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C-141 All Weather Landing System Engineering Support

Lear Siegler, Inc.

June 1976

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C-141 ALL WEATHER LANDING SYSTEM ENGINEERING SUPPORT

AWLS ENGINEERING SUPPORT GROUP
LEAR SIEGLER, INCORPORATED
MANAGEMENT SERVICES DIVISION
WRIGHT-PATTERSON AFB, OHIO 45433

JUNE 1976

FINAL REPORT: AUGUST 1972 – OCTOBER 1975

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WASHINGTON, D.C. 20590

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WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

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Approach and Landing Radar	

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The systems and equipment described herein augment the capability of the standard C-141 Category II All Weather Landing System (AWLS) to operate in Category III weather conditions. This final report describes the physical location, operation, and theory of operation of the individual systems. A description of the on-board test instrumentation used to document the results of the AWLS test flights is included. A simulation model of the NC-141A test aircraft was developed to facilitate analysis of an AWLS operation on a system mock-up. The aircraft equations of motion which were used and the results of the

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simulation model testing are a part of this report.

FOREWORD

This report describes the All Weather Landing System (AWLS) installed in the program test bed aircraft, NC-141A/61-2775, assigned to the 4950th Test Wing (AFSC) at Wright-Patterson AFB, Ohio. The AWLS Program is a joint United States Air Force/Federal Aviation Administration (FAA) effort for gathering data on the psychological, physiological, and procedural aspects of landing a large turbojet aircraft in actual low visibility weather conditions down to and including Category IIIc weather (zero ceiling, zero visibility). Program management is provided by the Air Force Flight Dynamics Laboratory (AFFDL/FGT) under Project 2187, Low Visibility Terminal Area Operations, with engineering support services provided by Lear Siegler, Inc/Management Services Division (LSI/MSD) under contract F33615-72-C-1358.

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LIST OF ABBREVIATIONS

Abbreviation	Term
A	Ampere
AA	Approach Arm
AA*	Approach Arm Star Term - Enable FID logic chains
AAFF	Approach Arm Flip-Flop
AAI	Approach Arm Internal (determine whether TPLC is in preland or en route test
AC	Alternating Current
ADI	Attitude Director Indicator
AFCS	Automatic Flight Control System
AGC	Automatic Gain Control
AGL	Absolute Ground Level
AIL	Aileron
a_l	Longitudinal Acceleration
ALR	Approach and Landing Radar
ALT HLD	Altitude Hold
AM	Amplitude Modulation
a_n	Normal Acceleration
AP	Autopilot Pitch
A/P	Autopilot
APCP	Autopilot Control Panel
AR	Autopilot Roll
ASN-24	Navigation Computer
ASN-35	Doppler Radar Navigation Computer
ATS	Automatic Throttle System
AVRI	Altitude and Vertical Rate Indicator
AWLS	All Weather Landing System
BSB	Bank Steering Bar
BITE	Built In Test Equipment
C	Comparator
CADC	Central Air Data Computer
C_D	Drag Coefficient
CG	Center of Gravity
C_L	Lift Coefficient
CLK	Clock
COUP	Coupler
CPLR	Coupler Autopilot
cps	cycles per second (Hz)
CRT	Cathode Ray Tube
CW	Continuous Wave
CWS	Control Wheel Steering
D	Drag
dB	decibel
DC	Direct Current
deg	Degree, angular, when <u>not</u> following a number
o	Degree, angular, when following a number
DETMAN	Detent Manual (switching function)
DG	Directional Gyro
DTL	Diode Transistor Logic
EADI	Electronic Attitude Director Indicator
ELEV	Elevator
EPR	Exhaust Pressure Ratio
ERT	En Route Test
ET	Elapsed Time
FAA	Federal Aviation Administration

FD	Flight Director
FDC	Flight Director Computer
$\underline{\text{FDS}}$	Flight Director System
$\overline{\text{FDV}}$	Flight Director not valid
FE	Flare Engage
FID	Fault Identification - lagic chain which may or may not terminate at the FIP
FIP	Fault Indicator Panel
FLAG	Fail Safe Latch AND Gate
FLT DIR	Flight Director
FM	Frequency Modulation
FPAC	Flight Path Angle Computer
FPD	Flight Progress Display (panel)
FSTC	Flare (computer) Self-test command Glideslope
ft	foot
FTCM	Full Time Command Modifier
g	gravity
GHz	Gigahertz
GPIP	Glide Path Intercept Point
GS	Ground Speed
G/S	Glideslope
G/SI	Glideslope Indicator
G/S Desens	Glideslope Desensitizer (switching function)
h	Altitude
$\underline{H}$	The heal portion of an en route or preland test step
$\overline{H}$	The test portion of an en route or preland test step
HA	Horizontal Accelerometer
$\dot{h}_B$	Barometric Altitude Rate
$\dot{h}$	Complemented Altitude Rate
HDG	Heading
hr	hour
HSI	Horizontal Situation Indicator
Hz	Hertz
IAS	Indicated Airspeed
IASG	Intermediate amplitude Selector Gate
IF	Intermediate Frequency
ILM	Independent Landing Monitor
ILS	Instrument landing System
ILRM	Independent Landing Radar Monitor
in	inch
INS	Inertial Navigation System
INU	Inertial Navigation Unit
IRAN	
IRATE	Interim Remote Area Terminal Equipment
ISS	Intermediate Signal Selector
K	Gain
kV	kilovolt
kV·A	kilovolt ampere
kW	kilowatt
kW·h	kilowatt hour
LA	Land Arm
LALSCD	Low Altitude Low Speed Control Development
LAT	Lateral (A/P channel)
LAT OFF	Lateral off
LMV	LOC Manual (Flight Director System) Warning

LOC	Localizer
LOC COUP	Localizer Coupler (switching function)
LSI	Lateral Situation Indicator
LV	Latch Verify
LVFF	Latch Verify Flip-Flop
M	Mach
MC	Master Caution
MDA	Minimum Decision Altitude
MEPADD	Mechanical Path Angle Director Display
MHz	Megahertz
MLS	Microwave Landing System
mV	millivolt
MW	Megawatt
uA	Microampere
um	micron (micrometer)
uV	Microvolt
NAFEC	National Aviation Facilities Experimental Center
NAV SEL	Navigation Select
n mi	Nautical mile
ns	nanosecond
NSP	Navigation Selector Panel
OPSD	One Program Step Detector
OTCD	One Test Command Detector
P	Pitch
PC	Program Control
PCFF	Program Control Forced Fault
PCFH	Program Control Forced Heal
PCI	Peripheral Command Indicator
PCWS	Pitch Control Wheel Steering
PIFAX	Pilot Control Display Factors Program
PLT	Preland Test
PM	Power Monitor
PMW	Pitch Manual (FDS) Warning
qc	Dynamic Pressure
R	Roll
RCD	Rudder Command Display
RCWS	Roll Control Wheel Steering
RDR	Runway Distance Remaining
RDR ALT	Radar Altimeter
RF	Radio Frequency
R/G-A	Rotation/Go-Around
RI	Roll Input
RMW	Roll Manual (FDS) Warning
RO	Roll Output
RSFP	Roll Synchronizer Fixed Phase
R/T	Receiver/Transmitter
RVR	Runway Visual Range
S	La Place Transform Variable
SEL	Select
SIM	Simulated
STA or B	Strobe A or B
STATE	Simplified Tactical Approach and Terminal Equipment
SVF	Supervalidity Flare
SVP	Supervalidity Pitch
SVR	Supervalidity Roll
T	Thrust
TACAN	Tactical Air Navigation

TALAR	Tactical Approach and Landing Radar
TAS	True Airspeed
T. O.	Take-off
TPLC	Test Programmer and Logic Computer
$\dot{V}$	Acceleration
V	Volt
VAL	Validity
V·A	Volt Ampere
VERNAV	Vertical Navigation
VHF ANV	Very High Frequency Navigation System
V_{CO}	Climb out Velocity
V_{LOF}	Lift off Velocity
VOR	Very High Frequency Omnidirectional Radio Range
V_{ROT}	Rotation Velocity
V_S	Stall Velocity
VSI	Vertical Scale Flight Instrument
VRMS	Volt Root Meanc Square
VVI	Vertical Velocity Indicator
W	Weight
Weather	Minimums
ZERO	Visibility Landing Study

LIST OF SYMBOLS

SYMBOL	DIMENSION	DEFINITION
A1	inches	Roll Control (Lateral Stick Deflection)
B	Hertz	Bandwidth of Course/Path Noise
B1	inches	Pitch Control (Longitudinal Stick Deflection)
b	feet	Wing Span
C_D	rad^{-1}	Drag Coefficient
C_L	rad^{-1}	Lift Coefficient
C_m	rad^{-1}	Pitching Moment Coefficient
C_n	rad^{-1}	Yawing Moment Coefficient
C_y	rad^{-1}	Side Force Coefficient
C_α	rad^{-1}	Rolling Moment Coefficient
c	feet	Aerodynamic Chord Length
c.g.	% MAC	Center of Gravity in % Mean Aerodynamic Chord
DLR	inches	Yaw Control (Rudder Pedal Deflection)
g	ft/sec^2	Acceleration Due to Gravity
h	feet	Altitude Pertubation
I_x, I_y,	$slug\text{-}ft^2$	Moments of Inertia About the x, y, and z Axes
I_{xz}	$slug\text{-}ft^2$	Product of Inertia
K_D, K_I		Gain Parameters
K_θ	volts	Pitch Gain
K_Θ	volts	Roll Gain
K_ψ	volts	Yaw Gain
M	ft-lb	Pitching Moment
M_{B1}		$\partial M/\partial B1$
M_q		$\partial M/\partial q$
M_u		$\partial M/\partial u$

SYMBOL	DIMENSION	DEFINITION
M_w		$\partial M/\partial w$
MM		Middle Marker
m	lb-sec^2/ft	Mass of the Aircraft
m_N	ua	Mean Value of Course/Path Noise
N	ft-lb	Yawing Moment
N_{DLR}		$\partial N/\partial DLR$
N_p		$\partial N/\partial p$
N_r		$\partial N/\partial r$
N_v		$\partial N/\partial v$
n		Number of Degrees of Statistical Freedom
OM		Outer Marker
p	rad/sec.	Roll Rate
q	rad/sec.	Pitch Rate
R		Desensitization Ratio
Rwy		Runway Threshold
r	rad/sec.	Yaw Rate
s		Laplace Operator ($s = \sigma + j\omega$)
TD		Touchdown Point
t	sec	Time
U_o	ft/sec.	Steady State Velocity (x-Axis)
u	ft/sec	Perturbation Velocity (x-Axis)
u_g	ft/sec.	Gust Velocity (x-Axis)
V	ft/sec.	Trim Airspeed
v	ft/sec	Perturbation Velocity (y-Axis)
W	lb	Gross Weight

SYMBOL	DIMENSION	DEFINITION
w	ft/sec	Perturbation Velocity (z-Axis)
X	lb	Force in the x-Direction
X_q		$\partial X/\partial q$
X_u		$\partial X/\partial u$
$X_{B}1$		$\partial X/\partial B1$
x		Horizontal Displacement (x-Axis)
Y	lb	Force in the y-Direction
Y_{A1}		$\partial Y/\partial A1$
Y_p		$\partial Y/\partial p$
Y_r		$\partial Y/\partial r$
Y_v		$\partial Y/\partial v$
y, $\bar{y}$		Lateral Displacement in the y-Axis
Z	lb	Force in the z-Direction
Z_{CO}		$\partial Z/\partial CO$
Z_q		$\partial Z/\partial q$
Z_u		$\partial Z/\partial u$
Z_w		$\partial Z/\partial w$
α	degree	Angle of Attack
$\dot{\alpha}$	degree/sec.	Angle of Attack Rate of Change
α_{LOF}	degree	Lift-Off Angle of Attack
α_a	degree	Augmented Angle of Attack
α_e	degree	Angle of Attack Error
α_p	degree	Porgrammed Angle of Attack
α_v	degree	Vane Angle of Attack
β	degree	Sideslip Angle
γ		Flight Path
δ	degree	Control Surface Angle

SYMBOL	DIMENSION	DEFINITION
δ_a	degree	Aileron Position
δ_e	degree	Elevator Position
δ_f	degree	Flap Position
δ_r	degree	Rudder Position
ζ		Damping Factor
η	degree	Glidepath Elevation Angle
θ	degree	Pitch
$\dot{\theta}$	degree	Pitch Rate
θ_c	degree	Pitch Command
$\theta_{g.s.\ trim}$	degree	Trim Pitch (Glideslope)
J		$\sqrt{-1}$
λ	inches	Bar Deflection
ξ		Expected Value $(\lvert m\rvert + \lvert\sigma\rvert)$
ξ_N	μa	Expected Value, Course/Path Noise
ξ_y	ft	Expected Value, Lateral Track Error
ρ	slug-ft^3	Atmospheric Density
σ		Standard Deviation
σ^2		Variance
σ_N	μa	Standard Deviation, Course/Path Noise
σ_N^2	μa^2	Variance of Course/Path Noise
$(\bar{\sigma}_N)^2$	a^2	Average Variance of Course/Path Noise
σ_y	feet	Standard Deviation, Lateral Track Error
$\sigma_{\delta a}$	degree	Standard Deviation, Aileron Position
σ_θ	degree	Standard Deviation, Roll Attitude
σ_ϕ	degree	" Deviation, Pitch Attitude

SYMBOL	DIMENSION	DEFINITION
σ_{Δ_h}	feet	Standard Deviation, Vertical Track Error
τ	second	Time Delay
ϕ	degree	Roll Attitude
ϕ_c	degree	Roll Command
ψ	degree	Yaw Angle
ω	rad/sec.	Natural Frequency
Δ		Transfer Function Denominator
Δ_h	feet	Vertical Off-set of Aircraft from Ideal Glideslope
Ω	rad/sec.	Rotational Speed
$<$		Less Than
$>$		Greater Than
$<<$		Much Less Than
$>>$		Much Greater Than
∂		Partial Derivitive
$=$		Equal
$\simeq$		Approximately Equal
$\mid \mid$		Absolute Value

SECTION I

INTRODUCTION

1.1 PURPOSE

The purpose of this final report is to describe the system operation and the equipment installed into the C-141A test aircraft, #61-2775 by Lear-Siegler Incorporated as pertaining to the development and implementation of the All Weather Landing System (AWLS) as specified by Contract F73615-726-1358.

1.2 REPORT BREAKDOWN

This report is divided into eight (8) sections as follows:

Section I - INTRODUCTION: This section provides the background information on the AWLS program.

Section II - AIRCRAFT SYSTEMS CONFIGURATION: This section provides the physical description and location as well as the functional description of the various equipment installed in the test aircraft.

Section III - SYSTEMS OPERATION: This section provides the overall operation and the landing profile operation of the AWLS equipment.

Section IV - AWLS SYSTEM DESCRIPTION: This section provides the fundamental block diagram theory of operation of the AWLS and the associated systems.

Section V - TEST INSTRUMENTATION DESCRIPTION: This section provides a description of the test instrumentation aboard the test aircraft as well as the parameters measured.

Section VI - EADI/ALR DESCRIPTION: This section provides the fundamental block diagram theory of operation of the EADI and the ALR systems.

Section VII - ANALYSIS: This section provides the data on the simulation model and documents the results of the simulation model testing.

Section VIII - CONCLUSION

1.3 BACKGROUND INFORMATION

Following the modifications (performed at Mobile, Alabama in 1972), the aircraft was returned to Wright-Patterson Air Force Base on 1 November 1972. The aircraft was then grounded for correction of numerous aircraft and AWLS problems, including an elusive fuel leak.

At this time, several efforts were simultaneously started as follows:

a. Optimization of the AWLS and associated instrumentation.

b. Starting of a failure/degraded performance computer simulation.

c. Integration of an Approach Landing Radar (ALR) and a Sperry Electronic Attitude Director Indicator (EADI) into the aircraft.

It was soon discovered that an electrical loading problem existed between the INS and the pilot's Bearing, Distance, Heading Indicator (BDHI) which required the installation of ME-1A Synchro Amplifiers. It was also found that the application of test power would cause the #1 Generator to drop off line. This problem was solved by rewiring the main test power source to the downstream side of the #1 Generator Current Sensor. These, along with several other problems, were corrected during the period of 18 January 1973 to 30 January 1973.

Following the installation of the ME-1A Amplifiers, it soon became apparent that the AWLS components were below the standards of Category III operation. The entire Category III (CAT III) system had been developed piece-meal by extensive modifications to the AWLS computers and the addition of a prototype Decrab Computer. At this time, modifications had been incorporated into the various sub-systems and the following items had become "one-of-a-kind" items:

a. The AFCS Coupler.

b. The AFCS Yaw Damper.

c. The AFCS Aileron Computer.

d. The pilot's and co-pilot's Flight Director Computer.

e. The pilot's and co-pilot's ADIs.

Maintenance on these items would have been extremely time consuming and expensive as the components had been in use since 1968, and both time and handling had taken their toll. The cost for repairs exceeded the purchase price of some items (ADI principally) and since some parts would have taken upward of 3 months to acquire, significant program delays would have resulted.

These factors prompted a second Class II Modification in which the following was accomplished:

a. The below listed items became standard CAT II AWLS components:

(1) The AFCS Coupler (modified for flare optimization).

(2) The AFCS Yaw Damper.

(3) The AFCS Aileron Computer.

(4) The pilot's and co-pilot's Flight Director Computers.

b. The pilot's and co-pilot's ADIs were replaced with Sperry AD 350B ADIs.

c. The Decrab Computer and associated equipment were removed.

d. The ADI interface and side-slip/roll-out functions were integrated into a newly designed CAT III Adapter that use State-of-the-Art electronics.

The entire modification was accomplished between 1 April 1973 and 18 April 1973 with a functional flight check being flown on 19 April 1973. That the successful completion of the modifications were completed in such a short period of time is testimony to the effort and co-operation between the AFFDL, 4950th TW and the support contractor, Lear-Siegler Incorporated.

During and following the Class II Modification, the AWLS was configured as separate but dual systems. In addition, an excessive lateral and vertical rate annunciation system was provided and a Runway Distance Remaining (RDR) Indicator was installed.

SECTION II

AIRCRAFT SYSTEMS CONFIGURATION

2.1 PURPOSE

The purpose of this section is to detail the physical location and the functional characteristics of the various systems that have been installed into the C-141 test aircraft (Figure 1) as part of the AWLS program.

(Figure 1) C-141 Aircraft

2.2 OVERALL CONFIGURATION

Numerous Class II Modifications have been performed during the course of the AWLS program. Figure 2 provides the overall physical layout of the equipment currently configured into the aircraft.

It is evident in this illustration that considerable component relocation has been done. The illustration also provides an overview as to the various stations and consoles that have been installed to facilitate the instrumentation and monitoring of the system performance.

Figure 3 provides an overall block diagram of the system that details the basic functional integration of the systems and computers.

2.3 DETAILED SYSTEMS CONFIGURATION

The following paragraphs describe the specific configuration changes with respect to location and function.

2.3.1 Cockpit Instrument Panel Configuration

2.3.1.1 C-12/INS Switch

This two (2) position switch is located on the pilot's side console to the rear. The C-12 position provides the compass reference data from the standard C-12 Compass System and attitude reference from the MD-1 Displacement Gyros. In the INS position, both attitude and heading reference are provided by the dual LTN-51 inertial navigation systems.

2.3.1.2 LOC G/S SMOOTH Indicator Lamp

This red indicator lamp is labeled LOC G/S SMOOTH and is located on the small indicator/switch panel as shown by Figure 4. During AWLS operations, the indicator lamp will illuminate when the AWLS is armed and will extinquish at Approach Arm (APP ARM) if the Inertial Smoothing Circuitry is activated. Illumination of this indicator lamp after APP ARM and while the AWLS is tracking the ILS beam will indicate that a failure in the Inertial Smoothing Circuitry has occured.

Control switches have been added to the AWLS Project Junction Box to enable the switching of the LOC and G/S Smoothing Circuits in and out of the system for flight test flexibility.

2.3.1.3 Excessive Rate Indicator Lamp

This amber indicator lamp is labeled EXCESS RATE and is located on the small indicator/switch panel as shown by Figure 4. The indicator lamp's circuitry is activated at Flare to provide a warning to the pilot when an unsatisfactory rate has developed relative to the runway (either laterally or vertically). The Lateral Rate Circuit will trip if the rate is 4 ft/sec or more and is sensed by the INS. The Rate Indicator Lamp will illuminate if the Vertical Rate Circuits detect a difference of greater than 1-degree of Pitch Attitude Command between the #1 and #2 flare computations or if the Flare Command exceeds 2 1/2-degrees pitch up or 1-degree pitch down.

2.3.2 Electronic Attitude Director Indicator (EADI) System Configuration

2.3.2.1 Symbol Generator Unit

The Symbol Generator Unit (SGU) is located in the cargo compartment at station 508. The function of the SGU is to receive analog and discrete data from the flight control and avionic systems, process it with internally generated symbology and feed it to the EADI display as synthetic display symbology.

2.3.2.2 Electronic Attitude Director Indicator Display (EADI)

The EADI (Figure 5) is located on the center cockpit instrument panel and provides a black and white presentation. The primary function of the EADI is to provide display of the aircraft's attitude, the navigational cues, and steering commands based on signals from the SGU.

Secondary functions include: failure warning annunciation, approach progress annunciations, a manual TEST switch and a manual brightness/contrast (BRT CONT) control to vary the CRT brightness.

2.3.2.3 EADI Control Unit (Figure 6)

The EADI Control Unit (located on the co-pilot's right side console) provides the control inputs to the SGU for mode selection (NORMAL, ATTITUDE, TV, and TEST), flight path, and radar altitude reference. The Control Unit also provides the Radar Altimeter (R/A TEST) Test Switch to test the Radar Altimeter.

2.3.3 Approach and Landing Radar (ALR)

The test aircraft has been equipped with an Approach Landing Radar (ALR) which is a high-resolution, short-range, KA band radar. The radar display is superimposed over the attitude/director information on the EADI to provide "real world" information to the pilot during the Final Approach, Landing and Roll-out/Taxi phases.

2.3.3.1 Antenna Assembly (Figure 7)

The Antenna is externally mounted beneath the aircraft's nose radome and contains the Phased Array Antenna and a high-power, micro-wave, receiver/transmitter.

2.3.3.2 ALR Equipment Rack (Figure 8)

The ALR Equipment Rack is located in the cargo compartment at station 1038. Included in this rack is the following:

a. The Display Signal Generator.

b. The ALR Control Panel.

c. The Video Processor.

d. The IF Signal Processor.

e. The Display.

2.3.4 Test Instrumentation

The test instrumentation is located throughout the aircraft and includes a Test Instrumentation Console at station 698 (Figure 9). This console contains the following:

a. An Automatic Digital Data Acquisition System (ADDAS)..

b. An eight (8) channel Brush Recorder.

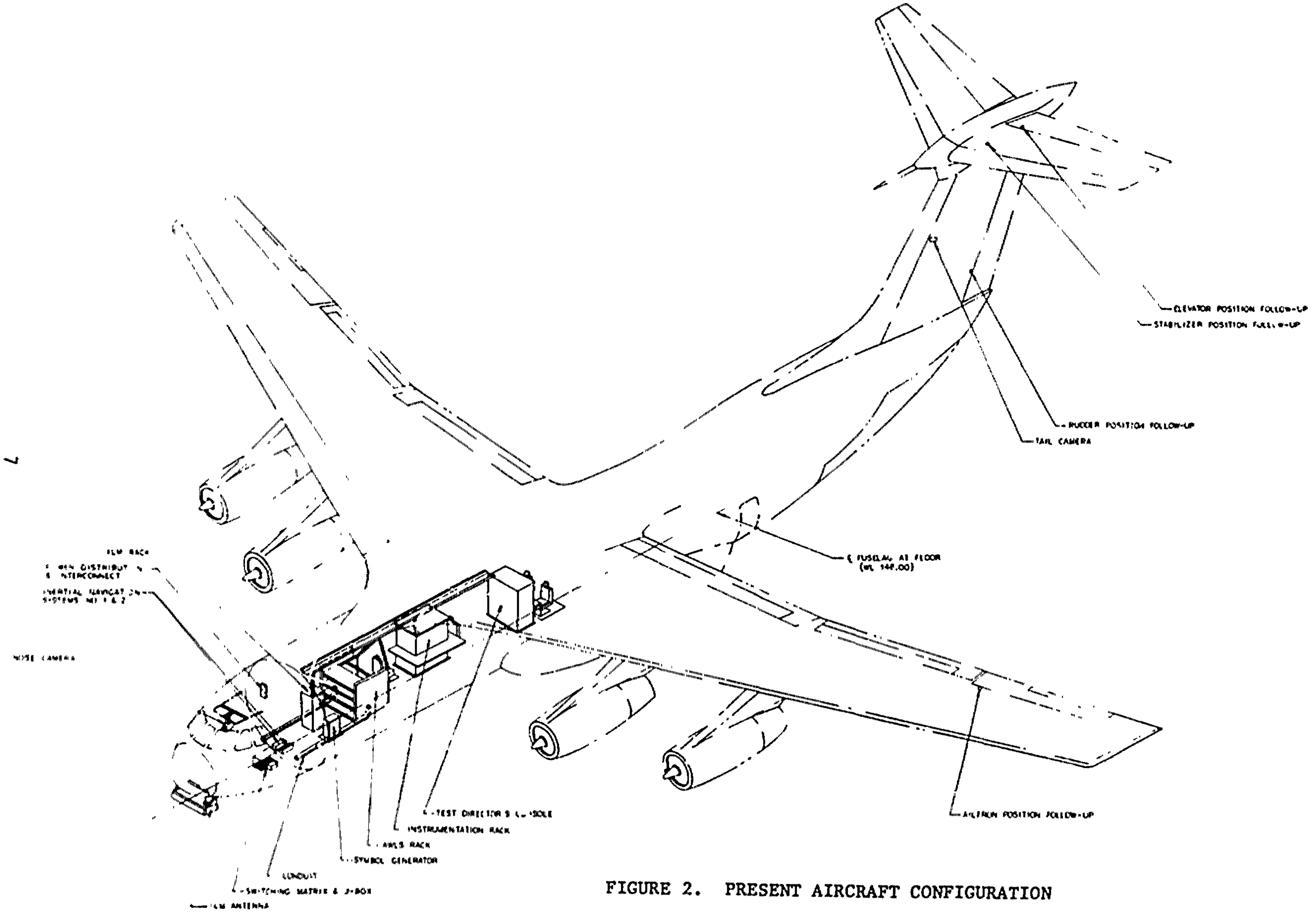


FIGURE 2. PRESENT AIRCRAFT CONFIGURATION

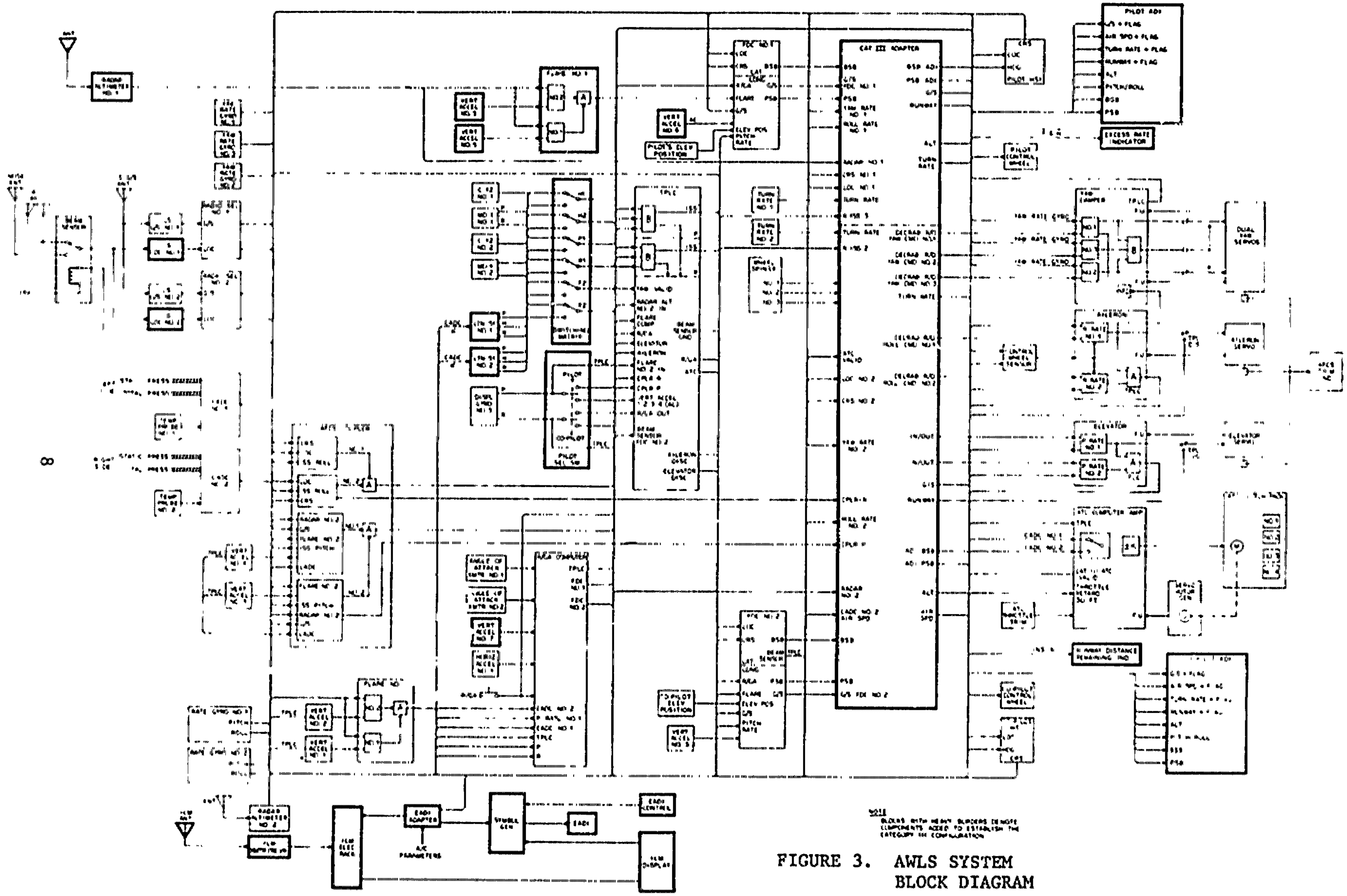
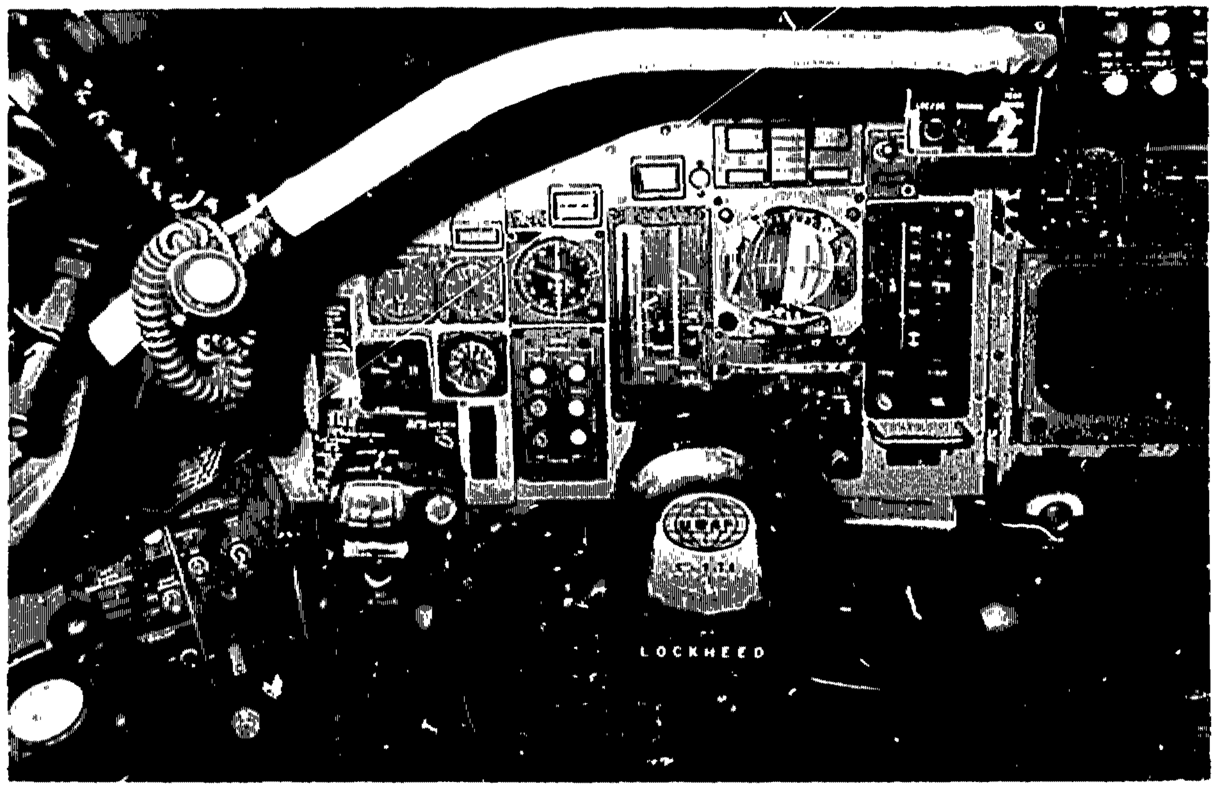
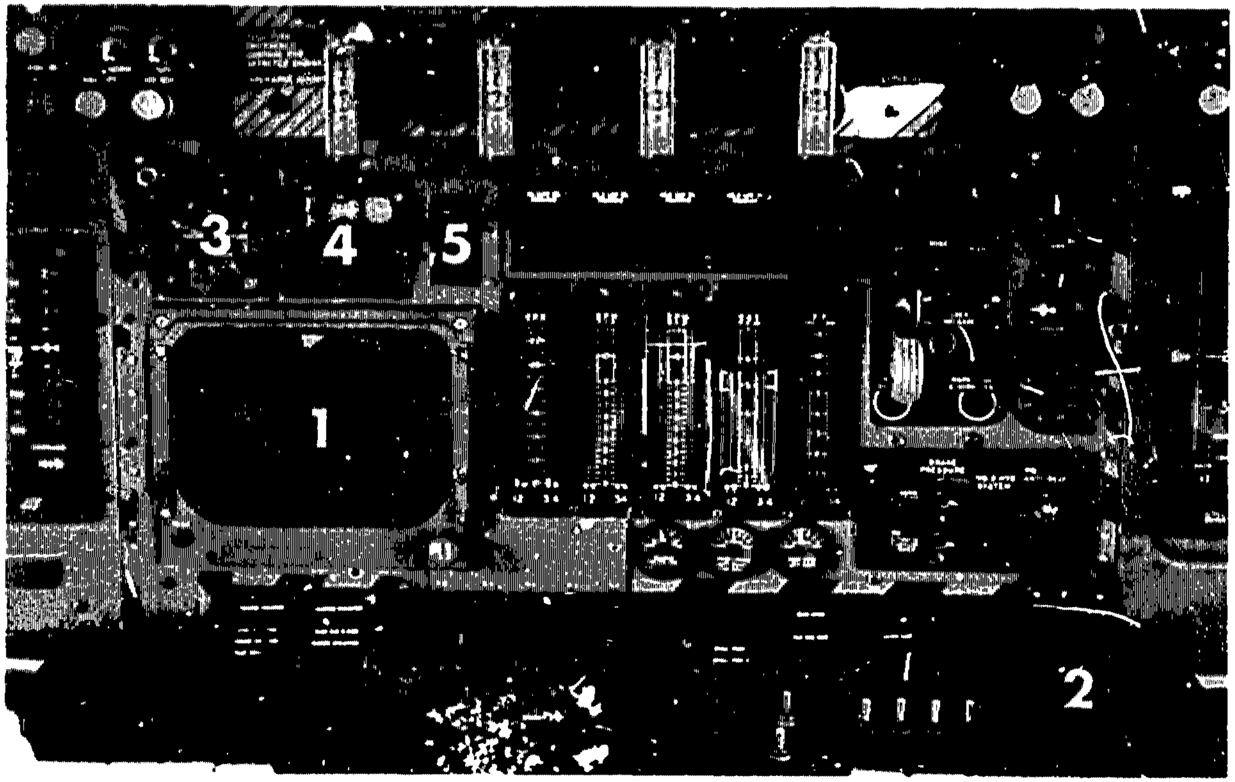


FIGURE 3. AWLS SYSTEM BLOCK DIAGRAM



1. LOC/GS SMOOTH INDICATOR
2. EXCESSIVE RATE INDICATOR

FIGURE 4. COCKPIT INSTRUMENT PANEL CONFIGURATION



1. EADI
2. AWLS FAULT PANEL
3. RADAR ALTIMETER
4. AUTO-PILOT SERVO EFFORT INDICATOR
5. FAULT PANEL (AWLS)

FIGURE 5. CENTER INSTRUMENT PANEL CONFIGURATION

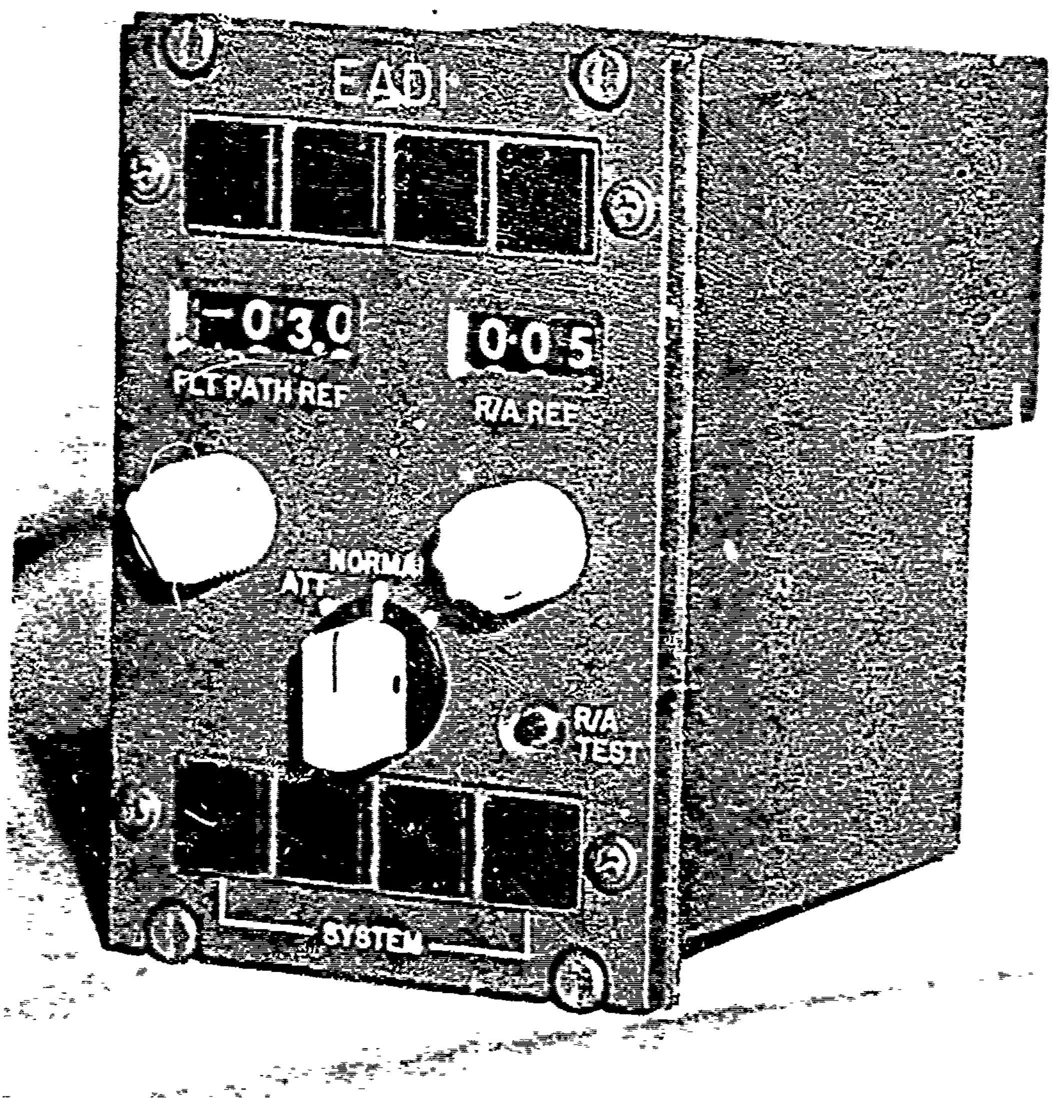


FIGURE 6. EADI CONTROL UNIT

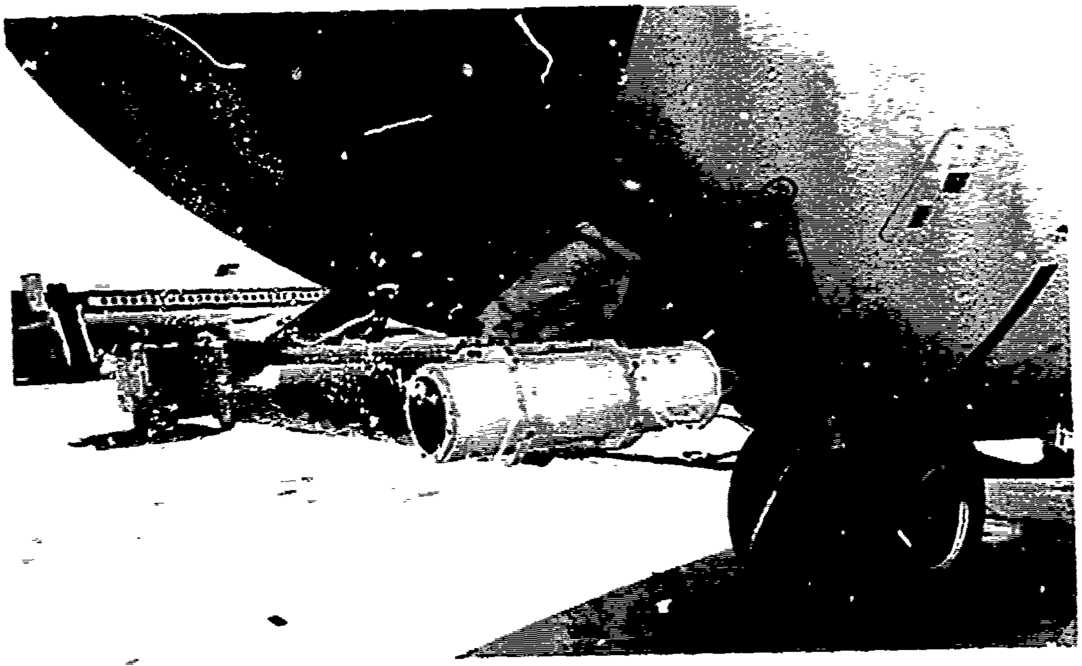


FIGURE 7. ALR ANTENNA

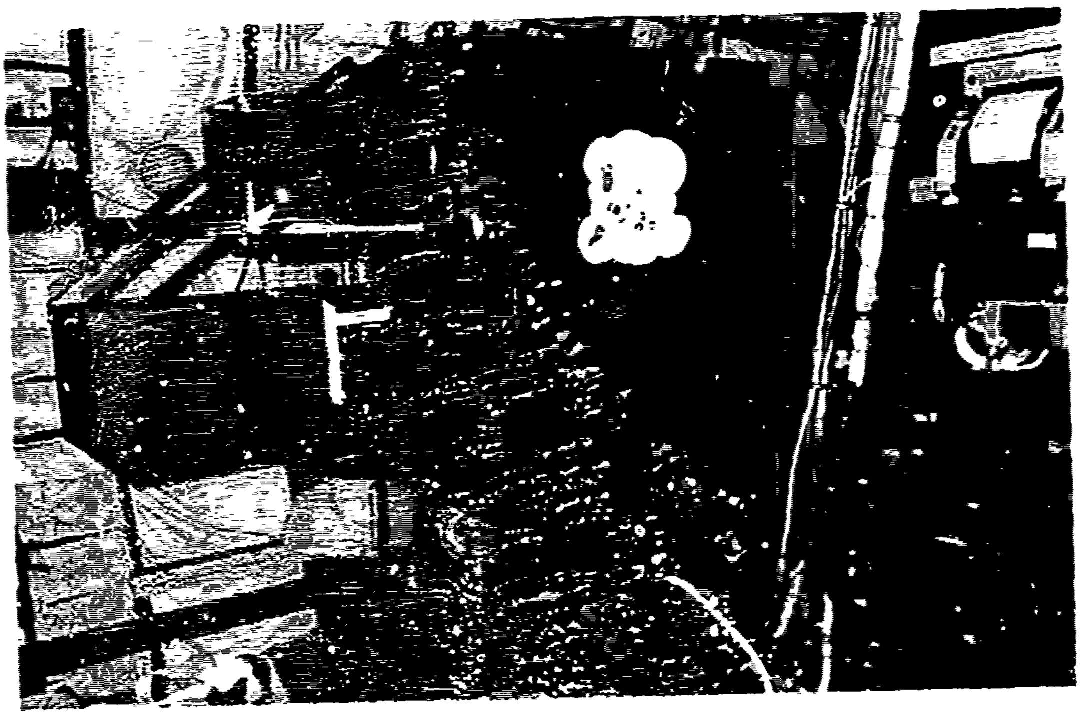


FIGURE 8. ALR EQUIPMENT RACK

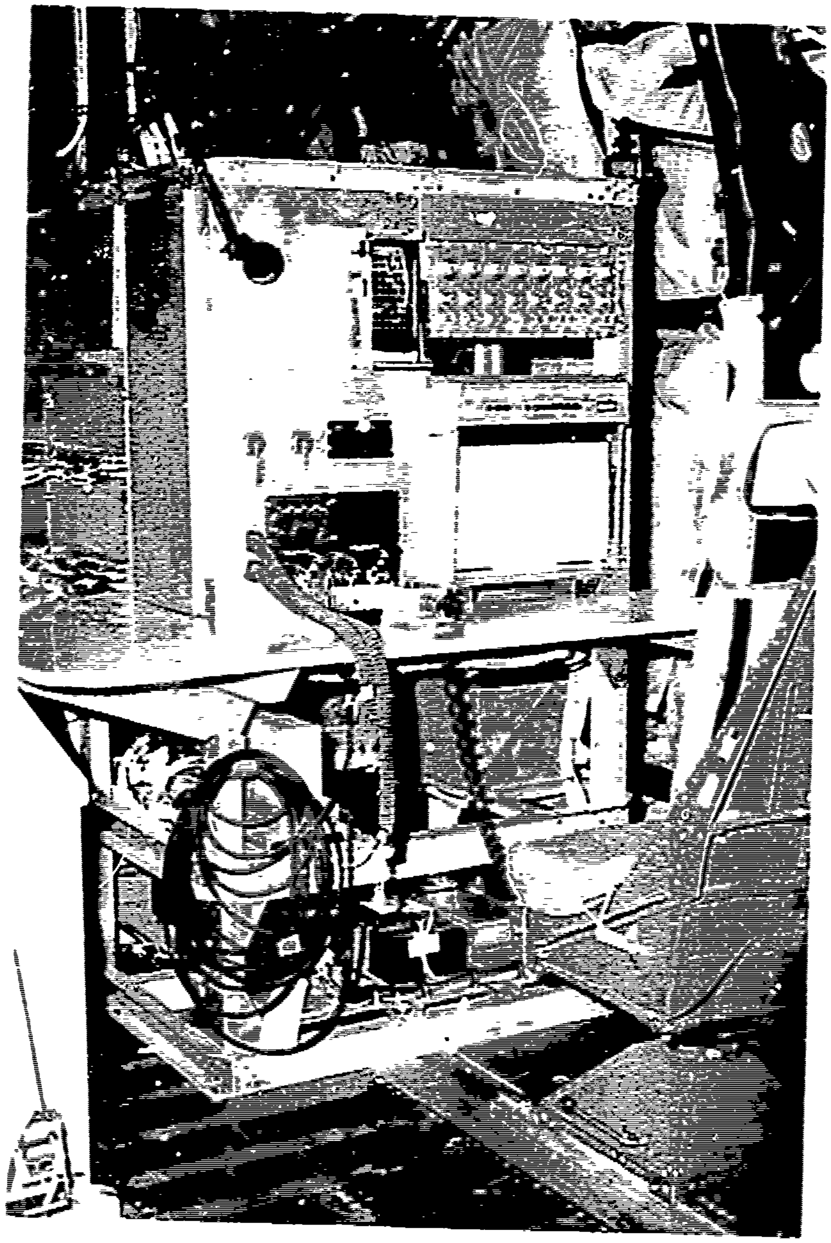


FIGURE 9. TEST INSTRUMENTATION CONSOLE CONFIGURATION

c. A Time Correlation Unit.

2.3.4.1 Automatic Digital Data Acquisition System (ADDAS)

The ADDAS is a magnetic tape recorder with 162 data channels. The controls are located on the front panel of the recorder. Associated to a specific value ($\pm$5 Vdc) to facilitate the analog-to-digital conversion performed by this unit.

2.3.4.2 Brush Recorder

The Brush Recorder is a hot-pen type unit with eight (8) channels. An associated patch panel provides the ability to receive a direct read-out of eight of the ADDAS channels.

2.3.4.3 Visicorder Oscillograph

This unit, mounted on the AWLS Rack, is a 24-channel, direct read-out, recorder. Seventeen (17) channels are used for monitoring system comparator performance and as an aid for system troubleshooting.

2.3.4.4 Closed-Circuit Color Television System

A complete closed-circuit Color Television System is installed aboard the test aircraft and consists of the following:

a. Color Camera - The Camera is a tri-color, balanced, two-tube, type with an f 1.8 lens coupled to a zoom-type lens. The Camera is located at station 345.

b. Sychronizing Generator - This unit, mounted at the Test Director's Console, accepts the output of the Camera and syncs the signal into the Recorder and/or Monitor.

c. Television Monitor - This unit is mounted on the Test Director's Console to display the events being seen by the Camera or the EADI presentation to the test director.

d. Video Recorder - The Video Recorder is located at the Test Director's Console and is a solid-state, color, video cassette that is equipped with auto-tape threading, footage counter and V/AGC. This unit will instantaneously record and/or play back through the Monitor, any presentation from either the Camera or the EADI as selected by the test director.

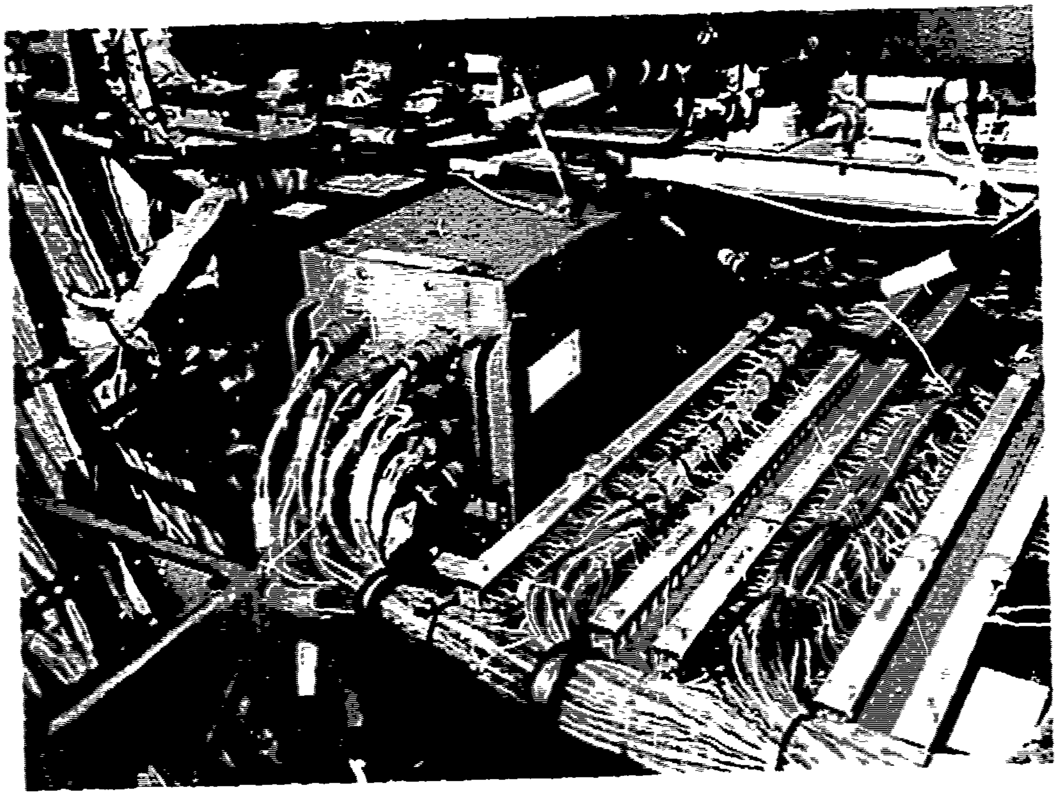
2.3.5 Switching Matrix

The Switching Matrix (Figure 10) is located in the underdeck area and provides the switching for the following:

a. Attitude reference signals between the INS #1/MD-1 Vertical

Gyro #1 and the INS #2/MD-1 Vertical Gyro #2.

b. Heading reference between the INS #1/C-12 Compass #1 and the INS #2/C-12 Compass #2.



(Figure 10) Switching Matrix

2.3.6 Test Director's Console

The Test Director's Console (Figure 11) is located between stations 879 and 898 and will provide the test director with the following monitors that have not previously been described:

a. An ADI.

b. An HSI.

c. An AWLS Progress and Failure Indicator.

d. An LTN-51 #1 Control Display Unit.

e. An Altitude-Vertical Velocity Indicator.

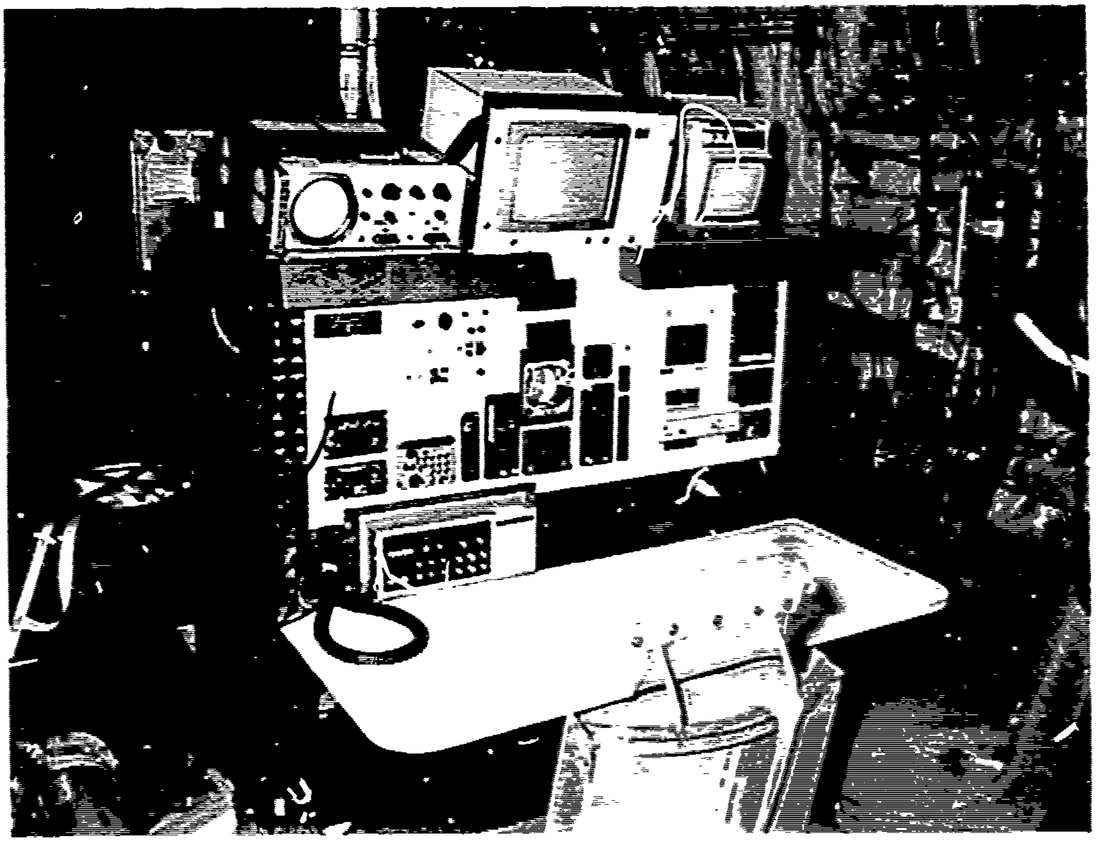
f. A Radar Altimeter.

g. A Digital Time Indicator.

h. A Mach/Airspeed Indicator.

i. An EPR Indicator.

j. An EADI.



(Figure 11) Test Director's Console

2.3.7 Runway Distance Remaining Indicator (RDR)

The RDR (Figures 12 and 13) accepts data from the INS and converts it to an analog display form. The Radar Altimeter triggers the RDR at Flare when the aircraft is essentially over the runway threshold prior to Touchdown. The RDR will then indicate the amount of runway remaining.

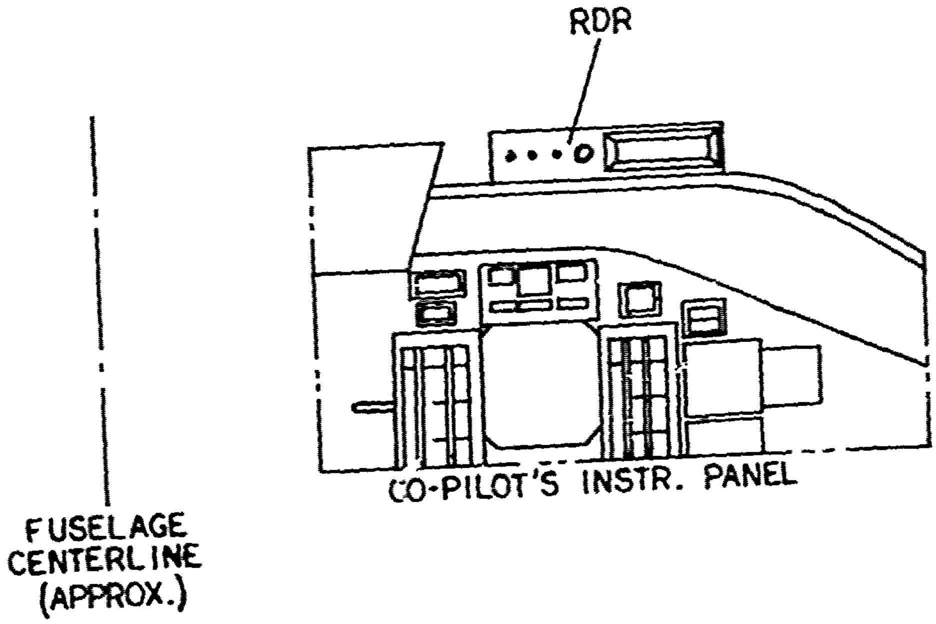


FIGURE 12. RDR LOCATION

FIGURE 13. RUNWAY DISTANCE REMAINING INDICATOR (RDR)

2.3.8 AWLS Rack

Located in the cargo compartment at station 598, this installation was designed to allow for the relocation of critical computers to a more accessable area. In addition, it provides junction boxes which enable ready access to critical parameters, interconnect points and logic. The Rack (Figure 14) contains the following:

a. The AFCS Yaw Damper Computer.

b. The AFCS Aileron Computer.

c. The Elevator Computer.

d. The AFCS Coupler.

e. The #1 and #2 Flare Computers.

f. The Rotate and Go-Around Computers.

g. The #1 and #2 Flight Director Computers.

h. The #1 and #2 Magnetic Auxiliary Heading Units.

i. The CAT III Adapter.

j. The CAT III Logic Box.

k. The EADI Adapter.

l. The ADI Power Transformer.

m. The Precision Altitude and Airspeed Transducers.

n. The 60 Hz. Converter.

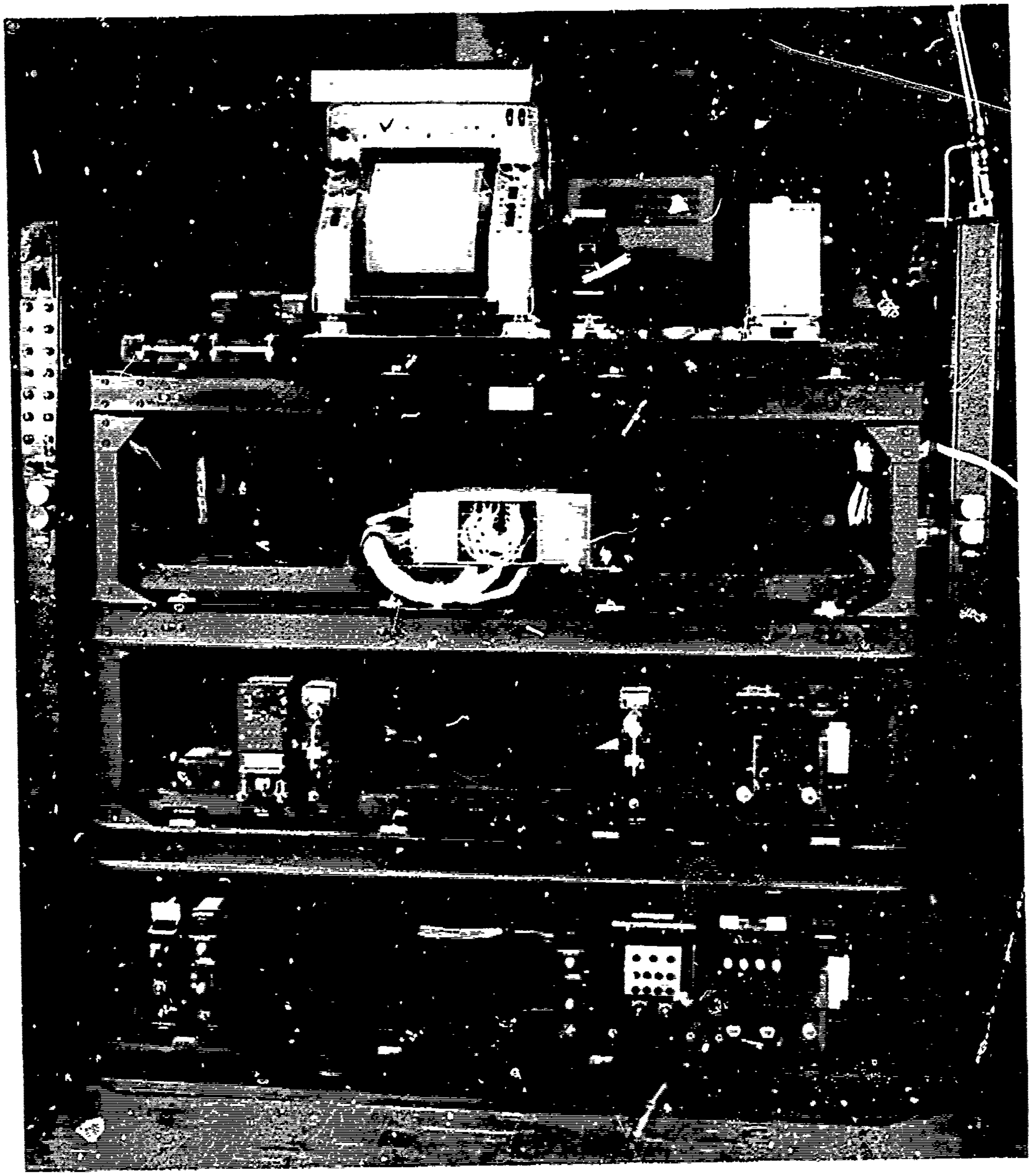


FIGURE 14. AWLS RACK

SECTION III

SYSTEMS OPERATION

3.1 PURPOSE

The purpose of this section is to provide an operational overview of the AWLS and the associated systems. The GENERAL SYSTEMS OPERATION is based upon the operation of the system during Approach and Landing. A landing profile (Figure 15) is used in support of the text. Photographs of the critical pilot instruments are superimposed at the key points of the Approach and Landing. Representative airspeed and time intervals are shown and the system switching functions are explained as the Approach progresses.

3.2 GENERAL SYSTEMS OPERATION

To accomplish an AWLS Approach (Category III), the pilot will establish the aircraft on a 22-degree beam intercept heading and prepare for Localizer Capture ("1" on the profile) as follows:

a. Set the intercept heading with the pilot's HSI HEADING SET knob.

b. Tune the ILS for the airfield frequency and set localizer course.

c. Extend the flaps to the "Approach" position.

d. Lower the landing gear.

e. Set the AUTOPILOT switch to "on".

f. Set the NAV SEL/LAT OFF switch to NAV SEL.

g. Set the AWLS switch to ARMED.

h. Set the HDG SELECT/NAV switch to NAV.

Localizer Capture begins at approximately 2 dots deviation ("2" on the profile). At Beam Capture, the Preset Course and Localizer Deviation signals are fed to the roll channel and the Preset or Compass Heading Signal is removed. The abrupt level changes that occur are smoothed by a Command Modifier in the AFCS. A Roll Command Crossfeed is then integrated into the yaw axis and the roll gain is doubled. This crossfeed ensures proper turn coordination (coupled with the increased roll gain) which results in a precise localizer track.

When the autopilot set for the 22-degree intercept heading, the system will maintain this heading at Beam Capture. As Beam Closure continues, the Localizer Signal decreases and the Preset Course signal dominates as the aircraft heading moves off the 22-degree limit toward alignment with the runway. At a value of $\leq$ 15-degrees of Δ Preset Course and $\leq$ 1/3 dot deviation, the system switches to the LOC Track mode and causes the LOC Progress Indicator Lamp to illuminate. This is also an indication that the LOC system has switched from the tail antenna to the nose antenna.

At this point ("3" on the profile), the Preset Course Error is washed out and fed to the LOC Track Circuits as a damping signal. The Localizer Deviation becomes the predominate signal and the aircraft is allowed to establish any heading consistant with the flying of the center of the Localizer Beam (cross-wind correction).

Just prior to Glideslope Intercept (1/3 dot), the pilot performs the following:

a. Extends the flaps to the "Landing" position.

b. Reduces the airspeed to the "Approach" speed.

c. Re-engages the Auto-Throttle System (ATS) if desired.

Glideslope Intercept occurs at 8-uA (near beam center) at an altitude of approximately 1500-feet Above Ground Level (AGL) ("4" on the profile). The following events then occur at Glideslope Intercept:

a. The G/S Progress Indicator Lamp will illuminate.

b. The G/A ARM Indicator Lamp will extinguish.

c. The Altitude Hold Switch drops out.

d. The ADI Pitch Steering Bar enters the field-of-view.

e. The bank limits are re-set from 30-degrees to 7.5-degrees.

f. The roll rate limits are re-set from 4.8-degrees to 20-degrees per second.

A Pre-Land Test is automatically initiated by either Flight Director's Glideslope Engage Logic and will cause the following:

a. The faulting and healing of the system comparators.

b. Initiation of the Flare Computer Self-Test.

Changes in the autopilot will be initiated and consist of the following:

a. The Land and Land Model functions are activated.

b. The washed out Preset Course Signal is sharply attenuated.

c. A Lag Roll Term is introduced as the beam damping function.

d. The LOC Integrator is enabled.

e. All previously selected modes in the pitch channel are replaced by the Glideslope Command (with acceleration damping).

Upon successful completion of the Pre-Land Test (30-seconds duration), the pilot's Approach Arm (AA) Indicator Lamp will illuminate ("5" on the profile) and the following will occur:

a. The aircraft is considered on-track for both localizer and glideslope.

b. The Model Channel (for monitoring) is on-line.

c. The TPLC (Test Programmer and Logic Computer) will use any faulted comparator condition for purposes of pilot display and automatic disengagement of the faulted AFCS axis.

d. The two (2) comparators in the CAT III Adapter are on-line.

e. The dual side-slip computation is continously monitored and will illuminate the DECRAB Fault and AUTO MASTER CAUTION Indicator Lamps if a difference equivalent to 2-degree Δ Preset Course is detected between the two channels.

f. The dual Left/Right (L/R) Runway computations are continously monitored. If a difference of 15 uA of raw localizer is detected, the L/R Fault and MANUAL MASTER CAUTION Indicator Lamps are illuminated.

g. Torque limiting is removed from the Roll and Pitch Servos.

At AA, the co-pilot sets and arms the Runway Distance Remaining Indicator (RDR). By arming the RDR (after manually setting in the runway length), the Flare Trip Circuit is enabled and thus ensures system operation through Roll-out. At this time, the Touchdown Footprint Scale is set over the nominal touchdown point on the Runway Scale.

At 1000-feet of radar altitude, beam de-senitization begins ("6" on the profile). A back-up control exists as a function of time and begins at Glideslope Engage if the Radar Altitude Mode is inoperative.

Split axis control is possible from this point on (i. e. : one axis is on automatic, the other on manual). Tests have shown that the pilots prefer to leave the Control Wheel Steering (CWS) off in the operable axis due to a disproportionate feel between the two (2) axes. Thus, the use of CWS during an AWLS Approach is considered controversial and over use will prevent the system from stabilizing on track. Its usage is usually limited to assisting the autopilot during large course corrections. Normally, with the good tracking capability of the system, CWS is not required.

As the aircraft continues tracking from the MM to the DH point, the continuous monitoring and course correcting continue. While most of these computations are conventional, there are some (RDR, Side-slip, Roll-out) which exist for Category III only).

At 200-feet AGL ("7" on the profile), the Rising Runway comes into view and will linearly provide a quantitative value of radar altitude. In addition, this same display provides a quantitative value of localizer error; displacing left or right on an expanded scale.

At 100-feet radar altitude, ("8" on the profile), the LAND ARM and MDA Indicator Lamps will illuminate. The pilot must ascertain that the aircraft is in position by being in a "window" established as ±27 feet from the localizer centerline. Assuming that the aircraft is within this limit, and that the ground track is either parallel to or converging with the localizer centerline at a pre-determined cross-track rate, the descent is allowed to continue.

At 45-feet radar altitude ("9" on the profile), Flare Engage occurs. This couples the Flare Error signal to the Elevator Channel and causes the Flare Progress Indicator Lamp to illuminate. The ADI's G/S Deviation Pointer is biased from view. The Glideslope Signal is completely de-sensitized to ensure that the Outer Loop Pitch Command is that derived from the flare maneuver. The localizer has, at this time, reached a maximum de-senitized gain of 45 per cent. The RDR is triggered and starts counting down the remaining runway distance.

The Excess Rate Circuits are enabled and will illuminate the EXCESS RATE Indicator Lamp if any of the following conditions exist:

a. A lateral rate across the runway in excess of 4-feet/second.

b. A Flare Command discrepancy between the #1 and #2 Flare Systems in excess of 2-feet/second.

c. A Flare Command in excess of 2 1/2-degrees nose-up or 1-degree nose-down.

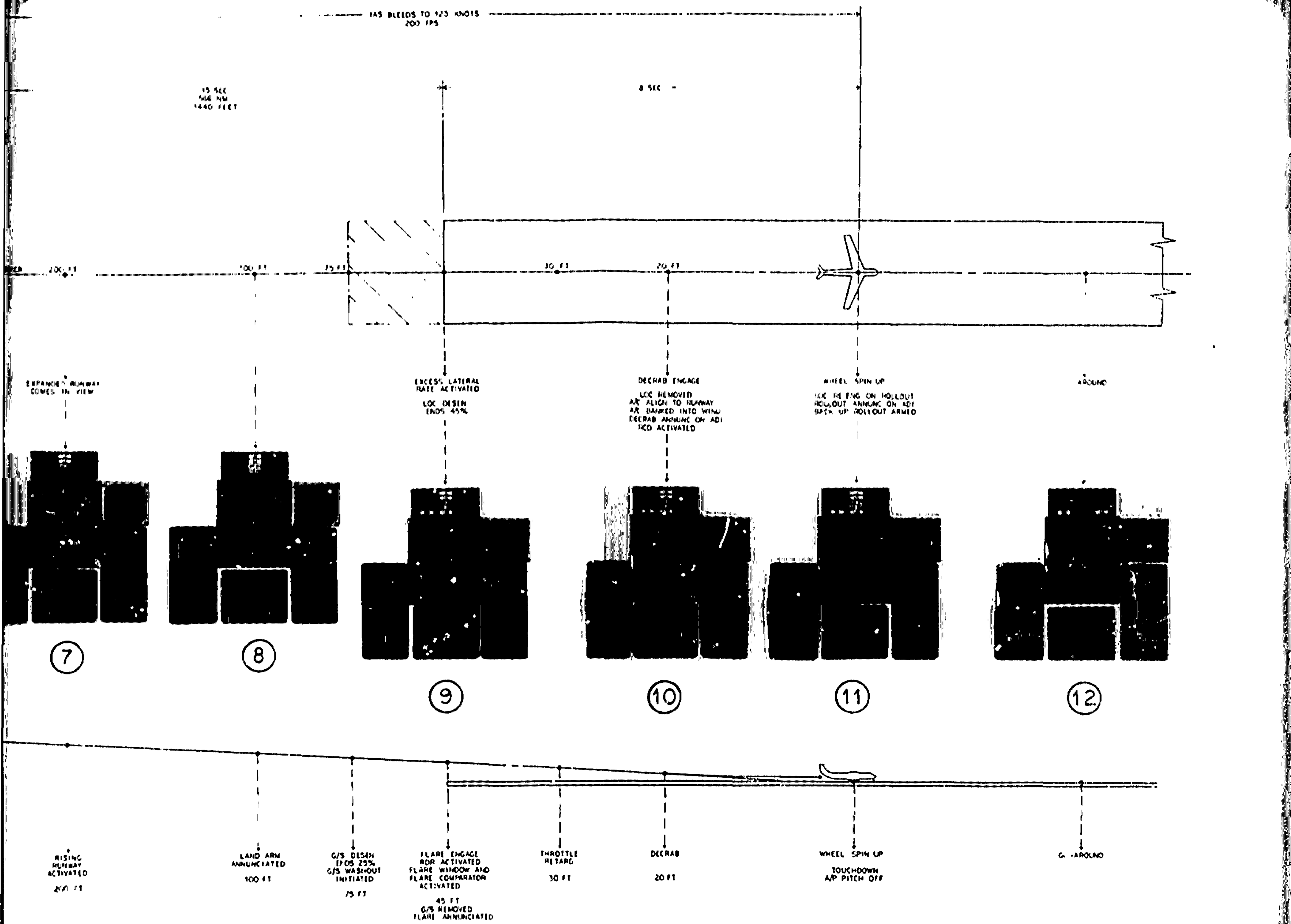
Illumination of the EXCESS RATE Indicator Lamp requires a go-around.

At 30-feet radar altitude, the Flare Computer sends a signal to the ATS for throttle retardation. The final flare maneuver is executed with the aircraft approaching Touchdown at a sink rate of about 2-feet/second.

At 20-feet radar altitude ("10" on the profile), a side-slip decrab maneuver is executed, that is proportional to the cross-wind and is held through Touchdown.

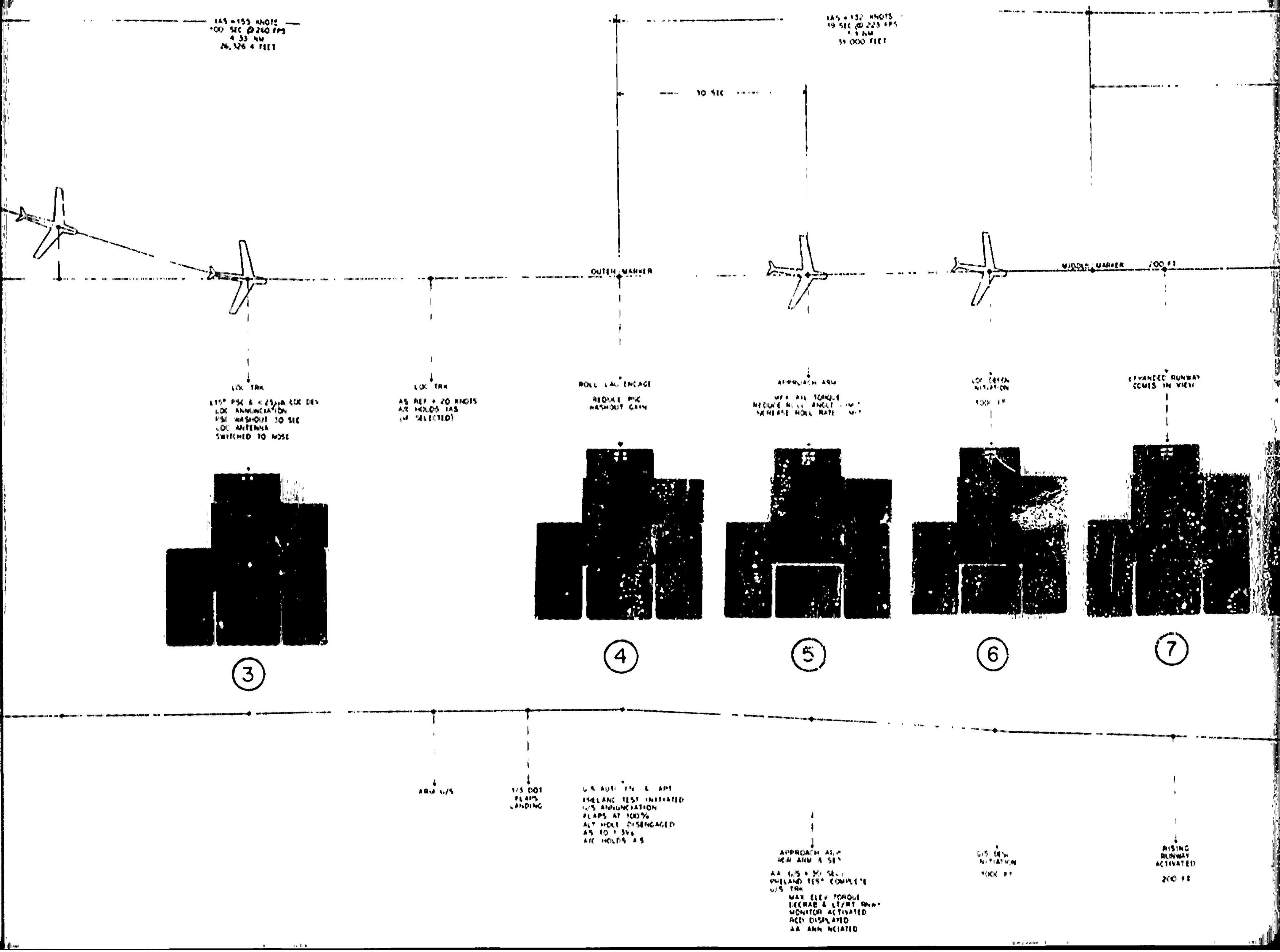
At Touchdown and with Main Gear Wheel Spin-up ("11" on the profile), the Wing'down Signal is reduced significantly. The localizer is introduced into the rudder computations to provide tracking on the runway. Full manual back-up indications are given to both pilots for an automatic cross-check and to allow assistance to be rendered to the automatics when required, particularly during the latter portion of the Roll-out when the rudder has lost authority and Nose Wheel Steering must be employed to maintain localizer center.

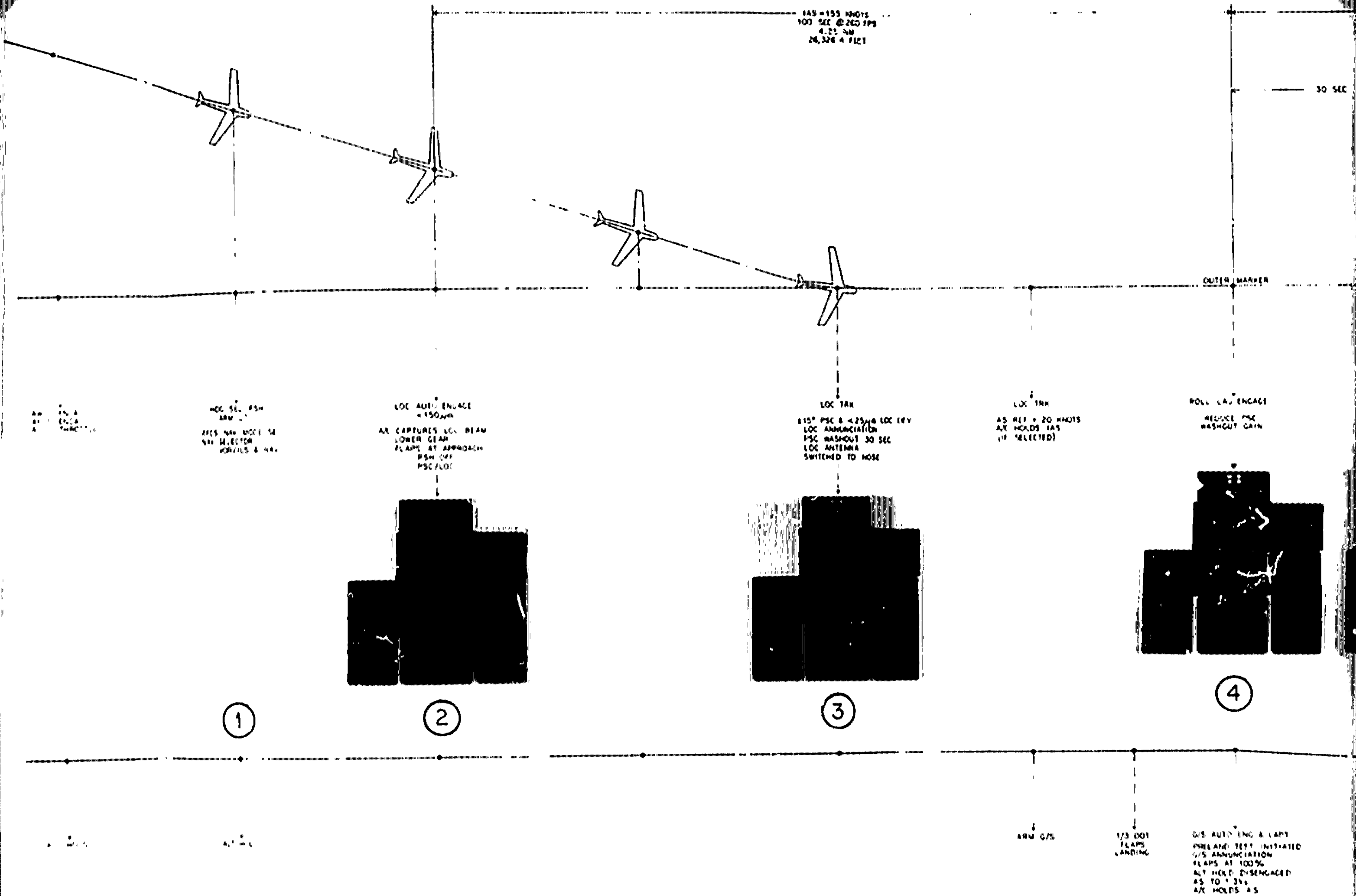
When required, a rotation/go-around may be initiated at anytime during the AWLS Approach. By pressing the GO-AROUND button on either control wheel, the AWLS (including autopilot and auto-throttle) is disengaged. The aircraft is thus returned to manual with the pilot flying to the Angle-of-Attack Command presented on the ADI. If initiated during Roll-out as indicated ("12" on the profile), lateral guidance is presented until Lift-off.



IAS BLEEDS TO 123 KNOTS
200 FPS
15 SEC
.568 NM
1440 FEET
8 SEC
200 FT
100 FT
75 FT
30 FT
20 FT
EXPANDED RUNWAY COMES IN VIEW
EXCESS LATERAL RATE ACTIVATED
LOC DESEN ENDS 45%
DECRAB ENGAGE
LOC REMOVED
A/C ALIGN TO RUNWAY
A/C BANKED INTO WIND
DECRAB ANNUNC ON ADI
RCD ACTIVATED
WHEEL SPIN UP
LOC RE ENG ON ROLLOUT
ROLLOUT ANNUNC ON ADI
BACK UP ROLLOUT ARMED
AROUND
7
8
9
10
11
12
RISING RUNWAY ACTIVATED
200 FT
LAND ARM ANNUNCIATED
100 FT
G/S DESEN
EPOS 25%
G/S WASHOUT INITIATED
75 FT
FLARE ENGAGE
RDR ACTIVATED
FLARE WINDOW AND
FLARE COMPARATOR
ACTIVATED
45 FT
G/S REMOVED
FLARE ANNUNCIATED
THROTTLE RETARD
30 FT
DECRAB
20 FT
WHEEL SPIN UP
TOUCHDOWN
A/P PITCH OFF
G. AROUND

FIGURE 15


IAS = 155 KNOTS
100 SEC @ 260 FPS
4.33 NM
26,326.4 FEET
IAS = 132 KNOTS
SEC @ 223 FPS
30 SEC
OUTER MARKER
MIDDLE MARKER
200 FT
LOC TRK
LOC ANNUNCIATION
PSC WASHOUT 30 SEC
LOC ANTENNA
SWITCHED TO NOSE
LOC TRK
AS REF + 20 KNOTS
A/C HOLDS IAS
(IF SELECTED)
REDUCE PSC
WASHOUT GAIN
APPROACH ARM
COMES IN VIEW
3
4
5
6
7
ARM G/S
1/3 DOT
FLAPS
LANDING
G/S ANNUNCIATION
FLAPS AT 100%
ALT HOLD DISENGAGED
APPROACH ARM
MAX ELEV TORQUE
MONITOR ACTIVATED
RCD DISPLAYED
RISING
RUNWAY
ACTIVATED
200 FT


IAS = 155 KNOTS
100 SEC @ 260 FPS
4.25 NM
26,326 4 FEET
30 SEC
OUTER MARKER
AFCS NAV MODE SE
NAV SELECTOR
VOR/ILS & NAV
LOC AUTO ENGAGE
< 150 μA
A/C CAPTURES LOC BEAM
LOWER GEAR
FLAPS AT APPROACH
PSH OFF
PSC/LOC
LOC TRK
±15° PSC & <25 μA LOC DEV
LOC ANNUNCIATION
PSC WASHOUT 30 SEC
LOC ANTENNA
SWITCHED TO NOSE
LOC TRK
AS REF + 20 KNOTS
A/C HOLDS IAS
(IF SELECTED)
ROLL CAU ENGAGE
REDUCE PSC
WASHOUT GAIN
1
2
3
4
ARM G/S
1/3 DOT
FLAPS
LANDING
G/S AUTO ENG & CAPT
PRELAND TEST INITIATED
G/S ANNUNCIATION
FLAPS AT 100%
ALT HOLD DISENGAGED
AS TO 1.3Vs
A/C HOLDS AS

SECTION IV

AWLS SYSTEM DESCRIPTION

4.1 PURPOSE

The purpose of this section is to describe the characteristics of the systems that directly relate to the operation of the AWLS. Block diagrams are provided in support of the text. Refer to the Appendix for system gradients, signal ratios, and transfer functions.

4.2 ATTITUDE AND HEADING SWITCHING MATRIX

4.2.1 General Description

The equipment presently configured into the test aircraft that is used in conjunction with the Switching Matrix consists of the following:

a. Dual LTN-51 Inertial Navigation Systems (INS) to provide the basic navigation requirements.

b. Switching circuits to distribute the aircraft attitude/heading data from the INS to the proper processors and/or displays.

c. Provisions for the use of the standard C-141 attitude and heading reference as a back-up system.

4.2.2 Detailed Description

Refer to Figure 16 for the Switching Matrix Block Diagram, which illustrates signal origins, distributions, and destinations.

The aircraft roll and pitch attitude are taken directly from the INS and distributed to all applicable systems. The Switching Matrix is commanded by the pilot's C-12/INS switch. Aircraft heading is supplied by the Magnetic Heading Auxiliary Units (MAHU #1 and #2). These will feed through the Switching Matrix to a Repeater Amplifier. The Repeater Amplifier is used to feed the analog signal to the required aircraft systems as the MAHU has a very low loading capacity. Complete separation of the two systems (attitude and heading) has been maintained. With the exception of the signal source selection, no operational procedure or redundancy has been altered.

An additional switching circuit was installed to enable the pilot(s) to select the AFCS (#3) Gyro as a third back-up for the Attitude Sphere in the ADI. Thus, if an INS failure did occur, the pilot dependent on the failed system would not be forced to go to the back-up MD-1 System and will thus preserve the accuracy of the operational INS. In practice, a failure of this nature would require a go-around unless the aircraft was committed to a full stop on the runway.

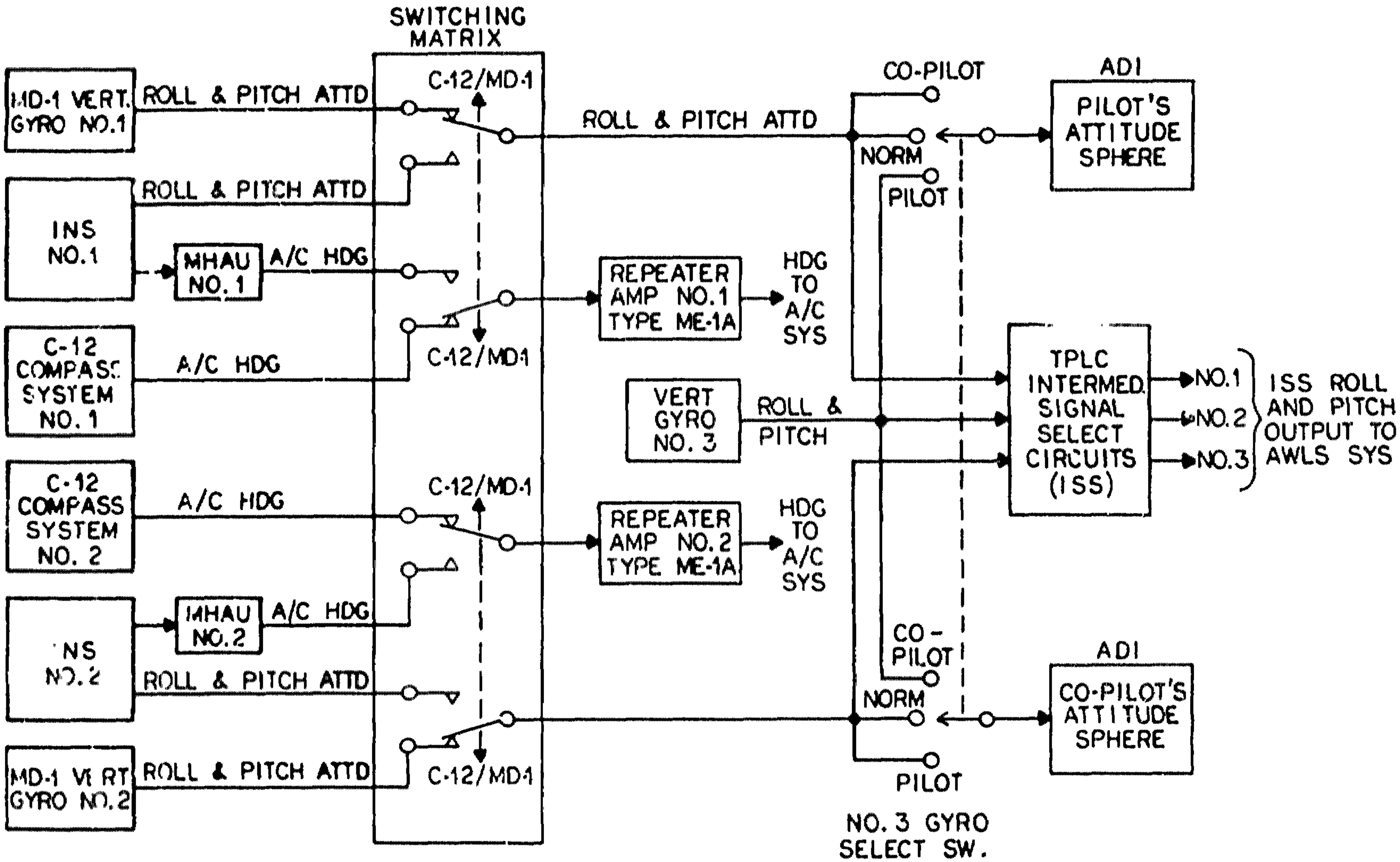


FIGURE 16. SWITCHING MATRIX BLOCK DIAGRAM

4.3 CATEGORY III ADAPTER

4.3.1 General Description

The Category III (CAT III) Adapter was designed, fabricated, and interfaced into the aircraft in-house. It contains the circuitry to allow the proper operation of the non-standard Sperry AD 350B ADIs, plus all manual and automatic computations required for Category III operations. Incorporation of this unit also enabled the restoration of the standard AWLS Category II components, eliminating the cumbersome adapters, modifiers, computers, and prototype ADIs that were previously installed in the aircraft.

Flight testing has validated the design objectives and has demonstrated that the basic concept of the CAT III Adapter for the C-141 is valid. The test pilots that flew the test system agree that the CAT III Adapter significantly assists the manual/automatic control throughout the side-slip/roll-out envelope.

The CAT III Adapter breaks down into two functional areas as follows:

a. The Attitude Director Indocator (ADI) Interface.

b. The Side-slip/Roll-out Command and Logic.

The Adapter (Figure 17) contains 26 printed circuit cards, two ±15 Vdc power supplies, and 2 reference spike generators. The circuit cards are of the universal function type which thus provides a high degree of reliability. Each card contains several circuits or combinations of circuits which perform the standard functions of system design (i.e.: summing, modulation, demodulation, amplification, synchronization, integration, and comparison).

Careful consideration was given to ensure that each circuit would function through a wide dynamic range by the simple selection of resistors/capacitors. This ensures compatability with a large variance of design requirements. In only one instance was it found that modification of an existing circuit was required to properly interface with the sensor and the display.

4.3.2 Attitude Director Indicator (ADI) Interface Description

Two (2) Sperry AD 350B ADIs (Figure 18) are installed in place of the standard ADIs on the instrument panel. An additional ADI is installed at the Test Director's Console. These ADIs serve as the basic attitude indicators and, in addition, provide the following:

a. The computed steering information.

b. The turn and bank information.

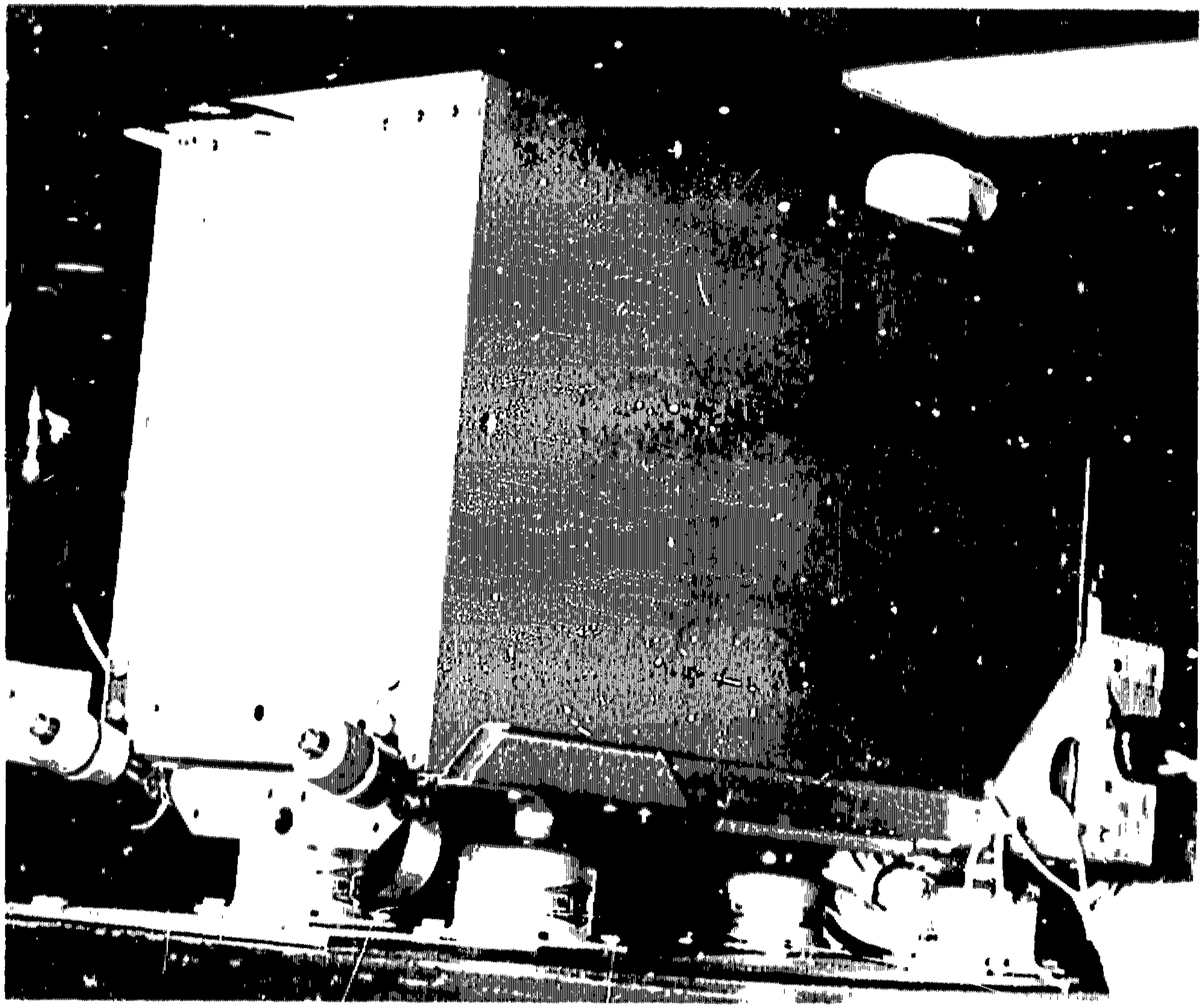


FIGURE 17. CATEGORY III ADAPTER

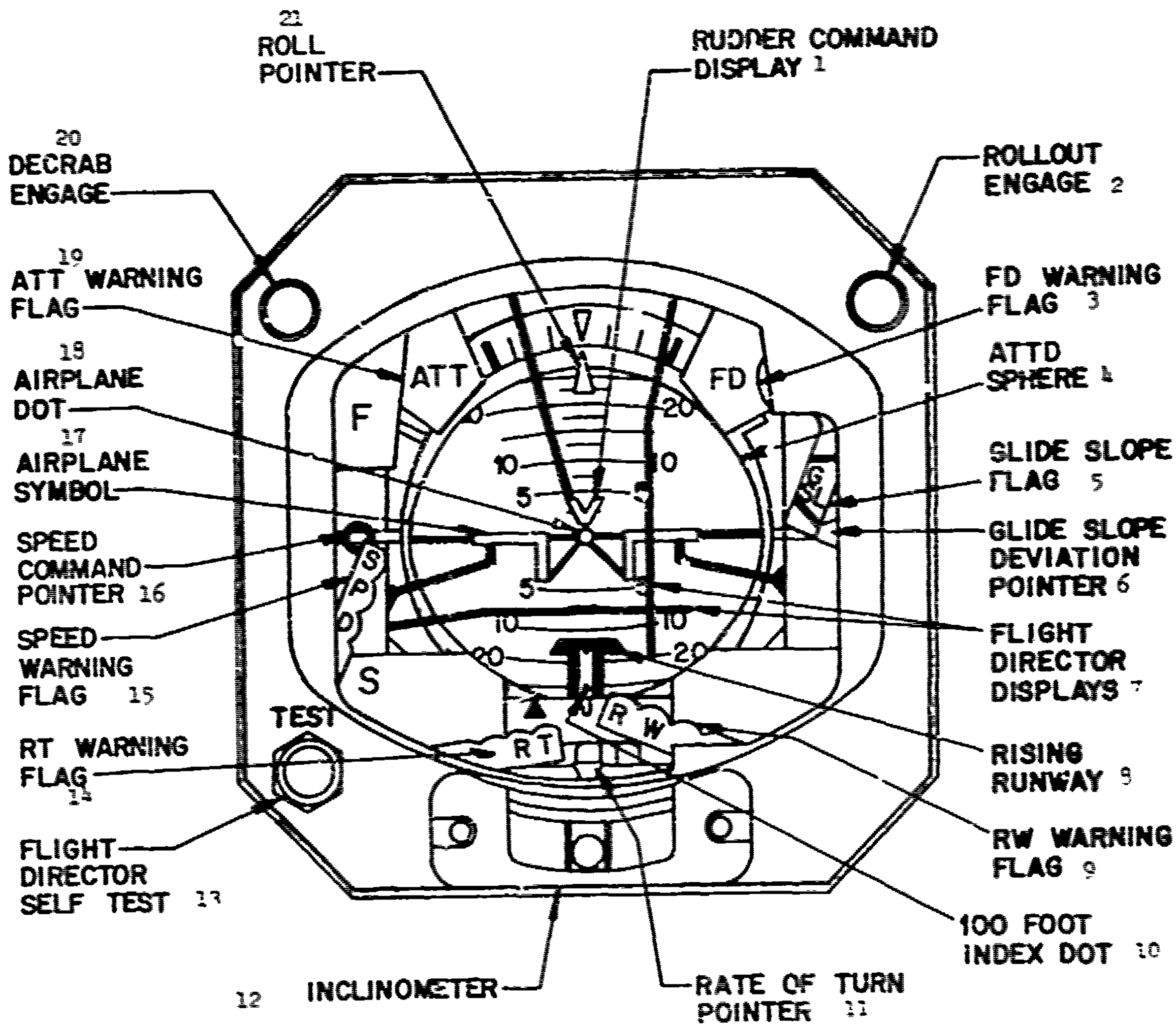


FIGURE 18. AD 350B ADI FRONT PANEL DISPLAY FEATURES AND CONTROLS

c. The rising runway with expanded localizer data for ILS position display.

d. The vertical deviation from the ILS glideslope or VERNAV angle.

e. The airspeed error.

f. The roll-out guidance.

g. The failure warning flags.

The CAT III Adapter contains the circuitry required to provide validity signals, stow logic, and signal conditioning to the failure warning flags, deviation pointers and the steering bars. Two (2) completely independent channels are incorporated into the unit for the ADIs. One channel is used for the pilot's and test director's ADIs and the other channel is used for the co-pilot's ADI. In addition, a protective circuit is provided to prevent the power being applied to the ADIs before the Vertical Gyro has completed its fast erection cycle. It had been possible for the Pitch Servo to drive the Attitude Sphere through the mechanical stops as the sphere is positioned to the random position of the gyro during its erection cycle. Figure 19 illustrates the protection circuit.

4.3.3 Attitude Sphere

The Attitude Sphere is positioned directly by the Vertical Gyro or the INS and is read relative to a fixed miniature aircraft (17)* to display roll and pitch attitudes. The Roll Pointer (21) and the Bank Scale show degrees of bank angle in 10-degree increments up to 30 degrees with additional markings at 60 and 90 degrees. The Pitch Reference Scale shows the aircraft pitch attitude in increments of 2.5 degrees to 40 degrees pitch with additional markings at 60 and 90 degrees. The Angular Pitch Reference Scale is non-linear starting at approximately 2.1:1 at 0 degrees and continually decreasing to 0.7:1 at 90 degrees of pitch. The Flight Director Warning Flag (3) indicates a loss of power or mechanical/electrical failures in the Flight Director System. The Attitude Warning Flag (19) indicates a loss of power in the ADI or Vertical Gyro or an internal failure in the ADI. Upon loss of the Attitude Input or power to the Attitude Gyro, the Attitude Sphere will rotate to indicate an approximate 85 degree bank in addition to the Attitude Warning Flag. Comparators within the ADIs monitor the relative position of the Attitude Spheres to that of the Vertical Gyro and will display the Attitude Warning Flag if a variation of more than 5 degrees exists in either the pitch or roll axis.

4.3.4 Steering Bars

Both Steering Bars are positioned by the CAT III Adapter. The Bank Steering Bar (BSB) indicates the bank correction commanded by the standard Flight Director Computer or the commanded wing down during the Side-slip maneuver as computed by the Cat III Adapter. The Pitch Steering

*This and all subsequent numbers refer to Figure 18.

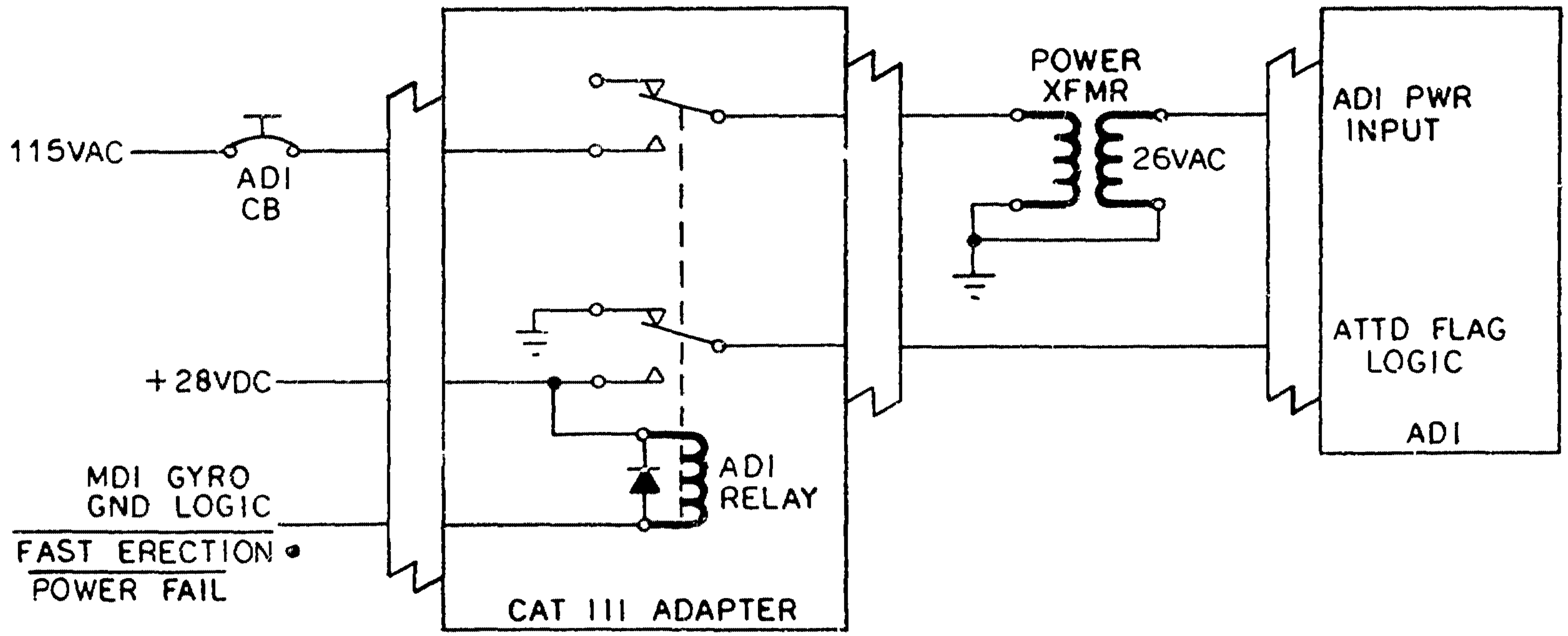


FIGURE 19. ADI POWER INTERLOCK BLOCK DIAGRAM

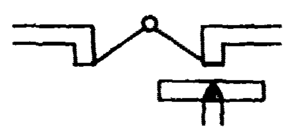
Bar (PSB) is commanded directly from the Flight Director Computer, except for an isolation circuit in the CAT III Adapter. The AD 350B Steering Bars are stowed in the opposite direction from that in the standard C-141 ADI. Refer to Figure 20. The PSB control during the R/G-A (PSB in view) and Roll-out (PSB stowed) is incorporated.

The Glideslope (G/S) Deviation Pointer (6) presents the aircraft displacement either above or below the ILS G/S or VERNAV angle on a two (2) dot basis. The CAT III Adapter converts the standard G/S Deviation Signal of ±150 mV to the ±2 Vdc level required by the AD 350B for a ±2 dot indication of the G/S Deviation Pointer.

The G/S Warning Flag (5) indicates the lack of valid G/S information when the ILS frequency is selected on the corresponding VHF/NAV receiver. The CAT III Adapter converts this low level validity (+350 mV max) to the +28 Vdc required by the AD 350B for stowage of the G/S Deviation Pointer. Refer to Figure 21.

The Rising Runway (8) moves vertically to display the radar altitude from 200 feet AGL to touchdown and laterally to indicate the aircraft's position as to localizer on an expanded scale. The Rising Runway is biased from view and the Runway (RW) Warning Flag (9) enters the field-of-view to indicate an invalid localizer or radar altimeter signal. The CAT III Adapter converts the normal radar altimeter signal of .2V ±10 mV/ft to .4 V ±20 mV/ft as required by the AD 350B (Figure 20). The Altitude Validity from the radar altimeter (+28 Vdc) does not require conversion and is applied directly to the validity input of the AD 350B. The Left/Right Displacement feature of this indication is used as a Lateral Situation Indicator. The presentation is raw Localizer Deviation, amplified by the Adapter such that ±12.4-uA of deviation will displace the runway either left or right as shown below:

Aircraft is left of runway.



This presentation becomes the lateral window for continuing the approach from flare altitude and indicates approximately 25 feet of lateral displacement from localizer at the touchdown point of the approach. This figure is derived from the FAA criteria for a tailored beam (Category II or III). In part, the specifications dictate that ±150-uA of localizer will equal ±350 feet of lateral dispersion at the approach end of the runway. This equates to 2.33 ft/uA. The Left/Right Display is disabled until AA to protect it from excessive voltages due to the amplification and large localizer deviations normally encountered during the capture portion of the approach. At AA, the aircraft has established localizer track and thus deviations are minor.

The Left/Right Runway Validity is supplied by the adapter. To accomplish this, the adapter converts the low level Localizer Flag Logic

(350 mV) to the +28 Vdc required by the AD 350B. The Flag Logic is further interlocked by the normal AWLS logic in that the logic will remain high whenever the AWLS is disengaged. This will preclude nuisance flags/runway stows during non-AWLS flights (Figure 21).

The Rudder Command Indicator (1) will provide rudder steering information for decrabbing the aircraft as well as directional guidance during the landing roll-out. The Rudder Command Indicator enters the field-of-view at the completion of the AWLS Pre-land Test. However, it will remain inactive until 18 feet AGL, at which time it is activated to provide decrab guidance. When Main Landing Gear Wheel spin-up occurs after touchdown, modes change to provide roll-out guidance to the localizer centerline. The Rudder Command Indicator does not have a warning flag, nor does it bias from view in the event it receives invalid data or if a failure occurs. Rather, the DECRAB Indicator Lamp on the AWLS panel will illuminate to warn the pilot that the rudder command display may be invalid. The indicator lamp circuitry appears in Figure 22; however, the computational circuits are described in the Roll-out/Side-slip discussions.

The Speed Command Pointer (16) displays airspeed deviation from the airspeed selected whenever the auto-throttles are engaged. The Speed Warning Flag (15), after engagement, is controlled by the auto-throttles validity. Both indicators and the flag are stowed when the auto-throttles are disengaged. In addition, the Fast/Slow Indicator is stowed during the Roll-out mode. The computation and logic circuits are shown in Figure 21.

The Turn and Slip Indicators (11 and 12) indicate the rate of turn and turn coordination of the aircraft. The Rate of Turn (RT) Warning Flag (14) comes into view to indicate a power failure to the Rate of Turn Sensor. The adapter uses an amplifier which converts the signal gradient of the standard turn rate into the gradient required by the AD 350B (Figure 20).

Also included on the front panel of the ADI are 2 indicator lamps: the Decrab (DE) Progress Indicator Lamp (20) and the Roll-out Engage (RO) Indicator Lamp (2). The DE Indicator Lamp illuminates at Side-slip Engagement while the RO Indicator Lamp illuminates when the AWLS system has entered the Roll-out mode after touchdown. All switching logic is contained in the CAT III Adapter.

The Flight Director Warning Flag (3) Logic is altered by the CAT III Adapter. The standard Flight Director Computers do not have the decrab function, and thus, are not used beyond Decrab Engage. The CAT III Adapter assumes all lateral computations at this time. In the event the Flight Director On Logic is removed after Decrab, the CAT III Adapter will keep the flag biased from view.

A Flight Director Self-Test (13) Button activates the Attitude Sphere, Flight Director and the Rudder Self-Test Circuits. When the button is pressed, the Attitude Sphere will rotate to indicate 10 degrees climb, 20 degrees of right bank, the Attitude and Flight Director Warning Flags appear, the Flight Director BSB will be positioned to the right edge of the miniature aircraft symbol, and the PSB will be positioned in the fly-up position.

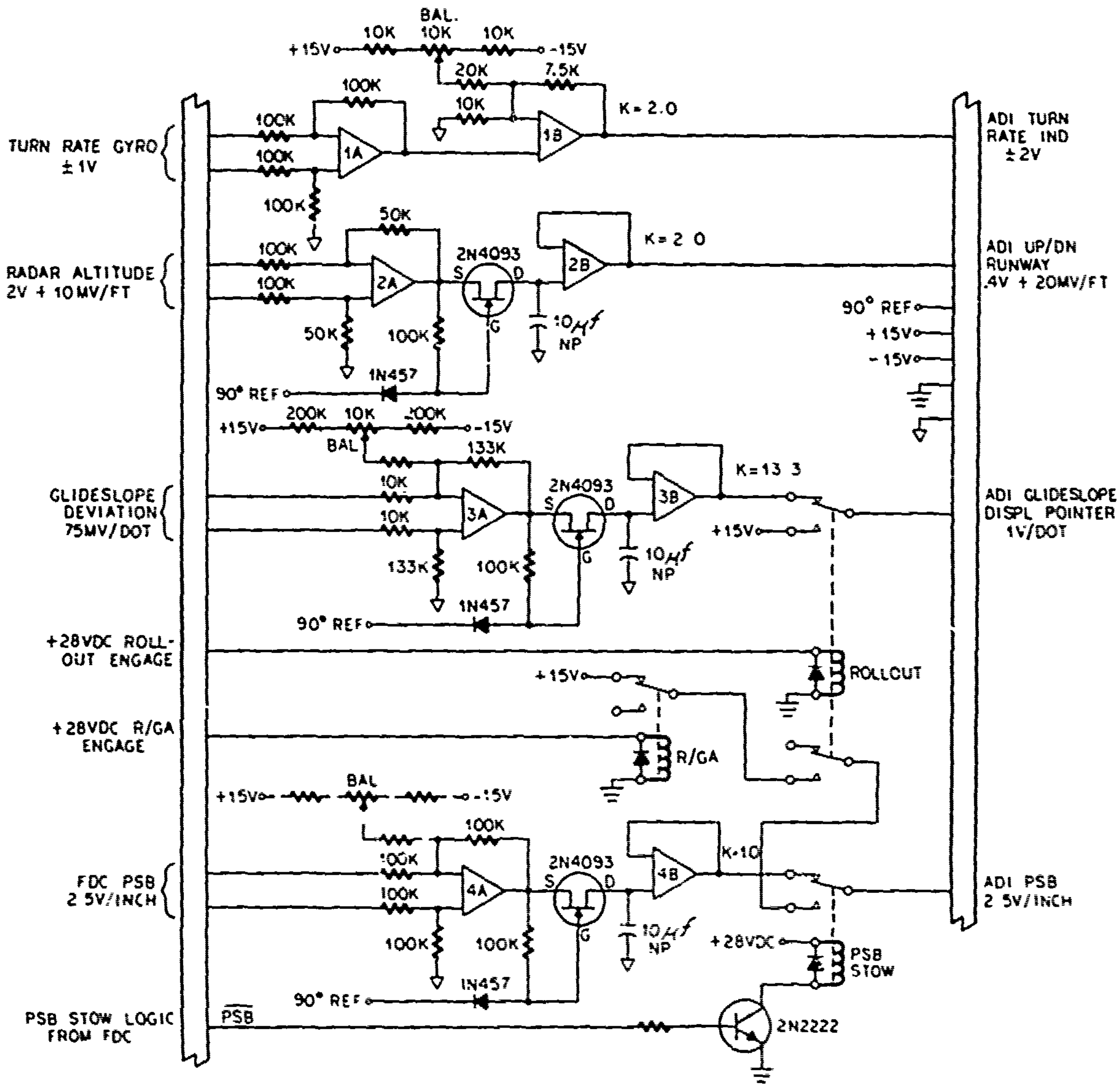


FIGURE 20. ADI SIGNAL INTERFACE

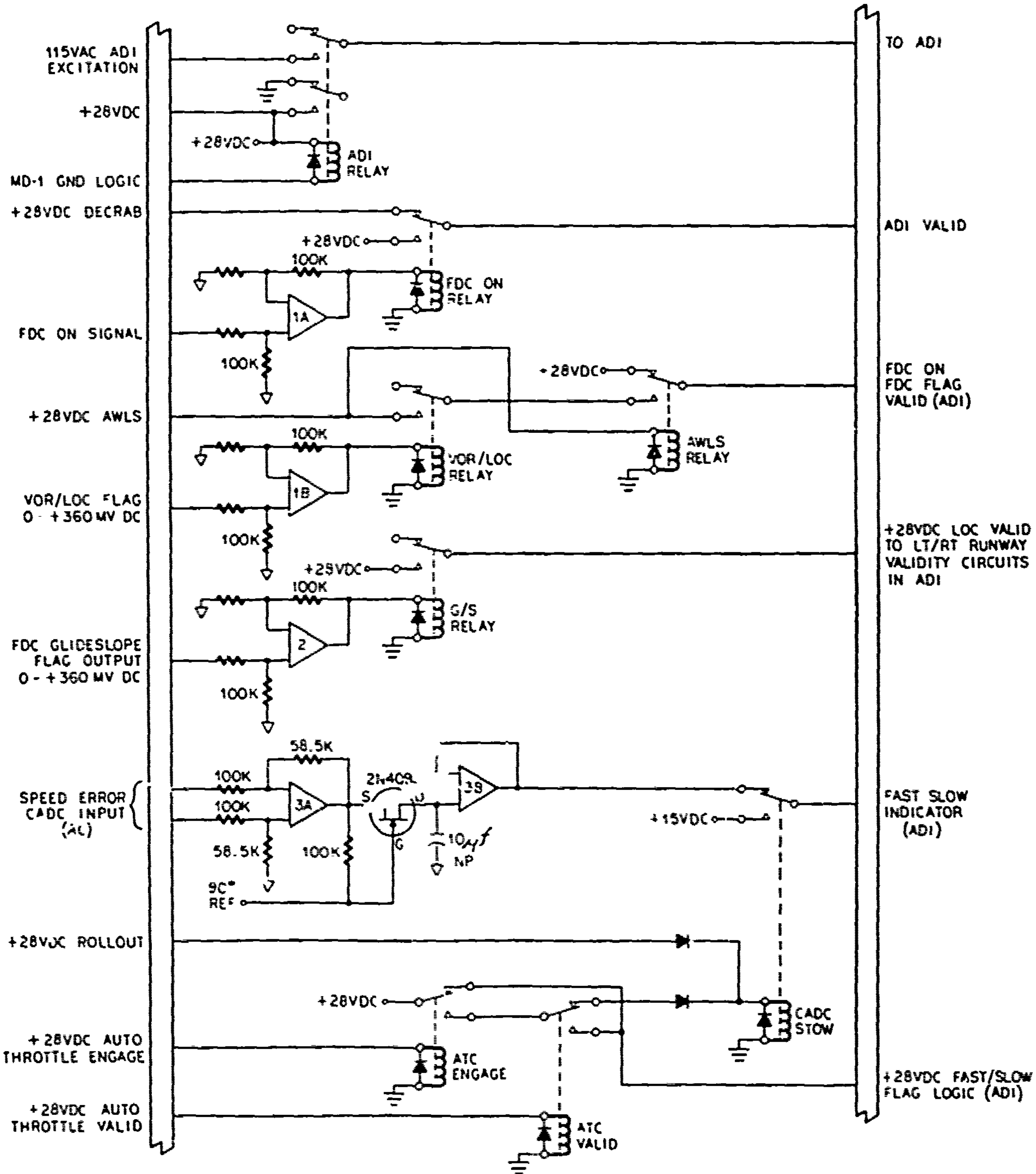


FIGURE 21. ADI LOGIC INTERFACE

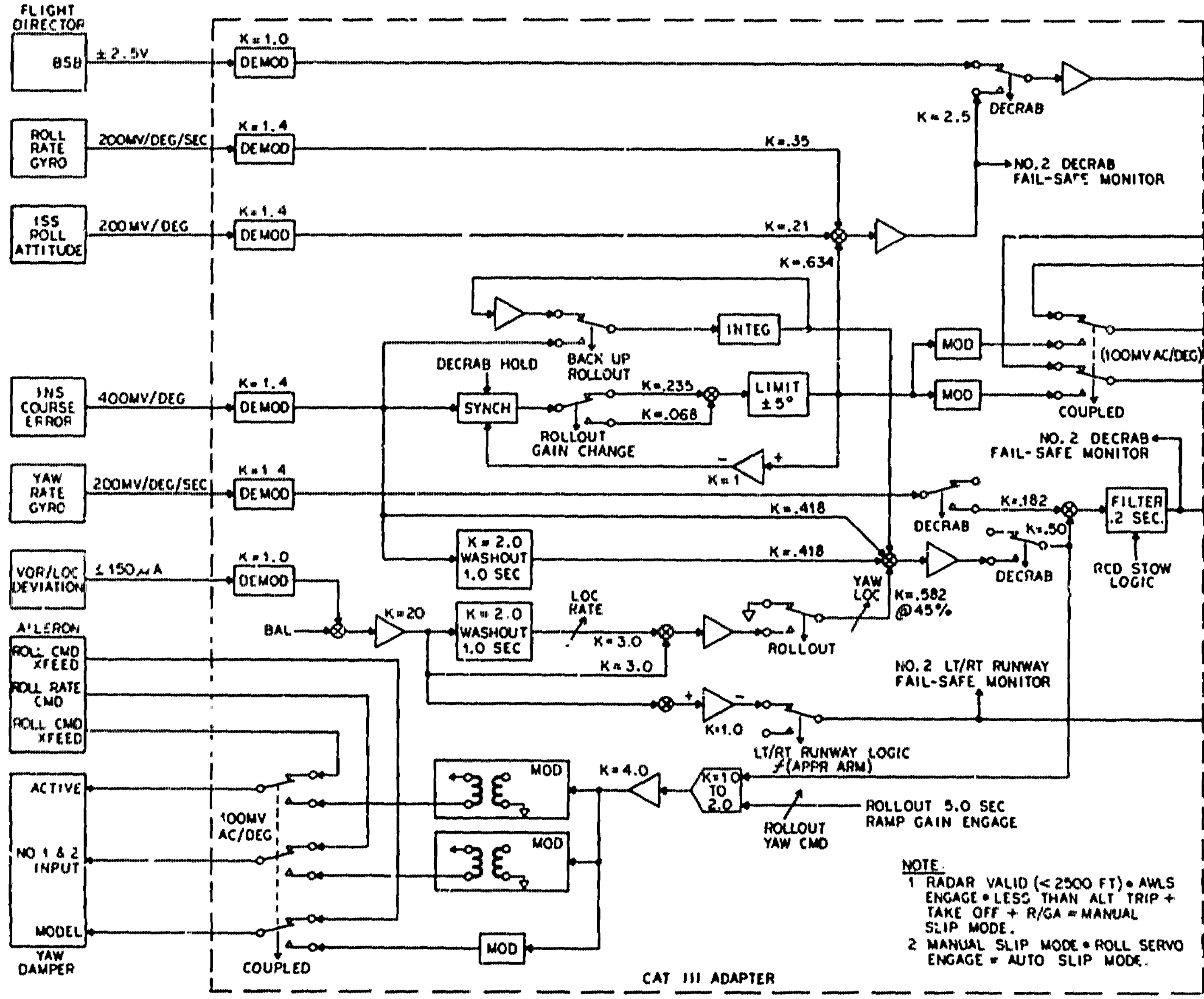


FIGURE 22
SIDE-SLIP/
ROLLOUT CIRCUITS

4.4 AWLS FLIGHT DIRECTOR SYSTEM

4.4.1 General Description

The Flight Director System consists of the following items:

a. The dual Flight Director Computers.

b. The dual ADIs.

c. The dual Horizontal Situation Indicators (HSI).

d. The Navigation Selector Panels.

e. The CAT III Adapter.

f. The dual Flare Computers.

This system provides computed Roll, Pitch, and Rudder Commands for manual operation and system back-up monitoring during automatic operation. Complete instrument displays are provided during the enroute, approach, and landing phases.

The Flight Director Computers and the CAT III Adapter use radio deviation signals, heading or course error, pitch and roll attitude reference, normal acceleration, pitch rate, and elevator position to generate the steering commands for the ADI presentation. Zeroing of these commands results in a flight path or roll-out conforming to the modes and switch setting selected. Logic circuitry is incorporated to perform the mode switching of the signals, monitoring of the incoming signal validity, and control of the ADI flag and pointer biasing.

4.4.2 Lateral Channel Description

The Flight Director Computers generate Lateral Steering Commands in either the manual heading mode or in a selected radio/navigation mode. Figure 23 provides a detailed block diagram of the lateral channel with parameters and signal path switching data. The following paragraphs detail the functional mechanization of the lateral channel on the basis of each operational mode.

4.4.2.1 Manual Heading Operation

This mode is activated by either the Heading Select Input or the lack of Lateral Beam Sense Logic in one of the radio track modes. Except for a bank command limit of 23 degrees in the heading select mode as opposed to a 30 degree limit in the radio track modes, the manual heading modes function the same.

The Heading Error Signal (Input) is routed through a Bank Command Limiter and in turn through the Command Rate Limiter to the Meter Drive Summing Amplifier. At this point, direct Bank Attitude Feeding is

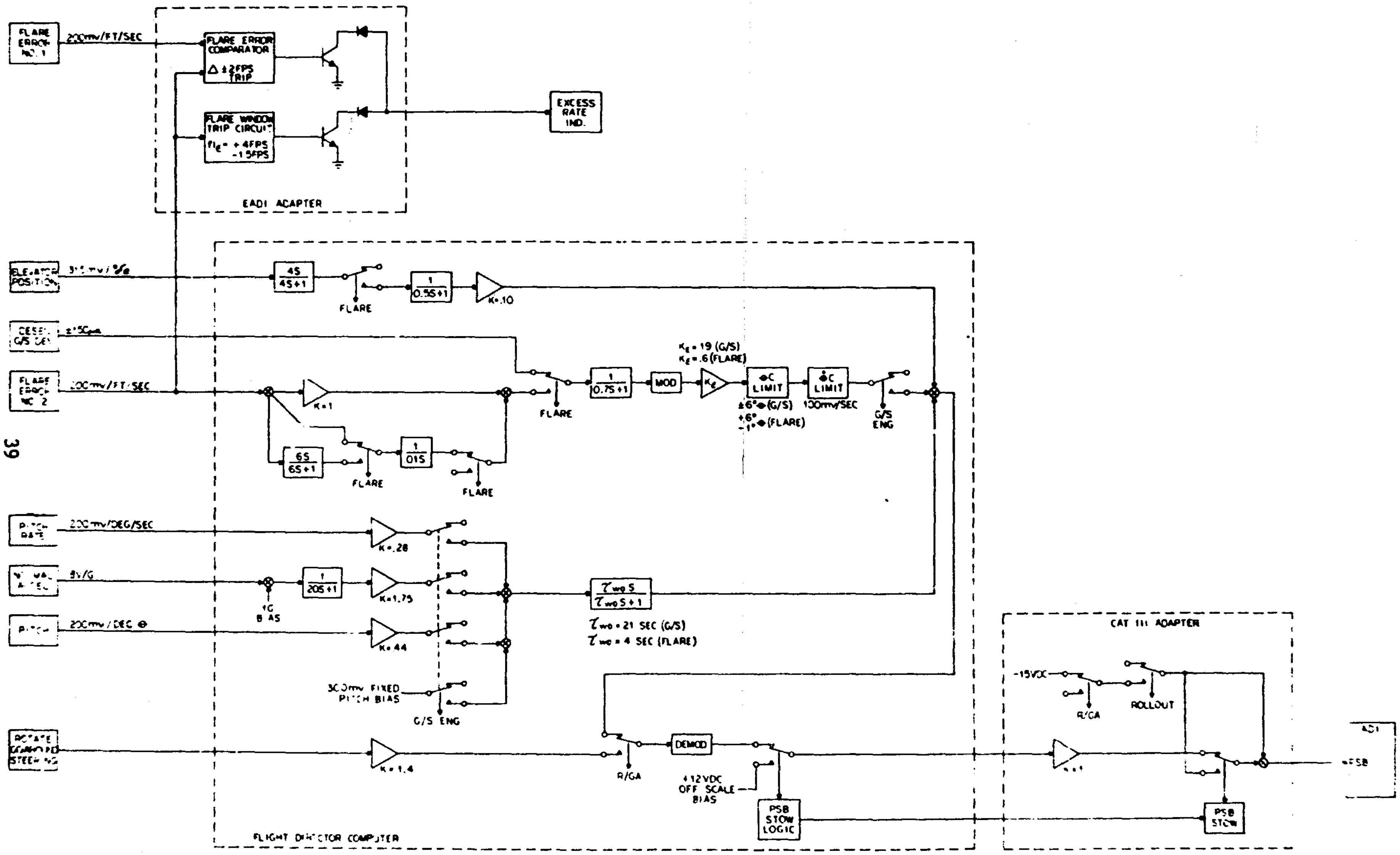


FIGURE 23. FLIGHT DIRECTOR/CAT III ADAPTER LATERAL CHANNEL

applied to permit nulling of the steering signal when the appropriate roll correction has been made. As the Heading Command diminishes, the Bank Attitude Signal results in a Roll Command. Thus, whenever the heading error is null, a Wings-level Command predominates.

4.4.2.2 Radio/Navigation Mode Operation

The specific radio/navigation mode activated is dependent upon the Navigation Selector Panel switch settings and the computer Track Logic state. The mode input signals affect the parameter switching appropriately to the characteristics and sensitivity of the input deviation signal. Track Logic establishes whether Beam Capture and/or Track Steering is used. Lateral Deviation is the fundamental reference or input signal in a radio/navigation mode. In the capture sub-mode, however, course error serves as the predominant input error signal. Bank Attitude Feedback is employed to relate the aircraft attitude to the commanded roll correction.

4.4.2.3 Capture Sub-mode Operation

This sub-mode is mechanized to reduce the course error input during closure to the beam such that the zeroing of the Lateral Deviation and Course Error occur simultaneously. For a course error of greater than 45 degrees, the Bank Command is obtained solely form the Direct Course Path. If the lateral deivation is high, the Course Cut Limiter saturates at a level equivalent to $K_{\psi ACC}$ times 45 degrees. This contribution subtracts from that of the Direct Course Path around the Limiter whose level is $K_{\psi ACC}$ times the full value of Course Error. Thus, a 45 degrees Course Error is approached, Roll Down Steering is obtained, provided that Lateral Deviation remains high enough to hold the Course Cut Limiter in saturation. The limiting characteristic is "soft" to preclude abrupt steering. When the Lateral Deviation is at a relatively low level, the Course Cut Limiter is not saturated, and Roll Down Steering is not pronounced. Also, a high rate of Beam Closure tends to bring the Course Cut Limiter out of saturation sooner since $K_{\dot{R}C}$ provides steering of the same sense as $K_{\psi C}$. The resultant effect is to maintain the Command Bank Attitude where Lateral Deviation is low and the rate of closure high.

Below 45 degrees of course error, the Direct Course Gain switches to $K_{\psi BCC}$. This effective gain is formed by the difference between $K_{\psi ACC}$ and $K_{\psi C}$, the direct course gain through the Course Cut Limiter. Since $K_{\psi ACC}$ is always larger than $K_{\psi C}$, the sense of the $K_{\psi BCC}$ signal maintains the Roll-up Steering to turn onto course, provided the Course Cut Limiter is out of saturation. K_{RC} and $K_{\dot{R}C}$ contributions modify the degree of Roll Command depending upon the proximity and rate of closure on beam center. Following summation of the course and radio terms, the Capture Command Signal passes through a 30 degree Bank Limit and then through the Command Rate Limiter. Direct Bank Feedback is summed with the Bank Command just prior to the Meter Driver. Note that the Course Rate Path and the Lagged Roll Path are disabled during the capture sub-mode.

4.4.2.4 Track Sub-mode Operation

Upon satisfaction of the track criteria, appropriate logic is applied to the lateral channel to effect the required parameter and signal path switching. Both direct Course Paths are disabled and the Course Cut Limiter ceases to be a factor. The bank limit is reduced to 15 degrees and the lagged roll and course rate contributions are activated.

The tracking sub-mode is designed to facilitate the manual flying of the aircraft on a selected radio/navigation course with a minimum of pilot effort. This entails holding the Lateral Deviation essentially at zero using a minimum of roll correction and heading change. Since the Lateral Deviation Signal is subject to beam noise and disturbance/step changes, the input radio signal is passed through an Input Lag Filter to virtually preclude such transients from affecting the Output Steering Signal. The Deviation Signal then branches out through 2 paths: K_{RT} and $K_{\dot{RT}}$. The radio path sums with the course rate and a roll signal prior to passing through the Rate Filter. The Roll Rate Signal is then subtracted from an associated Direct Roll Path to yield the Lagged Roll contribution K_{ϕ}. Thus, 3 damping terms are summed (radio, course, and lagged roll rate) with the Direct Radio Gain prior to the Bank Command Limiter. Roll Attitude Feedback is summed with the Bank Command at the input to the Meter Driver.

To obtain the higher radio gain required for Category III AWLS performance, the input relays switch (at AA) from the desensitized localizer to the raw localizer available at the Lateral Drviation Input. Not only does this result in tighter tracking at the low end of the approach, it permits looser tracking at the high end where pilot effort can be properly reduced. The Bank Command in either the manual or a radio/navigation mode is rate limited to 300 mV/sec at the Meter Driver Output.

4.4.2.5 Side-slip/Roll-out Mode Operation

At a radar altitude of 20 feet, the Flight Director Computer computations are completely removed and the terminal guidance is supplied by the CAT III Adapter.

Two independent computational channels in the CAT III Adapter provide the Side-slip/Roll-out guidance with both model and active channels being incorporated. The active channel is displayed on the co-pilot's ADI and provides the computational input to the autopilot. The model channel is displayed on the pilot's and test director's ADIs. For manual operation, the Aileron Wing-down Command is displayed on the BSB and the Decrab/Roll-out Command is displayed on the Rudder Command Indicator. The computational channels are cross-monitored and will illuminate the DECRAB Fault Indicator

Lamp if the channels differ by a pre-set amount. Since the computational channels are identical, only one will be described.

4.4.2.5.1 Side-slip. The Side-slip Computational Channels of the CAT III Adapter provide for the operation of the manual side-slip mode. The Side-slip Circuits provide the displays required to reduce the crab angle and align the aircraft track to Runway Heading. The point at which the Side-slip is initiated is nominally set at 20 feet radar altitude. The Runway Heading Signal is taken directly from the INS for the accuracy of both computations.

The maneuver is best described as 2 independent maneuvers: decrab and and wing-down. At Side-slip Engagement, the following occurs simultaneously:

a. The Rudder Command Outputs of the CAT III Adapter are enabled to provide command signals to the RCD.

b. The Flight Director Bank Steering Bar output is removed and switched to the computational output of the CAT III Adapter.

The computation for the decrab maneuver serves only to bring the course error (Runway Heading Error) to zero. Thus, when the computation is satisfied, the aircraft will be aligned with the runway. For the manual mode, an aircraft Yaw Rate Signal is introduced for damping the manual control inputs.

The wing-down computation is derived directly from the heading change accomplished by the aircraft during the decrab maneuver. This is accomplished by synchronizing the Runway Heading Error to zero until Decrab Engage, and any heading change passes directly to the output through a ± 15 degree Bank Limiter. In the manual mode, the aircraft roll attitude is used to provide the Hard Bank Limit with the aircraft roll rate used to damp the manual control inputs.

Computationally, 1/2 degree of roll attitude is commanded to the Roll Axis for every degree of heading change generated by the decrab maneuver. The reason for the side-slip maneuver is to eliminate or reduce the downwind drift generated by the decrab maneuver.

4.4.2.5.2 Roll-out. The roll-out mode is initiated after Touchdown and Main Landing Gear Wheel spin-up. The roll-out channel command computation uses both the runway heading error and the localizer to provide the required steering commands to acquire and maintain the localizer centerline. Manual steering commands are displayed by the RCD and the BSB of the ADIs.

Localizer error into the rudder command channel becomes the primary guidance signal. Runway heading and localizer rate become course damping terms to prevent excessive cross track rate build-ups and to maintain reasonable course cut angles back to localizer centerline. In the yaw channel, no further changes are involved.

In the aileron channel, the wing-down command is reduced by 75% to eliminate a bicycling tendency of the aircraft for both manual and automatic channels.

4.4.2.6 Roll-out Back-up System

Perhaps the most serious failure that can occur is the loss of the localizer after the aircraft is on the runway and committed to a full stop. The loss of this primary guidance signal could be caused by ILS shutdown or NAV receiver failure. In the early mechanization of the C-141 AWLS, loss of this signal would leave the runway heading as the only roll-out guidance signal. This signal does not contain position information and thus can be considered divergent due primarily to the reaction lag of the pilot in correcting an error in heading. It is not adequate in itself to ensure a reasonable lateral position on the runway. Thus, a circuit was designed to overcome this.

The circuit computation (in the CAT III Adapter) remains synchronized to zero until the following conditions are met:

a. Main Landing Gear Wheel spin-up.

b. LOC Flag in view.

If this situation arises, circuit reverts to an integrator from a synchronizer. It then provides an integram of aircraft heading changes about the runway heading and displays it as rudder command on the RCD. Through proper phasing of this signal, it directs the pilot to change the aircraft heading in the opposite direction for the same amount of time that the original error was present. This will return the aircraft to its original position on the runway and eliminate the divergent tendency experienced with the basic runway heading signal.

Operationally, the system worked extremely well under little or no wind conditions. Two shortcomings were found:

a. Under strong wind conditions, a drift with the wind across the runway would develop as the aircraft would literally be blown across the runway.

b. The runway heading information was not always accurate.

Neither shortcoming was considered unsafe. In the case of the crosswind, it was surmised that the element from Touchdown to Full Stop was such that even under severe conditions, the distance of crosswind drift would still keep the aircraft on the runway if a footprint of ± 25 feet of centerline was assumed at Touchdown and during that portion of the Roll-out prior to localizer loss. In addition, under the severest of visibility conditions (fog), crosswinds of any magnitude are seldom, if ever, experienced. In the case of the runway heading accuracy, this signal was taken directly from the INS system and thus eliminated the HSI stack-up of 1/4 to 1 1/2 degrees normally encountered.

4.4.2.7 Lateral Excess Rate Detetector

The Lateral Excess Rate Indicator Lamp (located on the pilot's instrument panel) illuminates when a lateral rate of greater than 4 feet/second is attained, thereby warning the pilot not to proceed to touchdown. The control signal for this indicator comes from the INS with the circuit being enabled at Flare Engage (45 feet AGL).

4.4.3 Longitudinal Channel Description (Figure 24)

4.4.3.1 ILS Approach Mode

Upon satisfaction of the mode logic, the PSB comes into view commanding a capture of the glideslope beam. The initial command is formed by the pitch attitude and fixed pitch bias. In the typical beam capture (level flight and below the beam), the capture command is derived from a slight pitch-up G/S Command plus a 3 degree fixed pitch-down bias. The fixed bias is effective at capture only, then washes out. Once the capture is initiated, the various damping terms then tend to produce a smooth, asymtotic capture. With proper flap deployment, power settings, and pitch trim, the glideslope is easily captured.

Glideslope tracking is referenced to the beam with desensitized radio deviation furnished from the autopilot coupler. Beam error is introduced through a Lag Filter with a 0.7 second time constant. After passing through a ± 6 degree Pitch Command Limiter, the glideslope signal is applied to the Pitch Command Rate Limiter and then summed with the various damping terms. These terms (pitch and pitch rate) are for long and short term damping and are routed through a common path with a 21 second Wash-out Filter. Following the removal of the 1-g bias, the Normal Accelerometer Input is passed through a Complementary Filter with a 20 second lag and a 21 second lead time to approximate Vertical Velocity. The Accelerometer Input is particularly effective in turbulence. The Composite Glideslope Steering Signal is summed just forward of the Meter Driver. Note that the VERNAV, Rotation/Go-around and Autopilot Pitch Signal paths are disabled during the ILS approach mode.

4.4.3.2 Flare Sub-Mode Operation

Upon reception of the Flare Mode Input in the ILS approach mode, the Flare Error Input Relay switches for Glideslope Beam Error to Flare Error. To preclude the possibility of a steering transient at mode switching, the Flare Error Signal is synchronized prior to Flare Engage. Upon entering Flare, any offset sychronized is washed out with a 6 second time constant. The Flare Error, less any offset, is passed through the 0.7 second Lag Filter and then to the Pitch Command Limiter, which assumes +6 and -1 degree pitch command limits. Following the Pitch Command Limiter, the Flare Error is summed with the washed out summation of Pitch, Pitch Rate, Normal Acceleration and Elevator Position Feedback. For the flare maneuver the wash out time constant is reduced to 4 seconds. The Elevator Feedback is added during the flare maneuver to reduce the tendency to over-flare.

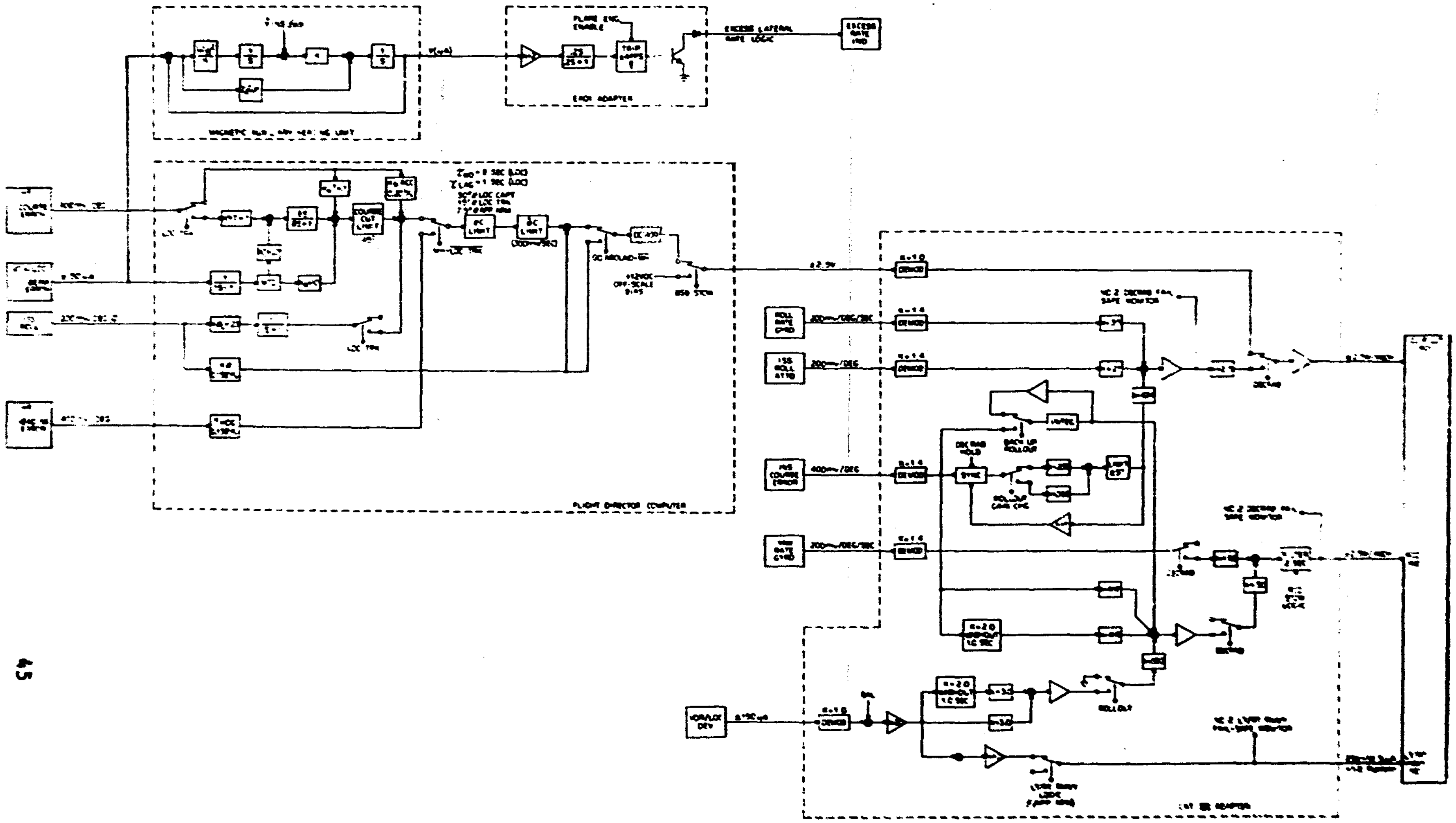


FIGURE 24. FLIGHT DIRECTOR/CAT III ADAPTER LONGITUDINAL CHANNEL

4.4.3.3 Rotation/Go-Around Mode Operation

Activation of this mode disables all signal paths in the longitudinal channel except that of the Rotation/Go-Around input. This steering signal is routed directly to the Pitch Meter Driver and thus displaces the PSB.

4.4.3.4 PSB Stow Logic Operation

Alterations of the PSB stow logic were required. In the standard C-141, stowage of the PSB is down. In the Sperry AD 350B ADI, stowage is up, although the fly-up/fly-down sensing is identical. A stow circuit in the CAT III adapter was implemented to invert the biasing voltage to insure proper stow a ection of the PSB.

4.4.3.5 Vertical Rate Detectors Operation

Due to the subtlety of the basic exponential flare maneuver, it was extremely difficult for the pilots to detect abnormalities during the flare maneuver or to visually discern that high rates were building up and still have time to initiate corrective action. In weather, without visual cues, it would be impossible. Thus, two detection circuits were implemented in the vertical axis to forewarn the pilots that a problem was developing.

4.4.3.5.1 Flare Error Window. The Flare Error Window Detector measures the output of the #2 Flare Computer. If a Pitch-Up Command of more than 4 feet/second or a Pitch-Down Command of more than 1 1/2 feet/second is evident, the circuit trips.

4.4.3.5.2 Flare Comparator. The Flare Comparator is unique to this aircraft. Two completely independent Flare Systems are installed with a cross comparator installed between them. If the two Flare Systems differ in their outputs by an amount exceeding 2 feet/second, the circuit trips.

Both of the Excessive Vertical Rate Indicators are enabled at Flare Engage. They illuminate the same indicator lamp as the Lateral Rate Circuit. In any case, a rotation/go-around is required.

4.5 AWLS AFCS SYSTEM

4.5.1 General Description

The AFCS consists of the following components:

a. The AFCS Coupler

b. The Yaw Damper Computer

c. The Aileron Computer

d. The Elevator Computer

e. The Flare Co

f. The CAT III Adap r

g. The AFCS Control Panel

h. The Test Programmer and Logic Computer (TPLC).

The system is comprised of the basic C-141A components (Category II) listed, but totally integrated with the CAT III Adapter which will provide all required computations to enter the Category III weather environment. Thus, during the initial phases of the approach, basic procedures and system operation are identical to the standard C-141A CAT II AWLS. To maintain continuity, some of these details are described herein.

The integrated system maintains the dual channel, fail-passive characteristics of the standard C-141A AWLS, whereby an active computational channel is constantly compared with a model channel. Differences exceeding a specific level will disengage the affected axis and illuminate an appropriate fault indicator lamp. These tolerances in terms of aircraft parameters are listed in the Appendix. The text will describe only the active channel although a redundant channel does exist.

4.5.2 Lateral Channel Description (Refer to Figure 25)

4.5.2.1 AFCS Coupler Description

The AFCS Coupler contains separate pitch and roll channels. The roll channel inputs for an AWLS Approach are as follows:

a. The Aircraft Heading

b. The Preset Course and Heading from the HSI

c. The #1 and #2 LOC Deviation

d. The #1 and #2 Radar Altitudes

e. The Roll Attitude from the TPLC (ISS #2 and #3).

These inputs are processed by the Coupler (depending upon the NAV mode) to develop a steering command signal to the Aileron Computer.

4.5.2.2 Aileron Computer Description

The Aileron Computer processes roll signals from the Coupler. Input signals to the Aileron Computer for an AWLS approach consist of the following:

a. The roll rate from the #1 and #2 rate gyros

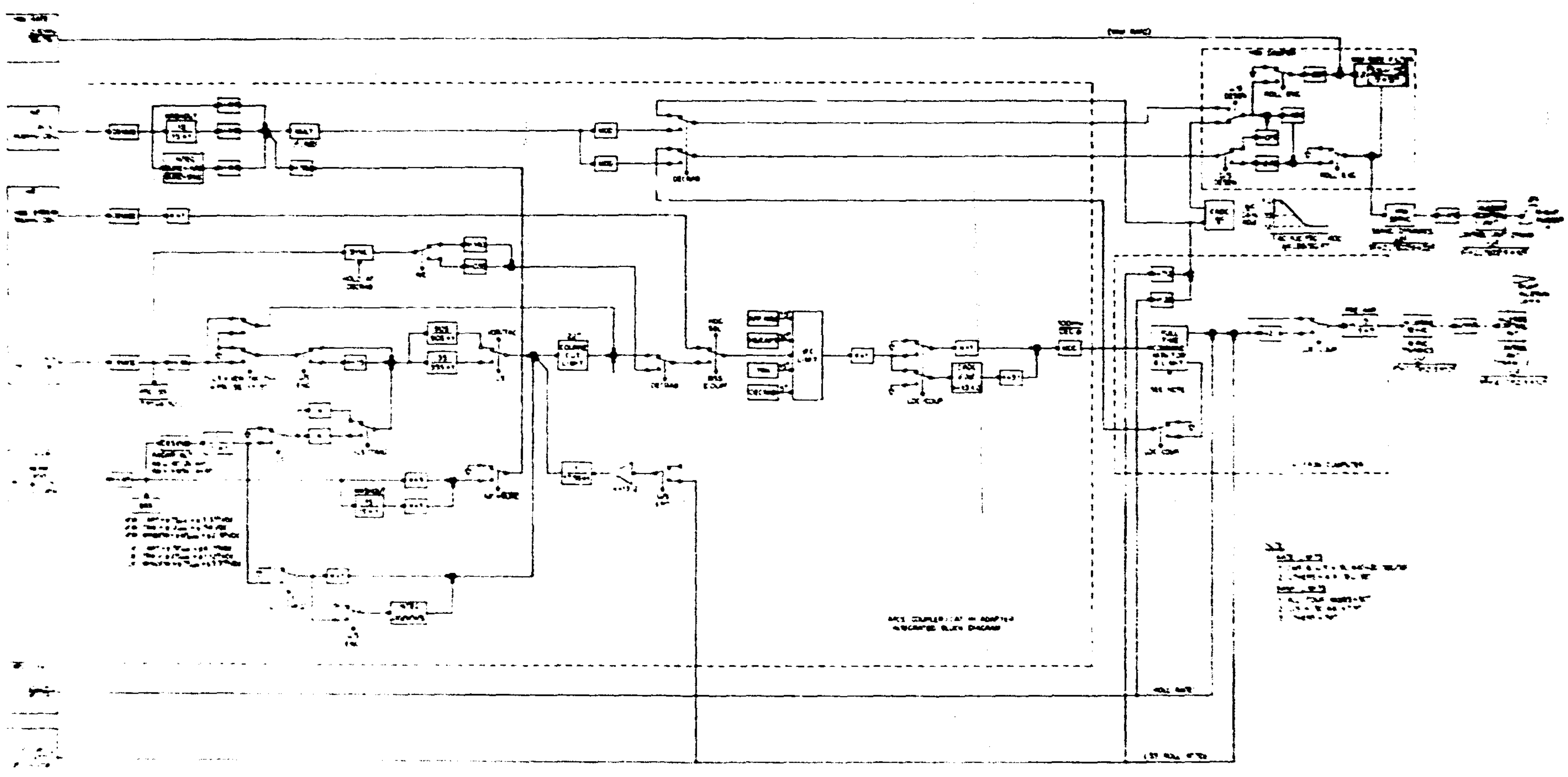


FIGURE 25. AFCS LATERAL AXIS BLOCK DIAGRAM

b. The roll attitude from the TPLC (ISS #1 and #2)

c. The active and model channel commands from the Coupler

d. The Wind-Down Command from the CAT III Adapter.

4.5.2.3 Yaw Damper Computer Description

The following are the inputs to the Yaw Damper Computer:

a. The yaw rate from the #1, #2, and #3 Yaw Rate Gyros

b. The turn coordination commands from the Aileron Computer

c. The Roll Command Crossfeed from the Aileron Computer

d. The Decrab/Roll-Out Rudder Command from the CAT III Adapter.

4.5.2.4 CAT III Adapter Description

The inputs to the CAT III Adapter are as follows:

a. #1 LOC Deviation from the pilot's NAV Receiver and #2 LOC Deviation from the co-pilot's NAV Receiver

b. #1 INS preset course from the pilot's LTN-51 INS System and #2 INS preset course from the co-pilot's LTN-51 INS System.

4.5.3 Lateral Channel Operation

4.5.3.1 General Operation

The two lateral modes normally associated with an automatic AWLS approach are Heading Select and NAV. An appropriate Localizer Frequency must be selected on both Navigation Receivers. Additionally, the Yaw Damper and the AFCS must be engaged, the AWLS Switch set to ARMED, and the Lateral Mode selected on the AFCS Controller.

Positioning and establishment of an intercept heading for Localizer Capture is normally accomplished in the Heading Select Mode. Although this may be done manually through the operation of the Turn Knob by using the Heading Select function, the AWLS is placed in a coupled state and interception of the beam can be accomplished automatically and with a minimum of switching.

Computationally, the Heading Select Signal is taken directly from the co-pilot's HSI and is the difference between the aircraft heading and the heading desired as set into the pilot's HSI with the HEADING SET knob. The ac signal is demodulated and limited in the AFCS Coupler and then sent for processing to the Central Air Data Computer (CADC) as a function inversely proportional to the Mach Number. The signal is then modulated and applied to the Full Time Command Modifier (FTCM) in the Aileron Computer.

The Full Time Command Modifier (FTCM) establishes additional bank limits and modifies abrupt signal changes by rate limiting its output to specific values.

In the Heading Select Mode, the rate of change of the output signal is limited to degrees/second/volt value equal to 4.8 degrees/second when summed with the Roll Rate Gyro Signal.

Roll Attitude is summed with the composite FTCM Roll Rate Signal to provide a heading error/roll attitude ratio which approximates 1:2 at the low Q conditions normally encountered in the approach environment.

Turn coordination for this mode is in the form of a composite Roll Attitude/Roll Rate Signal applied from the Aileron Computer, through the CADC which sets the gain inversely proportional to Mach Number and applied to the Yaw Damper as a Displacement Command Signal with a lead term since it is also applied to the Yaw Rate Filter (Wash-Out) circuits.

Having established an intercept heading to the localizer, the pilot now needs to assure himself that his proximity to the localizer is such that no false nulls will be encountered and then place the HEADING SELECT/NAV switch on the NAV Selector Panel to the NAV position. With this action, a Localizer Signal Switch Circuit (BSS) in the AFCS Coupler will engage the Localizer Capture Mode when the localizer signal becomes less than 150 uA and remove the Heading Select Signal.

During the Localizer Capture Mode, Localizer Deviation (limited in magnitude equivalent to 22 degrees of Preset Course Error) is summed with that term. The composite signal is applied to the Roll Command Limiter which limits the maximum value to an equivalent of ± 30 degrees bank attitude and applies it directly to the full time Command Modifier in the Aileron Computer after modulation. The Localizer Capture Mode shunts the Mach Compensation Circuit in the CADC.

An additional roll-to-yaw crossfeed is applied to the Yaw Damper from the Aileron Computer for tighter turn coordination. This additional signal is derived from the Coupler Roll Command and is additive to the original term. It does not contain a lead term since it bypasses the Yaw Rate Filters. At the same time, the minor loop gain of the Aileron Computer Servo Loop is doubled.

No other changes occur during Localizer Capture until the Localizer Beam Signal Switch of less than 25 uA and the Preset Course Signal Switch of less than ± 17.5 degrees are both satisfied. At this time, several events occur as follows:

a. The Hard Preset Course Signal is removed

b. A Washed-out Preset Course Term is introduced.

c. A Localizer Rate Signal is introduced.

d. The Roll Command Limit is reduced from ±30 degrees to ±15 degrees of roll attitude.

Since the Localizer Deviation is the only hard signal, it is predominate and the aircraft will be allowed to establish whatever heading or crab angle into the wind is required to maintain localizer center (crosswind correction).

No additional switching occurs in the Aileron Computer or the Yaw Damper at this time.

At Glideslope Intercept, considerable changes to the lateral computation occur. The following occurs in the AFCS Coupler:

a. The Washed-out Preset Course Term is sharply attenuated.

b. The Localizer Rate Term is increased.

c. A Lag Roll Attitude Term is introduced and replaces the washed-out course signal as the primary damping term.

d. A Localizer Beam Integrator is enabled.

e. The bank command limit is reduced to ±7.5 degrees of roll attitude 30 seconds after Glideslope Engage.

In the Yaw Damper, the following occurs:

a. The Mach Compensation for the Turn Coordination Signal is removed and is applied directly without attenuation.

b. The roll-to-yaw crossfeed derived from the Coupler Roll Command is sharply amplified. Thus, a significant portion of localizer tracking is transferred to the yaw axis which results in reduced bank angles and flatter turns.

Glideslope Engage Logic also initiates a Pre-Land Test generated by the TPLC. This test (30 second duration) accomplishes the following:

a. Fails and heals all critical comparators in the standard AWLS in a rigid time sequence. Both failure and healing conditions must be satisfied or a fault will be registered in the TPLC.

b. Generates a Flare System Self-Test.

Following the 30 second Pre-Land Test, the TPLC will generate an Approach Arm Logic. Four events occur:

a. Torque limiting is removed from the Roll and Pitch AFCS Servo Actuators.

b. The AWLS Master Caution circuits are enabled and faults registered during the Pre-Land Test are displayed.

c. The 2 comparators in the CAT III Adapter are on-line.

d. An Approach Arm annunciator is illuminated to advise the pilots that the Pre-Land Test is completed.

Except for Localizer Beam Desensitization which is designed to minimize beam convergence, no further switching occurs until Side-slip Initiation at 20 feet AGL.

4.5.3.2 Side-slip Computation

The Side-slip Computation provides the required signals to the roll and yaw axis of the autopilot to reduce the crab angle and align the aircraft with the runway (decrab) plus establish the bank angle to maintain the aircraft track to touchdown.

For automatic operation, the Rudder Command is applied to the Yaw Damper Crossfeed circuits and the Wind-Down Command is used to replace the AFCS Coupler Input to the Aileron Computer. The point at which the Side-slip Mode is engaged is nominally set at 20 feet radar altitude with the Runway Heading Signal being taken directly from the INS for precision.

At Side-slip Engage, the Rudder Commands of the CAT III Adapter are enabled and thus provide commands directly to the Yaw Damper thereby replacing the original crossfeed commands. The computation for the decrab maneuver serves to bring the course error (Runway Heading Error) to zero. The Decrab Command Signal is applied to the Yaw Damper's Rate and Displacement Circuits. Thus, a Displacement Command plus a Lead Term is realized.

The Wind-Down Computation is derived directly from the heading change that is accomplished during the decrab maneuver by synchronizing the Runway Heading Error to zero until Decrab Engage. After Decrab Engage, any heading change will pass directly to the input circuits of the Aileron Computer through a ± 5 degree Bank Limiter.

The roll attitude and roll rate terms in the Aileron Computer provide the attitude limit and damping functions. Computationally, 1/2 degree of roll attitude is commanded to the roll axis for every degree of change generated by the decrab maneuver. The computation used by the CAT III Adapter will eliminate (or reduce) the downwind drift of the aircraft as generated by the decrab maneuver. Removal of the localizer tracking during this maneuver has little effect upon the final track of the aircraft. The aircraft will essentially maintain the track it had at DECRAB ENGAGE.

4.5.3.3 Roll-Out Computation

The Roll-out Mode is initiated after touchdown and Main Landing Gear Wheel spin-up. The roll-out channel computation uses both the Heading Error and localizer to provide the required steering commands to the aileron and rudder axes of the AFCS to acquire and maintain the localizer centerline.

Localizer is reintroduced into the rudder computation as the primary guidance signal. Runway heading and localizer rate now become a course damping term to prevent excessive cross-track rate build-ups and to maintain reasonable course angles back to localizer centerline. A gain change doubles the rudder authority and its effectiveness over a 2 second period.

In the aileron channel, the Wing-Down Command is reduced by 75% to eliminate a tendency of the aircraft to lean upwind for both the manual and automatic modes.

4.5.3.4 Back-Up Roll-Out System (BURO)

As in the Manual Mode, the BURO circuits computations remain synchronized to zero until the following conditions are met:

a. AWLS and Main Landing Gear Wheel Spin-Up.

b. LOC Flag in View - indicates a non-valid localizer signal.

If this situation arises, the circuit reverts from a synchronizer to an integrator. It then provides an integral of aircraft heading changes about the runway heading. This becomes the primary Rudder Command to the Yaw Damper. Through proper phasing of the signal, it directs the Yaw Damper to change the aircraft's heading in the opposite direction for the same amount of time that the original heading error was present. This, in essence, will return the aircraft to its original position on the runway and eliminate the divergent tendency experienced with only the basic Runway Heading Signal. With the Back-Up Roll-Out system in operation, the localizer signal path is grounded.

In either Roll-Out Mode, the pilot must assist the automatics as rudder effectiveness diminishes when the aircraft slows down and stops.

4.5.4 Longitudinal Channel Description (Refer to Figure 26)

4.5.4.1 General Description

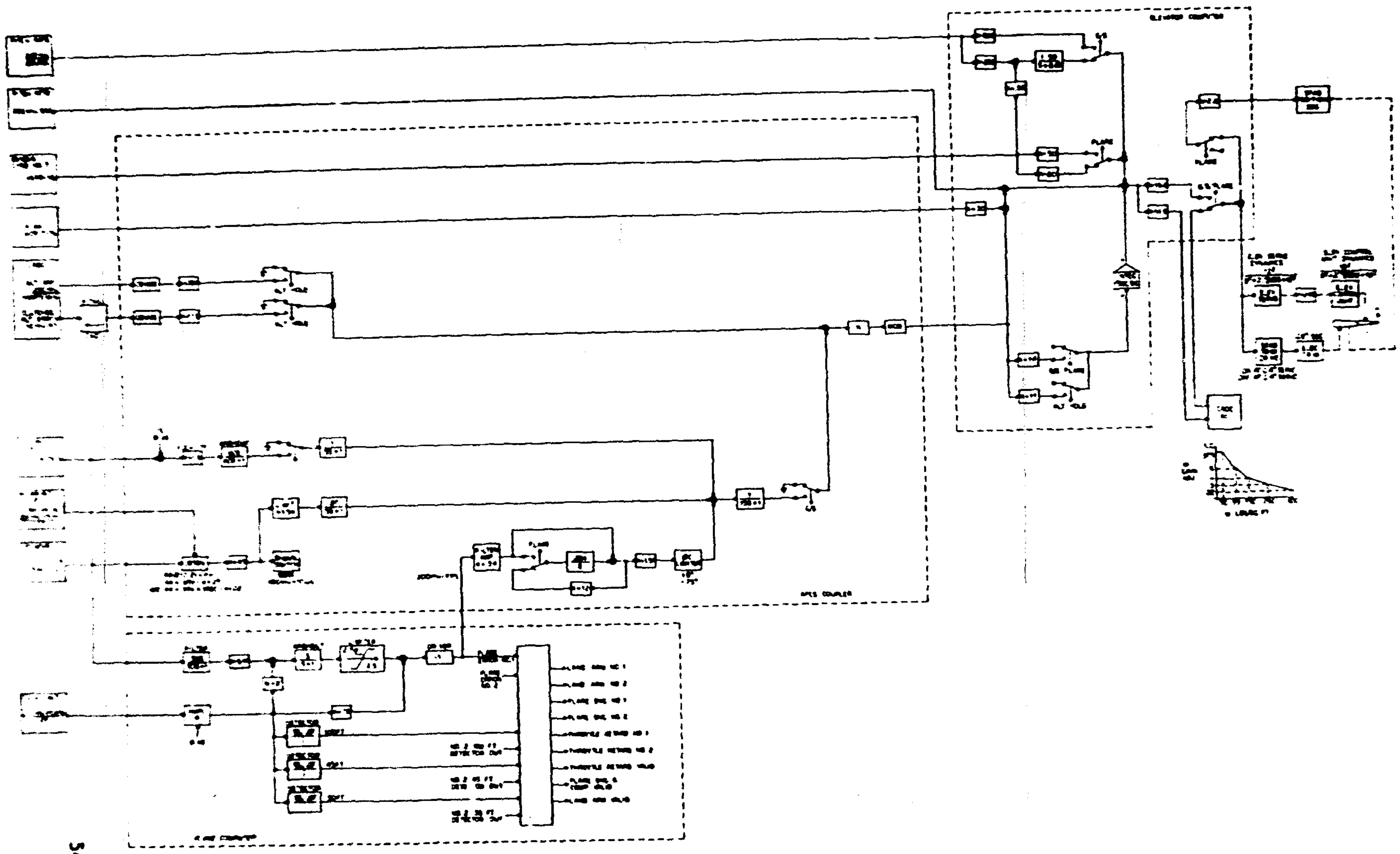


FIGURE 26. AFCS LONGITUDINAL AXIS BLOCK DIAGRAM

The longitudinal channel of the AFCS remains relatively unchanged in the Category III System as compared to the standard Category II System. Minor modifications were perfromed in the Flare Circuits and are described in detail.

4.5.4.2 AFCS Coupler Description

Pitch Channel inputs to the Coupler for an AWLS Approach are as follows:

a. The #2 CADC Altitude Rate and Altitude error.

b. The Vertical Acceleration Signal.

c. The #2 Radar Altitude.

d. The #2 Glideslope Deviation.

e. The Flare Error form the #2 Flare Computer.

These input signals are processed by the Coupler (depending on mode) to develop a Pitch Command for the Elevator Computer.

4.5.4.3 Elevator Computer Description

The input signals to the Elevator Computer for an AWLS Approach are as follows:

a. The Pitch Rate form the #1 and #2 Rate Gyros.

b. The Pitch Attitude form the TPLC (ISS #2 and #3).

c. The Band Versine form the #3 Vertical Gyro.

d. The position of the flaps.

e. The Horizontal Stabilizer Position Signal.

f. The AFCS Coupler Pitch Command.

Depending upon the autopilot mode, the Elevator Computer will drive the elevator control surfaces and produce 3 outputs as follows:

a. The Up and Down Stabilizer Trim to the Horizontal Stabilizer Actuator.

b. Up and Down Stabilizer Trim Cut-out to the Horizontal Stabilizer Actuator.

c. The Servo Effort to the Trim Indicator.

4.5.4.4 Flare Computer Description

The Inputs to the Flare Computer are as follows:

a. The Radar Altitude.

b. The Normal Acceleration.

The output of the Flare Computer is applied to the AFCS Coupler as a Flare Command. The unit also supplies the critical Flare Engage and Throttle Retard Logic.

4.5.4.5 Altitude Hold Mode Description

The Altitude Hold Mode is selected by placing the ALT HOLD/ PITCH OFF switch on the AFCS/AWLS Control Panel to ALT HOLD. In this mode, the Pitch Knob is disengaged from the potentiometer. The Altitude Synchro in the #2 CADC is now clutched in and deviations from the Engaged Attitude will result in a signal output. This output is adjusted as a function of true airspeed and then used to change the pitch attitude to return the aircraft to the referenced altitude. The signal is adjusted to keep the gain as high a possible over the entire performance range without introducing oscillations or overshoot. After the TAS gain adjustment of the Altitude Error Signal, the Altitude Rate Signal is added to it to provide added damping thereby allowing high gains to be used without causing oscillations. When the ALT HOLD is engaged, the Pitch Integrator is operating. The Integrator is an electro-mechanical device (without position feedback) that will drive at a rate proportional to the applied input signal until the signal is removed. The Coupler output signal (whose only steady-state component is Altitude Error) is the pitch attitude and will continue to do so until the input is at null. When the ALT HOLD is disengaged, a centering clutch restores the Integrator to its null position.

4.5.4.6 Automatic Glideslope Mode Description

The AFCS is designed to automatically engage and fly the glideslope. This may be done whether or not LOC is used to command the lateral axis of the system. Engagement of the G/S beam is subject to the following:

a. An ILS frequency must be selected on the co-pilot's Navigation Receiver.

b. The VOR/ILS Button of the co-pilot's Navigation Selector Panel must be depressed and the HDG SELECT/NAV Switch must be set to NAV.

c. Glideslope Deviation greater than 30 mV in either direction must be encountered to arm the signal switch, followed by deviation of less than 10 mV fly-up or any fly-down deviation to capture.

When the GS/VERBAV switch is placed in the G/S position, a green G/S ARMED Indicator Lamp on the AFCS Trim Indicator will be illuminated to indicate completion of the cockpit switching and a reminder that the aircraft will automatically fly-down the glideslope upon interception of beam center. At Beam Engagement, the G/S ARMED Indicator Lamp will extinguish and Fly-down Command will be initially smoothed.

The Glideslope Signal is de-sensitized as is the LOC Mode with the gain changing from 1 at 1000-feet to .25 at 75-feet and then to 0 as a function of time (3.0-second time constant). Pitch commands from the glideslope are always limited to ±7.5-degrees, when the Pitch Integrator is active.

Additional damping in the G/S is provided by the Normal Acceleration Signals. The Accelerometer output at 1g is 8 volts. A bias is therefore, provided such that the summation of the accelerometer signal and the bias vary about 0 volts. This signal is then washed out by a long time constant (over 50-seconds) wash out to eliminate any steady-state condition, lagged in a 5-second Lag Circuit, and then summed with the Glideslope Signal. The composite signal is applied to the Elevator Computer as a Pitch Displacement Command. Major loop computations such as Beam Integration, Pitch Rate Damping, and the Pitch Attitude Reference are contained in the Elevator Computer.

4.5.4.7 Flare Mode Description

Dual Flare Signal Inputs are provided to the Coupler. The Altitude Rate Error Signal from the Flare Computer is the input to the Flare Synchronizer Integrator. Prior to Flare Engage, the device is operating as a synchronizer, and keeps the input to the Unsymmetrical Limiter at null. At Flare Engage, the device becomes an Integrator, with the input being the Flare Error Signal. Thus, a combination of Flare Error and Integrated Flare Error will be summed and routed to the Unsymmetrical Limiter which will allow all nose-up commands but limit all nose-down commands to 3/4-degree. The Limiter output is then summed with the Normal Acceleration Signal (G/S is 0), modulated, and routed to the Elevator Computer. The Pitch Integrator is locked during the Flare. At Touchdown (Nose Gear squatted), the AFCS pitch axis is disengaged.

Basic design mechanization of the Auto Flare Control Loop dictated that the Coupler output must increase to and maintain a Nose-up Command to compensate for the nose-down attitude stored in the Pitch Integrator of the Elevator Computer. The Pitch Integrator establishes a Nose-down Command consistant with the flight path angle assumed while flying the glideslope (a nominal -3-degrees). At the Flare Engage point, the Pitch Integrator is disabled; thus, the stored nose-down attitude remains as the primary pitch attitude reference to the Elevator Computer.

Investigation revealed that in the original mechanization of the Flare Error Integrator in the AFCS Coupler, a diode had been inserted into the Integration Loop. The function of the diode was to prevent the Integrator from generating a Nose-down Command after Flare

Engage. The concept is logical; however, the Integrator is synchronized to a nominal zero. At Flare Engage, the Flare Error must exceed the diode's forward voltage drop of about 600 mV before the Integrator can function (Refer to Figure 27). To correct this and other deficiencies, the following modifications were made on the Flare Integration Card:

a. The diode was removed to eliminate the dead zone.

b. The Direct Flare Error Gain was reduced 25% to bring the initial aircraft rotation to a reasonable level.

c. The Nose-up Ramp was adjusted to provide a 2-degree Pitch-up Command over 8-seconds.

d. The Flare Integrator Gain was unaltered and remains at 2.94 volts/second output for every volt applied.

After making minor modifications (to increase the Touchdown rate), the automatic flares were observed to be consistant, well within the vertical velocity specification range, and with minimal longitudinal dispersion. On board recordings of the Flares are shown by Figures 28 and 29. Note that the Coupler output builds up and maintains an output of about +3-degrees attitude command. Simultaneously, the Flare Error is reduced and maintained near null.

4.6 RUNWAY DISTANCE REMAINING INDICATOR

4.6.1 Purpose

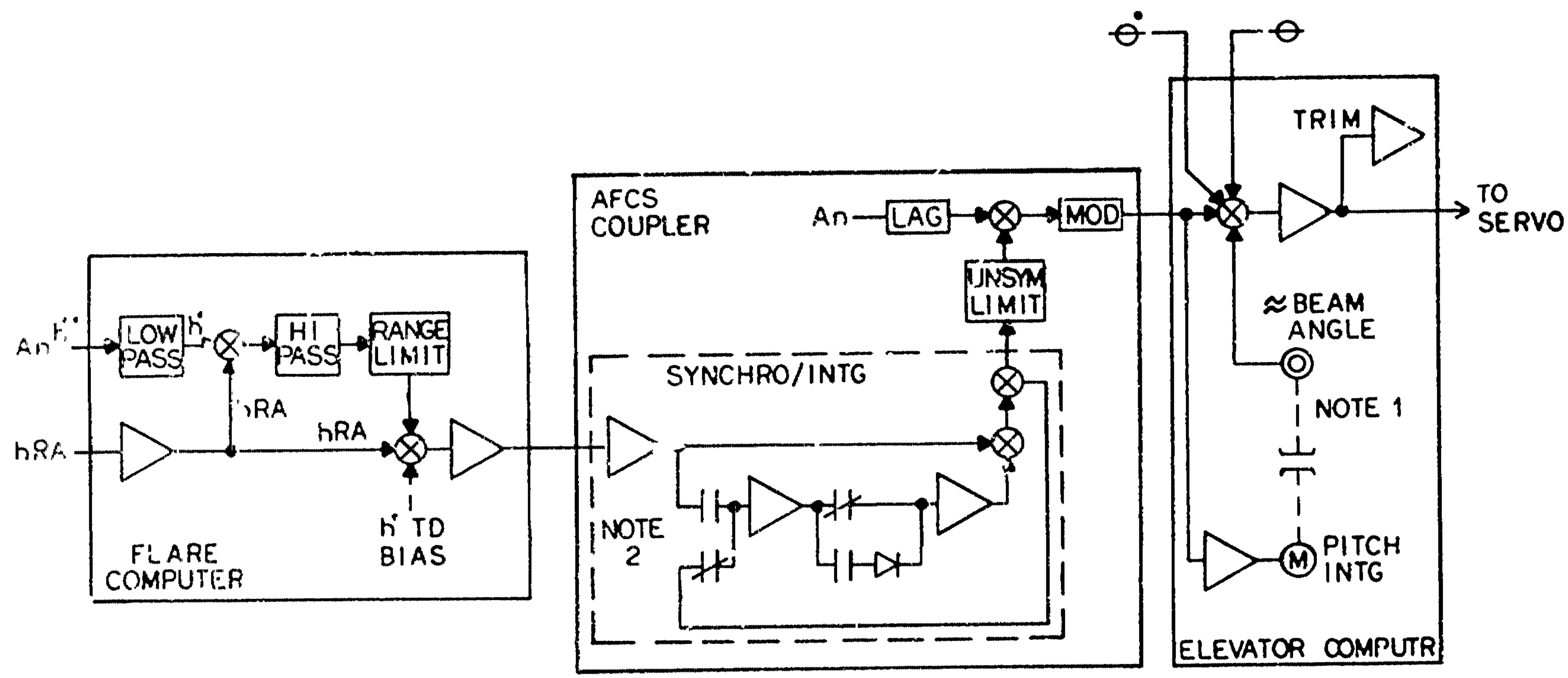
Early in the flight test phase of the C-141 AWLS, it became apparent that the crew would need accurate information as to the position of the aircraft along the runway. In low visibility weather, the only way a pilot could be assured of having sufficient runway length for Take-off Roll and a possible abort was to taxi to the beginning of the runway for every Take-off. Runway distance remaining was also required to perform Touch-and-Go Landings in case the pilot elected to abort during the Ground Roll. Since restricitng the test operations to full-stop, taxi-back landings would severely limit the number of approaches that could be made, a Runway Distance Remaining (RDR) Indicator was developed.

4.6.2 Physical and Functional Characteristics

The unit was designed with the indicators and switches mounted as a single device (Figure 12) on the glare shield in front of the co-pilot (Figure 11). The unit is 7.5-inches long, 10-inches wide and 2-inches high and weights 3-pounds.

The indicator is a Weston, Model 1201, flat-scale, edge light projected moving pointer, 0-10 Vdc. The scale is calibrated in 250-foot increments up to 15,000-feet maximum.

The analog computation of runway distance is an integral of runway



NOTE:

1. DECLUTCHED AT FLARE ENCAGE. G/S INTG SIGNAL ≈ TO G/S BEAM ANGLE.
2. FLARE ENGAGE RELAYS SHOWN IN PRE-ENGAGED POSITION.

FIGURE 27. FLARE COMPUTATION

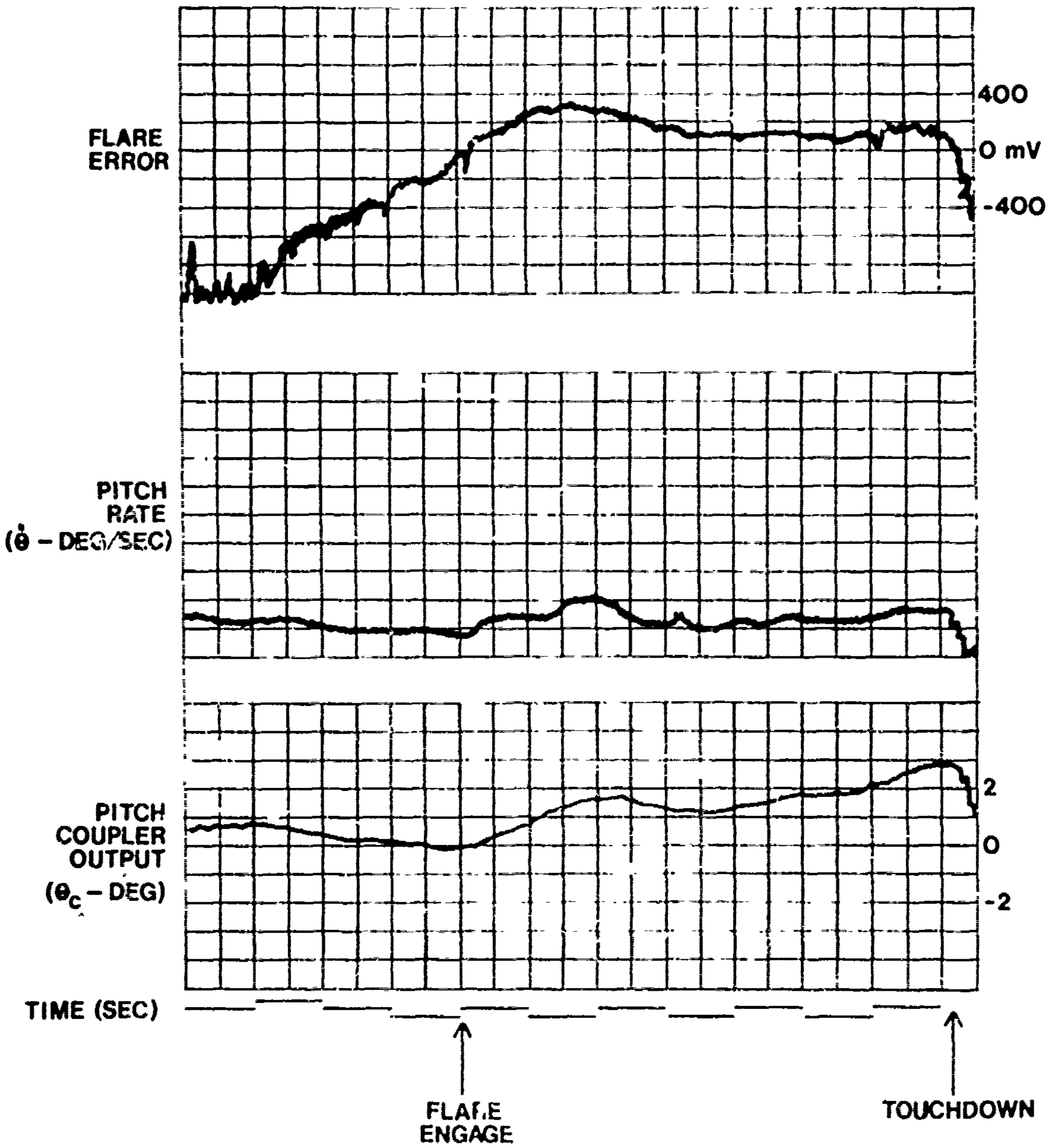


FIGURE 28. MODIFIED AIRCRAFT RESPONSE

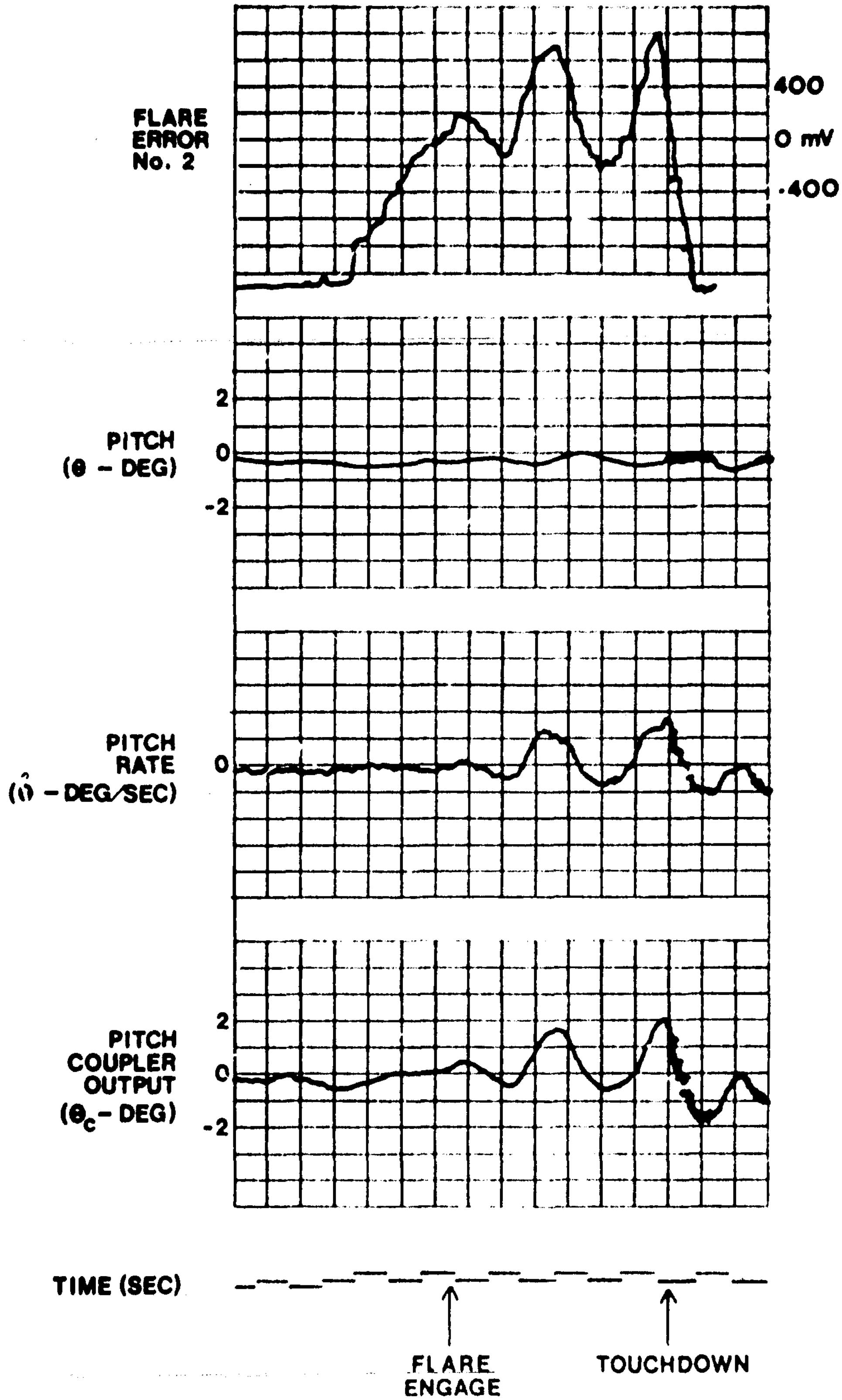


FIGURE 29. UNMODIFIED AIRCRAFT RESPONSE

speed. The LTN-51 (INS) supplies the speed signal and requires the proper runway heading selection for correct RDR operation. The runway speed integration is differentially summed with the runway distance set into the unit, thereby resulting in an indication of the runway distance remaining at any given point (Figure 30).

4.6.3 RDR Operation

The RDR may be used either for an AWLS Take-off or an AWLS Approach and Landing. In either case, the system power must be turned on. For an AWLS Take-off, the RDR ARM/RESET Switch is momentarily set to the RESET position. Using the RUNWAY DISTANCE SET Knob, the RDR is adjusted to indicate the number of feet of runway remaining at commencement of the Take-off Roll. Prior to beginning the roll, the T. O. (MANUAL) Button is pressed to engage the system. Take-off then proceeds with the RDR functioning from the INS inputs.

For an Approach and Landing, the power is turned on, the RDR ARM/RESET is placed to RESET and the RUNWAY DISTANCE SET Knob is adjusted to the runway length. After completion of the Approach Arm System Self-Test, the RDR ARM/RESET Switch is placed to the RDR ARM position. With the RDR on and armed, the indicator will be activated by Flare Engage (45-feet of radar altitude) and will function from the INS inputs.

4.7 MODIFIED RGA COMPUTER DESCRIPTION

In response to the pilot comments that the climb attitude in the RGA Mode was too steep, the RGA Computer was modified to lower the pitch limit from the nominal +15 degrees to a more comfortable +10 degrees limit. At the test aircraft's gross weight, acceleration to 1.3 Vs is extremely rapid which results in the Computer commands for the maximum pitch attitude very shortly after rotate speed is reached.

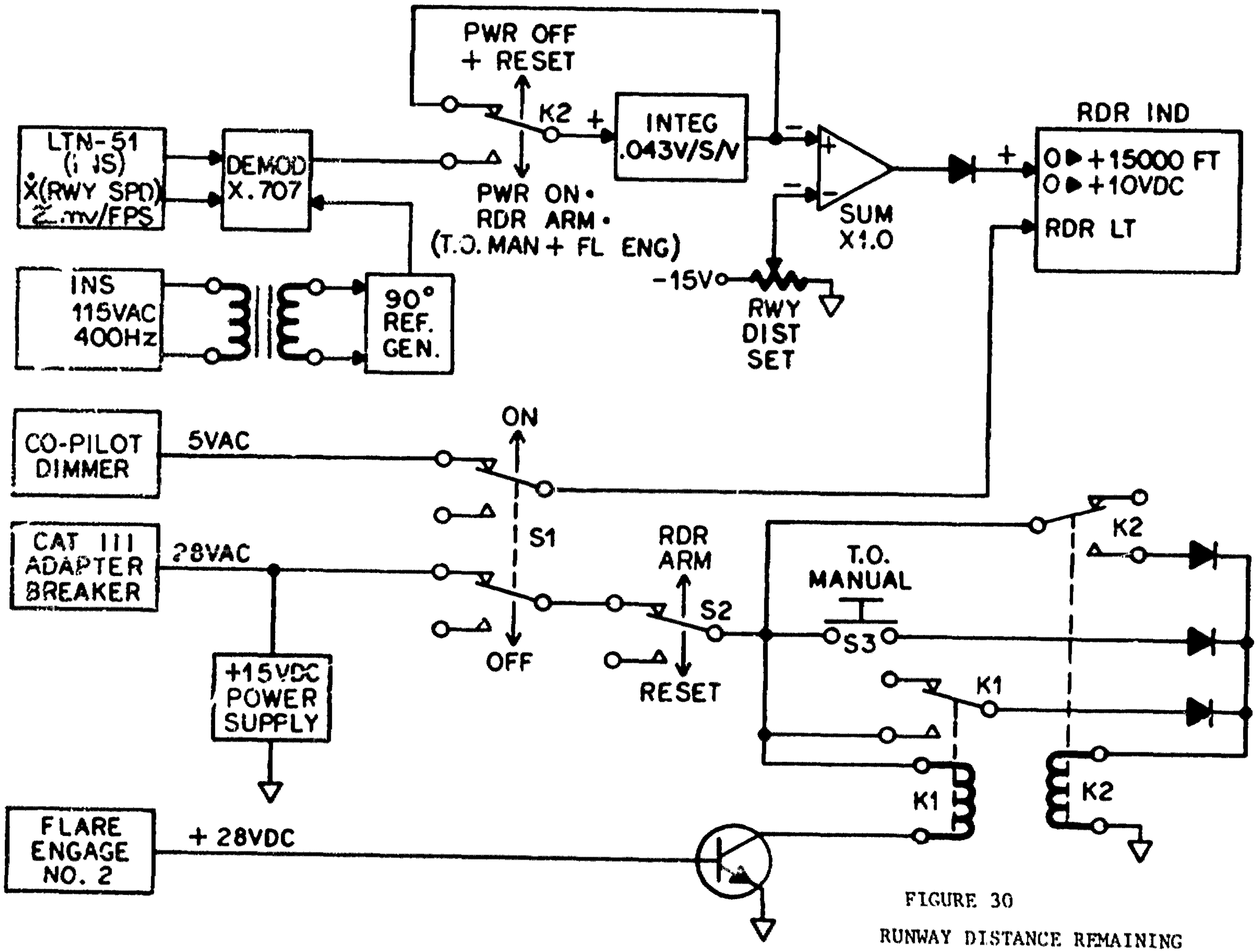


FIGURE 30

RUNWAY DISTANCE REMAINING
SIGNAL FLOW

SECTION V

TEST INSTRUMENTATION DESCRIPTION

5.1 PURPOSE

The purpose of this section is to provide a description of the test instrumentation aboard the project aircraft.

5.2 GENERAL DESCRIPTION

Instrumentation is extensive and breakd down into the following four major areas:

a. The Primary Data Acquisition System associated with the Instrumentation Rack.

b. A closed-circuit, Color Television System with video recording abilities.

c. Separate direct read-out recording of all critical system comparators.

d. Photographic coverage in 3 separate areas of the aircraft.

5.2.1 Instrumentation Rack Description

The Instrumentation Rack is installed in the cargo compartment and contains the following items:

a. A Teledyne Automatic Digital Data Acquisition System (ADDAS).

b. Signal Conditioners.

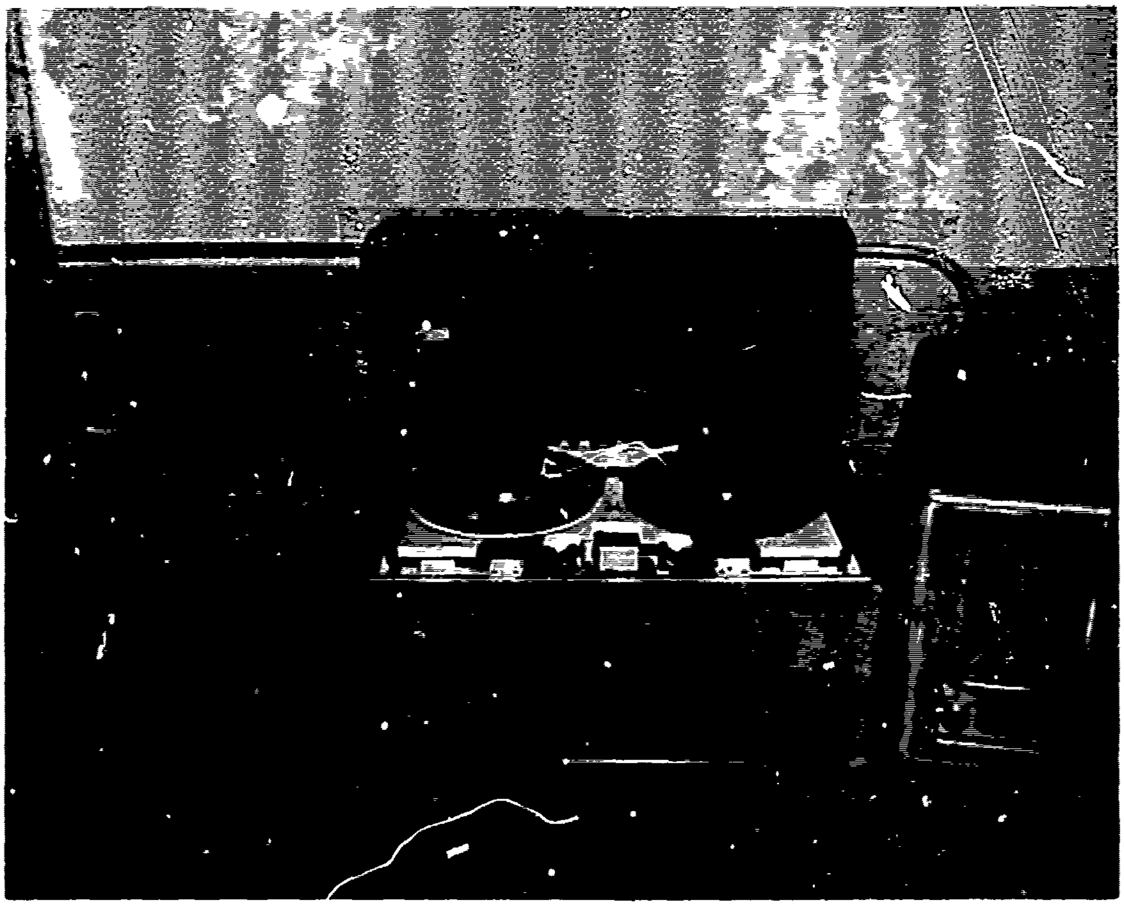
c. The Brush Recorder Patch Panel.

d. The Brush Recorder.

e. Clock

5.2.2 Automatic Digital Data Acquisition System Description

The Automatic Digital Data Acquisition System (ADDAS shown in Figure 31) is a magnetic tape recorder capable of recording 162 data channels. Controls for the ADDAS are mounted on the recorder front panel. Circuit protection is on the Instrumentation Rack CB Panel. A master instrumentation data link is contained in the Appendix and defines all parameters recorded by the ADDAS.



(Figure 31) Automatic Digital Data Acquisition System

5.2.3 Signal Conditioning (Figure 32)

A separate Signal Conditioner is installed for each parameter listed in the Instrumentation Data Sheet (Appendix E). The purpose of each Signal Conditioner is to convert the analog and discrete signals to working levels of ± 5 volts as required by the ADDAS for the digital conversion.

5.2.4 Brush Patch Panel

Every Signal Conditioner output is paralleled to a Patch Panel. Through proper programming of the panel, any 8 of the parameters can be applied to and read-out directly on the Brush Recorder.

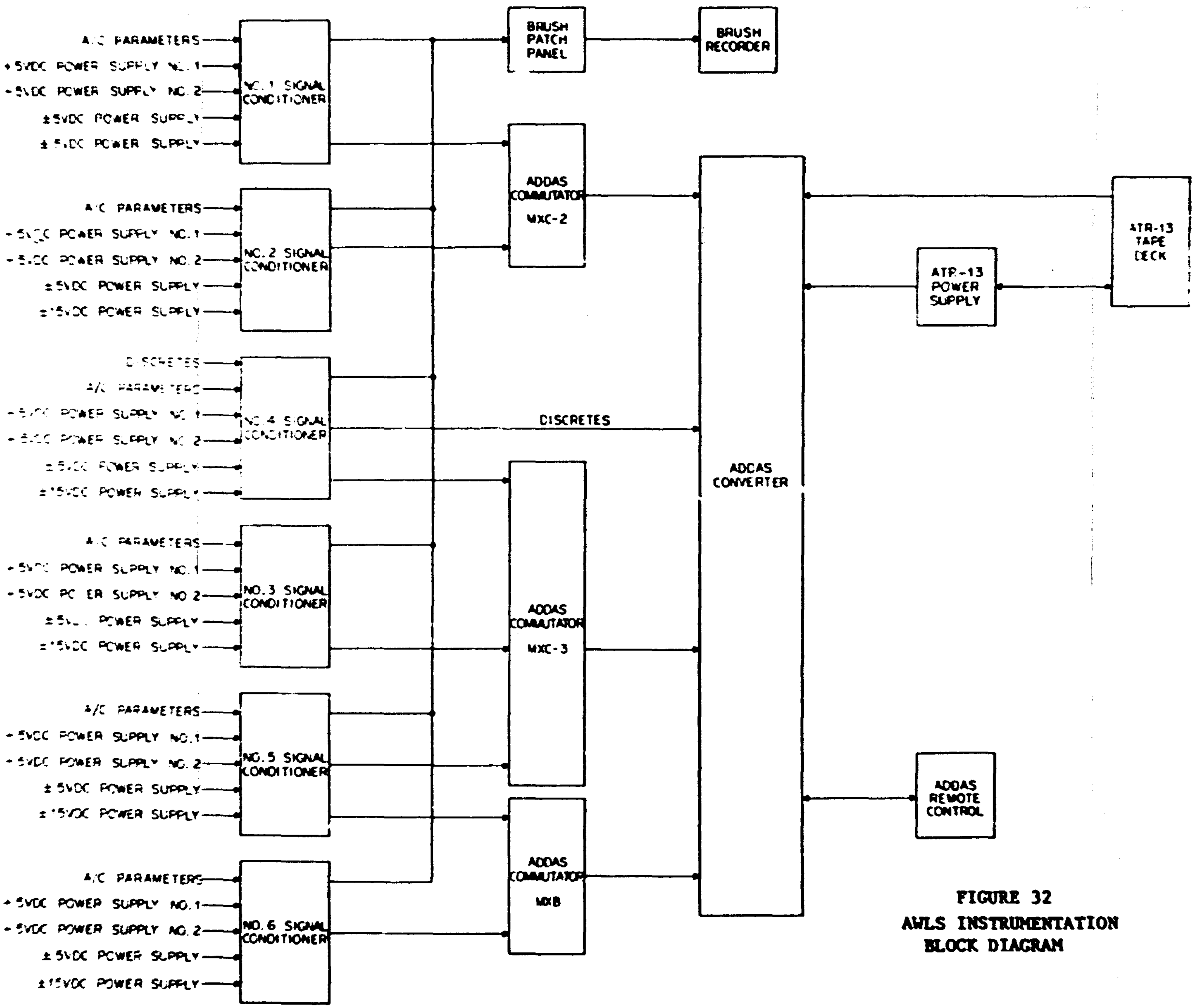


FIGURE 32
AWLS INSTRUMENTATION
BLOCK DIAGRAM

5.2.5 Brush Recorder

The Brush Recorder is a hot-pen type recorder, capable of providing a direct read-out of 8 data channels. Controls for the Brush Recorder are mounted on the front panel of the Recorder. Circuit protection is on the Instrumentation Rack CB Panel.

5.2.6 Closed Circuit Color TV System

The color TV system consits of the following:

a. The Color TV Camera.

b. A Color and Sync Control Unit.

c. A 19-inch Color TV Monitor.

d. The Cassette Video Recorder.

The Camera is installed in the cockpit directly above the jump seat between the pilots. The mount is adjustable and allows the Camera to look forward through the windshield or to be directed on either the pilot's, co-pilot's or the center instrument panel. Camera position is established by mission profile requirements.

The remaining units are installed at the Test Director's Console in the cargo area. All controls providing the Monitor and Record functions are thus available to the test director.

The TV Monitor and the Video Recorder are also utilized to display and/of record the EADI/ALR presentation by simply interchanging coaxial cables at the unit.

5.2.7 Visicorder Installation (Figures 33 and 34)

A 24-channel, direct read-out oscillograph is mounted on the AWLS Rack. This unit is used as a maintenance tool in that it records critical validities of the AWLS. Thus, the specific comparator and the time of failure are immediately available to the crew. Affected circuits and parameters can then be patched to the Brush Recorder with verification and isolation of the problem being possible while still in flight.

5.2.8 Photographic Coverage

Three separate movie camera installations are aboard the project aircraft. An additional installation is available at the cockpit TV camera mount. The 3 normal positions are as follows:

a. The Instrument Panel Glare Shield

b. The Test Director's Console.

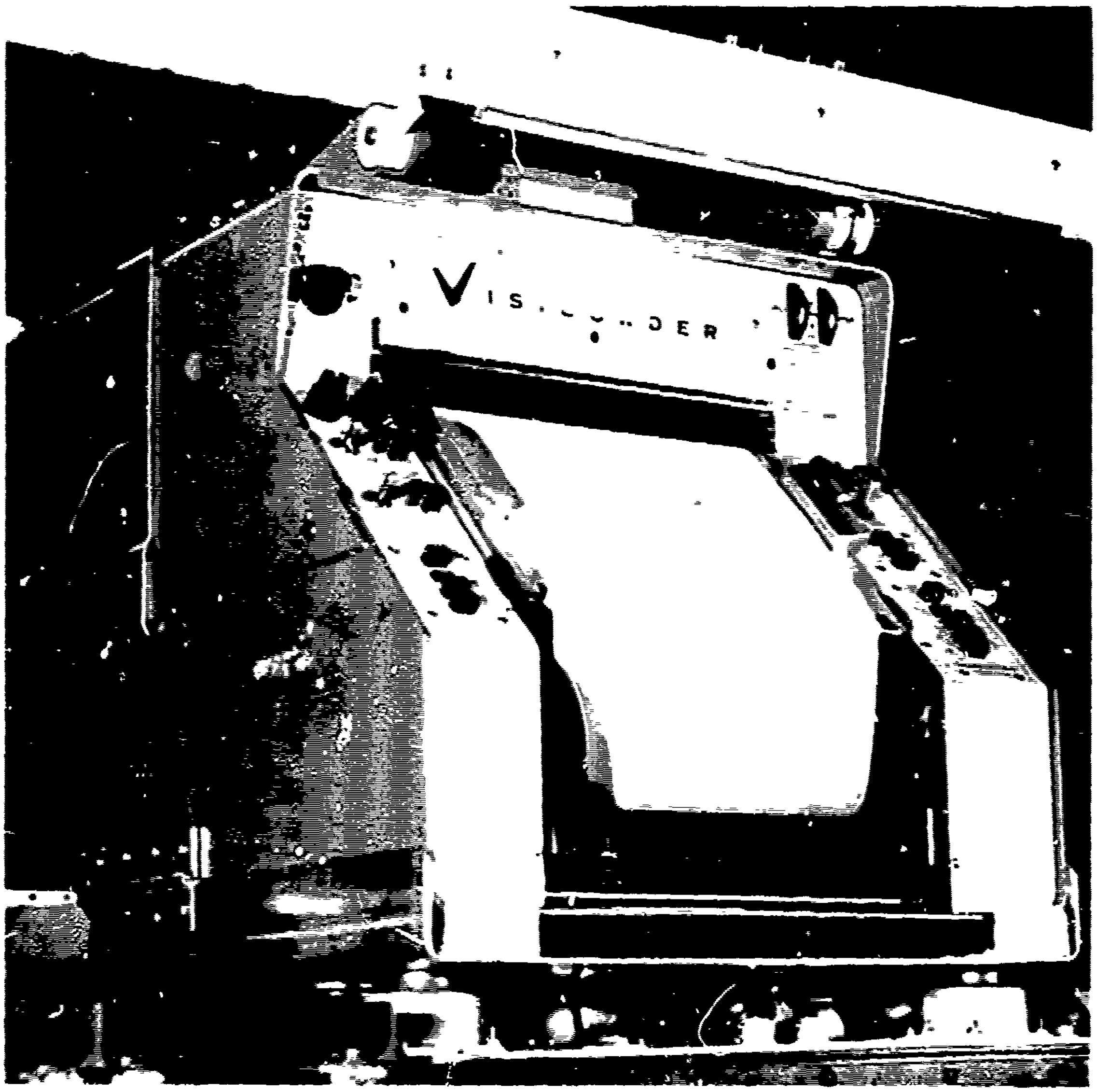
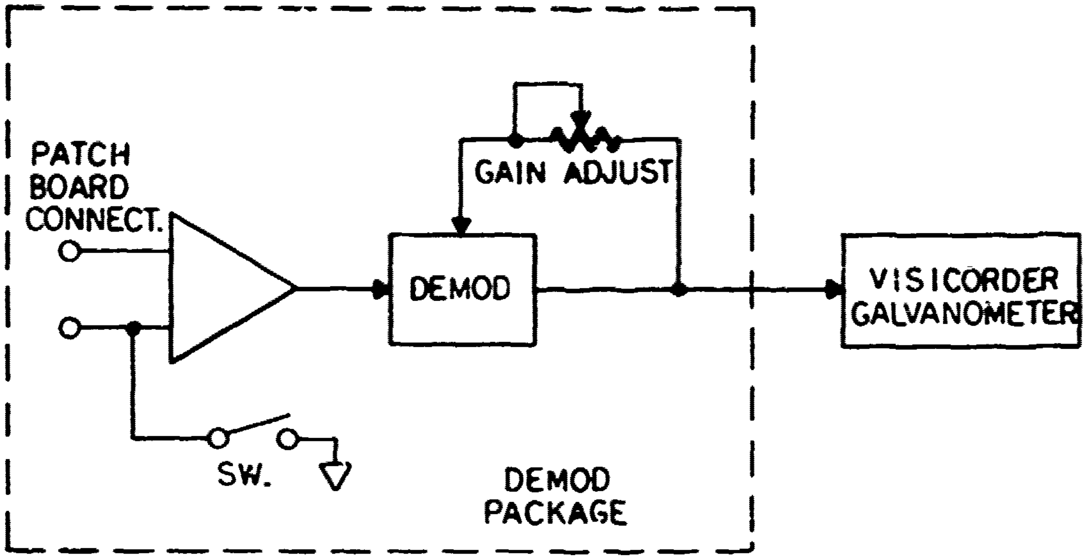
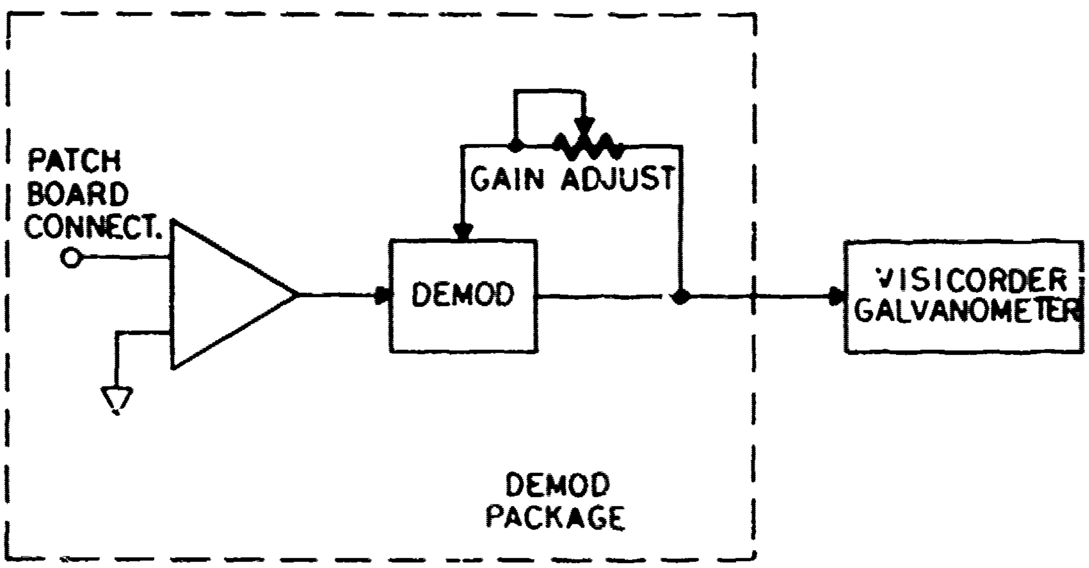


FIGURE 33. VISICORDER



TYPICAL DEMOD/VISICORDER
SIGNAL FLOW FOR CHANNELS 1-12.



TYPICAL DEMOD/VISICORDER
SIGNAL FLOW FOR CHANNELS 13-24

FIGURE 34. VISICORDER LOGIC

c. Externally, on the tail

The camera mounted on the Instrument Panel Glare Shield provides the primary forward looking documentation of all weather missions.

The camera installation at the Test Director's Console provides photographic coverage of those areas of the Test Director Panel dictated by the mission profile.

A camera is mounted on the left side of the vertical stabilizer just below the horizontal stabilizer. This camera provides over-the-nose photographic coverage.

All cameras are remotely controlled from the Test Director's Console.

SECTION VI

EADI/ALR DESCRIPTION

6.1 PURPOSE

The purpose of this section is to provide the fundamental theory of operation as well as the systems description of the Electronic Attitude Director Indicator (EADI), Aircraft Landing Radar (ALR) systems.

6.2 EADI DESCRIPTION

6.2.1 General Description

6.2.1.1 System Description

The primary function of the EADI system is to display the aircraft attitude, steering commands, and navigational cues based on data received from the external avionics systems and internally derived signals. The display will present the following representative symbology:

a. Reference Aircraft.

b. Radar Altitude Reference Height.

c. Artificial Horizon Line.

d. Auxiliary Pitch Lines.

e. Roll Pointer and Indices.

f. Sky/Ground Shading.

g. Flight Path Reference Line.

h. Roll and Pitch Command Bars.

i. Speed Error.

j. Rate of Turn.

k. Radar Altitude.

l. Localizer Deviation.

m. Glideslope Deviation.

n. Tele-Visual Displays.

o. Decrab Commands.

Symbology that is fixed or moves within the aircraf reference frame is generated by raster modulation. A Roll Stabilized Symbology is generated by using the stroke writing technique, except for the sky/ground shading. Priority of the symbology is shown by Table 1.

TABLE 1. SYMBOL PRIORITY

Priority	Symbol	Raster/Stroke	Shade
Top Priority (Stroke written symbology enhances raster at coincidence)	Flight Path Acceleration Flight Path Angle Artificial Horizon Auxiliary Pitch Lines Roll Pointer Flight Path Reference	Stroke	White
I	Reference Aircraft Altitude Numerics	Raster	Peak White
II	Flight Director Commands Speed Error Roll Indices Altitude Window[1]	Raster	Black
III	ILS Symbol	Raster	White
IV	Sky Shading	Raster	Light Gray
V	Ground Shading	Raster	Dark Gray
VI	Television Scene[2]	Raster	White to Black

NOTES: 1. All windows override (blank) stroke written symbology.
2. Television Scene is an optional feature.

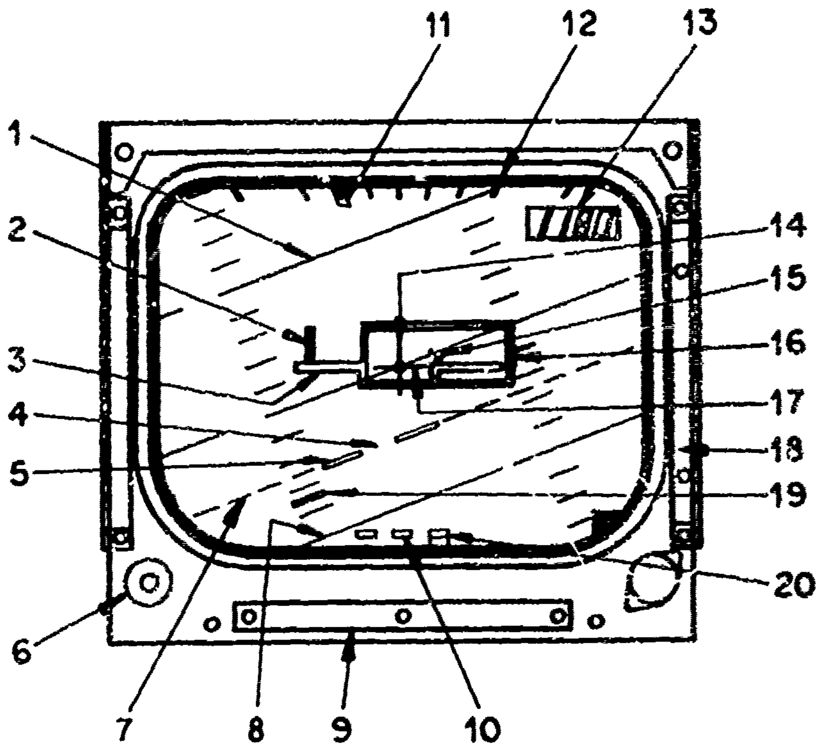
Table 2 shows the modes when the symbology is displayed and (Figure 35) shows the display presentation.

6.2.1.2 EADI Symbol Generator Unit

The primary funciton of the SGU is the receipt and processing of the analog and digital data into synthetic video display symbology and deflection signals. The entire interface between the EADI system and the other avionics equipment is provided by the SGU.

TABLE 2. MODE SYMBOLOGY

Symbol	Mode Select		
	Attitude	Normal	TV Only
Reference Airplane	Displayed	Displayed	Displayed
Artificial Horizon	Displayed	Displayed	Not Displayed
Auxiliary Pitch Lines	Displayed	Displayed	Not Displayed
Flight Path Reference	Displayed	Displayed	Not Displayed
Roll Pointer and Reference	Displayed	Displayed	Not Displayed
Altitude	Displayed below 2,500 feet	Displayed below 2,500 feet	Not Displayed
Roll Command and Pitch Command	Not Displayed	Displayed when F/D switch is turned on	Not Displayed
Speed Error	Not Displayed	Displayed when Speed Error switch turned on	Not Displayed
Flight Path, Flight Path Acceleration and Drift Angle	Not Displayed	Displayed	Not Displayed
ILS	Not Displayed	Displayed below 800 feet	Not Displayed
Television	Not Displayed	Displayed when Approach TV Enable switch turned on; TV system slaved to EADI scaling	Displayed



1. Pitch Scale, F/D Mode (shown at +10° Pitch).
2. Speed Error (shown at 5 knots "too fast")
3. Reference Aircraft
4. Drift Angle Gap (shown at -3.23° Drift Angle).
5. Velocity Vector (shown at +5° Flight Path Angle).
6. EADI Test Button
7. Flight Path Reference Line (shown at +5°)
8. Artificial Horizon Line (shown at +10° Pitch).
9. Fault Display Panel
10. Rate of Turn Scale
11. Roll Pointer (shown at +20° Roll).
12. Roll Indices
13. Radar Altitude
14. Roll Command Bar (shown in Null Position).
15. Rudder Command Display
16. ILS Window (shown 0.625° to left of center and 0.157 above the G/S Center.
17. Pitch Command Bar (shown in Null Position).
18. Flight Progress Display Panel
19. Flight Path Acceleration (shown at 277 Ft/Sec^2).
20. Turn Rate Pointer (shown at standard 2 Minute Right Turn).

FIGURE 35. EADI SYMBOLOGY

6.2.1.3 EADI (Figure 36)

The EADI is a cathode ray tube (CRT) display of the symbology generated by the SGU. In addition to providing the display information, the EADI unit provides the following warning annunciators:

a. Attitude (ATT) Warning.

b. Speed Error (SPD) Warning.

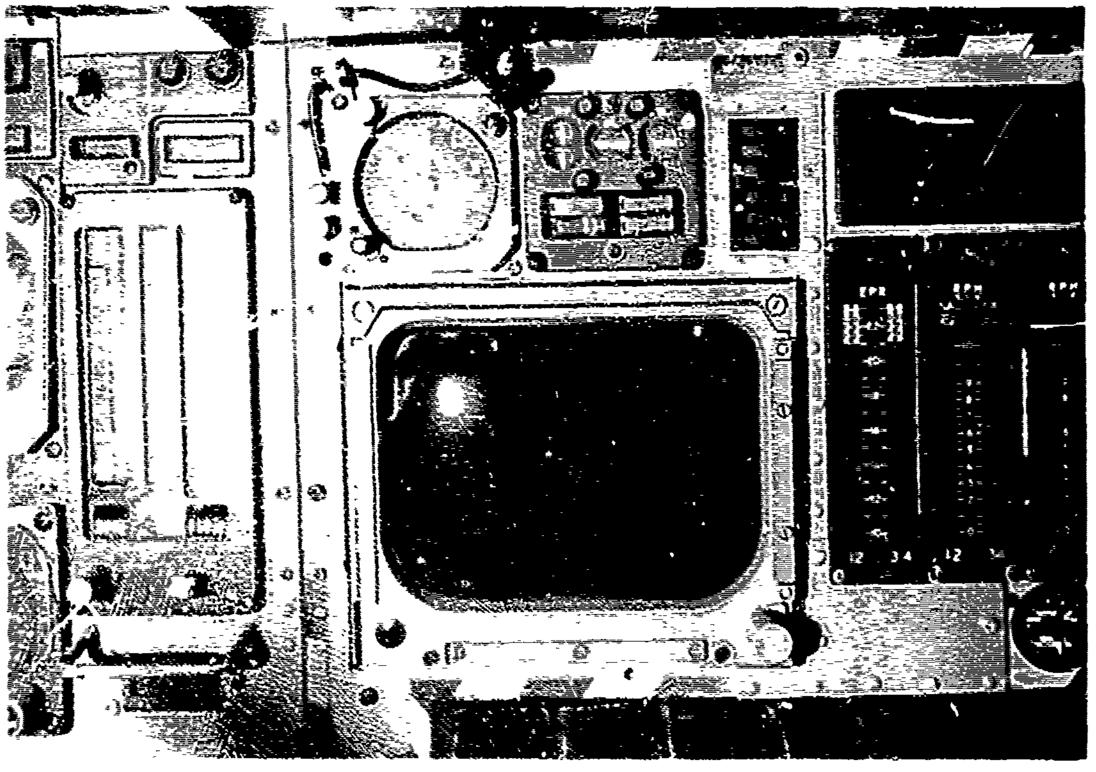
c. Radar Altitude (R-ALT) Warning.

d. Instrument Landing System (ILS) Warning.

e. Flight Director (F/D) Warning.

f. Flight Path Angle (FPA) Warning.

These annunciators are arranged left to right on the bezel beneath the screen as listed above. Whenever a discrete input indicates that the data is invalid, the associated failure warning annunciator is illuminated.



(Figure 36) Electronic Attitude Director Indicator

There are five (5) approach progress annunciators located on the right side of the bezel as follows:

a. Localizer (LOC) - Illuminates Amber or Green.

b. Glideslope (G/S) - Illuminates Amber or Green.

c. Decision Height (DH) - Illuminates Amber Only.

d. Flare (FLR) - Illuminates Amber or Green.

e. Go Around (G/A) - Illuminates Amber or Green.

The illumination of these annunciators is controlled externally of the EADI system. Amber indicates the mode is armed and green indicates the mode is energized.

A display brightness/contrast control and a test button are located on the front bezel.

6.2.1.4 EADI Control Unit (Figure 37)

The primary function of the panel mounted Control Unit is to provide mode selection, flight path reference setting, and radar altitude reference. The unit provides the following manual control inputs to the SGU:

a. Mode Selection - Attitude, Normal or TV.

b. Speed Error (On/Off).

c. Flight Director (On/Off).

d. Approach TV (On/Off).

e. Flight Path Reference (SET).

f. Radar Altitude Reference (SET).

g. Radar Altimeter Test.

The EADI receives the following inputs form the avionics:

a. The Aircraft Status which consists of the following:

(1) Pitch and Roll Attitude of the Aircraft.

(2) Rate of Turn.

(3) Glideslope and Localizer.

(4) Radar Altitude.

(5) Speed Error.

b. Control Commands as follows:

(1) Pitch and Roll.

(2) Decrab and Roll-out.

c. A TV input with the ELA SR170 modified for a 100 cycle field rate.

6.2.2 Display Symbology

6.2.2.1 Artificial Horizon, Auxiliary Pitch Lines.

The Artificial Horizon symbol is in the form of sky/ground shading across the full width of the display with a sharp boundary indicated by a single stroke-written line. This symbol rolls about the Aircraft Symbol in response to the roll inputs and is limited to within 0.20-inch of the edge of the display. The pitch sensitivity is 5.84-degrees per inch. Auxiliary Pitch Scale Lines are provided with major graduations at increments of 10-degrees to $\pm$90-degrees and minor graduations at 2-degree increments between -20 and +30-degrees. The major and minor graduations are 4.1-inches and 0.5-inches in length, respectively. The 10-degree lines are annotated with 2-digit numerics approximately 0.25-inch high and 0.15-inch wide. The thickness of the auxiliary pitch information is equivalent to a single stroke-written line.

The Pitch Lines are normally located on the vertical centerline of the screen. When either Pitch or Roll Steering Commands are displayed, the minor graduations are symmetrically positioned about the vertical centerline of the screen. The Auxiliary Pitch Lines revolve about the Reference Aircraft in response to the roll inputs.

6.2.2.2 Altitude

Altitude is presented as a 4-digit numeric read-out in the upper right-hand corner of the display. White numerals on a black background that are 0.307-in. high and 0.18-in. wide are used.

6.2.2.3 ILS Window

The ILS symbol is an open box 2.00-in. wide and 0.87-in. high. The line thickness is approximately 0.05-inch. Localizer deviation and glideslope deviation are used to position the ILS window. The zero position of the ILS window is centered about the Reference Aircraft symbol. The localizer deviation input will move the ILS left and right and the glideslope deviation signal will move the ILS up and down relative to the Aircraft Symbol.

6.2.2.4 Flight Director Steering Command

The Pitch and Roll Steering Commands appear as separate black and white bars which are generated by raster modulation. The size of each command bar is 0.75-inch in length and 0.024-inch thick. The Pitch Command Bar moves in a vertical direction while the Roll Command Bar moves horizontally.

6.2.2.5 Speed Error

Speed Error is displayed in the form of a bar whose length is proportional to the Speed Error Input. The bar is 0.19-inch in width and varies in length as a function of the input signal from 0.118-inch located symmetrically about the left wing of the Reference Aircraft. The symbol is generated by digital raster modulations and appears black.

6.2.2.6 Flight Path Reference

The Flight Path Reference symbol (stroke-written) is a broken line across the entire width of the display. The symbol is 0.01-inch thick and is made up of 0.4-inch wide segments, each separated by 0.4-in. The entire line is blanked from view when coincident with the Horizon Line.

This symbol is positioned by the pilot at any selected angle from -15 to +25-degrees. The manual control knob for this symbol is located on the Control Unit. The symbol remains parallel to the Artificial Horizon during the aircraft's attitude changes. It is adjusted from the Horizon Line at the same angular scale factor as pitch.

6.2.2.7 Roll Pointer and Scale

The Roll Pointer is stroke-written and rotates about the Reference Aircraft symbol in response to the roll input. The Roll Pointer is approximately 0.01-in thick and extends down 0.50-in from the top of the screen at 0-degree roll angle. The roll angle graduations are digitally generated and appear white. Major indices are provided at 0 and ±30-deg. while monor indices are provided at ±10, ±20, ±45-deg.

6.2.3 Display Characteristics

6.2.3.1 Visibility

The display symbology is clearly readable under all ambient lighting conditions ranging from night conditions up to and including 7,000 foot-lamberts.

6.2.3.2 Line Definition

With the manual Brightness/Contrast control adjusted for maximum display brightness, the display has a line width of 0.015-in (normally) measured at the 50 percent (3 dB) maximum brightness level.

6.2.3.3 Display Refresh Rate

The displayed symbology is refreshed at a 50-cycle frame rate.

6.2.3.4 Writing Speed

The writing speeds for the raster and stroke-written information are as follows:

a. Raster - 225,000-inches/second.

b. Stroke - 75,000-inches/second.

6.2.3.5 Positional Stability

The screen center position does not change by more than ±0.05-inch while the display image does not vary by more than ±3% of full scale deflection.

6.2.3.6 Accuracy

The deflection non-linearity is not greater than ±3% while the linearity correlation between the raster and stroke symbology is matched to ±1.5%.

6.2.3.7 Resolution

The horizontal and vertical resolution are as follows:

X Resolution - 81 lines per inch for Raster and Stroke.

Y Resolution - 81 lines per inch for Raster and 162 lines per inch for Stroke.

6.2.3.8 Contrast Ratio

Four levels of symbol brightness are provided.

6.2.3.9 Controls.

The indicator (EADI) unit contains the following controls:

a. BRIGHTNESS/CONTRAST - A control posentiometer, located on the lower right corner of the bezel, permits the manual setting of the display overall brightness/contrast ratio.

b. BRIGHTNESS TRIM - A trim control is provided on the rear plate of the EADI for maintenance adjustments.

6.2.3.10 PRESS-TO-TEST Switch

The PRESS-TO-TEST Switch is located on the lower left hand corner of the bezel and is protected by a rubber boot. When activated, the test pattern will be displayed.

6.2.4 Viewing Angle

The EADI is designed for optimum viewing from an angle of 15-degrees above the line normal to the centerline of the cathode ray tube.

6.2.5 Symbol Generator Electronics Rack

The Rack is 21-inches wide, 15-inches deep and 36-inches high. It is a completely enclosed unit with interface connection and all electronics are of the plug-in printed circuit board type.

The bottom section of the Rack contains the power supplies and deflection amplifiers. The middle section contains the 43 circuit cards which comprise all the symbol generation, video and deflection drive circuitry. The Upper section contains all the analog-to-digital and synchro-to-digital converters required to interface the SGU with the aircraft avionics inputs.

6.2.5.1 Construction

The SGU is of modular construction with a functional break down of replaceable circuit cards and modules.

6.2.5.2 Self-Test

The circuit design of the SGU is such that there is a self-test function that will provide both qualitative and quantitative testing of the internal functions. Up to 95% of all SGU malfunctions will be identified by this test.

6.2.5.3 Failure Warning

The self-test circuitry provides blanking of the appropriate symbol accompanied by the generation of the failure warning annunciator discrete signal when a failure is detected.

6.3 EADI ADAPTER DESCRIPTION (Figures 37, 38, 39, and 40)

The EADI Adapter incorporates the Barometric/Radar Altitude Rate In/Fade Out circuitry for the EADI/ALR system plus converting the standard AWLS Logic and Validities into the levels required for the EADI System.

6.4 ALR SYSTEM DESCRIPTION

6.4.1 General Description

The system referred to as the High Resolution, Phased Array Antenna System consists of a radar system (Approach and Landing Radar or ALR) operating in conjunction with a forward looking altitude sensor (Independent Altitude Determination System or IAD).

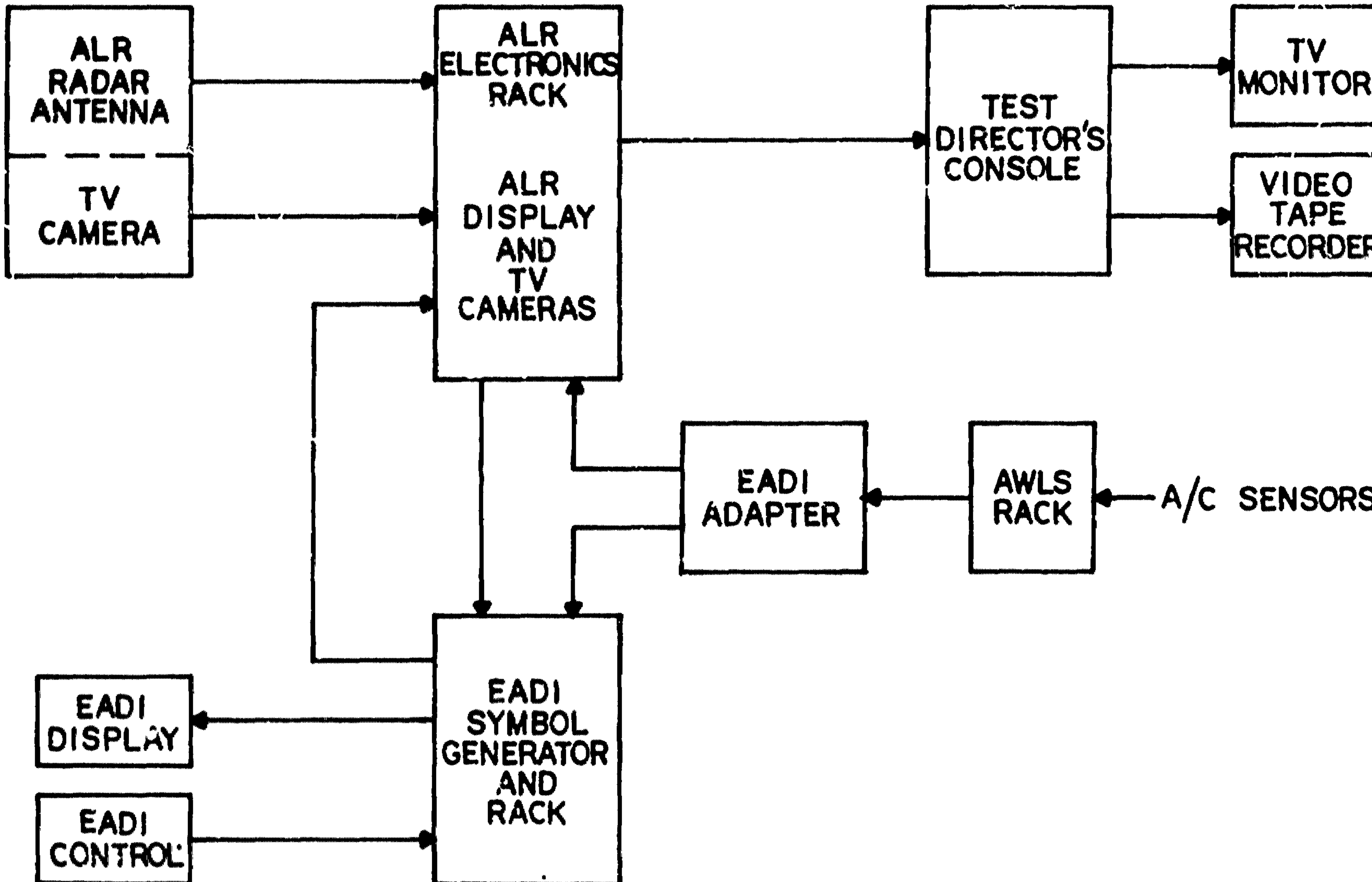


FIGURE 37.
ALR/EADI INTERFACE
BLOCK DIAGRAM

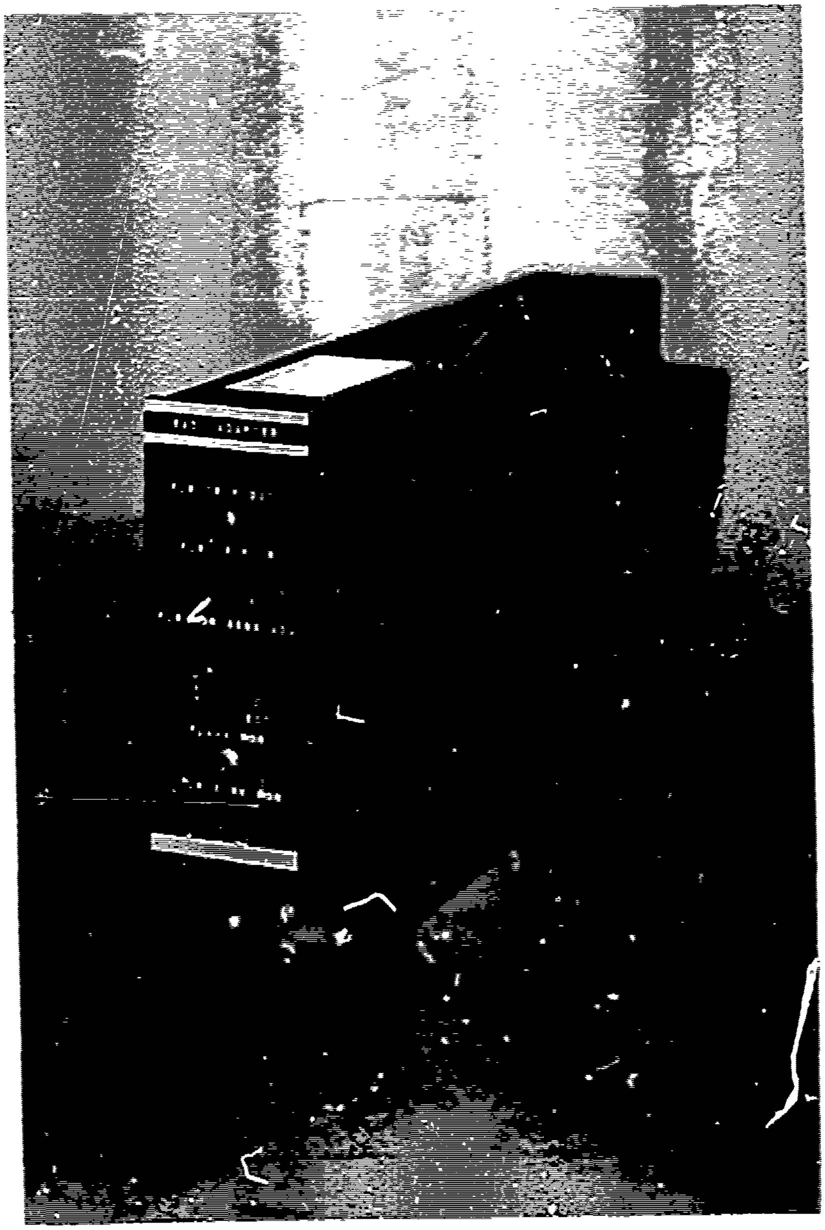


FIGURE 38. EADI ADAPTER

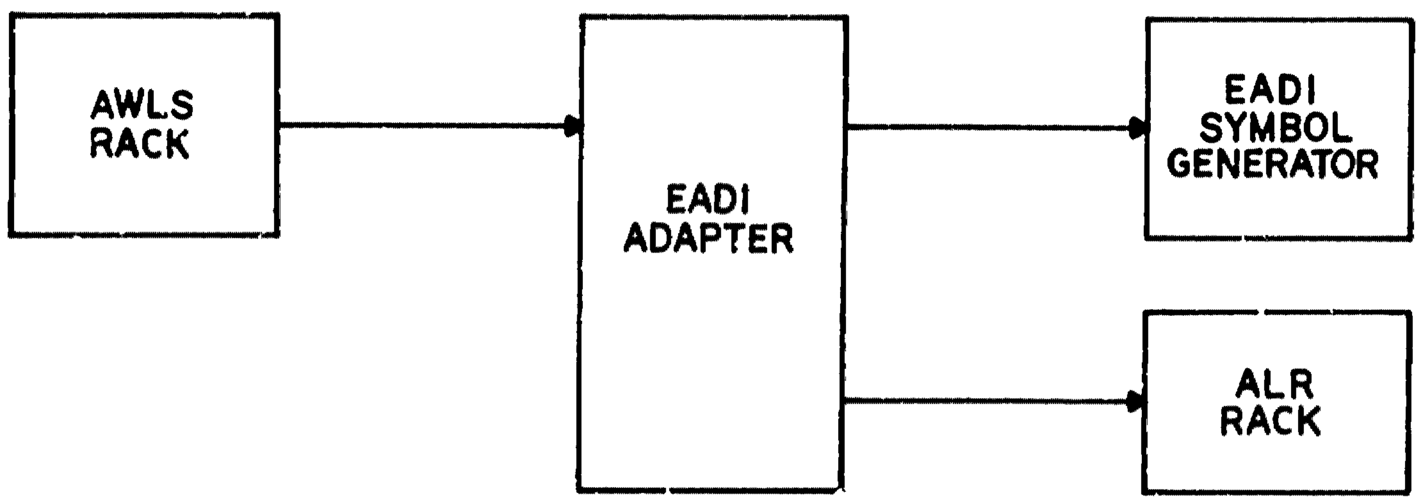


FIGURE 39. EADI ADAPTER BLOCK DIAGRAM

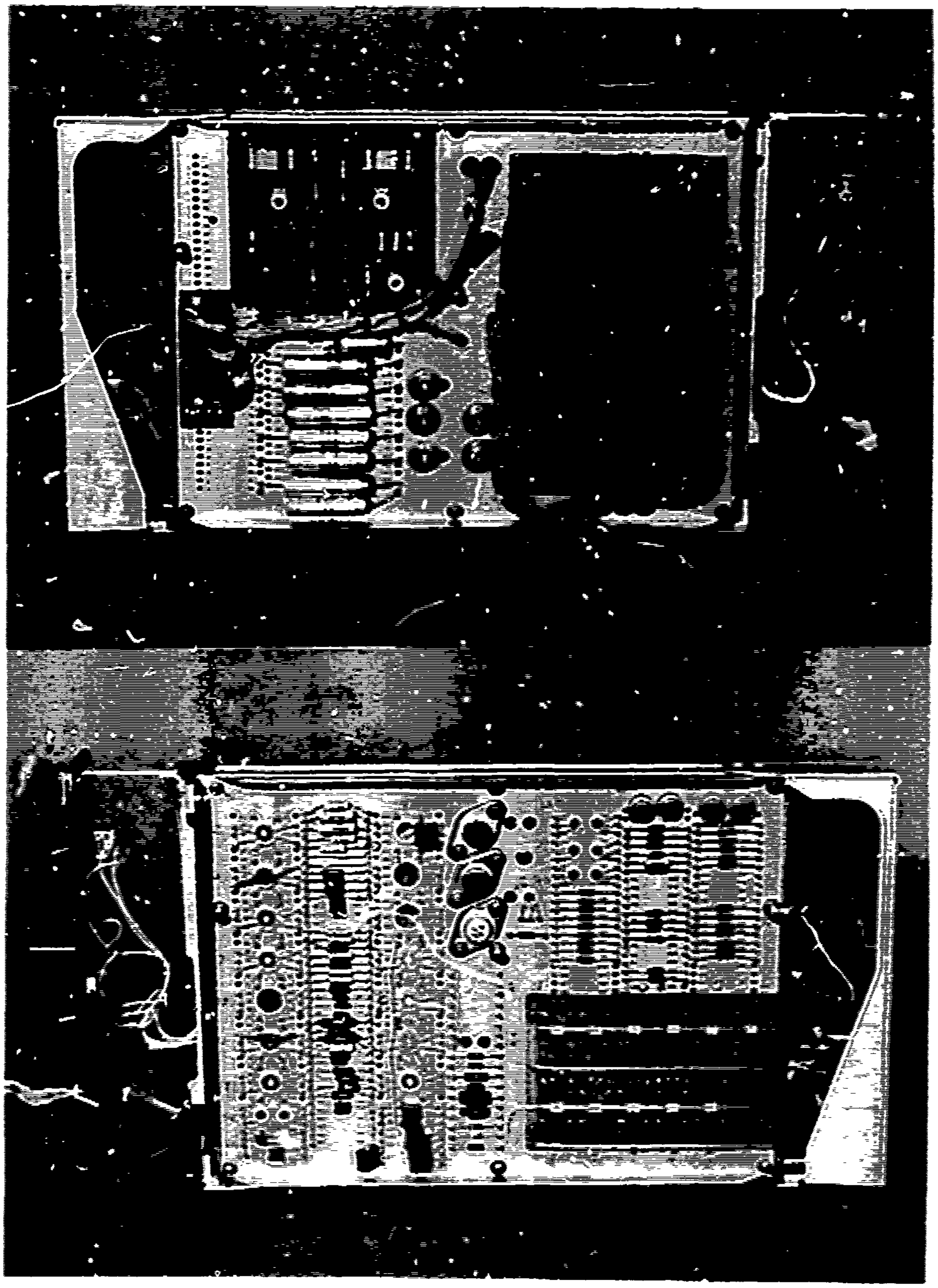


FIGURE 40. EADI ADAPTER (INTERNAL)

6.4.2 Approach and Landing Radar (ALR) (Figure 41)

The purpose of the ALR is to provide the pilots with an autonomous, visual confirmation of the runway location when landing during IFR conditions. The ALR is an airborne, high resolution, forward looking radar system capable of detecting airport lighting fixtures, runway obstructions, and the runway with surrounding terrain.

Visual Cues normally viewed through the windshield are presented in perspective on a CRT display. The aircraft stabilized display maintains 1:1 correspondence with the real world thus minimizing the need for interpretation.

The system can be broken down into 5 subassemblies: the Antenna, Transmitter/Receiver, Signal Processor, Display and Power Supply. The Antenna and the Transmitter/Receiver are physically located in an externally mounted unit. The remaining subassemblies are located in a standard 19-inch electronic rack. Auxiliary equipment consists of a sulfer hexaflouride supply and video recording equipment.

6.4.2.1 Antenna

The antenna is located on the underside of the aircraft's nose. During operation, it remains stationary (does not physically scan to achieve its $\pm$16.5-degree aximuth coverage). The Antenna consists of a 185-element, electro-mechanically scanned, phased arrry. The beam is positioned by a continously rotating drum with non-contacting cams which turns on an axis that is parallell to the array. The elements are fed by a dual series feed that provides sum and difference outputs required for signal processing.

6.4.2.2 Transmitter/Feceiver

The Litton Model L-4516 transmitter magnetron is tunable over a frequency range of 34.85 $\pm$0.15 GHz. A HY-11 Hydrogen Thyratron Switch is used in conjunction with a line type modulator enabling the transmitter to deliver a 50-ns, 80-Kw, peak pulse at a 10-KHz PRF.

Duplexing is accomplished with a circulator and latching ferrite switch combination.

The local oscillator is a solid-state, Gunn-Effect Oscillator with a voltage tuned AFC and has a range of 50-MHz.

Matched, linear sum and difference IF Amplifiers with 25-MHz bank widths are controlled by a common AGC circuit that responds to maximum signal in the sum or difference line.

6.4.2.3 Signal Processor

The Signal Processor's prime funcion is to distinguish between individual radar targets (point targets) and ground clutter without sacrificing the useful information contained in each. In doing so, it enhances range and azimuth resolution of point targets by about 30%.

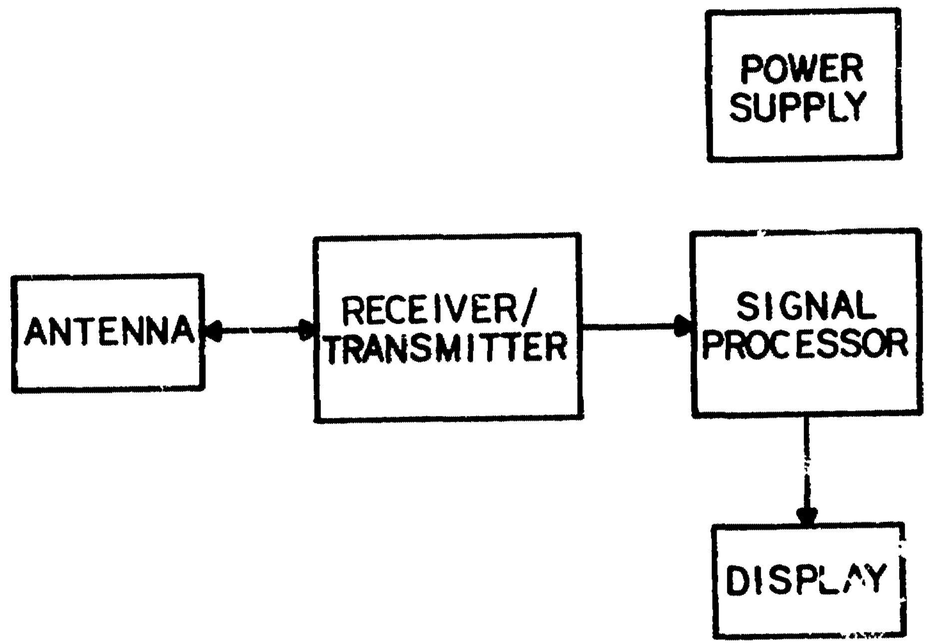


FIGURE 41. ALR SYSTEM BLOCK DIAGRAM

The basic approach is to divide the sum and difference signals into several channels in a way that yields a number of independent estimations of ground clutter. They are then re-combined into two lines called the point target and clutter channels. As a result of this process, the required signal-to-noise ratio for any given probability of detection is higher, range and azimuth resolution of point targets are enhanced, and the data provided by ground clutter is smoothed and preserved.

6.4.2.4 Perspective Display

The primary display mode is called the perspective mode. When it's used, radar returns form the terrain features are displayed to coincide with a pilot's cockpit view of the same features. This is true even as the aircraft pitches and rolls because the display is stabilized. The resultant image fidelity makes the system adaptable to any head-down or head-up display.

The ALR provides substitute visual approach information (i. e. : the basically qualitative data the pilot obtains via his windshield view). It consists of aim point, velocity, distance, sink rate, runway alignment, terrain features, relative size, and separation of ground objects, attitude and altitude.

Radar returns are input to the display via 2 separate lines (point target and clutter target).

The point target channel handles returns form objects having large radar cross sections. These are displayed at maximum intensity. The clutter channel is a saturable analog channel with its peak intensity maintained at a gray scale below the point target channel. Returns from each are superimposed to provide an overall clutter which is highlighted by point targets.

A maximum flexibility for flight evaluation is achieved by scan-converting the CRT radar display to TV format. This is done by using the output of a TV Camera directed at the face of the CRT display.

To provide an understanding of the nature of the display, the basic elements of the displayed information have been plotted by a computer for a variety of situations that the pilot will encounter during approach and landing.

6.4.2.5 Additional Display Modes

There are 2 additional display modes: a conventional point position indicator (PPI) and a "B scope" presentation. "B scope" depicts radar returns as azimuth vs. range. It is possible that the system's capability will be enhanced (in terms of lateral guidance) by using this mode.

6.4.3 Independent Altitude Determination System (IAD)

The RF section of the IAD is mounted in the Antenna housing. From this unit a signal corresponding to the measured change in phase is transmitted to the electronics for digital processing. The IAD is adaptable to any forward looking, narrow pulse radar. It can measure, within 4%, the altitude with respect to Touchdown, even if the aircraft is out beyond the Middle Marker.

When installed in the ALR, it will work with a KA Band Interferometer. Each antenna feeds a pre-comparator receiver that uses the ALR Local Oscillator and AGC. By means of appropriate frequency conversion and phase comparasion with delayed samples, the time rate of change of phase is measured and modified by system constants to provide an altitude output. The measured altitude is displayed numerically on the face of the EADI tube.

6.4.4 Application to AWLS

6.4.4.1 Crab Angle Information

The pilot's assessment of the crab angle is considered during the period in which the decrab procedure is performed at about 18-feet AGL. The runway edges converge at a point non-coincident with the boresight line. This easily descernable situation is directly comparable to the pilot's conventional windshield view with the visual cues having a directly proportional relationship.

6.4.4.2 Lateral Guidance

Lateral offset is a situation similar to crabbing in the respect that the runway is not centered in the pilot's field of view. During the approach phase, the difference is easily recognizable. The runway and its approach light pattern appear along a line perpendicular to the horizon in a crab situation, whereas in lateral offset the pattern is skewed.

At shorter ranges, when the plane passes the threshold, the difference is more subtle. The altitude is 10-feet and the lateral offset is 12-feet. The most notable clue is the position at which the runway edges intercept the screen.

6.4.4.3 Reduction of Tunnel Vision

The tunnel vision problem is the tendency of the pilot to concentrate on extremely small independent visual indicators to the exclusion of all surrounding information. This problem is associated with artificial displays and does not normally occur when the pilot looks out the window at the real world. The apparent difference is that the real world view visual cues are not concentrated at any single point. If this is so, it can be expected that pilot will tend to expand his field of view, thus suppressing tunnel vision tendencies as he views the EADI with a superimposed pictoral presentation.

6.4.4.4 Forward Looking Altitude Information

Forward looking altitude information is provided by the IAD. It is unique because it can read altitude with respect to the projected touchdown point unlike the radar and barometric altimeters.

Measurement is taken with respect to patches to the left and right of touchdown extending for approximately 500-feet in the direction of the runway. The altitude of the touchdown point is of prime importance to the pilot in low visibility conditions. Thus, it is proposed that the information be presented in numeric form in lieu of the standard altitude reading that appears on the EADI. This can be accomplished by making the IAD output compatible with altitude input of the EADI symbol generator.

6.4.4.5 Glideslope Reference

The ratio of measured altitude and range corresponds to the sine of the declination angle to the point of measurement.

A circuit is provided to permit the pilot to select his glideslope angle. The range measurements will converge to the projected touchdown point by successively comparing the sine of the angle to the point of measured altitude and range with the sine of the glideslope angle.

Coincident with the touchdown point, a glideslope reference marker will be displayed. It will consist of a horizontal line at the nominal glideslope angle below the horizon.

SECTION VII

ANALYSIS

7.1 GENERAL

Work was done to develop simulation models of the C-141A aircraft and its associated avionics systems. The simulation models that have been developed have been used as follows:

a. To analyze the effect of hardward changes, thereby minimizing the use of costly flight time for trial and error.

b. To have a before and after comparison of old and new subsystems.

c. To provide a combination of hardware and simulation for bench testing a complete system.

Because of this, the following has been incorporated into the design and testing of AWLS equipment:

a. Inclusion of a means to electrically "disturb" the aircraft during final track. This will aid in evaluating system performance and thus determine whether the system is operating normally during flight testing.

b. Both the lateral (initial and final) track and longitudinal final track configurations appear overly complex. Consideration is being given to replacing these three (3) control methods with less complex but equal performing methods.

c. A lateral bench test set-up is now operational. This set-up will be an invaluable aid after its accuracy has been validated by correlation with the flight tests. A more complete test set-up (hardware/simulation) is under construction for lateral and longitudinal testing of future hardware.

7.2 AIRCRAFT EQUATIONS OF MOTION

The C-141A equations of motion used in the simulation were taken from Technical Report, AFFDL-TR-74-52, Volume 1*, dated April, 1974. The flight conditions used herein were for light and heavy landing/approach conditions. Four (4) other sets of landing/approach stability derivatives (i. e.: forward and after c. g. for each of the weight configurations) were supplied by Captain Lawrence D. Roberts.

7.2.1 Lateral Equations of Motion

The lateral equations of motion used are as follows:

$$\dot{p} = L_B'B + L_p'p + L_r'r + L_{\delta_a}'\delta_a + L_{\delta_r}'\delta_r \qquad (7\text{-}1)$$

*Military Transport (C-141) Fly-By-Wire Program

$$\dot{r} = N_B{}'B + N_p{}'p + N_r{}'r + N_{\delta a}{}'\,\delta a + N_{\delta r}{}'\,\delta r \qquad (7\text{-}2)$$

$$\dot{v} = -U_o r + g\,\phi + Y_v v + Y_{\delta_r}\,\delta r \qquad (7\text{-}3)$$

Stability derivatives for equations 7-1 through 7-3 are shown in Table 3.

	FCI	FC2	FC3	FC4	FC5	FC6
$L_B{}'$	-.658			-.885		
$L_p{}'$	-.750			-1.106		
$L_r{}'$	.790			1.044		
$L_{\delta a}{}'$	-.231			-.318		
$L_{\delta r}{}'$	.039			.058		
$N_B{}'$	.201			.230		
$N_p{}'$	-.130			-.151		
$N_r{}'$	-.727	-.691	-.714	-.856	-.813	-.834
$N_{\delta a}{}'$	-.0071			-.0058		
$N_{\delta r}{}'$	-.172			-.192		
Y_v	-.055			-.060		
Y_{δ_r}	3.07			3.07		

TABLE 3. LATERAL STABILITY DERIVATIVES

7.2.2 Longitudinal Equations of Motion

The longitudinal equations of motion used are as follows:

$$\ddot{\theta} = M_{\delta e}\,\delta e + M_{\dot{w}}\dot{w} + M_w\bar{w} + M_q\theta \qquad (7\text{-}4)$$

$$\dot{w} = Z_{\delta e}\,\delta e + Z_w\bar{w} + U_o\theta + Z_u U \qquad (7\text{-}5)$$

$$\dot{u} = X_w\bar{w} - g\,\theta + X_u U \qquad (7\text{-}6)$$

Where: $\bar{w} = \int w\,dt + w_{ge}$ (7-7)

And, $W_{ge} = (Uo)\ (^{\Delta}_{\alpha})\ (1 - h_r/75)$ (7-8)

And where $h_r = 1$ above 75 feet.

Stability derivatives for equations 7-4 through 7-6 are shown in Table 4.

	FC1	FC2	FC3	FC4	FC5	FC6
M_{δ_e}	-.743			-.657		
$M_{\dot{w}}$	-.00136	-.00097	-.00102	-.0011	-.00102	-.00107
M_w	-.0064	-.0029	-.0051	-.0063	-.0029	-.0051
M_q	-.673	-.608	-.645	-.646	-.583	-.619
Z_{δ_e}	5.36			-5.41		
Z_w	-.558			-.617		
U_o	201.0			184.0		
Z_u	-.321			-.349		
X_w	+.083			+.089		
g	32.2			32.2		
X_u	-.044			-.048		

TABLE 4. LONGITUDINAL STABILITY DERIVATIVES

7.3 ILS EQUATIONS

The Instrument Landing System (ILS) simulation assumed smooth and straight guidance beams in the lateral and longitudinal planes.

7.3.1 Localizer Beam Equations

The localizer beam equations are as follows:

$$\dot{Y}_a = U_o \text{ SIN } (\Psi + B) \quad U_o (\Psi + B) = U_o \Psi + V \qquad (7\text{-}9)$$

$$\dot{Y}_{cg} = \dot{Y}_a + \dot{Y}_w \qquad (7\text{-}10)$$

$$\dot{Y}_{an} = \dot{Y}_{cg} - l_{an} r/57.3 \qquad (7\text{-}11)$$

$$Y = \int \dot{Y}_{an} dt \qquad (7\text{-}12)$$

In equation 7-12, the lateral deviation (Y) is expressed in terms of feet from beam center.

The range equation is given by:

$$R = R_o - \int U_o \cos(\Psi + B)\, dt \qquad (7\text{-}13)$$

7.3.2 Glideslope Beam Equations

1. glideslope beam equations are as follows:

$$h_B = \tan(\eta/57.3)\,(X)$$
$$\approx (\eta/57.3)\left(\int_{t_o}^{t} \dot{X}\, dt\right)$$
$$\approx (\eta/57.3)\left(h_B(o) - \int_o^t \dot{X}\, dt\right) \qquad (7\text{-}14)$$

7.4 LONGITUDINAL AVIONICS

Figure 42 is a block diagram of the longitudinal AWLS which shows the three avionics boxes that contain the required glideslope tracking and flare circuitry.

The Flare Computer uses normal acceleration ($\ddot{h}$) and radar altitude (h) to generate scaled vertical rate and altitude signals. These signals, plus a fixed bias, are used for open loop (no beam guidance) landing of the aircraft from a radar altitude of 45 feet. The fixed bias acts as an altitude command which will cause the aircraft to seek an altitude approximately 14 feet below the level of the runway. The flare is therefore, an approximate exponential (in altitude) path of the aircraft wherein the steady state level of the exponential is 14-feet lower than the runway. The derived vertical rate term is used for damping of the radar altitude loop.

The Elevator Computer uses the pitch command from the Coupler, along with pitch rate and pitch attitude to control the pitch attitude of the aircraft. The Elevator Computer also contains the elevator servo controls and stabilizer trim drivers. The outputs of the Elevator computer are command signals to the elevator servo and the stabilizer trim motor.

The pitch portion of the coupler (for AWLS) processes normal acceleration, glideslope error, radar altitude and the Flare Computer output to generate a pitch command signal. The radar altitude signal is used to change the gain of the glideslope signal as shown by Figure 42. The glideslope signal gain change will compensate for the glideslope beam gain changing as a function of range. The gain change will maintain a relatively constant total loop gain (results in an approximately constant product of beam gain and de-sensitizer gain) as the range changes. Table 5 lists the product of beam gain and de-sensitizer gain changes with respect to radar altitude.

The lagged normal acceleration is used as a beam damping term. The long term wash out removes the steady state accelerometer output due to the aircraft pitch trim attitude.

RADAR ALTITUDE (FT)	BEAM GAIN*	DESENSITIZER	BEAM GAIN X DESENS. GAIN
1500	.4	1	.4
1200	.5	1	.5
1000	.6	1	.6
800	.75	.90	.68
600	1.0	.74	.74
400	1.5	.57	.86
200	3.0	.41	1.23
120	5.0	.29	1.45
100	6.0	.27	1.62
75	8.0	.25	2.00
60	10.0	.125	1.25

$$*\text{Beam Gain} = \frac{214 \text{ X Beam Angle (Deg.)}}{\text{Radar Altitude}}$$

TABLE 5. GLIDESLOPE BEAM AND DESENSITIZER GAINS

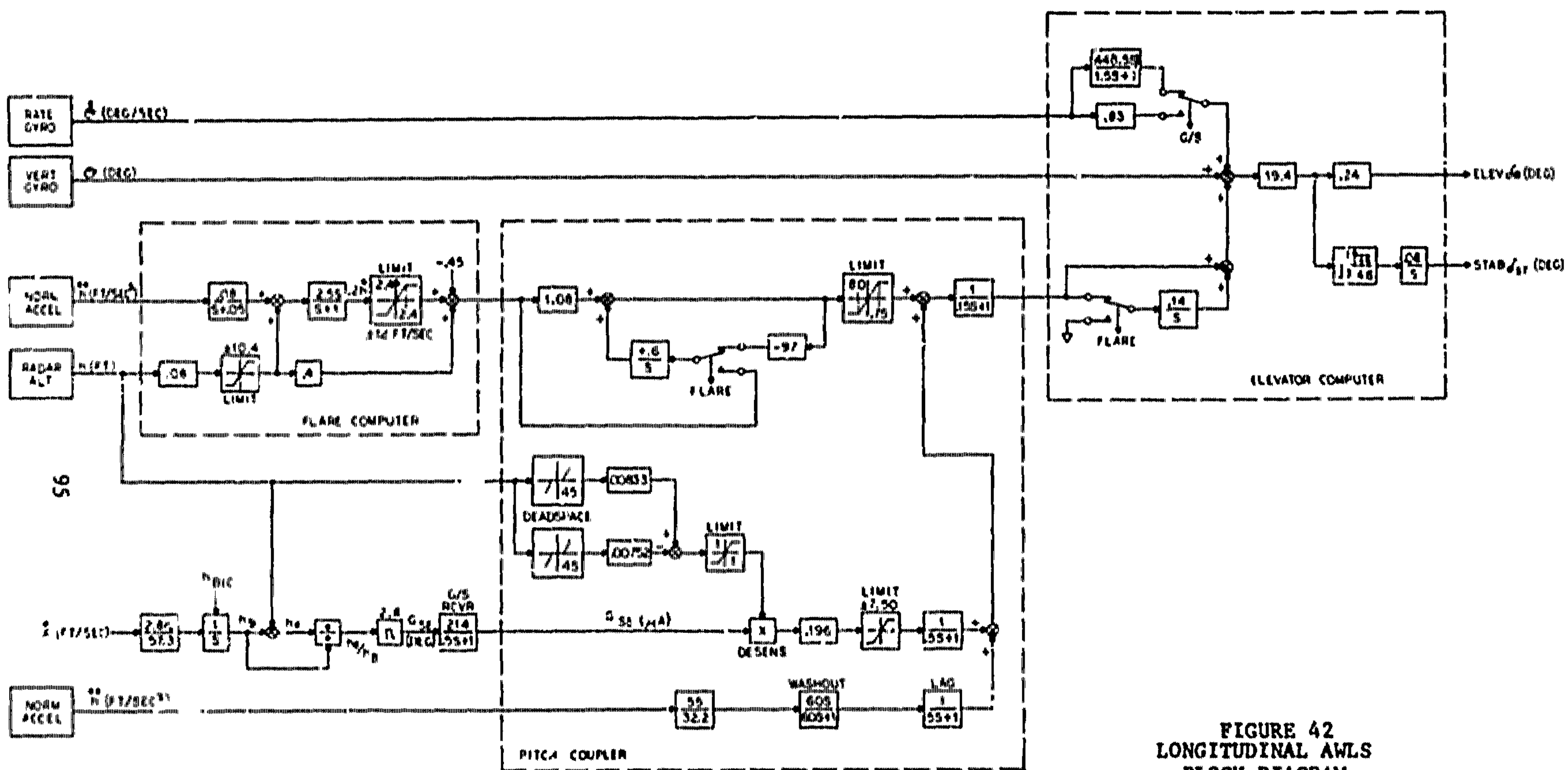


FIGURE 42
LONGITUDINAL AWLS
BLOCK DIAGRAM

The Flare Computer output is synchronized until the aircraft reaches the flare altitude of 45-feet. The Flare Computer signal is then processed by generating straight gain and integral terms to produce a pitch command. The sum of the straight gain and integral terms will be limited to prevent excessive pitch down (.75 degree) and pitch up (8.0 degrees) commands.

7.5 LATERAL AVIONICS

Figure 43 shows the avionics required for lateral axis localizer tracking and decrab.

The Coupler uses localizer error, radar altitude, roll attitude, and preselect heading to generate initial and final track commands that are fed to both the aileron and yaw computers.

Localizer capture is performed using straight gain localizer error and preselect heading signals. At localizer track, (LOC < 25 μA) the straight gain preselect heading signal is replaced with a washed out preselect heading signal. This signal is used as the localizer damping signal until Glideslope capture (G/S). At G/S, the washed out preselect heading signal is attenuated and the localizer damping is then performed by lagged bank angle. At Approach Arm (AA), the localizer error signal is integrated to minimize the effects of wind shear during final track. The localizer is also de-sensitized to maintain a constant total loop gain despite geometry gain changes due to the changing range with respect to the transmitter.

The Yaw Computer contains a high pass filter for yaw rate damping. Also injected into the Yaw Computer are roll and roll rate signals for improving turn coordination, a coupler command signal for localizer tracking and decrab signals for decrabbing the aircraft.

The Aileron Computer contains the circuitry required for a roll attitude control loop. The processed feedback signals are roll attitude and roll rate. The roll command signal (from the Coupler) is processed to contain attitude command limiting and attitude rate command limiting.

The CAT III Adapter supplies signals to the Aileron and Yaw Computers below a radar altitude of 18 feet (decrab). The yaw signal is used to align the nose of the aircraft parallel to the heading of the runway centerline.

7.6 DIGITAL SIMULATIONS

MIMIC and CSMP digital simulations have been generated to simulate AWLS operation from capture to touchdown. The simulations are split into lateral and longitudinal channels that contain all of the significant signal shaping, beam signal limiting and de-sensitization, and mode switching in an AWLS approach.

7.6.1 Longitudinal Simulation

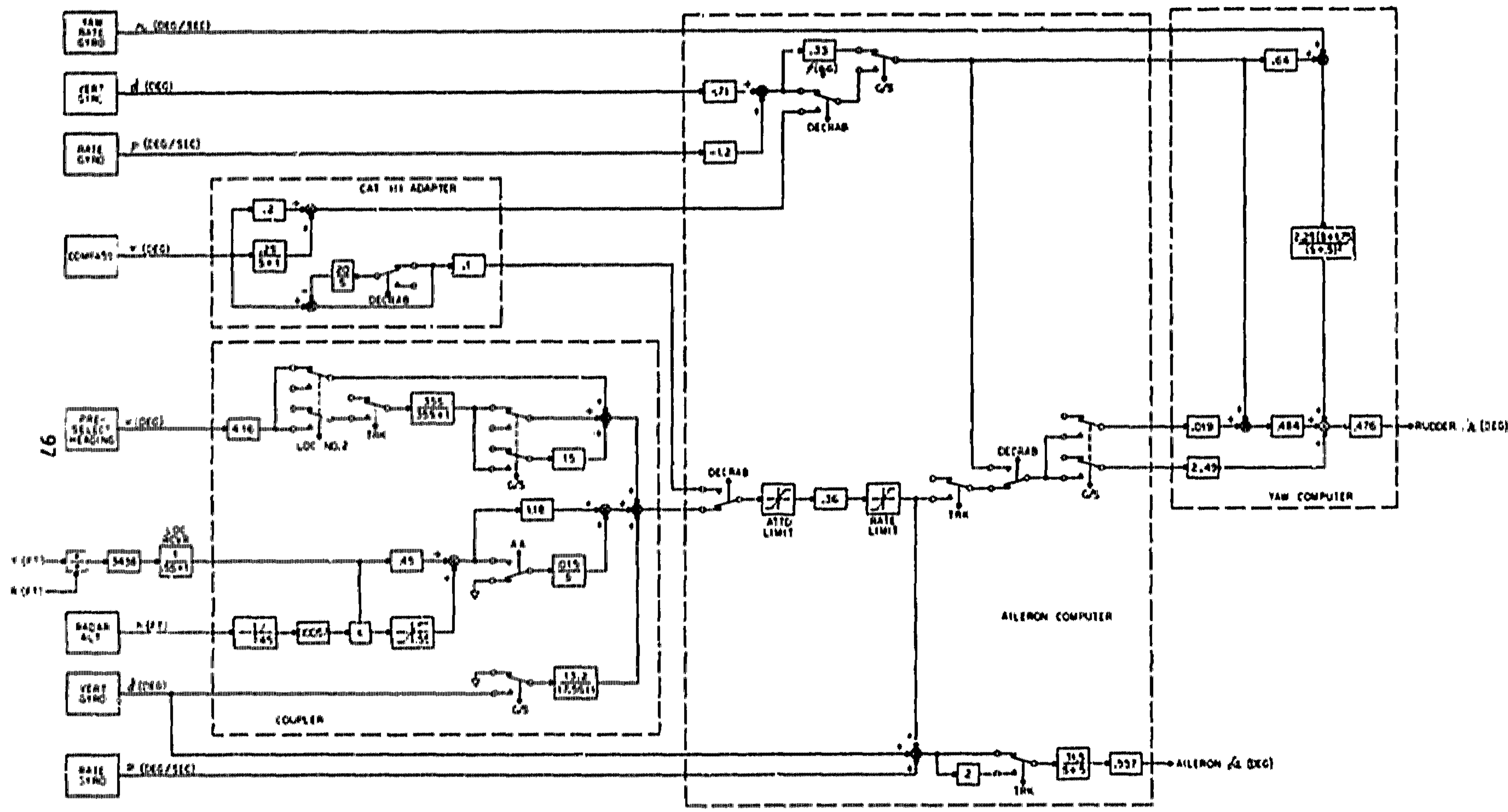


FIGURE 43. LATERAL AWLS BLOCK DIAGRAM

POT.	VARIABLE	F.C.I.	F.C.4
R50	$-Yu$	.055	.060
R51	$Uo/200$	1.000	.920
R53	$Y\delta_r/4$	.767	.767
R61	$50N_B'/Uo$	.05	.07
R62	$-N_r'$	.73	.86
R63	$-10N_{\delta a}'$	.07	.06
R64	$-2.5N_{\delta r}'$	.43	.48
R65	$-N_p'$	.130	.151
R73	$-.5L_p'$	.375	.555
R74	$-.50L_B'/Uo$	.16	.24
R75	$.5L_r'$	.395	.520
R76	$-L_{\delta a}'$	.23	.32
R77	$2.5L_{\delta r}'$	.10	.15

TABLE 6. LATERAL POTENTIOMETER SETTINGS

Figure 44 is a block diagram of the longitudinal simulation. This includes test inputs (C20 and C21) for disturbing the system. An Altitude Hold mode (CRUISE) is included to simulate capturing the glide-slope beam from Altitude Hold. Simulation of capture from a vertical rate mode can also be done by summing an altitude ramp command with the sensed altitude signal.

7.6.2 Lateral Simulation

Figure 45 is a block diagram of the lateral simulation. This includes test inputs (C20, C21, C22, and C23) for disturbing the system. A Heading Mode (CRUISE) is included to simulate capturing the beam from Heading Hold.

7.7 ANALOG/HARDWARE SIMULATION

The Analog Computer/hardware bench set-up was constructed to test system (AWLS) operation in the laboratory. The Analog is a special purpose computer that simulates the aircraft, sensors and beam equations. The hardware used consists of the Coupler and a test unit for powering up the Coupler. The Coupler inputs and outputs are provide at a patch panel. Reference flight data was not available (for making performance comparisons) since the present AWLS can not be easily disturbed without tripping the system monitors and thereby cause a disconnect.

7.7.1 Lateral Test Set-Up

Figure 46 is a diagram of the lateral Analog/hardware test set-up. The Yaw and Aileron Computer signal processing (without switching functions) was approximated for the final track (after AA) configuration and simulated on the Analog Computer. There was good correlation between the Analog/hardware and the digital simulations. Qualitative comparison with the flight test data did not appear to disclose gross discrepancies. Table 6 lists the potentiometer settings for the two approach mode dynamics that were used.

Figure 47 shows the simulation of the control inputs and the lateral 3 degrees-of-freedom aerodynamics. Figure 48 illustrates the simulation of the lateral sensors and generation of the localizer beam signal.

7.7.2 Longitudinal Test Set-Up

Figure 49 is a flow diagram of the longitudinal Analog/hardware test set-up. This test set-up was not implemented due to a shortage of time. Table 7 lists the potentiometer settings for two sets of approach mode dynamics. Figure 50 illustrates the simulation of the control input and longitudinal 3 degrees-of-freedom aerodynamics. Figure 51 shows the simulation of the longitudinal sensors and generation of glideslope beam signal.

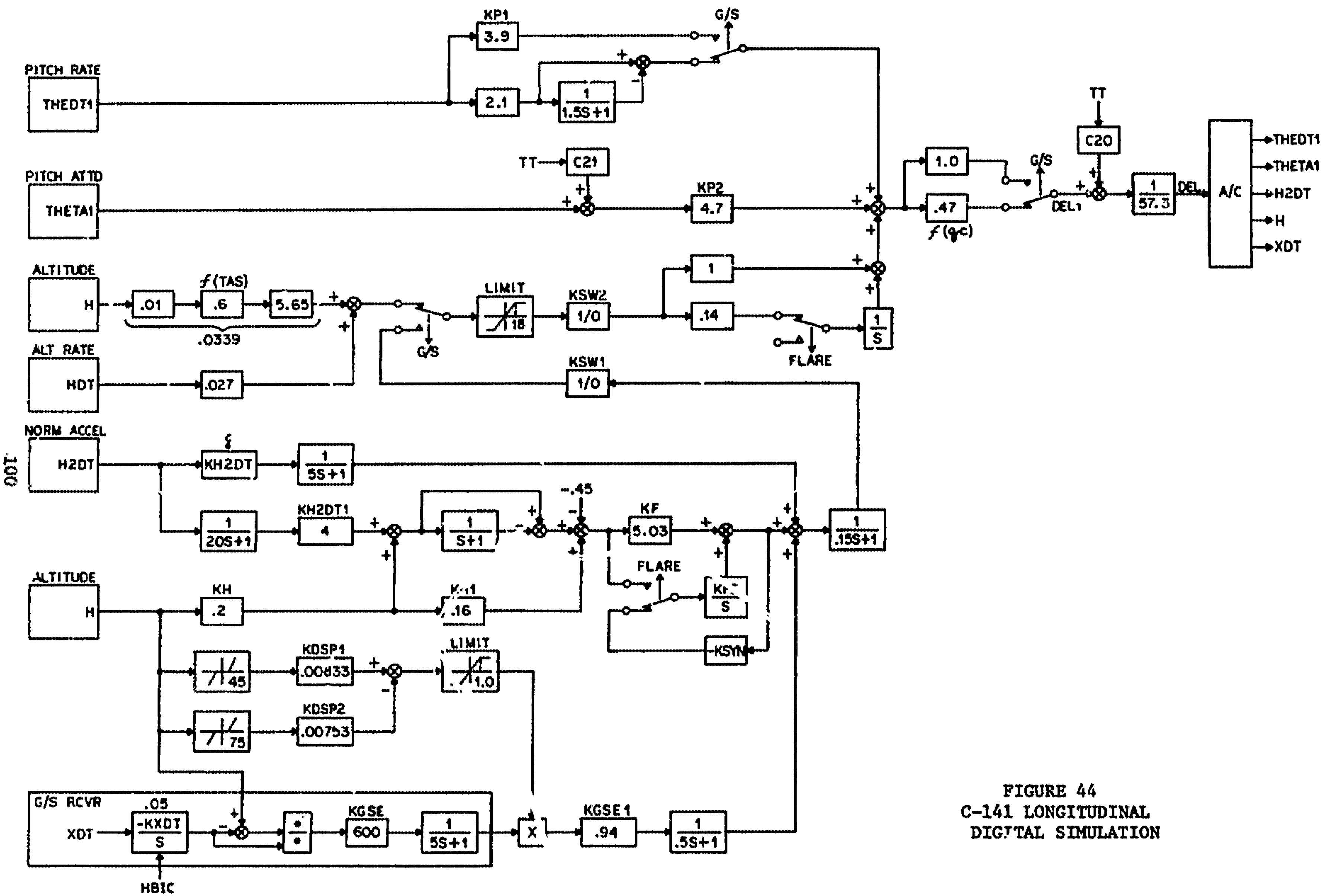


FIGURE 44
C-141 LONGITUDINAL
DIGITAL SIMULATION

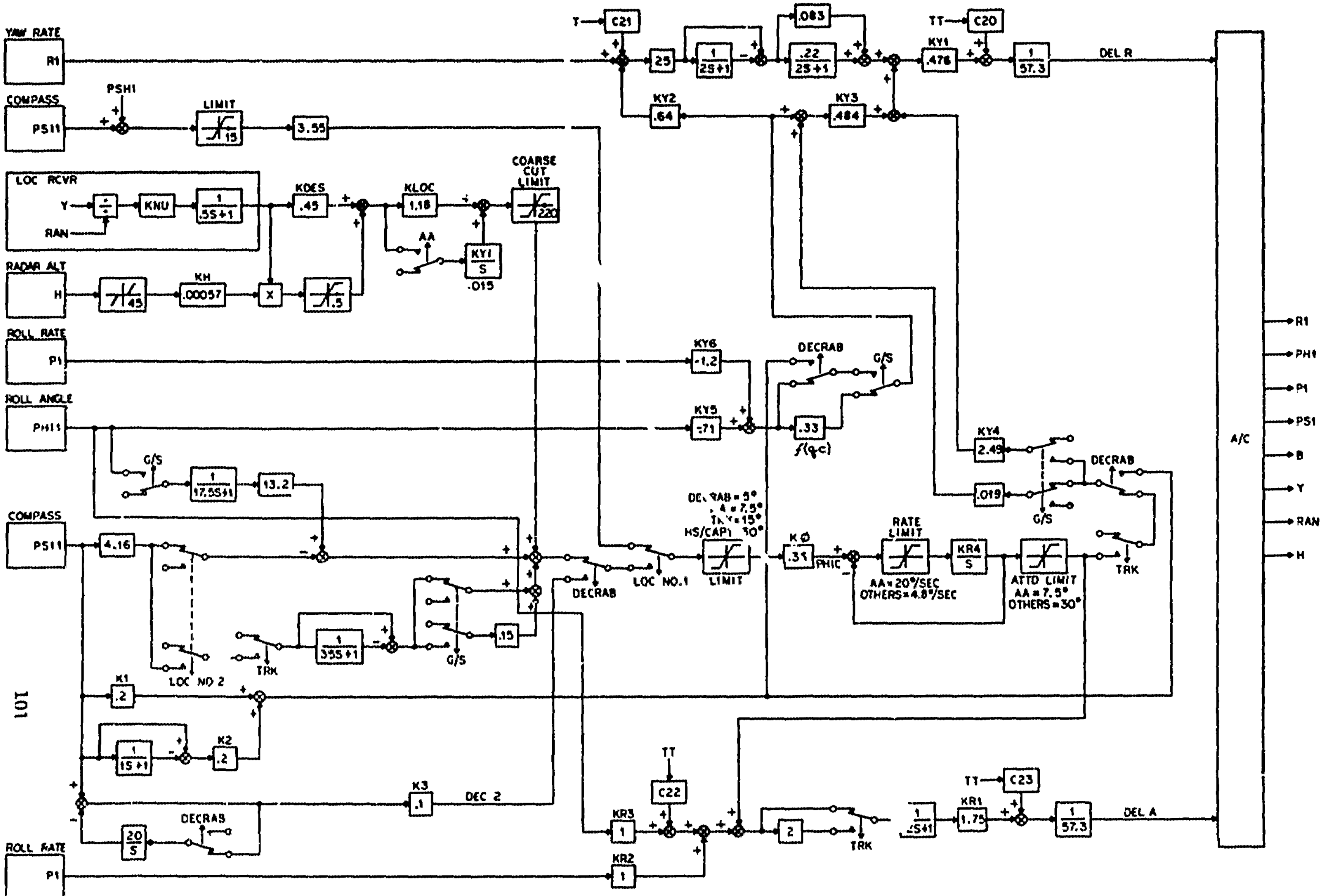


FIGURE 45. C-141 LATERAL DIGITAL SIMULATION

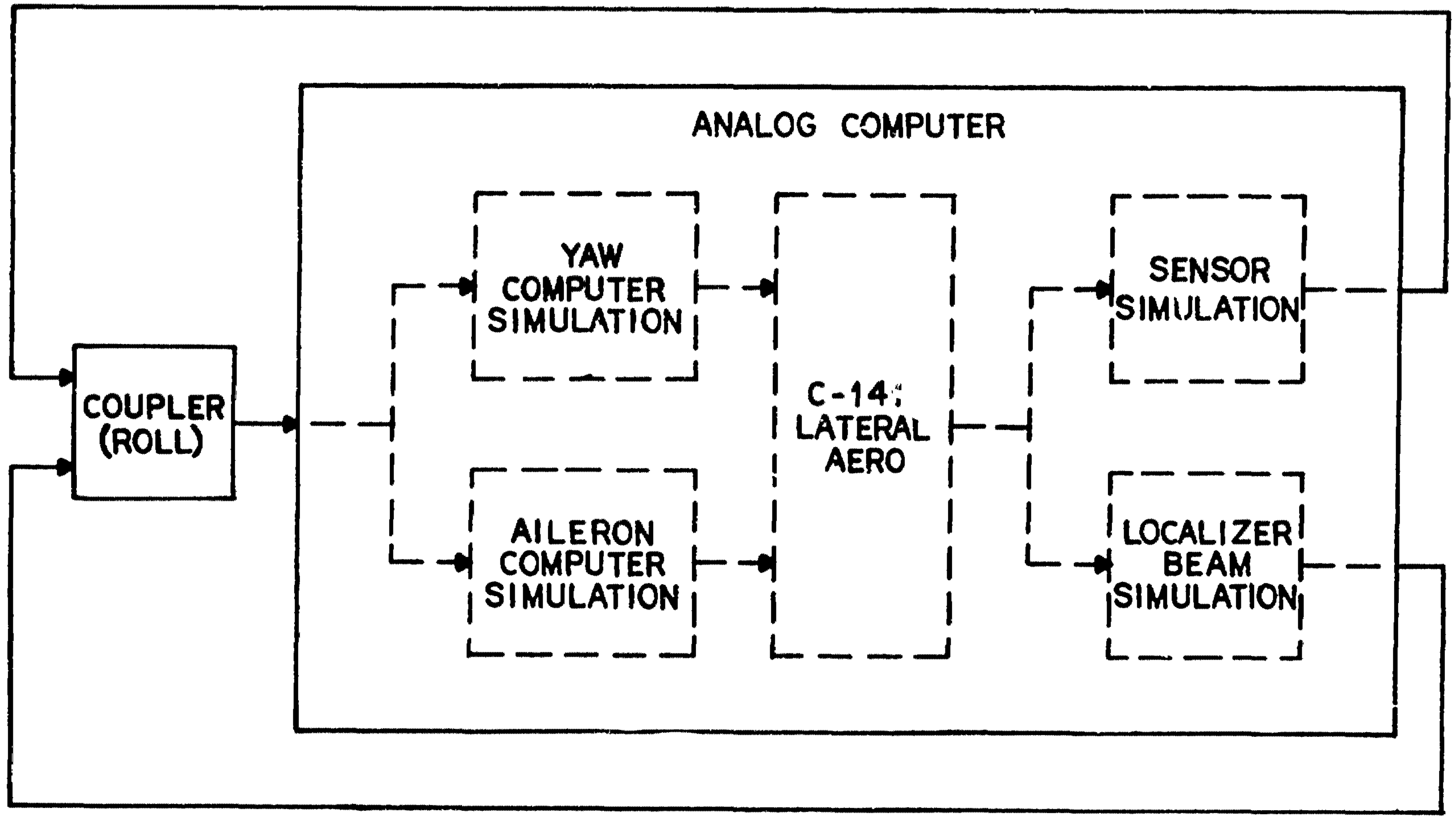


FIGURE 46. LATERAL ANALOG/HARDWARE FLOW DIAGRAM

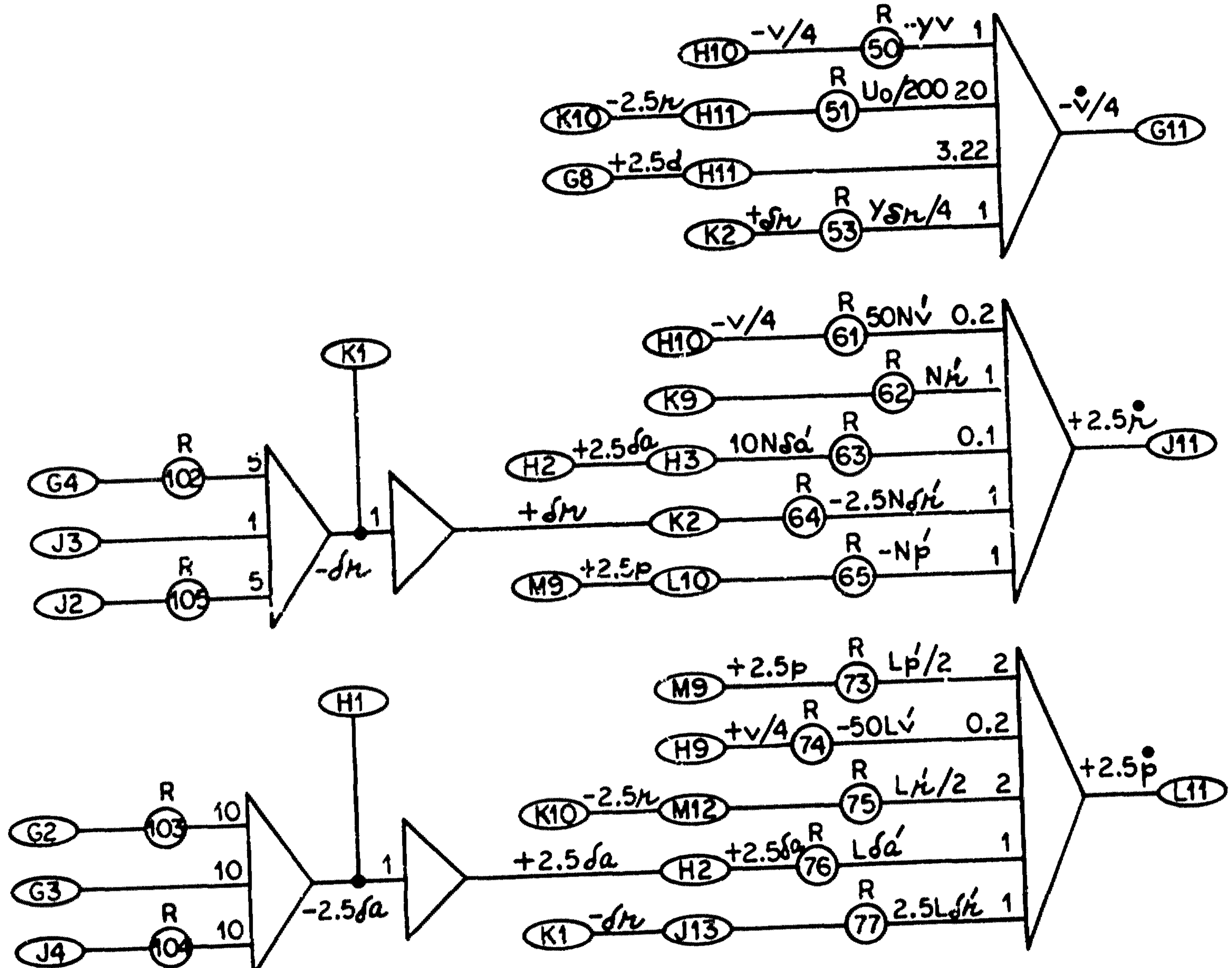


FIGURE 47. ANALOG C-141 LATERAL AERODYNAMICS

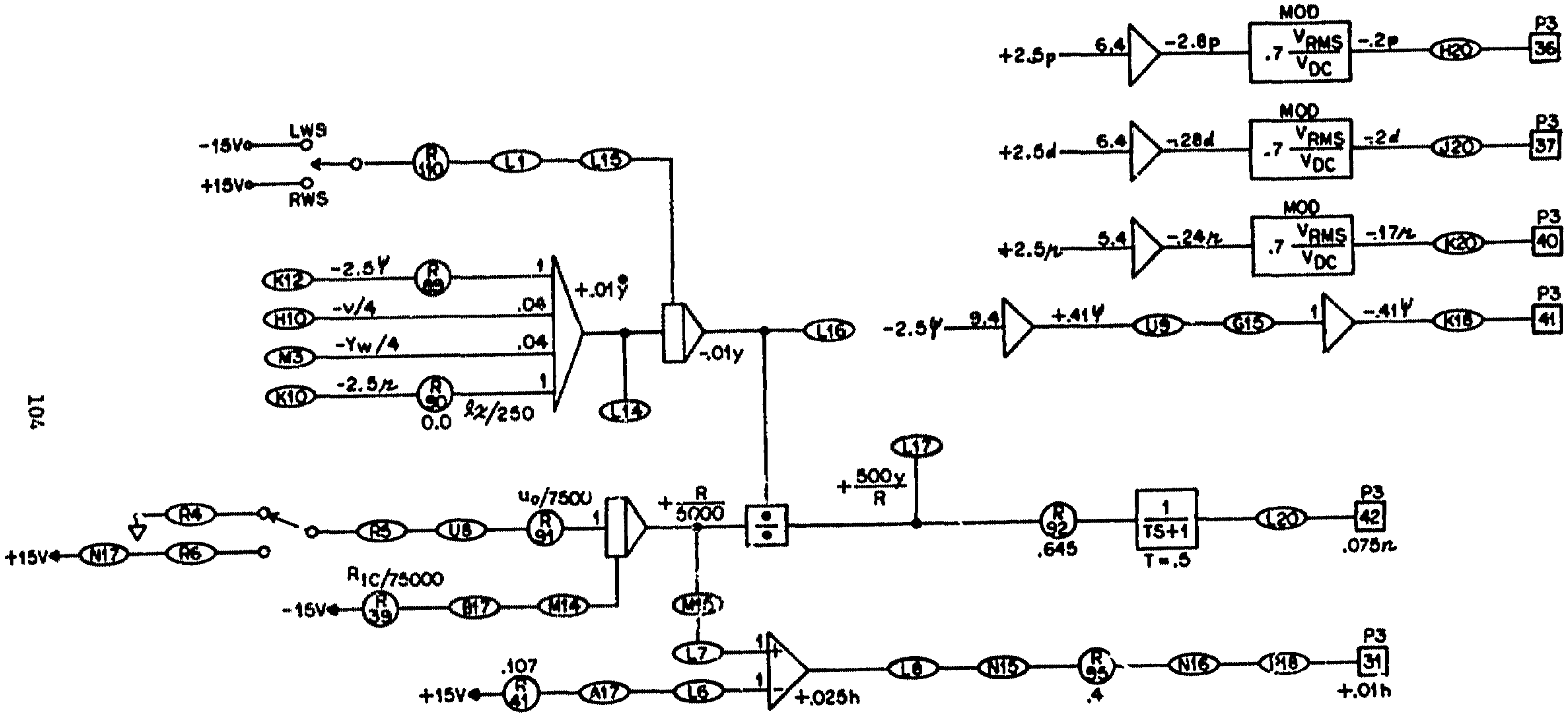


FIGURE 48. ANALOG C-141 LATERAL SENSORS AND LOCALIZER BEAM

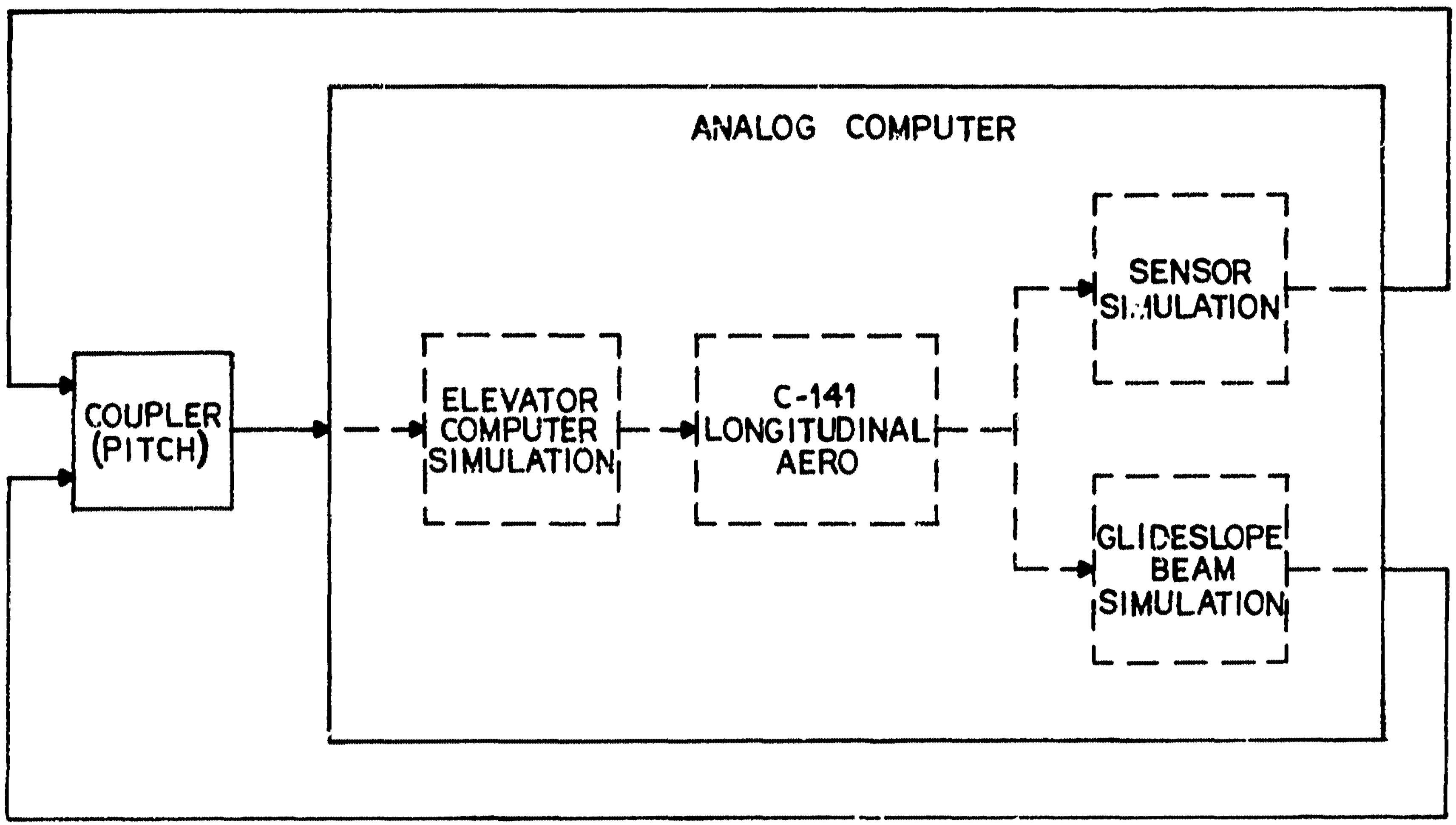


FIGURE 49. LONGITUDINAL ANALOG/HARDWARE FLOW DIAGRAM

POT.	VARIABLE	F.C.1.	F.C.4
R3	$X_w/2$	.042	.045
R4	$-X_u$	.044	.048
R13	$-Z_w/10$	.056	.062
R14	$U_o/400$	.500	.460
R15	$-2Z_u$	.134	.135
R25	$-M_q/10$	.067	.065
R26	$-20M_w$	.120	.012
R28	$-20M_{\dot{w}}$	.02	.002
R29	$-M_{\delta e}/2$	.372	.329
R40	$U_o/200$	1.00	.920

TABLE 7. LONGITUDINAL POTENTIOMETER SETTINGS

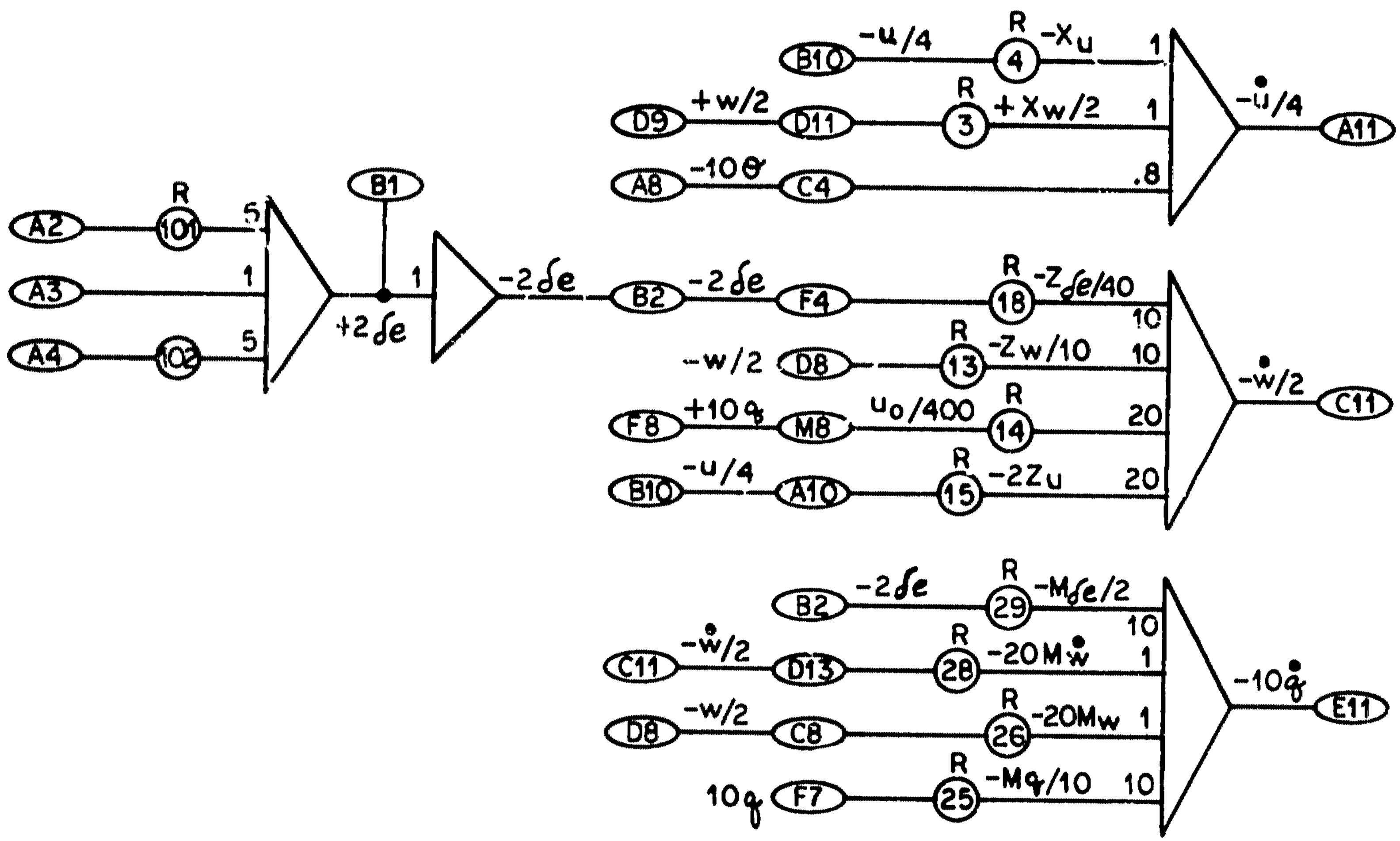


FIGURE 50. ANALOG C-141 LONGITUDINAL AERODYNAMICS

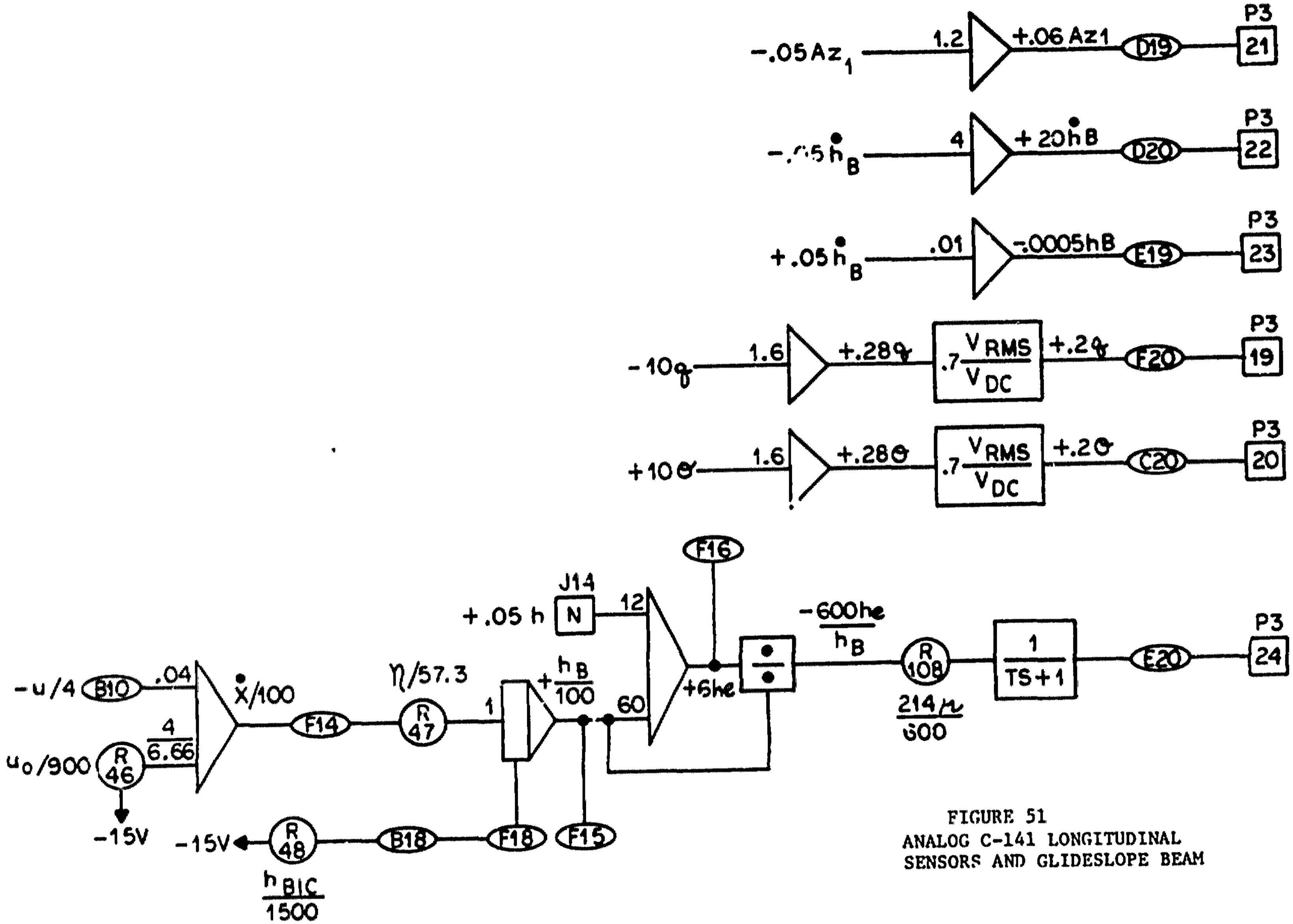


FIGURE 51
ANALOG C-141 LONGITUDINAL
SENSORS AND GLIDESLOPE BEAM

SIGNAL SWITCHING - ROLL, YAW

CWS On	1.3lb			
CWS Off	1.15lb			
LOC Engage	175 μA			
VT Engage	75 μA + 17.5°			
Sta. Entry	65 μA			
Sta. Exit	65 μA + 14 sec			

Doppler Engage	124	μA
PSC Washout-LOC	75	μA + 17.5°PSC
PSC Washout-VT	12	μA + 17.5°PSC
PSC Washout-Dop	124	μA + 17.5°PSC
Roll Down		$\phi \leqslant 6°$
RCWS No Force will hold		if $\phi \geqslant 3°$

SIGNAL SWITCHING PITCH

GS Engage Below Beam	< 8μA
GS Engage Above Beam	> 50μA
Trim Cut off	7.61lb
CWS ON	2.51lb
CWS OFF	1.71lb

Flare Engage	≈45 ft
Throttle Retard	≈30 ft
Land Arm	≈100 ft

ROLL/YAW

Limits

H. S. or Coup	30° ϕ
Roll Limit LOC	7.5°ϕ
ϕRoll Limit Sideslip	5.0° ϕ
Other	36° ϕ
Loc	22°ψ
VOR/TAC	32°ψ
VOR/TAC/TRK	12°ψ
Doppler	30°ψ
Comm. Mod. Rate	
Pre-engage	22° ϕ/sec
ψRoll engage	4.8°ϕ/sec
Loc Roll Rate	22° ϕ/sec
RCWS	22°ϕ/sec
δ_R from Coup	4.5°δ_R
XFD in GS Des	1.2°δ_R

CWS Deadspot ~3 lb

Roll to Yaw XFD Com. Soft τ = 9 sec

Limits

Up Attitude	6° θ
Glideslope	7.5°θ
Flare	8.0°θ Nose Up
Flare	0.75°θ Nose Down
Mach Beeper	0.045 Mach

Flare

Synch τ = 0.034

Command TD Rate $\dot{h}_{TD}$ = -2.25 ft/sec

Time Constant τ_F = 6.43

Misc. Lags

β 0.48 sec + 0.11 sec

Flare 0.11 sec

COMPARATORS

COUPLER

C-14 LOC RCVR 0.2°η

C-13 Des 2-1 0.2°ηNon-Des

C-9 3.3°ϕ

AILERON

C-2 Surface 2.8°ϕ

C-3 CWS 13.8lb

YAW DAMPER

C-1 XFD 4°δ

Yaw Rate 1.8°ψsec (Y. R. Filter)
Surface 1.8°δ_R

COUPLER

C-18 G/S RCVR 0.1°β

C-16 Des 3-2

C-15 Des 2-1 0.1°β

C-17 Des 3-1 Non-Des

C-19 1°θ

ELEVATOR

C-4 Surface 1.27°θ

C-5 CWS 17.2lb

FLARE

C-20 Sink Rate 4 ft/sec

CAT III Adapter
Decrab/Rollout

L/R Runway

$\pm$20% Power Supply Error
$\pm$10% RCD Displacement Error
$\pm$10% BSB Displacement Error
$\pm$15 A

EADI Adapter
XTrack Rate
Flare 1 & 2
Flare Error Detector

+4'/Sec
$\pm$1/2°θcommand
$\pm$2 1/2°θcommand
-1°

	PITCH AUTO	G/S MAN 1	G/S MAN 2	FLARE	ROLL AUTO	LOC AUTO AND MAN	THRESHOLD
C-1 Roll XFD	-	-	-	-	x	-	4.0 deg rudder
C-2 Ail Sig Ch	-	-	-	-	x	-	2.8 deg attitude
C-3 RCWS	-	-	-	-	x	-	13.8 lb
C-4 Elev Sig Ch	x	-	-	-	-	-	1.3 deg attitude
C-5 PCWS	x	-	-	-	-	-	17.2 lb
C-9 Roll Coup Sig	-	-	-	-	x	-	3.3 deg attitude
*C-10 Radar Alt	-	-	-	x	-	-	100 feet
C-13 LOC Des	-	-	-	-	x	-	0.2 deg beam
C-14 LOC RCVR	-	-	-	-	x	x	0.2 deg beam
C-15, C-17 G/S Des 1	-	x	-	-	-	-	0.1 deg beam
C-16, C-15 G/S Des 2	-	-	x	-	-	-	0.1 deg beam
C-16, C-17 G/S Des 3	x	-	-	-	-	-	0.1 deg beam
C-18 G/S RCVR	x	x	x	By-passed	-	-	0.1 deg beam
C-19 Pitch Coup Sig	x	-	-	x	-	-	1.0 deg attitude
*C-20 Flare	-	-	-	x	-	-	4 feet/second

*Faults in the Flare System are announced when detected, and result in autopilot pitch disconnect and Flight Director pitch warning at 100 feet.

PITCH

Aircraft Linkage	δ_e/δServo	=0.24 deg/deg LO Q = 0.21 deg/deg HI Q
Glideslope	δ_e/θ	=4.52 deg/deg LO Q
$\overline{\text{Glideslope}}$	δ_e/θ	=3.04 deg/deg LO Q = 0.64 deg/deg HI Q
Glideslope	$\delta_e\theta$	=3.75 deg/deg/sec LO Q
$\overline{\text{Glideslope}}$	$\delta_e\theta$	=1.34 deg/deg/sec LO Q = 0.28 deg/deg/sec HI Q

$$\frac{S}{S + 0.67}$$

Flaps Up	δStab/θ	=1.56 deg/deg LO Q = 0.37 deg/deg HI Q
CWS-Anticipation	δ_e/lb	=0.38 deg/lb LO Q = 0.033 deg/lb HI Q
CWS-Anticipation	θ /lb	=3.48 deg/lb $\frac{0.135}{S + 0.135}$
CWS-$\overline{\text{Glideslope}}$	θ/lb	=0.15 deg/sec/lb
ALT Hold	$\theta/_h$	=0.057 deg/ft + 0.0015 deg/sec/ft
ALT Hold (Rate)	$\theta/_{\dot{h}}$	=0.027 deg/ft/sec
VER NAV	$\theta/_h$	=0.024 deg/ft + 0.087 deg/sec/ft
VN TRK	$\theta/_h$	=0.024 deg/ft + 0.002 deg/sec/ft
VN ALT Hold	$\theta/_h$	=0.0578 deg/ft + 0.002 deg/sec/ft
VN AH Capture	$\theta/_h$	=0.024 deg/ft + 0.087 deg/sec/ft
Glideslope	$\theta/_\beta$	=42 deg/deg + 2.7 deg/sec/deg Integ. in Elev. Comp.

DESENSITIZER GAIN PROGRAM

		*Des 2 AP, FD Des 1 FD
f (t)	t = o	1.0
	t = 120	0.45
$f(h_{RA})$	1000 ft	1.0
	45 ft	0.45

Gain = 0.45 at t = , h_{RA} = 0

*FD Des not used on AWLS

		Des 3 AP	Des 2, 1 FD
f(t)	t = o	1.0	1.0
	t = 120	.25	.18
$f(h_{RA})$	1000 ft	1.0	1.0
	75 ft	125	.18
	3 sec		
	EZ Off		
	45 ft	0	0

Gain at t = 120

Same at t =

Gain = 0 at h_{RA} = o

Aircraft Linkage - Roll $\frac{\delta_A}{\delta_{Servo}}$ = 0.436 deg/deg

Loc/Coup $\frac{\delta_A}{\phi}$ = 3.26 deg/deg

$\overline{\text{Loc/Coup}}$ $\frac{\delta_A}{\phi}$ = 1.63 deg/deg

Loc/Coup $\frac{\delta_A}{\phi}$ = 3.64 deg/deg

$\overline{\text{Loc/Coup}}$ $\frac{\delta_A}{\dot{\phi}}$ = 1.82 deg/deg/sec

Aircraft Linkage - Yaw $\frac{\delta_R}{\psi}$ = 0.476 deg/deg

Yaw Damper $\frac{\delta_R}{\dot{\psi}}$ = 1.04 deg/deg/sec $\frac{S\,(S + 1.75)}{(S + 0.5)^2}$

*Roll XFD-L/C GS Des $\frac{\delta_R}{\dot{\psi}}$ = 0.165 deg/deg $\frac{\delta_R}{\dot{\phi}}$ = 0.28 deg/deg/sec

$\frac{\delta_R}{\phi_c}$ = 0.62 deg/deg

*Roll XFD-L/C $\overline{\text{GS Des}}$ $\frac{\delta_R}{\phi}$ = 0.165 deg/deg $\frac{\delta_R}{\dot{\phi}}$ = 0.23 deg/deg/sec

$\frac{\delta_R}{\phi_c}$ = 0.043 deg/deg

CWS - Anticipation $\frac{\delta_A}{lb}$ = 0.26 deg/lb

CWS - Loc/Coup $\frac{\phi}{lb}$ = 20 deg/lb

CWS - $\overline{\text{Loc/Coup}}$ $\frac{\dot{\phi}}{lb}$ = 0.88 deg/sec/lb

Roll and Yaw Symbol Symbol	Nomenclature	Interface Source	Sensitivity	Comments
-	Turn Cont.	Control Panel	$5.4V(1-\cos(0.643_T))$, $_T$ = Degrees of Controller	
-	Roll Wheel Force	Wheel Force Sensor	48.3mV/lb.	
ψ	Heading	Compass	360mV/deg.	3 wire source
ψ	Pre-set course	HSI	360mV/deg.	
ψ	Pre-select Heading	HSI	360mV/deg.	
ψ	Yaw Rate	Rate Gyro	200mV/deg./sec.	
δ_A	Aileron Surface	Airframe	168mV/deg.	As read at Servo F. U. 100%
δ_R	Rudder	Airframe	430mV/deg.	As read at Servo F. U. 100%
ϕ	Roll Attitude	Vertical Gyro	200mV/deg.	
ϕ	Roll Rate	Rate Gyro	200mV/deg/sec.	
η	Nav Beam	Nav Revrs	75µA/deg. Loc 15µA/deg/VOR 50µA/mile/Doppler, 500µA/mile Doppler Air Drop-All into a 1000Ω load	

PITCH

Mach Hold	θ/M	$= 1.9$ deg/0.0.M + 0.34 deg/sec/0.01M
Flare	$\theta/\dot{h}_{\varepsilon}$	$= 0.85$ deg/ft/sec + 0.25 deg/sec/ft/sec sec
Accelerometer	θ/a_n	$= 65 \text{ deg/g} \dfrac{52.5 \quad S}{52.5 \; S + 1} \; \dfrac{1}{5S + 1}$
Controller	θ/Pitch Wheel	$= 0.039$ deg/deg
Flaps	$\theta \; \delta_{Flap}$	$= 0.27$ deg/deg
Up Attitude	θ/ϕ	$= \dfrac{(1 - \cos\phi)}{11.5 \sin\theta} \; \dfrac{6 \times 0.5}{0.85}$

Couplerθ_c = 200mV/°θ

ROLL - YAW

Heading Hold	ϕ/ψ	=3.5 deg/deg
PSH	ϕ/ψ	=3.83 deg/deg LO Q
PSH	ϕ/ψ	= .5 deg/deg HI Q
PSC LOC $\overline{AA}$	ϕ/ψ	=1.4 $\frac{35S}{35S + 1}$ deg/deg
PSC LOC AA	ϕ/ψ	=0.21
PSC $\overline{LOC}$ (VOR TRK)	ϕ/ψ	=4.23 $\frac{90S}{90S + 1}$ deg/deg LO Q
		= .55 $\frac{90S}{9S + 1}$ deg/deg HI Q
*Roll Lag AA	ϕ_c/ϕ	=4.85 deg/deg $\frac{1}{17.5S + 1}$
**Controller	ϕ/Cont	=10 deg/V
PSC VOR CAPT	$\phi\ /\psi$	=4.23 deg/deg LO Q
		= .55 deg/deg HI Q
LOC CAPTURE	ϕ/η	=32 deg/deg
LOC TRACK	ϕ/η	=32 deg/deg + 0.38 deg/sec/deg
VOR/TAC	ϕ/η	=51 deg/deg LO Q
		=6.5 deg/deg HI Q
VOR/TAC/TRK	ϕ/η	=16 deg/deg LO Q
		= 2 deg/deg HI Q $\frac{1}{17.5S + 1}$
STA PASS	ϕ/η	=2.3 deg/deg
Doppler or Air Drop	ϕ/η	=52 deg/mile

* = Coupler Output = 100mV/deg Roll Command

SECTION VIII

CONCLUSION

The modified AWLS System installed in the test aircraft NC141A, #61-2775 did demonstrate the capability to provide the required automatic system operation and manual automatic system operation and manual backup displays to successfully operate in Category III weather conditions.

This above system capability was established as a result of the Class II Modifications accomplished to the basic C141A AWLS. (The modifications are covered generally in paragraph 1.3, Background Information.)

Specifically, the following functions performed in the Category III Adapter were considered essential to the mission:

a. The Left/Right runway display coupled with the Excessive Lateral Rate Indication served to insure a specific lateral footprint and a lateral drift rate within known limits.

b. The forward slip maneuver insured the aircraft's alignment to the runway prior to touchdown with assurance that any change in lateral drift rates (as a result of the decrab maneuver) were minimized by a compensating wing down maneuver.

c. The vertical excess rate indicator insured the flare maneuver was being performed within reasonable limits and that the major redundant sensors were operating within close tolerances.

d. The automatic rollout circuits provided continuous lateral control. The Left/Right runway continued to display the aircraft's lateral situation on the runway with respect to the localizer centerline. Procedures called for the pilot to provide manual assistance (as required) to the autopilot. Thus, he was "in the loop" at all times.

e. Backup rollout guidance circuits were available in the event the primary guidance signal (Localizer) was lost during rollout. Though never required during actual weather missions, numerous simulations of such an occurrence were accomplished during test missions with considerable success in maintaining the aircraft's lateral position on the runway.

Not considered essential to Category III operation but highly desired by the pilots during test missions was the Runway Distance Remaining Indicator. This instrument proved very accurate and was very beneficial in "Stop-and-Go" approaches.

Last, it is the intent of this section to disseminate those functions which were considered essential to the success of this program. No recommendation is implied as to specific requirements for Category III landing systems.

APPENDIX A

AFCS GAINS, GRADIENTS AND RATIOS
SYSTEM GRADIENTS

Pitch Symbol	Nomenclature	Interface Source	Sensitivity	Comments
θ	Pitch Attitude	Vertical	200mV/deg.	
θ	Pitch Rate	Rate Gyro	200mV/deg/sec.	
β	Glide Slope Dev.	Glide Slope Rec.	215μA/deg.	Into a 1000Ω load
h	Altitude Rate	CADC	250mV/1000ft/min.	
h	Altitude	CADC	10mV/ft	Same for Ver Nav Source
M	Mach	CADC	IV/0.01M	
MB	Mach Beeper	Control Panel	.3V/sec.	
h_{RA}	Radar Altitude	Radar Altitude	0.2V+10mV/ft.	
h_e	Flare Command	Flare Computer	200mV/ft/sec.	
a_n	Acceleration	Accelerometer	8V/g	
-	Pitch Cont. Comm.	Control Panel	12.5mV/deg. wheel	
-	Pitch Wheel Force	Wheel Force Sensor	29mV/lb	
δFLAP	Elev. Surface	Airframe	325mV/deg.	As read at Servo F. U.100%
δFLAP	Flap Surface	Airframe	200mV/deg.	As read at Elev. Comp. Input
δ FLAP	Stabilizer	Airframe	1.5V/deg.	As read at Elev. Comp. Input
-	Output to Flight Director	Coupler	215mV/deg.	Into a 500 Ω load
-	ADI Input	Coupler	75mV/DOT	

APPENDIX B - FLIGHT DIRECTOR GAINS & RATIOS

Constants:

BSB = 2.5V/inch

PSB = 2.5V/inch

Output rate limiter = $\dot{\phi}c$ = 300mV/sec

= .125"BSB/sec

Heading Select Mode:

BSB/ψPSH = 50mV/° ψPSH = .02"BSB/°ψ PSH

BSB/°ϕ = 30mV/°ϕ = .012" BSB/°ψ PSH

ϕ/ψPSH ϕLimit = $\pm 30°$

VOR or TACAN (capture)

$$°\eta/\text{BSB} = 1°\eta/500\text{mV}\ \frac{1}{10S+1} = 15\text{mV/ degree Beam} = .2\text{"BSB}$$

$$°\eta/\text{BSB} = 1°\ s\ \eta/147\text{mV}\ \frac{1}{10S+1} + \frac{52S}{52S+1} + 1 = .06\text{" BSB}$$

°ϕ /BSB = 1°ϕ/30mV = .012"BSB

°ψ PSC/BSB = 1° ψ/40mV = .016"BSB

°ϕ/°η = 16.3°ϕ/degη

°ψPSC/°η = 12.5°ψ /deg η

°ϕ /°ψ PSC = 1.3°ϕ /°ψ

Course Cut Limit =45°ψ PSC

Bank Limit =30°ϕ

ILS (Glide Slope)

$\beta = 215\text{mV/degree}$

$\beta/PSB = 1°\beta/4V\ PSB = 2.5''PSB$

$An/PSB = 1GAn/14V\ [\frac{1}{20S+1} + \frac{20S}{20S+1}]\ PSB = 5.6''PSB$

$\theta/PSB = 1°\theta/88mV\ [\frac{20S}{20S+1}]\ PSB = .035''PSB$

$\beta/An = 1°\beta/.25\ [\frac{1}{20S+1} + \frac{20S}{20S+1}]\ G$

$\beta/\theta = 1°\beta/70\ [\frac{20S}{20S+1}]°\theta$

$\theta/An = 1°\theta/.006\ [\frac{1}{20S+1}]G$

Flare

$fl\varepsilon = 200\text{mV/fPS}$

$fl\varepsilon/PSB = 1fPS\ fl\varepsilon/120mV\ PSB = .045''PSB$

$\theta/PSB = 1°\theta/88\ [\frac{4S}{4S+1}]mV\ PSB = .035''PSB$

$An/PSB = 1GAn/14\ [\frac{1}{20S+1} + \frac{4S}{4S+1}]V\ PSB = 5.6''PSB$

$\delta e/PSB = 1°\delta e/33\ [\frac{4S}{4S+1}]mV\ PSB = .013''PSB$

$fl\varepsilon/An = 1fPS/.008\ [\frac{1}{20S+1} + \frac{4S}{4S+1}]G$

$fl\varepsilon/\theta = 1fPS/1.36\ [\frac{4S}{4S+1}]deg\ \theta$

$fl\varepsilon/\delta e = 1fPS/3.6\ [\frac{4S}{4S+1}]deg\delta e$

$\theta/An = 1°\theta/.006\ [\frac{1}{20S+1}]G$

$\theta/\delta e = 1°\theta/2.6°\delta e$

$\delta e/An = 1°\delta e/.002\ [\frac{1}{20S+1}]G$

ILS (Track)

$$\eta = 75\text{mV/degree}$$

$$\eta/BSB = 1°\eta/750\text{mV}BSB = .3''BSB$$

$$\eta/BSB = 1°\eta/1.5V\ [\tfrac{8S}{8S+1}]\ BSB = .6''BSB$$

$$\psi/BSB = 1°\psi/36\text{mV}\ [\tfrac{8S}{8S+1}]\ BSB = .144''BSB$$

$$\phi/BSB = 1°\phi/30\text{mV}\ BSB = .012''BSB$$

$$\phi/BSB = 1°\phi/46\text{mV}\ [\tfrac{8S}{8S+1}]\ BSB = .02''BSB$$

$$\eta/\psi_{WO} = 1°\eta/21°\psi_{WO}$$

$$\eta/\phi = 1°\eta/25°\phi$$

$$\eta/\phi = 1°\eta/16°\phi$$

$$\psi_{WO}/\phi = 1°\psi/1°\phi$$

$$\psi_{WO}/\phi = 1.2°\psi_{WO}/1°\phi$$

VOR or TACAN (Track)

$$°\eta/BSB = 1°\eta/183mV \quad \frac{1}{(10S+1} \quad = .07''BSB$$

$$°\dot{\eta}/BSB = 1°\ sec\eta/220mV \left(\frac{1}{10S+1} + \frac{52S}{52S+1}\right. \quad = .088''BSB$$

$$°\phi/BSB = 1°\phi/40mV \frac{52S}{52S+1} \quad = .016''BSB$$

$$°\psi PSC/BSB = 1°\psi/40mv \frac{52S}{52S+1} \quad = .016''BSB$$

$\psi PSC/\ \eta = 4.57°\psi/°\eta$

$\phi/\eta = 4.57°\phi/°\ \eta$

$\phi/\psi PSC = 1°\phi/1°\psi_{PSC}$

Bank Limit = 15°ϕ

ILS (Capture)

= 75mV/degree

$°\eta/BSB = 1°\ \eta/1.0VBSB$ = .4"BSB

$\psi_{PSC}/BSB = 1°\psi/40mVBSB$ = .016"BSB

$\phi/BSB = 1°\phi/30mVBSB$ =.012"BSB

$\eta/\psi_{PSC} = 1°\eta/25°\psi_{PSC}$

$\eta/\phi = 1°\ /33.3°$

$\psi_{PSC}/\phi = 1°\eta\ _{PSC}/1.33°\phi$

45° Course Cut Limit

30° Bank Limit

C141A, No. 61-2775

APPENDIX C MASTER INSTRUMENTATION DATA

PARAMETER	SIGNAL TYPE	VOLTS	GRADIENT									
Pilot Event	DC	DISCRETE										
Event Mark	DC	DISCRETE										
AFSC Engage Elev. Servo	DC	DISCRETE										
AWLS Engage	DC	DISCRETE										
Pilot's RCD	DC	DISCRETE										
Left Decrab	DC	DISCRETE										
LH Stomp Lt.	AC	DISCRETE										
RH Stomp Lt.	AC	DISCRETE										
Nose Gear TD	DC	DISCRETE										
#1 Wheel Spin up	DC	DISCRETE										
#2 Wheel Spin up	DC	DISCRETE										
#3 Wheel Spin up	DC	DISCRETE										
#1 Yaw Servo Clutch	DC	DISCRETE										
#2 Yaw Servo Clutch	DC	DISCRETE										
Ail. Servo Clutch Eng.	DC	DISCRETE										
Elev. Servo Clutch Eng.	DC	DISCRETE										
AFCS Pitch Trim Disc.	DC	DISCRETE										
Flare Eng. #2	DC	DISCRETE										
Flare Eng. #2	DC	DISCRETE										
R/GA Engage	DC	DISCRETE										
Loc. Beam Sensor #1	DC	DISCRETE										
Loc. Beam Sensor #2	DC	DISCRETE										
GS. Beam Sensor #1	DC	DISCRETE										
GS. Beam Sensor #2	DC	DISCRETE										
#1 INS Comp. Warn.	DC	DISCRETE										

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PARAMETER	SIGNAL TYPE	VOLTS	GRADIENT									
#2 INS Comp. Warn	DC	DISCRETE										
#1 P/R Warn.	DC	DISCRETE										
#2 P/R Warn.	DC	DISCRETE										
Throttle Retard	DC	DISCRETE										
Ø1 Yaw Servo Var Ø	AC	36										
#2 Yaw Servo Var Ø	AC											
Aileron Servo Var Ø	AC											
Elev Servo Var Ø	AC											
X° (Runway Speed)	AC	22mV	FPS									
Ø1 Yaw Servo Tach	AC	2.2										
#2 Yaw Servo Tach	AC											
Aileron Servo Tach	AC											
Elev. Servo-Tach	AC											
Throttle Servo. Tach.	AC	50mV	Max rate									
Rudder Position F. U.	AC											
Aileron Position LH	AC											
Aileron Position RH	AC	320mV	°δ_A									
Elev. Position	AC	300mV	°α_e									
#2 Upper RH Spoiler Pos.	AC	320mV										
Horiz. Stab. Pos.	AC	1.5V	°α_e									
Flap Position R. H.	AC	200mV	°α_{flap}									
Throttle Position	AC											
Ø1 Yaw Servo F. U.	AC	430mV	°δ_R									
#2 Yaw Servo F. U.	AC	430mV	°δ_R									
Aileron Servo-F. U.	AC	168mV	°α_A									
Elev. Servo-F. U.	AC	325mV	°δ_e									
Pilots CWS Roll	AC	48.3mV	lb.									
Pilots CWS Pitch	AC	30mV	lb.									
TPLC IN #1 Roll	AC	200mV	DEG.									
TPLC IN Ø1 Pitch	AC	200mV	DEG.									
TPLC IN Ø2 Roll	AC	200mV	DEG.									
TPLC IN Ø2 Pitch	AC	200mV	DEG.									

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PARAMETER	SIGNAL TYPE	VOLTS	GRADIENT
TPLC IN #3 Roll	AC	200mV	DEG.
TPLC IN #3 Pitch	AC	200mV	DEG.
TPLC OUT ISS Roll	AC	200mV	DEG.
TPLC OUT ISS Pitch	AC	200mV	DEG.
#1 Yaw Rate	AC	200mV	Deg/Sec
#2 Yaw Rate	AC	200mV	Deg/Sec
#3 Yaw Rate	AC	200mV	Deg/Sec
#1 Roll Rate	AC	200mV	Deg/SEc
#2 Roll Rate	AC	200mV	Deg/Sec
#2 Pitch Rate	AC	200mV	Deg/Sec
#1 Pitch Rate	AC	200mV	Deg/Sec
#1 FDC CRS Error	AC	360mV	Deg.
#2 FDC CRS Error	AC	360mV	Deg.
#1 CADC V IAS	AC	250mV	Knot
#2 CADC V IAS	AC	250mV	Knot
#1 INS Roll	AC	200mV	Deg.
#2 INS Roll	AC	200mV	Deg.
#1 INS Pitch	AC	200mV	Deg.
#2 INS Pitch	AC	200mV	Deg.
#1 INS $\ddot{h}$	DC	+6	G
#2 INS $\ddot{h}$	DC	+6	G
#1 INS VDrift			
#2 INS VDrift			
Wings Level #1	AC	123mV	°error
Wings Level #2	AC	123mV	°error
Pilots BSB	DC	31mV	Deg
Co-pilots BSB	DC	31mV	Deg
Pilots PSB	DC	90mV	Deg
Co-pilots PSB	DC	90mV	Deg
#1 Decrab Comm	DC	108mV	° error

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PARAMETER	SIGNAL TYPE	VOLTS	GRADIENT									
IAS XDucer	DC	≈5mV	knot									
#1 INS MAG HDG	AC	200mV	Deg									
#2 INS MAG HDG	AC	200mV	Deg									
#1 INS PLTFRM HDG	AC	200mV	Deg									
#2 INS PLTFRM HDG	AC	200mV	Deg									
#1 INS Preset CRS	AC	450mV	Deg									
#2 INS Preset CRS	AC	450mV	Deg									
#1 INS Y Across R/W	AC	59mV	Ft/Sec									
#2 INS Y Across R/W	AC	59mV	Ft/Sec									
Desen. Loc #1	DC	60mV	Degree									
Desen. Loc #2	DC	60mV	Degree									
Loc Flag #1	DC	DISCRETE										
Loc Flag #2	DC	DISCRETE										
Desen. GS. #2	DC	215mV	Deg									
Radar Alt. #1	DC	10mV/	Ft.									
Radar Alt. #2	DC	10mV/	Ft.									
Nav. RX #1 Chan. A	DC	60mV	Deg.									
Nav. RX#1B INS LOC	DC	60mV	Deg.									
Nav. RX#2 Chan. A	DC	60mV	Deg.									
Nav. RX#2 Chan. B	DC	60mV	Deg.									
GS RX#1 Chan. A	DC	215mV	Deg.									
GS RX#1B INS GS	DC	215mV	Deg.									
GS RX#2 Chan.A	DC	215mV	Deg.									
GS RX#2 Chan. B	DC	215mV	Deg.									
Decrab Comm	AC											
CPLR Roll Comm	AC	100mV	Deg.									

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PARAMETER	SIGNAL TYPE	VOLTS	GRADIENTS									
CPLR Pitch Comm	AC	200mV	Deg									
Flare Error #1	DC	200mV	Ft/Sec									
Flare Error #2	DC	200mV	Ft/Sec									
Rudder Pedal Position	DC											
#1 Engine RPM	DC	81.8mV	1%RPM									
#2 Engine RPM	DC	81.8mV	1%RPM									
#3 Engine RPM	DC	81.8mV	1%RPM									
#4 Engine PPM	DC	81.8mV	1%RPM									
#1 Engine EPR	AC											
#2 Engine EPR	AC											
#3 Engine EPR	AC											
#4 Engine EPR	AC											
Throttle LImit Min.	DC	DISCRETE										
Throttle Limit Max.	DC	DISCRETE										
Horiz. Stab. FB	AC											
Long, Accel. #1	DC	8.0	G									
Vert. Accel #1	DC	8.0	G									
Vert. Accel#2	DC	8.0	G									
Vert. Accel #3	DC	8.0	G									
Vert. Accel #4	DC	8.0	G									
Vert. Accel #5	DC	8.0	G									
Vert. Accel. #6	DC	8.0	G									
Vert. Accel. #7	DC	8.0	G									
Vert. Accel. #8	DC	8.0	G									
#1 Roll XFeed	AC	58mV	Deg									
#2 Roll XFeed	AC	58mV	Deg									
Roll SFeed Model	AC	58mV	Deg									
#1 Lateral Accel	DC	1.0	G									
#2 Lateral Accel	DC	1.0	G									
H° BARO Comp #1	AC	250mV	FT/Min									
H°BARO CADC #1	AC	250mV	Ft/Min									
A/A Vane #1	AC	250mV	Deg									
A/A Vane #2	AC	250mV	Deg									