1) The LOC active mode provides aileron commands to acquire and track the centerline of the localizer beam during all phases of single or multi-channel ILS approach. The A/P yaw channel provides rudder commands during multi-channel approaches to maintain runway alignment during rollout.

(l) GUI 001-114, 116-999;
Backcourse

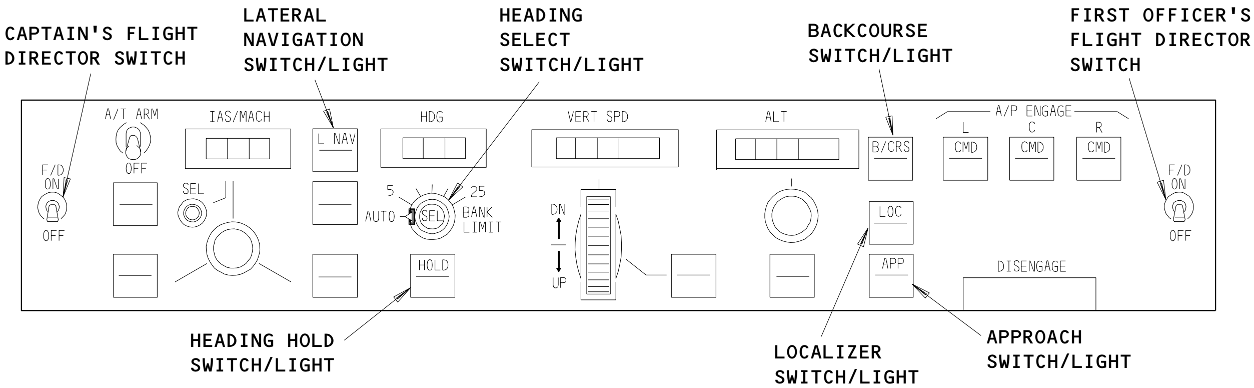
1) The B/CRS switch operates independently of A/P or F/D to control ILS displays on the ADI except after LOC capture. After LOC capture, the roll channel provides commands to track the B/CRS LOC centerline. The pitch channel provides commands to track the G/S centerline if G/S is available. The pitch channel will stay in the previously engaged mode after the B/CRS switch is pushed.

(m) GUI 001-114, 116-999;
Backcourse Localizer Mode

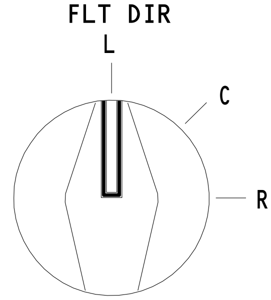
1) The B/CRS LOC mode provides corrected A/P and F/D commands to acquire and track the centerline of the backcourse localizer beam. It is automatically initiated whenever the airplane is within the computed capture band and the B/CRS LOC ARM state has been activated in the HDG HLD, HDG SEL or LNAV modes.

(3) Roll Flight Director Operation (Fig. 8)

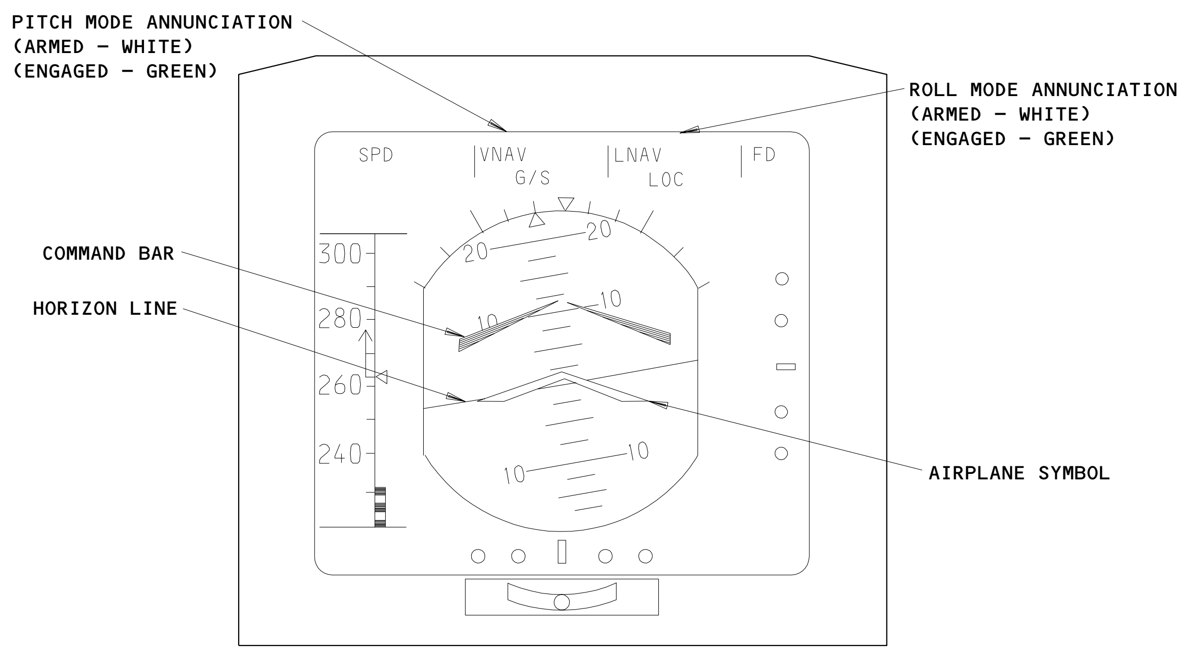
(a) The Flight Director (F/D) permits the pilot to fly the airplane without engaging the autopilot (A/P). The FCC computes F/D commands and displays these on the EADIs. The pilot controls the airplane by control column and control wheel movements in response to these commands.



MODE CONTROL PANEL

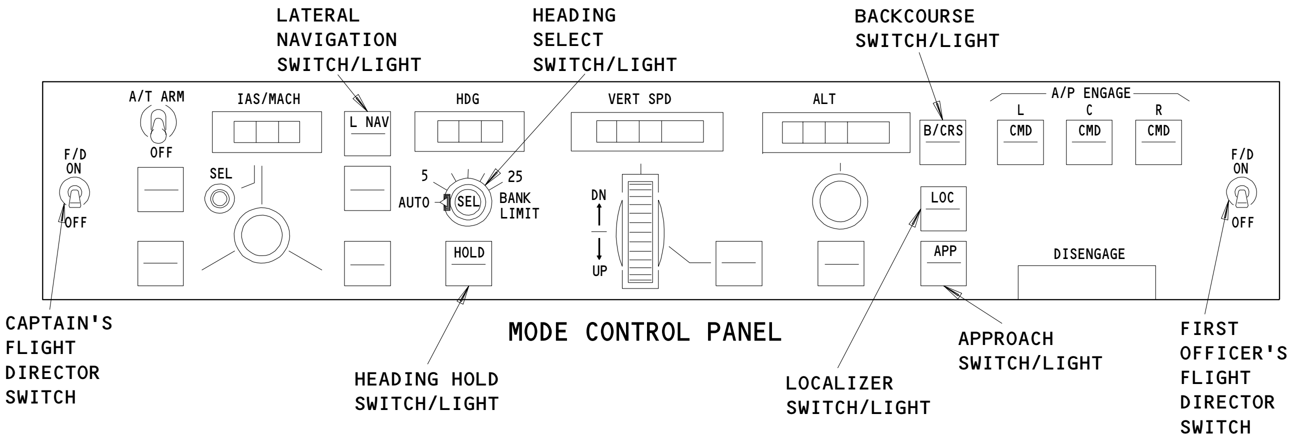


LEFT FLIGHT DIRECTOR SWITCH
(RIGHT SWITCH IS EQUIVALENT)

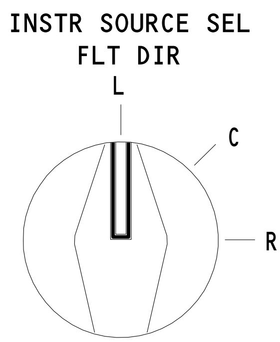


ELECTRONIC ATTITUDE DIRECTOR INDICATOR

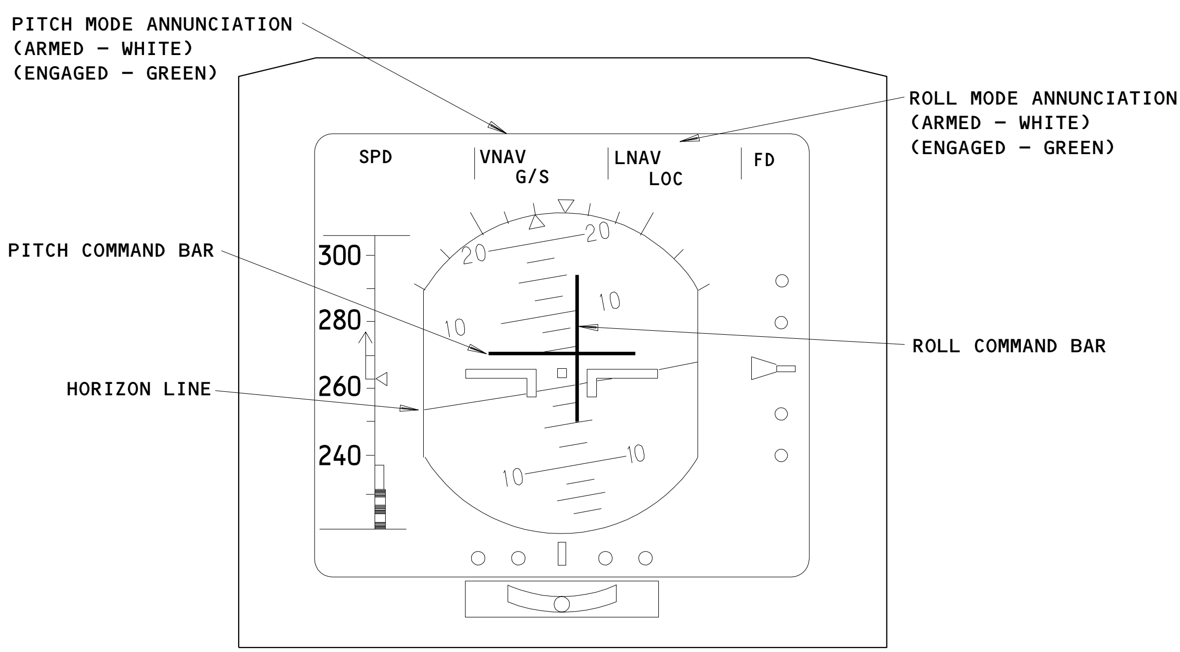
Flight Director Controls and Displays
Figure 8 (Sheet 1)



MODE CONTROL PANEL

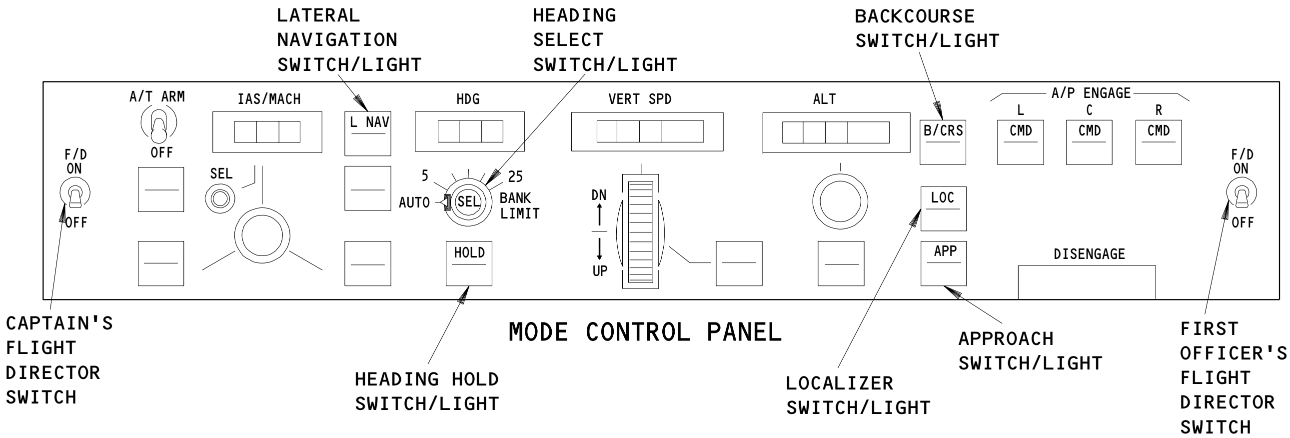


LEFT FLIGHT DIRECTOR SWITCH
(RIGHT SWITCH IS EQUIVALENT)

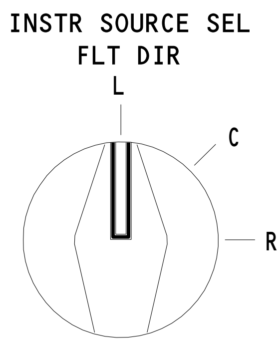


ELECTRONIC ATTITUDE DIRECTOR INDICATOR

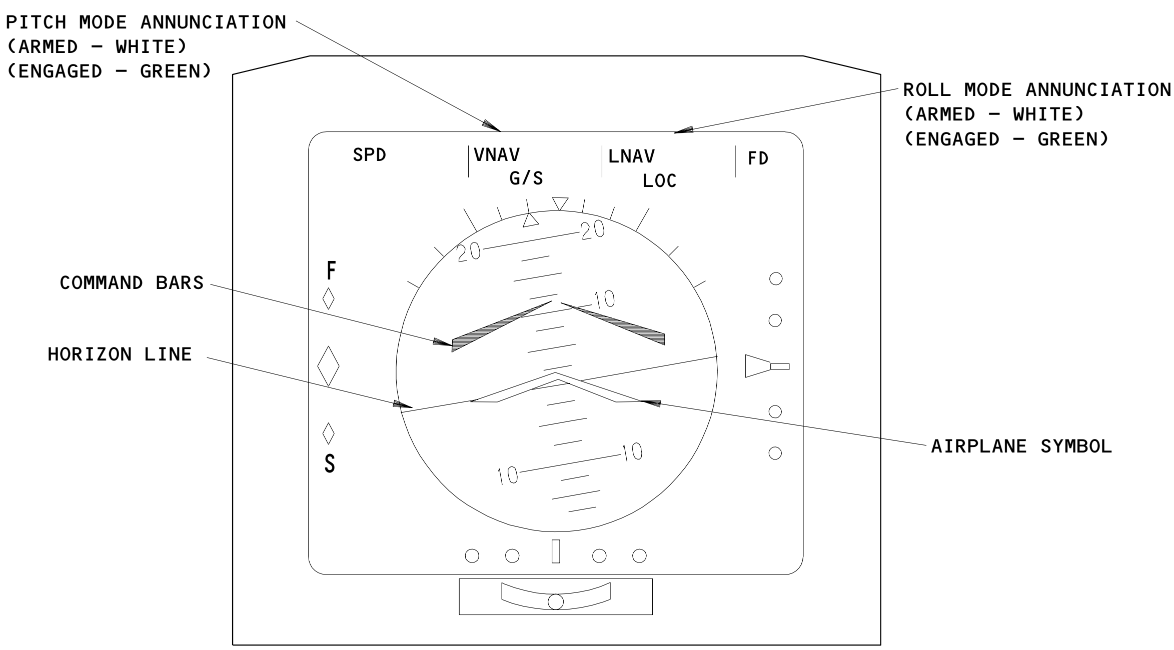
Flight Director Controls and Displays
Figure 8 (Sheet 2)



MODE CONTROL PANEL

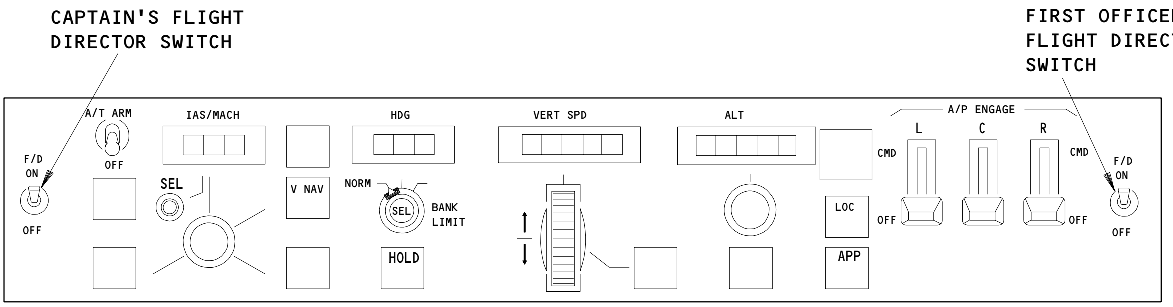


LEFT FLIGHT DIRECTOR SWITCH
(RIGHT SWITCH IS EQUIVALENT)



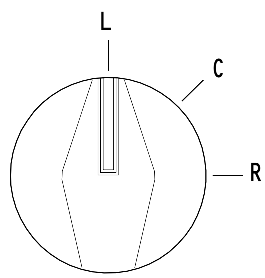
ELECTRONIC ATTITUDE DIRECTOR INDICATOR

Flight Director Controls and Displays
Figure 8 (Sheet 3)

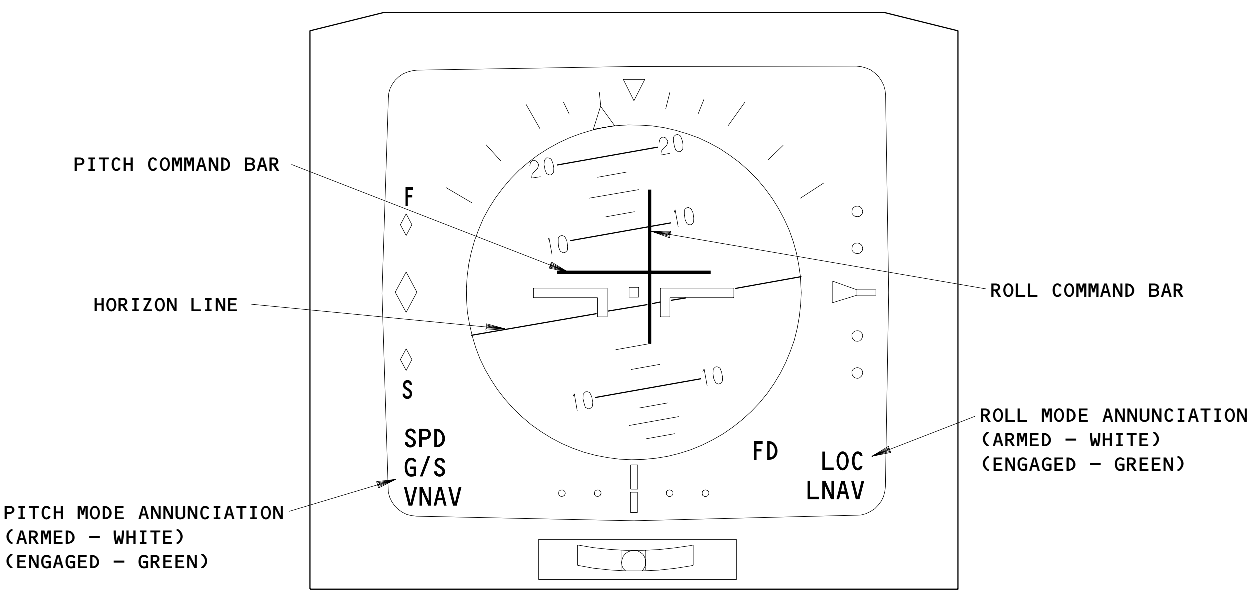


MODE CONTROL PANEL

INSTRUMENT SOURCE SEL
FLIGHT DIRECTOR



LEFT FLIGHT DIRECTOR SWITCH
(RIGHT SWITCH IS EQUIVALENT)



ELECTRONIC ATTITUDE DIRECTOR INDICATOR

Flight Director Controls and Displays
Figure 8 (Sheet 4)

(b) When on the ground, the F/D automatically engages into the takeoff-wings level mode when it is first switched on. It changes to the takeoff-track hold mode when the airplane reaches five feet radio altitude. During flight, if the A/P is engaged when the F/D is first switched on, the F/D engages into the current A/P mode.

(c) Manually selectable F/D modes are:

1) Heading Select (HDG SEL)
2) Heading Hold (HDG HLD)
3) Lateral Navigation (LNAV)
4) Localizer (LOC)
5) Approach (APP)
6) Go-Around (G/A)
7) GUI 001-114, 116-999;
Backcourse Localizer (B/CRS LOC)

(d) Flight Director Controls and Displays

1) Flight Director operation and mode selection are accomplished by switches and switch/lights on the Mode Control Panel. The captain's and first officer's F/D switches are paralleled, so either can be used to turn the F/D on. Both must be off to disable the F/D. Each F/D switch, however, independently controls the F/D displays on the corresponding EADI. The L, C, or R FCC may be selected as data source for the captain's or first officer's F/D displays by the respective FLT DIR switch (P1-1 or P3-3 panels).

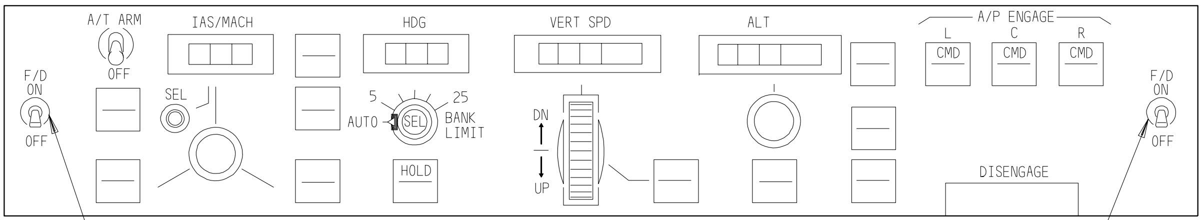
(4) Takeoff Mode (F/D only) (Fig. 9)

(a) Preparatory Operations

1) Before the takeoff mode can be entered, either or both flight directors must be on and all A/P CMD ENGAGE switches must be off. The F/D only configuration is verified by the AFDS status FMA on the EADI. The IAS/MACH indicator is set at V2 (climb speed); HDG is set to the heading of the initial flight leg after takeoff; and ALT is set for the initial clearance altitude.

(b) Flight Director Display

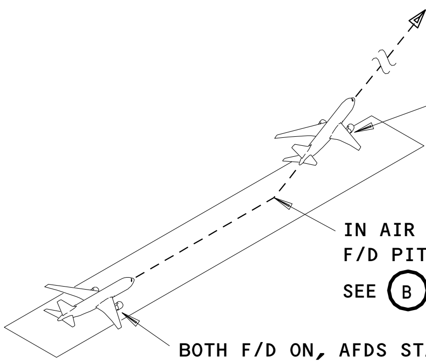
1) Initially (A) the EADI shows wings level and rotation attitude of 8 degrees nose up. The status FMA shows F/D and both the pitch and roll operative mode FMAs show TO.
2) On reaching 5 feet altitude (B), roll channel commands provide track hold and the pitch rotation attitude command reduces linearly toward zero within 3 seconds. The roll operative mode FMA changes to TRK HLD.



CAPTAIN'S FLIGHT DIRECTOR SWITCH

FIRST OFFICER'S FLIGHT DIRECTOR SWITCH

MODE CONTROL PANEL

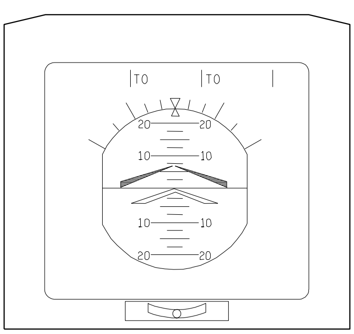


A/P MAY BE ENGAGED IN CMD
ROLL FUNCTIONAL MODE MAY BE HDG HOLD, HDG SELECT, ATT HLD OR LNAV

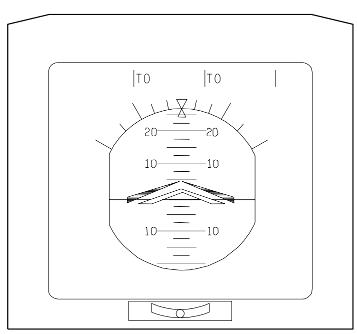
TAKEOFF IN AIR PROVIDES ROLL CMDS FOR GROUND TRACK HOLD
PITCH F/D CMDS TO MAINTAIN ATTITUDE FOR SAFE CLIMB SPEED
ROLL OPERATIVE FMA SHOWS TRK HOLD
SEE C

IN AIR CONDITION DURING ROTATION WHEN RA SENSES HR ABOVE 5 FT CLIMBING
F/D PITCH CMD ROTATION ATTITUDE EASY OFF TO ZERO CMD AFTER 3 SEC
SEE B

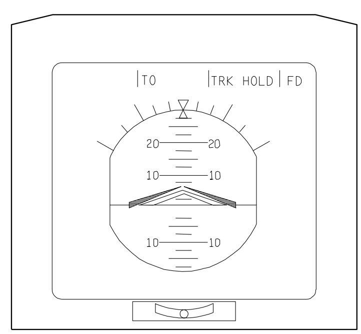
BOTH F/D ON, AFDS STATUS FMA SHOWS F/D, ROLL OPERATIVE FMA SHOWS T/O
ROLL F/D CMDS FOR WINGS LEVEL, SEL HDG TO INITIAL DEPARTURE HEADING
PITCH F/D CMD SHOWS 8 DEGREE NOSE UP ROTATION ATTITUDE
SEE A



EADI DISPLAY
A

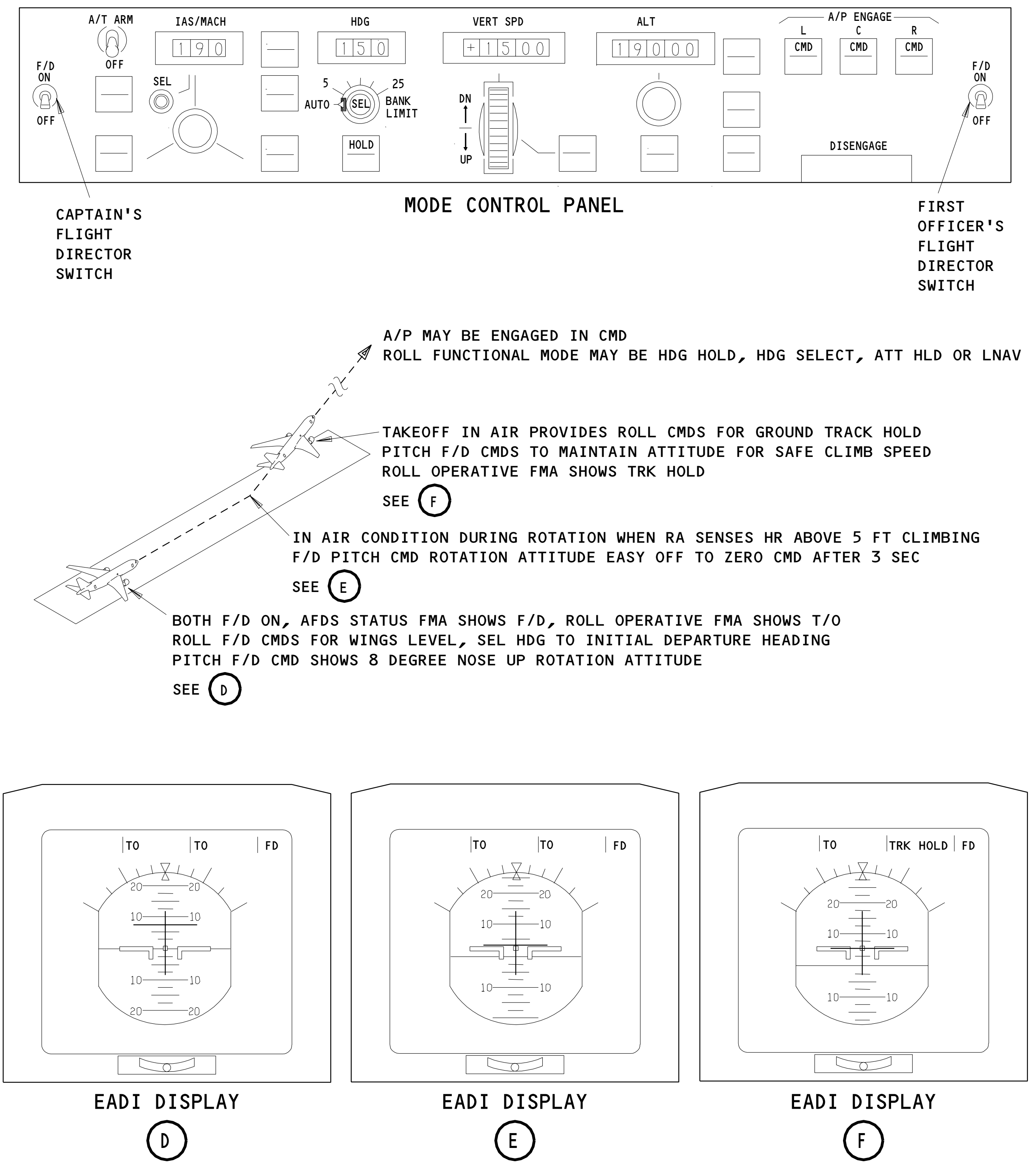


EADI DISPLAY
B

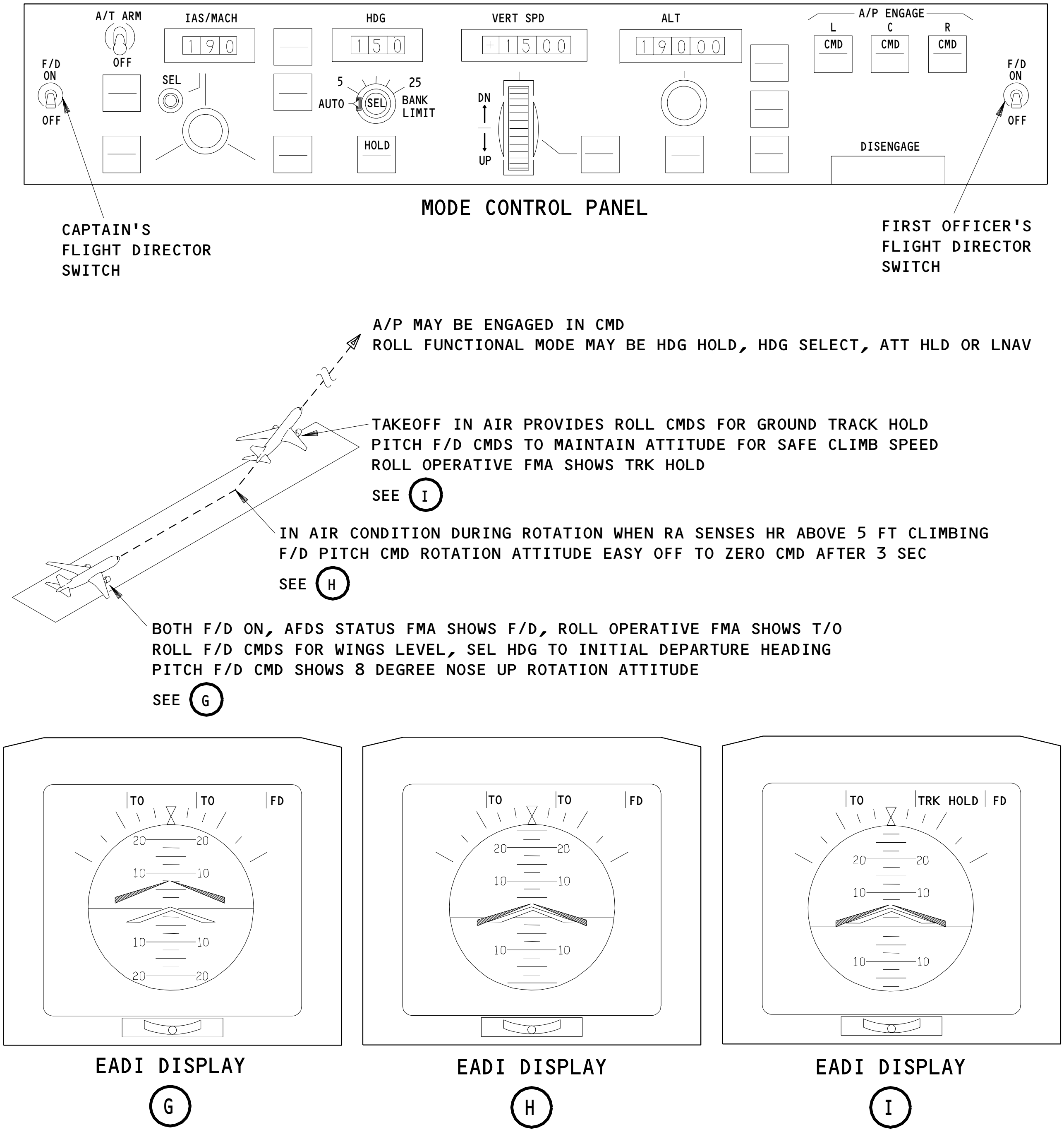


EADI DISPLAY
C

Takeoff Mode (F/D Only)
Figure 9 (Sheet 1)

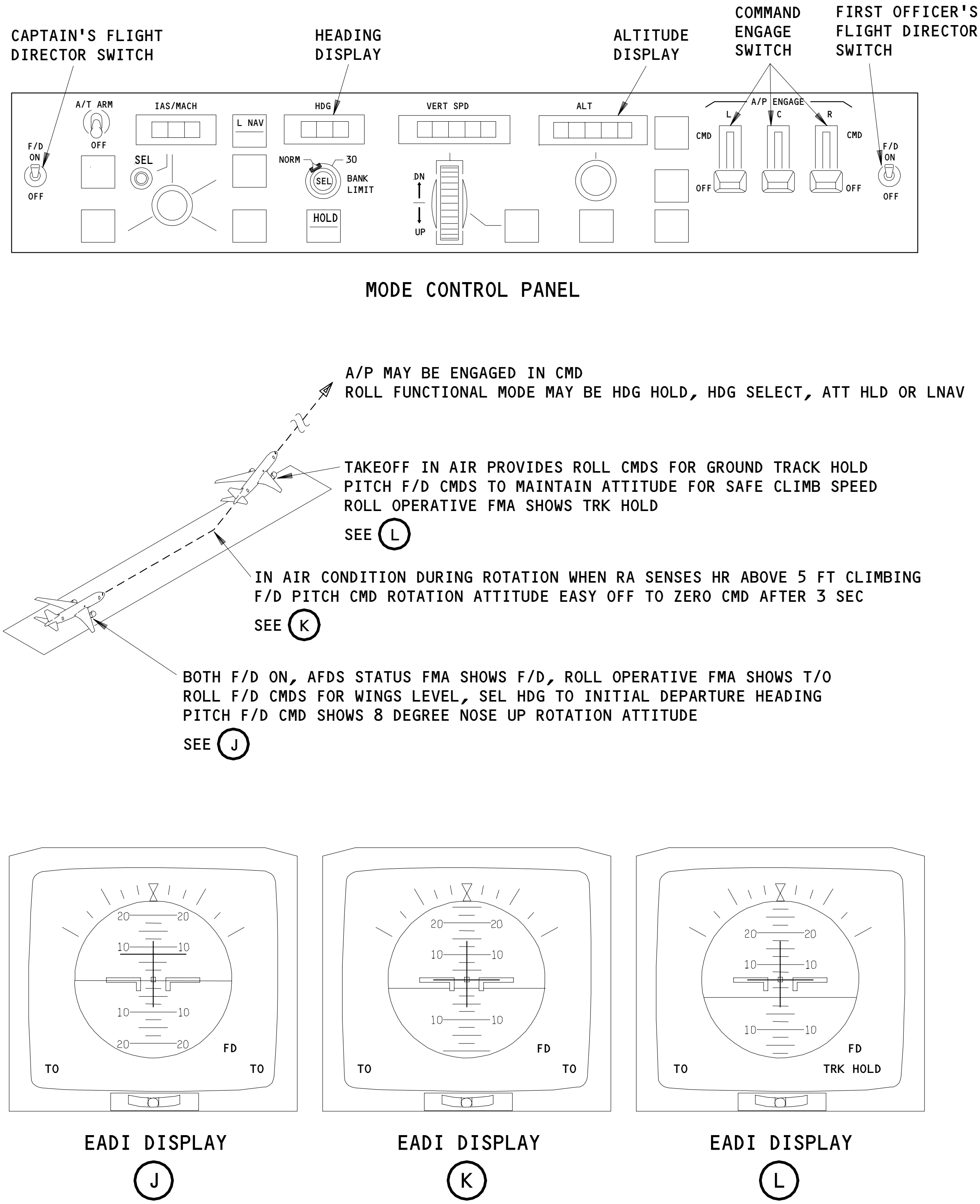


Takeoff Mode (F/D Only)
Figure 9 (Sheet 2)



Takeoff Mode (F/D Only)
Figure 9 (Sheet 3)

C33008



Takeoff Mode (F/D Only)
Figure 9 (Sheet 4)

3) The F/D enters the takeoff speed engage mode (C) which produces pitch attitude commands to maintain a safe climbout speed.

(c) Mode Transitions

1) From the takeoff mode, transition into the following modes is possible: roll channel, HDG HLD, HDG SEL, LNAV ARM, and LNAV capture; pitch channel, ALT HLD, V/S, FLCH, and VNAV. The takeoff mode ends when the A/P is engaged, when both F/Ds are switched off, or when transition to the above modes are accomplished.

(5) Autopilot/Flight Director System Roll and Yaw Modes Logic (Fig. 10)

(a) Takeoff Mode Logic

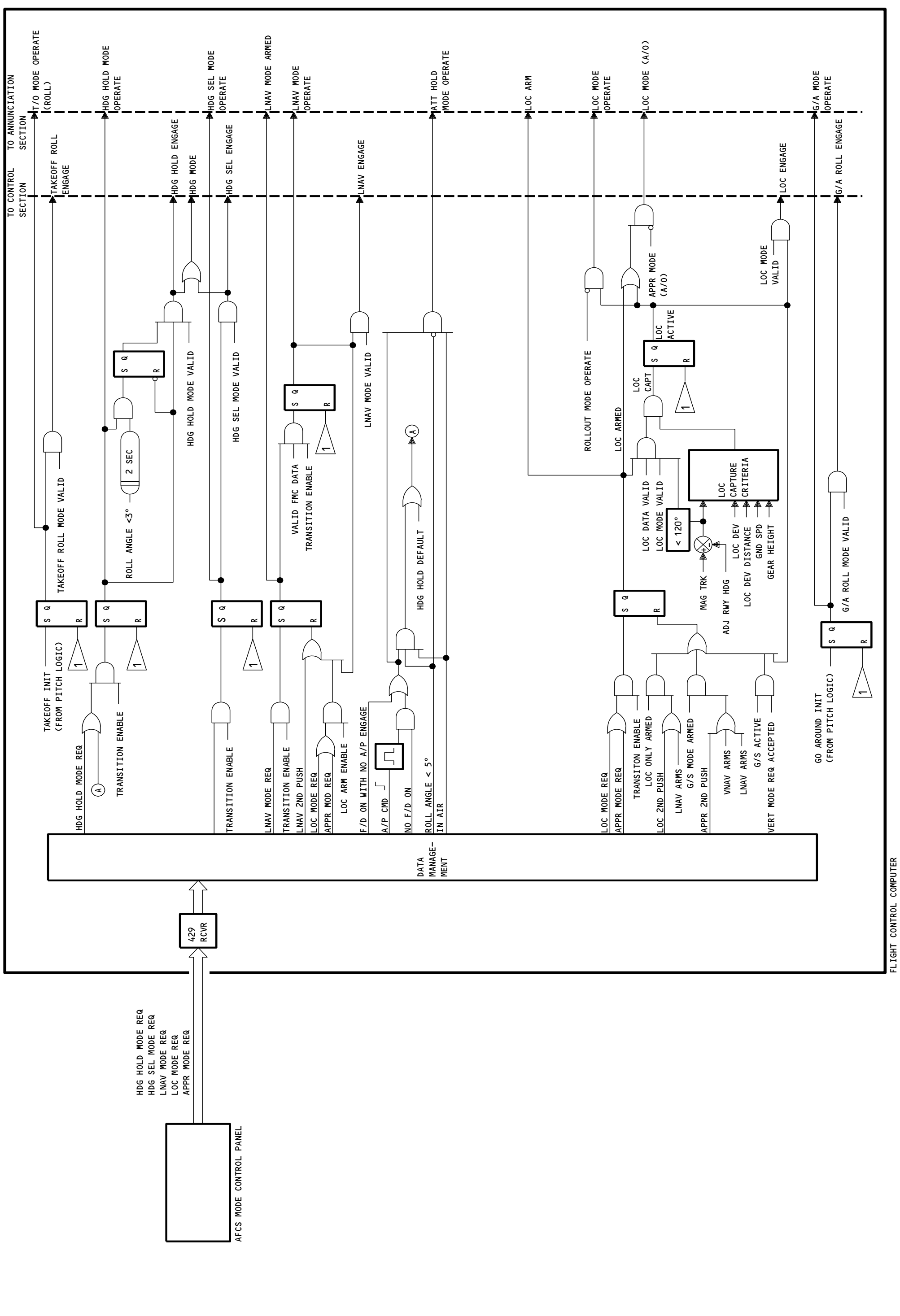
(b) When either Flight Director is switched ON, F/D engage logic enables the takeoff mode. The on-ground condition generates roll attitude washout logic which produces a roll F/D command of wings-level. At five feet radio altitude, the Track Hold mode is automatically engaged. This changes the roll F/D commands from wings-level to track hold.

(6) Autopilot/Flight Director system Roll and Yaw Control Signals Flow (Fig. 11)

(a) Takeoff Mode Control Signal Flow

(b) When the F/D is turned ON and the airplane is on the ground, the F/D command processor integrator is set to wings-level. The difference between roll attitude and the integrator output is roll attitude error. The roll error is combined with body roll rate and pitch attitude to produce the roll F/D bar command which is applied to the EADI. The wings-level command is provided until the in-air condition is sensed.

(c) The in-air condition occurs when the radio altitude is five feet or greater. At this time, adjusted track is used as the input to the heading hold synchronizer and the turn direction command processor. However, heading hold logic disables the integrator in the heading hold synchronizer so that only adjusted track is used to produce the F/D heading command. The F/D heading command is applied to the roll F/D command processor when the roll attitude washout logic goes to zero at five feet radio altitude. The input is limited by the position of the bank angle (B/A) limit switch on the MCP.

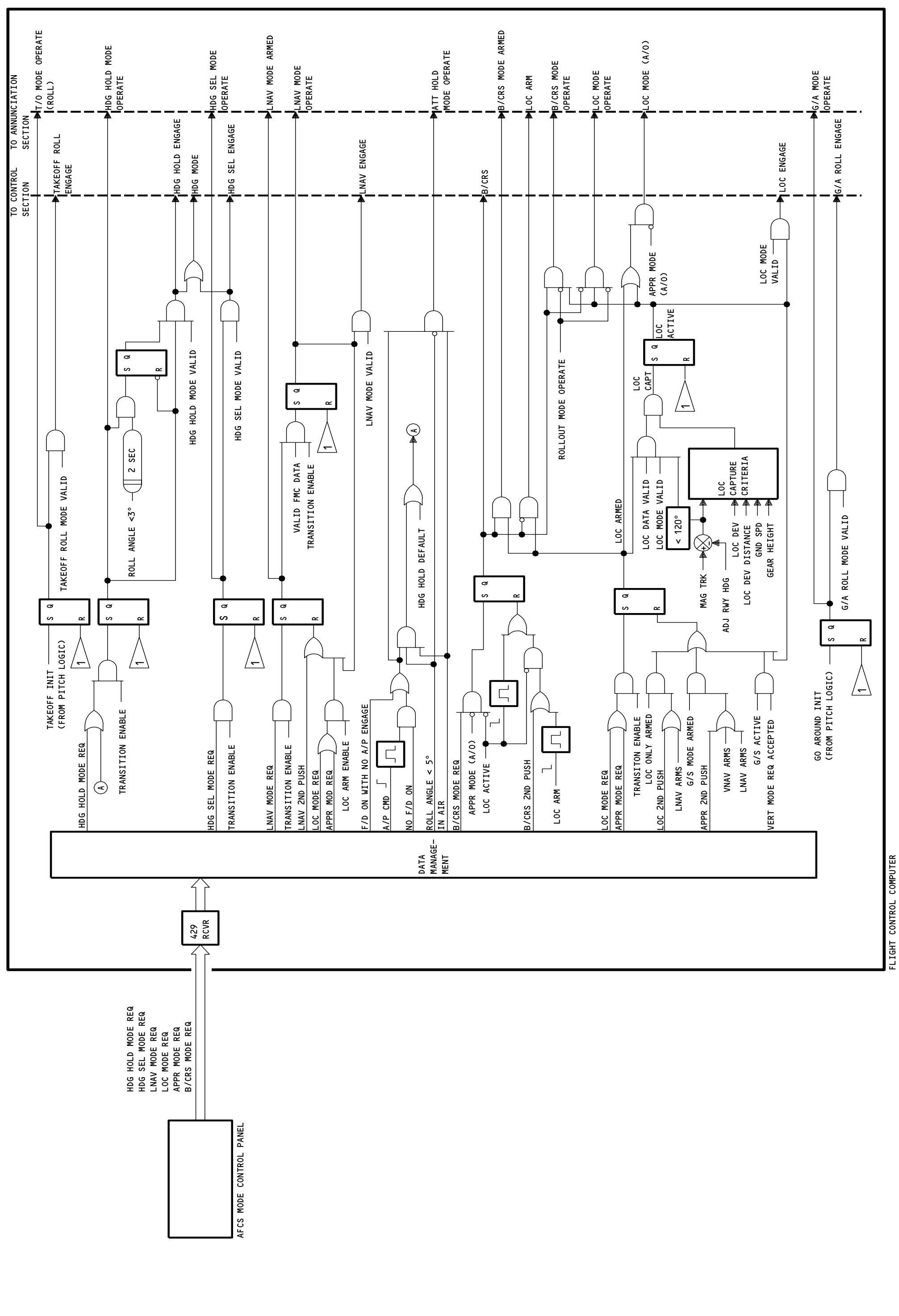


1 SYSTEM OFF OR ENTER ANOTHER MODE

HDG SEL MODE REQ

Autopilot/Flight Director System Roll and Yaw Modes Logic
Figure 10 (Sheet 1)

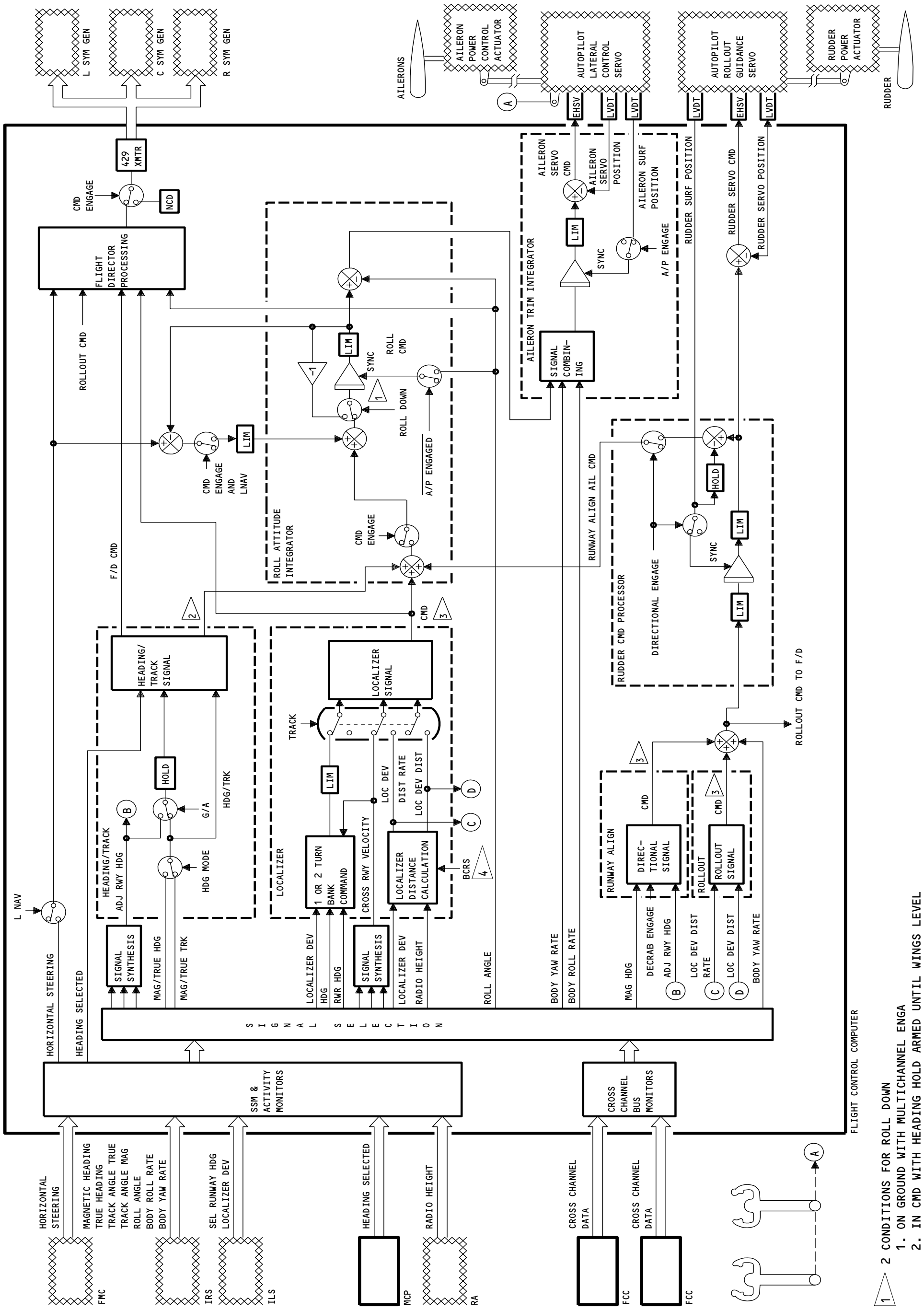
303416



1 SYSTEM OFF OR ENTER ANOTHER MODE

Autopilot/Flight Director System Roll and Yaw Modes Logic
Figure 10 (Sheet 2)

303417



1 2 CONDITIONS FOR ROLL DOWN
1. ON GROUND WITH MULTICHANNEL ENGA
2. IN CMD WITH HEADING HOLD ARMED UNTIL WINGS LEVEL

2 SYNC UNTIL HDG HOLD, HDG SEL, OR G/A

3 SYNC UNTIL MODE ENGA

4 GUI 001-114,116-999

Autopilot/Flight Director System Roll and Yaw Control Signal Flow Schematic
Figure 11

A70861

(7) Output Signal Synchronization

(a) Aileron Servo Command

1) Aileron surface position from the ALCS is applied to the Aileron Trim Integrator (ATI). A synchronization loop established when the A/P is not engaged, synchronizes the roll servo command to ±65 degrees of wheel rotation. With the A/P engaged, the limiter is set at ±18 degrees in single channel operation and ±40 degrees in multi-channel operation. Any servo command in response to aileron position change is supplied to the EHSV. Since the ALCS is hydraulically armed, the servo actuator moves in response to the EHSV. The actuator piston LVDT output nulls the command. The EHSV centers but the servo actuator piston is off center and the detent pistons are retracted. When APENG occurs, detents clamp the crank at the off-center position.

(b) Flight Director Roll Command

1) Flight Director roll commands are computed continuously whether the A/P is engaged or not and whether the F/D is on or off.

2) The computed F/D roll command is transmitted to the EFIS. The F/D ON discrete is not present when F/D switches are OFF, therefore, EFIS displays are inhibited.

(c) Rudder Servo Command

1) Rudder surface position from the ARGS is applied to the rudder command processor. A synchronization loop is established when directional yaw steering (DYS) is not engaged. This synchronizes the rudder commands up to ±25 degrees. With DYS engaged, servo commands are applied to the EHSV. Servo actuator operation and LVDT output is similar to that described above for the ALCS. Output to the Runway Alignment Command Signal Processor (RACSP) is held at zero by synchronizing loop. When DYS is engaged, the RACSP receives only the rudder servo command.

(d) Computed Commands

1) The roll F/D command for aileron and the rudder command are individually synchronized in appropriate sections of the FCC. The source for F/D commands depends on the mode. Computed F/D commands are applied to the symbol generators and EADIs of the EFIS. During CMD ENG, a fixed bias is applied and the F/D command bars on the EADI are biased out of view.

(8) Heading Hold and Heading Select Modes (Fig. 12)

(a) The MCP controls and displays associated with the heading modes are identified on figure 12.

(b) The heading hold mode can be engaged by pressing the HDG HOLD switch/light. If the bank angle (B/A) is greater than the heading capture criteria threshold, the AFDS commands a roll to wings-level and then commands to heading hold. If the B/A is within the heading capture criteria threshold, the AFDS commands to heading hold when the HDG switch/light is pressed.

(c) Heading select mode is used to change to a new heading. The HDG SEL knob and switch on the MCP control the selected heading and the heading select mode. The AFDS commands generated depend on whether the SEL switch is pressed before or after the SEL knob is turned to provide the selected heading.

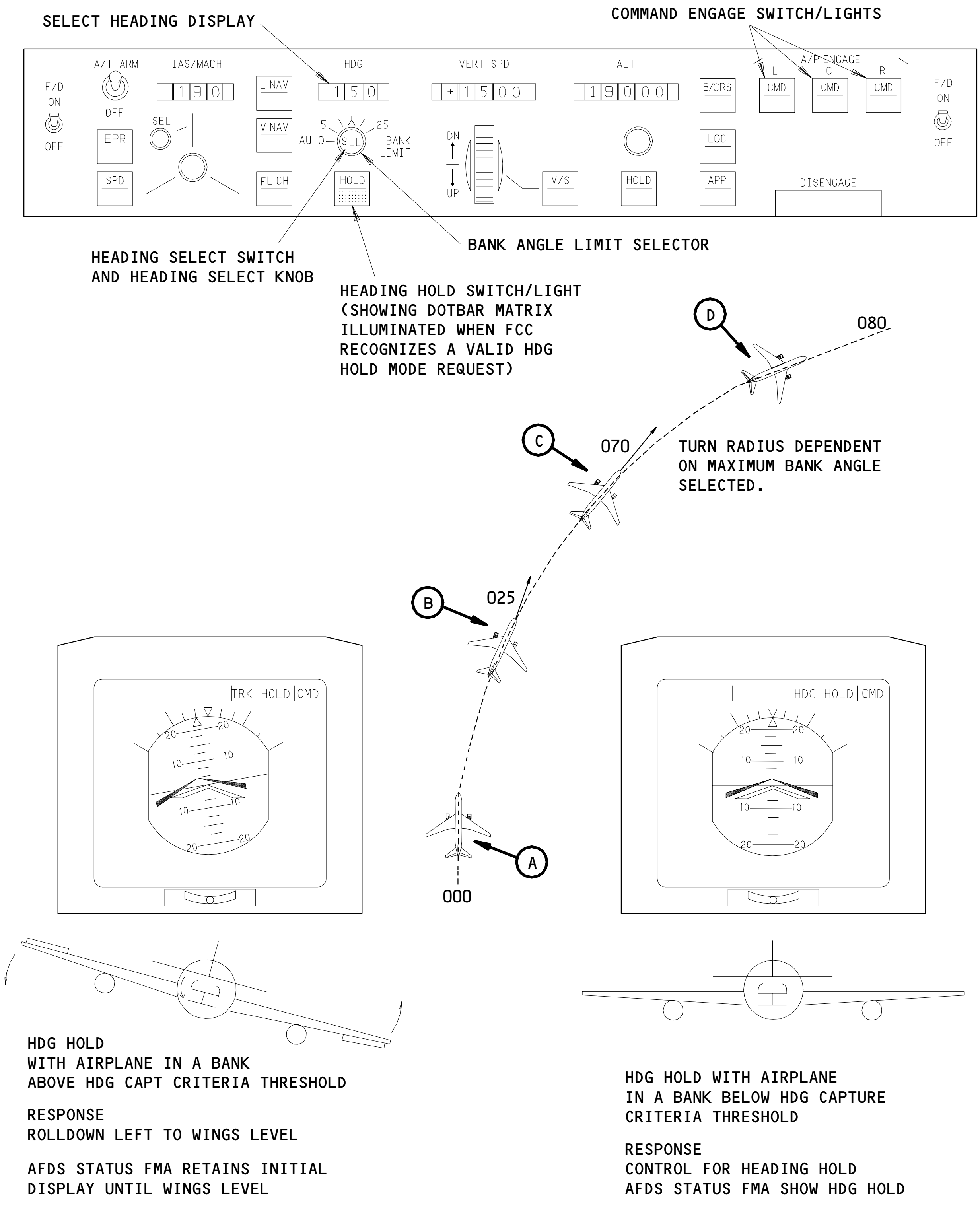
1) If the HDG SEL knob is rotated to the desired heading before the SEL switch is pressed, the AFDS commands a turn towards that heading in the direction of the smallest turn angle.

2) If the SEL switch is pressed first, the AFDS commands generated are always in the direction the HDG SEL knob is turned.

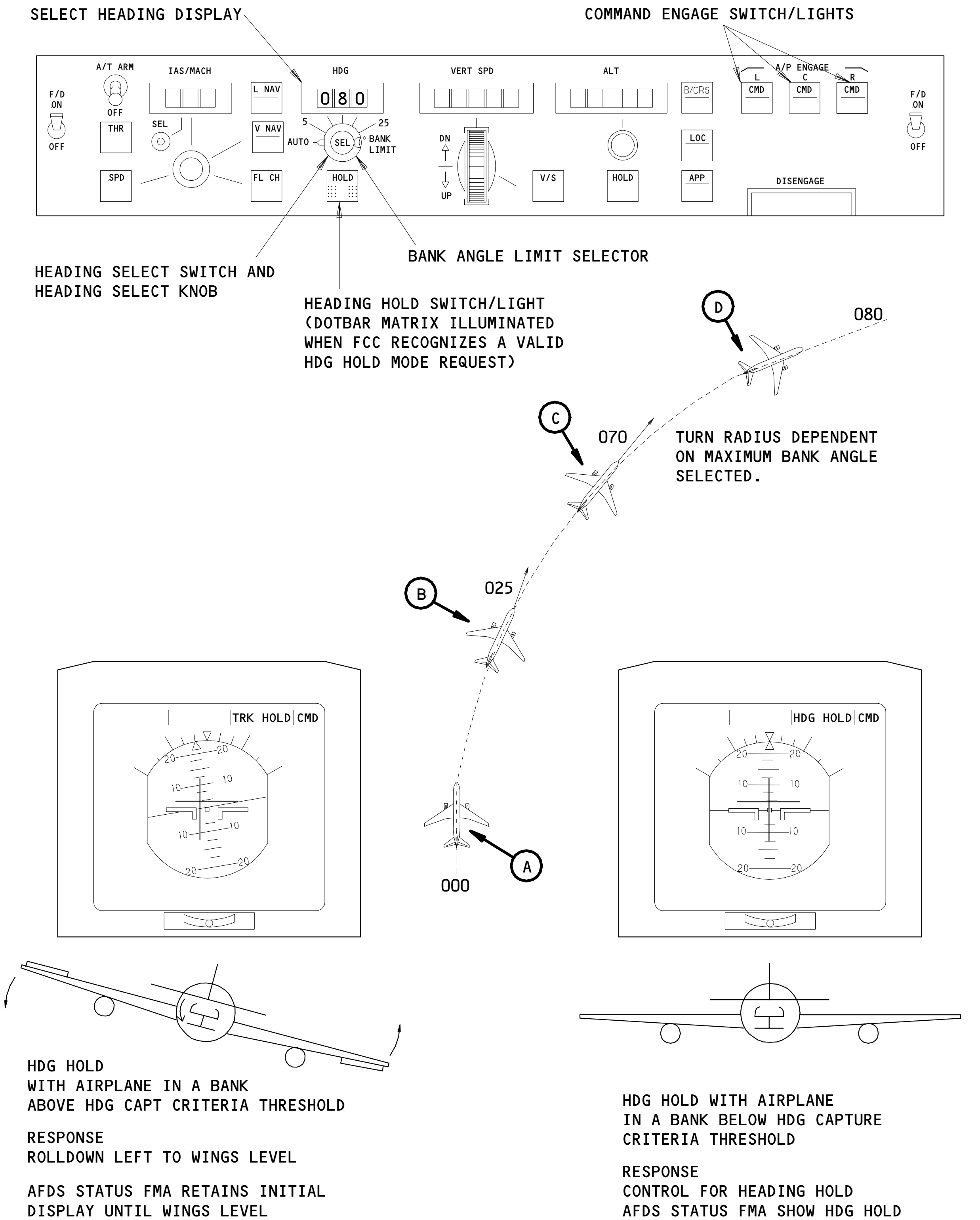
(d) Figure 12 shows an airplane at heading 000 (A) turning to a heading of 080 (D). After the SEL switch has been pressed, the airplane (B) rolls to a B/A proportional to the heading error but is limited by the B/A limit switch on the MCP. The F/D command bars on the EADI are centered indicating that correct roll command is being applied. At C, the heading error has reduced to 10 degrees right requiring a reduction in B/A to permit smooth rollout at 80 degrees. The EADI shows reduced roll attitude and roll left command. The heading of 080 at D is equal to the selected heading so the AFDS automatically engages into heading hold and maintains the selected heading.

(9) Heading Hold/Heading Select Mode Logic

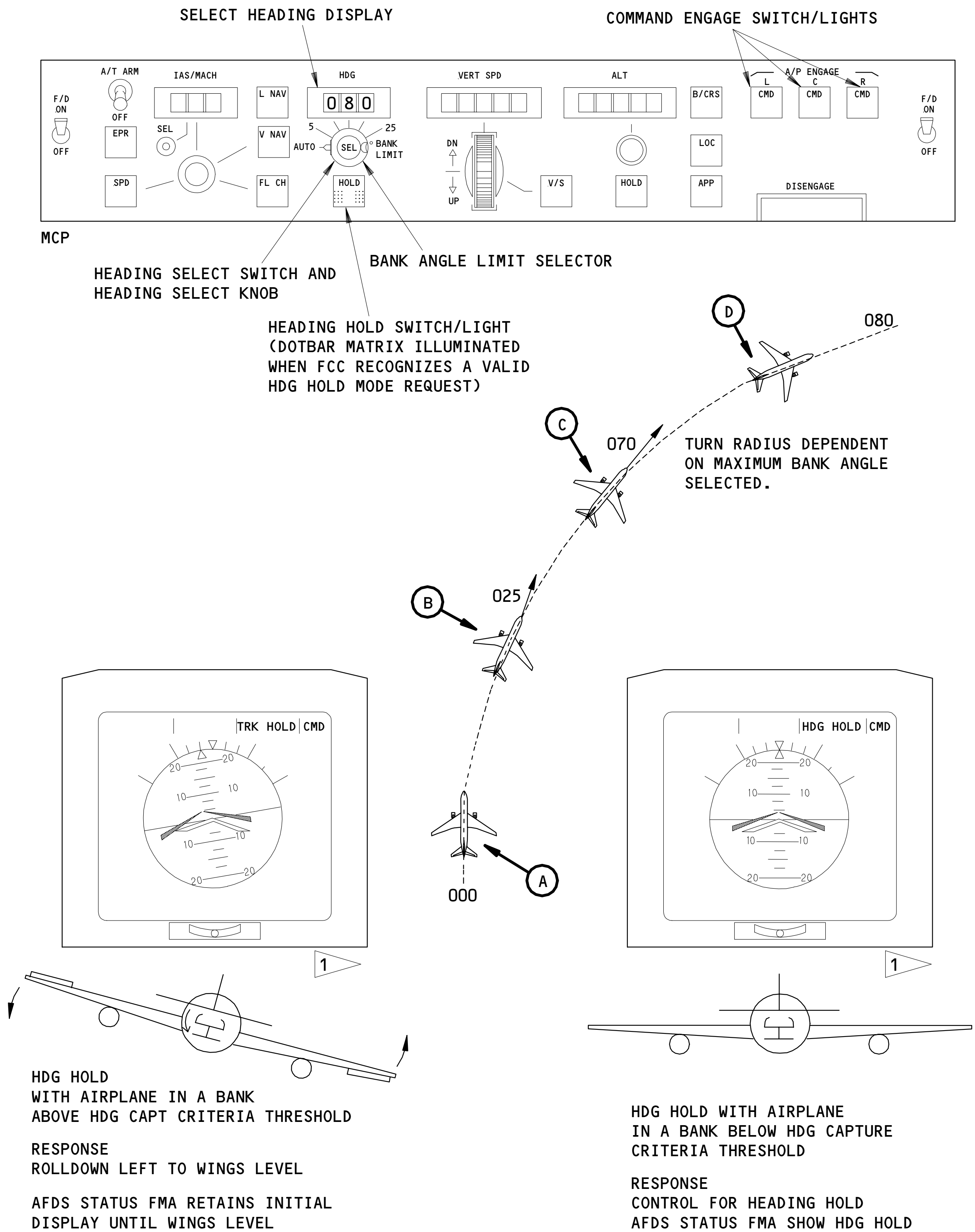
(a) The heading hold mode is initiated by pressing the HDG HOLD switch/light. If adjusted heading is valid, the heading hold mode armed logic is generated (Fig. 10). When the heading hold capture criteria logic is zero, indicating a B/A greater than the threshold, the roll attitude washout logic is generated. When the heading capture criteria logic goes high, a heading hold request is generated. One second later, heading hold engage is generated and the washout logic goes to zero. The heading hold mode logic is generated and this initiates the command heading/track control and the heading proportional command logic signals.



Heading Select/Heading Hold Mode
Figure 12 (Sheet 1)

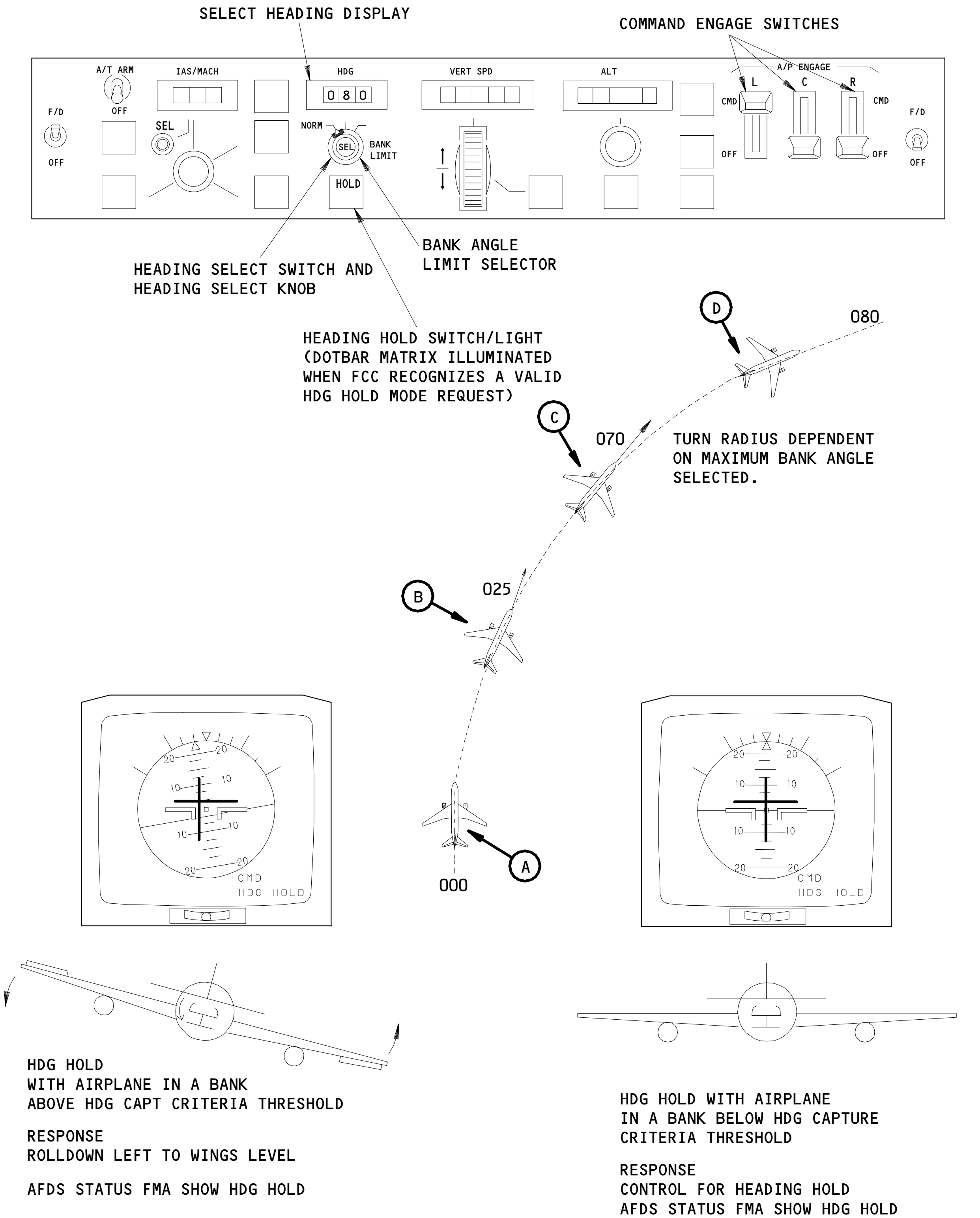


Heading Select/Heading Hold Mode
Figure 12 (Sheet 2)



1 THE EADI CONFIGURATION SHOWN IS AN EXAMPLE OF AN AWW EADI

Heading Select/Heading Hold Mode
Figure 12 (Sheet 3)



Heading Select/Heading Hold Mode
Figure 12 (Sheet 4)

(b) Heading select mode enable logic is generated by pressing the HDG SEL switch. The heading select engage valid logic is generated when a selected heading is entered through the HDG SEL knob. These two logic signals produce the heading select mode engaged logic which then produces the heading/track and heading proportional logic.

(10) Heading Hold/Heading Select Signal Processing

(a) Heading/Track command processing computes the heading proportional command, the heading integral command, and the flight director heading command. It permits synchronization of heading computation software when the A/P is not engaged in a heading control mode. It selects the correct heading reference for the heading control mode in use and ensures that turn commands produce the correct turn direction for the mode in use. Gain programming is applied as required (Fig. 11).

(b) Initial Conditions - Heading Hold Mode

1) When the airplane is on a given heading, magnetic heading is applied as roll attitude reference to both the autopilot and flight director turn direction controls. The heading command feedback from the heading command processor (the heading proportional command) is also magnetic heading. The output of both turn direction controls is zero and the heading integral command and the flight director command are both zero (i.e., no aileron command and F/D roll bar is centered).

(c) Roll F/D Command - Heading Select Mode

1) When the HDG SEL button is pressed, the heading select mode is engaged and the selected heading from the MCP HDG SEL knob is compared with the magnetic heading. The difference is applied to both turn direction controls. Output from the F/D turn direction control is scaled to produce the F/D heading command. This is then modified by other inputs: roll attitude; roll attitude rate; and, if LNAV is engaged, horizontal steering commands. This produces the Roll F/D command which drives the roll command bar on the EADI.

(d) Aileron Servo Command

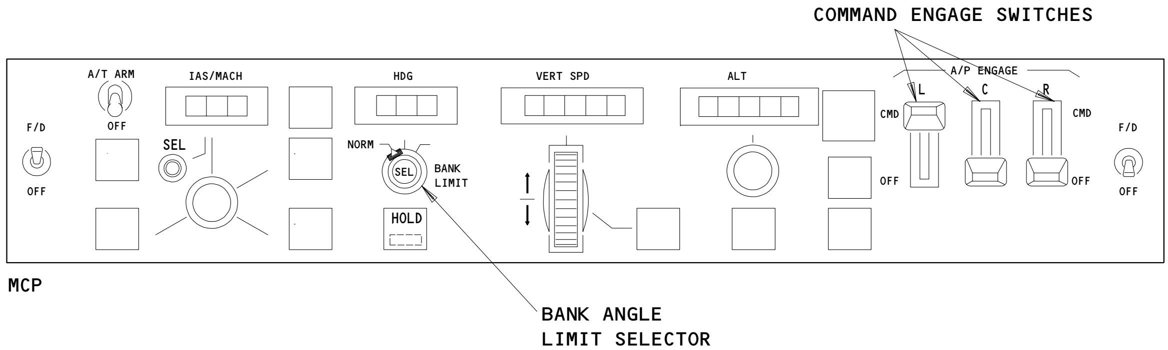
1) Selected heading is integrated and limited by the B/A switch setting and the heading rate command filter. This output is summed with heading angle command and becomes the heading proportional command. The integrator output is also summed with magnetic heading and developed into the heading integral command. Localizer and runway alignment commands are added, when appropriate, and the roll rate integral command and delta roll attitude command are produced. The latter is differentiated and added to the former to produce the roll attitude command.

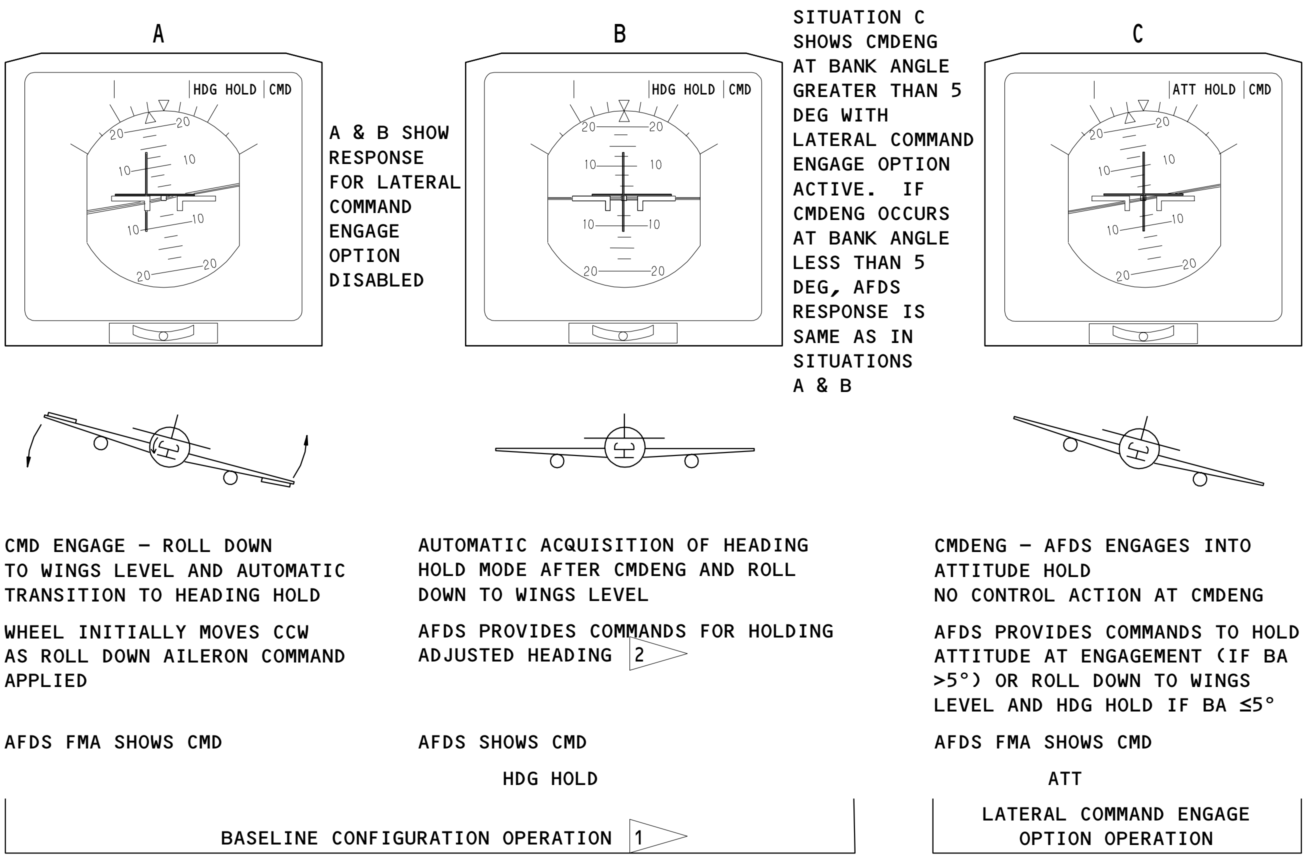
2) Roll attitude and roll attitude command are summed to produce roll attitude error. This error signal is summed with body roll rate, differentiated and summed with aileron servo position feedback to produce wheel rate. This is then applied to the aileron trim integrator which sets max and min values for surface deflection. After integration, wheel rate becomes the aileron servo command which is applied to the electrohydraulic servo valve (transfer valve). The amplitude of the aileron servo command determines the rate of response (roll rate) while the duration of the command determines the bank angle achieved.

(e) As the airplane turns toward the selected heading, the difference between magnetic and selected heading is reduced. Both the aileron servo command and the roll F/D command are reduced in magnitude. When the heading is within capture limits, the roll attitude washout logic is generated and the airplane rolls to wings level. The heading hold made is then engaged.

(11) GUI 115;
Attitude Hold Mode (Fig. 13)

(a) The roll mode engaged after CMD ENGAGE depends on airplane attitude, airplane dynamics, and the Lateral Command Engage Option (LCEO). The baseline configuration is LCEO inoperative.





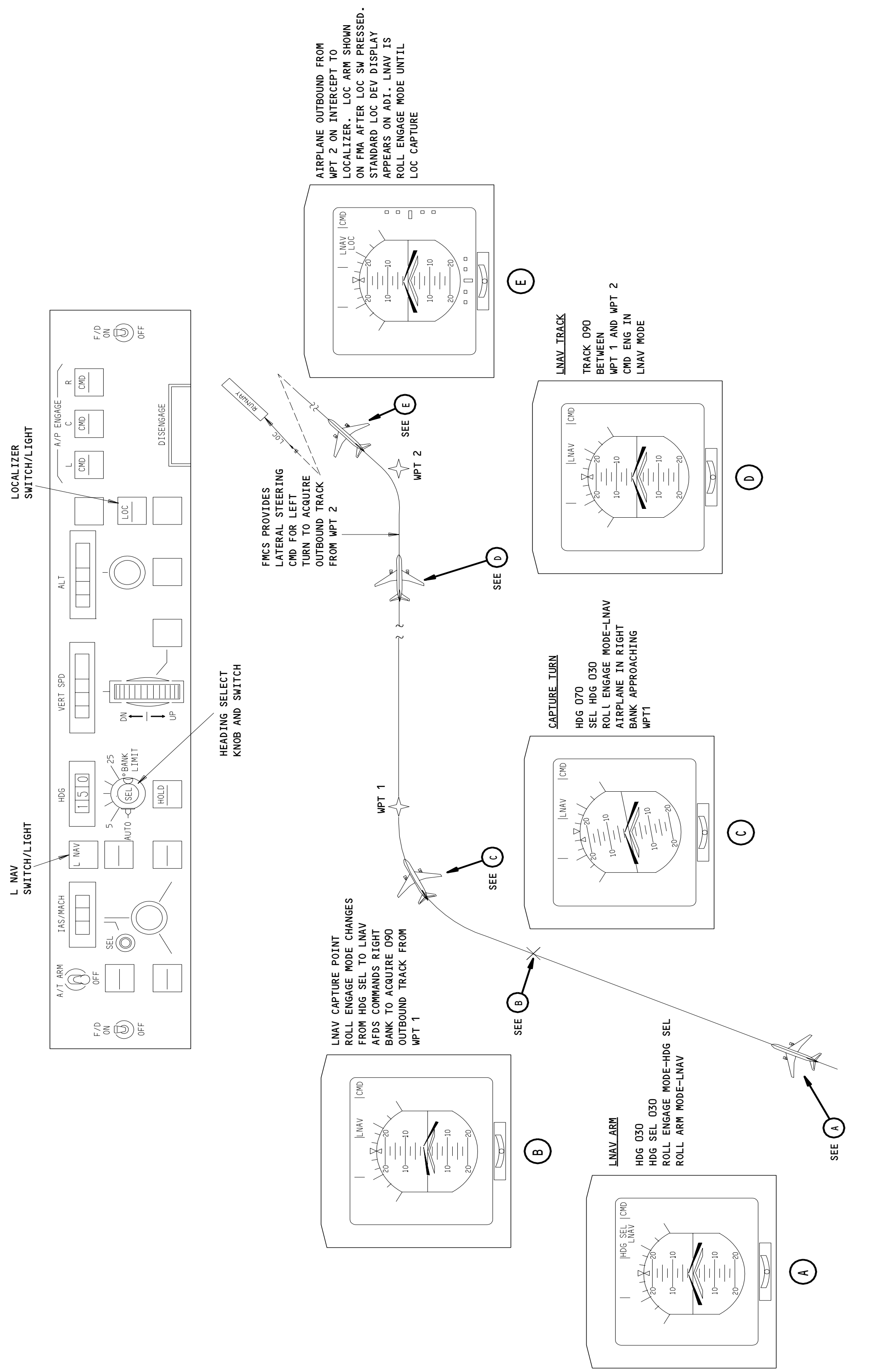
1 RESPONSE FOR BOTH FLT DIRECTORS OFF, OR EITHER FD ON IN T/O OR G/A MODE
IF FD ON AT CMDENG, AP ENGAGES INTO CURRENT FD MODE (SEE AP MODE ENGAGE OPTION)

2 NORMALLY HDGMAG = HDGTRU IF HDGTRU SELECTED

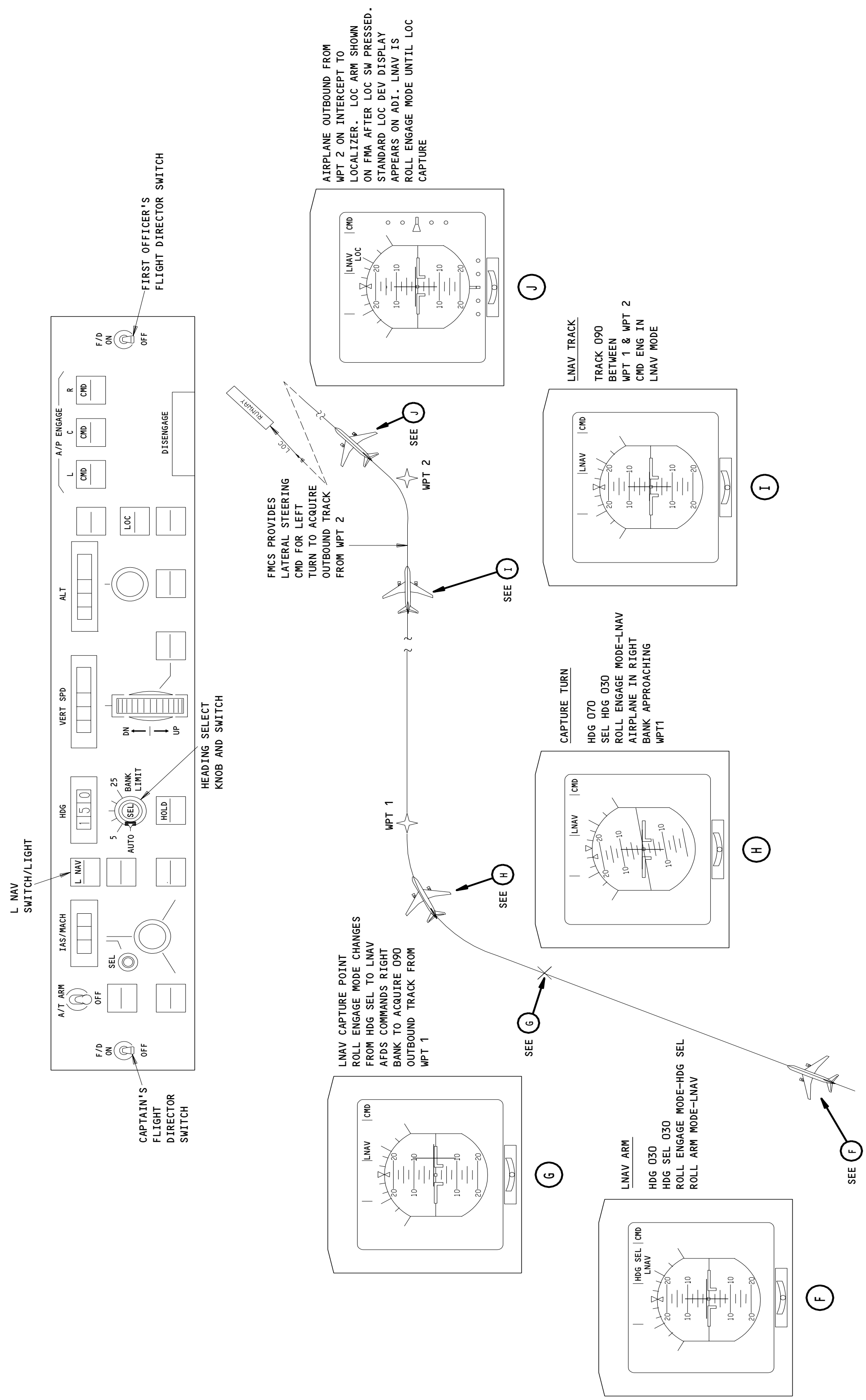
Attitude Hold Mode
Figure 13

(b) GUI 115;
With LCEO operative, three variations exist:
1) If the bank angle is less than 5 degres, the airplane rolls to wings level and engages in heading hold mode.
2) If the bank angle is between 5 and 30 degrees, the airplane engages in roll attitude hold.
3) If the bank angle is greater than 30 degrees, the airplane is rolled to a 30-degree bank angle and engaged in the attitude hold mode.

(12) Lateral Navigation (LNAV) Mode (Fig. 14)
(a) In the LNAV mode, the FMC provides lateral steering commands to the FCC to follow a selected flight plan. Applicable controls on the MCP for this mode are the LNAV switch/light and the HDG display.
(b) LNAV ARM Mode (A)
1) The LNAV ARM mode is initiated during HDG SEL (roll engaged mode) by pressing the LNAV switch/light. The mode is annunciated on the EADI. The AFDS commands to hold selected heading until a valid LNAV steering command is received from the FMC at LNAV capture.
(c) LNAV Capture (B) and Capture Turn (C)
1) At LNAV capture, the FMC supplies valid lateral steering command (LNAVC). Detection of LNAVC changes the roll active mode to LNAV and the EADI roll engaged annunciations change to LNAV and LNAV ARM. The LNAVC commands a right turn to acquire flight plan track at WPT 1.
(d) LNAV Mode (D)
1) The FMC supplies LNAVC to maintain the FMC flight plan track from WPT 1 to WPT 2. Automatic compensation is applied to compensate for wind drift. There is no change in EADI indications. The FMC provides lateral steering commands at WPT 2 for a left turn to acquire the outboard track.
(e) LNAV and LOC ARM (E)
1) The LOC ARM submode is initiated by pressing the LOC switch/light on the MCP. The roll channel is armed for capture of selected ILS frequency. The EADI displays LOC as the roll armed mode but the AFDS responds to LNAVC until LOC capture. The EADI displays LOC and GS. Localizer mode will be discussed later.

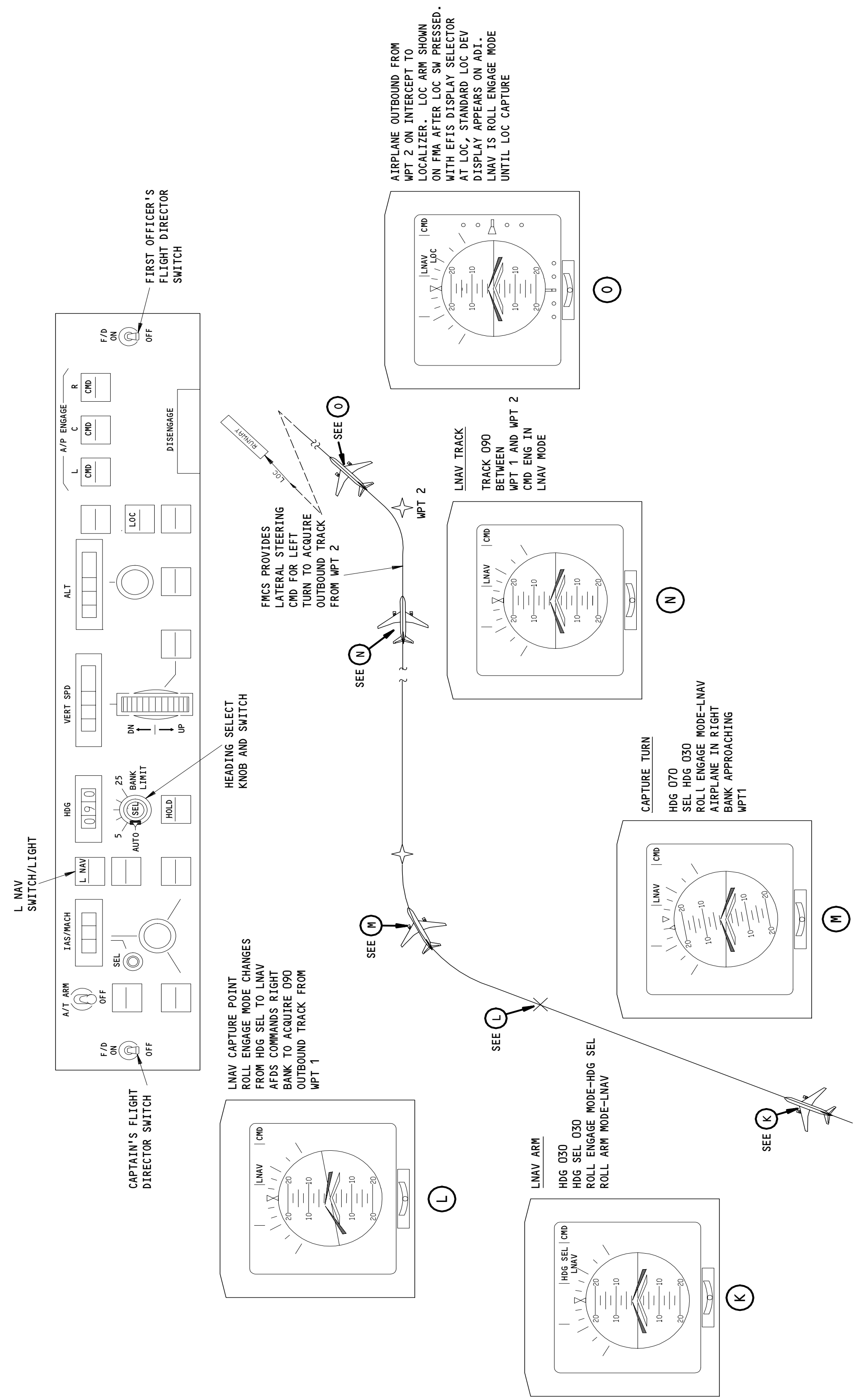


LNAV Mode
Figure 14 (Sheet 1)



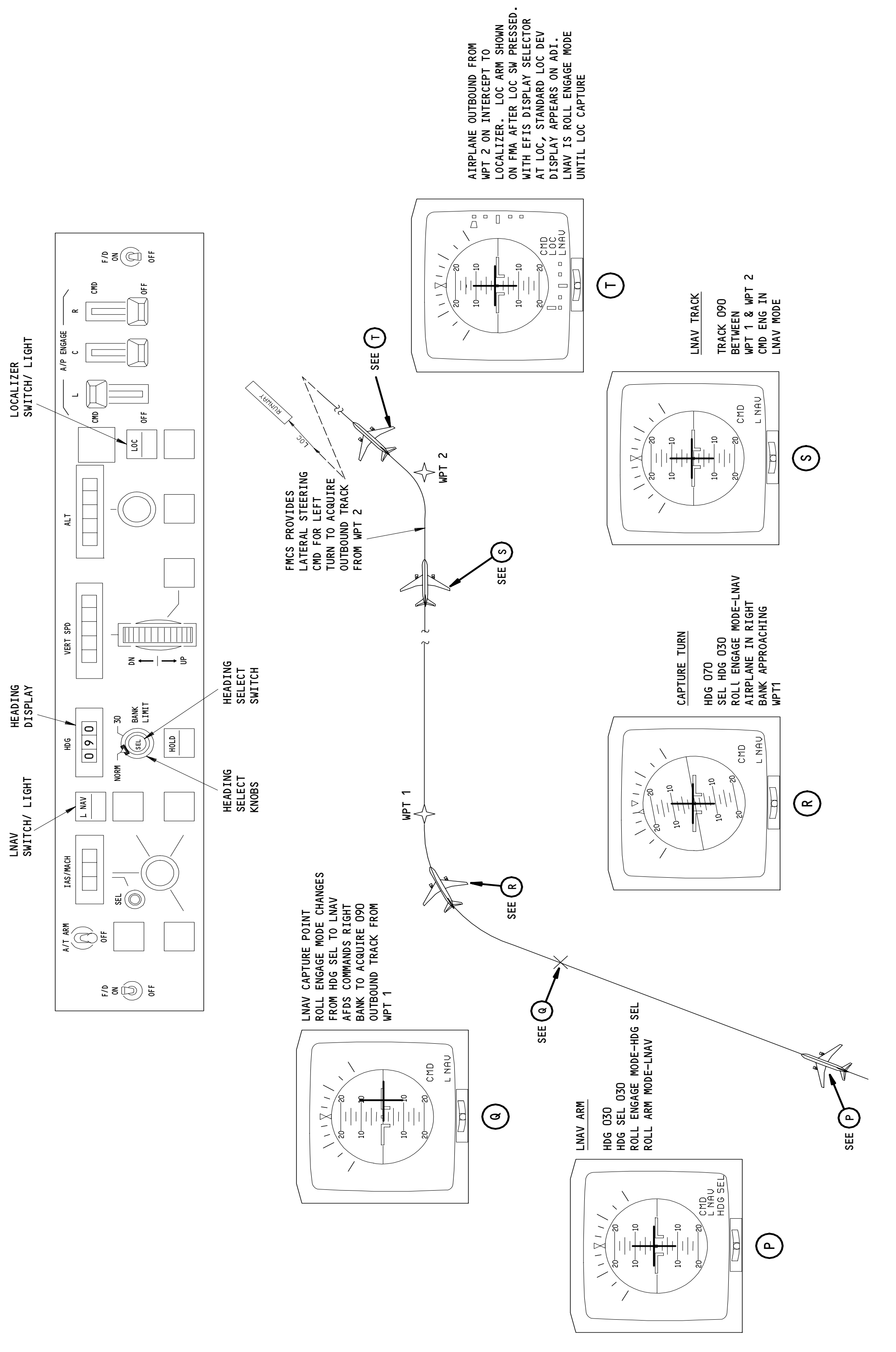
LNAV Mode
Figure 14 (Sheet 2)

A47885



LNAV Mode
Figure 14 (Sheet 3)

C33052



LNAV Mode
Figure 14 (Sheet 4)

A70886

(13) LNAV Mode Logic (Fig. 10)

(a) When the LNAV switch/light is pressed and accepted, this request is processed by mode transition logic to produce the LAT NAV logic. When the FMC indicates a valid horizontal steering command (LNAV capture) the LAT NAV engage valid logic is produced. This develops the 2 NAV engaged logic. If one or more channels are in CMD ENG, the LNAV command input logic (LNAVIN) is produced. The LNAV engage and LNAVIN logic signals are used in signal processing.

(14) LNAV Mode Signal Processing (Fig. 11)

(a) The LNAV engage logic applies the horiziontal steering command from the FMC to a summing point where it is summed with the F/D heading command in the development of the roll F/D commands. The LNAVIN logic applies the horizontal steering command to the roll attitude processing chain which generates the aileron servo command. Aileron servo position and surface position LVDTs provide feedback to the processor for nulling the servo command.

(15) Localizer (LOC) Mode (Fig. 15)

(a) The MCP controls and displays associated with the LOC mode are the HDG display, the LOC switch/light and the APP switch/light.

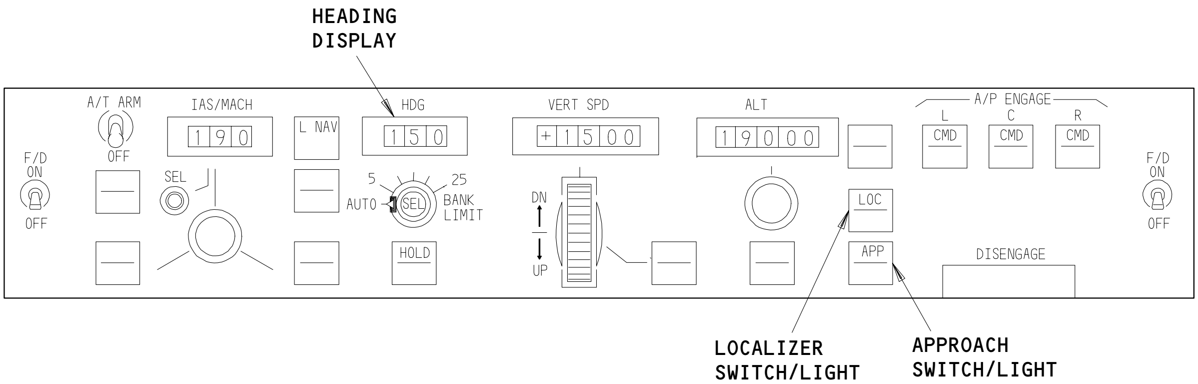
(b) The AFDS acquires and maintains the front course LOC centerline during single or multichannel ILS approach. Localizer control laws generate AFDS commands for control of optimum flight path to the LOC centerline. At LOC capture, continuous tests are started to determine when transition from LOC capture to LOC track should occur. In LOC track, the AFDS commands for optimum tracking of the LOC centerline.

(c) The FMA on the EADI displays LOC in white as the roll armed mode before LOC capture and LOC in green as the roll engaged mode after LOC capture. At LOC capture, the heading display on the MCP changes from the last selected heading to the selected course (runway heading) on the ILS control panel.

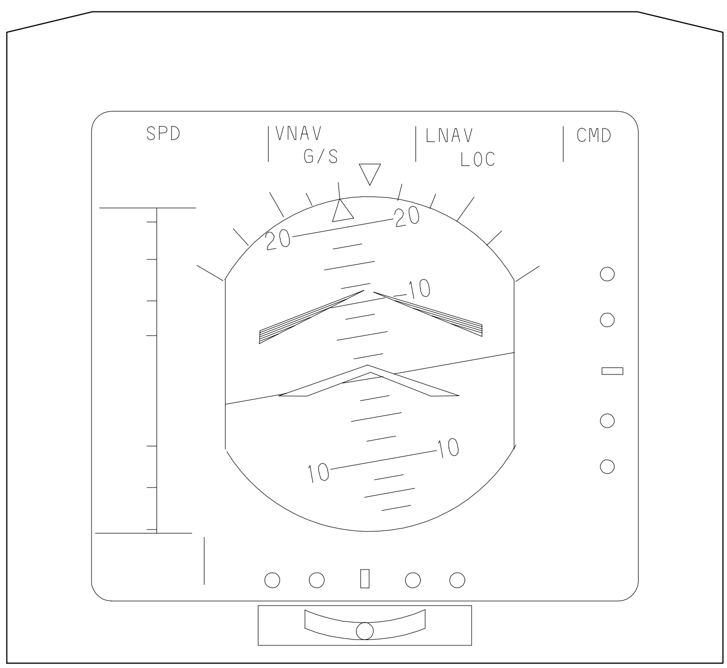
(d) Single Channel LOC Approach

1) A single channel LOC approach is enabled by pressing the LOC switch. This mode is used when the pilot wishes to fly a different vertical path than that dictated by the glideslope beam, or where no G/S is available. The pitch axis may be controlled using any single channel vertical control mode.

2) The single channel LOC approach mode can be disengaged by: selecting heading hold; arming the approach mode and capturing G/S; sensing an on-ground condition; A/P disconnect; or switching the F/D off.

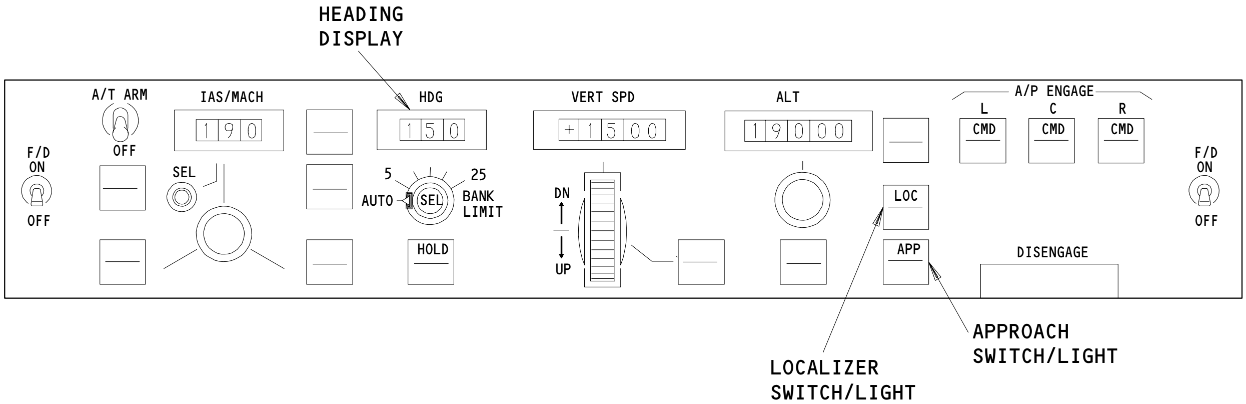


MODE CONTROL PANEL

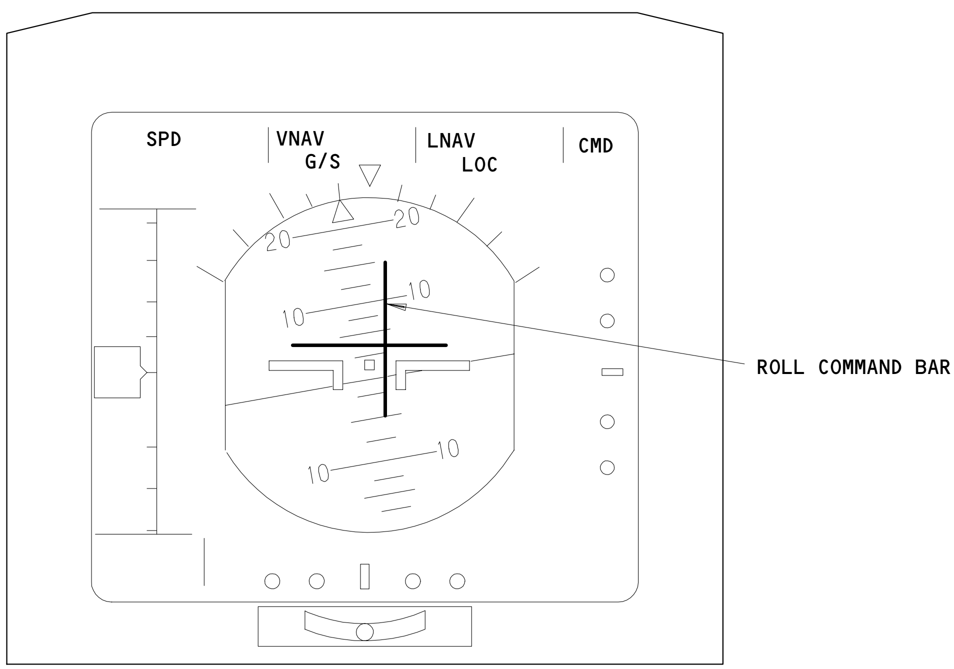


EADI

Localizer Mode
Figure 15 (Sheet 1)

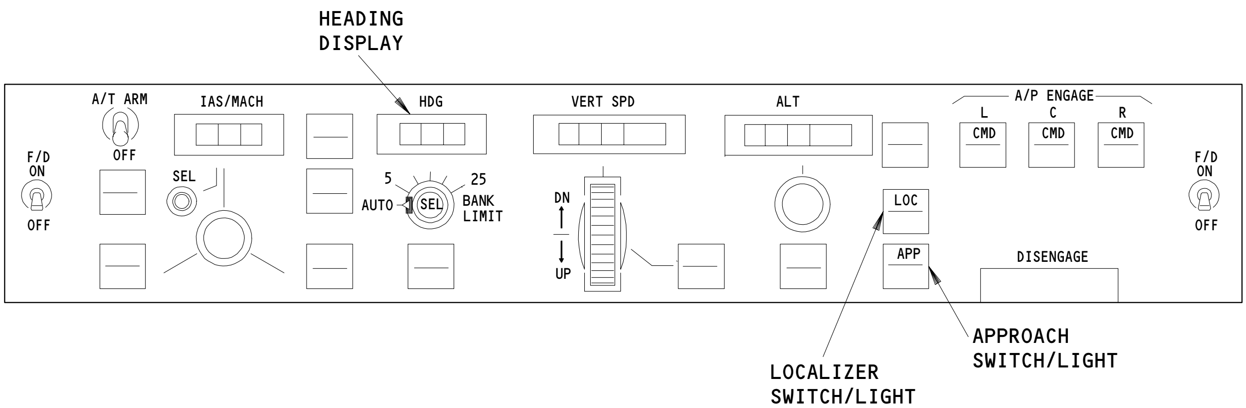


MODE CONTROL PANEL

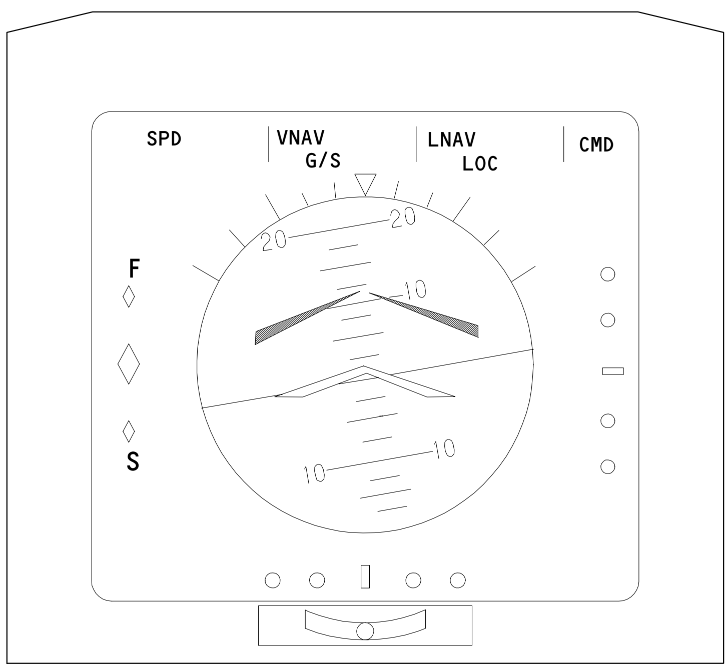


EADI

Localizer Mode
Figure 15 (Sheet 2)

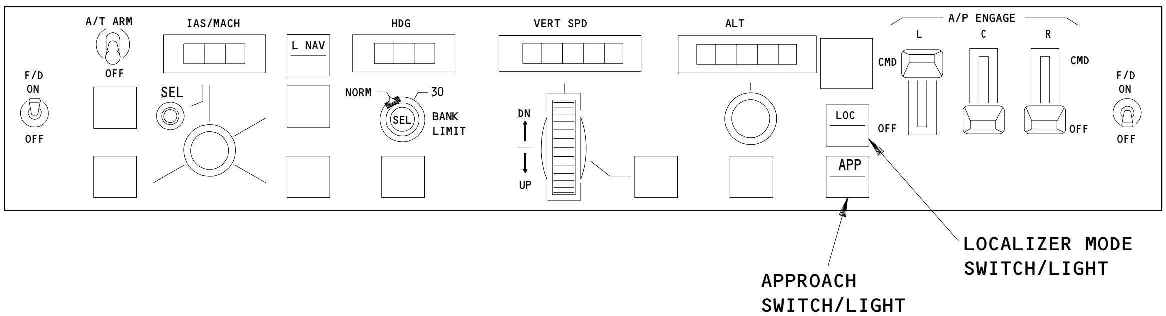


MODE CONTROL PANEL

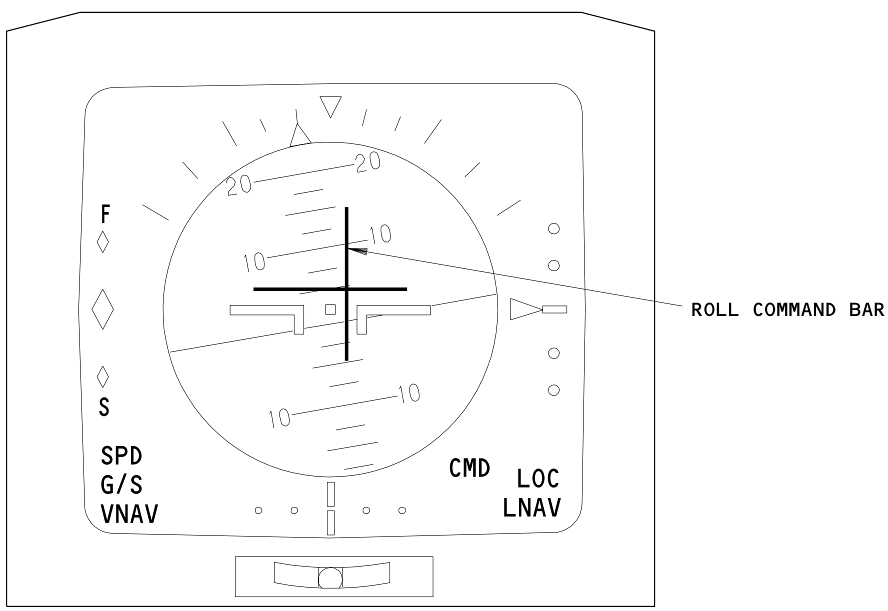


EADI

Localizer Mode
Figure 15 (Sheet 3)



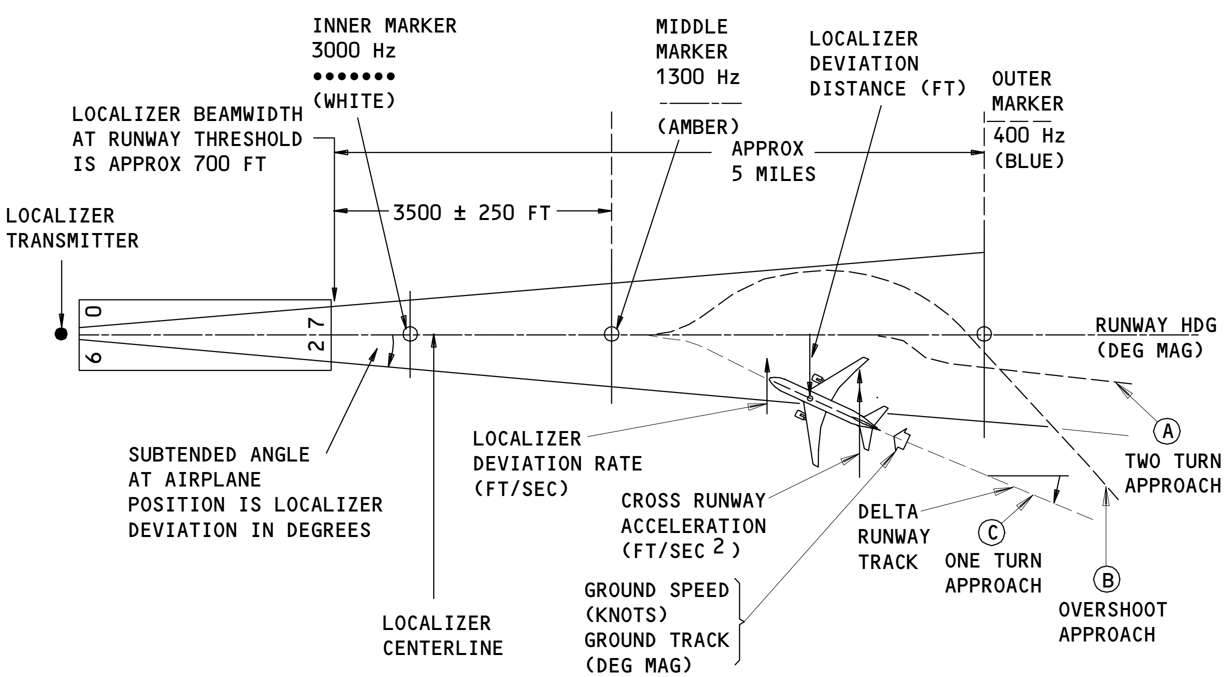
MODE CONTROL PANEL



EADI

Localizer Mode
Figure 15 (Sheet 4)

(e) Single Channel ILS Mode
1) GUI 115;
the single channel ILS mode is enabled by pressing the APP switch when the CMD switches of non-engaged channels are off. This provides LOC and G/S control but no runway alignment, flare or rollout control. The mode is disabled by selecting APP and subsequent multi-channel engage, or by A/P disconnect.
2) GUI 001-114, 116-999;
single channel ILS operation is not possible, since all channels are automatically armed when the APP switch/light is pressed and any one channel is in CMD.
(f) Multi-Channel ILS Approach (Autoland)
1) The multi-channel ILS mode is enabled by pressing the APP switch and then engaging the CMD switch of any non-engaged channel. Multi-channel operation provides full LOC and G/S AUTOLAND control to CAT III B standards. The mode is disabled by A/P disconnect or by default to single channel A/P approach.
(16) Localizer Approach Geometry (Fig. 16)



Localizer Approach Geometry
Figure 16

(a) Controlling Factors
  1) Localizer deviation distance (ft) is the horizontal distance from the localizer centerline to the airplane. Localizer deviation rate (ft/sec) is a function of the cross-localizer centerline velocity. Cross-runway acceleration is in ft/sec^2. The angle between ground track and runway heading is called delta runway track.

(b) Raw Data Inputs
  1) Runway heading information comes from the ILS control panel. The Inertial Reference System (IRS) provides ground speed and ground track data.

(c) LOC Capture
  1) Localizer capture occurs when LOC capture criteria are satisfied. Capture criteria threshold is a variable point depending on values of the controlling factors listed above. It is calculated so that a smooth capture is achieved with minimum control activity and without overshoot beyond beam threshold. Localizer capture is inhibited when delta runway track exceeds 120 degrees.

(d) Capture Flight Paths
  1) If the delta runway track angle is so small that maintaining present track would cause intersecting the beam centerline too late or not at all, or if the angle is divergent up to 120 degrees, a two-turn capture (A) is commanded. A one-turn capture (C) is commanded if a sufficiently early beam center intersection is possible and a bank angle is computed to fly a circular flight path. One turn capture causes a beam center overshoot if the computed bank angle is greater than max bank angle allowed (B). The start of the capture turn on the flight path (C) is delayed until the minimum LOC capture bank angle is computed.
  2) The capture turn is maintained at the commanded bank angle as long as possible, until computed rolldown rate to wings level is at the maximum permissible value. Control laws apply inertial smoothing to reduce effects of beam noise and propagation irregularities. Signal compensation is applied for gust filtering and crosswind shear effects.

(e) Localizer Tracking

1) Control laws are modified when LOCLIN (LOC linear) criteria are satisfied, when lateral position and deviation rate have primary control. The LOCLIN criteria are functions of cross runway acceleration and LOC processing signal levels.

(f) Control Law Objectives for LOC Operation

1) The objectives of localizer control laws are zero steady state error (eliminate LOC standoff error); max B/A commanded, 30 deg.; max roll rate commanded, 7 deg/sec, maximize disturbance rejection; minimize control activity, and avoid unnecessary overshoots.

(17) Localizer Mode Logic (Fig. 10)

(a) LOC Mode Arm

1) The LOC mode arm logic is generated by pressing the LOC switch/light on the MCP, but is inhibited during the F/D disconnect pulse. This permits synchronization of the LOC signal and command processing so that the LOC integral and proportional roll commands are zero.

(b) LOC Mode Enable

1) The LOC enable logic is generated when LOC mode arm or APP arm and LOC deviation filter engage are a logic one. The LOC deviation filter engage is initially dependent on LOC in-range logic. Synchronization continues as in LOC mode arm. When runway correction enable is logic one, computation of filtered deviation distance is modified to be primarily controlled by cross runway acceleration.

(c) Two-Turn Capture (LOC2TC)

1) The following conditions determine the start of a two-turn capture. The threshold value of delta ground track (DGTRK) is the value below which LOC2TC becomes logic one (DPSIWN). It is the lesser of 20 degrees and a computed value which is a function of G/S attitude and filtered LOC deviation distance (DYLOCF). A divergent capture situation exists when the signs of DGTRK and DYLOCF are equal. When LOC2TC is logic one, LOC1TC is inhibited. Localizer capture (LOC CAPT) is generated by LOC 2 (LOC second phase).

(d) One-Turn Capture (LOC1TC)
1) When the computed B/A exceeds the minimum LOC capture turn B/A, LOC1TC logic is generated and LOC CAPT is generated. A latch maintains LOC CAPT and generates LOC CAPP (past value of LOC CAPT). Localizer signal and command synchronization is disabled to permit capture turns to be generated.

(e) LOC Linear
1) After the airplane has become aligned to the LOC centerline, LOC LINEAR is generated and the LOC TRACK submode is enabled. Delta runway track must be less than 90 degrees but final control in developing LOC LINEAR is the output of the localizer deviation rate computation. This computation is sensitive to all critical variables for LOC centerline acquisition. The LOC LINEAR logic modifies the control laws for improved centerline tracking.

(f) LOC Engaged (LOC ENG)
1) The LOC ENG logic is generated by LOC CAPT. At the transition to LOC LINEAR, LOC ENG momentarily goes to zero. This supresses transients to the LOC command processor during entry into LOC LINEAR and permits synchronizing for final capture.

(18) Localizer Mode Control Signal Flow (Fig. 11)
(a) The localizer control laws produce commands to fly the necessary path to capture the beam center. The system initially controls ground track angle until an initial capture turn is commanded at the computed bank angle. Final capture of the beam centerline then starts using lateral position control. Overshoot and control activity is minimized by flying as much of the capture turn as possible at the computed bank angle. Compensation is provided for effects of turbulence, beam noise, and beam propagation irregularities.

(b) LOC Mode Arm
1) The LOC command processor synchronizes to LOC displacement rate error (DYDOTE) and filtered LOC deviation distance (DYLOCF). The cross-runway integrator synchronizes to roll attitude (PHI). The DGTRK is multiplied by the sign of DYLOCF (±1) to generate the two turn bank angle command curve. The minimum bank angle is calculated as a function of complimentary filtered gear altitude. This sets the minimum value of the P2TURN limiter. The P2TURN signal is not used, however, if LOC2TC logic is zero. The P1TURN signal can only be computed when LOC ENG is logic one, but until LOC CAPT occurs, P1TURN is not used in processing.

(c) LOC One-Turn Capture

1) The Sync-loops are opened by LOC ENG and LOC CAPT. At LOC CAP, the stored value of P1TURN (LOC Proportional roll command) is held through the signal latch. Subsequent changes of P1TURN have no effect. The P1TURN signal is applied through a ± 5.0 deg bank cmd/sec limiter and then converted into cross-runway acceleration. This develops cross runway velocity. This velocity is compared with LOC displacement rate (DYDOT) and LOC displacement rate error (DYDOTE) is produced. This error is converted to a proportional roll command and applied to a summing point. The other input to the summing point is zero, so the LOC proportional command is the only command generated at the initial capture turn.

(d) LOC Two-Turn Capture

1) When conditions cause LOC2TC to be logic 1, P2TURN is selected for processing. Since P2TURN has been continuously computed during LOC mode arm, LOC prop roll assumes the value of P2TURN at LOC CAPT and is maintained at that value during the turn towards the beam. System response downstream from this summing point is the same as during P1TURN operation. When LOC2TC conditions no longer exist due to increased DGTRK, P1TURN is selected for the second capture turn.

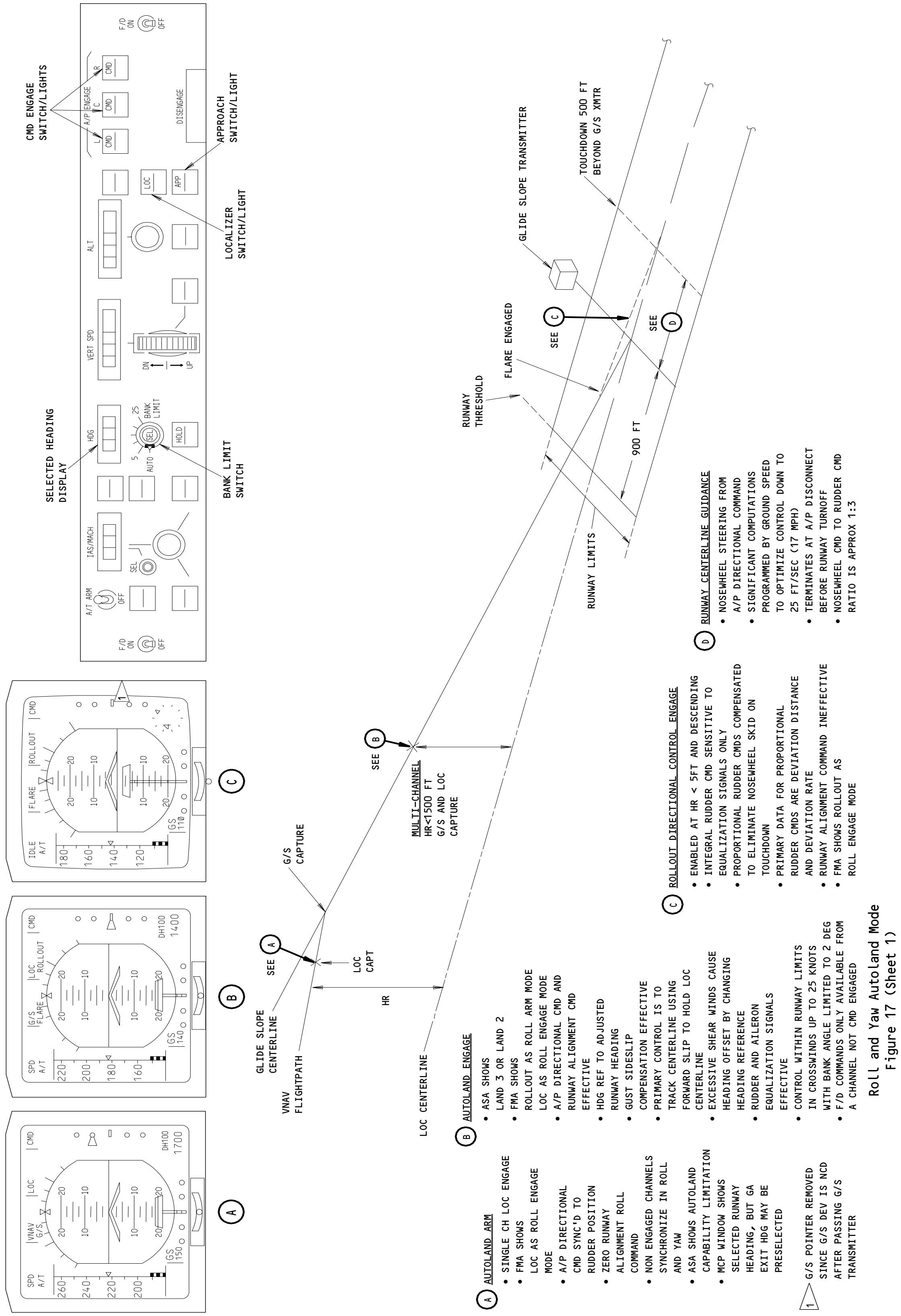
(e) LOC Linear Conditions

1) Final capture criteria is determined by the need to minimize overshoot and to avoid excessive rolldown rates. Continuous computation of lateral deviation and deviation rate occurs during the initial capture turn established by LOC proportional roll command. The LOCLIN logic is generated after LOCCAPT when LOC deviation rate is within the capture band in the LOC CMD processor filter frequency computation. The maximum rolldown rate command is P2MAX. When computations detect that rolldown rate equals P2MAX, the LOCLIN transition occurs and establishes filter values in the LOC CMD processor.

2) Localizer command integrator already has an output derived from cross runway acceleration command during the initial capture turn. This is now modified at the summing point and fed back to the integrator. This output is of opposite sign to the original input, so it causes a rundown of the integrator output. This establishes a maximum DYC at LOCLIN initiation.
3) The LOC displacement error (DYE) which reduces as DYLOCF feedback reduces, is converted into a proportional roll command and summed with LOC displacement rate error to modify PCLOC as a function of DYLOCF. The gain schedule generates a dependent LOC integral command which eliminates any standoff error.

(f) Final Capture of LOC Centerline
1) The final capture flight path follows an exponented law. When the computed rolldown command is less than P2MAX, a new value is generated to control the final rolldown command.

(19) Autoland Mode - Roll and Yaw (Fig. 17)
(a) The MCP controls and displays related to the Autoland Mode are identified on figure 17.
(b) The selected heading display shows the selected runway heading from the ILS control panel after LOC capture and maintains this display until the A/P is disconnected after landing or after the roll go-around mode is terminated. The B/A limit selector is usually left in the NORM position during autoland. When pressed, the approach mode switch/light (APP) initiates the transition into the autoland configuration. The other two engage switches are armed after the APP mode request has been accepted by the FCC. The other two engage switches can then be manually positioned to CMD and the FCC provides a logic out to hold the switches in CMD. This arms the multi-channel engage logic in the corresponding FCC pitch and roll channels.
(c) Autoland Roll/Yaw Flight Path
1) Four conditions of rollout control are shown: rollout arm, rollout engage, rollout directional control engage, and runway centerline guidance.



A AUTOLAND ARM
- SINGLE CH LOC ENGAGE
- FMA SHOWS LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD SYNC'D TO RUDDER POSITION
- ZERO RUNWAY ALIGNMENT ROLL COMMAND
- NON ENGAGED CHANNELS SYNCHRONIZE IN ROLL AND YAW
- ASA SHOWS AUTOLAND CAPABILITY LIMITATION
- MCP WINDOW SHOWS SELECTED RUNWAY HEADING, BUT GA EXIT HDG MAY BE PRESELECTED

1 G/S POINTER REMOVED SINCE G/S DEV IS NCD AFTER PASSING G/S TRANSMITTER

B AUTOLAND ENGAGE
- ASA SHOWS LAND 3 OR LAND 2
- FMA SHOWS ROLLOUT AS ROLL ARM MODE LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD AND RUNWAY ALIGNMENT CMD EFFECTIVE
- HDG REF TO ADJUSTED RUNWAY HEADING
- GUST SIDESLIP COMPENSATION EFFECTIVE
- PRIMARY CONTROL IS TO TRACK CENTERLINE USING FORWARD SLIP TO HOLD LOC CENTERLINE
- EXCESSIVE SHEAR WINDS CAUSE HEADING OFFSET BY CHANGING HEADING REFERENCE
- RUDDER AND AILERON EQUALIZATION SIGNALS EFFECTIVE
- CONTROL WITHIN RUNWAY LIMITS IN CROSSWINDS UP TO 25 KNOTS WITH BANK ANGLE LIMITED TO 2 DEG
- F/D COMMANDS ONLY AVAILABLE FROM A CHANNEL NOT CMD ENGAGED

C ROLLOUT DIRECTIONAL CONTROL ENGAGE
- ENABLED AT HR < 5FT AND DESCENDING
- INTEGRAL RUDDER CMD SENSITIVE TO EQUALIZATION SIGNALS ONLY
- PROPORTIONAL RUDDER CMDS COMPENSATED TO ELIMINATE NOSEWHEEL SKID ON TOUCHDOWN
- PRIMARY DATA FOR PROPORTIONAL RUDDER CMDS ARE DEVIATION DISTANCE AND DEVIATION RATE
- RUNWAY ALIGNMENT COMMAND INEFFECTIVE
- FMA SHOWS ROLLOUT AS ROLL ENGAGE MODE

D RUNWAY CENTERLINE GUIDANCE
- NOSEWHEEL STEERING FROM A/P DIRECTIONAL COMMAND
- SIGNIFICANT COMPUTATIONS PROGRAMMED BY GROUND SPEED TO OPTIMIZE CONTROL DOWN TO 25 FT/SEC (17 MPH)
- TERMINATES AT A/P DISCONNECT BEFORE RUNWAY TURNOFF
- NOSEWHEEL CMD TO RUDDER CMD RATIO IS APPROX 1:3

Roll and Yaw Autoland Mode
Figure 17 (Sheet 1)

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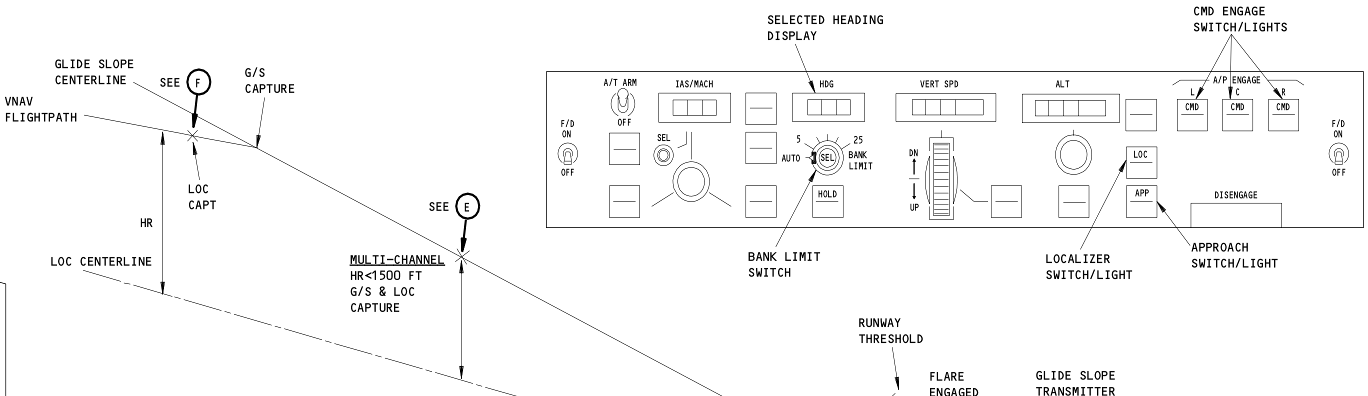
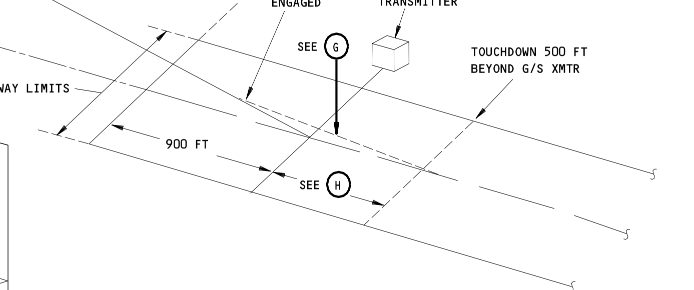
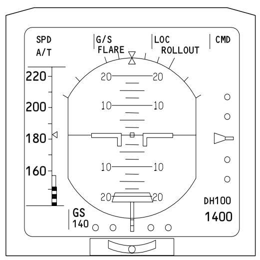


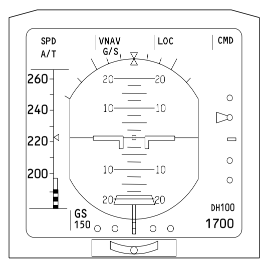




AUTOLAND ENGAGE

- ASA SHOWS
  LAND 3 OR LAND 2
- FMA SHOWS
  ROLLOUT AS ROLL ARM MODE
  LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD &
  RUNWAY ALIGNMENT CMD
  EFFECTIVE
- HDG REF TO ADJUSTED
  RUNWAY HEADING
- GUST SIDESLIP
  COMPENSATION EFFECTIVE
- PRIMARY CONTROL IS TO
  TRACK CENTERLINE USING
  FORWARD SLIP TO HOLD LOC
  CENTERLINE
- EXCESSIVE SHEAR WINDS CAUSE
  HEADING OFFSET BY CHANGING
  HEADING REFERENCE
- RUDDER & AILERON
  EQUALIZATION SIGNALS
  EFFECTIVE
- CONTROL WITHIN RUNWAY LIMITS
  IN CROSSWINDS UP TO 25 KNOTS
  WITH BANK ANGLE LIMITED TO 2 DEG
- GUI 002-006;
  F/D CMDS ONLY AVAILABLE FROM
  A CHANNEL NOT CMD ENGAGED

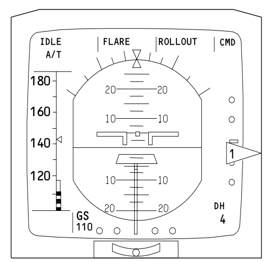
E



AUTOLAND ARM

- SINGLE CH LOC ENGAGE
- FMA SHOWS
  LOC AS ROLL ENGAGE
  MODE
- A/P DIRECTIONAL
  CMD SYNC'D TO
  RUDDER POSITION
- ZERO RUNWAY
  ALIGNMENT ROLL
  COMMAND
- NON ENGAGED CHANNELS
  SYNCHRONIZE IN ROLL &
  YAW
- ASA SHOWS AUTOLAND
  CAPABILITY LIMITATION
- MCP WINDOW SHOWS
  SELECTED RUNWAY
  HEADING, BUT GA
  EXIT HDG MAY BE
  PRESELECTED

F



ROLLOUT DIRECTIONAL CONTROL ENGAGE

- ENABLED AT HR < 5FT & DESCENDING
- INTEGRAL RUDDER CMD SENSITIVE TO
  EQUALIZATION SIGNALS ONLY
- PROPORTIONAL RUDDER CMDS COMPENSATED
  TO ELIMINATE NOSEWHEEL SKID ON
  TOUCHDOWN
- PRIMARY DATA FOR PROPORTIONAL
  RUDDER CMDS ARE DEVIATION DISTANCE &
  DEVIATION RATE
- RUNWAY ALIGNMENT COMMAND INEFFECTIVE
- FMA SHOWS ROLLOUT AS
  ROLL ENGAGE MODE

G

RUNWAY CENTERLINE GUIDANCE

- NOSEWHEEL STEERING FROM
  A/P DIRECTIONAL COMMAND
- SIGNIFICANT COMPUTATIONS
  PROGRAMMED BY GROUND SPEED
  TO OPTIMIZE CONTROL DOWN TO
  25 FT/SEC (17 MPH)
- TERMINATES AT A/P DISCONNECT
  BEFORE RUNWAY TURNOFF
- NOSEWHEEL CMD TO RUDDER CMD
  RATIO IS APPROX 1:3

H

1 G/S POINTER REMOVED
SINCE G/S DEV IS NCD
AFTER PASSING G/S
TRANSMITTER

Roll and Yaw Autoland Mode
Figure 17 (Sheet 2)

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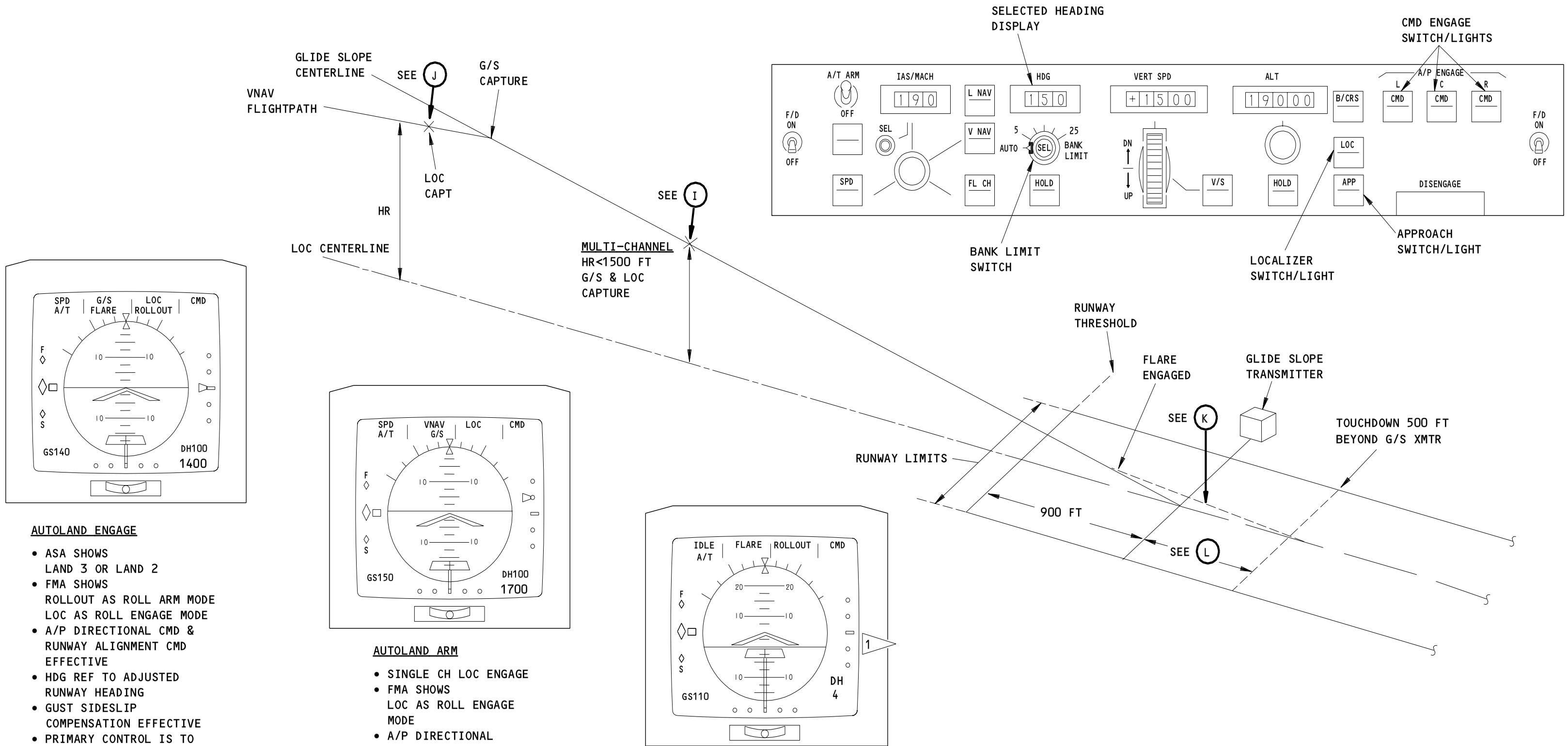

AUTOLAND ENGAGE

- ASA SHOWS LAND 3 OR LAND 2
- FMA SHOWS ROLLOUT AS ROLL ARM MODE LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD & RUNWAY ALIGNMENT CMD EFFECTIVE
- HDG REF TO ADJUSTED RUNWAY HEADING
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- F/D CMDS ONLY AVAILABLE FROM ANY CHANNEL NOT CMD ENGAGE

I

AUTOLAND ARM

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- A/P DIRECTIONAL CMD SYNC'D TO RUDDER POSITION
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J

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- PROPORTIONAL RUDDER CMDS COMPENSATED TO ELIMINATE NOSEWHEEL SKID ON TOUCHDOWN
- PRIMARY DATA FOR PROPORTIONAL RUDDER CMDS ARE DEVIATION DISTANCE & DEVIATION RATE
- RUNWAY ALIGNMENT COMMAND INEFFECTIVE
- FMA SHOWS ROLLOUT AS ROLL ENGAGE MODE

K

RUNWAY CENTERLINE GUIDANCE

- NOSEWHEEL STEERING FROM A/P DIRECTIONAL COMMAND
- SIGNIFICANT COMPUTATIONS PROGRAMMED BY GROUND SPEED TO OPTIMIZE CONTROL DOWN TO 25 FT/SEC (17 MPH)
- TERMINATES AT A/P DISCONNECT BEFORE RUNWAY TURNOFF
- NOSEWHEEL CMD TO RUDDER CMD RATIO IS APPROX 1:3

L

1 G/S POINTER REMOVED SINCE G/S DEV IS NCD AFTER PASSING G/S TRANSMITTER

Roll and Yaw Autoland Mode
Figure 17 (Sheet 3)

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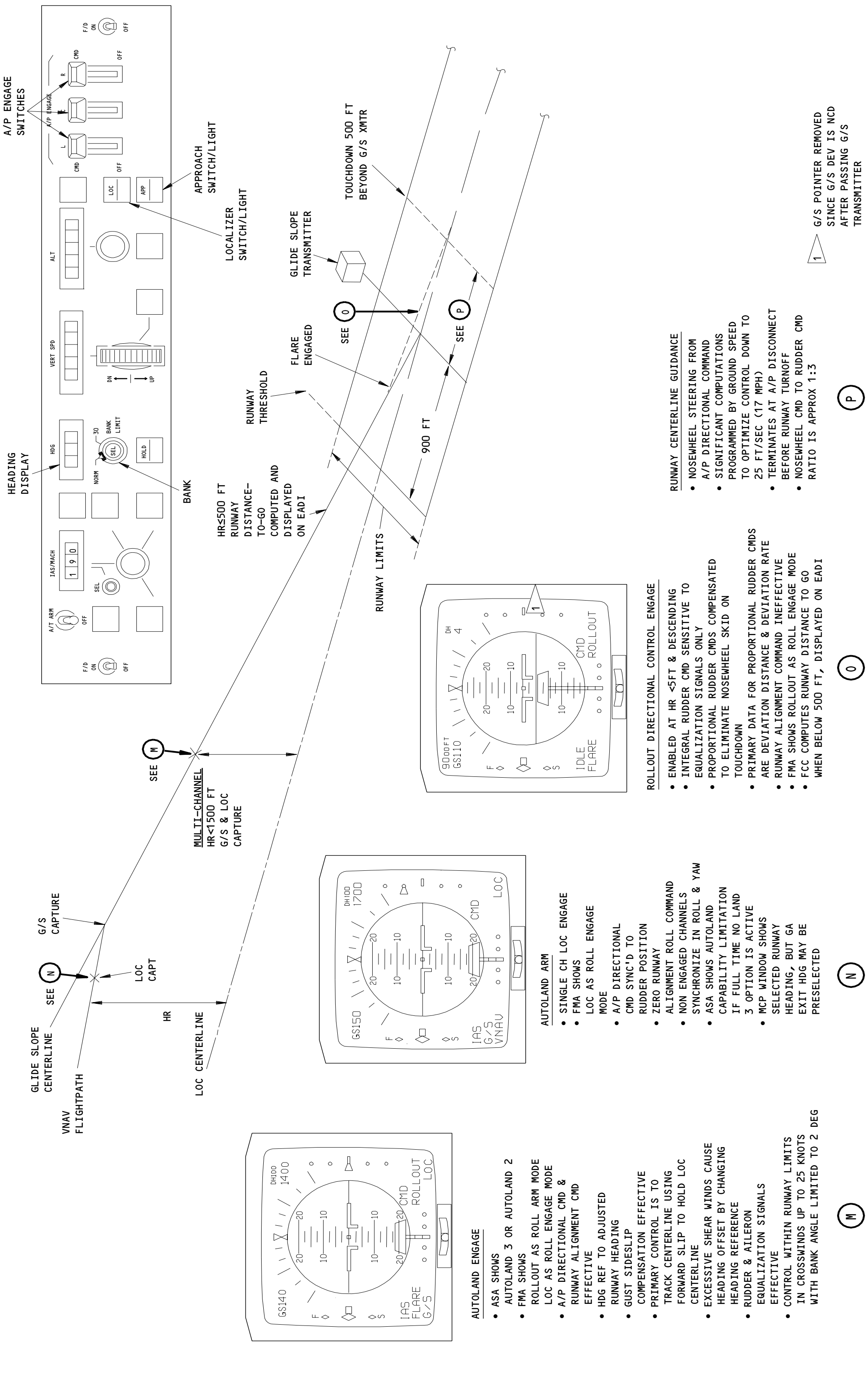

Roll and Yaw Autoland Mode
Figure 17 (Sheet 4)

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(d) Rollout Arm (A)
1) Rollout arm is single channel in CMD with APP selected. The EADI displays LOC as the roll engaged mode. The A/P directional CMD is synchronized to rudder position. All non-engaged channels synchronize in roll and yaw.

(e) Rollout Engage (B)
1) All channels are CMDENG in roll, pitch and yaw. The EADI displays rollout white as the roll armed mode. The A/P directional CMD and runway alignment command are active. Heading reference is to adjusted runway heading. Primary control is to track centerline using forward slip to hold the LOC centerline.

(f) Rollout Directional Control Engage (C)
1) Rollout directional control engage is enabled when radio altitude is less than 5 feet and descending. The EADI displays rollout green as the roll engaged mode. The integral rudder command is sensitive to equalization signals only. Rudder proportional commands are compensated to eliminate nose wheel skid on touchdown and body yaw rate. Deviation distance and deviation rate serve as primary data for the rudder proportional commands.

(g) Runway Centerline Guidance (D)
1) Nose wheel steering is enabled after the nose wheel touches the runway. It maintains the airplane on the centerline of the runway. Significant computations are programmed by ground speed to optimize control down to 25 ft/sec (17 mph). Control terminates at A/P disconnect before runway turnoff.

(20) Autoland Mode Logic (Fig. 10)

(a) Upon receipt of the APP mode ARMED or OPERATIVE discrete, the APP switch dot-bar matrix is illuminated. Each CMD ENGAGE switch initiates a CMDENG request discrete which goes to the corresponding FCC. A valid request generates a REQUEST ACCEPTED discrete which locks the CMD switch in the up position. The APP switch generates a similar request when pressed. Each F/D switch generates an ON/OFF discrete but there is no visual feedback to the switches.

(b) Command Engage Sequence Detection Logic
1) Each FCC receives CMDENG REQUEST ACCEPTANCE logic from the other two FCCs via the cross-channel bus and records the order in which each become CMDENG. One of the discretes (L FIRST, C FIRST, R FIRST) is supplied to the first-in-command detection logic. This logic determines which FCC will supply FMA signals to the EFIS. It identifies LOCAL FIRST IN COMMAND and FOREIGN channels during multi-channel engage (MCHENG). On detection of a failure in the first-in-command channel, the fail sequence logic selects a different channel to be first-in-command.

(c) Autoland Processing Logic
1) The APP mode request generates the APPROACH ARM logic when the F/D disengage logic is zero. This is applied to the localizer logic as an alternate to the LOC mode arm logic (bubble B on the illustration), and is also used in autoland logic.
2) When APP ARM, MCHENG, Runway alignment enable, and IN-AIR are all logic one, the generation of runway align engage logic is enabled. This logic goes high when roll attitude is less than 10 degrees, delta runway track is less than 2 degrees and the LOC LINEAR logic has been generated. Runway align enable logic enables the runway alignment command processor. This logic one is latched after 2 seconds to prevent loss of the signal due to out-of-tolerance inputs that would cause a zero logic from the preceeding AND gate. The runway align engage logic also produces the delta runway engage and delta runway path logic.
3) When radio height is less than five feet, Rollout logic is produced and runway alignment logic goes to zero. The delta runway engage and delta runway path logic remain logic ones.

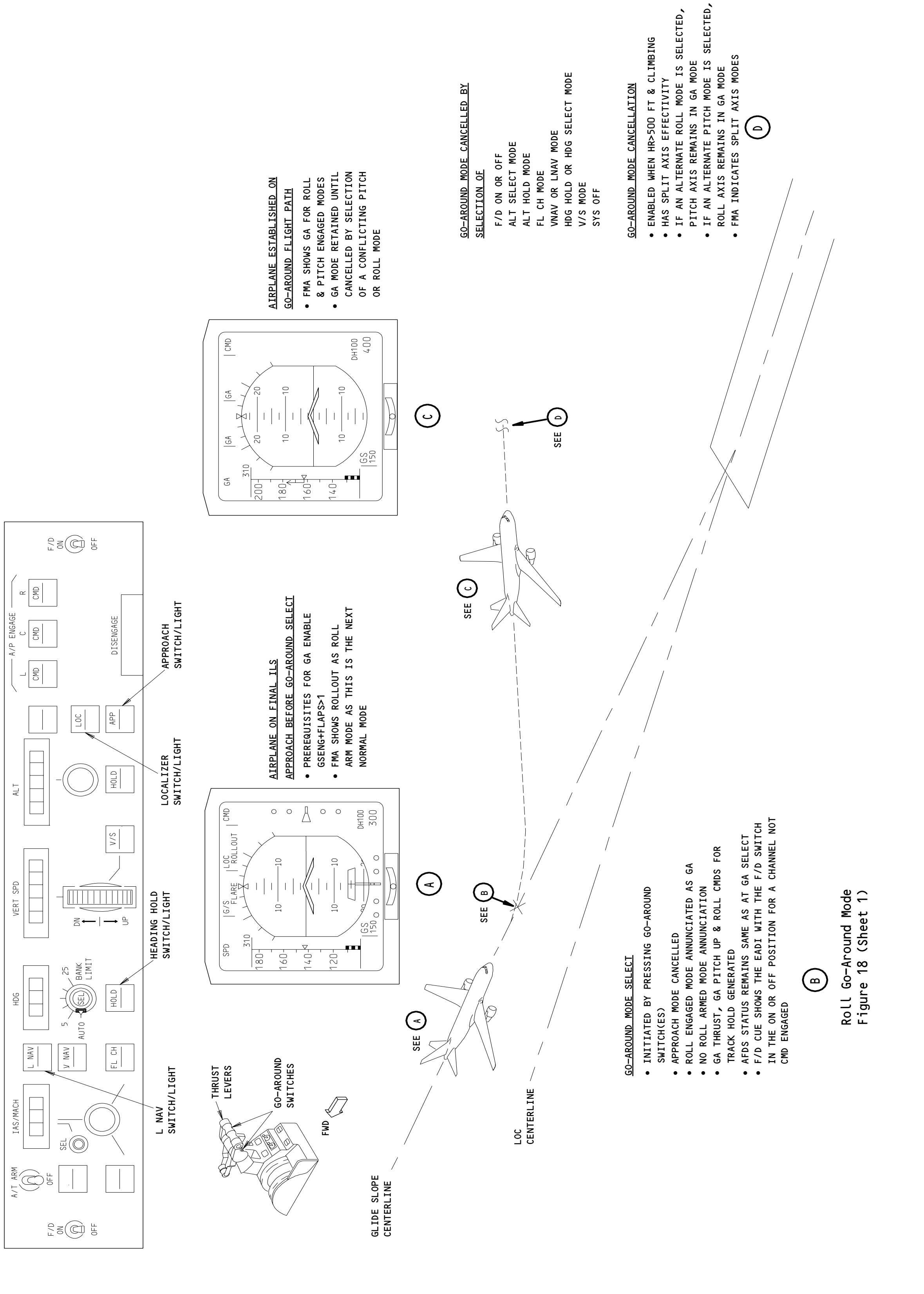
(21) Autoland Mode Control Signal Flow (Fig. 11)
(a) Rollout Arm Conditions
1) Delta heading runway is synchronized to adjusted heading and is zero. Delta LOC deviation distance is synchronized to filtered LOC deviation distance and is also zero. The rudder servo command is synchronized to zero to maintain the rudder crossfeed command at zero so that the runway alignment roll command will be held at zero.

(b) Rollout Engage
1) At multi-channel engage, command equilization must occur to ensure that all FCCs deliver identical value commands to the flight control systems. All FCCs are synchronizing to rudder position.
2) The first channel in command engages into AUTOLAND CMD as soon as any other channel has a command request accepted. Any difference between servo actuator position and rudder position is the rudder equilization command (normally very small) and is routed to the max/min selectors. The max/min feature prevents excessive deflection from the rudder's present position.
3) The multi-channel engage logic one enables the generation of runway align engage when roll angle and delta ground track are less than 10 degrees and 2 degrees, respectively, and the LOC LINEAR logic is one.

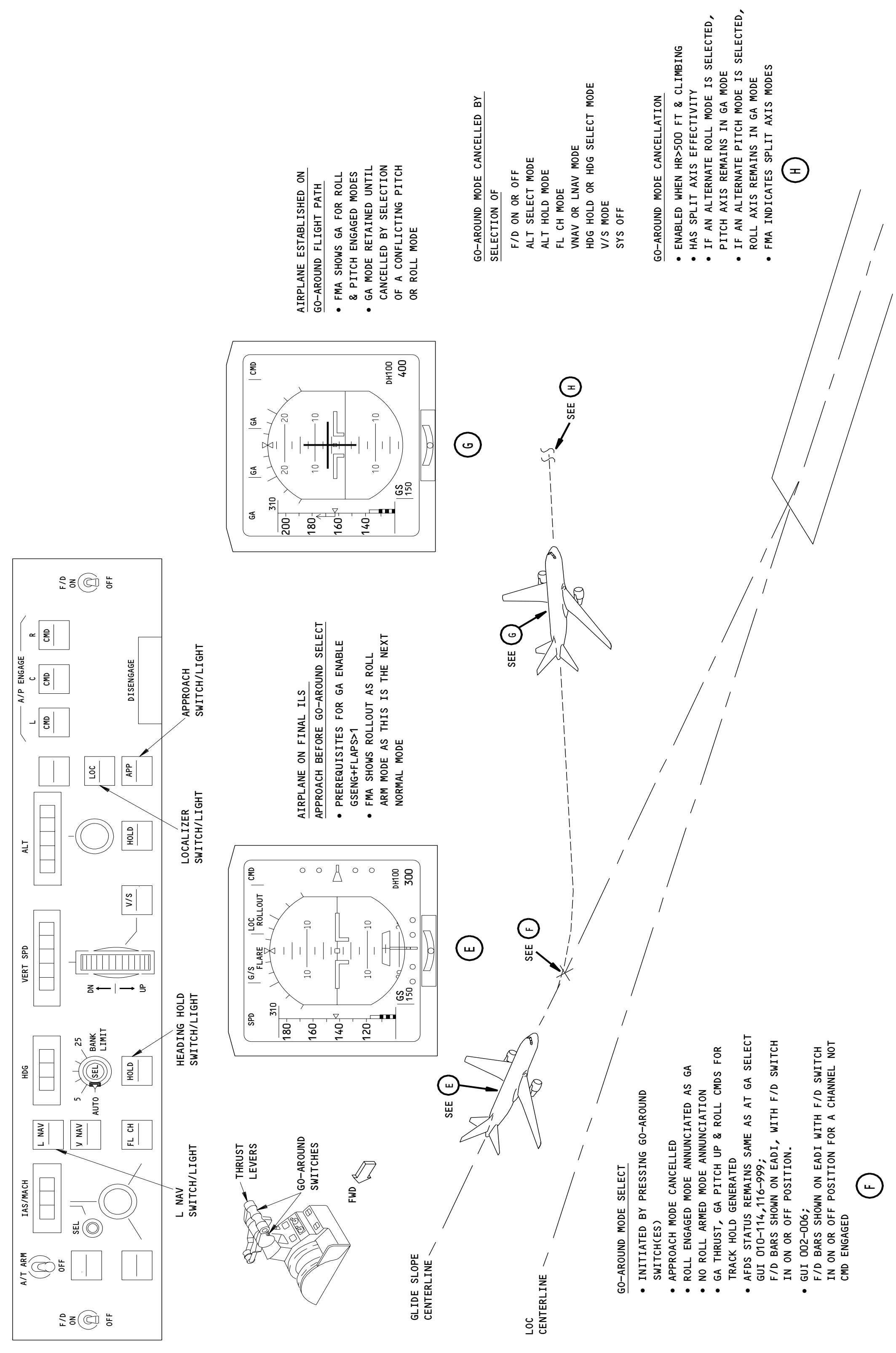
(c) Rollout Engage
1) Rollout directional control engage logic is generated when radio height is five feet. This maintains delta runway engage and delta runway path at logic one while runway align engage goes to zero. Delta LOC deviation distance, which was synchronized to zero, is now increased, at a rate limited to 2 ft/sec, to the current value of filtered LOC deviation distance. This is converted into deg/sec and applied to the rudder integral command signal chain.
2) When rollout directional control engage becomes logic one, additional feedback is added. Body yaw rate is summed with pseudo body rate computed from ground speed and rudder position, and multiplied by a linear constant. This eleminates nose wheel skid on touchdown and thus ensures effective nose wheel steering as soon as the nose wheel touches the ground. As ground speed reduces, rudder control becomes less effective and nose wheel steering provides primary rollout direction control.
3) When ground speed reduces to below 25 ft/sec, inverse gain schedules cause increased command response as rollout speed reduces thus improving LOC centerline tracking.

(d) Rollout Command Processing
1) Runway alignment command in degrees of roll is summed with LOC proportional command. The result is applied to the roll flight director. This signal is also differentiated and added to the LOC integral command. This combined signal is the roll attitude command integrator input in the ROLLOUT engage mode.
2) Aileron or wheel position and ALCS servo position are summed to generate the roll equalization signal. This signal is limited to ±2 degrees and converted into a deg/sec roll command. At MCHENG the equalization roll rate signal is added to the roll command signal. This signal modifies the response of an FCC coming online at MCHENG to compensate for electrical and mechanical hysteresis which results in small variations of LVDT outputs for the same surface position.
3) At MCHENG ±40 degrees becomes the reference for max/min select limiters. The aileron is also limited to servo command ±40 degrees. These larger control limits are required at lower speeds during approach when rapid airplane response is necessary.

(22) Roll Go-Around Mode (Fig. 18)
(a) Significant MCP Features
1) GUI 001-114, 116-999;
the MCP may show any of the following switch dot-bar matrices illuminated before entry into the go-around mode: HDG HOLD, LNAV, LOC APP, CMD. The F/D switches may be ON or OFF. The HDG display shows selected runway heading before G/A and MCP selected heading during G/A.



Roll Go-Around Mode
Figure 18 (Sheet 1)



Roll Go-Around Mode
Figure 18 (Sheet 2)

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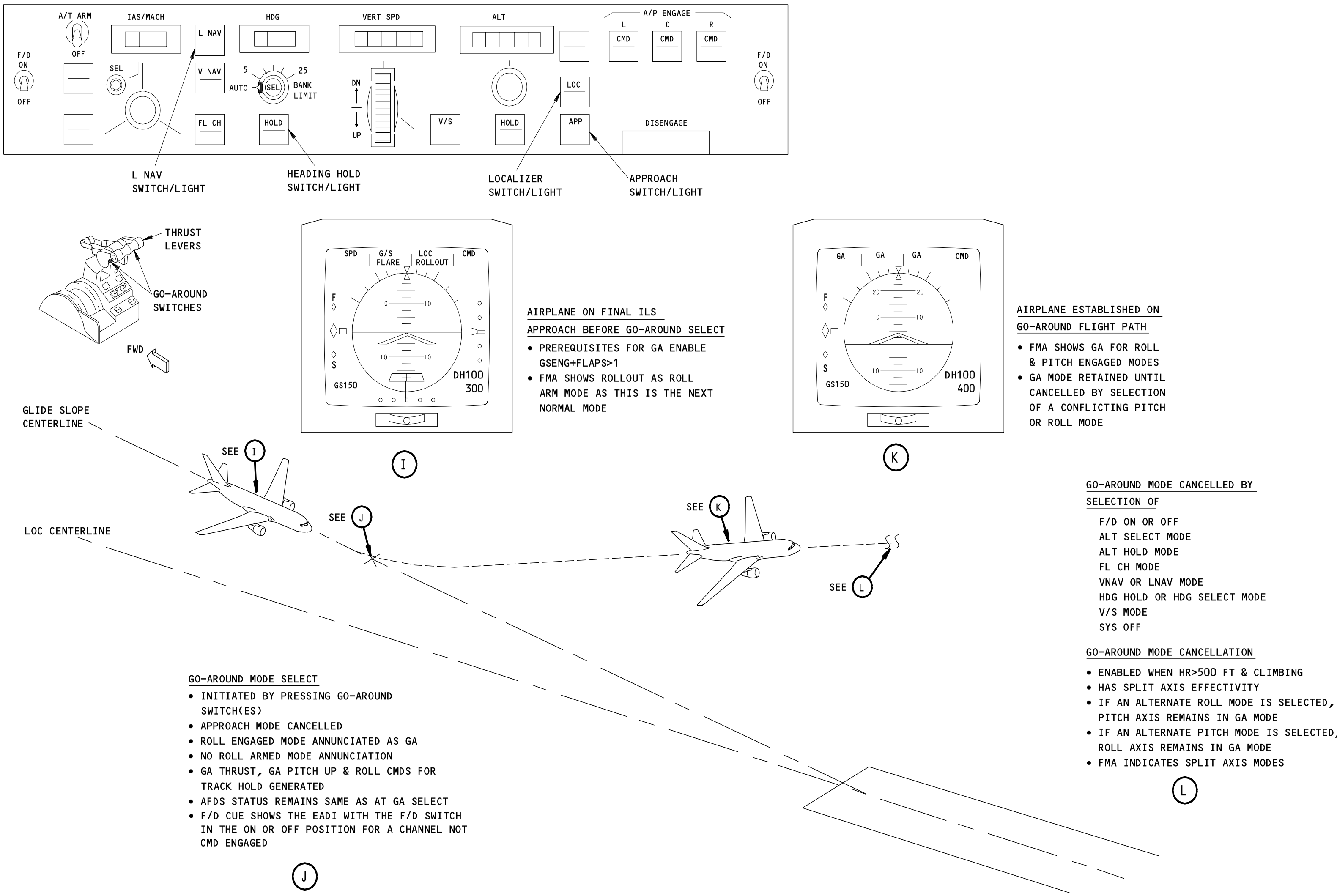

Roll Go-Around Mode
Figure 18 (Sheet 3)

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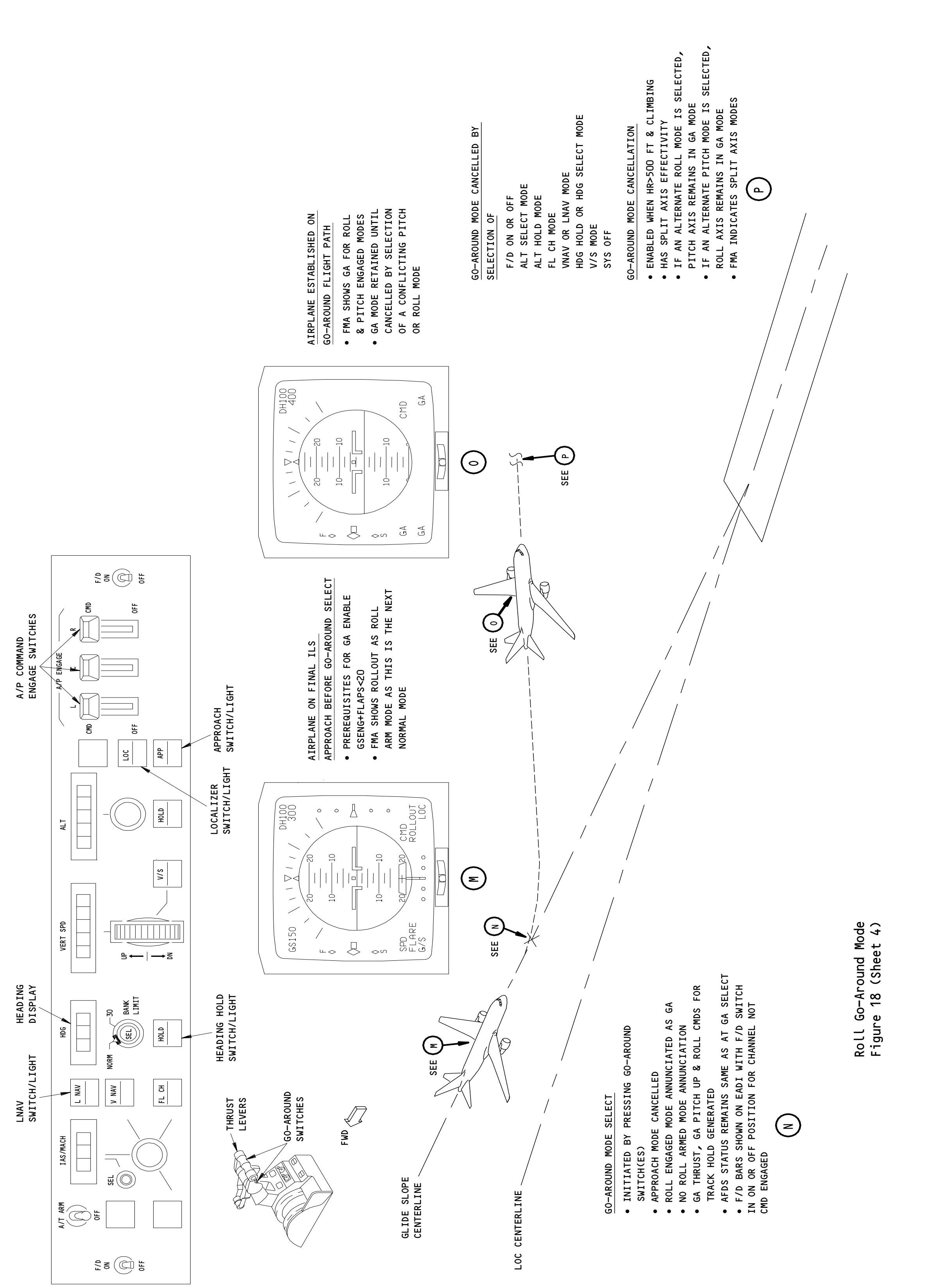


Roll Go-Around Mode
Figure 18 (Sheet 4)

2) GUI 115;
the MCP may show any of the following switch dot-bar matrices illuminated before entry into the go-around mode: HDG HOLD, LNAV, LOC APP. The F/D switches may be ON or OFF. The HDG display shows selected runway heading before G/A and MCP selected heading during G/A.

(b) Roll Go-Around Flight Path

1) The sequence of events if G/A is selected during an ILS approach is shown. Position A indicates pre-go-around conditions. The airplane is on final approach; G/S and flare are engaged; flaps less than 20 degrees. The EADI displays rollout white as the rolled armed mode since that would normally be the next mode.

2) Position B shows the G/A select conditions. Go-around is initiated by pressing the G/A switch on the thrust levers. The ILS guidance and annunciation are discontinued; G/A is shown as the roll engaged mode (with no roll armed mode annunciation); and G/A thrust, pitch-up commands and roll commands are generated. The AFDS status remains the same as before G/A thrust.

a) GUI 010-014, 016-999;
the flight director is shown on the EADI during G/A, with the F/D switch in the ON or OFF position.

b) GUI 001-009, 115;
the flight director is shown on the EADI during G/A, with the F/D switch in the ON or OFF position, for any channel not CMD engaged.

3) Position C indicates the G/A climbout. The EADI displays G/A green as the roll and pitch engaged modes. The G/A mode will be retained until cancelled by the selection of a conflicting pitch or roll mode.

4) Position D shows the G/A cancellation conditions. Cancellation is enabled when radio altitude is greater than 400 feet and the airplane is climbing. This produces a split axis condition. If an alternate roll mode is selected, the pitch axis remains in G/A mode and vice versa. The EADI FMA indicates split axis modes. Go-around can be cancelled by selection of F/D ON or OFF; altitude select or altitude hold; FLCH; VNAV or LNAV; HDG hold or select; V/S; or system OFF.

(23) Go-Around Mode Logic

(a) Go-around engage logic is generated by pressing the go-around switch on the thrust levers (Fig. 18). Go-around engage logic generates track mode select logic and go-around runway logic. After a one-frame delay, heading hold is generated. If the A/P is still CMD ENG, heading/track control and heading proportional command/track mode reference are generated.

(24) Go-Around Mode Signal Processing

(a) At go-around engage, the heading reference is synchronized to the adjusted runway heading which is compensated for drift angle (Fig. 11). The delta heading command from the A/P turn direction control is supplied to the roll F/D command processor. It is also used to generate the delta roll command. This is further processed to generate delta wheel command to the ALCS.

(b) Track mode select logic selects adjusted track as input to the F/D turn direction control. The F/D heading command changes slightly to generate roll commands reflecting this change of reference from heading to track. The A/P turn direction control also senses this change in reference. The delta heading command then becomes runway heading minus adjusted track in degrees roll. This output is then processed and waits until heading track control logic is generated one frame later.

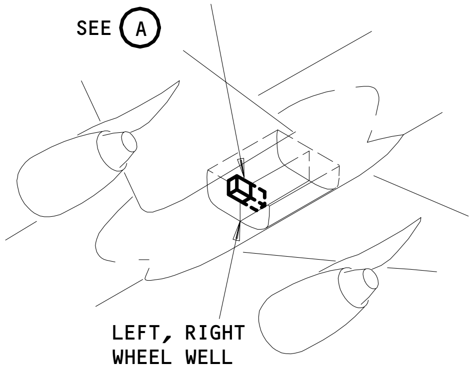
(c) After heading track control is generated, the delta heading command from the A/P turn direction control generates the heading proportional command and roll commands to hold ground track.

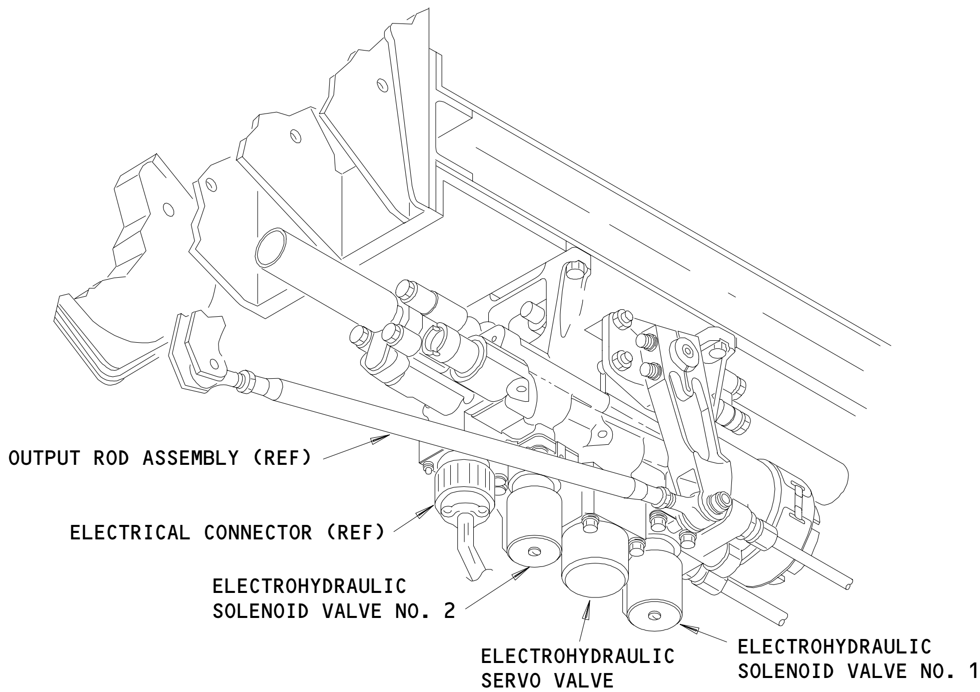
AUTOPILOT/FLIGHT DIRECTOR ROLL AND YAW CHANNEL

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
SERVO - C AUTOPILOT LATERAL CONTROL, M10041	1	1	L/R WHEEL WELL	22-13-03
SERVO - C AUTOPILOT ROLLOUT GUIDANCE, M278	2	1	VERT STAB., 324AL	22-13-01
SERVO - L AUTOPILOT LATERAL CONTROL, M10040	1	1	L/R WHEEL WELL	22-13-03
SERVO - L AUTOPILOT ROLLOUT GUIDANCE, M277	2	1	VERT STAB., 324AL	22-13-01
SERVO - R AUTOPILOT LATERAL CONTROL, M10042	1	1	L/R WHEEL WELL	22-13-03
SERVO - R AUTOPILOT ROLLOUT GUIDANCE, M279	2	1	VERT STAB., 324AL	22-13-01
VALVE - ALCS ELECTROHYDRAULIC SERVO	1	1	L/R WHEEL WELL, EA AUTOPILOT LATERAL CONTROL SERVO	22-13-04
VALVE - ALCS ELECTROHYDRAULIC SOLENOID	1	2	L/R WHEEL WELL, EA AUTOPILOT LATERAL CONTROL SERVO	22-13-04
VALVE - ARGS ELECTROHYDRAULIC SERVO	2	1	VERT STAB., 324AL, EA AUTOPILOT ROLLOUT GUIDANCE SERVO	22-13-02
VALVE - ARGS ELECTROHYDRAULIC SOLENOID	2	2	VERT STAB., 324AL, EA AUTOPILOT ROLLOUT GUIDANCE SERVO	22-13-02

* SEE THE WDM EQUIPMENT LIST

Autopilot/Flight Director Roll and Yaw Channel - Component Index
Figure 101

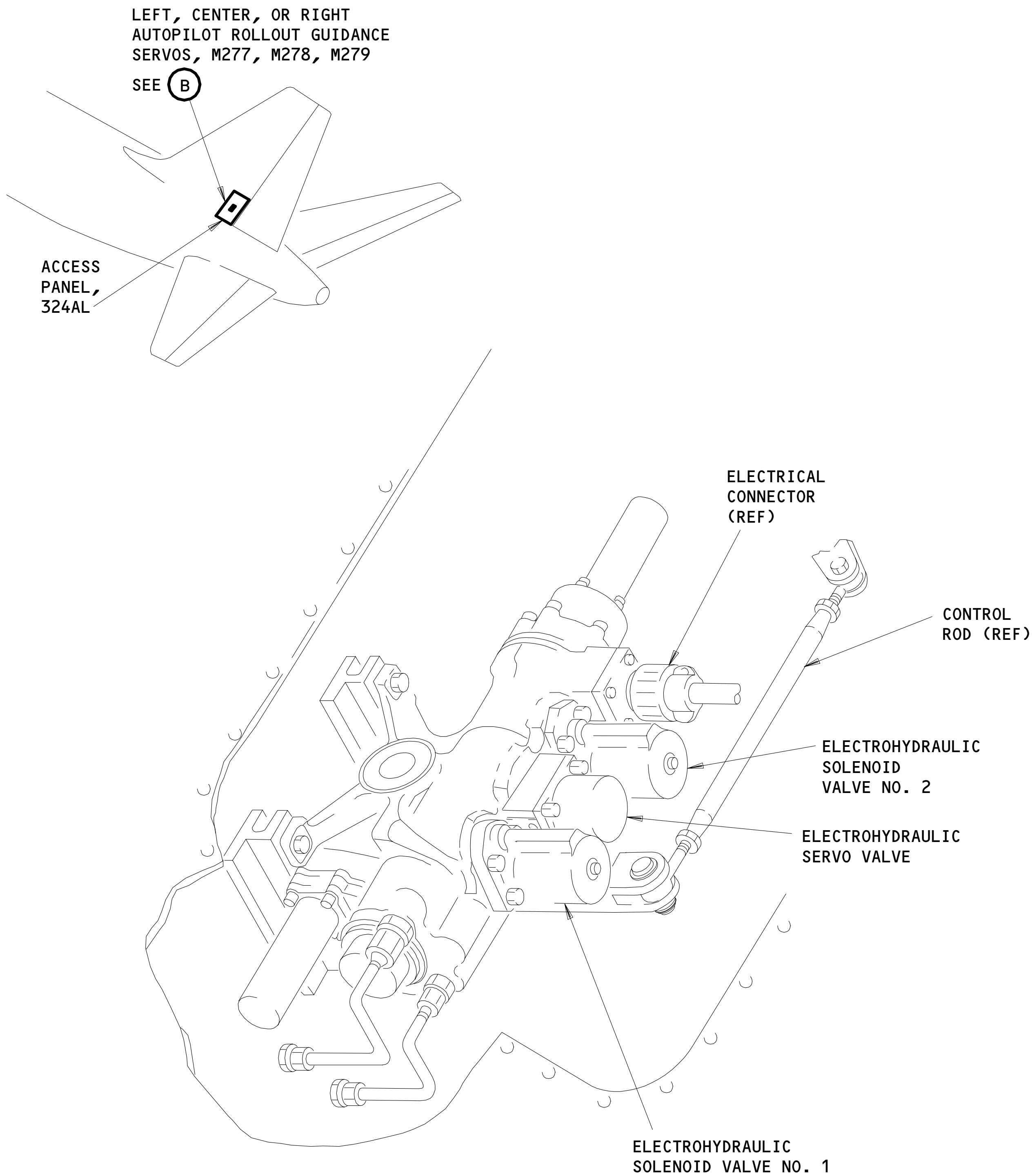




LEFT, CENTER, OR RIGHT AUTOPILOT LATERAL CONTROL SERVO, M10040,M10041 OR M10042

A

Autopilot/Flight Director Roll and Yaw Channel - Component Location
Figure 102 (Sheet 1)



LEFT, CENTER, OR RIGHT AUTOPILOT
ROLLOUT GUIDANCE SERVO, M277, M278, OR M279

B

Autopilot/Flight Director Roll and Yaw Channel - Component Location
Figure 102 (Sheet 2)

AUTOPILOT ROLLOUT GUIDANCE SERVO - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. The three Autopilot Rollout Guidance Servos (ARGSs) are adjacent to each other at the bottom of the vertical stabilizer. They are immediately in front of the rudder hinge line. Each servo has an output rod connected to the rudder feel, centering, and trim mechanism.
   B. This procedure includes tasks to remove, install, adjust, and test the Autopilot Rollout Guidance Servos.
   C. There are two adjustment tasks for the Autopilot Rollout Guidance Servo:
      (1) Adjustment with the Maintenance Control Display Panel (MCDP)
      (2) Adjustment with the Phase Synchronous Voltmeter Model 101A

TASK 22-13-01-014-001

2. Remove the Autopilot Rollout Guidance Servo
   A. Equipment
      (1) Rig Pin R3-P/N B20003-16, part of set B20003-XX (AMM 20-10-24/201)
   B. References
      (1) AMM 06-42-00/201, Empennage and Section 48 (Major Zone 300) Access Doors and Panels
      (2) AMM 20-10-24/201, Rig Pins
      (3) AMM 24-22-00/201, Electrical Power - Control
      (4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   C. Access
      (1) Location Zone
          324 Vertical Stabilizer, Rear Spar to Trailing Edge

      (2) Access Panel
          324AL Body to Fin Seal

   D. Prepare for Removal

      S 864-007
      (1) Supply electrical power (AMM 24-22-00/201).

      S 864-006
      (2) Remove the pressure from the left, center and right hydraulic systems (AMM 29-11-00/201).

S 864-005

(3) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the OFF position.

S 864-004

(4) Set the two STAB TRIM switches on control stand panel P10 in the CUTOUT position.

S 864-003

(5) Set the two F/D switches on the MCP in the OFF position.

S 864-002

(6) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF LEFT
   (b) 11C13, STAB TRIM SHUTOFF RIGHT
   (c) 11H17, FLT CONT SHUTOFF LEFT
   (d) 11H18, FLT CONT SHUTOFF CENTER
   (e) 11H28, FLT CONT SHUTOFF RIGHT
   (f) For the left servo:
      1) 11E17, FLT CONT CMPTR PWR LEFT
      2) 11E18, FLT CONT CMPTR SERVO LEFT
   (g) For the center servo:
      1) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
      2) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER
   (h) For the right servo:
      1) 11E35, FLT CONT CMPTR PWR RIGHT
      2) 11E36, FLT CONT CMPTR SERVO RIGHT

S 014-008

(7) Remove the access panel 324AL, on the base of the vertical fin (left side), to get access to the Autopilot Rollout Guidance Servos (AMM 06-42-00/201).

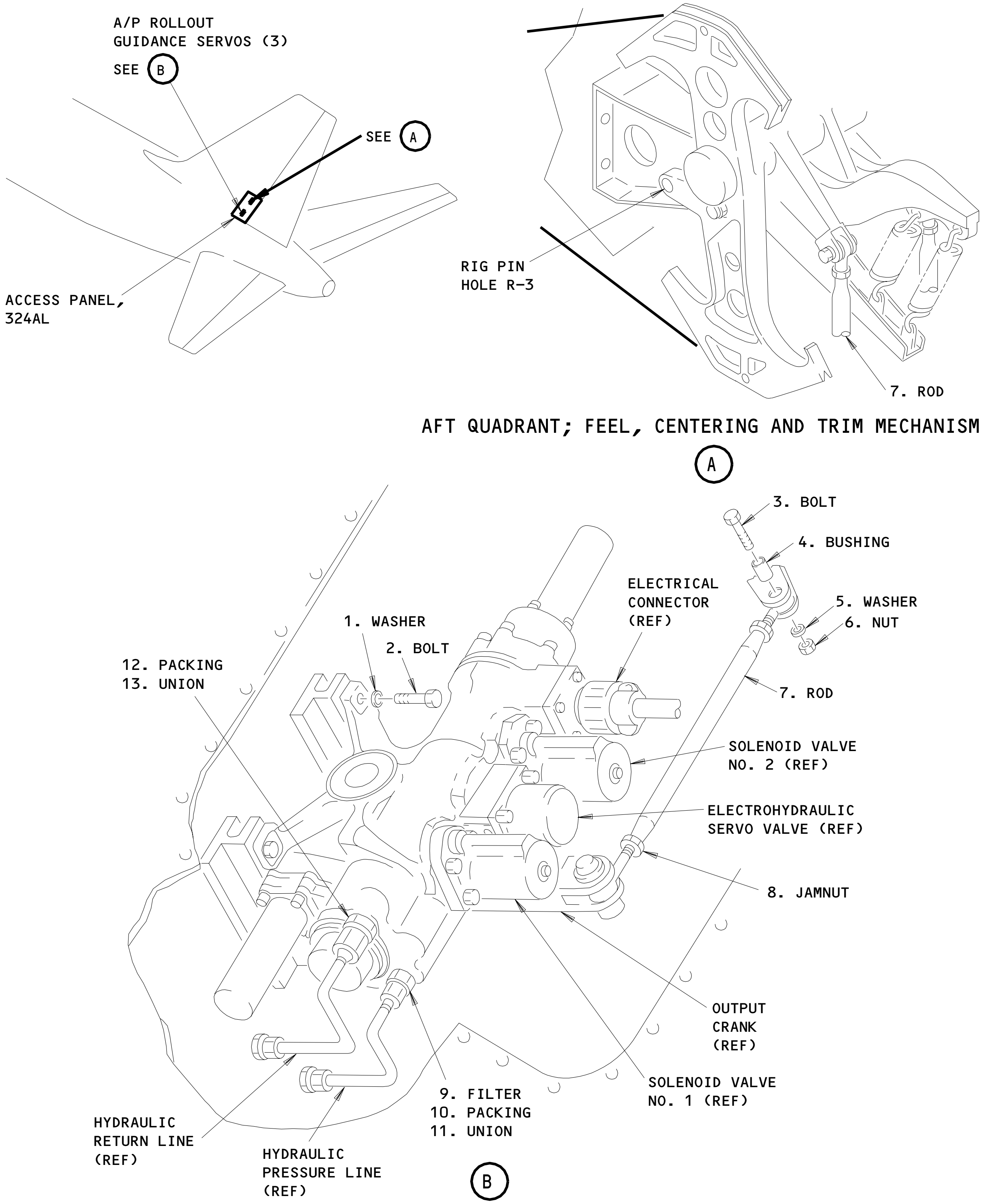
E. Remove the Autopilot Rollout Guidance Servo

S 484-017

(1) Install the rig pin R3 in the aft quadrant (Refer to Fig. 401).

S 034-016

(2) Disconnect the electrical connector from the Autopilot Rollout Guidance Servo.



Autopilot Rollout Guidance Servo (ARGS)
Figure 401

S 034-015
(3) Disconnect the hydraulic lines from the servo.

S 034-014
(4) Loosen the jamnuts (8) on the control rod (7).

S 034-013
(5) Disconnect the control rod (7) by removing the bolt (3), the nut (6), the washer (5), and the bushing (4) from the quadrant assembly crank.

S 034-012
(6) Remove the control rod (7) from the end of the servo arm rod.

S 034-011
(7) Remove the bolts (2) and washers (1) from the servo unit.

S 024-010
(8) Remove the servo from the vertical stabilizer.

S 394-009
(9) Seal the hydraulic lines with caps.

TASK 22-13-01-404-018
3. Install the Autopilot Rollout Guidance Servo
A. Equipment
(1) Rig Pin R3-P/N B20003-16, part of set A20003-XX (AMM 20-10-24/201)
B. Parts

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401		Servo Assembly	22-13-01	01	50

C. Reference
   (1) AMM 20-10-24/201, Rig Pins
D. Access
   (1) Location Zone
       324 Vertical Stabilizer, Rear Spar to Trailing Edge

   (2) Access Panel
       324AL Body to Fin Seal

E. Install the Autopilot Rollout Control Servo.

   S 414-020
   (1) Install the rig pin R3 in the aft quadrant if it is not installed (AMM 20-10-24/201).

   S 014-021
   (2) Remove the caps from the hydraulic lines.

   S 414-022
   (3) Put the servo in its position on the holes for the bolts.

   S 434-023
   (4) Put the bolts (2) and washers (1) in the holes and tighten them by hand.

   S 434-024
   (5) Connect the hydraulic lines to the servo and tighten.

   S 434-025
   (6) Tighten the bolts (2) that hold the servo.

   S 434-026
   (7) Connect the electrical connector to the servo.

   S 434-027
   (8) Put the jamnut (8) on the servo arm rod end.
      (a) Turn the jamnut until it gets to the end of the threads.

S 434-028
(9) Install the control rod assembly (7) on the servo arm rod end.
(a) Turn the control rod assembly until there is 18.25 inches from bolt center to bolt center.

S 434-077
(10) Use the bolt (3), the bushing (4), the washer (5), and the nut (6) to attach the control rod (7) to the quadrant assembly crank.

S 084-089
(11) Remove rig pin R3.

S 824-029
(12) Adjust the Servo with the Maintenance Control Display Panel (MCDP) or Model 101A Phase Synchronous Voltmeter.

TASK 22-13-01-824-019
4. Adjust the Servo with the Maintenance Control Display Panel (MCDP)
A. Equipment
(1) Rig Pin R3-P/N B20003-16, part of set B20003-XX (AMM 20-10-24/201)
B. References
(1) AMM 20-10-24/201, Rig Pins
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
C. Access
(1) Location Zone
324 Vertical Stabilizer, Rear Spar to Trailing Edge

(2) Access Panel
324AL Body to Fin Seal

D. Adjust the Servo.

S 864-031
(1) Supply electrical power (AMM 24-22-00/201).

S 864-030

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C12, STAB TRIM SHUTOFF LEFT
   (b) 11C13, STAB TRIM SHUTOFF RIGHT
   (c) 11E17, FLT CONT CMPTR PWR LEFT
   (d) 11E18, FLT CONT CMPTR SERVO LEFT
   (e) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
   (f) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER
   (g) 11E35, FLT CONT CMPTR PWR RIGHT
   (h) 11E36, FLT CONT CMPTR SERVO RIGHT
   (i) 11H17, FLT CONT SHUTOFF TAIL LEFT
   (j) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (k) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-084

WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF CONTROL SURFACES AND CONTROL COLUMNS BEFORE HYDRUALIC SYSTEMS ARE PRESSURIZED. THE CONTROL SURFACES CAN MOVE WHEN A HYDRAULIC SYSTEM IS PRESSURIZED. THIS COULD CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the ON position.

S 864-085

(4) Set the L and R STAB TRIM switches on the P10 panel in the NORMAL position.

S 864-086

(5) Pressurize the left, center, and right hydraulic systems (AMM 29-11-00/201).

S 824-038

(6) Use the rudder trim switch to center the rudder until the trailing edge is less than ±.15 inches from the index mark.

S 434-076
(7) Put rig pin R3 into its hole.
    (a) Make sure the rig pin is put into its hole freely.
    (b) Rig pin R3 will stay installed during the Autopilot Rollout Guidance Servo adjustment.

S 864-098
(8) Remove hydraulic pressure (AMM 29-11-00/201).

S 744-037
(9) Do the MCDP Ground Test 66-XDCR OUTPUTS (AMM 22-00-02/201).
    (a) Use the YES/ADV switch to move the test forward until the top line of the MCDP display shows 66 RUD SURF DEG.

S 824-036
(10) Adjust the control rod (7) until the MCDP display shows 0.00 ±.6 degree with the control rod in its installed position in the quadrant assembly crank.

S 434-035
(11) Tighten the control rod jamnuts (8).

S 214-090
(12) Make sure the MCDP display shows 0.00 ±.6 degrees.

S 214-091
(13) Make sure there is a sufficient length of thread engaged at each rod end.

S 084-033
(14) Remove rig pin R3.

S 714-032
(15) Do the "Test the Servo Installation" steps.

TASK 22-13-01-824-039
5. <u>Adjust the Servo with the Model 101A Phase Synchronous Voltmeter</u>
  A. Equipment
    (1) Rig Pin R3-P/N B20003-16, part of set B20003-XX (AMM 20-10-24/201)
    (2) Voltmeter, Phase Synchronous Model 101A Kit - 101AC5
        Electronic Aviation Systems,
        300 Airport Way,
        Renton, WA 98005
        (a) Voltmeter, Phase Synchronous
        (b) Adapter Cable Assembly 1014002-5

B. References
  (1) AMM 20-10-24/201, Rig Pins
  (2) AMM 24-22-00/201, Electrical Power - Control
  (3) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

C. Access
  (1) Location Zone
      324 Vertical Stabilizer, Rear Spar to Trailing Edge

  (2) Access Panel
      324AL Body to Fin Seal

D. Prepare To Adjust the Servo

S 864-092

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
  (a) 11H17, FLT CONT SHUTOFF TAIL LEFT
  (b) 11H18, FLT CONT SHUTOFF TAIL CENTER
  (c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-093

WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF CONTROL SURFACES AND CONTROL COLUMNS BEFORE HYDRUALIC SYSTEMS ARE PRESSURIZED. THE CONTROL SURFACES CAN MOVE WHEN A HYDRAULIC SYSTEM IS PRESSURIZED. THIS COULD CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the ON position.

S 864-094

(3) Pressurize the left, center, and right hydraulic systems (AMM 29-11-00/201).

S 824-095

(4) Use the rudder trim switch to center the rudder until the trailing edge is less than ±.15 inches from the index mark.

S 434-096
(5) Put rig pin R3 into its hole.
  (a) Make sure the rig pin is put into its hole freely.
  (b) Rig pin R3 will stay installed during the Autopilot Rollout Guidance Servo adjustment.

S 864-097
(6) Remove hydraulic pressure (AMM 29-11-00/201).

S 484-044
(7) Connect the Phase Synchronous Voltmeter to the power source given on the voltmeter placard.

S 864-043
(8) Set the ON/OFF (S1) switch in the ON position.
  (a) Permit 10 minutes for the voltmeter to become warm.

S 864-042
(9) Set the CAL/RUN (S2) switch in the CAL CHK position.
  (a) Make sure the IN PHASE VOLTS are 9.990 to 10.010.

S 864-041
(10) Set the CAL/RUN (S2) switch in the RUN position.

S 034-040
(11) Disconnect the electrical connector from the servo.

S 434-047
(12) Connect the -5 cable assembly to the Autopilot Rollout Guidance Servo and the Phase Synchronous Voltmeter.

E. Adjust the Servo

S 824-051
(1) Adjust the control rod (7) until the Phase Syncronous Voltmeter shows 0 ±.05 IN PHASE VOLTS.

S 434-049
(2) Tighten the control rod jamnuts (8).

S 214-100
(3) Make sure the Phase Syncronous Voltmeter shows 0 ±.05 IN PHASE VOLTS.

S 214-099
(4) Make sure there is a sufficient length of thread engaged at each rod end.

S 084-048
(5) Disconnect the Phase Synchronous Voltmeter.
  (a) Set the ON/OFF switch in the OFF position.

(b) Disconnect the -5 cable assembly from the autopilot servo.
(c) Disconnect the voltmeter from the power source.

S 434-054
(6) Connect the airplane electrical connector to the autopilot servo.

S 084-053
(7) Remove rig pin R3.

S 714-052
(8) Do the "Test the Servo Installation" steps.

TASK 22-13-01-714-055
6. Test the Servo Installation
A. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
B. References
(1) AMM 09-11-00/201, Towing
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
C. Access
(1) Location Zone
324 Vertical Stabilizer, Rear Spar to Trailing Edge

(2) Access Panel
324AL Body to Fin Seal

D. Test the Servo

S 864-074

WARNING: LOCK THE NOSE GEAR STEERING WHEN MOVEMENT OF THE RUDDER IS SCHEDULED. THIS WILL PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Move the towing lever on the nose gear metering valve module to the TOW position.

S 434-075
(2) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 864-057
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT
(c) 11E17, FLT CONT CMPTR PWR LEFT
(d) 11E18, FLT CONT CMPTR SERVO LEFT
(e) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
(f) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER

(g) 11E35, FLT CONT CMPTR PWR RIGHT
(h) 11E36, FLT CONT CMPTR SERVO RIGHT
(i) 11H17, FLT CONT SHUTOFF TAIL LEFT
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-058

WARNING: MAKE SURE ALL PERSONS AND STANDS ARE CLEAR OF CONTROL SURFACES AND CONTROL COLUMNS BEFORE HYDRUALIC SYSTEMS ARE PRESSURIZED. THE CONTROL SURFACES CAN MOVE WHEN A HYDRAULIC SYSTEM IS PRESSURIZED. THIS COULD CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the ON position.

S 864-059
(5) Set the L and R STAB TRIM switches on the P10 panel in the NORMAL position.

S 864-060
(6) Pressurize the left, center, and right hydraulic systems (AMM 29-11-00/201).

S 984-068
(7) Push the left and right rudder pedals through full travel three times to operate the rudder system.

S 824-080
(8) Use the rudder trim switch to center the rudder until the trailing edge is less than ±.15 inches from the index mark.
(a) Make sure you can freely put rig pin R3 into its hole.
(b) Remove rig pin R3.

S 744-067
(9) Do the MCDP Ground Test 66-XDCR OUTPUTS (AMM 22-00-02/201).
(a) Use the YES/ADV switch to move the test forward until the top line of the MCDP shows 66 RUD SURF DEG.
(b) With the rudder pedals released, make sure the MCDP shows 00.0 ±0.6 degree on the bottom line of the display.

S 744-063
(10) Do the MCDP Ground Test 09-SERVO RUD (AMM 22-00-02/201).
(a) Make sure no MCDP fault messages show during the test.

S 744-083
(11) Do the MCDP Ground Test 69-RUD SURF LIM (AMM 22-00-02/201).
(a) Make sure the rudder positive and negative limits are 22.8 degrees minimum for the left, center, and right servos.

S 214-061

(12) Examine the Autopilot Rollout Control Servos for leaks.

(a) Repair leaks that you find.

E. Put the Airplane Back to Its Usual Condition

S 034-073

(1) Remove the DO-NOT-OPERATE tags.

S 084-072

<u>WARNING</u>: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN YOU REMOVE LOCKPIN. THE NOSE WHEELS CAN MOVE TO THE CENTER POSITION QUICKLY. THIS CAN CAUSE INJURY TO PERSONS OR CAUSE DAMAGE TO EQUIPMENT.

(2) Put the nose gear wheels in the center position.

S 084-101

(3) Remove the nose gear steering valve lockpin.

S 864-071

(4) Set the MCDP to off.

S 864-070

(5) Remove the hydraulic power if it is not necessary (AMM 29-11-00/201).

S 864-069

(6) Remove the electrical power if is not necessary (AMM 24-22-00/201).

ARGS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - MAINTENANCE PRACTICES

1. General
   A. The three Autopilot Rollout Guidance Servos (ARGS) are in the base of the vertical stabilizer, immediately forward of the rudder hinge line. Each servo unit has one electrohydraulic servovalve (EHSV) and two solenoid valves. The servos can be accessed through panel 324AL on the lower left side of the vertical stabilizer (Ref 6-42-00).

TASK 22-13-02-002-001

2. Remove the ARGS Electrohydraulic Servovalve or Solenoid Valves (Fig. 201)
   A. References
      (1) 06-42-00/201, Empennage and Section 48 (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   B. Access
      (1) Location Zone
          324 Vertical Stabilizer, Rear Spar to Trailing Edge

      (2) Access Panel
          324AL Body to Fin Seal

   C. Prepare for Removal

      S 862-002
      (1) Supply electrical power (Ref 24-22-00).

      S 862-003
      (2) Remove the pressure from the L, C, and R hydraulic system (Ref 29-11-00).

      S 862-004
      (3) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel to OFF.

      S 862-005
      (4) Set the L and R STAB TRIM switches on the control stand panel P10 to CUTOUT.

      S 862-006
      (5) Set the two F/D switches on the MCP to OFF.

S 862-007

(6) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF LEFT
   (b) 11C13, STAB TRIM SHUTOFF RIGHT
   (c) 11H17, FLT CONT SHUTOFF LEFT
   (d) 11H18, FLT CONT SHUTOFF CENTER
   (e) 11H28, FLT CONT SHUTOFF RIGHT
   (f) For the left servo:
      1) 11E17, FLT CONT COMPUTER POWER LEFT
      2) 11E18, FLT CONT COMPUTER SERVO LEFT
   (g) For the center servo:
      1) 11E19 or 11E20, FLT CONT CMPTR PWR CENTER
      2) 11E20 or 11E21, FLT CONT CMPTR SERVO CENTER
   (h) For the right servo:
      1) 11E35, FLT CONT CMPTR PWR RIGHT
      2) 11E36, FLT CONT CMPTR SERVO RIGHT

S 862-008

(7) Remove the access panel 324AL at the base of the vertical fin (left side) to gain access to the rudder control servos (Ref 06-42-00).

D. Remove the Valve

S 142-009

CAUTION: DO NOT LET CONTAMINATION GET INTO THE AUTO PILOT ROLLOUT GUIDANCE SERVO. THE AUTOPILOT ROLLOUT GUIDANCE SERVO IS A VERY SENSITIVE ELECTROHYDRAULIC DEVICE. CONTAMINATION IN THE SYSTEM CAN CAUSE DAMAGE TO THE EQUIPMENT OR FAILURE.

(1) Carefully clean the external areas of the valve and around the valve mounting surface.

S 032-010

(2) Remove the lockwire.

S 022-011

(3) Remove the screws, the valve with O-ring, and the seal plate from the servo assembly.

S 432-012

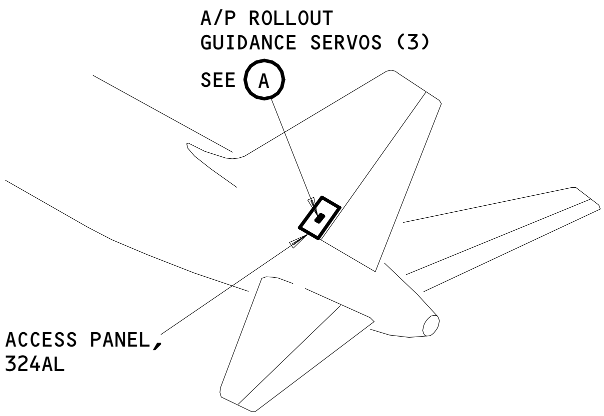
(4) Put covers on the hydraulic and the electrical servo openings to make sure contamination does not get into them.

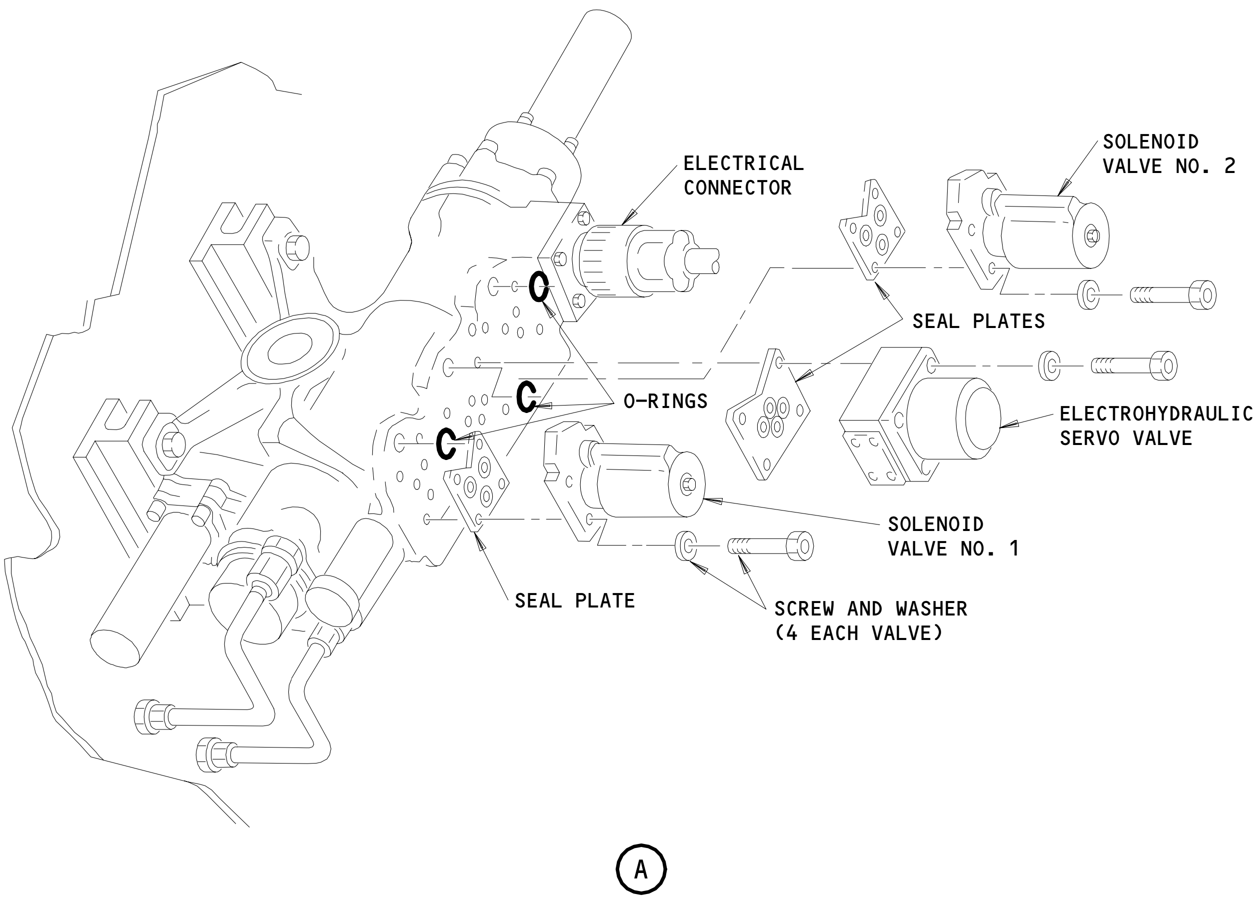
TASK 22-13-02-402-013

3. Install the ARGS Electrohydraulic Servovalve or Solenoid Valve (Fig. 201)

A. Equipment

(1) Nose Gear Steering Valve Lockpin - A09003-1





ARGS Electrohydraulic Servo Valve and Solenoid Valves
Figure 201

B. Consumable Materials
   (1) D00148 Hydraulic Fluid - fire resistant, BMS 3-11
C. References
   (1) 06-42-00/201, Empennage and Section 48 (Major Zone 300) Access Doors and Panels
   (2) 09-11-00/201, Towing
   (3) 22-00-02/201, Autoflight BITE
   (4) 24-22-00/201, Electrical Power - Control
   (5) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
D. Access
   (1) Location Zone
       324 Vertical Stabilizer, Rear Spar to Trailing Edge

   (2) Access Panel
       324AL Body to Fin Seal

E. Install the Valve

   S 032-014
   (1) Remove the covers from the openings.

   S 142-015
   (2) Clean the mating surfaces of the valve, the new seal plate, and the servo assembly with hydraulic fluid. Do not let hydraulic fluid go into the electrical connector.

   S 432-016
   (3) Install a new O-ring on the valve electrical connector.

   S 432-017
   (4) Install the valve and the new seal plate.

   S 432-018
   (5) Install the screws with grease.
      (a) Tighten the screws to 60 pound-inches.

S 432-019

(6) Put a lockwire on the screws.

F. Test the ARGS Electrohydraulic Servovalve and Solenoid Valves

S 982-020

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY RUDDER MOVEMENT IS SCHEDULED. THIS MUST BE DONE TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Move the towing lever on the nose gear metering valve module to the TOW position.

S 862-022

(2) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 862-023

(3) Supply electrical power (Ref 24-22-00).

S 862-024

(4) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT
(c) 11E17, FLT CONT COMPUTER POWER LEFT
(d) 11E18, FLT CONT COMPUTER SERVO LEFT
(e) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(f) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(g) 11E35, FLT CONT CMPTR PWR RIGHT
(h) 11E36, FLT CONT CMPTR SERVO RIGHT
(i) 11H17, FLT CONT SHUTOFF TAIL LEFT
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 862-025

(5) Set the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel to ON.

S 862-026
(6) Set the STAB TRIM switches on the P10 panel to NORMAL.

S 862-027

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(7) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 982-029
(8) Manually operate the rudder system three or more times.

S 722-030
(9) Do MCDP Ground Test 09 - SERVO RUD (Ref 22-00-02).

S 862-041
(10) Set the L, C, or R FLT CONTROL SHUTOFF switches to OFF and attach DO-NOT-OPERATE tags.

S 862-043
(11) Set the L and R STAB TRIM switches to CUTOUT and attach DO-NOT-OPERATE tags.

S 212-023
(12) Examine the ARGS valves for hydraulic leakage and correct as necessary.

G. Put the Airplane Back to Its Usual Condition

S 032-032
(1) Remove the DO-NOT-OPERATE tags.

S 862-033
(2) Set the L, C, and R FLT CONTROL SHUTOFF switches to ON.

S 862-034
(3) Set the L and C STAB TRIM switches to NORMAL.

S 212-035

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN THE LOCK PIN IS THE NOSE WHEELS CAN MOVE QUICKLY TO THE CENTERED POSITION. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Make sure the nose gear wheels are centered and remove the nose gear steering valve lock pin (Ref 09-11-00).

S 862-036
(5) Set the MCDP to off.

S 412-037
(6) Install the access panel 324AL (Ref 06-42-00).

S 862-038
(7) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 862-039
(8) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT LATERAL CONTROL SERVO - REMOVAL/INSTALLATION

1. General
   A. The left, center, and right autopilot lateral control servos (ALCSs) are in the main landing gear wheel well. The left and center ALCSs are left of the keel beam, with the left in the forward position and the center in the aft position. The right ALCS is to the right of the keel beam.
   B. You must fully remove the control rod before you remove the servo. After the installation of the servo, the control rod is installed again.
   C. Two adjustment procedures are given for the Autopilot Lateral Control Servo:
      (1) Adjustment with the Maintenance Control Display Panel (MCDP)
      (2) Adjustment with the Phase Synchronous Voltmeter Model 101A

TASK 22-13-03-004-001

2. Remove the Autopilot Lateral Control Servo
   A. Equipment
      (1) Rig Pins from Set B20003-XX (Ref 20-10-24)
         (a) A1-P/N B20003-27
         (b) A2-P/N B20003-27
   B. References
      (1) 20-10-24/201, Rig Pins
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   C. Access
      (1) Location Zones
         732/742   Main Landing Gear Body Door

      (2) Access Panels
         732/742   Main Landing Gear Body Door

   D. Prepare for Removal

      S 864-002
      (1) Supply electrical power (Ref 24-22-00).

S 864-003
(2) Remove pressure from the L, C, and R hydraulic systems (Ref 29-11-00).

S 864-004
(3) Set the two F/D switches on the MCP to OFF.

S 864-005
(4) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
(a) 11E17, FLT CONT COMPUTER POWER LEFT
(b) 11E18, FLT CONT COMPUTER SERVO LEFT
(c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(e) 11E35, FLT CONT CMPTR PWR RIGHT
(f) 11E36, FLT CONT CMPTR SERVO RIGHT

E. Remove the Servo

S 984-006
(1) Rotate the quadrants until the rig pins A1 and A2 can be freely put in the left and right quadrant assemblies.

S 434-007
(2) Install the rig pins.

S 034-008
(3) Remove the electrical connector from the ALCS.

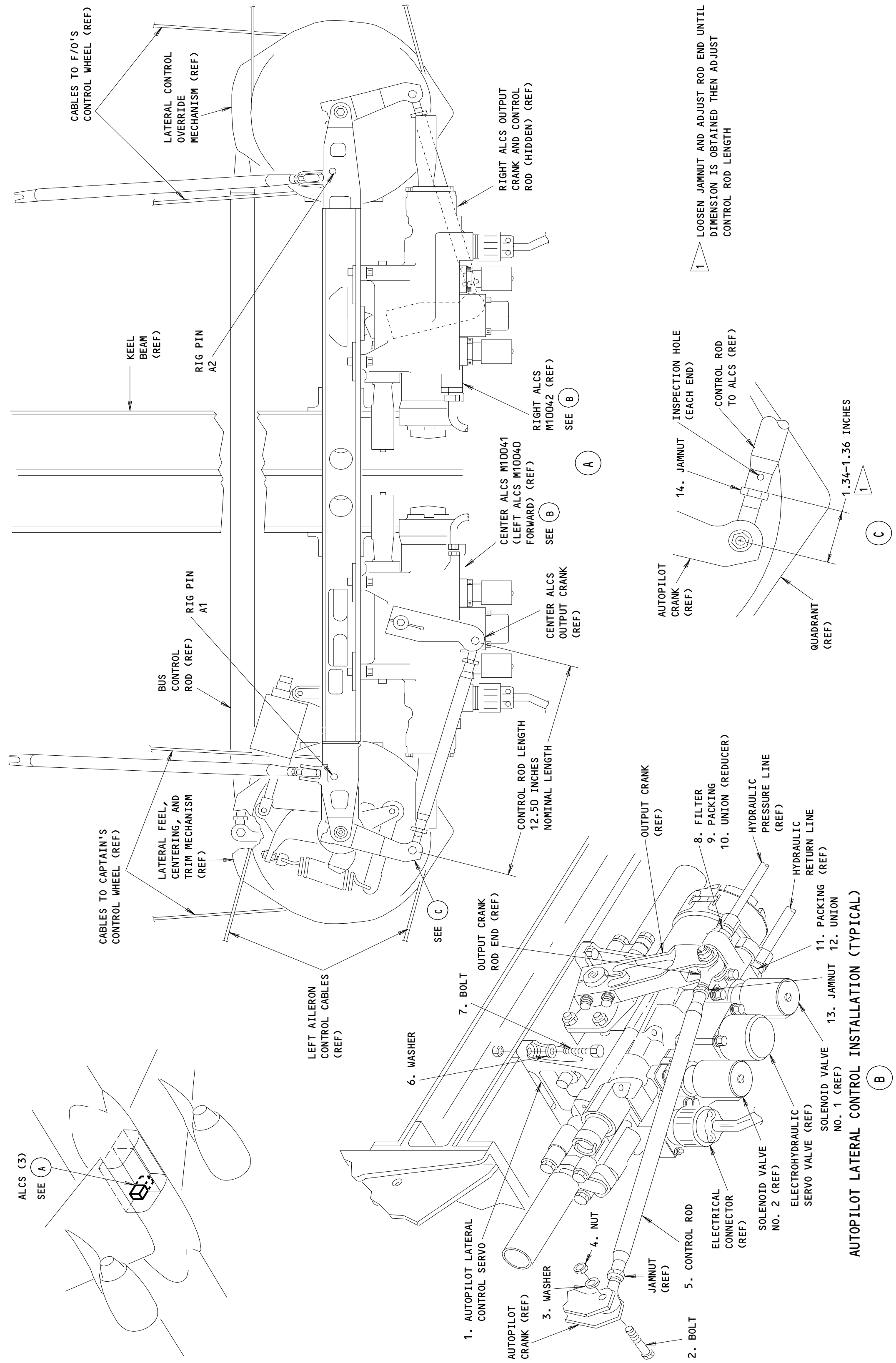
S 034-009
(4) Disconnect the hydraulic lines from the servo.

S 034-010
(5) Loosen the jamnut (13) on the control rod at the servo output crank rod end.

NOTE: The servo output crank rod end is part of the servo output crank.

S 034-011
(6) Remove the bolt (2), the nut (4), and the washer (3) to disconnect the control rod (5) from the quadrant assembly autopilot crank.

S 034-012
(7) Remove the control rod (5) from the servo output crank rod end.



Autopilot Lateral Control Servo Installation (ALCS)
Figure 401

S 034-068

WARNING: HOLD THE SERVO WHEN THE BOLTS AND THE WASHERS THAT ATTACH THE SERVO TO THE AIRPLANE ARE REMOVED. THE SERVO ASSEMBLY WEIGHS 17 POUNDS. IF THE SERVO ASSEMBLY FALLS ON A PERSON INJURY CAN OCCUR.

(8) Remove the bolts (7) and the washers (6) from the servo unit.

S 034-013
(9) Remove the servo from the airplane structure.

S 434-014
(10) Put caps on the hydraulic lines.

TASK 22-13-03-404-015
3. Install the Autopilot Lateral Control Servo (Fig 401)
A. Equipment
(1) Rig Pins from Set B20003-XX (Ref 20-10-24)
(a) A1-P/N B20003-27
(b) A2-P/N B20003-27
(2) Voltmeter, Phase Synchronous Model 101A kit - 101AC5
Electronic Aviation Systems
300 Airport Way
Renton, WA 98005
(Required when not using MCDP for rigging)
(a) Voltmeter, Phase Synchronous
(b) Adapter Cable Assemblies 1014002-1, -5
B. Consumable Materials
(1) D00148 Hydraulic Fluid - fire resistant, BMS 3-11
C. Parts

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401	1	Lateral Servo Assembly	27-11-14	02	110
	1	Lateral Servo Assembly	22-13-03	01	5
	8	Filter	22-11-51	01	265
	8	Filter	22-13-03	10	60
	9	Packing	22-11-51	01	260
	9	Packing	22-13-03	10	50
	11	Packing	22-11-51	01	275
	11	Packing	22-13-03	10	70

D. References
   (1) 20-10-24/201, Rig Pins
   (2) 20-30-04/201, Lubricants
   (3) 22-00-02/201, Autoflight BITE
   (4) 24-22-00/201, Electrical Power - Control
   (5) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

E. Access
   (1) Location Zones
       732/742   Main Landing Gear Body Door

   (2) Access Panels
       732/742   Main Landing Gear Body Door

F. Install the servo.

   S 434-016
   (1) Install rig pins A1 and A2 if they are not installed.

   S 034-017
   (2) Remove the caps from the hydraulic lines.

   S 434-018
   (3) Put the servo in position and install the bolts (7) and the washers (6) by hand.

   S 434-019
   (4) Connect the hydraulic lines and tighten the fittings by hand.

   S 434-020
   (5) Tighten the servo bolts (7).

   S 434-021
   (6) Tighten the hydraulic line fittings.

   S 434-022
   (7) Install the jamnut (13) onto the rod end on the servo output crank.

S 434-023
(8) Adjust the rod end on the autopilot crank end of the control rod to 1.34 to 1.36 inches. Refer to Fig. 401, item C.
(a) Tighten the jamnut.

S 434-024
(9) Put the control rod assembly (5) onto the servo output crank rod end.

S 824-025
(10) Adjust the output crank end of the control rod nominal length to 1.34 to 1.36 inches.

NOTE: The nominal length is used to reduce the amount of adjustment required to adjust the servo to null.

(a) Tighten the jamnut (13).

S 434-026
(11) Attach the control rod assembly (5) to the quadrant autopilot crank with the bolt (2).

S 434-027
(12) Connect the electrical connector to the servo.

S 824-028
(13) Adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

G. Adjust the Servo with the Maintenance Control Display Panel (MCDP)

S 434-029
(1) Install rig pins A1 and A2 if they are not installed.

S 864-030
(2) Supply electrical power (Ref 24-22-00).

S 864-031

(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E17, FLT CONT COMPUTER POWER LEFT
(b) 11E18, FLT CONT COMPUTER SERVO LEFT
(c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(e) 11E35, FLT CONT CMPTR PWR RIGHT
(f) 11E36, FLT CONT CMPTR SERVO RIGHT

S 864-032

(4) Make sure these circuit breakers on the P11 panel are closed:

(a) 11C30, LANDING GEAR POS SYS 1
(b) 11E16, MODE CONT PNL LEFT
(c) 11E34, MODE CONT PNL RIGHT
(d) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(e) GUI 115;
11S6, MAINT CONT DSPL
(f) 11S15, AIR/GND SYS 1
(g) 11S19, AIR/GND SYS 2
(h) 11S23, POS SYS 2

S 714-033

(5) Start MCDP Ground Test 66 - XDCR OUTPUTS (Ref 22-00-02) with hydraulic power off and rig pins A1 and A2 installed.

S 984-034

(6) Push YES/ADV until 66 AIL SURF DEG, ±XX.X ±XX.X ±XX.X shows.

S 824-035

(7) Adjust the servo control rod assembly until the applicable servo display is 0.00 ±0.5 degrees. Make sure the difference between the channels is not more than 0.4 degree.

S 214-036
(8) Make sure the threads on the adjustable rod ends are correctly installed. Rod ends will cover at least one-half of the applicable inspection hole.

S 434-037
(9) Tighten the jamnuts (13).

S 434-038
(10) Install the nut (4) and washer (3) on the bolt (2) that attaches the rod end to the quadrant autopilot crank.
(a) Tighten the nut (4).

S 094-039
(11) Remove the rig pins A1 and A2.

S 714-040
(12) Do the Test the Servo Installation steps.

H. Adjust the Servo with a Model 101A Phase Synchronous Voltmeter

S 494-041
(1) Install rig pins A1 and A2 if they are not installed.

S 484-042
(2) Connect the Phase Synchronous voltmeter to the power source shown on the voltmeter placard.

S 484-043
(3) Set the ON/OFF (S1) switch to ON.

S 484-070
(4) Stop for 10 minutes to let the voltmeter prepare for operation.

S 824-044
(5) Set the CAL/RUN (S2) switch to CAL CHK. Make sure the IN-PHASE VOLTS are 9.990 to 10.010.

S 864-045
(6) Set the CAL/RUN (S2) switch to RUN.

S 034-046
(7) Disconnect the electrical connector from the servo.

S 484-047
(8) Connect the -1 and -5 cable assemblies to the Autopilot Pitch Lateral Servo and Phase Synchronous Voltmeter.

S 824-048
(9) Adjust the Autopilot Servo control rod (5) until the Phase Synchronous voltmeter shows 0.00 ± 0.03 IN-PHASE VOLTS. Make sure the control rod (5) is connected to the Servo.

S 434-049
(10) Install the nut (4) and washer (3) on the bolt (2) that attaches the rod end to the quadrant autopilot crank.
(a) Tighten the nut (4).

S 434-050
(11) Tighten the control rod (5) jamnut.

S 084-051
(12) Disconnect the Phase Synchronous Voltmeter
(a) Set the ON/OFF switch to OFF.
(b) Disconnect the -1 and -5 cable assemblies from the Autopilot Servo.
(c) Disconnect the voltmeter from the power source.

S 434-052
(13) Connect the airplane electrical connector to the autopilot servo.

S 034-053
(14) Remove rig pins A1 and A2.

S 714-054
(15) Do the Test the Servo Installation steps

I. Test the Servo Installation

S 864-055

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(1) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 984-056

(2) Manually operate the ailerons several times.

S 714-057

(3) Start MCDP Ground Test 66-XDCR OUTPUTS (Ref 22-00-02).

S 714-052

(4) Push YES/ADV until MCDP top line message 66 AIL SURF DEG shows.

S 214-058

(5) With the control wheel released, make sure the MCDP bottom line shows 00.0 ±0.5 degrees. Make sure the difference between the channels (L-R, R-C, L-C) is not more than 0.4 degrees.

S 714-053

(6) Do MCDP Ground Test 67-AIL SURF LIM (AMM 22-00-02/201) for all three autopilot channels.
   (a) Answer YES for the "67 AIL TO POS LIM ?" message.
      1) Make sure the MCDP bottom line value for the left, center, and right channels is 19.0 degrees minimum and the A/P DISC light remains off.
   (b) Answer YES for the "67 AIL TO NEG LIM ?" message.
      1) Make sure the MCDP bottom line value for the left, center, and right channels is 19.0 degrees minimum and the A/P DISC light remains off.

S 714-062

(7) Do MCDP Ground Test 07-SERVO AIL (Ref 22-00-02).

(a) Make sure no fault messages show during the test.

S 214-064

(8) Inspect the installation for hydraulic leaks and fix them as necessary.

J. Put the Airplane Back to Its Usual Condition

S 864-065

(1) Set the MCDP to off.

S 864-066

(2) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 864-067

(3) Remove electrical power if it is not necessary (Ref 24-22-00).

ALCS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - MAINTENANCE PRACTICES

1. General
   A. Two Autopilot Lateral Control Servos (ALCSs) are in the left wing root and one is in the right wing root. Get access to the servos through the main landing gear wheel wells.

TASK 22-13-04-002-001

2. Remove the ALCS Electrohydraulic Servovalve or Solenoid Valves (Fig. 201)
   A. References
      (1) 24-22-00/201, Electrical Power-Control
      (2) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   B. Access
      (1) Location Zones
          732/742   Main Landing Gear Body Door

      (2) Access Panels
          732/742   Main Landing Gear Body Door

   C. Prepare for Removal

      S 862-002
      (1) Supply electrical power (Ref 24-22-00).

      S 862-003
      (2) Remove pressure from the L, C, and R hydraulic systems (Ref 29-11-00).

      S 862-004
      (3) Set the two F/D switches on the MCP to OFF.

      S 862-013
      (4) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11E17, FLT CONT COMPUTER POWER LEFT
          (b) 11E18, FLT CONT COMPUTER SERVO LEFT
          (c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
          (d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
          (e) 11E35, FLT CONT CMPTR PWR RIGHT
          (f) 11E36, FLT CONT CMPTR SERVO RIGHT
   D. Remove the Valve

S 112-005

CAUTION: DO NOT LET CONTAMINATION GET INTO THE AUTOPILOT LATERAL CONTROL SERVO AND VALVES. THE AUTOPILOT LATERAL CONTROL SERVO AND VALVES ARE VERY SENSITIVE ELECTROHYDRAULIC DEVICES. CONTAMINATION IN THE SYSTEM CAN CAUSE DAMAGE TO THE EQUIPMENT.

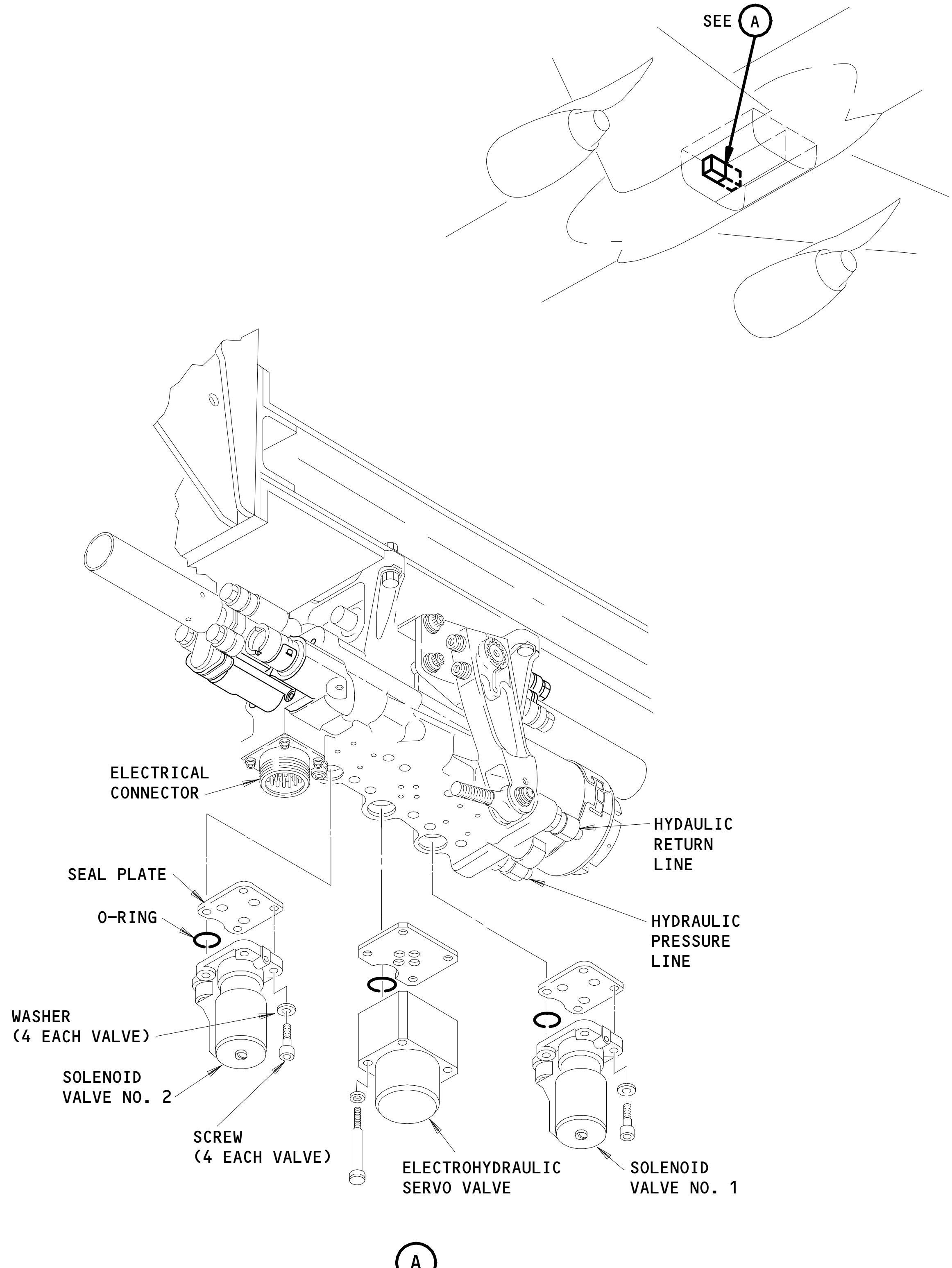
(1) Carefully clean the external areas of the valve and around the valve mounting surface.

S 032-006
(2) Remove the lockwire.

S 022-007
(3) Remove the mounting screws, the valve with the O-ring, and the seal plate from the servo assembly.

S 862-008
(4) Cover the servo openings to make sure contamination does not get into the hydraulic electrical passages.

TASK 22-13-04-402-009
3. Install the ALCS Electrohydraulic Servovalve and Solenoid Valves (Fig. 201)
A. Consumable Materials
(1) D00148 Hydraulic Fluid - Fire Resistant, BMS 3-11
B. References
(1) 22-00-02/201, Autoflight BITE
(2) 24-22-00/201, Electrical Power-Control
(3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
C. Access
(1) Location Zones
732/742 Main Landing Gear Body Door



ALCS Electrohydraulic Servo Valve and Solenoid Valves
Figure 201

(2) Access Panels
732/742 Main Landing Gear Body Door

D. Install the Valve.

S 032-010
(1) Remove the covers from the openings and clean the mating surfaces of the valve, the new seal plates, and the assembly unit with hydraulic fluid. Do not let hydraulic fluid to get into the connector.

S 432-011
(2) Install the valve and the new seal plate on the servo unit.

S 432-012
(3) Install the mounting screws with grease.
(a) Tighten the screws to 60 inch-pounds.

S 432-013
(4) Put a lockwire on the mounting screws.

E. Test the ALCS Electrohydraulic Servovalve and Solenoid Valves

S 862-014
(1) Supply electrical power (Ref 24-22-00).

S 862-015
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E17, FLT CONT COMPUTER POWER LEFT
(b) 11E18, FLT CONT COMPUTER SERVO LEFT
(c) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
(d) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
(e) 11E35, FLT CONT CMPTR PWR RIGHT
(f) 11E36, FLT CONT CMPTR SERVO RIGHT

S 862-016

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 982-017

(4) Manually operate the aileron three or more times.

S 862-018

(5) Make sure these circuit breakers on the P11 panel are also closed:

(a) 11C30, LANDING GEAR POS SYS 1
(b) 11E16, MODE CONT PNL LEFT
(c) 11E34, MODE CONT PNL RIGHT
(d) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(e) GUI 115;
11S6, MAINT CONT DSPL
(f) 11S15, AIR/GND SYS 1
(g) 11S19, AIR/GND SYS 2
(h) 11S23, POS SYS 2

S 712-019

(6) Do MCDP Ground Test 07 - SERVO AIL (Ref 22-00-02).

S 862-012

(7) Remove the pressure from the L, C, and R hydraulic systems.

S 212-011

(8) Examine the valves for hydraulic leaks and fix them as necessary.

F. Put the Airplane Back to its Usual Condition

S 862-021

(1) Set the MCDP to off.

S 862-022

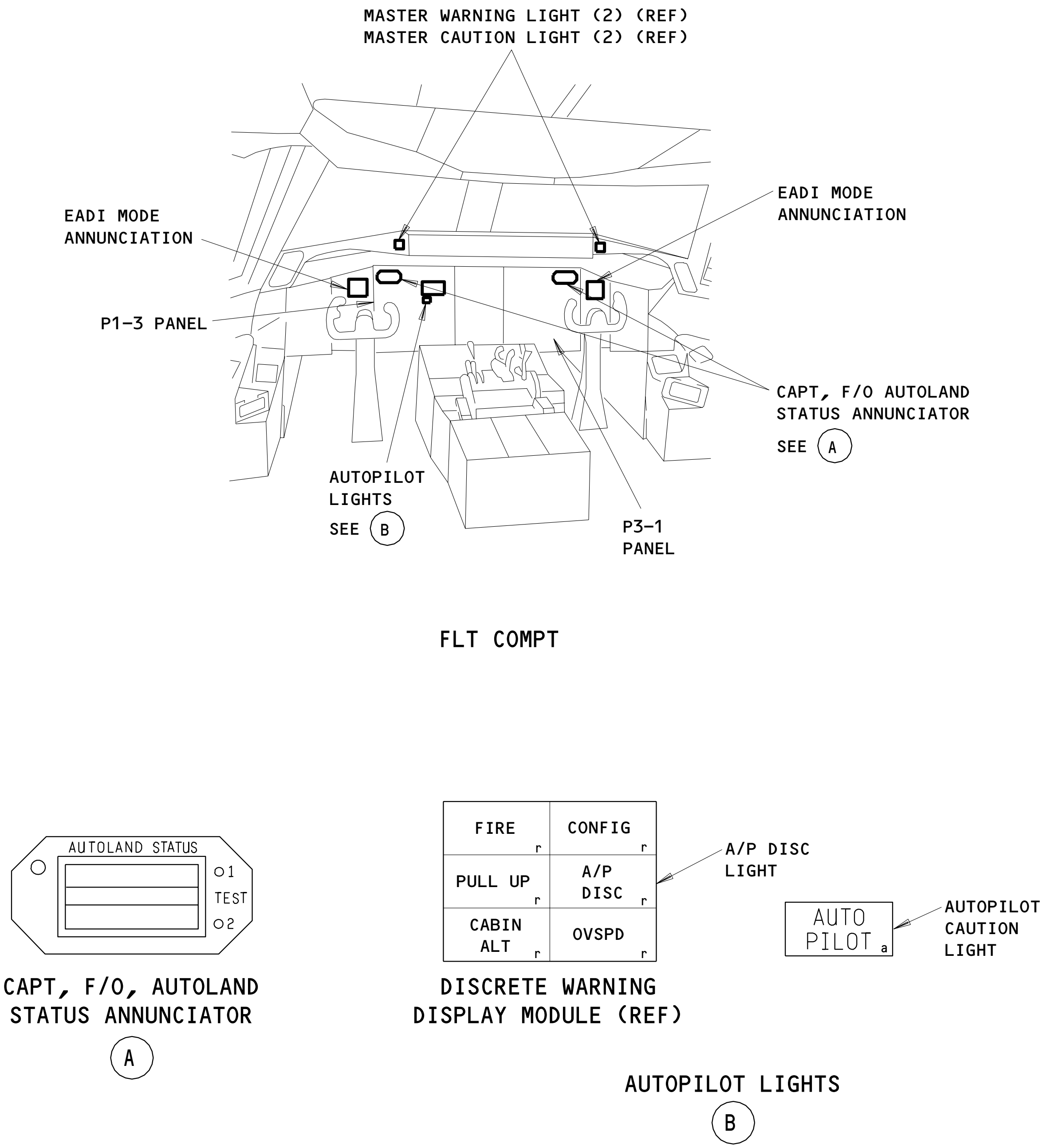
(2) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 862-023

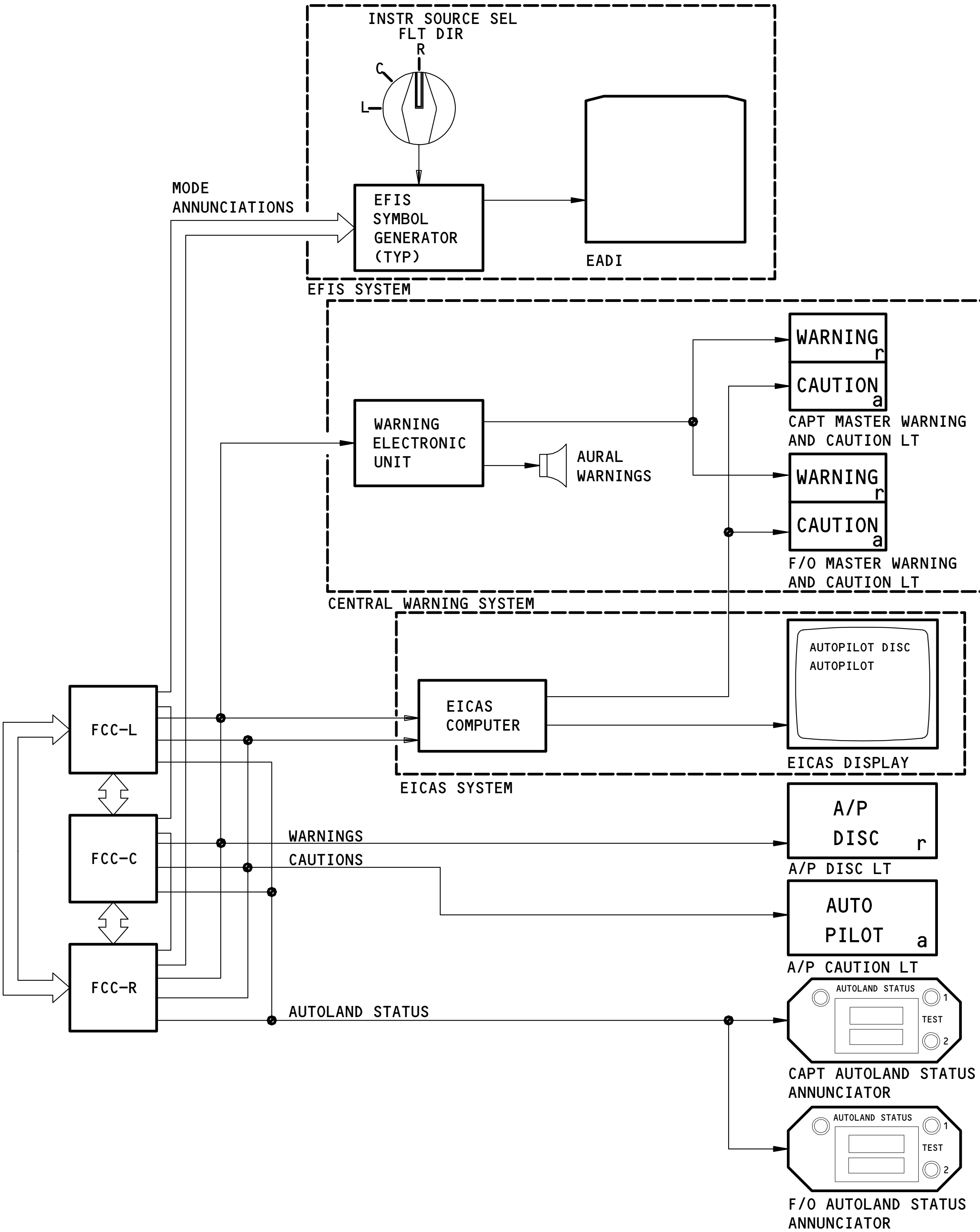
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR WARNING AND ANNUNCIATION - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The warning and annunciation functions of the Flight Control Computers (FCCs) provide Autopilot/Flight Director System (AFDS) status and alerts the crew to system failures. The following displays are provided.
      (1) Autopilot (A/P) caution and warning lights located on the P1 panel.
      (2) Two master caution and warning lights located on the pilot's glareshield (P7).
      (3) Two Electronic Attitude Direction Indicators (EADI), one on the P1 panel and one on the P3 panel. The EADIs display the following annunciations:
         (a) Flight mode annunciation
         (b) Flight mode failure annunciation
         (c) Autopilot engage status annunciation (Ref 22-11-00)
         (d) Flight Director (FD) engage status annunciation (Ref 22-11-00)
      (4) Two Autoland Status Annunciators (ASA), one on the P1 and one on the P3 panel.
   B. AFDS Warning and Annunciation (Fig. 2)
      (1) Annunciation
         (a) The operational (armed and engaged) flight modes are annunciated on the Electronic Attitude Director Indicators (EADIs). The displays include autopilot and flight director engage status. The armed and engaged modes for pitch and roll are also displayed. Status and mode failures add a yellow line through the annunciation.
      (2) Autoland Status
         (a) GUI 001-114, 116-999;
             Autoland status is provided by two Autoland Status Annunciators (ASAs) connected in parallel. They are controlled by the three Flight Control Computers (FCCs). The ASAs display the loss of autoland capability in all phases of flight, and display the autoland status during the approach mode.
         (b) GUI 115;
             Autoland status is provided by two Autoland Status Annunciators (ASAs) connected in parallel. They are controlled by the three Flight Control Computers (FCCs). The ASAs only display the autoland status during the approach mode.



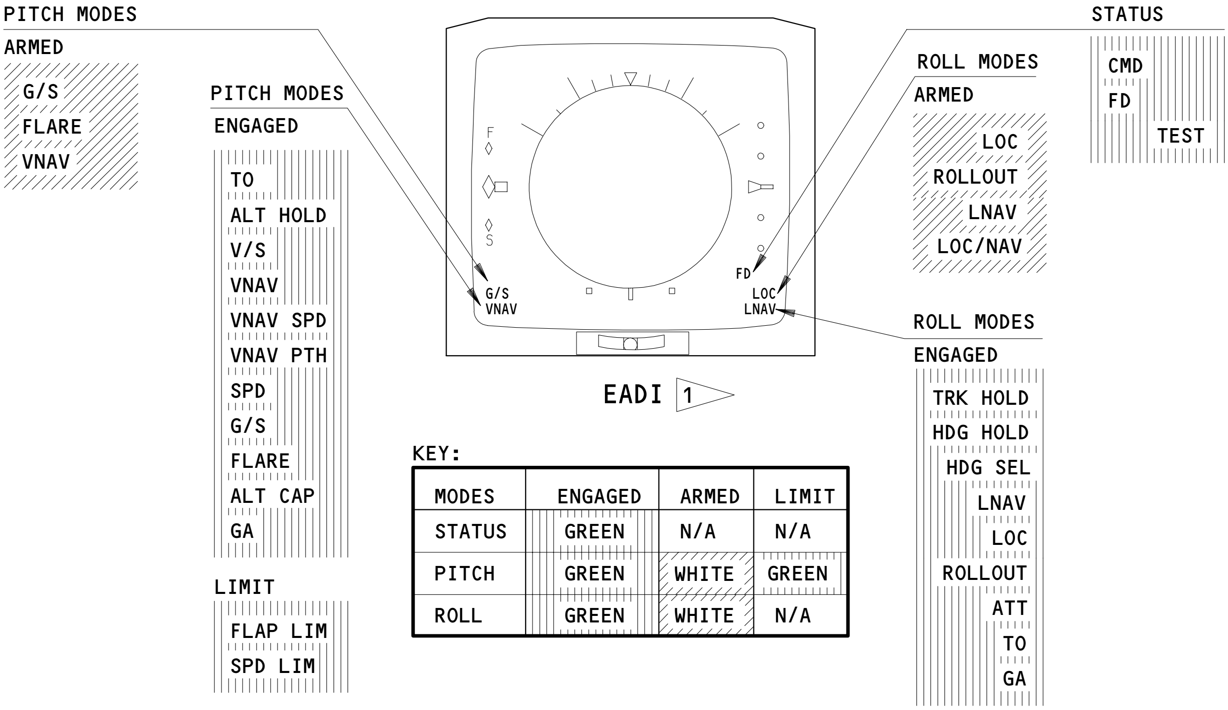
AFDS Warning and Caution Annunciation - Component Location
Figure 1



AFDS Warning and Annunciation
Figure 2

(3) Autopilot Caution
(a) One autopilot caution light is connected in parallel with the three FCCs. The light indicates a problem requiring crew attention. A separate output turns on the master CAUTION lights (2), an owl aural tone, and the Engine Indicating and Crew Alerting System (EICAS) computer displays an AUTOPILOT message (Ref 31-51-00, Warning System).
(4) Autopilot Disengage Warning
(a) GUI 001-008, 010-114, 116-999;
autopilot disengage warning is provided by a dual lamp indicator connected in parallel with the three FCCs. The light indicates disengagement of an FCC when no other FCCs are engaged. One lamp is controlled by normal power, the other by standby power. Separate outputs from the FCCs turn on the master WARNING lights (2), a siren aural warning, and the EICAS computer displays an AUTOPILOT DISC message (Ref 31-41-00, 31-51-00).
(b) GUI 009, 115;
autopilot disengage warning is provided by a dual lamp indicator connected in parallel with the three FCCs. The light indicates disengagement of an FCC when no other FCCs are engaged. One lamp is controlled by normal power, the other by standby power. Separate outputs from the FCCs turn on the master WARNING lights (2), a wailer aural warning, and the EICAS computer displays an AUTOPILOT DISC message (Ref 31-41-00, 31-51-00).
(5) Cross-Channel Data
(a) Digital buses and analog discrete connections interface with the three FCCs. The computers share sensor input data and engage status information.

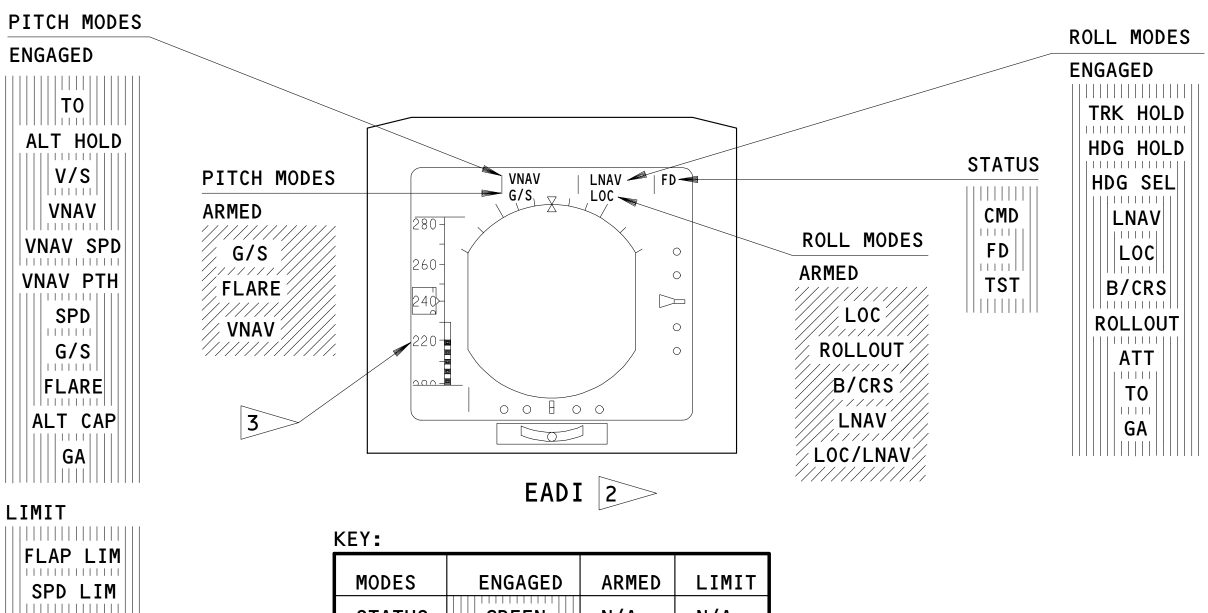
2. Component Details
A. Autopilot/Flight Director Flight Mode Annunciation (Fig. 3)
(1) Status
(a) GUI 115;
The autopilot engage status (CMD) is the top line of the display in the lower right corner of the EADI. The flight director engaged status (FD) is also annunciated.
(b) GUI 001-114, 116-999;
The autopilot engage status (CMD) is the top line of the display in the upper right corner of the EADI. The flight director engaged status (FD) is also annunciated.
(2) Roll Modes
(a) GUI 115;
The armed and engaged roll modes are indicated below the status display.



EADI [1]

KEY:

MODES	ENGAGED	ARMED	LIMIT
STATUS	GREEN	N/A	N/A
PITCH	GREEN	WHITE	GREEN
ROLL	GREEN	WHITE	N/A



EADI [2]

KEY:

MODES	ENGAGED	ARMED	LIMIT
STATUS	GREEN	N/A	N/A
PITCH	GREEN	WHITE	GREEN
ROLL	GREEN	WHITE	N/A

[1] GUI 115
[2] GUI 001-114,116-999
[3] GUI 009;
THE FAST/SLOW DISPLAY SHOWS

Autopilot/Flight Director System Mode Annunciation
Figure 3

(b) GUI 001-114, 116-999;
the armed and engaged roll modes are indicated to the left of the status display.

(3) Pitch Modes

(a) GUI 115;
the armed and engaged pitch modes are displayed in the lower left corner of the EADI. Overspeed or minimum speed limit displays override normal annunciation.

(b) GUI 001-114, 116-999;
the armed and engaged pitch modes are displayed in the upper left center of the EADI. Overspeed or minimum speed limit displays override normal annunciation.

(4) Fail Display

(a) Status or mode failures place a yellow line through the status or mode communication.

B. Autoland Status Annunciator (Fig. 4)

(1) Displays

(a) The Autoland Status Annunciator (ASA) consists of two mechanical displays driven by magnetic coils. The coils use 28 vdc to operate the display. Each display has three faces. One face is blank; the other two faces are designated as A and B. Figure 4 shows the displays and associated color.

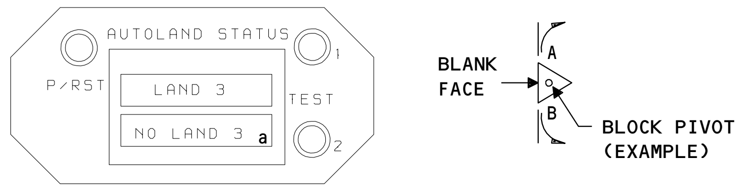
(2) Switches

(a) GUI 001-114, 116-999;
Three momentary pushbutton switches are on the ASA panel. Pressing the reset (P/RST) switch clears the bottom display, which indicates the failure mode. Two Press-To-Test (PTT) switches are on the right side of the annunciator. Test 1 shows both A faces. Test 2 shows both B faces.

(b) GUI 115;
Three momentary pushbutton switches are on the ASA panel. Pressing the reset (STATUS) switch clears the bottom display, which indicates the failure mode. Two Press-To-Test (PTT) switches are on the right side of the annunciator. Test 1 shows the top A face. Test 2 shows both B faces. The bottom A face is not used.

PTT 1 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE A

PTT 2 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE B



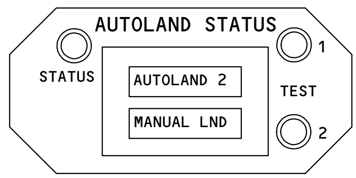
SYSTEM CONFIGURATION	DISPLAY
FOLLOWING FIRST FAIL (FAIL PASSIVE, BEFORE MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO LAND 3 (AMBER) A
AFTER 2ND FAILURE, BEFORE MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO AUTOLND (AMBER) B
FAIL OPERATIONAL, AFTER MULTICHANNEL ENGAGEMENT	LAND 3 (GREEN) A BLANK (BLACK)
FOLLOWING FIRST FAIL (FAIL PASSIVE), AFTER MULTICHANNEL ENGAGEMENT	LAND 2 (GREEN B /WHITE) NO LAND 3 (AMBER) A
AFTER PILOT PRESSES P/RST SW DISPLAY CHANGES	LAND 2 (GREEN/ B WHITE) BLANK (BLACK)
SINGLE CHANNEL AUTOPILOT OPERATION (2ND FAILURE) AFTER MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO AUTOLND (AMBER) B

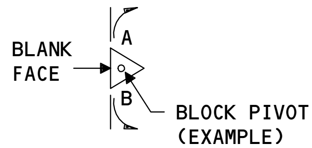
SYSTEM STATUS	DISPLAY
ON GROUND	READS OUT DEGRADATION OF AUTOLAND STATUS (NON-LATCHING)
IN ROUTE	READS OUT DEGRADATION OF AUTOLAND STATUS (LATCHING UNTIL MANUAL RESET OR APPROACH)
AT APPROACH SELECT	AUTOMATIC RESET PULSE READS OUT CURRENT AUTOLAND SYSTEM STATUS (LATCHING UNTIL MANUAL RESET)
APPROACH	READS OUT CURRENT AUTOLAND SYSTEM STATUS ABOVE ALERT HEIGHT (LATCHING UNTIL MANUAL RESET)

Autoland Status Annunciator
Figure 4 (Sheet 1)

PTT 1 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE A

PTT 2 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE B





SYSTEM STATUS	DISPLAY
AT APPROACH SELECT	AUTOMATIC RESET PULSE READS OUT CURRENT AUTOLAND SYSTEM STATUS (LATCHING UNTIL MANUAL RESET)
APPROACH	READS OUT CURRENT AUTOLAND SYSTEM STATUS ABOVE ALERT HEIGHT (LATCHING UNTIL MANUAL RESET)

SYSTEM CONFIGURATION	DISPLAY
FAIL OPERATION	AUTOLAND 3 (GREEN) A BLANK (BLACK)
FOLLOWING 1ST FAIL (FAIL PASSIVE)	AUTOLAND 2 (WHITE) B BLANK (BLACK)
SINGLE CHANNEL OPERATION AFTER 2ND FAIL	BLANK (BLACK) MANUAL LND (AMBER) B

Autoland Status Annunciator
Figure 4 (Sheet 2)

C. Autopilot Warning and Caution Lights and Annunciations (Fig. 5)

(1) Warning Light

(a) The A/P DISC light is a dual-lamp indicator controlled by the FCCs. One lamp uses normal 28 vdc power. The other lamp uses 28 vdc standby power. The color of the indicator is red. The light indicates disengagement of an FCC when no other FCCs are engaged. In ground test mode, only the right lamp (powered by standby power) illuminates when disengagement of an FCC occurs. The A/P DISC light is part of the Discrete Warning Display Module (Ref 33-16-00).

(2) Caution Light

(a) The AUTOPILOT caution light is a dual-lamp indictor controlled by a single output from the FCCs. The lamps use 28 vdc power. The color of the indicator is amber.

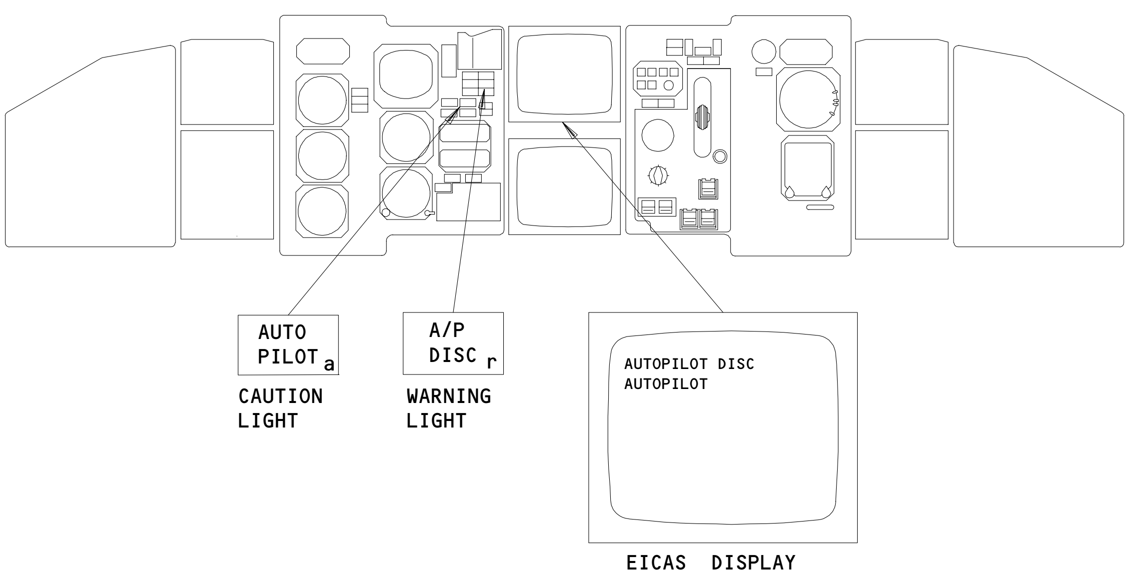
(3) Warning and Caution Annunciations

(a) The EICAS computer receives a ground from the FCCs when an autopilot disengage or autopilot failure occurs. This ground causes the EICAS computer to display a red AUTOPILOT DISC or amber AUTOPILOT message (Ref 31-41-00).

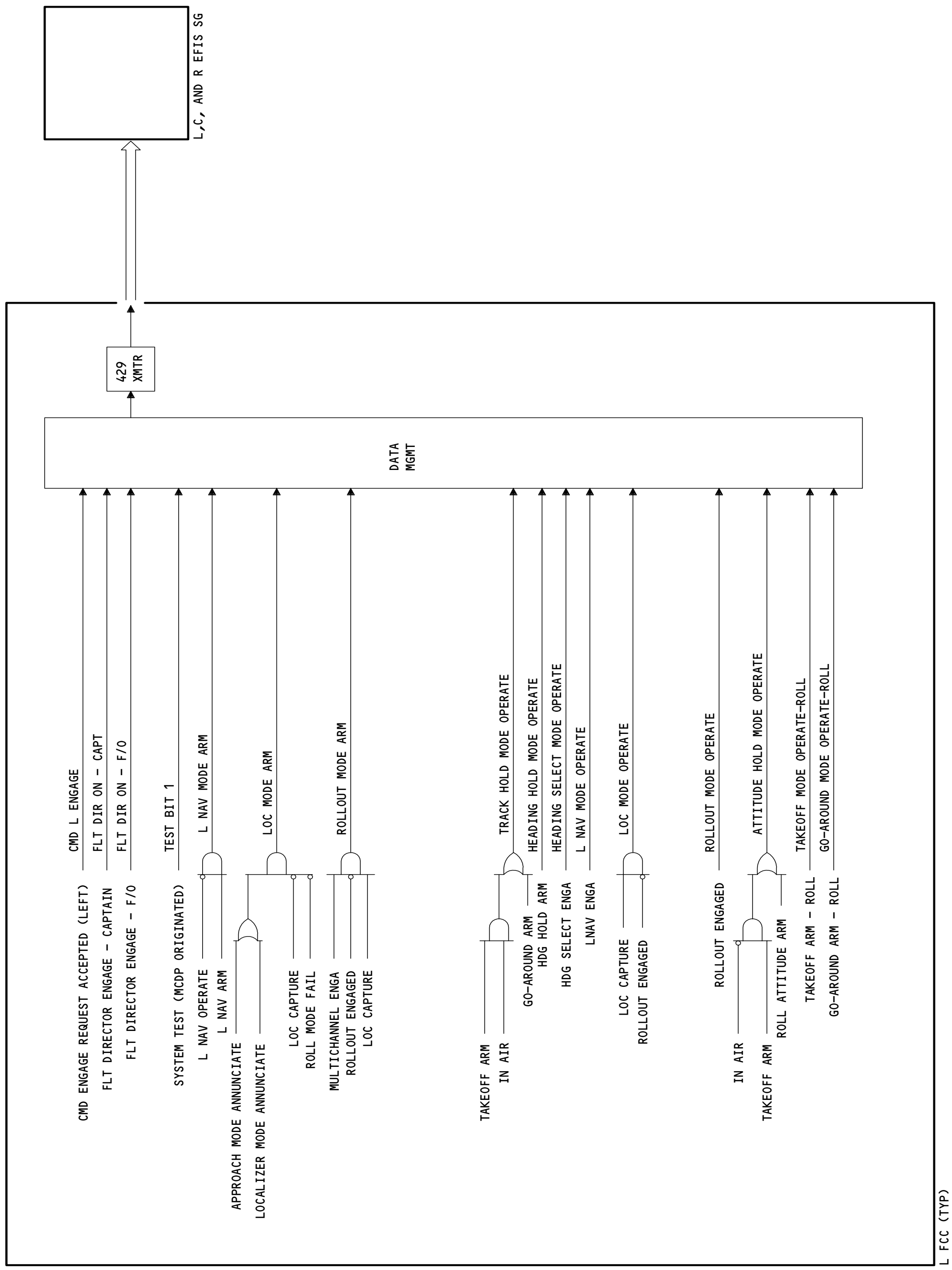
3. Operation

A. Functional Description

(1) EADI AFDS Status and Roll Mode Annunciation Signals (Fig. 6)

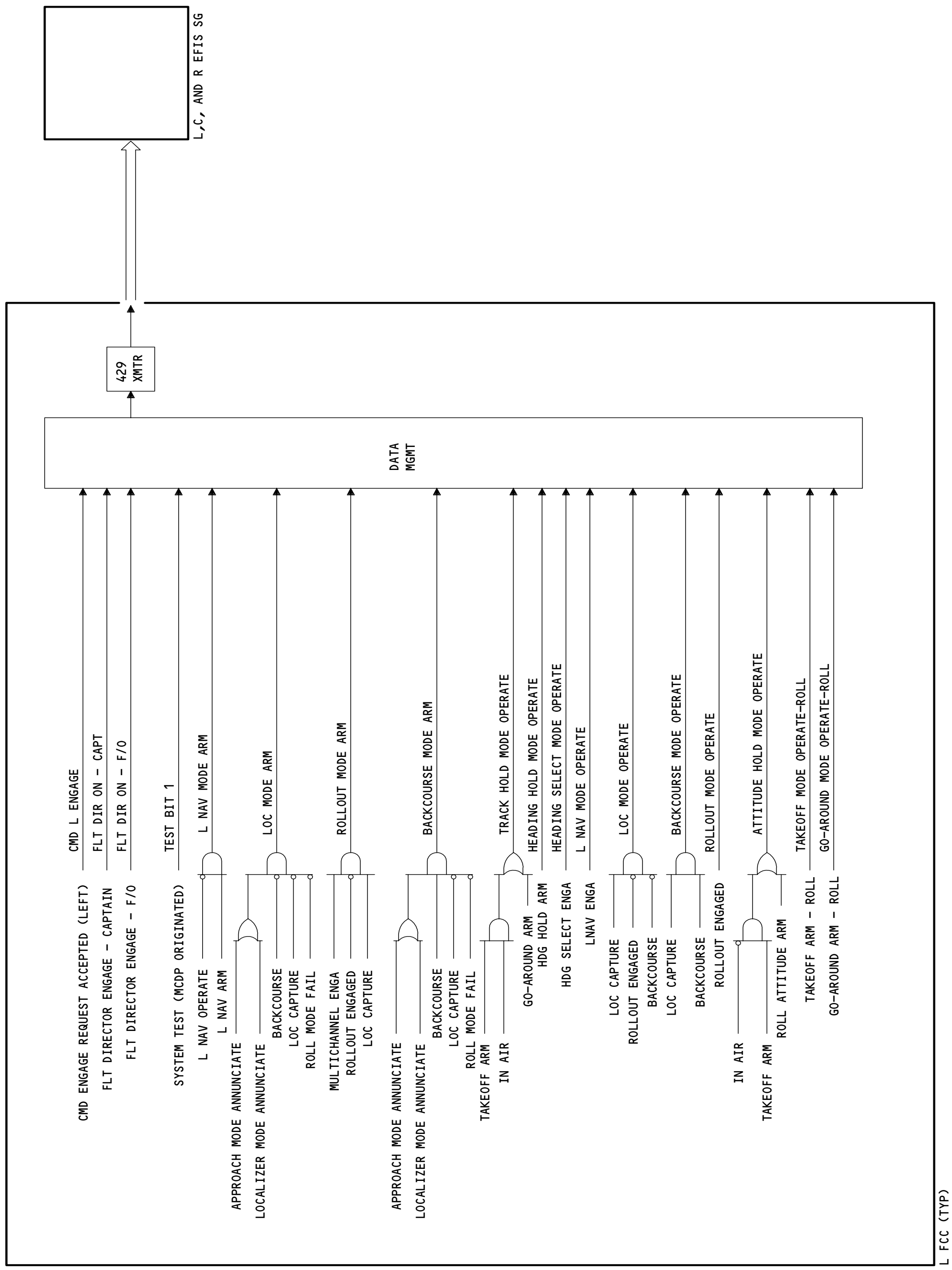


Autopilot Warning and Caution Annunciations
Figure 5



EADI AFDS Status and Roll Mode Annunciation Signals
Figure 6 (Sheet 1)

305577



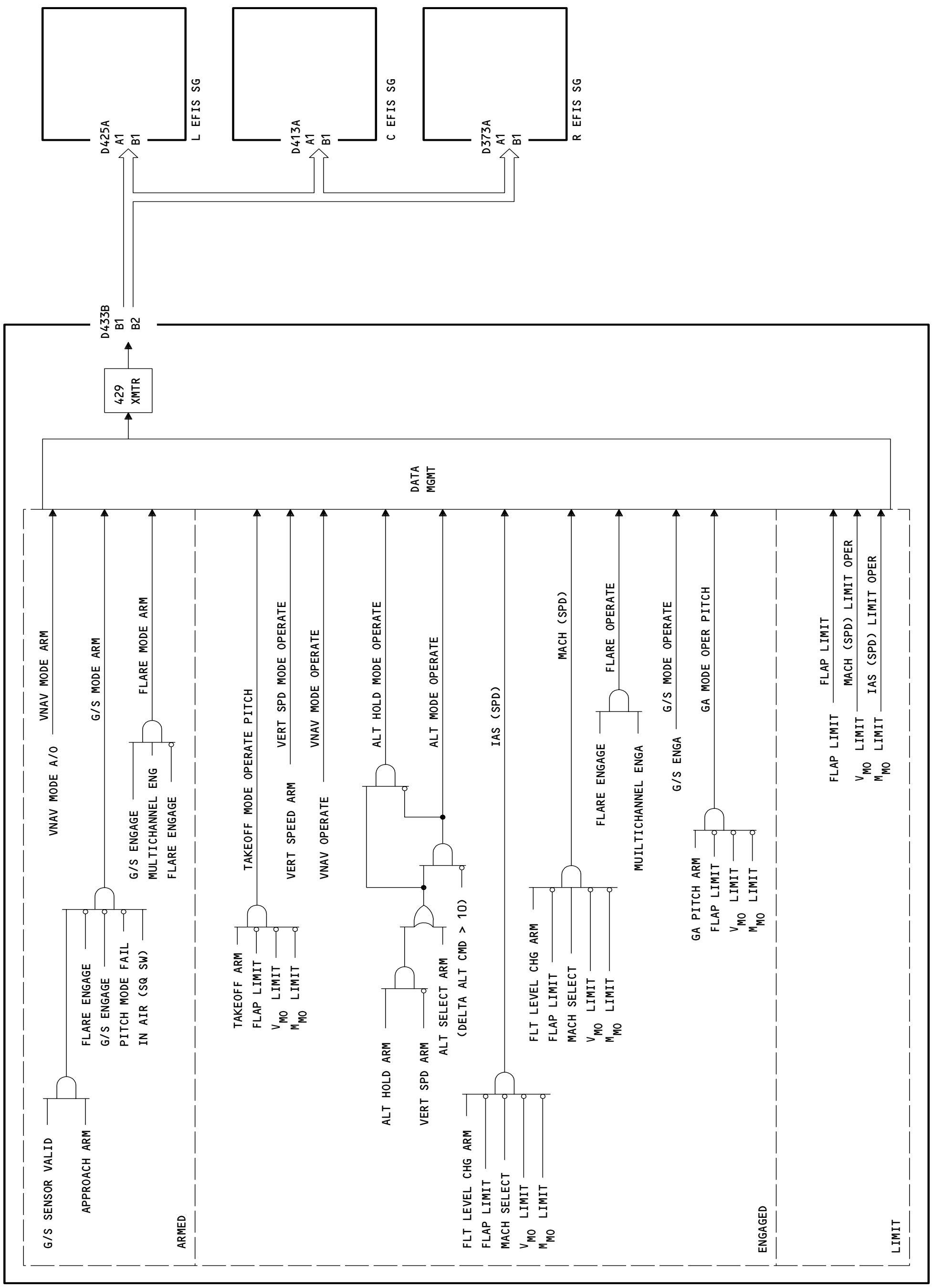
EADI AFDS Status and Roll Mode Annunciation Signals
Figure 6 (Sheet 2)

A71079

(a) Status Annunciation
1) Autopilot status messages are sent from the FCC that has an autopilot or flight director channel engaged. Four status signals are supplied by the FCC. These are as follows:
a) CMD ENGAGE
b) FLT DIR ON - CAPT
c) FLT DIR ON - F/O
d) TEST BIT 1 - when the Maintenance Control and Display Panel (MCDP) is conducting a ground test (Ref 22-41-00)

(b) Roll Mode Annunciation
1) The FCC roll mode logic determines the following armed and engaged modes:
a) LNAV Mode ARM - LNAV armed but not engaged
b) LOC MODE ARM - Approach or localizer mode annunciate and no backcourse, no localizer capture, and no roll mode fail
c) ROLLOUT MODE ARM - Multichannel engaged, localizer captured, and rollout not engaged
d) GUI 001-114, 116-999;
BACKCOURSE MODE ARM - Approach or localizer mode annunciate, backcourse selected, and no localizer capture or roll mode fail
e) LOC/NAV MODE ARM - LOC and LNAV modes armed
f) TRACK HOLD MODE OPERATE - Takeoff arm and in air, or go around mode arm
g) HEADING HOLD MODE OPERATE - Heading hold selected
h) HEADING SELECT MODE OPERATE - Heading select engaged
i) LNAV MODE OPERATE - LNAV engaged
j) LOC MODE OPERATE - Localizer capture, no backcourse selected, and rollout disengaged
k) GUI 001-114, 116-999;
BACKCOURSE MODE OPERATE - Localizer capture and backcourse selected
l) ROLLOUT MODE OPERATE - Rollout engaged
m) ATTITUDE HOLD MODE OPERATE - Takeoff arm and on ground, or roll attitude arm
n) TAKEOFF MODE OPERATE (ROLL) - Takeoff arm (roll)
o) GO AROUND MODE OPERATE (ROLL) - Go around arm (roll)

(2) EADI Pitch Mode Annunciation Signals - Armed, Engaged, and Limit (Fig. 7)
(a) Armed and Engaged Modes
1) The FCC pitch mode logic determines the following armed and engaged modes:
a) GS MODE ARM - The (GS) glide slope sensor must be valid and approach mode armed. Also, the airplane must be in air with the flare mode disengaged, GS disengaged, and no pitch mode fail
b) FLARE MODE ARM - GS engaged, multichannel engaged, and the flare mode disengaged
c) VNAV MODE ARM - VNAV armed but not engaged



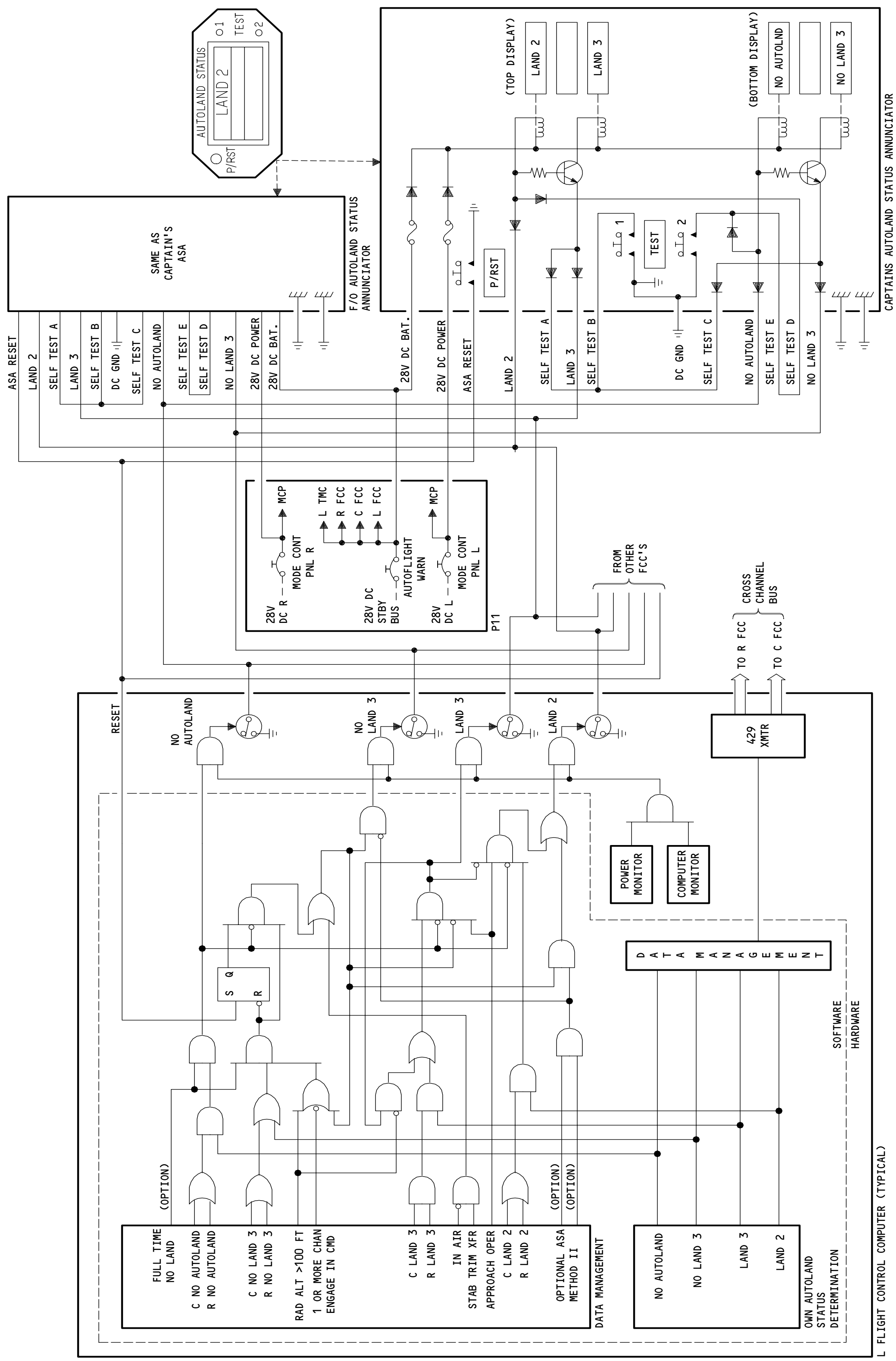
EADI Pitch Mode Annunciation Signals - Armed, Engaged and Limit
Figure 7

334789

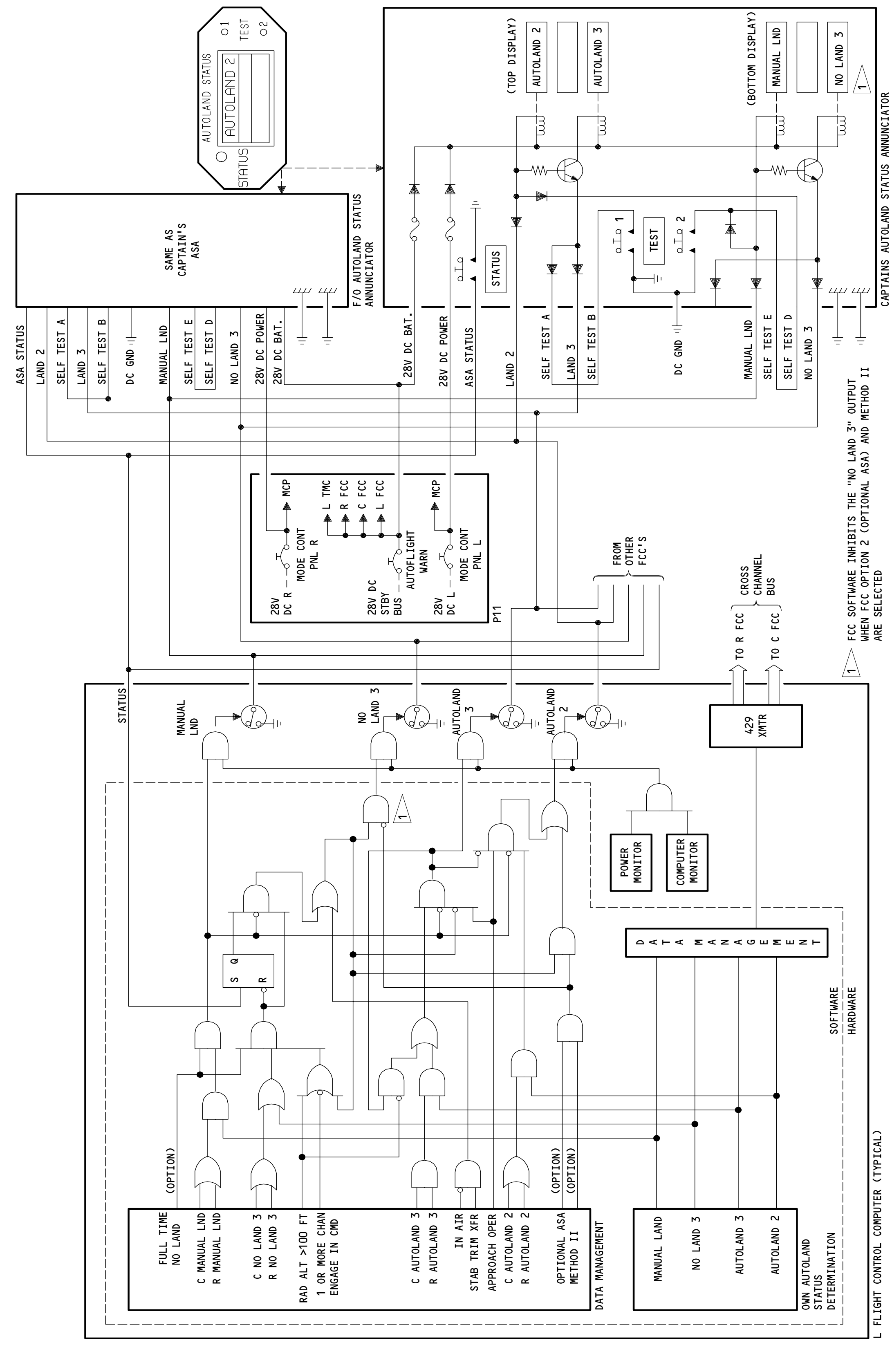
d) TAKEOFF MODE OPERATE (PITCH) - Takeoff armed (limit modes override this display)
e) VERT SPD MODE OPERATE - Vertical speed armed
f) VNAV MODE OPERATE - VNAV engaged
g) ALT HOLD MODE OPERATE - Valid only when ALT MODE OPERATE is not valid. Also, altitude hold must be armed with no vertical speed arm, or altitude select arm must be valid
h) SPD - Flight level change armed and Mach or IAS selected on MCP (limit modes override this display)
i) GS MODE OPERATE - Glide slope engaged
j) FLARE OPERATE - Multichannel engaged and flare engaged
k) ALT MODE OPERATE - Valid only when the delta altitude command is greater than 10. Also, altitude hold must be armed with no vertical speed arm, or altitude select arm must be valid
l) GA MODE OPERATE (PITCH) - Go-around pitch arm (limit modes override this display)

(b) Limit Modes
1) When the pitch channel of the AFDS is controlling speed through use of the elevator, overspeed and minimum speed limits override the engaged mode. The normal annunciation is inhibited and the operating limit mode is displayed. The following limit modes are displayed:
a) FLAP LIMIT
b) SPD LIMIT OPER - Maximum operating airspeed or Mach limit

(3) AFCS Autoland Status Annunciator Signal Development and Control (Fig. 8)
(a) General
1) GUI 001-114, 116-999;
Each FCC independently determines its own autoland status. Through cross-channel data transfer, the FCCs determine the proper display. Only one display output is possible unless a NO LAND 3 condition occurs. Both the NO LAND 3 and the NO LAND 2 displays will be shown during multi-channel approach. In decreasing order of autoland capability, the displays are LAND 3, NO LAND 3, LAND 2, and NO AUTOLAND.
2) GUI 115;
Each FCC independently determines its own autoland status. Through cross-channel data transfer, the FCC's determine the proper display. Only one display output is possible. In decreasing order of autoland capability, the displays are AUTOLAND 3, AUTOLAND 2, and MANUAL LND.



AFCS Autoland Status Annunciator Signal Development and Control Schematic
Figure 8 (Sheet 1)



AFCS Autoland Status Annunciator Signal Development and Control Schematic
Figure 8 (Sheet 2)

(b) GUI 001-114, 116-999;
LAND 3
1) The LAND 3 display is enabled only when the approach mode is operational. The display requires that all three FCCs have determined LAND 3 is valid. LAND 3 is valid when the FCCS, A/P servos, and outer loop data are normal. The display latches at 200 ft. of radio altitude.

(c) GUI 115;
AUTOLAND 3
1) The AUTOLAND 3 display is enabled only when the approach mode is operational. The display requires that all three FCCs have determined AUTOLAND 3 is valid. AUTOLAND 3 is valid when the FCCs, A/P servos and outer loop data are normal. The display latches at 200 ft. of radio altitude.

(d) GUI 001-114, 116-999;
NO LAND 3
1) The NO LAND 3 display is enabled during the entire flight. The NO LAND 3 is displayed when one or more of the FCCs determine a NO LAND 3 status exists. This display is inhibited below 200 ft. unless already displayed, or no FCC is in the CMD mode. Transfer of the automatic stabilizer trim function from the first channel in command to another channel causes NO LAND 3 to be displayed after touchdown. When the P/RST switch is pressed, the reset flip-flop is set. This clears the display if the fail status is corrected.

(e) GUI 001-114, 116-999;
LAND 2
1) The LAND 2 display is enabled only when the approach mode is operational. The display requires an FCCs' own determination of LAND 2, plus one other FCC with a LAND 2 status. LAND 2 is valid when the remaining FCCs, A/P servos, and outer loop data are normal.

(f) GUI 115;
AUTOLAND 2
1) The AUTOLAND 2 display is enabled only when the approach mode is operational. The display requires an FCCs' own determination of AUTOLAND 2, plus one other FCC with a AUTOLAND 2 status. AUTOLAND 2 is valid when the remaining FCCs, A/P servos, and outer loop data are normal.

(g) GUI 001-114, 116-999;
NO AUTOLAND
1) The NO AUTOLAND display is enabled during the entire flight. To receive a NO AUTOLAND display, at least two FCCs must determine that a NO AUTOLAND status exists. When the P/RST switched is pressed, the reset flip-flop sets. This clears the display if the fail status is corrected.

(h) GUI 115;
MANUAL LND

1) The MANUAL LND display is enabled only during the approach mode. To receive a MANUAL LND display, at least two FCCs must determine that a MANUAL LND status exists. When the STATUS switched is pressed, the reset flip-flop sets. This clears the display if the fail status is corrected.

(4) Autoland Status Annunciator Drive

(a) FCC Outputs

1) Software digital output signals in the FCC are converted to analog and supplied to hardware drivers. The drivers are inhibited during a power loss or computer failure, to prevent a false annunciation. The outputs of the three FCCs are connected in parallel to the ASAs.

(b) Display Drive

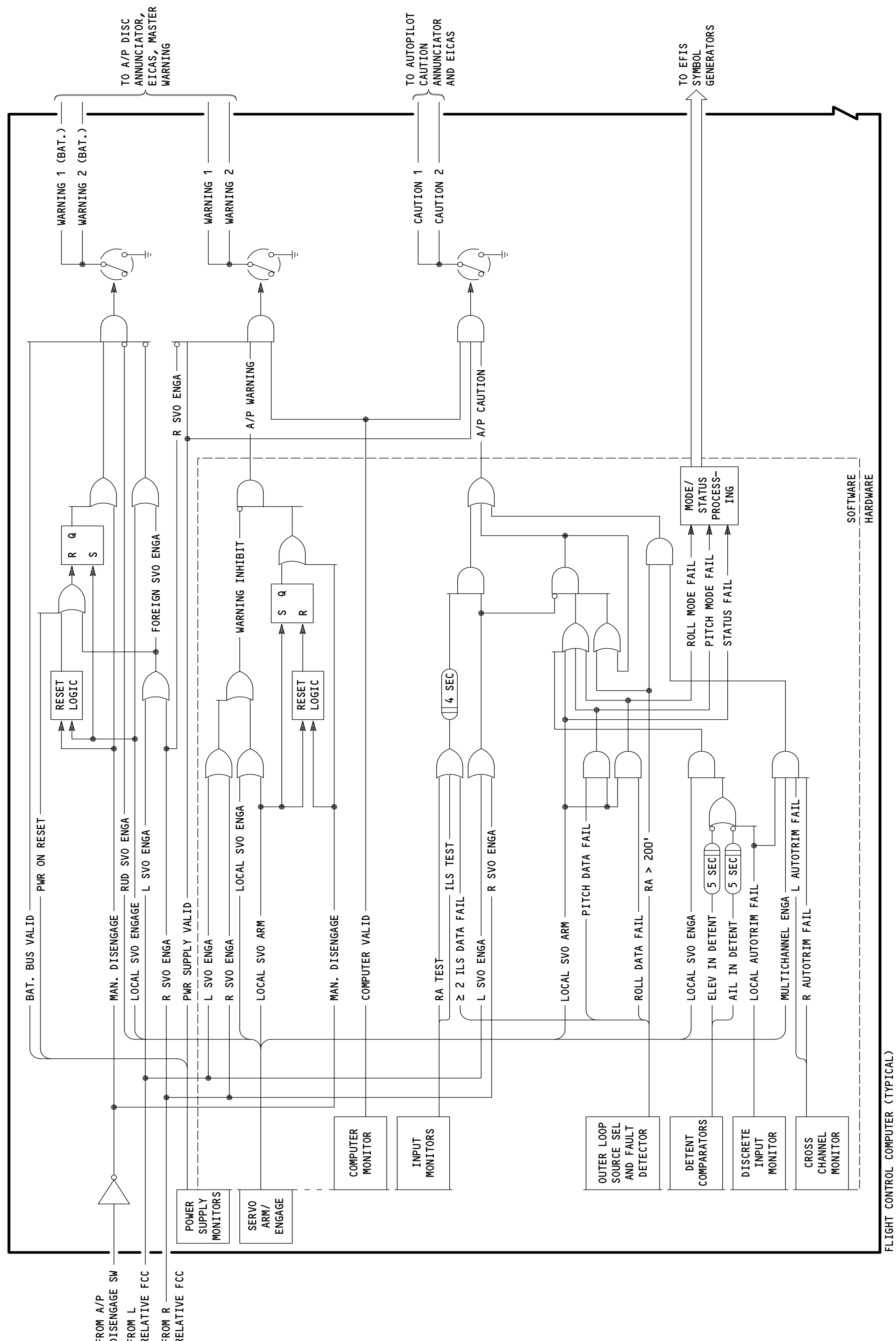
1) The top and bottom displays are driven when a ground is provided by the FCC. The ground completes the 28 vdc circuit to energize the magnetic coils and operate the mechanical display. Standby 28 vdc power connects in parallel with normal 28 vdc power. This provides a back-up source of power. A ground to the B side of the display overrides a ground to the A side.

(c) Switches

1) GUI 001-114, 116-999;
Diodes isolate the primary control from the test switches and allow testing the displays. Operation of the test switches checks power and display operation. Pressing TEST 1 displays both A faces. Pressing TEST 2 displays both B faces. The P/RST switches are paralleled and allow resetting the software logic.

2) GUI 115;
Diodes isolate the primary control from the test switches and allow testing the displays. Operation of the test switches checks power and display operation. Pressing TEST 1 displays the top A face. The bottom A face is not tested since it is not used. Pressing TEST 2 displays both B faces. The STATUS switches are paralleled and allow resetting the software logic.

(5) Autopilot/Flight Director System Caution and Warning Annunciation (Fig. 9)



Autopilot/Flight Director System Caution and Warning Schematic
Figure 9

263307

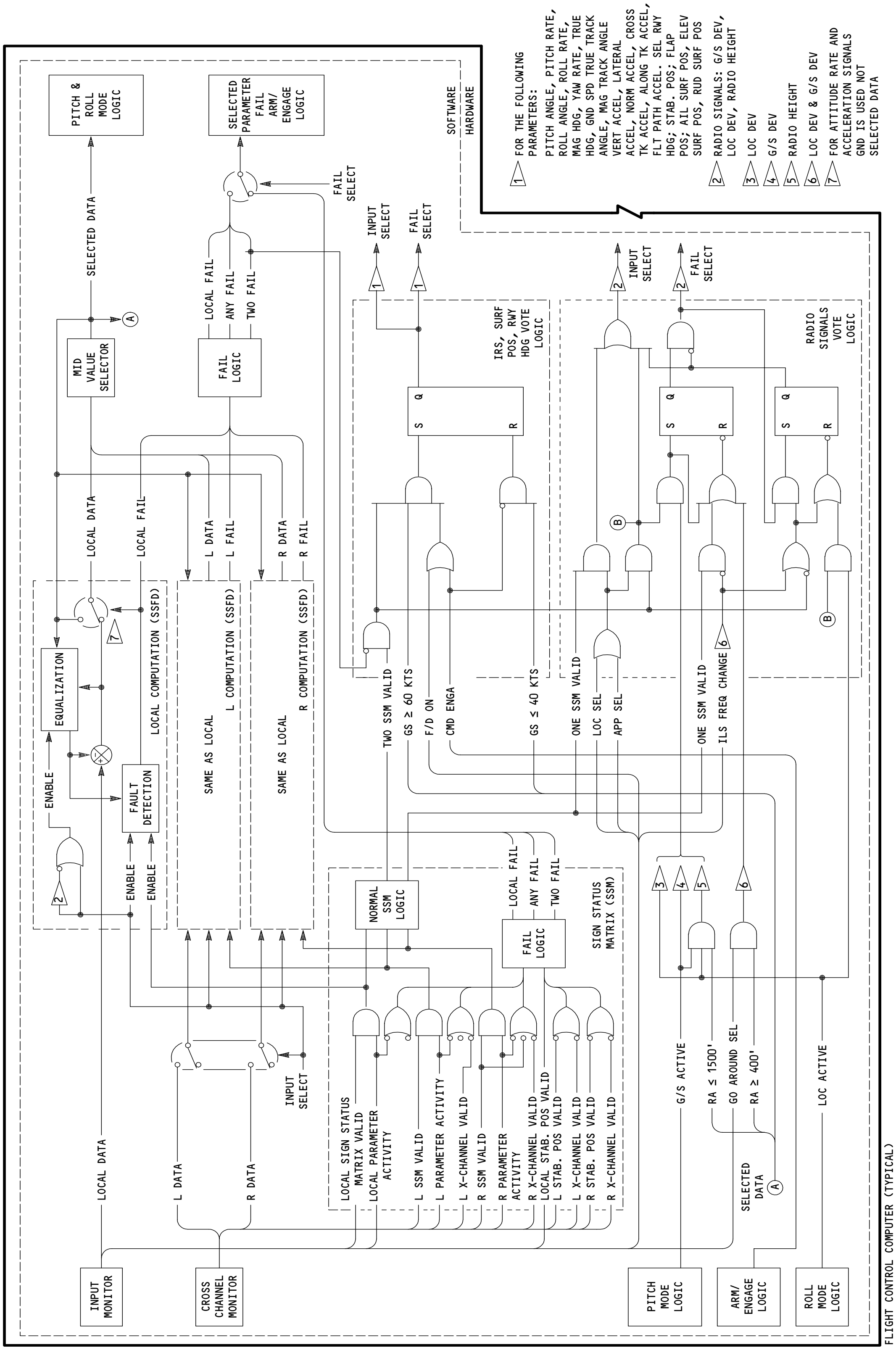
(a) Autopilot Caution
1) Any autopilot failure requiring crew attention illuminates the AUTOPILOT caution light. A master CAUTION light is also illuminated, the EICAS computer displays an AUTOPILOT message, and an owl aural warning sounds (Ref 31-51-00). A power or computer failure inhibits the caution output to prevent a false annunciation. The following conditions cause a caution output signal prior to multi-channel engage only:
a) Failure of Air Data Computer, Radio Altimeter, and Instrument Landing System (outer loop) data when the pitch or roll servos are armed.
b) An elevator or aileron servo out-of-detent for more than five seconds when the servos are engaged.
c) An automatic stabilizer trim failure when the servos are engaged.
d) An CMD engage inhibit signal when the servos are armed.
e) Autopilot in flare mode with only one channel engaged in CMD mode.
2) During multi-channel engage, the caution output signal is inhibited except for the following conditions:
a) Simultaneous test on two or more ILS receivers (ILS test requests are normally inhibited during multi-channel engage)
b) Simultaneous ILS data failure for more than four seconds
c) Automatic stabilizer trim failure in all three FCCs

(b) EADI Failure Annunciation
1) The Electronic Flight Instrument System (EFIS) EADI autopilot failure is indicated with a yellow line through the engaged mode display. A line through the CMD annunciation on the EADI occurs when the following conditions exist:
a) GUI 001-114, 116-999;
A NO AUTOLAND or NO LAND 3 determination
b) GUI 115;
A MANUAL LND determination
c) Loss of local ILS data during a single channel approach
2) A line through the active pitch or roll mode displayed on the EADI occurs when the following conditions exist:
a) A failure of applicable outer-loop data when the servos are armed
b) An elevator or aileron servo out-of-detent for more than five seconds when the servos are engaged
c) An automatic stabilizer trim failure when the servos are engaged (pitch only)
d) An engage inhibit signal when the servos are armed
3) Invalid local input signals required for any F/D mode causes the flight director bars to be removed from view on the EADI.

(6) Autopilot Warning Annunciation
(a) General
1) Disengagement of the only engaged channel causes the following:
a) The A/P DISC warning light illuminates
NOTE: In ground test mode, only the right half of the light illuminates.
b) The master WARNING light illuminates
c) The EICAS computer displays an AUTOPILOT DISC message.
d) GUI 001-008, 010-114, 116-999;
a siren sounds
e) GUI 009, 115;
a wailer aural warning sounds
2) Two WARN outputs are generated for each of these warning functions. One WARN output uses software logic and is inhibited by a loss of normal power or a computer failure. The second WARN output uses hardware logic that is powered by 28 vac standby power. This ensures a valid warning if normal power is lost.
(b) Signal Generation
1) Latches are set when the FCC is engaged. If the channel subsequently disengages, the warning is issued providing no other channels are engaged. All outputs are generated anytime either autopilot disengage switch is operated.
(c) Reset
1) When the autopilot is disengaged with the A/P disengage switch, the warnings are reset on the second press of the switch. Reset of an autopilot disengage warning due to any other means of disconnect is achieved by pressing the control wheel disengage switch once.

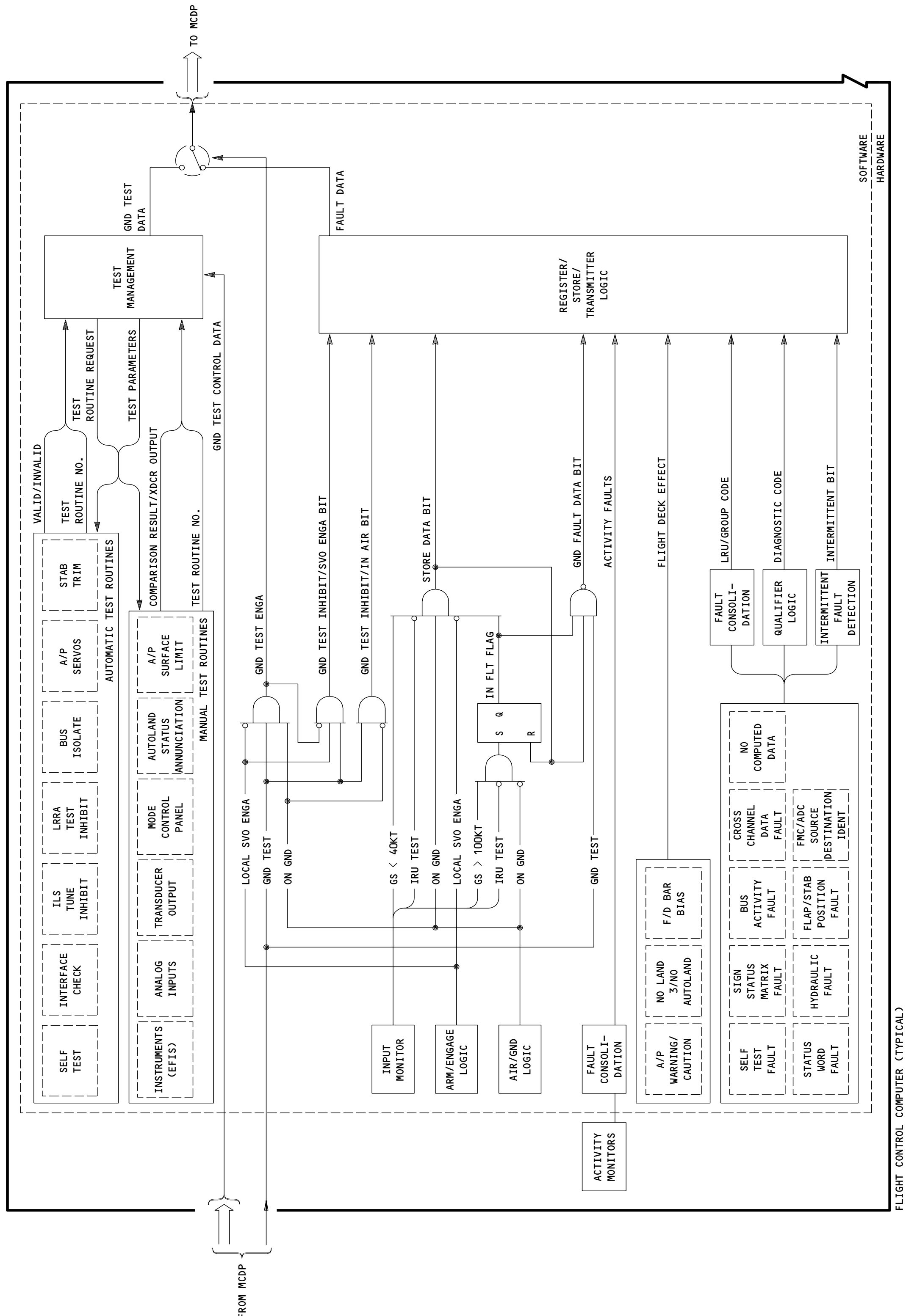
B. Autopilot/Flight Director System Signal Source Fault Detection (SSFD) (Fig. 10)
(1) Signal Source and Fault Detection provides independent selection and monitoring of sensor input data in each FCC. Signal Selection is provided to maintain servo tracking during multi-channel operation and when dual flight directors are selected. Monitoring sensor data allows detection of a failed sensor and permits exclusion of the invalid data from autopilot servo and flight director command computations. The SSFD also provides failure flags for ASA, Caution, and mode fail annunciations. There are three types of SSFD algorithms utilized for triplex, discrete, and dual/single source sensors.
(a) Mediam Select SSFD Process computes the mid-value of triplex (3) sensor inputs.
(b) Discrete SSFD Process perform a majority vote of three discretes.
(c) Force Select SSFD process selects one data source for use by all FCCs. All ADC data and ADC source IRU data (inertial altitude and inertial vertical speed) are force selected.

C. Autopilot/Flight Director System MCDP Monitor and Test (Fig. 11)



Autopilot/Flight Director System Signal Source Fault Detection (SSFD) Schematic
Figure 10

263312



Autopilot/Flight Director System MCDP Monitor and Test Schematic
Figure 11

(1) The MCDP provides a manual test of the ASAs. The ASA test verifies the proper operation of the front panel test and reset switches.

(2) The FCC provides the following system interface tests:

(a) ARINC 429 inputs from system units are checked in two ways. First, the FCC checks the sign status matrix to verify the validity of the incoming data. The buffer memory locations on the I/O data path assembly are also periodically checked to determine if the data words have been updated. If the words have not been updated, then a fault is assumed and the data words at the tested memory locations are not used.

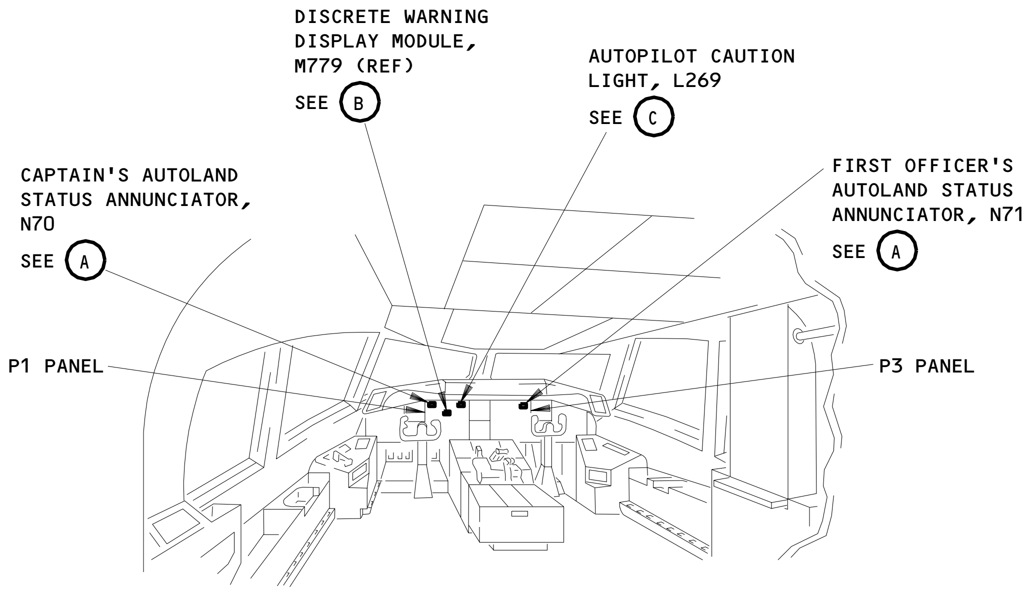
(b) Discrete input and output interfaces are monitored by software which detects shorted wires. These monitors are used for the caution/warning and autoland status annunciator inputs and outputs as well as the engage and arm solenoid power, trim outputs, engage and arm commands and inputs. The interface monitor circuits are tested, as appropriate, at power up or prior to engagement of autopilot/autoland.

AUTOPILOT/FLIGHT DIRECTOR WARNING AND ANNUNCIATION

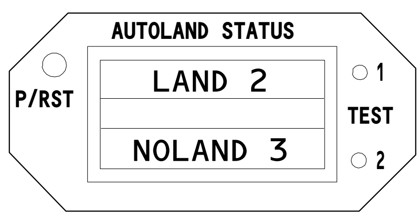
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
ANNUNCIATOR - CAPT AUTOLAND STATUS, N70	—	1	FLT COMPT, P1	22-14-01
ANNUNCIATOR - F/O AUTOLAND STATUS, N71	—	1	FLT COMPT, P3	22-14-01
LIGHT - AUTOPILOT DISC	—	1	FLT COMPT, P1, DISCRETE WARNING DISPLAY MODULE, M779 (REF)	*
LIGHT - AUTOPILOT CAUTION, L269	—	1	FLT COMPT, P1	
MODULE - (FIM 33-16-00/101) DISCRETE WARNING DISPLAY				*

* SEE WDM EQUIPMENT LIST

Autopilot/Flight Director Warning and Annunciation - Component Index
Figure 101

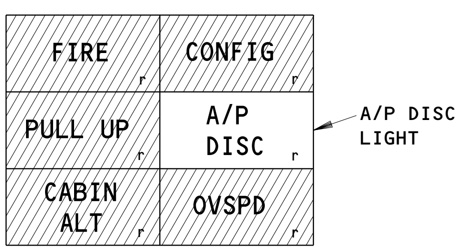


FLIGHT COMPARTMENT



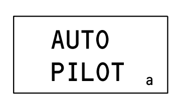
CAPTAIN'S OR FIRST OFFICER'S AUTOLAND
STATUS ANNUNCIATOR, N70 OR N71

A



DISCRETE WARNING DISPLAY
MODULE, M779 (REF)

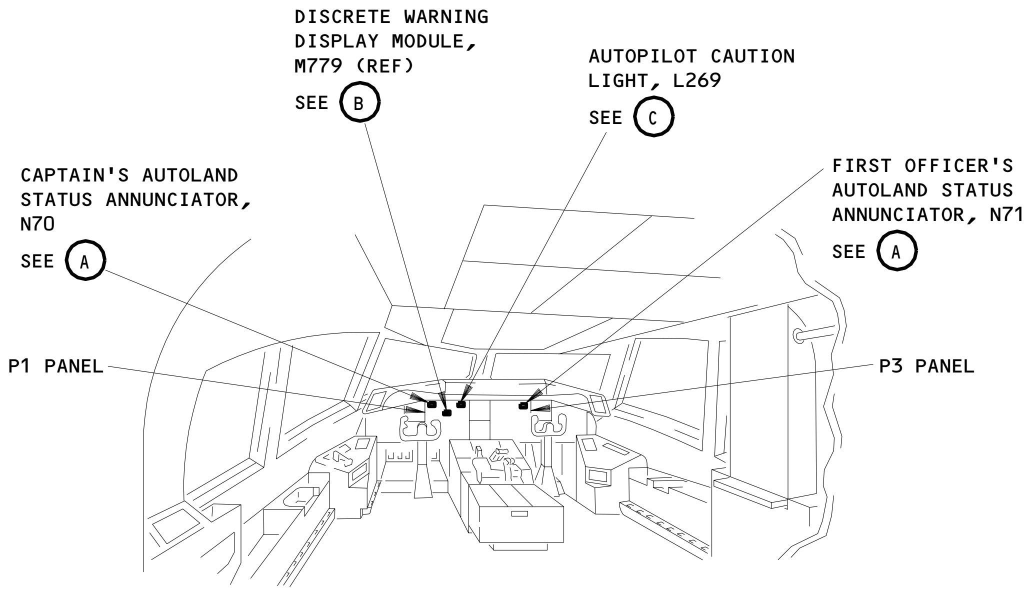
B



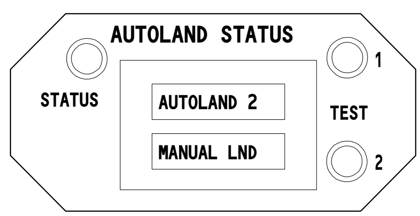
AUTOPILOT CAUTION
LIGHT, L269

C

Autopilot/Flight Director Warning and Annunciation - Component Location
Figure 102 (Sheet 1)

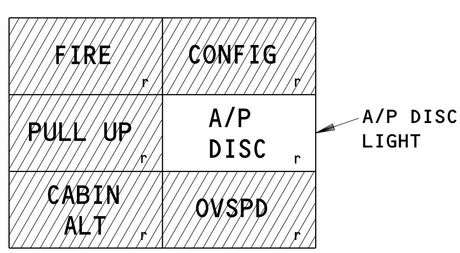


FLIGHT COMPARTMENT



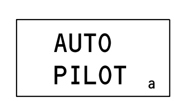
CAPTAIN'S OR FIRST OFFICER'S AUTOLAND
STATUS ANNUNCIATOR, N70 OR N71

A



DISCRETE WARNING DISPLAY
MODULE, M779 (REF)

B



AUTOPILOT CAUTION
LIGHT, L269

C

Autopilot/Flight Director Warning and Annunciation - Component Location
Figure 102 (Sheet 2)

AUTOLAND STATUS ANNUNCIATOR - REMOVAL/INSTALLATION

1. General
   A. Two Autoland Status Annunciators (ASAs) are found in the flight compartment. The left (captain's) ASA is on the P1 panel. The right (first officer's) ASA is on the P3 panel. Each ASA is attached to the panel with a clamp. Two electrical connectors are on the rear of each ASA.

TASK 22-14-01-004-001

2. Remove the Autoland Status Annunciator
   A. Access
      (1) Location Zone
          211/212 Flight Compartment

   B. Prepare for Removal

      S 864-003
      (1) For the left ASA, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E16, MODE CONT PNL LEFT

      S 864-004
      (2) For the right ASA, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E34, MODE CONT PNL RIGHT

   C. Remove the Autoland Status Annunciator

      S 034-005
      (1) Loosen the two screws that attach the clamp to the ASA. The screws are found in the upper right and the lower left corners.

      S 024-006

      CAUTION: CAREFULLY REMOVE THE ASA FROM THE PANEL TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO THE ELECTRICAL CABLES COULD OCCUR.

      (2) Pull the ASA out of the panel.

S 864-023

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

(3) Disconnect the electrical connectors.

S 434-002
(4) Apply protective covers to the electrical connectors.

TASK 22-14-01-404-008
3. Install the Autoland Status Annunciator
A. References
(1) AMM 22-00-02/201, Autoflight BITE
(2) AMM 24-22-00/201, Electrical Power - Control
B. Access
(1) Location Zone
211/212 Flight Compartment

C. Install the Autoland Status Annunciator

S 034-009
(1) Remove the protective covers from the electrical connectors.

S 864-025

CAUTION: CROSS CONNECTION POSSIBILITY WNEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

(2) Connect the electrical connectors.

S 424-011

CAUTION: CAREFULLY PUT THE ASA INTO THE PANEL TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO THE ELECTRICAL CABLES COULD OCCUR.

(3) Put the ASA into the panel opening.

S 434-012

(4) Tighten the two screws, in the upper right and lower left corners, that attach the clamp to the ASA.

D. Test the autoland status annunciator.

S 864-013

(1) Supply electrical power (AMM 24-22-00/201).

S 864-014

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11A17, AUTOFLIGHT WARN
   (b) 11E16, MODE CONT PNL LEFT
   (c) 11E34, MODE CONT PNL RIGHT

S 714-015

(3) Make sure that the ASA lightplate comes on.

S 714-016

(4) Push the TEST 1 switch on the applicable ASA.
   (a) GUI 001-114, 116-999;
   make sure that the top display shows LAND 3 and the bottom display shows NO LAND 3.
   (b) GUI 115;
   make sure that the top display shows AUTOLND 3 and the bottom display stays blank.

S 714-017

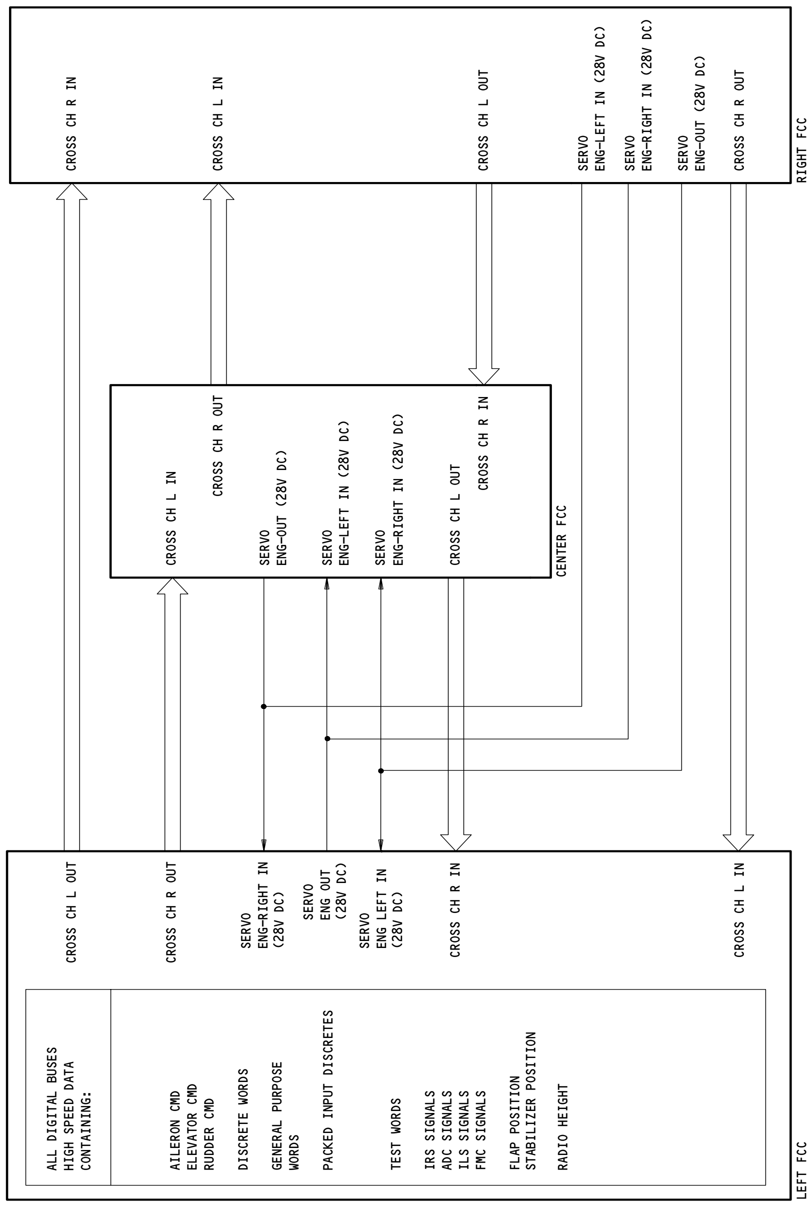
(5) Push the TEST 2 switch on the applicable ASA.
   (a) GUI 001-114, 116-999;
   make sure that the top display shows LAND 2 and the bottom display shows NO AUTOLND.
   (b) GUI 115;
   make sure that the top display shows AUTOLND 2 and the bottom display shows MANUAL LND.

S 714-018
(6) Do MCDP Ground Test 06-ASA (AMM 22-00-02/201).
    (a) Make sure that no failures occur during the test.

S 864-019
(7) Remove electrical power if not necessary (AMM 24-22-00/201).

AUTOPILOT/FLIGHT DIRECTOR INTERCHANNEL DATA - DESCRIPTION AND OPERATION

1. General
   A. The three Flight Control Computers (FCCs) communicate with each other via interchannel digital data buses and analog discretes. There are two input and two output interchannel high speed ARINC 429 buses connected to each FCC. There are two input and one output discrete data lines for analog discretes. Sensor signals, engage, and autoland status are shared to allow signal voting, fault detection, and determination of proper annunciations based on the status of all FCCs.
2. Operation
   A. Functional Description
      (1) Autopilot/Flight Director Interchannel Data (Fig. 1)
         (a) General
            1) All three FCCs communicate with each other via the four ARINC 429 high speed interchannel data buses and the three analog discretes data lines. The buses operate at 100 Kilobits Per Sec (100KBPS). The FCCs share sensor signals on the interchannel digital data buses. This allows signal voting and fault detection through comparison. The three FCCs share the interchannel discrete data lines containing engage and autoland status. This allows each FCC to determine proper annunciations based on the status of all FCCs.
         (b) Digital Buses
            1) All cross-channel data is transmitted as non-standard words. The ARINC word is reformated so that data in the most significant bit is right shifted one position. This makes bit 29 a SSM bit. The right shifting makes bit 28 the most significant bit. This causes the full scale input range to double since the maximum input range is .5. Therefore, test equipment for a standard 429 bus cannot be used without manual conversion and interpretation of the data.
            2) Sensor inputs, surface commands, input discretes, and status are carried on the buses as follows:
               a) Aileron, elevator, and rudder surface positions in Binary Numerical Representation (BNR) format.
               b) Four discrete words contain past value, autoland, armed, and active mode status.
               c) Two general purpose words are BNR spares.
               d) Two packed input discretes words. One word contains go-around and slats extended mode status. The second word reserved for strapping pins.
               e) Two test words are generated by each FCC. They verify the validity of the intercommunication paths.
               f) Nineteen Inertial Reference Unit (IRU) signals contain vertical and lateral navigation data in BNR format.
               g) Eight Air Data Computer (ADC) signals contain altitude, angle of attack, mach, airspeed, and impact pressure in BNR format.



Autopilot/Flight Director Interchannel Data Schematic
Figure 1

h) Three Instrument Landing System (ILS) signals contain selected runway heading, localizer, and glideslope deviation in BNR format.

i) Three Flight Management Computer (FMC) signals contain vertical and lateral navigation in BNR format.

j) Flap position, stabilizer (stab) position, and radio height data in BNR format.

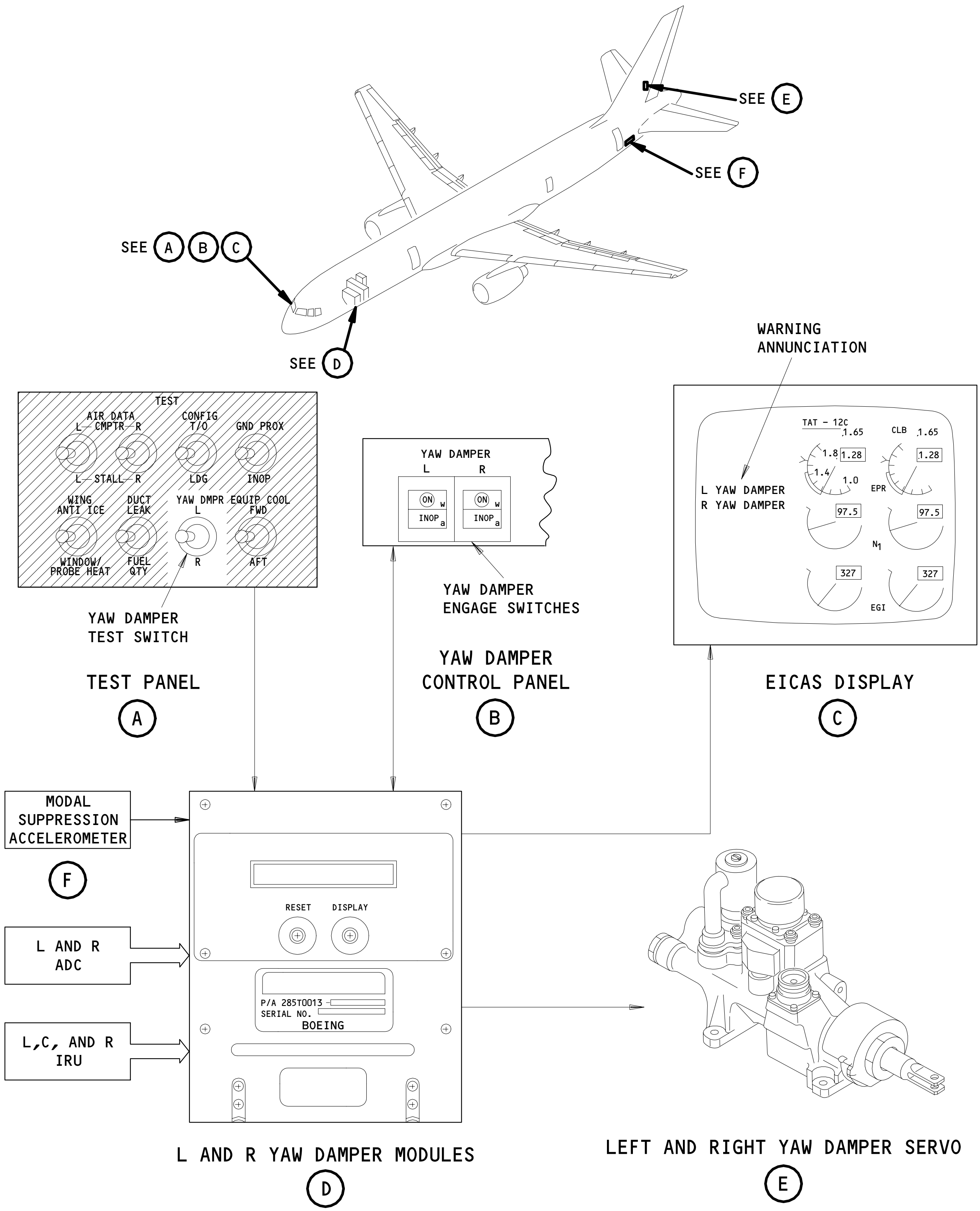
k) Heading and altitude reference data in BNR format for power-up synchronization.

(c) Discretes

1) Pitch and roll servo engage discretes are generated by center, left, and right FCCs. The FCCs use this as part of the information to establish autopilot disconnect warnings.

YAW DAMPER SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. An inherent characteristic of all swept back wing airplanes is a yawing motion that induces sideslip and roll. The Yaw Damper System (YDS) uses the following inputs to dampen undesirable sideslip and rolling motions:
      (1) Air Data Computer
      (2) Inertial Reference Unit
      (3) Modal Suppression Accelerometer
   B. The YDS commands drive yaw damper servos which control the rudder mechanical linkage and power control actuators. Rudder displacement commands are proportional to yaw rate. Turn coordination is also provided by the YDS. Roll attitude inputs are used to compute turn coordination commands.
   C. Control Panels
      (1) The yaw damper panel on overhead panel P5 provides control and indication of yaw damper engagement status. The panel indicates a yaw damper ON or inoperative (INOP) state. The YAW DMPR test switch is located on the P61 panel and initiates the system operational test.
   D. Yaw Damper Modules (2)
      (1) The yaw damper modules receive sensor inputs from the following systems to compute rudder commands appropriate to flight conditions.
         (a) Air Data Computers (ADCs)
         (b) Inertial Reference Units (IRUs)
         (c) Modal Suppression Accelerometer
      (2) These commands drive the yaw damper servos, which control the rudder through power control actuators. The modules monitor system operation and indicate system faults through automatic and manually initiated system testing. Yaw damper fault indications consist of the following (Ref 31-41-00):
         (a) Yaw damper panel INOP light
         (b) EICAS display unit annunciation
         (c) Fault message display
   E. Yaw Damper Servos (2)
      (1) The yaw damper servos use electrical command inputs from the yaw damper modules. The electrical input commands control hydraulic flow to actuator pistons, which provide a mechanical output. This output is connected in series with manual and autopilot rudder inputs supplied to rudder power control actuators, which actually move the rudder.



Yaw Damper System
Figure 1

(2) The maximum rudder authority for each SERVO is 6 degrees (3 degrees in each direction). If both SERVOS are operating, the rudder authority is 12 degrees. The yaw damper servo command is zero while the airplane is on ground, except during preflight testing. The yaw damper command is provided to the servos via an easy-on/easy-off function that reduces transients during ground/air transitions. If a system fault is detected, the servos are disengaged.

F. Modal Suppression Accelerometer

(1) The modal suppression accelerometers provide the yaw damper modules with lateral acceleration data. Each yaw damper module has its own modal suppression accelerometer.

G. Yaw Damper Component Locations (Fig. 2)

(1) Flight Deck

(a) The pilots overhead panel (P5) contains the yaw damper controls and indicators. Each channel (left and right) has a separate yaw damper ON switch/light and INOP indicator. The YAW DMPR test switch is on the P61 panel.

(2) Main Equipment Center

(a) The yaw damper modules are located in the main electrical/electronic equipment center. The right module is on the E4-1 shelf. The left module is on the E3-1 shelf.

(3) Vertical Stabilizer

(a) The two yaw damper electrohydraulic servos are connected to the mechanical rudder linkage. The mechanical linkage is the interface between the yaw damper servos and the power control actuators, which move the rudders.

(4) Modal Suppression Accelerometer

(a) The two modal suppression accelerometers are located in the aft cargo compartment ceiling and are accessed through the aft cargo compartment door (822).

H. Yaw Damper Block Diagram (Fig. 3)

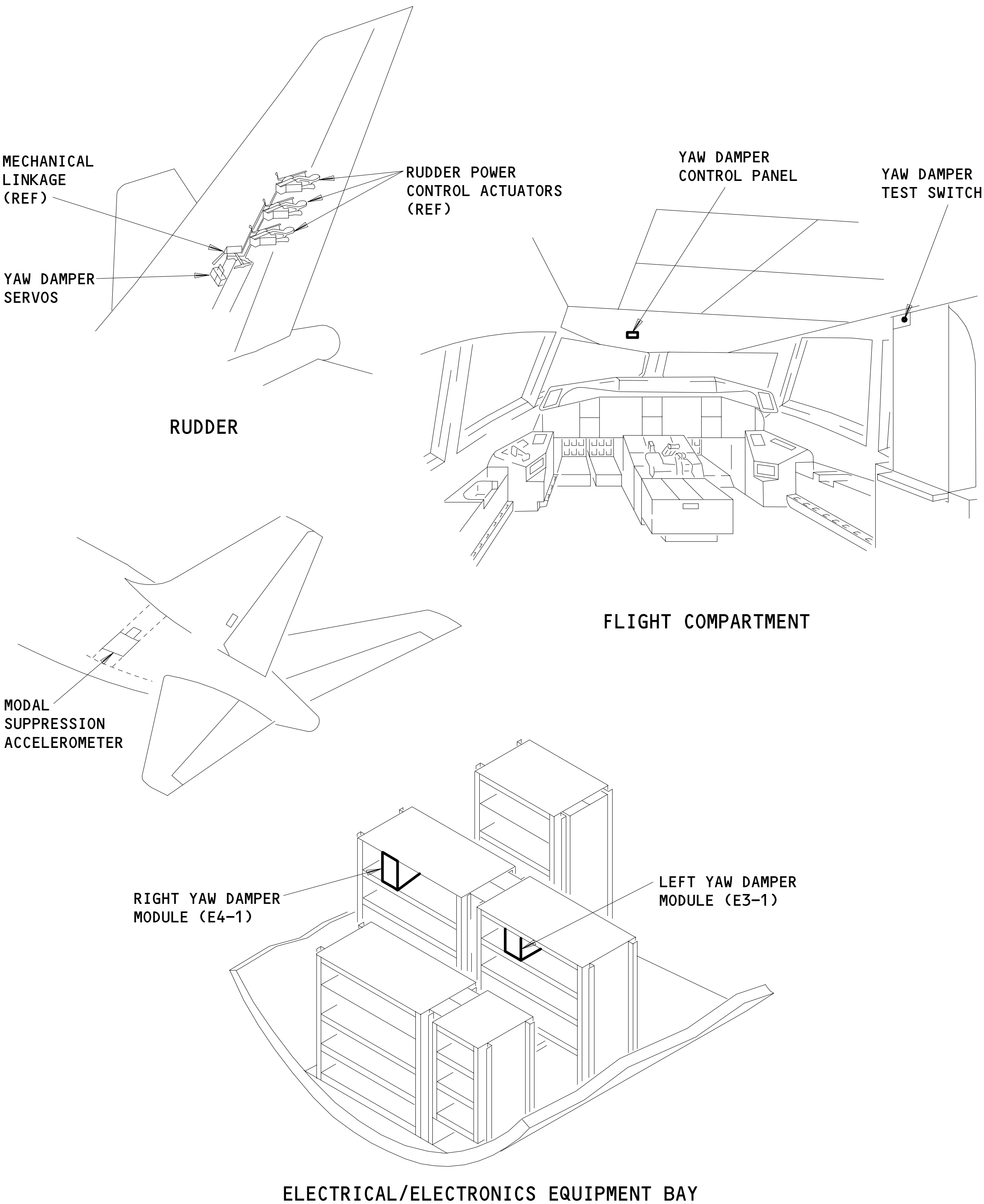
(1) System Inputs

(a) The YDS uses the following inputs to determine existing conditions, rudder commands, and system operation.

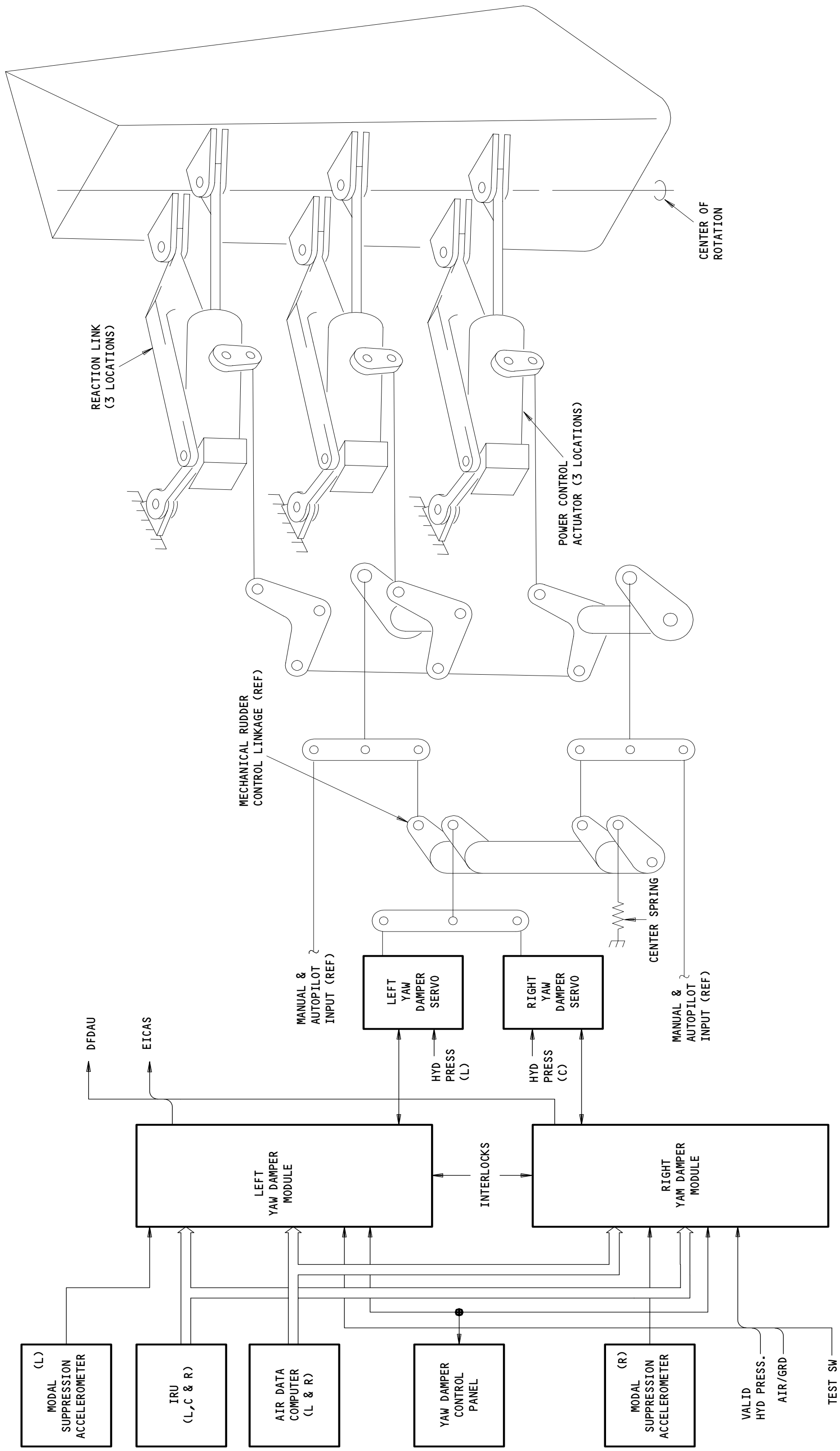
1) The left, right, and center IRUs each provide roll attitude, roll rate, yaw rate, lateral acceleration, and ground speed. The left and right ADCs each provide true airspeed (TAS), impact pressure, and angle of attack (AOA).

2) The left and center hydraulic pressure switches sense normal hydraulic pressure to the left and right yaw damper servos, respectively. The air/ground relay provides an analog discrete signal indicating the airplane is airborne or on the ground.

3) Airplane type code is provided by program pins selectively grounded through airplane wiring. The yaw damper channel (left or right) code is also provided by program pins selectively grounded through airplane wiring.



Yaw Damper Component Locations
Figure 2



Yaw Damper Block Diagram
Figure 3

276506

4) The modal suppression accelerometers provide lateral acceleration data to their corresponding yaw damper modules.

(2) Yaw Damper Modules

(a) The yaw damper modules contain ARINC 429 receivers (data bus input) plus software signal check and selection programmed routines. As multi-input parameters are received, a median value selector provides single valid input parameters for microprocessor calculations.

(b) The yaw damper modules provide mode control for various operational conditions encountered over the full flight regime. Rudder commands computed by the modules assist turn coordination and damp undesirable yaw. Internal circuits in the modules conduct tests and exercise monitoring circuits. These provide failure indications and YDS disengagement if a fault is detected.

(c) The yaw damper modules perform signal conditioning for analog rudder command outputs to the servos. Servo Linear Variable Differential Transducer (LVDT) feedback monitored by the modules determines rudder position. The two yaw damper modules are connected by an interlock to determine if both modules are installed. If one module is not installed, the interlock is open circuited. This causes the installed module to illuminate an INOP annunciator (for the absent module) on the yaw damper panel.

(3) Yaw Damper Servos

(a) Two yaw damper servos are connected in series to the rudder actuating linkage. The yaw damper servos provide the mechanical output to the rudder power control actuator input linkage. The servos receive electrical commands from the yaw damper modules and convert these signals to mechanical commands via an electrohydraulic servo valve (EHSV). The EHSV responds to electrical commands and delivers hydraulic flow proportional to the input signals. Hydraulic fluid then drives the servo actuator piston. The actuator piston drives the LVDT, which provides position feedback to the yaw damper module.

(4) Modal Suppression Accelerometer

(a) The modal suppression accelerometers measure the lateral acceleration of the aft of the airplane. This information is provided to their corresponding yaw damper modules as an analog signal.

(5) System Annunciation

(a) The yaw damper ON and INOP annunciation is on the yaw damper panel. System failure status is also provided to the EICAS computers for display on the EICAS display unit (Ref 31-41-00).

(6) Digital Flight Data Acquisition Unit (DFDAU) interface
(a) The left and right yaw damper modules provide left and right yaw damper engagement status to the DFDAU (Ref 31-31-00).

2. Component Details (Fig. 4)

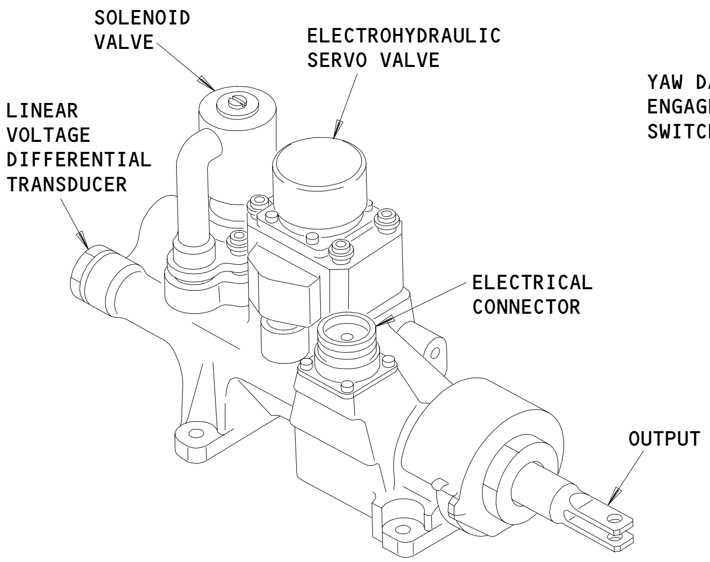
A. Yaw Damper Panel Assembly

(1) The YDS controls and indicators are on the overhead panel (P5). The left and right ON and INOP indicator lamps use 28 vdc provided by the standby and right main bus, respectively. Case cooling is provided by forced ambient cooling air. Two electrical connectors (one for each yaw damper module) are on the back side of the panel.

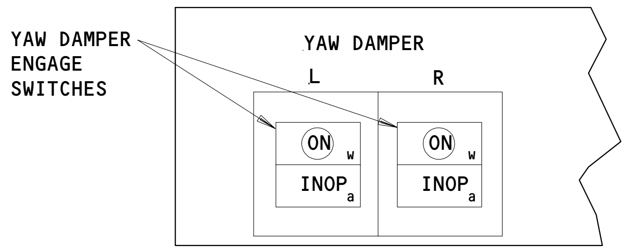
(2) Each yaw damper module has its own set of controls and indicators on the panel.
(a) The ON switch/light controls engagement of the YDS by controlling 28 vdc to the servo solenoid valve on the yaw damper servos. When the switch/light is first pressed, the ON legend is illuminated. Pressing the switch/light a second time turns off the YDS.
(b) The INOP indicator illuminates under the following conditions:
1) When the yaw damper module associated with the INOP is not installed (the opposite module must be installed and functioning properly)
2) When the yaw damping function is lost
3) During the pre-flight test (the YAW DMPR test switch initiates a pre-flight YDS test, provided the airplane is on ground)
4) During the power-up test

B. Yaw Damper Module

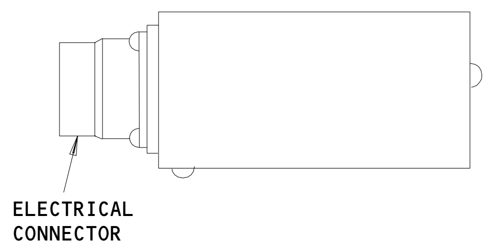
(1) Each yaw damper module consists of two digital computing channels (control and monitor) and a servo amplifier. The module consists of the following functional components:
(a) Fault Display - provides the logic to display fault messages on the front panel of the module.
(b) Monitor CPU - provides the digital computing function for the yaw damper, duplicating the computing of the control channel.
(c) Monitor Receiver - provides three high-speed and two low-speed ARINC 429 receiver interfaces for the control and monitor microprocessors.



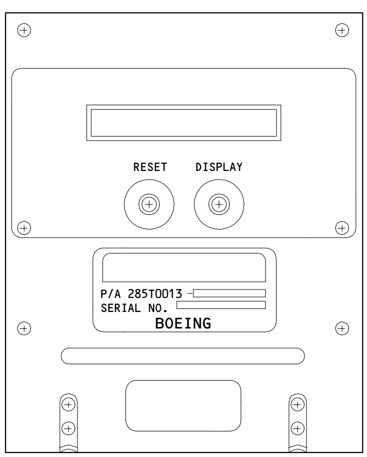
YAW DAMPER SERVO



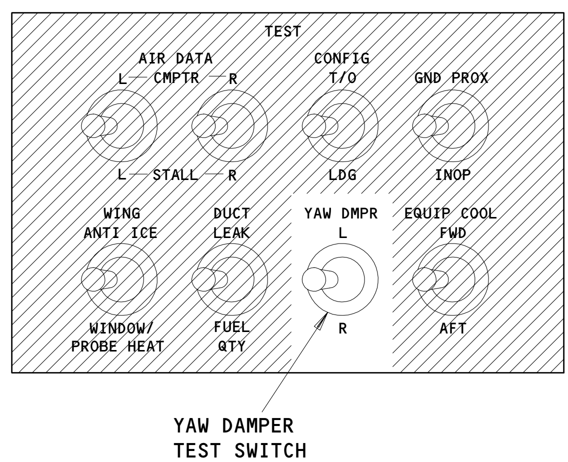
YAW DAMPER PANEL



MODAL SUPPRESSION
ACCELEROMETER



YAW DAMPER MODULE



MISC TEST PANEL

Yaw Damper System Components
Figure 4

(d) Analog - provides the output to the yaw damper servos and analog to digital and digital to analog signal conversion for both channels.
(e) Input/Output - provides input and output discrete signal conditioning for the control and monitor microprocessors.
(f) Control CPU - provides the digital computing function for the yaw damper.
(g) EMI Filter - provides EMI filtering for signal inputs, signal outputs and electrical power inputs for the control and monitor channels.

(2) The servo amplifier provides rudder command signals to the yaw damper electrohydraulic servo-valve. Feedback from the Linear Variable Differential Transducer (LVDT) position sensor is received by the actuator position demodulator assembly to close the servo loop. The assembly also provides servo loop monitoring which operates the automatic solenoid disengagement relay when a failure affecting operation is detected.

(3) One connector with three inserts is on the rear panel of each module. Two inserts contain the YDS signal pins and low voltage power pins. One insert contains the ground reference pins. Cooling is provided by air from the rack plenum being forced from the bottom to the top of the modules. A hinged front provides access to the interior of the module.

(4) The 28v dc engage servo solenoid power is routed from the control panel, through the module auto-disengage relay contacts, to the servo. Module operating power (+5v dc and ±15v dc) and the 26v ac LVDT demodulator reference is provided by the Control System Electronics Unit (CSEU) power supply modules (Ref 27-09-00).

(5) The fault detection logic in the modules isolates faults to the LRU level within the YDS and interfacing LRUs. The faults are divided into FAULTS NOW and LAST LEG FAULTS. FAULTS NOW corresponds to the fault monitors currently indicating a fault. LAST LEG FAULTS are the faults that occurred prior to the last air/ground transition.

(6) The following messages are displayed by the yaw damper module:
(a) NO FAULTS - There are no FAULTS NOW and no LAST LEG FAULTS.

(b) NO FAULT NOW - There are no FAULTS NOW but there are LAST LEG FAULTS.
(c) NO LEG FAULT - There are FAULTS NOW but there are no LAST LEG FAULTS.
(d) FAULTS NOW - There are one or more FAULTS NOW faults, and they are displayed by pressing the DISPLAY button on the faceplate of the yaw damper module.
(e) LAST LEG - There are one or more LAST LEG FAULTS, and they are displayed by pressing the DISPLAY button on the faceplate of the yaw damper module.
(f) TEST RUNNING - Power Up Test (PUT) or Pre-Flight Test (PFT) is running on the yaw damper module.
(g) ACCEL - The yaw damper module has determined that there is a fault with the modal suppression accelerometer.
(h) L ADC, R ADC - The yaw damper module has determined that the data from the indicated air data computer is invalid.
(i) ADC DIFF - The yaw damper module has detected an invalid input from an air data computer but is unable to determine which one is at fault.
(j) AIRGND 1, AIRGND 1/2, AIRGND 2 - The yaw damper module has determined that the indicated air/ground discrete is invalid.
(k) AIR DIFF - The yaw damper module has detected an invalid air/ground discrete but is unable to determine which air/ground relay is at fault.
(l) 26 VAC - The yaw damper module has detected the loss of or the invalid level of the 26V RMS reference voltage.
(m) HYD SWITCH - The hydraulic pressure high signal agreed with the hydraulic pressure low signal for more than 2 seconds.
(n) L IRU, C IRU, R IRU - The yaw damper module has determined that the data from the indicated inertial reference unit is invalid.
(o) IRU OFF - The IRUs are not on or no data is transmitted.
(p) IRU NCD - IRUs are on but no data or bad data is transmitted.
(q) IRU DIFF - The yaw damper module has detected an invalid input from an inertial reference unit but is unable to determine which one is at fault.

(r) LVDT - The yaw damper module has detected that the LVDT common mode voltage is low. During the preflight test, an LVDT fault is recorded if the common mode voltage is low and a position test fault is detected (valid high hydraulic pressure must be present).

(s) YD ACT - During normal operation, a yaw damper actuator fault occurs when the yaw damper module detects a servo loop fault. During the preflight test, a yaw damper actuator fault is recorded if a position test fault occurs. To record a YD ACT fault, valid high hydraulic pressure must be present and the LVDT common mode voltage must be normal.

(t) YD TEST - Failure of PFT (Pre-Flight Test).

(u) YDM - A yaw damper module fault is recorded and latched with the following faults: power up test fault, preflight test fault, A/C type code input fault, control or monitor channel command coincidence fault, or a computing channel fault. A YDM fault is also recorded when either control or monitor channel detected auto-disengage relay faults.

1) If no YDM fault exists, the YDM fault latch resets for the following conditions:

a) Successful completion of power up test or maintenance reset on ground.

b) Power up reset, manual engage or YDM autoreset in air or on ground.

(7) Fault messages are displayed on the 12 character alphanumeric LED display located on the faceplate of the yaw damper module. To view the fault messages, the DISPLAY button is pressed on the faceplate of the yaw damper module until all the fault messages are displayed. To display the fault messages one at a time, the DISPLAY button is pressed and released to display each message. The fault messages automatically extinguish from the display after 30 seconds. LAST LEG FAULT messages are distinguished from FAULTS NOW messages by an asterick (*) preceeding the fault message. To view the first fault message, the DISPLAY button is pressed and held for over 2 seconds.

(8) When the RESET button on the faceplate is pressed, all yaw damper module monitors and control logic are reset. To inhibit the LAST LEG FAULT messages from being displayed, the DISPLAY and RESET buttons are pressed at the same time.

C. Yaw Damper Servo

(1) Each yaw damper servo drives the mechanical linkage to the power control actuators. Servo weight is approximately 7.5 pounds and the piston stroke is ±0.75 inches. Normal operating pressure at the pressure port is 3000 psi. The maximum return pressure is 100 psi. The servo rated load at 3000 psi is 1700 pounds maximum with the piston extended or retracted.

(2) Servo component parts consist of the input filter, electrohydraulic solenoid valve, electrohydraulic servo valve, output piston and shaft, and LVDT. A single electrical connector completes the servo engage voltage, command signal voltage, LVDT excitation voltage, and LVDT feedback voltage.

(3) Yaw Damper Servo Schematic (Fig. 5)

(a) Hydraulic pressure is maintained at the servo input port by the airplane hydraulic system. The hydraulic fluid is filtered by a filter screen before entering the servo.

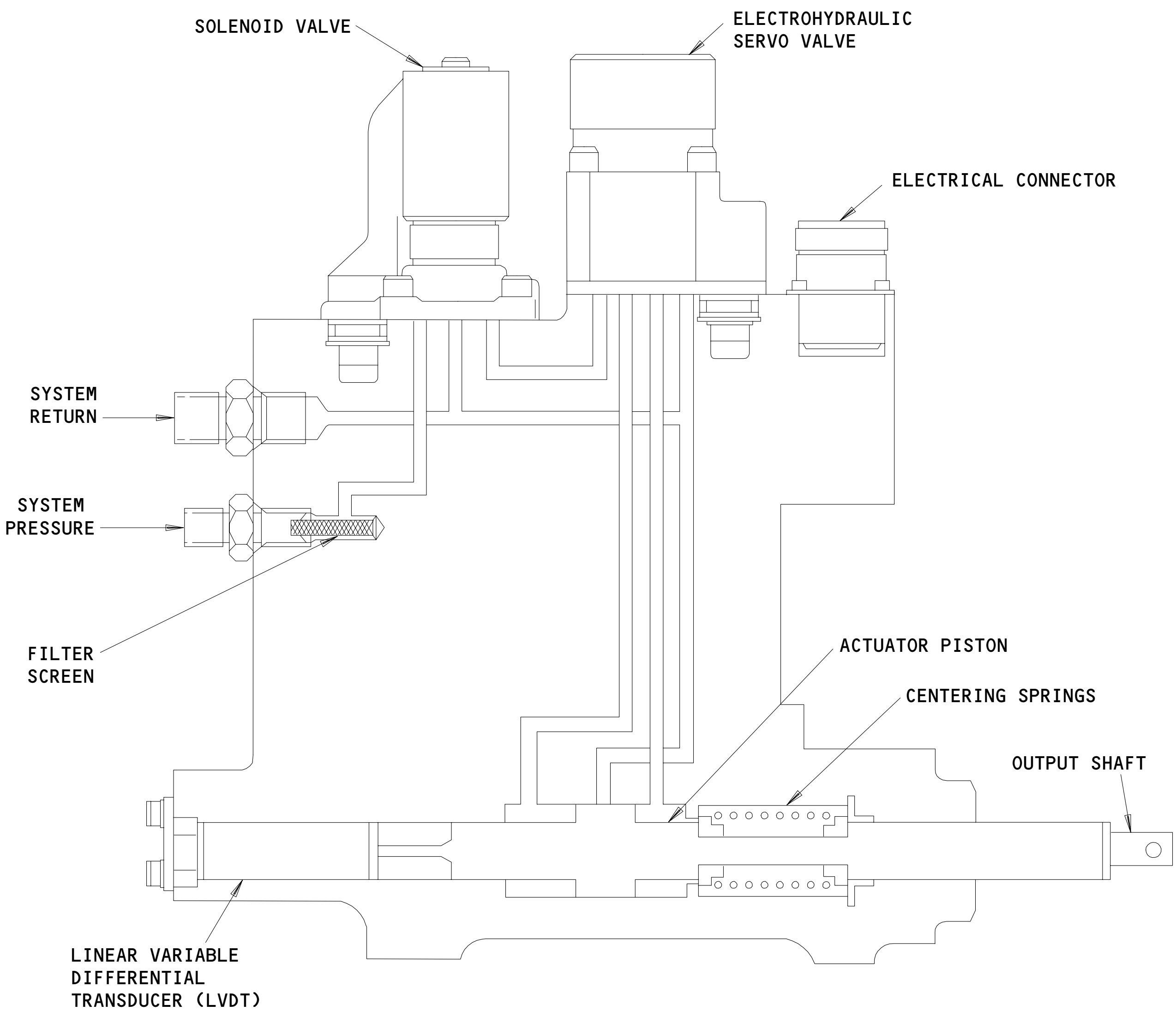
(b) Solenoid Valve

1) The solenoid valve is an electrically operated open-close valve that completes hydraulic pressure through the servo when the yaw damper system is armed. The 28 vdc valve-actuating voltage is controlled by the auto-disengage relay in the yaw damper module. The valve is installed with 4 bolts and sealed with a gasket plate.

(c) Electrohydraulic Servo Valve (EHSV)

1) One EHSV is installed on each yaw damper servo. It contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydraulic pressure through the valve can be applied to either of two output ports. When no error signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor and out the projector jet. From the jet, equal pressure is applied to opposite ends of the piston, holding it at center and closing both output ports.

2) When an error signal is supplied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston causing it to move. This opens the corresponding output port to complete hydraulic fluid to the servo. When the error signal is nulled out, the motor armature and jet return to center. This equalizes the pressures on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.



Yaw Damper Servo Schematic
Figure 5

3) The EHSV is controlled by a rudder command signal from the yaw damper modules. In response to this command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the LVDT nulls out the module error signal and piston movement stops. The EHSV is installed with 4 bolts and sealed with a gasket plate.

(d) Actuator Piston Assembly

1) The actuator piston translates yaw damper input commands through the EHSV into mechanical positioning of the rudder. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV. The springs center the piston when pressurization is released.

(e) Linear Variable Differential Transducer (LVDT)

1) The actuator piston LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The LVDT provides a feedback voltage to the yaw damper modules that varies directly with linear motion. This feedback is used to null the servo loop. The LVDT is not a line replaceable unit.

(f) The electrical connection to the yaw damper servo is provided by one external and two internal connectors. One internal connector is used for the solenoid valve and the other for the EHSV. These electrical pins are mated when the solenoid or EHSV is bolted in position.

(g) The solenoid valve on each servo is controlled by the auto-disengage relay in the yaw damper modules. When 28 vdc is applied, the valve opens and arms the yaw damper system. The servo valve opens in proportion to a dc command signal from the yaw damper modules. The LVDT is excited by 26 vac, 400 Hz and provides a feedback voltage signal proportional to the linear position of the actuator piston.

D. Modal Suppression Accelerometer

(1) The modal suppression accelerometer provides the yaw damper module with lateral acceleration data of the aft end of the aircraft. This data aids the yaw damper module in controlling the yaw and turn coordination of the aircraft.

(2) The modal suppression accelerometer is composed of a detection circuit that consists of a cantilever surrounded on both sides by metal plates. The cantilever moves with the changing lateral velocity of the aircraft. An output signal is produced by comparing the difference of the spacial gaps between the metal plates and the cantiliver.

3. Operation

A. Functional Description

(1) Yaw Correction Requirements (Fig. 6)

(a) Sideslip

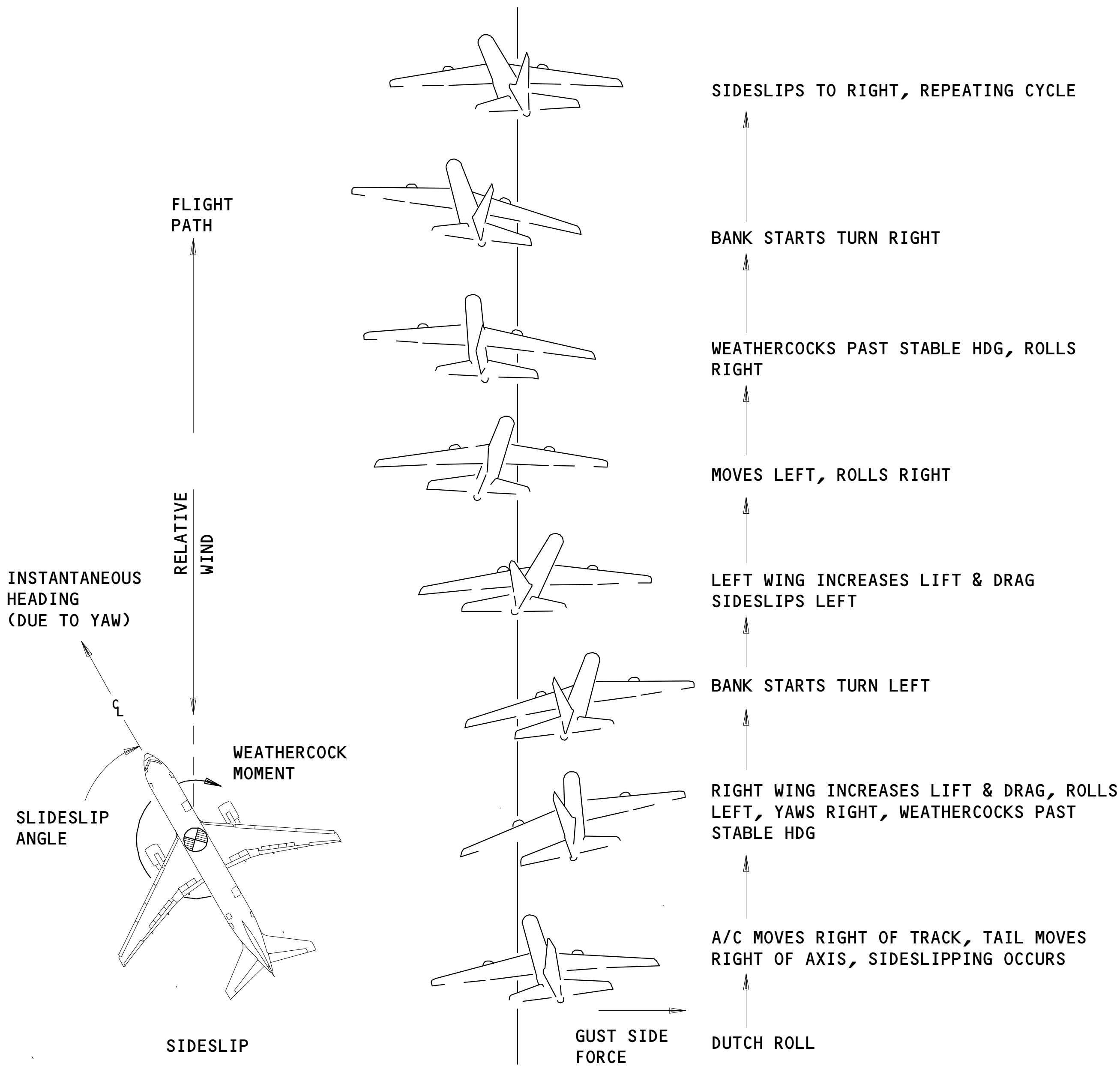
1) An airplane is defined to be in sideslip when the airplane heading is not aligned with the flight path. The sideslip angle is the angle between airplane heading and the flight path. Airplane sideslip is caused by unstable air conditions, aerodynamic instability, or banking without coordinating rudder movement. When the airplane sideslips, it flies yawed and a side force tends to retard the sideways velocity. This damping force comes largely from increased drag on the faster moving wing, the fuselage, and vertical stabilizer. Damping and the vertical fin tends to correct the sideslip by "weathercocking" the airplane.

2) When a sideslip condition is present, the airflow on the forward wing goes from an outboard point on the leading edge to an inboard point on the trailing edge. The airflow on the trailing wing goes from an inboard point on the leading edge to an outboard point on the trailing edge. This results in an increase in angle of attack of the forward wing relative to that of the trailing wing. A difference in angle means a difference in lift on the two sides of the airplane that induces a rolling moment.

(b) Dutch Roll

1) Dutch roll is a common oscillatory condition encountered due to low drag high speed aerodynamic design and turbulance which can be created by air mass instability. If this condition is allowed to persist, it causes considerable discomfort. If the airplane experiences a side force causing yaw relative to the flight path, sideslip is produced. The change in direction of relative wind causes one wing to produce more lift than the other, thus inducing roll.

2) The yaw induced roll and weathercocking moment combine to induce yaw in the opposite direction of the roll. This produces an uncoordinated turn. The change in direction of relative wind causes one wing to produce more lift than the other, causing the wings to level. The airplane then banks back in the other direction and repeats the cycle with oscillating motion. The frequency of this oscillation is in the range from 0.5 to 0.1 Hz.



Yaw Correction Requirements
Figure 6

3) The yaw damper provides damping via the rudder to eliminate dutch roll.

(c) Turn Coordination

1) When the airplane is rolled into a turn, a yawing moment is generated by changing lift vectors due to the changes in relative wind. This moment produces a forward component (reduced drag) on the down-going wing and a rearward component (increased drag) on the up-going wing. The result of the yaw moments produce yaw opposite to the intended turn direction. Rudder deflections to counter roll induced yaw are required to achieve a coordinated turn.

(2) Yaw Damper Operational Modes (Fig. 7)

(a) The yaw damper modules provide self-tests, full flight regime yaw damping control, and turn coordination. To accomplish this, each module has eight opertional modes described in the following paragraphs.

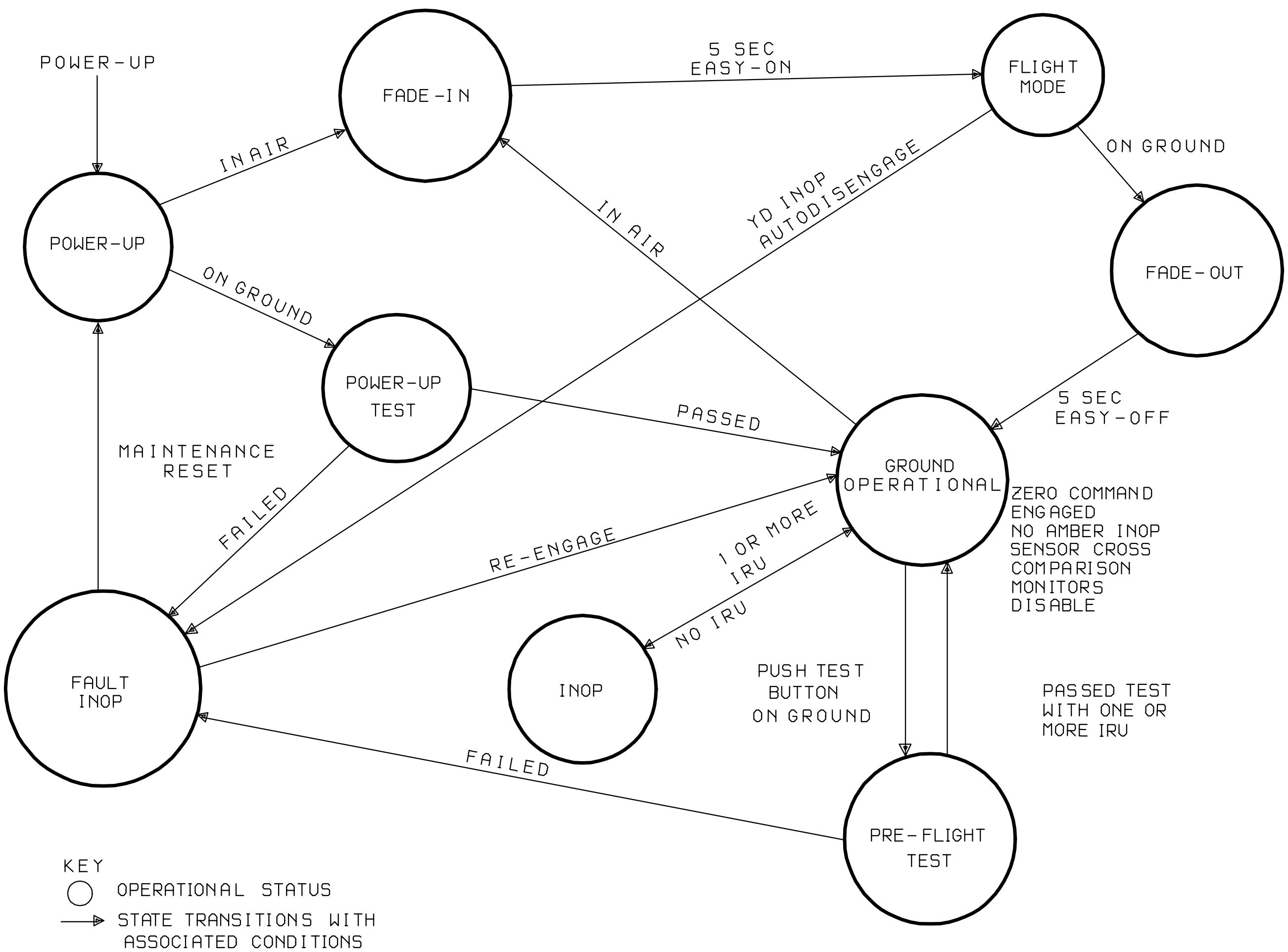
(b) Power-Up

1) The power-up mode is entered when power is supplied to the yaw damper modules (power applied to bus). The CSEU power supply modules must be operative and the bus circuit breakers must be closed. The mode can be entered when on the ground or in the air. If the airplane is on the ground, the system proceeds to the power-up test. If power-up is entered in the air, the system proceeds to the flight mode via a 5 second easy-on function. Upon power-up, the yaw damper modules initialize all interfaces, latches, command, and state variables to a safe state. The YDS monitors are reset; however, the fault ball indicators are latched and do not reset during power-up.

(c) Power-Up Test

1) The power-up test is performed when the YDS is powered up on the ground. The test lasts approximately 20 seconds. The yaw damper INOP amber light is turned on and an EICAS message is displayed during the power-up test. If a fault is detected, the INOP light remains on. The following tests are conducted during the power-up test:

a) The channel fault disengagement test is performed for each channel (control and monitor). It monitors disengage relay status and reports a fault if the relay does not disengage.



Yaw Damper Operational Modes
Figure 7

b) This causes a YDM fault message to be recorded.
c) The servo loop monitor test verifies the ability of the monitor to detect loop faults that cause auto-disengagement.
d) The command coincidence monitor test (for both channels) verifies YDM fault signal latching and automatic disengagement. The YDM fault signal is removed following successful completion of the test.
e) The test resets the YDM fault latch following the successful completion of the preceding tests. It also verifies that the yaw damper is not automatically disengaged and that no fault reset signals are active.
f) The test performs a Random Access Memory (RAM) read/write test.

2) During the power-up test there is no rudder actuator movement. If a test failure is detected, the system enters the FAULT/INOP mode. (a fault message is recorded and the INOP light illuminates).
If the test is successful, the system proceeds to the ground operational mode.

(d) Ground Operational
1) The ground operational mode is entered when one of the following conditions is valid:
a) Successful completion of power-up or preflight test
b) Successful fault recovery and subsequent re-engagement
c) Sensor loss recovery to at least one IRU
d) Upon landing

2) During this mode, the yaw damper control law computes commands. However, the airplane on ground status causes the control law output to convert commands to zero. Even though no output commands are possible, the INOP lamps do not illuminate during the ground operational mode. The sensor cross-comparison monitors are also disabled during this mode. The ground operational mode is exited when the airplane becomes airborne, all IRUs are lost, or when the preflight test is initiated.

(e) Preflight Test

1) The preflight test is entered when the ground operational mode is valid and the YAW DMPR test switch on P61 is activated. The INOP indicator is illuminated throughout the test. The preflight test also initiates the IRS self-test. During the IRS self-test, the four annunciator lights for each IRS channel illuminate. The preflight test has the following automatic test sequence:
   a) Moves the rudder to 3 degrees each direction and returns to center.
   b) The rudder holds in each of the three positions for one second and a position test is performed. The position test determines the difference between the commanded position and the feedback position. The position test fails if the difference exceeds 0.5 degree.
   c) After 10 seconds, the test verifies proper IRU input to the system. If any IRU input is not within limits, the corresponding IRU fault message is recorded.
2) If all the position tests and any one IRU data are satisfactory, the INOP indication is cancelled and the system returns to the ground operation mode. If any position test or all three IRU data tests fail, the INOP indication persists and the auto-disengage relay is actuated. If any IRU channel self-test fails, the FAULT annunciator light for that channel remains illuminated.

(f) Fade-In

1) The fade-in mode is entered from the power-up mode or ground operational mode. The mode is entered if the airplane is airborne when power is first applied or after the airplane becomes airborne. The mode prevents a sudden rudder transient response. The easy-on control law provides a linear rise in gain from zero to one in 5 seconds. When a gain of one is reached, the full flight mode is engaged.

(g) Flight Mode

1) The full flight mode is entered when the airplane is airborne and the fade-in function is completed. The flight mode performs turn coordination and yaw damping. Yaw correction and turn coordination are computed from yaw rate, lateral acceleration, indicated angle of attack, roll angle, roll rate, and true airspeed (or ground speed when true airspeed is not available). The flight mode also provides fault monitoring, self checks, command limiting, degraded operation as required, auto-disengage, and fault indications.

2) When no IRU data is available or a servo loop fault is detected, a YDS fault is latched and the associated channel automatically disengages. Each channel may also be engaged and disengaged by the YDS manual controls. The flight mode automatically exits to the ground operational mode upon landing via the fade-out function.

(h) Fade-Out

1) The fade-out mode is entered directly upon airplane landing. The mode prevents a sudden stop of rudder control. The easy-off control law provides a linear decrease in gain from one to zero in a 5-second time period. The transition begins as soon as a ground state is established.

(i) Fault

1) The fault mode is entered as a result of the following:
   a) Failure of the power-up test
   b) Failure of the preflight test
   c) INOP type failure resulting in automatic disengagement

2) When the fault mode is entered, the associated channel is disengaged, the INOP lamp is illuminated, and the corresponding fault message is recorded.

3) The fault mode exits to the power up mode by a maintenance reset, or to the ground operational mode by automatic re-engagement if no INOP faults remain.

(3) Yaw Damper Schematic (Fig. 8)

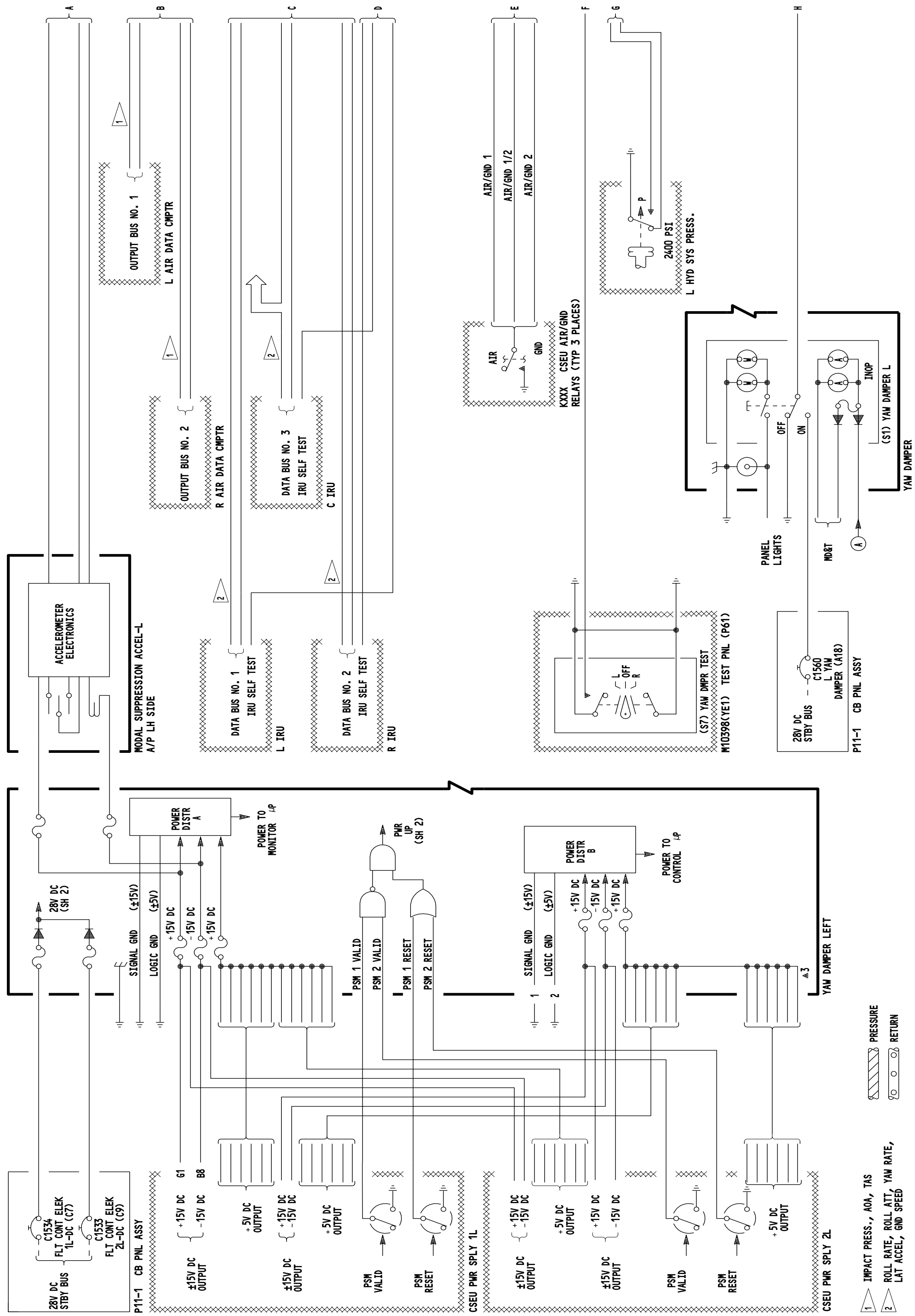
(a) Electrical Power

1) Yaw damper module operating power consists of +5 and ±15 vdc parallel inputs from CSEU power supply modules 1L and 2L.

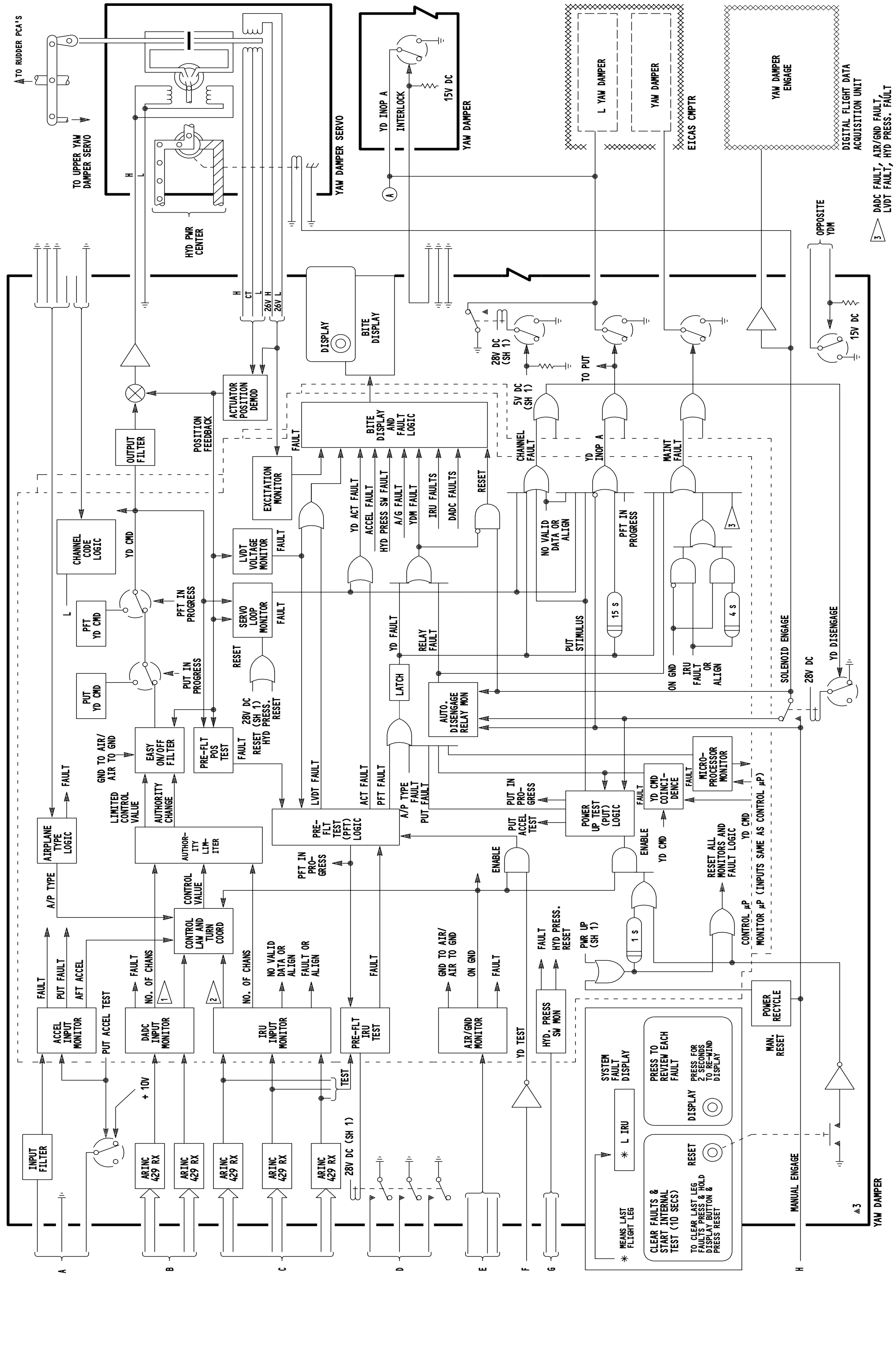
2) The 28 vdc engage solenoid power is supplied by a YAW DAMPER circuit breaker via the ON switch. The 26 vac, 400 Hz servo LVDT excitation is supplied by the CSEU power supply modules.

(b) Inputs

1) Airplane Type Code
   a) The airplane type code is a five wire ground combination with a sixth wire for parity.



Yaw Damper Schematic
Figure 8 (Sheet 1)



Yaw Damper Schematic
Figure 8 (Sheet 2)

b) The parity monitor checks parity and compares type code with the software issue code. This ensures that the correct software is used for the airplane type in which the yaw damper module is installed. If installed correctly, an APL TYPE VALID signal is issued. A detected failure disengages the yaw damper, illuminates the INOP light, provides an EICAS yaw damper message, and the YDM fault message is recorded.

2) Channel Code
   a) The channel code input is a two wire ground/open combination. One pin is grounded for the left installation of the module and the other pin is grounded for right module installation. The monitor checks to ensure only one channel code line is grounded.
   b) A YDM fault message is recorded if a failure is detected.
3) Hydraulic Pressure Switch
   a) The hydraulic pressure switch is a two wire input. One wire is normally grounded at a time, indicating the hydraulic condition (high or low hydraulic pressure state). The failure mode is set if the hydraulic pressure HIGH signal agrees with the hydraulic pressure LOW signal for two seconds.
   b) This causes a HYD SWITCH fault to be recorded.
4) Air/Ground Relays
   a) Air/ground information is received on three status lines from the air/ground relays. The status lines indicate: right gear aft tilt switch level (airplane on ground - through master A/G relay), right gear aft tilt switch level (airplane on ground - direct to YDM), and right gear forward tilt switch level (airplane on ground - direct to YDM).
   b) A triplex majority vote signal selector/monitor is used to determine A/G status. The monitor detects when one of three previously valid inputs is different from the selected value for more than 12 seconds. When the two remaining valid inputs disagree for more than 12 seconds following a first failure, a second failure is detected.
5) ADC
   a) The ADC parameter values for impact pressure, indicated angle of attack, and true airspeed are received on the ARINC 429 bus. Input signal processing includes active monitoring, parity monitoring, sign status matrix monitoring, and value substitution. Excessive signal failure rates or lack of activity result in an ADC input fault.

b) A median value for each parameter is selected for use in the control law. Incoming parameter values are compared with the median value and are fault registered if the comparison is beyond the maximum limits.

6) IRU
   a) The IRU parameter values for roll rate, roll attitude, yaw rate, lateral acceleration, and ground speed are received on the ARINC 429 bus. Input signal processing includes: activity monitoring, parity monitoring, sign status matrix monitoring, and value substitution. Excessive failure rates or lack of activity results in an IRU input fault.
   b) A median value for each parameter is selected for use in the control law. Incoming parameter values are compared with the median value and are registered as faults if the comparison is beyond the maximum limits.
7) Power Supply Modules (PSMs)
   a) The CSEU PSMs (No. 1 and 2 left) each furnish power up reset and PSM valid signals. All four signals must be true to initiate a power-up reset of the yaw damper module.
8) Preflight Test Monitor
   a) The IRU parameter test values are monitored during the preflight test to ensure correct test values. Failure of this test generates an IRU input fault.

(c) Computing Channels
1) Each yaw damper module consists of two independent microprocessors with dissimilar software and instruction sets. Each microprocessor and associated software form a separate computing channel. These are a MONITOR channel and a CONTROL channel. Each microprocessor is programmed with the following software functions:
   a) Executive
   b) Power-up test and preflight test
   c) Input, output, and initialization
   d) Foreground software that includes input monitors, signal selectors, redundancy management, comparison monitors, degraded mode logic, control laws, and annunciation logic.
   e) Background monitors that conduct watchdog monitor tests and Read Only Memory (ROM) tests.
2) Control Law Computations
   a) The control law computations are enabled only if the APL type code is valid. The code ensures that the software in the module is correct for the airplane in which it is installed. The control law continually computes output commands. The output is zero on the ground (except during the preflight test) due to the disabled easy-on/easy-off function.

b) The control law computes output commands based on the data from the ADCs and IRUs. The command outputs from the two microprocessors are compared in coincidence monitors to detect when the two channel outputs do not agree. If non-coincidence is detected, the YDM fault is set.

c) Each computing channel of a yaw damper module performs self monitoring. The channel self monitoring consists of a background ROM check, executive activity monitor, and a watchdog monitor. The watchdog monitor checks for total Central Processing Unit (CPU) loss, operating speed, and erroneous ROM sum check values. If the watchdog monitor detects a failure, the YDM fault is set.

(d) Authority Limiting

1) The control law output command may never exceed ±3 degrees rudder. The authority is further limited as the number of valid ADC and IRU inputs decreases.

(e) Easy-On/Easy-Off

1) The control law output command is routed through a ramp gain schedule stage. The gain is zero when on the ground and one when in-air. With an air/ground relay indication that the airplane has taken off, the gain linearly rises from zero to one in 5 seconds. With an air/ground relay indication that the airplane has landed, the gain linearly decreases from one to zero in a 5-second time period. If an in-air power-up occurs, the fade-in begins immediately following the power-up.

(f) Command Output

1) The control microprocessor output is converted to analog, smoothed, and routed to the yaw damper servo for rudder control. The output command operates the yaw damper electrohydraulic servo valve (EHSV). The EHSV directs hydraulic pressure flow to the servo actuator piston in proportion to the magnitude and sign of the command signal.

2) The EHSV receives hydraulic pressure when the engage solenoid is actuated by 28 vdc from the control panel ON switch (via the engage relay in the yaw damper module). The engage relay is closed unless an auto-disengage signal is generated, or there is a loss of operating power to the yaw damper module.

3) A Linear Variable Differential Transducer (LVDT) is physically attached to the actuator piston. The LVDT uses 400 Hz, 26 vac power for excitation. The LVDT output voltage reflects the position of the servo actuator piston. This data is routed back to the yaw damper module to complete the servo loop.

4) The LVDT feedback signal and the command output signal are compared in the yaw damper module. If the LVDT signal does not track the command signal, the servo fault latch is set.

5) This causes a YD ACT fault to be recorded.

6) The common mode sum voltage from the LVDT is also monitored. If it is less than 3 vrms for more than 2 seconds, an LVDT fault message will be shown.

(g) INOP Indication
1) The yaw damper control panel indicates INOP with the following conditions present:
a) During the power-up test
b) During the preflight test
c) With disengagement of a yaw damper channel (yaw damper servo solenoid valve not actuated)
2) When yaw damper INOP conditions are valid, a caution annunciation (L or R YAW DAMPER) appears on the EICAS display unit (Ref 31-41-00).
3) Disengagement of a yaw damper channel occurs under the following conditions:
a) Manual disengagement from the control panel ON switch
b) Auto-disengagement resulting from: any YDM fault, power-up test failure, preflight test actuator failure, preflight test LVDT failure, no IRU data valid, or a servo fault during normal operation.
c) The interlock between modules is broken by one yaw damper module not being installed.

<u>NOTE</u>: One YDM must be installed to convert the opposite YDM open circuit to a ground for the INOP lamp.

(h) Each yaw damper also sends a discrete signal (yaw damper engage) to the digital flight data acquisition unit (DFDAU).
(i) Maintenance Reset
1) The yaw damper module front panel contains a RESET switch. Pressing the switch initiates the power-up test if the airplane is on the ground, pushing the RESET button resets all the yaw damper module monitors and control logic.
(j) Yaw Damper Module Auto-reset.
1) The YDM fault latch is responsive to transient fault recovery of YDM fault monitors. YDM fault monitors capable of transient fault recovery are as follows:
a) Airplane type code
b) Yaw damper command coincidence
c) Activity monitor
d) Monitor or control watchdog monitor fault
2) The YDM fault latch resets if all YDM monitors exhibit no failures for 10 seconds continuously and within 15 seconds of detecting a YDM fault. Autoreset is limited to a maximum of four resets per flight. There is no limit on autorest when the airplane is on ground.
3) The following faults are incapable to automatic recovery:
a) Power-up reset YDM fault
b) Preflight test YDM fault
c) Background Read Only Memory (ROM) fault

(4) Yaw Damper Control Law Schematic (Fig. 9)

(a) Yaw Damping Command

1) Lateral acceleration from the IRUs is filtered and corrected for the effect of yaw rate and IRU displacement from the center of gravity. This value is gain scheduled by true airspeed to obtain the sideslip angle rate. The yaw rate required for roll coordination is computed from roll attitude and true airspeed. The actual yaw rate is subtracted from this value to give the total yaw rate due to roll that must be damped. Roll rate is multiplied by a body angle of attack factor to form the third damping command factor. The above factors are summed and gain scheduled by impact pressure to generate the yaw damping command (CMD).

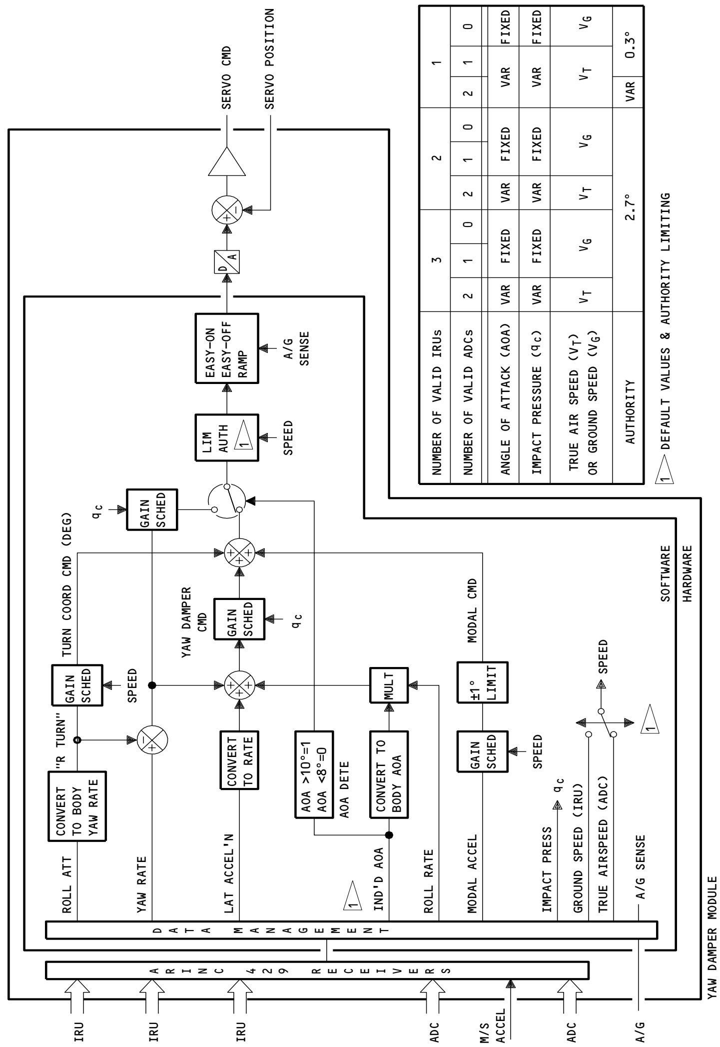
2) Sideslip angle rate is synthesised by summing together yaw rate, lateral acceleration scheduled by true airspeed, roll attitude scheduled by true airspeed, and roll rate multiplied by body angle of attack. The lateral acceleration signal from the IRU's, is filtered and corrected for the effect of yaw rate and IRU displacement from center of gravity. Sideslip angle rate is gain scheduled by impact pressure to generate the yaw damping command.

(b) Turn Coordination Command

1) The turn coordination command is derived from the roll attitude gain scheduled by true airspeed. This value is summed with the yaw damping CMD to form the output command.

(c) Output Command Processing

1) The output command is authority limited to fixed or variable amounts, depending on the availability of ADC and IRU inputs. Authority is in accordance with the table shown. The variable authority is programmed as a function of true airspeed. True airspeed is used when both ADCs are valid and at least one IRU is valid. For other combinations, ground speed is substituted for true airspeed. The authority is either variable, ±3.0 degrees, or ±0.3 degrees rudder as shown in the table. Commands are applied through an easy-on/easy-off gain schedule (5 second, ramp) when the air/ground discrete indicates in-air (takeoff), or on-ground (landing), respectively.



NUMBER OF VALID IRUs	3			2			1		
NUMBER OF VALID ADCs	2	1	0	2	1	0	2	1	0
ANGLE OF ATTACK (AOA)	VAR	FIXED		VAR	FIXED		VAR		FIXED
IMPACT PRESSURE (q_c)	VAR	FIXED		VAR	FIXED		VAR		FIXED
TRUE AIR SPEED (V_T) OR GROUND SPEED (V_G)	V_T	V_G		V_T	V_G		V_T		V_G
AUTHORITY	2.7°						VAR	0.3°	

1 DEFAULT VALUES & AUTHORITY LIMITING

Yaw Damper Control Law Schematic
Figure 9

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2) Default values for indicated angle of attack (AOA), impact pressure, and true airspeed are used depending upon ADC and IRU availability. As an example, with one ADC and two IRUs available, indicated AOA becomes a constant 7.88 degrees, impact pressure becomes a constant 270 millibars, and ground speed is substituted for true airspeed.

(5) ADC Input Signal Monitoring (Fig. 10)

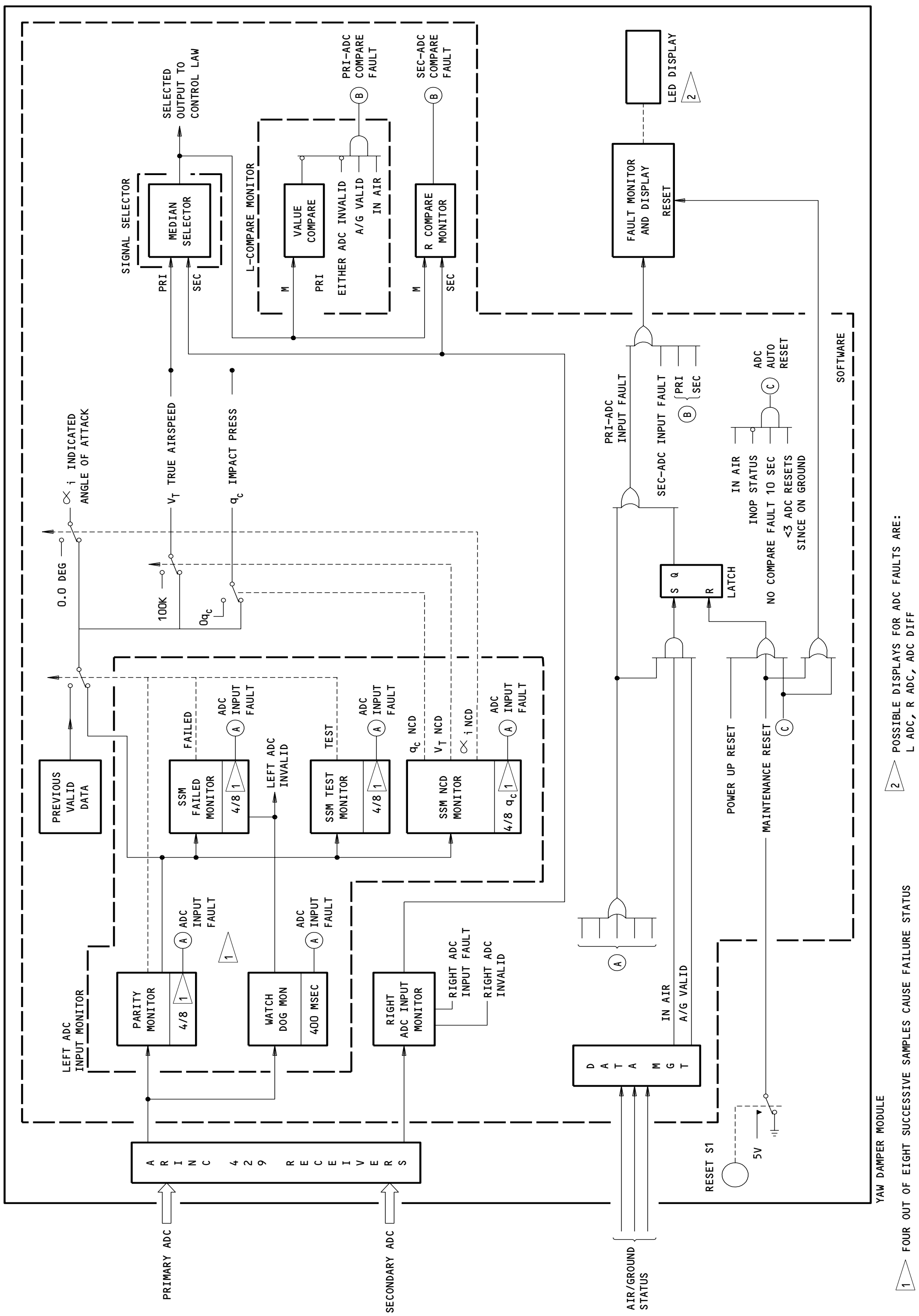
(a) Input Monitor

1) The ADC input monitor consists of a watchdog monitor, parity monitor, and Sign Status Matrix (SSM) monitor. The watchdog monitor transmits an input fault signal if no air data information is received for 400 msec when required by the YDM. The parity monitor transmits a fault signal if a parity error is detected in 4 out of any 8 successive samples. For all samples with parity error, the incoming values are discarded and the last previous valid values are used.

2) The SSM monitor provides an input fault signal if incoming data is indicated as FAILED for 4 out of 8 successive samples. For all samples indicating FAILED, the incoming values are discarded and the last previous valid value is used. A fault signal is also provided if impact pressure is indicated as No Computed Data (NCD) for 4 out of 8 successive samples. For all samples indicating NCD, the incoming value is discarded and the last previous valid value is used. If any indicated angle of attack sample is indicated as NCD, 0.0 degrees is substituted for the sample value. If a true airspeed sample is indicated as NCD, 100 knots is substituted for the sample value. If incoming data is indicated as TEST for 4 out of 8 successive samples, a fault signal is transmitted. For all samples indicating TEST, the incoming values are discarded and the last previous valid value is used.

(b) Signal Selector

1) A median value of parameters from both ADCs is selected for use in the control law computations. Incoming parameter values are compared to the respective signal selector output. If the difference is greater than the specified values shown below for one second, the compare fault is set. (Airplane must be 200 milliseconds after the ground to air transition, A/G discrete valid, and neither ADC invalid.) Following a ADC fault, data from the remaining valid ADC may be used.



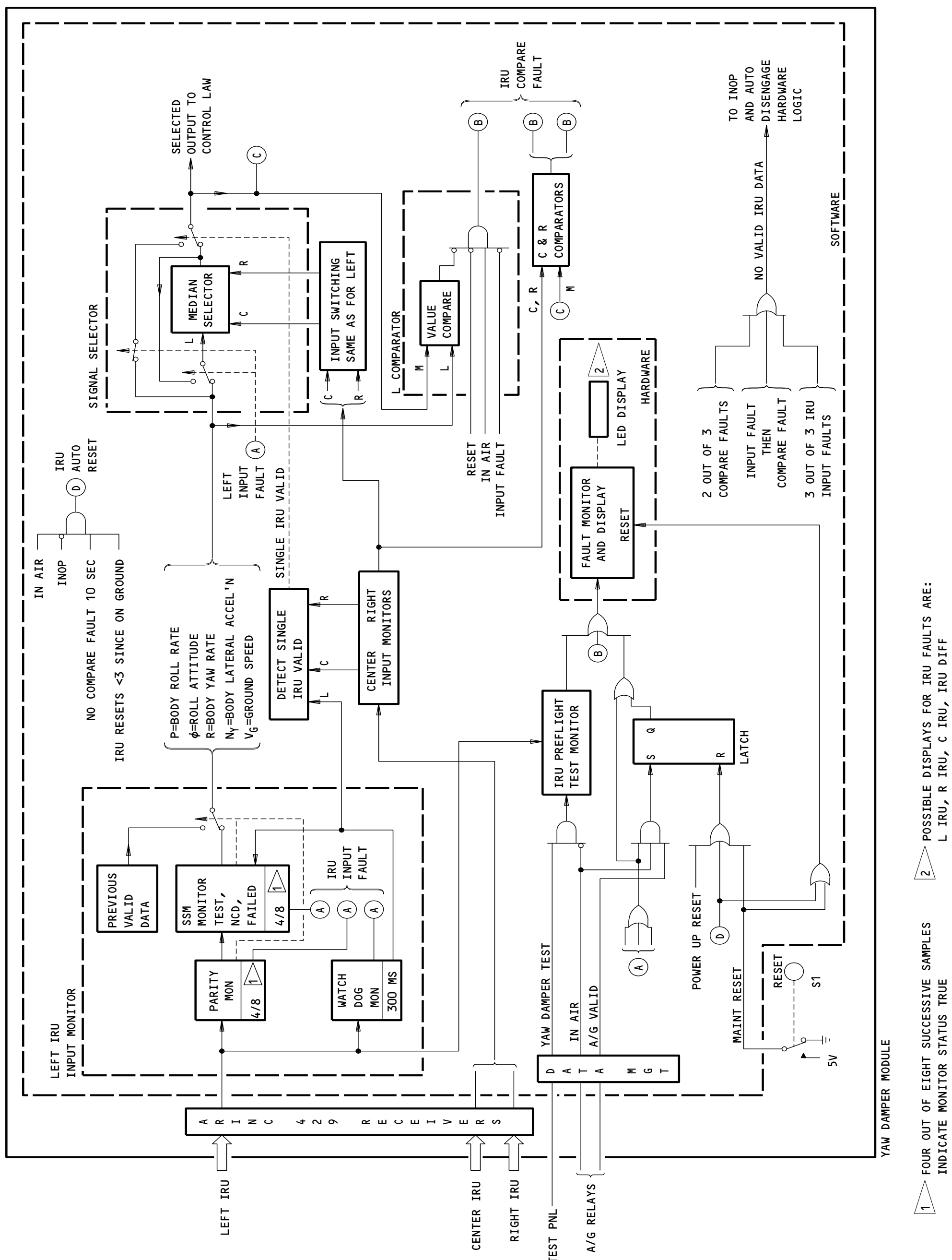
ADC Input Signal Processing
Figure 10

2) The maximum deviation allowed between the ADC input values and the median ADC value follows:
   a) True airspeed - 60 knots
   b) Angle of attack - 5 degrees
   c) Impact pressure - 27 millibars

(c) Fault Annunciation
1) Any ADC input or comparison fault causes an ADC fault message to be recorded.
2) If the airplane is in the air and the air/ground (A/G) discrete is valid, input faults are latched. The latch is reset by the power-up reset, maintenance reset, or ADC automatic reset.
3) The ADC auto-reset is generated with the airplane in air and the yaw damper not in an INOP status. If sensor data within monitor threshold is detected for 10 seconds, the ADC fault latch is reset, provided there have been no more than four resets since the airplane was last on ground. Maintenance reset is generated by pressing the RESET switch on the yaw damper module.

(6) IRU Input Signal Processing (Fig. 11)
(a) Input Monitor
1) The IRU input monitor consists of a watchdog monitor, parity monitor, preflight test monitor, and Sign Status Matrix (SSM) monitor. The watchdog monitor input fault is set when no IRU information is received for 300 msec when required by the YDM. The parity monitor input fault is set if a parity error is detected in four out of any 8 successive samples. For all samples with a parity error, the incoming values are discarded and the last previous valid values are used.
2) The SSM monitor input fault is set when incoming data is indicated as FAILED, TEST or NCD for 4 out of 8 successive samples. For all samples indicating FAILED, TEST or NCD, the incoming values are discarded and the last previous valid values are used. The preflight test inputs monitor fault is set if the test values from the IRUs are incorrect. The monitor is active only during the airplane preflight test.

(b) Signal Selector
1) A median value of parameters from all three IRUs is selected for use in control law computations. If one IRU fails, that input is not used. The two remaining IRU inputs and the last valid median value are used for median selection. With only one IRU remaining valid, median selection is avoided and that single input is used. Each parameter input value is compared to the selected median value. Differences exceeding specified amounts for more than one second result in a comparison fault latch and the affected IRU being declared failed. (The airplane must be 200 milliseconds after the ground to air transition, no input fault for that IRU and A/G discrete valid).



IRU Input Signal Processing
Figure 11

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2) The maximum deviation allowed between IRU input values and the median IRU value follows:
   a) Roll attitude - 5 degrees
   b) Body axis roll rate - 2.5 deg/sec
   c) Body axis yaw rate - 2.5 deg/sec
   d) Body axis lateral acceleration - 0.1 g
   e) Ground speed - 60 knots

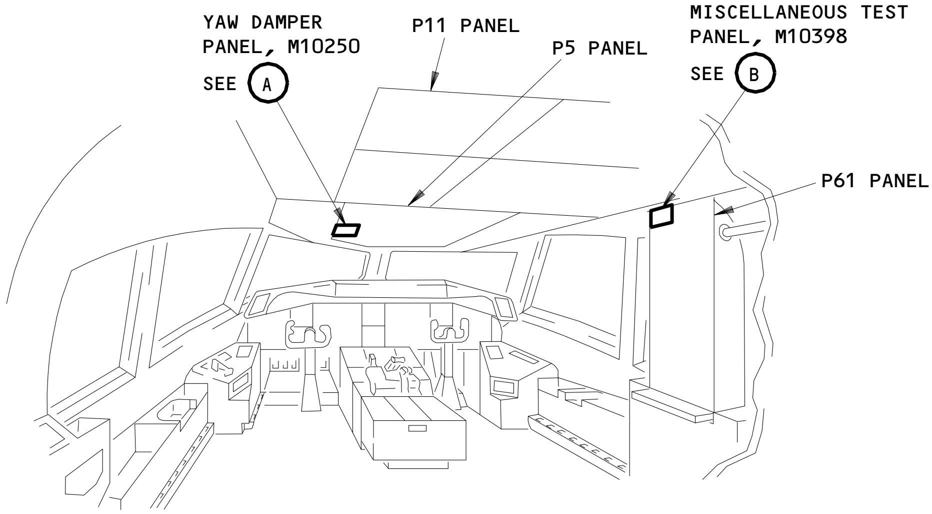
(c) Fault Annunciation
1) If any IRU input, comparison, or preflight test monitor fault is detected, the corresponding IRU fault message is recorded.
2) If the airplane is airborne, input faults are latched. The monitors are reset by the power-up reset, IRU auto-reset, or the maintenance reset.
3) IRU data is declared unacceptable if:
   a) Two out of three IRUs have comparison faults
   b) An input fault and a comparison fault are detected.
   c) All IRUs have input faults
4) When IRU data is declared unacceptable, the yaw damper channel goes INOP and is auto-disengaged. An IRU auto-reset is generated with the airplane in-air and the yaw damper not in an INOP status. If sensor data within monitor threshold is detected for 10 seconds, the IRU fault latch is reset, provided there have been no more than four resets since the airplane was on the ground. The maintenance reset is generated by the YDM front panel RESET switch.

YAW DAMPER SYSTEM

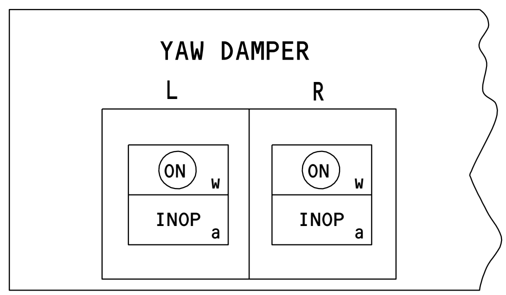
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	REFERENCE
ACCELEROMETER - MODAL SUPPRESSION L, M10734	2	1	822, AFT CARGO DOOR, AFT CARGO COMPT CEILING	22-21-05
ACCELEROMETER - MODAL SUPPRESSION R, M10735	2	1	822, AFT CARGO DOOR, AFT CARGO COMPT CEILING	22-21-05
CIRCUIT BREAKERS -	1		FLT COMPT, P11	*
CSEU 1L AC (OR FLT CTRL ELEC 1L AC), C1538		1	11C6	*
CSEU 1L DC (OR FLT CTRL ELEC 1L DC), C1534		1	11C7	*
CSEU 1R AC (OR FLT CTRL ELEC 1R AC), C1536		1	11G17	*
CSEU 1R DC (OR FLT CTRL ELEC 1R DC), C1531		1	11G18	*
CSEU 2L AC (OR FLT CTRL ELEC 2L AC), C1537		1	11C8	*
CSEU 2L DC (OR FLT CTRL ELEC 2L DC), C1533		1	11C9	*
CSEU 2R AC (OR FLT CTRL ELEC 2R AC), C1535		1	11G27	*
CSEU 2R DC (OR FLT CTRL ELEC 2R DC), C1532		1	11G28	*
YAW DAMPER L, C1560		1	11A18	*
YAW DAMPER R, C1561		1	11F34	*
COMPUTERS - (31-41-00/101)				
EICAS L, M10181				
EICAS R, M10182				
COMPUTERS - (34-12-00/101)				
AIR DATA L, M100				
AIR DATA R, M101				
MODULES - (27-09-00/101)				
LEFT POWER SUPPLY 1, M536				
LEFT POWER SUPPLY 2, M537				
RIGHT POWER SUPPLY 1, M538				
RIGHT POWER SUPPLY 2, M539				
MODULE - YAW DAMPER L, M522	2	1	119BL, MAIN EQUIP CTR, E3-1	22-21-04
MODULE - YAW DAMPER R, M523	2	1	119BL, MAIN EQUIP CTR, E4-1	22-21-04
PANEL - YAW DAMPER, M10250	1	1	FLT COMPT, P5	22-21-01
RELAYS - (31-01-36/101)				
AIR/GND SYS NO. 1, K135				
AIR/GND SYS NO. 1, K10384				
RELAYS - (31-01-37/101)				
AIR/GND SYS NO. 2, K215				
AIR/GND SYS NO. 2, K10387				
SERVO - LEFT YAW DAMPER, M509	2	1	324BL, VERT STAB (APL L SIDE)	22-21-02
SERVO - RIGHT YAW DAMPER, M510	2	1	324BL, VERT STAB (APL R SIDE)	22-21-02
SWITCHES - (29-31-00/101)				
SYS C HYDRAULIC PRESSURE, S10002				
SYS L EDP CONTROL PRESSURE, S27				
SWITCH - YAW DMPR TEST, YPHS7	1	1	FLT COMPT, P61, MISC TEST PANEL, M10398	*
UNITS - (34-21-00/101)				
INTERTIAL REFERENCE C, M160				
INTERTIAL REFERENCE L, M159				
INTERTIAL REFERENCE R, M161				
VALVE - YDS ELECTROHYDRAULIC SERVO	2	2	324BL, VERTICAL STABILIZER, EACH YAW DAMPER SERVO	22-21-03
VALVE - YDS ELECTROHYDRAULIC SOLENOID	2	2	324BL, VERTICAL STABILIZER, EACH YAW DAMPER SERVO	22-21-03

* SEE THE WDM EQUIPMENT LIST

Yaw Damper System - Component Index
Figure 101

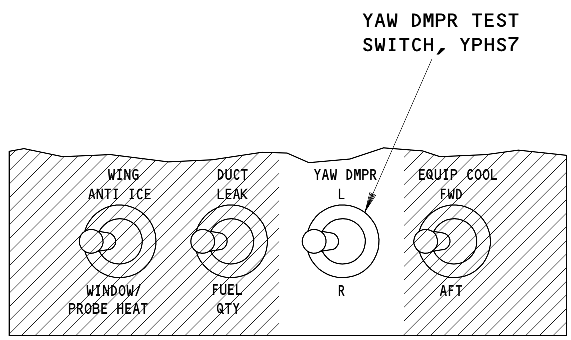


FLIGHT COMPARTMENT



YAW DAMPER PANEL, M10250

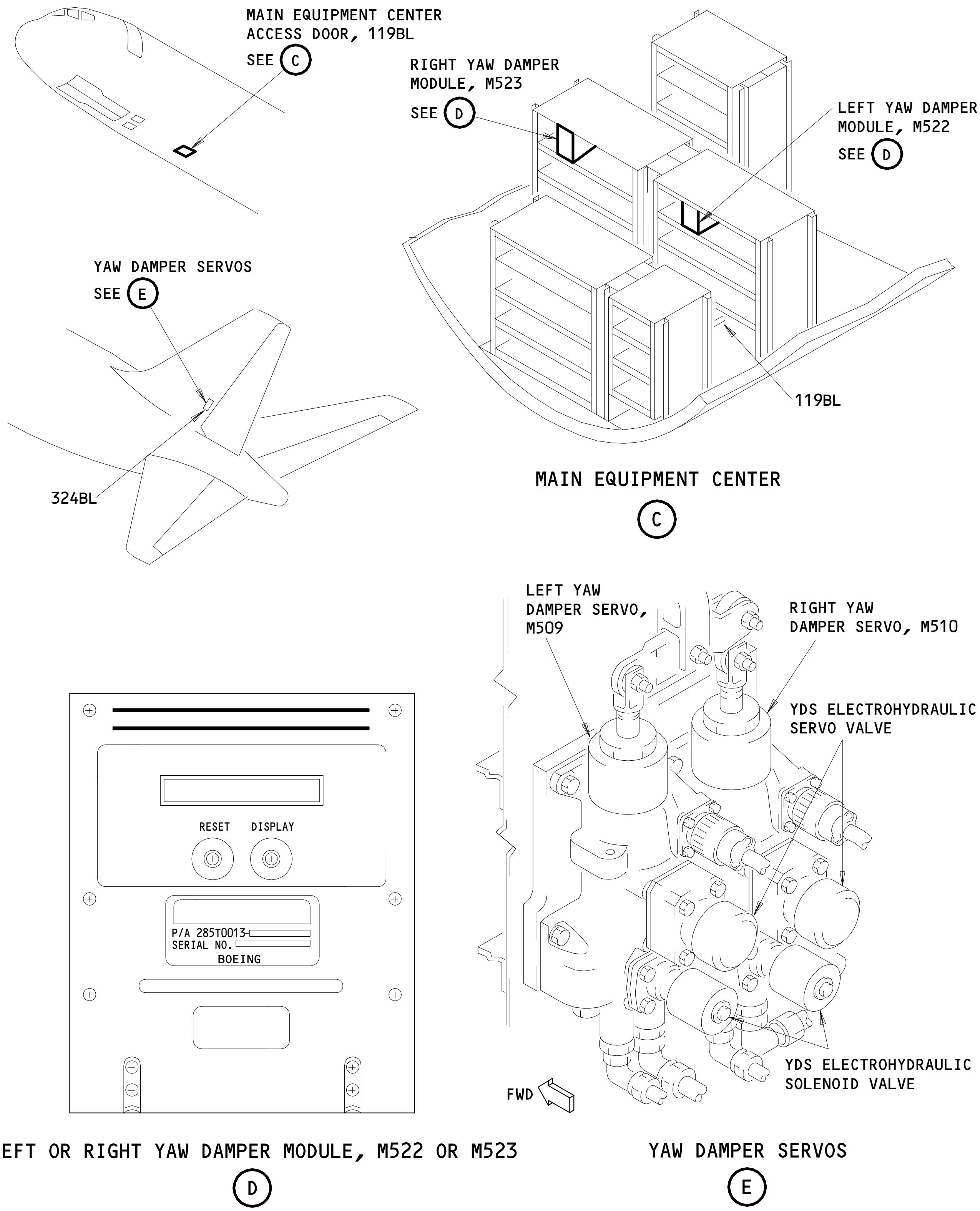
A



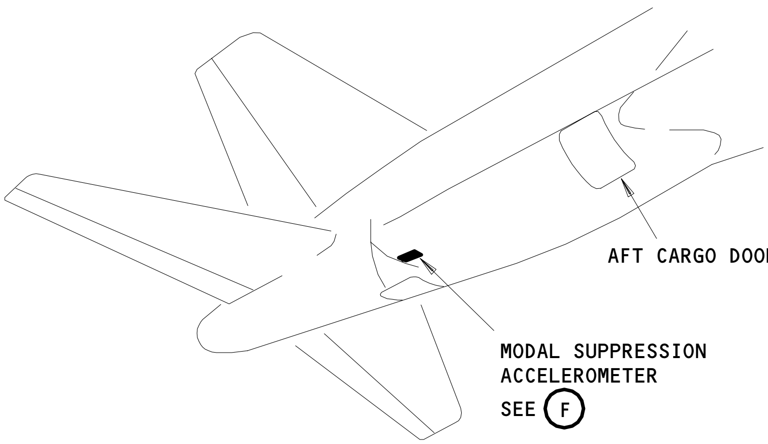
MISCELLANEOUS TEST PANEL, M10398

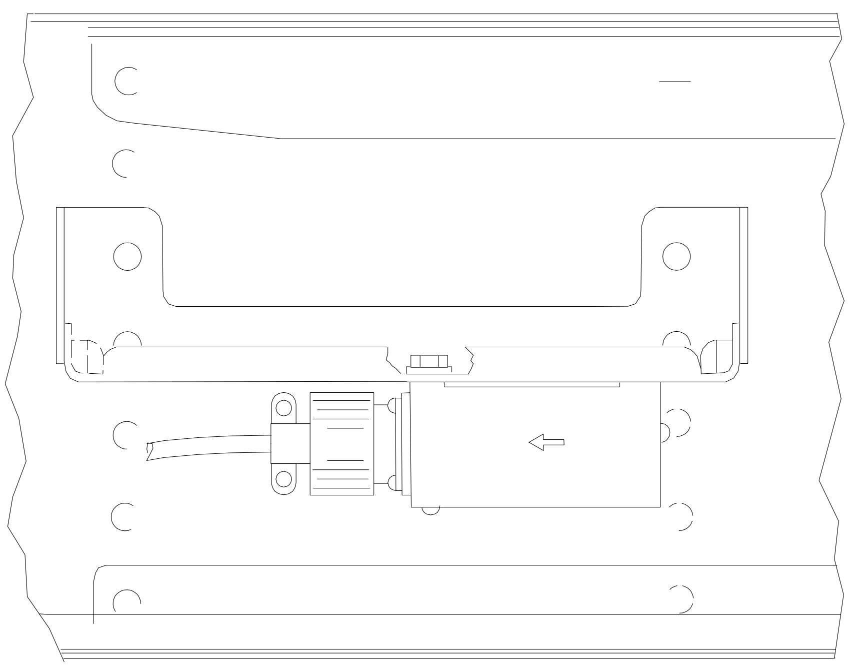
B

Yaw Damper System - Component Location
Figure 102 (Sheet 1)



Component Location
Figure 102 (Sheet 2)





MODAL SUPPRESSION ACCELEROMETER
F

Component Location
Figure 102 (Sheet 3)

YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General
   A. The paragraphs that follow give an operational test and a system test of the yaw damper system. The operational test and system test do not use external test equipment. The operational test gives a preflight test of the yaw damper system. The system test examines the yaw damper system interfaces to make sure the system operates correctly.

TASK 22-21-00-715-005

2. AIRPLANES WITH YDM -101 THRU -126;
   Operational Test - Yaw Damper System
   A. General
      (1) The operational test uses only the built-in test features of the yaw damper system. To start the operational test, push the YAM DMPR test switch on the P61 panel. The test makes sure that the yaw damper modules, the yaw damper servos, and the switches on the yaw damper panel operate correctly.
   B. References
      (1) 24-22-00/201, Electrical Power - Control
      (2) 27-21-00/501, Rudder and Rudder Trim Control System
      (3) 27-61-00/201, Spoiler/Speedbrake Control System
      (4) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (5) 31-41-00/501, Engine Indication and Crew Alerting System
      (6) 32-09-02/201, Air/Ground Relays
      (7) 33-16-00/501, Master Dim and Test
      (8) 34-12-00/501, Air Data Computing System
      (9) 34-21-00/501, Inertial Reference System
   C. Access
      (1) Location Zones
          119/120 Main Equipment Center
          211/212 Control Cabin

      (2) Access Panel
          119BL Main Equipment Center

   D. Prepare for Test

      S 865-006
      (1) Supply electrical power (Ref 24-22-00).

S 865-007

(2) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
   (a) 11A18, YAW DAMPER LEFT
   (b) 11C6, CSEU 1L AC
   or
   FLT CONT ELEC 1L AC
   (c) 11C7, CSEU 1L DC
   or
   FLT CONT ELEC 1L DC
   (d) 11C8, CSEU 2L AC
   or
   FLT CONT ELEC 2L AC
   (e) 11C9, CSEU 2L DC
   or
   FLT CONT ELEC 2L DC
   (f) 11F34, YAW DAMPER RIGHT
   (g) 11G17, CSEU 1R AC
   or
   FLT CONT ELEC 1R AC
   (h) 11G18, CSEU 1R DC
   or
   FLT CONT ELEC 1R DC
   (i) 11G27, CSEU 2R AC
   or
   FLT CONT ELEC 2R AC
   (j) 11G28, CSEU 2R DC
   or
   FLT CONT ELEC 2R DC

S 865-001

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 865-008

(4) Make sure that these circuit breakers on the P11 panel are closed:
   (a) 11H17, FLT CONT SHUTOFF TAIL LEFT
   (b) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 985-338

(5) Put the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the ON position.

S 865-339

(6) Make sure these systems operate:
   (a) Rudder and Rudder Trim Control System (Ref 27-21-00)
   (b) Air Data Computing System (Ref 34-12-00)
   (c) Inertial Reference System (aligned and in NAV mode) (Ref 34-21-00)
   (d) Air/Ground Relays (Ref 32-09-02)
   (e) Master Dim and Test System (Ref 33-16-00)
   (f) Engine Indication and Crew Alerting System (Ref 31-41-00).
   (g) Select STATUS on EICAS display select panel on panel P9.

E. Test Yaw Damper System Operation

S 985-010

(1) Push the RESET button on the left and right yaw damper modules. Stop for 30 seconds before you continue.

S 985-011

(2) Push the left yaw damper engage switch on the yaw damper panel and make sure that the ON light comes on.
   (a) Make sure that the left yaw damper INOP light is off.

S 985-013

(3) Put the YAW DMPR test switch on the right side panel, P61, in the L position, then put the switch back to the center.
   (a) Make sure that the left yaw damper INOP light comes on.

      NOTE: The right yaw damper INOP light can also come on.
   (b) Make sure that the rudder position indicator on the lower EICAS display (Ref 31-41-00) shows this sequence of rudder movement in less than 10 seconds:
      1) the rudder moves approximately 3 degrees trailing edge right
      2) the rudder moves approximately 3 degrees trailing edge left
      3) the rudder goes back to the center.
   (c) Make sure that the left yaw damper INOP light goes off in less than 15 seconds.

S 985-014

(4) Push the right yaw damper engage switch and make sure that the ON light comes on.
   (a) Make sure that the right yaw damper INOP light is off.

S 985-016

(5) Put the YAW DMPR test switch in the R position then put the switch back to the center.

(a) Make sure that the right yaw damper INOP light comes on.

NOTE: The left yaw damper INOP light can also come on.

(b) Make sure that the rudder position indicator on the lower EICAS display (Ref 31-41-00) shows this sequence of rudder movement in less than 10 seconds:

1) the rudder moves approximately 3 degrees trailing edge right

2) the rudder moves approximately 3 degrees trailing edge left

3) the rudder goes back to the center.

(c) Make sure that the right yaw damper INOP light goes off in less than 15 seconds.

S 985-019

(6) Push the DISPLAY button on the left and right yaw damper modules.

S 285-020

(7) Make sure that the message NO FAULTS or NO FAULT NOW shows on the display of the two modules.

S 865-347

(8) Push and release the left and right yaw damper engage switches to disengage the yaw damper system.

F. Put the Airplane Back to Its Initial Condition.

S 865-023

(1) Remove power from the left, center and right hydraulic systems (Ref 29-11-00).

S 865-024

(2) Remove electrical power if it is not necessary (Ref 24-22-00).

S 865-922

(3) Put the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel in the OFF position.

S 845-368

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

TASK 22-21-00-735-143

3. System Test - Yaw Damper System

A. General

(1) The system test makes sure that these system interfaces operate correctly:
   (a) Modal Suppression Accelerometer
   (b) Yaw Damper servos
   (c) Inertial Reference Units
   (d) Air Data Computers
   (e) EICAS
   (f) Air/Ground relays
   (g) Hydraulic pressure switches and sources
   (h) System electrical power sources

(2) The system test also makes sure that these functions operate correctly:
   (a) Channel fault disengagement
   (b) Disengage relay operation
   (c) Servo loop fault detection
   (d) Monitor operation

B. References

(1) 24-22-00/201, Electrical Power - Control
(2) 27-21-00/501, Rudder and Rudder Trim Control System
(3) 27-61-00/201, Spoiler/Speedbrake Control System
(4) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

(5) 31-41-00/501, Engine Indication and Crew Alerting System
(6) 32-09-02/201, Air/Ground Relays
(7) 33-16-00/501, Master Dim and Test
(8) 34-12-00/501, Air Data Computing System
(9) 34-21-00/501, Inertial Reference System

C. Access

(1) Location Zones

119/120 Main Equipment Center
211/212 Control Cabin

(2) Access Panel

119BL Main Equipment Center

D. Prepare for Test

S 715-146

(1) Do the Prepare for Test procedures in the Yaw Damper Systems Operational Test.

E. Power-Up-Test

S 985-147

(1) Push the two YAW DAMPER-ON switch/lights and make sure the ON light on the yaw damper panel comes on.

S 845-380

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-148

(3) Open for 1 second and close these circuit breakers on the P11 panel:

(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC

(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 285-149
(4) Make sure that the YAW DAMPER INOP (L, R) lights on the overhead panel, P5, come on momentarily yellow and then go off within 30 seconds.

S 985-150
(5) Push the RESET button on the left and right yaw damper modules.

S 285-151
(6) Stop for a minimum of 30 seconds then make sure that the YAW DAMPER INOP L, R lights are off.

S 985-152
(7) Push the DISPLAY button on the left YAW DAMPER module.

S 285-153
(8) Make sure the the message NO FAULT is shown on the left YAW DAMPER module fault display.

S 985-154
(9) Push the DISPLAY button on the right YAW DAMPER module.

S 285-155
(10) Make sure that the message NO FAULTS is shown on the right YAW DAMPER module fault display.

F. Power Supply Interface Test

S 845-355

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-371

(2) Open these circuit breakers on the P11 panel:
  (a) 11G17, CSEU 1R AC
  or
  FLT CONT ELEC 1R AC
  (b) 11G27, CSEU 2R AC
  or
  FLT CONT ELEC 2R AC

S 285-156

(3) Make sure the YAW DAMPER INOP R light is on.

S 865-157

(4) Close this circuit breaker on the P11 panel:
  (a) 11G27, CSEU 2R AC
  or
  FLT CONT ELEC 2R AC

S 285-158

(5) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP R light is off.

S 845-356

<u>WARNING</u>: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-159

(7) Open this circuit breaker on the P11 panel:
  (a) 11G27, CSEU 2R AC
  or
  FLT CONT ELEC 2R AC

S 285-160

(8) Make sure that the YAW DAMPER INOP R light is on.

S 865-161

(9) Close this circuit breaker on the P11 panel:
  (a) 11G17, CSEU 1R AC
  or
  FLT CONT ELEC 1R AC

S 285-162

(10) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP R light is off.

S 845-357

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(11) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-163

(12) Close this circuit breaker on the P11 panel:
- (a) 11G27, CSEU 2R AC
  or
  FLT CONT ELEC 2R AC

S 865-164

(13) Open these circuit breakers on the P11 panel:
- (a) 11C6, CSEU 1L AC
  or
  FLT CONT ELEC 1L AC
- (b) 11C8, CSEU 2L AC
  or
  FLT CONT ELEC 2L AC

S 285-165

(14) Make sure that the YAW DAMPER INOP L light is on.

S 865-166

(15) Close this circuit breaker on the P11 panel:
- (a) 11C8, CSEU 2L AC
  or
  FLT CONT ELEC 2L AC

S 285-167

(16) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP L light is off.

S 845-358

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(17) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 985-168
(18) Put the EICAS computer select switch, on the pilots EICAS DISPLAY SELECT PANEL, in the L position.

S 865-169
(19) Open this circuit breaker on the P11 panel:
(a) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

S 285-170
(20) Make sure that the YAW DAMPER INOP L light is on.

S 865-171
(21) Close this circuit breaker on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC

S 285-172
(22) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP L light is off.

S 865-173
(23) Close this circuit breaker on the P11 panel:
(a) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

G. 28V DC Power Test

S 845-359

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-174
(2) Open these circuit breakers on the P11 panel:
(a) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(b) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

S 285-175
(3) Make sure that the YAW DAMPER INOP R light is on.

S 865-176
(4) Close this circuit breaker on the P11 panel:
(a) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC

S 285-177
(5) Make sure that the YAW DAMPER INOP R light is off.

S 845-360

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-178
(7) Open this circuit breaker on the P11 panel:
(a) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC

S 285-179
(8) Make sure that the YAW DAMPER INOP R light is on.

S 865-180
(9) Close this circuit breaker on the P11 panel:
(a) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

S 285-181
(10) Make sure that the YAW DAMPER INOP R light is off.

S 845-361

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(11) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-182
(12) Close this circuit breaker on the P11 panel:
(a) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC

S 865-183
(13) Open these circuit breakers on the P11 panel:
(a) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC
(b) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

S 285-184
(14) Make sure that the YAW DAMPER INOP L light is on.

S 865-185
(15) Close this circuit breaker on the P11 panel:
(a) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

S 285-372
(16) Make sure that the YAW DAMPER INOP L light is off

S 845-362

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(17) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-187
(18) Open this circuit breaker on the P11 panel:
(a) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

S 285-188
(19) Make sure that the YAW DAMPER INOP L light is on.

S 865-189
(20) Close this circuit breaker on the P11 panel:
(a) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC

S 285-190
(21) Make sure that the YAW DAMPER INOP L light is off.

S 865-191
(22) Close this circuit breaker on the P11 panel:
(a) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

H. Yaw Damper Engage Test

S 865-192
(1) Open this circuit breaker on the P11 panel:
(a) 11A18, YAW DAMPER LEFT

S 985-193
(2) Put the EICAS computer select switch, on the pilots EICAS DISPLAY SELECT PANEL, in the L position.

S 285-194
(3) Make sure that the YAW DAMPER INOP L light is on, and that the yellow L YAW DAMPER message is showm on the upper EICAS display.

S 985-195
(4) Put the EICAS computer select switch in the R position.

S 285-196
(5) Make sure that the upper EICAS display shows the L YAW DAMPER message in yellow.

S 865-197
(6) Close this circuit breaker on the P11 panel:
(a) 11A18, YAW DAMPER LEFT

S 985-198
(7) Stop for a minimum of 30 seconds then make sure that YAW DAMPER INOP L light is off and L YAW DAMPER message is not shown on upper EICAS display.

S 985-199
(8) Put the EICAS computer select switch in the L position.

S 285-200
(9) Make sure that the L YAW DAMPER message is not shown on the upper EICAS display.

S 865-201

(10) Open this circuit breaker on the P11 panel:
  (a) 11F34, YAW DAMPER RIGHT

S 285-202

(11) Make sure that the YAW DAMPER INOP R light is on, and that the yellow R YAW DAMPER message is shown on the upper EICAS display.

S 985-203

(12) Put the EICAS computer select switch in the R position.

S 285-204

(13) Make sure that the upper EICAS display shows the R YAW DAMPER message in yellow.

S 865-205

(14) Close this circuit breaker on the P11 panel:
  (a) 11F34, YAW DAMPER RIGHT

S 285-206

(15) Stop for a minimum of 30 seconds then make sure that the YAW DAMPER INOP R light is off, and the R YAW DAMPER message is not shown on the upper EICAS display.

S 985-207

(16) Put the EICAS computer select switch in the L position.

S 285-208

(17) Make sure that the R YAW DAMPER message is not shown on the upper EICAS display.

I. Air/Ground System Interface Test

<u>NOTE</u>: There shall be no Rudder Pedal movement or Rudder Trim Switch operation during this test.

S 865-381

<u>WARNING</u>: PREPARE THE SAFETY-SENSITIVE SYSTEMS FOR THE AIR MODE BEFORE YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS. IN THE AIR MODE, MANY OF THE AIRPLANE SYSTEMS CAN OPERATE AND CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(1) Prepare the safety sensitive systems for air mode simulation (Ref 32-09-02/201).

S 865-783
(2) Make sure that the AOA Vanes are set to equal postions.

NOTE: If the AOA Vane's are not equal:
- The "YAW DAMPER" EICAS maintenance message will show.
- A "ADC DIF" YDM fault meassage will occur.

S 985-385
(3) Push the ECS/MSG switch on the EICAS Maintenance Panel (P61).

S 285-386
(4) Make sure that the "YAW DAMPER" message is not displayed on the Lower EICAS Display - Maintenance Page.

S 285-387
(5) Make sure that the "L YAW DAMPER" and "R YAW DAMPER" messages are not displayed on the Upper EICAS Display.

S 865-209
(6) Open these circuit breakers on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1
(b) 11S19, AIR/GND SYS 2

S 985-210
(7) Put the YAW DMPR test switch on the P61 panel in the L position, then the R position, and then put it back to the center position.

S 285-211
(8) Make sure that the YAW DAMPER L and R INOP lights on the YAW DAMPER Control Panel (P5) are off.

S 985-388
(9) Push the DISPLAY STATUS switch on the EICAS Display Control Panel (P9).

S 285-212
(10) Make sure that the Rudder Position Indicator, on the Lower EICAS Display - Status Page, shows no rudder movement for 10 seconds.

S 985-213
(11) Push the DISPLAY switch on the Left YAW DAMPER MODULE.

S 285-214
(12) Make sure that these fault messages are not shown on the LEFT YAW DAMPER MODULE fault display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 985-215
(13) Push the DISPLAY SWITCH on the Right YAW DAMPER module.

S 285-217
(14) Make sure that these fault messages are not shown on the Right YAW DAMPER MODULE fault display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 865-216
(15) Open these circuit breakers on the P11 panel:
(a) 11A18, YAW DAMPER LEFT
(b) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(c) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

S 865-218
(16) Close this circuit breaker on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1

S 285-219
(17) Make sure that the YAW DAMPER-R INOP light, on the Yaw Damper Control Panel (P5), is off.

S 285-389
(18) Make sure that the "L YAW DAMPER " message is displayed on the Upper EICAS Display.

S 985-220
(19) Stop for 2 seconds minimum and then push the DISPLAY switch on the Right YAW DAMPER MODULE.

S 285-221
(20) Make sure that this fault message is shown on the Right YAW DAMPER MODULE fault display:
(a) AIRGND 2

S 985-222
(21) Put the COMPUTER Select switch, on the EICAS Display Select Panel (P9), in the L position.

S 985-391
(22) Push the ECS/MSG switch on the EICAS Maintenance Panel (P61).

S 285-223
(23) Make sure that the "YAW DAMPER" message is shown on the Lower EICAS Display - Maintenance Page.

S 985-224
(24) Put the COMPUTER Select switch, on the EICAS Display Select Panel, in the R position.

S 285-225
(25) Make sure that the "YAW DAMPER" message is shown on the Lower EICAS Display - Maintenance Page.

S 865-226
(26) Open this circuit breaker on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1

S 985-227
(27) Stop for 10 seconds minimum and then push the DISPLAY switch on the Right YAW DAMPER MODULE.

S 285-228
(28) Make sure that these fault messages are not shown on the Right YAW DAMPER MODULE Fault Display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 865-229
(29) Close these circuit breakers on the P11 panel:
(a) 11A18, YAW DAMPER LEFT
(b) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(c) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

S 865-230
(30) Close this circuit breaker on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1

S 865-231
(31) Open these circuit breakers on the P11 panel:
(a) 11F34, YAW DAMPER RIGHT
(b) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC

(c) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 285-232
(32) Make sure that the YAW DAMPER-L INOP light, on YAW DAMPER Control Panel, is off.

S 285-392
(33) Make sure that the "R YAW DAMPER" message is shown on the Upper EICAS Display.

S 985-233
(34) Stop for 2 seconds and then push the DISPLAY switch on the Left YAW DAMPER MODULE.

S 285-234
(35) Make sure that this message is shown on the Left YAW DAMPER MODULE fault display:
(a) AIRGND 2

S 285-235
(36) Make sure that the "YAW DAMPER" message is shown on the Lower EICAS Display - Maintenance Page.

S 865-236
(37) Open this circuit breaker on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1

S 985-237
(38) Stop for 10 seconds and then push the DISPLAY switch on the Left YAW DAMPER MODULE.

S 285-238
(39) Make sure that these fault messages are not shown on the Left YAW DAMPER MODULE Fault Display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 865-239
(40) Close these circuit breakers on the P11 panel:
(a) 11F34, YAW DAMPER RIGHT
(b) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(c) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(d) 11S19, AIR/GND SYS 2

S 985-240
(41) Stop for a minimum of 2 seconds and then push the DISPLAY switch on the Left YAW DAMPER MODULE.

S 285-241
(42) Make sure that this message is shown on the Left YAW DAMPER MODULE fault display:
(a) AIRGND 1

S 985-242
(43) Push the DISPLAY switch on the Right YAW DAMPER MODULE.

S 285-243
(44) Make sure that this fault message is displayed on the RIght YAW DAMPER MODULE Fault Display:
(a) AIRGND 1

S 865-244
(45) Open this circuit breaker on the P11 panel:
(a) 11S19, AIR/GND SYS 2

S 985-246
(46) Stop for a minimum of 10 seconds and then then push the DISPLAY switch on the Left YAW DAMPER MODULE.

S 285-245
(47) Make sure that these fault messages are not shown on the Left YAW DAMPER MODULE Fault Display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 865-373
(48) Push the DISPLAY button on the Right YAW DAMPER MODULE.

S 285-393
(49) Make sure that these fault messages are not shown on the Right YAW DAMPER MODULE Fault Display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 865-247
(50) Close these circuit breakers on the P11 panel:
(a) 11C30, LANDING GEAR POS SYS 1
(b) 11S19, AIR/GND SYS 2

S 865-354
(51) Put the safety sensitive systems back to their initial conditions (Ref 32-09-02/201).

J. Hydraulic Interface Test

S 865-248
(1) Remove pressure from the left hydraulic system (Ref 29-11-00).

S 985-249
(2) Put the YAW DMPR test switch on the P61 panel in the L position, then put the switch back to the center.

S 285-250
(3) Make sure that the rudder position indicator on the lower EICAS display shows no rudder movement.

S 865-251
(4) Stop for a minimum of 15 seconds after you do the two steps before.

S 285-252
(5) Make sure that the YAW DAMPER INOP L light is on.

S 985-253
(6) Push the DISPLAY button on the left YAW DAMPER module.

S 285-254
(7) Make sure that this message is not shown on the left YAW DAMPER module fault display:
(a) HYD SWITCH

S 985-255
(8) Push the left YAW DAMPER switch/light on the overhead panel, P5, twice to re-engage left system.

S 865-256
(9) Pressurize the left hydraulic system (Ref 29-11-00).

S 285-257
(10) Stop for a minimum of 30 seconds then make sure the YAW DAMPER INOP L light is off.

S 985-258
(11) Push and release the RESET button on the left YAW DAMPER module.

S 985-259
(12) Push the DISPLAY button on the left YAW DAMPER module.

S 285-260
(13) Make sure that this message is not shown on the left yaw damper module fault display:
(a) HYD SWITCH

S 865-261
(14) Remove pressure from the center hydraulic system (Ref 29-11-00).

S 985-262
(15) Put the YAW DMPR test switch on the P61 panel in the R position, then put the switch back to the center.

S 285-263
(16) Make sure that the rudder position indicator on the lower EICAS display shows no rudder movement.

S 755-374
(17) Stop for a minimum of 15 seconds after you do the two steps before.

S 285-264
(18) Make sure that YAW DAMPER INOP R light is on.

S 985-265
(19) Push the DISPLAY button on the right YAW DAMPER module.

S 285-266

(20) Make sure that this message is not shown on the right yaw damper module fault display:
(a) HYD SWITCH

S 985-267

(21) Push the right YAW DAMPER switch/light twice to re-engage right system.

S 865-268

(22) Pressurize the center hydraulic system (Ref 29-11-00).

S 285-269

(23) Stop for a minimum of 30 seconds then make sure that YAW DAMPER INOP R light is off.

S 985-270

(24) Push the RESET switch on right yaw damper module.

S 985-271

(25) Push the DISPLAY button on the right YAW DAMPER module.

S 285-272

(26) Make sure that these fault messages are not shown on the right yaw damper module fault display:
(a) HYD SWITCH
(b) YD ACT

K. Air Data Computer Interface Test

S 845-363

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-375
(2) Push and hold the test button on the L ADC.

<u>NOTE</u>: When you push the test button on the ADC, the ADC will send a self-test signal and a failure signal to the YDM. The failure signal is stored by the YDM as a fault message that can be viewed when you push the YDM's DISPLAY button.

S 985-273
(3) Push the DISPLAY button on the left YAW DAMPER module.

S 285-274
(4) Make sure that this message is shown on the left YAW DAMPER module fault display:
(a) L ADC

S 985-275
(5) Push the DISPLAY button on the right YAW DAMPER module.

S 285-276
(6) Make sure that this message is shown on the right yaw damper module fault display:
(a) L ADC

S 985-277
(7) Release the test button on the L ADC.

S 985-278
(8) Push the DISPLAY button on the left YAW DAMPER module.

S 285-279
(9) Make sure that these fault messages are not shown on the left yaw damper module fault display:
(a) L ADC
(b) R ADC

S 985-280
(10) Push the DISPLAY button on the right YAW DAMPER module.

S 285-281
(11) Make sure that these fault messages are not shown on the right yaw damper module fault display:
(a) L ADC
(b) R ADC

S 985-282
(12) Push and hold the test button on the R ADC.

S 985-283
(13) Push the DISPLAY button on the left YAW DAMPER module.

S 285-284
(14) Make sure that this message is shown on the left yaw damper module fault display:
(a) R ADC

S 865-376
(15) Push the DISPLAY button on the right YAW DAMPER module.

S 285-285
(16) Make sure that this message is shown on the right yaw damper module fault display:
(a) R ADC

S 985-286
(17) Release the test button on the R ADC.

S 985-287
(18) Push the DISPLAY button on the left YAW DAMPER module.

S 285-288
(19) Make sure that these fault messages are not shown on the left yaw damper module fault display:
(a) L ADC

(b) R ADC

S 985-290
(20) Push the DISPLAY button on the right YAW DAMPER module.

S 285-289
(21) Make sure that these fault messages are not shown on the right yaw damper module fault display:
(a) L ADC
(b) R ADC

L. Inertial Reference Unit (IRU) Interface Test

<u>NOTE</u>: When you push the TEST button on the IRU, the IRU starts a 10 second self-test. The yaw damper response to this is to store a fault message that can be shown on the fault YDM's display.

S 985-291
(1) Push and release the TEST button on front of the left IRU.

S 985-292
(2) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-293
(3) Make sure that this message is shown on the left yaw damper module fault display:
(a) L IRU

S 985-294
(4) Push the DISPLAY button on the right YAW DAMPER module.

S 285-295
(5) Make sure that this message is shown on the right YAW DAMPER module fault display:
(a) L IRU

S 985-296
(6) Stop for a minimum of 10 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-297
(7) Make sure that this message is not shown on the left YAW DAMPER module fault display:
(a) L IRU

S 985-298
(8) Push the DISPLAY button on the right YAW DAMPER module.

S 285-299
(9) Make sure that this message is not shown on the right YAW DAMPER module fault display:
(a) L IRU

S 985-300
(10) Push and release the TEST button on front of the right IRU.

S 985-301
(11) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-302
(12) Make sure that this message is shown on the left YAW DAMPER module fault display:
(a) R IRU

S 985-303
(13) Push the DISPLAY button on the right YAW DAMPER module.

S 285-304
(14) Make sure that this message is shown on the right YAW DAMPER module fault display:
(a) R IRU

S 985-305
(15) Stop for a minimum of 10 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-306
(16) Make sure that this message is not shown on the left YAW DAMPER module fault display.
(a) R IRU

S 985-307
(17) Push the DISPLAY button on the right YAW DAMPER module.

S 285-308
(18) Make sure that this message is not shown on the right YAW DAMPER module fault display:
(a) R IRU

S 985-309
(19) Push and release the TEST button on front of center the IRU.

S 985-310
(20) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-311
(21) Make sure that this message is shown on the left YAW DAMPER module fault display:
(a) C IRU

S 985-312
(22) Push the DISPLAY button on the right YAW DAMPER module.

S 285-313
(23) Make sure that this message is shown on the right YAW DAMPER module fault display:
(a) C IRU

S 985-314
(24) Stop for a minimum of 10 seconds, then psuh the DISPLAY button on the left YAW DAMPER module.

S 285-315
(25) Make sure that this message is not shown on the left YAW DAMPER module fault display.
(a) C IRU

S 985-316
(26) Push the DISPLAY button on the right YAW DAMPER module.

S 285-317
(27) Make sure that this message is not shown on the right YAW DAMPER module fault display:
(a) C IRU

M. Yaw Damper Modal Suppression Accelerometer Interface Test

S 985-318
(1) Open these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 035-319
(2) Disconnect the accelerometer electrical connectors D1357 and D1359.

S 865-320
(3) Close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 985-321
(4) Push the reset button on the left yaw damper module.

S 985-322
(5) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 295-323
(6) Make sure that this message is shown on the left yaw damper module fault display:
(a) ACCEL

S 985-324
(7) Push the reset button on the right yaw damper module.

S 985-325
(8) Stop for a minimum of 15 seconds, then push the DISPLAY button on the right YAW DAMPER module.

S 285-326
(9) Make sure that this message is shown on the right yaw damper module fault display:
(a) ACCEL

S 865-327
(10) Open these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 435-328
(11) Connect the accelerometer electrical connectors, D1357 and D1359.

S 865-329
(12) Close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 985-330
(13) Push the reset button on the left yaw damper module.

S 985-331
(14) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module.

S 285-332
(15) Make sure that this message NO FAULTS is shown on the left yaw damper module fault display.

S 985-333
(16) Push the reset button on the right yaw damper module.

S 985-334
(17) Stop for a minimum of 15 seconds then push the DISPLAY button on the right YAW DAMPER module.

S 285-335
(18) Make sure that this message NO FAULTS is shown on the right yaw damper module fault display.

N. Put the airplane back to its initial condition.

S 865-336
(1) Remove the power from the left and center systems (Ref 29-11-00).

S 865-337
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YAW DAMPER PANEL - REMOVAL/INSTALLATION

1. General
   A. The yaw damper panel is on the left side of the overhead panel, P5. The yaw damper panel is held in place by four 1/4 turn panel screws. Two electrical connectors are at the rear of the unit.

TASK 22-21-01-024-001

2. Remove the Yaw Damper Panel
   A. Reference
      (1) AMM 24-22-00/201, Electrical Power - Control
   B. Access
      (1) Location Zone
          211 Control Cabin (left)

   C. Prepare for Removal

      S 864-003
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A18, YAW DAMPER LEFT
          (b) 11A34, INDICATOR LIGHTS 3
          (c) 11F34, YAW DAMPER RIGHT
          (d) 11P28, R IND LTS 1
   D. Remove Yaw Damper Panel

      S 034-004
      (1) Hold the control panel in position and loosen the four quick-release screws.

      S 034-005
      (2) Carefully lower the yaw damper panel from the overhead panel.

      S 034-006
      (3) Remove the electrical connectors attached to the rear of the panel.

TASK 22-21-01-424-007

3. Install Yaw Damper Panel
   A. Reference
      (1) AMM 24-22-00/201, Electrical Power - Control

B. Access
   (1) Location Zone
       211 Control Cabin (left)

C. Prepare for Removal

   S 284-008
   (1) Make sure that the panel is aligned correctly and connect the two electrical connectors to the back of the unit.

D. Install the Yaw Damper Panel

   S 434-009
   (1) Raise the panel into position and attach it with the four quick-release screws.

E. Yaw Damper Panel Test

   S 864-010
   (1) Supply electrical power (AMM 24-22-00/201).

   S 864-011
   (2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel.
       (a) 11A18, YAW DAMPER LEFT
       (b) 11A34, INDICATOR LIGHTS 3
       (c) 11F34, YAW DAMPER RIGHT
       (d) 11P28, R IND LTS 1

   S 984-012
   (3) Push the IND LTS TEST button on the P5 panel, and hold if necessary, and make sure the light on the panel is on.

   S 984-021
   (4) Push the IND LTS TEST button to stop the test, if necessary.

F. Put the Airplane Back to Its Initial Condition

   S 864-013
   (1) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YAW DAMPER SERVO - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. Two yaw damper servos are on the rear spar of the vertical stabilizer. You can access the servos through the trailing edge service access panel, 324BL, of the vertical stabilizer (AMM 06-42-00/201).

TASK 22-21-02-024-057

2. Remove Yaw Damper Servo
   A. References
      (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels.
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 27-21-00/501, Rudder and Rudder Trim Control System
      (4) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (5) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (6) AMM 31-41-00/501, Engine Indication and Crew Alerting System
      (7) AMM 32-09-02/201, Air/Ground Relays
      (8) AMM 33-16-00/501, Master Dim and Test
      (9) AMM 34-12-00/501, Air Data Computing System
      (10) AMM 34-21-00/501, Inertial Reference System
   B. Access
      (1) Location Zones

211/212	Flight Compartment
324	Vertical Stabilizer - Rear Spar to Trailing Edge

      (2) Access Panel

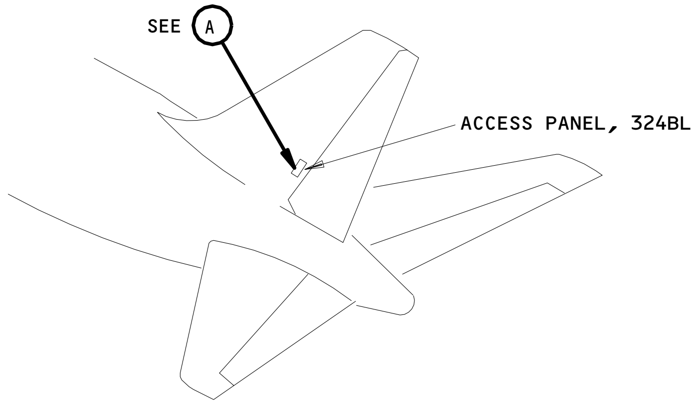
324BL	Vertical Stabilizer - Trailing Edge

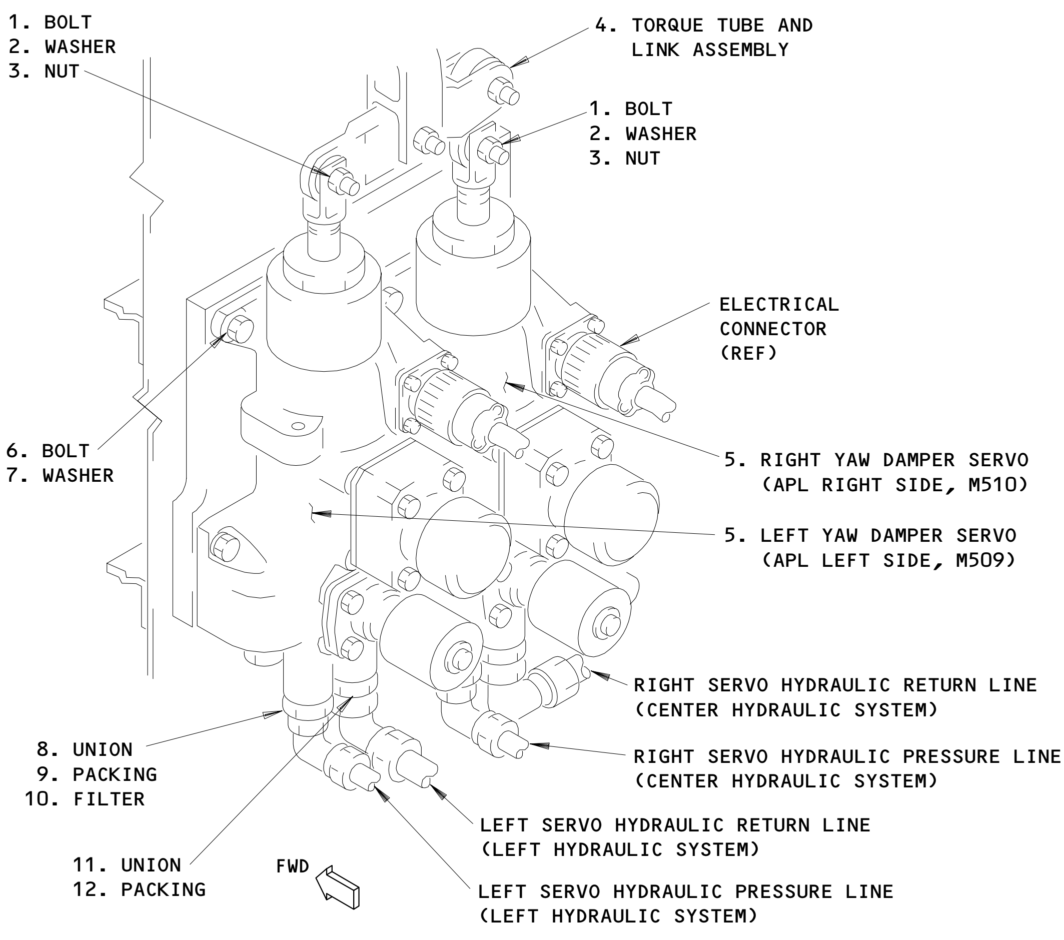
   C. Prepare For Removal

S 844-058

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.





YAW DAMPER SERVOS
A

Yaw Damper Servo Installation
Figure 401

S 864-001

(2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11A18, YAW DAMPER LEFT
   (b) 11C6, CSEU 1L AC
   or
   FLT CONT ELEC 1L AC
   (c) 11C7, CSEU 1L DC
   or
   FLT CONT ELEC 1L DC
   (d) 11C8, CSEU 2L AC
   or
   FLT CONT ELEC 2L AC
   (e) 11C9, CSEU 2L DC
   or
   FLT CONT ELEC 2L DC
   (f) 11F34, YAW DAMPER RIGHT
   (g) 11G17, CSEU 1R AC
   or
   FLT CONT ELEC 1R AC
   (h) 11G18, CSEU 1R DC
   or
   FLT CONT ELEC 1R DC
   (i) 11G27, CSEU 2R AC
   or
   FLT CONT ELEC 2R AC
   (j) 11G28, CSEU 2R DC
   or
   FLT CONT ELEC 2R DC

S 864-004

(3) Supply electrical power (AMM 24-22-00/201).

S 864-005

(4) For the left servo, remove pressure from the left hydraulic system (AMM 29-11-00/201).

S 864-006

(5) For the right servo, remove pressure from the center hydraulic system (AMM 29-11-00/201).

S 984-007

(6) Make sure the L, C and R FLT CONTROL SHUTOFF switches on panel P61 are in the OFF position.

S 984-008

(7) Put the C and R STAB TRIM switches on the control stand, P10, in the CUT OUT position.

S 864-009

(8) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
- (a) 11C12, STAB TRIM SHUTOFF LEFT
- (b) 11C13, STAB TRIM SHUTOFF RIGHT
- (c) 11H17, FLT CONT SHUTOFF TAIL LEFT
- (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
- (e) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-067

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(9) Make sure the HF system does not transmit.
- (a) Open these circuit breakers (if installed) on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
  - 1) 11G8, HF COMM LEFT
  - 2) 11G34, HF COMM RIGHT

S 014-010

(10) Open the service access panel, 324BL (AMM 06-42-00/201).

D. Procedure

S 034-011

(1) Remove the electrical connector from the yaw damper servo.

S 034-012

(2) Remove the hydraulic lines from the yaw damper servo and seal the lines with a cap.

S 394-013

(3) Seal the hydraulic ports on the yaw damper servo with caps.

S 034-014

(4) Remove bolt (1), washer (2) and nut (3) to disconnect the servo (5) output from the torque tube and link assembly (4).

S 034-015

(5) Hold the yaw damper servo (5) in position and remove mounting bolts (6) and washers (7).

S 024-016

(6) Remove the servo (5) from the vertical stabilizer rear spar.

TASK 22-21-02-424-017

3. Install Yaw Damper Servo

A. Consumable Materials

(1) C00308 Compound, Corrosion Prevention MIL-C-11796

B. Parts

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401	4	Tube and Link Assy - Torque	27-21-17	01	125
	5	Yaw Damper Servo	22-21-02	01	10
	9	Packing	22-21-02	02	10
	10	Filter			15
	12	Packing			25

C. References

(1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels.
(2) AMM 20-30-03/201, Finishing Materials
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(5) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(6) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(7) AMM 31-41-00/501, Engine Indication and Crew Alerting System
(8) AMM 32-09-02/201, Air/Ground Relays
(9) AMM 33-16-00/501, Master Dim and Test
(10) AMM 34-12-00/501, Air Data Computing System
(11) AMM 34-21-00/501, Inertial Reference System

D. Access

(1) Location Zones

211/212 Flight Compartment
324 Vertical Stabilizer - Rear Spar to Trailing Edge

(2) Access Panel

324BL Vertical Stabilizer - Trailing Edge

E. Procedure

S 864-068

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(1) Make sure the HF system does not transmit.

S 394-018
(2) Apply a layer of Corrosive Preventive Compound MIL-C-11796 (AMM 20-30-03/201) to all smooth surfaces of bolts (6).

S 984-019
(3) Put the servo (5) into mounting position on the vertical stabilizer.

S 424-020
(4) Install the servo with washers (7) and bolts (6).

S 034-021
(5) Remove the caps from the hydraulic ports on the servo and the hydraulic lines.

S 434-023
(6) Install new packing (9, 12) in the hydraulic pressure and hydraulic return lines.

S 434-022
(7) Connect the hydraulic lines to the servo and tighten by hand.

S 434-025
(8) Connect the servo output to the torque tube and link assembly (4) with bolt (1), washer (2), and nut (3).

S 494-059
(9) Set the torque wrench to 100-120 pound-inches.

S 434-060
(10) Tighten the hydraulic pressure line union (8) to 100-120 pound-inches.

S 494-061
(11) Set the torque wrench to 125-150 pound-inches.

S 434-062
(12) Tighten the hydraulic return line union to 125-150 pound-inches.

S 434-026

(13) Connect the electrical connector to the servo.

F. Test the Yaw Damper Servo

S 864-028

(1) Supply electrical power (AMM 24-22-00/201).

S 864-029

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
  (a) 11A18, YAW DAMPER LEFT
  (b) 11F34, YAW DAMPER RIGHT
  (c) 11C6, CSEU 1L AC
      or
      FLT CONT ELEC 1L AC
  (d) 11C7, CSEU 1L DC
      or
      FLT CONT ELEC 1L DC
  (e) 11C8, CSEU 2L AC
      or
      FLT CONT ELEC 2L AC
  (f) 11C9, CSEU 2L DC
      or
      FLT CONT ELEC 2L DC
  (g) 11G17, CSEU 1R AC
      or
      FLT CONT ELEC 1R AC
  (h) 11G18, CSEU 1R DC
      or
      FLT CONT ELEC 1R DC
  (i) 11G27, CSEU 1R AC
      or
      FLT CONT ELEC 2R AC
  (j) 11G28, CSEU 2R DC
      or
      FLT CONT ELEC 2R DC

S 864-063

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
  (a) 11H17, FLT CONT SHUTOFF TAIL LEFT
  (b) 11H18, FLT CONT SHUTOFF TAIL CTR
  (c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 984-031
(4) Put the L, C, and R FLT CONTROL SHUTOFF switches in the ON position.

S 984-032
(5) Put the STAB TRIM switches on panel P10 in the NORM position.

S 284-033
(6) Make sure these systems operate.
(a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
(b) Air Data Computing System (AMM 34-12-00/501)
(c) Inertial Reference System (AMM 34-21-00/501)
(d) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
(e) Air/Ground Relays (AMM 32-09-02/201)
(f) Master Dim and Test System (AMM 33-16-00/501)

S 984-034
(7) Push the STATUS switch on the EICAS display select panel on the control stand panel, P10:

S 864-035
(8) For the left yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 864-036
(9) For the right yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

S 984-037
(10) Push the left or right YAW DAMPER switch/light on P5 and make sure that ON comes on.

S 284-038
(11) Make sure that the applicable YAW DAMPER L or R INOP light is off.

S 984-039
(12) For left servo, put the YAW DMPR test switch on P61 in the L position, then put the switch back to the center.

S 284-066
(13) Make sure that the YAW DAMPER L INOP light comes on.

S 284-065
(14) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

(d) Make sure that the YAW DAMPER L INOP light goes off in less than 15 seconds after you put the switch back to center.

S 984-040
(15) For the right servo, put the YAW DMPR test switch on P61 in the R position, then put the switch back to center.

S 284-041
(16) Make sure that the YAW DAMPER R INOP light comes on.

S 284-064
(17) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.
(d) Make sure that the YAW DAMPER R INOP light goes off in less than 15 seconds after you put the switch back to center.

S 984-045
(18) Push the DISPLAY button on the left and right yaw damper modules and make sure that the message NO FAULTS shows on the two yaw damper modules.

G. Inspect the valves for hydraulic leaks.

S 984-047
(1) Put the L, C and R FLT CONTROL SHUTOFF switches in the OFF position.

S 984-048
(2) Put the STAB TRIM switches in the CUT OUT position.

S 864-049
(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11H17, FLT CONT SHUTOFF TAIL LEFT
(b) 11H18, FLT CONT SHUTOFF TAIL CENTER
(c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 864-069

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(4) Make sure the HF system does not transmit.

S 284-050
(5) Look to see if hydraulic fluid leaks from the screws and correct as necessary.

H. Return the Airplane to Its Initial Condition

S 464-070

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(1) Make sure the HF system does not transmit.

S 414-051
(2) Close the service access panel, 324BL.

S 864-052
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT
(c) 11G8, HF COMM LEFT (if installed)
(d) 11G34, HF COMM RIGHT (if installed)
(e) 11H17, FLT CONT SHUTOFF TAIL LEFT
(f) 11H18, FLT CONT SHUTOFF TAIL CENTER
(g) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 984-053
(4) Put the L, C, and R FLT CONTROL SHUTOFF switches in the OFF position.

S 984-054
(5) Put the R and C STAB TRIM switches in the NORM position.

S 864-055
(6) Remove the pressure from the left and center hydraulic systems (AMM 29-11-00/201).

S 864-056
(7) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YDS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - MAINTENANCE PRACTICES

1. General (Fig. 201)
   A. The two Yaw Damper Servos are on the rear spar of the vertical stabilizer. Each servo unit has one electrohydraulic servovalve (EHSV) and one solenoid valve. Removal/Installation procedures are the same for the two valves. You can access the servos through the trailing edge service access panel, 324BL, of the vertical stabilizer (AMM 06-42-00/201).

TASK 22-21-03-022-059

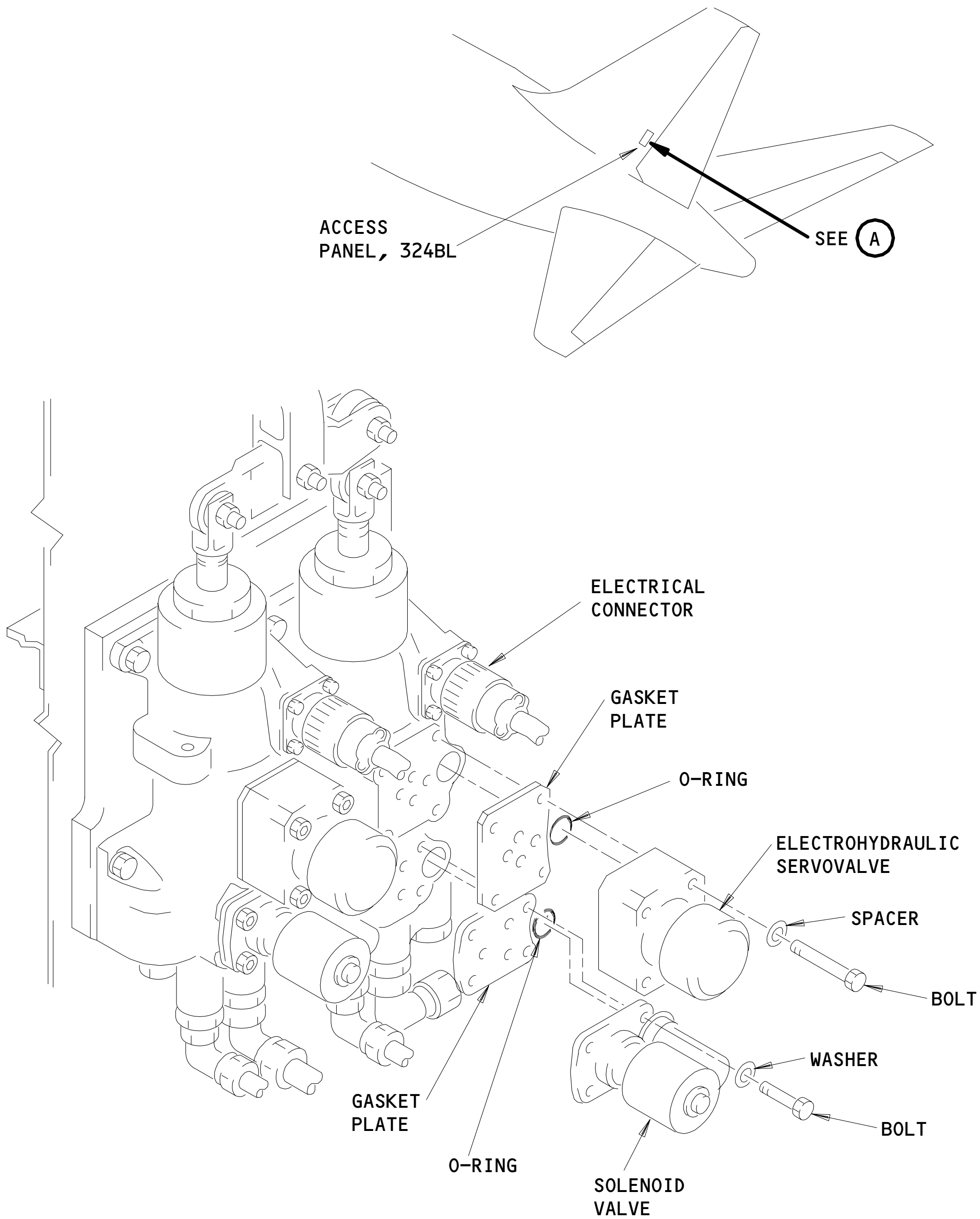
2. Remove Electrohydraulic Servovalve and Solenoid Valve
   A. Equipment
      (1) Torque Wrench - Commercially available (For torque range, Ref Fig. 201)
   B. Consumable Materials
      (1) D00148 Hydraulic Fluid, Fire Resistant, BMS 3-11
   C. References
      (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 27-21-00/501, Rudder and Rudder Trim Control System
      (4) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (5) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (6) AMM 31-41-00/501, Engine Indication and Crew Alerting System
      (7) AMM 32-09-02/201, Air/Ground Relays
      (8) AMM 33-16-00/501, Master Dim and Test
      (9) AMM 34-12-00/501, Air Data Computing System
      (10) AMM 34-21-00/501, Inertial Reference System
   D. Access
      (1) Location Zones

211/212	Flight Compartment
324	Vertical Stabilizer - rear spar to trailing edge

      (2) Access Panel

324BL	Vertical Stabilizer - trailing edge

   E. Prepare For Removal



YDS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES

A

YDS Electrohydraulic Servovalve and Solenoid Valves
Figure 201

S 862-001

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(1) Supply electrical power (AMM 24-22-00/201).

S 842-003

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 862-004

(3) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11A18, YAW DAMPER LEFT
   (b) 11C6, CSEU 1L AC
or
FLT CONT FLEC 1L AC
   (c) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC
   (d) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
   (e) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

(f) 11F34, YAW DAMPER RIGHT
(g) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(h) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(i) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(j) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

S 862-005
(4) For the left servo valves, remove pressure from the left hydraulic system.

S 862-006
(5) For the right servo valves, remove pressure from the center hydraulic system (AMM 29-11-00/201).

S 982-007
(6) Make sure the L, C, and R FLT CONTROL SHUTOFF switches on the panel P61 are in the OFF position.

S 982-008
(7) Put the STAB TRIM switches on control stand panel, P10, in the CUT OUT position.

S 862-009
(8) Open these circuit breakers on the panel P11 and attach DO-NOT-CLOSE tags:
(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT
(c) 11H17, FLT CONT SHUTOFF TAIL LEFT
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 862-092

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(9) Make sure the HF system does not transmit.
(a) Open these circuit breakers (if installed) on the overhead circuit breaker panel, P11, and install DO-NOT-CLOSE tags:
1) 11G8, HF COMM LEFT
2) 11G34, HF COMM RIGHT

S 862-010
(10) Open the service access panel, 324BL (AMM 06-42-00/201).

F. Remove Valve

S 162-011
(1) Fully clean the external areas of the valve and around the valve mounting surface.

S 032-012
(2) Remove the lockwire from the mounting bolts.

S 032-013
(3) Remove the mounting bolts from the valve assembly.

S 022-014
(4) Remove the valve assembly.

S 032-015
(5) Remove the O-ring.

S 032-016
(6) Remove the gasket plate from the servo unit.

S 392-017
(7) Cover all the openings.

TASK 22-21-03-422-062

3. Install Valve

A. Equipment
(1) Torque Wrench - Commercially available (For torque range, Ref Fig. 201)

B. Consumable Materials
(1) D00148 Hydraulic Fluid, Fire Resistant, BMS 3-11

C. References
(1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels

(2) AMM 20-10-23/401, Lockwires
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(5) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(6) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(7) AMM 31-41-00/501, Engine Indication and Crew Alerting System
(8) AMM 32-09-02/201, Air/Ground Relays
(9) AMM 33-16-00/501, Master Dim and Test
(10) AMM 34-12-00/501, Air Data Computing System
(11) AMM 34-21-00/501, Inertial Reference System

D. Access

(1) Location Zones

211/212	Flight Compartment
324	Vertical Stabilizer - rear spar to trailing edge

(2) Access Panel

324BL	Vertical Stabilizer - trailing edge

E. Procedure

S 862-093

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(1) Make sure the HF system does not transmit.

S 012-018

(2) Remove the covers from the openings and clean the mating surfaces of the valve, the gasket plate and servo assembly with clean hydraulic fluid. Do not allow the hydraulic fluid to go into the electrical connector.

S 432-019

(3) Install the new gasket plate on the servo unit.

S 642-020

(4) Lubricate the new O-ring with BMS 3-11 and install it.

S 422-021

(5) Install the valve assembly with the mounting bolts and the washer/spacer.

NOTE: The countersink on the washer/spacer must be towards the bolthead.

S 492-022

(6) Set the torque wrench to 50-70 pound-inches.

S 432-023
(7) Tighten the mounting bolts to 50-70 pound-inches.

S 432-024
(8) Install the lockwire on the mounting bolts (AMM 20-10-23/401).

TASK 22-21-03-722-063
4. Test Yaw Damper Electrohydraulic Servovalve And Solenoid Valve
A. Prepare for test.

S 862-025
(1) Supply electrical power (AMM 24-22-00/201).

S 862-026
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11A18, YAW DAMPER LEFT
(b) 11C6, CSEU 1L AC
or
FLT CONT FLEC 1L AC
(c) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC
(d) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(e) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER RIGHT
(g) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(h) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(i) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(j) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

S 862-027
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11H17, FLT CONT SHUTOFF TAIL LEFT
(b) 11H18, FLT CONT SHUTOFF TAIL CENTER
(c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 982-028
(4) Put the L, C, and R FLT CONTROL SHUTOFF switches on the panel P61 in the ON position.

S 862-029
(5) Make sure these systems operate:
(a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
(b) Air Data Computing System (AMM 34-12-00/501)
(c) Inertial Reference System (AMM 34-21-00/501)
(d) Air/Ground Relays (AMM 32-09-02/501)
(e) Master Dim and Test System (AMM 33-16-00/501)
(f) Engine Indication and Crew Alerting System (AMM 31-41-00/501)

S 982-030
(6) Push the STATUS switch on the EICAS display select panel, on panel P9, to the flight control positions.

S 862-060

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(7) For the left yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 862-032
(8) For the right yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

B. Test valves

S 982-033
(1) Push the YAW DAMPER L or R switch/light on the P5 panel make sure that ON light comes on.

S 282-034
(2) Make sure that the applicable YAW DAMPER L or R INOP light is off.

NOTE: Two persons are necessary for the steps below. One is necessary in the flight compartment and one in the main electrical/electronic equipment center.

S 862-035
(3) Open this circuit breaker on the P11 panel to stop the operation of the left (right) servo.
(a) 11A18, YAW DAMPER LEFT
(11F34, YAW DAMPER RIGHT)

S 282-036
(4) Make sure that the YAW DAMPER L (R) INOP light is on.

S 982-037
(5) Put the YAW DMPR test switch on P61 in the L (R) position, then put the switch back to the center position.

S 282-038
(6) Make sure that the rudder position indicator on the EICAS display unit shows no rudder movement.

NOTE: During this test a YD ACT fault may be shown and a YDM fault will be shown by the left (right) yaw damper module.

Faults set during this test should be clear after you push the YDM RESET button in the step that follows.

S 862-061
(7) Close these circuit breakers on the P11 panel and push the RESET button on the front of left (right) yaw damper module:
(a) 11A18, YAW DAMPER LEFT
(b) 11F34, YAW DAMPER RIGHT

S 982-042
(8) Push the DISPLAY button on the left (right) yaw damper module. If the message NO FAULTS is not displayed on the module, there is a fault.

S 982-043
(9) Put the YAW DMPR test switch on the P61 panel in the L (R) position, then put it back to the center. Make sure that the YAW DAMPER L INOP light comes on in less than 15 seconds.

S 282-044
(10) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

S 282-045
(11) Make sure that the YAW DAMPER L (R) INOP light goes off in less than 15 seconds after the switch is put back to center.

S 282-049
(12) The NO FAULTS message should be the only message shown on the yaw damper modules after completion of this test if there are no faults.

C. Inspect for leakage.

S 982-050
(1) Put the L, C, and R FLT CONTROL SHUTOFF switches in the OFF position.

S 862-051
(2) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11H17, FLT CONT SHUTOFF TAIL LEFT
(b) 11H18, FLT CONT SHUTOFF TAIL
(c) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 862-094

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(3) Make sure the HF system does not transmit.

S 282-052
(4) Look for hydraulic leakage around the valves and correct as necessary.

S 862-053
(5) Close the service access panel, 324BL (AMM 06-42-00/201).

D. Restore Airplane To Normal

S 862-054
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF LEFT
(b) 11C13, STAB TRIM SHUTOFF RIGHT

(c) 11G8, HF COMM LEFT
(d) 11G34, HF COMM RIGHT
(e) 11H17, FLT CONT SHUTOFF TAIL LEFT
(f) 11H18, FLT CONT SHUTOFF TAIL CENTER
(g) 11H28, FLT CONT SHUTOFF TAIL RIGHT

S 982-055
(2) Put the L, C, and R FLT CONTROL SHUTOFF switches in the OFF position.

S 982-056
(3) Put the R and C STAB TRIM switches in the NORM position.

S 862-057
(4) Remove the power from the left and center hydraulic systems (AMM 29-11-00/201).

S 862-058
(5) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-21-03-712-064
5. Yaw Damper Electrohydraulic Servo Valve and Solenoid Valve Operational Check

NOTE: This is a scheduled maintenance task.

A. References
(1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
(2) AMM 24-22-00/201, Electrical Power - Control
(3) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(5) AMM 31-41-00/501, Engine Indication and Crew Alerting System
(6) AMM 32-09-02/201, Air/Ground Relays
(7) AMM 33-16-00/501, Master Dim and Test
(8) AMM 34-12-00/501, Air Data Computing System
(9) AMM 34-21-00/501, Inertial Reference System

B. Access
(1) Location Zones

119	Main Equipment Center
211/212	Flight Compartment
324	Vertical Stabilizer - rear spar to trailing edge

C. Prepare for Test

S 862-065

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 862-066
(2) Supply electrical power (AMM 24-22-00/201).

S 862-067
(3) Put the L, C, and R FLT CONTROL SHUTOFF switches on the panel P61 in the ON position.

S 862-068
(4) Make sure these systems operate:
  (a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
  (b) Air Data Computing System (AMM 34-12-00/501)
  (c) Inertial Reference System (AMM 34-21-00/501)
  (d) Air/Ground Relays (AMM 32-09-02/501)
  (e) Master Dim and Test System (AMM 33-16-00/501)
  (f) Engine Indication and Crew Alerting System (AMM 31-41-00/501)

S 862-072
(5) Push the STATUS switch on the EICAS display select panel, on panel P9, to the flight control positions.

S 862-073

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) For the left yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 862-074
(7) For the right yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

D. Valves Test

S 862-069
(1) Push the YAW DAMPER L or R switch/light on the P5 panel make sure that ON light comes on.

S 862-070
(2) Make sure that the applicable YAW DAMPER L or R INOP light is off.

NOTE: Two persons are necessary for the steps below. One is necessary in the flight compartment and one in the main electrical/electronic equipment center.

S 862-071
(3) Open this circuit breaker on the P11 panel to stop the operation of the left (right) servo.
(a) 11A18, YAW DAMPER LEFT
(11F34, YAW DAMPER RIGHT)

S 862-075
(4) Make sure that the YAW DAMPER L (R) INOP light is on.

S 862-076
(5) Put the YAW DMPR test switch on P61 in the L (R) position, then put the switch back to the center position.

S 862-077
(6) Make sure that the rudder position indicator on the EICAS display unit shows no rudder movement.

NOTE: During this test a YD ACT fault may be shown and a YDM fault will be shown by the left (right) yaw damper module.

Faults set during this test should be clear after you push the YDM RESET button in the step that follows.

S 862-078
(7) Close these circuit breakers on the P11 panel and push the RESET button on the front of left (right) yaw damper module:
(a) 11A18, YAW DAMPER LEFT
(b) 11F34, YAW DAMPER RIGHT

S 862-082
(8) Push the DISPLAY button on the left (right) yaw damper module. If the message NO FAULTS is not displayed on the module, there is a fault.

S 862-083
(9) Put the YAW DMPR test switch on the P61 panel in the L (R) position, then put it back to the center. Make sure that the YAW DAMPER L INOP light comes on in less than 15 seconds.

S 862-084
(10) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

S 862-085
(11) Make sure that the YAW DAMPER L (R) INOP light goes off in less than 15 seconds after the switch is put back to center.

S 862-089
(12) The NO FAULTS message should be the only message shown on the yaw damper modules after completion of this test if there are no faults.

E. Put the Airplane Back to It's Usual Condition.

S 862-149
(1) Put the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel to the OFF position.

S 862-090
(2) Remove the power from the left and center hydraulic systems (AMM 29-11-00/201).

S 862-091
(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YAW DAMPER MODULE - REMOVAL/INSTALLATION

1. General
   A. Two yaw damper modules are in shelves E3-1 and E4-1 of the electrical/electronic equipment center. Each module has two hold down hooks on the front of the module and one multiple pin connector with three inserts at the rear. The removal/installation steps are the same for each yaw damper module.

TASK 22-21-04-024-032

2. Remove Yaw Damper Module
   A. References
      (1) AMM 20-10-01/401, E/E Rack Mounted Components
      (2) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
      (3) AMM 24-22-00/201, Electrical Power - Control
      (4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
      (5) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (6) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (7) AMM 31-41-00/501, Engine Indication and Crew Alerting System
      (8) AMM 32-09-02/201, Air/Ground Relays
      (9) AMM 33-16-00/501, Master Dim and Test
      (10) AMM 34-12-00/501, Air Data Computing System
      (11) AMM 34-21-00/501, Inertial Reference System
   B. Access
      (1) Location Zones
          119/120 Main Equipment Center
          211/212 Control Cabin

      (2) Access Panel
          119BL Main Equipment Center

   C. Prepare for Removal

      S 844-001

      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

      S 864-002

      (2) For the left yaw damper module, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A18, YAW DAMPER LEFT

(b) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC

(c) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC

(d) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

(e) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

S 864-003

(3) For the right yaw damper module, open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE TAGS:

(a) 11F34, YAW DAMPER RIGHT

(b) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC

(c) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC

(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

(e) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

D. Procedure

S 844-004

<u>CAUTION</u>: DO NOT TOUCH THE YAW DAMPER MODULE BEFORE YOU DO THE PROCEDURE FOR ELECTROSTATIC DISCHARGE SENSITIVE DEVICES (AMM 20-41-01/201). ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE YAW DAMPER MODULE.

(1) Do the procedure for electrostatic discharge sensitive devices (AMM 20-41-01/201).

S 024-035

(2) Remove the yaw damper module (AMM 20-10-01/401).

TASK 22-21-04-424-033

3. <u>Install Yaw Damper Module</u>

A. References

(1) AMM 20-10-01/401, E/E Rack Mounted Components

(2) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices

(3) AMM 22-21-00/501, Yaw Damaper System
(4) AMM 24-22-00/201, Electrical Power - Control
(5) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(6) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(7) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(8) AMM 31-41-00/501, Engine Indication and Crew Alerting System
(9) AMM 32-09-02/201, Air/Ground Relays
(10) AMM 33-16-00/501, Master Dim and Test
(11) AMM 34-12-00/501, Air Data Computing System
(12) AMM 34-21-00/501, Inertial Reference System

B. Access
(1) Location Zones
119/120 Main Equipment Center
211/212 Control Cabin

(2) Access Panel
119BL Main Equipment Center

C. Procedure

S 844-005

CAUTION: DO NOT TOUCH THE YAW DAMPER MODULE BEFORE YOU DO THE PROCEDURE FOR ELECTROSTATIC DISCHARGE SENSITIVE DEVICES (AMM 20-41-01/201). ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE YAW DAMPER MODULE.

(1) Do the procedure for electrostatic discharge sensitive devices (AMM 20-41-01/201).

S 424-036
(2) Install the yaw damper module (AMM 20-10-01/401).

D. Yaw Damper Module Test

S 864-006
(1) Apply electrical power (AMM 24-22-00/201).

S 864-007
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel for left module:
(a) 11A18, YAW DAMPER LEFT
(b) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC

(c) 11C7, CSEU 1L DC
or
FLT CONT ELEC 1L DC
(d) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(e) 11C9, CSEU 2L DC
or
FLT CONT ELEC 2L DC

S 864-008
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel for right module:
(a) 11F34, YAW DAMPER RIGHT
(b) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(c) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(e) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC

S 284-009
(4) Make sure that the engine indication and crew alerting system is operational (AMM 31-41-00/501).

S 984-010
(5) Push the STATUS switch on the EICAS display select panel on the control stand panel, P9.

S 864-011

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) For the left yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 864-013
(7) For the right yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

S 284-014

(8) Make sure these systems operate.
   (a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
   (b) Air Data Computing System (AMM 34-12-00/501)
   (c) Inertial Reference System (AMM 34-21-00/501)
   (d) Air/Ground Relays (AMM 32-09-02/201)
   (e) Master Dim and Test System (AMM 33-16-00/501)

S 984-015

(9) Push the left or right YAW DAMPER switch/light on the P5 panel as necessary.
   (a) Make sure that ON is on.
   (b) Make sure that the applicable YAW DAMPER L or R INOP light is off.

S 984-018

(10) For the left yaw damper module, put the YAW DMPR test switch on the P61 panel in the L position, then put the switch back to center.
   (a) Make sure that the YAW DAMPER L INOP light is on.
   (b) Make sure that the rudder position indicator on the lower EICAS display (AMM 31-41-00/501) shows this sequence of rudder movement in less than 10 seconds.
      1) The rudder moves approximately 3 degrees trailing edge right
      2) The rudder moves approximately 3 degrees trailing edge left
      3) The rudder goes back to the center.
   (c) Make sure that the YAW DAMPER L INOP light goes off in less than 15 seconds.

S 984-022

(11) For the right yaw damper module, put the YAW DMPR test switch on the P61 panel in the R position, then put the switch back to center.
   (a) Make sure that the YAW DAMPER R INOP light is on.
   (b) Make sure that the rudder position indicator on the lower EICAS display (AMM 31-41-00/501) shows this sequence of rudder movement in less than 10 seconds:
      1) The rudder moves approximately 3 degrees trailing edge right

2) The rudder moves approximately 3 degrees trailing edge left
3) The rudder goes back to the center.

(c) Make sure that the YAW DAMPER R INOP light goes off in less than 15 seconds.

S 984-028

(12) Push the DISPLAY button on the left and right yaw damper modules.

S 284-029

(13) Make sure that the message NO FAULTS is shown on the display of the two modules.

E. Put the Airplane Back to Its Initial Condition

S 864-042

(1) Push the left or right YAW DAMPER switch/light on the P5 panel, as applicable, to disengage the yaw damper system.

S 864-030

(2) Remove the pressure from the left and center hydraulic systems (AMM 29-11-00/201).

S 864-031

(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

MODAL SUPPRESSION ACCELEROMETER - REMOVAL/INSTALLATION

1. General
   A. Two modal suppression accelerometers are in the ceiling in the area aft of the bulk cargo compartment. They can be accessed through the aft cargo door.

TASK 22-21-05-024-021

2. Remove the Modal Suppression Accelerometer
   A. References
      (1) AMM 24-22-00/201, Electrical Power - Control
      (2) AMM 25-50-03/401, Bulkhead Lining
      (3) AMM 32-09-02/201, Air/Ground Relays
      (4) AMM 34-12-00/501, Air Data Computing System
      (5) AMM 34-21-00/501, Inertial Reference System
   B. Access
      (1) Location Zones
          165/166 Area Aft of Bulk Cargo Compartment
          211/212 Control Cabin

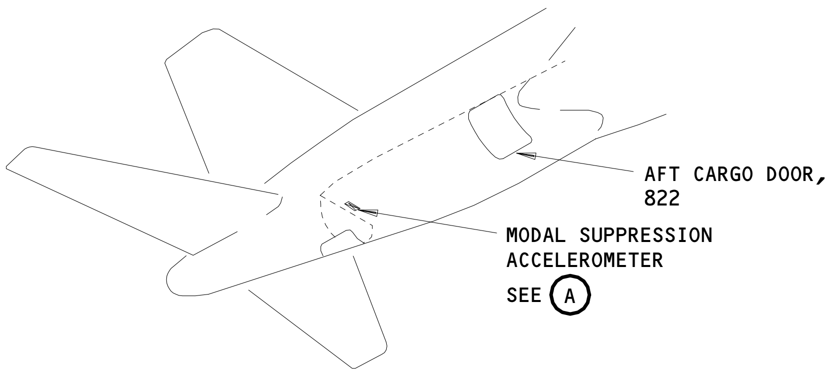
      (2) Access Door
          822 Aft Cargo Compartment Door

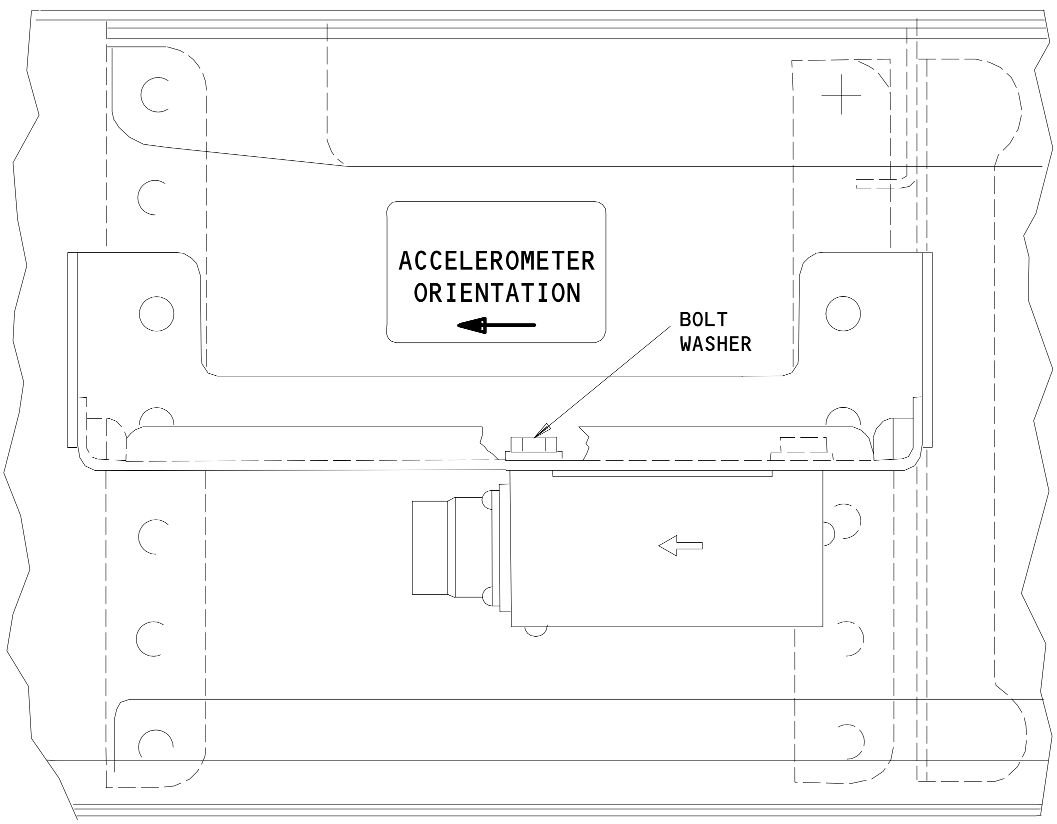
   C. Prepare for Removal

      S 864-001
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11C6, CSEU 1L AC
              or
              FLT CONT ELEC 1L AC
          (b) 11C8, CSEU 2L AC
              or
              FLT CONT ELEC 2L AC
          (c) 11G17, CSEU 1R AC
              or
              FLT CONT ELEC 1R AC
          (d) 11G27, CSEU 2R AC
              or
              FLT CONT ELEC 2R AC

      S 014-020
      (2) Open the aft cargo door and remove bulkhead lining in the bulk cargo compartment (AMM 25-50-03/401).
   D. Remove Modal Suppression Accelerometer

      S 864-003
      (1) Remove the electrical connector from the modal suppression accelerometer.





MODAL SUPPRESSION ACCELEROMETER
A

Modal Suppression Accelerometer Installation
Figure 401

S 984-004
(2) Hold the modal suppression accelerometer in position.

S 034-005
(3) Remove mounting bolts and washers.

S 024-006
(4) Remove the modal suppression accelerometer from the area aft of the bulk cargo compartment.

TASK 22-21-05-424-007

3. Install Modal Suppression Accelerometer

A. References
   (1) AMM 24-22-00/201, Electrical Power - Control
   (2) AMM 25-50-03/401, Bulkhead Lining
   (3) AMM 32-09-02/201, Air/Ground Relays
   (4) AMM 34-12-00/501, Air Data Computing System
   (5) AMM 34-21-00/501, Inertial Reference System

B. Access
   (1) Location Zones
       165/166 Area Aft of Bulk Cargo Compartment
       211/212 Control Cabin

   (2) Access Door
       822 Aft Cargo Compartment Door

C. Install Modal Suppression Accelerometer

S 424-022
(1) Put the modal suppression accelerometer into mounting position in the ceiling in the area aft of bulk cargo compartment.

S 424-009
(2) Install the modal suppression accelerometer with washers and bolts.

S 434-010
(3) Connect the electrical connector to the modal suppression accelerometer.

D. Model Suppression Accelermoter Test

S 864-012
(1) Remove DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 864-013
(2) Supply electrical power (AMM 24-22-00/201).

S 284-014
(3) Make sure that these systems operate:
(a) Air/Ground Relays (AMM 32-09-02/201)
(b) Air Data Computing System (AMM 34-12-00/501)
(c) Inertial Reference System (AMM 34-21-00/501)

S 984-015
(4) Push the RESET button on the front panel of the yaw damper module.

S 984-016
(5) Stop for 10 seconds, then push the display button on the front panel of the yaw damper module.

S 284-017
(6) Make sure that the fault message ACCEL is not shown on the front panel of the yaw damper module.

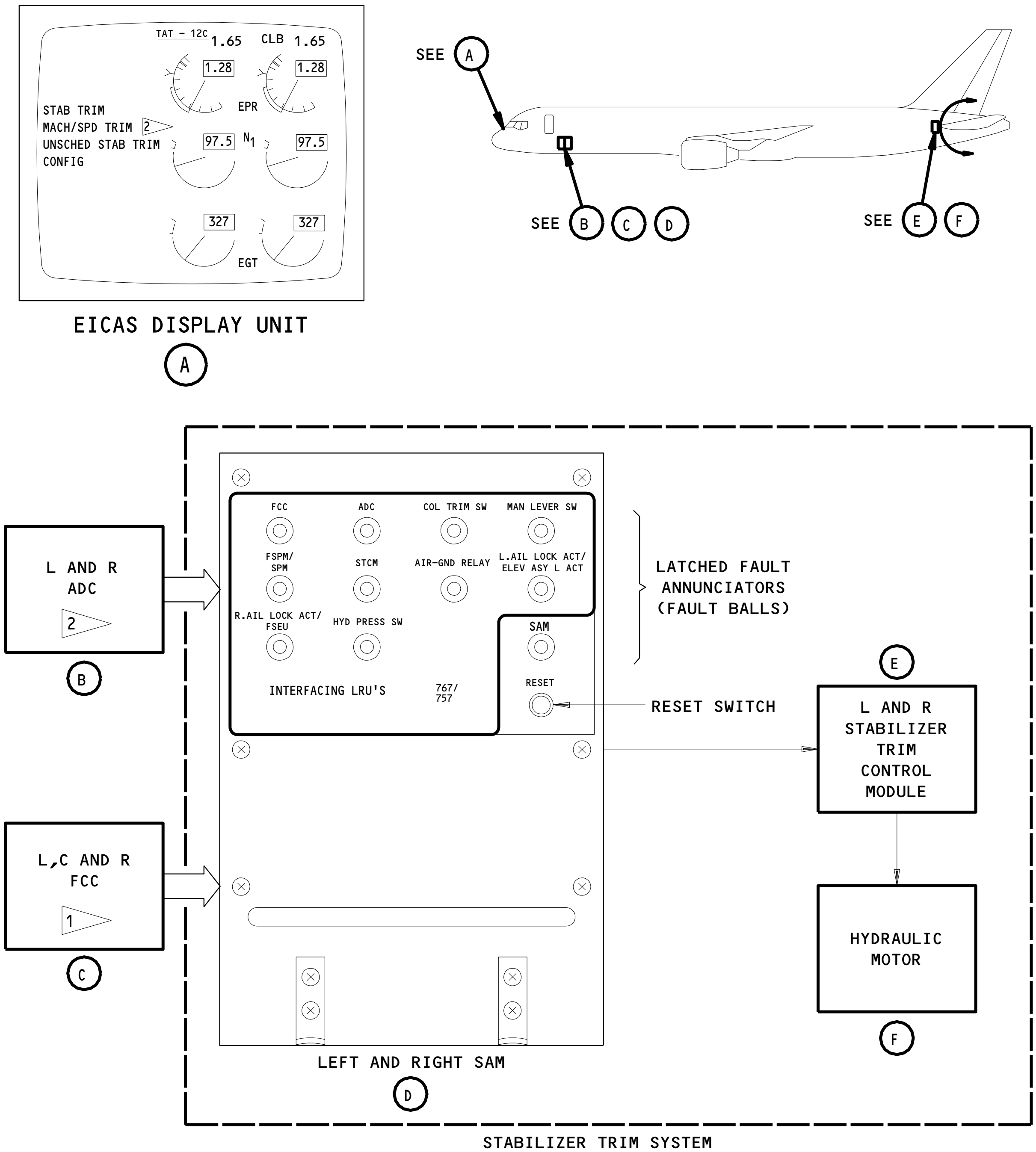
E. Put the airplane back to its initial condition.

S 414-018
(1) Install the bulkhead lining in the bulk cargo compartment and close the aft cargo door (AMM 25-50-03/401).

S 864-019
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

AUTOMATIC STABILIZER TRIM SYSTEM - DESCRIPTION AND OPERATION

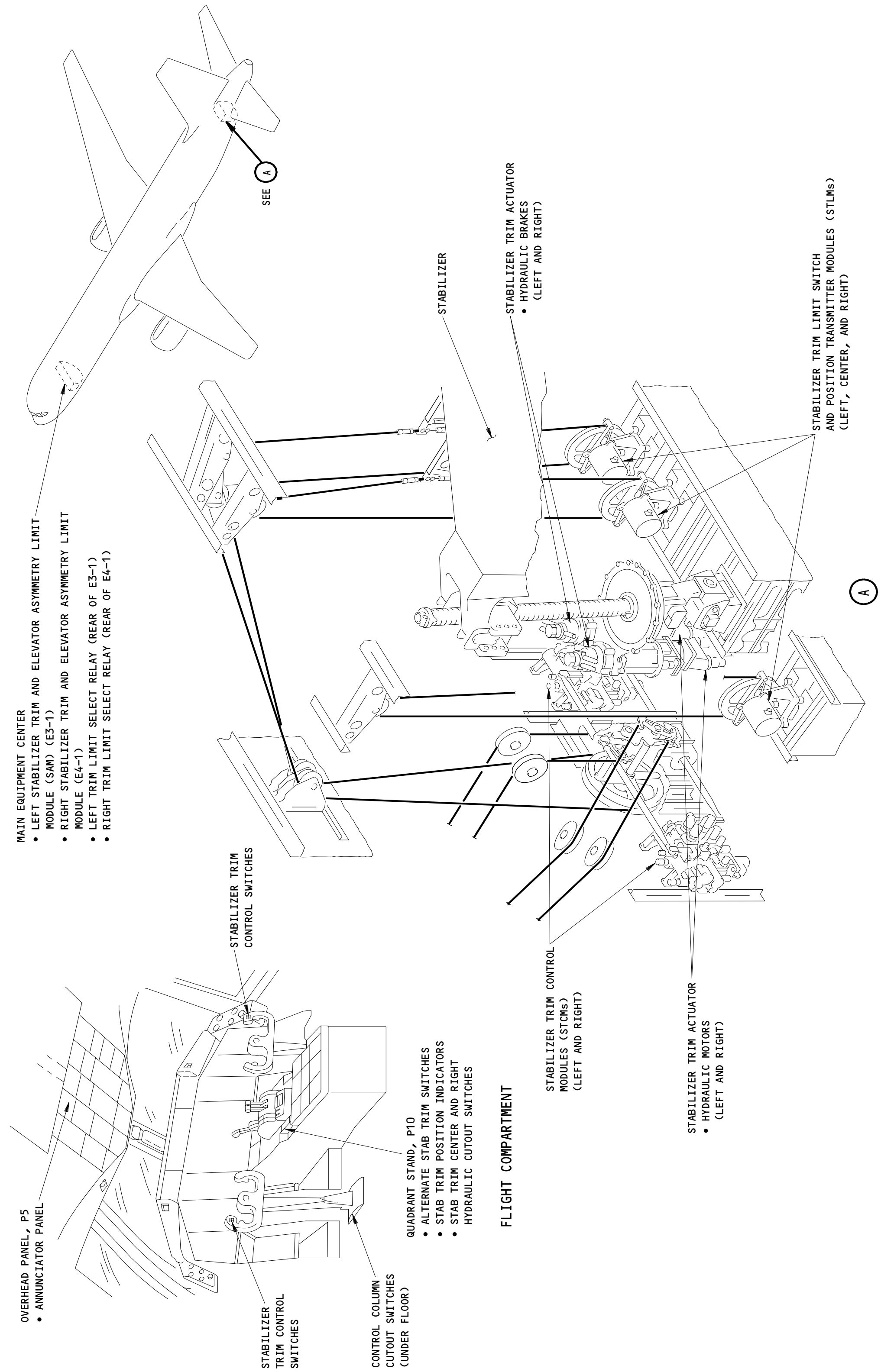
1. General (Fig. 1)
   A. The automatic stabilizer trim system provides automatic control of horizontal stabilizer position. When the autopilot is engaged, the flight control computer outputs stabilizer position commands. There are no line replaceable units, only automatic stabilizer trim components. All components are part of the Horizontal Stabilizer Trim Control System (Ref Chapter 27).
   B. This section covers autotrim operation. A general description of horizontal stabilizer trim system components is added to help understand operation. For component details and complete system description, refer to 27-41-00/001, Horizontal Stabilizer Trim Control System.
   C. The stabilizer trim system maintains the airplane in a longitudinally trimmed condition by varying the incidence of the horizontal stabilizer and provides automatic Mach and speed stability. The optimum pitch attitude is maintained during various flight conditions. The stabilizer may be trimmed manually or automatically. The stabilizer trim system has the following operational modes.
      (1) Manual and manual electric modes - the stabilizer is trimmed by pilot commands.
      (2) Autotrim mode - The autopilot Flight Control Computers (FCCs) trim the stabilizer to relieve sustained elevator loads.
      (3) Mach trim mode - The stabilizer is trimmed as Mach increases.
      (4) Speed trim mode - The stabilizer is trimmed at lower airspeeds when the flaps are down.
   D. Stabilizer Trim Mach Speed Stability Component Locations (Fig. 2)
      (1) Flight Deck Components
         (a) The following stabilizer trim system components are in the control cabin. Stabilizer trim (STAB TRIM) manual trim levers, position indicators, and hydraulic cutout switches are adjacent to the throttle quadrant on the control stand (P10). The annunciator panel on overhead panel P5 contains the amber STAB TRIM, UNSCHED STAB TRIM and MACH/SPD TRIM lights. The stabilizer trim control switches are on the outboard horn of the captain's and first officer's control wheels. The control column stabilizer trim cutout switches are under the cabin floor.



1 AUTOMATIC STABILIZER TRIM SYSTEM ONLY

2 MACH SPEED STABILITY SYSTEM ONLY

Automatic Stabilizer Trim System
Figure 1



Stabilizer Trim Mach Speed Stability Component Locations
Figure 2

A07552

(2) Main Equipment Center

(a) The main equipment center contains the following stabilizer trim system components. Two Stabilizer Trim and Elevator Asymmetry Limit Modules (SAMs) are in the left E3-1 and right E4-1 shelves, respectively. The left and right trim limit select relays are on the aft end of shelves E3-1 and E4-1. The SAM interfacing components in the main Elec/Elex equipment center are the Flight Control Computers (FCCs), Air Data Computers (ADCs), Flap Slat Electronic units (FSEUs), Flap/Slat Accessory Module (FSAM) (if installed), and the Stabilizer Position Modules (SPMs). The left, right, and center FCCs are in shelves E2-1, E2-2, and E2-3, respectively. The left and right ADCs are installed on shelves E2-1 and E2-2, respectively. The No. 1, 3 and 2 FSEUs are installed on shelves E5-1, E5-2 and E5-3, respectively. The FSAM (if installed) is on the E5-2 shelf in place of FSEU 3. The left, center, and right SPMs are installed in the card file of equipment rack P50.

(3) Stabilizer Jackscrew Access Area

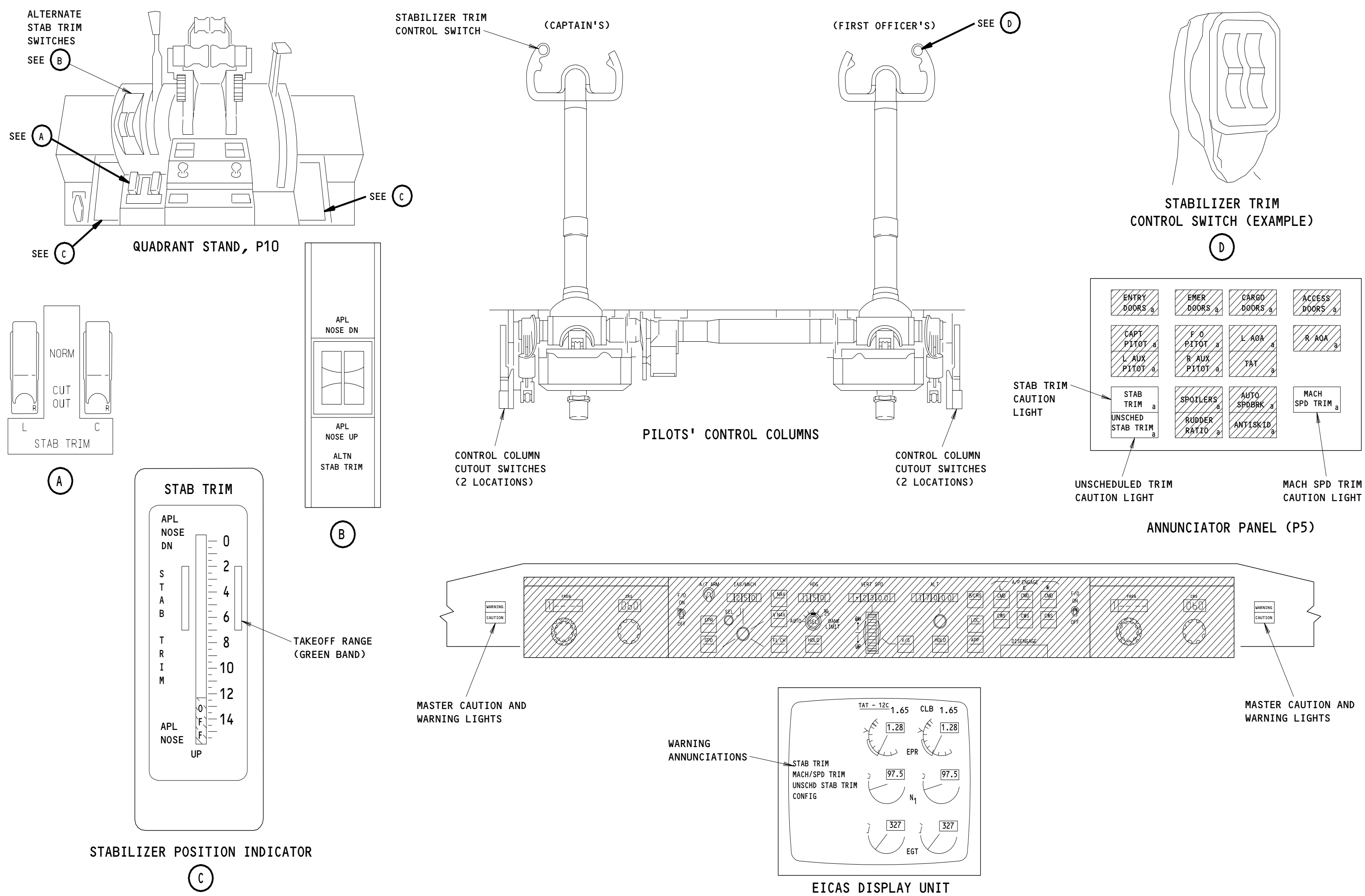
(a) The stabilizer jackscrew area behind the rear pressure bulkhead is accessible through the aft service access door (AMM 52-49-00/001, Exterior Service Doors). The area contains the Stabilizer Trim Actuator Assembly (STAA), the left and right Stabilizer Trim Control Modules (STCMs), and the left, center and right Stabilizer Trim Limit Switch and Position Transmitter Modules (STLMs). The STAA consists of two independent hydraulic motors and brakes coupled to a mechanical differential drive device.

E. Stabilizer Trim Mach Speed Flight Deck Components (Fig. 3)

(1) Stab Trim Position Indicators

(a) The control stand contains two linear indicators that display stabilizer trim position. One is driven by the left STLM. The other is driven by the right STLM.

(b) The size of the indicator panel is 6.5 inches long and 1.5 inches wide. The weight is approximately 2 pounds. The indicator has a vertical scale that displays stabilizer position. The scale reads from 0 units to 16 units. A 0 units indication represents a +4 degree, stabilizer leading edge up (airplane nose down) condition. A 15.5 unit reading represents a -11.5 degree, stabilizer leading edge down (airplane nose up) condition. A green band on each side of the scale (2 to 7 units) indicates the allowable takeoff range. When the system is off, a black and amber striped area is in view at the bottom of the scale.



Stabilizer Trim Mach Speed Stability Flight Deck Components
Figure 3

A07579

(2) Stab Trim Cutout Switches
   (a) The cutout switches enable system hydraulic pressure to be removed from the Stabilizer Trim Control Modules (STCMs). Two guarded switches are on the aisle stand, one for the left and one for the right STCMs. In the ON position, 28 vdc battery bus power activates a motor-operated valve in the STCM that completes hydraulic pressure to the STCM solenoid valves. The CUT OUT position shuts off hydraulic pressure.

(3) Stab Trim Control Switches
   (a) The stab trim control switches enable the pilot or fist officer to input manual trim commands to both SAMs. Manual electric stabilizer trim control is provided by two single pole, three position, center off, thumb-operated rocker switches. The switches are on the outboard horn of each control wheel. Actuating both switches in the up direction issues simultaneous airplane nose up ARM and CONTROL commands. Down actuation issues airplane nose down ARM and CONTROL commands. When the trim switches are energized, both trim channels are activated and the stabilizer is driven at full-rate trim.

(4) Control Column Cutout and Asymmetric Switches
   (a) Each control column operates a set of four switches under the cabin floor, which allow electric trim commands to be interrupted by column action. The switches are wired in series with the SAMs arm command outputs. A column forward action greater than 4.0 degrees opens the switch in the airplane nose-up stabilizer trim command circuit. A column aft action greater than 2.7 degree opens a switch in the airplane nose-down stabilizer trim circuit. Two column asymmetric switches are wired in parallel with the column cutout switches. These switches provide an alternate path for trim commands when the control columns are operated asymmetrically after a system jam.

(5) Failure Indicators
   (a) When stabilizer movement of more than 0.3 degree occurs without a valid SAM output command in autopilot or mach/speed trim mode, the following indications are activated.
      1) An owl aural warning sound and discrete amber UNSCHED STAB TRIM light on the annunciator panel illuminates.
      2) An amber UNSCHD STAB TRIM caution message is displayed on the Engine Indication and Crew Alerting System (EICAS) display unit.
      3) Master caution light on glareshield panel P7 illuminates.
   (b) When faults are detected in the mach/speed circuits of both SAM's or their inputs, the following indications are activated:
      1) A discrete amber MACH/SPD TRIM light on the annunciator panel illuminates.
      2) An amber MACH/SPD TRIM advisory message is displayed on the EICAS display unit.

(c) When a loss of full rate trim occurs during manual electric trim due to SAM, actuation or airplane wiring faults the following indications are activated:
1) A discrete amber STAB TRIM light on the annunciator panel illuminates.
2) An amber STAB TRIM advisory message is displayed on the EICAS display unit.

(d) When the stabilizer is positioned outside the takeoff greenband range and the throttles are advanced, the following indications are activated:
1) Warning siren and discrete CONFIG red warning light on the P1 panel illuminates.
2) A red STABILIZER warning message is displayed on the EICAS display unit.
3) Master warning light in panel P7 illuminates.

(e) When SAM failure monitors detect a fault a first fail discrete is provided to the EICAS computer. This will cause a STAB TRIM message to be generated on the EICAS maintenance page.

(f) When a single autopilot is engaged and an autotrim failure (dead trim, unscheduled trim or AUTOTRIM signal from SAM is invalid as detected by the FCC) cannot be cleared by autosequencing, the following indications are activated.

NOTE: Autosequencing can occur only when the center FCC is engaged.

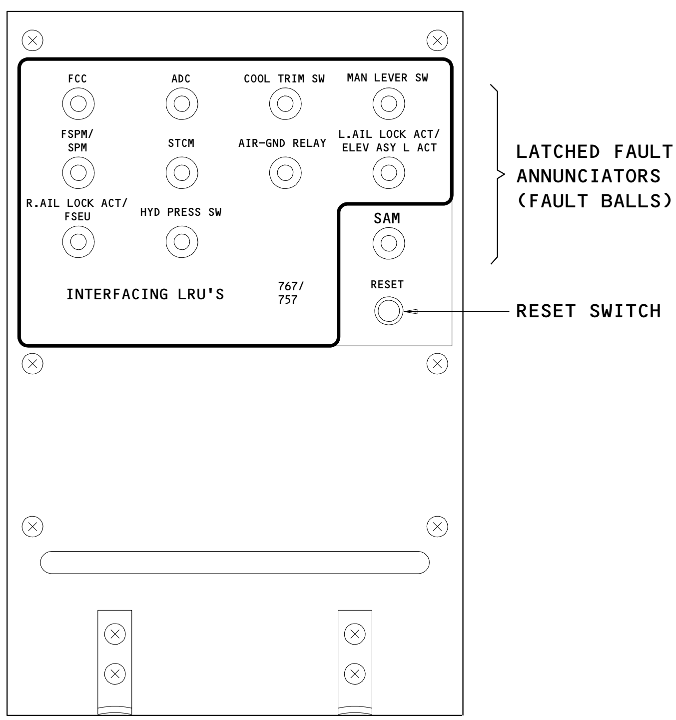
1) A discrete amber AUTOPILOT caution light on P1 panel illuminates.
2) Master caution light on panel P7 illuminates and an aural tone (owl) is sounded.
3) A pitch mode fail (EFIS mode lined - out) is displayed on EADI display units.

(g) When two or more FCCs are engaged and an autotrim failure (detected by the FCC during an approach) is cleared by autosequencing, NO LAND 3 is displayed on the Autoland Status Annunciators (ASA) after touchdown and disconnect of the autopilot. The ASAs are on the P1 and P3 panels.

(h) When two or more FCCS are engaged and an autotrim failure (detected by the FCC) cannot be cleared by autosequencing while the airplane is above Decision Height (DH), the following indications are activated.
1) NO AUTOLAND is displayed on each ASA.
2) A pitch mode fail (EFIS mode lined - out) is displayed on EADI display units.
3) A discrete amber AUTOPILOT light on the panel P1 illuminates.
4) Master caution light on panel P7 illuminates and the owl aural warning sounds.

(i) When two or more FCCs are engaged and an autotrim failure (detected by the FCC) cannot be cleared by autosequencing while the airplane is below Decision Height (DH), the following indictions are activated after touchdown and disconnect of the autopilot.
1) NO AUTOLAND is displayed on ASAs.
2) A pitch mode fail (EFIS mode - out) is displayed on EADIs.
3) A discrete amber AUTOPILOT light on panel P1 illuminates.
4) Master caution light on panel P7 illuminates and the owl aural warning sounds.

F. Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) (Fig. 4)
(1) SAM Description
(a) Each SAM is part of a dual redundant system. The SAM performs the following functions:
1) Generates Mach/speed trim commmands in response to input Mach number and computed airspeed data.
2) Selects the active trim source; either the Flight Control Computers (FCCs), Mach/Speed function of the SAM, or the column-mounted trim switches.
3) Outputs both ARM and CONTROL electrical commands to the Stabilizer Trim Control Modules (STCMs).
4) Provides redundancy management, failure monitoring, and annunciation.



Stabilizer Trim and Elevator Asymmetry Limit Module (SAM)
Figure 4

5) Each module includes two parallel, non-identical microprocessors (ARM and CONTROL) that develop simultaneous trim commands. These commands are based on input data from the three trim sources.
6) The SAM provides logic circuits for system engage/disengage, stabilizer trim mode, and trim-source command selection. Other internal circuits are digital data ARINC 429 receivers, analog discrete input and output buffers, and cross-channel and self monitoring. Built-In-Test (BIT) fault indications are output to the Engine indicator and Crew Alerting System (EICAS) display unit and annunciator panel.
7) The front panel of the SAM contains 12 fault balls that indicate specific failures within the stabilizer trim system. All except the SAM fault ball are set by faults identified within the CONTROL microprocessor. The SAM fault ball is set by either ARM or CONTROL microprocessor. Fault balls cannot be set if the SAM fault ball is set. The fault balls are black when not set, and yellow when set. Fault balls do not latch for input faults when airplane is on ground. The following fault ball indications are provided:
   a) FCC - set by a left, center, or right flight control computer input fault.
   b) ADC - set by a left or right Air Data Computer (ADC) fault discrete, or ADC input comparator fault.
   c) STCM - set by a fault detected by hydraulic pressure switch monitor, loop function monitor or unscheduled trim monitor. (Fault ball can only be set if either of the hydraulic pressure discrete from the associated hydraulic system pressure switch indicates high hydraulic pressure.)
   d) COL trim SW - set by a manual-trim coincidence monitor detected failure (manual electric trim switches fault).
   e) MAN LEVER SW - set by a standby manual-trim lever switch fault during power-up test (the trim lever operated at power-up or the switch is closed).
   f) FSPM/SPM - set by a failed output from the stabilizer position monitor for more than 2 seconds.
   g) AIR-GND RELAY - set by first or second air/ground fault.
   h) L. AIL LOCK ACT/ELEV ASY L. ACT - set by an elevator assymmetry limit actuator fault.
   i) R. AIL LOCK ACT/FSEU - set by a first or second flaps retracted fault.
   j) HYD PRESS SW - set by a system hydraulic pressure switch fault.

8) The SAM monitor sets the SAM faultball when the following faults are detected:
   a) Channel code discrete and airplane type code discrete faults (If discrepancy persists for more than 2 seconds).
   b) Mode coincidence, background, and ARM/CONTROL watchdog faults
   c) Elevator asymmetry limit actuator and power-up test faults
9) The SAM is able to reset itself if it is not in control and a SAM fault occurs. The SAM can also reset itself if a SAM fault has occured and the other SAM picks up control. If a SAM fault occurs again within one minute of an autoreset, the SAM will not be able to perform another autoreset. The SAM fault ball can reset itself if an autoreset is successful and no other SAM faults occur within one minute.

(b) When pressed, the RESET switch on the front panel of the SAM clears (resets) all fault balls, unless the fault is still present.

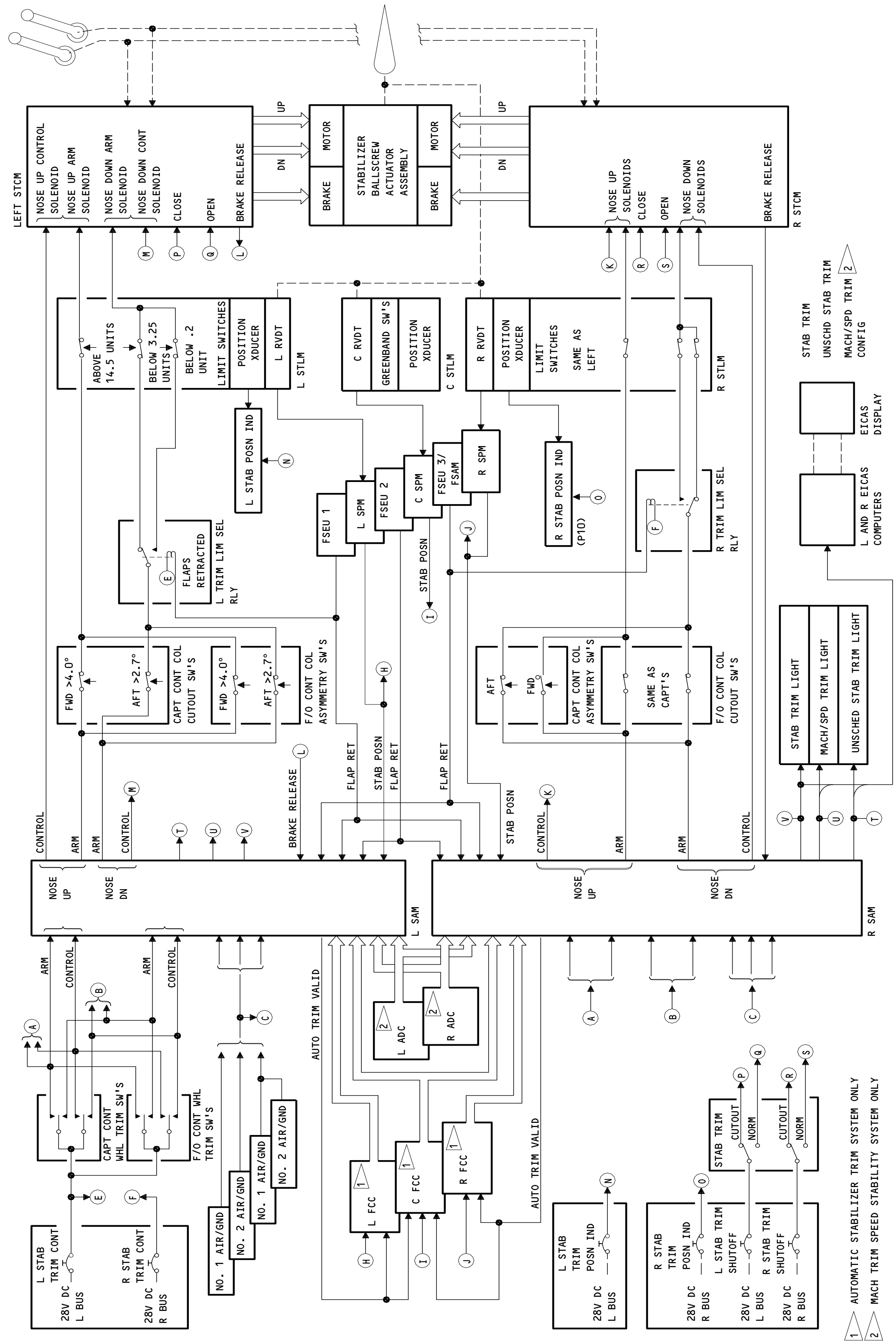
G. Stabilizer Trim System Block Diagram (Fig. 5)

(1) Stab Trim Manual Control

(a) The stabilizer ball-screw actuator consists of two independent hydraulic motors and brakes coupled to a mechanical drive device. The assembly moves the stabilizer at half-rate trim when powered by either the left or right Stabilizer Trim Control Module (STCM). Pressure supplied simultaneously from both STCMs produces full-rate trim. The brakes prevent air loads on the stabilizer from either back driving or overruning the actuator.

(b) Each STCM has an ARM and a CONTROL hydraulic valve. The manual stabilizer trim controls (P10) are connected to the STCMs. One control activates the ARM valve in both STCMs and the other control activates the CONTROL valves. Actuation of the manual trim controls trim the stabilizer at full rate and overrides all electric trim commands.

(c) The arm lever is equipped with a switch that transmits a discrete signal to the SAM when the trim levers are engaged. This synchronizes the Mach/speed trim function to the stabilizer position commanded by the manual control system.



Stabilizer Trim System Block Diagram
Figure 5

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(d) When the electric trim switches are engaged, both trim channels are activated, and the stabilizer is driven at full-rate trim. The STCM responds to electrical trim commands by applying hydraulic pressure to the motors on the actuator. A hydraulic brake on the actuator assembly is released by the same hydraulic pressure. A hydraulic pressure signal from the elevator feel computer modulates the flow rate of fluid to the actuator via a rate valve within the STCM. The flow rate is reduced at high speed and at aft center of gravity. This varies the trim rate between 0.1 and 0.25 degree/sec for one active STCM, and between 0.2 and 0.5 degree/sec for two active STCMs. Between high and low trim rates, the rate varies linearly as a function of elevator feel pressure. Actuation of the manual electric trim switches overrides all other electric trim modes, except when in Autoland mode.

(2) Cutout, Asymmetric and Stabilizer Limit Switch Activation

(a) Each control column is coupled to a set of four switches that enable electric trim commands to be interrupted (solenoid valve voltage removed) by column action. A column forward action opens the switch in the airplane nose-up stabilizer command circuit. A column aft action opens the switch in the airplane nose-down command circuit. Limit switches remove solenoid valve voltage when stabilizer position limits are reached. Two swithes are implemented such that if the columns are operated asymmetrically, as would occur after a system jam, both trim modules remain operative. This allows full rate trim capability after a jam.

(b) The forward and aft control column cutout switches and the nose-up and nose-down limit switches operate as follows:

1) A control column forward movement from neutral greater than 4.0 degrees opens the nose-up ARM inputs to the STCMs. If the leading edge stabilizer position is more than 14.5 units, the limit switch opens the nose-up ARM input to the STCM.

2) A control column aft movement from neutral greater than 2.7 degrees opens the nose-down ARM inputs to the STCMs. The limit switch opens the nose-down ARM input to the STCM if the leading edge stabilizer position is less than +3.25 units (flaps retracted) or +0.2 units (flaps extended).

3) Two asymmetric switches on each column are also wired in series with the up and down ARM channel output commands from both SAMs. Either set of trim switches are provided even when one column is jammed past the cutout switch interrupt point. The asymmetric switches allow up to 24 degrees left to right elevator asymmetry for computed airspeed less than 194 ±3 knots. The switches allow up to 8 degrees asymmetry for computed airspeed greater than 210 ±3 knots.

(c) Stabilizer mechanical stops are at +4 and -11.5 degrees. These limits control the maximum range that the stabilizer may travel.

2. Operation

A. Functional Description

(1) Automatic Stabilizer Trim Block Diagram (Fig. 5)

(a) The purpose of the Flight Control Computers (FCCs) in the stabilizer trim system is to provide trim commands when at least one channel of the autopilot is engaged. All three FCCs may be used; any one when in the cruise mode, or two or more when in the autoland mode. The left and right FCCs provide inputs to their respective SAMs. The center FCC provides inputs to both SAMs. Only one SAM is engaged at any time, with the first FCC engaged selected as the autotrim source. Priority for simultaneous FCC engagement is left, center, right.

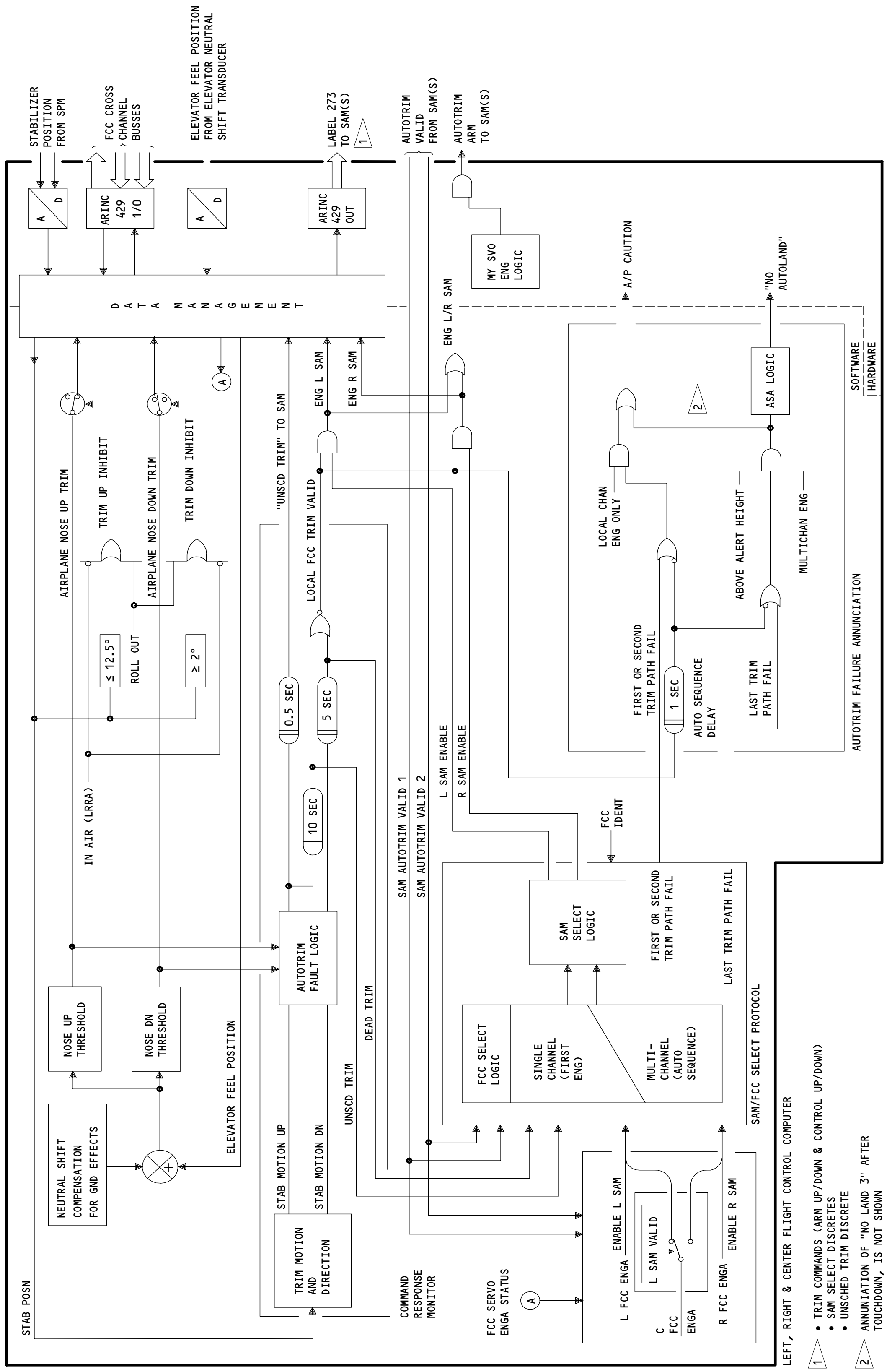
(b) The FCCs generate trim commands in response to an elevator deflection exceeding a preset authority threshold. Trim commands applied to the SAM cause the stabilizer to assume the elevator trim load. Stabilizer position signals are fed back to the FCCs from the associated Stabilizer Position Modules (SPM). Trim commands stop when the elevator is within the elevator authority threshold limits.

(c) The FCC generates a digital trim engage discrete and a separate analog autotrim discrete. These cause the SAM to select the FCC as the active trim source. With any one FCC controlling, a single STCM hydraulic output drives the stabilizer motor at half-rate speed (0.1 degrees/sec to 0.25 degrees/sec). The elevator feel computer determines the pressure applied at the STCM, which determines stabilizer rate.

(2) Automatic Stabilizer Trim FCC Logic (Fig. 6)

(a) Autotrim FCC Operation

1) The FCC provides trim commands when the autopilot is engaged and manual trim mode is not selected. The engaged FCC generates a digital trim-engage discrete and a separate analog autotrim arm discrete. The ENG L SAM or ENG R SAM discretes are separate bits in ARINC 429 word 273. The AUTOTRIM ARM signal is the analog discrete. Both signals are provided to each channel of the SAM. When the SAM accepts the autotrim engage request, an AUTOTRIM VALID analog discrete signal is transmitted to the FCC.



Automatic Stabilizer Trim FCC Logic
Figure 6

2) The three FCCS communicate with the SAM via ARINC 429 buses. Protocol logic within each FCC determines which computer trims the stabilizer. The FCCS monitor each other. In the autotrim mode, the command response monitor in FCC performs loop monitoring to detect autotrim failures. Autotrim failure occurs when stabilizer does not respond to computed commands, stabilizer movement is inconsistent with computed commands or when analog AUTOTRIM VALID from SAM becomes invalid. When a single autopilot is engaged, a detected autotrim failure which cannot be cleared by internal autosequencing (center FCC only) disables the active trim control path by disengaging the appropriate SAM. Since the center FCC interfaces with both SAMs, sequencing to another trim channel following a trim failure can occur only if the center FCC is engaged. When the center autopilot is engaged, the left SAM is always selected. A subsequent trim failure requires the FCC to inititate sequencing to the right SAM. When more than one autopilot is engageed, sequencing to another trim channel is attempted when the current SAM-driving autopilot disconnects, or the active trim control path fails as detected by the local command-response monitor. The FCC autosequencing cycle is left - Right - Center - Left.

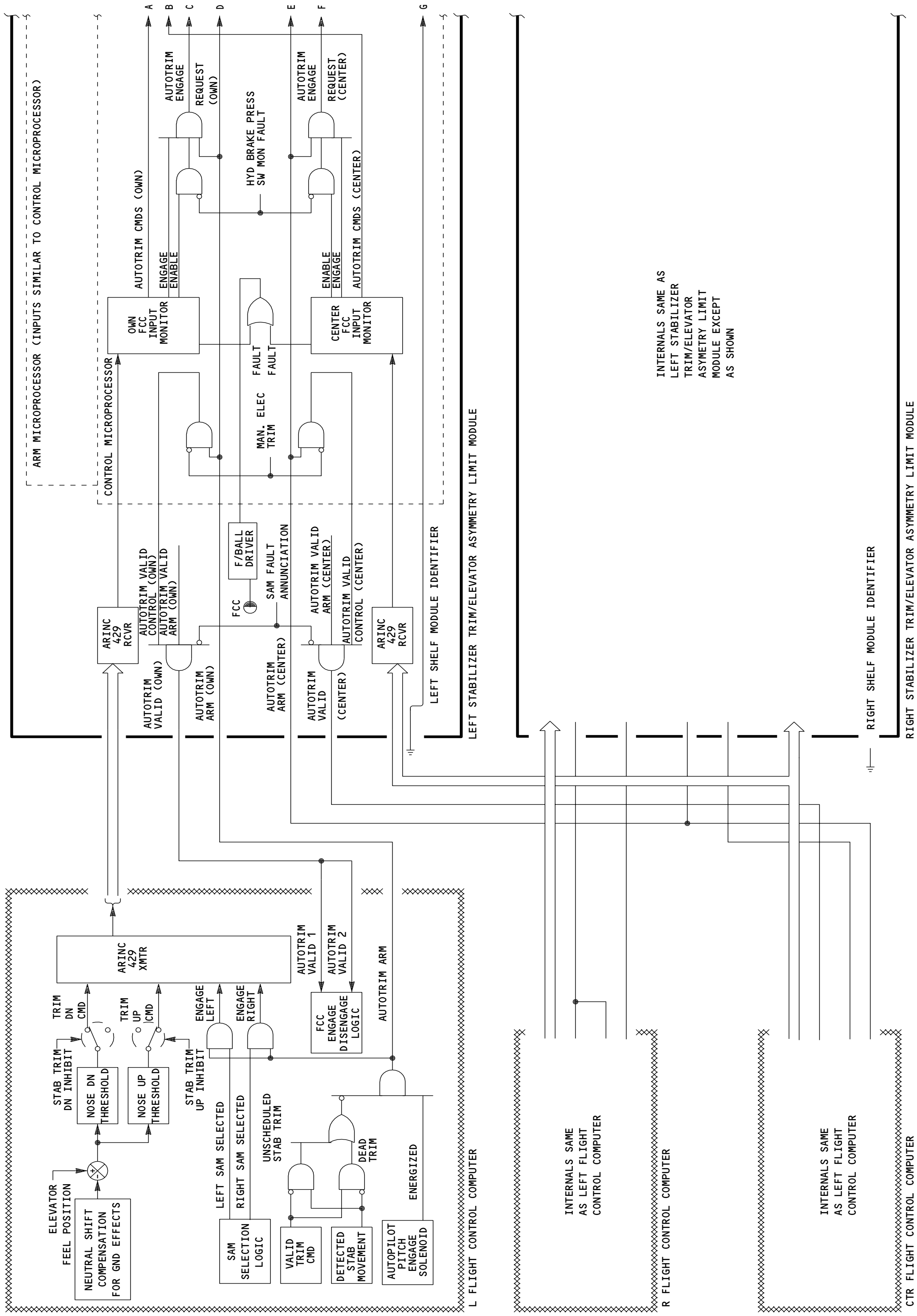
3) Elevator feel position and neutral shift compensation for ground effects determine trim commands. Trim up or trim down ARM and CONTROL signals are digitally outputed to the SAM. Trim motion is then monitored by command - response monitor in each FCC. If the TRIM VALID signal is not valid, AUTOTRIM ARM, and ENG R SAM or ENG L SAM signals are not transmitted.

(3) Automatic Stabilizer Trim Schematic (Fig. 7)

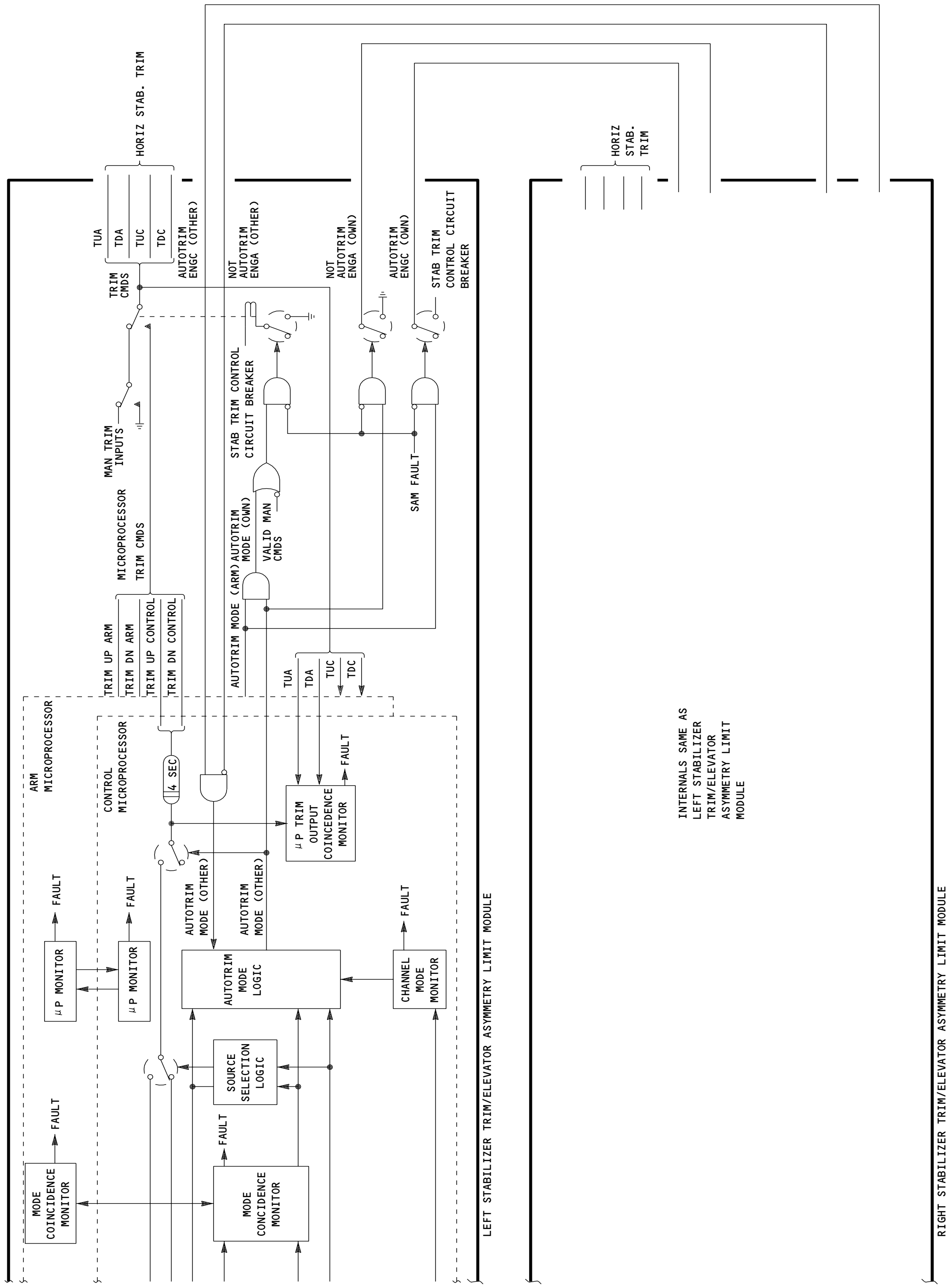
(a) Autotrim SAM Operation

1) The SAM contains ARM and CONTROL microprocessors to form a redundant system. Each processor receives inputs from its corresponding FCC and center FCC. The AUTOTRIM ARM input is an analog discrete signal from the FCC. Digital inputs from the FCC are the following:

a) TRIM UP or TRIM DN ARM and CONTROL CMDS inputs



Automatic Stabilizer Trim Schematic
Figure 7 (Sheet 1)



Automatic Stabilizer Trim Schematic
Figure 7 (Sheet 2)

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b) Left or right SAM ENG inputs

2) If unscheduled stabilizer motion is detected by the corresponding SAM (stabilizer moves more than .3 degree in either direction without a valid FCC autotrim command), a ground is provided by the SAM to turn on the UNSCHED STAB TRIM light on panel P5. An UNSCHD STAB TRIM message is also displayed on EICAS display unit to alert the flight crews.
3) The FCC inputs are monitored to determine which FCCs are engaged. The input monitors also detect faults that set the FCC fault ball as follows:
   a) If four or more parity errors occur within eight successive samples, a fault is set for the corresponding FCC.
   b) Autotrim command data which is identified as FAILED by the ARINC 429 status matrix is discarded and no trim data substituted. If four or more autotrim command inputs within eight successive samples are indicated as FAILED, a fault is set for the corresponding FCC (detected by the ARINC 429 sign status matrix).
   c) If an autotrim command word is not received for 125 milliseconds for 2 consecutive frames, a fault is set for the corresponding FCC. Old data is used when new data is not received for one frame.
4) Input coincidence monitoring detects invalid autotrim commands. An invalid command is defined as non-coincidence of ARM and CONTROL autotrim digital discretes, coincidence of opposite direction ARM or CONTROL autotrim digital discretes, or an ARM or CONTROL autotrim command without ENGAGE digital discrete or analog ARM discretes. If an invalid command is received for two or more successive samples, a fault is set for the corresponding FCC.
5) An FCC fault is cleared in air if valid data is continuously received for two seconds from the corresponding FCC.

6) Autotrim inputs identified by the status matrix as TEST or NO COMPUTED DATA are ignored by the SAM. The corresponding autotrim and enable valid discretes are set invalid. The corresponding FCC is not considered as fail unless other monitors are tripped. If enable conditions are met, an AUTOTRIM VALID signal is transmitted to the engaged FCC.
7) Autotrim UP and DOWN (ARM and CONTROL) commands inputs are delayed five seconds. This allows for mode switching and intermittent trim signals. Autotrim commands for the stabilizer trim control modules are output via the autotrim output coincidence monitor. The monitor compares CONTROL and ARM trim commands. The SAM fault ball is set if a difference is detected for longer than two seconds.
8) The SAM monitor sets the SAM fault ball when the following faults are detected:
   a) Channel code discrete and airplane type code discrete faults (If discrepancy persists for more than 2 seconds).
   b) Mode coincidence, background, and ARM/CONTROL watchdog faults
   c) Elevator asymmetry limit actuator and power-up test faults
9) Autotrim commands have priority over Mach/speed commands. Only manual electric and manual trim commands override the autotrim commands. Manual electric trim will not override autotrim when all three FCCs are engaged. Use of manual trim levers while in autotrim mode will cause UNSCHED STAB TRIM annunciation.

AUTOMATIC STABILIZER TRIM SYSTEM - ADJUSTMENT/TEST

1. General
   A. The paragraphs that follow are an operational test and a system test of the Automatic Stabilizer Trim System. The operational test and the system test require no external test equipment.
      (1) The operational test makes sure that the FCC can command the Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) to move the horizontal stabilizer up or down. It makes sure the FCC-to-SAM interface operates correctly.
      (2) The system test makes sure that the interfaces between equipment items and the automatic stabilizer trim system operate correctly.
      (3) The MCDP is on shelf E3-2 of the main equipment center. It supplies switches for test control, test direction and it shows any failure. The MCDP front panel shows general operation instructions. A connection, from the MCDP, is in the control cabin for a hand-held MCDP remote control unit. The connector for the remote unit is on the forward side of the P61 panel. Use of the remote unit decreases test time. The remote unit lets messages displayed on the MCDP in the main equipment center to be shown on the bottom EICAS display unit.

TASK 22-22-00-715-011

2. Operational Test - Automatic Stabilizer Trim System

   A. General
      (1) This test makes sure that the auto trim mode of the Stabilizer Trim/Elevator Asymmetry Limit Modules (SAMs) operates. The test uses the Maintenance Control Display Panel Ground Test 65-STAB TRIM.
   B. Equipment
      (1) Remote Control Unit, Maintenance Control Display Panel -
          A22001-22 (preferred),
          A22001-15 (optional).
   C. References
      (1) 22-00-02/201, Autoflight Bite
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   D. Access
      (1) Location Zones

119/120	Main Equipment Center
211/212	Control Cabin
312/313	Stabilizer Center Section Compartment

      (2) Access Panel

119BL	Main Equipment Center

   E. Prepare for Test

      S 865-177
      (1) Supply electrical power (Ref 24-22-00).

S 865-001

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Pressurize the left, right, and center hydraulic systems (Ref 29-11-00).

S 285-013

(3) Make sure that these systems operate:
    (a) Horizontal Stabilizer Trim Control System (Ref 27-41-00).
    (b) Autopilot (Flight Control) (Ref 22-10-00).
    (c) Engine Indication and Crew Alerting System (Ref 31-41-00).

S 285-014

(4) Make sure that the control column is in the neutral position and not restrained.

S 865-015

(5) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
    (a) 11A33, INDICATOR LIGHTS 2
    (b) 11C6, CSEU 1L AC
        or
        FLT CONT ELEC 1L AC
    (c) 11C7, CSEU 1L DC
        or
        FLT CONT ELEC 1L DC
    (d) 11C8, CSEU 2L AC
        or
        FLT CONT ELEC 2L AC
    (e) 11C9, CSEU 2L DC
        or
        FLT CONT ELEC 2L DC

(f) 11C12, STAB TRIM SHUTOFF LEFT
(g) 11C13, STAB TRIM SHUTOFF RIGHT
(h) 11C14, FSEU 2 PWR
or
FLAP SLAT ELEC UNIT 2 POWER
(i) 11C15, FSEU-2 CONT
or
FLAP SLAT ELEC UNIT 2 CONT
(j) 11C16, FSEU-2 SENSOR
or
FLAP SLAT ELEC UNIT 2 SENSOR
(k) 11C30, LANDING GEAR POS SYS 1
(l) 11F19, STAB POS MOD CENTER
(m) 11G12, FSEU-1 PWR
or
FLAP SLAT ELEC UNIT 1 POWER
(n) 11G13, FSEU-1 CONT
or
FLAP SLAT ELEC UNIT 1 CONT
(o) 11G15, STAB POS MOD LEFT
(p) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(q) 11G18, CSEU 1R DC
or
FLT CONT ELEC 1R DC
(r) 11G21, FSEU-3 PWR
or
FLAP SLAT ELEC UNIT 3 POWER
(s) 11G22, FSEU-3 CONT, or
FLAP SLAT ELEC UNIT 3 CONT, or
FSAM CONTROL
(t) 11G23, FSEU-3 SENSOR, or
FLAT SLAT ELEC UNIT 3 SENSOR, or
FSAM SENSOR

(u) 11G24, STAB POS MOD RIGHT
(v) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC
(w) 11G28, CSEU 2R DC
or
FLT CONT ELEC 2R DC
(x) 11H10, STAB TRIM POS IND L
or
STAB TRIM LEFT POS IND
(y) 11H11 or 11C05, STAB TRIM CONT L
or
STAB TRIM LEFT CONT
(z) 11H19, STAB TRIM POS IND RIGHT
(aa) 11H20, STAB TRIM CONT RIGHT
(ab) 11P28, R IND LTS 1
(ac) 11S19, AIR/GND SYS 2

S 485-229
(6) If the remote MCDP is to be used, do the steps that follow:
(a) Install the MCDP remote control unit as follows:
1) Remove the dust cover from connector D41101J found behind the P6-5 panel.
2) Connect the MCDP remote control unit to connector D41101J.
(b) Make sure the EICAS circuit breakers on the P11 panel are closed.
(c) On the EICAS maintenance panel on the P61 panel, push the CONF/MCDP switch to show MCDP data on the EICAS bottom display.

NOTE: The same messages that show on the MCDP in the main equipment center will show on the EICAS bottom display. The indication MCDP FLT FAULTS or MCDP GRD TEST shows to identify which mode the MCDP is in.

(d) Make sure EICAS bottom display shows the message MCDP OFF.

S 715-230
(7) Do MCDP Power-Up and Self Test (AMM 22-00-02/201).

F. Test Automatic Stabilizer Trim System

S 285-018
(1) Make sure that the MCDP shows the message LAST FLT FAULTS?.

S 985-019
(2) Push the GRD TEST switch and make sure that the GRD TEST mode is engaged.

S 985-020
(3) Use the GRD TEST SEL UP and DOWN switches or the NO/SKIP switch to set the MCDP for test 65-STAB TRIM.

S 285-021
(4) Make sure that the display shows the message 65-STAB TRIM TEST?.

S 985-022
(5) Push the YES/ADV switch to arm the test.

S 285-023
(6) Make sure that the MCDP shows the message 65 IN PROGRESS.

S 285-024
(7) Make sure that the MCDP shows the message 65 VFY HYD ON - ADV.

S 985-025
(8) Operate the YES/ADV switch.

S 285-026
(9) Make sure that the MCDP shows the message 65 A/P ENG L TO CMD.

S 985-180
(10) GUI 115;
Put the A/P ENGAGE L switch on the AFCS mode control panel in the CMD position.

S 985-183
(11) GUI 001-114, 116-999;
Push the L CMD button on the AFCS mode control panel.

S 285-031
(12) Make sure that the MCDP shows the message 65 IN PROGRESS.

S 285-032
(13) Make sure that this sequence of stabilizer indicator and stabilizer movement occurs:
- (a) The stabilizer indicator is in the TO CG position.
- (b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
- (c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.

(d) The stabilizer indicator moves back into the TO CG position.

S 285-033
(14) Make sure that the MCDP shows the message 65 A/P ENG R TO CMD.

S 985-185
(15) GUI 115;
Put the A/P ENGAGE R switch on the AFCS mode control panel in the CMD position.

S 985-187
(16) GUI 001-114, 116-999;
Push the R CMD button on the AFCS mode control panel.

S 285-038
(17) Make sure that the MCDP shows the message 65 IN PROGRESS.

S 285-039
(18) Make sure that this sequence of stabilizer indicator and stabilizer movement occurs:
(a) The stabilizer indicator is in the TO CG position.
(b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
(c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.
(d) The stabilizer indicator moves back into the TO CG position.

S 285-040
(19) Make sure that the MCDP shows the message 65 A/P ENG C TO CMD.

S 985-188
(20) GUI 115;
Put the A/P ENGAGE C switch on the AFCS mode control panel in the CMD position.

S 985-189
(21) GUI 001-114, 116-999;
Push the C CMD button on the AFCS mode control panel.

S 285-045
(22) Make sure that the MCDP shows the message 65 IN PROGRESS.

S 285-046
(23) Make sure that this sequence of stabilizer indicator and stabilizer movement occurs:
(a) The stabilizer indicator is in the TO CG position.
(b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
(c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.
(d) The stabilizer indicator moves back into the TO CG position.

S 285-047
(24) Make sure that the MCDP shows the message TEST COMPLETE.

G. Put the Airplane back to its initial condition.

S 865-048
(1) Remove pressure from the left, right, and center hydraulic systems (Ref 29-11-00).

S 865-049
(2) Remove electrical power if it is not necessary.

TASK 22-22-00-725-050

3. System Test - Automatic Stabilizer Trim System

A. General
(1) The system test makes sure that the interfaces between equipment items and the automatic stabilizer trim system operate correctly.

B. References
(1) 22-00-02/201, Autoflight Bite
(2) 22-10-00/501, Autopilot (Flight Control)
(3) 24-22-00/201, Electrical Power - Control
(4) 27-41-00/501, Horizontal Stabilizer Trim Control System
(5) 27-51-00/201, Trailing Edge Flap System
(6) 27-61-00/201, Spoiler/Speedbrake Control System
(7) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(8) 31-41-00/501, Engine Indication and Crew Alerting System
(9) 34-12-00/501, Air Data Computing System
(10) 34-21-00/501, Inertial Reference System

C. Access
(1) Location Zones

119/120	Main Equipment Center
211/212	Control Cabin
312/313	Stabilizer Center Section Compartment

(2) Access Panel

119BL	Main Equipment Center

D. Prepare For Test

S 865-002
(1) If any fault indications occur accidentally during any of the tests that follow, the SAM must be reset. Use any of these actions as necessary.
(a) Push the reset button on the front panel of the SAM

(b) Open and close these circuit breakers on the P11 overhead circuit breaker panel:
   1) 11C6, CSEU 1L AC
      or
      FLT CONT ELEC 1L AC
   2) 11C8, CSEU 2L AC
      or
      FLT CONT ELEC 2L AC
   3) 11G17, CSEU 1R AC
      or
      FLT CONT ELEC 1R AC
   4) 11G27, CSEU 2R AC
      or
      FLT CONT ELEC 2R AC

S 865-054

(2) When you push the RESET switch on the front panel of the SAM it clears (resets) all faultballs, unless the fault is still present.

S 845-004

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 845-007

WARNING: DO THE DEACTIVATION PROCEDURE FOR FLIGHT MODE SIMULATION BEFORE YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS. WHEN YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS THE AIRPLANE IS IN FLIGHT MODE. IN FLIGHT MODE, MANY OF THE AIRPLANE SYSTEMS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Do the deactivation procedure for flight mode simulation (Ref 32-09-02).

S 845-008

(5) Put the airplane back to the ground mode (Ref 32-09-02).

S 865-060

(6) The FSEU faultball will reset with the next flap cycle.

S 715-051

(7) Do the Operational Test - Automatic Stabilizer Trim System.

S 865-150

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(8) Pressurize the left, right, and center hydraulic systems (Ref 29-11-00).

S 985-063

(9) Operate the control wheel trim switches (either captain's or first officer's) to put the stabilizer at approximately five units of trim.

S 985-064

(10) Push the ECS/MSG switch on EICAS maintenance panel on P61.

S 985-065

(11) Put the computer selector switch on EICAS display select panel in the L position.

E. Test Maintenance Interface

S 285-066

(1) Make sure that the lower EICAS display unit does not show the STAB TRIM message.

S 985-067

(2) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-068

(3) Make sure that the lower EICAS display unit does not show the STAB TRIM message.

S 845-003

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-069
(5) Open these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 985-070
(6) Put the C and R STAB TRIM switches on the control stand, P10, in the CUT OUT position.

S 865-169
(7) Hold the alternate stab trim switches on the control stand, P10, out of the neutral position.

S 865-072
(8) Close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC
(c) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 285-073
(9) Make sure that the STAB TRIM message is shown on the lower EICAS display unit after 30 seconds.

S 985-074
(10) Put the computer selector switch on the EICAS display select panel in the L position.

S 285-075
(11) Make sure that the STAB TRIM message is shown on the lower EICAS display unit.

S 865-170
(12) Move the alternate stab trim switches back to the neutral position.

S 985-077
(13) Put the C and R STAB TRIM switches in the NORM position.

F. Test the STAB TRIM Fault Light and the Stabilizer Trim Control Module (STCM) Brake Switch Interface

S 985-078
(1) Operate the control wheel trim switches to put the stabilizer to approximately four units of trim.

S 845-006

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-079
(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC

(b) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 985-080
(4) Put the R STAB TRIM switch in the CUT OUT position.

S 285-081
(5) Make sure that the yellow STAB TRIM light on the overhead panel, P5, and the STAB TRIM message on the upper EICAS display unit are out.

S 985-082
(6) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-083
(7) Make sure that the yellow STAB TRIM light on the P5 panel is out.

S 285-084
(8) Make sure that the STAB TRIM message on the upper EICAS display unit is out.

S 985-085
(9) Operate the control wheel trim switches in the up direction.

S 285-086
(10) Make sure that the yellow STAB TRIM light comes on and the upper EICAS display unit on the pilot's center instrument panel shows the STAB TRIM message.

S 985-087
(11) Put the R STAB TRIM switch in the NORM position.

S 985-088
(12) Operate the control wheel trim switches in the up direction.

S 285-089
(13) Make sure that the yellow STAB TRIM light and message are off.

S 865-090
(14) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11G17, CSEU 1R AC
or
FLT CONT ELEC 1R AC
(b) 11G27, CSEU 2R AC
or
FLT CONT ELEC 2R AC

S 845-005

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(15) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoiler panels.

S 865-091
(16) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

S 985-092
(17) Put the C STAB TRIM switch in the CUTOUT position.

S 985-093
(18) Put the computer selector switch on the EICAS display panel in the L position.

S 985-094
(19) Operate the control wheel trim switches in the down direction.

S 285-095
(20) Make sure that the yellow STAB TRIM light comes on and the upper EICAS display unit shows the STAB TRIM message.

S 985-096
(21) Put the C STAB TRIM switch in the NORM position.

S 985-097
(22) Operate the control wheel trim switches in the down direction.

S 285-098
(23) Make sure that the yellow STAB TRIM light and the STAB TRIM message are off.

S 985-099
(24) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC
or
FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC
or
FLT CONT ELEC 2L AC

G. Test Auto Trim Mode Crossfeed Interface

S 285-100
(1) Make sure that the Inertial Reference System is initialized and operational (Ref 34-21-00).

NOTE: Initialization of IRS requires about 10 minutes.

S 985-101
(2) Operate the control wheel trim switches to put the stabilizer to approximately seven units of trim.

S 985-102
(3) Put the F/D switch on the mode control panel in the ON position to start the Flight Director.

S 985-103
(4) Push the V/S switch/light on the mode control panel.

S 985-110
(5) Put the vertical speed dial to -2000 ft/min.

S 985-193
(6) GUI 115;
Put the A/P ENGAGE L switch on the AFCS mode control panel in the CMD position.

S 985-196
(7) GUI 001-114, 116-999;
Push the L CMD button on the AFCS mode control panel.

NOTE: GUI 001-009;
If the flight director is switched to the FCC that is engaged, the flight director bars will not show on the EADI display.

S 285-009
(8) Make sure that these actions occur:
(a) the stabilizer trim position indicator shows stabilizer movement
(b) the yellow UNSCHED STAB TRIM light is off
(c) the UNSCHED STAB TRIM message is not shown on the upper EICAS display unit.

S 985-112
(9) Push the autopilot wheel disconnect switch to disengage the autopilot.

S 985-113
(10) Push the A/P disconnect switch a second time to cancel aural and visual alerts.

S 985-114
(11) Put the computer selector switch on the EICAS display select panel in the R position.

S 985-227
(12) Put the FLAP handle at the 5 degree position.

S 985-115
(13) Operate the control wheel trim switches to put the stabilizer to approximately two units of trim.

S 985-122
(14) Put the vertical speed dial on the mode control panel to +2000 ft/min.

S 985-200
(15) GUI 115;
Put the A/P ENGAGE R switch on the AFCS mode control panel in the CMD position.

S 985-204
(16) GUI 001-114, 116-999;
Push the R CMD button on the AFCS mode control panel.

S 285-010
(17) Make sure that these actions occur:
(a) the stabilizer trim position indicator shows stabilizer movement
(b) the yellow UNSCHED STAB TRIM light is off
(c) the UNSCHED STAB TRIM message is not shown on the upper EICAS display unit.

S 985-124
(18) Push the autopilot wheel disconnect switch twice to disengage the autopilot and cancel the alerts.

S 985-125
(19) Put the vertical speed dial to 0000.

H. Test Unscheduled Trim Interface

S 985-126
(1) Operate the control wheel trim switches to put the stabilizer to approximately six units of trim.

S 865-171
(2) Make sure that the stab trim switches on the control stand, P10, are in the neutral position.

S 285-128
(3) Make sure that the yellow UNSCHED STAB TRIM light on the overhead panel, P5, is off.

S 985-207
(4) GUI 115;
Put the A/P ENGAGE L switch on the AFCS mode control panel in the CMD position.

S 985-210
(5) GUI 001-114, 116-999;
Push the L CMD button on the AFCS mode control panel.

S 865-172
(6) Move the alternate stab trim switches forward.

S 285-134
(7) Make sure that the yellow UNSCHED STAB TRIM light is on.

S 285-135
(8) Make sure that the UNSCHD STAB TRIM message is shown on the upper EICAS display unit.

S 865-173
(9) Move the alternate stab trim switches back to the neutral position.

S 285-137
(10) Make sure that the UNSCHED STAB TRIM light and message are off.

S 985-138
(11) Push the autopilot wheel disconnect switch twice to disengage the autopilot and cancel the alerts.

S 985-139
(12) Put the computer select switch on the EICAS display select panel in the L position.

S 985-213
(13) GUI 115;
Put the A/P ENGAGE R switch on the AFCS mode control panel in the CMD position.

S 985-216
(14) GUI 001-114, 116-999;
Push the R CMD button on the AFCS mode control panel.

S 865-174
(15) Move the alternate stab trim switches aft.

S 285-151
(16) Make sure that the yellow UNSCHED STAB TRIM light is on.

S 215-178
(17) Make sure that the UNSCHD STAB TRIM message is shown on the upper EICAS display unit.

S 865-175
(18) Move the alternate stab trim switches back to the neutral position.

S 285-146
(19) Make sure that the UNSCHED STAB TRIM light and message are off.

S 985-147
(20) Push the autopilot wheel disconnect switch twice.

I. Put the Airplane back to its initial condition.

S 985-228
(1) Put the FLAP handle to the 0 degree position.

S 085-231
(2) Do these steps to remove the MCDP remote control unit if it is installed:
(a) Disconnect the MCDP remote control panel from connector D41101J.

(b) Install the dust cover on connector D41101J.

S 865-148

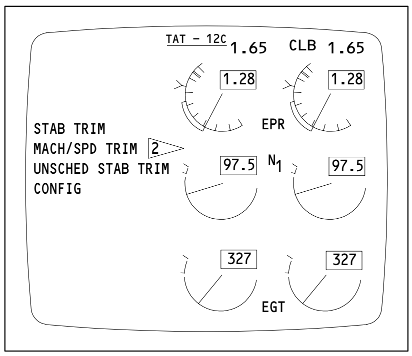
(3) Remove pressure from the left, right, and center hydraulic systems (Ref 29-11-00).

S 865-149

(4) Remove electrical power if it is necessary (Ref 24-22-00).

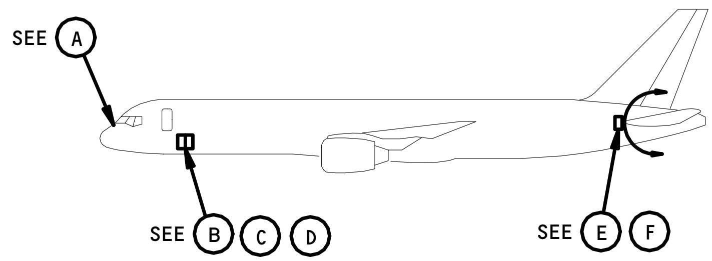
MACH TRIM/SPEED STABILITY - DESCRIPTION AND OPERATION

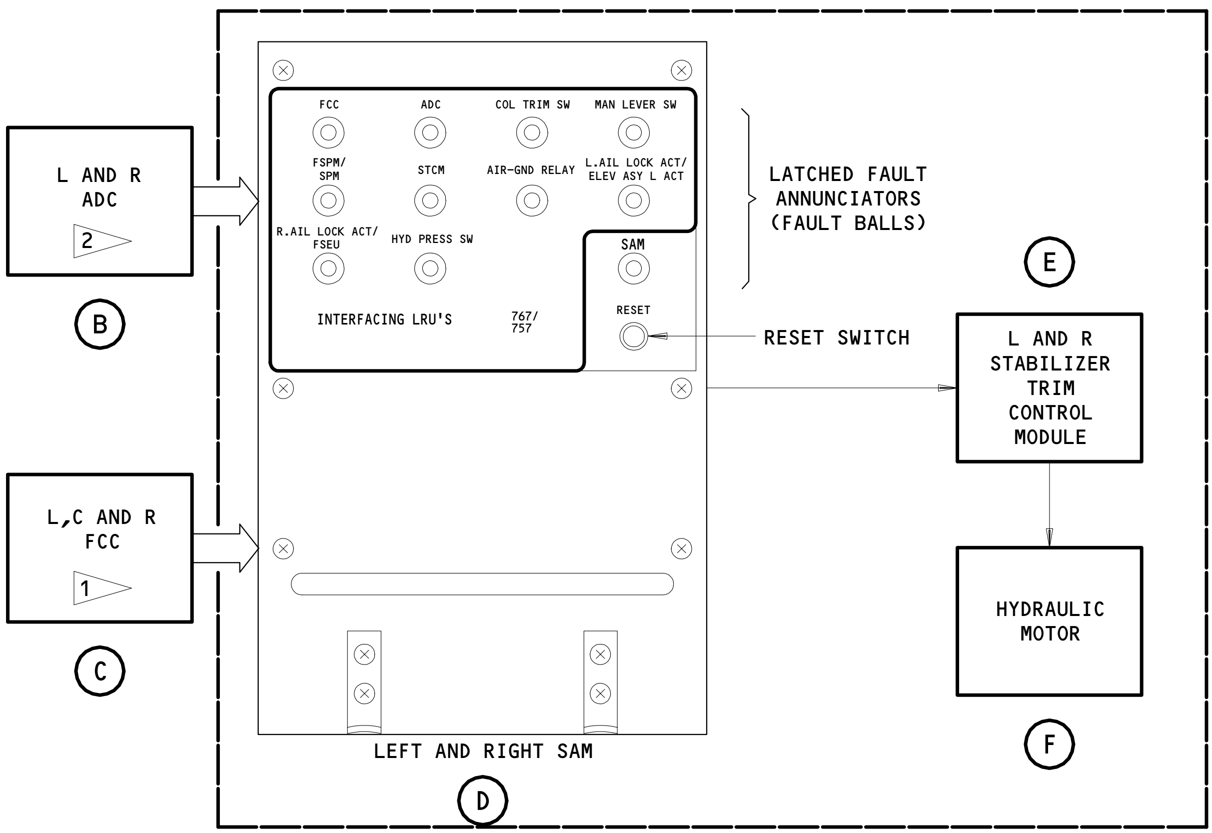
1. General (Fig. 1)
   A. The Mach trim/speed stability mode of the stabilizer trim system provides incremental horizontal stabilizer position in response to changes of Mach and airspeed. The mode is only engaged when airborne (after a 20 second delay) and the manual and autotrim functions are not engaged.
   B. This chapter contains Mach trim/speed stability operation. Refer to 22-22-00/001 for a general description of the Automatic Stabilizer Trim System. Refer to 27-41-00/001 for the component details and trim system operation of the Horizontal Stabilizer Trim Control System.
2. Operation
   A. Functional Description
      (1) Mach Trim Speed Stability Block Diagram (Fig. 2)
         (a) The Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) selects the Mach/speed function as the active trim command source when:
            1) The flight control computer (autotrim) function is not used.
            2) The control wheel mounted manual electric trim switches are not activated.
         (b) When an FCC or Mach/speed function is the trim command source, only one trim channel (left or right) is operated. The controlling SAM is randomly selected at power up. This provides half-rate trim (0.1 to 0.25 degrees/sec). During Mach/speed operation, the stabilizer position is related to airspeed or Mach. The SAM determines the desired stabilizer position command that operates the Stabilizer Trim Control Module (STCM) and ball-screw actuator. The SAM uses the following inputs to determine ARM and CONTROL output commands to the STCM:
            1) Digital airspeed and Mach data from the Air Data Computers (ADCs).
            2) Flap position (up or down) from the Flap/Slat Electronic Units (FSEUs) and Flap/Slat Accessory Module (FSAM) (if installed).
            3) Stabilizer position from the Rotary Variable Differential Transformer Transducers (RVDTs) via the Stabilizer Position Modules (SPMs).
         (c) If the Mach/speed function is operational, flap position determines which control law (Mach trim or speed stability) is used. When the flaps are extended, speed stability is in control. When the flaps are retracted, Mach trim is in control.
      (2) Mach Trim/Speed Stability Control Law Schedules (Fig. 3)



EICAS DISPLAY UNIT

A



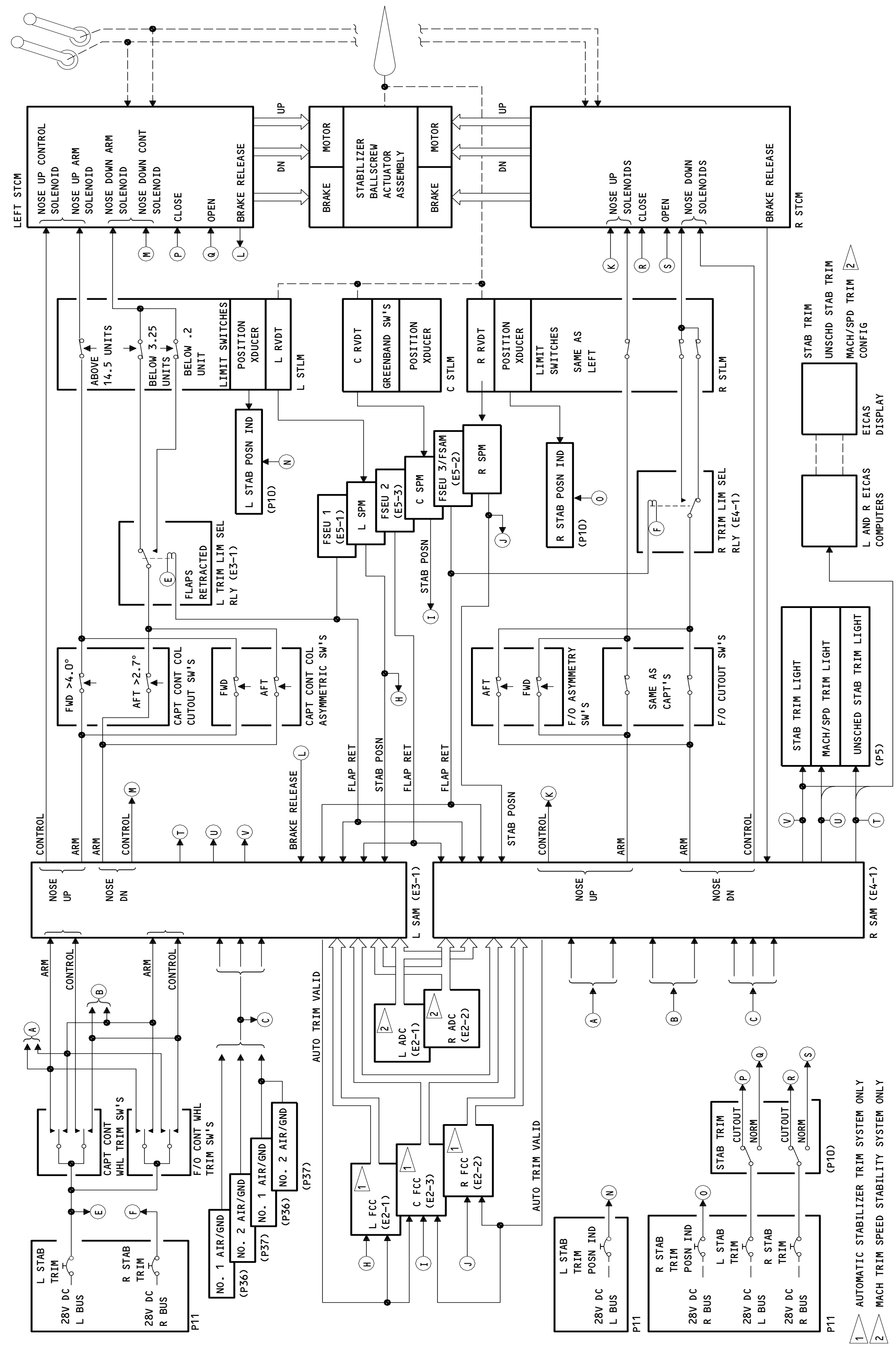


STABILIZER TRIM SYSTEM

1 AUTOMATIC STABILIZER TRIM SYSTEM ONLY

2 MACH SPEED STABILITY SYSTEM ONLY

Mach Trim/Speed Stability System
Figure 1

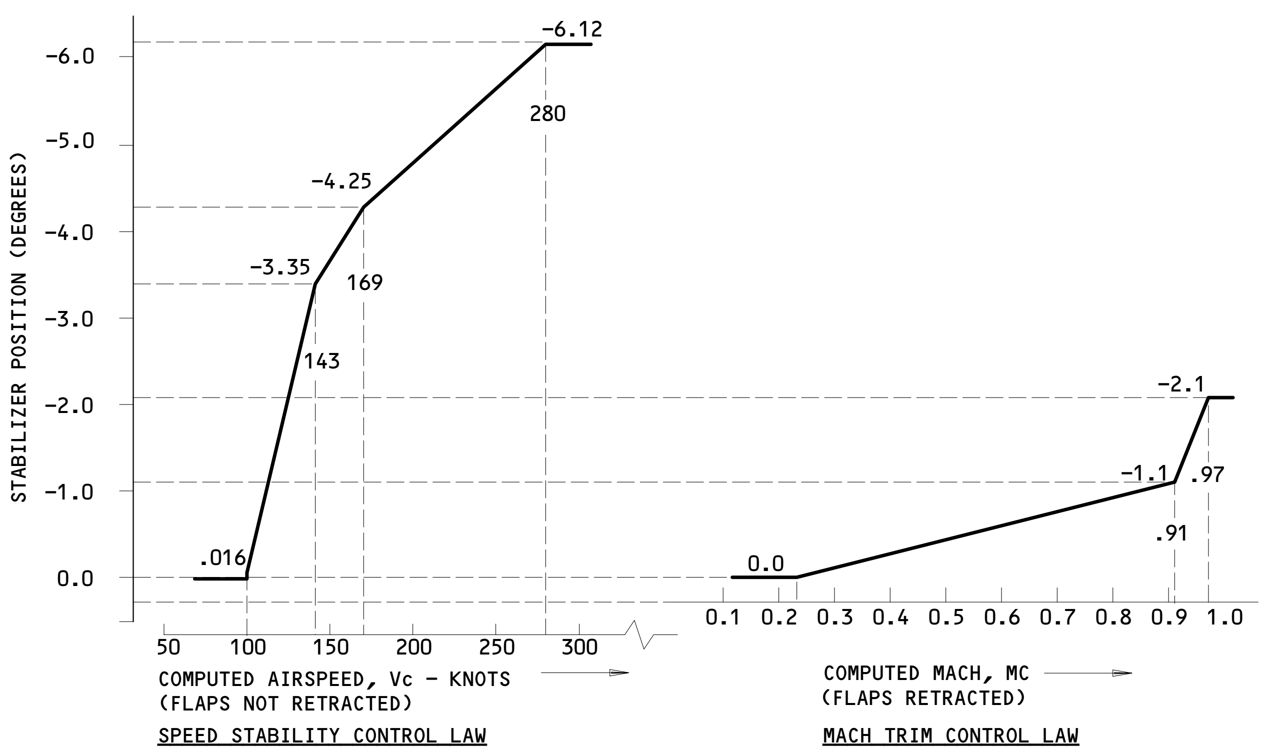


Mach Trim Speed Stability Block Diagram
Figure 2

28682

(a) Speed Stability Control Law

1) When the speed trim mode engages, the speed trim control law synchronizes to the stabilizer position. The control law provides incremental stabilizer position changes dependent upon computed airspeed-knots (Vc) when the flaps are extended. The Air Data Computers (ADCs) provide digital Vc inputs. Half-speed trim (0.1 to 0.25 degrees/sec) commands are provided by the selected SAM. The speed stability mode is disengaged when the airplane is on the ground and until 20 seconds into the in-air mode.



Mach Trim/Speed Control Law Schedule
Figure 3

2) After each trim event the stabilizer position change is equal to or above the speed control law (more negative) for increasing speeds, and equal to or below the control law (more positive) for decreasing speeds. {The threshold of speed change for the initial trim event following deviation from trim speed is nominally 5 knots. This is required to protect the system from unwanted trim activity caused by air stream turbulence. The threshold of speed change for subsequent trim events is approximately 2.5 knots. This requirement is based on the tending of the system to overtrim the schedule stabilizer position changes.

(b) Mach Trim Control Law

1) When the Mach trim mode engages, the Mach trim control law synchronizes to stabilizer position. When the flaps are retracted, the control law provides incremental stabilizer position changes dependent upon computed Mach (Mc). The ADCs provide digital Mc inputs. Half-speed trim commands are provided by the selected SAM. The Mach trim mode is disengaged on-ground and until 20 seconds into the in-air mode.

2) After each trim event the stabilizer position change is equal to or above the Mach trim control law (more negative) for increasing Mach and is equal to or below the control law (more positive) for decreasing Mach. The threshold of Mach change for the initial trim event following deviation from trim speed is nominally 0.0l Mach. The threshold of Mach change for subsequent trim events is approximately 0.005 Mach, based on system overtrim characteristics.

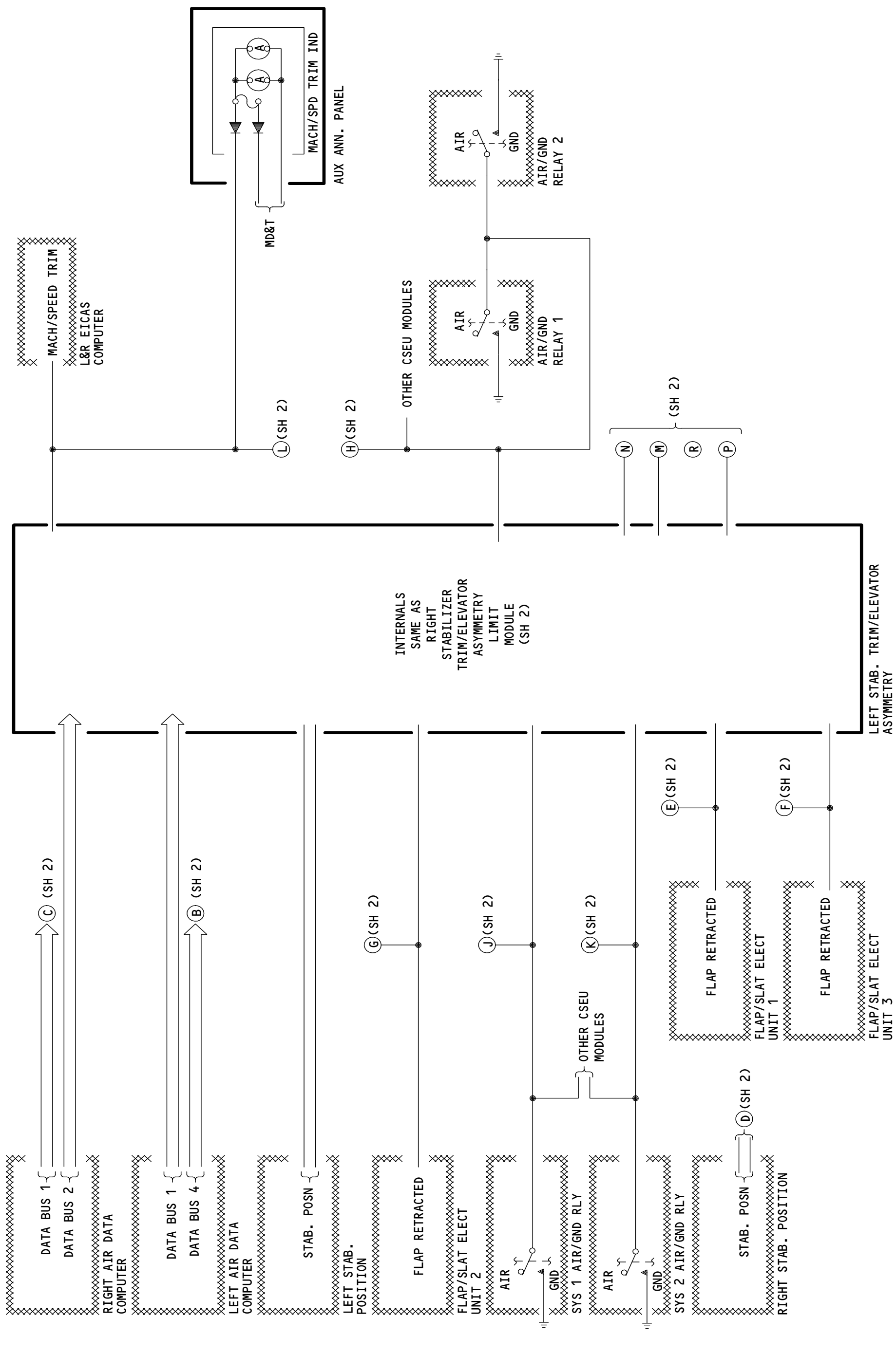
(3) Mach Trim/Speed Stability Schematic (Fig. 4)

(a) The Stabilizer Trim and Elevator Asymmerty Limit Module (SAM) uses the following inputs to determine Mach trim and speed stability commands.

1) The Air Data Computers (ADCs) provide digital airspeed and Mach inputs to both SAMs (Ref 34-12-00/001, Air Data Computing System).

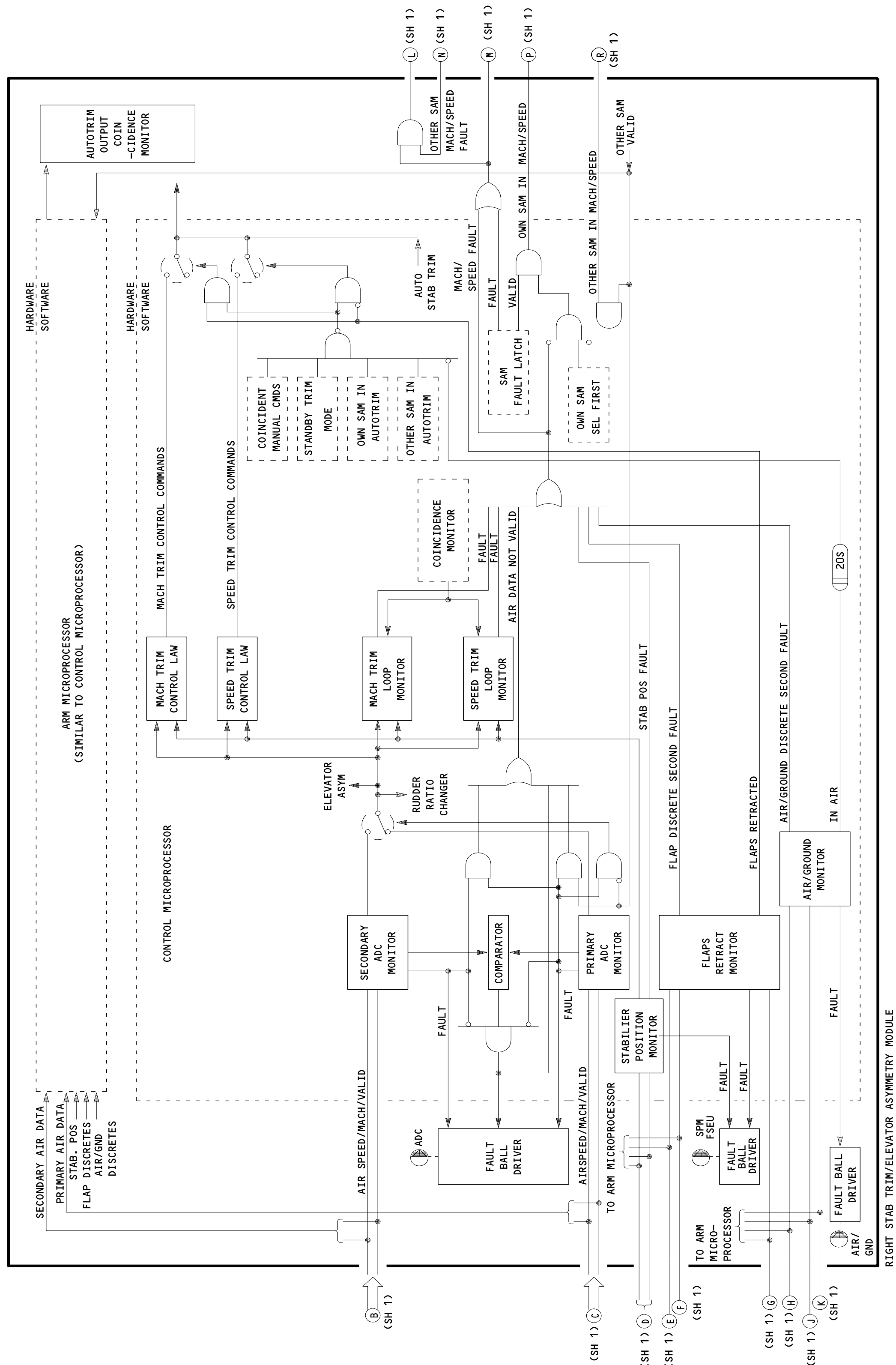
2) The Flap/Slat Electronic Units (FSEUs) and Flap/Slat Accessory Module (FSAM) (if installed) input a flap retracted signal to both SAMs (Ref 27-51-00/001, Trailing Edge Flap System). The left and right SPMs input stabilizer position to the corresponding SAM (Ref 27-48-00/001, Stabilizer Trim Positions Indicating System).

3) The air/ground relays provide airplane air/ground status. Each SAM receives three air/ground inputs from four relays (Ref 32-09-00/001, Landing Gear Multiple Use System/Components).



Mach Trim/Speed Stability Schematic
Figure 4 (Sheet 1)

S2595



Mach Trim/Speed Stability Schematic
Figure 4 (Sheet 2)

52623

(b) Each SAM contains ARM and CONTROL microprocessors to form a dual redundant system. Each processor receives digital inputs from the left and right ADCs. The inputs are Mach (Mc) and computed airspeed (Vc). Continuous monitoring is provided for both primary and secondary ADC inputs. The following input checks are provided:
  1) If a parity error is detected, the value is discarded and the previous valid value is used. Four or more parity errors within eight successive samples set an ADC fault.
  2) Data which is identified as FAILED by the ARINC 429 sign status matrix is discarded and the previous valid value is used. Four or more FAILED values of a parameter within eight successive samples sets an ADC fault.
  3) The Mc data identified as No Computed Data (NCD) by the sign status matrix is discarded. The value is replaced by 0.1 Mach. The Vc data identified as NCD is discarded and replaced by 30 knots.
  4) If either Mc or Vc is not received within 150 milliseconds for 2 consecutive frames, an ADC Fault is set. Old data is used when new data is not received within one frame.

(c) Cross comparison between primary and secondary Mc and Vc is provided. An ADC comparison fault is set if:
  1) Primary and secondary Mc differ by more than 0.05 Mach for more than five seconds.
  2) Primary and secondary Vc differ by more than 10 knots for more than five seconds.

(d) The cross comparison is inhibited if either ADC is invalid or in the TEST mode. An ADC fault is cleared if valid data is received continuously for 30 seconds and no more than four resets have occured.

(e) The flaps retract and air/ground monitor are majority vote (two out of three) monitors. Flaps-retract failures set the FSPM/SPM faultball. Air/ground failures set the AIR/GND fault ball.

(f) The stabilizer position is monitored and input to synchronize the control laws. A input level monitor checks for signal inputs less than +1.0 vdc or an open circuit. A detected failure sets the FSPM/SPM faultball. A stabilizer fault is cleared in the air if a valid input is received for five seconds and no more than four resets have occured.

(g) Mach or speed commands are generated in the control law. The command outputs depend on the following SAM modes and inputs:
  1) Flaps retracted for Mach command outputs.
  2) Flaps not retracted for speed command outputs
  3) Neither SAM in autotrim or manual trim modes (Ref 27-41-00/001, Horizontal Stabilizer Trim Control System).

(h) Stabilizer movement of more than 0.3 degree in either direction without a valid SAM command is detected by the Mach loop or speed loop monitor. An UNSCD TRIM discrete is output from the controlling SAM. The UNSCD TRIM discrete turns on the UNSCHED STAB TRIM light and causes the message UNSCHD STAB TRIM to appear on the upper EICAS display unit. The Mach loop or speed loop monitor is also active when the airplane is on ground.

(i) The internal monitors in each SAM determine a MACH/SPEED fault. The monitors detect the following conditions.

1) The stabilizer does not move in the correct direction within 3.5 seconds of a valid SAM command. (The loop fault latch resets after 5 seconds of no fault conditions).
2) Input stabilizer signal level is outside preset limits.
3) The control loop error for the CONTROL channel exceeds 0.3 degree or control loop error for the ARM channel exceeds 0.5 degree for longer than 3.5 seconds with a valid SAM command. (The loop fault latch resets after 5 seconds of no fault conditions).
4) Release of a hydraulic brake without a valid SAM output trim command.
5) A hydrualic brake not released within 2 seconds of a valid SAM output trim command.
6) Faults within ADCs or wiring to SAM's that cause air data to be monitored as invalid.
7) Following a second fault in either flap retracted or air/ground discrete signals majority vote.

(j) Switching to the other SAM occurs when a MACH/SPEED fault or SAM fault is detected within controlling SAM. If a MACH/SPEED fault or SAM fault is detected within both SAMs, the MACH/SPD TRIM light on panel P5 is illuminated and the message MACH/SPD TRIM is displayed on the upper EICAS display unit. This indicates a complete loss of the Mach/speed function.

(k) A SAM is able to reset itself if it is not in control (other SAM is selected for Mach/speed mode) and a SAM fault occurs. A SAM is also able to reset itself if a SAM fault has occured and the other SAM picks up control (Mach/speed mode select goes high on other SAM). If a SAM fault occurs again within one minute of an autoreset the SAM will not be able to perform another autoreset. (All monitors in the SAM are less than one minute in length.) The SAM fault ball will reset itself if an autoreset is successful and no other SAM faults occur within one minute.

MACH TRIM/SPEED STABILITY SYSTEM - ADJUSTMENT/TEST

1. General
   A. This procedure gives the system test for the Mach Trim/Speed Stability System. It is necessary to use external test equipment for the system test. This test examines the interfaces between the equipment in the Mach Trim/Speed Stability System.

TASK 22-24-00-735-008

2. System Test - Mach Trim/Speed Stability System
   A. References
      (1) 24-22-00/201, Electrical Power - Control
      (2) 27-41-00/501, Horizontal Stabilizer Trim Control System
      (3) 27-51-00/201, Trailing Edge Flap System
      (4) 27-61-00/201, Spoiler/Speedbrake Control System
      (5) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (6) 31-41-00/501, Engine Indication and Crew Alerting System
      (7) 32-09-02/201, Air/Ground Relays
      (8) 34-11-00/201, Pitot-Static System
      (9) 34-12-00/501, Air Data Computing System
   B. Access
      (1) Location Zones
          119/120 Main Equipment Center
          211/212 Flight Compartment

      (2) Access Panel
          119BL Main Equipment Center

   C. Prepare for the System Test

      S 865-009
      (1) Supply electrical power (Ref 24-22-00).

      S 285-010
      (2) Make sure that these systems operate correctly:
         (a) Horizontal Stabilizer Trim Control System (Ref 27-41-00).
         (b) Air/Ground Relays (Ref 32-09-02)
         (c) Air Data Computing System (Ref 34-12-00).
         (d) Engine Indication and Crew Alerting System (EICAS) (Ref 31-41-00).

      S 285-011
      (3) Make sure the control column is at its neutral position.

      NOTE: Do not hold the control column.

S 845-012

(4) If a fault ball is set or if an overhead panel light comes on during the tests, do one of the steps that follow:
   (a) Open and close these circuit breakers on the P11 overhead circuit breaker panel:
      1) 11C8, CSEU 2L AC
         or
         FLT CONT ELEC 2L AC
      2) 11G17, CSEU 1R AC
         or
         FLT CONT ELEC 1R AC
      3) 11G27, CSEU 2R AC
   (b) Push the RESET switch on each SAM.

S 865-016

(5) The RESET switch on the front panel of the SAM sets all of the fault balls. If there is a failure, the fault ball sets to yellow after you release the RESET switch.

<u>WARNING</u>: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

   (a) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoilers (Ref 27-61-00).

<u>WARNING</u>: DO THE DEACTIVATION PROCEDURE FOR FLIGHT MODE SIMULATION BEFORE YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS. WHEN YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS THE AIRPLANE IS IN FLIGHT MODE. IN FLIGHT MODE, MANY OF THE AIRPLANE SYSTEMS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

   (b) Do the deactivation procedure for flight mode simulation (Ref 32-09-02).

S 865-017

(6) Open these circuit breakers on the power distribution panel, P6, and attach DO-NOT-CLOSE tags:
   (a) 6L15, PITOT HEAT CAPT MAIN

(b) 6L16, PITOT HEAT AUX L
(c) 6L21, PITOT HEAT R AUX
(d) 6L22, PITOT HEAD F/O MAIN
(e) 6L23, R AOA HEAT
(f) 6L24, TAT PROBE HEAT

S 865-018

(7) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:
(a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC
(b) 11C7, CSEU 1L DC or FLT CONT ELEC 1L DC
(c) 11C8, CSEU 2L AC or FLT CONT ELEC 2L AC
(d) 11C9, CSEU 2L DC or FLT CONT ELEC 2L DC
(e) 11C12, STAB TRIM SHUTOFF LEFT
(f) 11C13, STAB TRIM SHUTOFF RIGHT
(g) 11C14, FSEU-2 PWR or FLAP SLAT ELEC UNIT 2 POWER
(h) 11C15, FSEU-2 CONT or FLAP SLAT ELEC UNIT 2 CONT
(i) 11C16, FSEU-2 SENSOR or FLAP SLAT ELEC UNIT 2 SENSOR
(j) 11C30, LANDING GEAR POS SYS 1
(k) 11F19, STAB POS MOD CENTER
(l) 11G12, FSEU-1 PWR or FLAP SLAT ELEC UNIT 1 POWER
(m) 11G13, FSEU-1 CONT or FLAP SLAT ELEC UNIT 1 CONT
(n) 11G14, FSEU-1 SENSOR or FLAP SLAT ELEC UNIT 1 SENSOR
(o) 11G15, STAB POS MOD LEFT
(p) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(q) 11G18, CSEU 1R DC or FLT CONT ELEC 1R DC
(r) 11G21, FSEU-3 PWR or FLAP SLAT ELEC UNIT 3 POWER
(s) 11G22, FSEU-3 CONT, or
FLAP SLAT ELEC UNIT 3 CONT, or
FSAM CONTROL
(t) 11G23, FSEU-3 SENSOR, or
FLAP SLAT ELEC UNIT 3 SENSOR, or
FSAM SENSOR
(u) 11G24, STAB POS MOD RIGHT
(v) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC
(w) 11G28, CSEU 2R DC or FLT CONT ELEC 2R DC
(x) 11H10, STAB TRIM LEFT POS IND
(y) 11H11 or 11C05, STAB TRIM CONT LEFT
(z) 11H19, STAB TRIM POS IND RIGHT
(aa) 11H20, STAB TRIM CONT RIGHT
(ab) 11B19, STAB TRIM ALT
(ac) 11P28, R IND LTS 1
(ad) 11S19, AIR/GND SYS 2

S 865-019

(8) Pressurize the Captain's and the F/O's pitot system (Ref 34-11-00).

NOTE: This will be referred to during the test as the air data computer (ADC) airspeed.

S 865-020

(9) Pressurize the captain's and the first officer's static system (Ref 34-11-00).

NOTE: This is referred to during the test as the pressure altitude.

S 865-021

(10) Pressurize the Auxiliary No. 1 and the No. 2 pitot system (Ref 34-11-00).

NOTE: This is referred to during the test as the elevator feel computer (EFC) airspeed.

S 985-022

(11) Set the ADC airspeed to 200 ±5 knots.

S 985-023

(12) Set the EFC airspeed to 224 ±20 knots.

NOTE: This value will not change during the test.

S 985-024

(13) Set the ADC pressure altitude to 30,000 ±1000 feet.

NOTE: This value will not change during the test.

S 865-003

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(14) Pressurize the right and the center hydraulic systems (Ref 29-11-00).

S 985-025

(15) Use the applicable (captain's, first officer's) control wheel trim switches to put the stabilizer at approximately five units of trim.

S 285-026

(16) Make sure the SAM fault ball is black on the applicable stabilizer trim and elevator asymmetry limit module (SAM).

S 285-027
(17) Make sure the HYD PRESS and ADC fault balls are black on the applicable SAM.

D. The Test of the STAB TRIM Fault Light and the Stabilizer Trim Control Module (STCM) Brake Switch Interface

S 985-028
(1) Use the control wheel trim switches to put the stabilizer to approximately four units of trim.

S 845-004

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoiler panels (Ref 27-61-00).

S 865-029
(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(b) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC

S 985-030
(4) Set R STAB TRIM switch to CUT OUT.

S 285-031
(5) Make sure the STAB TRIM amber light on the overhead panel, P5, is not on.

S 285-129
(6) Make sure STAB TRIM does not show on the top EICAS display.

S 985-032
(7) Set the computer selector switch on the EICAS display panel to the R position.

S 285-033
(8) Make sure the STAB TRIM amber light on the overhead panel, P5, is not on.

S 285-128
(9) Make sure STAB TRIM does not show on the top EICAS display.

S 985-034
(10) Push the control wheel trim switches in the UP direction.

S 285-035
(11) Make sure the STAB TRIM amber light comes on.

S 285-126
(12) Make sure STAB TRIM shows on the top EICAS display unit.

S 985-036
(13) Set the R STAB TRIM switch to the NORM position.

S 985-037
(14) Push the control wheel trim switches in the UP direction.

S 285-038
(15) Make sure the STAB TRIM light (yellow) is off.

S 865-127
(16) Make sure STAB TRIM does not show on the top EICAS display unit.

S 865-039
(17) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(b) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC

S 865-040
(18) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC or FLT CONT ELEC 2L AC
(c) 11C8, CSEU 1L, AC or FLT CONT ELEC 2L AC

S 985-041
(19) Set the C STAB TRIM switch to the CUT OUT position.

S 985-042
(20) Set the computer sector switch on the EICAS display select panel to the L position.

S 985-043
(21) Push the control wheel trim switch in the DOWN direction.

S 285-044
(22) Make sure the STAB TRIM light (yellow) comes on.

S 215-142
(23) Make sure STAB TRIM shows on the top EICAS display unit.

S 985-045
(24) Set the C STAB TRIM switch to the NORM position.

S 985-046
(25) Push the control wheel trim switches in the DOWN direction.

S 285-047
(26) Make sure the STAB TRIM light (yellow) on the P5 panel is off.

S 865-130
(27) Make sure the STAB TRIM does not show on the EICAS display unit.

S 865-048
(28) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC
(b) 11C8, CSEU 2L AC or FLT CONT ELEC 2L AC

E. The Test of the Mach/Speed Select Crossfeed and the Air/Ground Interface

S 845-138

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoiler panels (Ref 27-61-00).

S 845-005

WARNING: MAKE SURE YOU DO THE FLIGHT MODE SIMULATION CORRECTLY. IF THE PROCEDURE IS NOT DONE CORRECTLY, INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR.

(2) Do the Flight Mode Simulation procedure for the air/ground system (Ref 32-09-02).

S 865-049
(3) Open these circuit breakers on the panel P11 and attach DO-NOT-CLOSE tags:
(a) 11C30, LANDING GEAR POSITION SYS 1
(b) 11S19, AIR/GND SYS 2

S 285-050
(4) After 20 seconds, make sure that the SAM and AIR/GND fault balls stay black on the left and right SAMs.

S 985-051
(5) Use the control wheel trim switches to put the stabilizer to approximately five units of trim.

S 865-139

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoiler panels (Ref 27-61-00).

S 865-052

(7) Open these circuit breakers on the P11 panel:
   (a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC

S 865-053

(8) Close these circuit breakers on the P11 panel:
   (a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC

S 985-054

(9) Set the R STAB TRIM switch to the CUT OUT position.

S 985-055

(10) Increase the ADC airspeed until the captain's and first officer's airspeed indicators show 350 ±5 knots.

S 285-056

(11) Make sure the STCM fault ball on the right SAM is yellow.

S 285-057

(12) Let the stabilizer move until it stops.

S 865-058

(13) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C30, LANDING GEAR POSITION SYS 1
   (b) 11S19, AIR/GND SYS 2

S 985-059

(14) Push the RESET switch on the right SAM.

S 285-060

(15) Make sure the STCM fault ball on the right SAM is black.

S 865-061
(16) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C30, LANDING GEAR POSITION SYS 1
(b) 11S19, AIR/GND SYS 2

S 985-062
(17) Stop for 20 seconds.
(a) Set the R STAB TRIM switch to the NORM position.

S 865-063
(18) Open and close each of these circuit breakers on the P11 panel:
(a) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(b) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC

S 985-064
(19) Set the C STAB TRIM switch to the CUT OUT position.

S 985-065
(20) Decrease the ADC airspeed until the captain's and first officer's airspeed indicators show 200 ±5 knots.

S 285-066
(21) Make sure the STCM fault ball on the left SAM is yellow.

S 285-067
(22) Let the stabilizer move until it stops.

S 865-068
(23) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C30, LANDING GEAR POSITION SYS 1
(b) 11S19, AIR/GND SYS 2

S 985-069
(24) Push the RESET switch on the left SAM.

S 285-070
(25) Make sure the STCM fault ball on the left SAM is black.

S 985-071
(26) Set the C STAB TRIM switch to the NORM position.

F. The Test of the Mach/Speed Trim Indicator Interface

S 285-072
(1) Make sure the MACH/SPD TRIM yellow light is off.

S 285-073
(2) Make sure MACH/SPD TRIM does not show on the top EICAS display unit.

S 985-074
(3) Set the computer selector switch on the EICAS display select panel to the L position.

S 285-075
(4) Make sure the MACH/SPD TRIM yellow light is off.

S 285-076
(5) Make sure MACH/SPD TRIM does not show on the top EICAS display unit.

S 865-140

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoiler panels (Ref 27-61-00).

S 865-077
(7) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC
(b) 11G24, STAB POS MOD RIGHT
(c) 11C8, FLT CONT ELEC 2L AC

S 285-079
(8) Make sure MACH/SPD TRIM shows on the top EICAS display unit.

S 985-080
(9) Set the computer selector switch on the EICAS display select panel to the R position.

S 285-081
(10) Make sure the MACH/SPD TRIM light (yellow) stays on.

S 285-125
(11) Make sure MACH/SPD TRIM shows on the top EICAS display unit.

S 865-082
(12) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C8, FLT CONT ELEC 2L AC
(b) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC

S 865-083
(13) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(b) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC

S 285-084
(14) Make sure the MACH/SPD TRIM light (yellow) goes off.

S 285-085
(15) Make sure MACH/SPD TRIM does not shows on the top EICAS display unit.

S 865-086
(16) Open this circuit breaker on the P11 panel and attach a DO-NOT-CLOSE tag:
(a) 11G15, STAB POS MOD LEFT

S 285-087
(17) Make sure the MACH/SPD TRIM light (yellow) comes on.

S 285-088
(18) Make sure MACH/SPD TRIM shows on the top EICAS display unit.

S 985-089
(19) Set the computer selector switch on EICAS display select panel to the L position.

S 285-090
(20) Make sure the MACH/SPD TRIM light (yellow) stays on.

S 285-091
(21) Make sure MACH/SPD TRIM shows on the top EICAS display unit.

S 865-092
(22) Close these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C8, CSEU 2L AC or FLT CONT ELEC 2L AC
(b) 11C6, CSEU 1L AC or FLT CONT ELEC 1L AC
(c) 11G17, CSEU 1R AC or FLT CONT ELEC 1R AC
(d) 11G27, CSEU 2R AC or FLT CONT ELEC 2R AC

S 285-094
(23) Make sure the MACH/SPD TRIM light (yellow) goes off.

S 285-095
(24) Make sure MACH/SPD TRIM does not show on the top EICAS display unit.

S 865-148
(25) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11G15, STAB POS MOD LEFT

G. The Test of the Mach/Speed Fault Crossfeed Interface

S 865-096
(1) Open this circuit breaker on the P11 panel and attach a DO-NOT-CLOSE tag:
(a) 11G24, STAB POS MOD RIGHT

S 285-097
(2) Make sure the MACH/SPD TRIM light (yellow) stays off.

S 725-143
(3) Make sure MACH/SPD TRIM does not show on the top EICAS display unit.

NOTE: STAB TRIM shows on the top EICAS display unit while circuit breaker is open.

S 865-099

(4) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
   (a) 11G24, STAB POS MOD RIGHT

S 865-100

(5) Open this circuit breaker on the P11 panel:
   (a) 11G15, STAB POS MOD LEFT

S 285-101

(6) Make sure the MACH/SPD TRIM light (yellow) stays off.

S 725-144

(7) Make sure MACH/SPD TRIM does not show on the top EICAS display unit.

   NOTE: STAB TRIM shows on the top EICAS display unit while circuit breaker is open.

S 865-103

(8) Close this circuit breaker on the P11 panel:
   (a) 11G15, STAB POS MOD LEFT

H. The Test of the Standby Trim Switch Interface

S 985-131

(1) Move the alternate stab trim switches forward.

S 285-105

(2) Make sure the UNSCHED STAB TRIM light (yellow) stays off.

S 865-141

(3) Release the alternate stab trim switches.

I. The Test of the Flap Position Interface

S 865-107

(1) Open these circuit breakers on the P11 panel in less than one second of each other and attach DO-NOT-CLOSE tags:
   (a) 11C14, FSEU 2 PWR or FLAP SLAT ELEC UNIT 2 POWER

(b) 11G12, FSEU-1 PWR or FLAP SLAT ELEC UNIT 1 POWER
(c) 11G21, FSEU-3 PWR or FLAP SLAT ELEC UNIT 3 POWER

S 285-108
(2) Make sure the R. AIL LOCK ACT/FSEU fault balls on the left and right SAMs are black.

S 865-109
(3) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the panel P11.
(a) 11G12, FSEU-1 PWR or FLAP SLAT ELEC UNIT 1 POWER

S 285-110
(4) Make sure the R. AIL LOCK ACT/FSEU fault balls on the left and right SAMs are yellow.

S 865-111
(5) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C14, FSEU-2 PWR or FLAP SLAT ELEC UNIT 2 POWER
(b) 11G21, FSEU-3 PWR or FLAP SLAT ELEC UNIT 3 POWER

S 985-112
(6) Push the RESET switch on the left and right SAMs.

S 285-113
(7) Make sure the R. AIL LOCK ACT/FSEU fault balls on the left and right SAMs are black.

S 865-114
(8) Open this circuit breaker on the P11 panel and attach a DO-NOT-CLOSE tag:
(a) 11G12, FSEU-1 PWR or FLAP SLAT ELEC UNIT 1 POWER

S 285-115
(9) Make sure the R. AIL LOCK ACT/FSEU fault balls on the left and right SAMs are yellow.

S 865-116
(10) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11G12, FSEU-1 PWR or FLAP SLAT ELEC UNIT 1 POWER

S 985-117

(11) Push the RESET switch on the left and right SAMs.

J. Put the Airplane Back to Its Usual Condition

S 865-006

(1) Put the airplane back to the ground mode (Ref 32-09-02).

S 845-007

(2) Do the activation procedure for the spoilers if you did the deactivation procedure (Ref 27-61-00).

S 985-118

(3) Set the ADC airspeed to 0 knot.

S 985-119

(4) Set the EFC airspeed to 0 knot.

S 985-120

(5) Set the pressure altitude to the ambient position.

S 865-121

(6) Remove pressure from the pitot-static system.

S 865-122

(7) Remove the DO-NOT-CLOSE tags and close circuit breakers on the P6 panel:

(a) 6L15, PITOT HEAT CAPT MAIN

(b) 6L22, PITOT HEAT F/O MAIN

(c) 6L23, R AOA HEAT

(d) 6L24, TAT PROBE HEAT

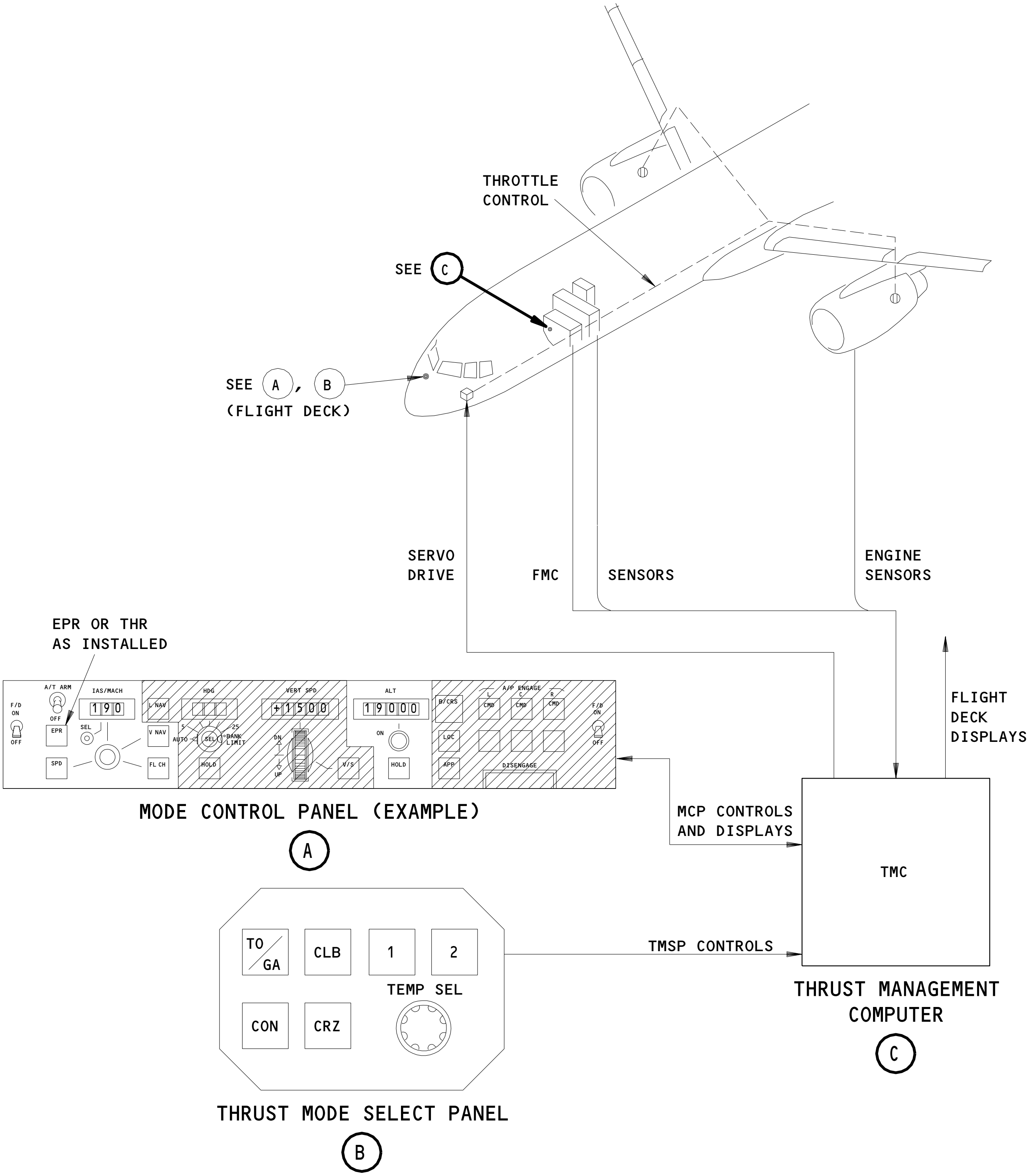
S 865-123

(8) Remove the pressure from the hydraulic system if it is not necessary (TASK 29-11-00-862-053).

S 865-124
(9) Remove electrical power if it is not necessary.

THRUST MANAGEMENT POWER - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Thrust Management System (TMS) computes and displays thrust limits and provides thrust lever control. The system interfaces with the engines through the servomotor generator, which drives thrust levers through "no-back" clutches. The Thrust Mode Select Panel (TMSP) and Thrust Management Computer (TMC) provide primary control of the servomotor generator.
   B. The TMSP allows crew selection of thrust limit modes, alternate thrust limit ratings, and assumed temperature. The Engine Indication and Crew Alerting System (EICAS) (Ref 31-41-00) display unit displays the selected limit mode or rating, corresponding limits, and assumed temperature.
   C. The TMC provides thrust limit computation and thrust lever control to Engine Pressure Ratio (EPR), airspeed, and Mach references. Minimum and maximum speed protection, thrust limit protection, and thrust lever retard during autoland are also functions of the TMC.
   D. The TMC uses single phase, 400Hz, 115v ac and 28v dc power. The 28v dc is used for external interlocks. The servomotor generator uses 115v ac from the left main bus. The TMC supplies +5v dc and ±15v dc regulated power to the TMSP.
   E. Major TMS Components
      (1) Thrust Management Computer (TMC)
         (a) One TMC is installed in the main electrical/electronic equipment compartment E2-3 shelf. Two EPR thrust limits are continuously computed by the TMC. Maximum EPR and reference EPR are functions of pressure altitude, Mach number, total air temperature, and engine air extraction. The TMC uses this data for control of the servomotor generator, which drives the thrust levers.
      (2) Thrust Mode Select Panel (TMSP)
         (a) The TMSP allows the crew to select thrust limit computation modes. A thrust rating mode and corresponding limit is selected on the panel for each stage of the flight profile. Selected thrust modes and limits are displayed on the EICAS display unit.
      (3) Autothrottle Servomotor Generator
         (a) The autothrottle servo provides the interface between the TMC and thrust levers. The servo drives the thrust levers and linkage through a gearbox and no-back clutch pack assembly.
      (4) Clutch Pack Assembly
         (a) The clutch pack assembly provides the interface between the autothrottle servo and the thrust levers. The clutches slip when the thrust levers are moved by the pilot. This allows the pilot to override the TMS without backdriving the autothrottle servo.



Thrust Management System
Figure 1

F. TMS Control Inputs

(1) The AFCS Mode Control Panel (MCP) allows manual selection of autothrottle control modes. The EPR (or THR) and SPD select controls are operable when the autothrottle switch on the MCP is in the ARM position. The SPD mode is operable only when the airplane is airborne. The Flight Management Computer (FMC) automatically selects autothrottle modes to follow the flight paths and speed profiles defined by FMC performance and vertical navigation functions.

G. Sensor Inputs

(1) The Air Data Computing System (ADCS) (34-12-00) consists of left and right Air Data Computers (ADCs), which provide environmental data to the airplane.

(2) The Inertial Reference System (IRS) (34-21-00) consists of left, right, and center IRUs which provide pitch and roll attitude; and longitudinal and vertical acceleration.

(3) The Electronic Engine Control (EEC) (Ref 73-21-00) provides thrust limiting for engine protection as measured by EPR.

(4) The autothrottle Power Lever Angle (PLA) transducer is a rotary variable differential transformer that provides an ac signal output proportional to thrust lever position. One sensor is installed on each engine. The autothrottle PLA is driven by inputs from the engine to strut control rod, which is connected to the control cables from the thrust levers (Ref 22-33-00).

H. TMS Outputs

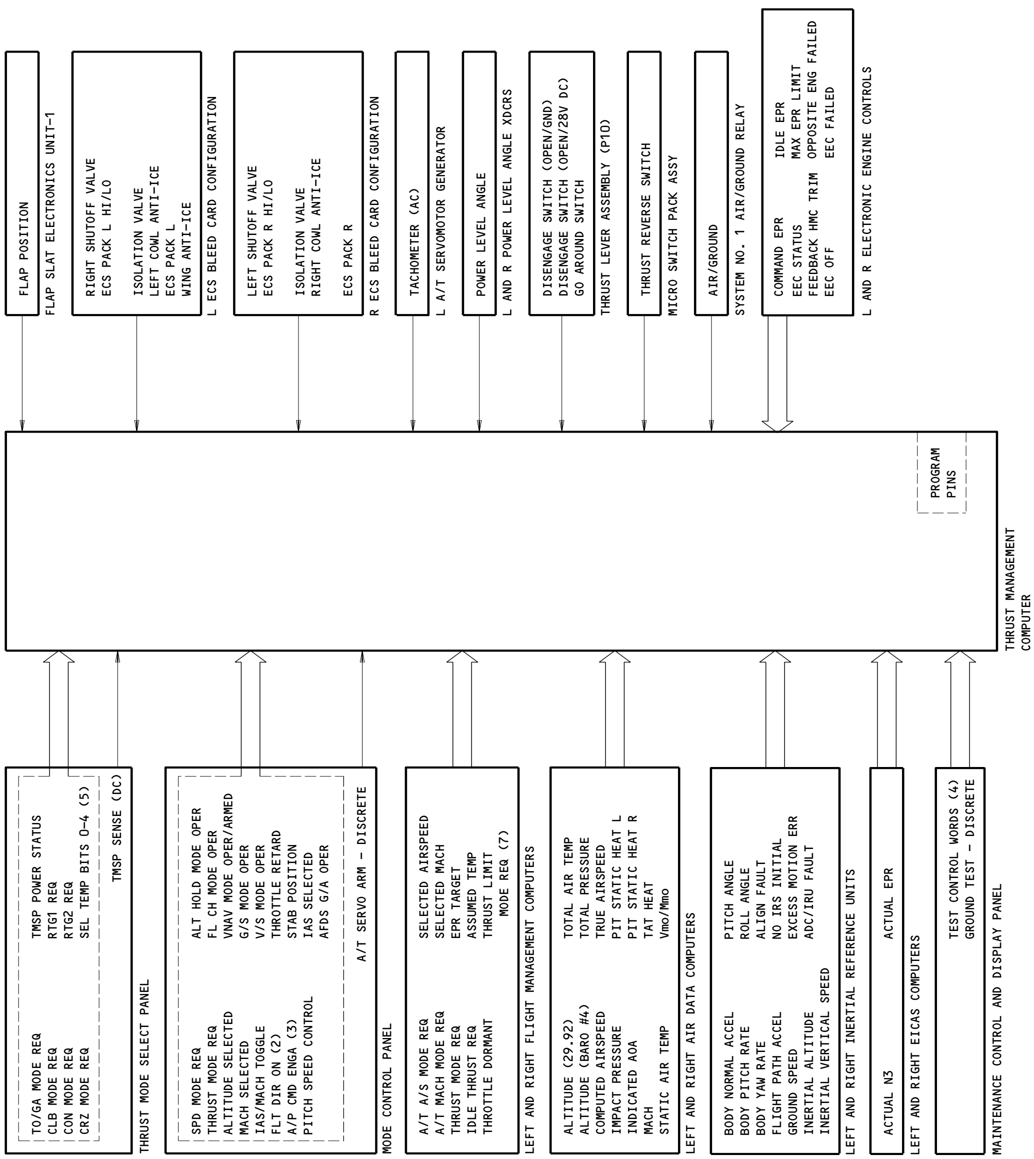
(1) The TMC computed thrust reference limits and maximum limits are displayed on the upper EICAS display unit.

I. TMC/Systems Interface Inputs (Fig. 2)

(1) Digital Data Bus Inputs

(a) Digital data bus inputs are received by the TMC from the following components.

1) Thrust Mode Select Panel (TMSP)
2) AFCS Mode Control Panel (MCP)
3) Maintenance Control Display Panel (MCDP)
4) Left and Right Air Data Computers (ADCs)
5) Left and Right Flight Management Computers (FMCs)



TMC System Interface Inputs
Figure 2

29088

6) Left and Right Inertial Reference Units (IRUs)
7) Left and Right EICAS Computers
8) Left and Right Electronic Engine Controls (EEC)

(2) Analog Inputs

(a) Analog inputs to the TMC are:

1) TMSP sense voltage for the TMC voltage regulator.
2) Flap position from the Flap/Slat Electronic Unit (FSEU).
3) Tachometer feedback from the servomotor generator and power lever angle from the autothrottle power lever angle transducers (L and R).

(3) Analog Discrete Inputs

(a) Analog discrete inputs to the TMC are:

1) A/T servo ARM from the MCP
2) In-air from air/ground relay
3) Ground test from the MCDP
4) Bleed discretes from the left and right ECS bleed configuration cards
5) Thrust reverse and A/T disconnect from respective switches
6) A/T disconnect reset and go around from respective switches

J. TMC Program Pin Data (Fig. 3)

(1) The TMC is designed to accomodate various airplane, engine, and customer selectable operational options. This is done by loading the computer with appropriate software and connecting designated external program pins. For specific pin connections on each airplane, refer to FIM 22-00-04/101, Figure 104. The hardwired pins establish the binary codes for software control.

(2) Engine Airframe and Thrust Limit Options

(a) Pin connections for the engine airframe options are open/ground. Thrust limit options are 28 vdc/ground. A parity pin is provided.

(3) Autothrottle and Aircraft Options

(a) Pin connections for autothrottle options are 28 vdc/ground. Aircraft options are open/ground. A parity pin is provided.

(4) Customer Option

(a) Pin connections are open/ground for customer options. There is no parity pin. If a ground is lost, that option is also lost.

K. TMC/System Interface Outputs (Fig. 4)

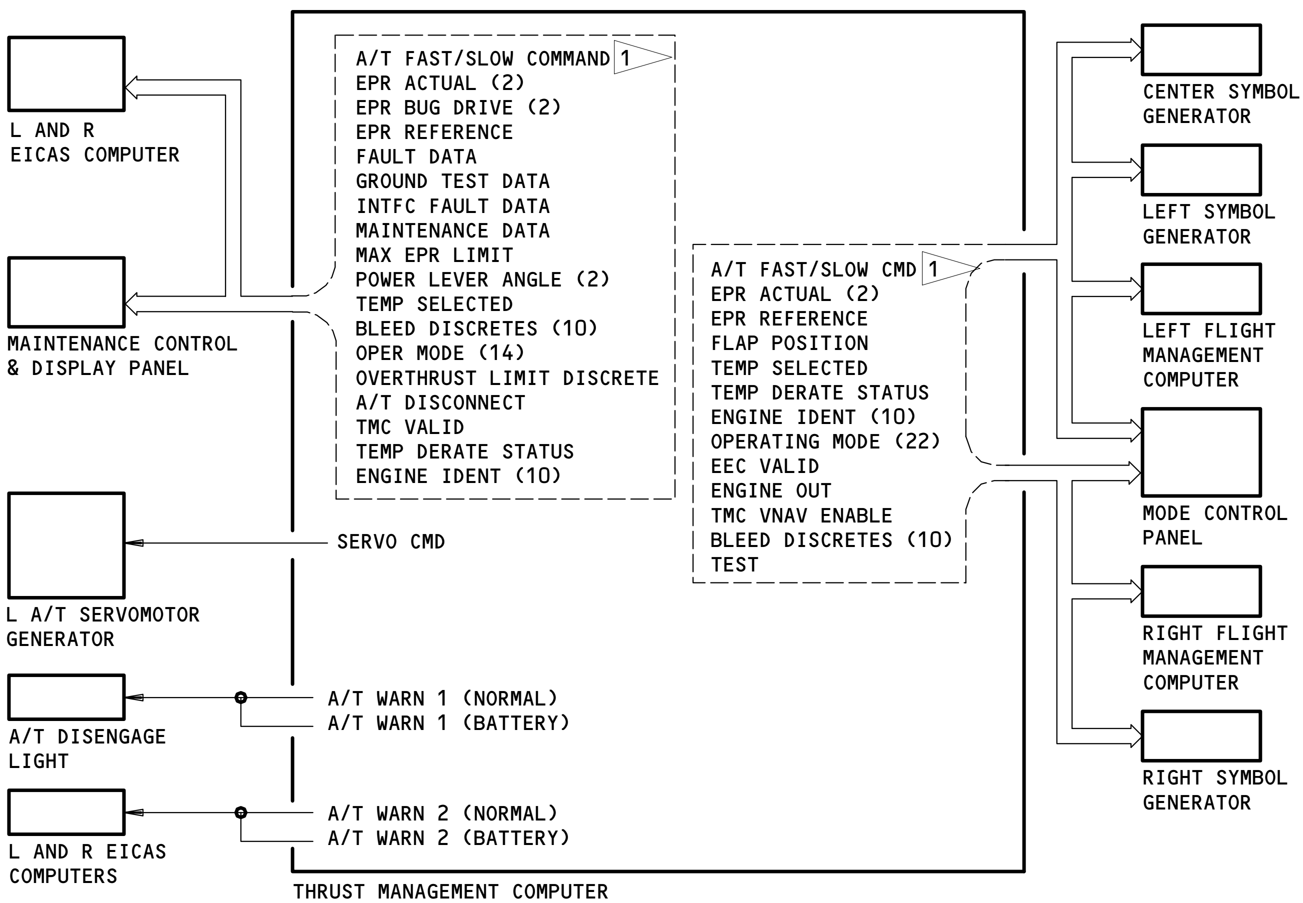
(1) Digital Data Buses

(a) One digital data bus is supplied for EICAS, MCDP and DFDAU outputs. There are two output buses for EFIS, FMC, and MCP signals. One bus contains MCP, left FMC; and left and center symbol generator signals. The other bus contains MCP, right FMC, and right symbol generator signals.

(2) Analog Outputs

(a) There are four analog servomotor generator outputs. These are SERVO CMD L-FWD and REV, and SERVO CMD R-FWD and REV. The SERVO CMD R-FWD and REV signals are not used.

Not Used
Figure 3



1 GUI 009,115

TMC Systems Interface Outputs
Figure 4

(3) Analog Discrete Outputs
  (a) Analog discretes that turn on the A/T disengage light are A/T WARN 1 (normal) and A/T WARN 1 (battery). Outputs to the EICAS computer and central warning system are A/T WARN 2 (normal) and A/T WARN 2 (battery) (Ref 22-34-00).

2. Component Details

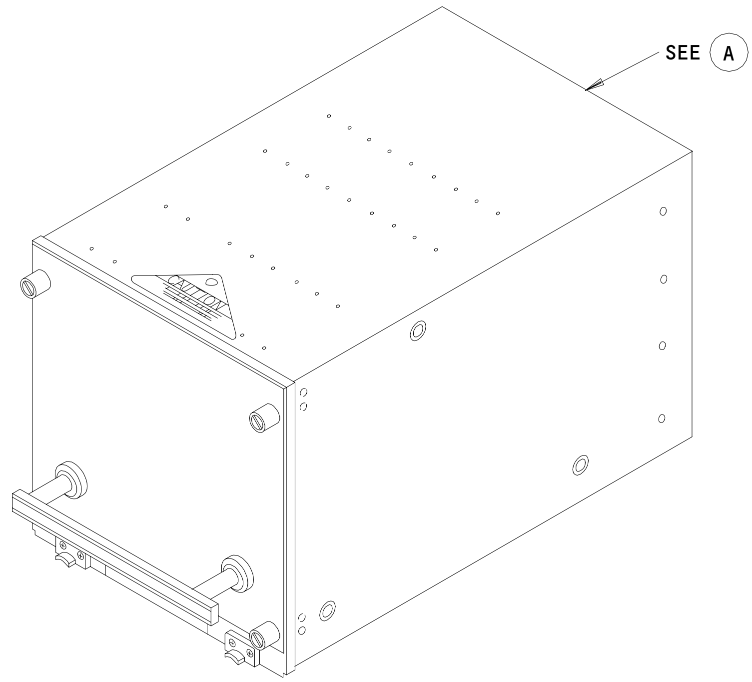
A. Thrust Management Computer (Fig. 5)

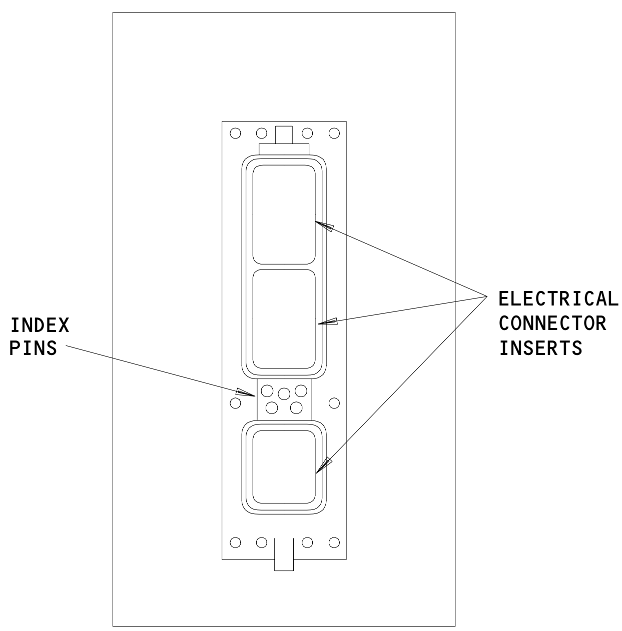
(1) The TMC consists of 8 modules in a 6 MCU case. The weight is 16.5 pounds. The TMC uses 115 vac and 28 vdc supplied through circuit breakers on P11.

(2) The TMC is a realtime, digital processor (MCP 701A). Functions are in independent physical modules which communicate via a digital bus structure. A basic description of each module is given in the following paragraphs:
  (a) The CPU/Timing module functions consist of the central processing unit, interrupt priority logic, real-time reference and watchdog monitor, memory parity generation and checking.
  (b) The memory module contains program storage and variable memory which are functionally independent. A growth location memory module is also provided.
  (c) The ARINC 429 DITS module interfaces all TMC digital inputs and outputs to the digital processor. The ARINC DITS expansion module contains the overflow and growth ARINC receivers.
  (d) The analog I/O module interfaces TMS external AC inputs and outputs and TMC internal DC inputs and outputs to the digital processor.
  (e) The System Interface Module (SIM) performs two functionally separate operations. It receives and transmits airplane system information in the form of analog discretes. It also provides non-volatile memory for registering the results of specific internal self-tests that generate a TMC INVALID digital discrete.
  (f) The power supply module provides all power conversion for the TMC and TMSP circuits.

(3) One connector with three inserts is on the back side of the TMC. One insert contains TMC power pins only. The other two inserts contain the TMC signal pins and low voltage power pins. Index pins ensure correct TMC installation.

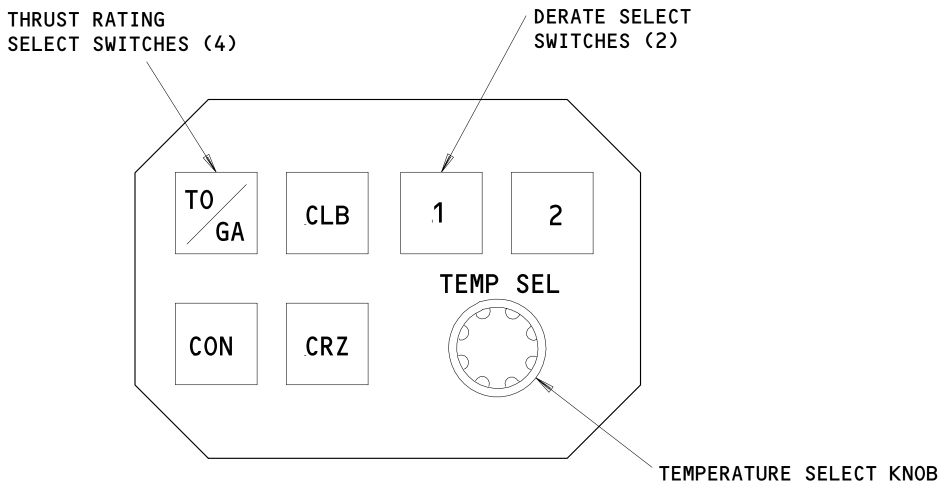
B. Thrust Mode Select Panel (TMSP) (Fig. 6)

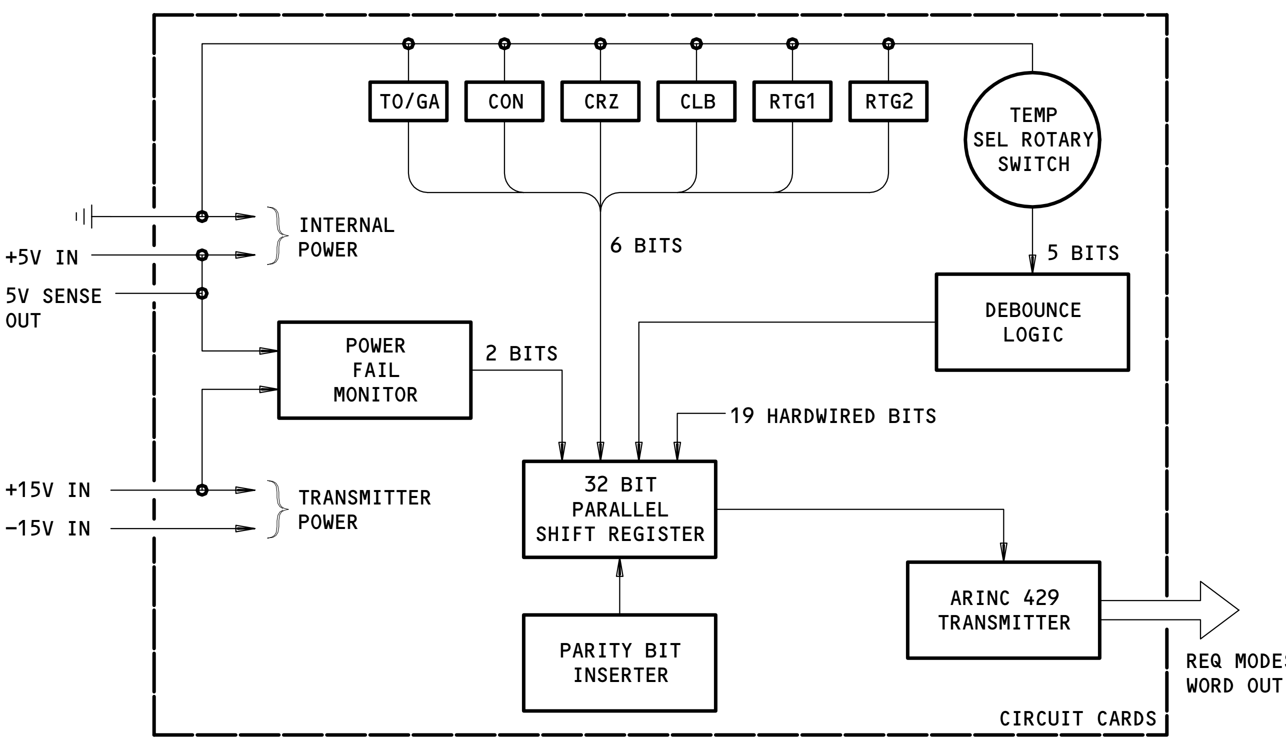




THRUST MANAGEMENT COMPUTER CONNECTIONS
A

Thrust Management Computer
Figure 5





THRUST MODE SELECT PANEL OPERATION

Thrust Mode Select Panel
Figure 6

(1) The TMSP consists of six pushbutton switches, one rotary switch, and two circuit cards mounted in a metal case. Panel switch illumination is provided by the edge-lit lightplate panel. The lightplate is powered by variable 0 to 5 vac provided by the airplane master dim and test circuits. A connector at the rear of the unit mates the TMSP with airplane wiring.

(2) Select Switches

(a) Manual thrust limit mode selection is accomplished by pressing the TMSP pushbutton for the desired mode. The switches ground the appropriate mode discrete bit line when activated. This informs the TMC that a particular mode has been selected. Pressing a pushbutton corresponding to the currently selected mode has no affect.

(b) The TMSP provides two derating schedule methods for reducing the thrust limit. Fixed derating schedules are selected by pressing TMSP derate pushbutton 1 or 2. Variable derating schedules are selected by using the TEMP SEL (temperature select) knob.

(c) The TEMP SEL knob controls a binary-coded, 22-position, continuous-rotation switch. The five bits that represent the position of the switch are transmitted to the TMC as part of the REQ MODES word. The TMC uses the switch position information (assumed temperature) to compute a derated value of thrust limit. Clockwise rotation of the knob by one detent position causes the flat rated temperature to be displayed on the upper EICAS display unit. Continued clockwise rotation changes the display in 1°C increments of the selected temperature. An increase in assumed temperature derates the thrust limit value.

(3) Module Description

(a) The circuit components of the TMSP are on two cards. The ARINC transmitter circuit card assembly contains the ARINC 429 serial data communication circuits. The clock generator circuit card assembly generates the timing signals required by the data communications circuits.

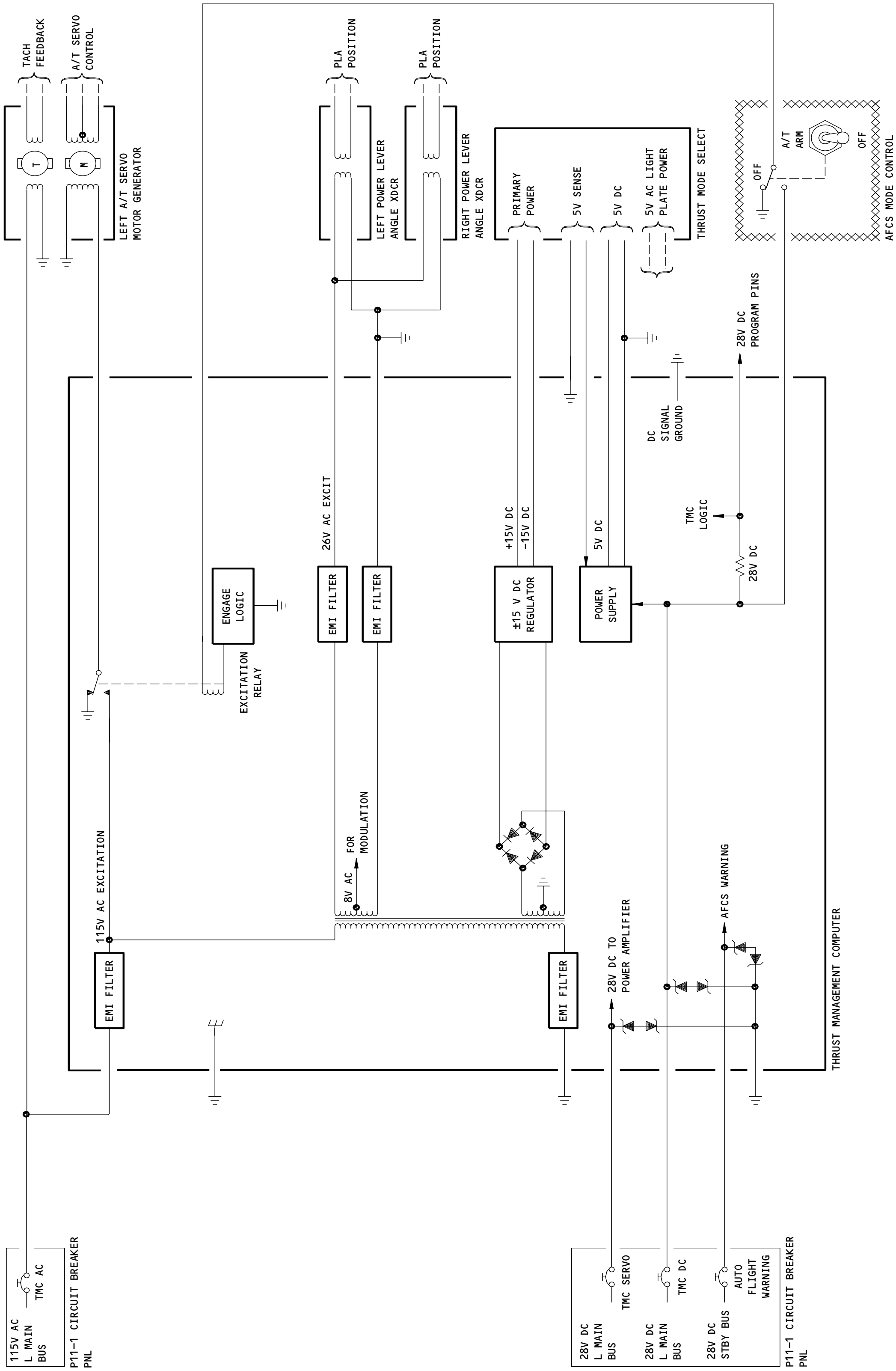
(b) The TMSP transmitter circuits inform the TMC of mode, derating factor, and assumed temperature selection. Data is transmitted as a single 32-bit digital word called the REQ MODES (request modes) word. The REQ MODES word informs the TMC of TMSP power status, mode switch selection, and TEMP SEL switch position. The REQ MODES word is transmitted to the TMC every 16 milliseconds.

(c) The clock generator circuit card consists of a crystal-controlled oscillator that generates a 2 MHz signal. The 2 MHz is divided to form 16 millisecond clock pulses used by the TSMP. The clock triggers the sequence of events required to transmit the 32-bit REQ MODES word.

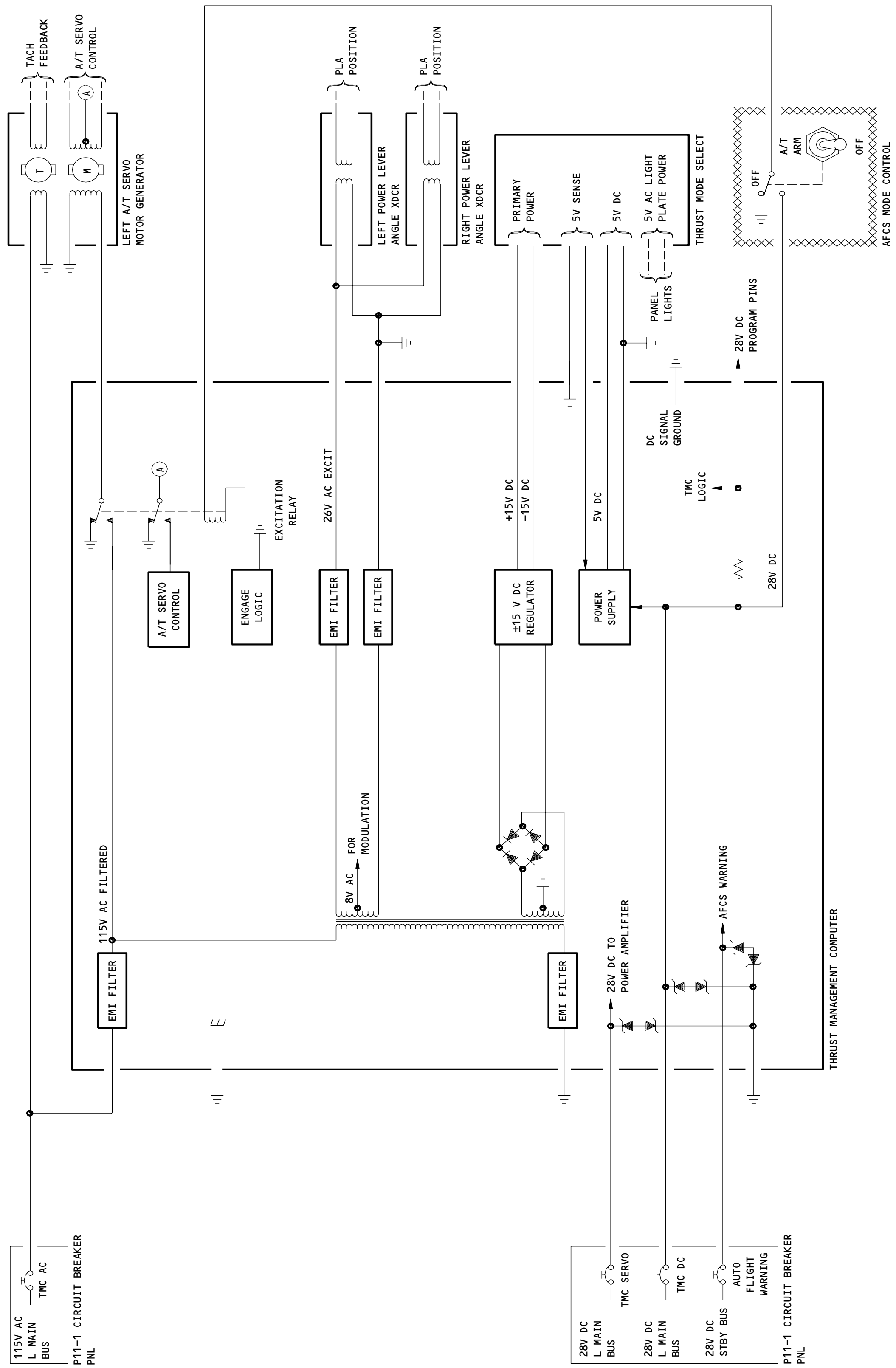
3. <u>Operation</u>

A. Functional Description

(1) Thrust Management System TMS Power Distribution (Fig. 7)



Thrust Management System Power Distribution - Schematic
Figure 7 (Sheet 1)



Thrust Management System Power Distribution - Schematic
Figure 7 (Sheet 2)

(a) TMC Power

1) The TMC power consists of 115v ac and 28v dc power. The 28v dc standby bus provides power for disconnect warning if TMC power fails. The TMC excitation relay connects 115 vac power and 36 vac control signals to the servomotor when engage conditions are valid. The EMI filters remove transients from the line and internal switching noise from the power supply.

2) The 115v ac aircraft power is transformed to appropriate levels in the TMC. The 8v ac is used for TMC demodulation/modulation reference, and 26v ac is used for PLA transducer excitation. Full wave rectification provides ±15v dc and +5v dc for TMSP internal power. The TMC provides voltage regulation on all dc outputs.

3) The power supply is monitored to provide a primary power fail discrete that results in a TMC INVALID when input power is low. The abnormal steady-state ac limit is 97v ac and the dc lower limit is 18v dc. The TMC is also monitored to provide a power fail discrete if the ±15v dc or +5v dc logic supply falls below ±14v dc or +4.5v dc, respectively.

(b) TMSP Power

1) The TMSP receives +5v dc from the TMC. In order to maintain this voltage within ±0.25v dc under varying conditions, remote sensing lines are provided (+5 volts sense and ground sense). When +5 volts changes, the TMC power supply/regulator responds, thus ensuring that normal operating limits are maintained. When the voltage drops below +4.3 volts, the TMSP power loss monitor provides a TMSP status fail signal to the TMC. The TMC also provides ±15v dc power that operates the TMSP transmitter.

(c) A/T Servomotor Generator

(d) The A/T servomotor generator has two main components:

- A two-phase motor that moves the throttles as commanded by the TMC
- A single phase generator (or tachometer) which provides servo rate feedback to the TMC.

1) The servomotor has a fixed field winding which uses 115 vac excitation and two control windings which use 36 vac excitation (provided by the TMC servo power amplifier). The servomotor drives at a rate proportional to the pulse width of the control voltage. Both excitation voltages are controlled by the TMC excitation relay.

2) The tachometer is an AC generator that uses 115 vac excitation. The tachometer output is a linear relationship between shaft rpm and AC output voltage. The excitation voltage is directly connected to airplane power.

(e) A/T ARM

1) The A/T ARM switch is on the MCP. When positioned to ARM, the A/T switch connects 28 vdc to arm the excitation relay.

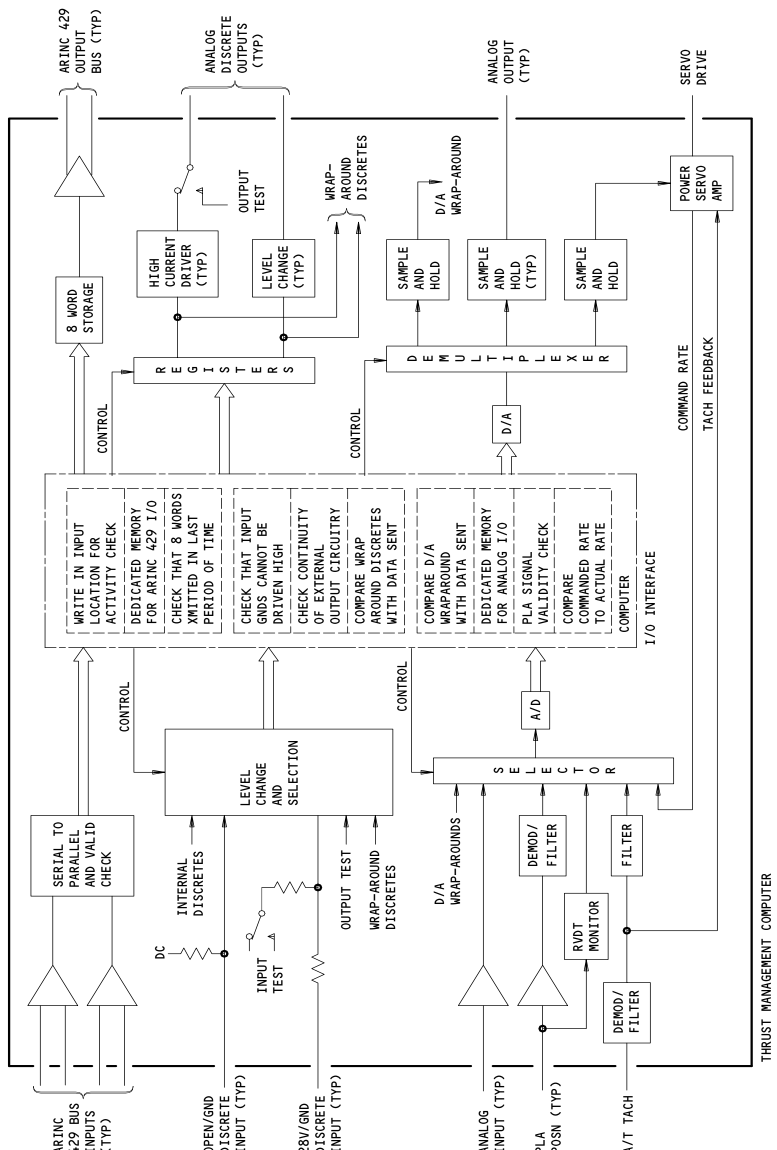
(2) TMC Operation (Fig. 8)

(a) Input Interface

1) Serial ARINC 429 bus inputs are converted to parallel and stored in dedicated memory. Analog discrete inputs consists of two types: open/ground and 28v dc/ground. The input level changers convert these signals into transistor to transistor logic compatible format. This is done by equating an open or 28v dc to a logical one and ground to a logical zero. The 48 level changed lines are formed into three 16-bit words. They are input along with a 16-bit internal discrete word into a one of four selector. The computer then selects the desired word for input by addressing the appropriate location. Analog inputs are selected one at a time. They are changed to a digital code in the hybrid Analog to Digital (A/D) converter for use in the digital processing section of the computer. When the conversion is complete, the digital value is stored in dedicated memory.

(b) Output Interface

1) The ARINC 429 transmitter receives 8 words at a time. These words are then transmitted serially on the 429 output bus. Analog discrete data is latched to the output register and then level changed to a voltage compatible to the airplane. These become the analog discrete outputs. Digital data is routed to the D/A converter and then multiplexed to one of eight Sample and Hold (S/H) circuits which provide the analog outputs. Two S/H provide A/T servo command outputs to the power servo amplifier on the power supply; however, only one is used. Two S/H are used for BIT (D/A wrap around) and two are used for testing. Two S/H are spares.



TMC Input/Output Interface and Monitors
Figure 8

28180

(c) Failure Protection and Maintenance

1) The TMC provides failure protection to assure system safety, and maintenance functions that isolate faults to a specific LRU of the TMS or interfacing system. Failure protection and maintenance activities of the TMC are classified as fault data management, interface verification test, and maintenance functional tests.

2) The primary function of fault data management is to support system safety. Fault data management runs continuously when the TMC is powered except during maintenance functional test. It interrogates all system monitors and runs self-test routines as required to prevent computing a false thrust limit or issuing an invalid servo command. When a fault is detected, fault data management responds appropriately to the type of fault and operational mode. It also suppots maintenance activities by performing fault isolation and by storing TMC fault information in nonvolatile memory. It continually transmits fault data to the Maintenance Control Display Panel (MCDP). When activated by touchdown, the MCDP stores the fault data for later display to maintenance personnel.

3) The interface verification test verifies that all input/output lines between the TMC and interfacing components are connected. It consists of passive and active tests. Passive tests are performed without interfering with the normal operation of the TMC. Active tests require the TMC to change the state of its outputs to verify connections. The active tests are part of the maintenance functional tests.

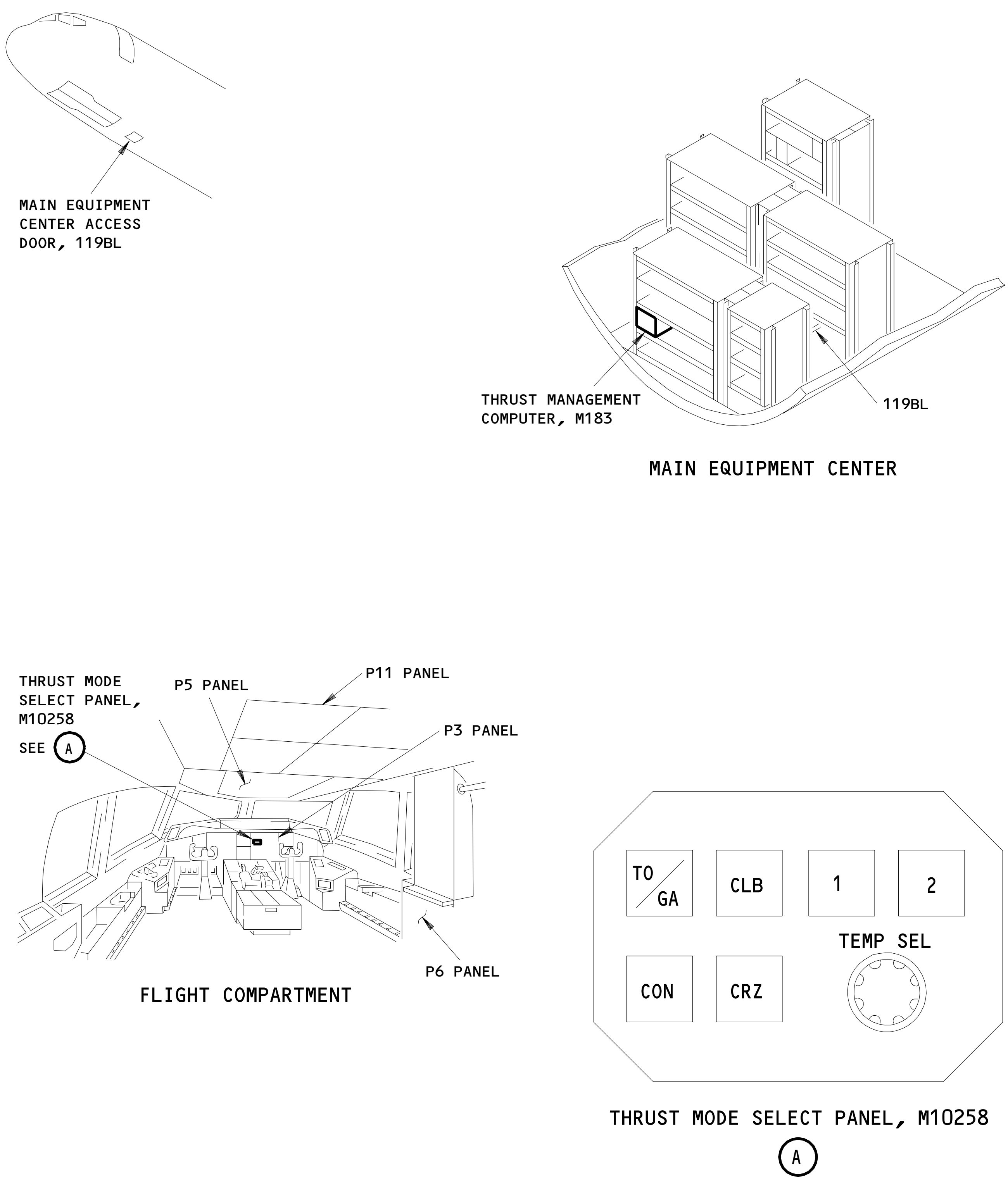
4) The maintenance functional tests provide the capability for performing TMS tests. The maintenance functional test mode is selected when the MCDP is placed in the GRD TEST mode. Maintenance personnel then select specific test(s) to be run via an MCDP input. The TMC then runs the test and transmits the results to the MCDP for display to the operator. The test may require operator action or verification of results. These tests can only be initiated when the airplane is on the ground.

THRUST MANAGEMENT POWER

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
CIRCUIT BREAKER -	--		FLT COMPT, P11	
AUTOFLIGHT WARN, C521		1	11A17	*
TMC AC, C501		1	11F14 OR 11F16	*
TMC DC, C525		1	11F15 OR 11F17	*
TMC SERVO, C512		1	11F16 OR 11F18	*
COMPUTER - THRUST MANAGEMENT, M183	--	1	119BL, MAIN EQUIP CTR, E2-3	22-31-01
PANEL - THRUST MODE SELECT, M10258	--	1	FLT COMPT, P3	22-31-02

* SEE THE WDM EQUIPMENT LIST

Thrust Management Power - Component Index
Figure 101



Thrust Management Power - Component Location
Figure 102

THRUST MANAGEMENT COMPUTER - REMOVAL/INSTALLATION

1. General
   A. The thrust management computer (TMC) is installed on the E2-3 shelf in the main equipment center. The TMC is held in position by two hold-down hooks and a lower extraction lip. One multi-pin connector with three inserts is at the rear of the unit.

TASK 22-31-01-004-002

2. Remove the Thrust Management Computer
   A. References
      (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) AMM 20-10-01/401, E/E Rack-Mounted Components
      (3) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

      (2) Access Panel

119BL	Main Equipment Center

   C. Prepare for Removal

S 864-001

      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11A17, AUTOFLIGHT WARN
         (b) 11F14 or 11F16, TMC AC
         (c) 11F15 or 11F17, TMC DC
         (d) 11F16 or 11F18, TMC SERVO

S 014-014

      (2) Open the access panel, 119BL, for the thrust management computer (AMM 06-41-00/201).

D. Remove the Thrust Management Computer

S 914-003

CAUTION: DO NOT TOUCH THE THRUST MANAGEMENT COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISHARGE CAN CAUSE DAMAGE TO THE THRUST MANAGEMENT COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 024-004

(2) Remove the thrust management computer (AMM 20-10-01/401).

TASK 22-31-01-404-005

3. Install the Thrust Management Computer

A. References
(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) AMM 20-10-01/401, E/E Rack-Mounted Components
(3) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
(4) AMM 22-00-02/201, Autoflight BITE
(5) AMM 24-22-00/201, Electrical Power - Control
(6) AMM 27-61-00/201, Spoiler/Speedbrake Control System

B. Access
(1) Location Zones
119/120 Main Equipment Center
211/212 Flight Compartment

(2) Access Panel
119BL Main Equipment Center

C. Install the Thrust Management Computer

S 914-010

CAUTION: DO NOT TOUCH THE THRUST MANAGEMENT COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE THRUST MANAGEMENT COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 424-006

(2) Install the thrust management computer (AMM 20-10-01/401).

D. Do a Test of the Thrust Management Computer

S 864-007

(1) Supply electrical power (AMM 24-22-00/201).

S 864-008
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11A17, AUTOFLIGHT WARN
(b) 11F14 or 11F16, TMC AC
(c) 11F15 or 11F17, TMC DC
(d) 11F16 or 11F18, TMC SERVO

S 864-015

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(3) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 864-016
(4) Make sure the engines are not on.

S 864-017
(5) Set the MCDP to OFF or go into Flight Faults Mode to make sure that the MCDP is out of Ground Test Mode.

NOTE: Before the MCDP Ground Test 02 TMC is done, the MCDP must be put out of Ground Test Mode to remove from the MCDP memory problems that have been corrected.

S 714-018
(6) Go into Ground Test Mode and do the MCDP Ground Test 02 TMC (AMM 22-00-02/201).

E. Put the Airplane Back to Its Initial Condition

S 864-013
(1) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 414-019
(2) Close the access panel, 119BL (AMM 06-41-00/201).

S 864-009
(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MODE SELECT PANEL - REMOVAL/INSTALLATION

1. General
   A. The thrust mode select panel (TMSP) is installed on the center instrument panel P3-1. The TMSP is held in position by a mounting clamp. One multi-pin connector is at the rear of the unit.

TASK 22-31-02-004-002

2. Remove the Thrust Mode Select Panel
   A. Access
      (1) Location Zone
          211/212 Flight Compartment

   B. Remove the Thrust Mode Select Panel

      S 864-001
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11F14 or 11F16, TMC AC
          (c) 11F15 or 11F17, TMC DC
          (d) 11F16 or 11F18, TMC SERVO

      S 034-013
      (2) Loosen the mounting clamp screws adjacent to the TMSP front panel.

      S 024-003

      CAUTION: YOU MUST CAREFULLY REMOVE THE TMSP FROM THE INSTRUMENT PANEL TO PREVENT DAMAGE TO THE ELECTRICAL CABLE.

      (3) Remove the TMSP from the instrument panel.

      S 034-004
      (4) Disconnect the electrical connector.

TASK 22-31-02-404-005

3. Install the Thrust Mode Select Panel
   A. References
      (1) AMM 22-00-02/201, Autoflight BITE

(2) AMM 24-22-00/201, Electrical Power - Control

B. Access

(1) Location Zone

211/212 Flight Compartment

C. Install the Thrust Mode Select Panel

S 434-008

(1) Connect the electrical connector to the TMSP.

S 424-009

CAUTION: YOU MUST CAREFULLY INSTALL THE TMSP IN THE INSTRUMENT PANEL TO PREVENT DAMAGE TO THE ELECTRICAL CABLE.

(2) Install the TMSP in the instrument panel.

S 434-010

(3) Tighten the mounting clamp screws.

D. Do a Test of the Thrust Mode Select Panel

S 864-011

(1) Supply electrical power (AMM 24-22-00/201).

S 864-012

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11A17, AUTOFLIGHT WARN

(b) 11F14 or 11F16, TMC AC

(c) 11F15 or 11F17, TMC DC

(d) 11F16 or 11F18, TMC SERVO

S 714-006

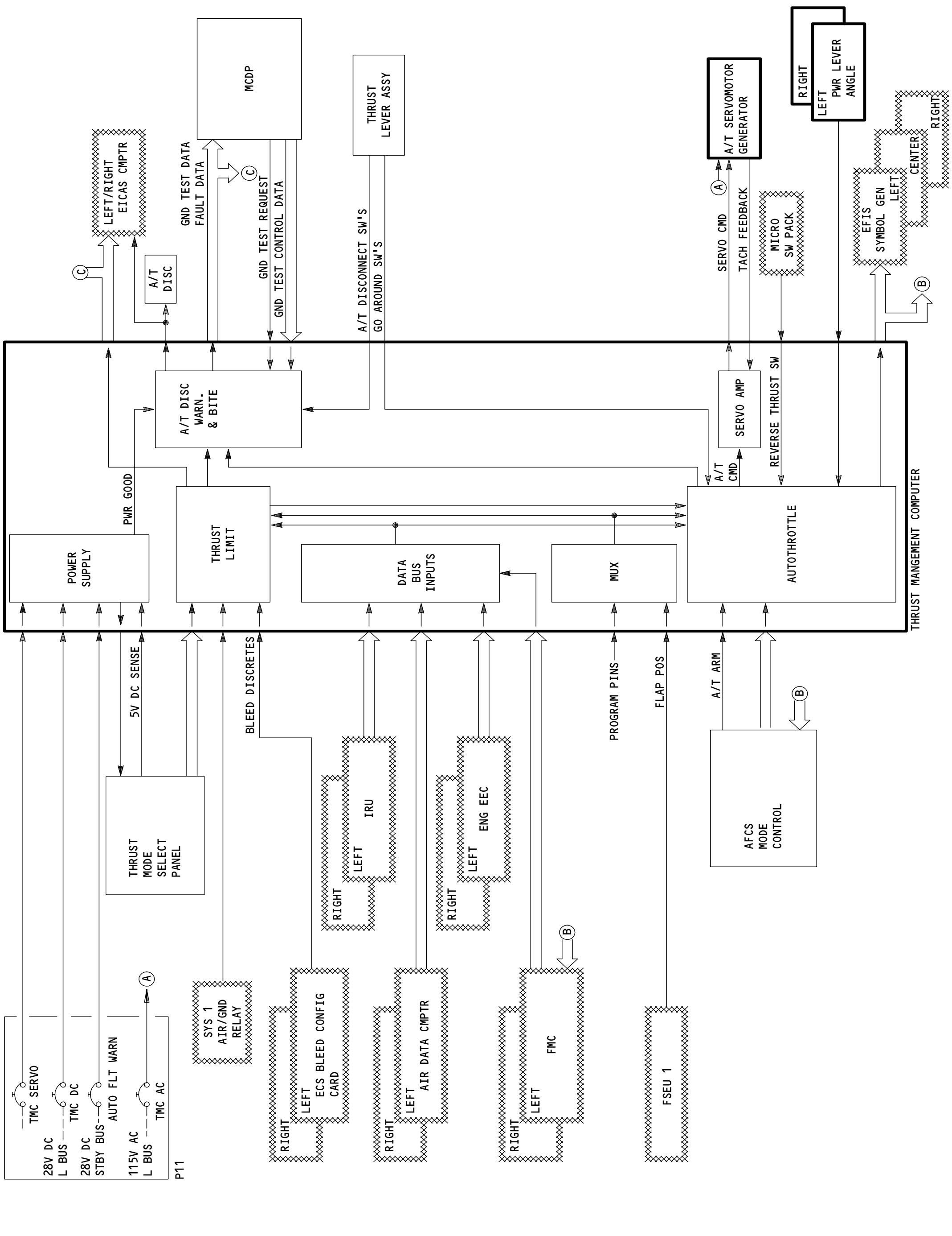
(3) Do the MCDP Ground Test 05 TMSP (AMM 22-00-02/201).

S 864-007

(4) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MANAGEMENT SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Thrust Management System (TMS) consists of a Thrust Management Computer (TMC) and Thrust Mode Select Panel (TMSP). It uses these to control a servomotor generator that drives the thrust levers. The TMS is a digital control system receiving inputs from the following primary interfaces:
      (1) Air Data Computers (ADCs)
      (2) Inertial Reference Units (IRUs)
      (3) Flight Management Computers (FMCs)
   B. The TMS functions include thrust limit computation, thrust lever control, and data display. Engine thrust limits are computed for the selected mode on the TMSP, then displayed on the EICAS display unit. The flight crew uses this data to monitor autothrottle (A/T) operations or to assist in manually setting engine thrust. The A/T servomotor generator is connected to thrust levers through a no-back clutch pack assembly. The clutch pack allows manual thrust commands without backdriving the servo. The autothrottle provides thrust lever control to selected inputs throughout the full flight regime. The control laws governing A/T operation are based on airspeed, Mach, vertical speed and thrust target limits.
   C. The autothrottle may operate independently or in a co-ordinated manner with the Flight Management Computer System (FMCS) and the Autopilot/Flight Director System (AFDS). The FMCS commands both the AFDS and TMS to follow flight path and speed commands as defined by FMC performance and vertical navigation functions (Ref 34-61-00, Flight Management Computer System, and 22-10-00, Autopilot (Flight Control)).
   D. Thrust Management Computer (TMC)
      (1) The TMC is on shelf E2-3 in the main electrical/electronic equipment bay. It provides thrust limit computation and thrust lever control to Engine Pressure Ratio (EPR), vertical speed, airspeed, and Mach references. Additional functions are provided for: minimum and maximum speed protection, thrust limit protection, and retarding thrust levers during autoland.
   E. Controls
      (1) The A/T ARM switch is on the left side of the Mode Control Panel (MCP). When armed, the MCP allows crew selection of autothrottle control modes. Engagement is accomplished directly through the EPR (or THR), SPD, and FLCH mode select switches, or indirectly through the VNAV mode select switch. When in the VNAV mode, the FMCS selects autothrottle modes and thrust limits.
      (2) The thrust mode select panel (TMSP) allows crew selection of thrust limit modes, alternate thrust limit ratings, and assumed temperature.



Thrust Management System - General
Figure 1 (Sheet 1)

TABLE 1 INPUT PARAMETERS

SOURCE	SIGNAL
MODE CONT PNL	ALT SEL
	ELEV SPD CMD
	MACH SEL
	SPD BRK HDL POS
	SPD SEL
	STAB POS
	DISCRETES:-
	IAS/MACH SEL
	F/D ON-F/O
	F/D ON-CAPT
	SPD MODE REQ
	THRUST MODE REQ
	ALT HOLD REQ
	FL CH MODE OPER
	V/NAV MODE
	G/S MODE OPER
	THROT RETARD
	PITCH SPD CONT
	A/P CMD C ENGA
	A/P CMD L ENGA
	A/P CMD R ENGA
	V/S MODE OPER
	AFDS G/A OPER
IRU	NORM ACCEL
	PITCH RATE
	ROLL RATE
	YAW RATE
	XTK HRZ ACCEL
	FPAC
	GS
	INS ALT
	INS V/S
	PITCH ANGLE
	ROLL ANGLE
	TKA TRUE
	TRUE HDG
	DISCRETES:-
	ALIGN FAULT
	NO IRS INITIAL
	EXCESS MOTION
	ADC/IRU FAULT
FMC	ASSUMED TEMP
	EPR TARGET
	SEL MACH
	TARGET AS
	A/T MACH MODE REQ
	A/T A/S MODE REQ
	CLB MODE REQ
	CON MODE REQ
	CRZ MODE REQ
	G/A MODE REQ
	T/O MODE REQ
	A/T LOW GAIN ARM
	IDLE THRUST REQ
	THROTTLE DORMANT
	THRUST MODE REQ
ADC	ALT (BARO 4)
	ALT (29,92)
	COMPUTED AS
	IMPACT PRESS
	IND AOA
	MACH
	MAX OPRTG SCHED
	SAT
	TAT
	TOTAL PRESS
	TAS
	DISCRETES:-
	P/S HT ON-L
	P/S HT ON-R
	TAT PROBE HT ON
EEC	EEC STATUS
	EPR COMMAND
	IDLE EPR
	MAX EPR LIMIT
TMSP	DISCRETES:-
	SEL TEMP BIT 0
	SEL TEMP BIT 1
	SEL TEMP BIT 2
	SEL TEMP BIT 3
	SEL TEMP BIT 4
	RATING 1 REQ
	RATING 2 REQ
	CON MODE REQ
	TO/GA MODE REQ
	CRZ MODE REQ
	CLB MODE REQ
MCDP	TEST CONT 13
	TEST CONT 14
	TEST CONT 15
	TEST CONT 16
EICAS	EPR ACTUAL-L
	EPR ACTUAL-R
	N3 ACTUAL-L
	N3 ACTUAL-R

TABLE 2 OUTPUT PARAMETERS

DESTINATION	SIGNAL
EFIS SYMBOL GENERATOR	A/T FAST/SLOW CMD [1]
	TMS FLT MODE ANN
FMC	TMS MODE STATUS
	ENG BLEED STATUS
	EPR ACTUAL - L
	EPR ACTUAL - R
	EPR REFERENCE
	FLAP POSITION
	TEMP SELECTED
	DISCRETE PARAMETERS 3
MODE CONT DNL (L AND R BUS)	TMS MODE STATUS
	VERT SPD CMD
EICAS CMPTR	EPR ACTUAL - L
	EPR ACTUAL - R
	EPR BUG DRIVE - L
	EPR BUG DRIVE - R
	MAX EPR LIMIT
	TAT
	TEMP SELECTED
	DISCRETE PARAMETERS 2
	DISCETE PARAMETERS 3
	EPR REFERENCE
	PLA-L
	PLA-R
MCDP	FAULT DATA
	GND TEST DATA
	INTERFACE FAULT DATA
DFDAU	FLIGHT RECORDER PARA-METERS

[1] GUI 009,115

TMS Digital Inputs and Outputs
Figure 1 (Sheet 2)

F. Thrust Lever Switches

(1) Disengagement of the autothrottle occurs if: either of two autothrottle disconnect switches (one on each thrust lever) is pressed, the A/T ARM switch on the MCP is positioned to OFF, or either thrust lever is set in the reverse thrust position. Pressing either of two go-around switches on the thrust levers causes the TMS to adjust thrust for a go-around.

G. Sensors

(1) The Inertial Reference Units (IRUs) and Air Data Computers (ADCs) provide environmental and attitude data to the TMS. The EPR transmitters provide actual EPR via the EICAS computers, and the autothrottle Power Lever Angle (PLA) transducers provide power lever angle to the TMS. The ECS bleed configuration cards provide engine bleed discrete signals.

H. Outputs

(1) TMS outputs consist of:

(a) TMC output to the servomotor generator which drives the thrust lever cables through no-back clutches.

(b) Fault data output to the MCDP.

(c) Display of the engaged autothrottle mode on the EADI.

(d) Display of thrust limit modes and calculated thrust limits on the EICAS display unit.

(e) Reference "bug" on EPR indicator (EICAS display unit) driven to calculated limit.

I. Annunciation

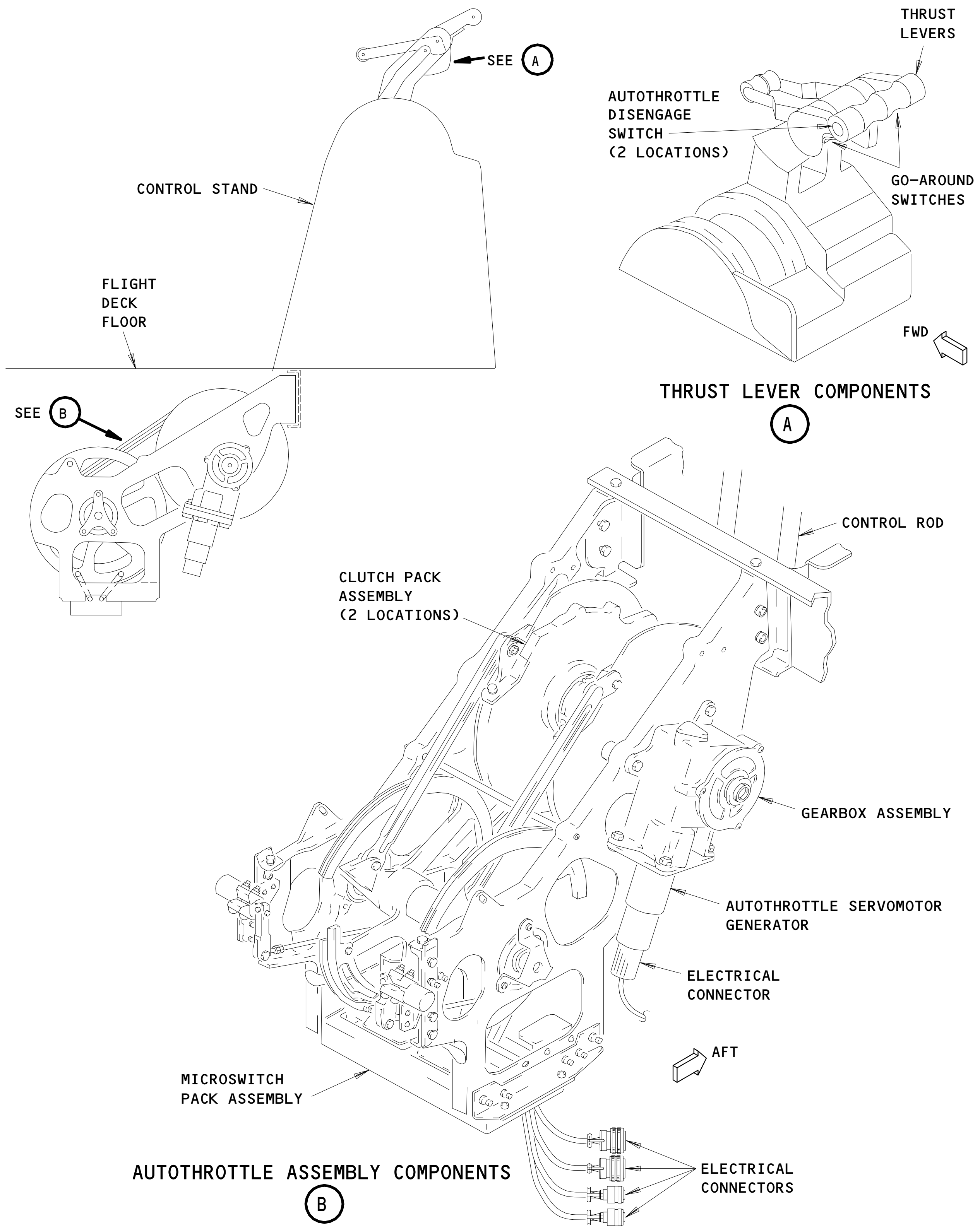
(1) Annunciation of selected autothrottle modes appears on the left side of the Electronic Attitude Director Indicator (EADI). Thrust limit modes are selected on the TMSP and displayed on the EICAS display unit. Upon autothrottle disengage, the A/T DISC light on the center instrument panel P1-3 and the Master Caution light on the glare shield illuminate. The A/T DISC light will also illuminate when the TMC is in ground test mode. The EICAS will display 'A/T DISC' and an aural caution will sound.

J. Test

(1) The Maintenance Control Display Panel (MCDP) records in-flight failures and initiates ground tests for the TMC. It identifies failures to the LRU level. In-flight failures are recorded for display during ground interrogation (Ref 22-41-00, Maintenance Monitor).

2. <u>Component Details</u>

A. Thrust Management System Components (Fig. 2)



Thrust Management System Components
Figure 2

(1) Autothrottle Servomotor Generator

(a) The autothrottle servomotor generator is located below the flight deck floor under the thrust lever control stand. This device automatically controls thrust lever position.

The A/T servomotor generator has two main components. A two-phase motor that moves the throttles as commanded by the TMC and a single phase generator (or tachometer) which provides servo rate feedback to the TMC.

- The servomotor has a fixed field winding which uses 115 vac excitation and two control windings which use 36 vac excitation (controlled by the servo power amplifier in the TMC). The servomotor drives at a rate proportional to the pulse width of the control voltage. Both excitation voltages are applied when the autothrottle is engaged.

- The tachometer is an AC generator with a linear relationship between shaft rpm and AC output voltage. The tachometer uses 115 vac excitation on the input winding and provides tachometer output for servo loop damping on the output winding.

(b) The splined output shaft of the servomotor generator mates with the gearbox assembly. Three bolts hold the servomotor generator to the gearbox. With the servomotor generator and gearbox reduction, the maximum thrust lever rate is 14 degrees/sec.

(2) A/T Disengage Switch

(a) An autothrottle disengage switch (double-push reset) is on the end of each thrust lever. It is a normally closed switch which, when depressed, disengages the autothrottle and provides an autothrottle disengage warning. A second press of the switch resets the warning circuits.

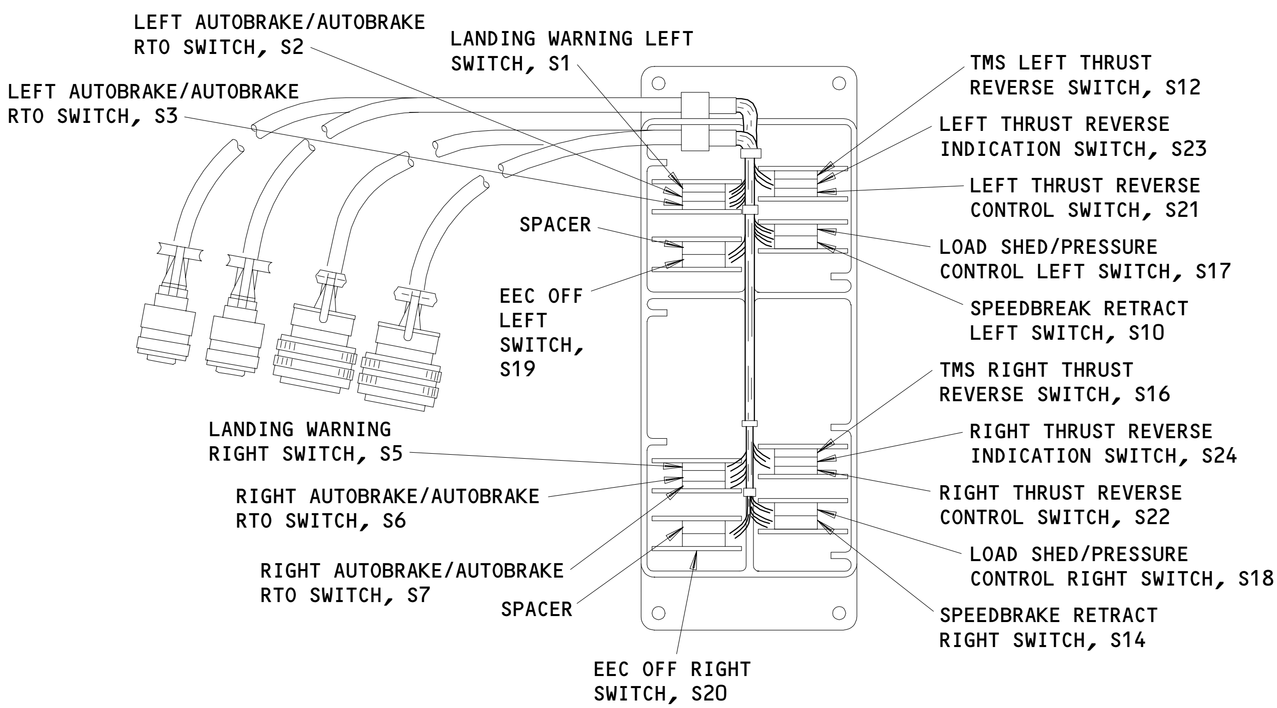
(3) Go-Around Switch

(a) One palm-operated go-around switch is at the base of each thrust lever. The go-around switches are wired in series with the go-around logic in the TMC. Actuating the switch engages the go-around mode if the go-around mode is armed (Ref 22-11-00, Autopilot/Flight Director Power for additional description).

(4) Clutch Pack Assembly

(a) The clutch pack assembly is directly under the thrust lever control stand. The clutch pack consists of two clutch assemblies mounted on a common shaft. One clutch is used by each thrust lever. Manual pressure applied to the thrust levers operates the cam mechanism that releases the clutch. The cam mechanism consists of normally closed brake shoes that transmit the autothrottle drive to the thrust lever linkage and autothrottle cable quadrant. Manually moving the thrust levers cams the brake shoes open and drives only the cable quadrant.

(5) Microswitch Pack Assembly (Fig. 3)



Microswitch Pack Assembly
Figure 3

(a) The microswitch pack assembly is directly under the clutch pack assembly. It is held in place by four bolts. The assembly is capable of containing 16 sets of switches. The number of switches installed depends on airplane configuration. The switches contained in the assembly are shown in Fig. 3.

(b) The thrust reverse switches are used by the TMS. One set is operated by the left thrust lever and the other set is operated by the right thrust lever. The switches are actuated when the reverse thrust levers are engaged. Thrust limit computation by the TMC is put in the reverse mode and the autothrottle is disengaged. The microswitch pack assembly includes wiring and electrical connectors that mate with airplane wiring.

(c) The arm assembly on the flap handle is moved by the flap lever quadrant in the flaps 1 or greater position. The switch assembly contains four sets of switches which are activated by the arm assembly movement. Three sets are used by the flight control computers. One set is used by the TMC.

3. Operation

A. Functional Description

(1) Thrust Management System - Block Diagram (Fig. 4)

(a) Thrust Control

1) Control of thrust is provided by the thrust levers (manually) and the A/T servomotor generator (automatically). They supply a mechanical input to the fuel flow governor, which is on the engine. The autothrottle PLA transducers on the engines provide power lever angle feedback to the TMC to close the servo loop. The EPR transmitters provide actual EPR to the Electronic Engine Control (EEC) and EICAS computers. The EEC uses this information along with throttle lever angle, air data, and bleed status to:

a) Control and limit thrust

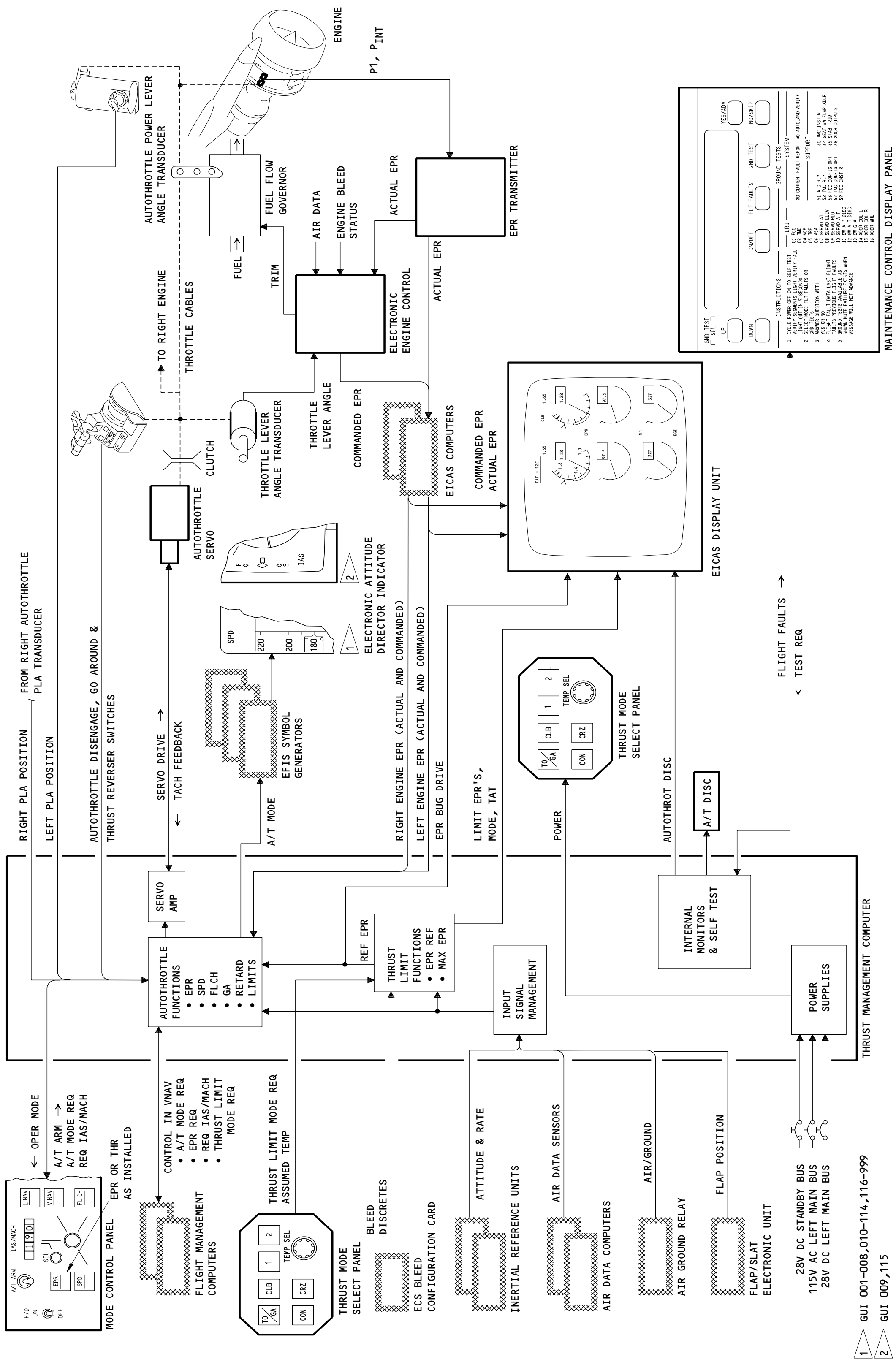
b) Linearize the throttle-to-thrust-rating relationship

c) Drive the engine thrust indicator command sector on the upper EICAS display unit.

(b) Flight Deck Indications

1) The primary engine format is shown on the upper EICAS display unit. Engine parameters shown on this format are those required to set and monitor engine thrust. Digital displays are provided for reference EPR, thrust limit mode, total air temperature, and selected temperature. The EPR indicators on the EICAS display unit provide an analog type round dial display for reference EPR, commanded EPR, and actual EPR. Caution advisory messages relating to the TMS are also displayed on the EICAS display unit.

2) GUI 009, 115;
the lower left side of the EADI displays the engaged autothrottle mode. The fast/slow display on the EADI indicates speed performance to the crew.



Thrust Management System - Block Diagram
Figure 4

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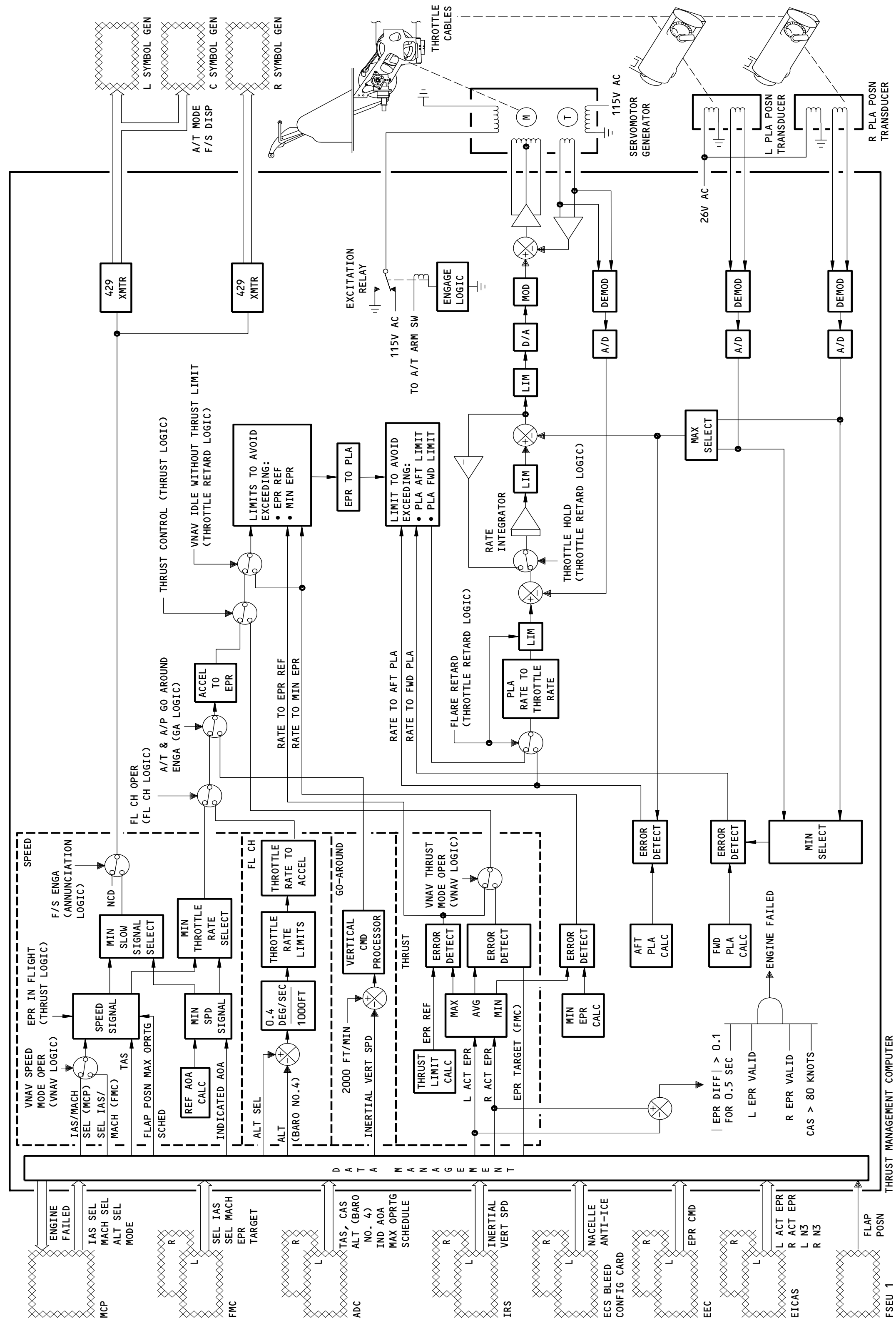
3) GUI 001-008, 010-114, 116-999;
   the upper left corner of the EADI displays the engaged autothrottle mode and autothrottle status.
   The fast/slow display on the EADI indicates speed performance to the crew.
4) The A/T DISC light on panel P1-3 turns on when the autothrottle disengages.

(c) Sensor Inputs
   1) Sensor inputs to the TMC are:
      a) Air data from ADCs
      b) Attitude and rate from IRS
      c) Aircraft configuration discretes

(d) Control Inputs
   1) Control inputs are received from three sources. The MCP provides A/T ARM, A/T mode select, and IAS/Mach select. The TMSP provides thrust limit mode requests and assumed temperature. Autothrottle mode requests; EPR, IAS, and Mach targets; and thrust limit mode request are supplied by the FMC when VNAV is engaged.

(e) TMC Internal Functions
   1) Autothrottle functions (Fig. 5):
      a) Thrust is controlled in the EPR (or THR) mode to the computed EPR REF target. When VNAV is engaged, the FMC computes the EPR REF target. If the SPD mode is selected on the MCP, speed is controlled to the selected target on the IAS/MACH display. The FMC selects the target speed in the VNAV mode. In the Flight Level Change (FLCH) mode, thrust is controlled to computed EPR REF for climb and to the idle limit for descent. The TMC sets thrust to the EPR REF for an autothrottle only Go-Around, or controls rate of climb during Go-Around with an autopilot channel engaged in CMD and rate of retard during flare. All of these functions are subject to EPR, PLA, and speed limits.
   2) Thrust Limit Functions:
      a) The REF and MAX EPR thrust limits are computed by the TMC. The limits are based on engine type, mode selected, flight conditions, and airplane configuration.



Autothrottle Block Diagram
Figure 5 (Sheet 1)

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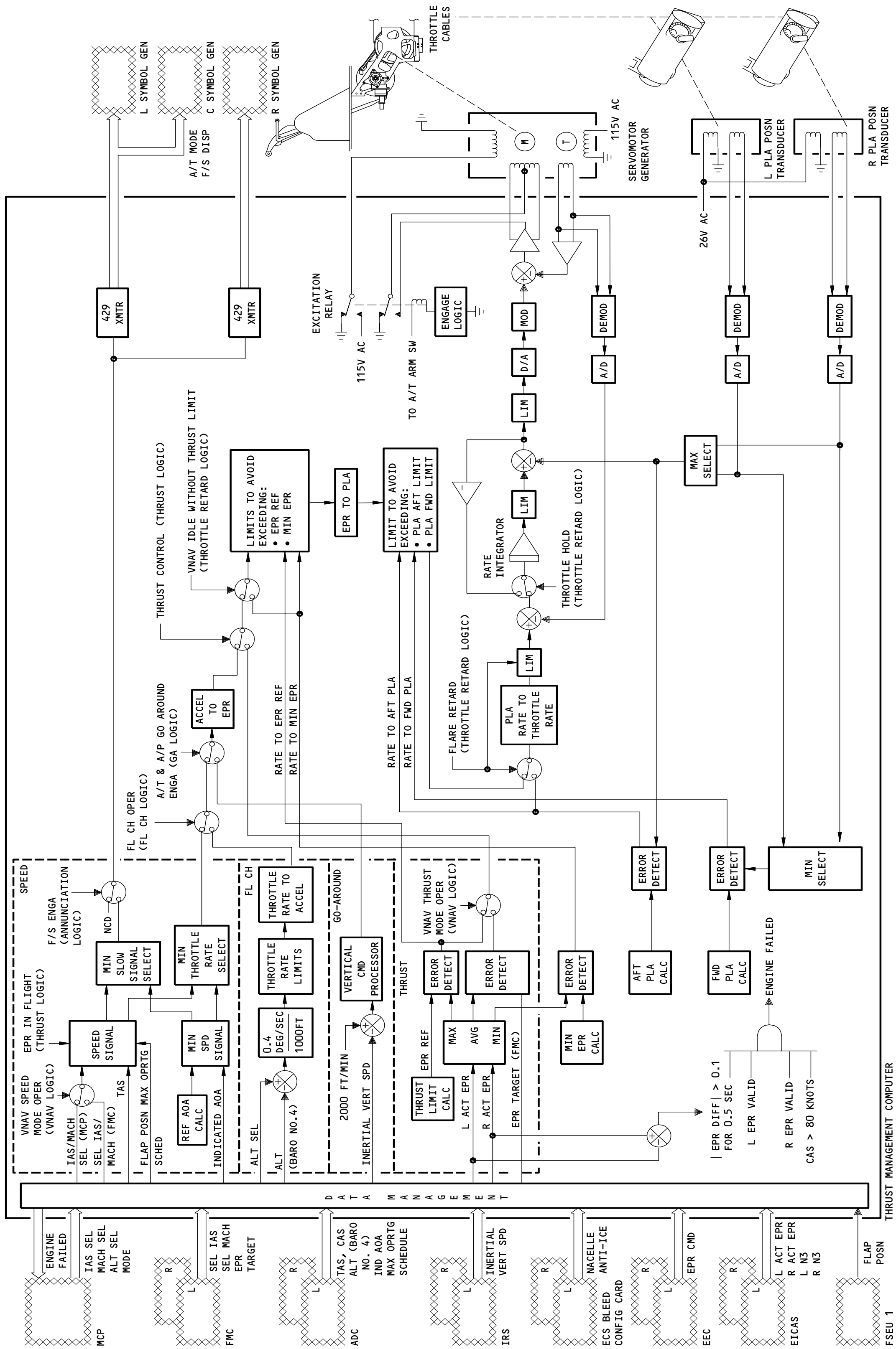
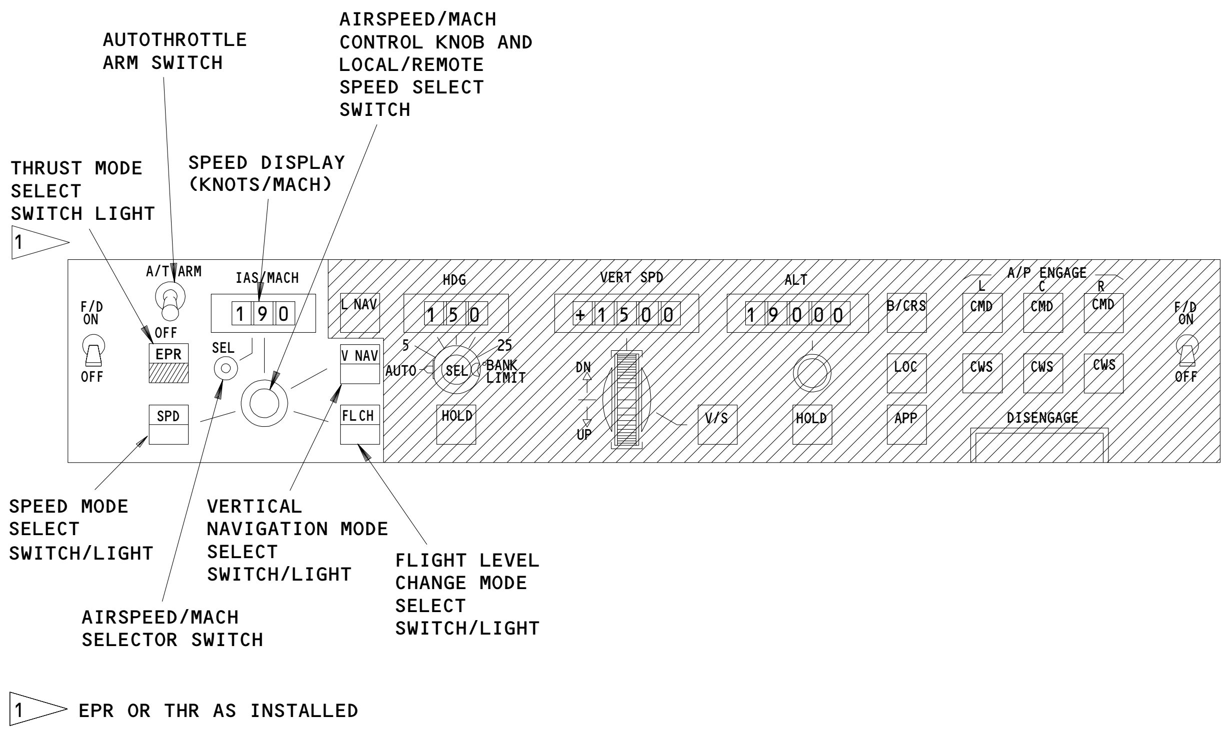


Autothrottle Block Diagram
Figure 5 (Sheet 2)

3) Input Signal Management:
   a) Selection of signal source is automatic. No external switching is required by the crew. Signals are monitored for validity. If the selected source becomes invalid, automatic transfer to the opposite source occurs.
4) Internal Monitors and Self Tests:
   a) The internal monitoring system of the TMC performs tests requested by the MCDP and senses autothrottle disconnect. All faults are recorded and transmitted to the MCDP.

(2) Mode Control Panel - Autothrottle Functions (Fig. 6)
   (a) Autothrottle Arm Switch
      1) The A/T arm switch is a two-position toggle switch that sends a 28 vdc analog discrete signal to the TMC. When positioned to ARM, it arms the excitation relay. The excitation relay connects the servomotor generator's 115 vac excitation and 36 vac servo control voltages, only after a mode is engaged.
   (b) Mode Select Switches (EPR/THR, SPD, FLCH, VNAV)
      1) The mode select switches provide mode engagement after arming has occurred.The switch dot-bar matrix illuminates when the mode request is accepted and operable.



Mode Control Panel - Autothrottle Functions
Figure 6

(c) Speed Switches and Display
1) Speed select is displayed in two forms:
a) 100-399 knots in increments of 1 knot.
b) 0.400-0.950 Mach in increments of 0.001 Mach.
2) An airspeed/Mach selector (SEL) switch transfers the speed display between airspeed and Mach. Airplane Mach must be at least 0.40 before transfer to the Mach display can be accomplished. Rotating the IAS/MACH control knob adjusts the speed/Mach display during local command. Pushing the knob transfers control of the speed reference bug on the MASI between the MCP and FMC. The speed/Mach display is blank when control is from the FMC.

(3) Electronic Attitude Director Indicator (EADI) - Autothrottle Displays (Fig. 7)

(a) Two types of status modes are displayed on the EADI:

1) A/T for autothrottle displayed.

2) F/S for autothrottle off, no autopilot speed control, and F/S reflecting a limit rather than airspeed bug.

(b) Two types of autothrottle modes are displayed on the EADI:

1) EPR, IAS, MACH, FL CH, GA, IDLE, TEST, THR HOLD for operating modes.

2) FLAP LIM, ALPHA, IAS LIM, MACH LIM for limit modes

(c) GUI 009, 115;
a fast/slow display on the left side of the EADI indicates speed performance to the crew. The display is active when within 10 knots of the reference airspeed.

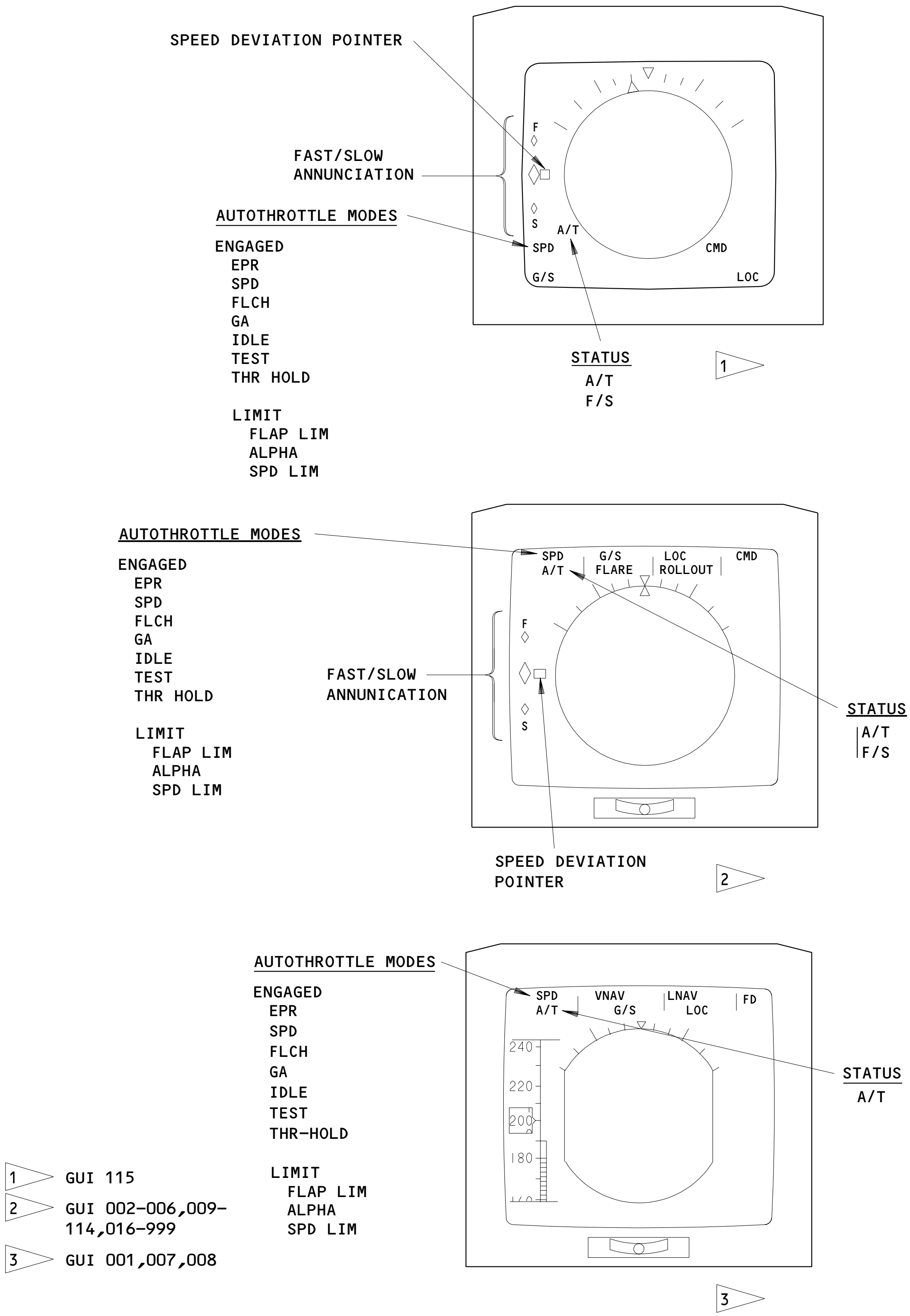
(4) EICAS Display Unit - Autothrottle Functions (Fig. 8)

(a) The upper EICAS display unit provides the primary engine parameters and TMC computed data required to set and monitor thrusts. These parameters are displayed continuously in both analog type round dial and digital readout for the left and right engines.

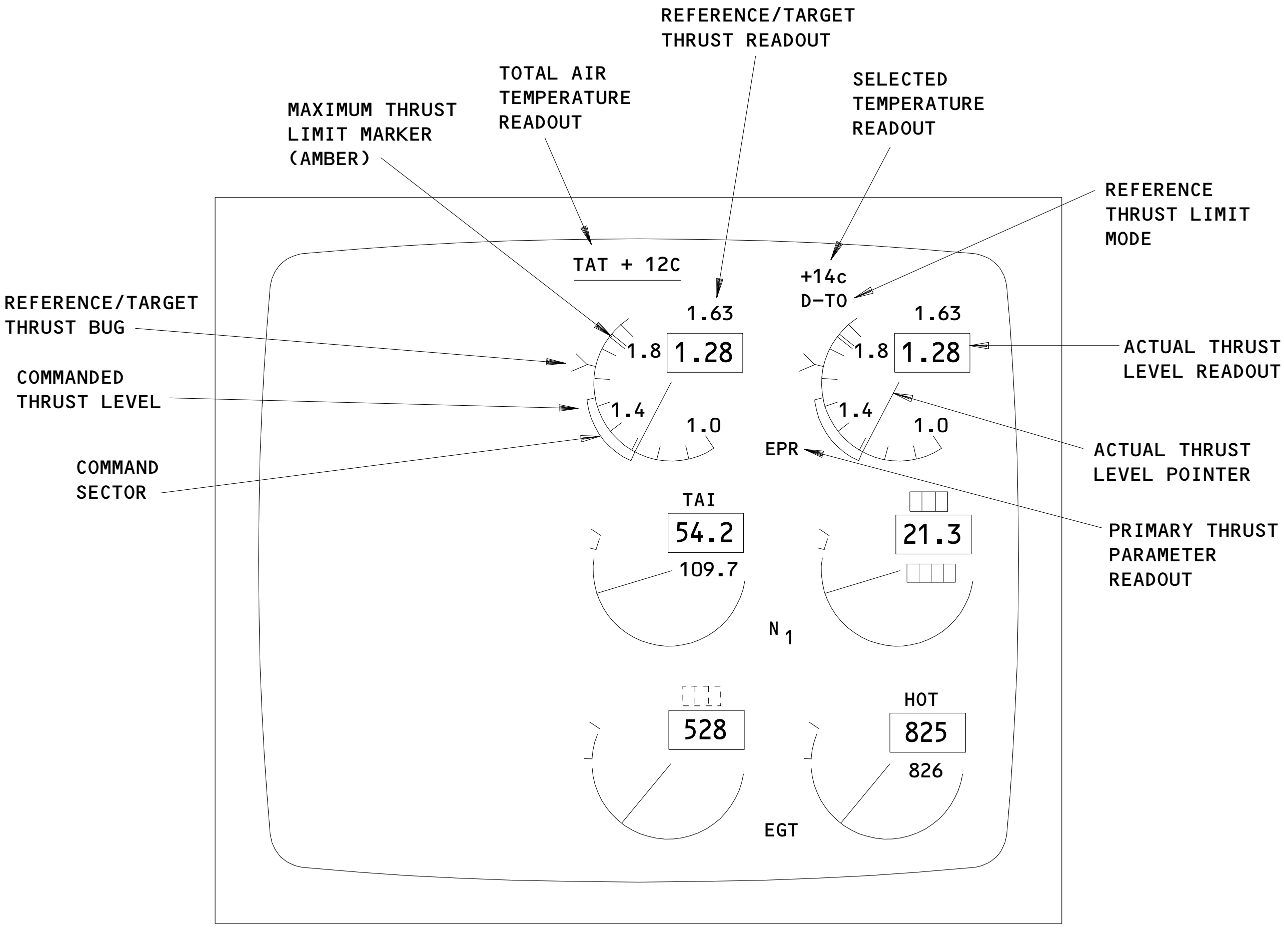
(b) Actual engine thrust is indicated by both actual thrust pointers and digital readout. Any momentary difference between actual thrust and commanded thrust is shown by the command sector. The maximum thrust limit as computed by the TMC or EEC is shown by an amber marker on the analog scale.

(c) Target thrust (reference EPR) is shown by a reference cursor on the analog scale and a digital readout above the actual thrust readout. The thrust limit mode and assumed temperature (if selected) are shown above the EPR indicators. When an assumed temperature is selected while the airplane is on the ground, a D appears in front of the thrust limit mode (only valid when in Take-Off EPR reference mode).

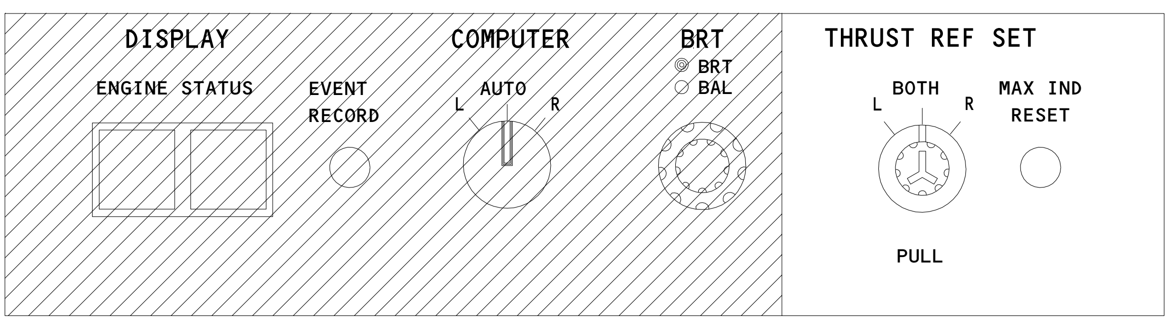
(d) Digital displays are also provided for total air temperature (TAT) and thrust reverser status. If a thrust reverse lever is fully deployed, the readout (REV) is green. The readout is yellow while the lever is in transit. For normal in-flight conditions, the readout is blank.



EADI - Autothrottle Displays
Figure 7



UPPER EICAS DISPLAY UNIT



EICAS CONTROL PANEL
(EXAMPLE)

EICAS Display - Autothrottle Functions
Figure 8

(e) The manual THRUST REF SET control on the EICAS control panel consists of a dual rotary switch combined with a push-pull switch. The inner control is a continuously rotating control with 12 detents per revolution. When initially set (pulled out), the readout corresponding to the engine selector switch (L, R, or BOTH) defaults to 1.55 EPR. Clockwise rotation of the control increases the readout and counter clockwise rotation decreases the readout. The rotation sensitivity is 0.01 EPR per detent. When a readout is modified manually its last input data is retained, regardless of engine selector switch position, until the inner control is set (pushed in). The outer control is a three-position rotary switch that selects the L (left), BOTH, or R (right) EPR readout.

(5) Thrust Management System - Engage/Disengage Logic (Fig. 9)

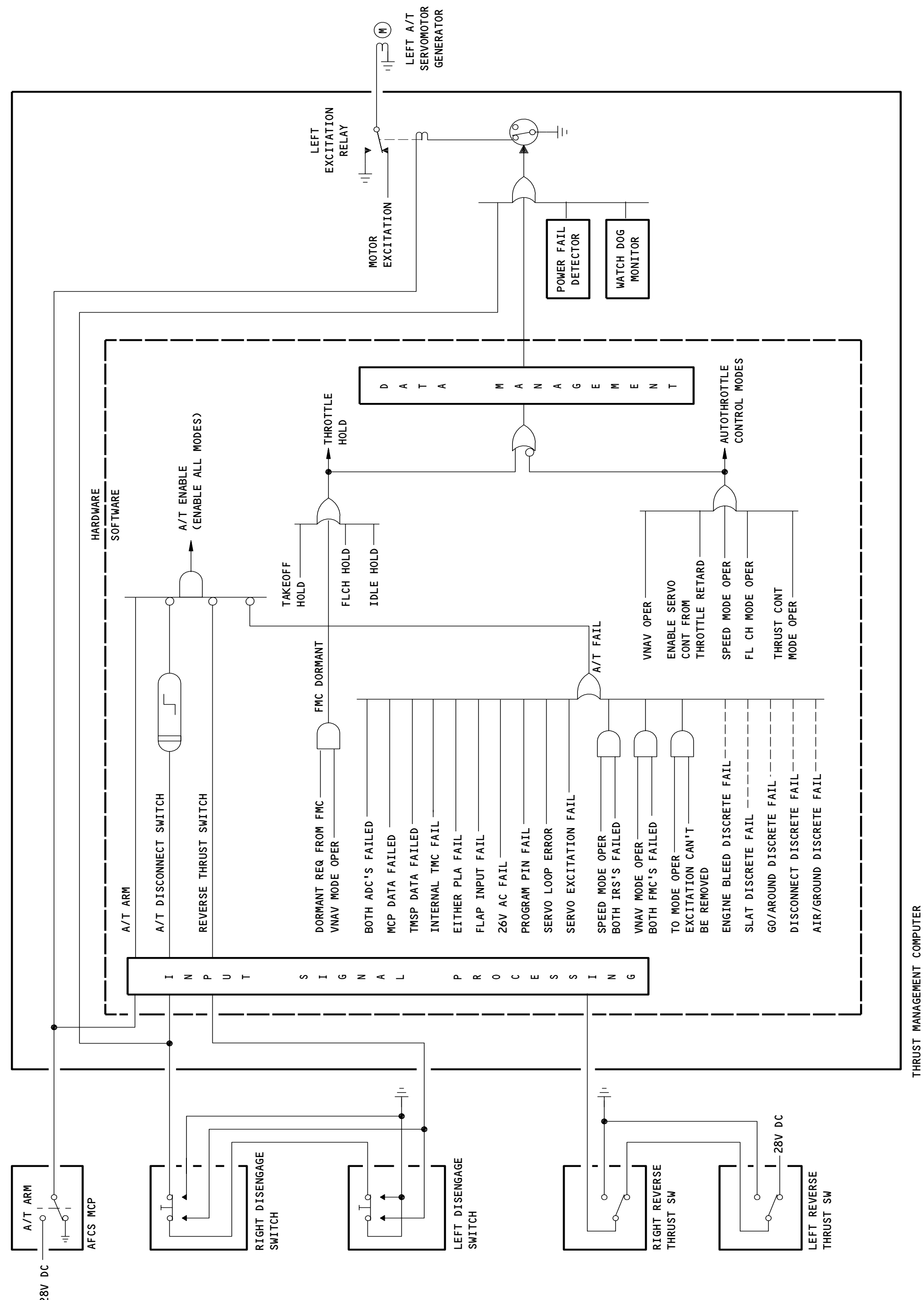
(a) The engage-disengage logic consists of both engage and mode logic. Major input signal sources supplying information for engage/disengage control laws are the FMC, MCP, EECs, and ADCs. All inputs are used to control the motor excitation relay, which connects excitation power and control inputs to the A/T servomotor generator.

(b) Input signals converted by software include digital and analog discretes, and digital variable signals. Analog discrete signals used for logic functions are sent to A/D converters before software usage.

(c) Analog discrete inputs:
1) A/T disconnect and reverse thrust
2) Air/ground
3) Autothrottle arm

(d) Digital variable inputs (from ADCs):
1) Calibrated airspeed
2) Filtered standard barometric altitude

(e) Digital discrete inputs:
1) VNAV mode operational, throttle retard, and thrust mode request (from MCP)
2) EEC off and EEC valid (from EECs)
3) A/T dormant, thrust request, and idle request (from FMCs)



Thrust Management System - Engage/Disengage Logic
Figure 9 (Sheet 1)

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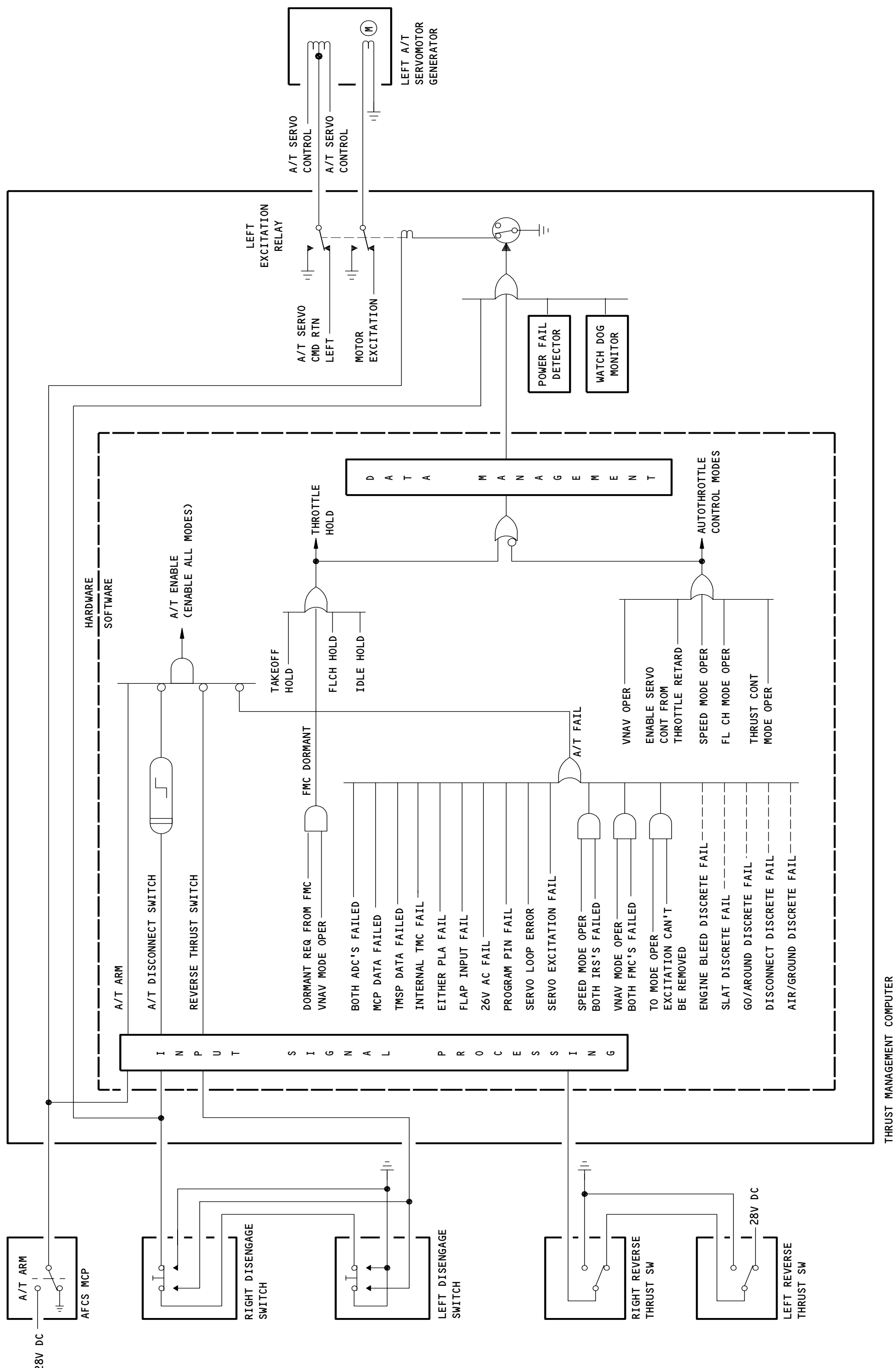

Thrust Management System - Engage/Disengage Logic
Figure 9 (Sheet 2)

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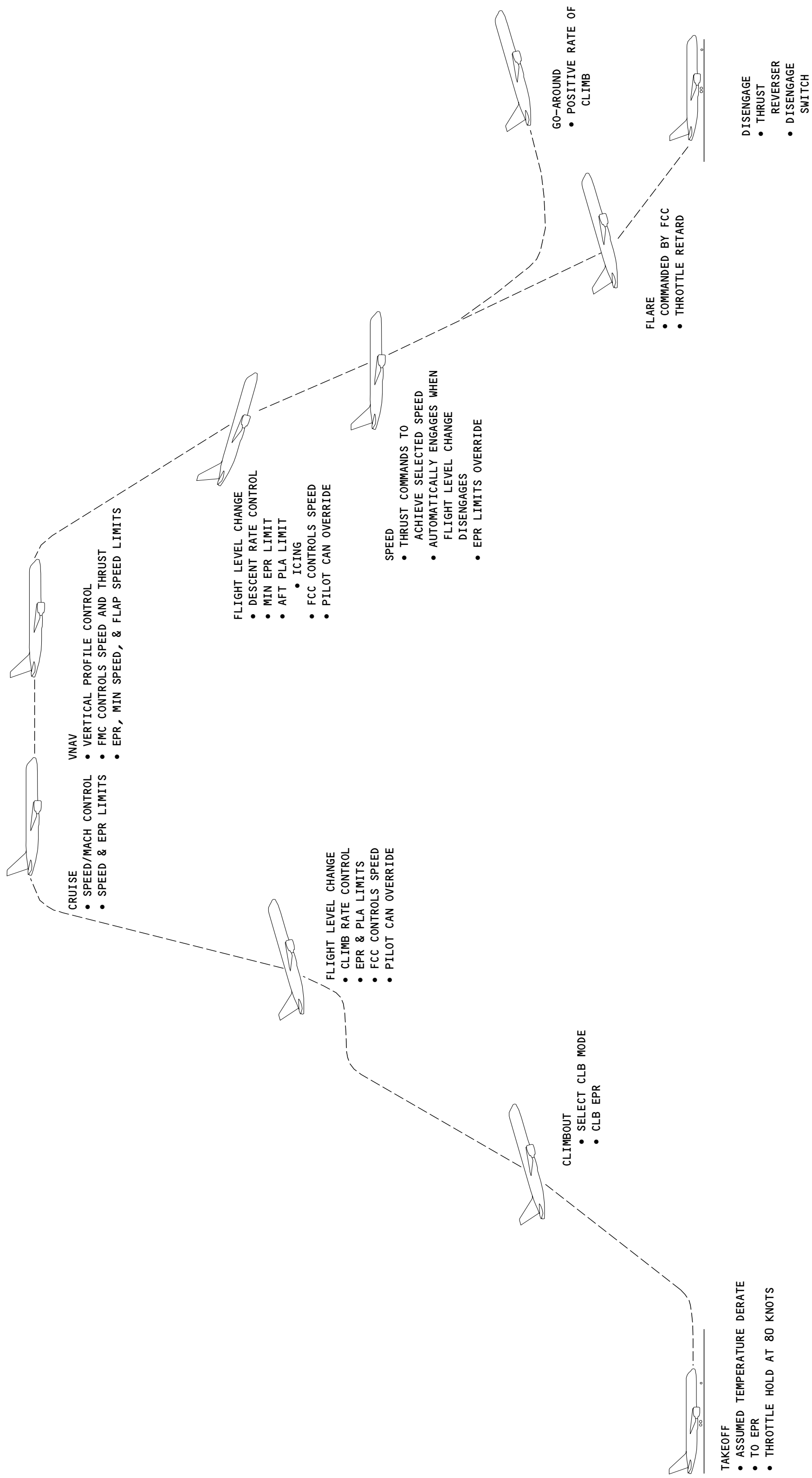
(f) When the A/T switch is in the ARM position, the TMC is enabled to engage the autothrottle servo. The servo controls thrust according to manually or automatically selected operating modes, provided there are no detected faults in the system. Any detected fault affecting autothrottle operation in the selected mode results in servo disengagement and corresponding annunciation on the EICAS display unit. Manual engagement into SPD or EPR (or THR) modes is obtained by pressing the corresponding switch on the MCP. Automatic engagement occurs when either the FLCH or VNAV mode is selected and is successfully engaged in the AFDS. When the autothrottle is engaged, the operating mode is displayed on the the EADI. Disengagement of the autothrottle occurs if:
1) A fault affecting autothrottle operation in the selected mode is detected.
2) An autothrottle disconnect switch on the thrust levers is pressed.
3) The A/T ARM switch is positioned to OFF.
4) The reverse thrust levers are set in the reverse thrust position.

(6) Typical Autothrottle Flight Profile (Fig. 10)
(a) Takeoff (T/O)
1) When TO is selected on the TMSP, a reference EPR is calculated based on temperature and altitude. A provision to insert artificially high temperatures for derated takeoff is included. When the EPR (or THR) mode on the MCP is engaged, the thrust levers are driven until the reference EPR is reached. The T/O mode deactivates the autothrottle servo by removing the 115v ac excitation and zeroing the command when the airplane reaches 80 knots CAS. This prevents the thrust levers from retarding after T/O thrust is set.
(b) Climbout (CLB)
1) When the airplane is more than 400 feet above ground level and the climb mode is selected on the TMSP, a new EPR REF is calculated by the TMC. The engines are retarded to the new EPR REF.
(c) Flight Level Change (FLCH) - Climb
1) Engagement of the FLCH mode on the MCP causes the TMS to accelerate the airplane by driving the thrust levers until the reference EPR is reached. The EPR REF and maximum PLA limits are not exceeded. Manual override of the thrust levers by the pilot causes the autothrottle servo to hold until a new mode is selected, capture of the selected altitude is achieved, or the REF EPR is exceeded.
(d) Cruise (CRZ)
1) When CRZ is selected on the TMSP, the thrust levers control to the speed/Mach reference on the MCP. The maximum/minimum speeds, EPR, and PLA limits are still observed.



Typical Autothrottle Flight Profile
Figure 10

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(e) Vertical Navigation (VNAV)
   1) The VNAV mode transfers both pitch and thrust control to the Flight Management Computer System (FMCS). The FMCS selects autothrottle modes (airspeed, Mach, and thrust) and provides the corresponding target reference values. The FMCS also commands throttle hold and retard to idle. All commands from the FMCS are subject to the thrust and speed limits computed by the TMS.

(f) Flight Level Change - Descend
   1) Selecting FLCH causes thrust lever retard to the computed aft limit until selected altitude is captured. Manual override of the thrust levers by a pilot causes autothrottle servo hold.

(g) Speed (SPD)
   1) The speed mode is selected manually by pressing the SPD pushbutton on the MCP. The autothrottle engages to control airspeed (or Mach) to the reference command on the IAS/MACH display on the MCP. The speed mode is automatically engaged upon capture of the selected altitude when in FLCH mode, or when the FLCH mode is exited.

(h) Go-Around (G/A)
   1) The thrust levers are automatically advanced for a positive rate-of-climb when a go-around switch on the thrust levers is pressed.

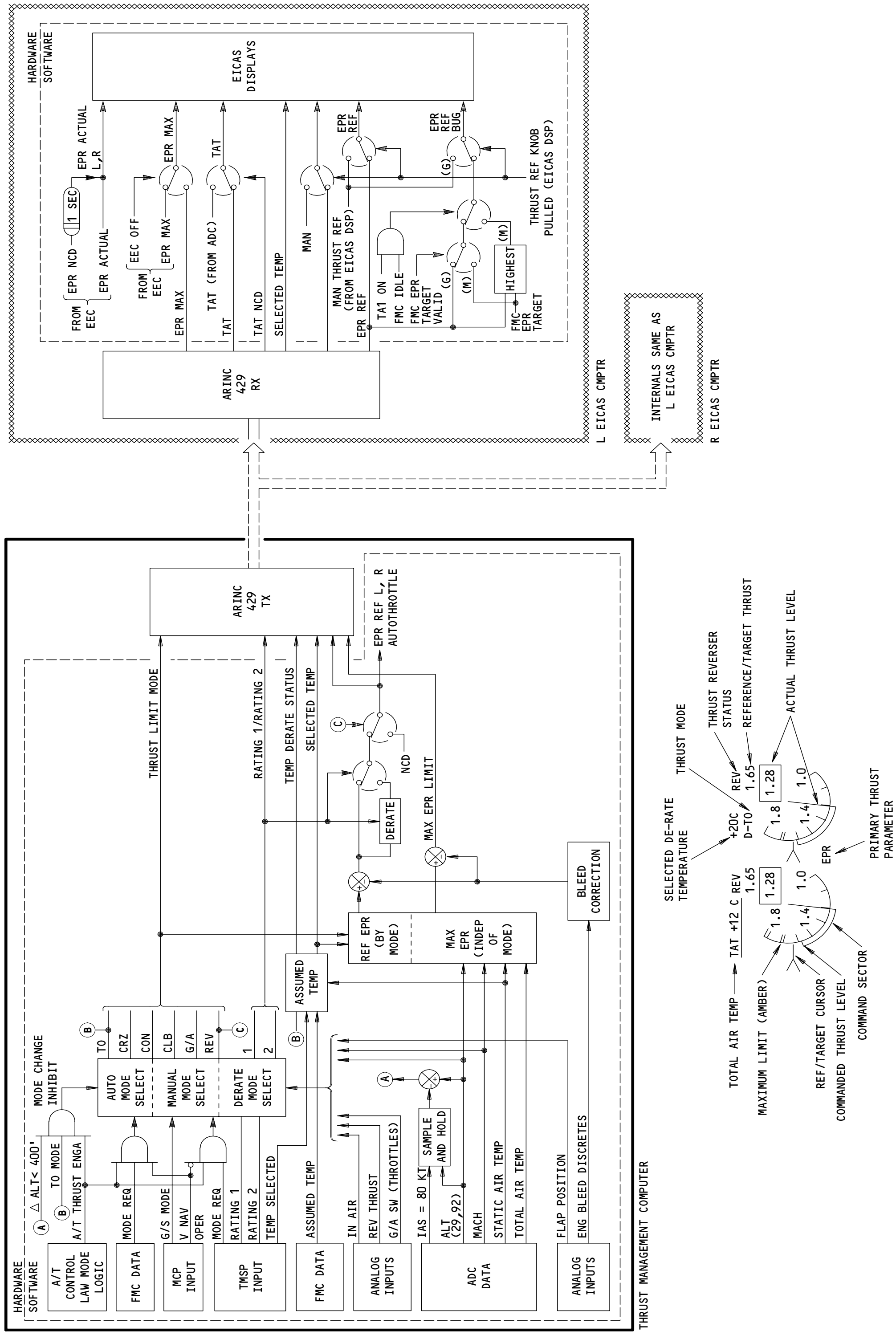
(i) Flare
   1) During an A/P multi-channel approach, the thrust levers retard to the aft limit on command from the FCCs.

(7) Thrust Management System - Mode Select Logic
   (a) The input signals shown are used for both mode and annunciation logic. With the autothrottle switch in the ARM position, selection of alternate modes is accomplished manually by actuating the proper controls, or automatically if proper conditions are met. The annunciation logic provides output signals for display on the EADI.
   (b) Digital data bus inputs:
      1) Autothrottle inputs from the MCP (discrete)
      2) Control inputs from the FMCs: thrust mode valid, thrust mode REQ, A/S mode REQ, and Mach mode REQ are discrete signals; and Mach selected and airspeed selected are variable signals.
      3) Environmental data from the ADCs (variable)
   (c) Analog inputs:
      1) Flap position from the flap/slat electronic unit
   (d) Analog discrete inputs:
      1) Autothrottle arm from the MCP, and ground test from the MCDP.
      2) Go-around engage from the go-around switches and in air from the air/ground relay.

(e) Logic Modes
   1) The autothrottle has three basic modes of operation: speed, thrust, and vertical speed. The speed (SPD) mode allows the autothrottle to control to either airspeed or Mach, depending on the control parameter selection. The thrust control mode provides control to a target EPR command, EPR reference limit, or MAX or MIN EPR limit depending on the flight condition. The Flight Level Change (FLCH) mode provides automatic control of thrust to EPR REF for climb and to the computed aft limit for descent. The Vertical Navigation (VNAV) mode transfers control of pitch and thrust to the FMCS. When the autothrottle is engaged, the operating mode is annunciated on the EADI.

(f) The following output signals to the EADI (for annunciation) are digital discretes:
   1) EPR, IAS, and MACH mode
   2) THR-HOLD, IDLE, and TEST mode
   3) FLAP LIM, ALPHA, IAS LIM, and MACH LIM mode
   4) FLCH and GA mode

(8) Thrust Management System - Thrust Limit Mode Logic (Fig. 11)
   (a) The thrust limit mode logic sets the mode for thrust limit calculations. Thrust modes are selected by TMSP switch engagement, FMC request when the VNAV mode is engaged, or upon power-up based on the flight condition.
   (b) Selection of a thrust limit mode disengages the mode that is in operation unless the conditions that follow occur:
      1) Reverse thrust or the landing flap is selected, or
      2) the glide slope is active from an Autopilot or a Flight Director, and
      3) the T/O thrust limit mode is not selected.
   (c) The thrust limit mode that is engaged continuously shows on the upper EICAS display unit.
   (d) When VNAV is engaged with the Autothrottle on, the TMSP switches are locked out.
   (e) T/O Mode
      1) The T/O thrust limit mode are entered four ways as follows:
         a) A TO/GA request from the TMSP when the airplane is on ground. This can only be used if the reverse thrust modes are not engaged.
         b) Automatically when the flap position is less than 23 degrees and the Mach is less than 0.12. This logic sets the T/O thrust limit mode after a landing without a use of the reverse thrust.
         c) Automatically upon power-up on ground unless reverse thrust is engaged.
         d) Automatically when reverse thrust is switched from on to off while the airplane is on the ground.
   (f) G/A Mode
      1) The G/A thrust limit mode is entered six ways as follows:
         a) A manual TO/GA request from the TMSP while in air and the VNAV mode is disengaged.



TMC Thrust Limit Schematic
Figure 11

b) An FMC G/A request when VNAV is engaged.
c) Automatically at the TMC power up initialization in air.
d) Automatically when the airplane is in the air, and the flap position is greater than 23 or the glide slope is captured. And also when the T/O thrust limit mode is not selected.
e) Automatically when flaps transition from 0 to 1.
f) Automatically when the Autothrottle G/A mode is engaged.

(g) CON Mode
1) The CON mode is entered two ways as follows:
a) A manual CON request from the TMSP when VNAV and reverse thrust modes are disengaged.
b) An FMC CON request when VNAV is engaged.

(h) CLB Mode
1) The CLB mode is entered two ways as follows:
a) A manual CLB request from the TMSP when the VNAV and the reverse thrust modes are disengaged. Also, the landing flaps are not set or the glide slope is not active.
b) A FMC CLB request when VNAV is engaged.

(i) CRZ Mode
1) The CRZ mode is entered two ways:
a) A manual CRZ request from the TMSP when the VNAV and the reverse thrust are disengaged and the aircraft is above 400 feet AGL. Also, the landing flaps are not set and the glide slope is not active from an Autopilot or a Flight Director.
b) A FMC CRZ request when the VNAV is engaged.

(j) REV Mode
1) The REV mode is engaged when the airplane is on the ground and the reverse thrust levers are moved to the reverse thrust position, and the autothrottle is disengaged.

(9) Thrust Limit Derate Logic
(a) Fixed Percentage Derates are available for CLB mode only. You can push the 1 or 2 button on the TMSP to preselect a CLB derate while in a mode other than CLB. When the CLB derate mode is preselected, T/O-X shows on the EICAS (where X shows the applicable derate number in white). When CLB mode becomes active, the preselected derate mode becomes the active derate. The EICAS shows CLB-X with the color of the derate number changed to green.

(b) When the airplane is on the ground and in the T/O mode, the TMC allows the assumed temperature entry from the TMSP or the FMC CDU for reduced T/O thrust.

(c) The TMC normally computes a thrust limit based on ambient temperature or flat rated temperature, whichever is greater. This value is used as the computation temperature. A further manual increase in computation temperature is referred to as assumed temperature and decrease thrust.

(d) Assumed temperature data is sent to the TMC for thrust limit calculation. If the assumed temperature is higher than flat rated and ambient temperature, the derated mode shows as a "D" adjacent to the mode display on the EICAS display unit. Assumed temperature can be set as follows:

1) Engage in a different limit mode to leave the T/O mode.
2) Push the GA switch on the throttle levers after lift off.
3) Cycle the TMC power.
4) Push the TO/GA button on the TMSP before lift off.
5) Push the 1 or 2 botton on the TMSP.

(e) When the CLB mode is engaged, the CLB derated thrust can be selected on the TMSP. The 1 or 2 buttons on the TMSP gives a fixed decrease in thrust.

(f) When a CLB derate is preselected and you engage the CLB thrust limit mode, it will provides the CLB derated thrust.

(10) Thrust Limit Calculation

(a) The TMC calculates EPR REF and MAX EPR based on:

1) Selected mode (effects EPR REF only)
2) Environmental data
3) Engine bleed status

(b) The TMC does these calculations regardless of A/T engagement status. Calculated limits show on the upper EICAS display unit. However, the TMC calculated MAX EPR limit only shows if the applicable EPR MAX limit from the EEC is not available.

(c) EPR Calculation

1) The EPR limit calculation alternates between MAX EPR and EPR REF on each iteration pass of the computer. The REF calculations are based on thrust limit mode, Mach, temperature, and altitude. The MAX calculations are based on Mach, temperature, and altitude only. The temperature used for T/O and G/A thrust limit calculation is the highest value of flat rated, ambient, or assumed temperature. The temperature used for all other thrust modes is the higher of Total Air Temperature (TAT) and Static Air Temperature (SAT). Standard barometric altitude is used for altitude correction.

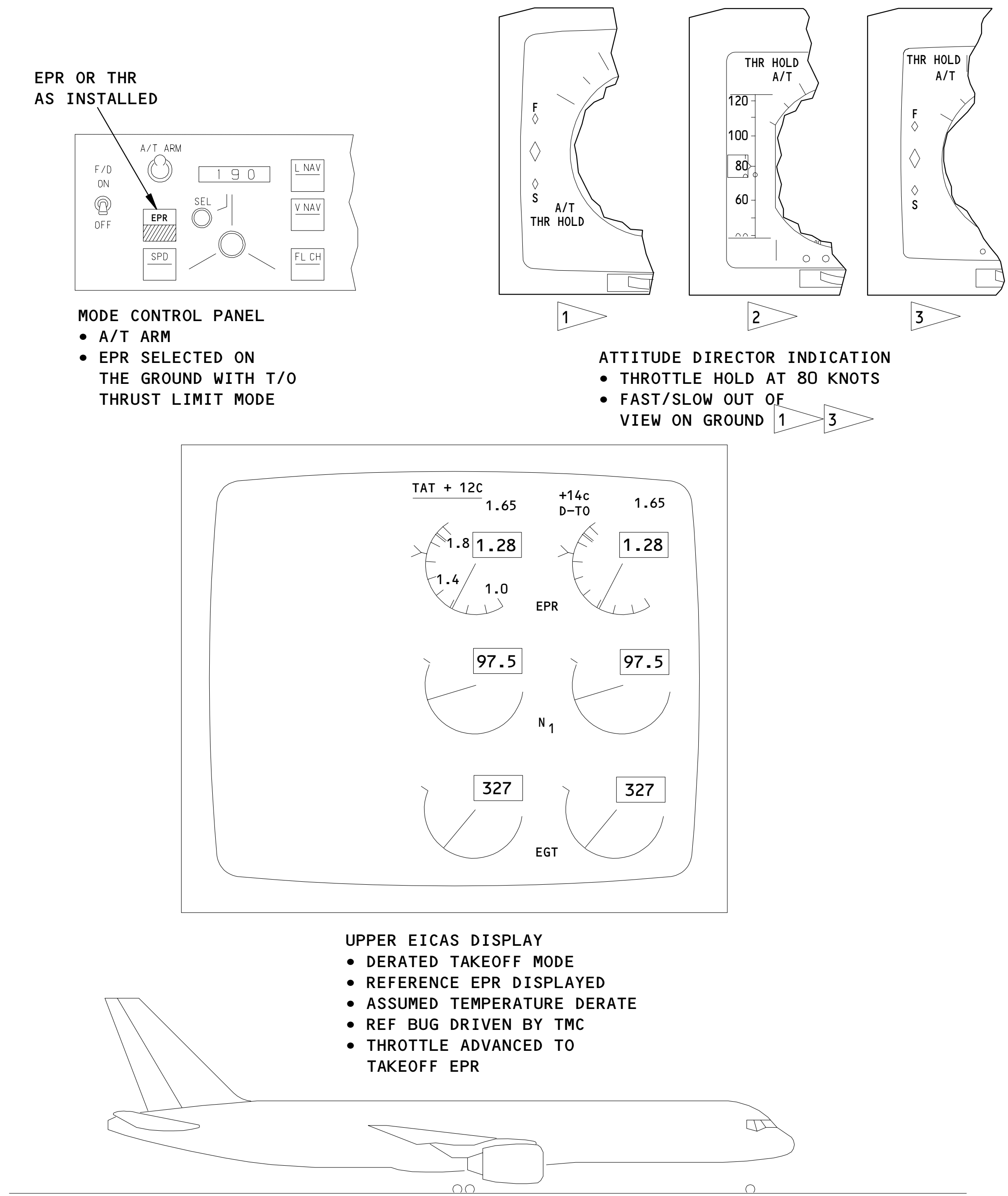
(d) Bleed Correction
1) If bleed air is being extracted, EPR limits are reduced. Discrete inputs select the amount of correction needed. No correction for cowl anti-ice is needed if the altitude is less than 8000 feet and the temperature is less than 7.8°C. If cowl anti-icing is operating under these conditions, the EPR correction is latched until an altitude of 14,000 feet or a speed of 0.45 Mach is obtained.

(e) MAX EPR
1) The MAX EPR limit is calculated, stored, and sent to EICAS. The calculated value is displayed as two amber lines on the EPR indicators on the EICAS upper display unit.

(f) REF EPR
1) The REF EPR limit is calculated and stored on the computer pass opposite of the MAX EPR pass. When the T/O mode is selected, the MAX select function chooses the highest value for REF EPR between REF calculated and MAX calculated minus 0.1. Thrust limit calculations are displayed on the EICAS display unit, and the EPR indicator "bug" is driven to the REF EPR. If the reverse mode is engaged, the REF EPR goes to No Computed Data (NCD) and the EICAS display is blank. During VNAV operation, the EPR indicator bug is driven to the FMC target EPR.

(11) Autothrottle - Takeoff Mode (Fig. 12)
(a) EPR TAKEOFF mode (T/O) is selectable on the TMSP only when the airplane is on the ground and the A/T ARM switch, located on the MCP, has been armed. Then, when the EPR (or THR) mode on the MCP is engaged on the ground, the thrust levers are driven until the (T/O) EPR REF is obtained.
(b) The T/O EPR REF, computed by the thrust limit function of the TMC, may be derated by assumed-temperature derating. The assumed temperature is inserted into the TMC either from the TMSP or from the FMC when VNAV is engaged. Assumed temperature derating of a takeoff thrust limit is achieved by inserting a temperature that is higher than both ambient and flat rated temperature.
(c) Following engagement in EPR (or THR) T/O mode and reaching 80 kts CAS, the A/T servo is deactivated and THR HOLD is displayed on the EADI. THR HOLD prevents the thrust levers from retarding after T/O EPR REF has been obtained.
(d) Annunciation of the EPR T/O mode engagement includes:
1) EPR (or THR) switch on MCP is illuminated.
2) EPR is displayed on EADI.

(12) Autothrottle - EPR Mode (Fig. 13)
(a) The A/T can engage in an EPR (or THR) mode of CLB, CON, or CRZ during flight. Manual selection of CLB, CON, or CRZ EPR thrust limit is made by pressing the respective switch on the TMSP. Manual selection is displayed on EICAS. If the A/T is engaged in SPD or FLCH, the EPR (or THR) mode must be engaged through the MCP following selection of the EPR thrust limit on the TMSP.

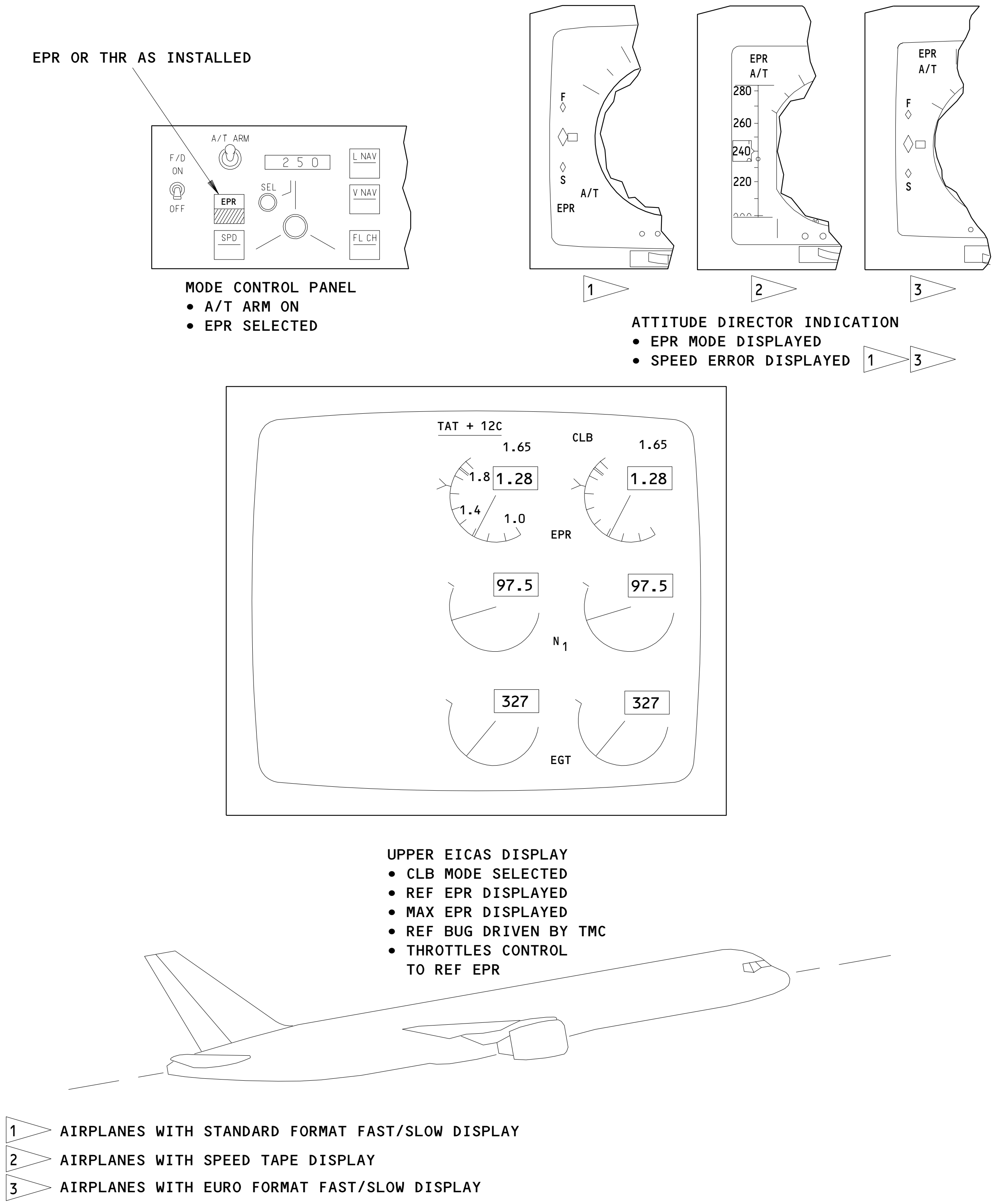


1 AIRPLANES WITH STANDARD FORMAT FAST/SLOW DISPLAY

2 AIRPLANES WITH SPEED TAPE DISPLAY

3 AIRPLANES WITH EURO FORMAT FAST/SLOW TAPE DISPLAY

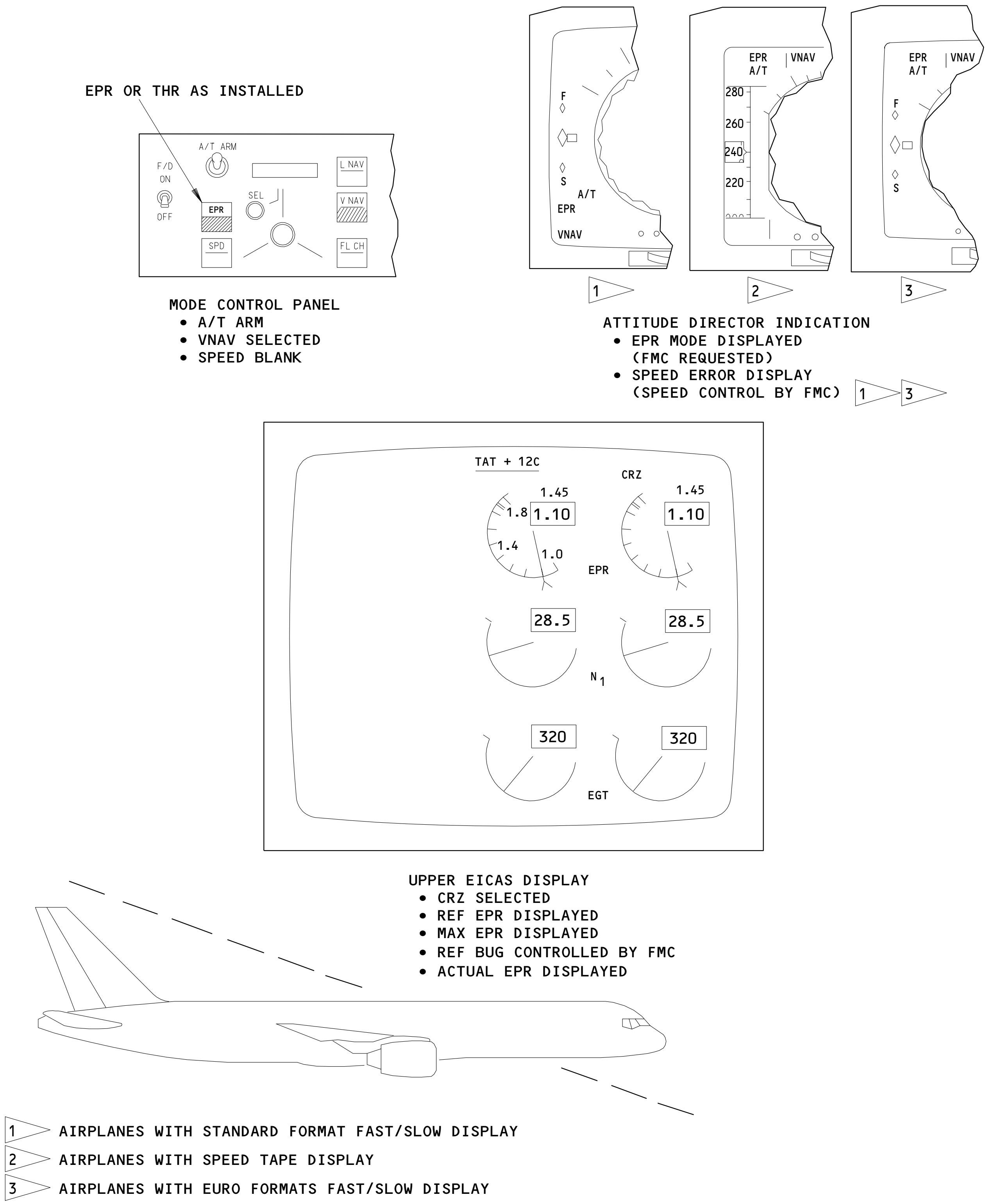
Autothrottle - Takeoff Mode
Figure 12



Autothrottle - EPR Mode
Figure 13

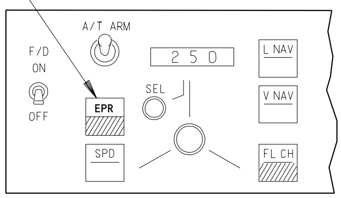
(13) Autothrottle - VNAV Mode (Fig. 14)
(a) The VNAV mode engages when the VNAV switch is pressed and the FCC accepts the VNAV request. The MCP VNAV switch/light dot-bar matrix then illuminates.
(b) Engaging the VNAV mode transfers control of pitch and thrust to the FMC. The TMC receives the VNAV mode engage status from the FCC. When VNAV is engaged, The TMC accepts mode requests along with target commands (EPR, Mach, or airspeed) from the FMC. The FMC request modes are EPR (or THR), SPD, Mach, idle, or dormant. The EPR (or THR) and SPD switch/lights do not illuminate during VNAV, and the IAS/Mach display on the MCP is blank. The EICAS display unit displays the TMC computed EPR REF and MAX EPR. The EPR indicator bug on the EICAS display is driven by the TMC to the FMC target EPR. During VNAV operation, the thrust levers are automatically driven to the FMC generated target EPR. The active autothrottle mode status is displayed on either the upper or lower left corner of the EADI.

(14) Autothrottle - Flight Level Change Mode (Fig. 15)
(a) The Flight Level Change (FLCH) mode is engaged when the A/T switch is in the ARM position and the MCP accepts the FLCH mode request (in air only).
(b) For a climb transition, throttle command to achieve the EPR REF is given to the A/T servomotor generator. For a descent transition, the aft PLA or MIN EPR, based on fuel flow or anti-ice requirements, controls the minimum throttle position. During descent in icing conditions, the approach idle power setting is used as minimum thrust setting.
(c) When the FLCH mode is engaged, annunciation of FLCH appears on the EADI. The reference bug on the EICAS EPR indicator is driven by the TMC to the EPR REF value. The fast/slow display on the EADI provides speed performance information as related to the MCP selected speed.
(d) If the selected altitude on the MCP is changed 500 feet or more, a new transition is commanded. Manual thrust lever movement by the pilot places the autothrottle servo in a hold state until a new mode is selected or the selected altitude is reached.

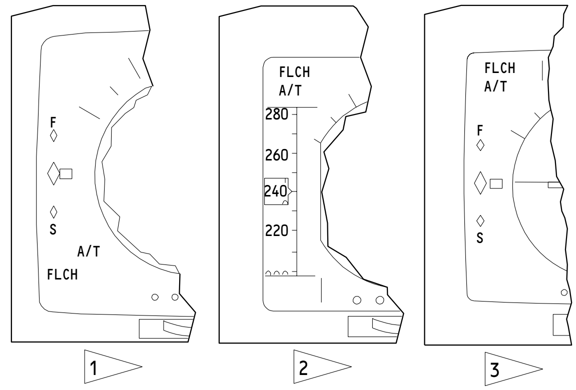


Authrottle - VNAV Mode
Figure 14

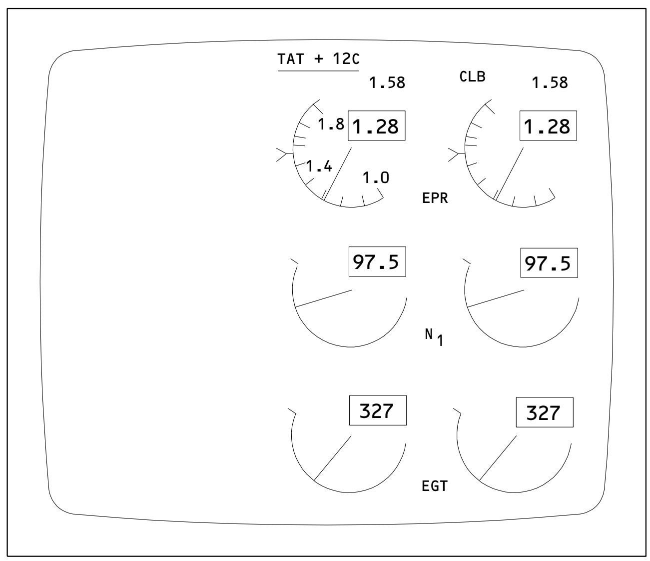
EPR OR THR AS INSTALLED



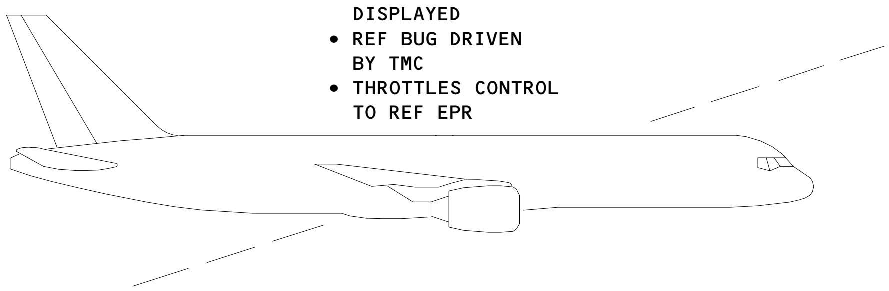
MODE CONTROL PANEL
- A/T ARM
- FL CH SELECTED
- EPR REF IS TARGET FOR CLIMB
- THROTTLES DRIVEN TO AFT STOP FOR DESCENT
- IAS/MACH IS TARGET FOR FCC PITCH CHANNEL



ATTITUDE DIRECTOR INDICATION
- FL CH MODE DISPLAYED
- FAST/SLOW DISPLAYED 1 3



UPPER EICAS DISPLAY
- LIMIT EPR'S DISPLAYED
- REF BUG DRIVEN BY TMC
- THROTTLES CONTROL TO REF EPR



1 AIRPLANES WITH STANDARD FORMAT FAST/SLOW DISPLAY
2 AIRPLANES WITH SPEED TAPE DISPLAY
3 AIRPLANES WITH EURO FORMAT FAST/SLOW DISPLAY

Autothrottle - FL CH Mode
Figure 15

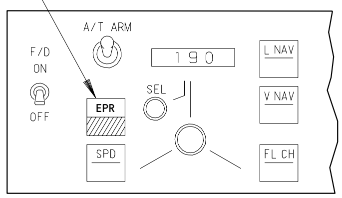
(15) Autothrottle - Go-Around Mode (Fig. 16)
    (a) The Go-Around (G/A) mode is automatically armed when the A/T switch is in the ARM position and; the glideslope is captured (from the FCC) or the flap position is greater than zero. Actual go-around engagement occurs when a go-around switch on the thrust levers is pressed.
    (b) When the G/A mode and an autopilot channel are engaged, the thrust levers advance to achieve a 2000 fpm rate of climb as limited by the EPR REF. When the G/A mode is engaged with no autopilot engaged, the thrust levers advance to achieve the EPR REF. When the go around mode is operational, annunciation of GA appears on the EADI. In addition, the reference bug on the EPR indicator is driven by the TMC to the EPR REF value.

(16) Autothrottle - Flare Retard (Fig. 17)
    (a) Autothrottle flare retard occurs upon request from the FCCs when two or more A/P channels are engaged. The autothrottle must be engaged in the SPD mode. Flare occurs at 50 feet radio altitude.
    (b) When flare retard is requested, the thrust levers retard to the flare idle limit at a controlled rate. For the autoland flare retard mode, the PLA is commanded to 2 degrees below the computed aft limit.

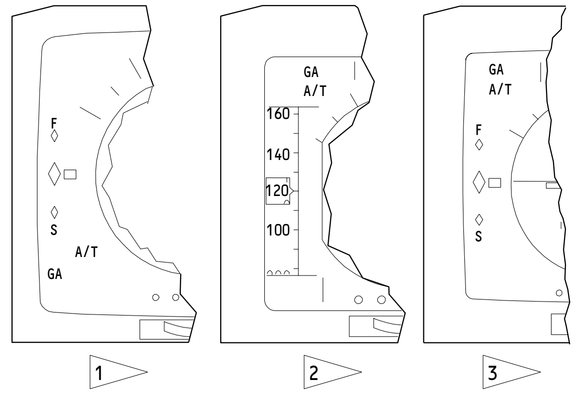
(17) Autothrottle - Idle Mode (Fig. 18)
    (a) Autothrottle idle occurs when requested by the flight management computer. The autothrottle must be armed and engaged in the VNAV mode to accept this request.
    (b) When the idle mode is engaged, the thrust levers retard to ground or flight idle limits at a controlled rate. While the thrust levers are moving to the idle positon, IDLE is annunicated on the EADI. When the thrust levers reach the idle position, throttle hold is engaged. This removes autothrottle servomotor generator excitation, and THR-HLD is annunicated on the EADI.

(18) Thrust Management System - EPR, FLCH Mode Signals and Servo Drive
    (a) Thrust lever position is determined in three steps:
        1) The TMC generates a desired throttle rate command signal in order to reach the EPR target.

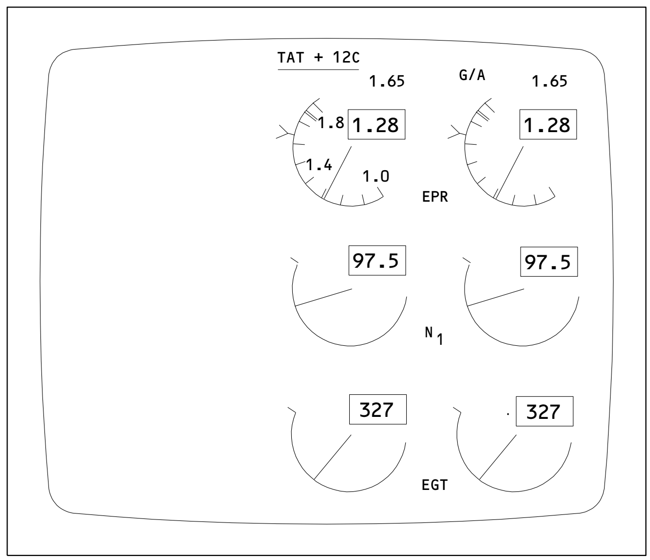
EPR OR THR AS INSTALLED



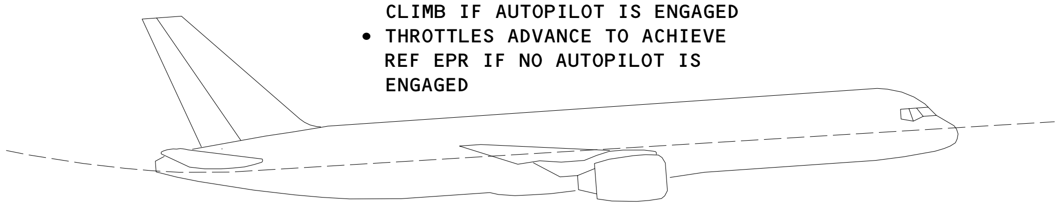
MODE CONTROL PANEL
- A/T ARM
- GA MANUALLY SELECTED BY GA SWITCH ON THRUST LEVERS



ATTITUDE DIRECTOR INDICATION
- GA ANNUNCIATED



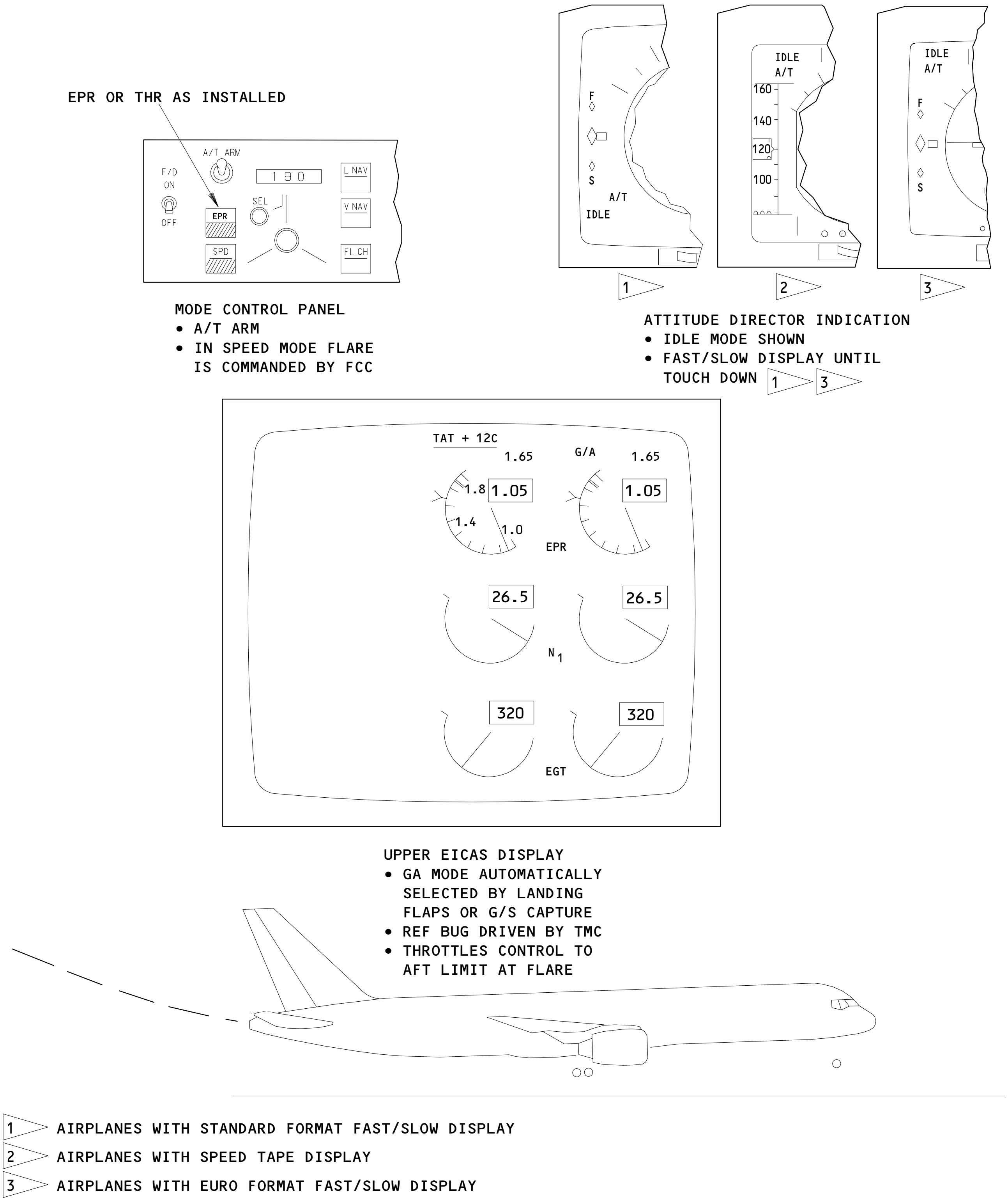
UPPER EICAS DISPLAY
- GA MODE ARMED MANUALLY (TMSP) OR AUTOMATICALLY ARMED BY FLAPS >"0" OR G/S CAPTURE
- EPR BUG DRIVEN TO REF EPR
- THROTTLES ADVANCED TO ACHIEVE 2000 FPM RATE OF CLIMB IF AUTOPILOT IS ENGAGED
- THROTTLES ADVANCE TO ACHIEVE REF EPR IF NO AUTOPILOT IS ENGAGED



1 AIRPLANES WITH STANDARD FORMAT FAST/SLOW DISPLAY
2 AIRPLANES WITH SPEED TAPE DISPLAY
3 AIRPLANES WITH EURO FORMAT FAST/SLOW DISPLAY

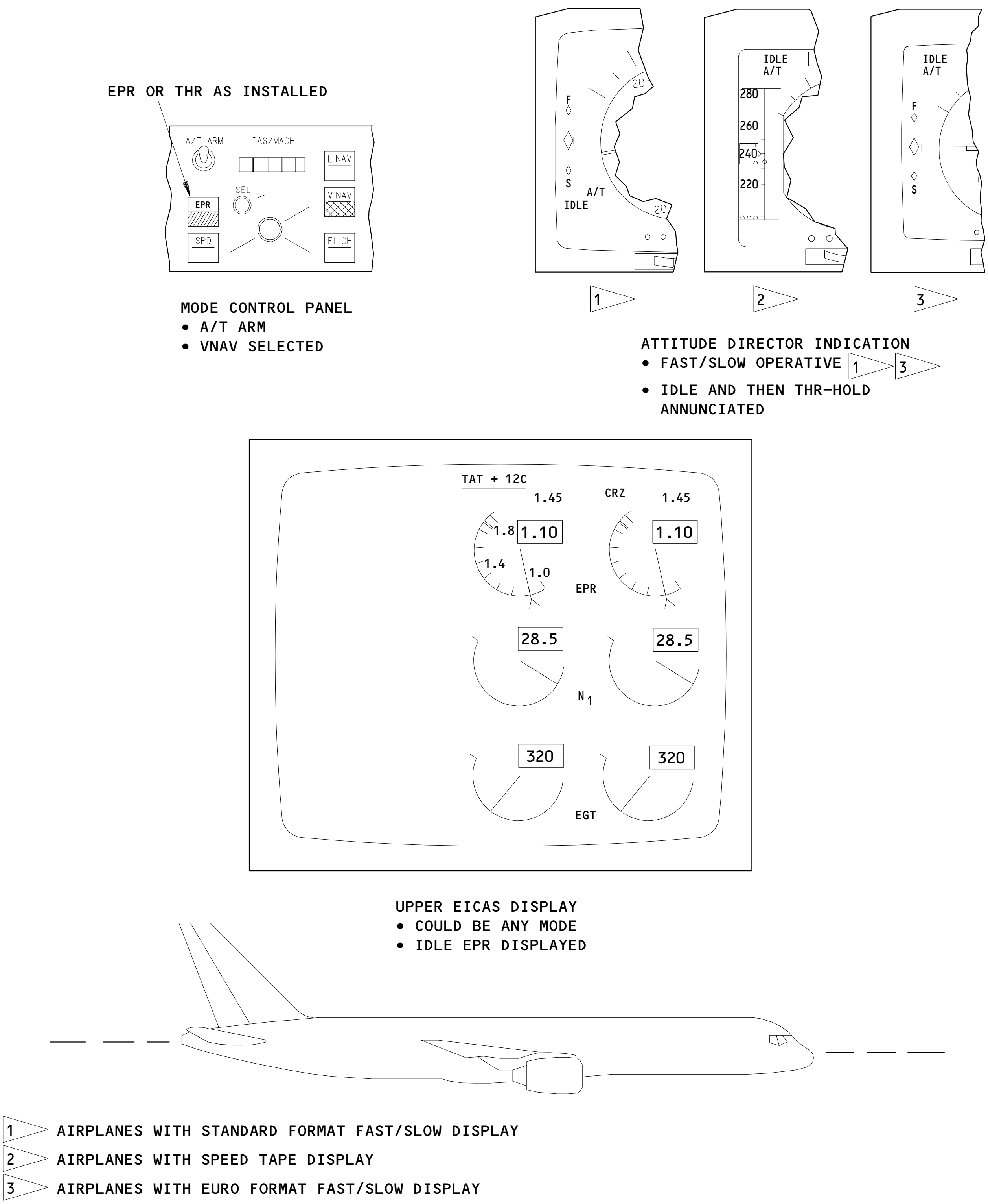
Autothrottle - Go-Around Mode
Figure 16

M21521



M21558

Autothrottle - Flare Retard
Figure 17



Autothrottle - Idle Mode
Figure 18

2) The TMC internal automatic limits override the rate command signal to prevent exceeding EPR or PLA limits.
3) The rate command signal drives the A/T servomotor generator at a desired rate. The signal is limited by a scheduled rate limit which does not allow the command to exceed the performance of the engine.

(b) EPR Control in TO or VNAV Mode
1) An ENABLE signal (SERVO CONTROL FROM THRUST) from the mode select logic allows selection of EPR error when the TO or VNAV mode is engaged. In the TO mode, the difference between the engine actual EPR and the EPR reference is used for the EPR error. When the VNAV mode is engaged and a valid SELECT FMC EPR TARGET CMD signal is received, actual EPR and FMC target EPR determine EPR error.

(c) EPR Control when not in TO or VNAV Mode
1) When in an EPR (or THR) mode other than TO or VNAV, the SPEED CONTROL THROTTLE RATE CMD signal provides initial rate. The rate is computed to increase airplane speed to the upper limit. With the flaps retracted, maximum operating airspeed (VMO) or maximum operating Mach (MMO) is taken as the limit. With the flaps extended, flap placard speed is taken as the limit. The airplane speed is at the upper speed limit unless actual EPR reaches the EPR REF limit. In this case, automatic override occurs and the airplane speed is limited to EPR REF.

(d) Minimum and Maximum EPR Protection and Power Lever Angle limits.
1) The approach idle power provides the minimum thrust setting needed to maintain the required air extraction for anti-icing. Approach idle power control is active only when anti-ice is ON and TAT is less than 10 degrees celsius.
2) The maximum EPR override logic limits the maximum EPR to the REF EPR value. The MAX EPR rate command used to control actual EPR to the EPR REF is compared to the throttle rate COMMAND. Override occurs if the throttle rate COMMAND is greater (more positive) than the MAX EPR rate command.
3) The aft PLA limit restricts aft travel of the power lever angle (PLA). The PLA aft limits are according to:
   a) The flight idle limit with flaps retracted
   b) The ground idle limit with flaps extended
   c) The flare limit when the flare mode is engaged
4) If the throttle rate COMMAND is greater (more negative) than the aft PLA limit rate command, control switches to the aft PLA limit.

5) The MAX PLA limit restricts forward travel of the power lever angle. The PLA maximum limit is the takeoff limit below 14,000 ft., and Mach and climb limits above 14,000 ft. If the throttle rate COMMAND is greater (more positive) than the MAX PLA rate command, control switches to the MAX PLA limit.

(e) Throttle Servo Rate Command Filter

1) The throttle rate COMMAND is compared to the servo tach feedback. The difference is filtered to form the THROTTLE SERVO RATE COMMAND.

(f) Power Amplifier

1) The THROTTLE SERVO RATE CMD is converted to an analog signal and amplified to drive the variable phase motor in the A/T servomotor generator. The rate command is compared to the tachometer feedback in order to reach the desired rate.

(g) Flight Level Change (FLCH) Control

1) The FLCH control law drives the thrust levers smoothly to the limit (maximum or idle), where they stay until the altitude is captured. The forward thrust limit is the reference thrust target. The aft limit is the throttle aft mechanical stop.

2) The actual PLA rate is compared to the command rate. If a difference is detected, pilot override is assumed to have occurred and FLCH hold is engaged. This removes the rate command and the excitation from the A/T servomotor generator.

(h) Power Lever Angle Retard Control

1) The PLA aft limits differ depending on the flight conditions:

a) Flare retard - 48 degrees (flare idle)

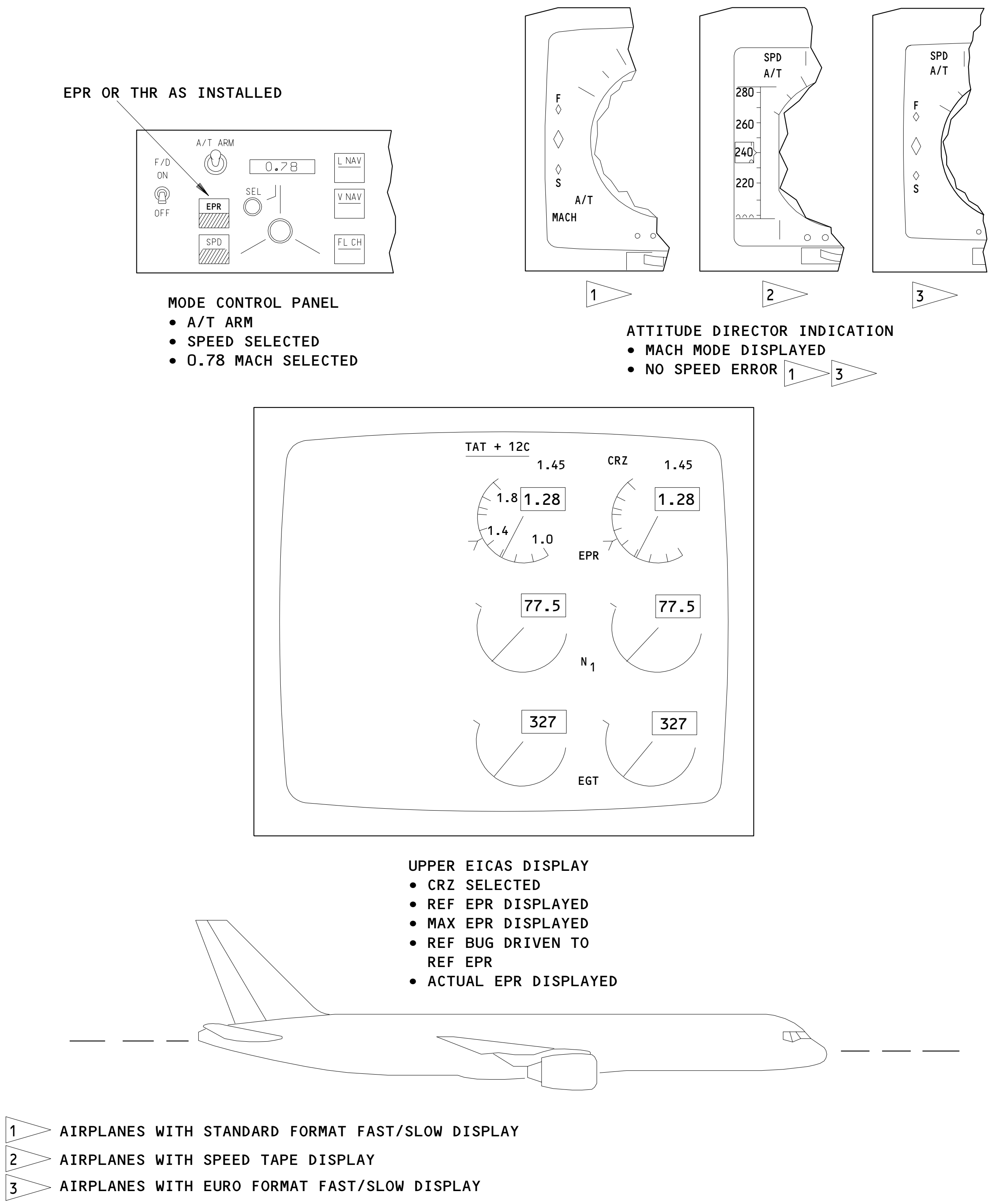
b) Landing flaps - 57.4 degrees (flight idle)

c) Ground idle - 53 degrees

2) The enable servo control signal from the retard logic switches control to the aft PLA limit. Upon reaching the aft limit, the autothrottle servo is placed in hold until a throttle advance is commanded.

(19) Autothrottle Speed Mode (Fig. 19)

(a) The speed mode is engaged when the A/T arm switch is in the ARM position and the MCP SPD request is valid. Automatic speed mode engage occurs when the VNAV or FLCH mode is exited.



Autothrottle - Speed Mode
Figure 19

(b) Depressing the SPD pushbutton sends the mode change request to the TMS and AFDS. If the autothrottle is armed and there are no faults affecting operation in the speed mode, the autothrottle engages and controls airspeed (or Mach) to the reference value selected in the IAS/MACH display on the MCP.

(c) The MCP IAS/MACH display is controlled by the SEL button and the unlabeled speed select knob located below the display window. The SEL button permits IAS or MACH reference speed to be displayed. Selection of the IAS or MACH reference speed is made by turning the unlabeled speed select knob. Pushing the center of the speed select knob transfers control of the speed reference from the MCP to the FMC. When a TMS control mode of SPD, FLCH or EPR (TO or GA) is engaged, pushing the speed select knob will not cause any speed reference change. When in VNAV mode, the IAS/MACH display will be blank when the FMC is in control of the reference and driving the airspeed bugs on the MACH/airspeed indicators. When the speed window is blank, pushing the speed select knob causes the current airspeed to be displayed in the window. If the airspeed reference is invalid, 200 knots is displayed.

(d) The ranges of the speed select knob are:
1) 0.4 to 0.95 for Mach
2) 100 to 399 knots for airspeed

(e) The SEL pushbutton allows selection of Mach or airspeed for display, provided it is within the allowable range.

(20) Thrust Management System - Speed Mode Signal

(a) Speed Mode Functions

1) The A/T COMMAND (SPEED CONTROL THR/RATE CMD) signal commands a throttle rate to capture and maintain the selected speed on the MCP. This command is automatically limited between minimum and maximum speed limits. The TMC fast/slow control law drives the fast/slow indicator on the EADI to indicate speed error of ± 10 knots full scale. When the minimum speed limit is reached, the fast/slow display is automatically referenced to and controlled by the minspeed limit function. When the EPR mode is engaged, the thrust levers advance toward the EPR REF but are limited to the flap limit speed or maximum speed (VMO). Normally the EPR REF limit overrides before the maximum speed is obtained.

(b) A/T COMMAND
1) The A/T COMMAND controls the thrust levers to maintain a target speed. The TMS uses true airspeed (TAS) in controlling to the CAS reference target speed. TAS is computed from the CAS output from the ADC. The commanded true airspeed is limited by the maximum speed limits to prevent the speed from violating MMO, VMO, and flap placard limits.

(c) Minimum Speed
1) The minimum speed limit automatically becomes active when the A/T command permits the speed to go below the critical minimum speed (30% above stall speed). A reference AOA is computed from Mach and airplane configuration data. A load factor is added to compensate for the change in AOA due to body loads.
2) The TMC causes an ALPHA speed message when the airplanes speed is at or below Vref. The TMC calculates Vref as a function of AOA and then compares the calculated Vref with actual speed. This eliminates nuisance ALPHA speed messages when the airplane AOA is high.

(d) GUI 009, 115;
Fast/Slow Display Command
1) The fast/slow control law drives the fast/slow speed indicator on the EADI. If the A/T is engaged or the FCC pitch channel is controlling speed, the true airspeed is compared to the limited selected speed to determine raw error. The turbulence cancelation factor is summed with this error to form the command that drives the display.
2) If no automatic mode is engaged, the fast/slow command is the greater of the proportional signals from the speed control or minimum speed control functions. The error between filtered true airspeed and process selected speed, along with the turbulence cancelation factor, combine to form the command signal when the speed control law is in control (proportional path).
3) The F/S MIN SPD ENGAGE logic automatically transfers the display drive signal to the minimum speed control law if the magnitude of slow command from the minimum speed control is greater than the magnitude of slow command from speed control.

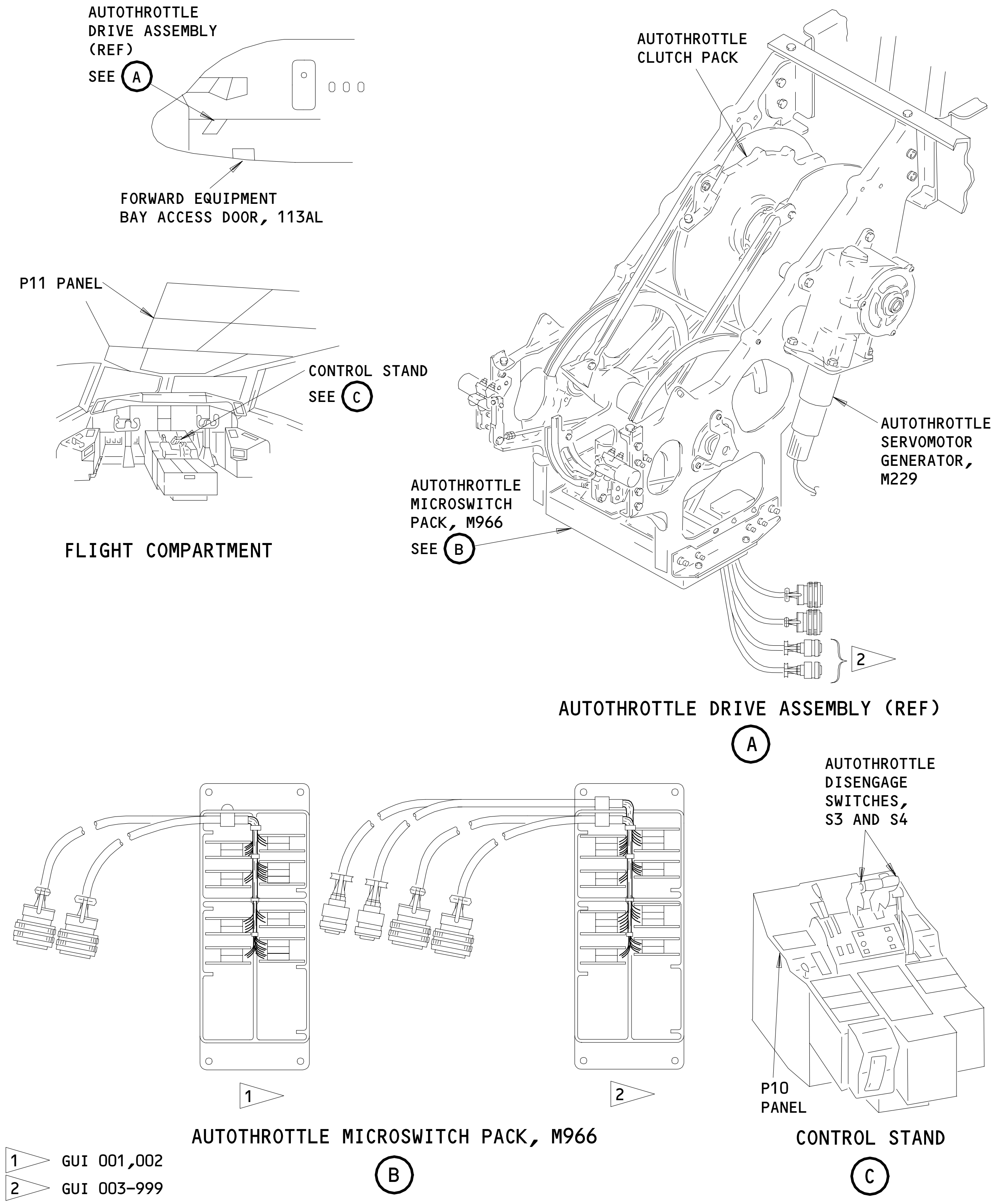
(e) Initial Target for EPR (or THR) Mode
1) When the A/T is in the EPR (or THR) mode and T/O is disengaged, the thrust levers track the new reference EPR. Flap placard and maximum airspeed and Mach protection is provided.

THRUST MANAGEMENT SYSTEM

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
CARD - (FIM 36-10-00/101)				
L ECS BLEED, M10313				
R ECS BLEED, M10312				
CIRCUIT BREAKER -	--		FLT COMPT, P11	
AUTOFLIGHT WARN, C521		1	11A17	*
FIRE DET ALTN PWR ENG LEFT, C763		1	11J26	*
FLAP SLAT ELEC UNIT 2 POWER, C1521		1	11C14	*
RUDDER TRIM POS, C1034		1	11J17	*
TMC AC, C501		1	11F14 OR 11F16	*
TMC DC, C525		1	11F15 OR 11F17	*
TMC SERVO, C512		1	11F16 OR 11F18	*
COMPUTER - (FIM 22-31-00/101)				
THRUST MANAGEMENT, M183				
COMPUTER - (FIM 31-41-00/101)				
EICAS L, M10181				
EICAS R, M10182				
COMPUTER - (FIM 34-12-00/101)				
AIR DATA L, M100				
AIR DATA R, M101				
COMPUTER - (FIM 34-61-00/101)				
FLIGHT MANAGEMENT L, M134				
FLIGHT MANAGEMENT R, M135				
CONTROL - (FIM 73-21-00/101)				
ELECTRONIC ENGINE L, M10391				
ELECTRONIC ENGINE R, M10392				
GENERATOR - AUTOTHROTTLE SERVOMOTOR, M229	--	1	113AL, FORWARD EQUIP BAY	22-32-01
PACK - AUTOTHROTTLE CLUTCH	--	1	113AL, FORWARD EQUIP BAY	22-32-05
PACK - AUTOTHROTTLE MICROSWITCH, M966	--	1	113AL, FORWARD EQUIP BAY	22-32-04
PANEL - (FIM 22-11-00/101)				
AFCS MODE CONTROL, M90				
PANEL - (FIM 22-41-00/101)				
MAINTENANCE CONTROL DISPLAY, M168				
RELAY - (FIM 31-01-36/101)				
SYSTEM 1 AIR/GROUND, K143				
SWITCH - L THRUST REVERSE, S12	--	1	113AL, FORWARD EQUIP BAY	22-32-04
SWITCH - R THRUST REVERSE, S16	--	1	113AL, FORWARD EQUIP BAY	22-32-04
SWITCH - SYS L AUTOTHROTTLE DISENGAGE, S3	--	1	FLT COMPT, P10	22-32-02
SWITCH - SYS R AUTOTHROTTLE DISENGAGE, S4	--	1	FLT COMPT, P10	22-32-02
SWITCH - (FIM 22-11-00/101)				
GO AROUND, S7, S8, OF M985				
TRANSDUCER - (FIM 22-33-00/101)				
L POWER LEVEL ANGLE, TS5029				
R POWER LEVEL ANGLE, TS5029				
UNIT - (FIM 27-51-00/101)				
FLAP/SLAT ELECTRONICS, M10331				
UNIT - (FIM 34-21-00/101)				
L INERTIAL REFERENCE, M159				
R INERTIAL REFERENCE, M161				

* SEE THE WDM EQUIPMENT LIST

Thrust Management System - Component Index
Figure 101



Thrust Management System - Component Location
Figure 102

THRUST MANAGEMENT SYSTEM - ADJUSTMENT/TEST

1. General
   A. This part of the Maintenance Manual contains a system test of the Thrust Management System (TMS). The test uses the Maintenance Control Display Panel (MCDP). Each test part may be completed independently if all the conditions in the "prepare for test" step have been completed.
   B. The test examines the Thrust Management System (TMS) to make sure of the best system performance. This test uses the not fully automatic maintenance functions in the TMS. These functions are controlled by the Maintenance Control and Display Panel (MCDP).
   C. The MCDP is on shelf E3-2 of the main equipment center. It supplies switches for test control, test direction and it shows any failure. The MCDP front panel shows general operation instructions. A connection, from the MCDP, is in the control cabin for a hand-held MCDP remote control unit. The connector for the remote unit is on the forward side of the P61 panel. Use of the remote unit decreases test time. The remote unit lets messages displayed on the MCDP in the main equipment center to be shown on the bottom EICAS display unit.

TASK 22-32-00-735-002

2. System Test - Thrust Management System
   A. Equipment
      (1) Remote Control Unit, Maintenance Control Display Panel -
          A22001-22 (preferred),
          A22001-15 (optional).
   B. References
      (1) 22-00-02/201, Autoflight BITE
      (2) 24-22-00/201, Electrical Power - Control
      (3) 27-61-00/201, Spoiler/Speedbrake Control System
      (4) 31-41-00/201, EICAS
      (5) 34-12-00/501, Air Data Computing System
      (6) 34-21-00/501, Inertial Reference System
      (7) 34-22-00/501, Flight Instrument System
      (8) 34-61-00/501, Flight Management Computer System
   C. Access
      (1) Location Zone
          211/212 Flight Compartment

   D. Prepare to Test

      S 865-003

      (1) Supply electrical power (AMM 24-22-00/201).

S 485-004
(2) Connect the MCDP remote control unit to connector D41101 on the P61 panel.

S 865-006
(3) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:
(a) 11D21, ENGINES HP BLD VLV L
(b) 11D22, ENGINES HP BLD VLV R
(c) 11E16, MODE CONT PNL LEFT
(d) 11F14 or 11F16, TMC AC
(e) 11F15 or 11F17, TMC DC
(f) 11F16 or 11F18, TMC SERVO
(g) 11G12, FLAP SLAT ELEC UNIT 1 POWER
(h) 11G13, FLAP SLAT ELEC UNIT 1 CONT
(i) 11G14, FLAP SLAT ELEC UNIT 1 SENSOR
(j) 11L4, LEFT ENGINE EEC LIMITER
(k) 11L5, LEFT ENGINE EEC SUPV
(l) 11L9, LEFT ENGINE EPR XMTR
(m) 11L31, RIGHT ENGINE EEC LIMITER
(n) 11L32, RIGHT ENGINE EEC SUPV
(o) 11L36, RIGHT ENGINE EPR XMTR
(p) 11Q10, ENG BLEED L
(q) 11Q19, R ENG BLEED CONT
(r) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(s) GUI 115;
11S6, MAINT CONT DSPL
(t) 11S15, AIR/GND-SYS 1
(u) 11S19, AIR/GND-SYS 2

S 865-070
(4) Make sure the systems that follow are on:
(a) Maintenance Control Display Panel (AMM 22-00-02/201).
(b) Engine Indication and Crew Alerting System (AMM 31-41-00/201).
(c) Air Data Computing System (AMM 34-12-00/501).
(d) Inertial Reference System (AMM 34-21-00/501).
(e) Flight Instrument System (AMM 34-22-00/501).
(f) Flight Management Computer System (AMM 34-61-00/501).

E. Test the Thrust Management System

S 865-091
(1) Set the ELEC ENG CONTROL switches on the P5 panel to on. This energizes the EEC power.

S 865-008
(2) Set the ENG LIMITER switches on the P5 panel to on.

S 865-011
(3) Set the EEC ground test switches below the EEC computers in the main equipment center to on.

S 865-151

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(4) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 755-015
(5) Make sure the engines are not on.

S 865-016
(6) Set the MCDP to test 02 TMC.

S 865-114
(7) Push the YES/ADV switch.

NOTE: The MCDP message IN PROGRESS is shown when an automatic test step is being done. An automatic move forward to the subsequent test step or a fail message is provided upon completion of the step.

S 755-018
(8) Make sure the MCDP message 02 VFY THRST LVR SYST CLR-ADV is shown.

S 755-019
(9) Make sure the throttle levers are clear.

S 865-020
(10) Push the YES/ADV switch.

S 755-021
(11) Make sure the MCDP message 02 SET MCP A/T SW TO ARM is shown.

S 865-022
(12) Set the A/T ARM switch on the MCP to the ARM position.

S 755-023
(13) Make sure the throttle levers move a small distance after the A/T ARM switch is set to ARM.

S 755-024
(14) Make sure that no MCDP fault messages are shown.

S 755-025
(15) Make sure the MCDP message 02 TEST COMPLETE is shown.

S 865-026
(16) Push the YES/ADV switch.

S 865-027
(17) Set the EEC ground test switches to off.

F. Test the Thrust Mode Select Panel (TMSP)

NOTE: This test makes sure the TMSP and its interface to the TMC operates correctly. The test includes operation of the TMSP mode on and off switches and operation of the TEMP SEL rotary switch.

S 865-038
(1) Set the MCDP to test 05 TMSP.

S 865-039
(2) Push the YES/ADV switch.

S 755-040
(3) Make sure the MCDP message 05 TMSP TEST? is shown.

S 865-041
(4) Push the YES/ADV switch.

S 755-042
(5) Make sure the MCDP message 05 MAN TEST? is shown.

S 865-043
(6) Push the YES/ADV switch.

S 755-044
(7) Follow the steps shown on the MCDP for the top EICAS TMS mode display.
    (a) When the MCDP message 05 PUSH OFF SW PER MODE DSPLY is shown, push the applicable MODE (the MODE is shown on top EICAS display) switch on the TMSP.

S 755-045
(8) Make sure the MCDP message TMSP 05 TURN TEMP SEL CW-DSPLY INC? is shown.

S 865-046
(9) Turn the TEMP SEL knob on the TMSP clockwise.

S 755-047
(10) Make sure the EICAS temperature select display on the top EICAS display increases.

S 865-048
(11) Push the YES/ADV switch.

S 755-049
(12) Make sure the MCDP message 05 TURN TEMP SEL CCW-DSPLY DEC? is shown.

S 865-050
(13) Turn the TEMP SEL knob counterclockwise.

S 755-051
(14) Make sure the EICAS temperature select display decreases.

S 865-052
(15) Push the YES/ADV switch.

S 755-053
(16) Make sure the MCDP message 05 TEST COMPLETE is shown.

S 755-179
(17) Do the MCDP Ground Test 52 TMC RLY/SW (AMM 22-00-02-5).

G. Put the Airplane Back to Its Initial Condition

S 865-054
(1) Set the MCDP to off.

S 865-092
(2) Set the ELEC ENG CONTROL switches on the P5 panel to off. This removes EEC power.

S 865-056
(3) Set the ENG LIMITER switches on the P5 panel to off.

S 085-060
(4) Disconnect the MCDP remote control unit from connector D41101 on the P61 panel.

S 865-167
(5) Set the Inertial Reference System (AMM 34-21-00/501) to off if it is not necessary.

S 445-151
(6) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 865-063
(7) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-32-00-755-141
3. <u>Test the Autothrottle Clutch Slip Load</u>
A. General
(1) This test does a check of the autothrottle clutch slippage through the full range of travel of the clutch.
B. Equipment
(1) Spring Scale 0-50 pounds, Chatillon Manufacturing Company Model DPP-100 or equivalent
C. Consumable Material
(1) 601505 Safety Wire, 0.020 inch-diameter minimum, stainless or Monel material
D. References
(1) AMM 06-41-00/201, Section 41 Access Doors and Panels
(2) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(3) AMM 78-31-00/201, Thrust Reverser System
E. Access
(1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/ 12	Flight Compartment

(2) Access Panel

113AL	Flight/Landing Gear/Engine Control Compartments

F. Prepare to Test the Autothrottle Clutch Slip

S 845-143

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or more all persons and equipment away from the spoilers.

S 845-144

WARNING: DO THE ACTIVATION PROCEDURE TO PREVENT THE OPERATION OF THE THRUST REVERSER. THE ACCIDENTAL OPERATION OF THE THRUST REVERSER CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the thrust reverser for ground maintenance (AMM 78-31-00/201).

S 015-140
(3) Open the access panel, 113AL, to get access to the autothrottle clutch assembly (AMM 06-41-00).

G. Procedure

S 355-131
(1) Remove the bolts, washers, and nuts to disconnect the clutch links from the clutch.

S 355-132
(2) Remove the bolts, washers, and nuts to disconnect the control rods from the clutch.

S 225-145
(3) Measure the clutch slip load.

CAUTION: DO NOT ATTACH THE SAFETY WIRE TO THE CLUTCH LINK ATTACH BOLT HOLE. THE BEARING CAN BE DAMAGED IF THE BOLT HOLE IS USED AS AN ATTACH POINT.

(a) Attach the safety wire to the top of the clutch near the clutch link attach bolt hole.
(b) Wind the safety wire around the clutch outer diameter in the CCW direction (Fig. 501).
(c) Attach the spring scale to the safety wire.

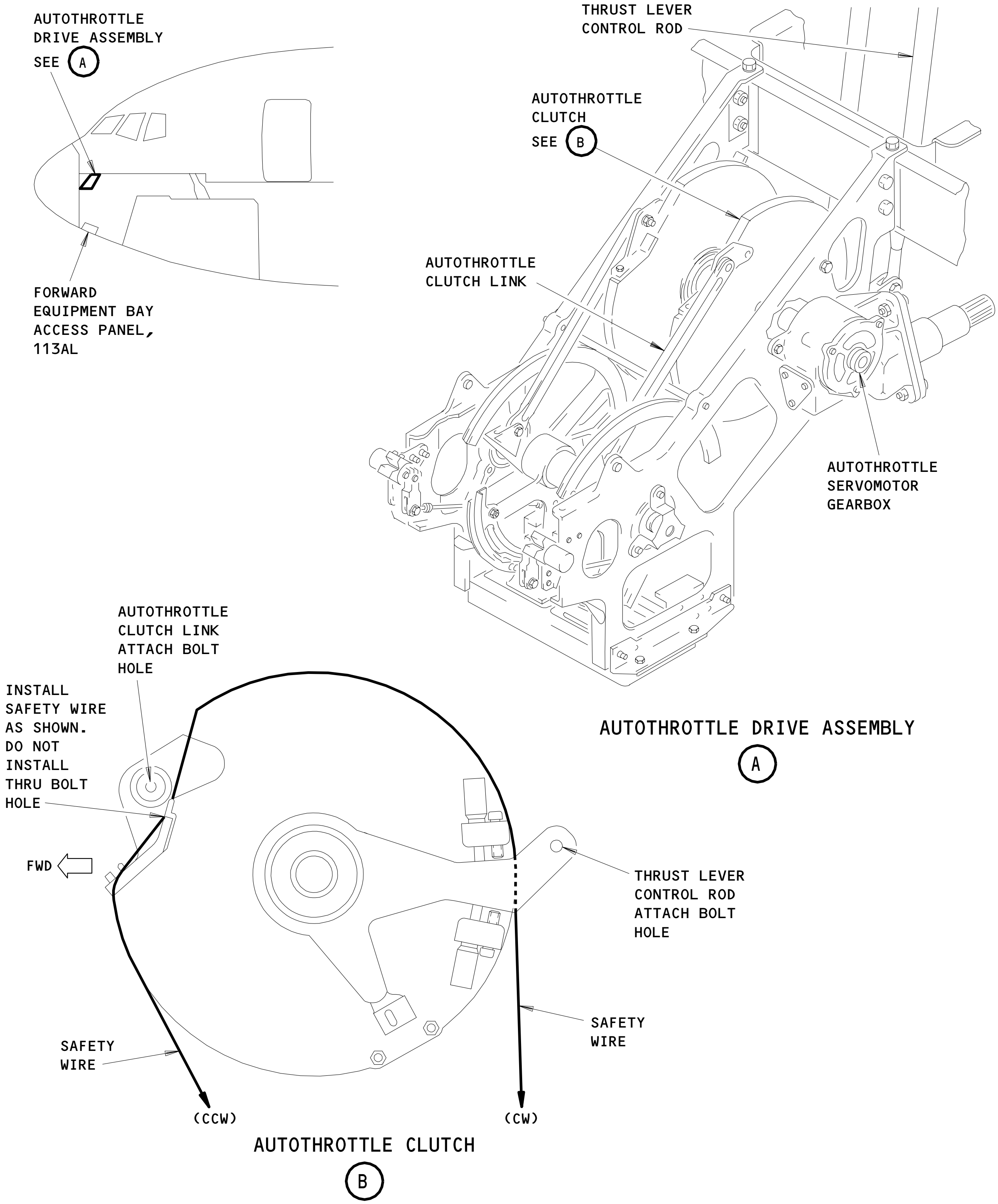
NOTE: Before applying a load, make sure that the safety wire is on the outer diameter of the clutch.

(d) Apply a load to the scale to cause the clutch to slip in the CCW direction. Record the clutch slip load measurement.

NOTE: The clutch slip load should be 18 pounds minimum.

(e) Wind the safety wire around the clutch outer diameter in the CW direction (Fig. 501).
(f) Attach the spring scale to the safety wire.

NOTE: Before applying a load, make sure that the safety wire is on the outer diameter of the clutch.



Autothrottle Clutch Slip Load - Test
Figure 501

(g) Apply a load to the scale to cause the clutch to slip in the CW direction. Record the clutch slip measurement.

NOTE: The clutch slip load should be 18 pounds minimum.

H. Put the Airplane Back to its Initial Condition

S 085-146
(1) Remove the spring scale from the attach point.

S 085-147
(2) Remove the safety wire from the clutch diameter.

S 435-137
(3) Connect the control rods to the clutch with the bolts, washers, and nuts.

S 435-138
(4) Connect the clutch links to the clutch with the bolts, washer, and nuts.

S 445-148
(5) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 445-149
(6) Do the activation procedure for the thrust reverser if you did the deactivation procedure (AMM 78-31-00/201).

AUTOTHROTTLE SERVOMOTOR GENERATOR AND GEARBOX - REMOVAL/INSTALLATION

1. General
   A. The autothrottle servomotor generator is installed on the gearbox of the autothrottle drive assembly. The servo is held in position by three bolts. The unit has one multi-pin electrical connector. Access to the servomotor generator/gearbox assembly is through the flight/landing gear/engine control components access panel 113AL. The servomotor generator and gearbox can be removed and installed as an assembly, to make other maintenance procedures easier.

TASK 22-32-01-004-001

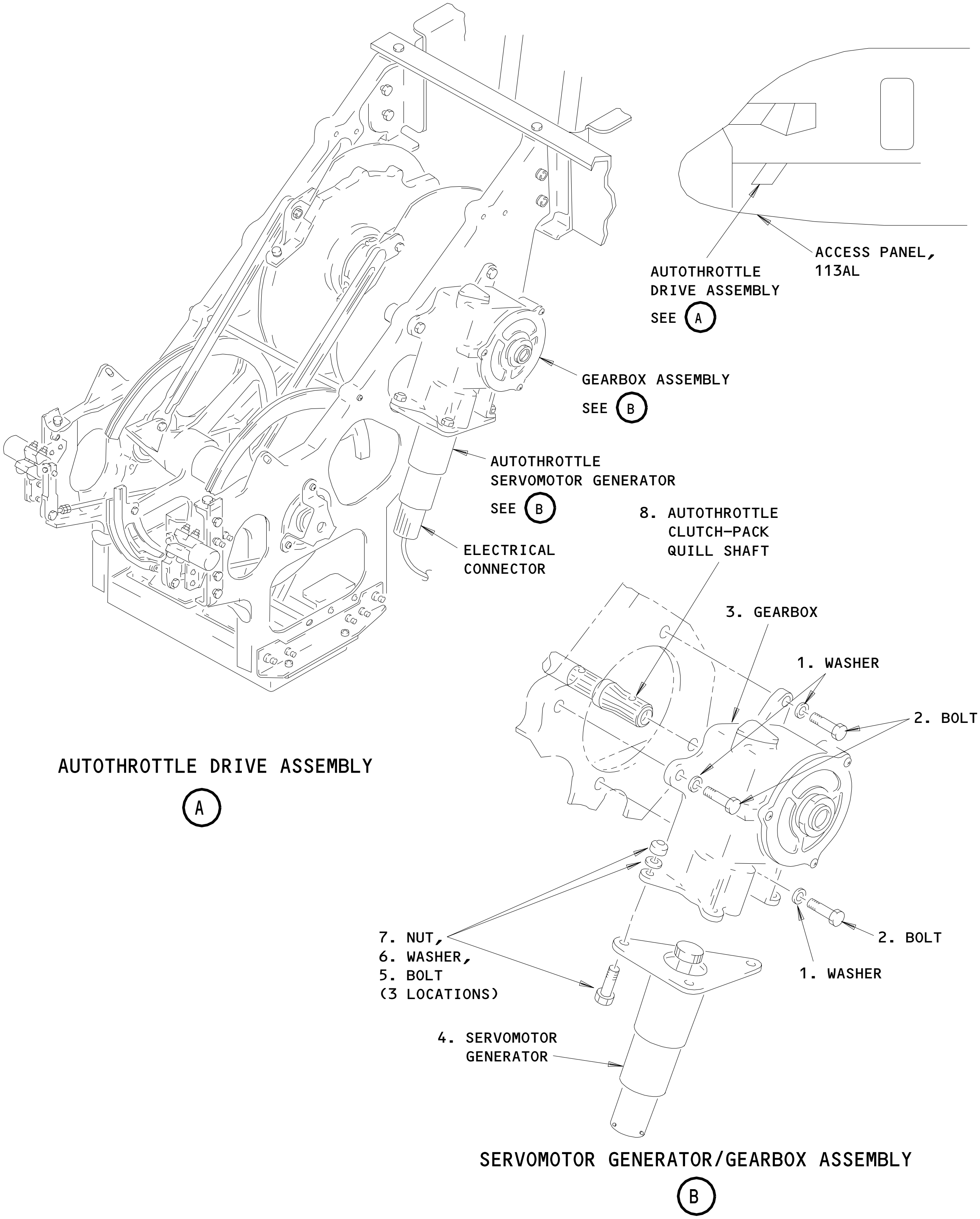
2. Remove the Servomotor Generator and/or the Gearbox (Fig. 401)
   A. Reference
      (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
   B. Access
      (1) Location Zones
          113/114 Area Forward of NLG Wheel Well
          211/212 Flight Compartment

      (2) Access Panel
          113AL Flight/Landing Gear/Engine Control Components

   C. Prepare for Removal

      S 864-002
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11F14 or 11F16, TMC AC
          (b) 11F16 or 11F18, TMC SERVO

      S 864-003
      (2) Attach DO-NOT-OPERATE tags on the left and right throttle levers.



Servomotor Generator/Gearbox - Installation
Figure 401

S 014-004
(3) Open the access panel, 113AL, for the autothrottle servomotor generator and gearbox (AMM 06-41-00/201).

D. Remove the Servomotor Generator and/or the Gearbox

S 034-005
(1) Disconnect the electrical connector.

S 024-006
(2) Remove the servomotor generator/gearbox assembly as follows:
(a) Hold the servomotor generator/gearbox assembly and remove the bolts (2) and washers (1) that attach the gearbox to the autothrottle drive assembly.

S 024-007
(3) Remove the servomotor generator (4) as follows:
(a) Hold the servomotor generator (4) and remove the bolts (5), washers (6), and nuts (7) that attach the generator to the gearbox (3).

TASK 22-32-01-404-008

3. Install the Servomotor Generator (Fig. 401)

A. Consumable Material
(1) D00633 Grease - BMS 3-33 (Preferred)
(2) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)

B. Access
(1) Location Zone
113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Flight/Landing Gear/Engine Control Components

C. Install the Servomotor Generator

S 644-009
(1) Apply a layer of grease on all the mating surfaces.

S 424-010

CAUTION: YOU MUST BE CAREFUL WHEN YOU INSTALL THE SERVOMOTOR GENERATOR AS THE SPLINE ON THE GEARBOX CAN EASILY BE DAMAGED.

(2) Hold the servomotor generator (4) on the gearbox (3) and install the bolts (5), washers (6), and nuts (7).

S 434-011
(3) Connect the electrical connector.

S 714-012
(4) Do a test of the servomotor generator/gearbox assembly.

TASK 22-32-01-404-013
4. Install the Servomotor Generator/Gearbox Assembly (Fig. 401)
A. Parts

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	3 4	Gearbox Servomotor Generator	22-32-01	01	30 25

B. Access
(1) Location Zone
113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Flight/Landing Gear/Engine Control Components

C. Install the Servomotor Generator/Gearbox Assembly

S 424-014
(1) Hold the gearbox assembly (3) and connect the gearbox assembly spline with the autothrottle clutch-pack quill shaft (8).
(a) Install the bolts (2) and washers (1).

S 434-015
(2) Connect the electrical connector.

S 714-016
(3) Do a test of the servomotor generator/gearbox assembly.

TASK 22-32-01-714-017
5. Test the Servomotor Generator/Gearbox Assembly
A. References
(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-61-00/201, Spoiler/Speedbrake Control System
B. Access
(1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/212	Flight Compartment

(2) Access Panel

113AL	Flight/Landing Gear/Engine Control Components

C. Prepare for Test

S 864-018
(1) Supply electrical power (AMM 24-22-00/201).

S 864-019
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11F14 or 11F16, TMC AC
(b) 11F16 or 11F18, TMC SERVO

S 864-020
(3) Remove the DO-NOT-OPERATE tags on the left and right throttle levers.

S 864-021
(4) Set the A/T switch on the AFCS mode control panel (on the P55 panel) to ARM.

S 864-028

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(5) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 864-029

(6) Make sure the engines are not on.

D. Procedure

S 714-023

(1) Do the MCDP Ground Test 10 SERVO A/T (AMM 22-00-02/201).

E. Put the Airplane Back to Its Initial Condition

S 864-024

(1) Set the A/T switch on the AFCS mode control panel to OFF.

S 864-025

(2) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 414-026

(3) Close the access panel, 113AL (AMM 06-41-00/201).

S 864-027

(4) Remove electrical power if it is not necessary (AMM 24-22-00/201).

DISENGAGE SWITCHES - REMOVAL/INSTALLATION

1. General
   A. One autothrottle disengage switch is installed on the outboard side of each thrust lever. The knob at the inboard end of each thrust lever is removed to access each switch. Electrical connections to the switches are screw-attached.

TASK 22-32-02-004-002

2. Remove the Disengage Switch (Fig. 401)
   A. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
   B. Access
      (1) Location Zone
          211/212 Flight Compartment

   C. Remove the Disengage Switch

      S 864-001
      (1) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tag:
         (a) 11F15 or 11F17, TMC DC

      S 014-003
      (2) Remove the thrust lever knob.

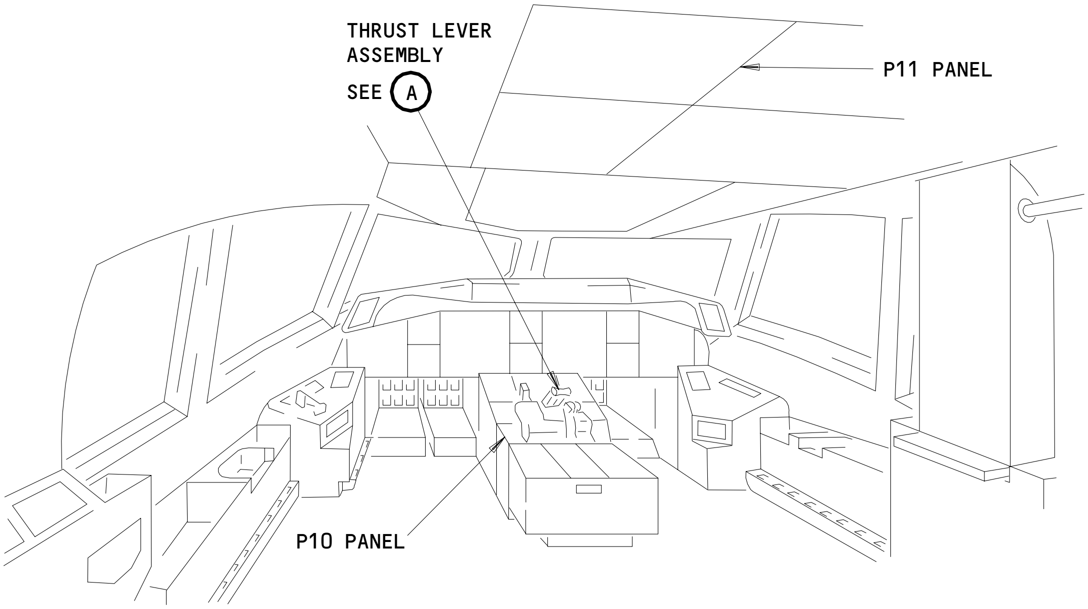
      S 034-004
      (3) Remove the heat-shrink tubing from the wire terminals.

      S 034-005
      (4) Remove the screw-attached wires from the disengage switch.

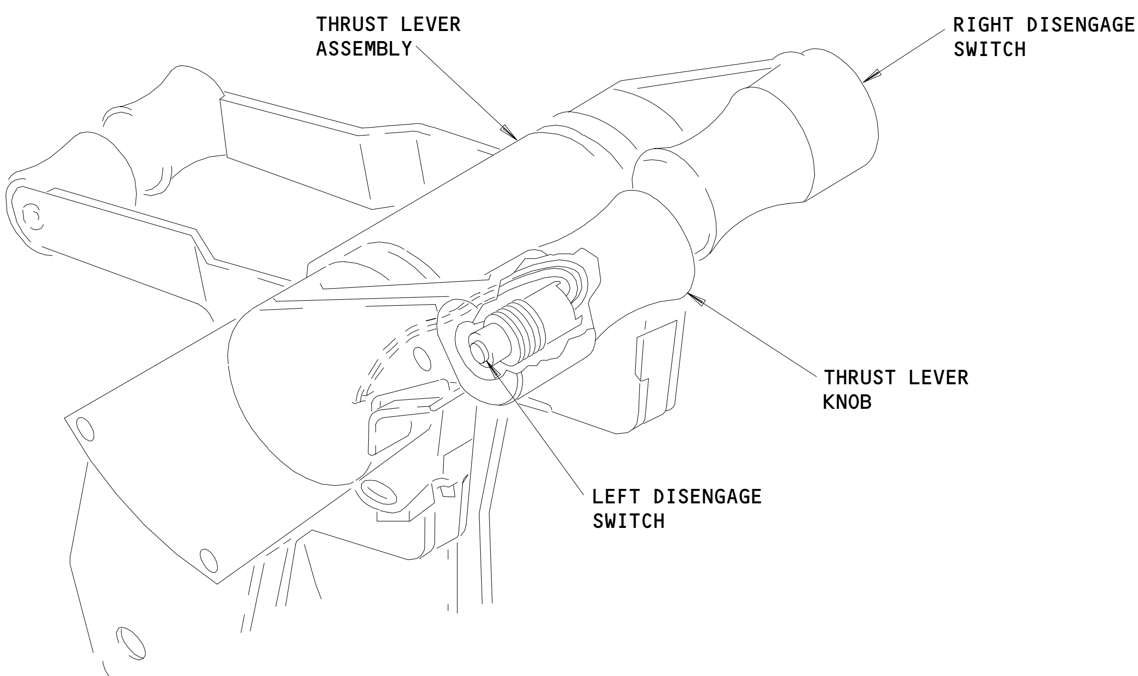
      S 024-006
      (5) Remove the disengage switch.

TASK 22-32-02-404-007

3. Install the Disengage Switch (Fig. 401)
   A. Consumable Materials
      (1) Heat Shrink Tubing - RT876, Color Yellow, Raychem Corp., Menlo Park, CA.



FLT COMPT



THRUST LEVER ASSEMBLY

A

Disengage Switches - Installation
Figure 401

(2) Locktite 222

B. References

(1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels

(2) 22-00-02/201, Autoflight BITE

(3) 24-22-00/201, Electrical Power - Control

C. Access

(1) Location Zone

211/212 Flight Compartment

D. Install the Disengage Switch

S 164-008

(1) Remove all of the remaining locktite from the thrust lever threads.

S 394-009

(2) Apply new locktite to the disengage switch threads.

S 424-010

(3) Install the disengage switch.

S 434-011

(4) Apply the heat-shrink tubing to the wire terminals (heat shrink after the wires are installed).

S 434-012

(5) Use the screws supplied with the switch to install the wires on the switch.

S 394-013

(6) Heat shrink the heat-shrink tubing.

S 414-014

(7) Install the thrust lever knob on the thrust lever.

E. Do a Test of the Disengage Switch

S 864-015

(1) Supply electrical power (Ref 24-22-00).

S 864-016
(2) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
    (a) 11F15 or 11F17, TMC DC

S 864-017
(3) Make sure these circuit breakers on the P11 panel are closed:
    (a) 11F14 or 11F16, TMC AC
    (b) 11F16 or 11F18, TMC SERVO

S 714-018
(4) Do the MCDP Ground Test 12 SW A/T DISC (Ref 22-00-02).

S 864-019
(5) Remove electrical power if it is not necessary (Ref 24-22-00).

MICROSWITCH PACK - MAINTENANCE PRACTICES

TASK 22-32-04-002-057

1. Remove the Microswitch Pack Assembly

   A. General

      (1) The microswitch pack assembly is attached to the bottom of the autothrottle clutch pack and cable quadrant frame. Access to the microswitch assembly is through access panel 113AL for the forward equipment bay (AMM 06-41-00/201).

      (2) A check of all the microswitches must be done after any microswitch pack assembly removal and installation. The switches are attached to movable mounting arms that can be adjusted to position the switches for correct activation (open/close points).

      To adjust the switches, you push down on the locking channel, which releases the adjusting bolt, and turn the adjusting bolt as necessary to correctly position the switch. You then continue to turn the adjusting bolt the minimum amount necessary to allow the locking channel to rise and lock the bolt in position.

   B. References

      (1) AMM 06-41-00/201, Section 41 Access Doors and Panels

      (2) AMM 27-61-00/201, Spoiler/Speedbrake Control System

      (3) AMM 78-31-00/201, Thrust Reverser System

   C. Access

      (1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/212	Flight Compartment

      (2) Access Panel

113AL	Flight/Landing Gear/Engine Control Components

   D. Prepare For Removal

      S 862-002

      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (Ref 27-61-00) or move all persons and equipment away from the spoilers.

S 862-003

(2) Open these circuit breakers on the overhead panel, P11 and attach DO-NOT-CLOSE tags:
- (a) 11F15 or 11F17, TMC DC
- (b) 11G11, AUTO SPEED BRAKE
- (c) 11S21, AUTOBK ANTISKID TEST IND 2

S 862-004

WARNING: DO THE DEACTIVATION PROCEDURE TO PREVENT THE OPERATION OF THE THRUST REVERSER. THE ACCIDENTAL OPERATION OF THE THRUST REVERSER CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(3) Do the deactivation procedure for the thrust reverser (forward thrust position) for ground maintenance (Ref 78-31-00/201).

S 012-039

(4) Open the access panel, 113AL, for the microswitch pack (Ref 06-41-00).

E. Remove the Microswitch Pack Assembly (Fig. 201)

S 032-081

(1) Disconnect the electrical connectors from the microswitch assembly.

S 032-006

(2) Remove the four bolts, washers, and nuts that hold the microswitch assembly to the autothrottle frame assembly.

S 032-007

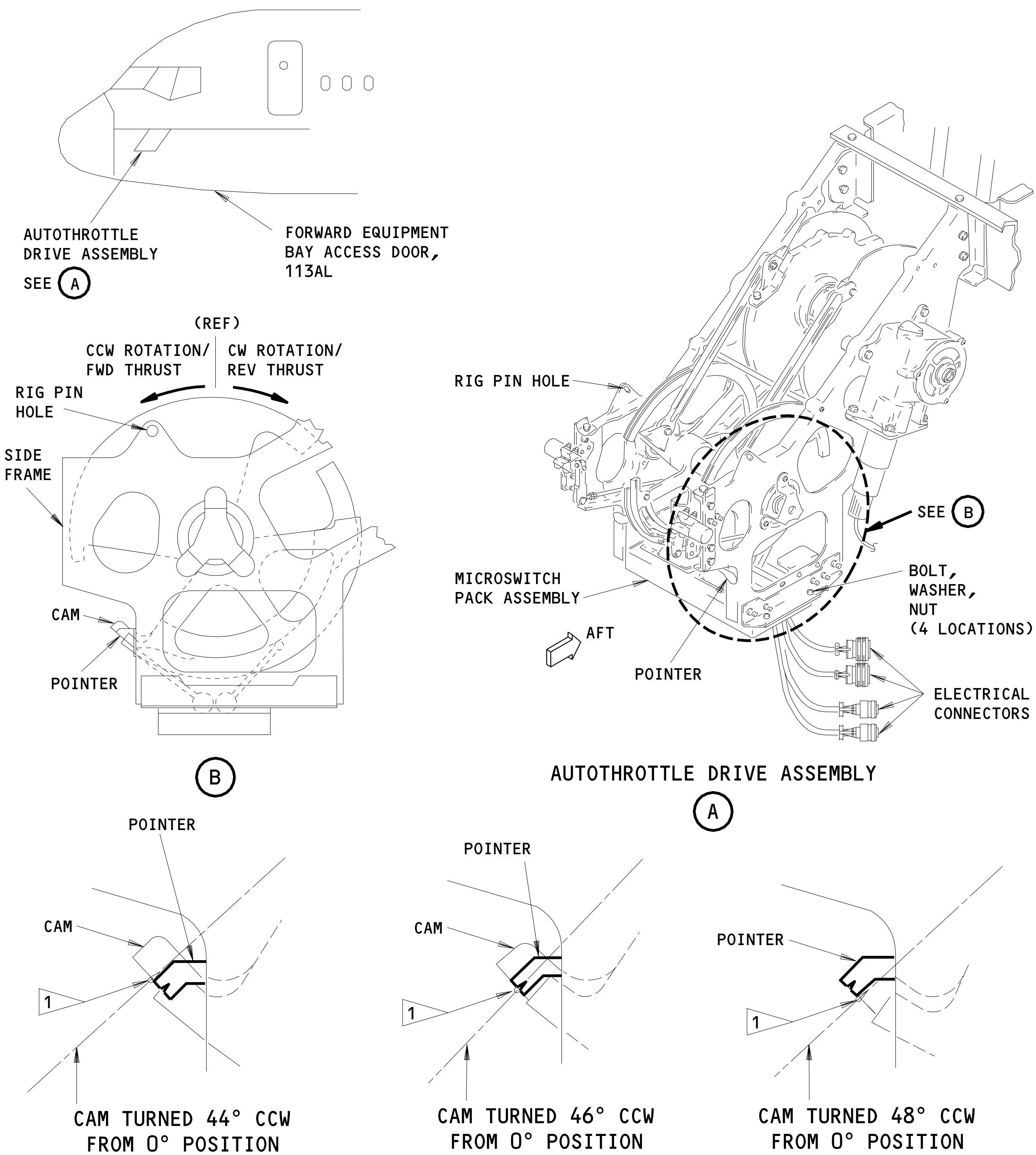
(3) Remove the microswitch pack assembly.

TASK 22-32-04-402-014

2. Microswitch Pack Assembly Installation

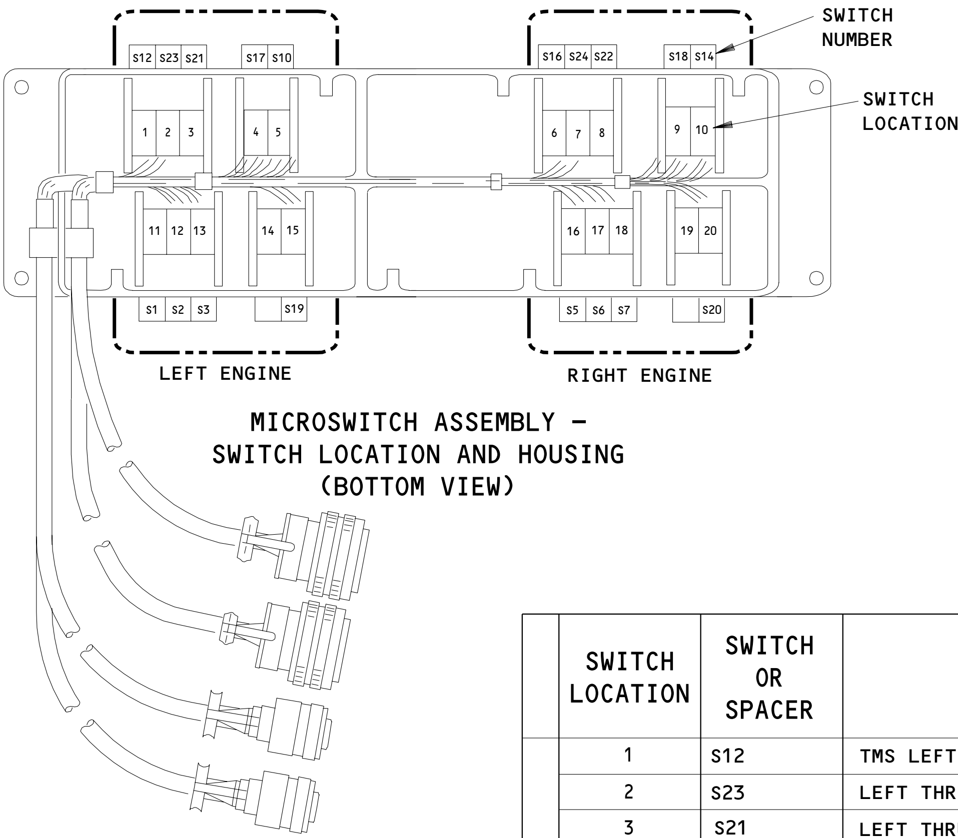
A. Equipment

(1) VTVM, Model 427A
Hewlett Packard
1501 Page Mill Rd.
Palo Alto, CA 94304



1 1/16-INCH DRILL BIT OR EQUIVALENT IS HELD IN THE CAM GROOVE TO HELP SET QUADRANT

Microswitch Pack - Installation
Figure 201



MICROSWITCH ASSEMBLY -
SWITCH LOCATION AND HOUSING
(BOTTOM VIEW)

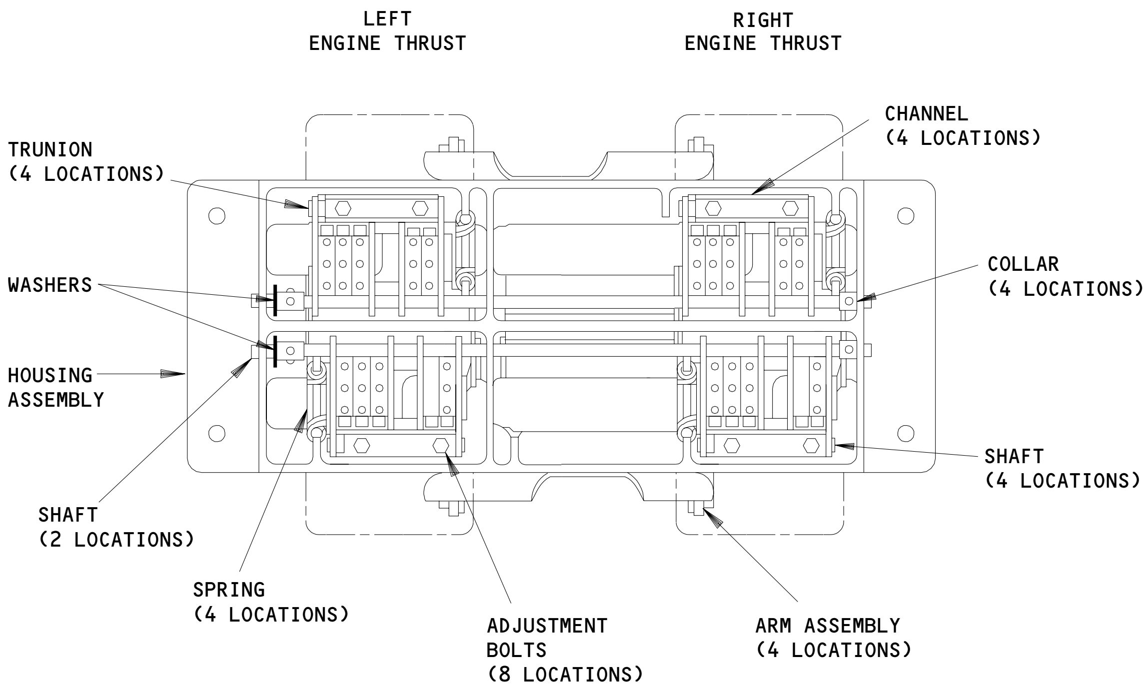
MICROSWITCH PACK -
WIRE BUNDLE ASSEMBLIES

	SWITCH LOCATION	SWITCH OR SPACER	FUNCTION
LEFT ENGINE	1	S12	TMS LEFT THRUST REVERSER
	2	S23	LEFT THRUST REVERSER INDICATION
	3	S21	LEFT THRUST REVERSER LOCK
	4	S17	LOAD SHED/PRESSURE CONTROL LEFT
	5	S10	SPEED BRAKE RETRACT LEFT
	11	S1	LANDING WARNING LEFT
	12	S2	LEFT AUTOBRAKE/AUTOBRAKE RTO
	13	S3	LEFT AUTOBRAKE/AUTOBRAKE RTO
	14	SPACER	NONE
	15	S19	EEC OFF LEFT
RIGHT ENGINE	6	S16	TMS RIGHT THRUST REVERSER
	7	S24	RIGHT THRUST REVERSER INDICATION
	8	S22	RIGHT THRUST REVERSER LOCK
	9	S18	LOAD SHED/PRESSURE CONTROL RIGHT
	10	S14	SPEED BRAKE RETRACT RIGHT
	16	S5	LANDING WARNING RIGHT
	17	S6	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	18	S7	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	19	SPACER	NONE
	20	S20	EEC OFF RIGHT

TMS: THRUST MANAGEMENT SYSTEM
RTO: REJECT TAKE-OFF

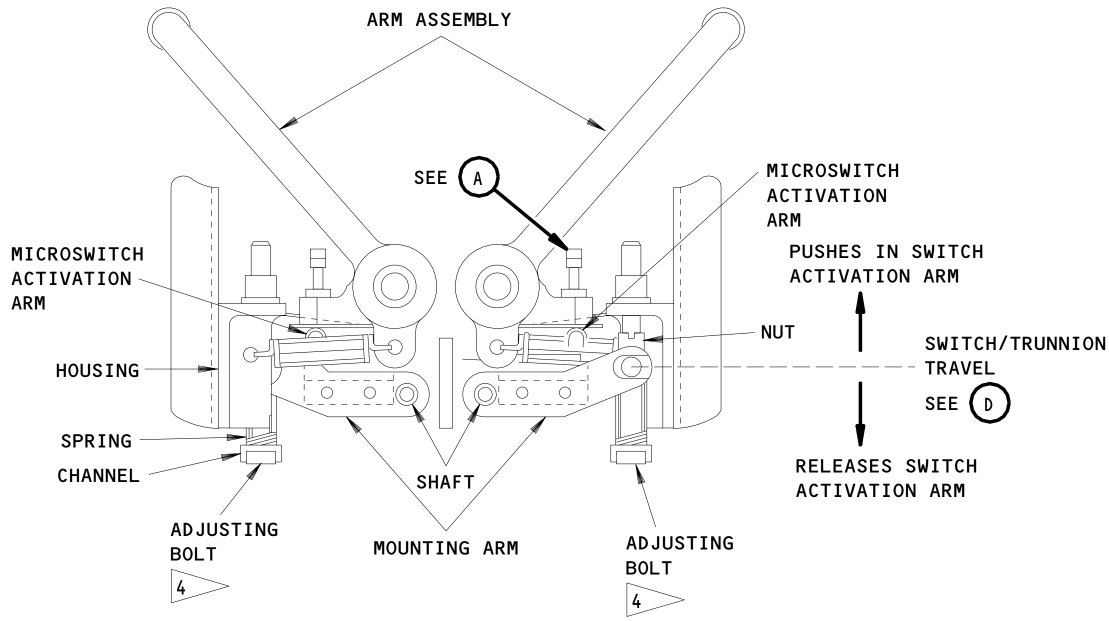
MICROSWITCH LOCATION AND FUNCTION

Microswitch Pack Assembly Switches
Figure 202 (Sheet 1)

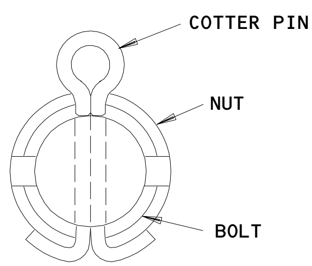


MICROSWITCH PACK
ASSEMBLY

Microswitch Pack Assembly Switches - Installation and Adjustment
Figure 202 (Sheet 2)



MICROSWITCH PACK ASSEMBLY



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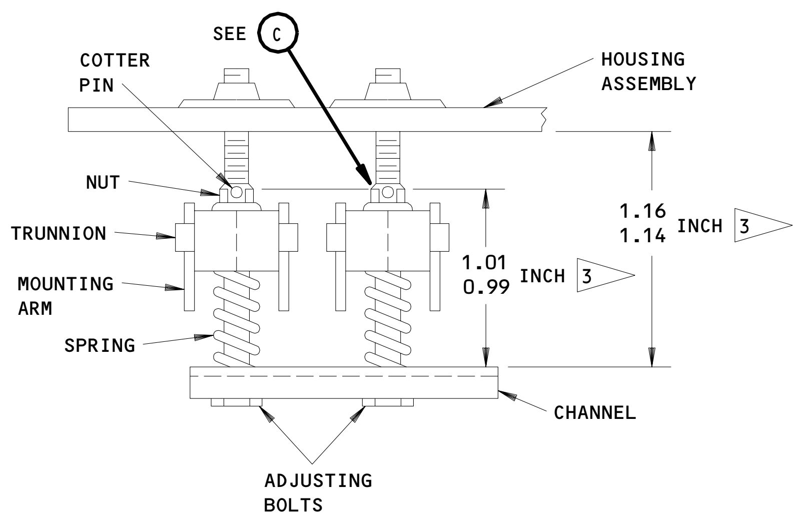


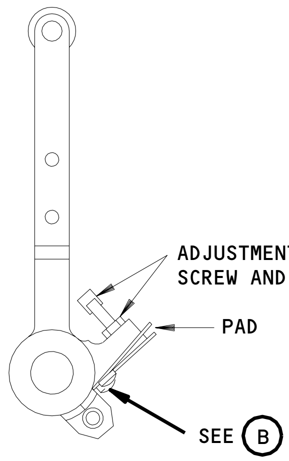
MICROSWITCH PACK - MOUNTING ARM/TRUNION DETAIL

D

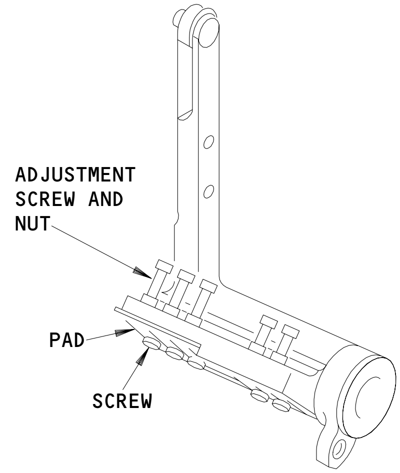
3 INITIAL ADJUSTMENT RANGE

4 ADJUSTING BOLT ADJUSTMENT DIRECTIONS:
- CW DIRECTION - PUSHES IN THE MICROSWITCH ACTIVATION ARM.
- CCW DIRECTION - RELEASES THE MICROSWITCH ACTIVATION ARM.

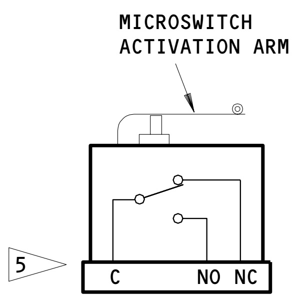
Microswitch Pack Assembly Switches - Adjustment and Operation
Figure 202 (Sheet 3)



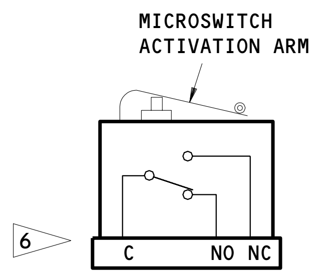
ARM ASSEMBLY
(EXAMPLE)
A



ADJUSTMENT SCREW
AND PAD DETAIL
B



RELEASED STATE
(NC - NORMALLY CLOSED)



PUSHED IN (ACTIVATED) STATE
(NO - NORMALLY OPEN)

MICROSWITCH PACK - SWITCH OPERATION MODES
(EXAMPLE)

5 POSITION OF MICROSWITCH WHEN THE ASSEMBLY ARM IS IN THE RELEASED STATE. WHEN THE ADJUSTING BOLT IS TURNED IN THE CCW DIRECTION, THE SWITCH WILL MOVE CLOSER TO THE RELEASED STATE.

6 POSITION OF MICROSWITCH WHEN THE ASSEMBLY ARM IS IN THE PUSHED IN (ACTIVATED) STATE. WHEN THE ADJUSTING BOLT IS TURNED IN THE CW DIRECTION, THE SWITCH WILL MOVE CLOSER TO THE PUSHED IN (ACTIVATED) STATE.

Microswitch Pack Assembly Switches - Operation and Adjustment
Figure 202 (Sheet 4)

(2) Rig Pin AT1 - P/N B20003-25, part of kit B20003-67 (Ref 20-10-24)

B. Access

(1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/212	Flight Compartment

(2) Access Panel

113AL	Flight/Landing Gear/Engine Control Components

C. Install the Microswitch Pack Assembly (Fig. 201)

S 432-042

(1) Put the microswitch pack in position and attach the four bolts, washers, and nuts.

D. Microswitch Adjustment (Fig. 204).

S 862-110

(1) Make sure the thrust lever rod is adjusted correctly (AMM 76-11-00/501).

S 482-111

(2) With the throttle in the idle position, put the rig pin AT1 through the autothrottle side frames and cams (Fig. 201).

S 482-132

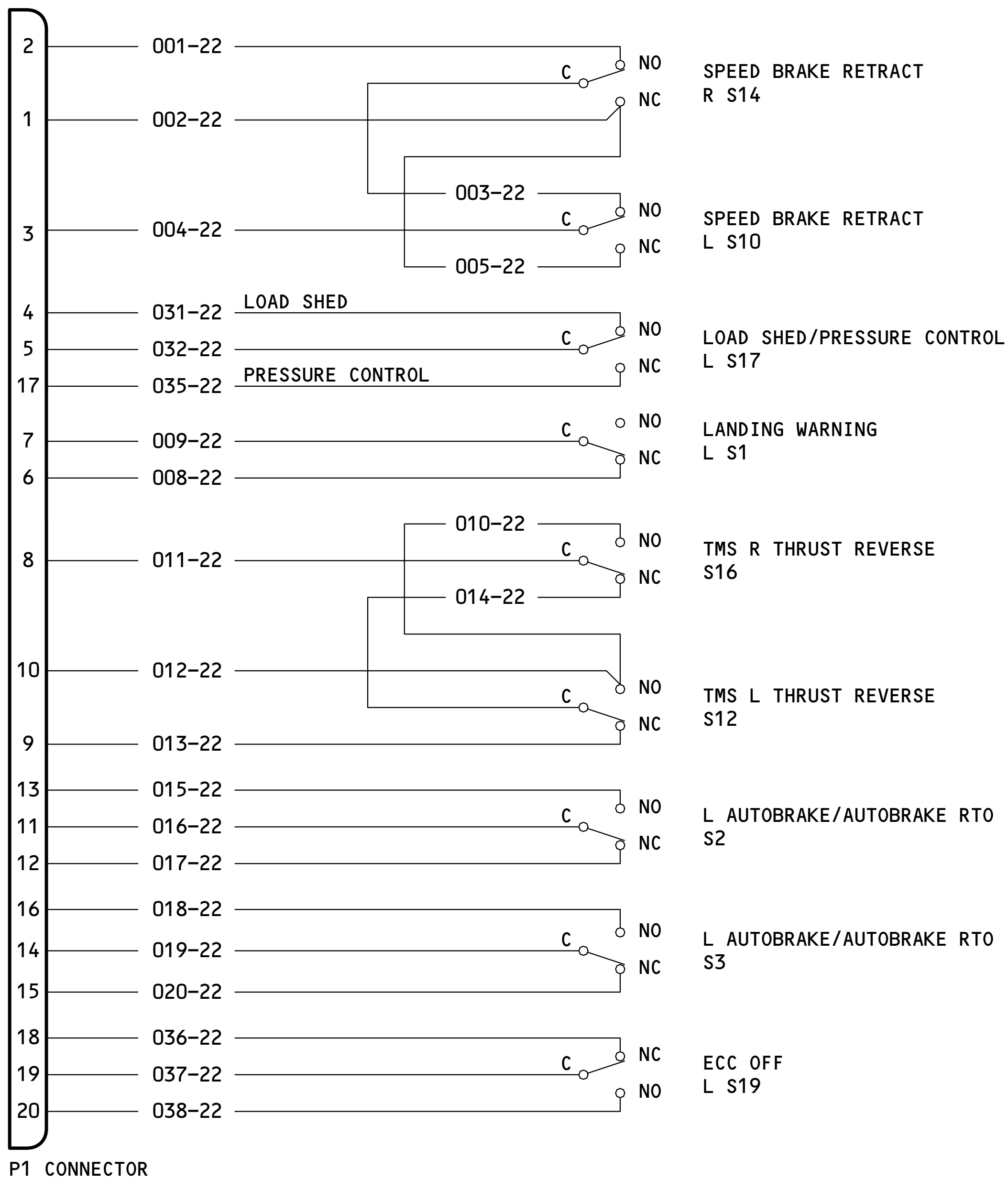
(3) The rig pin must be installed so that the autothrottle clutch link is in the center.

S 822-112

(4) Make sure that the "0" degree mark on the side of each cam aligns with the applicable pointer on the side frames. Bend the pointer as necessary to align with the "0" degree mark.

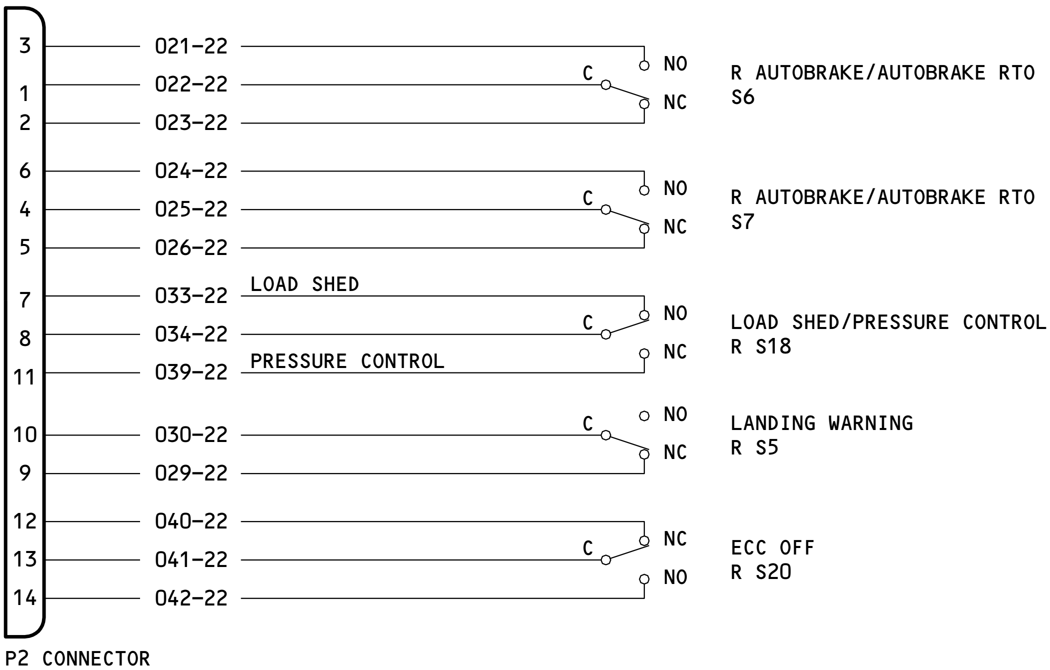
S 082-113

(5) Remove the rig pin AT1.



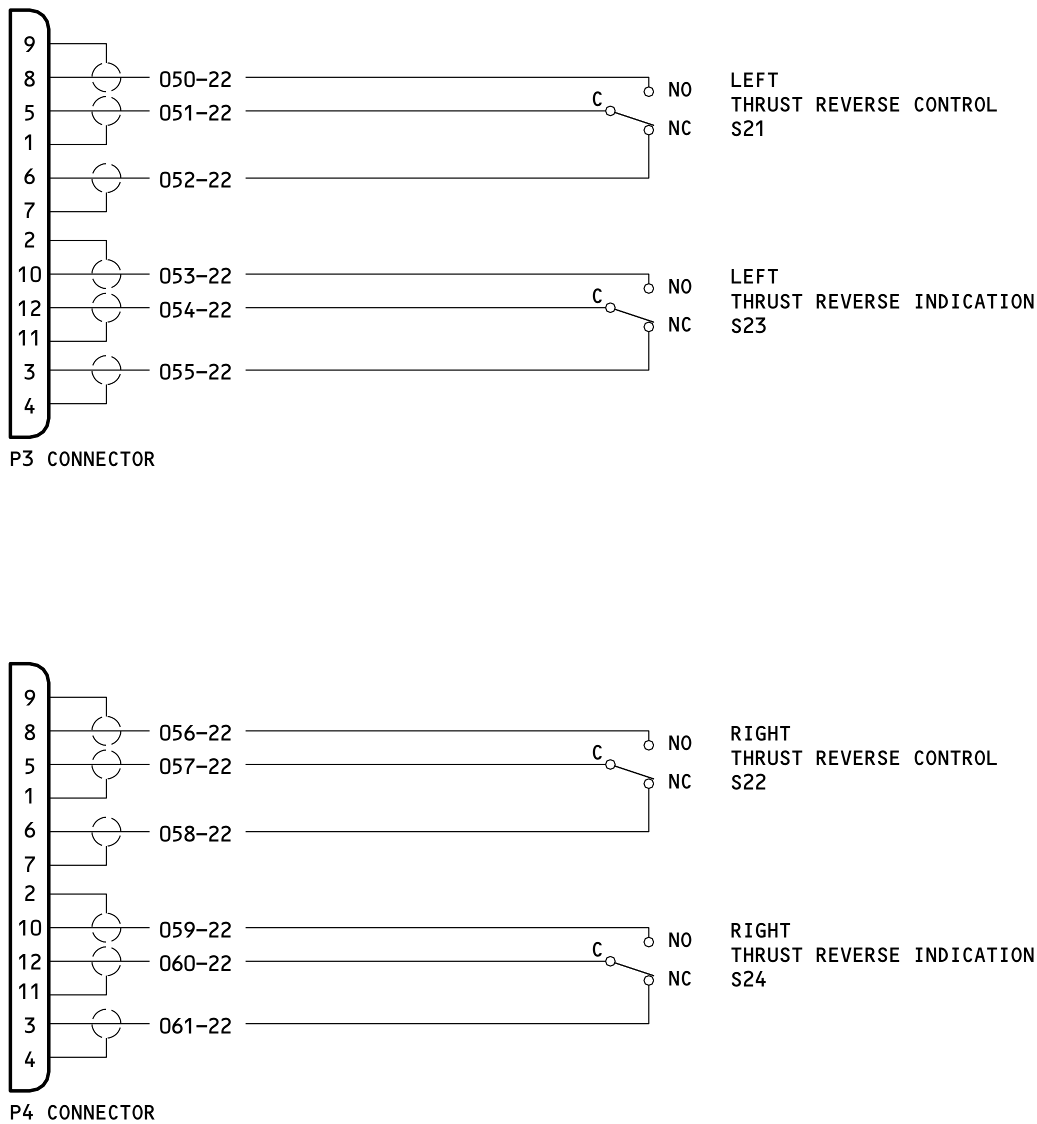
NOTE: SWITCH MODES SHOWN WITH THROTTLES AT IDLE POSITION (0° A/T CAM POSITION).

Microswitch Pack - Wire Schematic
Figure 203 (Sheet 1)



NOTE: SWITCH MODES SHOWN WITH THROTTLES AT IDLE POSITION (0° A/T CAM POSITION).

Microswitch Pack - Wire Schematic
Figure 203 (Sheet 2)



NOTE: SWITCH MODES SHOWN WITH THROTTLES AT IDLE POSITION (0° A/T CAM POSITION).

Microswitch Pack - Wire Schematic
Figure 203 (Sheet 3)

SWITCH	STEP 1	AUTOTHROTTLE CAM POSITION 2		CONNECTOR 3	PIN 3	PIN CONTINUITY CONDITION 4	SWITCH ASSEMBLY ARM CONDITION 5
		LEFT	RIGHT				
S1	1	0°		P1	6,7	CLOSED	RELEASED
S2	1	0°	-	P1	11,12	CLOSED	RELEASED
S3	1	0°	-	P1	14,15	CLOSED	RELEASED
S5	1	-	0°	P2	9,10	CLOSED	RELEASED
S6	1	-	0°	P2	1,2	CLOSED	RELEASED
S7	1	-	0°	P2	4,5	CLOSED	RELEASED
S1	2	3.5 ±0.5° (FWD)	-	P1	6,7	OPEN	PUSHED IN
S2	2	3.5 ±0.5° (FWD)	-	P1	11,12	OPEN	PUSHED IN
S3	2	3.5 ±0.5° (FWD)	-	P1	14,15	OPEN	PUSHED IN
S5	2	-	3.5 ±0.5° (FWD)	P2	9,10	OPEN	PUSHED IN
S6	2	-	3.5 ±0.5° (FWD)	P2	1,2	OPEN	PUSHED IN
S7	2	-	3.5 ±0.5° (FWD)	P2	4,5	OPEN	PUSHED IN
S17	1	0°	-	P1	4,5	CLOSED	PUSHED IN
S18	1	-	0°	P2	7,8	CLOSED	PUSHED IN
S10,S14	1	0°	0°	P1	2,3	CLOSED	PUSHED IN
S10	2	8.5 ±0.5° (FWD)	0°	P1	1,3	CLOSED	RELEASED
S17	2	8.5 ±0.5° (FWD)	-	P1	4,5	OPEN	RELEASED
S14	3	0°	8.5 ±0.5° (FWD)	P1	1,3	CLOSED	RELEASED
S18	3	-	8.5 ±0.5° (FWD)	P2	7,8	OPEN	RELEASED
S19	1	0°	-	P1	18,19	CLOSED	RELEASED
S20	1	-	0°	P2	12,13	CLOSED	RELEASED
S19	2	46 ±2° (FWD)	-	P1	18,19	OPEN	PUSHED IN
S20	2	-	46 ±2° (FWD)	P2	12,13	OPEN	PUSHED IN
S12,S16	1	0°	0°	P1	8,9	CLOSED	RELEASED
S21	1	0°	-	P3	5,6	CLOSED	RELEASED
S23	1	0°	-	P3	3,12	CLOSED	RELEASED
S22	1	-	0°	P4	5,6	CLOSED	RELEASED
S24	1	-	0°	P4	3,12	CLOSED	RELEASED
S12	2	4.5 ±0.5° (REV)	0°	P1	8,10	CLOSED	PUSHED IN
S21	2	4.5 ±0.5° (REV)	-	P3	5,8	CLOSED	PUSHED IN
S23	2	4.5 ±0.5° (REV)	-	P3	10,12	CLOSED	PUSHED IN
S16	3	0°	4.5 ±0.5° (REV)	P1	8,10	CLOSED	PUSHED IN
S22	3	-	4.5 ±0.5° (REV)	P4	5,8	CLOSED	PUSHED IN
S24	3	-	4.5 ±0.5° (REV)	P4	10,12	CLOSED	PUSHED IN

1 SWITCHES MUST BE ADJUSTED IN THE SEQUENCE SHOWN.

2 SEE FIG. 201 TO SET AUTOTHROTTLE QUADRANT:
- (FWD) FORWARD THRUST DIRECTION FROM 0° REFERENCE
- (REV) REVERSE THRUST DIRECTION FROM 0° REFERENCE
- (-) NOT APPLICABLE.

3 SEE FIG. 203 FOR WIRING SCHEMATIC.

4 ELECTRICAL CONDITION OF SWITCH CIRCUIT MEASURED BETWEEN INDICATED PINS.

5 MECHANICAL CONDITION OF MICROSWITCH ASSEMBLY ARM

Switch - Adjustment
Figure 204

S 822-114

(6) Do these steps to adjust switches S1, S2, S3, S5, S6, and S7.
 (a) Set the left and right autothrottle cam's to the position shown in step 1, Fig. 204, for switches S1, S2, S3, S5, S6, and S7.
 (b) Do a continuity check between the pins shown in step 1, Fig. 204, for each switch and do these adjustments as necessary.
  1) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.
  2) If a closed circuit exists, continue with the procedure.
 (c) Set the left and right autothrottle cam's to the position shown in step 2, Fig 204, for each switch.
 (d) Do a continuity check between the pins shown in step 2, Fig. 204, for each switch and do these adjustments as necessary.
  1) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until a closed circuit exists. Then turn the adjustment bolt clockwise just until an open circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.
  2) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until an open circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

S 822-116

(7) Do these steps to adjust switches S19 and S20.
 (a) Set the left and right autothrottle cam's to the position shown in step 1, Fig. 204, for switches S19 and S20.

(b) Do a continuity check between the pins shown in step 1, Fig. 204, for each switch and do these adjustments as necessary.
   1) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.
   2) If a closed circuit exists, continue with the procedure.

(c) Set the left and right autothrottle cam's to the position shown in step 2, Fig. 204, for each switch.

(d) Do a continuity check between the pins shown in step 2, Fig. 204, for each switch and do these adjustments as necessary.
   1) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until a closed circuit exists. Then turn the adjustment bolt clockwise just until an open circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.
   2) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until an open circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

S 822-117

(8) Do these steps to adjust switches S10, S14, S17 and S18.

NOTE: Switches S10, S14, S17, and S18 must be adjusted in the sequence shown.

NOTE: Switches S10/S17 and S14/S18 are mechanically paired together, use a common adjusting bolt (see Fig. 202).

NOTE: Switches S10/S14 are electrically paired together and must be adjusted in the sequence shown (see Fig. 203).

(a) Step 1:
  1) Set the left and right autothrottle cam's to the position shown in step 1, Fig. 204, for switches S17 and S18.
  2) Do a continuity check between the pins shown in step 1, Fig. 204, for switches S17, S18 and do these adjustments as necessary.
    a) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.
    b) If a closed circuit exists, continue with the procedure.
  3) Do a continuity check between the pins shown in step 1, Fig. 204, for switches S10, S14 and do these adjustments as necessary.
    a) If an open circuit exists, push the locking channel on the microswitch pack and turn the S10 adjustment bolt one turn and the S14 adjustment bolt one turn in the clockwise direction. Continue this sequence until a closed circuit exists for both switches. Continue to turn the adjustment bolts clockwise the minimum amount necessary to lock each bolt in their channel.
    b) If a closed circuit exists for both switches, go to step 2.

(b) Step 2:
  1) Set the left and right autothrottle cam's to the position shown in step 2, Fig. 204, for switch S10.
  2) Do a continuity check between the pins shown in step 2, Fig. 204, for switch S10 and do these adjustments as necessary.
    a) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise until an open circuit exists. Then turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

b) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

(c) Step 3:

1) Set the left and right autothrottle cam's to the position shown in step 3, Fig 204, for switch S14.

2) Do a continuity check between the pins shown in step 3, Fig. 204, for switch S14 and do these adjustments as necessary.

a) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise until an open circuit exists. Then turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

b) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

S 822-118

(9) Do these steps to adjust switches S12, S16, S21, S22, S23, and S24.

NOTE: Switches S12, S16, S21, S22, S23 and S24 must be adjusted in the sequence shown.

NOTE: Switch combinations S12/S21/S23 and S16/S22/S24 are mechanically connected together and use a common adjusting bolt (see Fig. 202).

NOTE: Switches S12/S16 are electrically paired together and must be adjusted in the sequence shown (see Fig. 203).

(a) Step 1:
1) Set the left and right autothrottle cam's to the position shown in step 1, fig. 204, for switches S12, S16, S21, S22, S23, and S24.
2) Do a continuity check between the pins shown in step 1, Fig. 204, for switches S12, S16, and do these adjustments as necessary.
a) If an open circuit exists, push the locking channel on the microswitch pack and turn the S12 adjustment bolt one turn then turn the S16 adjustment bolt one turn in the counterclockwise direction. Continue this sequence just until a closed circuit exists for both switches. Continue to turn each adjustment bolt counterclockwise the minimum amount necessary to lock the bolts in the channel.
b) If a closed circuit exists, continue with this procedure.
3) Do a continuity check between the pins shown in step 1, Fig. 204, for switches S21, S22, S23, S24, and do these adjustments as necessary.
a) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until an closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.
b) If a closed circuit exists, go to step 2.

(b) Step 2:
1) Set the left and right autothrottle cam's to the position shown in step 2, Fig. 204, for switches S12, S21, and S23.
2) Do a continuity check between the pins shown in step 2, Fig. 204, for switches S12, S21, S23, and do these adjustments as necessary.
a) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until an open circuit exists. Then turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

b) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

(c) Step 3:

1) Set the left and right autothrottle cam's to the position shown in step 3, Fig 204, for switches S16, S22, and S24.

2) Do a continuity check between the pins shown in step 3, Fig. 204, for switches S16, S22, S24, and do these adjustments as necessary.

a) If a closed circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until an open circuit exists. Then turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

b) If an open circuit exists, push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

E. Autothrottle Microswitch Pack - Installation Test

NOTE: You must do this test to check for correct microswitch operation.

S 222-053

(1) Do the continuity checks for each switch shown in Figure 205.

(a) Set the Autothrottle Cam to the indicated position and measure the pin continuity as shown for each switch.

NOTE: If the microswitches fail this test. Remove the microswitch pack and check the microswitches initial adjustment settings (Refer to the CMM).

SWITCHES TO EXAMINE	AUTOTHROTTLE QUADRANT POSITION		PIN CONTINUITY (CONNECTOR, PINS) [2]	
	LEFT [1]	RIGHT [1]	OPEN	CLOSED
S1 S2 S3 S5 S6 S7	4.5° ±0.5° (FWD)	4.5° ±0.5° (FWD)	P1 6,7 P1 11,12 P1 14,15 P2 9,10 P2 1,2 P2 4,5	----------------- P1 11,13 P1 14,16 ----------------- P2 1,3 P2 4,6
S1 S2 S3 S5 S6 S7	1.0° ±0.5° (FWD)	1.0° ±0.5° (FWD)	----------------- P1 11,13 P1 14,16 ----------------- P2 1,3 P2 4,6	P1 6,7 P1 11,12 P1 14,15 P2 9,10 P2 1,2 P2 4,5
S10 & S14 S17 S18	9.5° ±0.5° (FWD)	9.5° ±0.5° (FWD)	P1 2,3 P1 4,5 P2 7,8	P1 1,3 P1 5,17 P2 8,11
S10 & S14 S17	5.5° ±0.5° (FWD)	9.5° ±0.5° (FWD)	P1 2,3 P1 5,17	P1 1,3 P1 4,5
S10 & S14 S18	9.5° ±0.5° (FWD)	5.5° ±0.5° (FWD)	P1 2,3 P2 8,11	P1 1,3 P2 7,8
S10 & S14	5.5° ±0.5°	5.5° ±0.5°	P1 1,3	P1 2,3
S12 & S16	6.0° ±0.5° (REV)	6.0° ±0.5° (REV)	P1 8,9	P1 8,10
S12 & S16	6.0° ±0.5° (REV)	3.0° ±0.5° (REV)	P1 8,9	P1 8,10
S12 & S16	3.0° ±0.5° (REV)	6.0° ±0.5° (REV)	P1 8,9	P1 8,10
S12 & S16	3.0° ±0.5° (REV)	3.0° ±0.5° (REV)	P1 8,10	P1 8,9
S19 S20	42° ±2° (FWD) [1]	42° ±2° (FWD) [1]	P1 19,20 P2 13,14	P1 18,19 P2 12,13
S19 S20	50° ±2° (FWD) [1]	50° ±2° (FWD) [1]	P1 18,19 P2 12,13	P1 19,20 P2 13,14
S21 S22 S23 S24	6.0° ±0.5° (REV)	6.0° ±0.5° (REV)	P3 5,6 P4 5,6 P3 3,12 P4 3,12	P3 5,8 P4 5,8 P3 10,12 P4 10,12
S21 S22 S23 S24	3.0° ±0.5° (REV)	3.0° ±0.5° (REV)	P3 5,8 P4 5,8 P3 10,12 P4 10,12	P3 5,6 P4 5,6 P3 3,12 P4 3,12

[1] SEE FIG. 201 TO SET AUTOTHROTTLE QUADRANT:
- (FWD) FORWARD THRUST DIRECTION FROM 0° REFERENCE
- (REV) REVERSE THRUST DIRECTION FROM 0° REFERENCE

[2] SEE FIG. 203 FOR WIRING SCHEMATIC.

Switch Adjustment - Check
Figure 205

F. Autothrottle Microswitch Pack - Load Shed Test

NOTE: This test checks the load shed function of the microswitch pack assembly and the P1 and P2 connector wiring.

S 432-028

(1) Make sure that the electrical connectors to the microswitch pack assembly are connected.

S 622-149

(2) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:
 (a) 11R4, UTIL BUS L
 (b) 11R31, UTIL BUS RIGHT

S 622-150

(3) Make sure these circuit breakers on the main power distribution panel, P6, are closed:
 (a) 6B1, GEN CONT UNIT L
 (b) 6B2, GEN CONT UNIT R
 (c) 6B3, GEN CONT UNIT APU
 (d) 6B4, BUS PWR CONT UNIT

S 622-151

(4) Supply electrical power (AMM 24-22-00/201).
 (a) Supply electrical power to the airplane buses from a single electrical source, either the left IDG, right IDG or APU generator.

S 222-152

(5) Push the L and R UTILITY BUS switches on the electrical system control panel on the P5 panel to the ON position.

S 222-153

(6) Make sure the L and R UTILITY BUS yellow OFF annunciators are not on.

S 222-154
(7) Set the two thrust levers against the forward stops.

S 622-155
(8) Make sure the L and R UTILITY BUS yellow OFF annunciators come on.

G. Put the airplane back to its usual condition

S 862-030
(1) Put the thrust levers to the idle position.

S 862-101
(2) Push the L and R UTILITY BUS switches on the electrical system control panel on the P5 panel to the ON position.

S 862-031
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11F15 or 11F17, TMC DC
(b) 11G11, AUTO SPEED BRAKE
(c) 11S21, AUTOBK ANTISKID TEST IND 2

S 862-032
(4) Do the spoiler/speedbrake activation procedure (Ref 27-61-00).

S 862-033
(5) Do the activation procedure for the thrust reverser (Ref 78-31-00/201).

S 412-056
(6) Close the access panel, 113AL (Ref 06-41-00).

S 862-034
(7) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOTHROTTLE CLUTCH PACK - REMOVAL/INSTALLATION

1. General
   A. The autothrottle clutch pack assembly is below the flight compartment floor, below the control stand. The clutch pack has two clutch assemblies installed on one shaft. The shaft is connected to a gearbox and servomotor generator. Access to the clutch pack is through the flight/landing gear/engine control components access panel 113AL.

TASK 22-32-05-004-004

2. Remove the Autothrottle Clutch Pack Assembly (Fig. 401)
   A. References
      (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) AMM 22-32-01/401, Autothrottle Servomotor Generator and Gearbox
   B. Access
      (1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/212	Flight Compartment

      (2) Access Panel

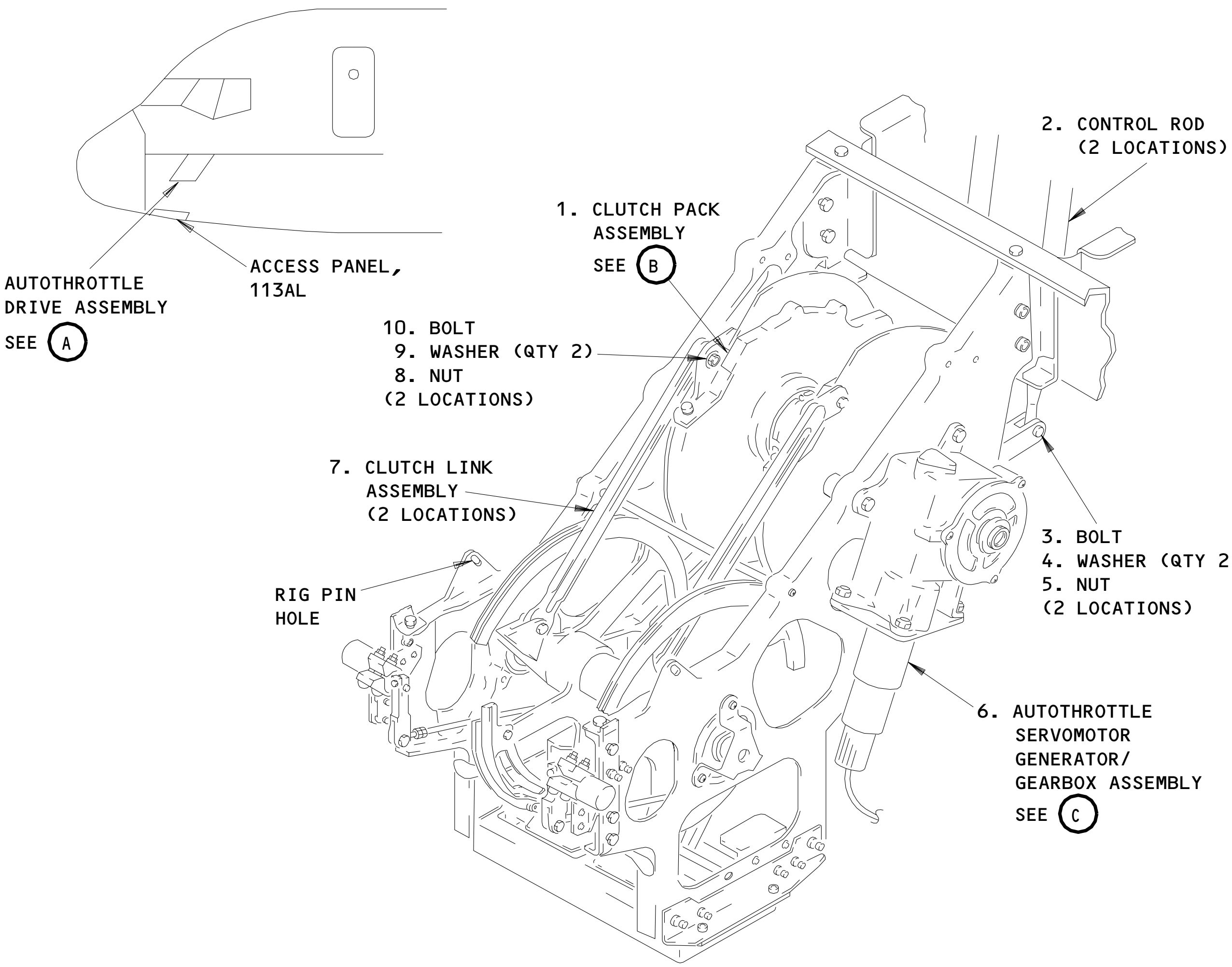
113AL	Flight/Landing Gear/Engine Control Components

   C. Prepare for Removal

      S 864-001
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11F14 or 11F16, TMC AC
         (b) 11F15 or 11F17, TMC DC
         (c) 11F16 or 11F18, TMC SERVO

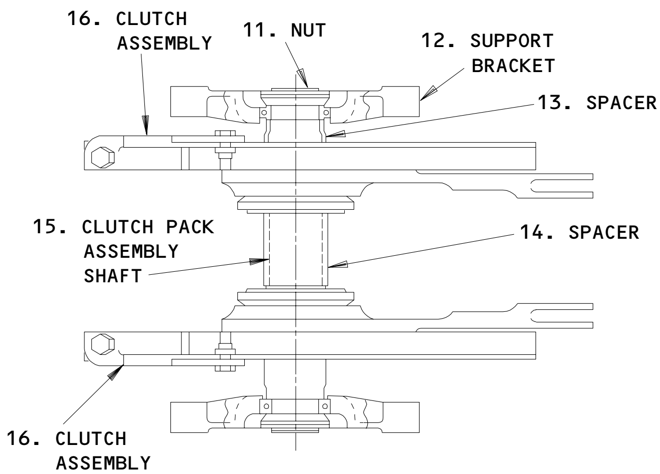
      S 864-002
      (2) Attach DO-NOT-OPERATE tags to the thrust levers.

      S 014-065
      (3) Open the access panel, 113AL, for the autothrottle clutch pack assembly (AMM 06-41-00/201).



AUTOTHROTTLE DRIVE ASSEMBLY

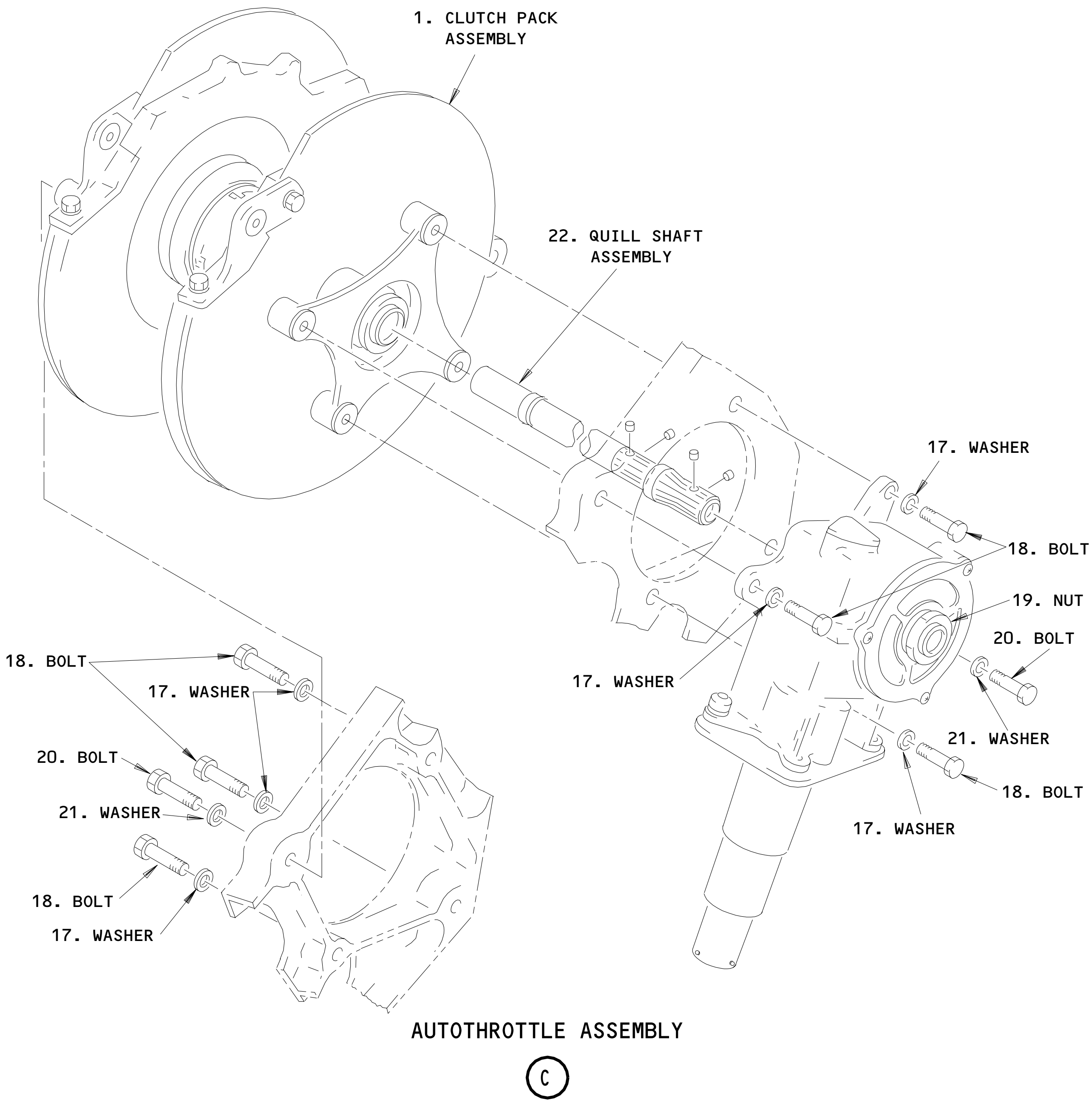
A



1. CLUTCH PACK ASSEMBLY
(TOP VIEW)

B

Autothrottle Clutch Pack Installation
Figure 401 (Sheet 1)



Autothrottle Clutch Pack Installation
Figure 401 (Sheet 2)

D. Remove the Autothrottle Clutch Pack Assembly

S 014-006

(1) Remove the autothrottle servomotor generator/gearbox assembly (6) (AMM 22-32-01/401).

S 034-009

(2) Remove the bolts (10), washers (9), and nuts (8) to disconnect the clutch link assembly (7) from each clutch assembly (16).

NOTE: Do not remove the bearing from the bracket assembly unless it is necessary for repair or replacement.

S 034-010

(3) Remove the bolts (3), washers (4), and nuts (5) to disconnect the control rod (2) (which is attached to each thrust lever) from each clutch assembly (16).

S 034-012

(4) Remove the quill shaft assembly (22) straight out of the clutch pack assembly (1).

S 034-014

(5) Hold the clutch pack assembly (1) and remove the bolts (18, 20) and washers (17, 21) that attach the clutch pack assembly to the frame assembly.

S 024-017

(6) Remove the clutch pack assembly (1) from the frame assembly.

TASK 22-32-05-004-018

3. Remove an Autothrottle Clutch Assembly (Fig. 401)

A. Access

(1) Location Zones

113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Flight/Landing Gear/Engine Control Components

B. Remove an Autothrottle Clutch Assembly

S 014-032
(1) Remove the autothrottle clutch pack assembly (1).

S 034-019
(2) Remove the nut (11) from the end of the clutch pack assembly shaft (15).

S 034-020
(3) Remove the support bracket (12) from the clutch pack assembly shaft (15).

S 034-021
(4) Remove the spacer (13, 14) from the clutch pack assembly shaft (15).

S 024-022
(5) Remove the clutch assembly (16) from the clutch pack assembly shaft (15).

TASK 22-32-05-404-023
4. Install an Autothrottle Clutch Assembly (Fig. 401)
A. Equipment
(1) Torque Rack - A22003-23

B. Parts

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401	1	Clutch Pack Assembly	22-32-51	05	202
	11	Nut			210
	12	Support Bracket *[1]			
	13	Spacer	22-32-51	05	206
	14	Spacer			205
	15	Clutch Pack Assembly Shaft			207
	16	Clutch Assembly			212
	16	Clutch Assembly			214

*[1] Refer to CMM 22-32-25.

C. Access
(1) Location Zones
113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Flight/Landing Gear/Engine Control Components

D. Install Autothrottle Clutch Assembly

S 424-024
(1) Install the clutch assembly (16) on the clutch pack assembly shaft (15).
(a) Make sure that the clutch assembly (16) is aligned with the center spacer (14).

S 434-025
(2) Install the spacer (13) on the clutch pack assembly shaft (15).

S 434-026
(3) Install the support bracket (12) on the clutch pack assembly shaft (15).

S 434-027
(4) Install a new nut (11) to the end of the clutch pack assembly shaft (15).

S 824-029
(5) Hold the clutch pack assembly (1) on a torque rack, and tighten the nut (11) to 200-220 in-lbs more than run-on torque.

NOTE: We recommend you replace the nut. The nut is a locknut with a nylon locking device. The run-on torque measured during locknut installation can be lower when you use the nut again.

S 414-030
(6) Install the autothrottle clutch pack assembly (1).

S 714-031
(7) Do a test of the autothrottle clutch pack assembly.

TASK 22-32-05-404-033
5. Install the Autothrottle Clutch Pack Assembly (Fig. 401)
A. Equipment

(1) Rig Pin - P/N P-7 A20004-23 from set A20004-2

B. Consumable Material

(1) D00013 Grease - MIL-G-23827

C. Parts

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Clutch Pack Assembly	22-32-51	05	202
	2	Control Rod	76-11-01	05	69
	3	Bolt			12
	4	Washer			22
	4	Washer			23
	5	Nut			25
	6	Servomotor Generator/Gearbox Assy.	22-32-51	05	45
	7	Clutch Link Assembly			195
	8	Nut			200
	9	Washer			198
	10	Bolt			196
	12	Support Bracket *[1]			
	15	Clutch Pack Assembly Shaft	22-32-51	05	207
	17	Washer			173
	18	Bolt			171
	19	Nut	22-32-51	06	170
	20	Bolt	22-32-51	05	168
	21	Washer			169
	22	Quill Shaft Assembly			185

*[1] Refer to CMM 22-32-25 Fig. 1

D. References

(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels

(2) AMM 22-32-01/401, Autothrottle Servomotor Generator and Gearbox

(3) AMM 76-11-00/501, Engine Control System

E. Access

(1) Location Zones

113/114 Area Forward of NLG Wheel Well
211/212 Flight Compartment

(2) Access Panel

113AL Flight/Landing Gear/Engine Control Components

F. Install the Autothrottle Clutch Pack Assembly

S 424-034

(1) Install and hold the clutch pack assembly (1) in the autothrottle drive frame assembly.

(a) Align the clutch pack assembly (1) boltholes with the frame assembly.

S 434-036

(2) Install the bolts (18, 20) and washers (17, 21) to attach the clutch pack assembly (1) to the right side of the frame assembly.

S 434-037

(3) Install the bolt (20) and washer (21) to attach the clutch pack assembly (1) to the left side of the frame assembly.

S 644-039

(4) Apply a layer of grease on the quill shaft assembly (22).

S 434-041

(5) Put the quill shaft assembly (22) in the clutch pack assembly shaft (15).

(a) Put the two quill shaft assemblies (20) in the clutch pack assembly shaft.

S 414-045

(6) Install the autothrottle servomotor generator/gearbox assembly (6) with the bolts (18) and washers (17) (AMM 22-32-01/401).

S 434-050

(7) Connect the clutch link assembly (7) to each clutch assembly (16) with the bolts (10), washers (9), and nuts (8).

S 834-058

(8) Install rig pin P-7 through the two thrust cable drums and the autothrottle side frames.

NOTE: You must turn the thrust cable drums clockwise to freely install rig pin P-7.

(a) Make sure that rig pin P-7 can turn easily with light pressure from your fingers.

S 834-059

(9) Have a person in the control cabin hold the forward thrust levers at the idle position.

S 434-060

(10) Connect the control rod (2) (from each thrust lever) to each clutch assembly (16) with the bolts (3), washers (4), and nuts (5).
  (a) Make sure the bolt that attaches the control rod to the release arm can be installed and turns freely.
  (b) Adjust the control rods if it is necessary (AMM 76-11-00/501).

S 834-051

(11) Remove the rig pin P-7 from the two thrust cable drums and the autothrottle side frames.

S 434-052

(12) Release the forward thrust levers.

NOTE: When the control system is correctly adjusted and the thrust levers are held against the idle stop, rig pin P-7 can not be freely installed. You must turn the thrust cable drums clockwise to freely install rig pin P-7.

S 714-052

(13) Do a test of the autothrottle clutch pack assembly.

TASK 22-32-05-714-053

6. Test the Autothrottle Clutch Pack Assembly

A. Equipment
  (1) Spring Scale, 0-20 lbs

B. References
  (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
  (2) AMM 22-00-02/201, Autoflight BITE
  (3) AMM 24-22-00/201, Electrical Power - Control
  (4) AMM 27-61-00/201, Spoiler/Speedbrake Control System

C. Access
  (1) Location Zones
    113/114 Area Forward of NLG Wheel Well
    211/212 Flight Compartment

  (2) Access Panel
    113AL Flight/Landing Gear/Engine Control Components

D. Prepare for Test

S 864-054

  (1) Supply electrical power (AMM 24-22-00/201).

S 864-055

  (2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
    (a) 11F14 or 11F16, TMC AC

(b) 11F15 or 11F17, TMC DC
(c) 11F16 or 11F18, TMC SERVO

S 864-058
(3) Remove the DO-NOT-OPERATE tags on the thrust levers.

S 864-059
(4) Set the A/T switch on the AFCS mode control panel (on the P55 panel) to ARM.

S 864-048

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(5) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 864-049
(6) Make sure the engines are not on.

E. Procedure

S 714-060
(1) Do the MCDP Ground Test 10 SERVO A/T (AMM 22-00-02/201).

S 864-061
(2) Set the A/T switch on the AFCS mode control panel to OFF.

S 484-066
(3) Attach a spring scale to the forward thrust lever knob.

S 724-067
(4) Move the thrust levers full forward, and make sure that the spring scale measures the following:
(a) A load of less than 4.5 pounds.

S 084-068
(5) Remove the spring scale.

F. Put the Airplane Back to Its Initial Condition

S 864-062
(1) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 414-063
(2) Close the access panel, 113AL (AMM 06-41-00/201).

S 864-064

(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MANAGEMENT COMPUTER ENGINE PROGRAM PINS - MAINTENANCE PRACTICES

1. General
   A. This procedure changes the engine program pin configuration to help with engine model changes. To set the engine model, the program pins are put to an electrical ground. These pins are in the connector behind the thrust management computer.

TASK 22-32-06-902-001

2. Change the Engine Program Pins
   A. References
      (1) SWPM 20-20-00, Standard Wiring Practices Manual
      (2) SWPM 20-71-14, Standard Wiring Practices Manual
      (3) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (4) 22-00-02/201, Autoflight BITE
      (5) 22-31-01/401, Thrust Management Computer
      (6) 24-22-00/201, Electrical Power - Control
   B. Access
      (1) Location Zones
         119/120 Main Equipment Center
         211/212 Flight Compartment

      (2) Access Panel
         119BL Main Equipment Center

   C. Change the Engine Program Pins

      S 022-002
      (1) Remove the thrust management computer (AMM 22-31-01/401).

      S 902-003
      (2) Change the program pin wires to agree with Table 1.
      (Ref SWPM 20-20-00, SWPM 20-71-14).

TABLE 1
THRUST MAMAGEMENT COMPUTER ENGINE PROGRAM PIN CONFIGURATION

			PIN CONNECTION					
CONFIGURATION OPTION	CONN NUM	PIN NUM	GND	OPEN	+28V	CODE	TMS FUNCTION SELECTED	MCDP TEST 57
ENGINE PROGRAM	D383B D383B D383B D383B D383A	D10 F6 G11 C1 E15	 X X X X	X		1 0 0 0 0	WITH RB211-535C ENGINES	ENGINE PROGRAM 01
ENGINE PROGRAM	D383B D383B D383B D383B D383A	D10 F6 G11 C1 E15	 X X X	X X		1 1 0 0 0	WITH RB211-535E4 ENGINES	ENGINE PROGRAM 03
ENGINE PROGRAM	D383B D383B D383B D383B D383A	D10 F6 G11 C1 E15	 X X X	X X		1 1 0 0 0	WITH RB211-535E4 OR RB211-535E4/535E4-B ENGINE INTERMIX	ENGINE PROGRAM 03
ENGINE PROGRAM	D383B D383B D383B D383B D383A	D10 F6 G11 C1 E15	X X X X	 X		0 0 1 0 0	WITH RB211-535E4-B ENGINES	ENGINE PROGRAM 04
PARITY (ENGINE PROGRAM AND THRUST LIMIT)	D383A	D13	?	?	AS REQUIRED FOR ODD PARITY			--

S 422-004
(3) Install the thrust management computer (AMM 22-31-01/401).

D. Do a Test of the Program Pin Configuration

S 862-005
(1) Supply electrical power (AMM 24-22-00/201).

S 712-006

(2) Do the MCDP Ground Test 57 TMC CONFIG/OPT (AMM 22-00-02/201).

(a) Make sure that the MCDP display agrees with Table 1.

E. Put the Airplane Back to Its Initial Condition

S 862-007

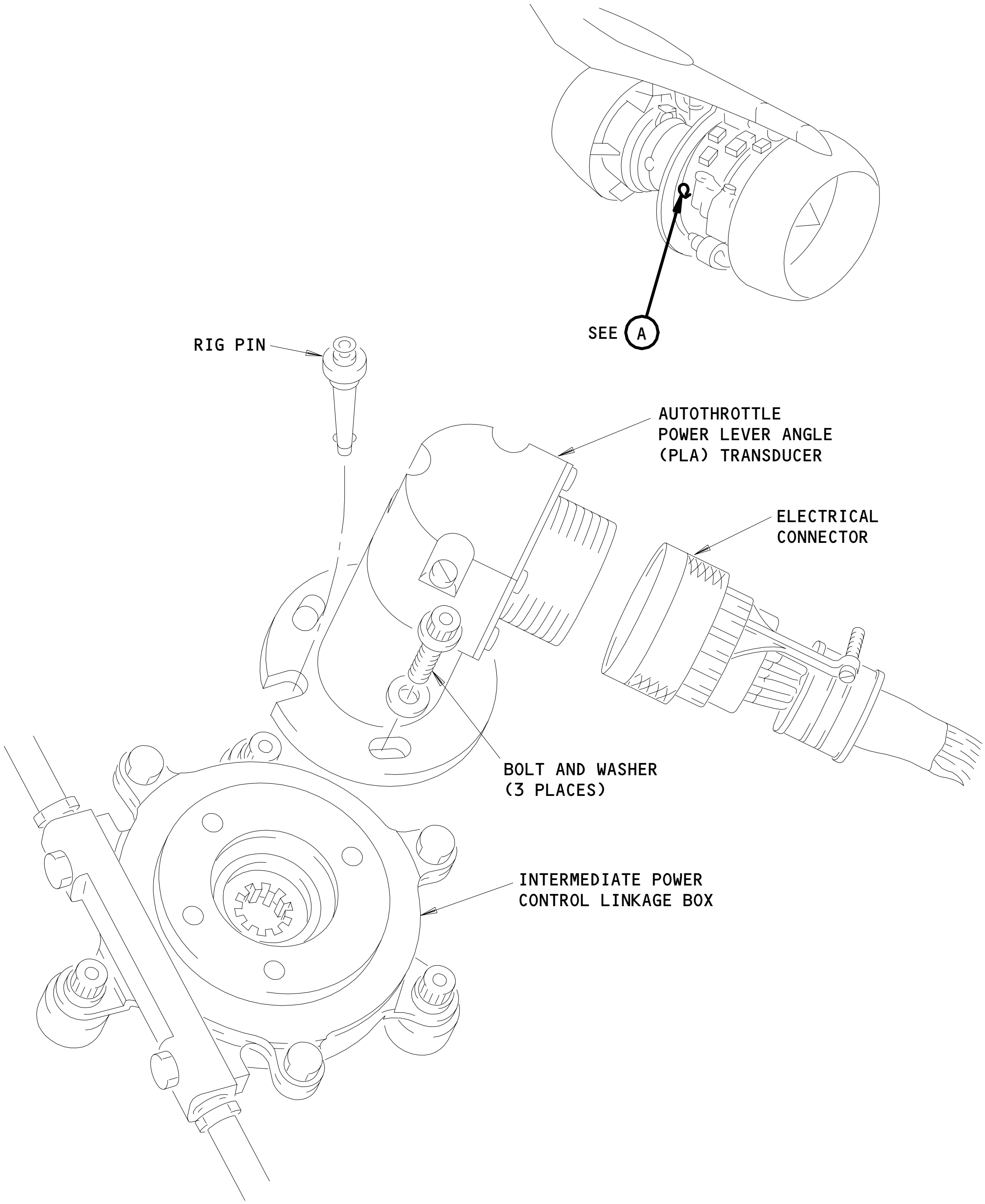
(1) Set the MCDP to off.

S 862-008

(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MANAGEMENT ENGINE - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The autothrottle Power Lever Angle (PLA) transducer is on the intermediate gearbox assembly (right side of each engine). This assembly is driven by a push pull cable which is controlled by the thrust lever.
2. Component Details
   A. Autothrottle Power Lever Angle Transducer
      (1) The autothrottle power lever angle transducer is a Rotary Variable Differential Transformer Transducer (RVDT) with a 80 degree rotary range. It is ac powered, moving a slug inductive device that outputs an ac voltage proportional to the angle displaced. The 26 volt ac, 400 Hz excitation is supplied by the TMC.
      (2) The autothrottle PLA transducer is approximately 2 inches in diameter and three inches high. One electrical connector on the side of the transducer housing contains the following pins:
         (a) 26 volts ac excitation (2 pins)
         (b) 2 wire output (2 pins)
         (c) Case ground (1 pin)
      (3) The autothrottle PLA transducer is mounted on the intermediate gearbox assembly with three bolts.
3. Operation
   A. Functional Description
      (1) The thrust levers move sheathed cables which in turn controls the intermediate gearbox assy. The intermediate gearbox assy drives the PLA by means of a gear on the PLA shaft. As the thrust (power) levers are moved the PLA outputs an ac voltage proportional to the angle moved by the thrust lever. The PLA transducer continually senses changes in the power lever angle and provides position feedback to the TMC. If the PLA aft or forward limits are reached, the TMC automatically switches to the respective PLA limit function in the TMC.



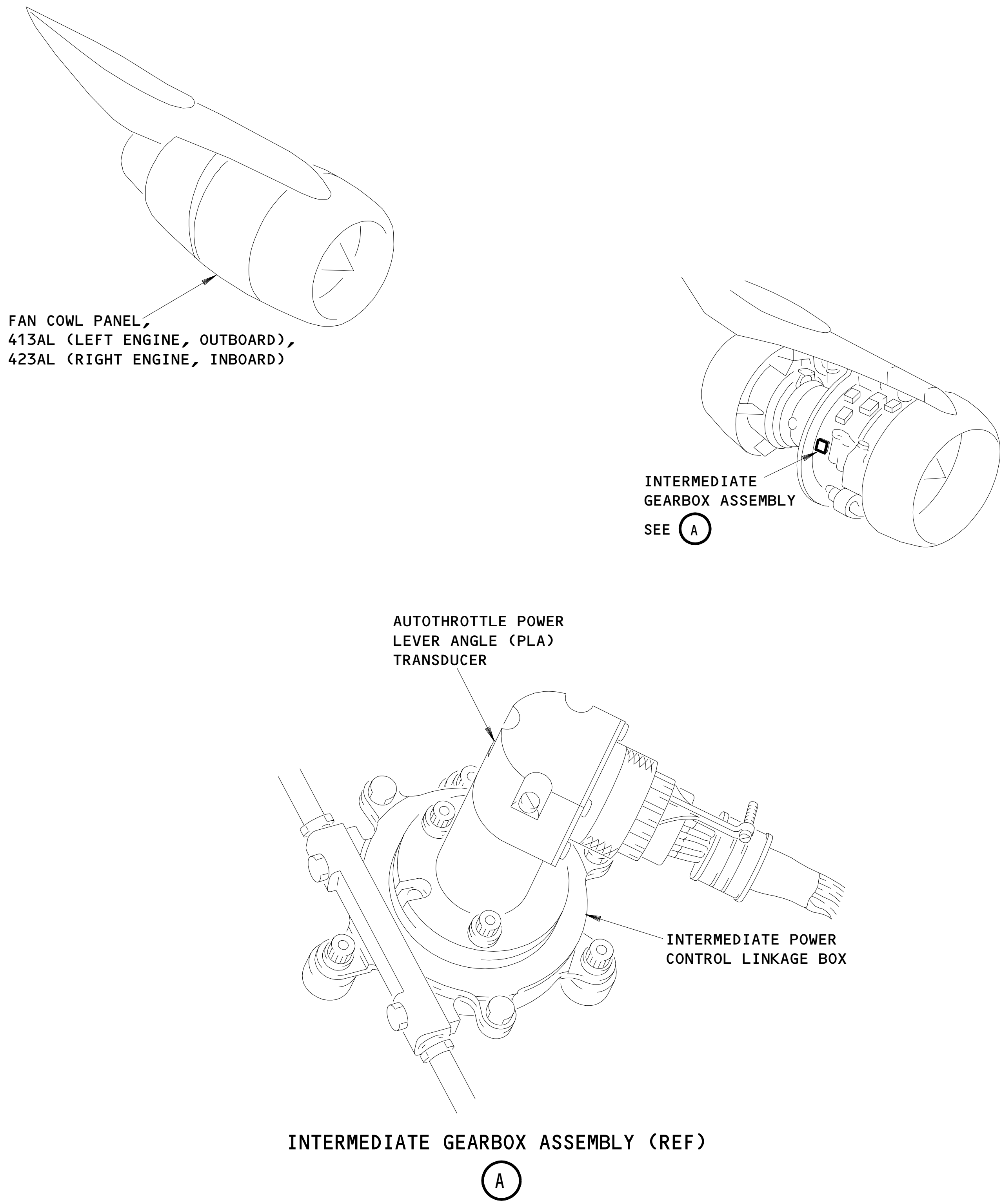
INTERMEDIATE GEAR BOX ASSEMBLY
A

Autothrottle Power Lever Angle Transducer Installation
Figure 1

THRUST MANAGEMENT ENGINE

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
TRANSDUCER - AUTOTHROTTLE POWER LEVER ANGLE, TS5029	--	1	413AL,423AL, FAN COWL PANEL INTERMEDIATE GEARBOX ASSY (REF)	22-33-01

Thrust Management Engine - Component Index
Figure 101



Thrust Management Engine - Component Location
Figure 102

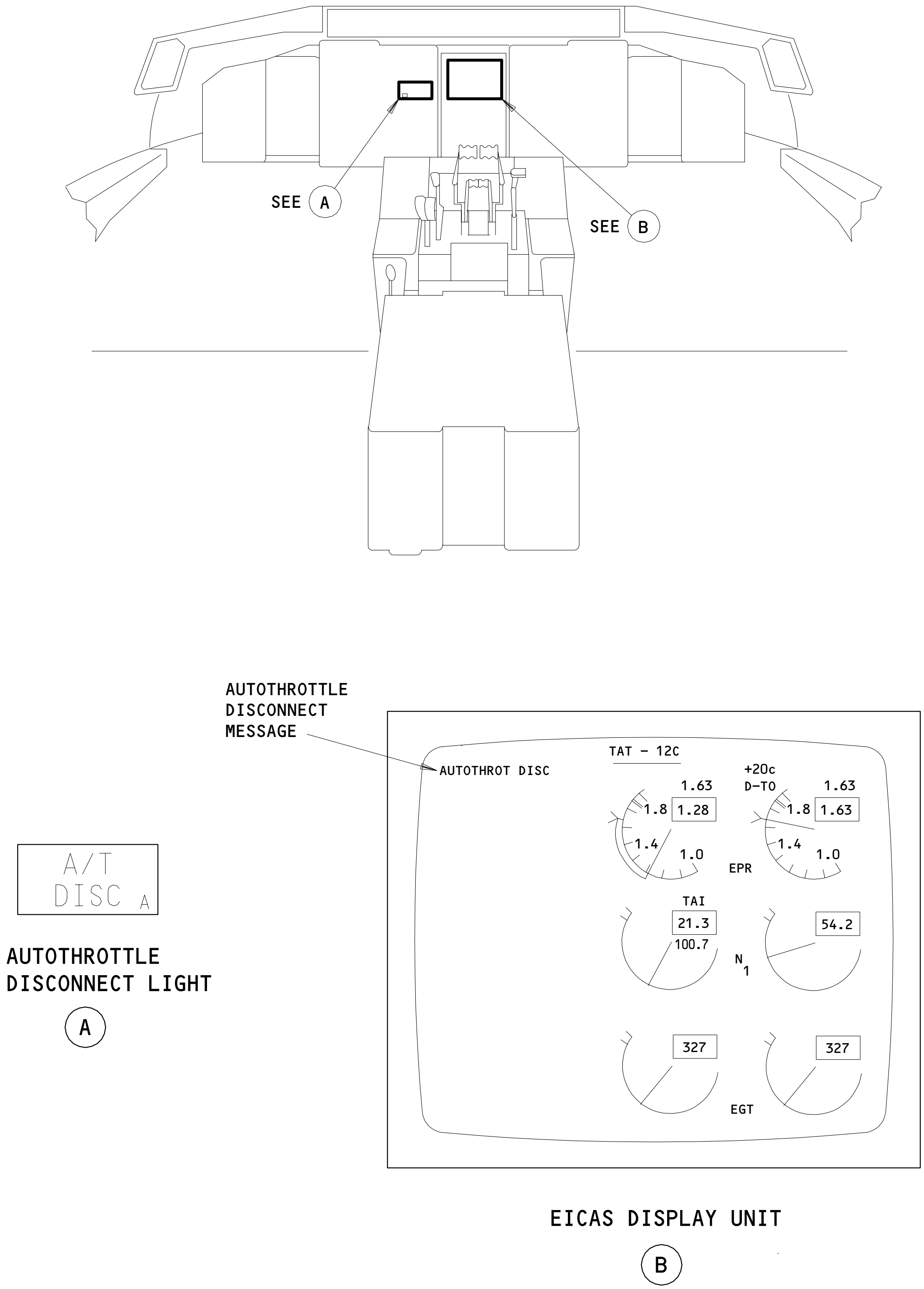
AUTOTHROTTLE POWER LEVER ANGLE TRANSDUCER - REMOVAL/INSTALLATION

TASK 22-33-01-004-043

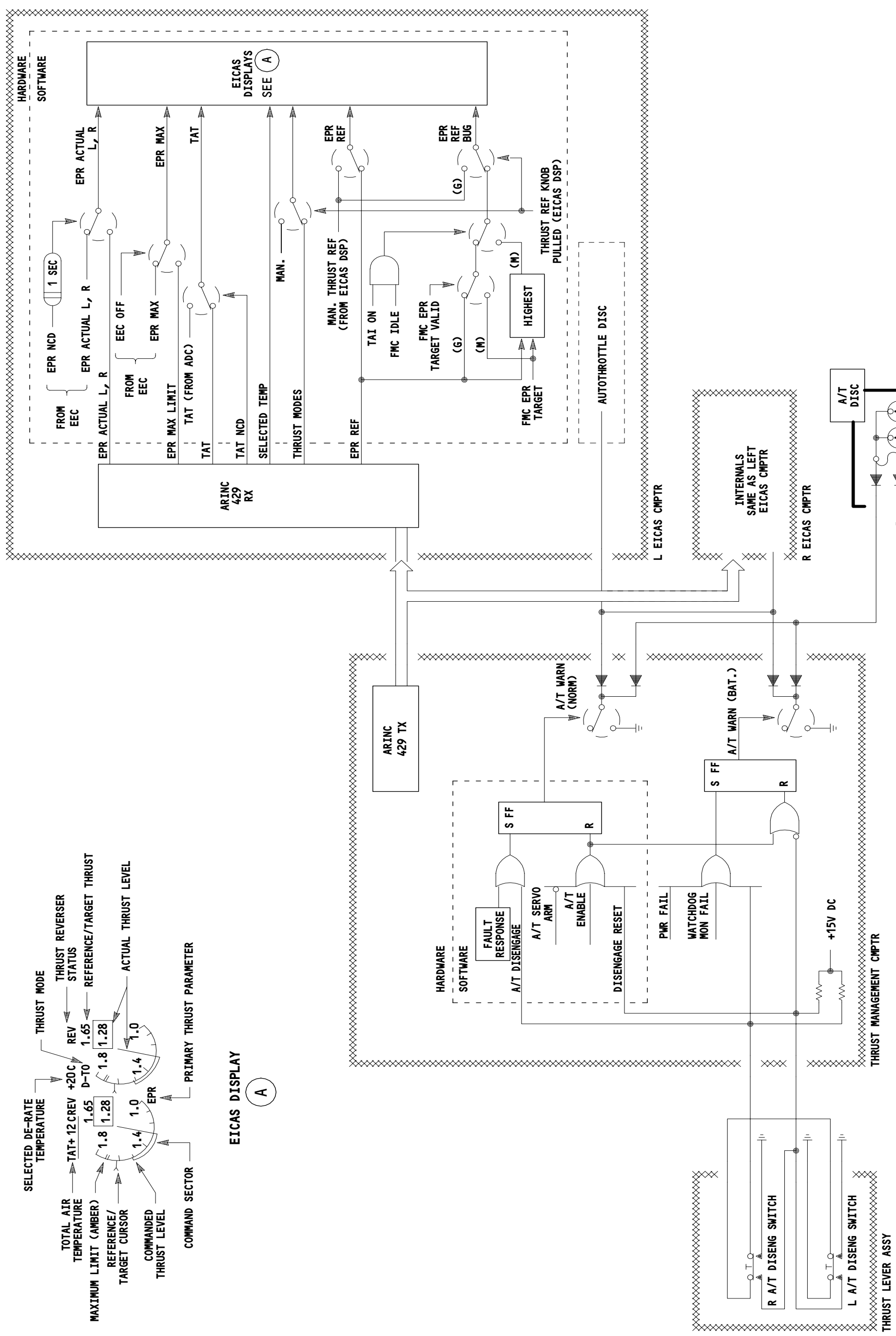
1. Autothrottle Power Lever Angle Transducer Removal and Installation
   A. General
      (1) The Autothrottle Power Lever Angle Transducer Removal and Installation is found in the Power Lever Angle Bleed Valve Control Unit and Autothrottle Transducer Removal and Installation (AMM 75-32-14/401).

THRUST MANAGEMENT WARNING AND ANNUNCIATION - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Thrust Management System (TMS) is monitored for autothrottle disengagement. Disengagement results in caution annunciations that include an amber A/T DISC light, an AUTOTHROT DISC message on the EICAS display unit (Ref 31-41-00), an aural warning, and flashing master CAUTION lights.
2. Component Details
   A. Autothrottle Disconnect Light
      (1) The A/T DISC light is on the P1-3 panel. It is an amber, dual-lamp indicator. The lamps use 28 vdc from the master dim and test system. They are turned on by application of a ground from the TMC or the master dim and test system (Ref AMM 33-16-00/001, Master Dim and Test). The A/T DISC light will also be illuminated whenever the TMC is in ground test mode.
3. Operation
   A. Functional Description
      (1) Thrust Management System Warning and Annunciation (Fig. 2)
         (a) When the autothrottle disengages, the A/T DISC light turns on. The ground to this light is controlled by both hardware and software flip-flop outputs. The software logic and output driver is powered by normal 28 vdc. The hardware logic and output driver is powered by 28 vdc from the standby bus.
         (b) The hardware flip-flop is set when the watchdog monitor senses a fault, power failure occurs, or an A/T disengage switch is pressed. The software flip-flop is set by either a software sensed failure (no A/T ENABLE signal from TMS engage-disengage logic) or when the A/T disengage switch is pressed. The flip-flops are reset by a second press of a disengage switch, or re-engagement of the autothrottle.
         (c) The left and right EICAS computers annunciate the autothrottle disengage signal one second after a ground is provided for the A/T DISC light. This results in an AUTOTHROT DISC message on the EICAS display unit, and operates the master CAUTION lights and aural warning. Pressing a master CAUTION light resets both master CAUTION lights. However, the amber A/T DISC light and EICAS message remain until both flip-flops are reset.



Flight Deck Indications
Figure 1

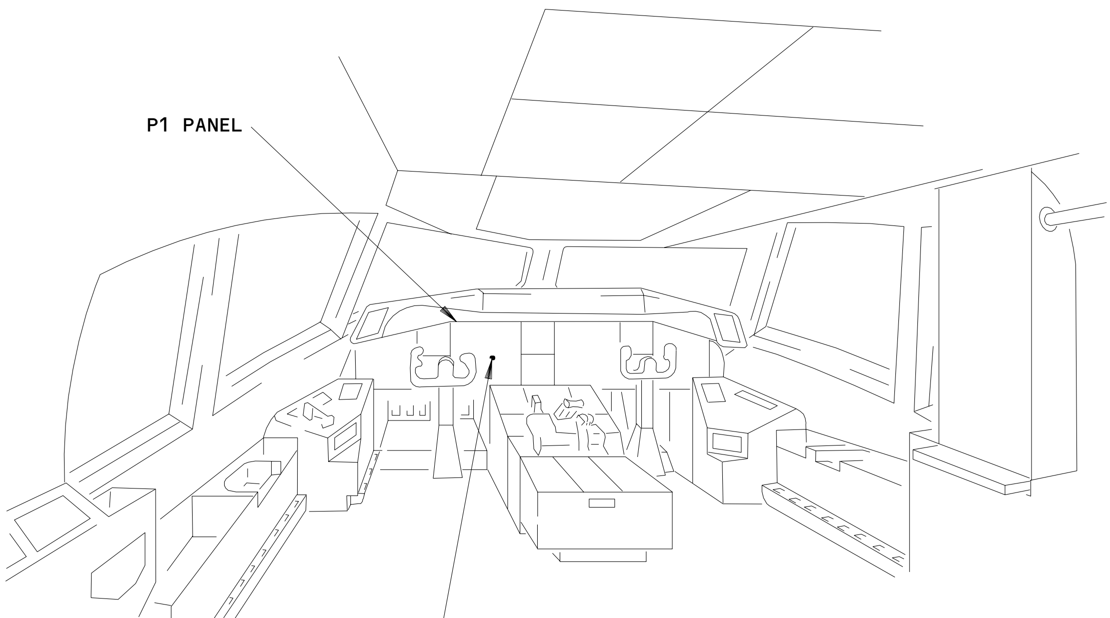


Thrust Management System - Warning and Annunciation
Figure 2

THRUST MANAGEMENT WARNING AND ANNUNCIATION

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
LIGHT - AUTOTHROTTLE DISCONNECT, L591	--	1	FLT COMPT, P1	22-34-00

Thrust Management Warning and Annunciation - Component Index
Figure 101



FLIGHT COMPARTMENT

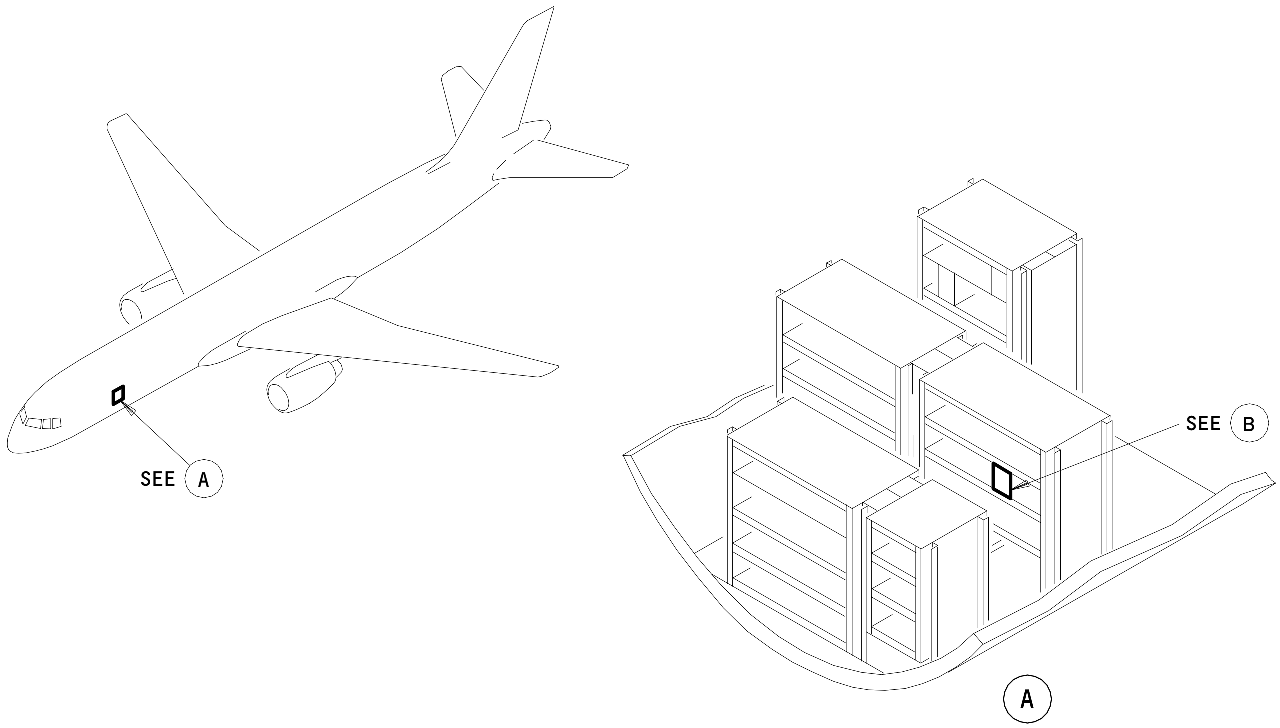
AUTOTHROTTLE DISCONNECT LIGHT, L591

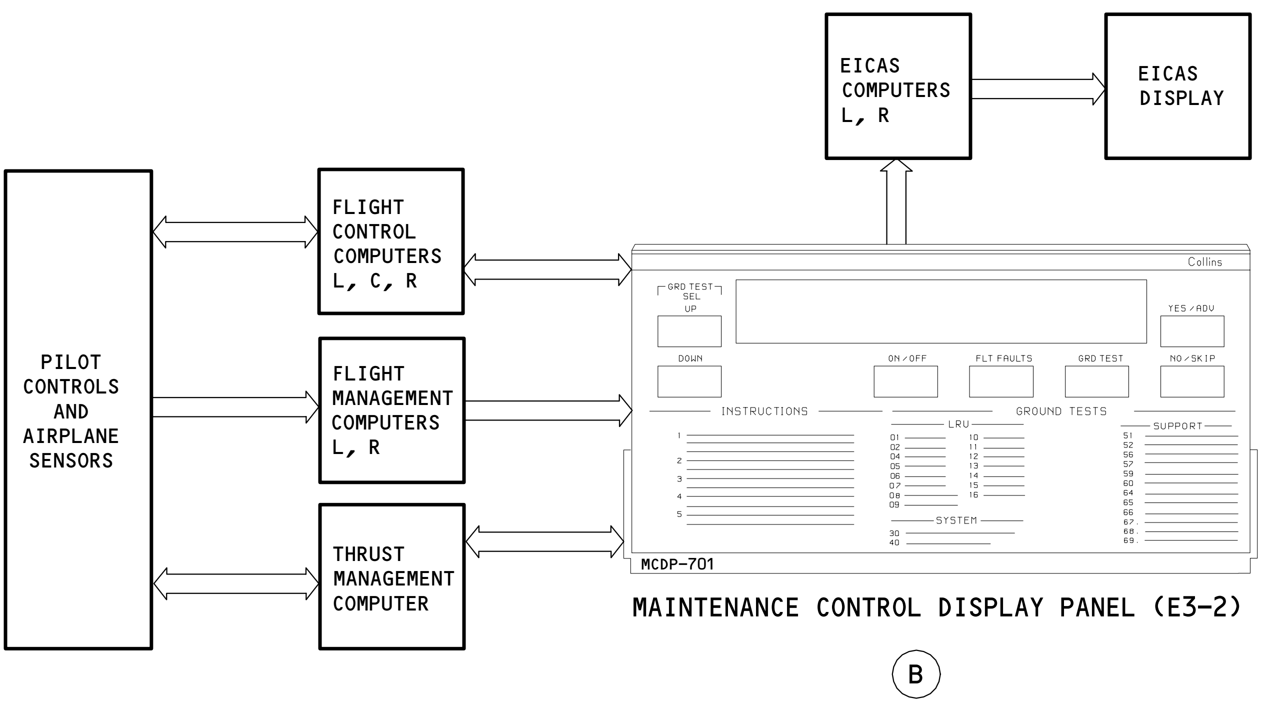
A

Thrust Management Warning and Annunciation - Component Location
Figure 102

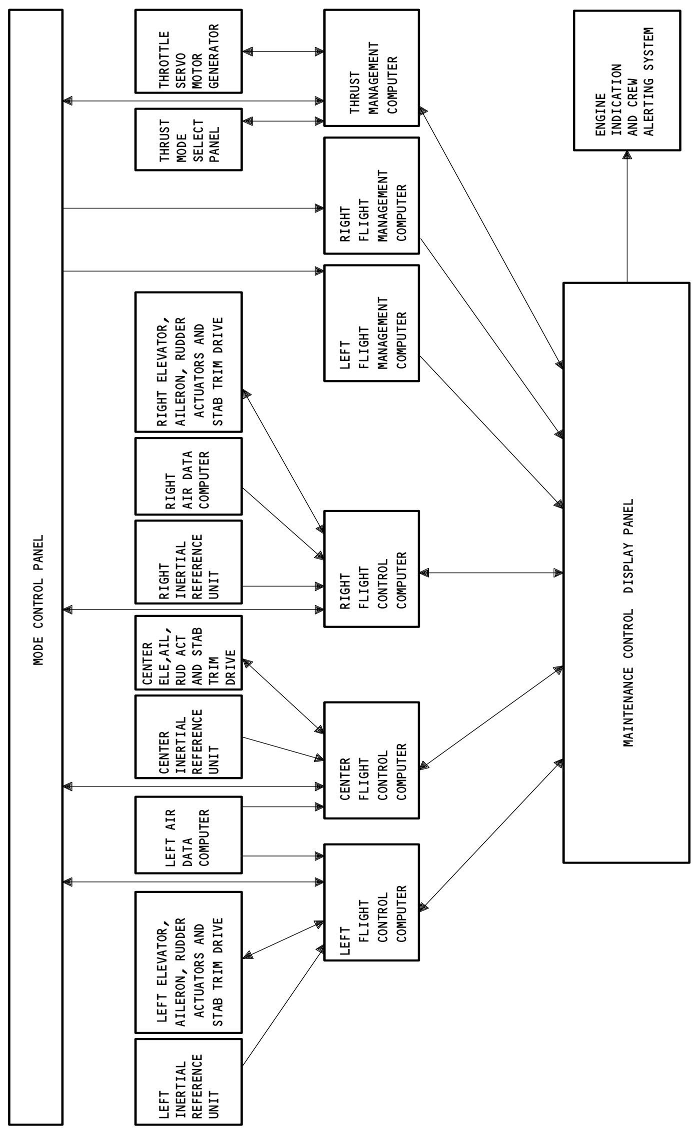
MAINTENANCE MONITOR - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The maintenance monitor system consists of a Maintenance Control Display Panel (MCDP) that monitors the status of the Flight Control Computers (FCC), Flight Management Computers (FMC), Thrust Management Computer (TMC), and their related sensors. The MCDP interrogates the computers after each landing, and stores up to five flight faults per computer in non-volatile memory for later interrogation by maintenance personnel.
   B. A central processing unit in the MCDP controls and processes ground test commands, ground test functions, ground, and flight faults. The MCDP displays flight and ground faults, test number, name or status, and operator instructions. The MCDP controls and display panel are located on the front of the unit. The controls enable maintenance personnel to display faults and ground tests that include operator instructions. The MCDP is located in the main Electrical/Electronic (E/E) equipment center.
   C. There are wiring and connector provisions for a remote control panel. The remote control panel is plugged into the connector on the forward side of the P61 panel and used for displaying flight faults and for running ground tests from the flight deck. The MCDP remote display is through the EICAS system maintenance panel in the P61 panel. The MCDP display can be displayed on the EICAS with or without the hand-held carry-on remote control panel, by pressing the CONF/MCDP switch of the EICAS maintenance panel.
   D. Maintenance Control Display Panel Interfacing Systems (Fig. 2)
      (1) The primary MCDP interfaces are with each FCC, FMC, and TMC. Secondary interfaces with the MCDP consist of FCC and TMC individual system sensors. Both primary and secondary interfaces are monitored by the MCDP during ground test functions.
      (2) Primary Interfaces
         (a) The left, center, and right flight control computers use ARINC 429 data buses for transmitting interface fault data to the MCDP and receiving ground test data from the MCDP. Analog discrete lines are used for ground test control.
         (b) The left and right flight management computers use ARINC 429 data buses to transmit fault data to the MCDP. The MCDP can receive FMC fault data but not transmit data to the FMC. The MCDP Ground Test 30 is the only ground test that displays FMC faults.
         (c) The thrust management computer uses an ARINC 429 data bus for transmitting interface fault data to the MCDP and receiving ground test data from the MCDP. An analog discrete line is used for ground test control of the TMC by the MCDP.





Maintenance Monitor System
Figure 1



Maintenance Control Display Panel Interfacing Systems
Figure 2

126038

(3) Secondary Interfaces

(a) The mode control panel supplies control signals and receives status data from each FCC, FMC, and TMC on ARINC 429 data buses.

(b) Each FCC supplies analog control signals to its control servos and receives analog servo position signals. Each FCC receives sensor data on ARINC 429 data buses from the associated Inertial Reference Unit (IRU) and air data computer.

(c) The TMC receives control signals on an ARINC 429 data bus from the thrust mode select panel. The TMC also transmits status data to the thrust mode select panel on an ARINC 429 data bus. The TMC supplies analog control signals to and receives analog position signals from the autothrottle servomotor generator.

2. Component Details

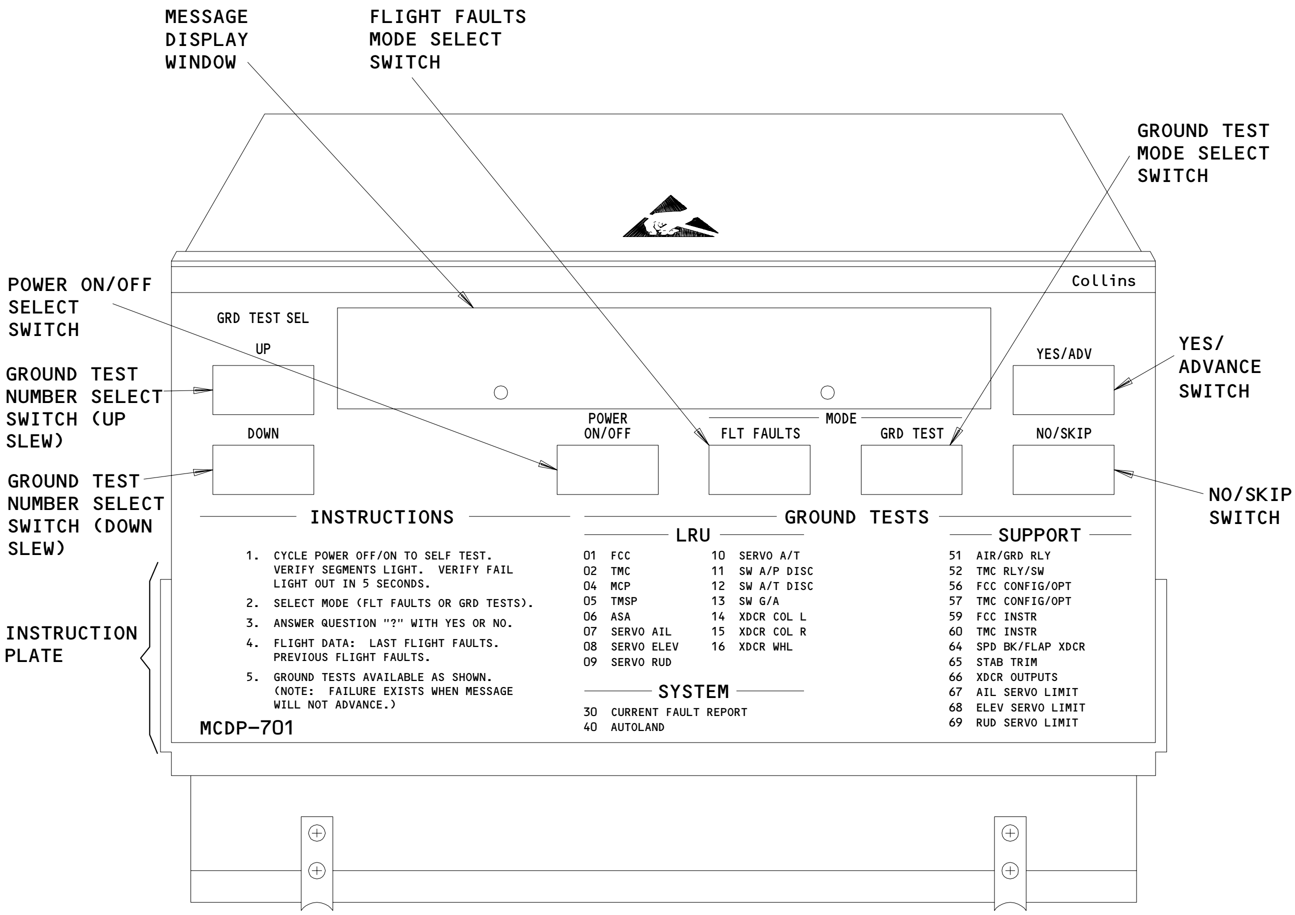
A. Maintenance Control Display Panel (Fig. 3)

(1) The MCDP automatically obtains and stores fault data after each landing from each FCC, FMC, and TMC. Maintenance personnel can then interrogate the MCDP to obtain last and previous flight fault data. They can also initiate, select, and conduct MCDP controlled ground tests.

NOTE: Do not transfer the MCDP directly from another airplane. Historical flight data is valid only for the original airplane installation. The flight data is recorded in the MCDP nonvolatile memory (EAROM) and can only be erased using shop procedures.

(2) The MCDP size is 6 MCU per ARINC 600. Weight is approximately 15 pounds. It uses bottom to top forced air cooling when installed on shelf E3-2 in the main E/E equipment center. The MCDP uses 115 vac, 400 Hz, single phase power from a circuit breaker on panel P11.

(3) The display window contains 32 characters arranged in 2 lines of 16 characters each. Each character, including decimal point, consists of 17 segmented gas discharge lamps. The display uses an English language format.



Maintenance Control Display Panel
Figure 3

(4) The two LEDs on the bottom edge of the display window indicate the Keep-Alive-Power (KAP) circuit is powered. Lights are on when there is power to the MCDP, whether the MCDP is on or off. These lights are covered, so are not very visible.

(5) Control Switches

(a) There are seven front panel control switches. Three of the switches are lighted switches with front panel replaceable lamps. All are momentary switches.

(b) The ON/OFF switch is a push ON/OFF light/switch. The switch initiates a power-up routine in the Central Processing Unit (CPU). The power-up routine initially lights the FAIL message in the ON/OFF switch. If the MCDP passes the self-test routine, the light turns off after 5 seconds.

(c) The FLT FAULTS (flight faults) switch is a push-on light/switch. The lamp lights when the function is engaged. The FLT FAULTS switch commands the CPU to enter the LAST FLIGHT FAULT ROUTINE. The FLT FAULTS MODE switch/light is also used to display the flight fault diagnostic code of the non-interface flight fault displayed.

(d) The GRD TEST (ground test) switch is a push-on light/switch. The lamp lights when the function is engaged. The GRD TEST switch commands the CPU to enter the ground test mode and initialize to the lowest available test number. The GRD TEST MODE switch/light is also used with ground test 30, CURRENT FAULT REPORT to display the ground fault diagnostic code of the non-interface ground fault displayed.

(e) The UP (slew up) switch increments the test number in a sequential loop each time the switch is pressed. Holding the switch on slews the test numbers at the rate of four test numbers per second. Unused test numbers are automatically skipped.

(f) The DOWN (slew down) switch decrements the test number in a sequential loop and slews at the same rate as the UP switch.

(g) The YES/ADV (yes/advance) switch is used to respond to messages ending with a question mark (?). Pressing the YES/ADV button in response to the question engages the displayed flight fault mode or ground test. The advance is used to advance to the next flight fault or ground test step. The operator must press YES/ADV to advance to the next test step if -ADV is displayed at end of display message whether test step passes or not. If -ADV is not displayed at end of display message and the test does not automatically advance to the next test step, then that test step has failed and the operator must push YES/ADV to continue the test.

(h) The NO/SKIP switch is to respond to messages ending with a question mark (?) when the operator does not want to display the particular flight faults or run the particular ground test. If the NO/SKIP switch is pressed in response to a question, then the MCDP will skip to the next test question. If the question is part of a sub-routine of a test then the MCDP will skip to the next sub-routine within that test if there are any. If NO/SKIP is pressed in response to a ground test message that does not end with a question mark, the MCDP will exit that ground test and display the next test listed on the front of the MCDP. The exit function of the NO/SKIP switch will not work if the message displayed is a fault message.

NOTE: The YES/ADV and NO/SKIP switch will not respond when the MCDP displays in test 40, SET R BUS TO ISOL FOR 10 SEC.

(6) The operating instruction plate is on the lower left side of the front panel. It contains the instructions for applying power and initializing the MCDP. It tells how to select each mode and respond to interrogative messages.

(7) The GROUND TEST Listing is located on the lower right and center of the front panel. It contains test numbers and corresponding names that can be selected. The ground tests are listed as LRU, SUPPORT, and SYSTEM groups.

(8) Internal Circuits

(a) Internal circuits in the MCDP consist of six functional elements on ten printed circuit boards.

(b) The power supply module consists of a power supply control card and a power supply card. The power supply converts the ac input to six dc voltages. It provides a separate unswitched (keep-alive-power) for the on/off logic circuitry. It has enough storage capacity to maintain its power output and normal activity during brief power interruptions of less than 200 milliseconds.

(c) The ARINC I/O (input/output) card receives data on six low-speed ARINC 429 buses (3 FCC, 2 FMC, 1 TMC). The card provides the buffering, level changing, and temporary storage of the input data. It transmits data on one high-speed and three low-speed ARINC buses. The high-speed bus transmits data to the EICAS display. The low-speed buses transmit data to the three flight control computers and the thrust management computer. The ARINC I/O card provides the timing and drive capability for the ARINC input and output buses.

(d) The I/O functions card provides the digital line buffers, latches, drivers, and decoders which interface discrete inputs and outputs to the microprocessor central bus. The card accepts 16 strapping inputs including four spares, two air/ground discretes, and seven panel switch inputs. It also has eight functional test outputs and three annunciator outputs.

(e) Program memory is supplied by two cards that provide up to 56K bytes of Ultra Violet Erasable PROM (UV EPROM). The EPROM contains the programming that is controlled by the executive routine in the central processing unit (CPU). The programming manages the power-up initialization, power interrupt management, mode switching, periodic functions management, and power-down routines. The mode switching routine controls the auto-on, power-up, self-test, flight faults, and ground test routines. During the power-up self-test, the MCDP does not accept flight fault data. It only accepts or displays flight faults after the a successful self-test.

(f) The processor card contains the CPU, scratch pad memory, and Electrically Alterable ROM (EAROM). The CPU's internal executive routine controls the programming in the program memory cards.

(g) The scratch pad memory is used for temporary data storage during program execution.

(h) The EAROM is used for non-volatile storage of flight faults. It can store up to ten flights with faults with each computer sending a maximum of five faults per flight.

(i) The display module consists of a display control card and a 32 character alpha-numeric gas discharge display. The display control stores the display data received from the CPU in its internal 96 x 8 bit RAM. The display control also contains the seven front panel switches and three annunciators that allow maintenance personnel to interface with the MCDP.

(j) There are two interconnect boards, rear and sideboard. The rear interconnect board routes incoming and outgoing signals between the ARINC 600 connector and the internal elements of the MCDP. The sideboard interconnect board contains the central bus. The central bus transfers the data between the internal elements of the MCDP and the front panel control board or the rear interconnect.

(9) Option pins allow a standard MCDP design to be customized by selecting applicable programming. The selecting is done by grounding (logic one) or leaving open (logic zero) the appropriate discrete option pins. These pins are located in the main connector on the back of the MCDP and are controlled with airplane wiring. Three pins are used to select the airplane configuration, one pin to select single FMC option, and another pin for dual FCC option. Five pins are reserved for customer options.

(a) Airplane Configuration Pin 1 is set to logic one and pins 2 and 3 are set to logic zero for 757 airplanes.

(b) The single FMC option pin set to logic zero indicates the dual FMC option.

(c) The dual FCC option pin set to logic zero indicates the three FCC option.

(d) Customer Option Pins:

1) Pin 1 set to logic zero indicates ADC Bus switching to the FCCs is not available on the pilots instrument select panel.

2) Pin 2 set to logic zero indicates FMC Bus switching to the FCCs is not available on the pilots instrument select panel.

3) Pin 3 set to logic zero indicates to the MCDP that the airplane has separate autopilot disconnect switches and autothrottle disconnect switches.

4) Pin 4 set to logic one enables the FMC fault reporting to the MCDP.

5) AIRPLANES WITHOUT ROTARY FMC SOURCE SELECT SWITCHES;
Pin 5 is a spare pin.

6) AIRPLANES WITH ROTARY FMC SOURCE SELECT SWITCHES;
pin 5 set to logic one indicates FMC switching on the instrument source select panel is a three position rotary switch.

7) MCDU/FMC SWITCHING (option 5) pin set to logic zero indicates FMC switching on the pilots instrument select panel is not a three position rotary switch and does not affect FMC inputs to the FCC's.

8) AIRPLANES WITH PUSHBUTTON FMC SOURCE SELECT SWITCHES; MCDU/FMC SWITCHING (option 5) pin set to logic zero indicates FMC switching on the pilots instrument select panel is not a three position rotary switch and does not affect FMC inputs to the FCC's.

(10) The MCDP is an electrostatic sensitive unit that requires special handling using authorized procedures (Ref Chapter 20).

3. Operation

A. Functional Description

(1) Maintenance Control Display Panel Schematic (Fig. 4)

(a) The CPU in the MCDP initiates an automatic power-up and self-test routine after each landing when an on-ground condition is sensed by either air/ground relay. The 5 vdc keep-alive-power on the CPU enables the power-up routine. A three minute timer is started at the same time the power-up routine is initiated. A successful self-test starts the flight faults routine. The flight faults routine requests fault data from the six computers (3 FCC, 2 FMC, 1 TMC). A failed self-test causes the CPU to power-down the MCDP. No flight fault data is transferred. The CPU normally powers-down the MCDP after all flight faults are received or after three minutes.

(b) The manual power-up and self-test routines are initiated by momentarily pressing the ON/OFF switch on the MCDP front panel. The FAIL light in the ON/OFF switch is turned on during the self-test routine. It turns off after five seconds if the self-test is successfully completed. During the self-test, a display test pattern is provide for three seconds so the operator can verify that all the switch/light lights and display light segments are working. After the FAIL light is turned off, the MCDP automatically enters FLT FAULT Mode and displays LAST FLT FAULTS?.

(c) With LAST FLT FAULTS? displayed after a successful self-test, pressing the YES/ADV switch causes a fault from the last flight to be displayed. Each time the YES/ADV switch is pressed, a different last flight fault is displayed until all last flight faults have been displayed. After all last flight faults have been displayed, the display reads ALL LAST FLT FAULTS READ. If there are no previous flight faults, then the display reads NO LAST FLT FAULTS after the YES/ADV switch is pressed the first time.

(d) With LAST FLT FAULTS? displayed after a successful self-test, pressing the NO/SKIP switch causes PREV FLT FAULT? to be displayed. Then pressing the YES/ADV switch causes the previous flight faults to be displayed. Each previous flight fault is prefaced by the associated flight number. The flight number represents the number of flights since the displayed fault occurred. Up to 99 flights can be counted, but only 10 fault flights are stored. Flight numbers are skipped if no fault is stored for that flight. The message ALL PREV FLT FAULTS DSPLY is displayed after all previous flight faults have been displayed. If there are no previous flight faults, the message NO PREV FLT FAULTS is displayed.

(e) If fault message is displayed, pressing FLT FAULTS MODE switch/light displays its fault message diagnostic code. Alternate pressing YES/ADV switch and FLT FAULTS switch/light to displays each fault message and diagnostic code. Repeated pressing of the FLT FAULTS switch/light also causes the flight faults and diagnostic codes to be alternately displayed. If interface faults are displayed, the YES/ADV switch must be pressed to display the next fault. Interface faults do not have diagnostic codes.

(f) Diagnostic code can be decoded as follows:

```
DATA SOURCE
-----------
FCC=1      |
FMC=2      |
TMC=3      |       INTERMITTENT BIT SET OR A SENSOR    1=YES
           |       --------------------
           |         |  FAULT (SSFDS) OCCURRED         0=NO
           |         |  DURING AUTOLAND AFTER APP
           |         |  SELECTED
           |         |
           X X X X X X
             | -----
             |  |
             |  DECIMAL CODE ASSIGNMENT (FIM 22-00-02 Fig.104
             |  -----------------------    FAULT ISOLATION
             |                              PROCEDURES)
             | CHANNEL
            ---------
                L=1
                R=2
                C=3

FLIGHT FAULT DIAGNOSTIC DISPLAY FORMAT
```

(g) After a successful self-test, pressing the GRD TEST switch causes self-tests to be run on all six computers (3 FCC, 2 FMC, 1 TMC) and interface check data to be collected from all valid computers. This data is stored in the MCDP for use in the ground tests. While the self-tests are being run, an IN PROGRESS message is displayed. After the self-tests are completed, the MCDP displays the first test, 01 FCC. If the test number displayed is the test desired, pressing the YES/ADV switch engages the test. If the test number displayed is not the desired test, pressing the ground test select UP switch or the NO/SKIP switch causes the test number to increase in a sequential loop. Pressing the ground test select DOWN switch causes the test number to decrease in a sequential loop. The MCDP front panel contains a list of available tests.

(h) Engaging the selected test causes the MCDP CPU to set the ground discrete line. This signals the selected computer(s) (FCCs and/or TMC) that a ground test is to be run. The selected computer checks that its servos are engaged, an in-flight flag indicates an on-ground condition, and it has no faults stored as in-flight fault data. The computers send the CPU their status and set the test required wrap-around bit if the IN-AIR bit, servos engaged, and GRD fault bits are set. The set wrap-around bit indicates to the CPU that the selected computer is ready for the ground test. The CPU then sends the selected computer the test routine number and the necessary parameters for the test. It completes the test routine and sends the results to the CPU. If there are no faults, the CPU sends the next test routine number and parameters. During the test routine, the MCDP displays XX IN PROGRESS when operator interaction is not required (XX denotes the test number being performed). If operator interaction is required, the MCDP displays the appropriate message.

(i) The test routine stops, and a fault message is displayed when a fault condition occurs. A fault during an operator response is indicated by the MCDP not advancing to the next step. Pressing the YES/ADV switch causes the CPU to continue the test. A XX TEST COMPLETE message is displayed when the test routine is completed. Pressing the YES/ADV or NO/SKIP switch causes the CPU to exit the test routine. Pressing the FLT FAULT switch interrupts the test routine that is in progress, and puts the MCDP into the in-flight mode. However, all information related to the interrupted test is lost.

(j) Diagnostic codes for ground tests are available for Test 30, CURRENT FAULT REPORT. If GROUND TEST 30 CURRENT FAULT REPORT is run, pressing GRD TEST MODE switch/light displays indicated faults diagnostic code. Alternately pressing YES/ADV switch and GRD TEST MODE switch/light displays each current fault and diagnostic code. Repeated pressing of the GRD TEST MODE switch/light also causes the current faults and diagnostic codes to be alternately displayed. The YES/ADV switch must be pressed to display the next fault, if an interface or group fault is displayed. Interface and group faults do not have diagnostic codes.

(k) Diagnostic code can be decoded as follows:

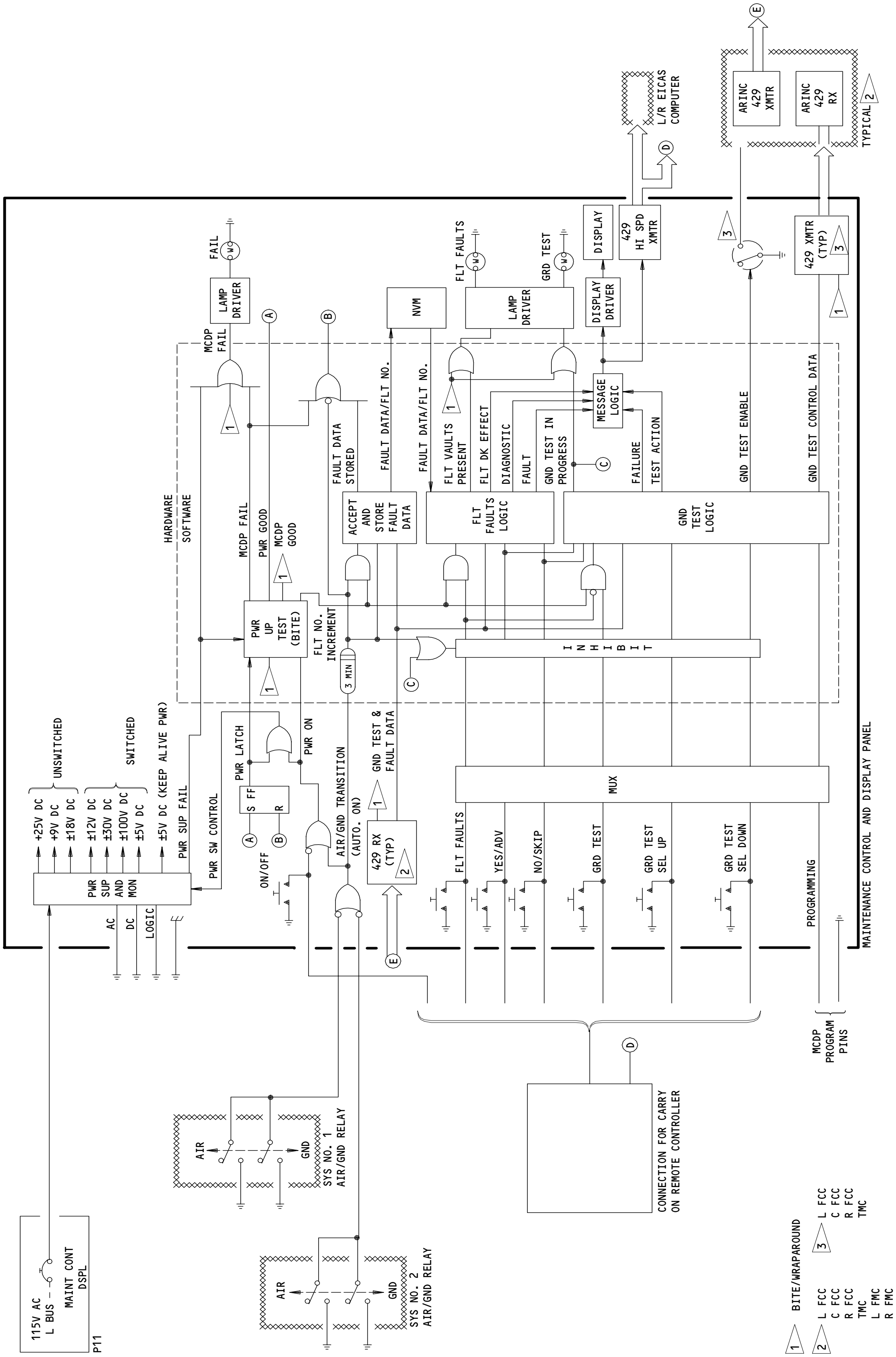
```
DATA SOURCE
-----------
FCC=1      |
FMC=2      |
TMC=3      |
          X X X X X
            | -----
            | |
            | | DECIMAL CODE ASSIGNMENT (FIM 22-00-02 Fig.104
            |   -------------------------  FAULT ISOLATION
            |                              PROCEDURES)
            | CHANNEL
           ---------
              L=1
              R=2
              C=3

GROUND FAULT DIAGNOSTIC DISPLAY FORMAT *

* NOTE:  Ground fault diagnostic display is the same as
         flight fault format except the intermittent bit is
         omitted.
```

(l) The MCDP remote control panel switches are in parallel with the switches on the MCDP. The MCDP display is duplicated on the EICAS display when the CONFIG/MCDP switch is selected on the EICAS Maintenance Panel (on the P61 panel). Operation and display using the remote control panel and EICAS display is the same as the standard MCDP control and display.

(2) MCDP Power-Up and Self-Test (Fig. 4)



Maintenance Control Display Panel Schematic
Figure 4

206097

(a) The MCDP power supply requires a 115v ± 10 vac, 400 ± 30 Hz, single phase power. It receives power from the left ac bus through a circuit breaker on panel P11-3. The MCDP has individual ac, dc, logic, and chassis grounding. The MCDP power supply is isolated from its ac source by a transformer. The ac to dc power conversion is accomplished with full wave, center tapped, rectifier circuits with noise suppression. Filter circuits limit output voltage ripple to approximately 5 percent.

(b) The dc outputs consist of four switched and three unswitched voltage sources and a separate unswitched 5 vdc Keep-Alive-Power (KAP). The 5 vdc KAP that is on whenever there is power on the airplane. The KAP is center tapped directly off the transformer and is regulated with a zener diode. The KAP enables the ON/OFF control logic, which causes full-power voltage regulators operation. All dc supply levels, except the 5 vdc, are monitored so that an out-of-tolerance voltage causes the FAIL annunciator to turn on. A second 5 vdc supply is a regulated switched supply. An unregulated 25 vdc supplies its regulator. A 25 vdc monitor detects impending power loss and generates an output interrupt signal to the CPU when unregulated voltage drops below 18 vdc. The switched 5 vdc voltage level is monitored so when the level falls below 4.75 vdc a HOLD signal is sent to the CPU. Operation of the CPU is then inhibited.

(c) Automatic Power-Up and Self-Test

1) Automatic power-up and self-test is initiated by an on-ground indication from either air/ground relay. This triggers two pulse circuits in the control logic. One pulse circuit turns on the power supply and instructs the CPU executive module to do the power-up and latch routine. The other pulse circuit starts the three minute timer and generates the AUTO ON signal to the self-test manager module.

2) The executive module (through the power-up and latch routine) checks that the regulated 5 vdc supply has a minimum level of 4.75 vdc. If the voltage is within limits the power latched-on flip-flop is set, and provides a POWER LATCHED ON signal to the self-test module.

3) The CPU reads the option pins to determine airplane configuration (757 or 767), number of FMCs and FCCs, customer options, and parity. The option pin data is stored for use in program routines.

4) The POWER LATCHED ON signal to the CPU enables the self-test routines. The AUTO ON signal selects which self-test routines are performed. When the AUTO ON SIGNAL is enabled, only the scratch pad memory check and program memory check are performed. The other five self-tests require operator interaction or cannot be accomplished when the airplane is in the air or rolling down the runway. A self-test failure causes the CPU to send a reset signal to the power latching flip-flop. The reset flip-flop signal removes the power regulators control signal. This switches the regulators off and removes operating power. A successful self-test allows the three minute timer to expire. This removes the AUTO ON signal and triggers a pulse circuit that sends a reset signal to the flip-flop after three minutes.

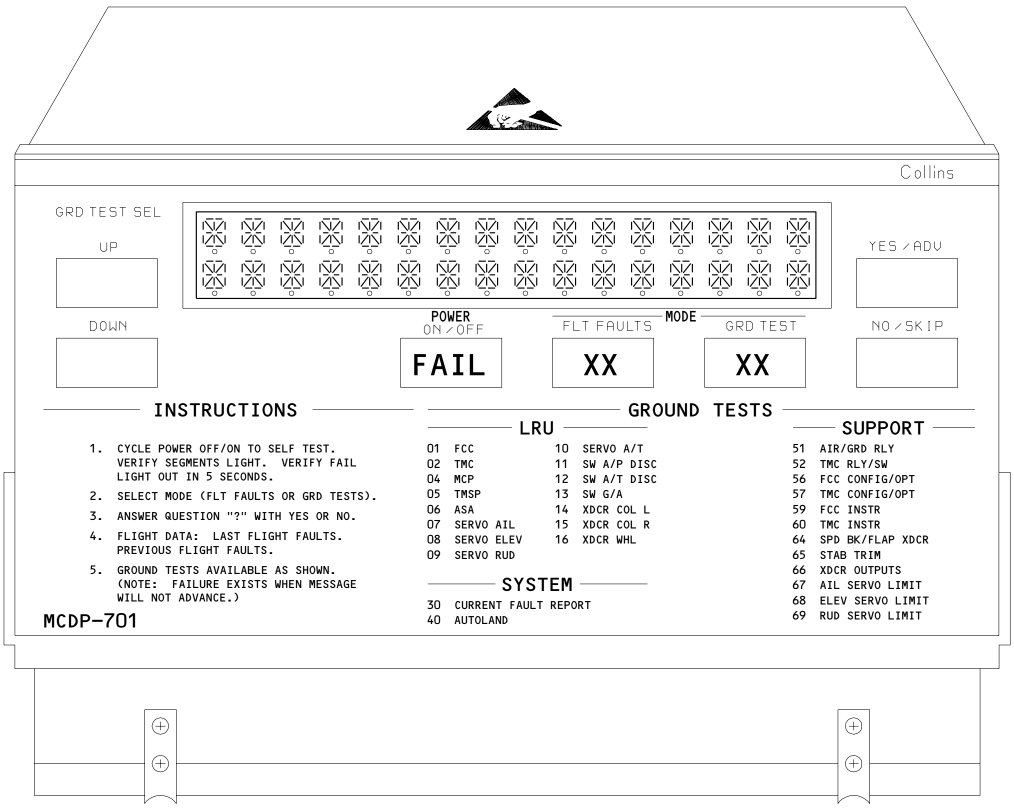
5) During an auto power-up, the displays PROM sequencer is in a standby state. This standby mode of the display reduces power consumption approximately 15 percent. During the autopower on mode, the MCDPs switches are deactivated.

(3) Manual Power-Up and Self-Test (Fig. 5)

(a) The manual power-up and self-test is initiated by pressing the ON/OFF light/switch. This sends a pulse signal to the power supply switching control and instructs the CPU to call up the power-up and latch routine.

(b) The power-up and latch routine checks that the regulated 5 vdc power level has a minimum level of 4.75 vdc. If the level is within limits, the power-up and latch routine triggers a pulse circuit that sets the power regulators latching flip-flop, and provides a POWER LATCHED ON signal to the self-test manager module.

(c) The POWER LATCHED ON signal to the CPU enables the self-test routines. The absence of an AUTO ON signal tells the CPU that a manual power-up has been done. Therefore, all of the following seven self-test functions are performed: 1) random access memory, 2) read only memory, 3) low-speed ARINC wrap-around, 4) high-speed ARINC wrap-around, 5) discrete wrap-around, 6) display, 7) lights. The self-test of the display and lights consists of turning on the annunciator lights and all segments of the 32 character display for approximately three seconds for operator verification. A successful self-test causes the CPU to enter FLIGHT FAULTS MODE. This is indicated by the FLT FAULTS annunciator lighting and a LAST FLT FAULTS ? message being displayed.



- INITIATE MANUAL POWER UP BY PRESSING ON/OFF BUTTON
- A SELF-TEST IS CONDUCTED AND THE ABOVE TEST PATTERN SHOWN
- AFTER 5 SECONDS, A SUCCESSFUL TEST IS INDICATED BY THE FAIL AND GRD TEST LIGHTS EXTINGUISH
- THE FLIGHT FAULTS MODE IS ENTERED

MCDP Manual Power Up and Self-Test
Figure 5

(d) If a fault is found during the self-tests, the CPU generates an MCDP failure signal that holds the FAIL annunciator on. The FAIL annunciator stays lighted until a manual power-off is done. In addition to the FAIL annunciator staying lighted, the MCDP displays MCDP FAIL on the top line of the display. On the bottom line, a diagnostic message for shop use is displayed. The diagnostic messages are stored on volatile memory. The message must be manually recorded prior to removing the MCDP for repair.

1) The option pins are read by the CPU to determine the airplane configuration, FMC, FCC, customer options, and parity. The option pin data is then stored for later use in the program routines.

(e) Manual Power-Down

1) A manual power-down signal is generated by pressing the ON/OFF lightswitch. This causes the CPU executive module to generate a TURN POWER OFF pulse. The pulse resets the power-on latch flip-flop to off. This enables the power-down routine which turns off the power supply switching control.

(4) Remote MCDP Operation (Fig. 6)

(a) Remote MCDP operation uses a hand held MCDP remote control panel that is connected at the forward side of the P61 panel in the flight compartment. The remote panel switches are connected in parallel with the control switches on the MCDP.

(b) The remote MCDP operation displays MCDP data on the EICAS display in the P-2 panel. The EICAS system accepts MCDP data when the CONFIG/MCDP switch is selected on the EICAS Maintenance Panel located on the P61 panel. The EICAS maintenance panel is disabled in flight through an air/ground relay.

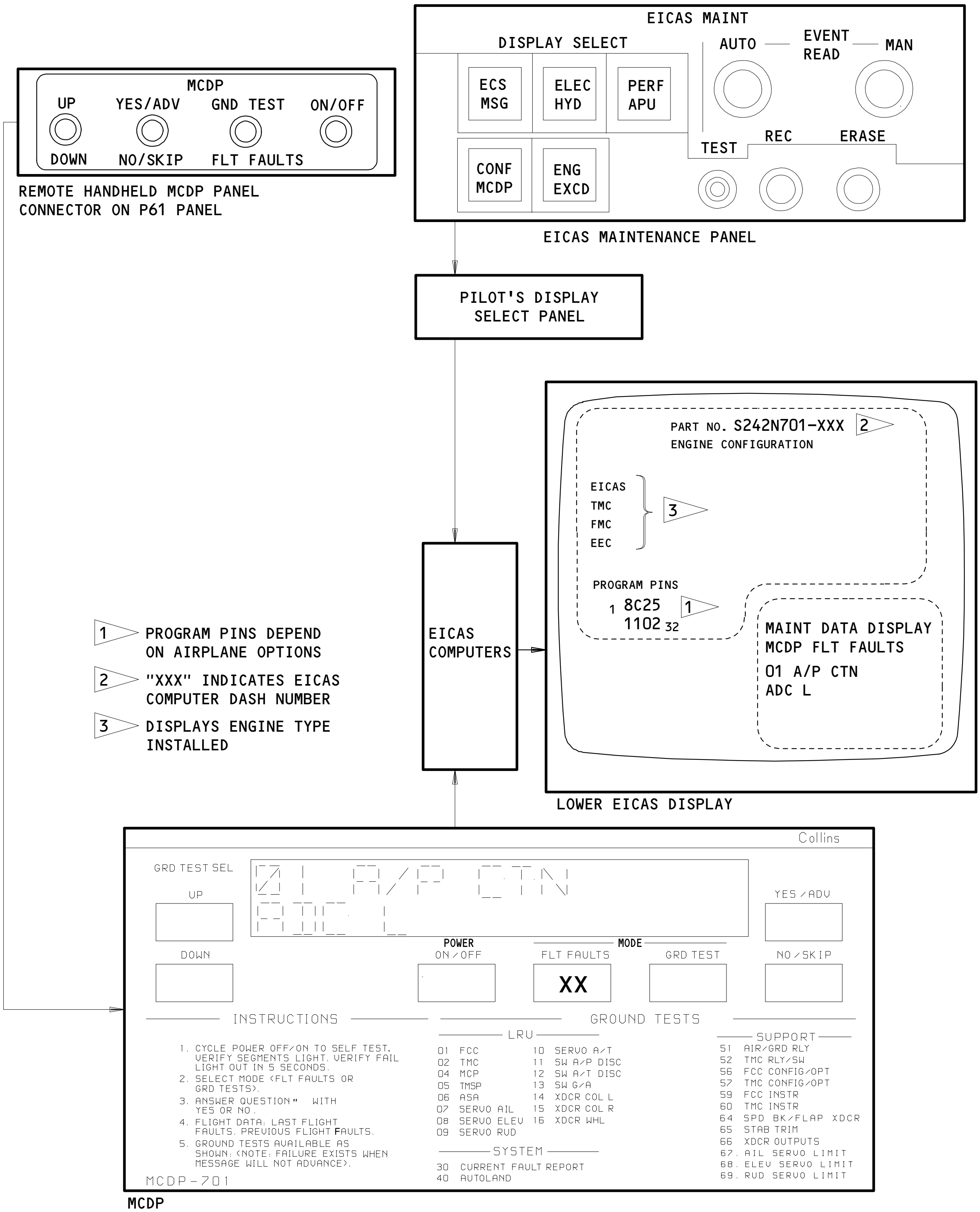
(c) The EICAS displays MCDP data the same as the MCDPs display. In addition to the normal display, the EICAS also displays the following:

1) MCDP OFF if CONFIG/MCDP switch is selected and the MCDP is off.

2) MAINT DATA DSPLY to indicate that maintenance data is being displayed. The MCDP mode message MCDP FLT FAULTS or MCDP GRD TEST is displayed below the MAINT DATA DSPLY message.

(5) MCDP Flight Faults Mode (Fig. 4)

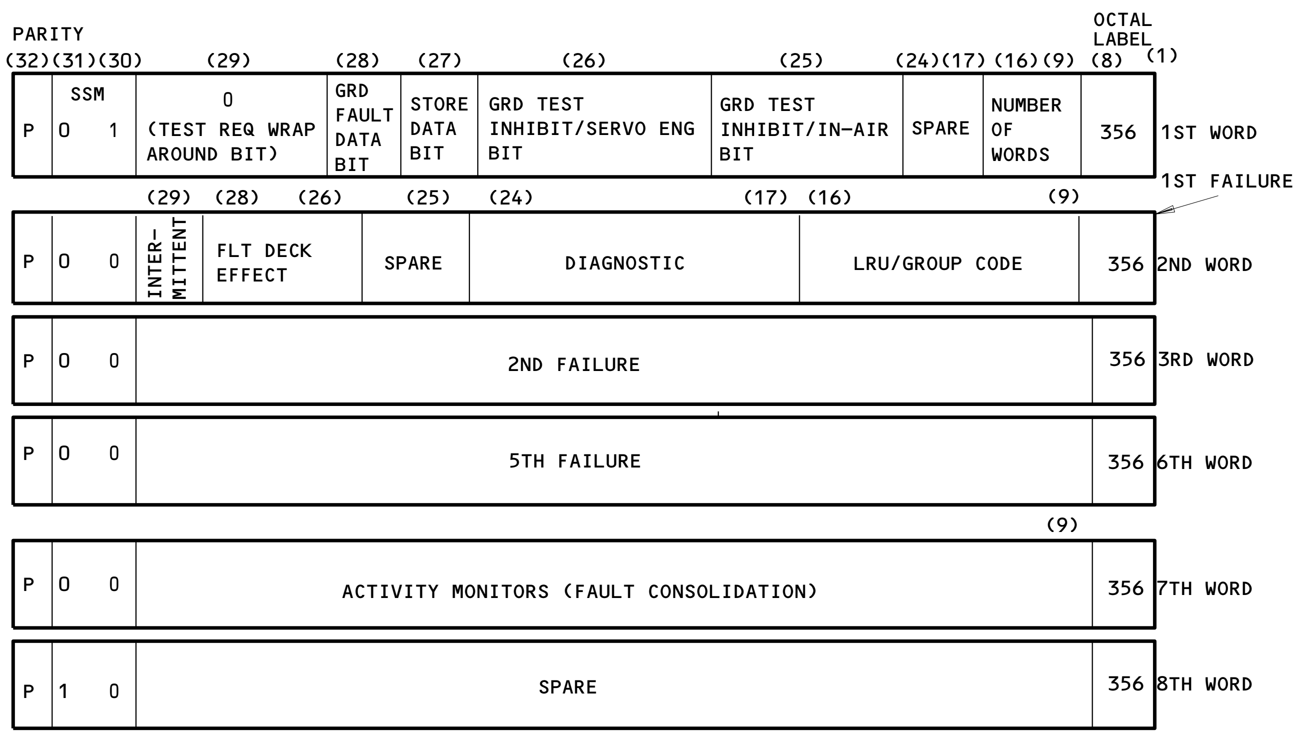
(a) The MCDP receives flight fault data from six computers (3 FCC, 2 FMC, 1 TMC). The data is received on ARINC 429 data buses. Automatic transfer of flight fault data to the MCDP occurs when the following conditions are met: 1) either air/ground relay indicates an on-ground condition, 2) a successful automatic power-up and self-test of the MCDP is completed, 3) the Inertial Reference Unit (IRU) senses a ground speed of less than 40 knots. The IRU cannot be in the functional test mode or have the in-flight flag set.



Remote MCDP Operation
Figure 6

(b) Each computer transmits flight fault data to the MCDP on an ARINC 429 data bus in a block of eight 32-bit words. The data block from each computer is initially stored in a Random Access Memory (RAM) until all valid computers have transmitted their fault data. After all the fault data is received, it is run through a fault consolidation routine and stored in non-volatile memory (EAROM). The MCDP can store up to 10 fault flight segments for a total of 350 flight faults. Each word in the data block is labeled either 356 for fault data and ground test data or 357 for interface fault data. The label 357 interface fault data is only acquired during ground test.

(6) MCDP Flight Fault Input Data Block (Fig. 7)

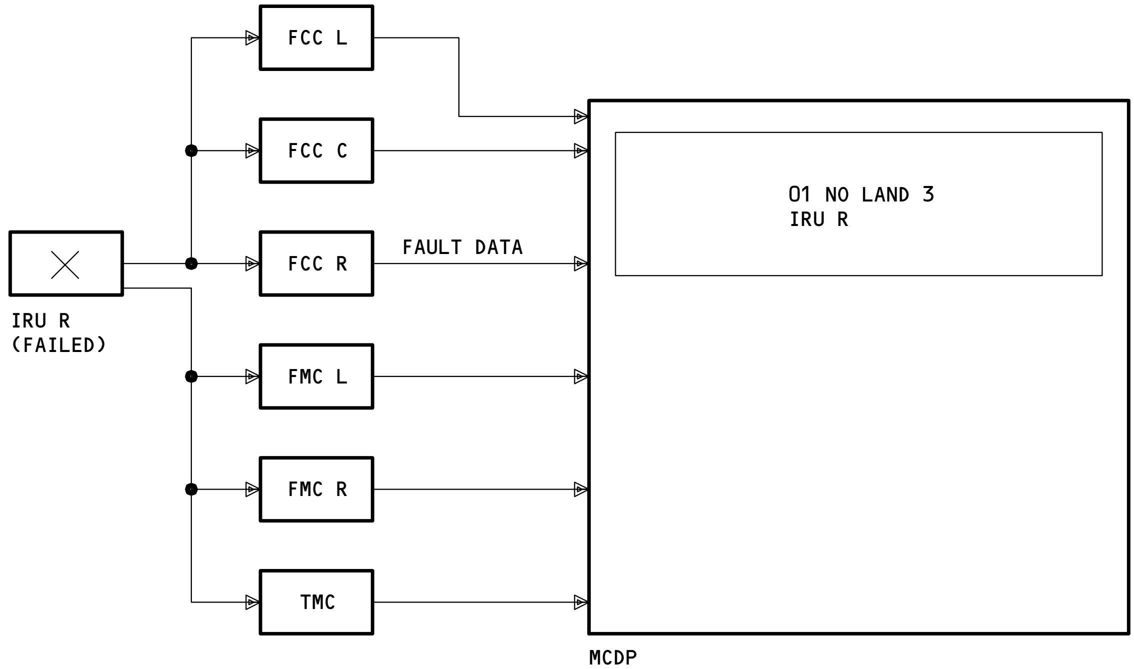


NOTE: CODES SHALL BE 2'S COMPLEMENT BINARY DATA WITH MSB ON LEFT

MCDP Flight Faults Input Data Block
Figure 7

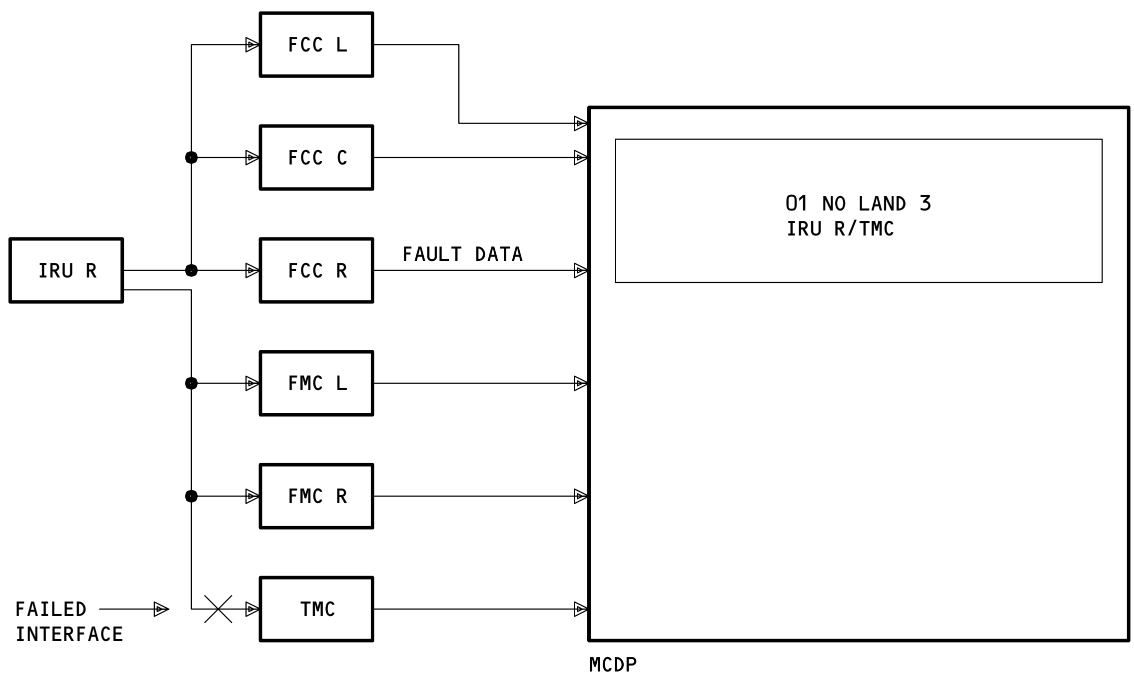
(a) The label 356 word format consists of a block of eight words. All eight words must be present for the MCDP to accept the data.
1) Word one is a general information word that informs the MCDP the computers status and what to do with the incoming data.
2) Words two through six contains one fault per word. Each word contains a LRU group code in bits 9 through 16, a diagnostic code in bits 17 through 24, a flight deck effect in bits 26 through 28 and intermittent bit 29. A total of five faults per computer per flight can be transmitted to the MCDP.
3) Word seven contains the status of all the activity monitors for fault consolidation in bits 9 through 29.
4) Word eight is not presently used.

(7) LRU Fault Consolidation (Fig. 8)
(a) The Fault Data Management (FDM) calls the AUTO ON MANAGER routine when the AUTO ON MANAGER ENGAGE logic is generated. The engage logic is generated when the following conditions are met: Automatic Power-On mode is enabled; a successful limited self-test; the AUTO ON signal is provided after the initialization of the three minute timer. The AUTO ON MANAGER routine performs three functions: a Data Input Routine, a Flight Consolidation Routine, and a Store Fault Data Routine.
(b) Data Input Routine monitors the option pins to determine the computer configuration (center FCC and right FMC). It checks for fault data from any installed computer that has the desired ARINC label and store data bit set. It continues the check for fault data until all the store data bits are set or the AUTO ON three minute timer expires.
(c) Flight Consolidation Routine determines the flight deck effect message and LRU fault message associated with faults received from the six computers. The flight deck effect code is bits 26 through 28 of ARINC label 356 words 2 through 6. The flight deck effect message displayed on line one of the MCDP display is based on priority grouping of the messages. The flight deck effect code is stored in the affected computer. The LRU group code is contained in bits 9 through 16 of ARINC label 356 words 2 through 6. The LRU message is displayed on line two of the MCDP display. The LRU message is based on 20 activity logic routines that monitor the incoming data. The activity logic routines determine when a fault is received from two or more computers. This prevents the fault from being displayed more than once per fault flight.



$(\text{FCC L} + \text{FCC C} + \text{FCC R}) \bullet (\text{FMC L} + \text{FMC R} + \text{TMC}) = \underline{\text{IRU R}}$ (SINGLE FAULT)

EXAMPLE - RIGHT IRU FAILURE



$(\overline{\text{FCC L}} + \overline{\text{FCC C}} + \overline{\text{FCC R}}) \bullet (\overline{\text{FMC L}} + \overline{\text{FMC R}} + \text{TMC}) = \underline{\text{IRU R/TMC}}$ (GROUP FAULT)

EXAMPLE - GROUP FAULT

LRU Fault Consolidation
Figure 8

(d) The MCDP consolidation routine uses twenty activity logic routines in processing the LRU/Group fault codes received from the six computers. The six computers receive sensor data on two separate buses. Each activity routine sets a combination of four logic flags which are used to isolate the fault. The logic flags are set based on which computers report the same LRU/Group fault code in words 2 through 6 and the status of the corresponding activity bits in word seven.

(e) The MCDP display a group fault when the fault cannot be isolated to a single LRU. To accurately determine a faulty LRU, the MCDP must receive the same LRU/Group fault code from two computers using separate buses from the same sensor. Therefore the MCDP displays a group fault message (i.e. IRU R/TMC). The unit on the right of the slash mark is the receiving unit. A group fault indicates a possible wiring problem. The corresponding failure word diagnostic can be displayed along with the intermittant bit.

(f) Store Fault Data Routine is used to store the flight faults in non-volatile memory (EAROM). The store routine monitors word one of the ARINC word group for STORE DATA bit to be set. When the bit is set, the fault data, LRU GROUP CODE, and flight deck effect for that computer is stored in EAROM. The store routine controls the flight segment index by updating the previous flight faults segment number to include the latest flight. The segment numbers are assigned 00 to 99. The flight segment after 99 goes to 00 and the fault data that was in segment 99 is erased. The completion of the store routine resets the power flip-flop signal. This causes a power-down routine. If the store routine does not reset the flip-flop within three minutes, the three minute timer causes a power-down.

(8) MCDP Flight Fault Display Message Format (Fig. 9)

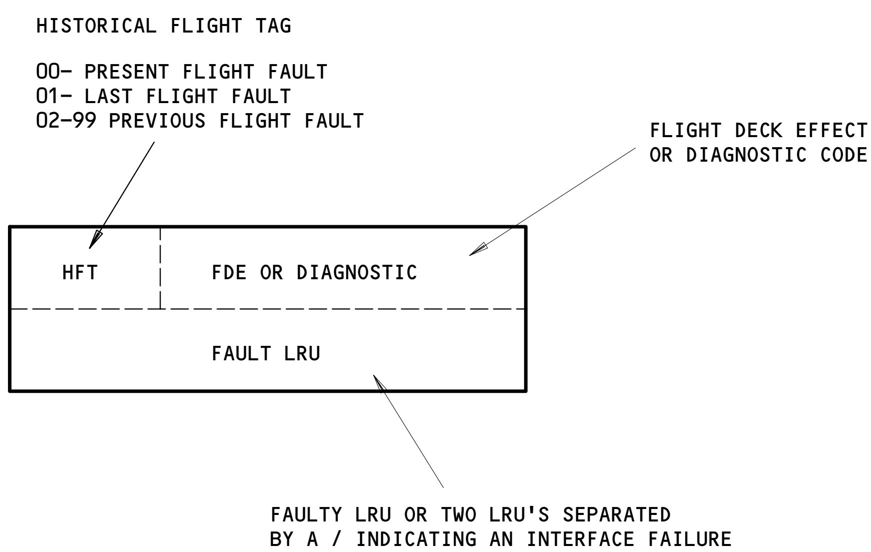
(a) The flight fault mode displays three groups of flight faults: LAST, PREVIOUS, and PRESENT. The PRESENT flight fault function is not usable because the EICAS Maintenance Panel is disabled in flight (Ref 31-41-00).

1) The historical flight tag is incremented during an auto power-on if the MCDP receives a store data bit from one or more FCCs, FMCs and TMC (FCCs channel not engaged). The store data bit is transmitted to the MCDP by the computers when the following sequence of events occur:

a) IRS Ground Speed is greater than 100 knots.

b) Air/ground relay equals IN AIR
c) IRS Ground Speed is less than 40 knots.
d) Air/ground relay equals ON GROUND

(b) The flight fault review manager routine controls the flight fault display. The executive routine enables the review manager routine when it receives the FLT FAULT REVIEW ENABLE signal. The signal is generated after a manual power-up or flight fault mode is selected while in the ground test mode. The flight fault review enable is held high when the following conditions are present: MCDP power OK; ground test not required; ground test not enabled; FLT FAULTS required selected, or power latched ON flip-flop set; and auto ON not engaged. The flight fault mode is disabled when the power is turned off or the ground test mode is selected.



MCDP Flight Fault Display Message Format
Figure 9

(c) Last Flight Faults

1) After the flight fault review routine is initialized, the FLT FAULT light/switch is turned on and the message LAST FLT FAULT? is displayed. The operator pushes the YES/ADV button to select the sub-mode being offered. Pushing the NO/SKIP pushbutton causes the MCDP to exit the LAST FLT FAULT routine and enter the PREVIOUS FLT FAULT routine.

2) Before displaying the last flight faults, the routine checks all ARINC 429 ports for valid interface data. The MCDP displays NO INFC + computer and position for each inactive port. Each computer interface fault message is displayed by pushing the YES/ADV button to progress to the next fault.

3) After all interface faults have been displayed, the Last Flight Faults of the valid computers are displayed. The YES/ADV pushbutton is used to review them. The message ALL LAST FLIGHT FAULTS DSPLY is displayed after all faults have been displayed. Exiting the Last Flight Fault mode, before the ALL LAST FLIGHT FAULTS DSPLY message is displayed, is done by pushing the NO/SKIP pushbutton. Pushing the NO/SKIP or YES/ADV pushbutton when the ALL LAST FLT FAULTS DSPLY message is displayed causes the routine to advance to the Previous Flight Fault routine. The GRD TEST switch selects the ground test mode only when the aircraft is on the ground. The ON/OFF pushbutton causes the MCDP to power-down.

4) The Last Flight Fault messages have the flight segment 01 preceding the flight deck effect message on the first line of the display. The flight fault is displayed on line two.

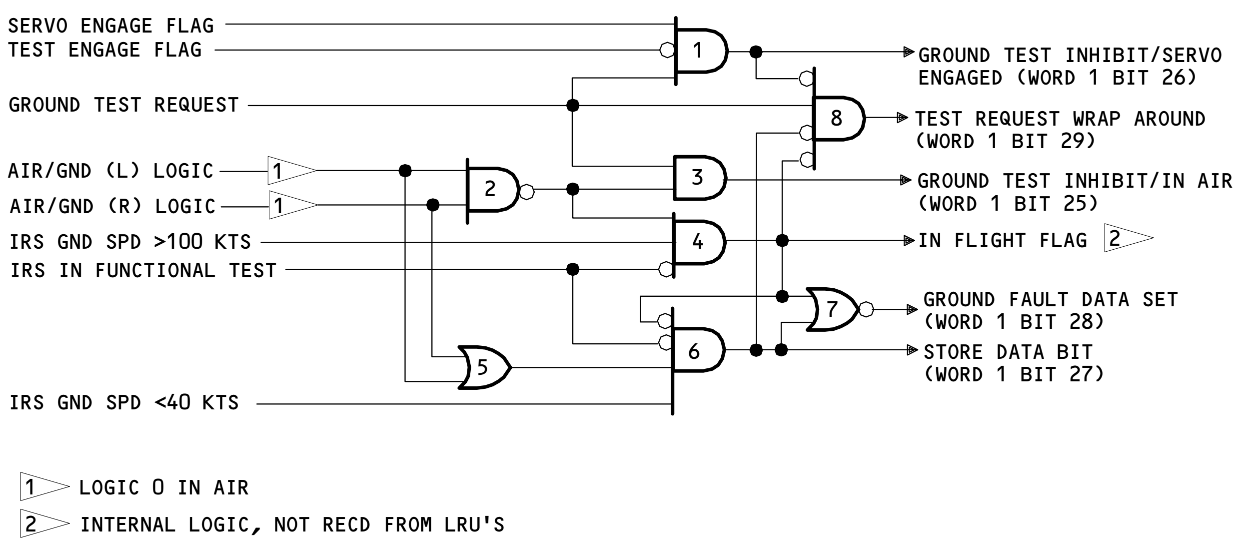
(d) Display Previous Flight Faults

1) The operation of the Previous Flight Fault routine is similar to the Last Flight Fault routine with the exception that the flight segment number ranges from 02 to 99. The flight segment number indicates how many flights ago the fault occurred.

2) The MCDP displays NO PRE FLT FAULTS message when there are no previous flight faults within the 02 to 99 flight segments. When all the previous flight faults have been displayed, the MCDP displays ALL PRE FLT FAULTS DSPLY. While the MCDP can store faults for up to 99 flight segments, it can only store the last 10 fault flights (flights with recorded faults).

(9) Flight Fault Control Logic (Fig. 10)

(a) The store data bit (word 1, bit 27 of the fault data block format) must be set to logic one before the MCDP stores fault data in non-volatile memory. The store data bit is set when: the inflight flag set to logic zero, the IRU is not in functional test mode, at least one air/ground relay indicating an on ground condition, and an IRU is indicating a ground speed of less than 40 knots.



Flight Fault Control Logic
Figure 10

(b) A test request wrap-around bit is used to check the interfaces between the MCDP and FCC, FMC, and TMC. To set the test request wrap-around bit to logic one (word 1, bit 29 of the fault data block format), the following conditions must be present: autopilot servos not engaged; ground test request analog discrete signals is received by the FCC, FMC, or TMC-- the store data bit is set to logic zero; and the inflight flag set is to logic zero. An interface fault is a fault that disables the interconnecting bus of two or more units.

(10) Flight Deck Effect Message Priorities (Fig. 11)

PRIORITY	FCC	TMC	FMC
0	SPARE	SPARE	SPARE
1	A/P DISC	A/T DISC	FMC FAIL LT
2	A/P CTN	SPARE	FMC CTN LT
3	NO LAND 3	TMC DSPLY	CDU DSPLY
4	NO A/L	SPARE	SPARE
5	F/D BAR		SPARE
6	SPARE		
7		F/S DSPLY	

Flight Deck Effect Message Priorities
Figure 11

(a) The flight deck effect messages have a priority code from one to five for each FCC, FMC, and TMC. Priority code one has the highest priority. The computers also have a priority order. The FCC has the highest priority; the FMC the lowest. The autopilot disconnect (A/P DISC) for an FCC has the highest priority and the flight director bar (F/D BAR) the lowest. Flight faults are listed in order of priority (highest to lowest). The FCC flight faults and flight deck effects are listed first, the TMC second, and FMC last.

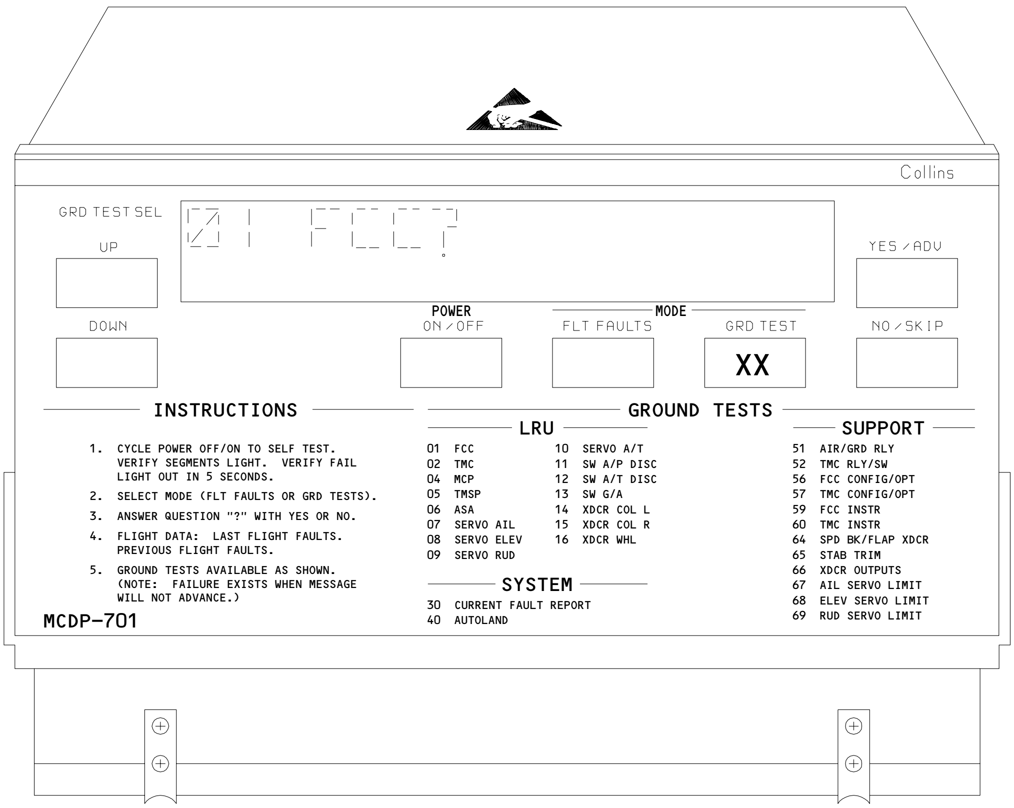
(11) Ground Test Mode (Fig. 12)

(a) The ground test mode augments the flight fault capability by providing tests to further isolate and verify flight faults. The ground test mode also verifies system operation after maintenance. The three ground test groups (LRU tests, system tests, and support tests) complete tests in the Autopilot/Flight Director System (AFDS) and Thrust Management System (TMS). The LRU tests provide tests of individual LRU's and require operator interaction. The system tests are used to test large sections of the AFDS and TMS at one time. The Autoland System test is both automatic and manual. The Current Fault Report System test is completely automatic. The support tests are used to test only functions that interface with the AFDS and TMS. For Ground tests selected that are not for the airplanes configuration, XX NO TEST THIS A/C CONFIG (XX = Test Number) is displayed.

(b) LRU Tests

1) The LRU tests provide a test armed interrogative message (e.g., 01 FCC TEST?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all prerequisites are available and automatically monitors the interfaces. Any missing prerequisite or interface fault is displayed.

2) Any missing prerequisite or interface fault ends the test. If all prerequisites and interfaces are present, the test executes the appropriate automatic and manual tests. Automatic tests being run are indicated by an IN PROGRESS message. A fault message is indicated for each invalid test. Manual tests provide action messages, verify responses, and display failures. At completion of the test, a TEST COMPLETE message is displayed. The test is exited when the YES/ADV or NO/SKIP switch is pressed.



Ground Tests Mode
Figure 12

(c) System Tests

1) The system tests provide a test armed interrogative message (e.g., 30 CURRENT FAULT REPORT?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all test prerequisites are available. Any missing prerequisite is displayed, and the test ends. The system test automatically arms and engages the appropriate LRU and support tests. A TEST COMPLETE message is displayed after all system tests have been completed and ground faults displayed. The test is exited when the YES/ADV or NO/SKIP switch is pressed.

(d) Support Tests

1) The support tests provide a test armed interrogative message (e.g., 52 TMC RLY/SW TEST?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all test prerequisites are available. Any missing prerequisite is displayed and the test ends. A TEST COMPLETE message is displayed after completion of all operator interaction tests. The test is exited when the YES/ADV or NO/SKIP switch is pressed.

(12) MCDP Ground Test Mode (Fig. 4)

(a) The ground test mode assists in isolating recorded flight faults to specific components, and is used to verify system operation after the removal and installation of components. Ground tests require that all the computers and interfacing systems required for a specific test must be installed before the test can be done. Electrical and hydraulic power sources must be supplied to the required components. Ground test mode can be entered anytime after a manual power up, the FAIL annunciator light is turned off, and both AIR/GRD relays indicating on-ground. Two to three operators may be required for certain ground tests interaction.

(b) Ground Test Entry

1) The ground test mode is entered by pressing the GRD TEST light switch. The closed switch connects a ground that triggers a pulse circuit which generates an interrupt signal to the executive routine (data management). The data management executive routine generates GRD TEST REQ logic that latches GND TEST ENABLE logic when the following conditions are met: Both AIR/GRD relays sense an on-ground condition; MCDP power circuits are operating properly; MCDP is not in automatic-on mode; flight faults are not requested; and ground test is not already engaged. The ground test enabled latch is removed when the flight faults mode is selected; power is removed; or either AIR/GRD relay indicates IN-AIR (relay position does not apply if GRD TEST 51 is enabled).

2) The ground test executive routine turns on the GRD TEST lightswitch and removes existing display message during initialization. The ground test executive routine calls up a ground-test engagement check routine which generates analog ground-test command discretes to all the FCCs and the TMC. During the ground-test engagement check routine, the message 00 SYST ENG VFY// IN PROGRESS is displayed. There is a two second delay after the ground test command is sent. This allows the computers to initialize into their ground-test modes and send ground-test data to the MCDP. The ground test command routine verifies each computer for a correct ground-test engagement response by checking bits 25, 26, 29, and 32 of word one of label 356 data block.
3) Each of the four computers does the following tests upon receiving the analog ground test discrete bit. They check the AIR/GRD sensors for an on-ground condition and if on-ground, sets TEST REQUEST WRAP-AROUND bit 29 to one. A valid response from the computer sets the ARINC INPUT ERROR flag to zero and TEST REQUEST WRAP-AROUND bit 29 to one. An invalid response from the computer causes the MCDP to display a fault message. The fault message is determined by the status of the ARINC INPUT ERROR flag and digital discrete bits 25, 26, 29, and 32. Refer to GROUND TEST ENGAGE FAULT MESSAGE MATRIX graph (Fig. 10) for details.
4) After the completion of the engagement check, the ground-test executive routine calls up the computer self-test routine. While this routine is run, the message 00 COMP TEST// IN PROGRESS is displayed. The computer self-test routine commands computers that passed the engagement check, to execute their built-in self-tests. Two test commands are sent to the FCC's. The first initiates the FCC self-test and the other initiates an interface check between the FCC and its sensors. The TMC has three test commands sent to it. The first test initiates the TMC self-test, the second initiates interface checks between the TMC and its sensor. The third test initiates an interface check between the TMC and the autothrottle servomotor generator. The MCDP computer self-test routine checks the response data from each computer's AIM word label 356. Word one must contain the test routine number (bits 17-24). Word two must have test valid (bit 29) set to one. Self-tests are automatically terminated after five seconds.

5) The completion of computer self-tests causes the microprocessor to collect and store word label 357 fault data for later use during ground testing. The label 357 word format is in an eight word block. Word one contains the number of words to follow. Words two through eight contain the discrete interface data with 21 discretes per word.

6) The interface faults are displayed by the MCDP according to the GROUND TEST ENGAGE FAULT MESSAGE MATRIX (Ref Fig. 10). During the self-test routine, the MCDP displays the messge 00 INFO DATA IN// IN PROGRESS. After all interface data has been stored and associated fault data messages displayed, the ground test routine transfers control of the microprocessor to the test select routine. The transfer is indicated when 01 FCC TEST? is displayed.

(c) Test Selection Process

1) The operator selects the desired test from the list on the instruction panel by pressing the UP or DOWN ground test select switches (incremental or fast slewing). The NO/SKIP switch increments the test selections up. During the selection routine, each test is displayed with a question mark. The operator presses the YES/ADV switch to engage the test displayed. The engaged test causes the test select routine to display any required computers that failed the ground entry checks and display interface faults for the functioning computers.

2) If at least one of the required computers and its interface is valid, the test number select routine for the front panel test is called. This loads the test routine number and test parameters into the MCDP output data block. The data block is transmitted to the test computer on the ARINC 429 data bus. The test computer responds by setting the test routine number in progress bit, and system status bits in word 356 (Ref to Fig. 12, MCDP Input Test Data Block).

(13) Ground Test Engage Fault Message Matrix (Fig. 13)

(a) The ground test engage executive routine checks the status of the ARINC INPUT ERROR FLAG, WRAP-AROUND bit, and SERVO ON bit. This determines whether there is a test engage fault. The ground test is engaged when wrap-around bit 29 is set to one and the ARINC INPUT ERROR flag is set to zero. The ARINC INPUT ERROR flag is set to one if any of the following fault conditions are present:
1) No activity is on the bus for a specific label.
2) The specific label is received but not the data block.
3) The word parity is incorrect.

ARINC INPUT ERROR FLAG	DIGITAL DISCRETE			FAULT MESSAGE
	W/A BIT 29	IN AIR BIT 25	SERVO ON BIT 26	
1	X	X	X	NO INFC (*) // MCDP
0	0	1	X	IN AIR-NO GRD TEST (*)
0	0	0	1	SERVO ON-NO GRD TEST (*)
0	0	0	0	MCDP / (*) TEST CMD FAIL
0	1	0	0	NO FAULT MESSAGE

* DENOTES AFFECTED COMPUTER

X INDICATES VALUE OF BIT NOT USED IN CONSIDERATION TO DISPLAY FAULT MESSAGE

Ground Test Engage Fault Message Matrix
Figure 13

4) The number of words is incorrect (more or less than eight).

(b) If the error flag is set to one, the fault message NO INFC (computer name)// MCDP is displayed.

(c) Ground test inhibit, in-air bit 25 (label 356) is set to one when the computer receives an in-air indication from the air/ground relays. The fault message IN AIR-NO GRD TEST (computer name) is displayed, provided ARINC INPUT ERROR flag bit 29 and IN AIR bit 25 are set to zero.

(d) Ground test inhibit, servo engaged bit 26 (label 356) is set to one when the analog ground test command discrete is generated due to an engaged autopilot control servo. Fault message SRVO ON-NO GRD TEST (computer name) is displayed, provided the ARINC INPUT ERROR flag, WARP-AROUND bit 29, and IN AIR bit 25 are set to zero. Fault message MCDP/(computer name) TEST CMD FAIL is displayed when the ARINC INPUT ERROR flag and all three digital DISCRETE bits are set to ZERO. This is an indication that the analog ground test command discrete was not received by the affected computer.

(14) MCDP Output Test Data Block (Fig. 14)

(32)	(31)(30)	5 BITS (29) (25)	8 BITS (24) (17)	8 BITS (16) (9)	FCC LABEL 8 BITS (8) (1)	TMC LABEL 8 BITS (8) (1)	
P	SSM	(P2) RESERVED	SIGN | PARAMETER P1	TEST ROUTINE NUMBER	OCTAL L 001 R 005 C 011	015	WORD 1
P	SSM	SIGN | (P5)	SIGN | (P4)	SIGN | (P3)	L 002 R 006 C 012	016	WORD 2
P	SSM	SIGN | (P8)	SIGN | (P7)	SIGN | (P6)	L 003 R 007 C 013	017	WORD 3
P	SSM	SIGN | (P11)	SIGN | (P10)	SIGN | (P9)	L 004 R 010 C 014	020	WORD 4

NOTE: PARAMETERS P1 THRU P9 & TEST ROUTINE NUMBER ARE BINARY DATA WITH MSB ON LEFT. SIGN BIT IS THE LEFT MOST BIT IN EACH PARAMETER.

MCDP Output Test Data Block
Figure 14

(a) The MCDP output test data block informs the FCC's and TMC which test routine to perform and what parameters to use. Test routine numbers are stored in MCDP CPU memory. Each test number is associated with a selected ground test. The test routine number is assigned to bits 9 through 16 of word one of each output test data block. The data block structure consists of four 32-bit words for each FCC and TMC. Each FCC and TMC is assigned a non-ARINC label to bits 1 through 8 for each word address in its data block as follows:

1) 001, 002, 003, 004 address left FCC
2) 005, 006, 007, 010 address right FCC
3) 011, 012, 013, 014 address center FCC
4) 015, 016, 017, 020 address TMC

(b) The test routine numbers and Parameters (P) are binary coded with the Most Significant Bit (MSB) on the left and the sign codes are word bits 16, 24, and 29. The output data block uses the standard AIM Sign/Status Matrix (SSM) code. Test parameter bits and word assignments are as follows:

1) P1, bits 17 through 24 of word 1
2) P2, bits 25 through 29 of word 1 (reserved)
3) P3, bits 9 through 16 of word 2
4) P4, bits 17 through 24 of word 2
5) P5, bits 25 through 29 of word 2
6) P6, bits 9 through 16 of word 3
7) P7, bits 17 through 24 of word 3
8) P8, bits 25 through 29 of word 3
9) P9, bits 9 through 16 of word 4
10) P10, bits 17 through 24 of word 4
11) P11, bits 25 through 29 of word 4

(15) MCDP Input Test Data Block (Fig. 15)

(a) The MCDP INPUT test data block sends the test results back to the MCDP. The data block consists of eight 32-bit words with each word containing ARINC label 356 in word bits 1 through 8. The SSM code in bits 30 and 31 uses standard AIM data coding. Bit 32 checks parity. The specific bit definitions for each word are as follows:

1) Word one - bits 9 through 16 specify the number of words (always 8) in the data block. Bits 17 through 24 contains the test routine numbers from the MCDP. Bits 25 through 28 are spares. Bit 29 contains the test request wrap-around used during ground test entry functions.

2) Word Two - bits 9 through 27 contain the ground test comparison results. The results are discretes that indicate whether the following are within tolerance; left and right column null; left column forward and aft; right column forward and aft; control wheel null; left and right control wheel rotation; flap 0, 10, and 30 degree position, elevator detent. Bit 28 contains the test in progress status. Bit 29 contains the test valid status.

3) Word Three - Bits 9 through 27 contain Binary Numerical Representation (BNR) data. The BNR data indicates position-data in response to the test number sent in word two. Bits 25 through 29 are spares.

4) Words Four through Seven - Bits 9 through 29 are discrete data defining the test response conditions.

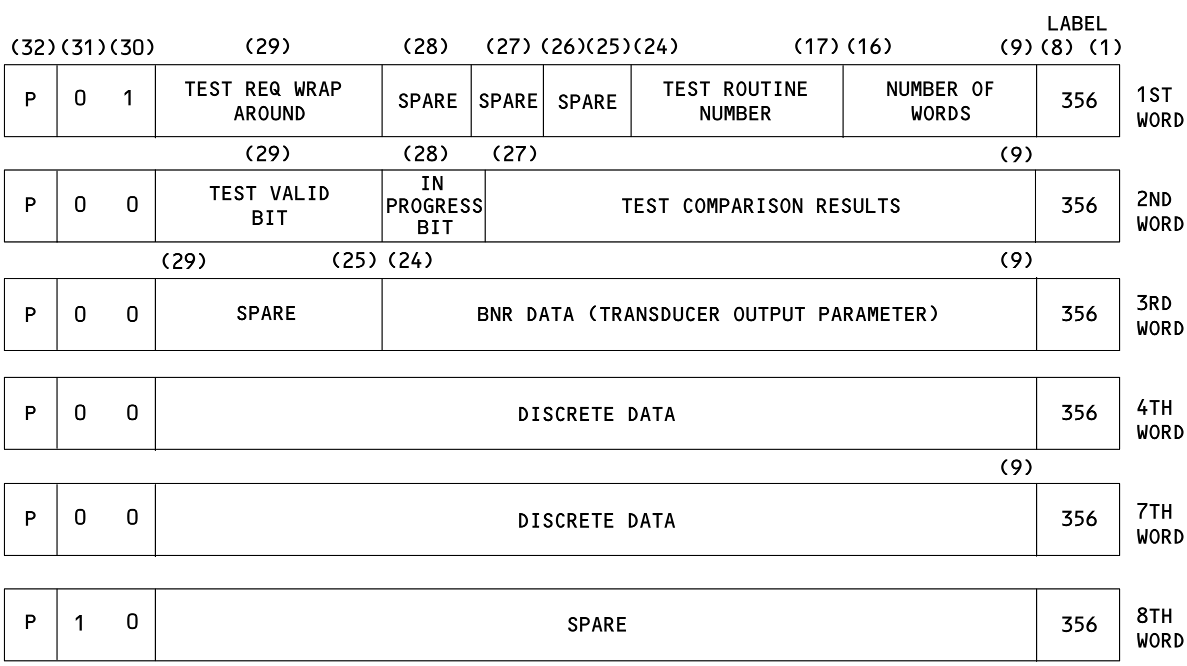
5) Word Eight - Bits 9 through 29 are spares.

(16) Ground Test Description (Fig. 16)

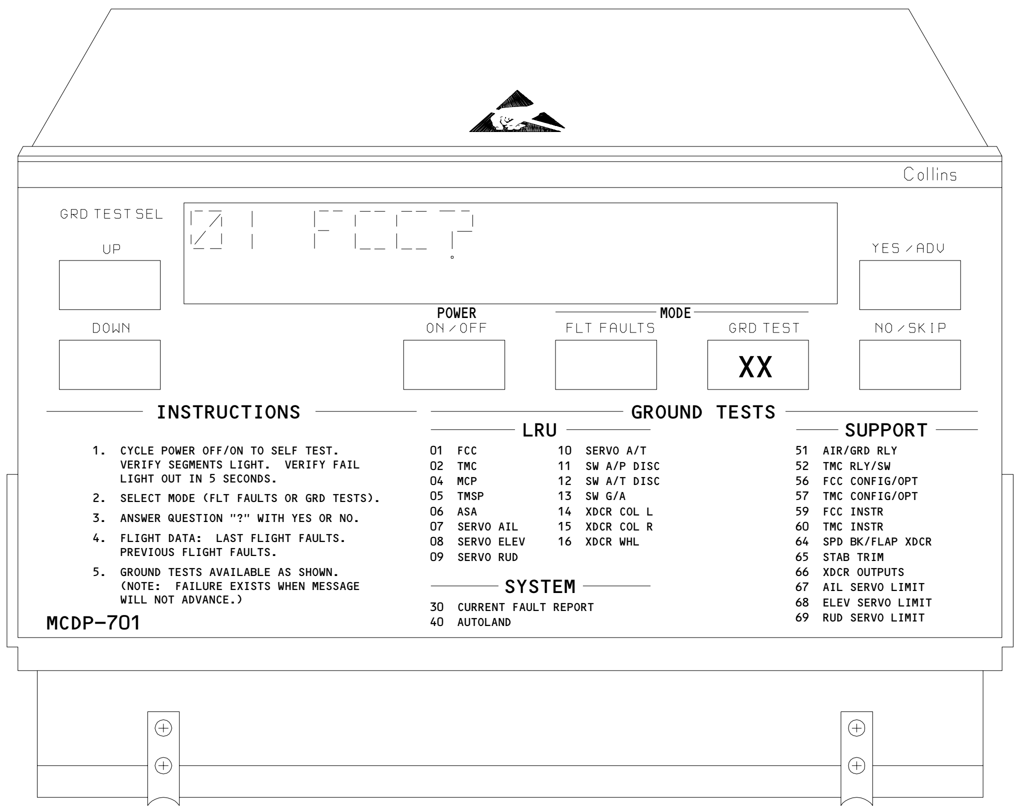
(a) LRU Tests Description

1) 01 FCC (Left, Right, and Center)

a) Displays any FCC self-test faults and FCC interface faults from the associated system interfaces.



MCDP Input Test Data Block
Figure 15



Ground Tests Description
Figure 16

2) 02 TMC
   a) Displays any TMC self-test faults and TMC interface faults from associated system interfaces.
3) 04 MCP
   a) Displays any FCC and TMC self-test faults. Displays FCC and TMC interface faults related to MCP.

      NOTE: In later MCP test steps failure of FCC L and FCC C, or FCC R will cause half lighting of lower half of MCP switch/light indicators. Failure of TMC will prevent EPR and SPD switch/lights from lighting.

   b) Displays the interaction message for checking the lower lights in the MCP lightswitches.
   c) Displays interaction messages for checking the on and off position of the captain's flight director switch.
   d) Displays interaction messages for checking the arm and off position of the autothrottle ARM switch.
   e) Displays the interaction message for checking the command function of the indicated airspeed/mach select switch.
   f) Displays the interaction message for checking the command function of heading select switch.
   g) Displays the interaction message for checking the ON and OFF positions of the first officer's flight director switch.
   h) GUI 115;
      displays interaction messages for checking the engaged and disengaged position of the bat handle switches.
   i) GUI 001-114, 116-999;
      displays interaction messages for checking the down and up position of the disengage bar.
   j) Displays interaction messages for checking the command function for each position of the bank limit switch.
   k) Displays interaction messages for checking the control function of the indicated airpseed/mach display rotary control.
   l) Displays interaction messages for checking the control function of heading display rotary control.
   m) Displays interaction messages for checking the control function of the vertical speed display rotary control.
   n) Displays interaction messages for checking the control functions of the altitude select display rotary control.
4) 05 TMSP
   a) Displays TMC self-test faults.
   b) Displays TMSP-to-TMC interface faults

c) Does a TMSP SW/MODE DISPLY test routine which checks the mode display and command function of continuous, climb, and cruise mode switching. Displays an interactive message at end of the test.
d) Does a TMSP TEMP SELECT test with interaction messages to check control function of temperature select control display.

5) 06 ASA
a) Displays any FCC self-test faults related to ASA.
b) Displays FCC interface faults related to ASA.
c) Checks all ASA displays and their reset function.

6) 07 SERVO AIL
a) Displays any FCC self-test faults and FCC interface faults related to aileron servos.
b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
c) Provides interaction message to verify aileron trim set to zero.
d) GUI 115;
provide interaction message to position MCP CMD bat handle switches up to CMD. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
e) GUI 001-114, 116-999;
provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
f) Wheel centers, then moves clockwise and then counterclockwise and back to center. Inboard ailerons are driven to a plus and minus 10 degree position. FCC's check that each position is within limits.

7) 08 SERVO ELEV
a) Displays any FCC self-test faults and FCC interface faults related to elevator servo.
b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
c) Provides interaction message to verify elevator trim set to zero.
d) Displays elevator feel position pressure for operator to verify pitot pressure is zero.

e) GUI 115;
provide interaction message to position MCP CMD bat handle switches up to CMD. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.

f) GUI 001-114, 116-999;
provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.

g) Column centers, then moves aft (trailing edge up), then forward (trailing edge down) and back to center. Elevators are driven to a plus 20 degrees (trailing edge up) and a minus 9 degrees (trailing edge down) and the FCCs check that each position is within limits.

8) 09 SERVO RUD
   a) Displays any FCC self-test faults and FCC interface faults related to rudder servo.
   b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
   c) Provides interaction message to verify rudder trim set to zero.
   d) Rudder centers, then moves rudder trailing edge right, then left and back to center. Rudder driven to a plus and minus 20 degrees and FCCs check that each position is within limits.

9) 10 SERVO A/T
   a) Displays TMC self-test faults and TMC interface faults related to autothrottle.
   b) Throttle is automatically initialized to aft position, then is driven at 10 degrees/second and checks for a Power Lever Angle (PLA) of 44 to 60 degrees. The test fails if a PLA of 55 degrees is not sensed within six seconds. The autothrottle is then driven forward for eight seconds at 10 degrees/second and checked for a PLA of 120 to 136 degrees, and a tachometer signal feedback of approximately 10 degrees/second. The autothrottle is driven aft for eight seconds at 10 degrees/second, and checked for a PLA of 44 to 60 degrees, and a tachometer feedback signal of approximately 10 degrees/second.

10) 11 SW A/P DISC
    a) Displays any FCC self-test faults, and FCC interface faults related to A/P Disc switch.
    b) Instructs the operator to close the captain's and first officer's autopilot disconnect switches. The switch output is monitored for autopilot disconnect command response.
11) 12 SW A/T DISC
    a) Displays any TMC self-test faults and TMC interface faults related to A/T disconnect switch.
    b) Instructs operator to close the left and right autothrottle disconnect switches. Switch output is monitored for autothrottle disconnect command response.
12) 13 SW G/A
    a) Displays any FCC and TMC self-test faults, FCC and TMC interface faults related to G/A switch.
    b) Instructs the operator to close the captain's and first officer's Go-Around (G/A) switches. The G/A switch command response is monitored at each FCC and TMC.
13) 14 XDCR COL L
    a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 14 NO TEST THIS A/C CONFIG.
14) 15 XDCR COL R
    a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 15 NO TEST THIS A/C CONFIG.
15) 16 XDCR WHL
    a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 16 NO TEST THIS A/C CONFIG.
16) 17 PVDC
    a) This test does not apply to this airplane configuration. If test selected, MCDP displays 17 NO TEST THIS A/C CONFIG.

(b) System Tests Description
1) 30 CURRENT FAULT REPORT
a) Displays all LRU and interface faults detected and stored in the FCCs and TMC while the MCDP operates in the ground test mode. Ground fault messages and diagnostic codes are the same as flight fault messages except the flight deck effect and intermittent bit status are not displayed.
b) Provides interaction message to allow operator to skip Interface Fault messages.
c) Provides interaction message to allow operator to select current Ground Fault display.
2) 40 AUTOLAND
a) Displays the current ground faults and interface faults related to the autoland system. The autoland related interfaces are automatically monitored. It automatically performs LRU and support tests related to the autoland system that do not require interaction with the operator.
b) Automatically performs LRU and support tests related to the autoland system that require no manual preconditioning except for applying electrical power.
c) Automatically performs LRU and support tests related to the autoland system that require manual preconditions but no hydraulic power.
d) Performs manual interface checks related to the autoland system.
e) Performs manual LRU and support tests related to the autoland system.
f) Performs A/P servo and stab trim tests.
(c) Support Test Description
1) 51 AIR/GRD RLY
a) Displays any FCC and TMC self-test faults and any interface faults related to air/ground relay.
b) Interaction test of the landing gear system 1 air/ground relay and its interface to the FCCs and TMC.
c) Interaction test of the landing gear system 2 air/ground relay and its interface to the FCCs.

2) 52 TMC RLY/SW
   a) Displays any TMC self-test faults and TMC interface faults related to Environmental Control System (ECS) cards, pneumatic system, anti-ice, and thrust reverser lever position.
   b) Tests for discrete output signals from the Electrical Systems Card File #1 (P-50 Panel) left and right Electronic Engine Controls (EEC) discrete cards, from the autothrottle microswitch pack assembly, pneumatic shutoff and isolation valves, and anti-ice valves.
3) 56 FCC CONFIG/OPT
   a) Displays any FCC self-test faults and FCC interface faults related to parity.
   b) Checks the status of the FCC program pin wiring for airplane and hardware configurations and customer selected options.
4) 57 TMC CONFIG/OPT
   a) Displays any TMC self-test faults and TMC interface faults related to parity.
   b) Checks status of TMC program strapping for engine and airplane configuration, autothrottle, and thrust limit customer selected options.
5) 59 FCC INSTR
   a) Displays any FCC self-test faults.
   b) Sends test routine command and parameters to FCCs, and displays operator interaction message.
   c) Tests operation of captain's and first officer's MCP flight director switches and the ability of FCCs to move the pitch command bar 10 degrees. Checks each valid FCC separately.
6) 60 TMC INSTR
   a) Displays any TMC self-test faults.
   b) Sends test routine number and parameters to TMC and displays operator action message.
   c) Tests the ability of the TMC to control the captain's and first officer's EPR instruments target index. EPR target index driven to a test value of 1.50.
7) 64 SPD BK/FLAP XDCR
   a) Displays any TMC and FCC self-test faults and FLAP XDCR related interface faults.
   b) Displays a VFY HYD ON message.
   c) Provides interaction messages to check the FCCs sensing of the speedbrakes down, armed, and up.
   d) Displays action messages to set flap lever to 25, 15, and 1.

e) Tests operation of flap position sensing transducer, and switch/transducer-to-FCC and TMC interfaces. The flap switch and flap transducer operation is checked when the flap lever is positioned to 30, 10, and 0 units.

8) 65 STAB TRIM
   a) Displays any FCC self-test faults and FCC interface faults related to stab trim.
   b) Provides action message VFY HYD ON.
   c) Tests the ability of the FCC to command the Stabilizer Trim/Elevator Asymmetry Limit Module (SAM) to drive the horizontal stabilizer UP and DOWN. It checks the FCC-to-SAM interface.
9) 66 XDCR OUTPUTS
   a) Displays any FCC and TMC self-test faults, and transducer interface faults related to the transducers.
   b) Displays position of horizontal stabilizer, aileron servo, aileron surface, elevator servo, elevator surface, rudder servo, rudder surface, and flaps to the FCC and TMC. It tests and displays the left and right power lever angle position sent to the TMC.
10) 67 AIL SURF LIMIT
   a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.
   b) Provides action messages VFY HYD ON and A/P ENG LCR TO CMD (for valid FCCs only) (LCR = Left, Center, Right).
   c) Drives aileron to its positive limit if operator responds YES/ADV to action message AIL POS LIM?
   d) Displays aileron surface position in degress.
      AIL SURF DEG
      ±XX.X ±XX.XX ±XX.XX   where XX.X = decimal number
11) 68 ELEV SURF LIM
   a) Elevator Surface limit test same as aileron limit test.
12) 69 RUD SURF LIM
   a) Rudder surface limit test same as aileron limit test.

(17) Ground Test - LRU - 04 MCP (Fig. 17)

(a) The MCP test checks the function of all lightswitches, controls, and displays on the AFCS mode control panel using a preprogrammed test routine with operator interaction. The tests are divided into three sections: testing the bottom half of the switch/lights, testing unlighted switches, and testing displays. A TEST COMPLETE message is displayed after all test steps are completed.

1) The MCP lighted switches are tested by the MCDP sending a test command to the FCC's on ARINC 429 data buses. The FCC's send necessary data (on ARINC 429 data buses) to the MCP's lightswitches. This turns on all switch/lights. The operator presses each lighted switch; this is sensed by each FCC, and each must send the MCP an off command to turn the light off.

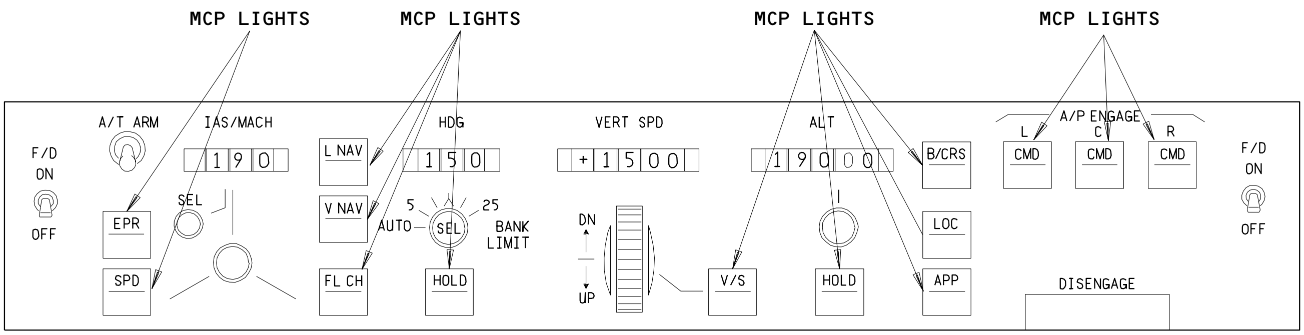
a) GUI 115;
the BARO switch/light is tested separately.

2) The testing of the unlighted switches is accomplished by the operator responding to instructions displayed by the MCDP. The test routine monitors AIM word 356 supplied from the FCC and detects the switch or control action. The test is automatically advanced to the next step after each correct test result. The test stops if the test result is incorrect. Pressing the YES/ADV switch continues the test after the test routine stops.

3) The MCP displays are tested by the operator responding to MCDP display messages and verifying that the display is in accordance with the MCDP message. Failure of the display to respond in accordance with the MCDP message indicates that the display is faulty. No fault messages are displayed on the MCDP. The operator must press the YES/ADV switch after each test, whether the result is correct or not, in order to continue the test.

(b) MCP Test Steps Description

1) Step 1 - The MCDP CPU test select routine monitors the GRD TEST SEL UP (or NO/SKIP) and DOWN switches for a command to select another test until the required test arm message 04 MCP TEST? is displayed. The operator presses the YES/ADV switch to engage the 04 MCP TEST.

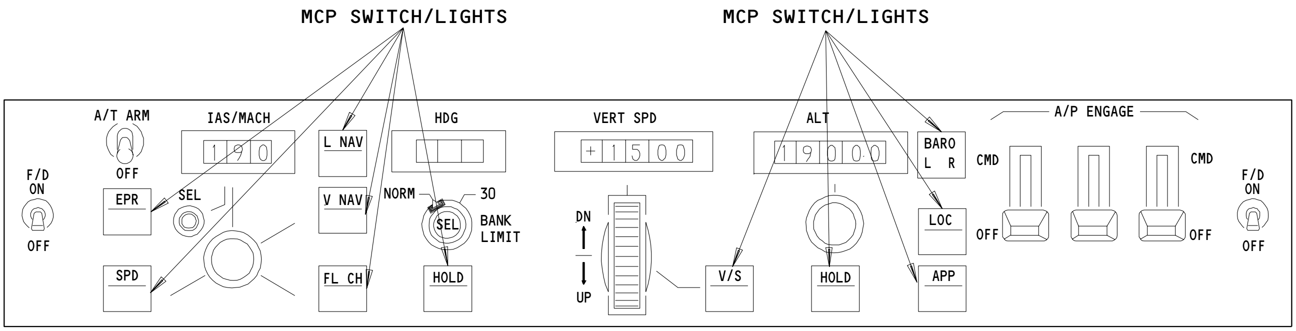


STEP MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
1. PROVIDE TEST ARMED MESSAGE ⇨	04 MCP TEST?	
2. DISPLAY ALL FAULTED FCC'S SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO COMPUTER FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	04 FCC [1] FAIL
3. DISPLAY MCP INTERFACE FAULTS SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO INTERFACE FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	MCP BUS IN A/P DISC SW MCP A/P ENG DISC MCP A/P ARM IN FCC TO MCP BUS TMC BUS IN
4. PROVIDE ACTION MESSAGE ⇨	04 MCP MAN TEST?	
5. PROVIDE ACTION MESSAGE AND ADVANCE AUTOMATICALLY WHEN IN PROGRESS BIT CHANGES FROM 1 TO 0 OR ⇨	04 PUSH OFF MCP LIGHTS - ADV	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

[1] DENOTES L, R, OR C COMPUTER

Ground Test - LRU - 04 MCP (Steps 1-5)
Figure 17 (Sheet 1)



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
1.	PROVIDE TEST ARMED MESSAGE ⇨	04 MCP TEST?	
2.	DISPLAY ALL FAULTED FCC'S SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO COMPUTER FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	04 FCC # FAIL
3.	DISPLAY MCP INTERFACE FAULTS SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO INTERFACE FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	MCP BUS IN A/P DISC MCP A/P ENG DISC MCP A/P ARM IN FCC TO MCP BUS TMC BUS IN
4.	PROVIDE ACTION MESSAGE ⇨	04 MCP MAN TEST?	
5.	PROVIDE ACTION MESSAGE AND VERIFY BARO SWITCH L/R LIGHTS AND ⇨	04 PUSH BARO-VFY L/R LIGHTS - ADV	1
6.	PROVIDE ACTION MESSAGE AND ADVANCE AUTOMATICALLY WHEN IN PROGRESS BIT CHANGES FROM 1 TO 0 OR ⇨	04 PUSH OFF MCP LIGHTS - ADV	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED

# DENOTES L, R, OR C COMPUTER

Ground Test - LRU-04 MCP (Steps 1-6)
Figure 17 (Sheet 2)

2) Step 2 - The MCDP displays 04 IN PROGRESS while checking its memory to see if any faulty FCCs and TMC were detected during the ground test entry function. Each fail message, 04 FCC (L, R, or C) FAIL, is displayed in sequence by pressing YES/ADV for the next fault. If there is no faulty TMC or FCC, the test automatically advances and checks for interface fault data stored in memory during the ground test entry functions. The 04 IN PROGRESS message is displayed while the CPU is checking for interface fault data.
3) Step 3 - An interface fault message is displayed for each faulty FCC-to-MCP and TMC-to-FCC interface. Subsequent interface faults are displayed sequentially by pressing the YES/ADV switch. The MCDP can display up to six of the following FCC interface fault messages:
   a) The MCP BUS IN message indicates that the ARINC 429 data bus from the MCP to the FCC is faulty.
   b) The MCP A/P ENG DISC message indicates that the 28 vdc servo power engage interface from the FCC to the MCDP is faulty.
   c) The A/P DISC SW message indicates that the control wheel A/P disconnect switch or interface is faulty.
   d) The MCP A/P ARM IN message indicates that the +28 vdc servo power arm interface from the FCC to the MCP is faulty.
   e) The FCC TO MCP BUS IN message indicates that the ARINC 429 data bus from the FCC to the MCP is faulty.
   f) The TMC BUS IN message indicates that the ARINC 429 data bus from the TMC to the FCC is faulty.
4) Step 4 - If there are no interface faults, the test routine automatically advances to the MCP test and displays 04 MCP MAN TEST?. Pressing the YES/ADV switch enables the 04 MCP manual test.
5) GUI 115;
   step 5 - Displays action message 04 PUSH BARO-VFY L/R LIGHTS-ADV for the operator to push the BARO switch and verify the alternate L and R light indication on the BARO switch.

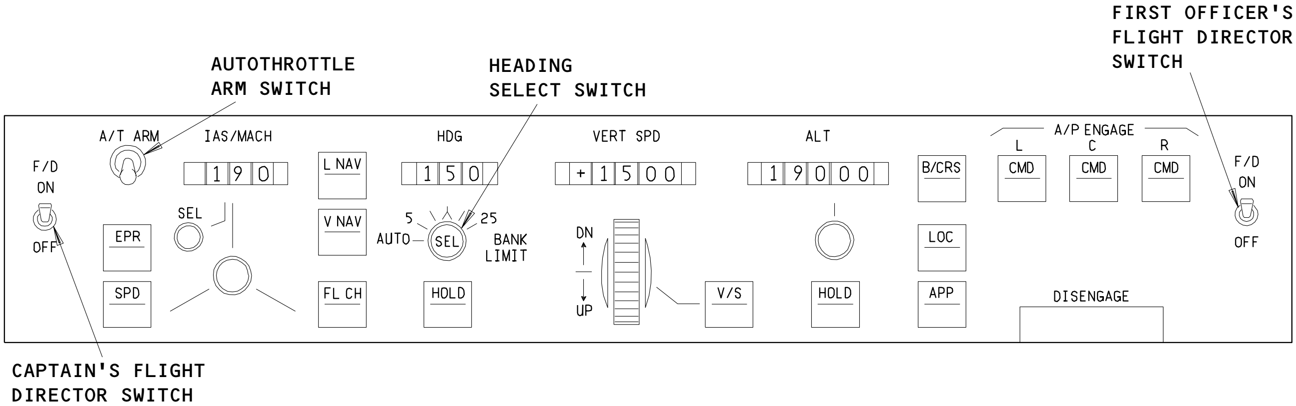
6) GUI 001-114, 116-999;
Step 5 - The 04 manual test starts by testing all lighted switches. The MCDP commands the FCCs to do the MCP lighted switch test. The FCC turns on all lightswitches that are controlled by the FCCs. The MCDP then displays action message 04 PUSH OFF MCP LIGHTS and the operator pushes each MCP lightswitch. If the lightswitch pushed is functioning, the FCC senses that the switch has been pressed and sends a signal back to the MCP to turn that lightswitch off. If pressing the lightswitch does not cause the light to turn off, it is an indication that the lightswitch circuit is faulty. No fault message is displayed. The YES/ADV switch must be pressed after all the switches have been tested to advance to the next step.

7) GUI 115;
step 6 - The 04 manual test starts by testing all lighted switches except the BARO switch. The MCDP commands the FCCs to do the MCP lighted switch test. The FCC turns on all lightswitches that are controlled by the FCCs. The MCDP then displays action message 04 PUSH OFF MCP LIGHTS and the operator pushes each MCP lightswitch. If the lightswitch pushed is functioning, the FCC senses that the switch has been pressed and sends a signal back to the MCP to turn that lightswitch off. If pressing the lightswitch does not cause the light to turn off, it is an indication that the lightswitch circuit is faulty. No fault message is displayed. The test advances to the next step when the FCC senses that all switches have been tested, or the YES/ADV switch is pressed.

(18) GUI 001-114, 116-999;
Ground Test - LRU - 04 MCP (Steps 6-12) (Fig. 18)

(a) MCP Test Steps 6-12 Description

1) Test steps (6-20) are verified by the CPU automatically advancing to the next test step. The test does not advance if the CPU does not verify that the switch tested is in the correct position. To continue the test past a fault, the operator presses the YES/ADV switch. There is no fault message for unlighted switch tests.

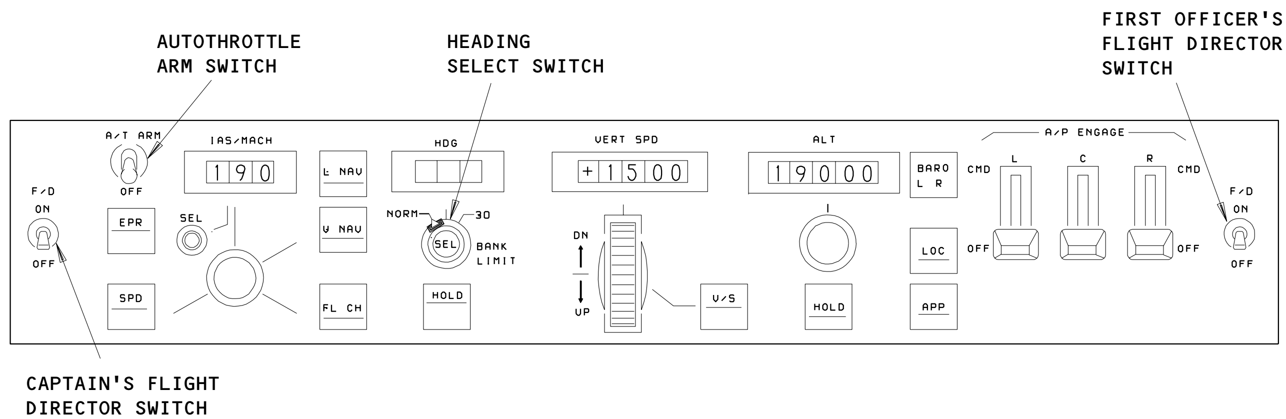


STEP MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
7. PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW ON	1
8. PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW OFF	1
9. PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW ARM	1
10.PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW OFF	1
11.PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF HDG SEL SWITCH VERIFIED OR ⇨	04 HOLD DN HDG SEL SW	1
12.PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW ON	1
13.PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW OFF	1

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITON

Ground Test - LRU - 04 MCP (Steps 6-12)
Figure 18 (Sheet 1)



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
7.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW ON	1
8.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW OFF	1
9.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW ARM	1
10.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW OFF	1
11.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF HDG SEL SWITCH VERIFIED OR ⇨	04 HOLD DN HDG SEL SW	1
12.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW ON	1
13.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW OFF	1

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITON

Ground Test - LRU - 04 MCP (Steps 7-13)
Figure 18 (Sheet 2)

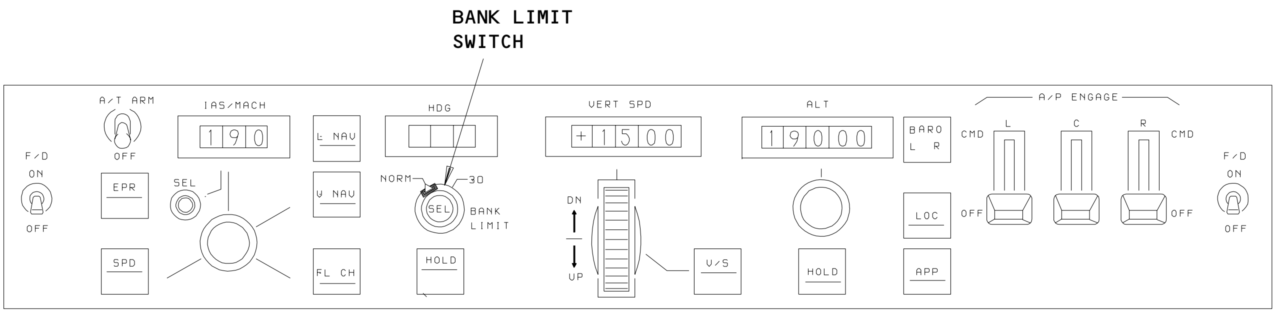
2) Step 6 - Action message 04 SET CAPT F/D SW ON instructs the operator to position the captain's flight director switch to ON. The CPU monitors AIM label 356, word 6, bit 9 to be set to logic one. Bit 9 set to one indicates to the CPU that the captain's F/D switch is in the ON position. The CPU then advances to the next test step.
3) Step 7 - Action message 04 SET CAPT F/D SW OFF instructs the operator to switch off the captain's flight director switch. The CPU monitors AIM label 356, word 6, bit 9 to be set to zero. Bit 9 set to zero indicates to the CPU that the captain's flight director is turned off. The CPU advances to the next step.
4) Step 8 - Action message 04 SET A/T SW ARM instructs the operator to position the autothrottle switch to A/T ARM. The CPU verifies that the switch position is A/T ARM when it senses that AIM label 356, word 6, bit 10 is set to logic one. When the switch position is verified, the CPU advances to the next test step.
5) Step 9 - Action message 04 SET A/T SW OFF instructs the operator to position the autothrottle switch to off. When the CPU verifies that AIM label 356, word 6, bit 10 has been set to logic zero, the CPU advances the test routine to the next step.
6) Step 10 - Action message 04 HOLD DN HDG SEL SW instructs the operator to press the heading select switch. The CPU verifies that AIM label 356, word 6, bit 24 is set to logic one. The CPU advances to the next test step.
7) Step 11 - Action message 04 SET F/O F/D SW ON instructs the operator to position the first officer's flight director switch to ON. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic one. The CPU advances to the next test step.
8) Step 12 - Action message 04 SET F/O F/D SW OFF instructs the operator to turn off the first officer's flight director switch. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic zero. The CPU advances to the next test step.

(19) GUI 115;
Ground Test - LRU - 04 MCP (Steps 7-13) (Fig. 18)
(a) MCP Test Steps 7-13 Description
1) Test steps (7-18) are verified by the CPU automatically advancing to the next test step. The test does not advance if the CPU does not verify that the switch tested is in the correct position. To continue the test past a fault, the operator presses the YES/ADV switch. There is no fault message for unlighted switch tests.
2) Step 7 - Action message 04 SET CAPT F/D SW ON instructs the operator to position the captain's flight director switch to ON. The CPU monitors AIM label 356, word 6, bit 9 to be set to logic one. Bit 9 set to one indicates to the CPU that the captain's F/D switch is in the ON position. The CPU then advances to the next test step.
3) Step 8 - Action message 04 SET CAPT F/D SW OFF instructs the operator to switch off the captain's flight director switch. The CPU monitors AIM label 356, word 6, bit 9 to be set to zero. Bit 9 set to zero indicates to the CPU that the captain's flight director is turned off. The CPU advances to the next step.
4) Step 9 - Action message 04 SET A/T SW ARM instructs the operator to position the autothrottle switch to A/T ARM. The CPU verifies that the switch position is A/T ARM when it senses that AIM label 356, word 6, bit 10 is set to logic one. When the switch position is verified, the CPU advances to the next test step.
5) Step 10 - Action message 04 SET A/T SW OFF instructs the operator to position the autothrottle switch to off. When the CPU verifies that AIM label 356, word 6, bit 10 has been set to logic zero, the CPU advances the test routine to the next step.
6) Step 11 - Action message 04 HOLD DN HDG SEL SW instructs the operator to press the heading select switch. The CPU verifies that AIM label 356, word 6, bit 24 is set to logic one. The CPU advances to the next test step.
7) Step 12 - Action message 04 SET F/O F/D SW ON instructs the operator to position the first officer's flight director switch to ON. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic one. The CPU advances to the next test step.
8) Step 13 - Action message 04 SET F/O F/D SW OFF instructs the operator to turn off the first officer's flight director switch. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic zero. The CPU advances to the next test step.

(20) Ground Test - LRU - 04 MCP (Steps 14-18) (Fig. 19)
(21) GUI 115;
Ground Test - LRU - 04 MCP (Steps 14-18) (Fig. 19)

(a) MCP Test Steps 14-18 Description
1) Step 14 - Action message 04 SET BANK LIM TO 30 instructs operator to position the bank limit selector switch to 30. The CPU verifies that the AIM label 356, word 6, bits 26 and 27 are set to logic one. The CPU advances to the next step.
2) Step 15 - Action message 04 SET BANK LIM TO NRM (Norm) instructs the operator to position the bank limit selector switch to NORM. The CPU verifies that the AIM label 356, word 6, bits 26, 27, and 28 are set to logic zero. The CPU advances to the next step.
3) Step 16 - Action message 04 SET A/P LCR TO CMD instructs the operator to engage the left, center, and right autopilot engage BAT HANDLES.
a) Displays 04 ENG # FAIL for each fault. Press YES/ADV to display each fault. Advances to next step for YES/ADV after last fault displayed.
4) Step 17 - Message 04 IN PROGRESS, CPU transmits null test to drop BAT HANDLES to Off and then advances to next step.
5) Step 18 - Action message 04 VFY A/P LCR TO OFF-ADV instructs the operator to verify that the left, center, and right BAT HANDLE switches are OFF. Press YES/ADV to advance to next step.

(22) GUI 001-114, 116-999;
Ground Test - LRU - 04 MCP (Steps 13-20) (Fig. 19)
(a) MCP Test Steps 13-20 Description
1) Step 13 - Action message 04 PUSH DISENG BAR DN instructs the operator to position the disengage bar down. The CPU verifies that the AIM label 356, word 6, bit 14 is set to logic one. The CPU advances to the next test step.
2) Step 14 - Action message 04 PUSH DISENG BAR UP instructs the operator to position the disengage bar up. The CPU verifies that AIM label 356, word 6, bit 14 is set to logic zero. The CPU advances to the next test step.
3) Step 15 - Action message 04 SET BANK LIM TO 25 instructs the operator to position the bank limit selector switch to 25. The CPU verifies that AIM label 356, word 6, bit 18 is set to logic one. The test advances to the next test step.

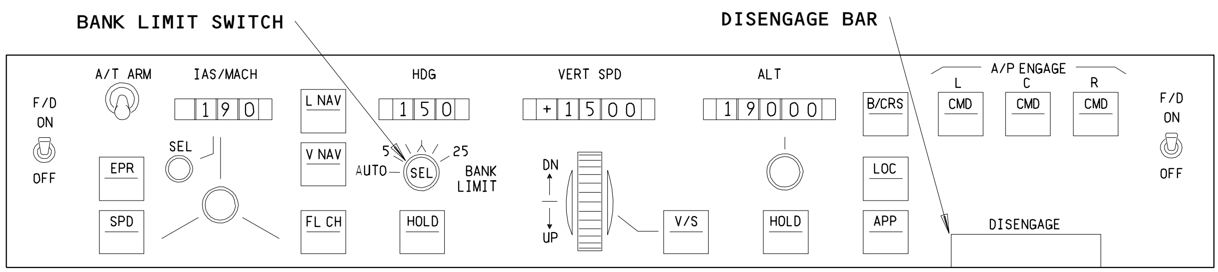


STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
14.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 30 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 30	1
15.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = NORM FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO NRM	1
16.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ALL VALID FCCS TO CMD VERIFIED OR ⇨	04 SET A/P LCR TO CMD	04 ENG #FAIL
17.	PROVIDE MESSAGE AND TRANSMITS MULL TEST TO DROP BAT HANDLES THEN ADVANCE TO NEXT STEP VERIFIED OR ⇨	04 IN PROGRESS	
18.	PROVIDE ACTION MESSAGE AND VERIFY BAT HANDLES OFF AND ⇨	04 VFY A/P LCR TO OFF - ADV	1

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 14-18)
Figure 19 (Sheet 1)



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
13.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF DISENGAGE BAR VERIFIED OR ⇨	04 PUSH DISENG BAR DN	[1]
14.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF DISENGAGE BAR VERIFIED OR ⇨	04 PUSH DISENG BAR UP	[1]
15.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 25 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM 0 25	[1]
16.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 20 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 20	[1]
17.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 15 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 15	[1]
18.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 10 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 10	[1]
19.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 5 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 5	[1]
20.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = AUTO FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO AUTO	[1]

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

[1] NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 13-20)
Figure 19 (Sheet 2)

4) Step 16 - Action message 04 SET BANK LIM TO 20 instructs the operator to position the bank limit selector switch to 20. The CPU verifies that AIM label 356, word 6, bit 19 is set to logic one. The test advances to the next test step.
5) Step 17 - Action message 04 SET BANK LIM TO 15 instructs the operator to position the bank limit selector switch to 15. The CPU verifies that AIM label 356, word 6, bit 20 is set to logic one. The test advances to the next step.
6) Step 18 - Action message 04 SET BANK LIM TO 10 instructs the operator to set the bank limit selector switch to 10. The CPU verifies that AIM label 356, word 6, bit 21 is set to logic one. The test advances to the next step.
7) Step 19 - Action message 04 SET BANK LIM TO 5 instructs the operator to set the bank limit selector switch to 5. The CPU verifies that AIM label 356, word 6, bit 22 is set to logic one. The test advances to the next test step.
8) Step 20 - Action message 04 SET BANK LIM TO AUTO instructs the operator to position the bank limit selector switch to AUTO. The CPU verifies that ARINC label 356, word 6, bit 23 is set to logic one. The test advances to the next test step.

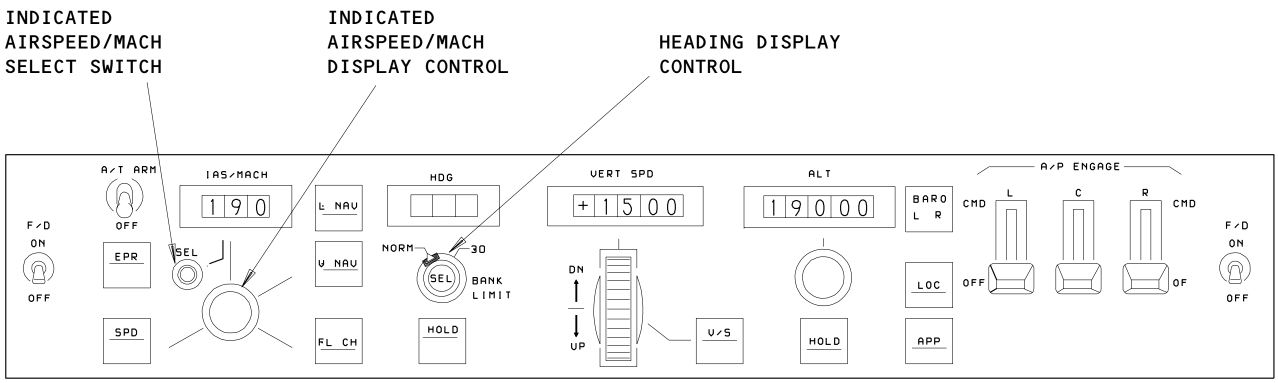
(23) GUI 115;
Ground Test - LRU - 04 MCP (Steps 19-24) (Fig. 20)

(a) MCP Test Steps 19-24 Description
1) For test steps 19 and on, whether the results are correct or not, the operator must press the YES/ADV switch to advance to the next test.
2) Step 19 - Action message 04 PUSH IAS/MACH SEL SW instructs the operator to push the IAS/MACH SEL switch. The CPU verifies that the AIM label 356, word 6, bit 11 is set to logic one. The CPU advances to the next test step.
3) Step 20 - Action message 04 TURN IAS/MACH CCW - DSPLY DEC? instructs the operator to rotate the indicated airspeed/mach display control counterclockwise, and verify that the display decreases in value.

4) Step 21 - Action message 04 TURN IAS/MACH CW - DSPLY INC? instructs the operator to rotate the indicated airspeed/mach display control clockwise, and verify that the display increases in value.
5) Step 22 - Action message 04 PUSH IAS/MACH - DSPLY BLANK? instructs the operator to push the IAS/MACH SEL switch, and verify that the IAS/MACH display window goes blank.
6) Step 23 - Action message 04 TURN HDG COW - DSPLAY DEC? instructs the operator to rotate the heading display control counterclockwise, and verify that the display decreases in value.
7) Step 24 - Action message 04 TURN HDG CW DSPLY INC? instructs the operator to rotate the heading display control clockwise, and verify that the display increases in value.

(24) GUI 001-114, 116-999;
Ground Test - LRU - 04 MCP (Steps 21-26) (Fig. 20)

(a) MCP Test Steps 21-26 Description
1) For test steps 21 and on, whether the results are correct or not, the operator must press the YES/ADV switch to advance to the next test.
2) Step 21 - Action message 04 PUSH IAS/MACH SEL SW instructs the operator to push the IAS/MACH SEL switch. The CPU verifies that the AIM label 356, word 6, bit 11 is set to logic one. The CPU advances to the next test step.
3) Step 22 - Action message 04 TURN IAS/MACH CCW - DSPLY DEC? instructs the operator to rotate the indicated airspeed/mach display control counterclockwise, and verify that the display decreases in value.
4) Step 23 - Action message 04 TURN IAS/MACH CW - DSPLY INC? instructs the operator to rotate the indicated airspeed/mach display control clockwise, and verify that the display increases in value.
5) Step 24 - Action message 04 PUSH IAS/MACH - DSPLY BLANK? instructs the operator to push the IAS/MACH SEL switch and verify that the IAS/MACH display window goes blank.

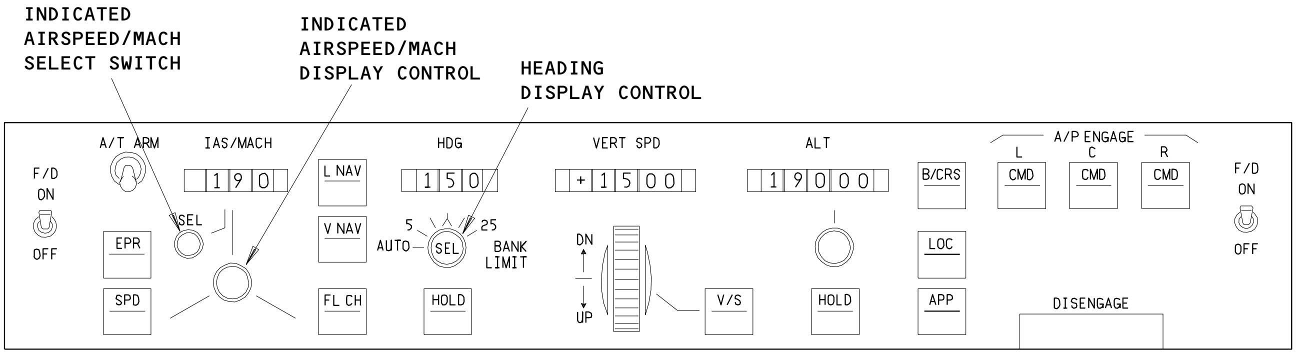


STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
19.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF IAS/MACH SELECT SWITCH VERIFIED OR ⇨	04 PUSH IAS/MACH SEL SW	1
20.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN IAS/MACH CCW-DSPLY DEC?	
21.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN IAS/MACH CW-DSPLY INC?	
22.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY BLANKS ⇨	04 PUSH IAS/MACH -DSPLY BLANK?	
23.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN HDG CCW-DSPLY DEC?	
24.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN HDG CW-DSPLY INC?	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 19-24)
Figure 20 (Sheet 1)



STEP MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
21. PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF IAS/MACH SELECT SWITCH VERIFIED OR ⇨	04 PUSH IAS/MACH SEL SW	[1]
22. PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN IAS/MACH CCW-DSPLY DEC?	
23. PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN IAS/MACH CW-DSPLY INC?	
24. PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 PUSH IAS/MACH -DSPLY BLANK?	
25. PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN HDG CCW-DSPLY DEC?	
26. PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN HDG CW-DSPLY INC?	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

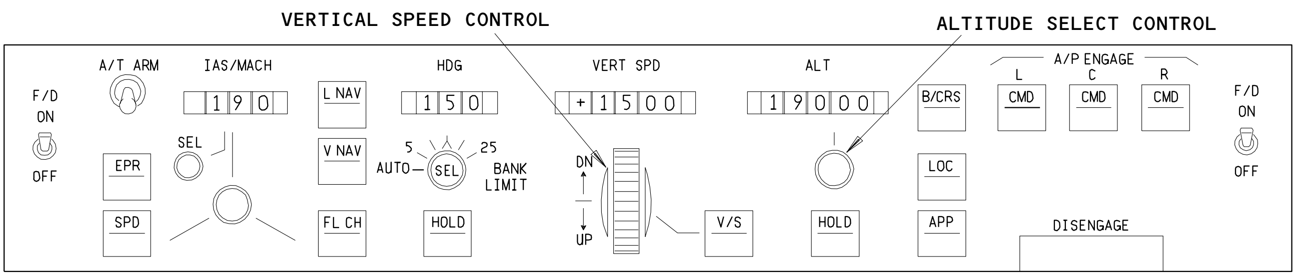
[1] NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 21-26)
Figure 20 (Sheet 2)

6) Step 25 - Action message 04 TURN HDG CCW - DSPLY DEC? instructs the operator to rotate the heading display control counterclockwise, and verify that the display decreases in value.
7) Step 26 - Action message 04 TURN HDG CW DSPLY INC? instructs the operator to rotate the heading display control clockwise, and verify that the display increases in value.

(25) GUI 001-114, 116-999;
Ground Test - LRU - 04 MCP (Steps 27-31) (Fig. 21)
(a) MCP Test Steps 27-31 Description
1) Step 27 - Action message 04 TURN VERT SPD UP - DSPLY INC? instructs the operator to rotate the vertical speed control toward UP and verify that the vertical speed display increases in value.
2) Step 28 - Action message 04 TURN VERT SPD DN - DSPLY DEC? instructs the operator to rotate the vertical speed control toward DN and verify that the vertical speed display decreases in value.
3) Step 29 - Action message 04 TURN ALT SEL CCW - DSPLY DEC? instructs the operator to rotate the altitude select control counterclockwise and verify that the altitude display decreases in value.
4) Step 30 - Action message 04 TURN ALT SEL CW - DSPLY INC? instructs the operator to rotate the altitude select control clockwise and verify that the altitude display increases in value.
5) Step 31 - Message 04 TEST COMPLETE is displayed when the 04 MCP test routine has ended. The completed test is exited and enters ground test select when either the YES/ADV or NO/SKIP switch is pressed. On entering the ground test select routine, the display shows the next ground test listed on the front panel.

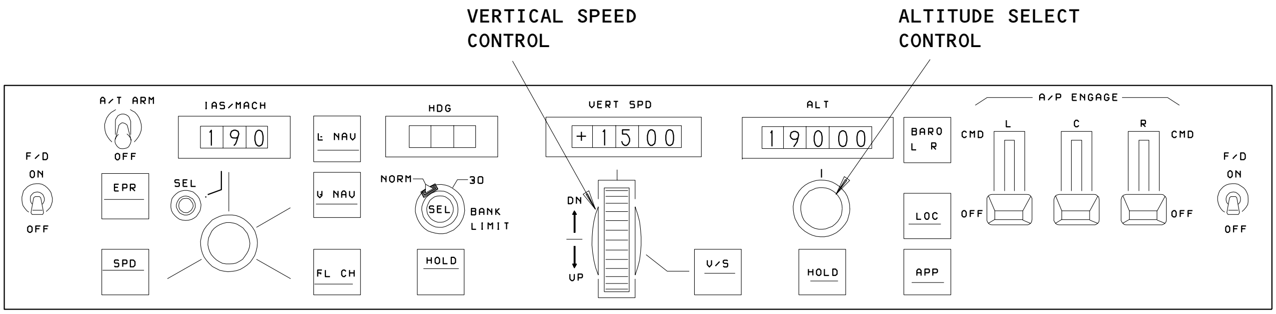
(26) GUI 115;
Ground Test - LRU - 04 MCP (Steps 25-29) (Fig. 21)



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
27.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN VERT SPD UP-DSPLY INC?	
28.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN VERT SPD DN-DSPLY DEC?	
29.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN ALT SEL CCW-DSPLY DEC?	
30.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN ALT SEL CW-DSPLY INC?	
31.	PROVIDE TEST COMPLETE MESSAGE	04 TEST COMPLETE	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

Ground Test - LRU - 04 MCP (Steps 27-31)
Figure 21 (Sheet 1)



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
25.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN VERT SPD UP-DSPLY INC?	
26.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN VERT SPD DN-DSPLY DEC?	
27.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN ALT SEL CCW-DSPLY DEC?	
28.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN ALT SEL CW-DSPLY INC?	
29.	PROVIDE TEST COMPLETE MESSAGE	04 TEST COMPLETE	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

Ground Test - LRU - 04 MCP (Steps 25-29)
Figure 21 (Sheet 2)

(a) MCP Test Steps 25-29 Description
   1) Step 25 - Action message 04 TURN VERT SPD UP - DSPLY INC? instructs the operator to rotate the vertical speed control toward UP and verify that the vertical speed display increases in value.
   2) Step 26 - Action message 04 TURN VERT SPD DN - DSPLY DEC? instructs the operator to rotate the vertical speed control toward DN and verify that the vertical speed display decreases in value.
   3) Step 27 - Action message 04 TURN ALT SEL CCW - DSPLY DEC? instructs the operator to rotate the altitude select control counterclockwise and verify that the altitude display decreases in value.
   4) Step 28 - Action message 04 TURN ALT SEL CW - DSPLY INC? instructs the operator to rotate the altitude select control clockwise and verify that the altitude display increases in value.
   5) Step 29 - Message 04 TEST COMPLETE is displayed when the 04 MCP test routine has ended. The completed test is exited and enters ground test select when either the YES/ADV or NO/SKIP switch is pressed. On entering the ground test select routine, the display shows the next ground test listed on the front panel.

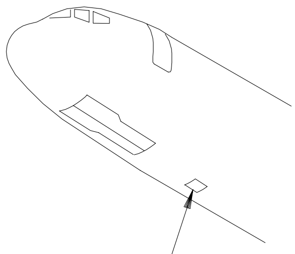
MAINTENANCE MONITOR

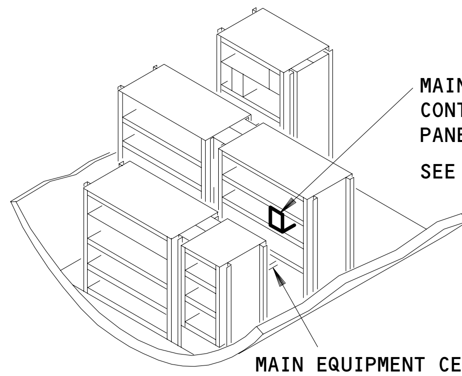
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
CIRCUIT BREAKER - MAINT CONT DISPLAY, C520	2	1	FLT COMPT, P11 11S3 [1] 11S6 [2]	*
CONNECTOR - MCDP REMOTE CONTROL PANEL, D4110J	2	1	FLT COMPT, P61	22-41-00
PANEL - MAINTENANCE CONTROL DISPLAY, M168	1	1	MAIN EQUIP CTR, 119BL, E3-2	22-41-01

* SEE THE WDM EQUIPMENT LIST

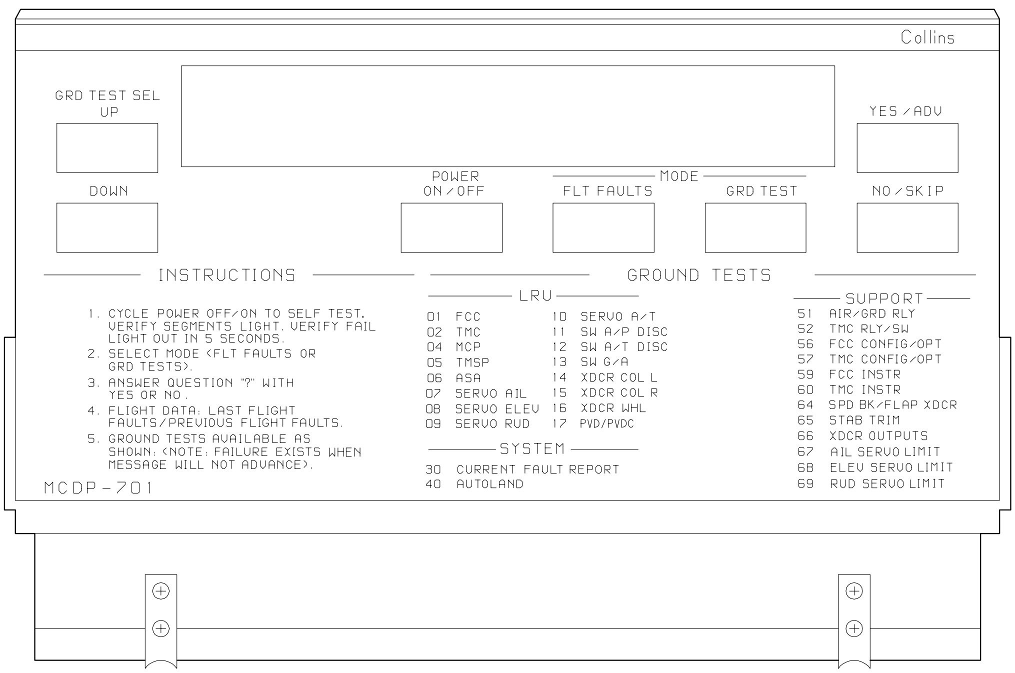
[1] GUI 001-114,116-999
[2] GUI 115

Maintenance Monitor - Component Index
Figure 101





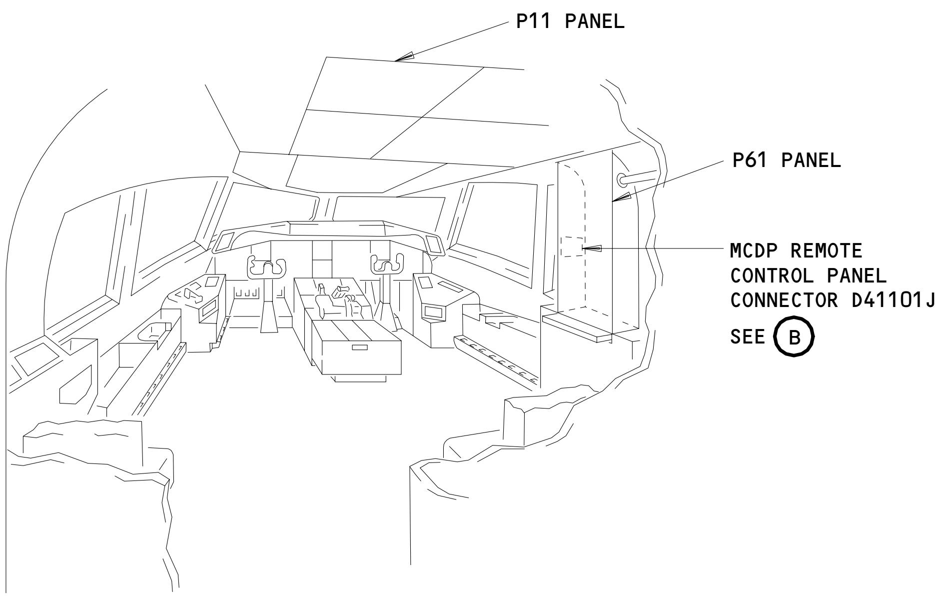
MAIN EQUIPMENT CENTER



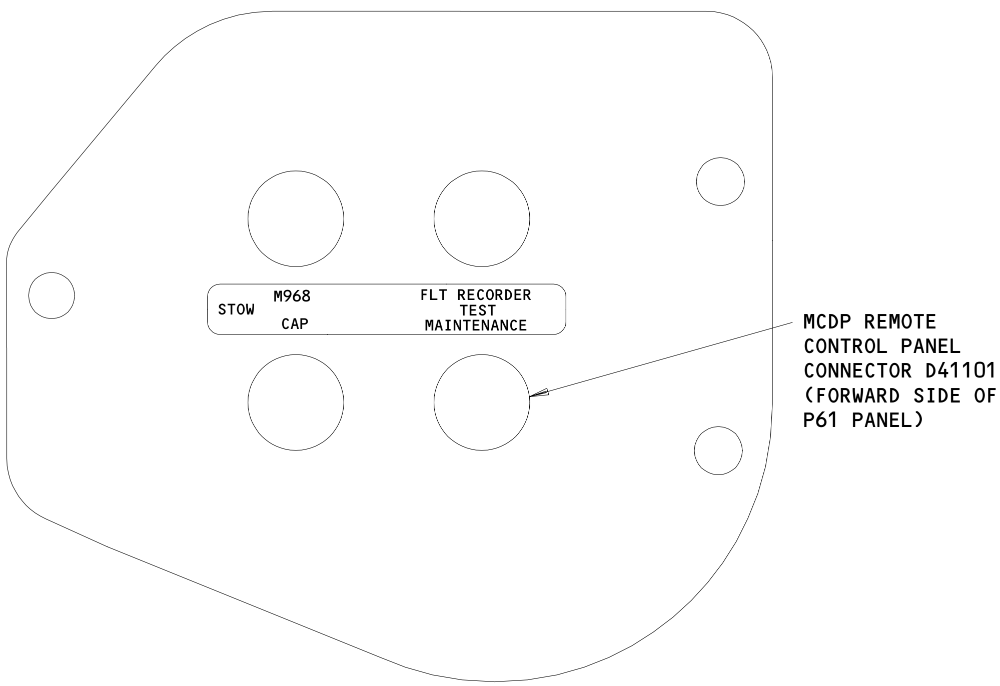
MAINTENANCE CONTROL DISPLAY PANEL, M168

A

Maintenance Monitor - Component Location
Figure 102 (Sheet 1)



FLIGHT COMPARTMENT



B

Maintenance Monitor - Component Location
Figure 102 (Sheet 2)

MAINTENANCE MONITOR - ADJUSTMENT/TEST

TASK 22-41-00-715-001

1. Operational Test - MCDP

   A. General

      (1) The Maintenance Control and Display Panel (MCDP) is found in the main equipment center. Do the procedures that follow to test the MCDP and use the flight faults displays and ground tests.

   B. Equipment

      (1) Remote Control Unit, Maintenance Control Display Panel - A22001-22 (preferred); A22001-15 (optional).

   C. References

      (1) 24-22-00/201, Electrical Power - Control
      (2) 22-00-02/201, Autoflight Bite
      (3) 31-41-00/501, Engine Indication and Crew Alerting System (EICAS)

   D. Access

      (1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

      (2) Access Panels

119AL	Main Equipment Center

   E. Prepare to do a Test

      S 865-002

      (1) Make sure these circuit breakers on the overhead circuit breaker panel, P11, are closed:
         (a) 11C30, LANDING GEAR POS SYS 1
         (b) 11E8, FMCS CDU LEFT
         (c) 11E9, FMCS CMPTR LEFT
         (d) 11E16, MODE CONT PNL LEFT
         (e) 11E17, FLT CONT COMPUTER POWER LEFT
         (f) 11E18, FLT CONT COMPUTER SERVO LEFT
         (g) 11E19 or 11E20, FLIGHT CONT CMPTR PWR CENTER
         (h) 11E20 or 11E21, FLIGHT CONT CMPTR SERVO CENTER
         (i) 11E29, FMCS CDU RIGHT
         (j) 11E30, FMCS CMPTR RIGHT
         (k) 11E34, MODE CONT PNL RIGHT
         (l) 11E35, FLT CONT CMPTR PWR RIGHT
         (m) 11E36, FLT CONT CMPTR SERVO RIGHT
         (n) 11F14 or 11F16, TMC AC
         (o) 11F15 or 11F17, TMC DC
         (p) 11F16 or 11F18, TMC SERVO

(q) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL
(r) GUI 115;
11S6, MAINT CONT DSPL
(s) 11S15, AIR/GND SYS 1
(t) 11S19, AIR/GND SYS 2
(u) 11S23, POS SYS 2

F. Do the MCDP Test

S 865-003

(1) If operation of the remote MCDP panel is to be done, see the Energize the MCDP with the Remote MCDP Control Panel procedure.

S 865-007

(2) Energize the MCDP without the Remote MCDP Control Panel.
(a) Supply electrical power (Ref 24-22-00)
(b) Push and hold the ON/OFF switch until the FAIL light in the ON/OFF switch comes on.
1) Make sure the FLT FAULTS and GRD TEST lights come on and then go off.
2) Make sure all the display segments come on.

NOTE: Push the MCDP ON/OFF switch off and then on again to get more time to make sure switch and display segments work correctly.

3) After 5 seconds, make sure the effects that follow occur.
a) The FAIL light in the ON/OFF switch goes off.
b) The FLT FAULTS MODE light comes on.
c) The message LAST FLT FAULT? is shown.

S 485-008

(3) Energize the MCDP with the Remote MCDP Control Panel.
(a) Supply electrical power (Ref 24-22-00).
(b) Make sure the EICAS operates (Ref 31-41-00).
(c) Connect the Remote MCDP control unit to the remote control connector found on the forward side of the P61 panel.
(d) Push the CONF/MCDP switch on EICAS MAINT panel on right side panel P61.
(e) Make sure the EICAS shows the message MCDP OFF.
(f) Push and hold the ON/OFF switch until an IN PROGRESS message shows on the EICAS.
1) Make sure the MCDP FLT FAULTS message replaces the MCDP OFF message on the EICAS display.
2) Make sure the LAST FLT FAULT? message shows below the MCDP FLT FAULTS message on the EICAS display.

S 865-009

(4) Ground Test Mode Operation

(a) Operate the GRD TEST switch.

1) Make sure the GRD TEST light on the EADI comes on and the message 01 FCC TEST? is shown on the MCDP.

2) If the remote MCDP control unit is used, then the MCDP GRD TEST message replaces the MCDP FLT FAULTS message and the message 01 FCC TEST? is shown on the EICAS display.

(b) Push the GRD TEST SEL UP, the GRD TEST SEL DOWN, or the NO/SKIP switch.

NOTE: See the front of the MCDP for the available tests. The NO/SKIP switch is used to go around a test that is shown but not engaged or to exit a test that has been engaged. The UP and DOWN switches show the test numbers at a rate of four for each second.

1) Make sure the test numbers increase when the GRD TEST SEL UP or the NO/SKIP switch is pushed.

2) Make sure the test numbers decrease when the GRD TEST SEL DOWN switch is pushed.

(c) If it is necessary to do a ground test, then go to the Autoflight Bite (Ref 22-00-02/201).

S 865-010

(5) Flight Faults Mode Operation

(a) If the FLT FAULT light is not on and the LAST FLT FAULT? message is not shown, push the FLT FAULT switch.

1) Make sure the message LAST FLT FAULT? is shown.

(b) If the LAST FLIGHT FAULTS are to be done, push the YES/ADV switch.

1) Make sure a 01 with a fault message or the NO LAST FLT FAULTS message is shown.

(c) Push the YES/ADV switch to show each fault message.

1) Make sure a different fault message is shown each time the YES/ADV switch is pushed until the ALL LAST FLT FAULT DSPLY message is shown.

(d) Push the YES/ADV or the NO/SKIP switch to go out of last flight faults after NO LAST FLT FAULTS or ALL LAST FLIGHT FAULT DSPLY message is shown.

1) Make sure the PREV FLT FAULTS? message is shown.

(e) If the PREVIOUS FLIGHT FAULTS are to be done and the PREV FLT FAULTS? message is not shown, push the NO/SKIP switch.

1) Make sure the PREV FLT FAULTS? message is shown.

(f) Push the YES/ADV switch to show the PREVIOUS FLIGHT FAULTS.
1) Make sure a number, XX, with a fault message is shown or the NO PREV FLT FAULTS message is shown.

NOTE: XX shows the number of flights before the last flight that the failure occurred.

2) Make sure a different fault message is shown each time the YES/ADV switch is pushed until the message ALL PREV FLT FAULT DSPLY is shown.

(g) Go out of PREVIOUS FLIGHT FAULTS when the NO PREV FLT FAULTS or the ALL PREV FLT FAULT DSPLY message shows. Push the YES/ADV or the NO/SKIPswitch to do this.

G. Put the Airplane Back to Its Usual Condition

S 865-014
(1) Set the MCDP to OFF if it is no longer necessary.

S 085-015
(2) Remove the remote MCDP panel if installed.

S 865-017
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

MAINTENANCE CONTROL DISPLAY PANEL - REMOVAL/INSTALLATION

1. General
   A. The Maintenance Control Display Panel (MCDP) is on the E3-2 shelf in the main equipment center. It is found through the main equipment center access door, 119BL.

TASK 22-41-01-024-001

2. Remove the Maintenance Control Display Panel
   A. References
      (1) 20-10-01/401, E/E Rack Mounted Components
      (2) 20-41-01/201, Electrostatic Device Sensitive Devices
   B. Access
      (1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

      (2) Access Panel

119BL	Main Equipment Center

   C. Procedure

S 864-002

      (1) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach a DO-NOT-CLOSE tag:
         (a) GUI 001-114, 116-999;
             11S3, MAINT CONT DSPL
         (b) GUI 115;
             11S6, MAINT CONT DSPL

S 914-003

CAUTION: DO NOT TOUCH THE MCDP BEFORE YOU DO THE PROCEDURE FOR ELECTROSTATIC DISCHARGE SENSITIVE DEVICES (AMM 20-41-01/201). ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCDP.

      (2) Do the procedure for electrostatic discharge sensitive devices (Ref 20-41-01).

S 024-004

      (3) Remove the MCDP (Ref 20-10-01).

TASK 22-41-01-424-005

3. Install the Maintenance Control Display Panel
   A. References
      (1) 20-10-01/401, E/E Rack Mounted Components

(2) 20-41-01/201, Electrostatic Discharge Sensitive Devices
(3) 24-22-00/201, Electrical Power - Control

B. Access

(1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

(2) Access Panel

119BL	Main Equipment Center

C. Procedure

S 914-006

CAUTION: DO NOT TOUCH THE MCDP BEFORE YOU DO THE PROCEDURE FOR ELECTROSTATIC DISCHARGE SENSITIVE DEVICES (AMM 20-41-01/201). ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCDP.

(1) Do the procedure for electrostatic discharge sensitive devices (Ref 20-41-01).

S 414-007

(2) Install the MCDP (Ref 20-10-01).

NOTE: Do not move the MCDP directly from a different airplane. Only shop procedures can erase flight data on the MCDP. The MCDP can keep flight data for many flights. Make sure that all flight data is erased, unless the MCDP will be installed on the same airplane.

S 864-008

(3) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:

(a) GUI 001-114, 116-999;
11S3, MAINT CONT DSPL

(b) GUI 115;
11S6, MAINT CONT DSPL

D. The MCDP Test

S 864-009
(1) Supply electrical power (Ref 24-22-00).

S 864-010
(2) Push the ON/OFF switch for a manual power-up and self-test.

S 864-011
(3) Make sure all the segments and switch lights come on during the test.

S 864-012
(4) Make sure the FAIL light goes off after 5 seconds.

S 864-013
(5) Make sure the FLT FAULTS annunciator light is on and the LAST FLT FAULTS? message is shown.

S 864-014
(6) Push the YES/ADV switch.

S 864-015
(7) Make sure the NO LAST FLT FAULTS message is on MCDP display.

S 864-016
(8) Push the YES/ADV switch.

S 864-017
(9) Make sure the PREV FLT FAULTS? message is on MCDP display.

S 864-018
(10) Push the YES/ADV switch.

S 864-019
(11) Make sure the NO PREV FLT FAULTS message is on MCDP display.

E. Put the Airplane Back to Its Usual Condition

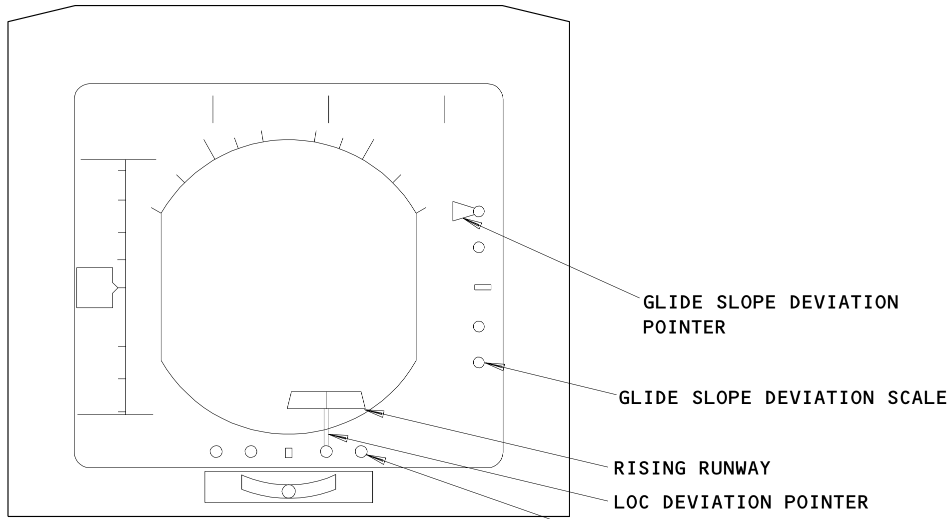
S 864-020
(1) Set the MCDP to OFF if it is no longer necessary.

S 864-021

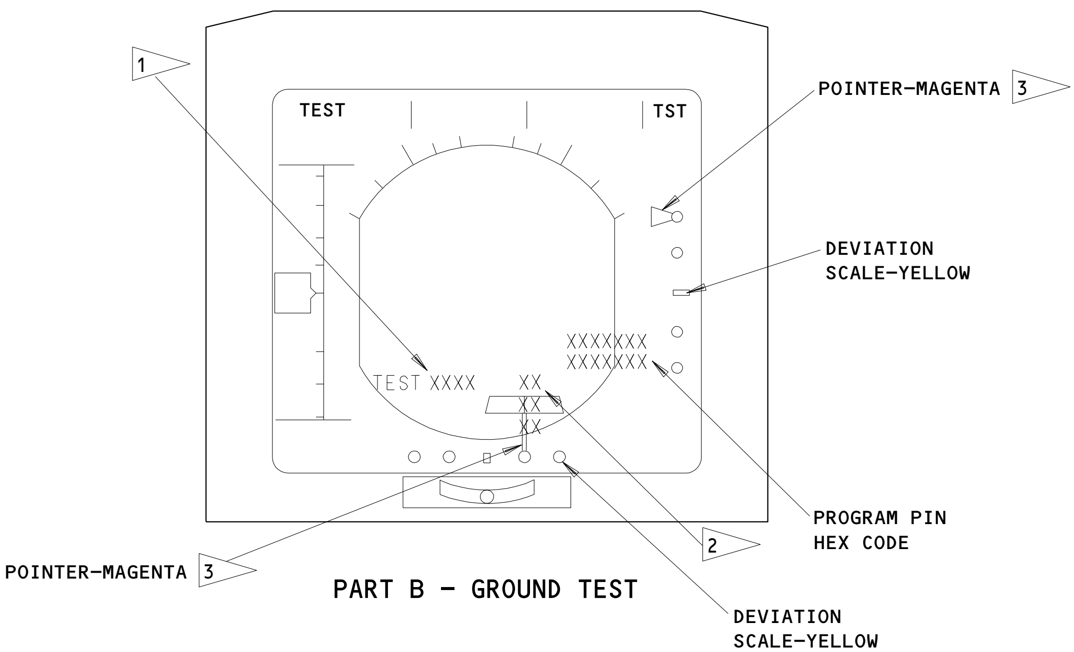
(2) Remove electrical power if it is not necessary (Ref 24-22-00).

ILS DEVIATION MONITOR - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The ILS deviation monitoring system consists of separate glideslope and localizer monitors. The monitors are an integral part of the EFIS symbol generators software. The monitors cause the EADI glideslope and localizer deviation scales to change from white to yellow and the deviation pointers to flash at four hertz. The monitors are enabled when the airplane deviates beyond prescribed limits from the center of the glideslope and localizer beams with the autopilot or flight director in approach mode.
   B. GUI 115;
      An expanded scale with a rising runway is displayed when localizer deviation is less than 5/8 of a dot; localizer or rollout mode is armed, and left, center, and/or right autopilot is engaged. The expanded display reverts back to standard scale if the deviation exceeds one dot, or if both modes are disengaged and ground speed is less than 30 knots or radio altitude is greater than 200 feet.
   C. GUI 001-114, 116-999;
      An expanded scale with a rising runway is displayed when localizer deviation is less than 5/8 of a dot and localizer or rollout mode is armed. The expanded display reverts back to standard scale if the deviation exceeds one dot; or if both modes are disengaged and ground speed is less than 30 knots or radio altitude is greater than 200 feet.
   D. All components used by the ILS deviation monitor are part of the Electronic Flight Instrument System (EFIS), (Ref 34-22-00).
   E. The ILS Deviation Monitor function is an option that is selected by hard wiring to ground the EFIS symbol generator ILS DEV WARN pin.
   F. The monitor function is tested during the EFIS system self-test. The self-test is active only when the airplane is on the ground.
2. Operation
   A. The ILS deviation warning option is available whenever the ILS DEV WARN pin (E8 on connector insert B) on the EFIS symbol generators is grounded.
   B. The ILS deviation warning is armed when the following conditions also exist:
      (1) EFIS symbol generator autopilot mode status input is set to logic one (Approach Mode Engaged).
      (2) Radio Altitude between 500 and 100 feet for glideslope.
      (3) Radio Altitude between 500 feet and touchdown for localizer.
   C. The ILS Deviation warning is issued when the above conditions exist and either or both of the following deviations exist for one second (Fig. 1, Part A).
      (1) Glideslope deviation pointer is greater than .87 dot above or below the glideslope beam center.
      (2) Localizer deviation pointer is greater than .27 dot left or right of the localizer beam center.
   D. The EFIS symbol generator changes the white deviation scale for either the EADI glideslope and/or localizer to yellow. The symbol generator also flashes the affected pointer at four hertz.
   E. The ILS deviation warning continues as long as the airplane exceeds the glideslope/localizer limits and the arming conditions exist.



PART A - IN FLIGHT



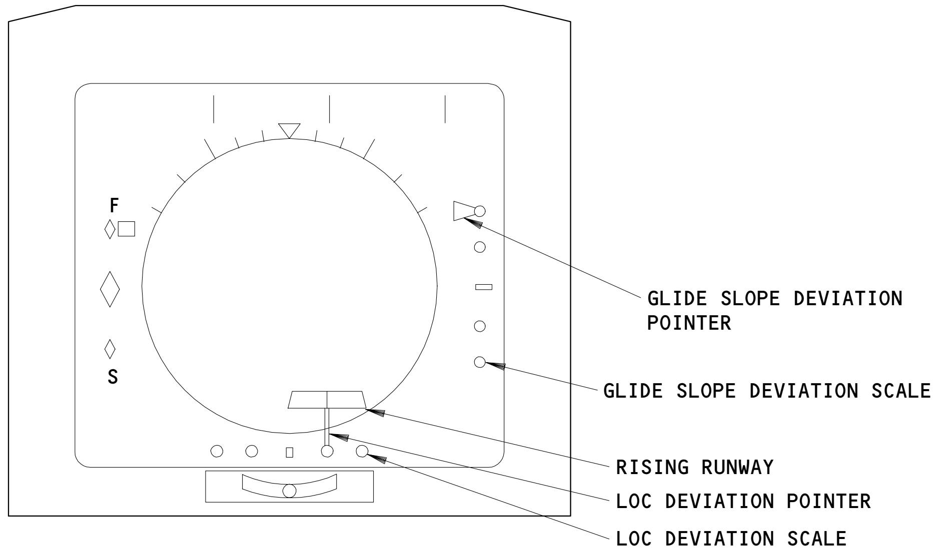
PART B - GROUND TEST

1 TEST MESSAGE "OK" OR "FAIL"

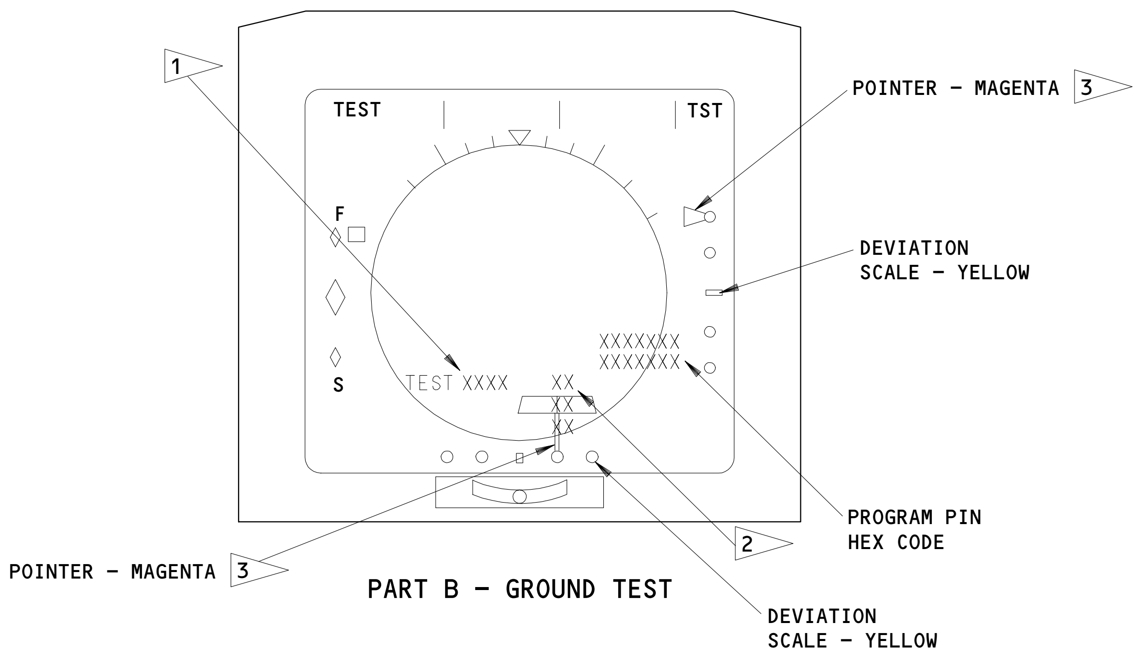
2 FAILURE MESSAGE-CP (CONTROL PANEL); DU (DISPLAY UNIT); AND SG (SYMBOL GENERATOR)

3 POINTERS BLINKING AT FOUR HERTZ

ILS Deviation Monitor - EADI Display
Figure 1 (Sheet 1)



PART A - IN FLIGHT



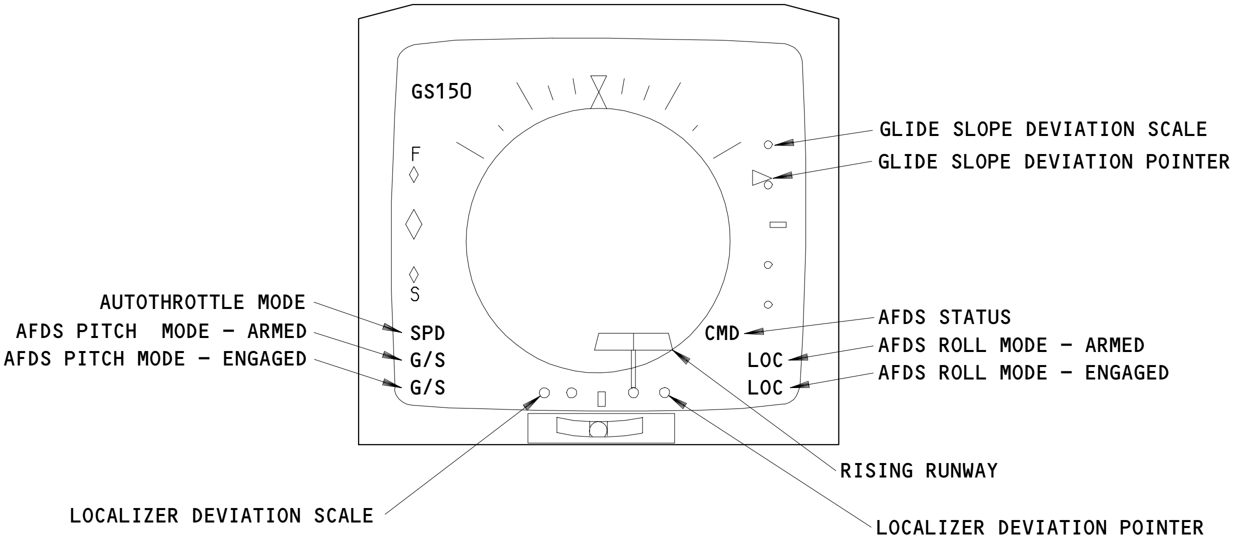
PART B - GROUND TEST

TEST MESSAGE "OK" OR "FAIL"

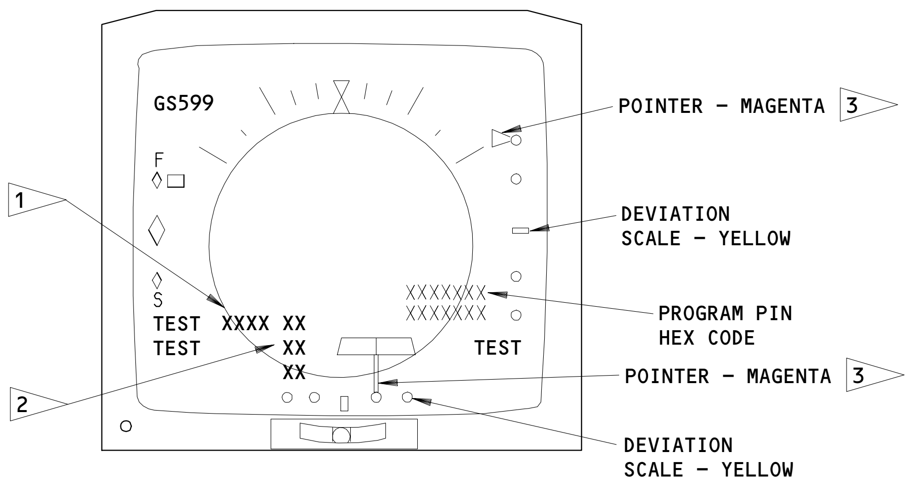
FAILURE MESSAGE - CP (CONTROL PANEL); DU (DISPLAY UNIT); AND SG (SYMBOL GENERATOR)

POINTERS BLINKING AT FOUR HERTZ

ILS Deviation Monitor - EADI Display
Figure 1 (Sheet 2)



PART A - IN FLIGHT



PART B - GROUND TEST

1 TEST MESSAGE "OK" OR "FAIL"

2 FAILURE MESSAGE-CP (CONTROL PANEL); DU (DISPLAY UNIT); AND SG (SYMBOL GENERATOR)

3 POINTERS BLINKING AT FOUR HERTZ

ILS Deviation Monitor - EADI Display
Figure 1 (Sheet 3)

F. Testing (Fig. 1, Part B)

(1) The ILS Deviation monitor can only be tested when the airplane is on the ground. The EFIS test function is disabled by an air/ground relay when the airplane is in flight.

(2) The ILS deviation monitors are tested by simulating the conditions required to enable the monitors during the EFIS self-test. The self-test is initiated by pressing the test button on overhead panel P5 for the master dim and test system, (Ref 33-16-00). When the test button is pressed, a test pattern is displayed on the EADI. The ILS Deviation monitor test causes the localizer pointer to deflect one dot right and the glideslope pointer to deflect two dots up. The deviation scales change from white to yellow and the pointers flash at four hertz. The test is displayed until the test button is released.

ILS DEVIATION MONITOR - ADJUSTMENT/TEST

TASK 22-42-00-705-001

1. ILS Deviation Monitor Test

   A. General

      (1) The ILS Deviation Monitor is part of the EFIS software. To do a test of the ILS Deviation Monitor, refer to AMM 34-22-00/501.