LM G&C DATA BOOK

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15 JULY 1967

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PREFACE

This LM Guidance and Control Data Book has been prepared for the Guidance and Control Division of the National Aeronautics and Space Administration, Manned Spacecraft Center, Houston, Texas, by TRW Systems, under Contract NAS 9-4810, ASPO Task 6B, "Maintenance and Operation of the Apollo Guidance and Control Data Exchange."

This book is Revision 2, and replaces Revision 1 of the LEM G&C Data Book, 1 June 1966 and Change A, 1 December 1966. In order to insure receipt of future changes, please notify the following if possession of this book is transferred from the original addressee:

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Any questions or comments regarding this data should be addressed to Larry Wilkerson at the above address. Any suggestions for improving the quality and usefulness of this document are welcomed. In the event this document is not used or retention is not desired, please return to the above address.

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REFERENCES

The following list of specifications and related documents is included to provide a reference to documents used in preparation of this data book. The specific revision letters listed below are effective as of this publication date. If more detail is desired, consult the applicable listed document.

LSP-300-3B	Abort Guidance Section; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-11E	Rate Gyro Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-13D	Descent Engine Control Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-14E	Control Assembly, Attitude and Translation; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-17C	Gimbal Drive Actuator Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-19B	Attitude Controller Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-33E	Abort Electronics Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-37E	Abort Sensor Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-39D	Data Entry and Display Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-300-180A	Thrust/Translation Controller Assembly; Guidance, Navigation, and Control Subsystem Design Control Specification
LSP-350-301B	Indicator, Attitude Displays Subsystem, Design Control Specification
LSP-350-302B	Transformation Assembly, Gimbal Angle Sequence, Design Control Specifications

4.0 LM GUIDANCE AND CONTROL DATA BOOK

The purpose of this document is to present data pertaining to the LM Guidance and Control Hardware. The information contained in this document was generated from master end item specifications, procurement specifications, drawings, status reports, study guides, and other documents provided to the data exchange by contractors in reply to data requests.

4.1 DEFINITIONS

Definitions for the LM Guidance and Control Data Book are defined as Acronyms, Assembly Levels, and the LM Coordinate System.

4.1.1 Apollo Acronyms

The following list of acronyms provides a selected few used in or associated with this document. For a more complete listing, refer to National Aeronautics and Space Administration document, "Apollo Acronym List," October 1966.

APOLLO ACRONYMS

AC	Alternating Current
ACA	Attitude Controller Assembly
ACE	Acceptance Checkout Equipment
ADA	Angular Differentiating Accelerometer
A/D	Analog to Digital (Conversion)
AE	Ascent Engine
AEA	Abort Electronics Assembly
AELD	Ascent Engine Latching Device
AGC	Apollo Guidance Computer (CSM)
AGS	Abort Guidance Section
AMPTF	Apollo Mission Planning Task Force
AOT	Alignment Optical Telescope
ASA	Abort Sensor Assembly
ATCA	Attitude and Translation Control Assembly
CCRD	Computer Control and Reticle Dimmer
CDU	Coupling Data Unit
CEA	Control Electronics Assembly
CES	Control Electronics Section
CM	Command Module
CPA	Control Panel Assembly
CSM	Command and Service Module
CSS	Computer Subsystem
D/A	Digital to Analog (Conversion)
DA	Decoder Assembly
D and C	Displays and Controls

DAP	Digital Autopilot
DC	Direct Current
DCA	Digital Command Assembly
DE	Descent Engine
DECA	Descent Engine Control Assembly
DEDA	Data Entry and Display Assembly
DIFF	Differential
DSEA	Data Storage Electronics Assembly
DSIF	Deep Space Instrumentation Facility
DSKY	Display and Keyboard
ECDU	Electronic Coupling Data Unit
ECS	Environmental Control Subsystem
EL	Electro Luminescence
EPS	Electrical Power Subsystem
FDAI	Flight Director Attitude Indicator
FOV	Field of View
GC	Guidance and Control
G&N	Guidance and Navigation
G&NS	Guidance and Navigation Subsystem
GAEC	Grumman Aircraft Engineering Corporation
GASTA	Gimbal Angle Sequence Transformation Assembly
GDA	Gimbal Drive Actuator
GSE	Ground Support Equipment
ICD	Interface Control Document
IGA	Inner Gimbal Axis

IMU	Inertial Measurement Unit
IRIG	Inertial Rate Integrating Gyroscope
JSL	Jet Select Logic
LM	Lunar Module
LES	Launch Escape System
LGC	LM Guidance Computer
LMP	LM Mission Programmer
LOR	Lunar Orbit Rendezvous
LOS	Line of Sight
LPD	Landing Point Designator
LR	Landing Radar
LRAA	Landing Radar Antenna Assembly
LREA	Landing Radar Electronics Assembly
MCT	Memory Cycle Time
MGA	Middle Gimbal Axis
MILA	Merrit Island Launch Area
MSC	Manned Spacecraft Center
MSFN	Manned Space Flight Network
NVB or NB	Navigation Base
NSIF	Near Space Instrumentation Facility
OGA	Outer Gimbal Assembly
OMSF	Office of Manned Space Flight
PCA	Program Coupler Assembly
PCMTE	Pulse Code Modulation and Timing Electronics
PDS	Power Distribution Subsystem

APOLLO ACRONYMS (Continued)

PGNCS	Primary Guidance, Navigation, and Control System
PGS	Power Generation Section
PIP	Pulse Integrating Pendulum
PIPA	Pulse Integrating Pendulous Accelerometer
PLSS	Portable Life Support System
PO	Parking Orbit
PPM	Pulse Position Modulation
PSA	Power and Servo Assembly
PTA	Pulse Torque Assembly
PTSA	Pulse Torquing Servo Amplifier
PVR	Precision Voltage Reference
RCS	Reaction Control System
RGA	Rate Gyro Assembly
RR	Rendezvous Radar
RRAA	Rendezvous Radar Antenna Assembly
RREA	Rendezvous Radar Electronics Assembly
S/C	Spacecraft
SCA	Signal Conditioning Assembly
SCS	Stabilization and Control Subsystem
S/C	Stabilization and Control Subsystem
SM	Service Module
SM	Stable Member
SPS	Service Module Propulsion System
TCE	Throttle Control Electronics
TCE	Temperature Control Electronics

APOLLO ACRONYMS (Continued)

TLI	Trans Lunar Insertion
TM	Telemetry
TOI	Transfer Orbit Insertion
T/TCA	Thrust Translation Controller Assembly
TVC	Thrust Vector Control
VAB	Vertical Assembly Building
VCO	Voltage Control Oscillator
ΔV	Velocity Differential

4.1.2 Assembly Levels

The following definitions of assembly levels are as defined in LSP-470-1, Contract Technical Specification for Lunar Excursion Module System.

System	Complete primary function
Subsystem	A major integral part of a system representing the first level of functional breakdown below the total system level.
Equipment	A generic term including sections, assemblies, subassemblies, and components.
Section	An arrangement of interrelated assemblies performing a certain function within the subsystem. A section can be classified in a general functional class.
Assembly	An arrangement of subassemblies, components, and parts, performing one or more functions. The assembly represents the "black box" equipment level and will usually define the first level of subsystem external interfaces.
Subassembly	The smallest grouping of components that can be physically separated from the assembly and defined in terms of its contributing functions.
Component	The smallest grouping of parts that performs a single elementary function within a subassembly.
Part (Element)	The smallest basic element found in a component. It is the lowest assembly level for which traceability is maintained.

4.1.3 LM Coordinate System

Figure 4.1.3-1 depicts the coordinate system referred to throughout this document.

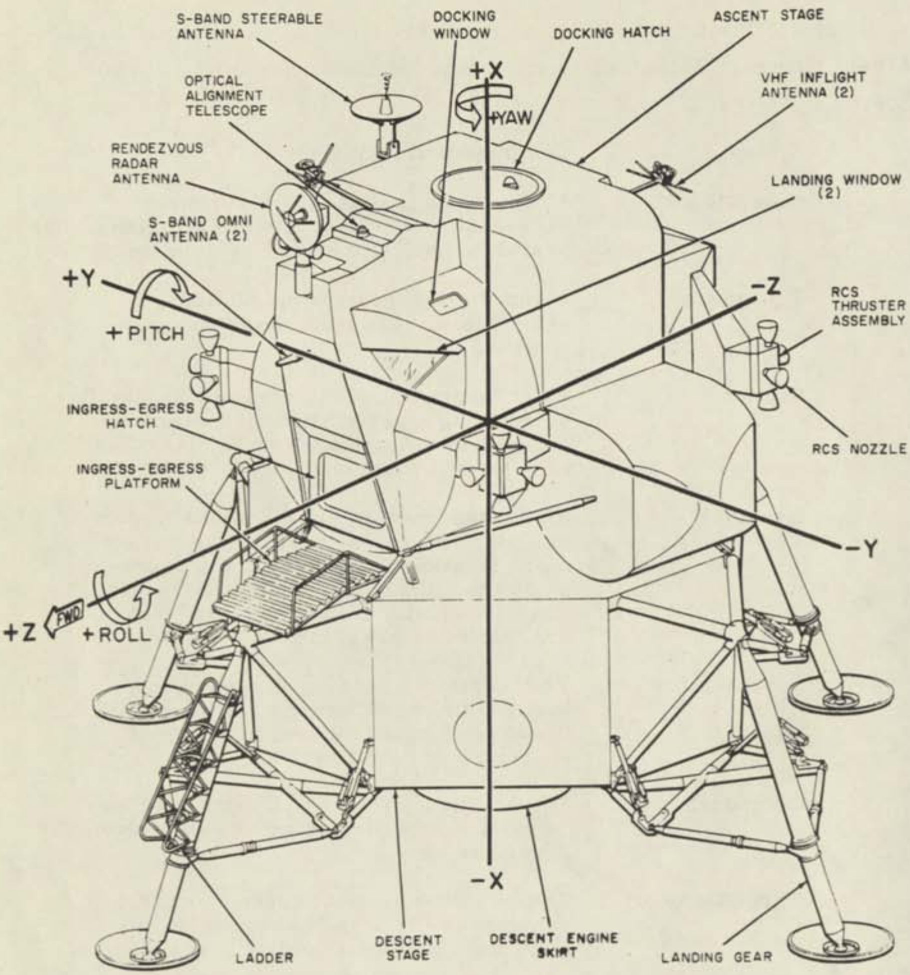


Figure 4.1.3-1. LM Coordinate System

4.2 GUIDANCE AND CONTROL SYSTEM CONFIGURATION (G&C)

The Guidance and Control System is comprised of the Guidance and Navigation Subsystem (G&NS) and the Stabilization and Control Subsystem (SCS).

4.2.1 Block Diagrams and Functional Descriptions

Figure 4.2.1-1 is an overall block diagram of the LM G&C System. A functional description of the G&C Subsystem is presented in the following paragraphs.

4.2.1.1 Guidance and Navigation Subsystem

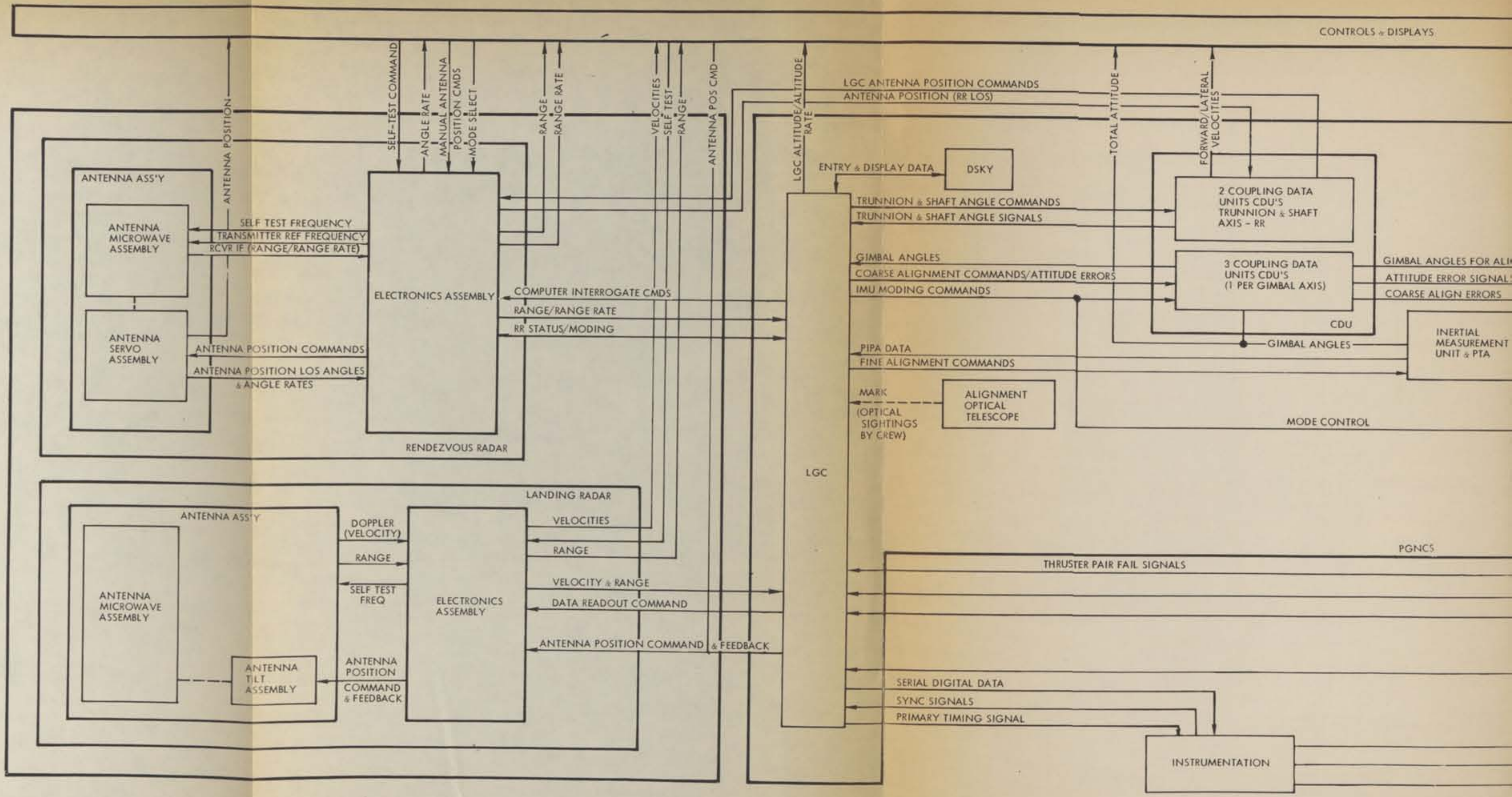
The G&N subsystem provides the measuring and data processing capabilities and control functions necessary to accomplish lunar landing and ascent and rendezvous and docking with the Command Service Module (CSM). The G&N subsystem is comprised of the Primary Guidance Navigation and Control Subsystem, which includes the inertial measurement unit (IMU), the LM guidance computer (LGC), the power and servo assembly (PSA), the coupling data units (CDU's), the display and keyboard (DSKY), the pulse torque assembly (PTA), the signal conditioner assembly (SCA), the navigation base (NB), the alignment optical telescope (AOT), and the radar sections, which includes the landing radar (LR) and the rendezvous radar (RR).

4.2.1.1.1 Primary Guidance Navigation and Control Subsystem (PGNCS). The PGNCS uses inertial components for guidance, an optical device and radar for navigation, and a digital computer for data processing and issuance of flight control signals.

The PGNCS is functionally divided into three major subsystems: inertial, optical, and computer. The PGNCS performs three basic functions: inertial guidance, navigation, and autopilot stabilization and control. Within these functions the subsystems, or combination of subsystems, with assistance from the astronaut, perform the following operations:

a) Establish an inertial reference, which is used for measurements and computations

b) Align the inertial reference by optical measurements and, through interface, align the inertial reference with the CSM PGNCS


CONTROLS & DISPLAYS
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ANGLE RATE
MANUAL ANTENNA POSITION CMDS
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RANGE RATE
VELOCITIES
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COMPUTER INTERROGATE CMDS
RANGE/RANGE RATE
RR STATUS/MODING
RENDEZVOUS RADAR
ENTRY & DISPLAY DATA
DSKY
TRUNNION & SHAFT ANGLE COMMANDS
TRUNNION & SHAFT ANGLE SIGNALS
2 COUPLING DATA UNITS CDU'S TRUNNION & SHAFT AXIS - RR
GIMBAL ANGLES
COARSE ALIGNMENT COMMANDS/ATTITUDE ERRORS
IMU MODING COMMANDS
3 COUPLING DATA UNITS CDU'S (1 PER GIMBAL AXIS)
CDU
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ATTITUDE ERROR SIGNAL
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GIMBAL ANGLES
INERTIAL MEASUREMENT UNIT & PTA
PIPA DATA
FINE ALIGNMENT COMMANDS
MARK
(OPTICAL SIGHTINGS BY CREW)
ALIGNMENT OPTICAL TELESCOPE
MODE CONTROL
LGC
LANDING RADAR
ANTENNA ASS'Y
ANTENNA MICROWAVE ASSEMBLY
ANTENNA TILT ASSEMBLY
DOPPLER (VELOCITY)
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SELF TEST FREQ
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ELECTRONICS ASSEMBLY
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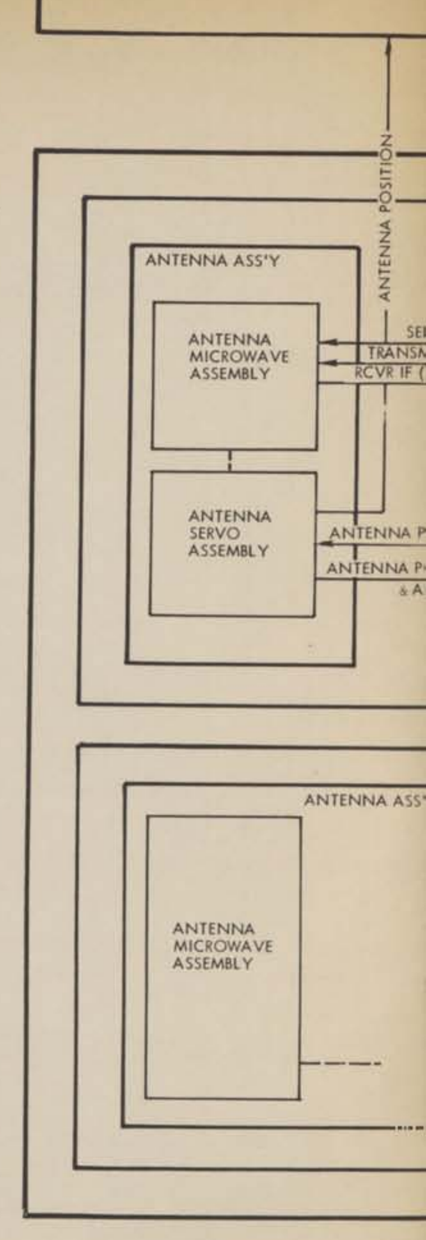
DATA READOUT COMMAND
ANTENNA POSITION COMMAND & FEEDBACK
PGNCS
THRUSTER PAIR FAIL SIGNALS
SERIAL DIGITAL DATA
SYNC SIGNALS
PRIMARY TIMING SIGNAL
INSTRUMENTATION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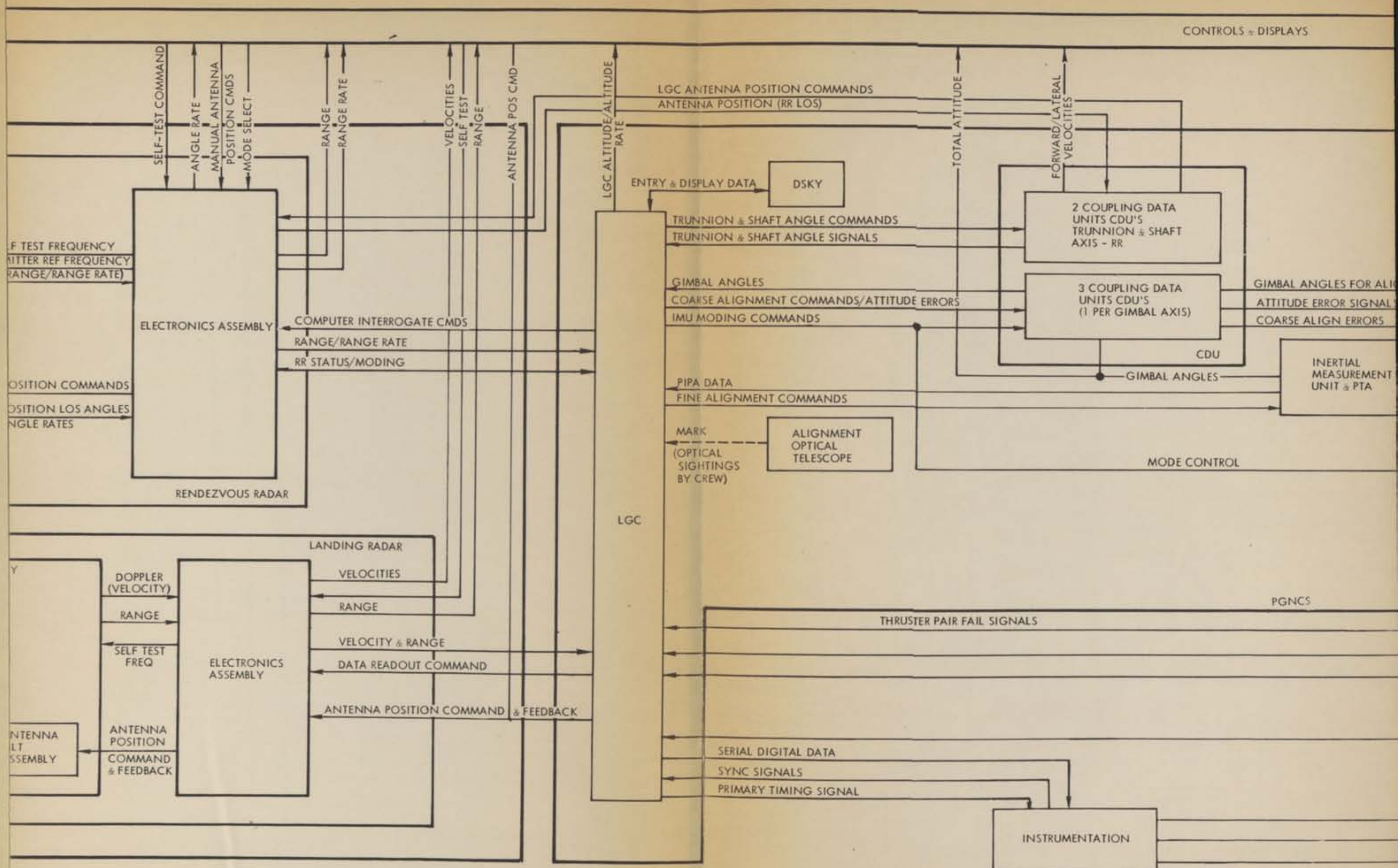
ANTENNA MICROWAVE ASSEMBLY


CONTROLS & DISPLAYS
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ANTENNA POS CMD
LGC ALTITUDE/ALTITUDE RATE
LGC ANTENNA POSITION COMMANDS
ANTENNA POSITION (RR LOS)
TOTAL ATTITUDE
FORWARD/LATERAL VELOCITIES
ENTRY & DISPLAY DATA
DSKY
TRUNNION & SHAFT ANGLE COMMANDS
TRUNNION & SHAFT ANGLE SIGNALS
2 COUPLING DATA UNITS CDU'S TRUNNION & SHAFT AXIS - RR
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GIMBAL ANGLES
COARSE ALIGNMENT COMMANDS/ATTITUDE ERRORS
IMU MODING COMMANDS
3 COUPLING DATA UNITS CDU'S (1 PER GIMBAL AXIS)
GIMBAL ANGLES FOR ALI
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ELECTRONICS ASSEMBLY
COMPUTER INTERROGATE CMDS
RANGE/RANGE RATE
RR STATUS/MODING
CDU
GIMBAL ANGLES
INERTIAL MEASUREMENT UNIT & PTA
OSITION COMMANDS
OSITION LOS ANGLES
NGLE RATES
PIPA DATA
FINE ALIGNMENT COMMANDS
MARK
(OPTICAL SIGHTINGS BY CREW)
ALIGNMENT OPTICAL TELESCOPE
MODE CONTROL
RENDEZVOUS RADAR
LGC
LANDING RADAR
DOPPLER (VELOCITY)
RANGE
SELF TEST FREQ
VELOCITIES
RANGE
ELECTRONICS ASSEMBLY
VELOCITY & RANGE
DATA READOUT COMMAND
PGNCS
THRUSTER PAIR FAIL SIGNALS
ANTENNA POSITION COMMAND & FEEDBACK
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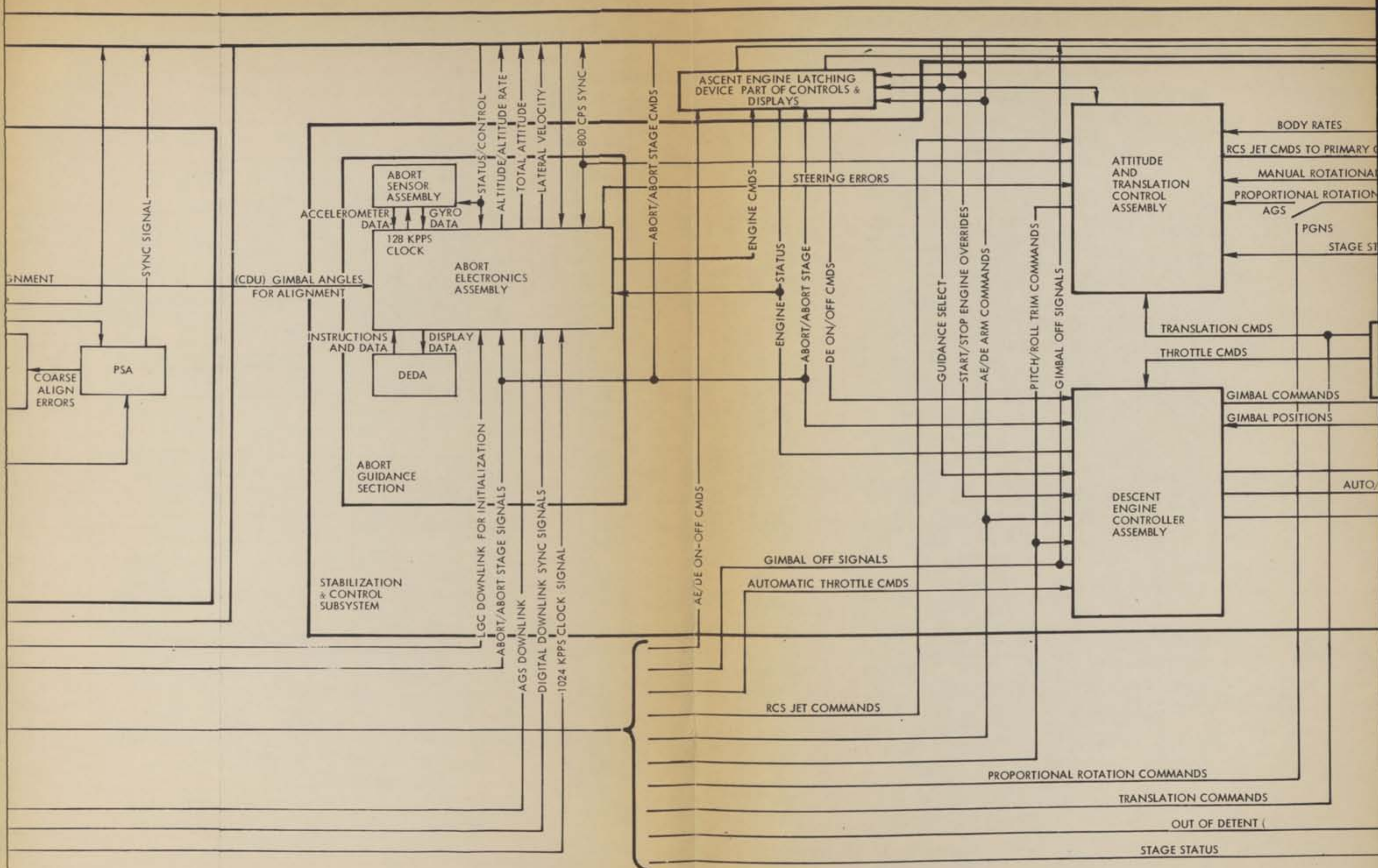
ANTENNA POSITION
COMMAND & FEEDBACK
SERIAL DIGITAL DATA
SYNC SIGNALS
PRIMARY TIMING SIGNAL
INSTRUMENTATION



ASCENT ENGINE LATCHING DEVICE PART OF CONTROLS & DISPLAYS
BODY RATES
RCS JET CMDS TO PRIMARY
MANUAL ROTATIONAL
PROPORTIONAL ROTATION
AGS
PGNS
STAGE ST
ATTITUDE AND TRANSLATION CONTROL ASSEMBLY
STEERING ERRORS
ABORT SENSOR ASSEMBLY
ACCELEROMETER DATA
GYRO DATA
128 KPPS CLOCK
ABORT ELECTRONICS ASSEMBLY
STATUS/CONTROL
ALTITUDE/ALTITUDE RATE
TOTAL ATTITUDE
LATERAL VELOCITY
800 CPS SYNC
ABORT/ABORT STAGE CMDS
ENGINE CMDS
ENGINE STATUS
ABORT/ABORT STAGE
DE ON/OFF CMDS
GUIDANCE SELECT
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AUTO/
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ABORT/ABORT STAGE SIGNALS
AGS DOWNLINK
DIGITAL DOWNLINK SYNC SIGNALS
1024 KPPS CLOCK SIGNAL
AE/DE ON-OFF CMDS
STABILIZATION & CONTROL SUBSYSTEM
GIMBAL OFF SIGNALS
AUTOMATIC THROTTLE CMDS
RCS JET COMMANDS
PROPORTIONAL ROTATION COMMANDS
TRANSLATION COMMANDS
OUT OF DETENT (
STAGE STATUS

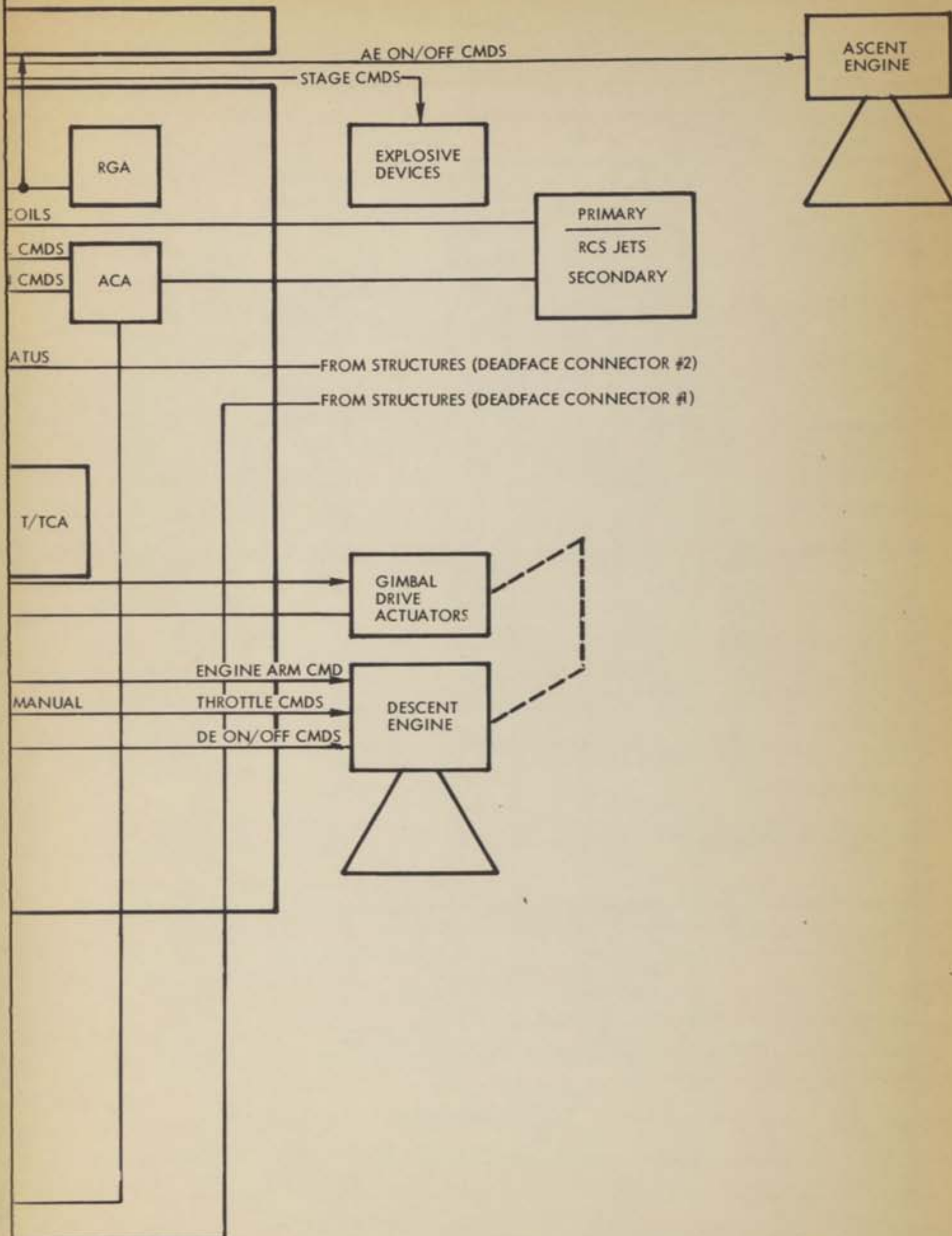


Figure 4.2.1-1. Block Diagram - LM G&C System

c) Calculate the position and velocity of the LM by inertial navigation

d) Accomplish a LM and CSM rendezvous by radar tracking, optical navigation, and inertial guidance

e) Generate attitude control and thrust commands to maintain the LM on a satisfactory trajectory

f) Control throttling of descent engine during lunar landing

g) Display pertinent data related to guidance status

h) Control ascent engine burn time to obtain proper velocity for rendezvous orbit

To perform its inertial guidance functions, the PGNCS employs an IMU containing accelerometers mounted on a gyro stabilized, gimbal-mounted platform (stable member). The IMU, three channels of the CDU, the pulse torque assembly, and the PSA form the Inertial Subsystem of the PGNCS.

To perform its navigational functions, the PGNCS employs the IMU and the Radar Sections. The AOT provides a means of manually taking direct visual sightings and precision angular measurements of preselected star targets. During the powered descent and landing phases, the PGNCS receives altitude and velocity data from the Landing Radar, which is used to update or check inertially derived data. During the coasting descent, lunar stay, and rendezvous phases, the Rendezvous Radar tracks the transponder in the orbiting CSM to provide range, range rate, and antenna angle measurements to the LGC.

The LGC is a digital computer which serves as both the control element and the primary data processing element of the PGNCS. The LGC and the display and keyboard form the computer subsystem of the PGNCS.

4.2.1.1.1.1 <u>LM and PGNCS Axes.</u> Several sets of axes are associated with the LM and PGNCS. Figure 4.2.1.1-1 illustrates these various orthogonal sets which are defined in the following paragraphs. Positive rotation about each axis is as defined by the right-hand rule.

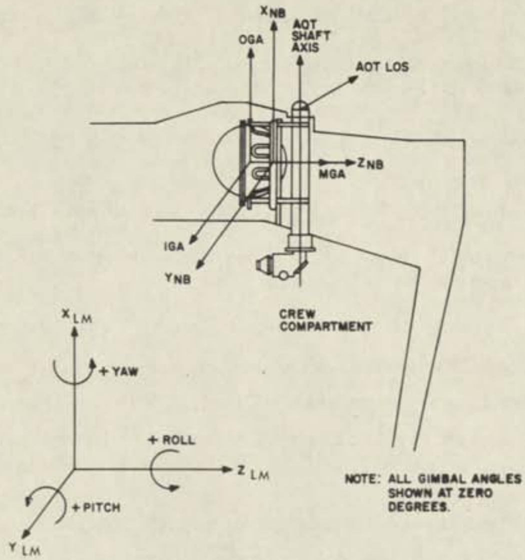


Figure 4. 2. 1. 1-1 LM and PGNCS Axes

4. 2. 1. 1. 1. 1. 1 LM Spacecraft Axes. The LM spacecraft axes provide a reference for all other sets of axes and define the point about which attitude maneuvers are performed. The LM spacecraft axes, designated X_{LM}, Y_{LM}, Z_{LM}, are referred to as the yaw, pitch, and roll axes, respectively. The X_{LM} axis points through the upper docking hatch, and the Z_{LM} axis points through the forward hatch. The Y_{LM} axis is perpendicular to the X_{LM}, and the Z_{LM} axis and can be considered to be pointing out of the astronaut's right shoulder as he faces toward the forward portion of the LM.

4. 2. 1. 1. 1. 1. 2 Navigation Base Axes. The navigation base provides a precise alignment of the IMU to the AOT and the ASA and a means of attaching all three units to the spacecraft. The navigation base is mounted to the LM structure so that a coordinate reference system is formed by its

mounting points. The Y_{NB} axis is defined by the centers of the two upper mounting points and is parallel to the Y_{LM} axis. The X_{NB} axis is defined by a line through the center of the lower mounting point, perpendicular to the Y_{NB} axis and parallel to the X_{LM} axis. The Z_{NB} axis is mutually perpendicular to the X_{NB} and Y_{NB} axes and is parallel to the Z_{LM} axis.

4.2.1.1.1.1.3 Inertial Axes. The inertial axes provide references for measuring changes in velocity and attitude. At zero gimbal angles, the inertial axes are parallel to the navigation base axes.

4.2.1.1.1.1.3.1 Gimbal Axes. The gimbal axes (outer, middle, and inner) are the axes of the movable gimbals. The axes are defined by the inter-gimbal assemblies which provide each gimbal with rotational freedom. The attitude of the spacecraft, with respect to the stable member, is measured by the gimbal resolvers located in the inter-gimbal assemblies.

4.2.1.1.1.1.3.2 Stable Member Axes. The stable member axes (X_{SM}, Y_{SM}, Z_{SM}) provide a reference for aligning the inertial components and for defining the angular orientation of the inertial axes during flight.

4.2.1.1.1.1.3.3 Accelerometer Axes. The accelerometer axes (X_a, Y_a, Z_a) are the positive input axes of the accelerometers and are parallel to the stable member axes. Velocity changes are measured along the accelerometer input axes. These velocity data are used to determine spacecraft position and velocity.

4.2.1.1.1.1.3.4 Gyro Axes. The gyro axes (X_g, Y_g, Z_g) are the positive input axes of the stabilization gyros and are parallel to the stable member axes. If the attitude of the stable member is changed with respect to inertial space, the gyro senses the change about its input axis and provides an error signal to a servo loop which realigns the stable member to its original orientation.

4.2.1.1.1.2 Inertial Subsystem (ISS). The ISS performs three major functions. It measures changes in LM attitude, assists in generating steering commands, and measures spacecraft velocity as a result of thrust. To accomplish these functions, the IMU provides an inertial reference consisting of a stable member with a three-degree-of-freedom gimbal system and stabilized by three rate-integrating gyros. Each time the inertial subsystem is energized, the stable member must be aligned with respect to a predetermined reference. During flight and prior to launch from the lunar surface, this alignment is accomplished by sighting the alignment optical telescope on star targets.

Once the ISS is energized and aligned, any rotational motion of the LM will be about the stable member, which remains fixed in space. Resolvers mounted on the gimbal axes act as angular sensing devices and measure the attitude of the LM, with respect to the stable member. These angular measurements are displayed by the flight director attitude indicator (FDAI), and angular changes are sent to the LGC via the CDU.

The desired LM attitude is calculated in the LGC and compared with the actual gimbal angles. Any difference between the actual and calculated angles results in the generation of attitude error signals by the ISS channels of the CDU, which are sent to the FDAI for display.

Vehicle acceleration is sensed by three pendulous accelerometers mounted on the stable member with their input axes orthogonal. The signals from the accelerometers are supplied to the LGC, which calculates the total vehicle velocity.

The modes of operation of the inertial subsystem can be initiated automatically by the LGC or by the astronaut selecting computer programs through the DSKY. The status or mode of operation is displayed on the DSKY. (Refer to Section 4. 2. 2 for detailed mode descriptions.)

4. 2. 1. 1. 1. 3 Computer Subsystem. The computer subsystem (CSS) is the control and processing center of the PGNCS. It consists of the LGC and a DSKY. The CSS processes data and issues discrete outputs and control pulses to the PGNCS and other LM systems. The LGC is a parallel digital control computer with many features of a general purpose computer. As a control computer, the LGC aligns the IMU, positions the RR antenna, and issues control commands to other LM systems. As a general purpose computer, the LGC solves the guidance and navigation equations required for the LM mission. In addition, the LGC monitors the operation of the LM including the CSS.

4. 2. 1. 1. 1. 4 Alignment Optical Telescope. The AOT provides a means of manually taking direct visual sightings and precision angular

measurements of preselected star targets. These measurements are manually transferred by the astronaut to the LGC through the DSKY. The LGC uses this angular information along with prestored data to compute the LM position and velocity and to accomplish a fine alignment of the IMU stable member.

4.2.1.1.2 Radar Subsystems. The rendezvous radar and the landing radar comprise the Radar Sections. The purpose of these systems is to provide range, range rate, and angular measurements to the PGNCS and AGS during rendezvous and to provide altitude and velocity to the PGNCS during lunar landing operations.

4.2.1.1.2.1 Rendezvous Radar. The Rendezvous Radar is a space-stabilized, continuous-wave tracking radar. It is an all solid-state, coherent tracking radar used in the LM for accomplishing rendezvous with the CSM in an 80-nautical mile orbit around the moon. Operating in conjunction with a transponder located in the CSM, the rendezvous radar measures range, range rate, angle, and angle rate with respect to the CSM. Both rendezvous radar and transponder use the solid-state varactor multipliers as transmitters. Transmission and reception are both performed in the CW mode. Gyros located on the rendezvous radar antenna assembly stabilize the aperture line-of-sight against variations in LM body motion, thus permitting accurate measurements of angular rate.

Angle tracking is accomplished by using an amplitude-comparison monopulse technique to obtain maximum angular sensitivity and boresight accuracy. Range rate is determined by measuring the two-way Doppler frequency shift on the signal received from the transponder. Range is determined by measuring the time delay between the transmitted signal modulation waveform and the received signal waveform. A three-tone modulation system is used to achieve range accuracy requirements.

The rendezvous radar comprises an antenna assembly (RRAA) and an electronic assembly (RREA). The antenna assembly converts vhf signals into microwave-modulated signals and propagates them through space. The return signal is converted into an IF signal and sent to the electronic assembly. The antenna assembly locks on and continually tracks the source of the return signal.

The electronic assembly furnishes crystal-controlled signals that drive the antenna assembly transmitter and provide a reference for receiving and processing the return signal. This assembly also supplies servo drive signals for antenna positioning.

4.2.1.1.2.2 Landing Radar. The landing radar senses the velocity and range of the LM relative to the lunar surface by means of a three-beam Doppler velocity sensor and a radar altimeter. The velocity and range information is processed and made available to the LM guidance computer in serial binary form and to the LM displays in the form of pulse trains and dc analog voltages.

The landing radar, located in the descent stage of the Lunar Module, is packaged in two replaceable assemblies. The LR Antenna Assembly (LRAA) serves to form, direct, transmit, and receive four narrow microwave beams. To perform this function the LRAA is composed of two interlaced phase arrays for transmission and four space-duplexed planar arrays for reception. The transmitting arrays form a platform on which are mounted four microwave mixers, four dual audio frequency preamplifiers, two solid-state microwave transmitters, an FM modulator, and an antenna pedestal tilt mechanism. The LR electronic assembly (LREA) contains the circuitry required to track, process, convert, and scale the Doppler and FM/CW returns which provide the velocity and slant-range information to the LGC and to the astronauts displays.

These data obtained by direct microwave contact with the lunar surface are used by the computer and the astronauts to achieve a controlled rate of descent, a hover at low altitude and to permit selection of a landing site, and a soft landing at the selected site.

4.2.1.2 Stabilization and Control System (SCS)

The SCS consists of two sections, the abort guidance section (AGS) and control electronics section (CES). The AGS provides the CES automatic steering commands derived from explicit guidance equations in the event of mission abort due to a PGNCS malfunction. The CES processes flight control signals during all mission phases.

4.2.1.2.1 Abort Guidance Section (AGS). AGS is a strapped-down inertial navigation and guidance system. The purpose of the AGS is to navigate and guide the LM back to a lunar rendezvous with the CSM in the event of a PGNCS failure. During an abort, the AGS determines the LM trajectory, or trajectories, required for rendezvous. Rendezvous from the abort point can be accomplished automatically under AGS control or manually by the astronaut, based on data displayed from the AGS. The abort trajectory is either of two paths; a path that intercepts the CSM within one orbit, or a path that places the LM in a parking orbit preparatory for a later rendezvous with the CSM.

The AGS is comprised of three assemblies: the abort electronics assembly (AEA), the data entry and display assembly (DEDA), and the abort sensor assembly (ASA). See Figure 4.3.3.1-1 and Section 4.3.3 for further breakdown of the AGS.

The AEA is mounted in the aft equipment bay; the ASA is mounted on the PGNCS navigation base on the AOT mounting pads; and the DEDA is located in the bottom side panel (Panel VI).

The ASA senses incremental angular displacement about the vehicle axes and velocity increments along the vehicle orthogonal axes. These angular and velocity increments are transmitted to the AEA in the form of pulses. An initial angle reference frame is established by aligning the AGS with the PGNCS. The state vector initialization is required to establish velocity, time, and position information. Velocity and position vectors and time may be manually entered into the computer via the DEDA or automatically from the PGNCS telemetry down link. Attitude alignment is accomplished by transferring PGNCS IMU gimbal angles to the AEA.

The outputs of the ASA go to the AEA, a 4,096 word capacity general purpose computer. Computations are performed using the inputs from the ASA. When the AGS is in control of the LM, the results are displayed, and control signals are issued to the vehicle's reaction control and propulsion systems.

After the AGS is initially aligned with the LM attitude data and initialized with LM and CSM position, velocity, and epoch time data, the AGS continuously computes attitude, position, and velocity of the LM and position and velocity of the CSM.

Control of the LM by the AGS is dependent upon control panel switch settings and input function selections to the AEA via the DEDA. In order to place the AGS in control, the guidance control switch (PGNCS, AGS) must be in the AGS position, the mode control selector (OFF, ATT HOLD, AUTO) may be in either the ATT HOLD or AUTO position. If the ABORT or ABORT STAGE pushbutton has not been depressed, the AGS will provide only attitude hold. In order that the AGS perform guidance steering (rather than just attitude hold) and engine control, the mode control selector must be in the AUTO position. In addition, the RCS attitude control must be under mode control direction. This is accomplished by setting the ROLL, PITCH, and YAW attitude control switches (MODE CONT, PULSE, DIR) to the MODE CONT position. Refer to Section 4.2.2 for a more detailed description of mode operations.

A detailed listing of AGS functions is presented in Table 4.2.1.2.1-1.

4.2.1.2.2 Control Electronics Section (CES). The CES is designed to control vehicle attitude and translation, about or along all vehicle axes, during the LM mission. Vehicle attitude and small translations are controlled by means of the 16 RCS jets mounted on the ascent stage. See Figure 4.1.3-1 for RCS jet locations. Major translations are accomplished by the main propulsion systems, i.e., ascent or descent engines.

The CES consists of an attitude and translation control assembly (ATCA), a descent engine control assembly (DECA), rate gyro assembly (RGA), two thrust/translation controller assemblies (T/TCA), and two attitude controller assemblies (ACA). The attitude and translational control inputs originate from any of three sources: (1) the PGNCS during

Table 4.2.1.2.1-1. Abort Guidance Section Functions

Mission Configuration	AGS Functions
A. EARTH PRELAUNCH	Preflight checkout and calibration
B. EARTH LAUNCH THROUGH TRANSLUNAR FLIGHT	Nonfunctioning
C. LUNAR ORBIT (PREPARATION FOR LANDING MISSION	Turn on of AGS (off, warmup, operate conditions)
During this phase, two astronauts are aboard the LM, and the LM and CSM are in lunar orbit approximately 80 nautical miles above the lunar surface.	Inflight self test
	Inflight gyro drift calibration
	Align inertial reference to PGNCS
	Initialize LM and CSM state vector using PGNCS downlink and absolute time via the DEDA
	Enter follow-up mode before descent orbit injection
D. LUNAR DESCENT ORBIT INJECTION (NONABORT CONDITION	Maintain inertial reference
During this phase, LM is inserted into Hohmann elliptical transfer orbit.	Maintain navigation
	Continually solve for abort trajectory which will either intercept the CSM in its orbit directly or execute concentric flight plan (depending on option selected)
	Provide engine-on command for mission abort
	Provide data for inflight monitoring
	Provide analog and digital telemetry
	Accept direct rendezvous or concentric commands
E. LUNAR DESCENT ORBIT INJECTION (ABORT CONDITION	Provide steering signals and engine-off commands as required to steer LM to rendezvous
During this phase, insertion into descent orbit is stopped, and LM rendezvous with CSM.	Maintain inertial reference and navigation
	Automatic mode of operation
	Continually solve for abort trajectory
	Provide engine-on command

Table 4.2.1.2.1-1. Abort Guidance Section Functions (Continued)

Mission Configuration	AGS Functions
	Provide data for inflight monitoring After engine shutdown, AGS functions as in Step O.
F. LUNAR DESCENT COAST (NONABORT CONDITION) During this phase, LM coasts to 50, 000 feet, approximately 225 nautical miles uprange from proposed landing sight.	Maintain inertial reference Follow-up mode of operation Maintain navigation Accept PGNCS alignment information for realignment of ASA Accept PGNCS downlink information (reinitialize LM and CSM) Continually solve the abort problem Provide data for inflight monitoring Provide digital telemetry Accept direct rendezvous or parking orbit commands via the DEDA
G. LUNAR DESCENT COAST (ABORT CONDITION) During this phase, descent orbit is terminated and LM rendezvous with CSM	Automatic submode Maintain inertial reference Maintain navigation Continually solve for abort trajectory Provide engine-on commands Provide steering signals and engine-off command as required to steer LM to rendezvous with the CSM or to a parking orbit After engine shutdown, AGS functions same as in Step E. Provide data for inflight monitoring
H. POWERED DESCENT (NONABORT CONDITION) During this phase, the LM is braking from 50, 000 feet to 10, 000 feet; final approach occurs to 700 feet from which a landing is made.	Same as Step D, Lunar Descent Orbit Injection (Nonabort Condition)

Table 4.2.1.2.1-1. Abort Guidance Section Functions (Continued)

Mission Configuration	AGS Functions
I. POWERED DESCENT (ABORT CONDITION)	Same as E, Lunar Descent Orbit Injection (Abort Condition)
J. LUNAR SURFACE (PRELAUNCH CHECKOUT; NONABORT CONDITION)	Off-standby-operate condition
	Lunar preflight checkout and calibration
	Self-test mode
	Align inertial reference to PGNCS
	Initialize LM via PGNCS downlink telemetry
	Initialize CSM and absolute time via the DEDA
	AGS in follow-up mode of operation prior to liftoff
K. LUNAR SURFACE (PRELAUNCH CHECKOUT; ABORT CONDITION)	Off-standby-operate condition
	Lunar preflight checkout and calibration/compensation
	Self-test mode
	Enter lunar alignment submode of operation
	Accept azimuth alignment data via the DEDA
	Accept LM, CSM, and absolute time data via DEDA
	Initialize LM, CSM, and absolute time via DEDA
	Enter Inertial Reference Mode
	Continually solve for abort problem from lunar surface
	Provide data for inflight monitoring
	Provide digital and analog telemetry
	On entry into automatic submode functions as described in Step E

Table 4.2.1.2.1-1. Abort Guidance Section Functions (Continued)

Mission Configuration	AGS Functions
L. LUNAR ASCENT BOOST (NONABORT CONDITION)	Same as Step D, Lunar Descent Orbit Injection (Nonabort Condition)
M. LUNAR ASCENT BOOST (ABORT CONDITION)	Same as Step E, Lunar Descent Orbit Injection (Abort Condition)
N. LUNAR ASCENT COAST (NONABORT CONDITION)	Same as Step F, Lunar Descent Coast (Nonabort Condition)
O. LUNAR ASCENT COAST (ABORT CONDITION)	Attitude hold submode
	Maintain inertial reference and navigation with Z-axis pointing for radar acquisition
	Acquisition submode
	Accept radar data to update navigation
	Solve for midcourse trajectory correction to enable rendezvous with CSM
	Provide inflight monitoring data
	Provide analog and digital telemetry
P. LUNAR ASCENT MIDCOURSE CORRECTION (ABORT CONDITION)	Maintain inertial reference
	Provide attitude reorientation commands on entry into semiautomatic mode of operation
	On entry into automatic submode, function as described in Step E
Q. CSM/LM RENDEZVOUS (ABORT AND NONABORT CONDITIONS)	Maintain inertial reference
	Maintain navigation
	Follow-up or attitude hold submodes of operation

normal automatic operation, (2) the ACA and TCA during manual operations, or (3) the AGS during an abort.

The CES operates in both the PGNCS mode and the AGS mode. When operating as part of the primary guidance control path in conjunction with the PGNCS, the CES provides implementation of the control signals originating from the LM guidance computer (LGC) as follows:

a) Converts reaction control jet commands to the required electrical power to operate the reaction control subsystem jet solenoid valves.

b) Accepts discrete (ON/OFF) descent engine rotation commands. Upon receipt of an ON command, the descent engine is rotated about its gimbal axes at a constant angular rate until the ON command is removed.

c) Accepts both LM Guidance Computer and manual engine ON/OFF commands and routes them to propulsion to start or stop the descent or ascent engine.

d) Accepts LM Guidance Computer and manual thrust commands to throttle the descent engine from 10 to 100 percent of its range.

When operating in the abort guidance control path, the CES performs the following functions:

a) Accepts attitude error signals from the Abort Guidance Section or manual attitude commands and fires the proper reaction control jets to achieve attitude control.

b) Accepts manual translation commands and fires RCS jets to accelerate the LM in the desired direction.

c) Automatically rotates the gimballed descent engine for trim control.

d) Accepts Abort Guidance Section and manual engine ON/OFF commands and routes them to the descent or ascent engine.

e) Accepts manual throttle commands to control the thrust of the descent engine.

4.2.2 Guidance and Control System Mode Descriptions

Guidance and control system mode descriptions are divided into two categories: (1) guidance and navigation subsystem modes, which include the primary guidance navigation control system and the radar systems (2) and Stabilization and Control System, which includes the abort guidance section and the control electronics section. For related information affecting mode operations, refer to the displays and controls section for additional information on individual switch and control operations. The data provided in the following paragraphs present a general description of mode functions. Related guidance and control documentation may contain material on this subject presented on a more detailed level which may present a semantics problem on mode definitions and functions. Data in this presentation were obtained primarily from LSP-470-2C. Modes described in this section should not be confused with modes discussed in Section 4.2.2.

4.2.2.1 PGNCS Modes

PGNCS modes consist of inertial subsystem and computer subsystem operations. The PGNCS has two primary operating modes (automatic and attitude hold) and three secondary operating modes (IMU temperature control, alignment, and LGC control). The primary operating modes are described in the control electronics section along with related abort guidance section functions. The secondary modes which pertain to the inertial and computer subsystems are described in the following sections.

4.2.2.1.1 IMU Temperature Control Mode. The IMU temperature control system maintains the temperature of the stabilization gyros and accelerometers within the required temperature limits during both standby and operating conditions of the IMU. The system supplies and removes heat to maintain the IMU heat balance. Heat is removed by convection, conduction, and radiation. The natural convection used during IMU nonoperating conditions changes to blower controlled, forced convection during IMU operating conditions. The IMU internal pressure is maintained between 3.5 and 15 pounds per square inch to enable the required force convection. To aid in removing heat, a water-glycol solution at approx-45 degrees Fahrenheit from the spacecraft coolant system passes through the coolant passages in the IMU case. The temperatures control system consists of three integrated circuits; temperature control circuit, temperature alarm circuit, and blower control circuit.

4.2.2.1.2 Alignment Mode. In the alignment mode, the stable member of the IMU is alignment using flight crew inputs via the DSKY to initiate the following:

a) CDU Zero - The zero mode sets the gimbal angle registers in the CDU and LGC to zero. These gimbal angle registers accept gimbal angle information in the incremental format. The signal initiating this mode is also sent to the AGS.

b) Coarse Align Mode - The coarse align mode is used for the initial coarse alignment of the IMU. The stable member of the IMU is coarse aligned by torquing the gimbal torquers with CDU error signals derived from LGC specified gimbal angles.

c) Fine Align Mode - In the fine align mode, the IMU stable member is positioned to the required inertial orientation. The LGC transforms star line-of-sight information into the required inertial reference frame. The difference between the required and the existing coarse aligned reference frame is used to compute the necessary gyro torquing signals to position the stable member to the desired orientation.

4.2.2.1.3 LGC Control Mode

a) Power Off - In this mode power is not applied to the computer.

b) Standby - In this mode data from higher mode of operation are retained in memory section.

c) Power On - Once the LGC is initialized, the power is not removed, but the standby mode is used for retention of stored data.

In the power on mode the Computer Subsystem (CSS) maintains vehicle control by means of one of the following submodes:

1) Manual Submode - In the manual submode the CSS generates vehicle rotation and translation commands as directed by the crew through displacement of the hand controllers.

2) Automatic Submode - In the automatic submode the CSS generates and issues all required commands to effect thrust vector control and vehicle stabilization and control.

4.2.2.2 Radar System Modes

The radar systems have basic modes of operation which correspond to the rendezvous radar and the landing radar. The rendezvous radar has related modes functioning in conjunction with the transponder located on the CSM.

4.2.2.2.1 Rendezvous Radar Modes

a) Designate Mode - The rendezvous radar antenna is moved as directed by either manual or LGC command.

b) Search Mode - The rendezvous radar is capable of receiving LGC commands to sweep out a prescribed search volume until the transponder is acquired.

c) Cooperative Track Mode - The rendezvous radar automatically tracks its cooperative transponder when it is located within the limits of the radar-transponder capability.

4.2.2.2.1.1 RR Cooperative Transponder Operating Modes

a) Beacon - The transponder provides a signal to assist acquisition of the signal from the RR.

b) Transponder - In this mode, the transponder receives the signal from the RR and retransmits a coherent signal that is frequency offset from the RR transmitted frequency.

4.2.2.2.2 Landing Radar Mode. The landing radar has a single automatic mode of operation. Once landing radar is initiated, it continues operation until turned off.

4.2.2.3 Abort Guidance Section Modes

The description of AGS operational functions are best classified as AGS conditions, modes, and submodes. All AGS conditions, modes, and submodes of operation, except for unpowered condition, off condition, standby condition, and the operate condition, shall be selectable via the DEDA keyboard. Inertial reference submodes are under AEA control and also may be selected by switch settings or DEDA operations. Selection of the unpowered, off, standby, and operate conditions shall be performed via the AGS status switch and the ASA and AEA circuit breakers as explained below.

The AGS is defined to be in the unpowered condition when both the ASA and AEA circuit breakers are open; the AGS is defined to be in the off condition when the AGS status switch is in the OFF position; and the ASA circuit breaker is closed. The AGS is defined to be in the standby condition when the AGS status switch is in the STANDBY position and the ASA and AEA circuit breakers are both closed. The AGS is defined to be in the operate condition when the AGS status switch is in the OPERATE position and the ASA and AEA circuit breakers are both closed.

a) Off Condition - In the off condition, the only function performed by the AGS is to apply excitation for protective heating of the ASA and inhibit the ASA standby condition.

b) Standby Condition - During the standby condition the AGS:

   1) Provides excitation and power for all subassemblies within the ASA.

   2) Provides "degraded operation" after a 10-minute period. All of the period may be spent in the standby condition, or five minutes of it may be spent in the warmup condition.

   3) Provide the capability of entering the AGS alignment mode after a 40-minute period. If 25 minutes or more are spent in the warmup condition, an additional 25 minutes shall be spent in the standby condition before the AGS alignment submode can be initiated.

   4) Provides the capability of accepting PGNCS alignment information twenty seconds after entering the standby condition.

c) Operate Condition - The operate condition has an alignment mode and an inertial reference mode which are defined as follows: (See Figure 4.2.2.3.-1. for the submode flow chart.)

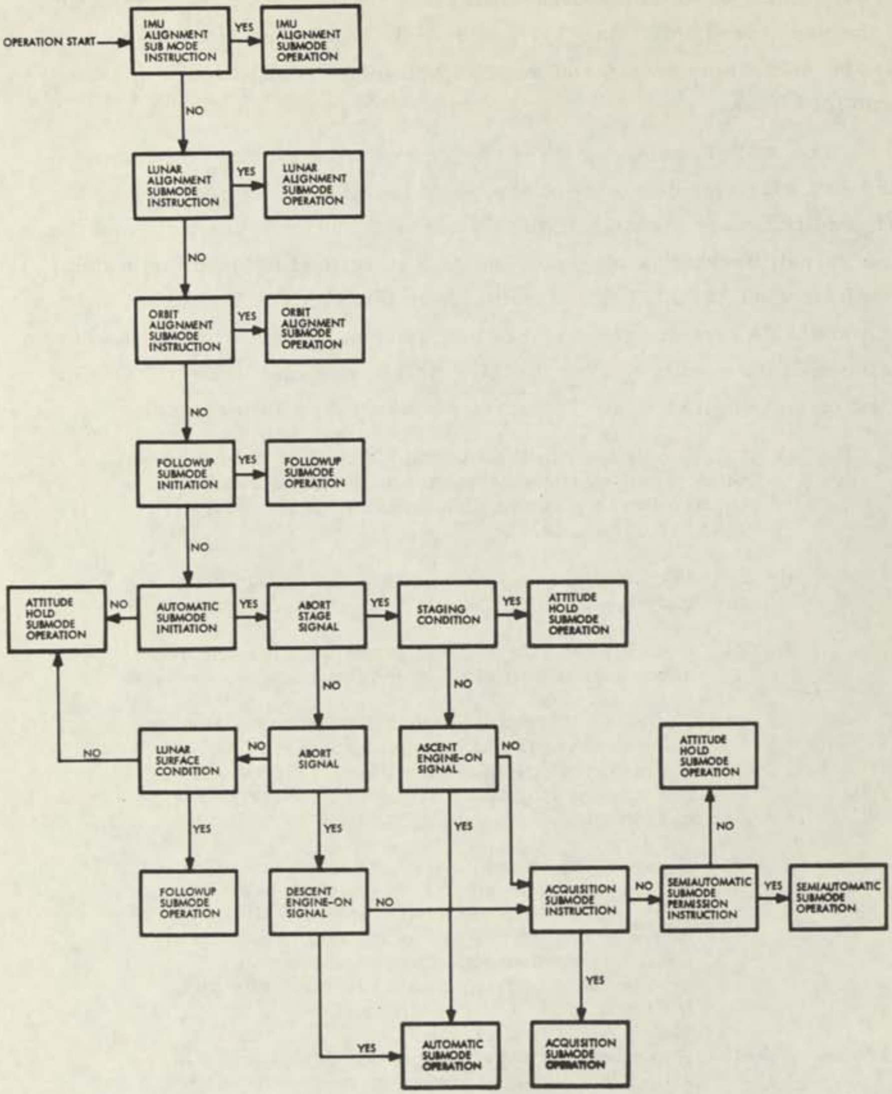


Figure 4.2.2.3-1. Submode Flow Chart

1) Alignment Mode - During this mode of operation, the AGS is capable of aligning to any inertial reference frame within a 3-minute period as determined by the alignment submode scheme selected via the DEDA. The alignment submodes are as follows:

   a) IMU Align - When AGS alignment is to be accomplished via the PGNCS, the IMU alignment submode is used. In this submode the AGS aligns to the inertial reference frame as determined by the PGNCS incremental alignment information.

   b) Lunar Align - When AGS alignment is accomplished on the lunar surface and the PGNCS is inoperative, the lunar alignment submode shall be used. In this submode, the AGS uses the local vertical and the CSM orbital plane as the inertial reference. The alignment is accomplished as follows:

      1. The ASA accelerometers are used to sense local vertical.

      2. The value of the vehicle azimuth angle with respect to CSM orbit plane at lunar touchdown is determined by one of the following methods:

         - If no IMU/CDU failure has occurred, IMU ALIGN is entered and the value of azimuth is stored in the AEA.

         - If an IMU/CDU failure does occur prior to touchdown, the azimuth information stored in the AEA is used. The instruction to store the information is entered through the DEDA at the time it is expected to be stored.

           A correction factor is added to compensate for the effects of lunar rotation during the lunar stay. This correction factor ($\Delta\sigma$) is supplied via the voice link and inserted using the DEDA, where $\Delta\sigma$ is the change in azimuths as a result of lunar rotation and CSM plane change.

- Body Axis Align - When AGS alignment is accomplished in orbit and no external information is available, body axis align submode is used. This submode is initiated by using the DEDA to apply an external instruction. The AGS aligns its inertial reference frame to the body reference axes as defined by the ASA input axes. During this submode, the vehicle motion is restricted to the following:

  - Vehicle limit cycling as indicated in Table 4.2.2.3-1.

  - Vehicle slewing rates up to 2 degrees per second.

2) Inertial Reference Mode - When the AGS is in the Inertial Reference Mode, it provides attitude error information and engine on/off commands which are used for vehicle stabilization and for directing vehicle abort. When in the Inertial Reference Mode, the AGS shall be in one of the following submodes:

   a. Follow-up Submode - While in the Follow-up Submode, the AGS performs the following functions:

      1. Maintain an inertial reference frame and outputs total attitude signals for display.

      2. Maintain the Attitude error signals to the CES at zero unless the display steering commands in follow-up and auto signals are present. In this case, error signals will be available on the FDAI for monitoring by the astronaut, but will not be used by CES.

      3. Use accelerometer inputs from the ASA to calculate LM vehicle present position and velocity in the inertial reference frame.

      4. Solve the LM abort guidance problem using an explicit guidance scheme, in accordance with the guidance option selection.

      5. Accept initializing information either from the PGNCS downlink or the DEDA.

Table 4.2.2.3-1. Body Axes Attitude Dynamics (Initial LM Weight 32,579 lbs)

	Axis	Separation	Start of Powered Descent	Hover	Liftoff	Docking
Limit Cycle Period ± 0.3-degree deadzone	t_p	62.7 sec	62 sec	35.4 sec	17.4 sec*	8.1 sec
	t_q	127	1.5-125	1.5-79	16.9	13.5
	t_r	127	1.5-125	1.5-89	28	8.2
Limit Cycle Period ± 5-degree deadzone	t_p	1050 sec				135 sec
	t_q	2110				223
	t_r	2110				137
Angular Acceleration Capabilities	$\dot{p}$	5.1 deg/deg^2				38.4 deg/sec^2
	$\dot{q}$	2.53				24
	$\dot{r}$	2.53				39
Attitude Rate Response Time for small (less than 0.5°/sec) commands 95% steady state)	T_p	0.2 sec				0.09 sec
	T_q	0.4				0.16
	T_r	0.4				0.10
Attitude Rate Response Time for large (e.g., 10 deg/sec) commands (95% Steady State)	T_p	1.86 sec				0.25 sec
	T_q	3.76				0.4
	T_r	3.76				0.24
Attitude Hold Time Constant	T_p	1.43 sec				0.36 sec
	T_q	1.34				0.33
	T_r	1.34				0.36

*Applies for no moment unbalance. Moment unbalances during powered ascent can cause limit cycle operation with a maximum frequency of 3.26 cps.

Table 4.2.2.3-1. Body Axes Attitude Dynamics (Initial LM Weight 32,579 lbs) (Continued)

	Axis	Separation	Start of Powered Descent	Hover	Liftoff	Docking
Attitude Rate Limits, (Upper)	Manual Control		All axes: 21 deg/sec ± 1 deg/sec			
	Automatic Control		Roll and Yaw: 5 deg/sec ± 5%, Pitch: 10 deg/sec ± 5%			

6. Command Engine-on if either the Ascent Engine-on signal or the Descent Engine-on signal are present and command Engine-off otherwise.

b. Attitude Hold Submode - The AGS is in the attitude hold submode conditions of Figure 4.2.2.3-1 are satisfied. In this submode the AGS outputs attitude error signals that will maintain the vehicle attitude which existed upon entering this submode. The attitude hold submode is entered in the absence of the other inertial reference submodes and at the following times during an abort operation:

   1. During staging as signaled by the abort stage signal and thereafter for a prescribed time between 1 and 10 seconds after receipt of the Ascent Engine-on signal.

   2. At liftoff from the lunar surface and thereafter for a prescribed time between 1 and 10 seconds after receipt of the Ascent Engine-on signal.

   3. Upon computer recognition of velocity to be gained threshold prior to termination of AGS controlled powered maneuvers.

   During the Attitude Hold submode, the AGS performs the functions detailed in steps 1, 3, 4, 5, and 6 under Follow-up Submode, except during and at liftoff when engine-on is commanded.

c. Automatic Submode - The AGS is in the automatic submode when the automatic submode conditions of Figure 4.2.2.3-1 are satisfied. In this submode the AGS outputs attitude error signals and engine commands that are necessary to guide the vehicle during the burns specified in the coelliptic flight plan. During this submode the AGS also performs the functions detailed in Steps 1, 3, 4 and 5 under Follow-up Submode.

d. Semiautomatic Submode - The AGS is in this submode when the semiautomatic submode conditions of Figure 4.2.2.3-1 are satisfied. When in this submode, the AGS generates attitude error signals to reorient the LM to the desired thrust direction. An engine-on command will be generated on completion of this maneuver provided ullage has been detected. During the semi-automatic submode, the AGS also performs the functions detailed in steps 1, 3, and 4 under Follow-up Submode.

e. CSM Acquisition Submode - The AGS is in this submode when the CSM acquisition submode conditions of Figure 4.2.2.3-1 are satisfied. During this submode, the AGS orients the LM vehicle Z-axis toward the AGS computed direction of the CSM. During this submode, the AGS also performs the functions detailed in Steps 1, 3, and 4 under Follow-up Submode. The X-axis is directly parallel to the CSM orbit plane.

f. Calibration Mode - The requirements of this mode are that it provides the capability for in-flight gyro drift and accelerometer bias calibration and lunar pre-launch gyro drift calibration. The calibration mode provides the following functions:

1. In-flight calibration/IMU align
2. Lunar surface calibration
3. Earth prelaunch calibration

4.2.2.4 Control Electronics System Modes

Three modes of operation are described in the control electronics area, two pertaining to PGNCS functions related to the CES, and one pertaining to AGS functions related to the CES. The first mode described pertains to thrust control mode (1S11) with the PGNCS operating in either automatic or manual mode. The remaining modes described pertain to automatic or attitude hold mode in either PGNCS control or AGS control.

4.2.2.4.1 Thrust Control Mode. During operation of descent engine, both manual and automatic thrust control signals are used to throttle the engine within 10- to 100-percent normalized range of operation. Both manual and automatic throttle signals generate engine throttle commands.

a) Automatic Control - Automatic throttle signals from the PGNCS will be processed in the CES and directed to the descent engine. Manual interruption of this path by means of controls and display switching will permit either crew member to fly manual thrust control by actuation of the translation controllers. Manual control inputs are summed with the LGC commands to determine total thrust commanded. Circuit design prevents the LGC from closing the thrust setting of less than that represented by the manual setting.

b) Manual Control - Throttle control signals proportional to the displacement of the X-axis or the thrust/translation controller shall generate descent engine throttling commands. Selection of the controller providing descent engine thrust commands shall be a crew function via displays and controls subsystem switching.

4.2.2.4.2 PGNCS Control. To execute the necessary guidance of the LM, the PGNCS shall provide for vehicle stabilization and control when operating in conjunction with the SCS in both automatic and attitude hold modes. The characteristics of the control signals provided to the SCS will be such as to permit spacecraft attitude control with adequate performance characteristics during either coasting or powered flight under either automatic or manual thrust control modes. In the attitude hold mode, the PGNCS accepts and executes manual rotation and translation command signals from the flight crew control. Attitude control is provided by individual RCS thruster commands and descent engine trim commands. In addition, the PGNCS provides descent engine throttling and engine on-off commands. Detailed functional requirements for the automatic and attitude hold modes are given below:

4.2.2.4.2.1 Automatic Mode

a) During thrusting phases, the PGNCS provides automatic command signals to the SCS to control thrust vector direction by commanding vehicle attitude changes. These control signals are in the form of RCS thruster commands and descent engine gimbal trim commands.

b) Automatic translation command capability is provided by the LGC along all three axes. Translation along the X-axis is performed using all four available thrusters.

c) The logic for firing operation contains provisions for use of only those thrusters providing force parallel to the X-axis in a "+X" direction during powered axcent and during powered abort using the ascent engine through selection of the proper jets. Automatic inhibit of the unbalanced couple restriction shall be provided when desired control torque exceeds the torque capability in the unbalanced couple mode.

d) This mode of yaw control overrides automatic attitude control about the X-axis during the powered descent maneuver. Proportioned yaw commands generated by crew action of the X-axis attitude controller are used to generate yaw rate commands as in the attitude hold mode. Automatic mode operation continues for Y- and Z-axis attitude control.

e) The LGC shall provide the SCS with automatic throttling signals representing positive and negative incremental changes in thrust. The incremental commands are summed and converted in the SCS to a form representing an automatic component of total thrust command. The automatic component is then summed with a manual component coming from the throttle controller. The throttle controller normally stays at a 10-percent thrust position but may be used to override the automatic thrust command in the positive direction.

f) The LGC provides engine ON-OFF commands at the beginning and end of all powered phases except lunar touchdown. At lunar touchdown the descent engine is shut off as a function of external touchdown sensors. The engine ON-OFF signals are directed to the SCS where they are processed and sent to the descent or ascent engines. In providing the thruster ON and OFF signals, the LGC provides for the ullage required for inflight thrusting and accounts for the effects of tailoff at shutdown.

g) The PGNCS computes and displays on the DSKY, the location of the landing site to which the PGNCS is automatically guiding the vehicle during the final approach phase of the powered descent. The PGNCS descent shall accept and process incremental attitude change commands. The number displayed on the DSKY for the landing site location and the markings on the landing point designator are compatible.

4.2.2.4.2.2 Attitude Hold Mode

a) Capability is provided to accept crew commands of vehicle attitude rates proportional to attitude controller displacement. When the controller is returned to the detent position, the vehicle attitude is maintained at the attitude existing at the time attitude rate reduces below a specified threshold. When large attitude changes are required, four jet couples are automatically commanded about the affected axis. The PGNCS provides the command signals to the SCS to perform the maneuvers and maintain the changed attitude. These command signals are in the same form as that specified for the automatic mode.

b) Capability is provided to accept crew commands for linear translational acceleration of the vehicle through on/off firing of the RCS thrusters via the LGC by means of the translation controller. This mode produces a two-jet response along all three axes.

c) Upon switching to the attitude hold mode, the vehicle rate of descent along the nominal landing site local vertical is maintained by means of incremental thrust commands issued from the LGC to the SCS. Upon receipt of momentary discretes from a control switch at the crew station, the LGC commands incremental changes in rate of descent in the positive and negative direction.

4.2.2.4.3 CES Control. When the abort guidance and control path is in operation, the CES provides stabilization and control using the following basic modes:

4.2.2.4.3.1 Automatic Mode

a) In the operation of this control mode, attitude steering errors are directed from the AGS to the CES where the appropriate RCS jet and descent engine gimbal commands are generated to achieve stable vehicle control response to the commanded maneuver. The steering signals are also directed to error needles on the control panel for monitoring by the crew. A wide and narrow dead zone capability is provided that is manually selectable from the displays and controls subsystem during coasting phases. An override of the automatic attitude control function is available in each of the three axes by moving the attitude controller to the hardover position.

When thus activated, it commands four jet couples directly via the secondary coils of the RCS thrusters. In addition, a two jet direct mode is available which is enabled on a per axis basis at the crew station. When in the two jet direct mode, automatic attitude control about the affected axis is disabled and the two jet couples are activated by switches in the attitude controller directly energizing the RCS thruster secondary coil. The switches are activated when the controller is displaced by 2-1/2 degrees.

b) A pulse mode, which is activated on a per axis basis by a switch at the crew station, is available. When the pulse mode is activated, automatic attitude control is disabled about the affected axis and a fixed train of torque impulses is commanded by means of the attitude controller when displaced 2-1/2 degrees.

c) The normal operation of the RCS jets for attitude control during operation of the ascent engine is such that jets providing force in the negative X-axis direction will not be fired.

d) Engine ON/OFF signals from the AGS shall be directed to sequencer logic circuitry of the stabilization and control (S&C) panel where they are combined with status signals regarding engine arm, engine firing, and abort conditions. Output signals are directed to latching relay devices for operating either the ascent or descent engine (depending on the phase of the mission) to start or stop the engine.

e) Pitch and roll gimbal trim signals shall be generated within the CES to assure that the descent engine thrust vector operates through the vehicle center of gravity and to trim out steady state attitude errors.

4.2.2.4.3.2 Attitude Hold Mode

a) The CES operates as a closed loop rate command servo for vehicle attitude maneuvering and has attitude hold capability in conjunction with the AGS during non-maneuvering periods.

b) Attitude error signals from the AGS computer are used to maintain present vehicle attitude within limit cycle regions when the attitude controller is in the neutral or detent position. When the controller is moved out of detent, the AGS is placed in follow-up about all vehicle axes and an attitude angular rate is commanded which is proportional to the controller displacement. Upon return of the attitude controller to the neutral position, the attitude error signals are reestablished from the AGS to control the vehicle to the attitude existing at the time of controller return to the detent position. Capability for wide and narrow deadband operation is provided in a manner identical to the automatic mode.

c) Four-jet direct, two-jet direct, and pulse modes are available; however, only four-jet provides for override of the attitude hold mode in a manner identical to the automatic mode. An open loop acceleration command system is available for translation by means of the reaction jets.

d) During thrusting periods of the descent engine, translations of the vehicle in the Y and Z directions is commanded by displacement of the translation controller in the appropriate directions, while throttling of the descent engine is accomplished by controller displacements in the X direction. During all other periods, translation controller displacement commands positive or negative translation in the X, Y, and Z vehicle directions. Crew selection of either two- or four-jet translation along the X-axis is provided by switches in the displays and controls subsystem.

e) The translation signlas are directed to jet select logic circuitry of the CES and are capable of commanding independent and simultaneous translation along the vehicle X, Y, and Z control axes in either a positive or negative direction by means of ON-OFF operation of the RCS jet engines. A manual +X axis translation capability is provided by means of a switch in the displays and controls subsystem that activates four-jet operation via the RCS secondary coils.

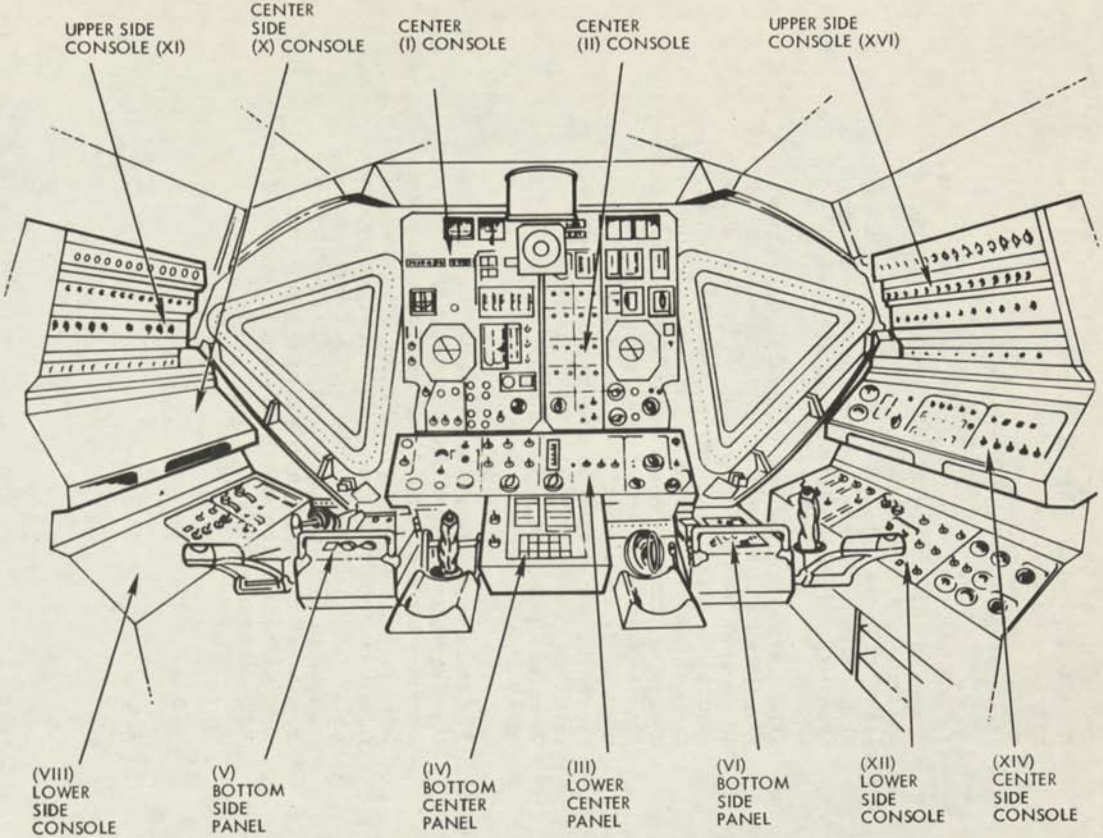


Figure 4.2.3-1. LM Cabin - Main Display Console

4.2.3 Displays and Controls, Functional Description (LM-4 Configuration).

(The following information is extracted from GAEC document LED 480-1E entitled "Functional Requirements for LEM Subsystems Controls and Displays." Only those portions of this document relating to guidance control and navigation are listed here. For information regarding controls and displays for other systems and subsystems refer to LED 480-1E and all subsequent revisions.)

The PGNCS controls and displays are shown in their respective locations within the LM cabin in Figure 4.2.3-1. A brief description of the function of each is contained in the balance of Section 4.2.3. Certain identifiable assemblies and sub-assemblies are treated in more detail in Section 4.3. A listing of the contents of Section 4.3 will be found in the index at the beginning of this book.

4.2.3.1 Engine/Thrust Control (See Figure 4.2.3-2.)

Code	Control/Indicator	Function
1-S-1	ENG ARM switch 3-position Toggle Lock-lock-lock	This three-position toggle (ASC-OFF-DES) provides arming signals to the engines. The ASC position provides an arming signal enabling firing of the ascent engine while simultaneously signalling the LGC that the engine is armed. In the OFF position the arming signals are removed from the engine valves and the LGC. In the DES position the descent engine is armed and the LGC is notified that the engine is armed. However, regardless of the position of this switch, the appropriate engine will be armed if 1-S-13 or 1-S-14 is actuated.
1-S-2	Engine START switch Guarded Pushbutton Maintain	This pushbutton, located at the Commander's station, provides a manual start capability for immediate firing of either the descent or ascent engine, depending upon the position of the 1-S-1. Once actuated, the engine fire command continues and the button is illuminated red. Activation of

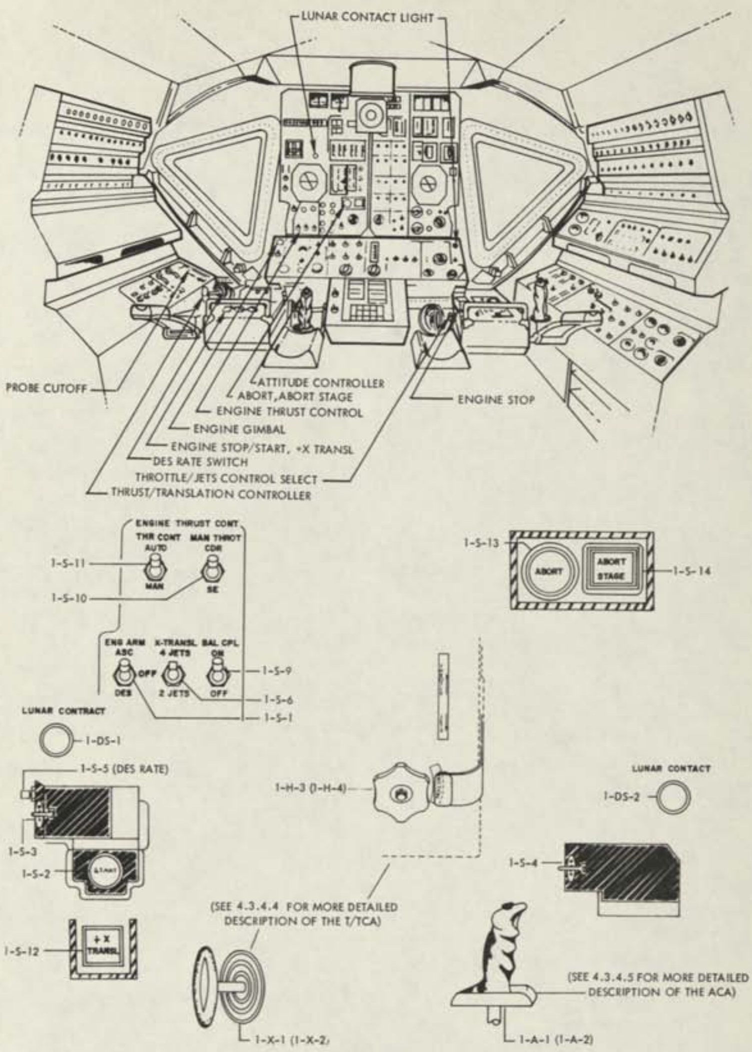


Figure 4.2.3-2. Engine/Thrust Control

Code	Control/Indicator	Function
		1-S-3 or 1-S-4 interrupts the override function and extinguishes the light.
1-S-3 1-S-4	Engine STOP switch (CDR) Engine STOP switch (SE) Guarded Push-buttons Maintain	One pushbutton (1-S-3) is located at the Commander's station and one (1-S-4) is located at the Systems Engineer's station. Each pushbutton sends a discrete stop signal to the descent and ascent engines independent of the position of 1-S-1. On a lunar landing mission (LM-4 and subsequent) it will be depressed on lunar landing after the lunar contact lights (1-DS-1 and 1-DS-2) are illuminated. When actuated, the button remains depressed and is illuminated red. It is reset by depressing it a second time.
1-S-5	RATE OF DESCENT switch 3-position Toggle Mom-main-mom	This switch enables the establishment of a vehicle rate of descent under PGNCS authority. Each switch actuation provides a discrete pulse changing the rate of descent by 1 foot per second. A switch actuation in the +X direction decreases the rate of descent by 1 foot per second. A switch actuation in the -X direction increases the rate of descent by 1 foot per second. Vehicle response to these commands is monitored on the Alt Rate meter (9-M-9B) or on DSKY.
1-S-6	X-TRANSL switch 2-position Toggle Main-main	This two-position toggle (4 JETS - two JETS) selects the number of jets that will be used in X-axis translation maneuvers. This control can only be used with the AGS system.
1-S-8	ENG GMBL switch 2-position Toggle Lock-lock	This two-position toggle (ENABLE - OFF) is located at the Commander's station. In the ENABLE position, the probe(s) signal will shut down the descent engine on contact with the lunar surface. In the OFF position, the probe(s) signal will not shut down the descent engine. In either case, manual shutdown of the descent engine through 1-S-3 or 1-S-4 is not affected.

Code	Control/Indicator	Function
1-S-9	BAL CPL switch 2-position Toggle Lock-lock	This two-position toggle (ON-OFF) selects either balanced pairs of RCS jets in a couple or unbalanced X-axis RCS jets for use in maintaining pitch and roll attitude during the ascent engine thrust phase when the AGS is in the guidance loop. This switch is normally in the ON position (balanced couples) during the initial phases of lunar ascent. This provides maximum stabilization and control over any center of gravity thrust vector misalignment. After some minimum burn time (to be determined) when the balanced couples operation is no longer required, the switch can be positioned to OFF to conserve fuel.
1-S-10	MAN THROT switch 2-position Toggle Lock-lock	This two-position toggle (CDR - SE) selects the controller (1-X-1 or 1-X-2) which will be used for manual descent engine thrust level adjustment if its corresponding select lever (1-H-1 or 1-H-2) is in the THROTTLE position. Thus, in the CDR position, only the Commander's Controller (1-X-1) is enabled to adjust descent engine thrust level, and in the SE position only the Systems Engineer's Controller (1-X-2) is enabled.
1-S-11	THR CONT switch 2-position Toggle Lock-lock	This two position toggle (AUTO-MAN) provides the capability for switching from automatic throttle control (LGC) to manual throttle control (selected T/TCA). In the AUTO position LGC thrust commands are summed with the manual command signals (T/TCA selected by MAN THROTTLE switch 1-S-10). The T/TCA always provides at least a 10 percent command, since it cannot be set below this level. In the AUTO position the THRUST indicator displays the LGC command plus a 10 percent fixed bias which is summed with the LGC signal to compensate for this 10 percent fixed command from the T/TCA. Manual throttle commands may be introduced by displacing the active T/TCA. This would cause the displayed percent

Code	Control/Indicator	Function
		THRUST CMD to decrease as only the LGC commands plus the fixed bias are displayed in the AUTO position. In the MAN position, LGC throttle commands to the descent engine are displayed on the THRUST indicator. The THRUST CMD display will indicate a minimum value of 10 percent when power is applied to the meter.
1-S-12	+X TRANS switch Guarded Push-Button Momentary	This pushbutton is located at the Commander's station. When it is held in the depressed position it provides four jet translation in the +X direction by energizing the RCS secondary coils. Upon release of the button, the signal is removed from the coils, terminating the jet firings.
1-S-13	ABORT switch Guarded Push-button Maintain	This pushbutton is actuated to initiate an abort using the descent engine. Actuation of this switch causes the following events to occur at the appropriate times: (a) a command signal is sent to arm the descent engine, (b) a signal is sent via instrumentation to telemetry indicating that the vehicle is preparing for an abort, and (c) a signal is sent to the LGC and AGS to compute and execute the abort trajectory by using the abort program. It is reset by depressing a second time. The indication that the button is set is provided by a yellow collar which is visible in the depressed condition. NOTE: THE ABORT SWITCH DOES NOT PERFORM DESCENT ENGINE PRESSURIZATION.
1-S-14	ABORT STAGE switch Guarded Pushbutton Maintain	This pushbutton is actuated to initiate an abort using only the ascent engine. Actuation of this switch will cause the following events to occur at the appropriate times: (a) a command signal is sent to the electro-explosive devices to pressurize the ascent engine, (b) a signal is sent to the LGC and AGS to compute and execute the abort trajectory using the abort stage program, (c) a signal is sent

Code	Control/Indicator	Function
		via instrumentation to telemetry indicating that the vehicle is preparing to stage for an abort, (d) the descent engine is shut down, and (e) an ASCENT ENGINE ON command is enabled which in turn fires the appropriate electro-explosive devices to initiate vehicle staging. At the same time the LGC will turn on the ascent engine and signal telemetry via the serial downlink that the ascent engine has been started. At some point in the abort stage sequence, electrical power is transferred automatically from descent batteries to ascent batteries. The Abort stage button is reset by depressing a second time. The indication that the button is set is provided by a yellow collar which is visible in the depressed condition.
1-S-15	DES ENG CMD OVRD switch 2-position Toggle Lock-lock	This two-position toggle (ON-OFF) in the ON position provides a 28 VDC alternate power source to the descent engine shut-off valves to prevent engine shutdown should there occur a loss of DECA power (4CB69) as the vehicle enters the deadmans zone. The OFF position interrupts this alternate power path.
1-A-1 1-A-2	ATTITUDE CONTROLLER (CDR) ATTITUDE CONTROLLER (SE)	There are two three-axis ATTITUDE CONTROLLERS; one at the Commander's station, the other at the Systems Engineer's station. Both provide attitude command signals to the vehicle as appropriate for the mode of stabilization and control selected by the attitude mode control (11-S-6) as modified by the GUID CONT (9-S-6) and ATTITUDE CONTROL (11-S-3, 4, 5) switches. The ATTITUDE CONTROLLER may always be thrown to the hardover position to provide four jet firing.
1-X-1 1-X-2	THRUST/ TRANSLATION CONTROLLER (CDR) THRUST/ TRANSLATION	One THRUST/TRANSLATION CONTROLLER is located at the Commander's station and one at the Systems Engineer's station. Each controller always provides the crew

Code	Control/Indicator	Function
	CONTROLLER (SE)	with translation capability along the Y and Z body axes. X-axis translation capability is added to the CDR's or SE's controller when the THROTTLE/ JETS SELECT lever (1-H-1 or 1-H-2) is positioned to JETS. Thrust control of the descent engine is provided to the CDR's controller when 1-H-1 is in the THROTTLE position and 1-S-10 is positioned to CDR. Thrust control of the engine is enabled with the SE's controller when 1-H-2 is in THROTTLE and 1-S-10 is positioned to SE. The force required to move the THRUST TRANSLATION CONTROLLER (1-X-1 or 1-X-2) in the ± direction when the THROTTLE position is determined by the setting of the THROTTLE FRICTION ROTARY CONTROLLER (1-H-3 or 1-H-4).
1-H-1	THROTTLE/JETS CONTROL SELECT Lever (CDR) THROTTLE/JETS CONTROL SELECT Lever (SE) Two 2-position Levers	Each of these two-position levers (THROTTLE-JETS) is associated with its respective THRUST/ TRANSLATION CONTROLLER (1-X-1 or 1-X-2). The THROTTLE position selects the manual descent engine throttling capability of the controller. The JETS position selects the RCS X-axis translation capability. However, if both 1-H-1 and 1-H-2 are placed in the THROTTLE position, only the controller selected on 1-S-10 will provide throttle commands.
1-DS-1 1-DS-2	LUNAR CONTACT light (CDR) LUNAR CONTACT light (SE)	One light is located at the Commander's station and one is located at the Systems Engineer's station. Both illuminate red when the probe(s) on the vehicle's landing gear touch the lunar surface. They provide positive feedback to the crew that the descent engine should be turned off. These lights will extinguish when either 1-S-3 or 1-S-4 is actuated which causes the stop logic to interrupt power to the light.

Code	Control/Indicator	Function
1-H-3	Throttle FRICTION Rotary Controller (CDR)	Each of these rotarys (FRICTION) varies the force required to move the associated THRUST TRANSLATION CONTROLLER (1-X-1 or 1-X-2) in the ±X direction. They are only operative when the associated THROTTLE/JETS control SELECT LEVER (1-H-1 or 1-H-2) is in the THROTTLE position. The force required to move the THRUST TRANSLATION CONTROLLER (1-X-1 or 1-X-2) varies from a minimum of between 1/4 and 1 pound to a maximum of between 2 1/2 and 3 1/2 pounds.
1-H-4	Throttle FRICTION Rotary Controller (SE)	

4.2.3.2 Caution and Warning. (See Figure 4.2.3-3)

Code	Control/Indicator	Function
6-S-2	MASTER ALARM switch Light (CDR)	These two pushbutton indicators, one located on each of the main panels, along with the caution/warning array light(s) 6-DS-1 through 6-DS-40 and the audible tone 6-T-1 are actuated by malfunction conditions, out-of-tolerance signals or by a loss of power to the CEWA. Depression of either switch will extinguish both red lights and terminate the audible tone, but will have no effect on the illuminated annunciator light(s).
6-S-3	MASTER ALARM switch Light (SE)	
6-T-1	AUDIBLE TONE	This tone is initiated in conjunction with the actuation of 6-S-2 and 6-S-3 and the caution/warning array lights. The tone is interrupted with the depression of either of these switch-lights.
6-DS-6	CES AC Section AC Voltage Failure Warning Light (RED)	The CES AC warning light is illuminated if the ac voltage (single phase 28 v 800 cps or three phase 26 v 800 cps) in the control electronics section of the STAB/CONT subsystem is out of tolerance. The light is extinguished when the gyro test control switch (11-S-1) is actuated to either the POS RT or NEG RT position.
6-DS-7	CES DC Section DC Voltage Failure Warning	The CES DC warning light is illuminated if the dc voltage of one or more of the dc power supplies (+15 v,

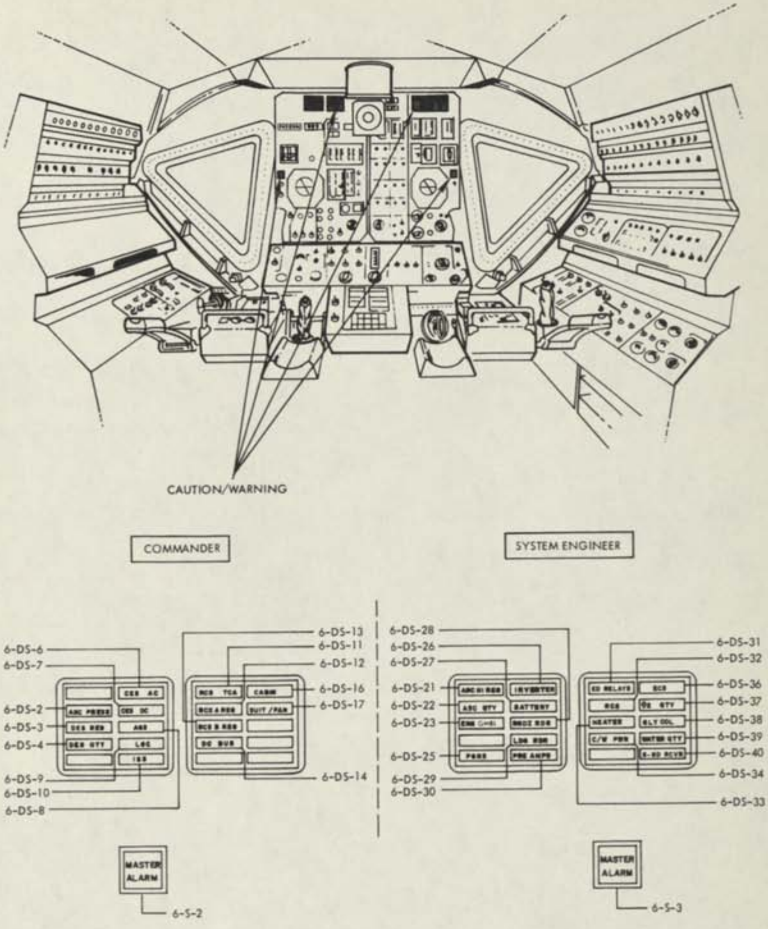


Figure 4.2.3-3. Caution/Warning

Code	Control/Indicator	Function
		-15v, +4.3v, +6v, -6v) in the control electronics section of the STAB/CONT subsystem is out of tolerance. The light is extinguished when 11-S-1 is actuated to either the POS RT or NEG RT position.
6-DS-8	AGS Failure Warning Light (RED)	The AGS warning light is illuminated if: (1) a failure is sensed in any of the following AGS power supplies: +28 v, +12 v, or 28 v 400 cps, or (2) if a fail signal is generated by the AEA test assembly. The signals to illuminate this light are inhibited if the AGS Status switch (12-S-7) is in the OFF position.
6-DS-9	LGC Failure Warning Light (RED)	The (LGC) warning light is illuminated in the event of an LGC power failure, two types of scaler failures, restart or counter failures during LGC operate periods, or in response to an alarm test program. The light is extinguished by either: (1) restoration of nominal voltage or (2) placing the GUID CONT switch (9-S-6) to the AGS position.
6-DS-10	ISS Failure Warning Light (RED)	The (ISS) warning light is illuminated if a failure of any of the following occurs: (1) failure of 3.2 KC power supply, (2) failure of 800-cps 5-percent power supply, (3) failure of PIPA during main engine thrusting, (4) gimbal servo error and tolerances, or (5) failure of the CDU.
6-DS-11	RCS TCA Warning Light (RED)	The RCS TCA warning light is illuminated if there is a primary coil command to fire a specific thruster but no chamber pressure at that thruster. A failure so that opposing colinear jets are on simultaneously illuminates the light. The light is extinguished when the isolation valve associated with the failed TCA is closed.
6-DS-23	ENG GIMBL Failure Caution	The ENG GIMBL caution light is illuminated if a difference between the

Code	Control/Indicator	Function
	Light (YELLOW)	gimbal drive signal and gimbal response from the Y-or Z-axis trim function is sensed. This signal is enabled only if the descent engine is armed and the ENG GMBL switch (1-S-8) is in the ENABLE position. Additionally, the signal is disabled following staging (deadface switch).
6-DS-25	PGNS Failure Caution Light (YELLOW)	The (PGNCS) caution is illuminated if any of the following occur: (1) gimbal lock, (2) LGC restart, (3) IMU temperature out of tolerance, (4) program alarm, or (5) tracker alarm. The light is estinguished either by selecting the AGS position on 9-S-6 or by depressing the DSKY RSET pushbutton once the fail condition has been corrected.
6-DS-26	INVERTER Failure Caution Light (YELLOW)	The INVERTER caution light is illuminated either (1) if the ac bus voltage is < 112 v or (2) if the ac bus frequency is < 398 cps or > 402 cps. The light is extinguished when within-tolerance conditions are restored or if the AC power switch (4-S-14) is turned off.
6-DS-27	BATTERY Failure Caution Light (YELLOW)	The BATTERY caution light is illuminated if an over-current, reverse-current, or over-temperature condition occurs in any of the four descent or two ascent batteries. The light is extinguished when nominal conditions are restored or if the affected battery is turned off.
6-DS-28	RNDZ RDR Data Failure Caution Light (YELLOW)	The RNDZ RDR caution light is illuminated if a rendezvous radar "data-not-good" condition occurs. The light is extinguished when a "data good" condition is restored, or if the rendezvous radar is turned off or if the rendezvous radar mode switch (17-S-3) is placed to any position other than AUTO TRACK. The RNDZ RDR caution is enabled only when the rendezvous radar mode switch is in the AUTO TRACK position.

Code	Control/Indicator	Function
6-DS-29	LDG RDR Failure Caution Light (YELLOW)	The LDG RDR caution light is illuminated if: (1) the landing radar range data are not good, or (2) if the velocity data are not good. This light is not enabled until both range and velocity "data good" signals have been achieved. See drawing LDW-370-28000A, Note 27.
6-DS-30	PRE AMPS Power Failure Caution Light (YELLOW)	The PRE AMPS caution light is illuminated by an out-of-tolerance condition in either of two parallel redundant -4.7-v regulated supplies which power the RCS jet preamplifiers. This light is extinguished either by the stage deadface switch or by actuation of the ABORT pushbutton (1-S-13).
6-DS-33	HEATER Failure Caution Light (YELLOW)	The HEATER caution light is illuminated by an out-of-tolerance temperature condition in: (1) any of the four RCS thruster quads, (2) the S-band antenna electronic drive assembly, (<-65 deg F or >160 deg F) (3) the rendezvous radar assembly, or (4) the landing radar antenna assembly. The light is extinguished by positioning the temperature monitor switch (18-S-10) to the affected heater assembly position.
6-DS-34	C/W PWR Failure Caution Light (YELLOW)	The C/W PWR caution light is illuminated if any one or more of the following regulated power supplies fail: +9 v, +23 v, +4 v, -3 v, -2.3 v, +2.3 v. The light is extinguished only by restoration of nominal power to the CWEA. Unlike the other C/W lights, this light is undimmable.

4.2.3.3 Flight Control (See Figure 4.2.3-4)

9-S-2	RATE/ERR MON switch (CDR) 2-position Toggle Main-main	Each of these two-position toggles (RNDZ RADAR - LDG RDR/CMPTR) enables signal inputs to the FDAI and the X-pointer displays. In the RNDZ RADAR position, shaft and trunnion angles from the rendezvous radar are displayed on the pitch and yaw error needles of the FDAI, and LOS azimuth

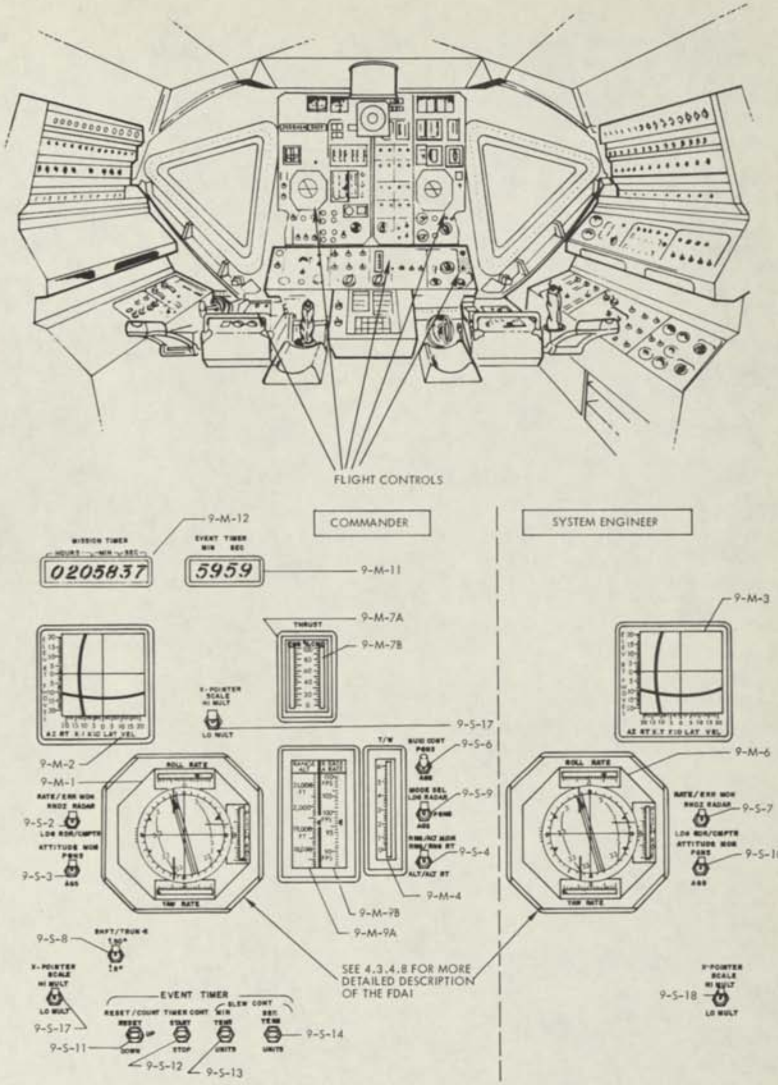


Figure 4.2.3-4. Flight Control

Code	Control/Indicator	Function
9-S-7	RATE/ERR MON switch (SE) 2-position Toggle Main-main	and elevation rates are displayed on the X-pointer display, (illuminating LOS AZ and LOS ELEV). In the LDG RDR/CMPTR position, vehicle attitude errors are displayed on the FDAI error needles, forward and lateral velocities are displayed on the X-pointer display, (illuminating LAT VEL and FWD VEL) 9-S-2 enables inputs to 9-M-1 and 9-M-2, and 9-S-7 enables inputs to 9-M-6 and 9-M-3.
9-S-3	ATTITUDE MON switch (CDR)	Each of these two-position toggles (PGNS -AGS) selects the source of attitude data to be displayed on the FDAI. In the PGNS position vehicle total attitude signals (after conditioning by GASTA) are sent to the FDAI ball, and LGC attitude error signals are sent to the FDAI pitch roll and yaw error needles. In the AGS position the AGS total vehicle attitude signals are sent to the FDAI ball, and AGS attitude error signals are sent to the FDAI pitch, yaw, and roll error needles. 9-S-3 enables signals to 9-M-1, and 9-S-10 enables signals to 9-M-6.
9-S-10	ATTITUDE MON switch (SE) Two 2-position Toggles Main-main	
9-S-4	RNG/ALT MON switch 2-position Toggle Main-main	This two-position toggle (RNG/RNG RT - ALT/ALT RT) selects the display legend on 9-M-9 and rendezvous radar range and range rate data or landing altitude and altitude rate data to be displayed. In the RNG/RNG RT position rendezvous radar range and range rate data are sent to the display. In the ALT/ALT RT position data from whatever source are selected by 9-S-9 will be sent to the display.
9-S-6	GUID CONT switch 2-position Toggle Lock-lock	This two-position toggle (PGNS - AGS) selects either PGNS or AGS for guidance control of the LM.
9-S-8	SHFT/TRUN switch 2-position Toggle Main-main	This two-position toggle (±50 deg ±5 deg) selects the range of rendezvous radar shaft and trunnion angles to be displayed on the pitch and yaw error needles of 9-M-1 and/or 9-M-6

Code	Control/Indicator	Function
		when 9-S-2 and/or 9-S-7 are placed in the RNDZ RADAR position. In the ±50-degree position a full deflection of the FDAI pitch and yaw error needles indicates shaft and trunnion angles of ±50 degrees (or greater). In the ±5-degree position a full deflection of the FDAI pitch and yaw error needles indicates shaft and trunnion angles of ±5 degrees (or greater). Less than full deflection needle positions are linearly proportional to shaft and trunnion angles of less than ±50 degrees or ±5 degrees, respectively.
9-S-9	MODE SEL switch 3-position Toggle Main-main-main	This three-position toggle (LDG RADAR - PGNS-AGS) selects radar or computer data for display on 9-M-2, 9-M-3, 9-M-9A and 9-M-9B. In the LDG RADAR position, radar altitude, altitude rate, and forward, and lateral velocity data are enabled for display. In the PGNS position LGC computed altitude, altitude rate, and forward and lateral velocity data. In the AGS position AGS computed altitude, altitude rate, and lateral velocity data are enabled for display. The data from the source selected by 9-S-9 are displayed on the X-pointers only when 9-S-2 and 9-S-7 are in the LDC RDR/CMPTR position, respectively. When 9-S-2 and 9-S-7 are in the RNDZ RADAR position, no inputs from 9-S-9 are accepted to the X-pointers. Data from the source selected by 9-S-9 displayed on the altitude/range indicator only when 9-S-4 is in the ALT/ALT RT position. When 9-S-4 is in the RNG/RNG RT position, no inputs from 9-S-9 are accepted to the altitude/range indicator, in which case it remains at zero until the START position is again selected.
9-S-11	RESET/COUNT switch 3-position Toggle Mom-main-mom	This three-position toggle (RESET-UP - DOWN) is used to reset the event timer and select the direction in which the timer will count. The

Code	Control/Indicator	Function
		momentary spring loaded RESET position immediately resets 9-M-11 to zero. Zero is maintained until a start pulse is initiated by actuation of 9-S-12. In the UP position the timer will count in an increasing direction to 59 minutes and 59 seconds, return to zero, and begin counting up again. In the DOWN position the timer will count in a decreasing direction from a preset value to 0 minutes and 0 seconds, then return to 59 minutes and 59 seconds, and count down again.
9-S-12	TIMER CONT switch Mom-main-mom	This three-position toggle (START-STOP) has two momentary spring loaded positions. In the START position the event timer will begin counting in a direction selected by 9-S-11. In the STOP position the count is stopped and frozen until either the START position is again selected or the RESET/COUNT switch is actuated.
9-S-13	MIN SLEW CONTROL switch	These toggles are used to set the four digits of 9-M-11. They have two momentary spring loaded positions (TENS-UNITS), which permit resetting each of the four digits independently. Positioning 9-S-13 to the TENS position slews the tens column digit of the minutes pair at the rate of two digits per second as long as the toggle is held in this position. In the UNITS position the unit column digit of the minutes pair is changed at a rate of two digits per second. All digits change in an increasing direction, regardless of the position of 9-S-11. In the center maintain position the slewing function is disabled. 9-S-14 is used in the same manner to set the seconds pair of digits on the events timer indicator.
9-S-14	SEC SLEW CONTROL switch Two 3-position Toggles Mom-main-mom	
9-S-17	X-POINTER SCALE switch (CDR)	These two-position toggles (HI MULT-LO MULT) control the scale range of 9-M-2 and 9-M-3.

Code	Control/Indicator	Function
9-S-18	X-POINTER SCALE switch (SE) Two 2-position Toggles Main-main	When LOS rates are displayed, the HI MULT position will provide a scale of ±20 mrad/second, and the LO MULT position will illuminate the X 1 multiplier to provide a scale of ±2 mrad/second. When horizontal velocities are displayed, the HI MULT position will illuminate the X 10 multiplier to provide a scale of ±200 ft/sec. The LO MULT position will provide a scale of ±20 ft/second. 9-S-17 controls scale range inputs to 9-M-2; and 9-S-18 controls scale range inputs to 9-M-3.
9-S-19	TIMER CONT 3-position Toggle Main-main-mom	This three-position toggle (START-STOP-RESET) is maintained in the START and STOP positions and momentary in the RESET position. The START position enables the clock to count in an upward direction from any number preset by the slew controls (9-S-20, 9-S-21, and 9-S-22) In the STOP position the count is stopped and frozen. Returning to the START position continues the count. The momentary spring loaded RESET position resets all numbers to zero.
9-S-20	SEC SLEW CONTROL	These three-position toggles have two momentary spring loaded positions (TENS-UNITS) which permit the selection of any digits for the mission timer. Positioning 9-S-22 to the TENS position slews the tens column digit of the hours at the rate of two digits per second as long as the toggle is held in this position. The hundreds digit of the hours is also controlled by the TENS position of 9-S-22, changing one digit for every 5 seconds of slew. In the UNITS position the unit column digit of the hours is changed at a rate of two digits per second as long as the toggle is held in this position. The hundreds digit of the hours is also controlled by the TENS position of 9-S-22, changing one digit for every 5 seconds of slew. In the UNITS position the unit column digit of the hours is changed at a rate
9-S-21	MIN SLEW CONTROL	
9-S-22	HOURS SLEW CONTROL	

Code	Control/Indicator	Function
		of two digits per second. All digits change in an increasing direction. In the center position the slewing function is stopped. 9-S-21 and 9-S-22 are used in the same manner to set the minutes and seconds digits and slew at the rate of two digits per second.
9-M-1 9-M-6	FLIGHT DIRECTOR ATTITUDE INDICATOR (CDR) FLIGHT DIRECTOR ATTITUDE INDICATOR (SE)	Each of these indicators displays: (1) vehicle attitude, attitude rates, and attitude errors or (2) vehicle attitude, attitude rates, and rendezvous radar shaft and trunnion angles, depending upon the position of 9-S-2 and 9-S-7. The position of 9-S-3 and 9-S-10 will select either PGNCS or AGS as the source of displayed vehicle attitude and attitude errors for 9-M-1 and 9-M-6, respectively. The attitude rate information displayed on the three meters surrounding the ball is always obtained from the CES rate gyros. The shaft and trunnion angles are displayed, respectively, on the pitch and yaw error needles of 9-M-1 and/or 9-M-6 when 9-S-2 and/or 9-S-7 is in the RNDZ RADAR position.
9-M-2 9-M-3	X-POINTER (CDR) X-POINTER (SE)	These indicators display: (1) forward and lateral velocities, (2) lateral velocities only, or (3) rendezvous radar LOS elevation and azimuth angular rates. What is displayed on 9-M-2 is a function of the position of 9-S-2 and 9-S-9, and what is displayed on 9-M-3 is a function of the position of 9-S-7 and 9-S-9. Forward and lateral velocities are measured in the horizontal plane along the LM Z-and Y-body axes when the source driving the display is the PGNCS. When the landing radar is the driving source, the forward and lateral velocities are coincident with LM Z-and Y-body axis velocities. This is true only when the radar beams are coincident with the vehicle body axis from hi-gate to touchdown. When the AGS system is the driving source, lateral velocity is the only information displayed and

Code	Control/Indicator	Function
		is vehicle Y-body axis velocity.
9-M-4	THRUST/WEIGHT Indicator	This indicator is a self-contained accelerometer which displays instantaneous X-axis acceleration in lunar g units (1 g = 5.23 ft/sec^2). The instrument may be used to provide a gross check on engine performance, since any given throttle setting will provide a specific acceleration when the vehicle has a given mass.
9-M-7A 9-M-7B	THRUST Indicator	This indicator displays descent engine chamber pressure, which corresponds to thrust (left pointer), and thrust commands to the engine (right pointer). Both scales read from 0 to 100 percent. Either automatic or manual throttle commands can be displayed, depending on the position of the THRUST CONTROL switch (1-S-11). The THRUST CMD indicator (right pointer) will indicate a 10 percent thrust command at all times, even when the engine is not firing. This is a result of the fact that the input is not the actual thrust command input to the engine, unless the engine is on and the T/TCA is at minimum. The actual thrust command to the engine when the GUID CONT switch is set to PGNS, is the sum of the thrust commands from the LGC and thrust commands from the T/TCA. The T/TCA provides a minimum thrust command of 10 percent at all times; it cannot be set to zero. When the THRUST CONTROL switch (1-S-11) is in the AUTO position and the T/TCA is in the minimum position, the LGC commands 10 percent less than required and is summated with the 10 percent command from the T/TCA to provide the required thrust level. When the THRUST CONTROL switch (1-S-11) is in the MAN position, the LGC commands are removed and all thrust commands originate

Code	Control/Indicator	Function
		from the T/TCA.

When the THRUST CONTROL switch (1-S-11) is in the AUTO position, the % THRUST CMD (right pointer) displays only the LGC thrust commands, plus a fixed bias (not the sum of the T/TCA and LGC commands). The fixed bias is to compensate for the absence of the T/TCA command. The bias is a signal equal to a 10 percent thrust command that has been introduced internally in the display. This results in equalizing the percent THRUST CMD and % ENG THRUST displays as long as the T/TCA is in the minimum thrust position. If the T/TCA should be moved from the minimum position, causing an increased T/TCA command level, the LGC would reduce it's command level by a corresponding amount. The engine thrust would remain unchanged. However, the signal displayed on the % THRUST CMD indicator is not the summated LGC and T/TCA; it is the LGC command plus a fixed bias equal to a 10 percent thrust command. Thus, the % THRUST CMD indicator would display a lower reading than the % ENG THRUST.

To facilitate a smooth transition from automatic throttle to manual throttle during a descent engine burn, the T/TCA setting should be gradually increased, causing a decrease in displayed % THRUST CMD (right pointer). The % ENG THRUST (left pointer) should remain unchanged. When the % THRUST CMD indicates 10 percent (further increases would cause the % ENG THRUST to begin to increase), full manual throttle authority has been established. When the THRUST CONTROL switch (1-S-11) is placed to MAN, both pointers will line-up, if the system responds properly. When the THRUST CONTROL switch (1-S-11) is in MAN position, manual throttle commands from

Code	Control/Indicator	Function
		either the CDR's T/TCA or SE's T/TCA will be displayed on the % THRUST CMD indicator, depending on the position of the MAN THROTTLE switch (1-S-10). This display has no input after LM has been staged. However, the display may show some fluctuations. This is a result of spurious signals and are to be ignored. When the GUID CONT switch is to AGS during a DPS burn, the LGC command is removed. Therefore, if a smooth transition to manual throttle control is desired, the T/TCA should be increased to the desired value (if conditions permit) before switchover to AGS.
9-M-9A 9-M-9B	ALTITUDE/ RANGE Indicator	This indicator will display either range/range rate information or altitude/altitude rate information, as selected by 9-S-4. The range/range rate data are from the rendezvous radar. The altitude/altitude rate information is from one of three sources: the landing radar, PGNS or AGS as selected by 9-S-9. When landing radar information is selected, true altitude and altitude rate data are available from low-gate to touchdown if the LM X-axis is vertical. When either PGNS or AGS is selected, inertially derived altitude and altitude rate data are available for display. Prior to hover, the most reliable altitude data will be available from PGNS, providing the radar is functioning accurately.
9-M-11	EVENT TIMER Indicator 4-place Digital Display	This four-digit indicator displays time in minutes and seconds. It is capable of counting up or down. The maximum time displayed is 59 minutes, 59 seconds. 9-S-11, 9-S-12, 9-S-13, and 9-S-14 control the operation and settings of this display.
9-M-12	MISSION TIMER	This seven-digit indicator displays time in hours, minutes and seconds. It is only capable of counting up. The maximum time displayed is 999

Code	Control/Indicator	Function
		hours, 59 minutes, and 59 seconds. 9-S-19, 9-S-20, 9-S-21, and 9-S-22 control the operation and settings of this display.

4.2.3.4 Primary Guidance and Navigation (AOT). (See Figure 4.2.3-5)

Code	Control/Indicator	Function
3-H-1	Reticle Control Continuous Rotary	This continuous rotary control rotates the AOT reticle. The AOT COUNTER (3-M-7) provides feedback of the position of the rotary. The rotary is used for the following: (1) to zero the AOT Counter, (2) to superimpose the X line on the target star, and (3) to superimpose the spiral on the target star. The time of acquisition is entered into LGC by depressing the respective MARK button on the AOT and the angle is read off the counter and entered via DSKY.
3-H-2	Azimuth Control 6-position Rotary	This four-position rotary switch (L-F-R-CL) controls a rotating prism in the AOT. Rotating the prism permits the viewer to view any of three 60 deg sectors, Left (L), Forward (F), and Right (R). In the closed (CL) position the rotating prism is stowed under a protective cover. NOTE: The position of the image on the screen will shift as the rotary is switched to a new position. This is normal as the AOT does not provide compensation for the prism movement.
3-H-3	Eyepiece Adjust Lever	This lever moves the eyepiece toward or away from the eye of the observer.
3-H-4	Eyepiece Lock	This lever locks the eyepiece in one of its positions, as selected by the crew.
3-S-23	MARK X switch Momentary Pushbutton	Depression of this pushbutton sends a signal to the LGC indicating that the star is aligned on the X-reticle position.
3-S-24	MARK Y switch Momentary Pushbutton	Depression of this pushbutton sends a signal to the LGC indicating that the star is aligned on the Y-reticle position.

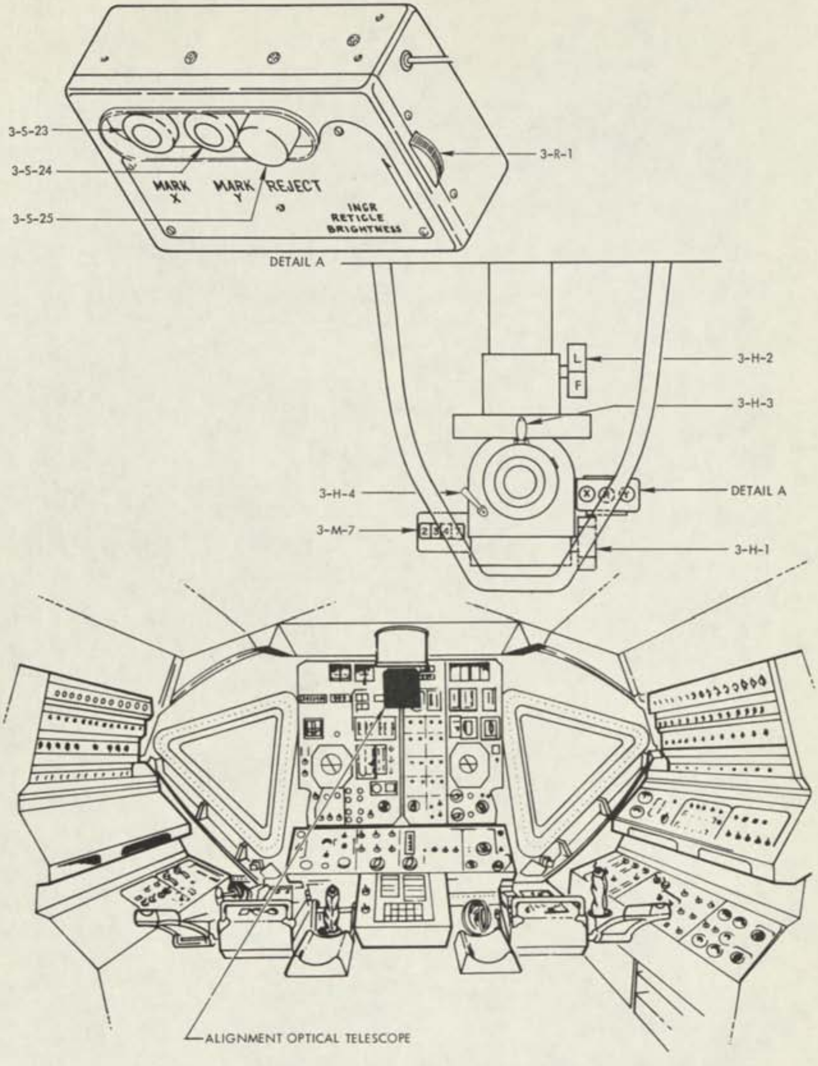


Figure 4.2.3-5. Alignment Optical Telescope

Code	Control/Indicator	Function
3-S-25	REJECT switch Momentary Pushbutton	Depression of this pushbutton sends a signal to the LGC indicating that the last MARK signal was incorrect and no action is to be taken until a new MARK signal is received.
3-R-1	INCR RETICLE BRIGHTNESS Control	This continuous rotary (OFF BRIGHT) controls the brightness of the AOT reticle.
3-M-7	AOT Counter 4-place Digital Display	This four-digit indicator displays, in degrees, the rotation of the reticle control knob. The extreme right digit displays tenths of degrees.
4.2.3.5	Primary Guidance and Navigation (DSKY) (See Figure 4.2.3-6)	
3-S-1	IMU CAGE switch 2-position Mom-lock	Aligns the inner, middle, and outer gimbals to the Y-, Z-, and X-body axes, respectively.
3-S-3	VERB switch Momentary Pushbutton	Pushing the button indicates that the next two numerical characters keyed in are to be interpreted as the verb code.
3-S-4	NOUN switch Momentary Pushbutton	Pushing the button indicates that the next two numerical characters keyed in are to be interpreted as the noun code.
3-S-5	CLR switch Momentary Pushbutton	Used during a data loading sequence to erase the information being loaded into register R1, R2, and R3. It allows the operator to clear the register in the event of an error in loading.
3-S-6	STBY switch Momentary Pushbutton	Depressing STBY when STBY is illuminated will place the computer in the Operate mode, and STBY will extinguish. Depressing STBY when STBY is not illuminated will place the computer in the Standby mode, and STBY will illuminate.
3-S-7	KEY REL switch Momentary Pushbutton	Depressing KEY REL in response to KEY REL (3-DS-7) illumination will permit the LGC to display information on the DSKY. The KEY REL button should be depressed subsequent to inserting information into DSKY.

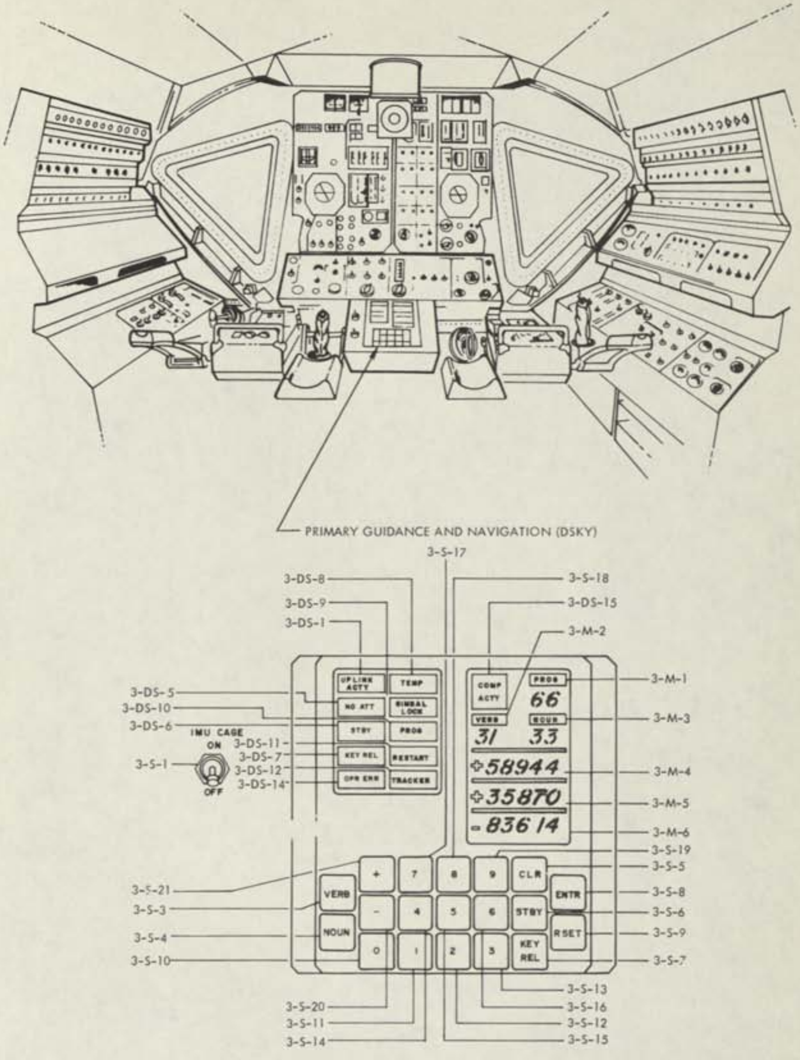


Figure 4.2.3-6. Primary Guidance and Navigation (DSKY)

Code	Control/Indicator	Function
3-S-8	ENTR switch Momentary Pushbutton	(1) To execute the verb/noun displayed on the VERB and NOUN registers. (2) To enter data words loaded into the data register (R1, R2, R3). (3) To request the computer to perform an action.
3-S-9	RSET switch Momentary Pushbutton	Enables turn off of alarm conditions on the DSKY, providing the alarm condition has been corrected.
3-S-10 through 3-S-21	Keyboard switches Momentary Pushbutton	Numerical keys 0 through 9 are self-explanatory, + and - sign keys are used for sign convention and to identify decimal data.
3-M-1	PROG Display	Displays code number 00 to 99 of current program. Note: A program is a block of logic sequences.
3-M-2	VERB Display	Displays code number 00 to 99 of current verb. Note: A verb indicates what action is to be taken.
3-M-3	NOUN Display	Displays code number 00 to 99 of current noun. NOTE: A noun indicates to what an action is applied.
3-M-4 through 3-M-6	DISPLAY (Row 1) (Row 2) (Row 3)	Displays decimal data (outputs from LGC or manually inserted data) plus a plus or minus sign. Octal data are presented without the sign. Every position in each data display need not be filled.
3-DS-1	UPLINK ACTY Status Light (WHITE)	Energized by the first character of a Digital Uplink message received by the LGC. If it is not extinguished by the termination of uplink transmission, it should be extinguished by crew use of the RSET or KEY REL buttons when the uplink transmission is complete.
3-DS-5	NO ATT Status Light (WHITE)	Illuminates to indicate that attitude reference data are not available from the IMU. This condition can result, for example, when the ISS is not

Code	Control/Indicator	Function
		powered or when the IMU is caged. This light illuminates during coarse alignment.
3-DS-6	STBY Status Light (WHITE)	Illuminates when LGC is in a standby mode, extinguishes when the LGC is in the operate mode.
3-DS-7	KEY REL Status Light (WHITE)	Illuminates when an internal sequence attempts to use the DSKY, but finds it busy due to independent operator action (any keyboard depression except RSET or KEY REL).
3-DS-8	TEMP Caution Light (YELLOW)	Illuminates to indicate that the IMU stable member temperature is outside of design limits (126.3 deg F to 134.3 deg F). The PGNS caution light will also illuminate.
3-DS-9	GIMBAL LOCK Caution Light (YELLOW)	Illuminates to indicate the IMU middle gimbal angle (MGA) is in excess of ± 70 degrees from its zero position. The NO ATT light and the PGNS caution light will also illuminate. NOTE: When the MGA exceeds ±85 degrees from its zero position, the LGC automatically commands a coarse alignment.
3-DS-10	PROG Caution Light (YELLOW)	Illuminates to indicate a program alarm condition exists. The PGNS caution light will also illuminate.
3-DS-11	RESTART Caution Light (YELLOW)	Illuminates if internal detection circuiting in LGC signifies a program abort. (A program abort is not to be confused with an LM abort; it is a LGC main alarm condition, which indicates that the program has stopped and LGC is attempting a restart.)
3-DS-12	TRACKER Caution Light (YELLOW)	Illuminates if one of the following conditions exist: (1) Rendezvous Radar CDU failure (when the rendezvous radar is in the LGC mode), (2) A RR "data no good" discrete is present, (3) Inability of the LGC to get coherent RR data, (4) A Landing Radar (LR) "data no good" discrete is present, (5) An ambiguous LR position

Code	Control/Indicator	Function
		status indication, (6) Inability of LGC to obtain LR data.
3-DS-14	OPR ERR Status Light (WHITE)	Illuminates to indicate that the astronaut has performed an improper sequence of key depressions on DSKY.
3-DS-15	COMP ACTY Status Light (WHITE)	Signifies computer activity in processing data.

4.2.3.6 Stabilization and Control (See Figure 4.2.3-7)

Code	Control/Indicator	Function
11-S-1	GYRO TEST control switch 3-position Toggle Mom-lock-mom	This three-position toggle (POS RT - OFF-NEG RT) enables a test signal to be applied to the rate gyro selected by 11-S-7. It is maintained in the OFF position and momentarily held in the other two. When this switch is held in the POS RT position a +5 degree/second attitude rate will be indicated on the appropriate rate needles of the flight director attitude indicators (9-M-1 and 9-M-6). The NEG RT position produces the opposite effect. In OFF, no test signal is produced.
11-S-2	DEADBAND switch 2-position Toggle Main-main	This two-position toggle (MAX - MIN) selects either a large amplitude limit cycle (MAX) for the attitude control system to conserve RCS fuel during coasting flight or a narrow deadband (MIN) for periods when accurate control is required. Regardless of this switch position, the minimum deadband will operate during main engine thrusting. This switch is not functional when the PGNCS is in use.
11-S-3 11-S-4 11-S-5	PITCH ATTITUDE CONTROL switch ROLL ATTITUDE CONTROL switch YAW ATTITUDE CONTROL switch Three 3-position Toggles Main-main-main	The three ATTITUDE CONTROL switches, one for each axis, each having three positions (MODE CONT - PULSE - DIR), are normally in the MODE CONT position while PGNCS is the guidance authority. This allows use of the proportional mode of the attitude controller while in ATT HOLD. When the MODE CONTROL switch (11-S-6) is in AUTO, the LGC supplies attitude commands to maneuver the vehicle. Placing the MODE CONTROL switch to ATT HOLD or AUTO allows use of the DIR mode in

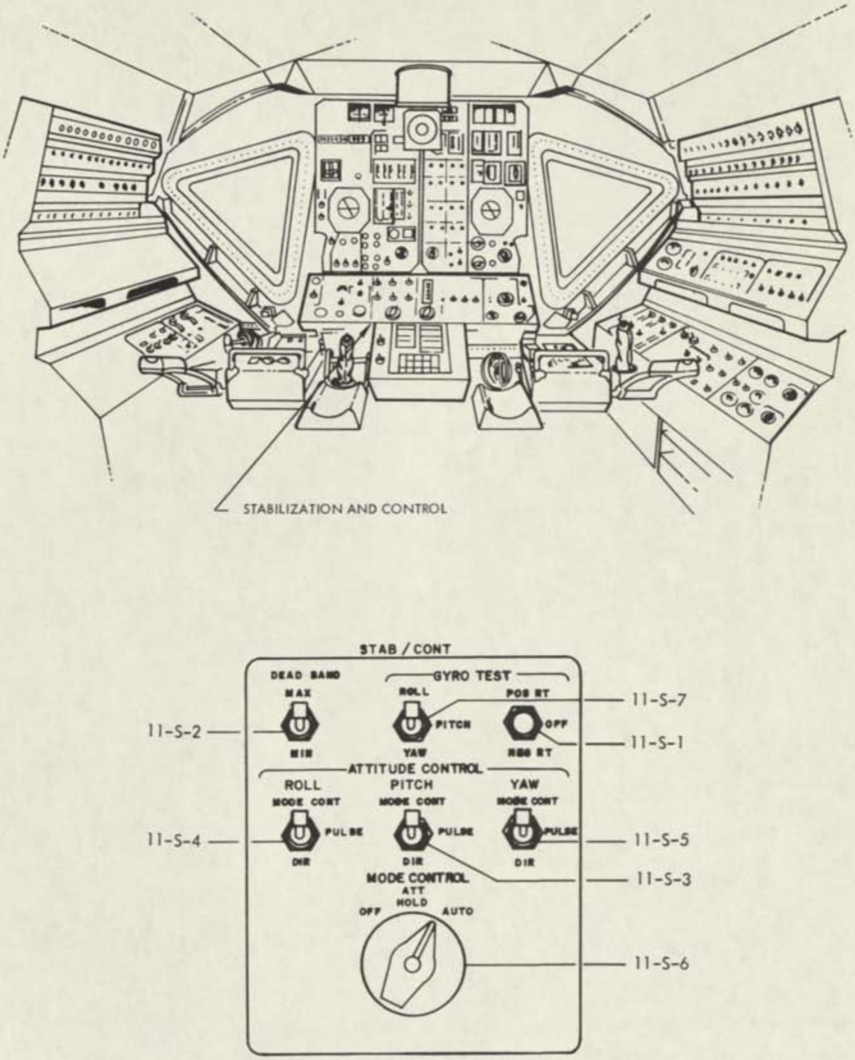


Figure 4.2.3-7. Stabilization and Control

Code	Control/Indicator	Function
		any axis. Each axis must be positioned to DIR independently. Displacing an attitude controller (1-A-1 or 1-A-2) one fourth of full throw commands a two-jet firing in the axis selected. This usage is not recommended, as it results in an open-loop acceleration that is opposed by the DAP above the command rate. The PULSE position is inoperative while under PGNCS guidance authority.
		When AGS is the guidance authority, the MODE CONT positions of the ATTITUDE CONTROL switches function in the same manner as in PGNCS. That is, when AUTO is selected on the MODE CONTROL switch, automatic attitude commands are issued by the guidance authority (PGNCS or AGS). When ATT HOLD is selected on the MODE CONTROL switch, displacing either attitude controller to the out of detent position commands an angular rate proportional to the displacement. Under AGS control, the PULSE and DIR modes differ only in the way the reaction jets fire.
		When PULSE is selected, two jets in the appropriate axis fire at a constant pulsed rate for as long as the attitude controller is displaced more than one fourth of full throw. When DIR is selected, two jets in the appropriate axis fire continuously for as long as the Attitude Controller is displaced more than one fourth of full throw. Both PULSE and DIR provide open-loop acceleration. The rates produced by PULSE or DIR commands must be nulled to zero by inducing, and the attitude is not automatically maintained until the ATTITUDE CONTROL switch is returned to MODE CONT.
		When the MODE CONT switch is in the OFF position, only the DIR positions of 11-S-3, 11-S-4, and 11-S-5 are operative. Selecting DIR produces two-jet open-loop accelerations when an Attitude Controller is displaced more than one fourth of full throw.

Code	Control/Indicator	Function
		This is irrespective of the guidance authority.
11-S-6	MODE CONTROL switch 3-position Rotary Main-main-main	This three-position rotary (OFF - ATT HOLD - AUTO) is associated with the ATTITUDE CONTROL switches (11-S-3, 11-S-4, and 11-S-5) and the GUID CONT switch (9-S-6). Thus, it affects the usage of the attitude controllers (1-A-1 and 1-A-2) and the firing of the RCS jets. When the OFF position is selected, neither guidance authority (as selected on 9-S-6) can command RCS firing to maintain or change attitude. The attitude controller cannot command proportional mode or pulse mode maneuvers. The attitude controllers can command four-jet hard over firing of the RCS, and if DIR is selected on 11-S-3, 11-S-4, or 11-S-5, two jet firing in the appropriate axis. The translation controllers are not enabled. In addition, main propulsion system burns must be manually initiated and shut down. The ATT HOLD position is used to maintain attitude and to allow manual adjustment of attitude to commanded rates, which will then be maintained automatically. Neither automatic attitude or translation maneuvers are provided in ATT HOLD, nor are automatic APS or DPS starts and stops. The attitude is changed by commanding an angular rate proportional to the displacement of an attitude controller (1-A-1 or 1-A-2). This capability exists in both PGNCS and AGS. (a) With AGS as the guidance authority, both PULSE and DIR (as selected on 11-S-3, 11-S-4, or 11-S-5) provide open-loop acceleration when an attitude controller is displaced more than one fourth of full throw. The rate created by a DIR or PULSE mode RCS firing must first be nulled to zero by inducing an opposite acceleration command. Attitude is not automatically maintained

Code	Control/Indicator	Function

until the ATTITUDE CONTROL switch is returned to the MODE CONT position.

(b) With PGNCS as the guidance authority, PULSE is inoperative, and while not recommended, DIR provides an open-loop acceleration. The translation controllers are enabled in both PGNCS and AGS while the MODE CONTROL switch is in ATT HOLD.

In the AUTO position the guidance authority (PGNCS or AGS) issues RCS commands to maintain or change the vehicles attitude, and the PGNCS performs RCS translation. The guidance authority will also stop and start the main propulsion systems, and in the case of the PGNCS, vary the thrust level of the descent engine.

(a) With PGNCS as the guidance authority and MODE CONTROL in AUTO, attitude controller proportional commands are normally ineffective, except for certain LGC determined periods of the mission. Selecting DIR on one or more of the ATTITUDE CONTROL switches (11-S-3, 11-S-4, or 11-S-5) will cause opposing jets to fire when the attitude controllers are displaced more than one fourth of full throw. The LGC will command jets to fire in opposition to both DIR and hard over commands.

(b) With AGS as the guidance authorith, the AUTO mode allows the AEA to initiate attitude maneuvers prior to a main engine burn must be commanded manually. This is also true for AGS controlled aborts that are initiated via the ABORT switch (1-S-12), or the ABORT STAGE switch (1-S-13). Assuming the ullage maneuver has occurred, AGS will

Code	Control/Indicator	Function
		start or stop a main propulsion system burn in AUTO mode. Displacing an attitude controller to the out-of-detent position results in conflicting commands. PULSE and DIR modes produce open-loop acceleration when an attitude controller is displaced more than one fourth of full throw, and drifting while in detent. The hard over mode will cause opposing jets to fire when attitude switches (11-S-3, 4, 5) are in MODE CONT. The translation controllers function normally.
11-S-7	GYRO TEST select switch 3-position Toggle Main-main-main	This three-position toggle (ROLL-PITCH-YAW) associated with 11-S-1, selects the rate gyro which is to be tested. The test output is displayed as described for switch 11-S-1.

4.2.3.7 Radar Control (See Figure 4.2.3-8)

Code	Control/Indicator	Function
17-S-1	SLEW switch Spring Loaded 4-way Center OFF Toggle	This four-way toggle (UP - RIGHT - DOWN - LEFT) slews the trunnion or shaft servos positioning the rendezvous radar antenna for manual acquisition of the target and subsequent automatic tracking. Two antenna slew rates are available through 17-S-2. Target acquisition is determined from 17-M-1 and AUTO TRACK is enabled through 17-S-3. The position of the rendezvous radar antenna (shaft and trunnion) can be determined from either the Commander's or Systems Engineer's FDAI (9-M-1 or 9-M-6) pitch and yaw error needles when the rate/error monitors switches (9-S-2 and 9-S-7) are positioned to RNDZ RADAR.
17-S-2	SLEW RATE switch 2-position Toggle Main-main	This two position toggle (HI-LO) provides two rates for manually slewing the rendezvous radar antenna. The high slew rate is 7 deg/sec and the low slew rate is 1 1/3 deg/sec. The high slew rate is used for quick, coarse adjustments to guide the antenna in the direction of the target. The

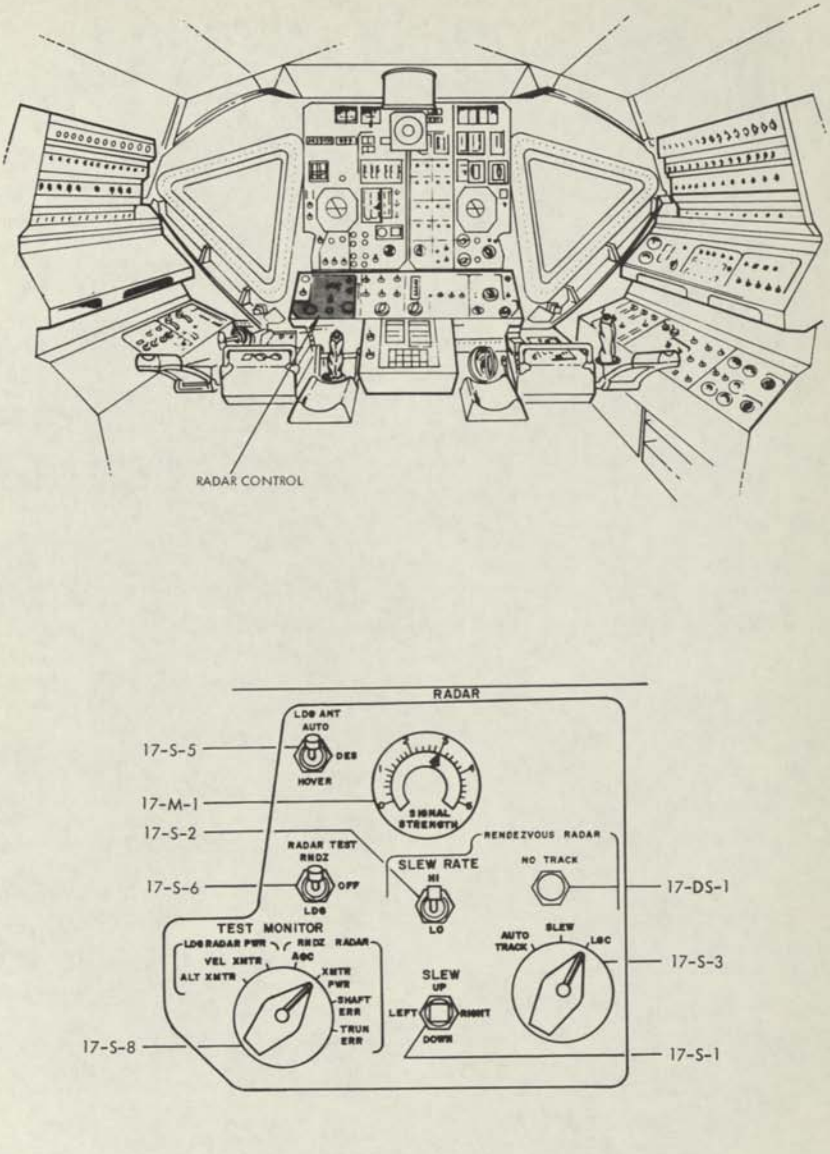


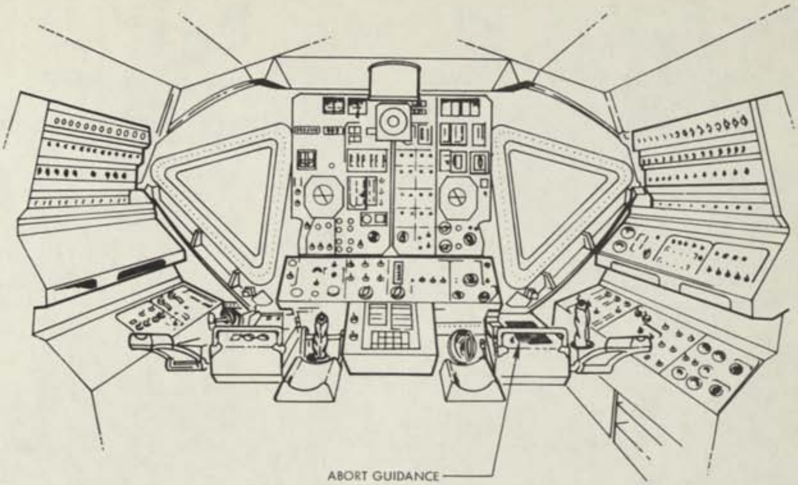
Figure 4.2.3-8. Radar Control

Code	Control/Indicator	Function
		low slew rate is used for slow, fine adjustments to drive the antenna toward maximum signal strength as monitored on 17-M-1 thus enabling the subsequent initiation of AUTO TRACK through 17-S-3.
17-S-3	RNDZ RADAR MODE switch 3-position Rotary	This trhee-position rotary (AUTO TRACK - SLEW - LGC) enables control of the rendezvous radar antenna drive. In the AUTO TRACK position, track is maintained by comparing the received radar signals from the shaft and trunnion channels; the resultant error condition signals are used to drive the antenna servos thus maintaining track. In the SLEW position the rendezvous radar antenna drive is controlled manually from 17-S-1. In the LGC position the computer automatically drives the antenna towards the target. Once the target is acquired the radar will automatically maintain track and the NO-TRACK component caution light (17-DS-1) will extinguish.
17-S-5	LDG ANT switch 3-position Toggle Lock-lock-lock	This three-position toggle (AUTO - DES - HOVER) in the AUTO position enables the LGC to position the landing radar antenna from position 1 (DESCENT) to position 2 (HOVER) at the appropriate mission time. The antenna is in the descent position until hi-gate; the LGC commands it to the HOVER position shortly thereafter. The LGC cannot return the antenna to the descent orientation. The DES position is also used to reposition the antenna after testing.
17-S-6	RADAR TEST switch 3-position Toggle Main-lock-main	This three-position toggle (RNDZ-OFF - LDG) provides signals to the rendezvous and landing radar's test circuitry. In the RNDZ position test circuitry is enabled, and rendezvous radar performance is determined by monitoring 17-M-1, the altitude/ range meter (9-M-9), the FDAI shaft and trunnion needles (9-M-1 and 9-M-6, and the X-pointers (9-M-2 and

Code	Control/Indicator	Function
		9-M-3) after the appropriate positioning of 17-S-8.
17-S-8	TEST/MONITOR switch 6-position Rotary	This six-position rotary (LDG RADAR PWR: ALT XMTR - VEL XMTR; RNDZ RADAR: AGC - XMTR PWR - SHAFT ERR - TRUN ERR) selects either landing or rendezvous radar test data or flight performance data for monitoring on meter 17-M-1. In the ALT XMTR position the altitude transmitter power output is displayed on 17-M-1 during testing and flight operations. In the VEL XMTR position the velocity transmitter power output is displayed on 17-M-1 during testing and flight operations. In the AGC position during testing, a specific value of receiver signal strength will be displayed on 17-M-1; however, during flight operations the signal will vary. In the XMTR PWR position, during flight operations, a specific value of transmitter output power is displayed on 17-M-1. In the SHAFT ERR or TRUN ERR positions, during testing, angle error signals will be displayed as needle fluctuations with equal amplitude on 17-M-1. During flight operations these error signals will be displayed as needle fluctuations which represent the smoothness of servo tracking, and will usually be unequal in amplitude.
17-DS-1	NO TRACK light Component Caution Light	The illumination of this light indicates that the rendezvous radar has broken track (data not good) in any one of the three tracking modes. It functions independently of the position of the RR mode switch (17-S-3).
17-M-1	SIGNAL STRENGTH METER	The Signal Strength meter displays landing radar and rendezvous radar data as selected by 17-S-8 for test and flight operations.

4.2.3.8 Abort Guidance (See Figure 4.2.3-9)

Code	Control/Indicator	Function
12-S-1 to 12-S-12	KEYBOARD switches Momentary Pushbuttons	These pushbuttons enable the crew to set up the necessary values (0 through 9) and signals (+ and -) on the address and data indicators. Depressing the pushbuttons successively fills 12-M-1 and 12-M-2, in that order, from left to right.
12-S-13	CLR switch Momentary Pushbutton	This pushbutton blanks 12-M-1 and 12-M-2 for reuse, and simultaneously cancels the effect of any depressions of 12-S-16.
12-S-14	ENTR switch Momentary Pushbutton	This pushbutton enables the crew to enter data which have been set up on the data indicator into the computer address set up on the address indicator. The indicators are then cleared for immediate reuse when the AEA computer has accepted the data.
12-S-15	READOUT switch Momentary Pushbutton	This pushbutton enables the crew to call up AEA computer data. The data will be displayed on 12-M-2 and will come from the AEA address displayed on 12-M-1. It also cancels the effect of 12-S-16.
12-S-16	HOLD switch Momentary Pushbutton	This pushbutton enables the crew to interrupt the updating of any continuously changing data displayed on 12-M-2. The data will remain unchanged until the readout or clear pushbutton is depressed. Depression of the readout pushbutton enables display of the present value of the data and continuance of updating. Depression of the CLR pushbutton (3-S-5) causes resetting of the HOLD condition as well as clearing of the display. A new display with updating can then be commanded.
12-S-17	AGS STATUS switch 3-position Toggle Lock-lock-lock	This three-position toggle (OPERATE-STANDBY - OFF) enables the crew to place the AGS in any one of three states. In the OFF position all AGS activity is inhibited except for the



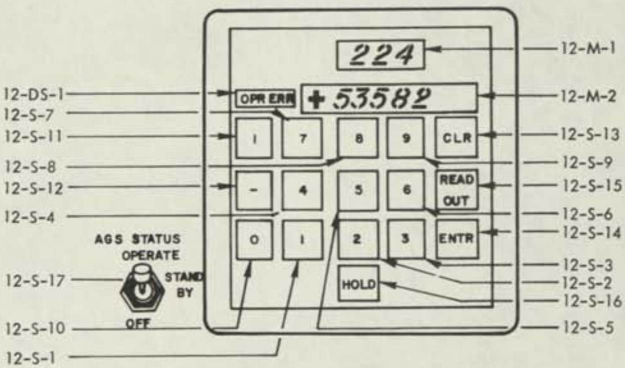


Figure 4.2.3-9. Abort Guidance

Code	Control/Indicator	Function
		Abort Sensor Assembly (ASA) heaters. In STANDBY full activity is enabled in the ASA, and the Abort Electronics Assembly (AEA) is placed in the STANDBY mode. In OPERATE full activity is enabled in the ASA and AEA. Thirty minutes of standby operation is required before OPERATE can be selected. The AEA DC circuit breaker must also be activated to provide power both for the heaters and for all AGS activity.
12-M-1	ADDRESS Indicator	This three-digit indicator displays the abort electronics assembly computer address selected by the crew via the keyboard. Three octal digits must be set up on this indicator by means of the keyboard prior to each ENTR or READOUT pushbutton depression.
12-M-2	DATA Indicator	This sign (+ or -) and five-digit indicator displays data to be entered into or read out from the AEA computer address displayed on 12-M-1. Data to be entered are selected by the crew via the keyboard and entered by depressing 12-S-14. Data selected for readout by the AEA computer address identified on 12-M-1 are displayed if 12-S-15 is depressed. The display updating rate of any changing data selected for readout is limited to two times per second.
12-DS-1	OPR ERR Status Light (WHITE)	This (OPR ERR) status light is illuminated if the crew depresses an incorrect number or sequence of pushbuttons for the operation being performed. If too few pushbuttons have been pressed the indicator illuminates after the ENTER or READOUT buttons have been depressed. The indicator is also illuminated if the plus or minus keys are not pressed first in entering data and if an eight or nine is used in entering an address (which is limited to octal digits). If too many depressions of the buttons have been made, the OPR ERR indicator will

Code	Control/Indicator	Function
		illuminate before the ENTER or READOUT buttons are depressed.

4.2.3.9 ORDEAL (See Figure 4.2.3-10)

Code	Control/Indicator	Function
9-R-1	ALT SET Continuous Rotary	This continuous rotary (10-to 310) nautical miles is used to insert the LM orbital altitude into the ORDEAL.
9-S-23 9-S-24	FDAI 1 switch FDAI 2 switch Main-main	These two-position toggles (ORB RATE - INRTL), when placed in the ORB RATE position, provide a pitch angle input to their respective FDAI, which causes the FDAI to display total attitude with respect to a local horizontal reference frame. When the toggle is placed in the INRTL position, the ORDEAL is bypassed and the FDAI displays total spacecraft attitude with respect to an inertial reference frame.
9-S-25	EARTH-PWR OFF-LUNAR	This three-position toggle (EARTH-PWR OFF-LUNAR) is used in conjunction with 9-R-1 to insert an appropriate scale factor into the ORDEAL electronics. The switch is always set to LUNAR or OFF. In OFF position, ac and dc power are removed from ORDEAL.
9-S-26	LIGHTING switch Main-main-main	This three-position toggle (BRT-OFF-DIM) controls the brightness of the ORDEAL EL panel lighting.
9-S-27	MODE switch Main-main	This two-position toggle (OPR/SLOW-HOLD/FAST) is used in conjunction with 9-S-28 to insert and maintain the proper angular offset into the FDAI's. The HOLD/FAST slew position in conjunction with the UP and DOWN positions permits slewing at an orbital rate x200 rate in the plus or minus direction. This is used as a coarse adjustment.

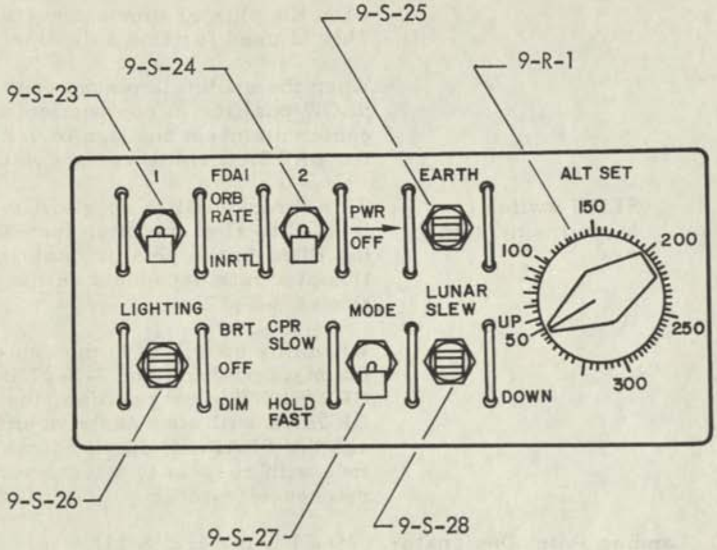


Figure 4.2.3-10. ORDEAL

Code	Control/Indicator	Function
		The OPR/SLOW position in conjunction with the UP and DOWN positions permits slewing at an orbital rate x 10 in the plus or minus direction. This is used for fine adjustment. When the switch is placed to the OPR/SLOW position in conjunction with the center maintain position of 9-S-28, the ORDEAL slews at the orbital rate.
9-S-28	SLEW switch Mom-main-mom	This three-position toggle (UP DOWN) is used to slew the pitch indication on the FDAI in the plus or minus direction at a rate dependent on the position of 9-S-27. When this switch is in the center, maintain position and 9-S-27 is in the OPR/SLOW/slew position, the ORDEAL will slew at the orbital rate and the FDAI will display total attitude with respect to a local horizontal reference frame.

4.2.3.10 <u>Landing Point Designator. (See Figure 4.2.3-11)</u>

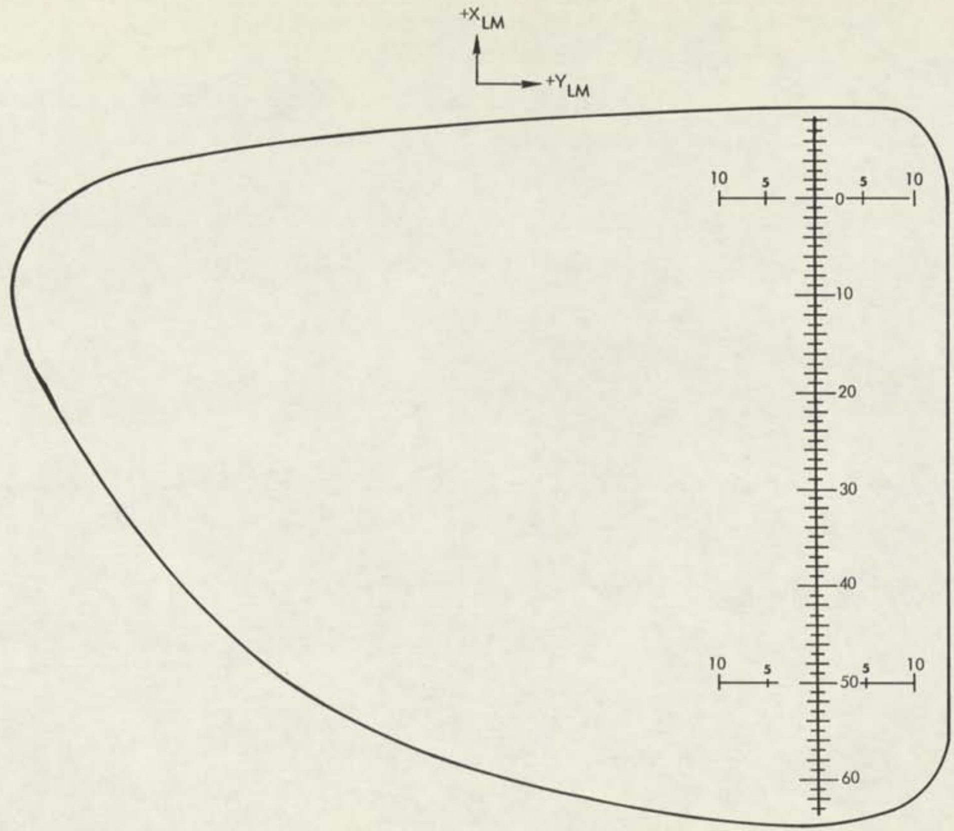


Figure 4.2.3-11. Landing Point Designator

4.3 IDENTIFICATION OF SYSTEM ELEMENTS

Identification of system elements is presented in the following sequence:

a) Primary Guidance Navigation and Control System*

1) IMU

2) LGC

3) DSKY

4) AOT

5) CDU (3)

6) PTA

7) PSA

8) NVB or NB

9) SCA

b) Radar Sections

1) LR Section

2) RR Section

c) Abort Guidance Section

1) ASA

2) AEA

3) DEDA

d) Control Electronics Section

1) ATCA

2) DECA

3) RGA

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*Description of Primary Guidance Navigation and Control System presented is of a limited nature. If further details are desired, refer to ND 1021042, Primary Guidance, Navigation, and Control System Manual (LM).

4) T/TCA (2)

5) ACA (2)

6) GDA (2)

7) GASTA**

8) FDAI (2)**

9) AELD**

10) ORDEAL**

e) LPD

4.3.1 PGNCS

Identification of PGNCS System Elements is presented in the sequence indicated in Paragraph 4.3.

4.3.1.1 Inertial Measuring Unit (IMU)

A brief description of Inertial Measuring Unit Component Identification, Function and Mechanization is presented in the following paragraphs.

4.3.1.1.1 Component Identification. The IMU is the principal component of the LM Primary Guidance, Navigation, and Control Subsystem. In conjunction with the NVB, PSA, PTA, and CDU, the IMU provides the primary inertial sensing system for the PGNCS. The LM IMU is essentially identical to the Block II CSM IMU.

The IMU provides an inertial reference consisting of a stable member with a three-degree-of-freedom gimbal system stabilized by three orthogonal mounted rate integrating gyros. Three accelerometers are orthogonally

An outline drawing of the IMU is shown in Figure 4.3.1.1-1. In the spacecraft the IMU is mounted on the navigation base, see details in Paragraph 4.3.1.8.

**These components are not packaged with the CES but are described in conjunction with the CES for clarity.

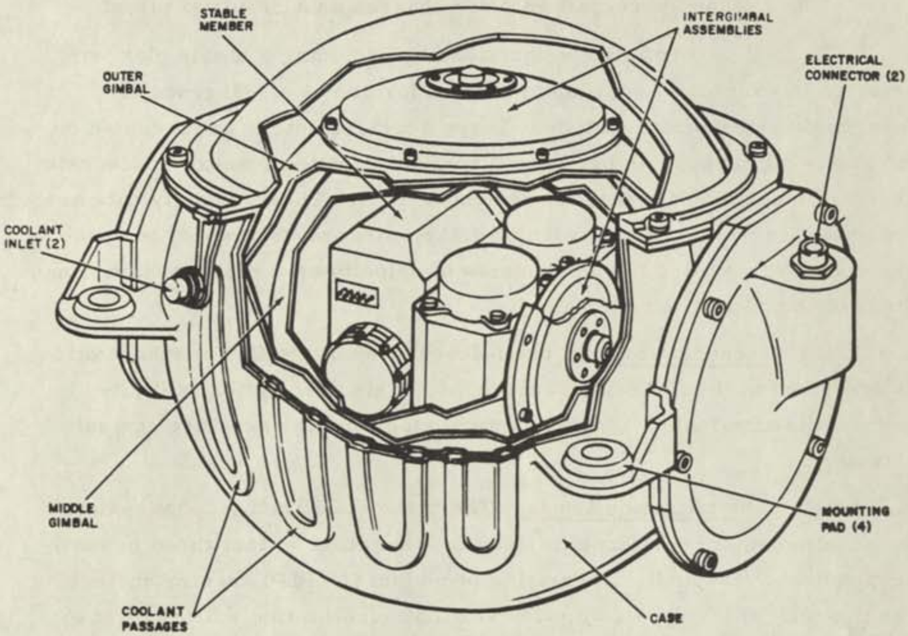

Figure 4.3.1.1-1. Inertial Measuring Unit

4.3.1.1.2 Function. The primary functions of the IMU are as follows:

a) Sense changes in spacecraft attitude

b) Sense spacecraft velocity changes as a result of thrust

The IMU performs these functions by providing a stable platform oriented in an inertial coordinate system for use as a reference for determining spacecraft attitude. Three accelerometers are mounted on the stable platform in an orthogonal configuration to measure spacecraft accelerations along the inertial reference axes. These velocity data are fed to the computer, which calculates the spacecraft's present position and velocity by adding the components of velocity as a result of thrust and the last known position and velocity.

4.3.1.1.3 Mechanization. A brief description of the IMU mechanization is described in the following paragraphs. This description is limited to the stabilization loops, accelerometer loops, and temperature control circuitry.

4.3.1.1.3.1 Stabilization Loops. The three stabilization loops maintain the stable member in a specific spatial orientation so that three mutually perpendicular 16-pulsed integrating pendulum (16 PIP) accelerometers can measure the proper components of LM acceleration with respect to the coordinate system established by the stable member orientation. See Figure 4.3.1.1-2 for the stabilization loop diagram. An input to the stabilization loops is created by any change in LM attitude with respect to the spatial orientation of the stable member. Because of gimbal friction and unbalances, motion of the LM structure relative to the stable member will produce a torque on the stable member which will tend to change its input, they issue error signals which are amplified, resolved, if necessary, into appropriate components, and applied to the gimbal torque motors. The gimbal torque motors then drive the gimbals until the stable member regains its original spatial orientation.

The stabilization loop consists of three prealigned Apollo II inertial reference integrating gyro (Apollo II IRIG) assemblies, a gyro error resolver, three gimbal servo amplifiers, three gimbal torque motors,

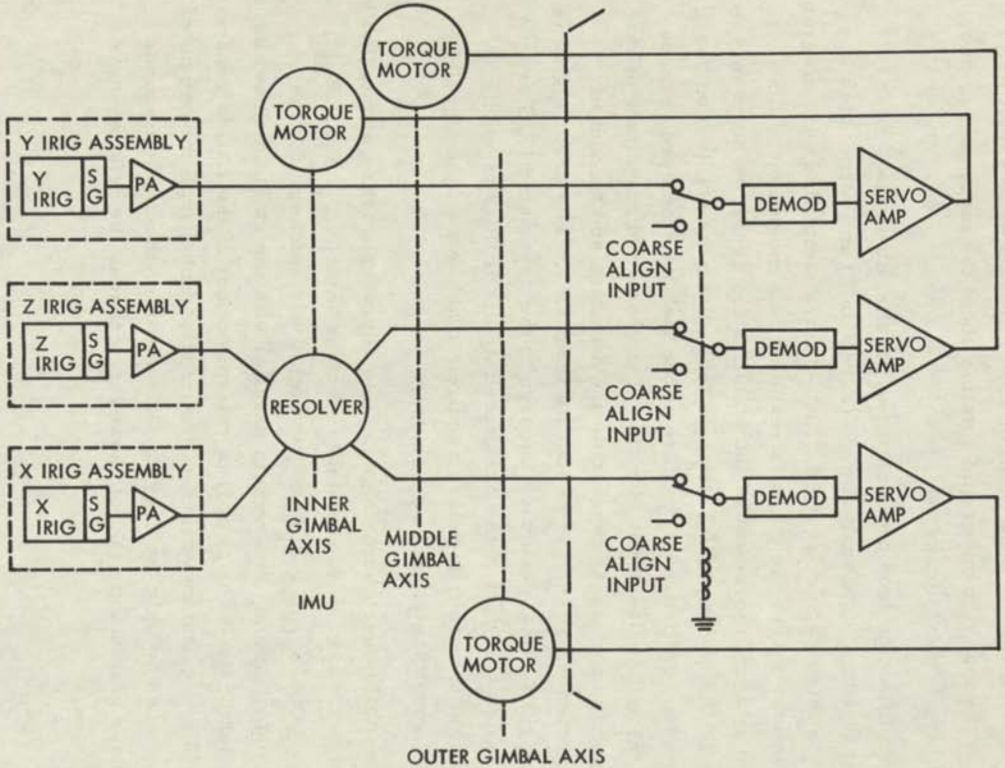

Figure 4.3.1.1-2. LM Stabilization Loops

three gimbals, and circuitry associated with these components. The inner gimbal is the stable member upon which the three stabilization gyros are mounted. The gyros are mounted with their input axes oriented in an orthogonal configuration. Movement of any gimbal tends to result in a movement of the stable member and rotation about the input axes of one or more of the stabilization gyros.

The stabilization loop contains three parallel channels. Each channel starts with a stabilization gyro (X, Y, or Z) and terminates in a gimbal torque motor. See Figure 4.3.1.1-3 for a simplified gyro diagram. The torque motor drives the gimbals resulting in a movement of the stable member and a movement of the stabilization gyros. When a movement of the IMU support gimbal attempts to displace the stable member from its erected position, one or more of the stabilization gyros senses the movement and issues error signals. The phase and magnitude of the 3,200 cps gyro error signal represents the direction and amount of rotation experienced by the gyro about its input axis. The error signal is fed from the gyro signal generator ducosyn to the associated IRIG preamplifier, which is a part of the prealigned Apollo II IRIG assembly. Amplification of the error signal is required to achieve a high signal-to-noise ratio through the gimbal slip rings.

The amplified gyro error signals also represent motion of the stable member about its axis since the stable member axes (X_{SM}, Y_{SM}, Z_{SM}) and the gyro axes (X_g, Y_g, Z_g) are parallel to one another.* If the middle and outer gimbal axes remain parallel with the stable member axes, then movement of the outer gimbal (a yaw movement of the LM) is sensed by only the X gyro, and movement of the middle gimbal (roll movement of the LM) is sensed by only the Z gyro. Movement of the stable member about the inner gimbal axis (Y_{SM}), however, changes the relationship of

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*The Z gyro has its positive input axis aligned to the $-Z_{SM}$ axis, but this is compensated for by reversing the polarity of the 3,200 cps excitation to the primary winding of the Z gyro signal generator ducosyn, which causes the Z gyro error signal to be representative of the direction and amount of motion about the Z_{SM} axis.

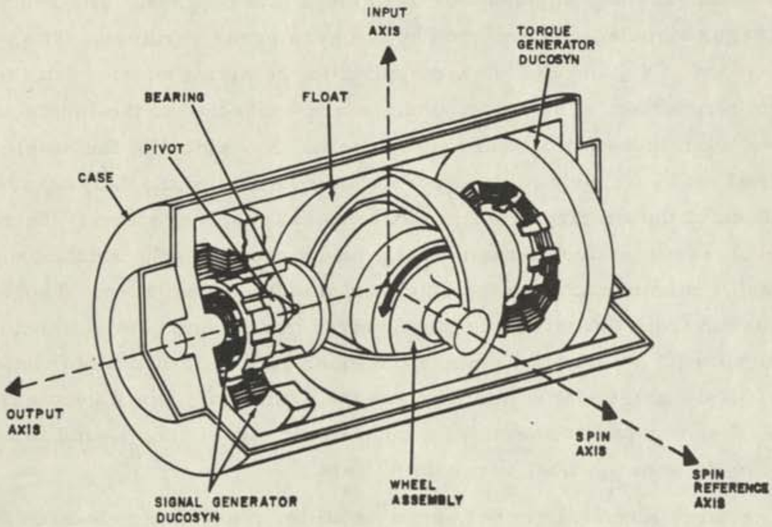


Figure 4.3.1.1-3. Simplified Gyro Diagram

the X and Z gyro input axes to the outer and middle gimbal axes. As a result, a movement of the middle or the outer gimbal is sensed by both X and Z gyros. The input required by the gimbal servo amplifiers to drive the gimbals and move the stable member back to its original position must be composed of components of both the X and Z gyros. The required gimbal error signals are developed by the gyro error resolver. The gyro error signals $E(X_g)$ and $E(Z_g)$, are applied to the stator windings of the gyro error resolver. The rotor windings are connected to the inputs of the outer and middle gimbal servo amplifiers. Movement of the stable member about the inner gimbal axis (pitch movement of the LM) changes the position of the resolver rotor relative to the resolver stator. This change corresponds electromagnetically to the change in the relationship of the stable member axes to the outer and middle gimbal axes. The outputs taken from the rotor are the required middle and outer gimbal error signals (E_{mg} and E_{og}). Since the inner gimbal torque motor axis and the Y-axis of the stable member are the same axis, the Y gyro error signal $E(Y_g)$, is equal to the inner gimbal error signal, (E_{ig}), and is fed directly to the inner gimbal servo amplifier.

The three identical gimbal servo amplifier modules are located in the PSA and contain a phase sensitive demodulator, a filter, and a dc operational power amplifier. The phase sensitive demodulator converts either the 3,200 cps gimbal error or 800 cps coarse align error, zero or pi phase, signals into a representative positive or negative dc signal. The dc signal is filtered and applied to a dc operational amplifier with current feedback. The compensation network in the feedback circuit of the amplifier controls the response characteristics of the entire stabilization loop. The output of the dc amplifier has an operating range between +28 vdc and -28 vdc and drives the respective gimbal torque motor directly in either angular direction.

The gain required for each stabilization loop differs. This difference compensates for the differences in gimbal inertia. The proper gain is selected by the connections to the gimbal servo amplifier module. A single motor is mounted on each gimbal at the positive end of the gimbal axis. The torque motors drive the gimbals to complete the stabilization loop.

The orientation of the stable member can be changed in either the coarse align, fine align, or IMU cage modes. Signals to reposition the gimbals are injected into the gimbal servo amplifiers from the CDU during the coarse align and IMU cage modes and into the stabilization gyros from the fine align electronics during the fine align mode. During the IMU cage mode and the coarse align mode, the reference signal for the demodulator in the gimbal servo amplifier is externally switched from 3,200 cps to 800 cps.

4.3.1.1.3.2 Accelerometer Loops. The pendulous accelerometer (PIPA) loops are described briefly in Paragraph 4.3.1.6, of the PTA section of this book. The PTA contains the supporting electronics for the PIPA loop. This section will refer to figures appearing in the PTA section. For detailed information of accelerometer loop circuitry, see Drawing 6015563 "Two-Wire Mechanization of Apollo PIPA Loops (LM)."

The three accelerometer loops measure the acceleration of the stable member along three mutually perpendicular axes and integrate these data to determine velocity. The velocity is used by the LGC to determine the LM velocity vector. Figure 4.3.1.6-1 is a functional diagram of an accelerometer loop.

The three accelerometer loops contain three prealigned 16-PIP assemblies, three PIP preamplifiers, three ac differential amplifier and interrogator modules, three binary current switches, three calibration modules, three dc differential amplifier and precision voltage reference modules, a pulse torque isolation transformer, and associated electronics.

The three mutually perpendicular PIP's are acceleration sensitive devices. When fixed in its associated accelerometer loop, the PIP becomes an integrating accelerometer. The PIP is basically a pendulum-type device consisting of a cylinder with a pendulous mass unbalance (pendulus float) pivoted with respect to a case. See Figure 4.3.1.1-4 for a simplified accelerometer diagram. The axis of the pivots defines the PIP output axis. A signal generator is located at the positive end of the output axis to provide electrical output signals indicative of the rotational position of the float. A torque generator located at the other end of the float acts as a

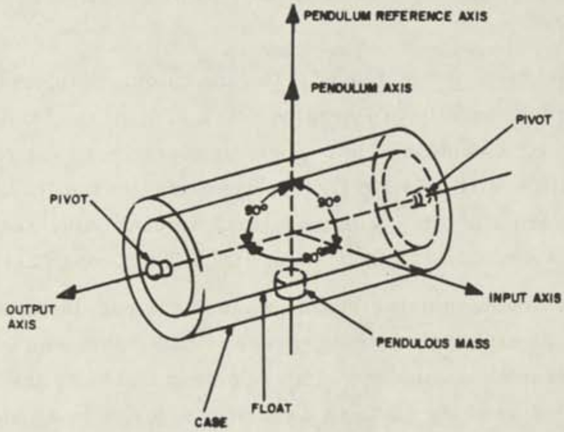


Figure 4.3.1.1-4. Simplified Accelerometer Diagram

transducer to convert electrical signals into mechanical torque about the float shaft. The accelerometer loop using a PIP is mechanized to operate in a binary (two-state) mode.

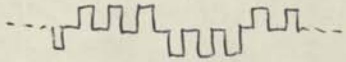
In the binary mode, the PIP pendulum is continually kept in an oscillatory motion. Thus, the two states: positive rotation or negative rotation. The rotation is accomplished by continuously routing torquing current through the torque generator plus or minus windings.

The torque generator has two windings, one to produce torque (rotation) in a positive direction, the other to produce torque (rotation) in a negative direction. Only one winding will have current in it at any one time. The torque winding selection is accomplished by the setting of a flip-flop in the binary current switch (Figure 4.3.1.6-3). When the loop is first energized, the interrogator sets the flip-flop to route the torquing current to one of the windings, which will rotate the float to null. As the float passes through null, the phase of the output signal of the signal generator changes, which causes the interrogator to issue pulses to reset the flip-flop in the binary current switch, and thus, route torquing current to the other torque winding. The float is then torqued in the opposite direction until the signal generator output again changes phase as the float passes through null, which reinstates the cycle.

The output of the signal generator, after being amplified by the PIP preamplifier is interrogated 3200 times a second by the interrogate pulse. The binary current switch flip-flop can be reset only when the interrogate pulse is present and the signal generator output is of the proper phase.

The PIP pendulum motion is an oscillatory motion about its null point and can be measured in cycles per second. As is characteristics of every electro-mechanical loop, there exists some natural resonant frequency. The natural frequency is dependent upon float damping, signal and torque generator sensitivities, and other loop characteristics. In the case of the accelerometer loop this natural frequency if approximately 500 cps, and the pendulum oscillates at a frequency close to that. At a torque winding selection rate of 3200 pulses per second, the value of this frequency can be any value equal to 3200 ÷ x where x is any even number.

number of pulses on one side, when accel.=0

 with accel = 0

Using the above ratio, it is possible for the pendulum to have a maximum frequency of 1600 cps (x equals 2). A frequency of 1600 cps means that for every torque selection pulse, the torque current would be routed to the opposite torque generator winding. Solving the equation $f = 3200 \div x$, the frequency closest to 500 is 533-1/3. In this case the value of x is six. Thus, one complete pendulum cycle will occur during six torque selection pulses. Dividing the time for the six pulses into positive and negative rotations, it is seen that the PIP functions in a 3-3 mode (positive rotation for three torque selection pulses, negative rotation for three torque selection pulses).

The physical configuration of the PIP is such that the float, when moding in its 3-3 state and sensing no acceleration, rotates an equal angular distance on both sides of an electrical and mechanical null.

The two-volt rms, 3200 cps, one-phase signal generator excitation voltage is synchronized with the LGC clock. The signal generator has a center tapped secondary winding which provides a double ended output, one side having a zero phase reference with respect to the 3200 cps excitation and the other side a pi phase reference. The center tap is connected to ground. The output signal is representative of the magnitude and direction of the rotation of the pendulous float about the output axis. The error signal is then routed to the preamplifier mounted on the stable member. The phase of the output signal from the preamplifier is -45 degrees from the reference excitation. The phase shifted zero or pi phase signals from the preamplifier are applied as separate inputs to the ac differential amplifier and further amplified. The two signals are then sent to the interrogator.

The ac differential amplifier and the interrogator are packaged in the same module, which is located in the pulse torque assembly (Figure 4.3.1.6-2). The interrogator analyzes the ac differential amplifier outputs to determine the direction of the 16-PIP float movement and generates appropriate torquing commands. The two amplified signals from the ac differential amplifier go to two summing networks and threshold amplifiers (represented in Figure 4.3.1.6-4 by AND gates). Interrogate pulses (IP) are continuously being received by the interrogator from the LGC. An interrogate pulse is a two-microsecond pulse occurring at 3200 pps and timed to occur 135 degrees after the positive going zero crossing of the reference excitation. With this phasing, the interrogate pulse occurs at the 90-degree peaks of the phase shifted zero or pi phase input signals from the PIP preamplifiers. The interrogate pulse occurs at a positive 90-degree peak of the zero phase signal if the float angle is positive and at a positive 90-degree peak of the pi phase signal if the float angle is negative. The zero and pi phase signals and the interrogate pulses are ANDed by the summing applied to a flip-flop as set or reset pulses. If the flip-flop is in the +set condition, a succession of set pulses will maintain the +set condition. The +set condition will remain until the float angle passes through null. At this time a reset pulse is produced to cause the flip-flop to go to the -set condition.

The outputs of the flip-flop are applied and two AND gates which are also driven by switch pulses received from the LGC. The switch pulses are a train of clock driven 3200-pps pulses, three microseconds in width, timed to occur three microseconds after the leading edge of the interrogate pulse. The flip-flop enables only one output gate at any switch pulse time. The outputs of the AND gates are called the TM +set and the TM -set pulse.

The binary current switch (Figure 4.3.1.6-3) uses the TM +set and TM -set outputs of the interrogator to generate 16-PIP torquing current. The TM +set and the TM -set pulses furnish the input to a flip-flop. If the flip-flop is in the +set condition, a succession of TM +set pulses will maintain the +set condition. The +set condition will persist until the float angle passes through null. The phase change will cause the flip-flop of the ac differential amplifier and interrogator module to reset to the -set condition. At this time, at TM -set pulse is developed and causes the binary current switch flip-flop to go to the -set condition. The outputs of the flip-flop control two transistor current switches. If the flip-flop is in the +set condition, the +set output will be at the base of the +torque current switch and will turn it on. The +torque current switch closes the path from the current regulated 120 vdc supply through the PIPA calibration module to the 16-PIP T +torque generator coils. If the flip-flop is in the -set condition, the -torque current switch will be turned on, closing the path through the T -torque generator coils.

An acceleration along the PIP input axis causes the pendulous mass to produce a torque which tends to rotate the float about the output axis. The torque produced by the acceleration is proportional to the magnitude of the acceleration. The acceleration produced torque aids and opposes the torque generator forces causing changes in the time required for the float to be torqued back through null. A change in velocity (ΔV) is the product of acceleration and incremental time (Δt); the torque is actually proportional to an incremental change in velocity (ΔT).

$$T_{ACCEL} = K_1 a \Delta t = K_1 \Delta V$$

The float is already in motion as a result of loop torquing; therefore, additional torque is required to overcome the acceleration torque and to keep the pendulum in its oscillatory motion. The additional torque is obtained by supplying torquing current for additional time through one of the torque windings. The current at any one time is a constant; therefore, the current must be presented for a longer period of time. Thus, to determine the amount of acceleration sensed by the PIP, it is necessary only to measure the length of time torquing current is applied to each torque winding.

$$ACCEL_{IND} = K_2 \Sigma \left[(T+) - (T-) \right] \Delta T$$

From the above identities, it is seen that torquing time (Δt) is proportional to the change in velocity (ΔT).

$$K_1 \Delta V = K_2 \Sigma \left[(T+) - (T-) \right] \Delta t$$

$$\Delta V = \frac{K_2}{K_1} \Sigma \left[(T+) - (T-) \right] \Delta t$$

The time (Δt), representative of the ΔV, is sent to the LGC in the form of P and N pulses (Figure 4.3.1.6-1).

In addition to selecting the proper torque generator winding, the outputs of the binary current switch flip-flop also go to two AND gates where they are ANDed with the 3200-cps data pulses from the LGC. The data pulse is three microseconds in width and is timed to occur two microseconds after the leading edge of the switch pulse. The data pulse and switch pulse are both 3200 cps; therefore, the LGC receives either a P pulse or an N pulse once every 1 ÷ 3200 seconds. When the PIP is sensing no acceleration, the pendulum is oscillating at a frequency of 533-1/3 cps, and the LGC is receiving three P pulses and three N pulses once every cycle or once every 1 ÷ 533-1/3 seconds. The LGC contains a forward-backward counter which receives the velocity pulses and detects any actual gain in velocity.

The counter counts forward on the three P pulses and then backward on the three N pulses. The counter continues this operation and generates no ΔV pulses. With an acceleration input to the PIP, however, the loop no longer operates at the 3-3 ratio, and the counter exceeds its capacity and reads out the plus or minus ΔV pulses, which are then stored and used by the LGC. The additional pulses above the 3-3 ratio are representative of the additional torque supplied by the torque generator to compensate for the acceleration felt by the LM. Each pulse indicates a known value of ΔV as a result of the loop scale factor.

The PIPA calibration module (Figure 4.3.1.6-5) compensates for the inductive load of the 16-PIP torque generator ducosyns and regulates the balance of the plus and minus torques. The calibration module consists of two load compensation networks for the torque generator coils of the 16 PIP. The load compensation networks tune the torque generator coils to make them appear as a pure resistive load to the binary current switch. A variable balance potentiometer regulates the amount of torque developed by the torque generator coils. Adjustment of this potentiometer precisely regulates and balances the amount of torque developed by the T+ and T- torque generator coils. This balancing insures that for a given torquing current an equal amount of torque will be developed in either direction.

4.3.1.1.3.3 Temperature Control System. The IMU temperature control system (Figure 4.3.1.1-5) maintains the temperature of the stabilization gyros and accelerometers within the required temperature limits during both standby and operating modes of the IMU. The system supplies and removes heat to maintain the IMU heat balance. Heat is removed by convection, conduction, and radiation. The natural convection used during IMU standby mode changes to blower controlled, forced convection during IMU operating modes. The IMU internal pressure is maintained between 3.5 and 15 psia to enable the required forced convection. To aid in removing heat, a water-glycol solution at approximately 45 degrees Fahrenheit from the spacecraft coolant system passes through the coolant passages in the IMU case.

4.3.1.1.3.3.1 Temperature Control Circuit. The temperature control circuit maintains the gyro and accelerometer temperature. The temperature control circuit consists of a temperature control thermostat and heater assembly, a temperature control module, three IRIG end mount heaters, three IRIG tapered mount heaters, two stable member heaters, and three accelerometer heaters. The thermostat and heater assembly is located on the stable member and contains a mercury-thallium thermostat, a bias heater, and an anticipatory heater. Except for the bias heater, all heaters (a total of 12) are connected in parallel and are energized by 28 vdc through a switching action of transistor Q2, which completes the dc return path. The thermostat acts as a control sensing element and senses the temperature of the stable member.

When the thermostat temperature falls below 130 (±0.2) degrees Fahrenheit, the thermostat opens and transistor Q1 conducts and drives transistor Q2 to conduction. When transistor Q2 conducts, current will flow through the twelve heaters. Because of the large mass of the stable member, its temperature will increase at a relatively slow rate as compared to the gyros, which have a heater in each end mount. The anticipatory heater improves the response of the thermostat to insure that the magnitude of the temperature cycling of the gyros and the accelerometers is as small as possible. When the thermostat temperature rises above 130 (±0.22) degrees Fahrenheit, the thermostat closes, and the base of transistor Q1 is shorted to ground, cutting off transistors Q1 and Q2 and

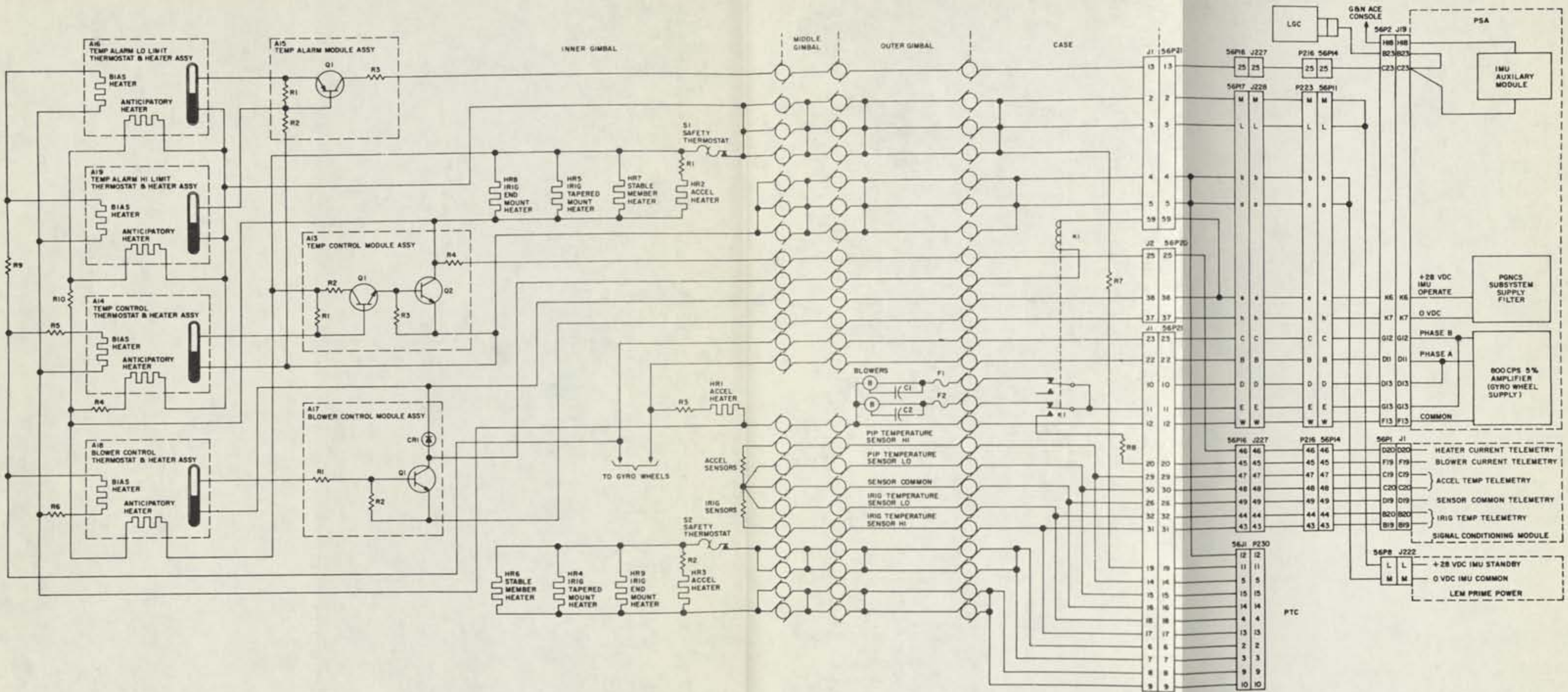


Figure 4.3.1.1-5. IMU Temperature Control System

de-energizing the heaters. The temperature control circuit will maintain the average of the gyro temperatures at 135 degrees Fahrenheit and the average of the accelerometer temperatures at 130 degrees Fahrenheit with the specified coolant temperature. The temperature difference between the gyros and the accelerometers is adjusted by properly proportioning the amount of power in each heater.

During IMU operation, power is applied to the fixed accelerometer heaters to compensate for the additional heat supplied to the gyros by the gyro wheel motor heat dissipation. Power is also applied to a bias heater on the control thermostat. The bias heater supplied a fixed amount of heat to the control thermostat to maintain the proper absolute temperature level of the gyros and accelerometers. The amount of bias heat is controlled by the selection of resistor R5. The power for the fixed accelerometer heaters and the thermostat bias heater are the -90-degree and -180-degree outputs, respectively, from the 28 vac power supplied which are also used for gyro wheel power.

The 28 vdc heater power is applied to the heaters through the contacts of a safety thermostat which will provide protection against an extreme overheat condition in case a malfunction occurs in the temperature control circuit. The safety thermostat contacts open at 139.5 (±3) degrees Fahrenheit.

4.3.1.1.3.3.2 Blower Control Circuit. The blower control circuit (Figure 4.3.1.1-5) maintains IMU heat balance by removing heat. The blower control circuit consists of a blower control thermostat and heater assembly, a blower control module assembly, two axial blowers, and a relay. The contacts of the thermostat contained in the blower control thermostat and heater assembly close at 139 (± 0.2) degrees Fahrenheit and remain closed at high temperatures. Resistor R6 is provided to limit the current through the bias heater in the blower control thermostat and heater assembly. The amount of heat supplied by the bias heater is a constant. If the duty cycle of the temperature control circuit exceeds 50 percent, enough additional heat will be provided by the anticipatory heater to increase the temperature of the blower control thermostat and heater assembly to 139 degrees Fahrenheit. When the thermostat contacts close, transistor Q1 conducts, and relay K1 is energized to remove the power from the blowers. The normal duty cycle of the temperature control circuit, with the IMU in a 75-degree Fahrenheit ambient temperature, is approximately 15 to 20 percent. Under this condition, the blowers will operate continuously. Only a very low ambient temperature will cause a blower off condition.

Power to the blowers is supplied from the -90-degree output of the 28 volt, 800 cps, 5-percent power supply which also provides gyro wheel motor power. Fused phase shift networks are associated with each blower so that excitation and control current can be supplied from the same source.

4.3.1.1.3.3.3 Temperature Alarm Circuit. The temperature alarm circuit, which monitors the temperature control system, consists of a temperature alarm high limit thermostat and heater assembly, a temperature alarm low limit thermostat and heater assembly, and a temperature alarm module assembly. If a high or low temperature is sensed by the thermostats located on the stable member, a discrete is sent to the LGC and to the IMU auxiliary module. When the temperature is within the normal range, the low limit thermostat contacts are closed and the high limit thermostat contacts are open. Transistor Q1 is then properly biased for conduction through a grounding system in the LGC.

At temperatures below 126.0 (± 0.2) degrees Fahrenheit, both the low limit thermostat contacts and the high limit thermostat contacts are open. At temperatures above 134.0 (± 0.2) degrees Fahrenheit, both the low limit thermostat contacts and the high limit thermostat contacts are closed. In either case, transistor Q1 is not able to conduct. Nonconduction of transistor Q1 signals the LGC of an alarm condition. There is no differentiation between a high or low temperature alarm. When the LGC senses a temperature alarm, it causes the TEMP and PGNCS lamps to light. When the IMU auxiliary module receives a temperature alarm, it sends the information to telemetry.

4.3.1.1.3.3.4 External Temperature Control. External temperature control of the IMU is provided by GSE control heater circuits in the IMU, which are controlled externally to the airborne equipment by the portable temperature controller (PTC) or the temperature monitor control panel of the optics-inertial analyzer (OIA). The ground support equipment (GSE) control heater circuitry consists of a safety thermostat, six gyro heaters, two stable member heaters, three accelerometer heaters, temperature indicating sensors, and an IMU standby power sensor, which disables the GSE when airborne power is on. The temperature indicating sensors act as the control sensing element of the external control and indicating circuitry. The heaters are connected in parallel. The six gyro temperature indicating sensors (two in each gyro) are connected in series to sense the average temperature of the gyros. The three accelerometer temperature indicating sensors (one in each accelerometer) are connected in series to sense the average temperature of the accelerometers. All of the GSE control heater circuitry is electrically independent of the airborne temperature control system and will not be used at the same time that the IMU temperature is being controlled by the airborne temperature control system. The GSE control heater circuitry cannot be used as a backup temperature control system during flight.

4.3.1.1.4 Mechanical Characteristics. Mechanical Characteristics are described under Components, Gimbal Arrangement, Size and Weight as follows:

4.3.1.1.4.1 Components. (See Figures 4.3.1.1.-1 and 4.3.1.1-6) The IMU consists of the following components in the indicated locations.

a) On the inner gimbal (stabilized member) (beryllium):

1) Three size-25 single-degree-of-freedom inertial reference integrating gyros

2) Three size-16 single-degree-of-freedom pulse integrating pendulums

3) Gimbal mounted electronics to service suspensions and issue signals of gyros and pendulous accelerometers

4) Temperature control heaters and sensors

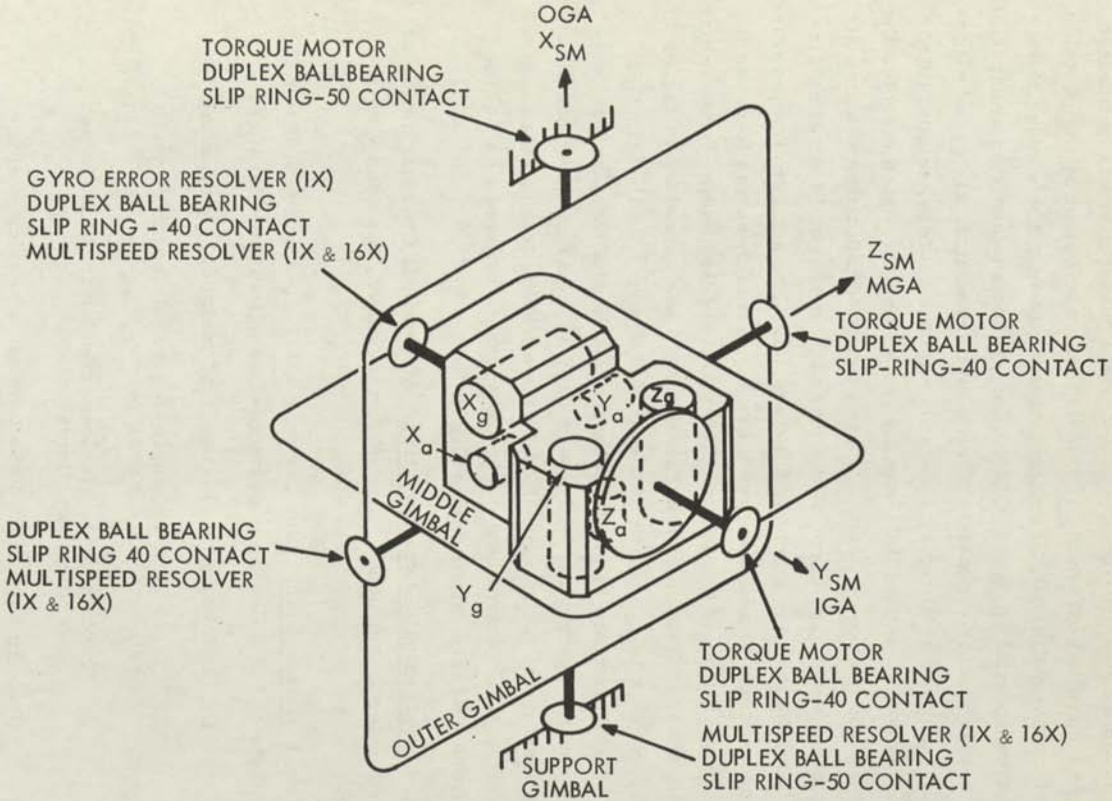


Figure 4.3.1.1-6. IMU Gimbal Assembly and Components

b) On the inner axis:

1) Two assemblies of slip rings, 40 rings each

2) Two duplex-pair bearings; one fixed, one floating

3) One combination 1-speed and 16-speed resolver transmitter

4) One single-speed gyro error signal resolver

5) One dc servo torque generator, 3.8-inch diameter air gap

c) On the middle gimbal (hydroformed aluminum hemispheres): No components

d) On the middle axis:

1) Two assemblies of slip rings, 40 rings each

2) Two duplex-pair bearings; one fixed, one floating

3) One combination 1-speed and 16-speed resolver transmitter

4) One dc servo torque generator, 3.8-inch diameter air gap

5) No gimbal stops, unlimited motion

e) On the outer gimbal (hydroformed aluminum hemispheres): blower motor and fan for heat transport from stable member to case

f) On the outer axis:

1) Two assemblies of slip rings, 50 rings each

2) Two duplex-pair bearings, one fixed, one floating

3) One combination 1-speed and 16-speed resolver transmitter

4) One dc servo torque generator, 3.125-inch diameter air gap

5) No gimbal stops, unlimited motion

g) IMU case (hydroformed aliminum hemisphere):

1) Roll-bonded coolant passage and quick disconnects

2) Electrical 16-speed zero adjustment module

3) Mounting pads for mounting to Navigation Base

4) Two 61-pin electrical connectors

5) Blower control relay

6) Insulation to control condensation on coolant passages

4.3.1.1.4.2 Gimbal Arrangement. The LM IMU is mounted in the spacecraft to provide the following relations between the IMU gimbal axes and the vehicle axes:

a) Outer axis: fixed to vehicle and parallel to and in same direction as vehicle X-axis

b) Middle axis: parallel to and in same direction as vehicle Z-axis when outer gimbal angle is zero

c) Inner axis: parallel to and in same direction as vehicle Y-axis when outer and middle gimbal angles are zero

When the IMU is caged, the middle and inner gimbal axes coincide with the Z- and Y-axes of the LM, respectively. This installation was chosen so that gimbal lock could normally be avoided during the LM mission phases. Gimbal lock occurs when the IMU outer and inner gimbal axes coincide or fall within 10 degrees of each other. This gimbal arrangement and the IMU components are shown in Figure 4.3.1.1-6 IMU Gimbal Assembly and Components.

4.3.1.1.4.3 Size. For overall outward appearance of the IMU, refer to Figure 4.3.1.1-1. The dimensions of the IMU are as follows:

a) On case diameter, 12.5 inches

b) Along outer axis, 14.0 inches

c) Volume, approximately, 1100 cubic inches

For detailed dimensions of the IMU, refer to Drawing ND 2018616 "Gimbal Assembly Outline - Block II IMU."

4.3.1.1.4.4 Weight. According to the MIT Report E-1142, April 1967, the weight is 42.4 pounds.

4.3.1.2 LM Guidance Computer

A brief description of the LM Guidance Computer Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.2.1 Component Identification. The LGC and the DSKY constitute the LM Computer Subsystem. The LGC is physically identical to the computer in the command module (AGC) with the exception of programming and fixed memory wiring. The outline drawing of the LGC is shown in Figure 4.3.1.2-1.

The LGC is the central data-processing device of the PGNCS. It is a parallel, fixed-point, one's complement digital computer with a fixed rope-core memory and an erasable ferrite-core memory. Inputs to the LGC are received from the landing and rendezvous radars through the radar/computer interface unit from the IMU, and from the navigator via the DSKY. Using these data and other system status data, the LGC computes flight and engine control commands and status information for display to the astronauts.

4.3.1.2.2 Function. The LGC in conjunction with the DSKY performs the following functions:

a) Solves guidance, navigation and control problems for all LM mission phases.

b) Generates attitude control and thrust commands to reaction control jets, ascent engine, and descent engine to maintain the LM on satisfactory trajectory for specific mission phases.

c) Displays pertinent information to the astronauts and to the earth when requested.

d) Provides a means by which the astronauts or ground control can directly communicate with the PGNCS system.

e) Provides control information to the PGNCS and other spacecraft systems.

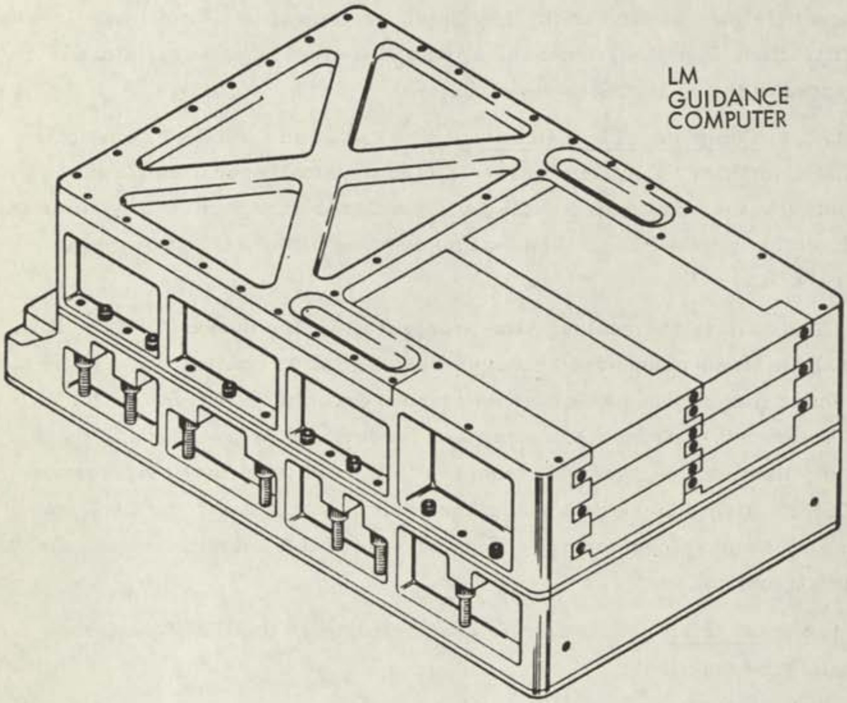


Figure 4.3.1.2-1. LM Guidance Computer - Outline

f) Provides timer which serves as basic frequency standard for LGC, PGNCS, and other LM systems.

g) Controls gimballing and throttling of descent engine during lunar landing.

h) Monitors its own operation and certain other LM system operations.

The main functions of the LGC are implemented through the execution of the programs stored in memory. Programs are written in a machine language called basic instructions. A basic instruction contains an operation (order) code and a relevant address. The order code defines the data flow within the LGC, and the relevant address selects the data that are to be used for computations. The order code of each instruction is entered into the sequence generator, which controls data flow and produces a different sequence of control pulses for each instruction. Each instruction is followed by another instruction. In order to specify the sequence in which consecutive instructions are to be executed, the instructions are normally stored in successive memory locations. By adding the quantity one to the address of an instruction being executed, the address of the instruction to be executed next is derived. Execution of an instruction is complete when the order code of the next instruction is transferred to the sequence generator and the relevant address is in the central processor.

The central processor consists of several flip-flop registers. It performs arithmetic operations and data manipulations on information accepted from memory, the input channels, and priority control. Arithmethic operations are performed using the ONE's complement number system. Values of 14 bits, excluding sign, (up to 28 bits during double precision operations) are processed with an additional bit produced for overflow or underflow. All operations within the central processor are performed under control of pulses generaged by the sequence generator. In addition, all words read out of memory are checked for correct parity, and a parity bit is generated within the central processor for all words written into memory. The LGC uses odd parity, that is, all words stored in memory contain an odd number of ONE's including the parity bit. The central processor also supplies data and control signals through the output channels and provides interface for the various spacecraft subsystems.

The LGC has ten program interrupt conditions. These ten interrupts are T6 RUPT, T5 RUPT, T3 RUPT, T4 RUPT, KYRPT 1, KYRPT 2 (or MKRPT), UPRUPT, CLKRPT, RADRPT, and HNDRPT. The T6 RUPT through T4 RUPT conditions are internal interrupts initiated by the LGC. The KYRPT 1 condition is initiated when a DSKY pushbutton is depressed. A MARK signal (discrete bit), indicating a sighting, initiates KYRPT 2. This interrupt shares the same priority as the KYRPT 2 interrupt associated with the navigation DSKY in the CSM application of the computer. UPRUPT indicates the completion of an uplink word. RADRPT is generated when a complete radar word is received. HNDRPT is initiated as soon as the hand controller is moved out of detent by the astronaut.

Before a priority program can be executed, the current program must be interrupted; however, certain information about the current program must be preserved. This information includes the program counter contents and any intermediate results contained in the central processor. The priority control produces an interrupt request signal, which is sent to the sequence generator. This signal, acting as an order code, causes the execution of an instruction that transfers the current contents of the program counter and any intermediate results to memory. In addition, the control pulses transfer the priority program address in priority control to the central processor, and then to memory through the write lines. As a result, the first basic instruction word of the priority program is entered into the central proccessor from memory, and execution of the priority program is begun. The last instruction of each priority restores the LGC to normal operation, provided no other interrupt request is present, by transferring the previous program counter and intermediate results from their storage locations in memory back to the central processor.

Certain data pertaining to the flight of the LM are used to solve the guidance and navigation problems required for the LM mission. These data, which includes real time, acceleration, and IMU gimbal angles, are stored in memory locations called counters. The counters are updated as soon as new data become available. An incrementing process which changes the contents of the counters is implemented by priority control between the execution of basic instructions. Data inputs to priority control are called incremental pulses. Each incremental pulse produces a counter address and a priority request. The priority request signal is sent to the sequence generator, where it functions as an order code. The control pulses produced by the sequence generator transfer the counter address to memory through the write lines of the central processor. In addition, the control pulses enter into the central processor the contents of the addressed counter to be incremented.

Real time plays a major role in solving guidance and navigation problems. Real time is maintained within the LGC in the main time counter of memory. The main time counter provides a 745.65-hour (approximately 31 days) clock. Incremental pulses are produced in the timer and sent to priority control for incrementing the main time counter. The LM mission requires that the LGC clock be synchronized with the KSC clock. The LGC time is transmitted once every second by downlink operation for comparison with the KSC clock.

Incremental transmissions occur in the form of pulse bursts from the output channels to the CDU, the gyro fine align electronics, the RCS of the spacecraft, and the radar. The number of pulses and the time at which they occur are controlled by the LGC program. Discrete outputs also originate in the output channels under program control. These outputs are sent to the DSKY and various other subsystems. Continuous pulse trains originate in the timing output logic for synchronization of other systems.

Through the DSKY, the astronaut can load information into the LGC, retrieve and display information contained in the LGC, and initiate any program stored in memory. A keycode is assigned to each keyboard pushbutton. When a keyboard pushbutton on the DSKY is pressed, the keycode is produced and sent to an input channel. A signal is also sent to priority control, where it produces both the address of a priority program stored in memory and a priority request signal, which is sent to the sequence generator. This operation results in an order code and initiates an instruction for interrupting the program in progress and executing the KEYRUPT priority program stored in memory. A function of this program is to transfer the keycode, temporarily stored in an input channel, to the central processor, where it is decoded and processed. A number of keycodes are required to specify an address, or data word. The program initiated by a keycode also converts the information from the DSKY keyboard to a coded display format. The coded display format is transferred by another program to an output channel and sent to the display portion of the DSKY. The display notifies the astronaut that the keycode was received, decoded, and processed properly by the LGC.

4.3.1.2.3 Mechanization. It is beyond the scope of this book to provide a detailed mechanization of the LGC hardware; however, the basic organization of the LGC and its functional components will be discussed briefly.

The LGC can be functionally divided into the following areas:

a) Timer

b) Sequence generator

c) Central processor

d) Priority control

e) Input-output

f) Memory

g) Power

These areas will be discussed in following paragraphs. For a block diagram of the basic flow of the computer subsystem refer to Figure 4.3.1.2-2.

4.3.1.2.3.1 Timer. The timer has three specific functions:

a) Generates all of the necessary synchronization pulses to insure a logical flow of data from one area to another within the LGC.

b) Generates waveforms which establish output rates to other spacecraft systems for control and synchronization purposes.

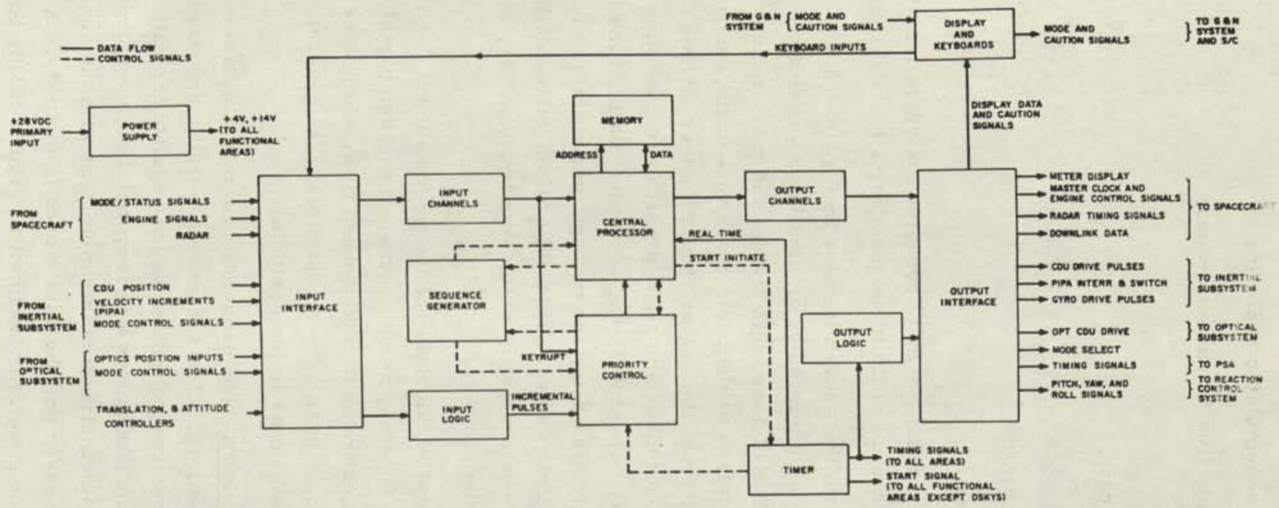


Figure 4.3.1.2-2. Flow Diagram - LGC

c) Generates and accumulates real time.

The timer is divided into five functional areas:

a) Oscillator.

b) Clock divider logic.

c) Scaler.

d) Time pulse generator.

e) Start-stop logic.

The timer generates the timing signals required for operation of the LGC and is the primary source of timing signals for all LM systems.

The timer is divided into the areas indicated in Figure 4.3.1.2-3. The master clock frequency is generated by an oscillator and is applied to the clock divider logic. The divider logic divides the master clock input into gating and timing pulses at the basic clock rate of the computer. Several outputs are available from the scaler, which further divides the divider logic output into output pulses and signals which are used for gating, for generating rate signal outputs, and for accumulating time. Outputs from the divider logic also drive the time pulse generator which produces a recurring set of time pulses. This set of time pulses defines a specific interval (memory cycle time) in which access to memory and word flow take place within the computer.

The start-stop logic senses the status of the power supplies and specific alarm conditions in the computer and generates a stop signal that is applied to the time pulse generator to inhibit word flow. Simultaneous with the generation of the stop signal, a fresh start signal is generated that is applied to all functional areas in the computer.

4.3.1.2.3.2 <u>Sequence Generator</u>. The sequence generator carries out the instructions stored in memory. The sequence generator processes insturction codes and produces control pulses which regulate the data flow of the computer. The control pulses are responsible for performing the operations assigned to each instruction in conjunction with the various registers in the central processor and the data stored in memory. A block diagram of the sequence generator is shown in Figure 4.3.1.2-4.

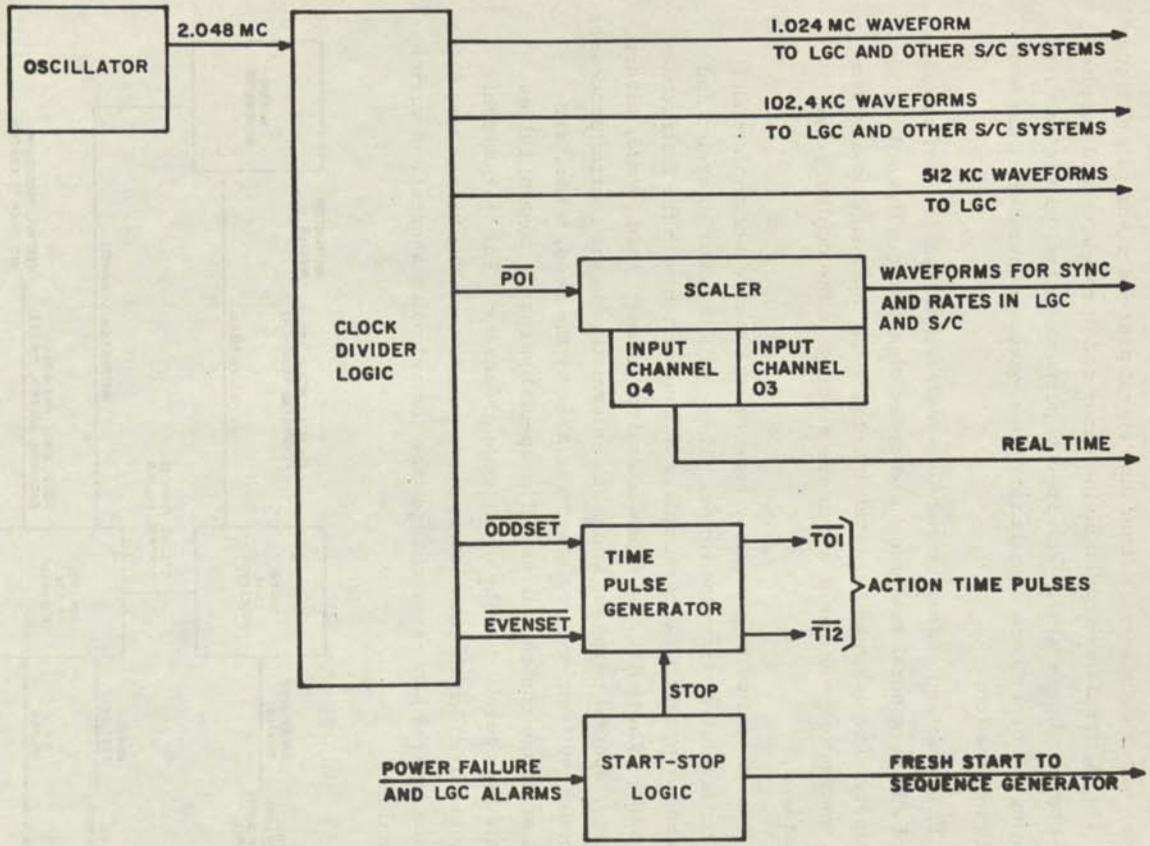


Figure 4.3.1.2-3. Timer, Block Diagram

The sequence generator consists of the order code processor, command generator, and control pulse generator. The sequence generator receives order code signals from the central processor and priority control. These signals are coded by the order code processor and supplied to the command generator. The special purpose control pulses are used for gating the order code signals into the sequence generator at the end of each instruction.

The command generator receives instruction signals from priority control and peripheral equipment and coded signals from the order code processor. The command generator decodes the input signals and produces instruction commands which are supplied to the control pulse generator.

The control pulse generator receives the twelve-action time and other timing pulses from the timer. These pulses occur in cycles and are used for producing control pulses in conjunction with the instruction commands. There are five types of control pulses: read, write, direct exchange, special purpose, and test. Information in the central processor is transferred from one register to another by the read, write, and direct exchange control pulses. The special purposes control pulses regulate the operation of the order code processor. The test control pulses are used within the control pulse generator. The branch test data from the central processor changes the control pulse sequence of various instructions.

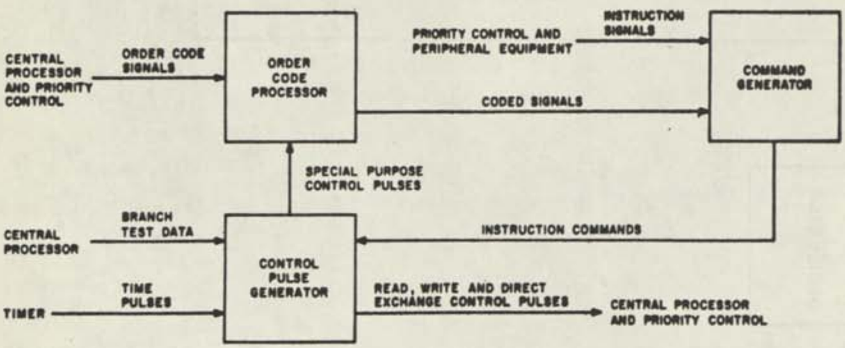


Figure 4.3.1.2-4 Sequence Generator, Block Diagram

4.3.1.2.3.3 Central Processor. The central processor, Figure 4.3.1.2-5 consists of the flip-flop registers, the write, clear, and read control logic, write amplifiers, memory buffer register, memory address register, and decoder and the parity logic. All data and arithmetic manipulations within the LGC take place in the central processor.

Primarily, the central processor performs operations indicated by the basic instructions of the program stored in memory. Communication within the central processor is accomplished through the write amplifiers. Data flows from memory to the flip-flop registers or vice-versa, between individual flip-flop registers, or into the central processor from external sources. In all instances, data are placed on the write lines and routed to a specific register or to another functional area under control of the write, clear, and read logic. This logic section accepts control pulses from the sequence generator and generates signals to read the content of a register onto the write lines and to write this content into another register of the central processor or to another functional area of the LGC. The particular memory location is specified by the content of the memory address register. The address is fed from the write lines into this register, the output of which is decoded by the address decoder logic. Data are subsequently transferred from memory to the memory buffer register. The decoded address outputs are also used as gating functions within the LGC.

The memory buffer register buffers all information read out or written into memory. During readout, parity is checked by the parity logic and an alarm is generated in case of incorrect parity. During write-in, the parity logic generates a parity bit for information being written into memory. The flip-flop registers perform the data manipulations and arithmetic operations. Each register is 16 bits or one computer word in length. Data flow into and out of each register as dictated by control pulses associated with each register. The control pulses are generated by the write, clear, and read control logic.

External inputs through the write amplifiers include the content of both the erasable and fixed-memory bank registers, all interrupt addresses from priority control, control pulses which are associated with specific arithmetic operations, and the start address for an initial start condition. Information from the input and output channels is placed on the write lines and routed to specific destinations either within or external to the central processor. The computer test set inputs allow a word to be placed on the write lines during system and subsystem tests.

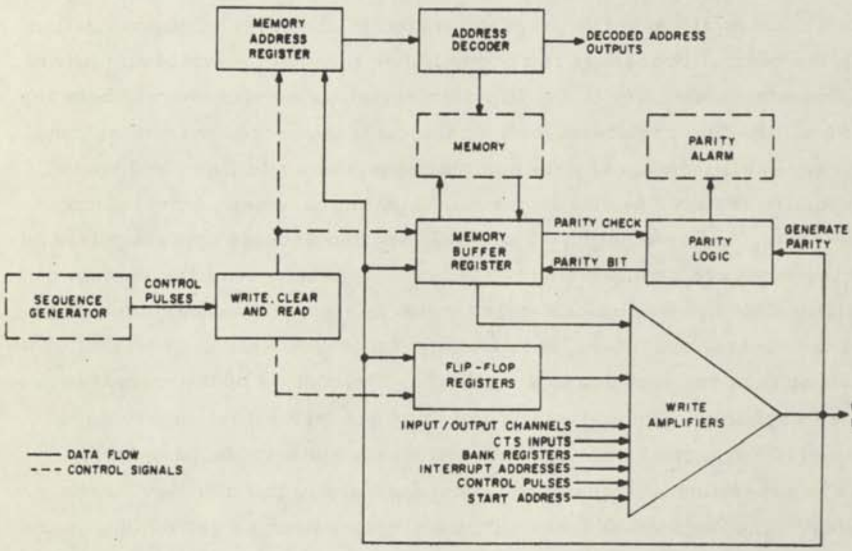


Figure 4.3.1.2-5 Central Processor, Block Diagram

4.3.1.2.3.4 Priority Control. Priority control is related to the sequence generator in that it controls all involuntary or priority instructions. The priority control processes input-output information, issues order code and instruction signals to the sequence generator, and issues twelve-bit addresses to the central processor.

The priority control (Figure 4.3.1.2-6) consists of the start, interrupt, and counter instruction control circuits. The start instruction control initializes the computer if the program works itself into a trap, if a transient power failure occurs, or if the interrupt instruction control is not functioning properly. The computer is initialized with the start order code signal, which not only forces the sequence generator to execute the start instruction, but also resets other computer circuits. When the start order code signal is being issued, the T12 stop signal is sent to the timer. This signal stops the time pulse generator until all essential circuits have been reset and the start instruction has been forced by the sequence generator. The computer may also be initialized manually when connected to the peripheral equipment and placed into the monitor stop mode. In this mode, the time pulse generator is held at the T12 position until the monitor stop signal is released.

The interrupt instruction control can force the execution of the interrupt instruction by sending the interrupt order code signal to the sequence generator and the twelve-bit address to the central processor. There are ten addresses, each of which accounts for a particular function that is regulated by the interrupt instruction control. The interrupt instruction control links the keyboard, telemetry, and time counters to program operations. The interrupt addresses are transferred to the central processor by read control pulses from the sequence generator. The source of the keyboard, telemetry, and time counter inputs is the input-output circuits. The interrupt instruction control has a built-in priority chain which allows sequential control of the ten interrupt addresses. The decoded interrupt addresses from the central processor are used to control the priority operation.

The counter instruction control is similar to the interrupt instruction control in that it links input-output functions to the program. It also supplies twelve-bit addresses to the central processor and instruction signals to the sequence generator. The instruction signals cause a delay (not an interruption) in the program by forcing the sequence generator to execute a counter instruction. The addresses are transferred to the central processor by read control pulses. The counter instruction control also has a built-in priority of the 29 addresses it can supply to the central processor. This priority is also controlled by decoded counter address signals from the central processor. The counter instruction control contains an alarm detector which produces an alarm if an incremental pulse is not processed properly.

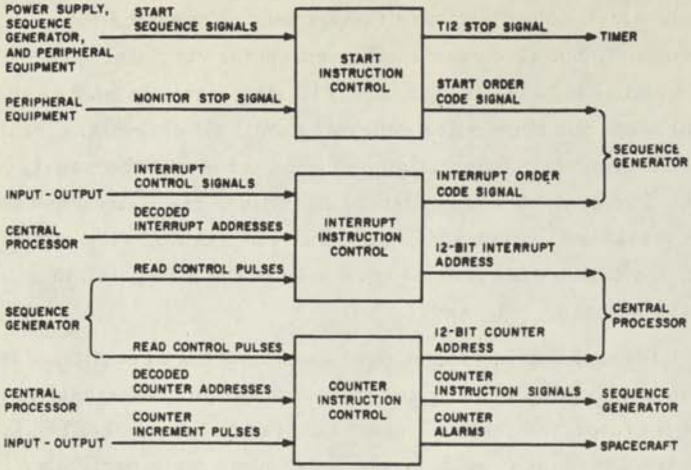


Figure 4.3.1.2-6. Priority Control, Block Diagram

4.3.1.2.3.5 Input-Output. The input-output section accepts all inputs to, and routes to other systems all outputs from, the computer. The input-output section (Figure 4.3.1.2-7) includes the interface circuits, input and output channels, input logic, output timing logic, and the downlink circuits.

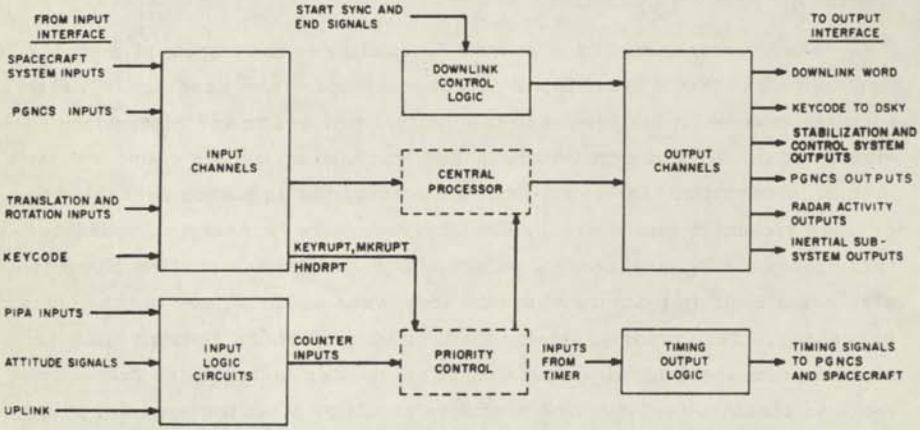


Figure 4.3.1.2-7. Input-Output, Block Diagram

Most of the input and output channels are flip-flop registers similar to the flip-flop registers of the central processor. Certain discrete inputs are applied to individual gating circuits which are part of the input channel structure. Typical inputs to the channels include keycodes from the DSKY and signals from the PGNCS proper and other LM systems. Input data are applied directly to the input channels; there is no write process as in the central processor. However, the data are read out to the central processor under program control. The input logic circuits accept inputs which cause interrupt sequences within the computer. These incremental inputs are applied to the priority control circuits and subsequently to associated counters in erasable memory.

Outputs from the computer are placed in the output channels and are routed to specific systems through the output interface circuits. The operation is identical to that in the central processor. Data are written into an output channel from the write lines and readout to the interface circuits under program control. Typically, these outputs include outputs to the stabilization and control system, the DSKY, and PGNCS, et cetera. The downlink word is also loaded into an output channel and routed to the LM spacecraft telemetry system by the downlink circuits.

The output timing logic gates synchronization pulses (fixed outputs) to the PGNCS and the LM spacecraft. These are continuous outputs since the logic is specifically powered during normal operation of the computer and during standby.

4.3.1.2.3.6 Memory. Memory (Figure 4.3.1.2-8) consists of an erasable memory with a storage capacity of 2,048 words and a fixed-core rope memory with a storage capacity of 36,864 words. Erasable memory is a random-access, destructive-readout storage device. Data stored in erasable memory can be altered or updated. Fixed memory is a nondestructive storage device. Data stored in fixed memory are unalterable since the data are wired in, and readout is nondestructive.

Both memories contain magnetic-core storage elements. In erasable memory the storage elements form a core array; in fixed memory, the storage elements form three core ropes. Erasable memory has a density of one word per 16 cores; fixed memory has a density of eight words per core. Each word is located by an address.

In fixed memory addresses are assigned to instruction words to specify the sequence in which they are to be executed; blocks of addresses are reserved for data, such as constants and tables. Information is placed into fixed memory permanently by weaving paterns through the magnetic cores. The information is written into assigned locations in erasable memory with the computer test set, the DSKY, uplink, or program operation.

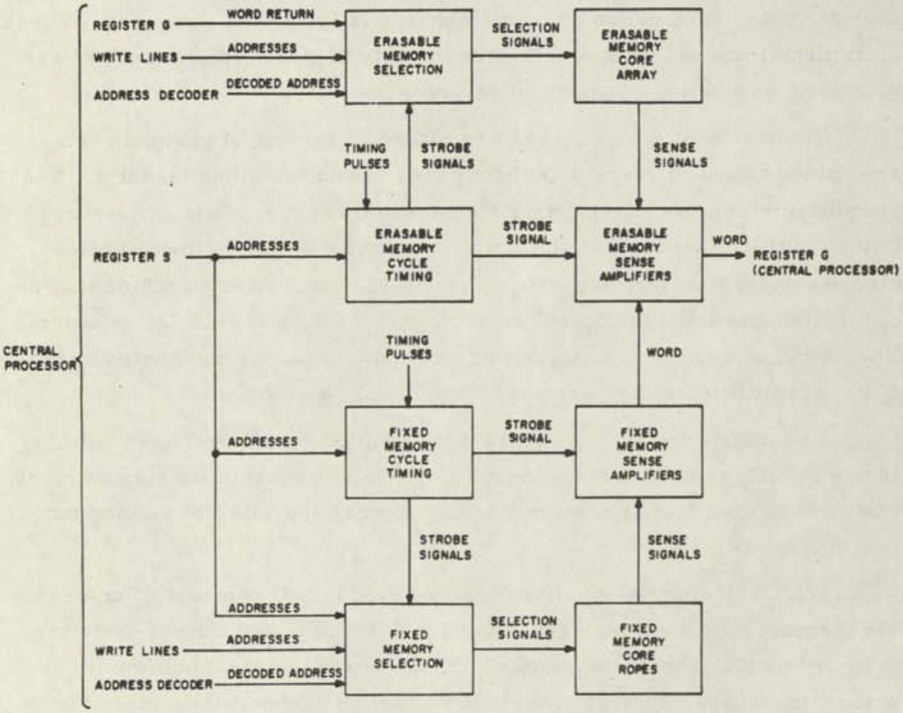
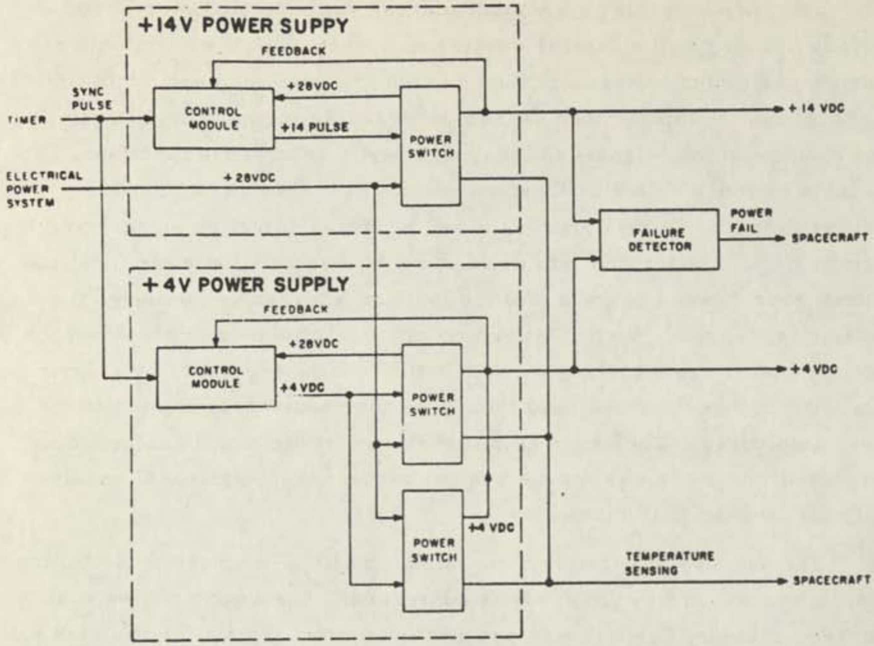


Figure 4.3.1.2-8. Memory, Block Diagram

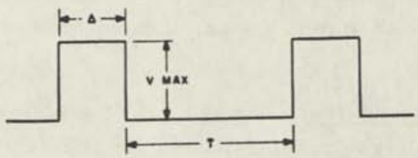
Both memories use a common address register (register S) and an address decoder in the central processor. When register S contains an address pertaining to erasable memory, the erasable memory cycle time is energized. Timing pulses sent to the erasable memory cycle timing then produce strobe signals for the read, write, and sense functions. The erasable memory selection logic receives an address and a decoded address from the central processor and produces selection signals which permits data to be written into or read out of a selected storage location. When a word is read out of a storage location in erasable memory, the location is cleared. A word is written into erasable memory through the memory buffer register (register G) in the central processor by a write strobe operation. A word read from a storage location is applied to the sense amplifiers. The sense amplifiers are strobed and the information is entered into register G of the central processor. Register G receives information from both memories.

The address in register S energizes the fixed-memory cycle timing when a location in fixed memory is addressed. The timing pulses sent to the fixed-memory cycle timing produce the strobe signals for the read and sense functions. The selection logic receives an address from the write lines, a decoded address and addresses from register S, and produces selection signals for the core rope. The content of a storage location in fixed memory is strobed from the fixed-memory sense amplifiers to the erasable memory sense amplifiers and then entered into register G of the central processor.

4.3.1.2.3.7 <u>Power Supplies</u>. The two power supplies (Figure 4.3.1.2-9) furnish operating voltages to the LGC and the DSKY. Primary power of 28 vdc from the spacecraft is applied to both power supplies. Regulator circuits maintain a constant output of +4 volts and +4 volts switched from one supply, and +14 volts and +14 volts switched from the other. The regulator circuits are driven by a sync signal input from the timer, each power supply having a different sync frequency. During system and subsystem tests, inputs from the computer test set can be used to simulate power supply failures.



(a)



(b)

Figure 4.3.1.2-9. Power Supply, Block Diagram

The standby mode of operation is initiated by pressing the standby (STBY) pushbutton on the DSKY. During standby, the LGC is put into a RESTART condition and the switchable +4 and +14 voltages are switched off, thus putting the LGC into a low power mode where only the timer and a few auxiliary signals are operative.

4.3.1.2.4 General Characteristics

a) Memory. The LGC has two memory sections, a fixed and an erasable section. The erasable section has 2,048 words of ferrite coincident current. The fixed or rope section has 36,864 words of wired transformer-type, wound cores.

b) Word Length. Sixteen bits (15 data + 1 parity)

c) Instruction Codes. Total of 35 as follows:

   1) 18 single precision

   2) 3 double precision

   3) 7 in/out servicing

   4) 7 branching and interrupt control

d) Clock Rate. 1.024 mc derived from 2.048 mc oscillator.

e) Significant Operating Times

1)	Memory cycle time	11.7 μ sec
2)	Normal instruction time	23.4 μ sec
3)	Add instruction time	23.4 μ sec
4)	Multiply time	48.0 μ sec
5)	Double precision addition time	36.0 μ sec
6)	Double precision multiply time	0.6 milli-sec
7)	Counter incrementing time	11.7 μ sec

4.3.1.2.4.1 Mechanical Characteristics

a) Construction. The computer is enclosed in a sealed metal package. The computer is assembled from an "A" tray and a "B" tray, each containing certain modules. The computer is mounted to the spacecraft with a 14 1/4-inch x 28 socket head cap screws.

b) Size. (Per MIT Drawing No. J121040, 21 January 1965) Maximum envelope is 6.000 x 12.438 x 24.250 inches. See Figure 4.3.1.2-1 for an outline drawing.

c) Weight. (Per MIT Report E-1142 April 1967) Weight is 69.0 pounds

4.3.1.2.4.2 Electrical Characteristics

a) Modular Components. The LGC is assembled from the following modules, plus housing and necessary connectors:

1) 24 logic modules containing 5400 NOR logic gates

2) 5 interface modules

3) 2 power supply modules

4) 6 rope memory modules

5) 2 rope driver modules

6) 1 strand select module

7) 2 sense amplifier modules

8) 1 erasable driver modules

9) 1 current switch module

10) 1 alarm module

11) 1 oscillator module

b) Power Consumption. Per MIT Report E-1142 Rev. 42 March 66, LGC operating power is 95 watts and LGC standby power is 10 watts from +28 vdc primary LM bus.

c) LGC Interface Signals. For details in this area refer to LGC-LM Electrical Interface ICD LIS 370-10004 and all current IRN's against this ICD.

Additional information is available in Paragraph 4.4 of this Data Book.

The LGC Continuous Drive Pulse Outputs are tabulated in Table 4.3.1.2-1.

Table 4.3.1.2-1. LM Continuous Drive Pulse Outputs

Signal	Frequency	Purpose
Master clock	1.024 mc	CTE frequency reference
LR reset	3200 pps	to LR transfer gates
RR reset	3200 pps	to RR transfer gates
CDU sync	51.2 kpps	to CDU interrogate mode to provide CDU timing
800 pps set 800 pps reset	800 pps	800 pps zero and pi phase supplied to the 800 cps 1 percent power supply
3200 pps set 3200 pps reset	3200 pps	3200 pps at phases of 0 deg and 180 deg supplied to the 3200 cps power supply
12.8 kpps	12.8 kpps	12.8 kpps zero phase supplied to the pulse torque power supply
25.6 kpps	25.6 kpps	25.6 kpps zero phase supplied to the -28 vdc vdc power supply
PIPA interrogate PIPA switching	3200 pps 3200 pps	provided to the AC differential amplifier and interrogator in the accelerometer loop
PIPA data pulse	3200 pps	provided to the binary current switch in the accelerometer loop

4.3.1.3 Display and Keyboard (DSKY)

A brief description of DSKY Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.3.1 Component Identification. The DSKY is an integral part of the computer subsystem. This device provides a communication channel between the LGC and the astronaut. It is located below the center panels of the LM controls and displays.

The DSKY consists of a keyboard, display panel, condition indicators, a power supply, and a relay package. The keyboard provides the astronaut with the capability of inserting data into the LGC and initiating LGC operations. Through the keyboard, the astronaut can also control the ISS moding. The DSKY display panel provides a visual indication of data being loaded into the LGC, the LGC activity, and LGC program. The display panel also provides the LGC with a means of displaying or requesting data. The condition indicators display PGNCS System status.

4.3.1.3.2 Function. The DSKY provides a two-way communications link between the astronaut, and the LGC and enables the following functions to be performed:

a) Loading of data into the LGC

b) Display of data from the LGC and data loaded into the LGC

c) Monitoring of data from the LGC

d) Display of the LGC program in operation

e) System control by the initiation of subsystem and system testing and control of the system's major modes of operation

f) Requests by the LGC to the system operator to perform actions

4.3.1.3.3 Mechanization.

4.3.1.3.3.1 DSKY Displays. DSKY displays are described in the following paragraphs.

4.3.1.3.3.1.1 Numerical Display. A numerical display is associated with the computer keyboard for display of called-up-data, verification of the inserted data, and computer requests for astronaut action or data. Three two-digit decimal readouts identify: (1) the current program (PROGRAM) being processed by the computer, (2) the current VERB being processed by the computer, and (3) the current NOUN being processed by the computer. Computer activated VERB-NOUN flashes indicate to the astronaut a request for data or action. Three five-digits plus an algebraic sign decimal readouts (REGISTER 1, 2 and 3) are provided for data display.

4.3.1.3.3.1.2 Condition Display. A condition display is associated with the computer keyboard for display (1) activity (UPLINK ACTIVITY), (2) status (OPERATOR ERROR, NO ATTITUDE, STANDBY, and KEY RELEASE), and (3) cautions (GIMBAL LOCK, TEMP PROGRAM, RESTSRT, and TRACKER). See Figure 4.3.1.3-1 for the caution and warning logic diagram.

The condition displays as used on the LM appear in Table 4.3.1.3-1.

4.3.1.3.3.2 DSKY Controls and Functions. The DSKY keyboard is used to manually insert or call up LGC data. The keyboard consists of ten numerical pushbuttons (0-9), two algebraic sign pushbuttons (+ (plus) and - (minus)) and seven instruction pushbuttons (ENTR, RSET, CLR, STBY, KEY REL, VERB, and NOUN). All the pushbuttons, except the STBY pushbutton, have five-bit codes associated with them and convey information to the LGC. These controls and their functions are listed in Table 4.3.1.3-2. The five-bit codes associated with the keyboard are tabulated in Table 4.3.1.3-3.

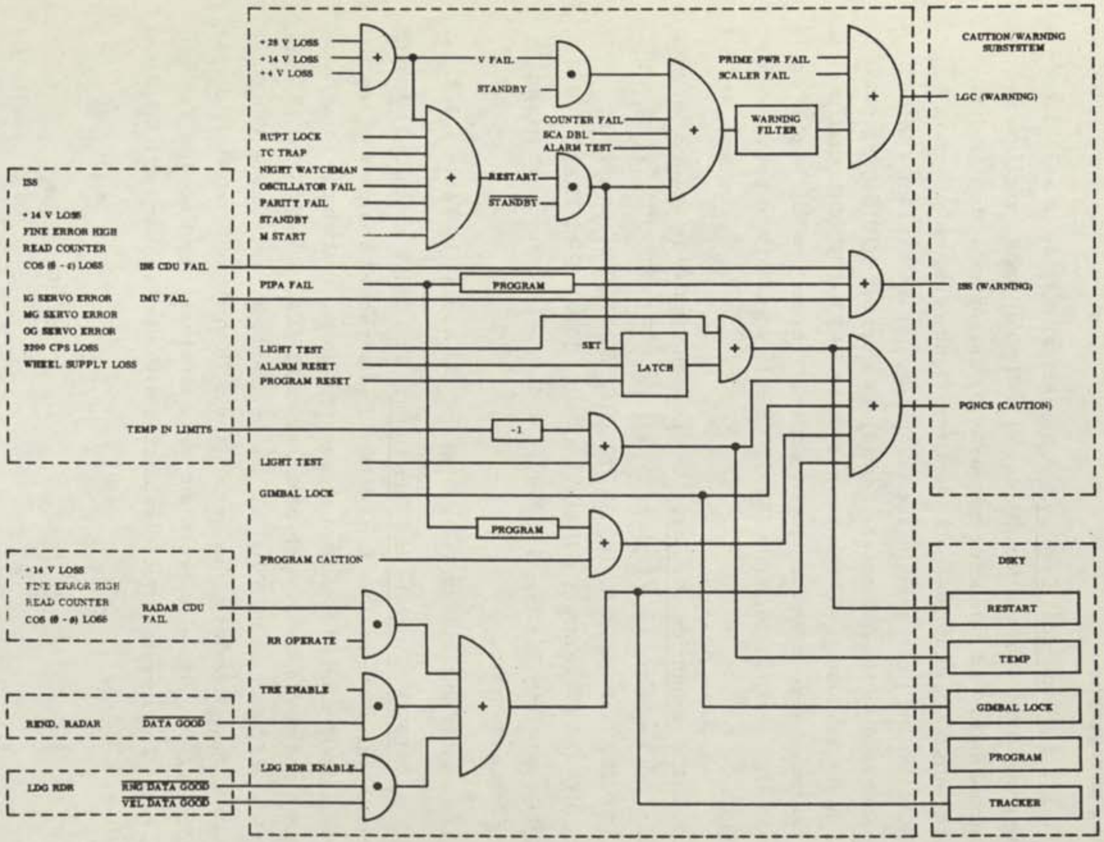


Figure 4.3.1.3-1. Caution and Warning Logic Diagram

Table 4.3.1.3-1. DSKY Condition Indicators

Indication	Function
UPLINK ACTY	Indicates that information is being received via UPLINK.
TEMP	Indicates that the stable member temperature has exceeded its design limits by ± 5 degrees Fahrenheit.
GIMBAL LOCK	Indicates that the middle gimbal has driven through an angle greater than ± 70 degrees from its zero position.
PROG	Indicates to the astronaut, that a program check has failed. This indicator is controlled by a computer program.
RESTART	Indicates: a) That a word has been incorrectly transferred from memory - Parity fail b) That the computer is in an endless control loop - TC Trap c) That the computer has been interrupted for 30 milliseconds - RUPT lock d) That the computer has not accomplished a CSS new job within 1.28 sec. (Night watchman) e) That a test alarm has been generated by program control
TRACKER	Indicates rendezvous radar CDU failure or improper data from rendezvous radar.
OPR ERR	Indicates that the Keyboard and Display program has encountered some improper operating conditions.
KEY REL	Indicates that the internal program has attempted to use the Keyboard and Display System and found it busy.
STBY	Indicates that the computer is in the standby condition.
NO ATT	Indicates to the astronaut that the ISS is not suitable for use as an attitude reference.

Table 4.3.1.3-2. DSKY Controls and Functions

Control	Function
0-9 keys	Used to enter data, address codes, and action request codes into the LGC.
VERB key	Conditions the LGC to interpret the following two numerical characters as an action request code and causes the verb display to be blanked.
NOUN key	Conditions the LGC to interpret the following two numerical characters as an address code and causes the noun display to be blanked.
ENTR key	Informs the LGC that the assembled data are complete and to execute the requested function or that requested action has been complied with.
(+) and (-) keys	Informs the LGC that the following data are decimal and indicates the sign of the data.
CLR key	Clears data contained in the data registers. Depressing the key clears whichever display register that is currently being used. Successive CLEARS clear the upper display register.
RSET key	Extinguishes the failure lights that are controlled by the LGC.
STBY	First depression puts computer in power-saving standby mode. Second depression causes the LGC to resume operation. (Depression of STBY will give an auto proceed except in PROG 06 which provides system deactivation.)
KEY REL key	Releases the DSKY displays initiated by keyboard action so that information supplied by the LGC program action may be displayed.

Table 4.3.1.3-3. Five-Bit Codes from DSKY

Key Identification	Code
0	00000
1	00001
2	00010
3	00011
4	00100
5	00101
6	00110
7	00111
8	01000
9	01001
Key release	11001
Verb	10001
Noun	11111
Enter	11100
Clear	11110
Reset	10010
+	11010
-	11011

4.3.1.3.4 General Characteristics

4.3.1.3.4.1 Mechanical Characteristics. The DSKY has an envelope 8.000 inches high by 8.124 inches wide by 6.900 inches deep. Physical appearance and front panel layout is shown in Figure 4.3.1.3-2. The weight of the DSKY, per MIT Report E-1142, April 1967, is 17.5 pounds.

4.3.1.3.4.2 Electrical Characteristics. The electrical characteristics are described under modules and power supplies as follows:

4.3.1.3.4.2.1 Modules. The DSKY is made of the following modular components.

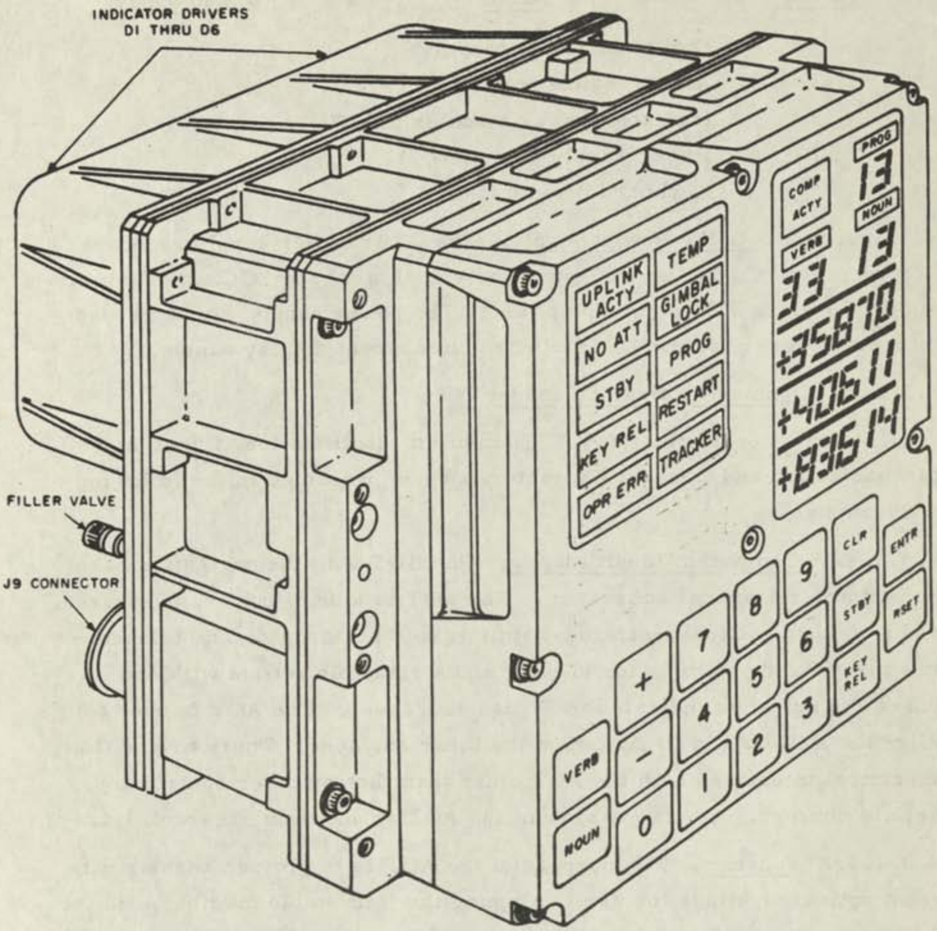


Figure 4.3.1.3-2. Display and Keyboard Assembly

Quantity	Name	Schematic
1	Digital indicator module	
1	Alarm indicator module	
6	Indicator driver modules (D1-D6)	
1	Power supply module (D7)	
1	Keyboard module (D8)	

4.3.1.3.4.2.2 DSKY Power Supply. The DSKY power supply receives +28 vdc, +14 vdc, and an 800-pps sync signal from the LGC. The power supply develops 275 v, 800 cps power. The power supply output is used to provide power for the DSKY electroluminescent display panels.

4.3.1.4 Alignment Optical Telescope

A brief description of AOT Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.4.1 Component Identification. The AOT with the navigation base constitutes the optical subsystem. The AOT is a unity power, 60-degree, field-of-view, optical instrument similar to a periscope. The telescope has three fixed positions for viewing and a rotatable reticle with cross-hairs and spiral as indicated in Figure 4.3.1.4-1. The AOT is used to align the IMU both in flight and on the lunar surfaces. There are no G&N electrical interfaces with the AOT other than the computer control and reticle dimmer. A cutaway view of the AOT is shown in Figure 4.3.1.4-2.

4.3.1.4.2 Function. The purpose of the AOT is to provide angular data from optical sightings for use in aligning the IMU stable member, via the LGC, to a reference coordinate system. These two alignment procedures are as follows.

4.3.1.4.2.1 Inflight Alignment. During inflight alignment, the AOT is placed in the right, left, or forward field-of-view position, and the reticle counter is set to the zero position. The LM vehicle is maneuvered in attitude until the target star is approximately centered in the AOT field of view. The attitude of the LM vehicle is then allowed to limit cycle so that the target star image crosses the reticle crosshairs as shown in view B of Figure 4.3.1.4-1. As the star image is coincident with the

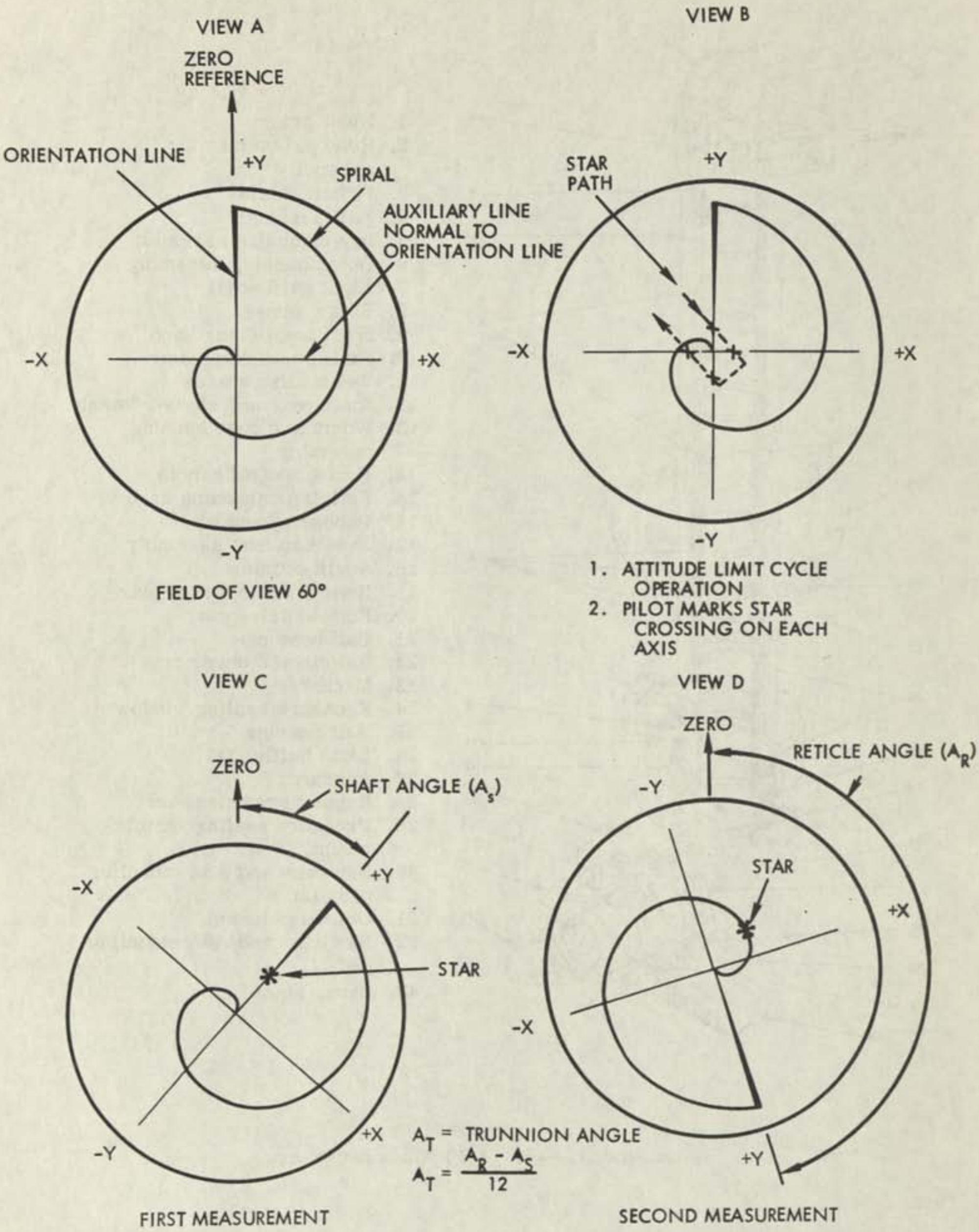
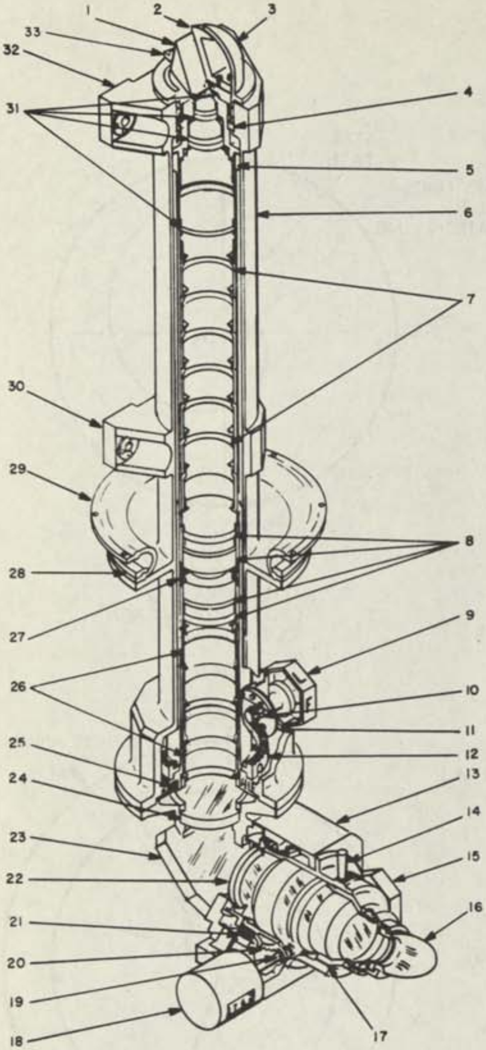


Figure 4.3.1.4-1. AOT Reticle Patterns



1. Head prism
2. Head prism housing assembly
3. Prism shield
4. Ball bearings
5. Inner housing assembly
6. Outer housing assembly
7. Light baffles (7)
8. Relay lenses
9. Shaft positioning knob
10. Shaft positioning gear
11. Detent disc spring
12. Shaft gear and slotted detent
13. Worm and gear housing assembly
14. Focus control handle
15. Reticle positioning knob
16. Rubber eyeguard
17. Eyepiece lens assembly
18. Angle counter
19. Reticle drive worm gear
20. Reticle drive gear
21. Ball bearings
22. Reticle and cover ring
23. Mirror
24. Pressure sealing window
25. Ball bearing
26. Light baffles (4)
27. Aperture
28. Rubber pressure seal
29. Pressure sealing vehicle mount
30. Nav base and ASA mounting pads (2)
31. Objective lenses
32. Nav base and ASA mounting pads (2)
33. Radar shield

Figure 4.3.1.4-2. AOT Cutaway View

Y-line, the astronaut presses the MARK-Y pushbutton on the computer control and reticle dimmer assembly located to the right of the AOT eyepiece. As the star image is coincident with the X-line, the astronaut presses the MARK-X pushbutton. The astronaut then inserts the star code number and the detent position into the LGC via the DSKY. Sighting must be taken on at least two stars to accomplish IMU alignment. The LGC then computes and issues the necessary gyro torquing signals to the IMU.

4.3.1.4.2.2 Lunar Surface Alignment. During lunar surface IMU alignment, the target star can be selected in either the AOT left, right, or forward positions. The astronaut using the manual drive knob adjusts the reticle to superimpose the orientation line of Y-line of the target star as shown in view C of Figure 4.3.1.4-1. This reticle angle displayed on the AOT counter is then inserted into the LGC via the DSKY. This provides the computer with star orientation angle (shaft angle). The astronaut then continues to rotate the reticle until a point on the spiral is superimposed on the target star. This second angular readout is then entered into the LGC via the DSKY. The AOT detent position and the star code number are also inserted into the LGC via the DSKY. The LGC can now calculate the angular displacement of the star from the center of the field of view by computing the difference between the two counter readings. As a result of the characteristics of the reticle spiral, this Δ angle is proportional to the distance of the star from the center of the field of view. Using this Δ angle and the proportionality equation, the computer is then able to calculate the trunnion angle. At least two star sightings are required for IMU coarse alignment through the CDU. Additional star sightings may be taken for IMU fine alignment, but these sightings may not be necessary. The LGC generates gyro torquing pulses to accomplish the IMU fine alignment.

4.3.1.4.3 Mechanization. The AOT is operated manually by the astronauts. The astronaut positions the prism and reticle with hand knobs and visually reads the angles of reticle rotation from the counter. Mark commands are initiated by the astronaut while viewing stars through the AOT. Reticle angle and shaft detent position for star sighting are manually inserted into the LGC via the DSKY. Mounted to the right side of the AOT is the computer control and reticle dimmer assembly. Located on this control box is a thumbwheel which provides the astronaut the means to adjust the brightness of the AOT reticle when taking a star sighting. The MARK-X and MARK-Y pushbuttons are also located on this assembly and are used by the astronaut to send discrete signals to the LGC when making star sightings for an IMU inflight alignment. A REJECT pushbutton is also available, should an invalid mark descrete be sent to the LGC.

4.3.1.4.3.1 Field of View. The shaft axis of the telescope is parallel to the X-axis of the LM. To insure that a sunlit lunar terrain does not interfere with star sightings when the LM is on the lunar surface, the bottom of the field of view is set 15 degrees above the LM vehicle Y-Z plane. Therefore, the center of the 60-degree field of view will form an angle of approximately 45 degrees with the LM thrust, or X-axis. By means of a detent knob, the astronaut may rotate the telescope head assembly about the shaft axis. This shaft angle rotation is shown in Figure 4.3.1.4-3 and is detented at four positions: (1) the vehicle X-Z plane, (2) 60 degrees to the left, (3) 60 degrees to the right, and (4) 180 degrees reversed (the latter for protection during non-use). By using detents, the three viewing positions are accurately known.

4.3.1.4.4 General Characteristics

4.3.1.4.4.1 Mechanical Details. The AOT is essentially an "L" shaped device approximately 36 inches in length, consisting of two perpendicular intersecting cylinders each of which is a major assembly, an upper section, and the eyepiece. Structural components such as housing and mounts are machined from beryllium, while spacers and similar parts are made of aluminum and stainless steel.

The upper section might be further broken down into the two housing assemblies. The inner housing assembly is mounted using ball bearings within the outer housing assembly. The outer housing consists of an outer tube and the positioning mechanism. The outer tube is basically a beryllium cylinder approximately 27 inches in length with a 3-inch internal diameter and a wall thickness of approximately 0.1 inch. The prism

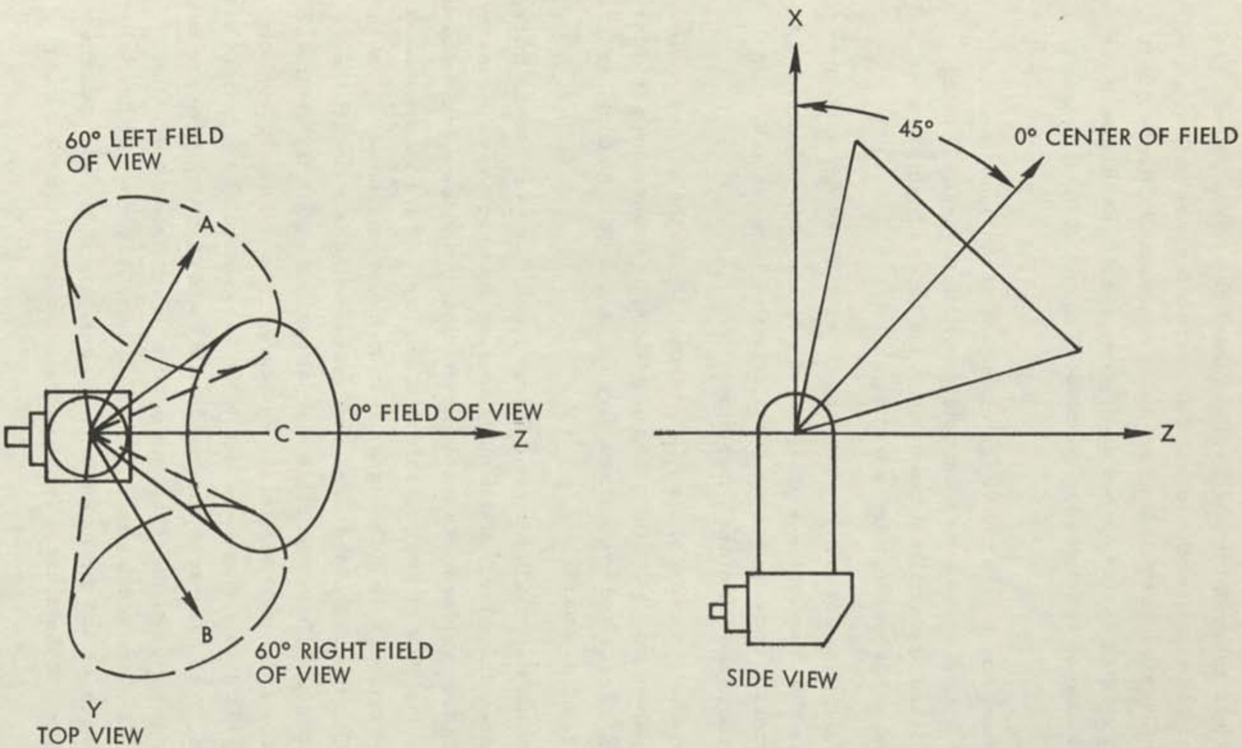


Figure 4. 3. 1. 4-3. AOT Articulation

shield, which is spherical in shape and cut back to allow a 180-degree view, is mounted on top of the tube. The positioning mechanism provides the means of rotating the prism about the shaft axis and detenting in it any of three viewing angles spaced 60 degrees apart. Each side of the hexagonal hand knob is lettered to correspond to one of these detent positions.

The inner housing, which is the main part of the upper section, contains the optical components and rotates within the outer housing. Centrally aligned and axially located within the inner housing are the objective and relay lens and the head prism.

The relay lens assembly is positioned near the bottom of the inner tube with the objective lens assembly above it. The near prism and its mounting form the uppermost part of the objective lens assembly and protrude through the top of the outer tupe.

The eyepiece section of the AOT, through which the astronaut views the superimposed images of the star and reticle, is broken down into three assemblies: mirror and window housing, worm and gear housing, and lens housing and eyeguard.

In addition to providing a precise location and support for the mirror and the window, the mirror and window housing also provides the means of connecting the eyepiece with the upper section, provides a seal between the two sections, and makes possible in a single structure the transition between the vertical axis of the upper section to the horizontal axis of the eyepiece. The function of the worm and gear housing is to locate the reticle and provide the means of its manual positioning via the control knob mounted on the right side of the eyepiece. The counter located on the left side displays a visual readout of the angle to which the reticle has been rotated. The purpose of the lens housing and eyeguard is to project the star image to the plane of the exit pupil where it is viewed by the astronaut. The lens housing is a cylindrical beryllium housing into which the eyepiece lenses and their aluminum spacers are clamped. Attached to the outer end of the housing are the focusing control and the rubber eyeguard.

4.3.1.4.4.1.1 Bellows Assembly. A bellows assembly formed from an elastomeric material in a semi-toroidal shape is used to form a pressure seal between the AOT and the LM hull. This seal also provides strain isolation between the AOT and the hull.

4.3.1.4.4.1.2 Weight. The weight of the AOT assembly, per MIT Status Report E-1142, April 1967, is 23.1 pounds plus 1.6 pounds for the computer control and reticle dimmer.

4.3.1.4.4.2 Optical Characteristics. The optical characteristics of the AOT are summarized as follows:

a)	Magnification:	unity
b)	Field of view:	60 degrees
c)	Field coverage in 3 positions:	180 degrees in azimuth x 60 degrees in elevation
d)	Eye relief:	from lens 1.45 inch from eyepiece 0.875 inch
e)	Rotating illuminated reticle with readout	

4.3.1.4.4.2.1 Optical Construction Details. A schematic diagram of the AOT optics is shown in Figure 4.3.1.4-4. The main components of the system are the head prism, objective lens assembly, relay lens assembly, and eyepiece assembly.

The head prism is fixed in elevation and, in conjunction with the inner tube, is movable in azimuth. Acting as a mirror, the prism through its internal reflections merely diverts the image into the telescope inner tube.

The objective lens assembly provides for focusing of the image at the eyepiece side of the field lens. Image diameter at the first focal plane is approximately 0.0006 inch. The six lens elements of the objective lens assembly are coated black on the periphery for eliminating scattered light in the optical path.

The purpose of the relay lens assembly is to transfer and focus the image at the second focal plane located at the AOT reticle. A glass window

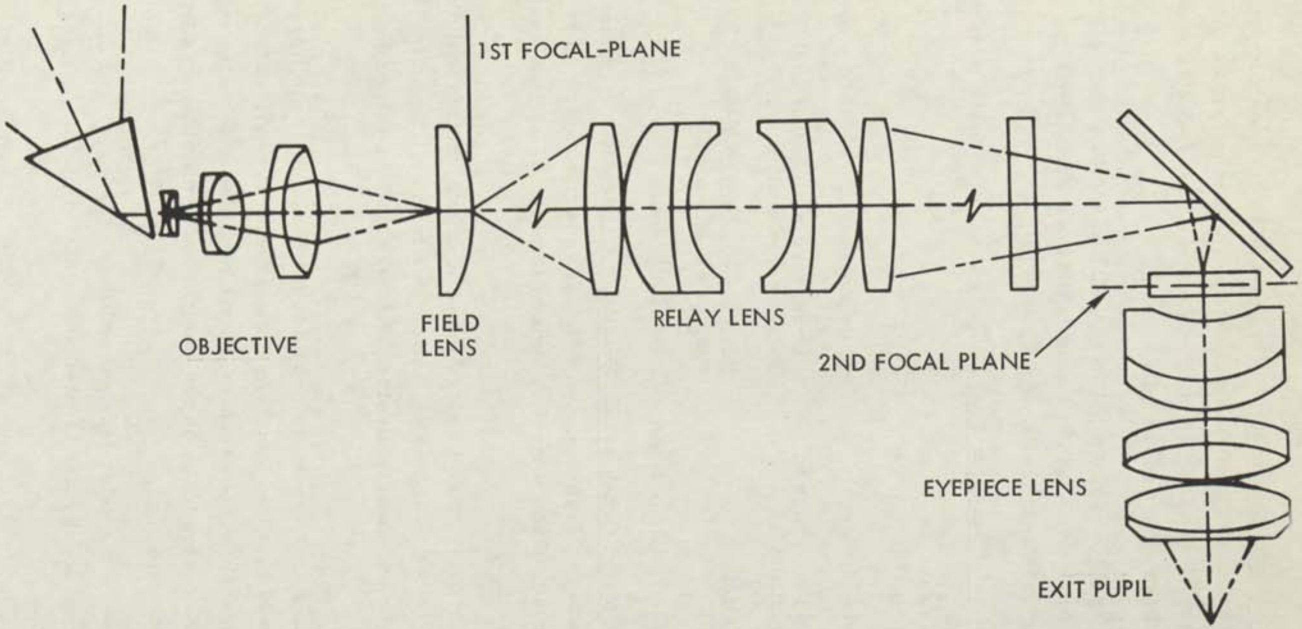


Figure 4.3.1.4-4. AOT Optical Schematic

is mounted between the relay lens and eyepiece to provide a seal between the two assemblies. Also located between the relay lens and eyepiece is a specially coated beryllium mirror fixed at an angle of 45 degrees. This mirror reflects the image from the relay lens into the eyepiece.

4.3.1.4.4.2.2 Reticle Pattern. The AOT reticle pattern is shown in Figure 4.3.1.4-1 and illustrates the manner in which it is used for IMU alignment on the lunar surface as described in Paragraph 4.3.1.4.2.2. The reticle pattern consists of crosshairs and a superimposed Archimedes spiral. During inflight alignment, the astronaut takes optical sightings by marking on a star viewed with respect to the crosshairs. During stay on the lunar surface, the astronaut uses the Archimedes spiral for alignment sightings. The spiral is constructed to depart radially from the center as a linear function of rotation about the center. By turning a knob near the eyepiece, the astronaut can rotate the entire reticle pattern about the center of the field of view. A micrometer-type readout is provided near the knob to indicate the angle of reticle rotation.

The vertical crosshair is an orientation line designated the Y-line and is parallel to the LM X-axis when the reticle is at the 0-degree reference position. The horizontal crosshair is an auxiliary line designated the X-line and is perpendicular to the orientation line. A one-turn spiral is superimposed from the center of the field of view to the top of the vertical crosshair. The reticle is edge illuminated during use by reticle lamps.

4.3.1.5 Coupling Data Unit

A brief description of CDU Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.5.1 Component Identification. The CDU is a solid state, digital device for transferring angular information from the IMU gimbal resolvers and the rendezvous radar shaft and trunnion resolvers to the LGC, to the SCS, and to the crew displays. The CDU also receives digital commands from the LGC and generates analog errors for positioning the IMU gimbals in the "coarse align" mode. There are five CDU's contained in the CDU package, with modules common to all five circuits.

4.3.1.5.2 Function. The five CDU's are the interface units between the digital computer and variables of the rest of the G&N. Each CDU has two sections.

a) Analog-to-digital section (A/D) accepting two-speed resolver data transmission and feeding angle incremental pulse to the appropriate computer counter.

b) A digital-to-analog section (D/A) accepting incremental pulse of the variables of concern and generating an 800-cps suppressed carrier voltage proportional to the count of the input pulses. Each D/A output has a synchronous demodulator associated with it to generate a ground isolated proportional DC voltage.

The five LM CDU's are used within the G&N system in the configuration shown in Figure 4.3.1.5-1. For a more detailed CDU system drawing refer to CDU two-wire mechanization diagrams, drawing numbers 2015566 and 2015567. The specific function for each CDU is as follows:

CDU #1

(A/D) Read IMU inner gimbal angles from 1- and 16-speed resolvers to LGC.
(D/A) Send pitch error from LGC to FDAI pitch attitude error needle. During coarse align mode of G&N send inner gimbal error signal to IMU servo.

CDU #2

(A/D) Read IMU middle gimbal angle from 1- and 16-speed resolvers to LGC.
(D/A) Send yaw error from AGC to FDAI yaw attitude error needle. During coarse align mode of G&N, send middle gimbal error signal to IMU servo.

CDU #3

(A/D) Read IMU outer gimbal angle from 1- and 16-speed resolvers to LGC.
(D/A) Send roll error from AGC to FDAI roll attitude error needle. During coarse align mode of G&N, send outer gimbal error signal to IMU servo.

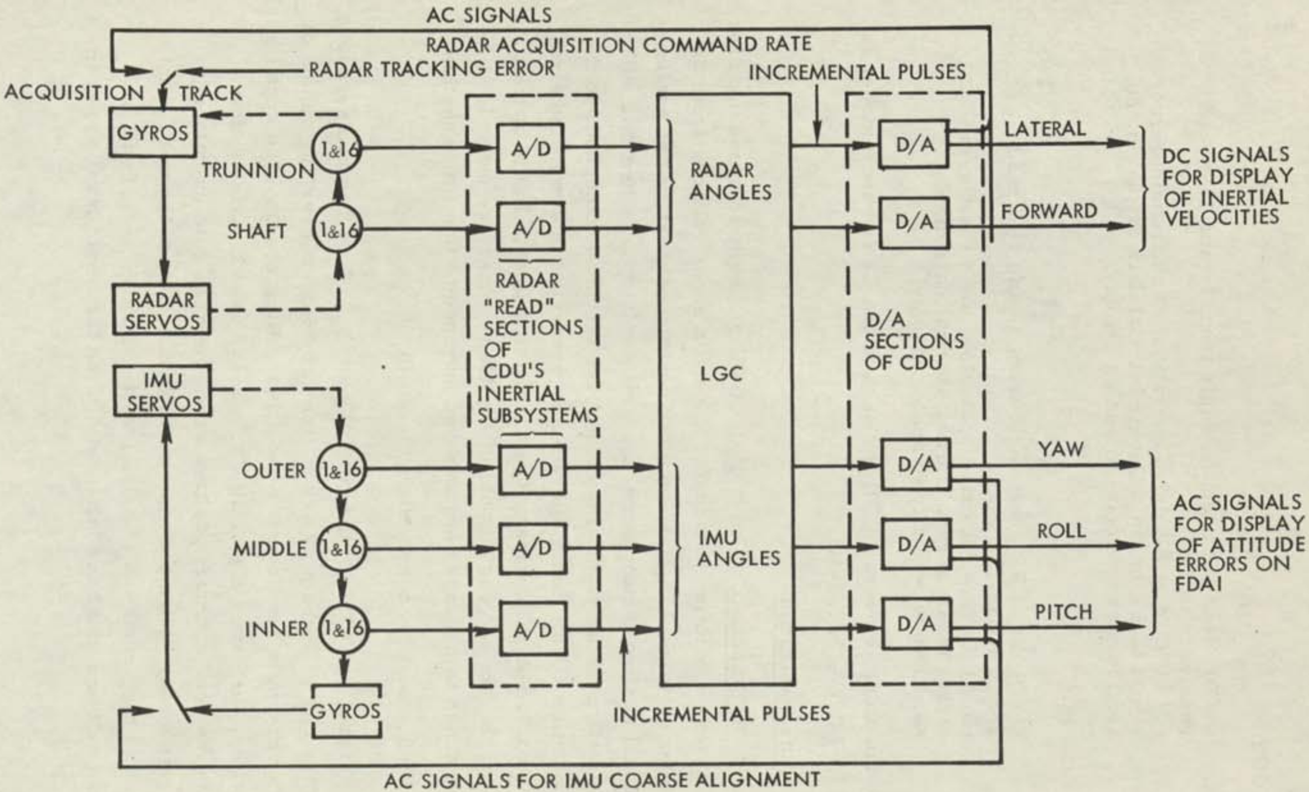


Figure 4.3.1.5-1. LM CDU Configuration

CDU #4

(A/D) Read RR trunnion angle from 1- and 16-speed resolvers to LGC.
(D/A) During RR drive operation, send trunnion error signal to RR servo. During inertial data display mode, send forward velocity to analog display.

CDU #5

(A/D) Read RR shaft angle from 1- and 16-speed resolvers to LGC.
(D/A) During RR drive operation, send shaft error signal to RR servo. During inertial data display, send lateral velocity to analog display.

The functions to overall CDU for each of the G&N system modes is tabulated in Table 4.3.1.5-1.

4.3.1.5.3 Mechanization

4.3.1.5.3.1 A/D Section. The digital - CDU principle is based upon the use of ac analog computing techniques to derive error voltages from an analog ladder network. Digital circuitry continuously alters the analog computational scaling factor to null the error voltages. When this null occurs, the state of the digital memory drive circuit corresponds to the resolver position. This basic operation is presented in simplified form in Figure 4.3.1.5-2, Elementary CDU Digitizing Loop. In the operation of this loop, the sin and cos outputs of the 1X resolver are passed through switched impedance networds and summed to form the function

$$\sin \theta_1 \cos \psi_1 - \cos \theta_1 \sin \psi_1 = \sin (\theta_1 - \psi_1).$$

The signal, $\sin (\theta_1 - \psi_1)$, will null when $\psi_1 = \theta_1$. The two networks representing $\cos \psi_1$ and $\sin \psi_1$ are switched through converging values of ψ_1 by the shaft-angle read counter until this nulling occurs. To accomplish this operation, the error signal $\sin (\theta_1 - \psi_1)$ is used to gate up-down pulses into the read counter; positive error voltages gate up pulses, negative gate down. When the most significant 7 bits of the read counter correspond to the actual shaft angle, the signal $\sin (\theta_1 - \psi_1)$ will have approached null and will have fallen within the threshold level shown in

Table 4.3.1.5-1. LM CDU System Functions

G&N System Modes	Description	Computer Discretes: ISS CDU Zero	Coarse Align Enable	ISS Error Counter Enable	Rendezvous Radar (RR) CDU-Zero	RR Errors Counter Enable	Display Inertial Data	IMU Cage
A. Inertial Subsystem (ISS) CDU-Zero	Initialized the ISS-computer interface by clearing all three IMU Gimbal "Read" (A/D) counters simultaneously, also includes an ambiguity control logic function.	X						
B. Coarse Alignment	Positioning of the IMU gimbals in response to computer incremental commands. The CDU includes rate limiting and position feedback.		X	X				
C. Fine Alignment	Precision Inertial member alignment by Gyro Pulse Torquing. The CDU A/D converters function to read the IMU gimbal angles and update the computer.							
D. Inertial Operation	ISS CDU A/D converters read IMU gimbal angles and update the computer.							
E. ISS Turn On	Initializes the Inertial Subsystem by orthogonalizing the gimbals, clearing the CDU read counters, and providing a delay for gyro run up and inertial component suspension settling. (The auto cage and delay feature is accomplished by PSA - computer discretes listed below. *)	X	X					
F. IMU Cage	An emergency recovery function for a tumbling IMU, allows orthogonalization of the IMU gimbals by activation of a manual control switch. Also allows for placing system on Inertial Control for Backup Total Attitude & Attitude Error Display when computer is inoperative.							X
G. Attitude Error Display	Provides D/A conversion for computer calculated attitude errors for display on the LM FDAI during G&C autopilot control of LM.			X				
H. Rendezvous Radar (RR) CDU-Zero	Clears both rendezvous radar "Read" counter simultaneously, also includes an ambiguity control function.				X			
I. RR Designate	Positioning of the radar antenna for automatic target acquisition.					X		
J. Display Inertial Data	Display of computer derived inertial forward and lateral velocities on LM control panel meters during landing operations.						X	

*ISS Turn On - Auto Cage Discretes:
a) ISS OPERATE POWER ON - sensed in PSA gimbals are automatically caged.
b) ISS TURN ON REQUEST TO COMPUTER - initiates time delay count down in computer.
c) TIME DELAY COMPLETE TO PSA - resets auto cage circuit allows ISS to be aligned and placed in Inertial Control.

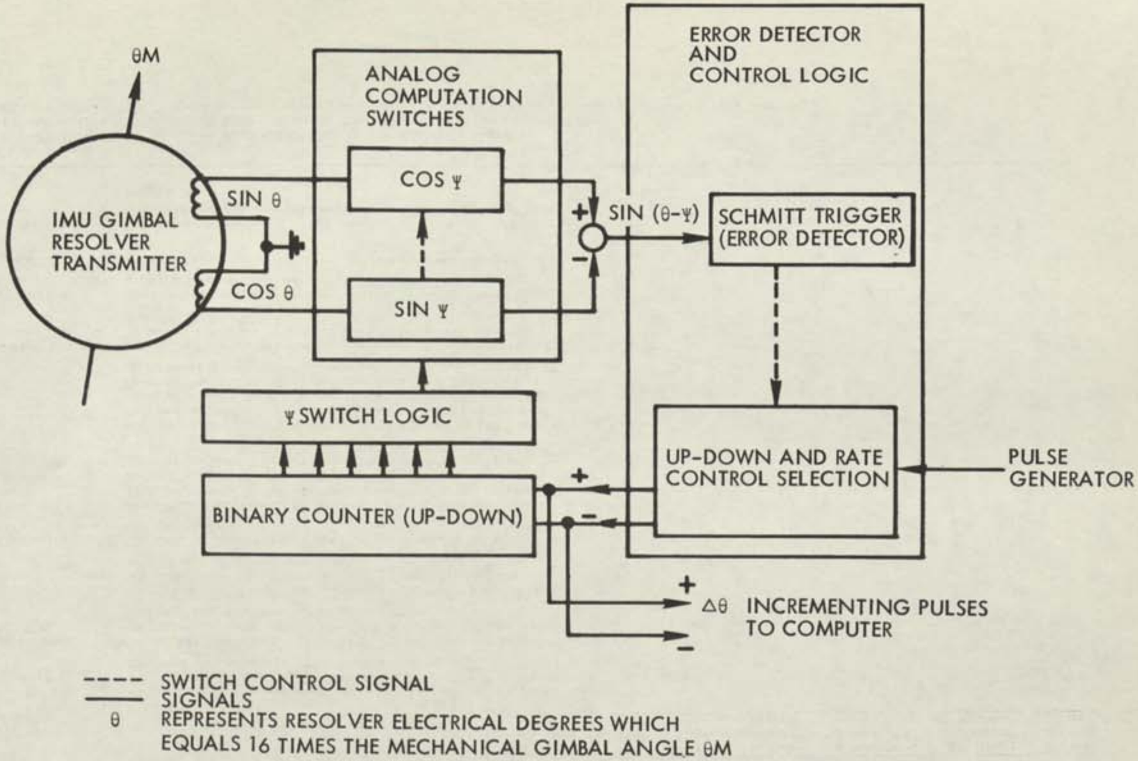


Figure 4.3.1.5-2. Elementary CDU Digitizing Loop

the error signal path. In order to obtain a better approximation to the true angle, this basic scheme is altered somewhat in the coarse and fine portions of the CDU. The coarse configuration is illustrated in Figure 4.3.1.5-3. This circuit is similar to Figure 4.3.1.5-2 with the addition of a circuit which achieves a final null by using $K\phi$, a small angle approximation to $\sin(\theta - \psi)$. This is implemented by using bits 9 through 12 to switch the $K\phi$ network to the appropriate value. The final output error of the coarse loop going to the error detector is

$$\epsilon = \sin(\theta_1 - \psi_1) - K\phi_1.$$

The fine system operates in a similar manner, only the signals are received from the 16X resolvers, and the final error nulling is more accurate. Bits 7 through 11 control the $\cos\psi_{16}$ and $\sin\psi_{16}$ networks to bring the read counter within 0.7 degree of the true shaft reading. The final null is achieved using a small angle approximation to $\sin(\theta_{16} - \psi_{16})$. This is implemented by switching the single network $K\phi_{16}$ using the least significant 7 bits of the read counter. This constant, $K\phi_{16}$, plus bias K_1 is multiplied by $\cos(\theta_{16} - \psi_{16})$ and nulled against $\sin(\theta_{16} - \psi_{16})$ to produce a final error signal

$$\epsilon = \sin(\theta_{16} - \psi_{16}) - \left[K\phi_{16} + K_1\right] \cos(\theta_{16} - \psi_{16}).$$

The total mechanization of the CDU read function is shown in Figure 4.3.1.5-4.

The pulse train into the read counter operates at 800 or 12,800 pps, depending upon the magnitude of the error signal. The pulse train entering the read counter is also sent to the computer. The pulses are accumulated in the computer in a similar manner, affecting shaft-angle reading. The contents of the shaft-angle read computer can be decoded for display on the DSKY, giving the crew a decimal display of the shaft angles.

4.3.1.5.3.2 D/A Section. Command pulses are received by the CDU's from the LGC in a serial pulse train. These pulses are accumulated in an error counter in the CDU. The error counter controls a ladder network which effects the D/A conversion. This basic D/A function is shown in Figure 4.3.1.5-5.

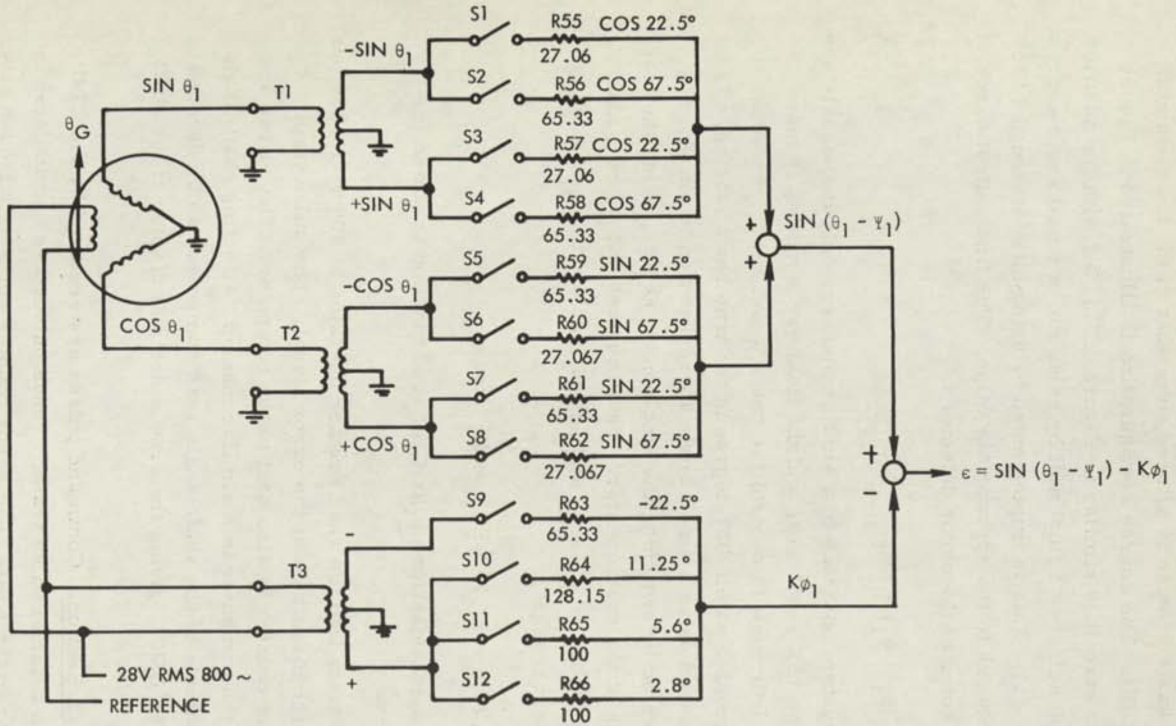


Figure 4. 3. 1. 5-3. CDU Coarse Error Mechanization

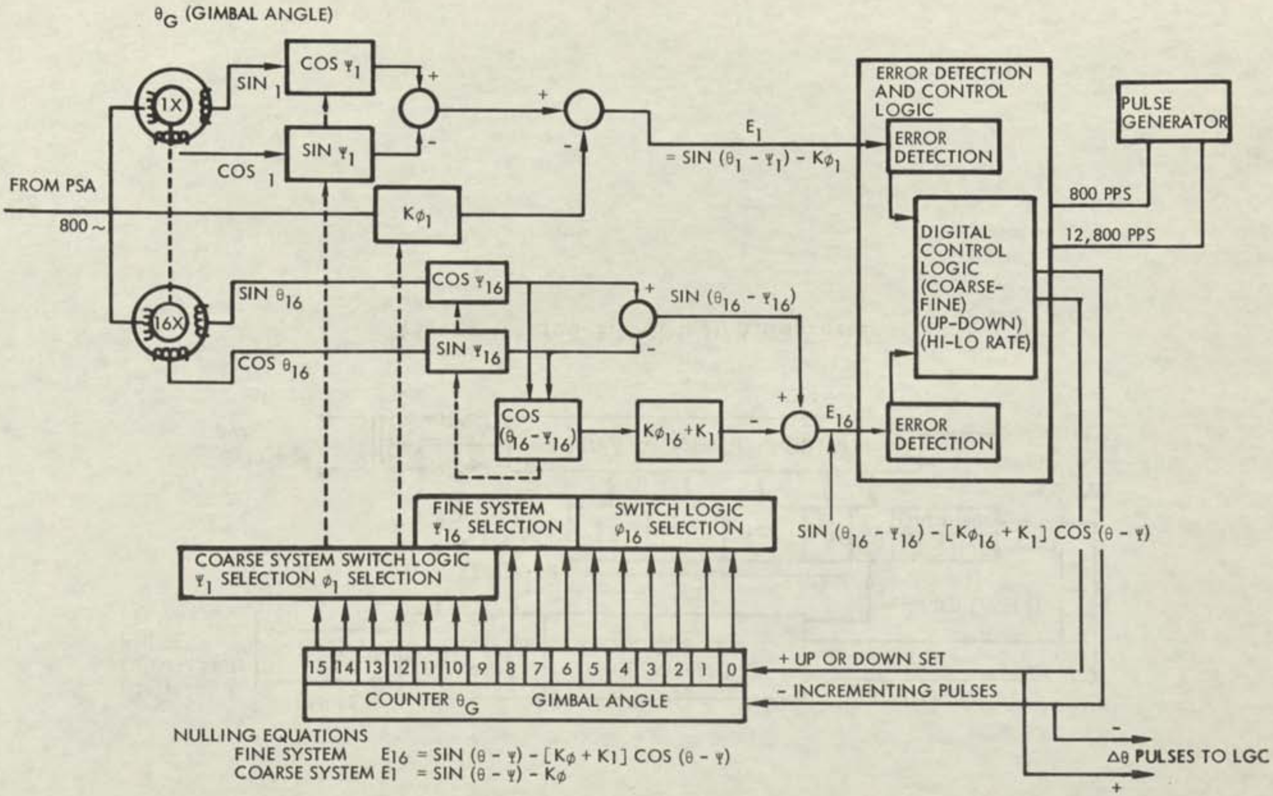


Figure 4.3.1.5-4. CDU Shaft-Angle Encoding (read) Logic

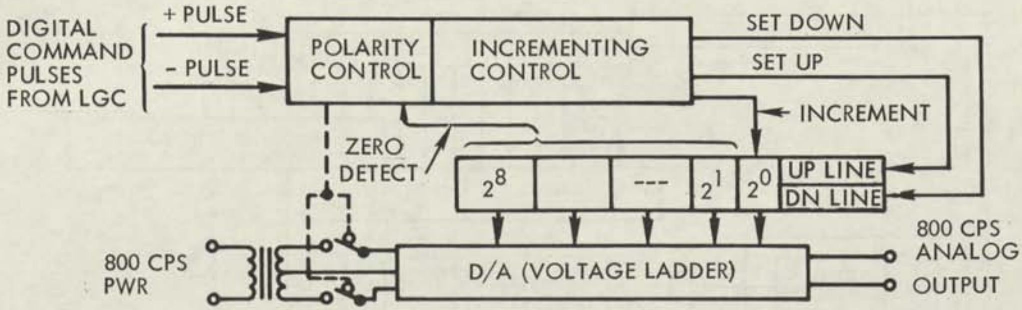


Figure 4.3.1.5-5. CDU D/A Converter

The integration of the D/A converter in the IMU gimbal loop is shown in Figure 4.3.1.5-6. This circuit shows the demodulator where a dc error signal can be derived from the 800-cps output.

4.3.1.5.4 General Characteristics

4.3.1.5.4.1 Mechanical Characteristics. The CDU is built from solid-state electronic modules. The modules are assembled and connected to wiring headers within a sealed metal package.

4.3.1.5.4.1.1 Weight. The weight of the complete CDU package, per MIT Status Report E-1142, April 1967, is 37.2 pounds. The outline of the CDU is shown in Figure 4.3.1.5-7.

4.3.1.5.4.2 Electrical Characteristics

4.3.1.5.4.2.1 Components. The 34 modules which make up the complete CDU package are as follows:

Quantity	Name	Part No.	Function
5	Coarse System	2007236	Provides coarse switching and attenuation circuitry necessary to increment angles.
5	Main Summing Amplifier and Quadrature Rejection	2007238	Provides fine switching and attenuation circuitry necessary to increment angles.
5	Quadrant Selector	2007243	Provides fine switching and attenuation circuitry necessary to increment angles.
5	Read Counter	2007140	Accumulates pulses representing angles and controls switching of coarse system module, quadrant selector module, and main summing amplifier.

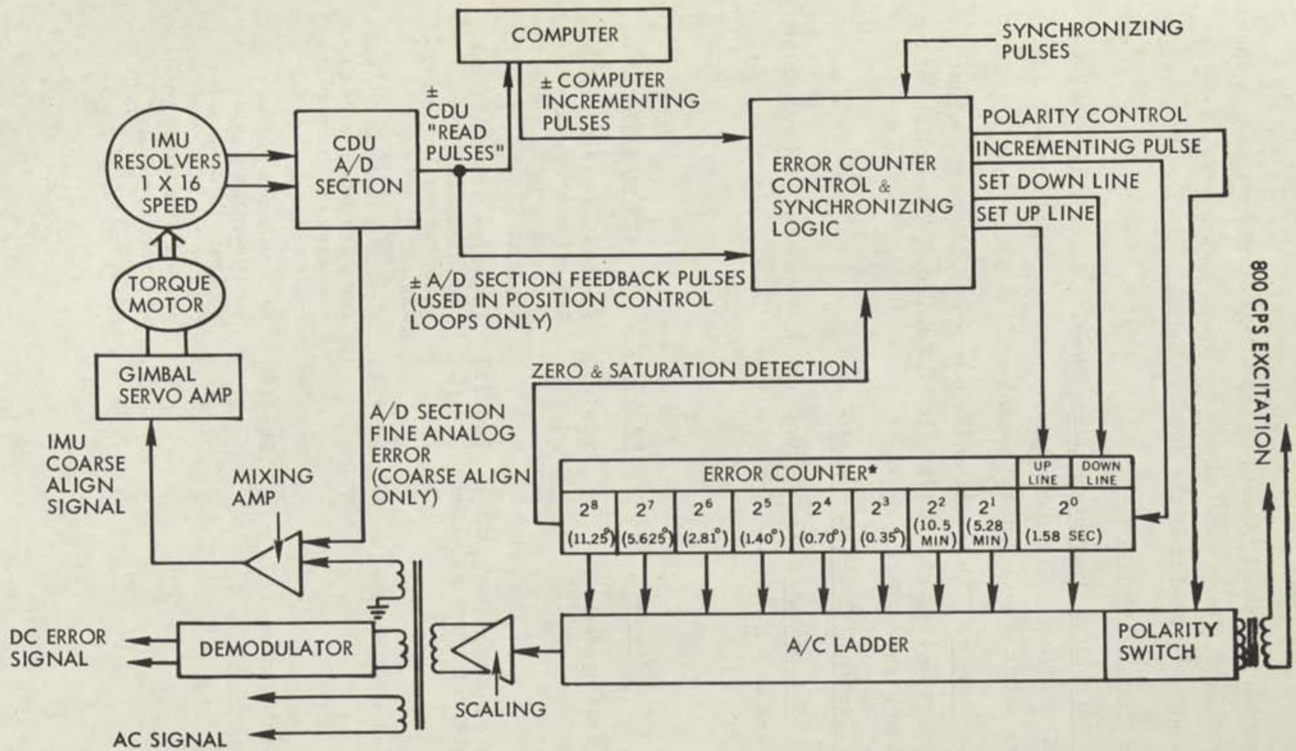


Figure 4.3.1.5-6. CDU D/A Section

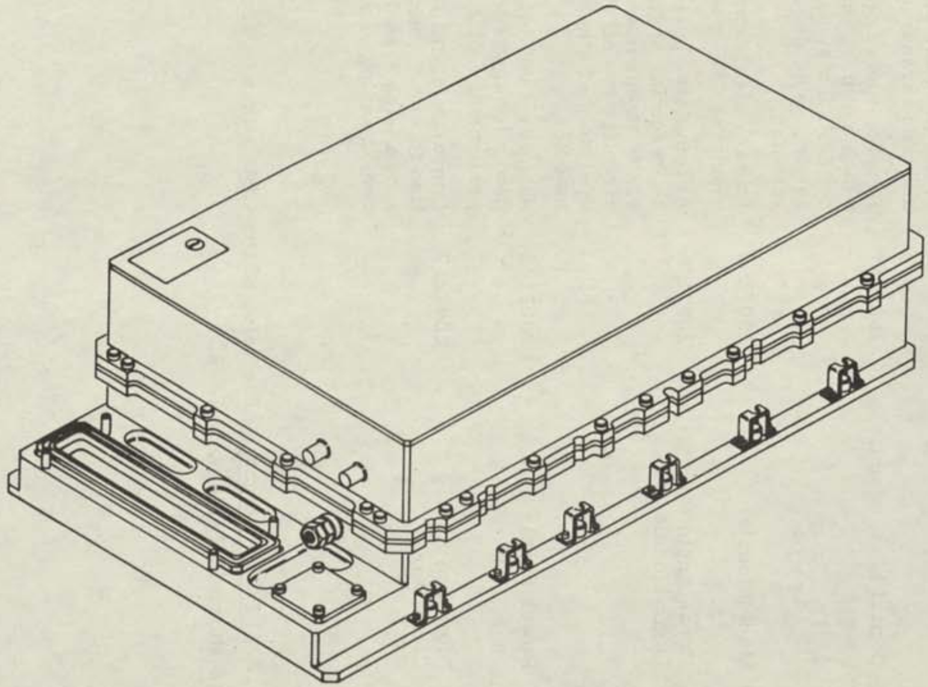


Figure 4.3.1.5-7. Coupling Data Unit

Quantity	Name	Part No.	Function
1	Interrogate Module	2007263	Generates interrogate pulses required for CDU operation, produces 14 vdc power, and provides circuitry for data and pulse transmission.
1	Digital Mode Module	2007141	Provides pulse commands which are used throughout CDU for synchronization, switching, and strobing.
1	Mode Module	2007254	Buffers signals and monitors CDU operations.
5	Error Angle Counter and Logic Module	2007139	Accumulates pulses representing angular error and provides logic circuitry to control operation of other CDU modules.
1	Power Supply	2007142	Supplies 4 vdc logic power to digital logic portions of CDU.
5	Digital to Analog Converter	2007237	Converts digital information in error counter into a dc analog signal and two ac analog signals.

4.3.1.5.4.2.2 Pulse Weights. The value of the CDU pulses to/from the LGC are as shown in Table 4.3.1.5-2.

Table 4.3.1.5-2. CDU Pulse Weights

a)	A/D OPERATIONS - (CDU TO COMPUTER)	
	1) Inertial Subsystem	39.6 sec
	2) Rendezvous Radar	39.6 sec
	3) Optics Shaft	39.6 sec
b)	D/A OPERATIONS (COMPUTER TO CDU)	
	1) IMU Coarse Alignment, IMU Gimbal Angle	158.4 sec
	2) Radar Acquisition, Antenna Rate	8.22 sec/sec
	3) Attitude Error Angle Display	158.4 sec
	4) Lateral and Forward Velocity Display	0.52 ft/sec

4.3.1.6 Pulse Torque Assembly

A brief description of PTA Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.6.1 Component Identification. The PTA is an electronics package, which contains serialized circuitry associated with particular serialized inertial components in the IMU. In addition, it contains the gyro calibration module portion of the stabilization loops.

4.3.1.6.2 Function. The electronics associated with the PIPA loops provide the torquing signals for the accelerometers and also provide the control and signal interface with the LGC.

The electronics associated with the stabilization loops perform a similar function on a time shared basis. The gyro calibration module provides torquing signals to the IRIG's and provides the control and signal interface with the LGC.

4.3.1.6.3 Mechanization

4.3.1.6.3.1 PIPA Loops. The PTA contains electronics for the three PIPA loops, one per each axis.

Figure 4.3.1.6-1, LM Accelerometer Loops, illustrates the configuration for each loop and also shows the interfaces of the PTA with the LGC and the IMU. Block diagrams of the individual PTA modules are shown in Figures 4.3.1.6-2, 3, 4 and 5. For detailed diagram of LM PIPA loops, refer to drawing 6015563, "Two-wire Mechanization of Apollo PIPA Loops (LM)."

4.3.1.6.3.2 Stabilization Loops. The PTA contains a gyro calibration module, a dc differential amplifier, the PVR, and a binary current switch for torquing the IRIG's in the IMU. These components are used on a time-shared basis under the control of the LGC. The configuration of these modules is illustrated in Figure 4.3.1.6-6. For a detailed diagram of LM stabilization loops refer to drawing 6015564, "Two-wire Mechanization of Apollo Stabilization Loops (LM)."

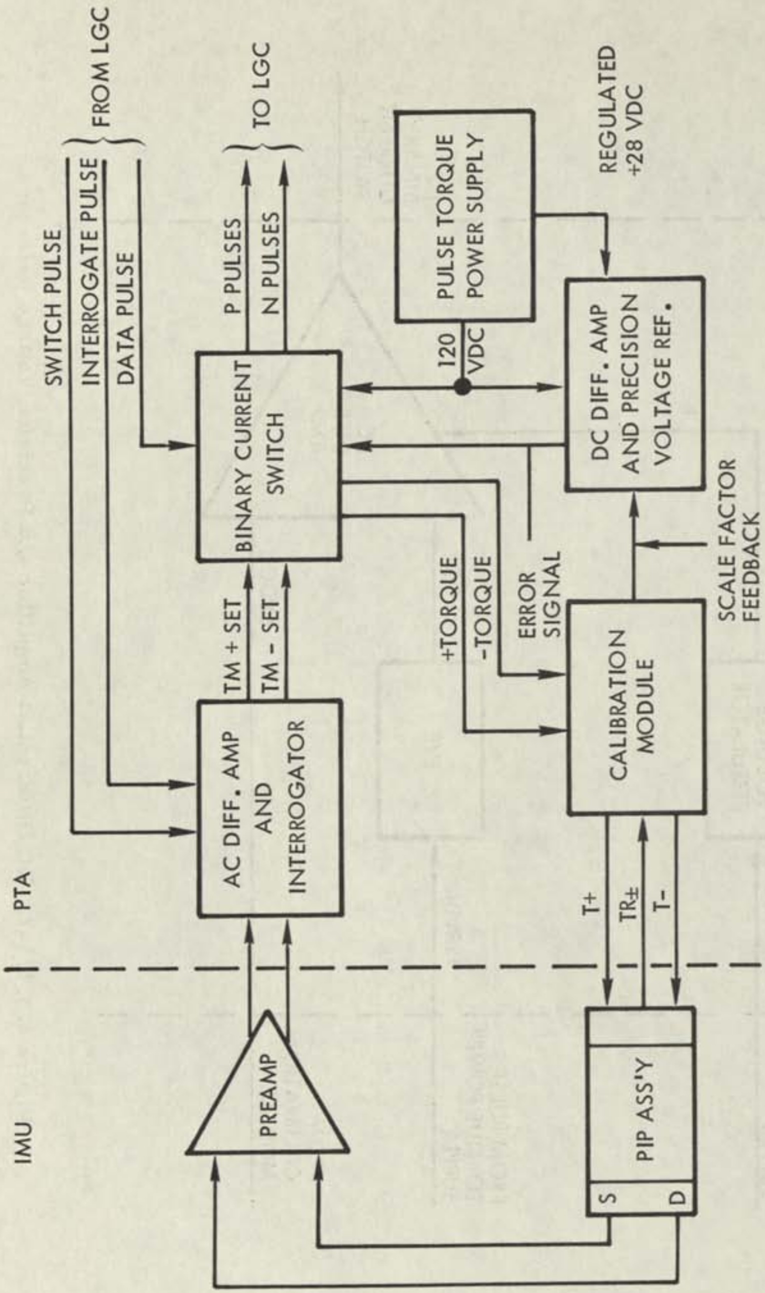


Figure 4.3.1.6-1. LM Accelerometer Loop

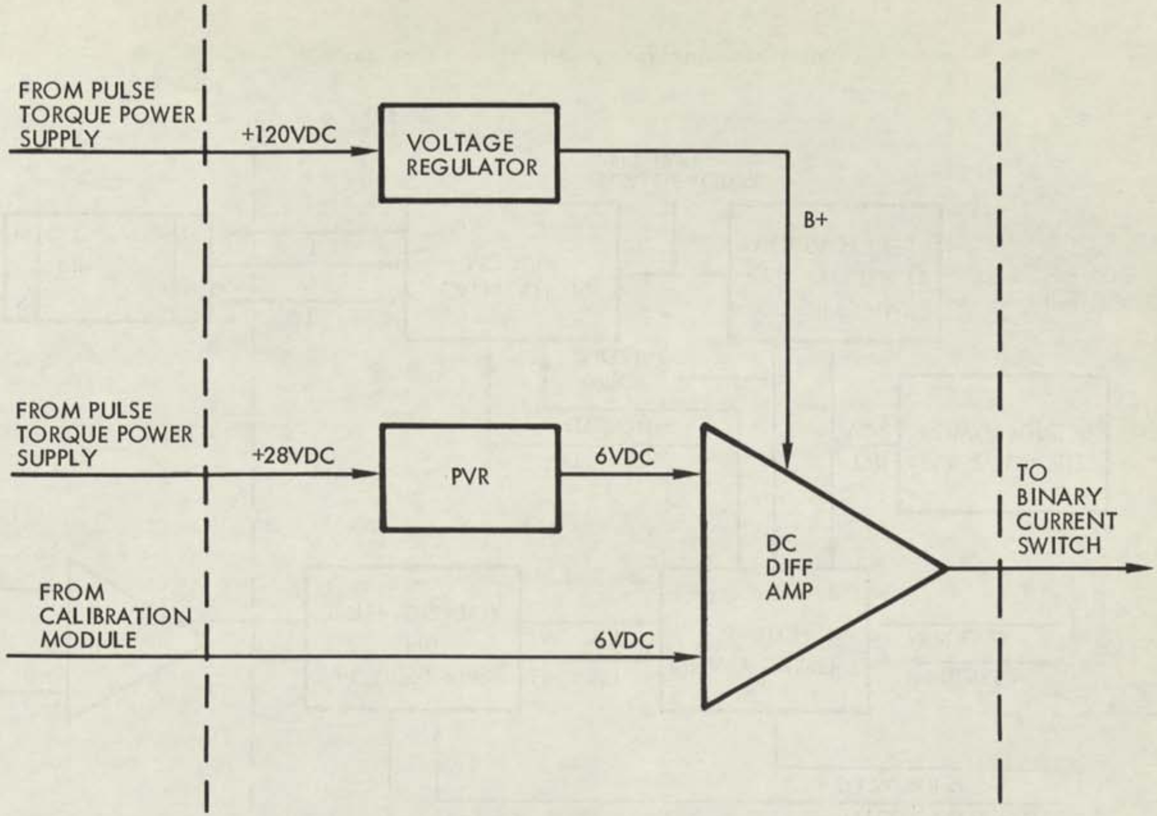


Figure 4.3.1.6-2. DC Differential Amplifier and Precision Voltage Reference

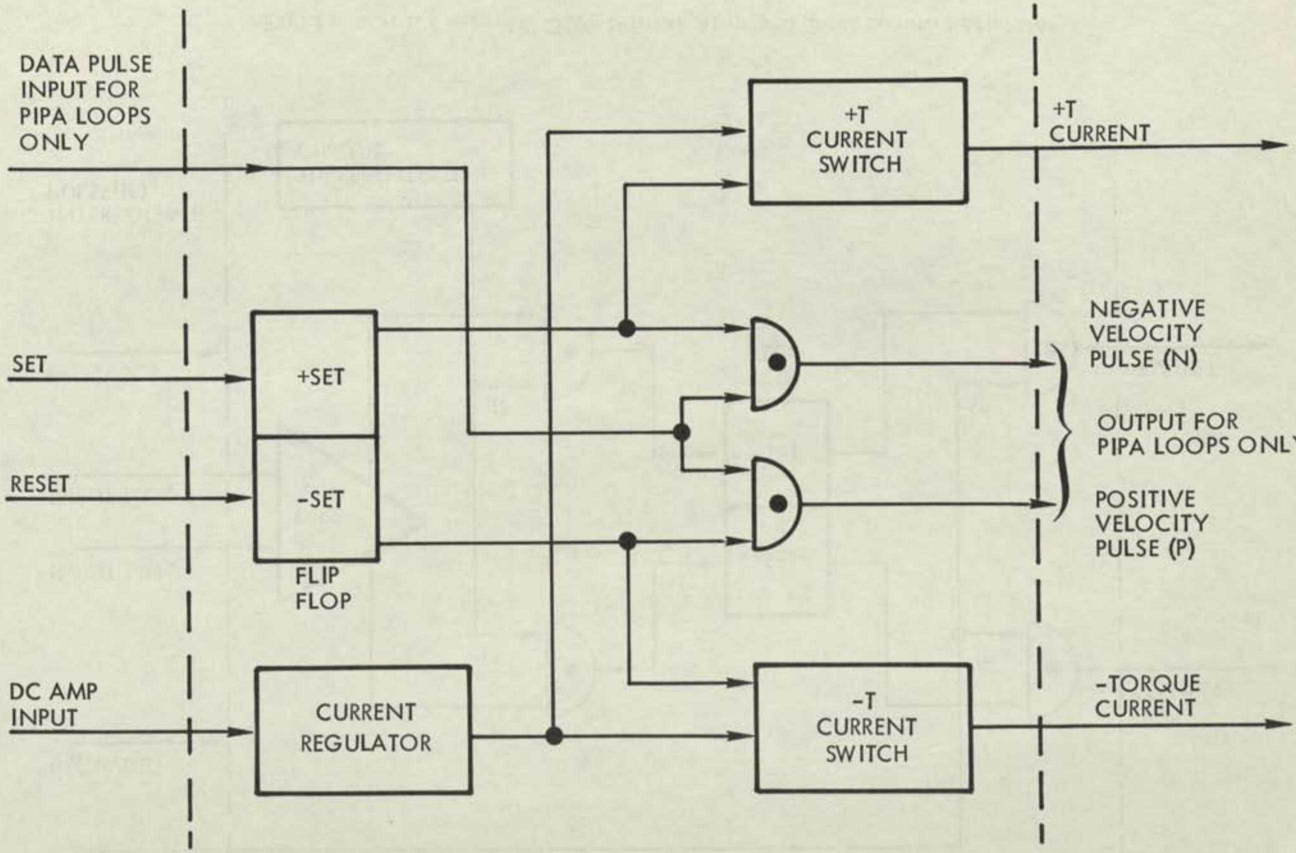


Figure 4.3.1.6-3. Binary Current Switch

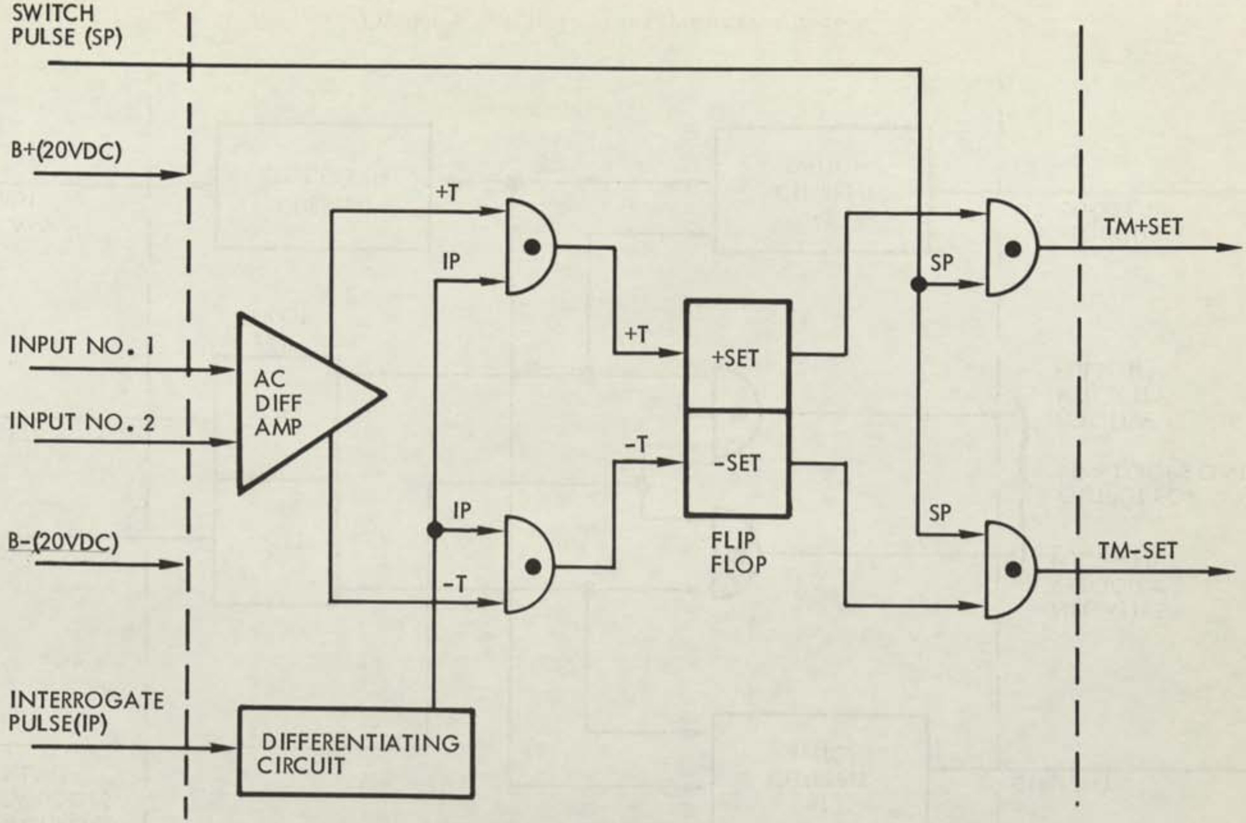


Figure 4.3.1.6-4. AC Differential Amp and Interrogator Module

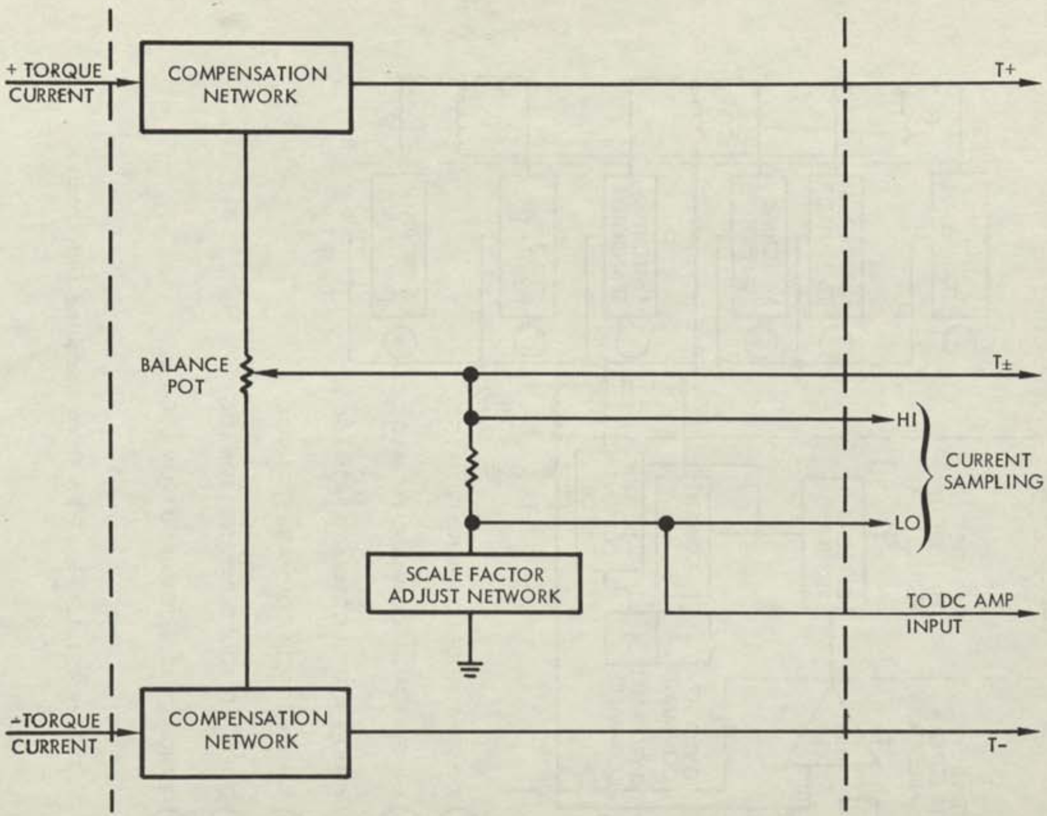


Figure 4. 3. 1. 6-5. PIPA Calibration Module

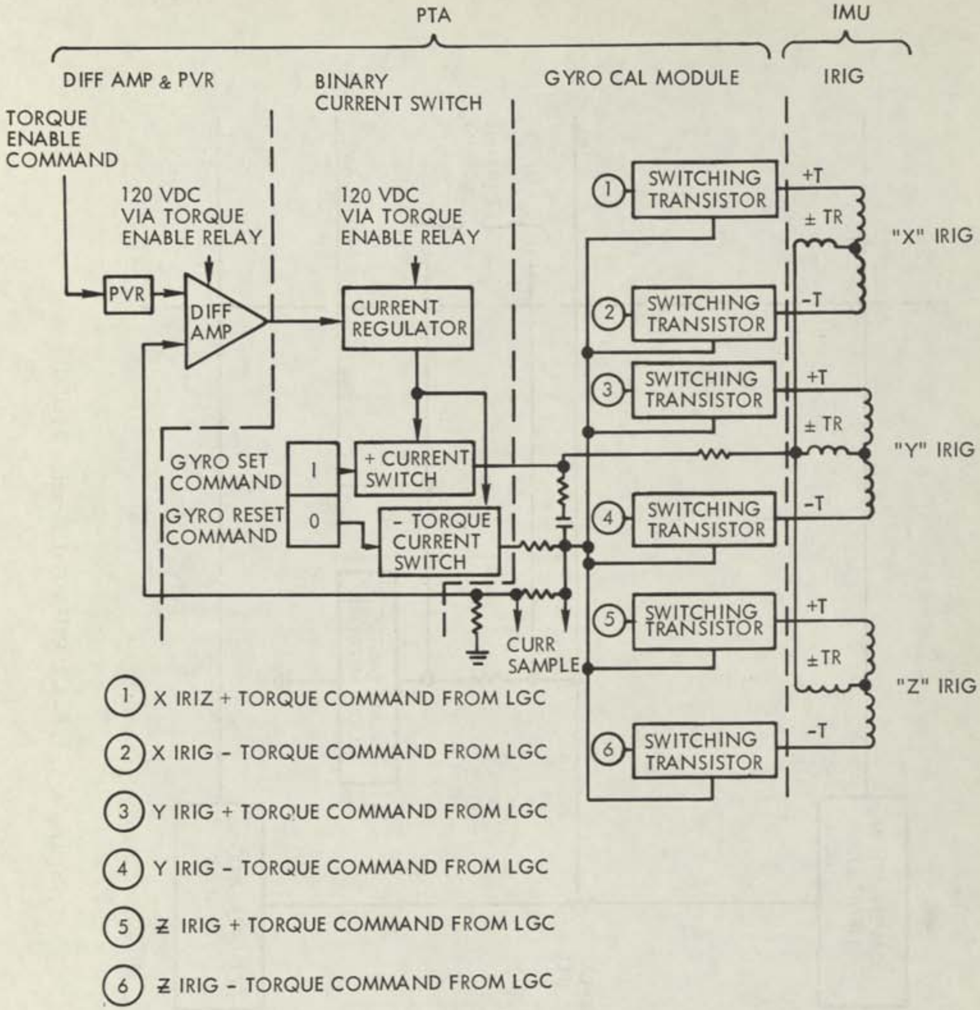


Figure 4.3.1.6-6. PTA Gyro Torquing Electronics

4.3.1.6.4 General Characteristics

4.3.1.6.4.1 Mechanical Characteristics. The PTA consists of modular electronic packages contained in a moisture proof sealed container. The packaging is similar to that for the PSA. An exploded view of the PTA is shown in Figure 4.3.1.6-7. An overall outline of the PTA is shown in Figure 4.3.1.6-8. The weight of the PTA, per MIT Status Report E-1142, April 1967, is 14.3 pounds.

4.3.1.6.4.2 Electrical Characteristics. The PTA contains 17 modules, which plug into a single-level interconnection header. (For details refer to interconnect list for PTA header, Drawing No. 6014522.) The PTA modules and their individual schematic references are listed below:

Quantity	Name	Part No.	Function
4	Binary Current Switch	2007103	One furnishes torquing current to the three Apollo II IRIG's, and three furnish torquing current to the individual 16 PIP's.
4	dc Differential Amplifier and Precision Voltage Reference	2007101	Regulate torquing current supplied through binary current switches.
3	ac Differential Amplifier and Interrogator	2007104	Amplify accelerometer signal generator signals and convert them to plus and minus torque pulses.
1	Gyro Calibration Module	2007102	Applies plus or minus torquing current to Apollo II IRIG's when directed by LGC.
1	Pulse Torque Power Supply	2007106 (used with system P/N 6015000-011 2007166 (used with system P/N 6015000-021 and above)	Supplies 120 vdc to dc differential amplifier and binary current switch during gyro torquing routines. Also supplies +20 and -20 vdc to the 16 PIP's and 28 vdc to the X, Y, and Z PIP PVR's.

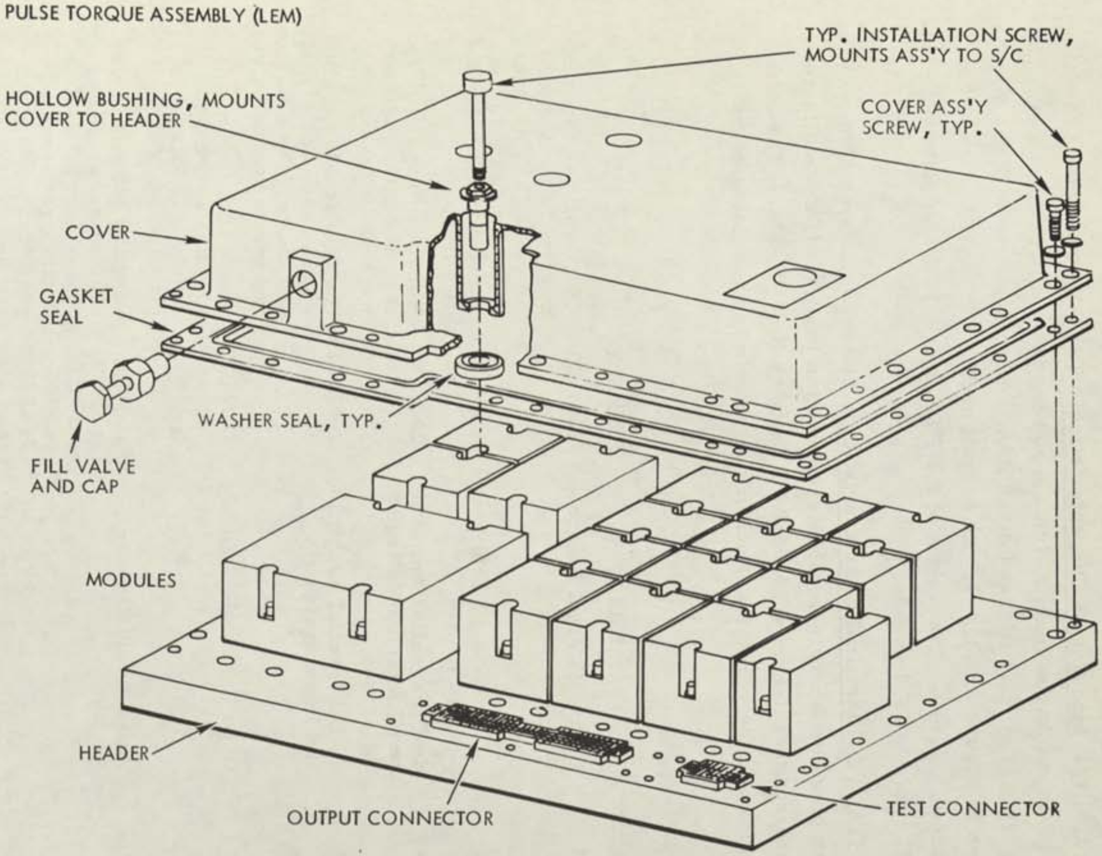


Figure 4.3.1.6-7. Exploded View, LM PTA

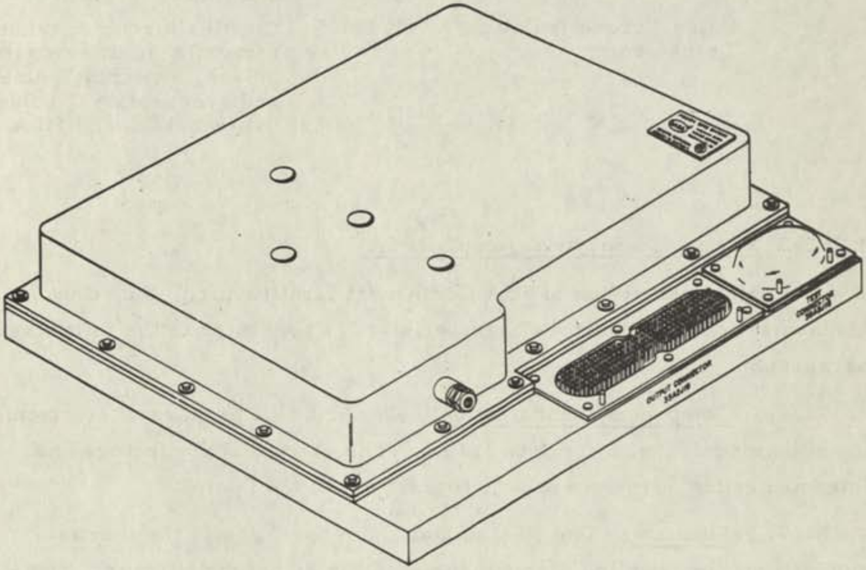


Figure 4.3.1.6-8. Outline of PTA

Quantity	Name	Part No.	Function
3	PIPA Calibration Module	6007105	Compensate for differences in inductive loading of accelerometer torque generator windings and regulate balance of plus and minus torques.
1	Pulse Torque Isolation Transformer	6007005	Couples torque commands, data pulses, interrogate pulses, switching pulses, and synchronizing pulses between LGC and PTA.

4.3.1.7 Power and Servo Assembly (PSA)

A brief description of PSA Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.7.1 Component Identification. The PSA is a package of electronics modules mounted on a common frame. The electrical connectors and interconnecting harness are an integral part of the frame.

4.3.1.7.2 Function. The PSA supports the operation of the inertial subsystem, the coupling display unit, and the signal conditioner. The PSA also furnishes 800 cps reference power to the GASTA and the FDAI's when switched to PGNCS data lines.

4.3.1.7.3 Mechanization. The power section of the PSA contains necessary multivibrators, phase and amplitude control, amplifiers and filters to supply regulated 800 cps, and 3200 cps power for the system. The balance of the power section contains a -28 vdc power supply.

The servo section of the PSA contains electronics to supply the operation of the inertial measurement unit. For mechanization of this section, refer to the Two-wire Mechanization Diagram for the LM IMU Stabilization Loops, Drawing No. 6015564, and the Two-wire Mechanization Diagram for the LM PIPA Loops, Drawing No. 6015563.

4.3.1.7.4 General Characteristics

4.3.1.7.4.1 Mechanical Characteristics. The PSA contains two types of modular electronic packages: (1) a heat sink type, where the components are surrounded by magnesium or aluminum heat sink material and (2) a frame type, where the components are in a more conventional "cordwood" arrangement. These modular packages plug into a single level header which provides necessary intermodular connections and also bring out signals through a wiring harness for external connection to other portions of the system. The PSA is enclosed in a moisture proof, sealed container. This packaging is illustrated by Figure 4.3.1.7-1.

4.3.1.7.4.1.1 Outline. See Figure 4.3.1.7-2.

4.3.1.7.4.1.2 Weight. The weight of the PSA, per MIT Status Report E-1142, April 1967, is 17.5 pounds.

4.3.1.7.4.2 Electrical Characteristics. There are 14 electronic modules in the PSA; these modules and their respective part numbers are identified as follows:

Quantity	Name	Part No.	Function
1	-28 vdc Power Supply	2007107	Supplies power to gimbal servo amplifiers and pulse torque power supply.
1	3200 cps, 1 Percent Amplifier	2007108	Supplies 28 volts, 3200 cps to ducosyn transformer on stable member and to the gimbal servo amplifiers.
1	3200 cps Automatic Amplitude Control (AAC), Filter and Multi-vibrator	2007109	Regulates operation of 3200 cps, 1 percent amplifier.
1	800 cps, 1 Percent Amplifier	2007110	Supplies 28 volts, 800 cps for IMU resolver excitation. (Also supplies this power for FDAI and SCS reference signals.) Provides reference signal for two 800 cps, 5 percent amplifiers.

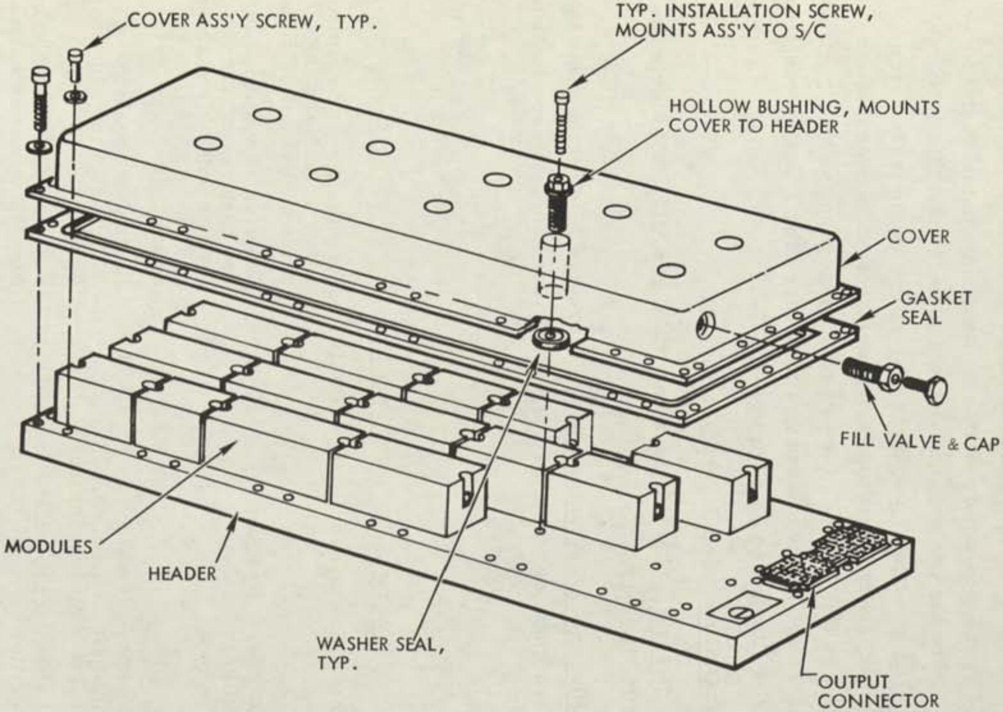


Figure 4.3.1.7-1. Exploded View LM PSA

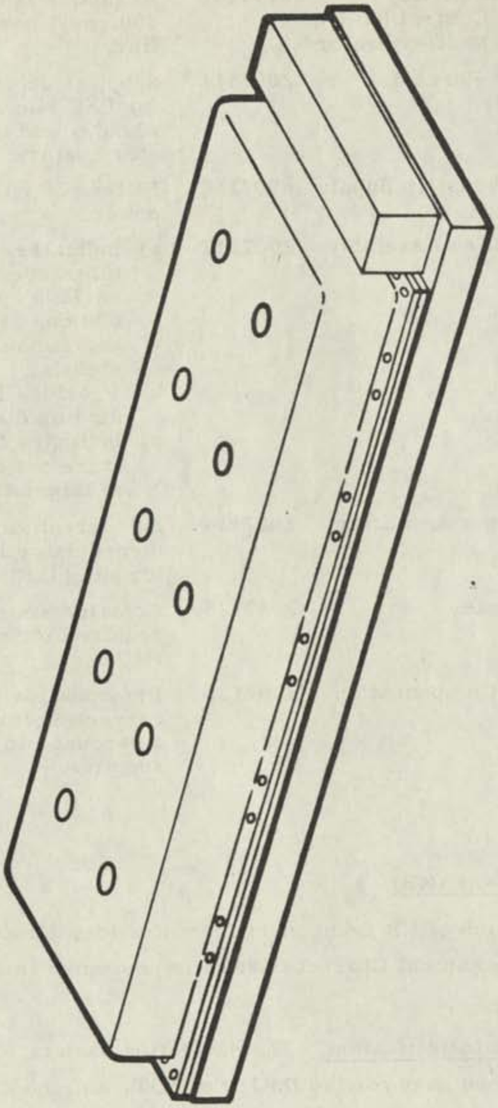


Figure 4.3.1.7-2. Power and Servo Assembly

Quantity	Name	Part No.	Function
1	800 cps Automatic Amplitude Control (AAC), Filter and Multi-vibrator	2007112	Regulates operation of 800 cps 1 percent amplifier.
2	800 cps, 5 Percent Amplifier	2007111	Supplies 28 volts, 800 cps for IMU blower, gyro wheels, and accelerometer heaters.
1	PGNCS Subsystem Supply Filter	6007114	Filters 28 volt prime power.
1	IMU Auxiliary Assembly	2007117	a) Indicates out of tolerance conditions on 3200 cps 28 v (rms), 800 cps 28 v (rms), and gimbal error signals. b) Provides IMU turn-on moding discrete. c) Indicates IMU temperature out of tolerance to telemetry.
3	Gimbal Servo Amplifier	2007114	dc current amplifier with demodulator input for torquing IMU gimbals.
1	Relay Module	2007123	Contains six relays required for moding of IMU.
1	IMU Load Compensation	2007132	Provides power factor correction for 800 cps, 1 percent and 5 percent supplies.

4.3.1.8 Navigation Base (NB)

A brief description of NB Component Identification, Function, Mechanization and Mechanical Characteristics is presented in the following paragraphs.

4.3.1.8.1 Component Identification. The navigation base is a fixture which provides a rigid common base for the IMU, the AOT, and the Abort Sensor Assembly. The NB is attached to the spacecraft by means of a navigation base support.

4.3.1.8.2 Function. The NB establishes a fixed coordinate system with respect to the LM body axis for IMU alignment.

4.3.1.8.3 Mechanization. The navigation base is attached to the LM vehicle so that a navigation base axes coordinate system is formed by its mounting points. The Y_{NB} axis is defined by the centers of the two upper mounting points and is parallel to the vehicle Y_{LM} axis. The X_{NB} axis is defined by a line through the center of the lower mounting pad and perpendicular to the Y_{NB} axis and is parallel to the vehicle X_{LM} axis. The Z_{NB} axis is mutually perpendicular to X_{NB} and Y_{NB} and is parallel to the vehicle Z_{LM} axis. This axes system is illustrated in Figure 4.2.1.1-1.

4.3.1.8.4 Mechanical Characteristics. The navigation base is a toroidal aluminum alloy ring with:

a) Four tabular aluminum posts to provide for IMU mounting

b) Four tabular aluminum posts for AOT and ASA mounting

c) Three aluminum inserts to provide strain isolation ball mounting to the Navigation Base Support

The navigation base is shown in Figure 4.3.1.8-1. The optics are mounted on the four posts on the opposite end of the IMU.

The navigation base support is shown in Figure 4.3.1.8-2.

The complete assembly of IMU, Optics, Navigation Base, and Navigation Base Support is illustrated in Figure 4.3.1.8-3.

4.3.1.8.4.1 Weight. The weight of the NB (not including support) per MIT Status Report E-1142 dated April 1967 is 5.1 pounds.

4.3.1.9 Signal Conditioner Assembly (SCA). A brief description of SCA component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs.

4.3.1.9.1 Component Identification. The SCA is an electronics package, mounted "piggyback" on the PSA, which is located on the back wall of the Lunar Module.

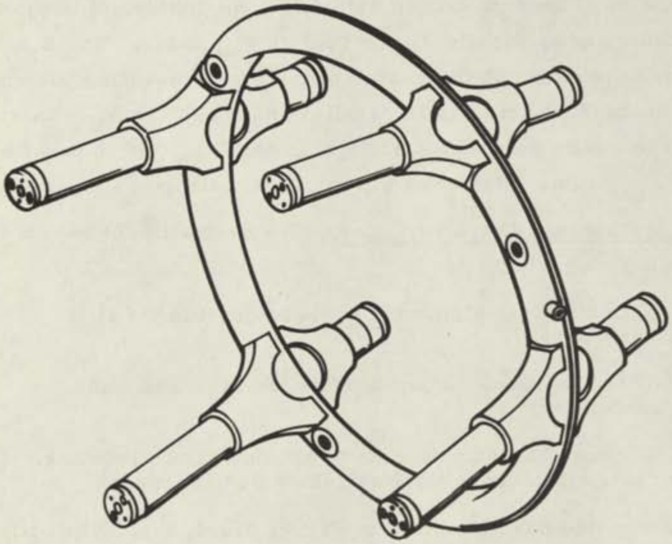


Figure 4.3.1.8-1. Navigation Base Assembly, P/N 6899980

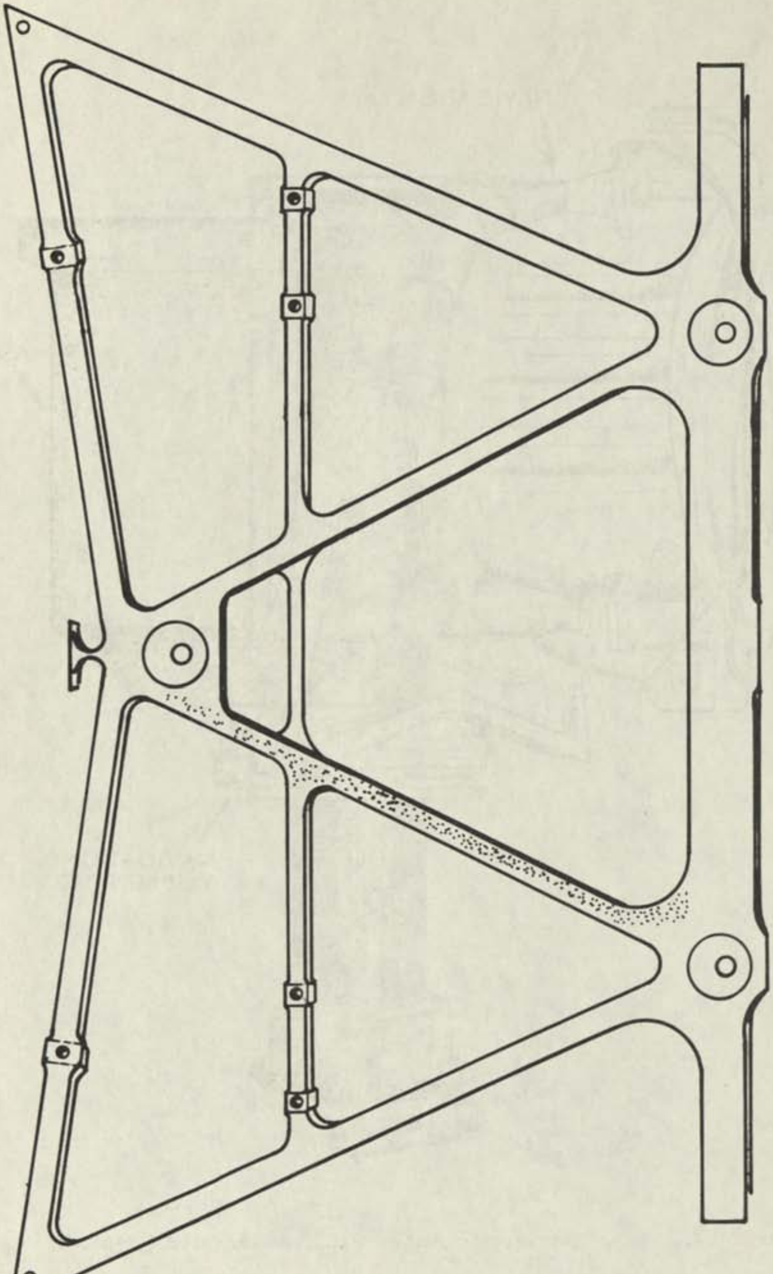


Figure 4. 3. 1. 8-2. Navigation Base Support

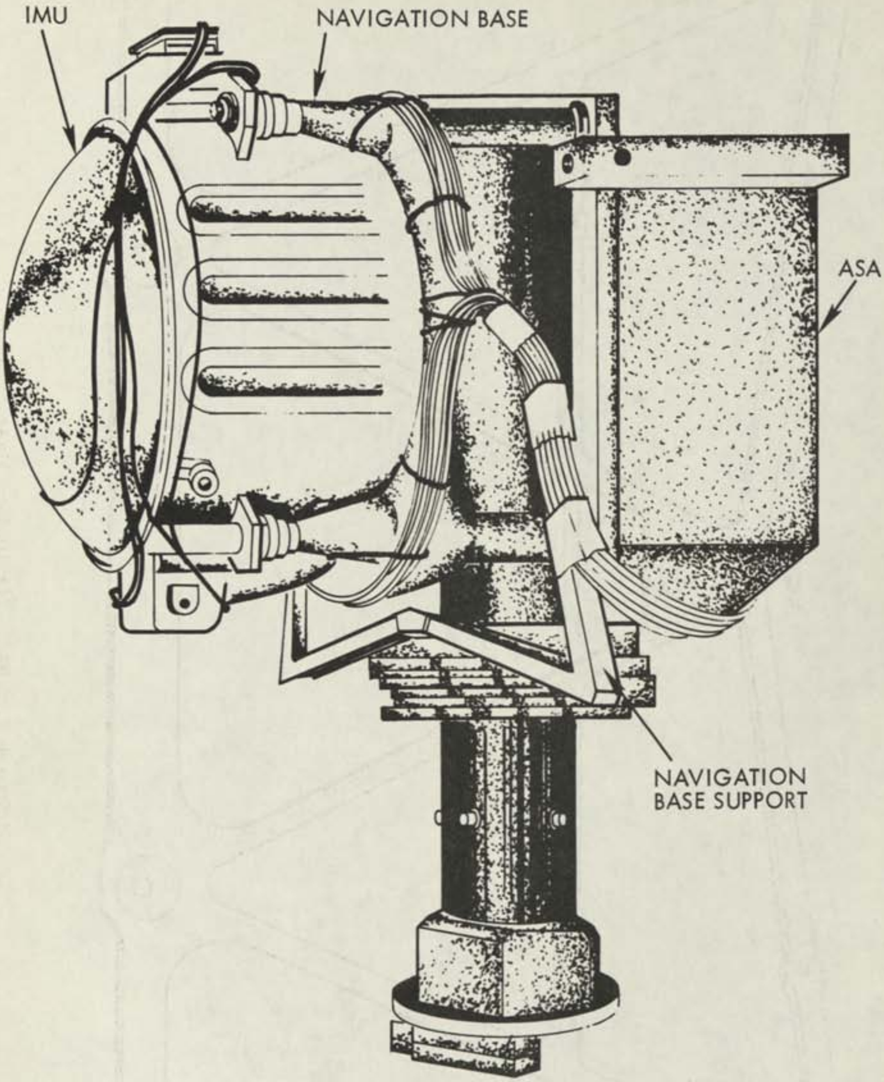


Figure 4.3.1.8-3. Navigation Base, Fully Assembled (Typical)

4.3.1.9.2 Function. The SCA receives PGNCS signals and converts them to a form acceptable to the airborne Pulse Code Modulation Telemetry Equipment. The SCA also provides isolation between the component or circuit being monitored and the telemetry system.

4.3.1.9.3 Mechanization. The SCA contains four modules: the DAC, PIPA Temp and 2.5V Bias module, the PIPA and IRIG module, the Gimbal Resolver module, and the Radar Resolver and 120V PIPA Supply module which have been mechanized to condition the corresponding PGNCS signals for telemetry.

4.3.1.9.4 General Characteristics

4.3.1.9.4.1 Mechanical Characteristics. The weight of the SCA, per MIT Status Report E-1142, April 1967, is 7.2 pounds. See Figure 4.3.1.9-1 for an outline drawing of the SCA.

4.3.1.9.4.2 Electrical Characteristics. For the electrical characteristics of the SCA refer to the schematics indicated below:

Module	Schematic
DAC, PIPA, Temp and 2.5V Bias	2007233
IRIG and PIPA	2007231
Gimbal Resolver	2007230
Radar Resolver and 120 V PIPA Supply	6007012

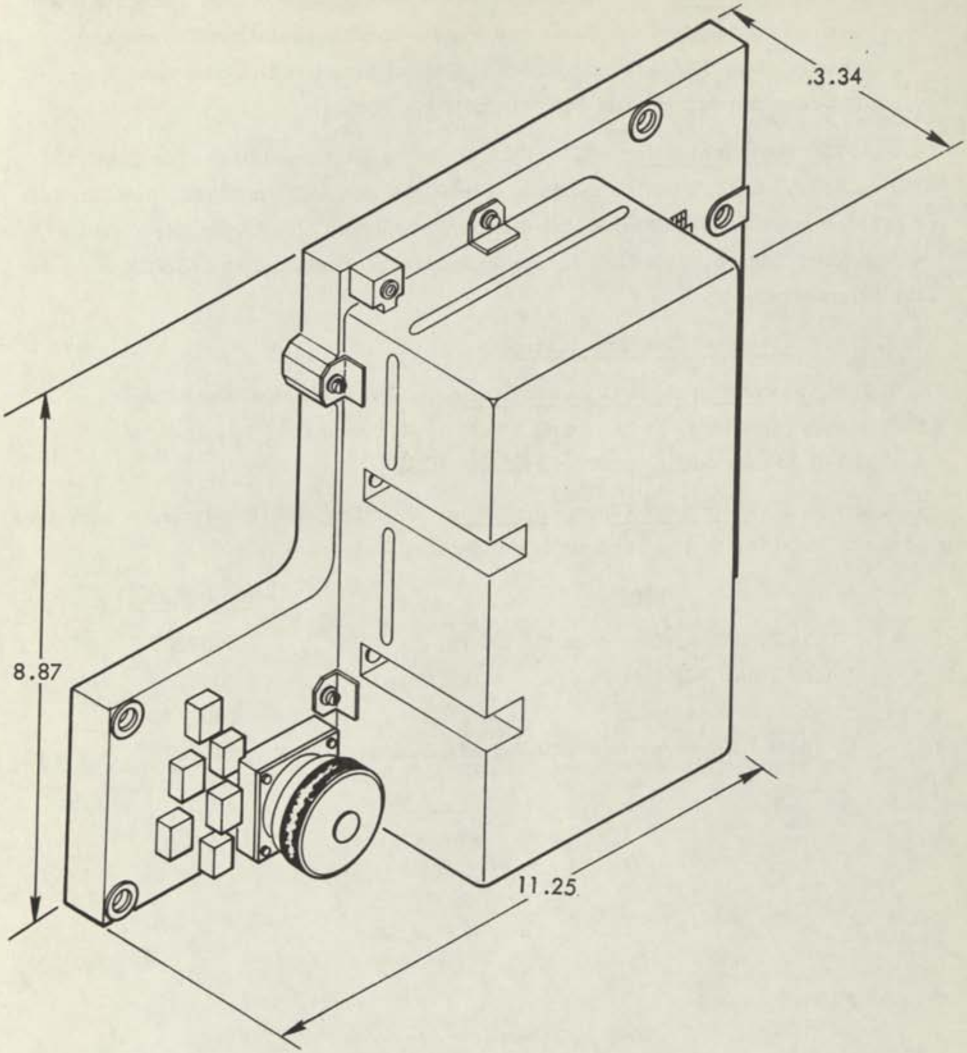


Figure 4.3.1.9-1. Signal Conditioner Assembly

4.3.2 Radar Subsystems

The Radar System is comprised of the landing radar and the rendezvous radar. See Figure 4.3.1-1 for a block diagram of the LR and RR interface with the G&C System.

4.3.2.1 Landing Radar

A brief description of LR Component Identification, Function, Mechanization, and General Characteristics is presented in the following paragraphs.

4.3.2.1.1 Component Identification. The LR consists of the following two assemblies:

a) Antenna assembly (LRAA)

b) Electronics assembly (LREA)

A block diagram of the LR is shown in Figure 4.3.2.1-1.

4.3.2.1.1.1 Antenna Assembly. The antenna assembly contains the three-beam CW velocity sensor and single beam FM/CW altimeter transmitters, altimeter microwave power leveler, antennas, waveguide, power monitor couplers, hybrids, crystal mixer assemblies, preamplifiers, and a-f limiters. For transmission the LRAA uses two interlaced phase arrays and for reception, uses four space-duplexed planar arrays. The transmitting arrays form a platform on which are mounted four microwave mixers, four dual audio frequency preamplifiers, two solid-state microwave transmitters, an FM modulator, and an antenna pedestal tilt mechanism. See Figures 4.3.2.1-2 and 4.3.2.1-3 for the antenna tilt positions and beam geometry.

4.3.2.1.1.2 Electronics Assembly. The electronics assembly contains the velocity and altimeter frequency trackers, velocity data converters, velocity data computer, signal data/range converter, data logic circuit, and power supply. The LREA contains the circuitry to track, process, convert, and scale the Doppler and FM/CW returns to provide the velocity and slant range information to the LGC and to the astronaut's display.

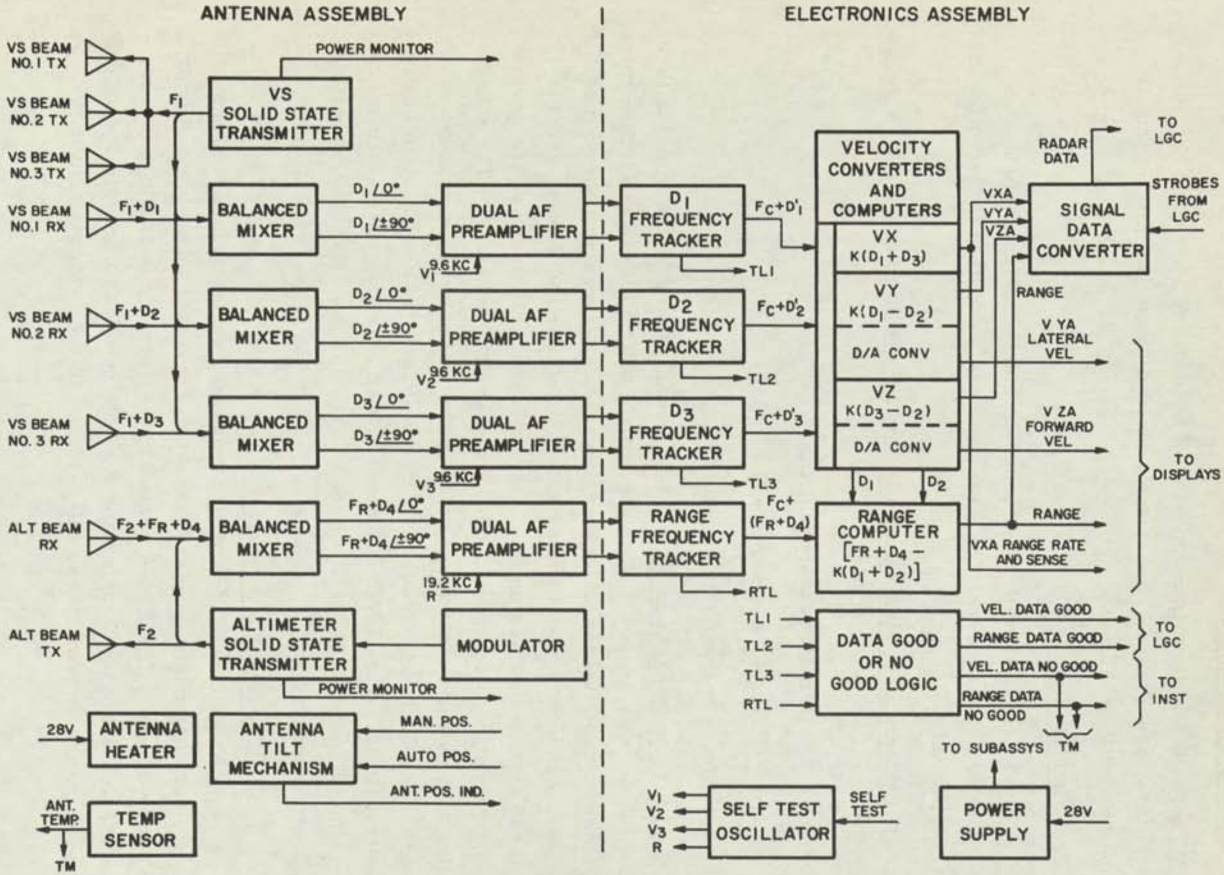


Figure 4.3.2.1-1. Landing Radar Block Diagram

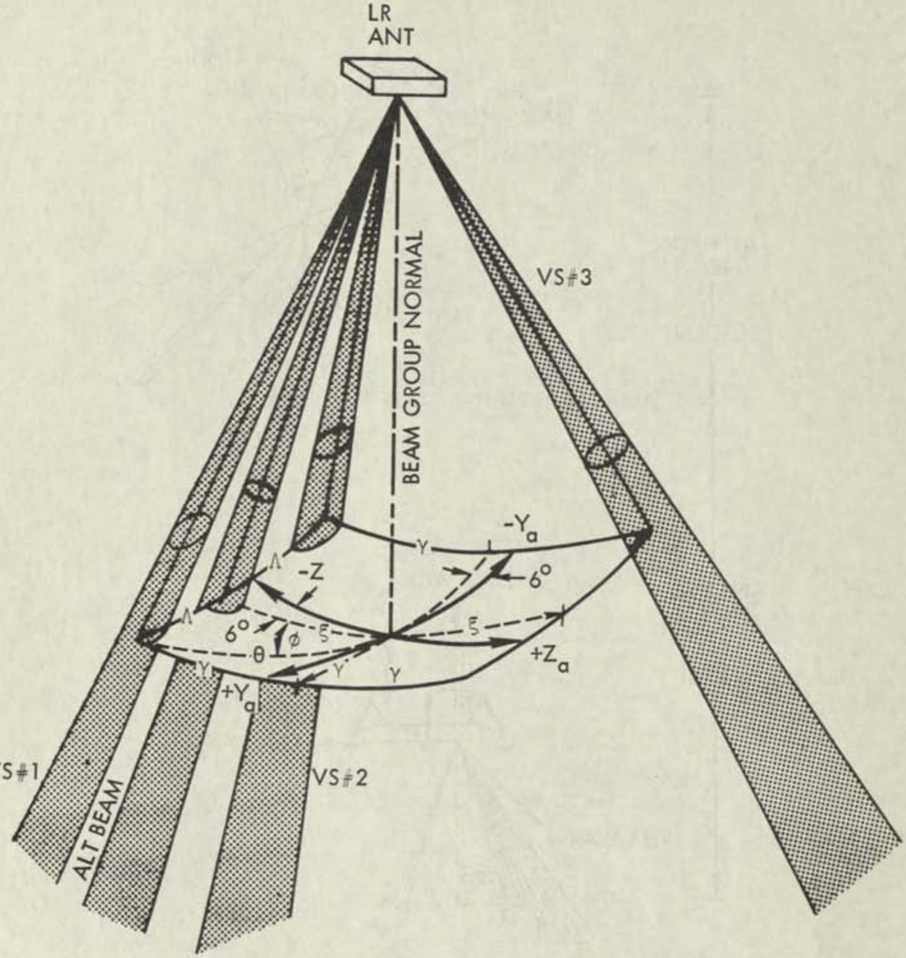


Figure 4.3.2.1-2. LR Antenna Beam Geometry

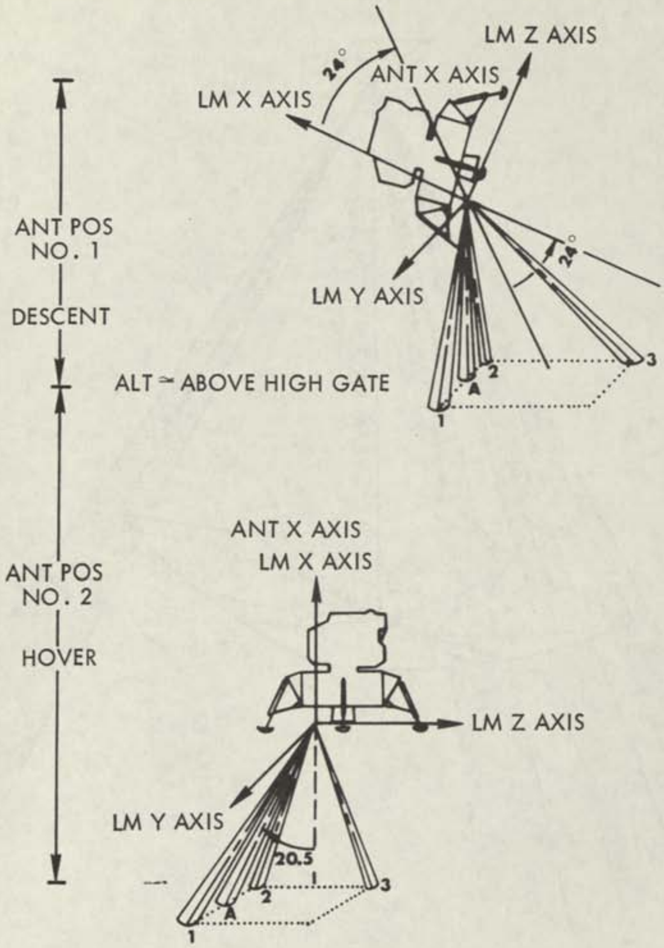


Figure 4.3.2.1-3. LR Antenna Tilt Positions

4.3.2.1.2 Function. The landing radar provides data to the LGC from which LM velocity and LM altitude is determined. The data are also made available for visual display to the astronaut.

4.3.2.1.3 Mechanization. The CW microwave energy from the velocity sensor solid state transmitter is radiated toward the moon by the transmitting antenna. Reflected energy, as a function of transmitted power, range, beam incidence angle, and lunar reflectivity, is received by three separate receiving antennas. The received Doppler shifted energy, split in the hybrids to form quadrature pairs, is mixed with a portion of the transmitted energy in microwave diode balanced mixers. The output of the balanced mixers is the difference frequency between the received and transmitted signals (i.e., Doppler frequency).

The altimeter transmitter frequency is modulated by a Frequency Modulator in a sawtooth manner at a repetition rate of 130 Hz. The altimeter microwave energy is applied to the altimeter transmitting antenna through a microwave power leveler to maintain incidental AM levels below 0.1 lb. The received energy from the receiving antenna is split in the hybrid to form a quadrature pair and is coupled to balanced microwave diode mixers along with a sample of the transmitted signal. The difference frequency at the output of the balanced mixers is proportional to the time difference of the transmitted and received modulated energy, plus the Doppler shift.

The quadrature outputs of the three velocity sensors and the altimeter balanced mixers are applied to the four audio frequency preamplifiers. Each of these amplifiers contains logic that switches the gain of the amplifier from 87 db to 55 db as the signal level increases. This prevents distortion of the signal as a result of the dynamic range of the amplifier. Limiters are incorporated at the output of the altimeter audio frequency preamplifiers, which minimize premature gain switching of the amplifiers and subsequent tracker acquisition of interference signals caused by the flyback pulses.

The broadband signals at the AF preamplifier outputs are applied to the inputs of frequency trackers which are located in the electronics assembly. The frequency trackers search the expected frequency range for the signal with a narrow band tracking filter, acquire, and track the signal. The tracker output is a frequency corresponding to the center of power of the Doppler or range signal spectrum plus a 614 kc reference (F_C). The tracker provides a dc step voltage to indicate tracker lock. A discrete signal is also supplied from the antenna assembly AF preamplifier signal level detection circuits to indicate the AF preamplifier gain state (i. e., 87 db or 55 db gain).

In the velocity sensor channels, the outputs of the AF preamplifiers are applied, in phase quadrature, to the single-sideband modulator (SSB). These outputs are $D_1 < 0$ and $D_1 < 90$. Four other inputs ($F_C + D_1'$) are applied to the SSB in phase quadrature, from a voltage control oscillater (VCO) through a quadrature generator. A phase-shift method is used to generate a SSB output signal for use in the frequency tracker section. The phase-shift method involves the use of two modulators both receiving the same a-f (Doppler D_1) and carrier (VCO) frequency ($F_C + D_1'$) inputs. A 90-degree phase-shift is introduced into both the Doppler voltage and the carrier voltage to one of the two modulators. Modulator A1 produces a double sideband suppressed-carrier output of arbitrary phase. The output of Modulator A2 contains sideband components identical with those in Modulator A1 except of such phase that phasor addition in the output circuit results in addition of one set of sidebands and cancellation of the other. The output of the SSB modulator consists of $F_C + D_1' - D_1$, where D_1' = Doppler frequency (D_1) plus VCO error frequency (D_C).

The modulator is arranged to provide the lower sideband of the mixing process. The action of the tuning loop is to maintain the frequency of the lower sideband at 614.4 kc plus the tuning loop error. The quadrature generator outputs are therefore at a frequency of 614.4 kc, plus the Doppler frequency, plus the tuning loop error. The lower sideband output of the SSB modulator ($F_C + D_1' - D_1$) is applied to an IF broad band amplifier having a flat response between 600 kc and 730 kc and a 3-db bandwidth of from 300 cps to 2.0 mc. The amplifier output is applied to two mixers which are excited in quadrature from a 614.4 kc clock reference. The difference frequency at the mixer output represents the tuning loop error frequency. The phase relationship between the two quadrature tuning loop error signals determines the sense of the error; i. e., whether the SSB modulator output is higher or lower in frequency than the 614.4 kc reference ($F_C + D_1' - D_1 \pm F_C$).

The difference frequencies at the two signal mixer outputs are filtered by step low pass filters. These filters are of two stage RC type with half power bandwidths at one of two values, (2800 cps or 400 cps) as selected by the state of the bandwidth control signal. The gain from the input of the IF amplifier to the output of the step low pass filter is approximately 90. The signals at the filter outputs are applied to limiters. The quadrature signals at the outputs of the limiters are applied to the frequency discriminator. The output of one of the limiters is applied to gates along with control signals of constant pulse width which are generated by a monostable multivibrator triggered from the other limiter input signal. The output of the discriminator consists of a train of pulses which are of constant amplitude and constant width. The pulse rate is equal to the tuning loop error frequency and the polarity of the pulses is determined by the sense of the tuning loop error.

The average value of the pulse train at the discriminator output is obtained by applying the pulses to an RC low pass filter. The smoothed output is used as control voltage for the VCO, which operates in the vicinity of 2.576 mc. The VCO output is applied to a quadrature generator, which consists of two crosscoupled flip flops. The frequency of the quadrature output is a factor of four lower than the VCO frequency. These quadrature signals are applied to the SSB modulator, thus closing the tracking loop.

The output of the SSB modulator ($F_C + D_1' - D_1$) is applied to an IF amplifier having a gain of 18 and a bandwidth (3 db) of 300 cps to 2.0 mc. The IF amplifier also supplies a Doppler signal level monitor output.

The signal acquisition process is based on comparison of energy in the two sidebands at the output of the SSB modulator. The upper sideband of the mixing process, which is at a frequency of the 614.4 kc reference frequency plus twice the Doppler frequency ($F_C + 2D_1$), is examined by applying the output of the sideband IF amplifier to a mixer and low pass filter. The mixer is excited from a reference signal equal to 614.4 kc plus twice the Doppler frequency, which is derived by scaling down the

VCO output frequency by a factor of two and mixing this signal with a 614.4 kc signal from the clock. The output of the low pass filter is compared to that of the tracking loop low pass filter in a threshold detector where the signals are rectified, filtered, and fed to a voltage comparator. In high-gain broadband operation, the threshold detector is energized when the signal channel output exceeds the noise channel output by 3 db. In high-gain narrow band operation, the threshold detector is biased so that the Doppler input signal must exceed a preset value, in addition to producing a signal to noise channel ratio of 3 db. In low-gain operation, the detector is biased so that a greater signal is required for the frequency tracker to "acquire" or lock onto the Doppler frequency. The output from the threshold detector is used to stop the search program.

The tracker determines signal acquisition by comparing the information at the input, which is in quadrature (signals), to that which is not in quadrature (noise). The SSB modulator translated quadrature information to one sideband ($f_C + D' - D$), where $f_C + D'$ is the quadrature generator frequency and D is the signal frequency. Noise (N) is translated into two sidebands $f_c + D' - N$ and $f_c + D' + N$. The acquisition circuit monitors both sidebands of the modulator output with two low pass filters. The signal channel monitors the lower sideband and is positioned at f_c with respect to the modulator output. When the output of the signal channel exceeds the output of the noise channel by a factor of "K", acquisition occurs and the tracker tracks the information in the signal channel. When the input contains a signal "D" in the presence of noise and the quadrature generator is positioned at $f_c + d' = f_c + D$, the signal appears in the signal channel. The noise is double sidebanded by the modulator, and therefore, appears in both channels. The signal appears in the noise channel but is translated 2D from its center. When the quadrature generator is positioned away from $f_c + D$, only noise appears in each channel, and the tracker searches. The value "K" is introduced to require signals to be 3 db above the noise level before acquisition can occur. The value of "K" is determined as follows: when the signal (S) appears in the signal channel, the output of this channel becomes $\sqrt{S^2 + N^2}$. The output of the noise channel is KN. Acquisition occurs when these two quantities are equal. Therefore $K = \sqrt{S^2/N^2 + 1} = 1.73$ for $S/N = 1.4 = 3.0$ db.

When the input frequency, D, approaches zero, it appears in both the signal and noise channels. Since the portion of signal seen in the noise channel is multiplied by "K", acquisition is impossible at frequencies near zero. Thus, the noise channel is purposely centered off the upper sideband in the low frequency region.

The function of the sweep limit switch is to detect when the VCO output frequency (divided by four by the quadrature generator and subtracted from 614.4 kc by the Doppler sign detector) reaches the lower sweep limit, and provide a flyback signal for the sweep control. The lower sweep limit in the broadband mose is -35.0 kcps for beams 1 and 2, and is -17.4 kcps for beam 3. The lower sweep limit in the narrowband mode is -4.5 kcps.

The primary input to the sweep limit switch is a square wave of frequency D'. The period of this signal is compared to an internal time interval reference, and a flyback pulse is generated whenever the signal period exceeds the reference interval.

The tracker outputs are applied through isolation amplifiers to the velocity and range data converters where the beam velocity information is resolved into velocity components in the antenna coordinate systems. The antenna coordinate system is referenced to the body coordinates of the antenna and is denoted V_{xa}, V_{ya}, V_{za}. This information as well as range information, is generated for use in the LM displays. Pulse train information is given for range and V_{xa} and dc analog voltages are provided for V_{ya} and V_{za}. A dc discrete signal is provided to indicate the sense of the V_{xa} pulse train.

The velocity data in the body coordinates of the antenna are given in pulse train form superimposed on a 153.0-kc reference frequency to facilitate indication of the sign of the velocity. These pulse trains along with the range pulse train are applied to the signal data converter. The signal data converter interfaces with the LM guidance computer by accepting strobe signals from the computer and using these to assemble and readout the range and velocity data in serial binary form. The serial binary radar output information is fed to the LGC.

Radar status signals which include Range Data Good, Velocity Data Good, Antenna Position Indicate, and Range Scale Factor are provided to

Table 4.3.2.1-1 Physical Data

Type of System		
Velocity Sensor	CW, 3-beam	
Radar Altimeter	FM/CW	
Altitude Capability		
Velocity	5 to 25,000 feet	
Altimeter	10 to 25,000 feet	
Velocity Capability	Hi Alt Mode	Lo Alt Mode
V_{xa}:	-2,000 to + 500 fps	-200 to +300 fps
V_{ya}:	-500 to + 500 fps	-200 to +300 fps
V_{za}:	-500 to + 3,000 fps	-200 to +300 fps
Weight		
Specified Weight	39.0 lbs	
Size (L x W x H)		
Antenna Assembly	20.0" x 24.6" x 6.5"	
Electronic Assembly	15.75" x 6.75" x 7.38"	
Power Consumption	147 watts dc max	
Altimeter Antenna		
Type	Planar array, space duplexed	
Gain (two-way)	50.4 db	
Beamwidth (two-way)	3.9 deg E plane	
	6.5 deg H plane	
Velocity Sensor Antenna		
Type	Planar array, space duplexed	
Gain (two-way)	49.2 db	
Beamwidth (two-way)	3.67 deg E plane	
	7.34 deg H plane	

Table 4.3.2.1-1. Physical Data (Continued)

Transmitters	
Type	Solid state
Frequency	
Velocity Sensor	10.51 gc
Radar Altimeter	9.58 gc
Output Power	
Velocity Sensor	200 mw minimum
Altimeter	100 mw minimum
Altimeter Modulation	Sawtooth FM
Modulation Frequency	130 cps
Deviations	8 mc > 2,500 ft
	40 mc < 2,500 ft
Response Time	0.1 sec nominal

Table 4.3.2.1-2 Outputs

Serial Binary to Computer

Parameter	Limits	Counting Time	Scale Factor
153.6 KC $-V_{xa}$	-2000 to +500 fps	100 m sec	0.5152 fps/count
153.6 KC $+V_{ya}$	-500 to +500 fps	100 m sec	0.9693 fps/count
153.6 KC $+V_{za}$	-500 to +3000 fps	100 m sec	0.6934 fps/count
Range	2500 to 25000 ft	200 m sec	2.1582 ft/count
Range	10 to 2500 ft	200 m sec	0.4316 ft/count

Display Outputs

Analog

V_{ya}	(-200 to +200 fps)	25 mv/fps
V_{za}	(-200 to +200 fps)	25 mv/fps

Pulse Trains

V_{xa}	(-500 to +500 fps)	19.41 pps/fps
Range	(2500 - 25000 ft)	2.322 pps/ft (nominal)
Range	(10 - 2500 ft)	11.6 pps/ft (nominal)

Status Indications - (Contact Closures Referenced to 28 ± 11V)

LGC

Range Data Good

Velocity Data Good

Range Low Scale Factor

Antenna Pos. 1

Antenna Pos. 2

Displays: (Isolated Contact Closures)

V_{xa} Sense (closed contact for opening or positive rates)

Table 4.3.2.1-2 Outputs (Continued)

Instrumentation: (Contact Closures)	
	Range Data No Good
	Velocity Data No Good
	LR Power On
	Antenna Pos. 1
	Antenna Pos. 2

Table 4.3.2.1-3. Accuracy

	Altitude (ft)	Accuracy
Range (digital and pulse train)	10 - 2000	±1.4% or ±5 ft
	2000 - 25000	±1.4% or ±15 ft
	25000 - 4000	±2% or ±15 ft
Velocity		
V_{xa} (digital)	5 - 25000	±1.5% or ±1.5 fps
V_{ya} (digital)	5 - 200	±2% or ±1.5 fps
	200 - 2000	±3.5% or ±3.5 fps
	2000 - 25000	±2% or ±2 fps
V_{za} (digital)	5 - 200	±2% or ±1.5 fps
	200 - 2000	±3% or ±3 fps
	2000 - 25000	±2% or ±2 fps
V_{xa} (pulse train)	5 - 50	±0.6 fps
	50 - 200	±0.7 fps
	200 - 2000	±1.4% or ±1.4 fps
	2000 - 25000	±1.3%
V_{ya} (analog)	5 - 50	±1.2 fps
	50 - 200	±1.6 fps
	200 - 2000	±2.8% or ±2.8 fps
	2000 - 25000	±1.7%
V_{za} (analog)	5 - 50	±1.0 fps
	50 - 200	+1.3 fps
	200 - 2000	3.0% or ±3.0 fps
	2000 - 25000	3%

Table 4.3.2.1-4. Interfaces

To LGC	
Range, V_{xa}	3.2 kc pulses 7 ± 3v (positive polarity)
V_{ya}, V_{za}	Pulse width 4 ± 2.0 μs
	Rise Time 0.2 μs
	Load Impedance 200 ohm (±10%)
	Source Impedance 100 ohm (max)
To LM Displays	
Range	0 to 70 kc pulses } parameters same as
V_{xa}	0 to 10 kc pulses } range and V_{xa} above
V_{ya} to V_{za}	Analog DC, 0 ± 5 v dc max
	Source Impedance 10 K ohm
	Load Impedance 50 to 100 K ohm

Table 4.3.2.1-5. Landing Radar Reference Data

a) Beam Angles

	(deg)	(min)	(sec)
γ	19	45	00
θ	24	33	00
Λ	13	59	22
ξ	20	22	48
Φ	35	34	47
ψ	14	53	00

b) Antenna Tilt Angles

Position 1	24 deg
Position 2	0 deg

c) Antenna Position

Location	LM - Y 56 in.
	LM - Z 56 in.
Rotation	6 deg (i.e., the XZ plane of antenna is rotated 6 deg in the +Y direction of the LM referenced to the LM XZ plane)

d) Range Scale Factors*

High Deviation	11.6116 cps/ft
Low Deviation	2.32232 cps/ft

e) Doppler Coefficients*

$\frac{2}{\lambda}\cos\Lambda\cos\xi$	$\frac{2}{\lambda}\sin\Lambda$	$\frac{2}{\lambda}\cos\Lambda\sin\xi$	$\frac{2}{\lambda}\cos\Lambda$
19.453 cps/fps	4.9968 cps/fps	7.3030 cps/fps	20.778 cps/fps

f) Velocity Coefficients*

$$\frac{\lambda}{2\cos\Lambda\cos\xi} \qquad \frac{\lambda}{4\sin\Lambda} \qquad \frac{\lambda}{4\cos\Lambda\sin\xi}$$

$$\frac{\lambda}{15\cos\Lambda} \qquad \frac{\lambda\ \mathrm{ctn}\ \xi}{4\cos\Lambda} \qquad \frac{\lambda\tan\xi}{2\cos\Lambda}$$

*Scale factors and coefficients contain no corrections for tracker gain, terrain bias, etc.

Table 4.3.2.1-5. Landing Radar Reference Data (Continued)

$\frac{1}{19.453}$ $\frac{\text{fps}}{\text{cps}}$	$\frac{1}{9.9935}$ $\frac{\text{fps}}{\text{cps}}$	$\frac{1}{14.606}$ $\frac{\text{fps}}{\text{cps}}$
$\frac{1}{19.479}$ $\frac{\text{fps}}{\text{cps}}$	$\frac{1}{15.602}$ $\frac{\text{fps}}{\text{cps}}$	$\frac{1}{55.34}$ $\frac{\text{fps}}{\text{cps}}$

g) Preamp Gain States

1) 87 db
2) 55 db

-160 dbw signal input at antenna gives 1 mv rms output signal from the preamp when in gain state of 87 db.

h) Preamp Filter Characteristics

1) <u>Velocity Channels</u>

low freq - 3000 cps (3 db point) roll-off at 6 db/octave

high freq - 250 kc (3 db point) roll-off at 6 db/octave

2) <u>Altimeter</u>

low freq = 12 kc (3 db point) roll-off at 6 db/octave

high freq = 250 kc (3 db point) roll-off at 6 db/octave

3) <u>Tracker Bandwidth</u>

Velocity wideband mode 2800 cps   Velocity narrowband mode 400 cps

Altimeter wideband mode 3200 cps   Altimeter narrowband mode 1000 cps

i) Tracker Bandwidth Switching Criteria

1) Wideband   Doppler freq > 1440 cps

and

Altitude > 2500 ft

2) Narrowband   Doppler freq ≤ 1440 cps

or

Altitude ≤ 2500 ft

Table 4.3.2.1-5. Landing Radar Reference Data (Continued)

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j) Altimeter Switching Hysteresis = 800 ft

i.e., switch to N-B mode at ≤ 2500 ft, switch back to W-B mode at 3300 ft.

k) Self Test Freq 9.6 kc Velocity Sensors

19.2 kc Range

l) Tracker Sweep Range

1) Altimerer

W-B + 1 to + 133 kc

N-B +1 to + 30 kc

2) Velocity

D1, D2 W-B -35 to +46 kc

D3 W-B -17 to +64 kc

D1, 2, 3 N-B -4.5 to + 6.5

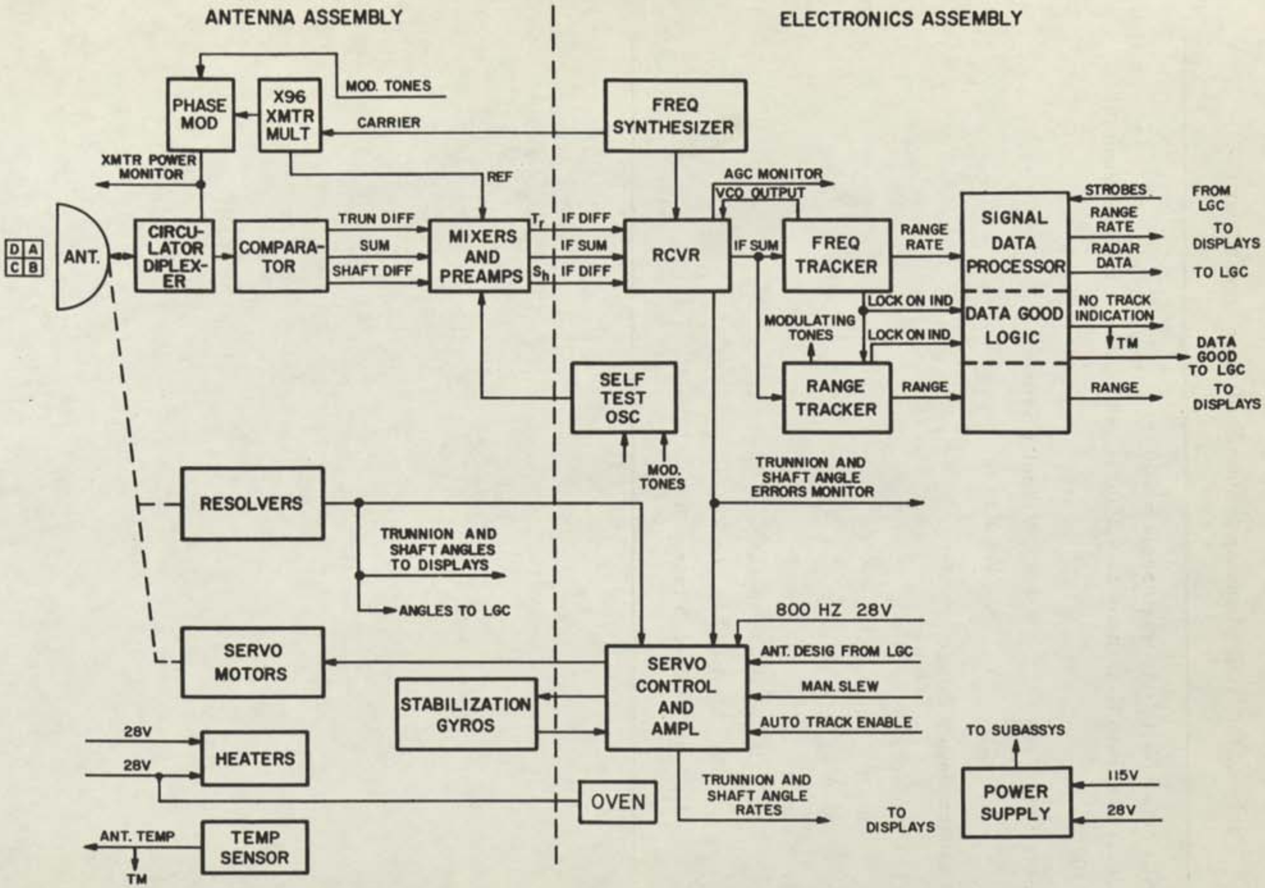


Figure 4.3.2.2-1. Rendezvous Radar Block Diagram

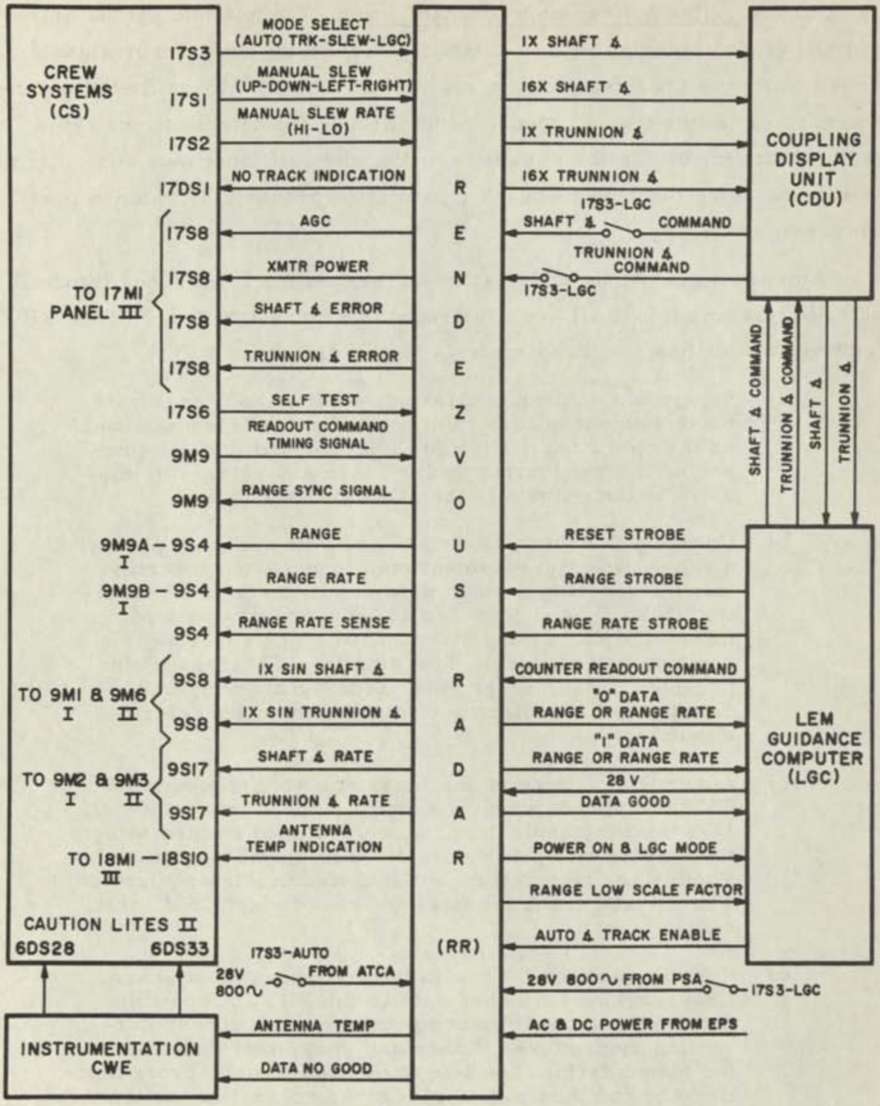


Figure 4.3.2.2-2. Rendezvous Radar Interface Diagram

4.3.2.2.3.1 Antenna Assembly Mechanization. The antenna can be driven in either (1) the tracking mode, in which the tracking-loop error signal is used to torque the integrating gyros in a direction which nulls the antenna error, or (2) the manual mode, controlled by the astronaut, or (3) in the designate mode. In the designate mode, the LGC positions and controls the antenna using position feedback information provided by antenna position resolvers.

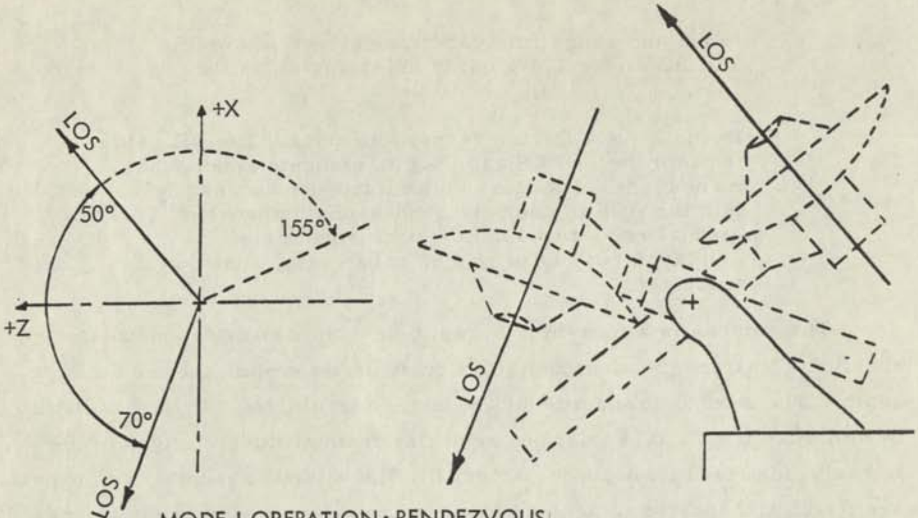
Antenna orientations and positioning ranges for rendezvous (Mode I) and CSM tracking (Mode II) are illustrated in Figure 4.3.2.2-3. The RR tracking capabilities for these modes are as follows:

a) Separation - After separation, a range and range rate measurement affords the crew a first independent check on the operation of the LM, PGNCS, and AGS by comparing the rendezvous radar range and range rate displays to the outputs of the other two sections.

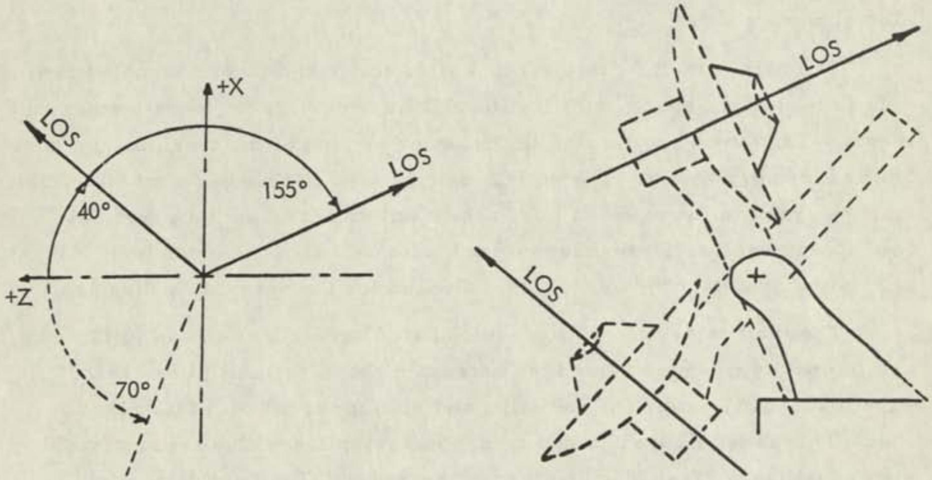
b) Coasting Descent - During the coasting descent phase, a range rate measurement made right after insertion into the coasting descent determines the applied insertion Delta V, and provides an independent check on orbit safety by giving an estimate of the expected pericynthion altitude. Tracking the CSM transponder provides range, range rate, and LOS tracking and data to the LM Guidance Computer for orbit determination.

c) Lunar Stay - Once on the lunar surface, tracking the CSM transponder provides range, range rate, and LOS tracking angle data, which are used to determine the landing site position relative to the CSM orbit. From this, launch time and launch azimuth are derived. The RR will track the first and next to last CSM orbit before launch.

d) Coasting Ascent - When in the coasting ascent phase, CSM tracking furnishes data to the PGNCS for orbit determination and for computation of midcourse correction maneuvers. Range and range rate data can be fed manually into the AGS to be used in orbit determination and computation of midcourse correction maneuvers. These data are inserted into the AGS through the DEDA while the LM +Z axis is pointed at the CSM and the RR shaft and trunnion angles are to zero degrees. For acquisition, LM +Z axis is pointed to estimated CSM position by AGS, RR antenna angles are manually nulled, and RR auto track is enabled. The LM +Z axis is manually rotated to renull the RR angles, and



MODE I OPERATION: RENDEZVOUS



MODE II OPERATION: CSM TRACKING

Figure 4.3.2.2-3. Rendezvous Radar Antenna Dual Mode Positions range and range rate data are entered into AGS. The AGS uses LM +Z axis euler angles as the LOS.

e) Rendezvous - During rendezvous phase, the RR data fed into the PGNCS are used to compute rendezvous maneuvers. RR data can be introduced manually into the AGS to compute rendezvous maneuvers. Manual rendezvous can be performed using the displayed outputs of range, range rate, angle, and angle rate.

The antenna is a four-horn, amplitude-comparison, monopulse-type apertur. A Cassegrainian configuration is used to minimize the total depth. The antenna transmits and receives circularly polarized radiation to minimize the signal variations resulting from attitude changes of the linearly-polarized transponder antenna. The various assembly components are physically located to achieve balance around each axis of the antenna. Each axis is controlled by a brushless motor that is driven by the servo electronics.

The four rate-integrating gyros (two active and two on standby) provide line-of-sight space stabilization and line-of-sight angle-rate measurements. The gyros are located in the lower section of the trunnion axis and act as a counterweight. The voting logic is used to transfer control to the standby gyros in the event of a failure in either of two active gyros. A two-speed resolver is mounted on each antenna axis to provide high accuracy angle data pickoff for the LGC and for the astronauts displays.

The multiplier chian, phase-modulated, and mixer-preamplifiers are internally mounted behind the antenna dish. The multiplier chain supplies X-band power for radiation and also for local oscillator excitation. This is feasible since the transponder replies with a frequency sidestep equal to the first IF frequency of the radar. The heat dissipated by the multiplier chain is radiated back into space by the dish. The phase-modulator uses a solenoid-controlled ferrite rod in the waveguide to provide the required modulation of the X-band carrier. Ranging tone signals are applied to the solenoid, which, in turn, varies the electrical length of the rod. The resulting changes in the rod's magnetic field phase modulates the carrier. Three balanced mixers and three preamplifiers are included, one for each of the three (reference, shaft error, and trunnion error) channels.

4.3.2.2.3.2 Electronics Assembly Mechanization. The return signal from the transponder is processed in a highly-stable, three-channel, triple-conversion superheterodyne receiver having intermediate frequencies of 40.8 mc, 6.8 mc, and 1.7 mc. The bandwidth of the first and second IF amplifiers is approximately 3 mc, and the bandwidth of the third IF amplifier is approximately 1 kc.

Two channels are provided for amplifying the antenna assembly shaft and trunnion axis error signals, and one channel is provided to amplify the sum or reference signal. The receiver also includes phase-sensitive detectors for generating angle error signals, an automatic gain control circuit for controlling the gain of the three receiver channels, an IF distribution amplifier unit for supplying reference channel signals to the range and frequency trackers, and a local oscillator-mixer for generating the second local oscillator signal. The second local oscillator frequency is obtained by heterodyning the frequency tracker VCO (voltage-controlled oscillator) output and a reference frequency. This produces a sum frequency exactly 6.8 mc lower than the incoming 40.8 mc Doppler-shifted frequency. At the second mixer, the Doppler frequency shift is removed, and all subsequent signal processing is accomplished at fixed-carrier frequencies.

The frequency synthesizer generates all the fixed frequencies required for coherent signal transmission and reception. A 1.7-mc stable crystal oscillator and a system of multiplication, division, and mixing are used to produce the required frequencies. The synthesizer generates a cw output signal for excitation of the transmitter multiplier chain in the antenna assembly from a separate 102.425-mc crystal oscillator. The synthesizer also generates various local oscillator, clock, and reference frequencies used by the receiver, signal-data converter, and frequency and range trackers.

The frequency tracker is used to phase-lock the VCO with the incoming narrowline spectrum. A sweep circuit is used to vary the VCO frequency across the expected Doppler frequency range of ±100 kc. The presence of carrier signal within locking range is sensed by a threshold circuit which stops the sweep and permits the VCO to phase-lock. The phase detector for the phase-locked loop uses a 6.8-mc reference signal supplied by the frequency synthesizer. The error signal drives the VCO to a frequency which, when mixed with a 27.2 mc synthesizer signal at the receiver second IF oscillator, removes the Doppler frequency shift from all signals in succeeding IF stages and assures passage of the signal through the 1.7-mc filters. These filters have a bandwidth of 1 kc.

The range tracker determines the range to the transponder by measuring the phase angle between the transmitted and received tones. The 6.8-mc signal received from the transponder is demodulated in a coherent product detector using a 6.8-mc quadrature reference. The individual sine-wave tones are extracted from the receiver signal using bandpass filters. Phase delay is measured independently for each of the three filtered tones in a closed tracking loop in the following manner. Three locally-generated square-wave reference signals, each having a variable phase with respect to the phase of the tones originally transmitted, are compared with their respective received tones in three separate phase detectors. The square waves are produced digitally by comparison between a running high-speed counter and a low-speed, forward/backward range counter. This counter is driven by incremental range pulses obtained from a dc-to-prf converter, which is, in turn, controlled by weighted integration of the three phase detector error signals. The low-speed range counter is driven forward to backward until phase null is achieved in each of three phase detectors.

The servo electronics contains amplifiers which drive the antenna shaft and trunnion axis servo motors and the gyro torquer coils. It also has the voting logic for selecting the correct gyro pair. The servo electronics, in conjunction with the antenna components and radar receiver, form an inner and outer closed servo loop for each axis. The inner, or stabilization loop, keeps the antenna boresight axis fixed in inertial space unaffected by the LM body motions. The outer, or tracking loop, keeps the antenna boresignt on the target in step with tracking error signals from the receiver. In the designate mode (Radar Mode Switch 17S3 in LGC position), this loop opens and the servo accepts the computer-designate data. The guidance computer designates the antenna boresight to the target, and then supplies an automatic track-enable signal for the rendezvous radar when within 1 degree of the computed target LOS. This signal and the frequency lock-on function causes the tracking loop to close. The antenna then continuously tracks the target in step with the receiver angle error signals. The antenna may also be manually slewed at fixed rates (Switch 17S3 in Slew position). The enable signal necessary to close the auto track loop is manually supplied in this case by the astronaut placing Switch 17S3 to the Auto Track position to acquire the CSM radar transponder.

Antenna shaft and trunnion positioning is accomplished by 32-pole, brushless, permanent-magnet, rotor-type motors. The motors are positioned by pulse-width modulation (pwm) drive signals applied to sine and and cosine widings of each motor. Reversal of the motor's direction is accomplushed by reversing the pwm voltage across the windings.

A gyro voting system, consisting of a performance comparison circuit and switching logic, is used to automatically detect and remove a malfunctioning gyro. Of the four available gyros, two are used to stabilize the antenna, the other two are redundant, and either pair can perform the control function. The voting system determines whether the active pair contains a failed gyro by comparing the resolved outputs of the four gyros, three at a time. If a failure or degradation occurs, the standby redundant pair is switched into the antenna stabilizing system.

The signal data converter accepts range and range-rate data from the range and frequency trackers and converts these data to the 15-bit serial format required by the guidance computer. Data are shifted out in digital form to the computer on output lines, as requested by the computer. The signal data converter also sends various discrete radar status indications to the computer, selects radar modes, and processes display data for activation of the astronaut display panels.

The self-test circuits permit testing of the radar without the use of the transponder. These circuits permit a check of transmitter power, phase-lock at nominal signal level, angle error detection, agc action, range, and range rate measurement. The self-test circuits are enabled when the astronaut places the Radar Test Switch 17S6 to the RNDZ position. The range meter 9M9 drives to 200 nautical miles and the range rate meter reads -500 feet per second.

The power supply is a highly efficient dc-to-dc converter which provides six regulated dc output voltages. The unit uses a method of switched-tap modulation for input regulation and series current regulation in each of the output lines. A chopping frequency of 20 kc is used to minimize the weight of transformer and ripple-filter components. Circuitry is provided to sense any overload current condition on the output lines and to deactivate the 20-kc chopping oscillator for a preset period in order to prevent damage as a result of a short circuit condition. If the overload has been removed after this time, normal operation is resumed; otherwise, the deactivation cycle will continue until the overload is removed.

4.3.2.2.3.3 Rendezvous Radar/Transponder Acquisition Sequence. A brief description of the normal acquisition sequence for the rendezvous radar and transponder is automatic and is as follows.

a) The radar antenna, under computer control, is designated in angle so that its transmitted CW radiation can be received at the transponder. The resolver angle readouts are available to the LGC and to the FDAI display.

b) The transponder, which was previously sweeping in frequency, senses this radiation, stops its sweep, and phase locks to the received radar signal. It then retransmits this signal sidestepped by 40.8 mc.

c) The radar receiver, which was previously sweeping in frequency, stops its sweep and phase locks to the received transponder signal. The maximum completion time of steps (b) through (c) is 4.5 seconds.

d) The computer transmits "auto track enable" when antenna LOS arrives within 1 degree of the computed target LOS.

e) The radar angle tracking loop is closed upon completion of steps (c) and (d).

f) The radar activates ranging modulation, and the range tracking error is nulled within a maximum of 7 seconds after step (d) is completed. The coherent loop is now closed.

g) The radar indicates a "data good" condition to the LM guidance computer, based on both range and range rate lock on completion. Range and range rate data are now available to the computer and astronaut display panel. Angle rate also is available to the display panel as well as shaft and trunnion angles.

4.3.2.2.3.4 Basic Signal Flow. A description of the basic signal flow is as follows:

The frequency synthesizer develops the required frequency for the transmitter (102.425 mc). This frequency is multiplied to "X" band (9832.8 mc) in the X96 multiplier chain and phase modulated by three tones (200 cps, 6.4 kc, and 204.8 kc) generated from a reference counter in the range tracker.

The phase modulation is applied to the carrier in the antenna for radiation to the transponder. The radiated energy incident upon the CSM transponder is processed so that the tones are removed, the frequency is sidestepped, the tones are replaced, and the energy is returned to the RR Antenna.

The return energy from the transponder is received by four feedhorns. If the transponder is directly in line with the RR antenna line-of-sight, the return energy is equally received by each of the four feedhorns. If the transponder is not directly in line, the amplitude of energy received by each feedhorn is not equal. The RF signal is passed to the comparator which processes the signal to develop the required sum and difference signals. The three comparator RF outputs are sent to the mixers where they are combined with the transmit frequency to develop three 40.8-mc IF signals.

The position of the transponder with respect to the RR line-of-sight determines the type of antenna pointing error generated. With the RR line-of-sight directly below the transponder, shaft pointing error is generated. Similarly, with the RR line-of-sight on either side of the transponder, trunnion pointing error is generated. The preamplifiers contain three channels: the trunnion channel, the shaft channel, and the sum (reference) channel. The trunnion channel input is the difference in received energy between the vertically adjacent feedhorns (A+B)-(C+D). The shaft channel input is the difference in received energy between the horizontally adjacent feedhorns (A+D)-(B+C). The sum channel input is the sum of the received energy of all four feedhorns (A+B+C+D).

The outputs of the preamplifiers are processed by the receiver in three separate channels. The sum signal now at 6.8 mc IF with the tones is sent to the range tracker subassembly. The range tracker compares sequentially the phase of the 200 cps, 6.4 kc, and 204.8 kc return modulation with the phase of the reference signal from the reference counter to provide range information to the signal data processor. Phase comparison between the return and reference signals provide unambiguous ranging from 80 feet to 400 nautical miles. The range output is sent to the LGC and to the astronauts displays.

The Sum signal output of the receiver is also sent to the frequency tracker. The received frequency from the transponder is not constant, but varies with the changing range between the LM and the CSM. The difference in frequency caused by the changing range is called the Doppler frequency. The Doppler frequency is a function of the rate of change of range. Thus, by measuring the Doppler frequency, the range rate may be determined. A frequency tracker subassembly detects the Doppler frequency by continually comparing the received frequency, after it has been converted to 6.8 mc IF in the receiver with a reference frequency. A voltage controlled oscillator in the frequency tracker is maintained at the reference frequency plus the Doppler frequency by means of an automatic phase lock loop. Any change in Doppler frequency then results in a corresponding change in VCO frequency. The instantaneous frequency of the VCO then indicates range rate. Range rate data are sent to the LGC and to the astronauts displays.

The Doppler frequency component has to be eliminated from the ranging and antenna positioning circuits (the shaft and trunnion receiver channels) because it would result in inaccurate information. Therefore, the loop controlling the VCO frequency encompasses part of the receiving circuits. In that way, this loop removes (washes out) the Doppler frequency from the ranging and antenna positioning circuits but still is effective in maintaining the VCO at the Doppler frequency plus the reference frequency.

After triple conversion in the receiver to 1.7 mc, the sum channel signal is compared with the shaft and trunnion channel signals to obtain drive signals for the antenna. The effect of these drive signals is to eliminate the antenna pointing error. This automatic antenna positioning loop then keeps the RR antenna line-of-sight directed at the transponder. Vehicle motion is compensated for in the antenna positioning loop by use of a gyro stabilization loop.

The position of the antenna, as a result of the antenna positioning and stabilization loops, indicates the direction of the transponder with respect to the LM. Resolver outputs indicating antenna position are applied to the LGC and to the astronauts displays.

4.3.2.2.4 General Characteristics. A brief list of characteristics is presented in Table 4.3.2.2-1.

Table 4.3.2.2-1. Rendezvous Radar Characteristics

Antenna Feed	Cassegrainian
Antenna Reflector Main	Parabolic Dish 24 in. Diameter
Antenna Reflector Sub	Hyperbolic 4.65 in. Diameter
Angle Track	Amplitude monopulse
Antenna Gain (transmit and sum receive)	32 db over isotropic
Modulation	3 Tone-phase
Polarization	Circular
Beamwidth (transmit and sum receive)	3.25 deg - 4.0 deg
Radiation Frequency	9832.8 mhz
Receiver Frequency (from transponder)	9792 mhz + doppler
Radiated Power	240 mw minimum
Receive Signal Level	-18 dbm to -122 dbm
Receiver Channels	3 (shaft, trunnion, sum)
IF Frequencies	40.8 mc, 6.8 mc, and 1.7 mc
Range	80 feet to 400 n mi
Range Rate	±4900 fps
Angular Accuracy:	

Table 4.3.2.2-1. Rendezvous Radar Characteristics (Continued)

Range (n mi)	Maximum Error (mr) Random	Bias
400	4	8
300	3	8
200	2	8
5	2	8
2	5	8
1	10	8
(80 ft)	10	8

Angle Rate Accuracy:

Range	Maximum Error
400 n mi	0.4 mr/sec
200 n mi to 1000 ft	0.2 mr/sec

Range Accuracy:

Range (n mi)	Random Error	Bias Error
400 - 5	1/4% or 300 ft	>50.8 n mi ±500 ft
5 nm - 80 ft	1% or 80 ft	<50.8 n mi ±80 ft
Range Rate Accuracy	±1 fps	± 1 fps

4.3.3 Abort Guidance Section Elements

The Abort Guidance Section is comprised of the abort sensor assembly, the abort electronics assembly, and the data entry and display assembly. A description of the assemblies is presented in the following paragraphs. See Figure 4.3.3.1-1 for a functional block diagram of the AGS.

4.3.3.1 Abort Sensor Assembly. A brief description of ASA component identification, function, mechanization, detailed performance requirements, electrical characteristics, ASA temperature control, power requirements, and mechanical characteristics is presented in the following paragraphs.

4.3.3.1.1 Component Identification. The principal components of the ASA are as follows:

a) Three gyro subassemblies

b) Three accelerometer subassemblies

c) One power supply subassembly

d) Six pulse torquing electronics

e) One frequency countdown unit

f) Warmup and fine temperature control amplifier

g) Interface electronic group

4.3.3.1.2 Function. The function of the ASA is to sense motion about and along its orthogonal axes and convert this sensed motion into a form useful to the AEA for computation. These sensed motions, derived from three gyros and three accelerometers, represent incremental angles and incremental velocity. The accelerometers sense change in motion along the vehicle orthogonal axes (mutually perpendicular) and the gyros sense motion around the orthogonal axes. For definition of these axes, see Figure 4.3.3.1-2.

4.3.3.1.3 Mechanization. The ASA is mechanized with three distinct modes of operation consisting of the off mode, warm-up mode, and standby/operate mode.

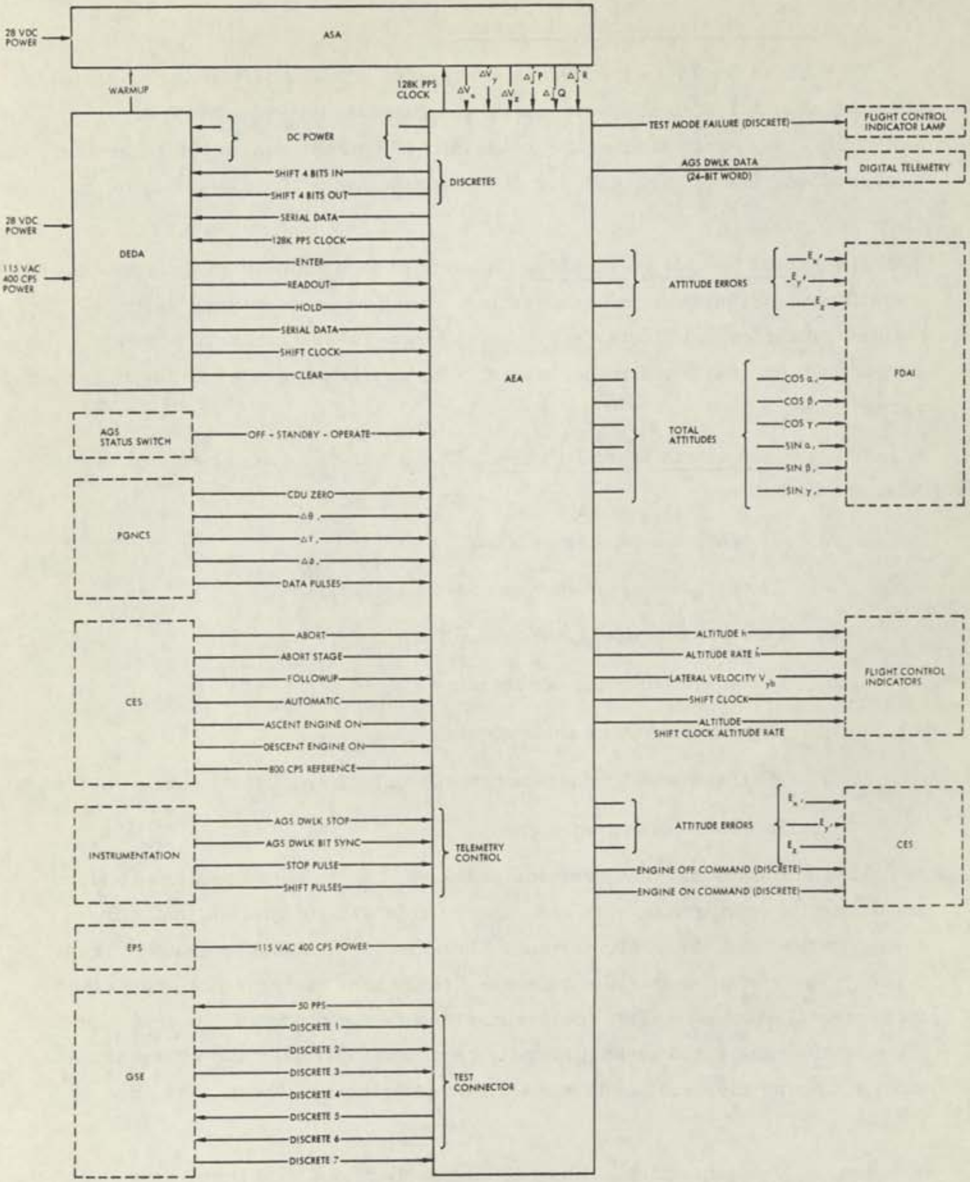


Figure 4.3.3.1-1. Functional Block Diagram of AGS and Interfaces

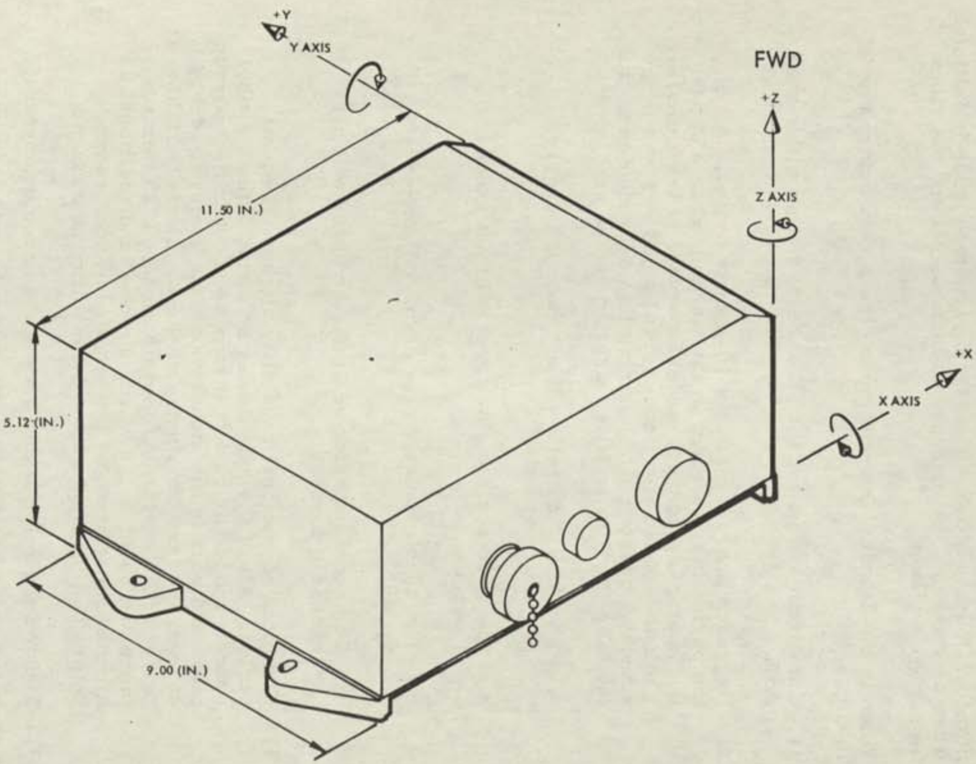


Figure 4. 3. 3. 1-2. Abort Sensor Assembly

a) Off Mode - In the off mode, no power is applied to the ASA.

b) Warmup Mode - The warmup mode is initiated when 28 volts dc and the external warmup signal are applied.

c) Standby/ Operate Mode - The standby/ operate mode is initiated when 28 volts dc, the external warmup signal and the clock signal are present.

When in the Standby/ Operate mode, the ASA provides the following:

1) Excitation and power for all subassemblies within the ASA.

2) Degraded operation after a 10-minute period. All of the period may be spent in the warmup mode and part in the standby/ operate mode; however, at least the last 5 minutes of the 10-minute period must be spent in the standby/ operate mode. During this degraded operation, the following limits of degradation can be predicted:

   a. The ASA drift rate will not exceed 6 degrees per hour per axis.

   b. The gyro scale factor error will not exceed 1 percent.

   c. The accelerometer bias error will not exceed 8×10^{-4} g.

   d. The accelerometer scale factor error will not exceed 1 percent.

3) Complete operational capability after a 40-minute period. All of the period may be spent in the standby/ operate mode, or part of it may be spent in the warmup mode and part in the standby/ operate mode; however, at least the last 25 minutes of the 40-minute period must be spent in the standby/ operate mode. If 25 minutes or more are spent in the warmup mode, and additional 25 minutes must be spent in the standby/ operate mode before complete operational capability is achieved.

Figure 4. 3. 3. 1-3 shows the relationship between the accelerometers, gyros, pulse torquing servo amplifiers, power supply, and frequency countdown subassembly. Only the X-axis accelerometer and gyro are shown; other accelerometers and gyros receive identical signals from the frequency countdown and power supply, with exception of signal conditioner gates A and B that are used by the X-axis accelerometer only.

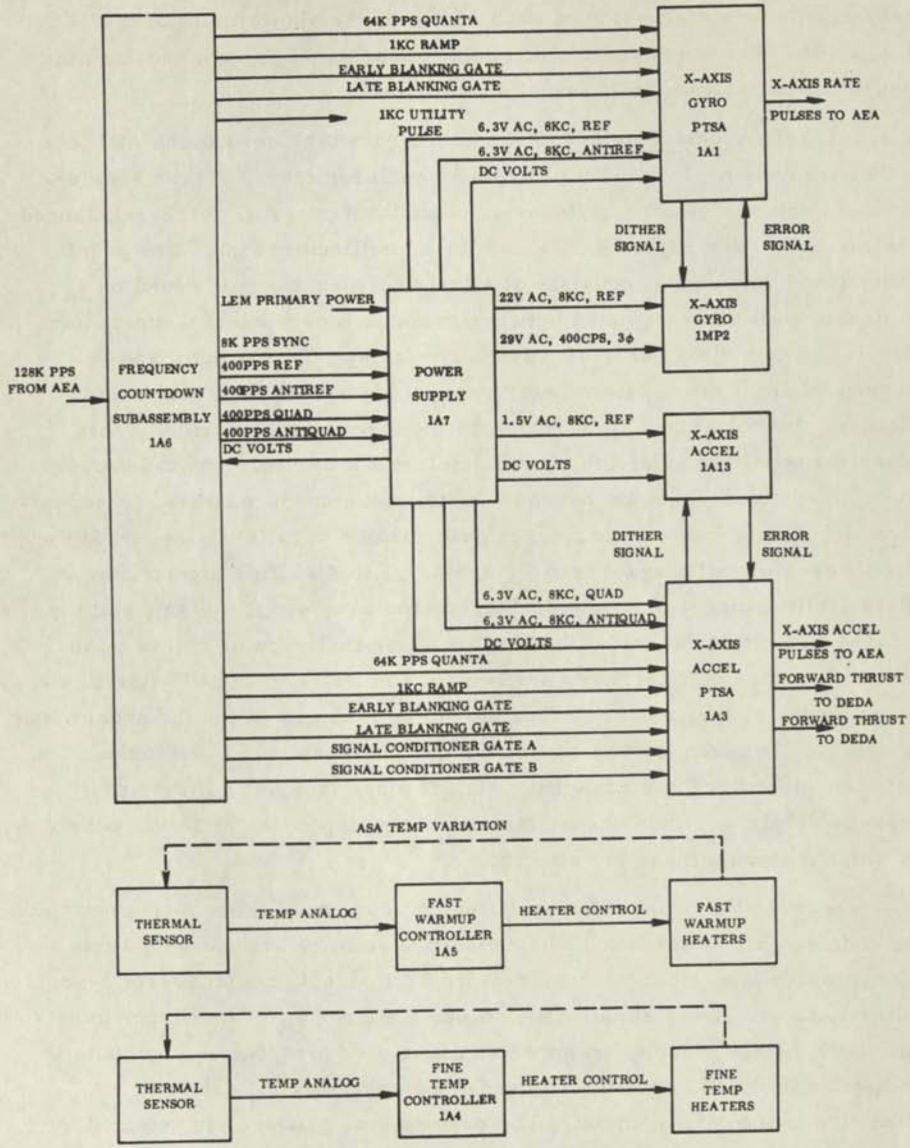


Figure 4.3.3.1-3. ASA, Simplified Block Diagram

ASA temperature control loops are also shown. The subsequent paragraphs provide a discussion of each of the blocks shown in Figure 4.3.3.1-3. Three interface electronics subassemblies, not shown in the figure but part of the ASA, are also described.

4.3.3.1.3.1 Accelerometers. The accelerometers used in the ASA are Bell Aerosystems Type VIIB pendulous accelerometers. These accelerometers are pendulous, single-axis, permanent-magnet, force-rebalanced instruments. See Figure 4.3.3.1-4 for an outline drawing. The proof mass (pendulous mass) consists of a force or torquing coil wound on a metallic spool and suspended between two alnico permanent magnets perpendicular to the axis of motion (sensitive axis). This proof mass is suspended from the accelerometer case by two springs held in place by clamps. In the absence of an accelerating force, which normally acts along the sensitive axis, the torquing coil is not excited, and the proof mass is suspended exactly between the two permanent magnets. A capacitive pickoff ring has an inner concentric surface parallel to the end of the proof-mass metallic spool (see Figure 4.3.3.1-4). This pickoff ring is electrically isolated from the accelerometer case and forms one plate of a large capacitor; the metallic spool on which the torquing coil is wound forms the other plate. There are two such rings, one on each side of the proof mass, hence, two such capacitors are formed. When the proof mass is centered between the two capacitive pickoff rings, the capacitance between each of the two capacitive pickoff rings is equal. If the proof mass is displaced, these capacitances change depending on the direction in which the proof mass is deflected.

Whenever the proof mass is deflected, one capacitance increases and its counterpart diminishes. The capacitive sensors are connected into a bridge which is excited by the 1.5-volt, 8-kc signal provided from 8-kc filter 1A7A6 in power supply 1A7. Since the change in capacitance unbalances the bridge circuit, an error signal is produced whose magnitude is indicative of the displacing force and whose phase is indicative of the direction of the displacement. The error signal produced is detected by the accelerometer electronics (not shown in Figure 4.3.3.1-4) and provided as an output for further signal processing. Under normal circumstances, this error signal is phase-demodulated in the pulse torquing

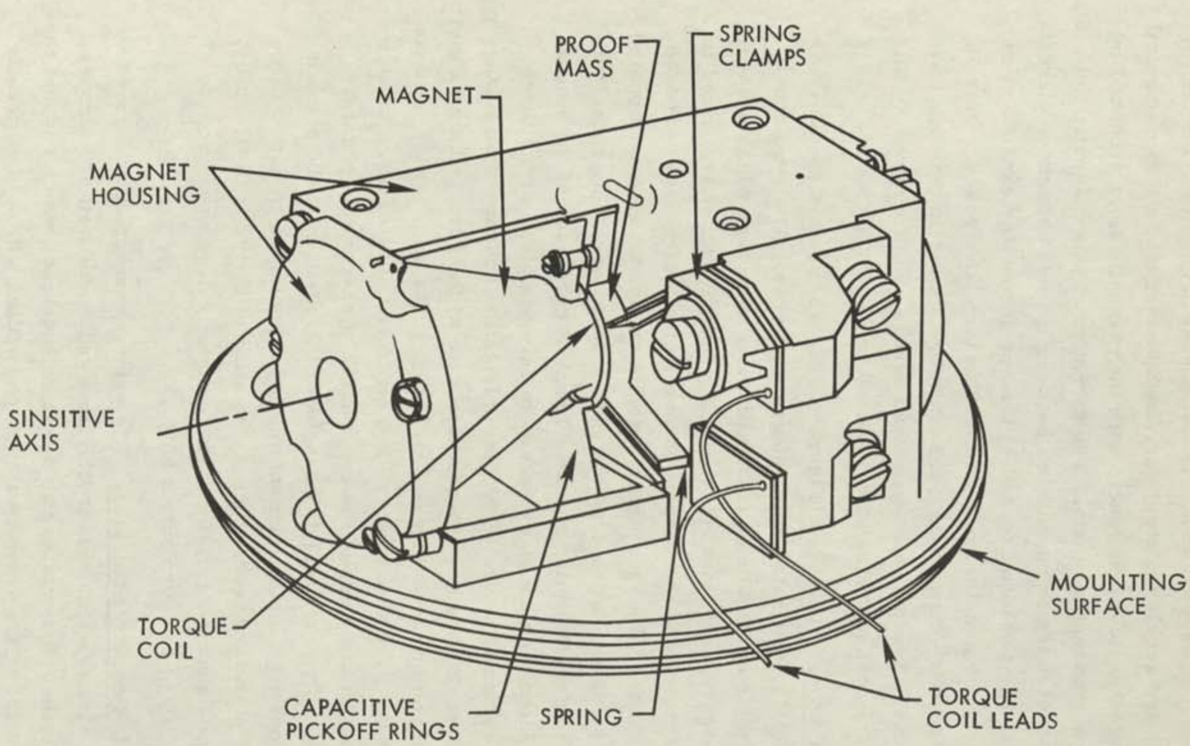


Figure 4.3.3.1-4. Pendulus Accelerometer Assembly

servo amplifier (PTSA) to produce a dc output whoe polarity is indicative of the amount of the displacement. The dc signal is used to drive the torquing coil in the accelerometer; current flow through this torquing coil produces a magnetic field that reacts with the magnetic field produced by the two alnico permanent magnets. The direction of this magnetic field is such that the proof mass is recentered between the two capacitive pickoffs. Current produced in the torquing coil of the accelerometer expends power that tends to heat the accelerometer which, in turn, increases errors in temperature-sensitive components; these errors appear in the output as displacing forces. Temperature-produced errors are minimized by the forced limit cycle concept which is explained in Paragraph 4.3.3.1.3.3.

4.3.3.1.3.2 Gyros. The gyros (Figure 4.3.3.1-5) used in the ASA are Norden Type R1 1139B floated rate-integrating gyros. The gyro comprises three major assemblies: (1) the primary structure which houses the pickoff, torquer, and float, (2) the float assembly which is pivoted in the primary structure and is suspended in a high-density fluid to reduce pivot friction, and (3) the spin motor assembly which is bearing-mounted within the float and which provides the gyro with its operating characteristics (ability to sense angular rates). The ASA gyros are of the single-degree-of-freedom type and can precess in only one plane of the three mutually perpendicular axes. A typical orientation scheme of an ASA gyro indicates the three mutually perpendicular axes of the gyro and shows their relation to one another (see Figure 4.3.3.1-5). The spin axis is the axis of rotation for the gyro spin motor. For any force exerted at right angles to the spin axis (in the plane of the input axis), a precession occurs, because the gyro flywheel resists changes in the position of the plane in which it develops its angular momentum. The developed resistive force acts in a third plane and tends to cause rotation of the gyro float. Thus, for any rotation about the gyro input axis, there is developed force to cause rotation about the gyro output axis.

4.3.3.1.3.2.1 Primary Structure. The primary structure is formed by two machined castings that support the float and the pickoff and torquer. The primary structure contains two sapphire endstones which support the float pivots. Part of the primary structure is filled with a high-density

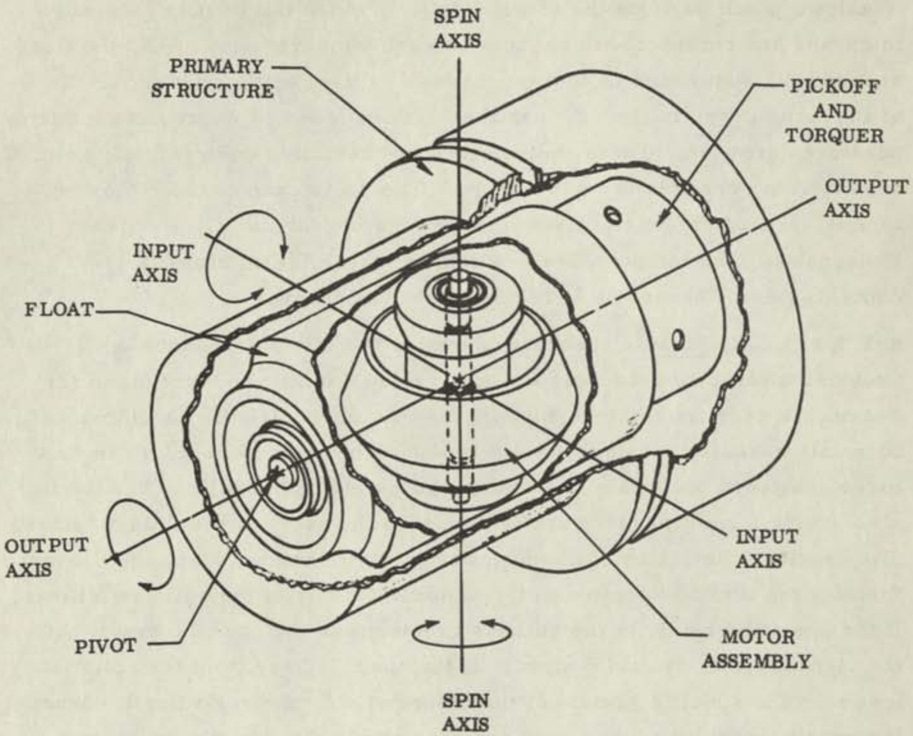


Figure 4.3.3.1-5. Integrating Gyro Assembly

viscous fluid that supports the float and completely fills the space between the primary structure and the float. One end of the primary structure has a bellows which permits the viscous fluid in which the float is suspended to expand and contract with changes in gyro temperature. Thus, the float assembly is suspended in, but permitted to rotate within, the fluid. Each of the three gyros in the ASA has thermistors mounted on its case for temperature detection. These thermistors are resistive devices having a high sensitivity to changes in temperature. The resistance of these thermistors is 5376 ± 100 ohms at a nominal gyro temperature of 120 degrees Fahrenheit. The temperature gradient for these thermistors is 112 ohms/degrees Fahrenheit at 120 degrees Fahrenheit.

4.3.3.1.3.2.2 Float. The float assembly has a two-fold function: (1) it provides a structural support for the gyro spin motor and wheel and (2) provides a near frictionless support for the output pickoff. Archimedes' principle asserts that when a solid is placed in liquid, the solid exerts a force downward on the liquid equal to the weight of the solid. The liquid also exerts a buoyant force upward equal to the mass of the fluid displaced. The result of the action of the downward force of the solid and the buoyant force of the liquid is related to the specific gravity of the solid as follows. If the specific gravity of the solid is greater than the specific gravity of the viscous fluid, the solid sinks. If the specific gravity of the solid is less than the specific gravity of the viscous fluid, the solid floats. When the specific gravity of the liquid and the solid are equal, the solid is suspended at any level in the liquid at which it is placed. The specific gravity of the viscous fluid in the gyro changes with changes in temperature, and, for this reason, the temperature of the gyros must be closely maintained. An advantage of flotation is viscous damping, which provides an integrating action that makes the gyro float less susceptible to vibration and shock along its input axis and cushioning of the float against shock and vibration along its output axis. The chief drawback of the flotation scheme is that the temperature of the viscous fluid must be carefully maintained, and hence, a temperature control system is needed. The temperature control circuit requires large amounts of power with an attendant sacrifice of space and weight in the system in which the gyro is installed.

4.3.3.1.3.2.3 Spin Motor Assembly. The spin motor assembly is a four-pole, synchronous hysteresis motor housed within the float and includes a wound pole stator, rigidly affixed to the gyro float by stator shaft supports; a magnetic hysteresis ring having four induced poles, which is attached to the flywheel; and the flywheel, supported by two bearings on the stator shaft, which forms the rotor of the hysteresis motor. Three-phase excitation of the stator produces a rotating magnetic field. The rotating magnetic field drags the hysteresis ring and the flywheel around with it, since this ring is attached to the flywheel. The hysteresis ring reacts with and follows the rotating stator field.

The hysteresis motor has the property of maintaining at synchronous speed any load that it can accelerate from a dead standstill. The flywheel rotates at 12,000 revolutions per minute and imparts the required angular momentum to the flywheel, which imparts gyroscopic action. The float chamber housing is filled with helium. Helium is a good conductor of heat and carries the heat away from the rotor and spin motor to the float wall into the viscous fluid in the primary structure. Each gyro has an auxiliary inductive pickoff in proximity to its spin motor. This inductive pickoff is excited by a small magnet each time the magnet passes the pickoff, so that for each revolution of the spin motor flywheel, a pulse is induced in the pickoff. The output from the pickoffs is routed to the test connector so that during test gyro spin motor velocity can be measured.

4.3.3.1.3.2.4 Pickoff and Torquer. The pickoff and torquer are installed in one end of the primary structure. The pickoff is an air-core differential transformer having a four-pole primary and a two-pole secondary winding. The torquer is an eight-pole D'Arsonval torque generator and operates much like a dc meter movement which reacts with a permanent magnet attached to the primary structure.

The pickoff can be considered as a transformer having a stationary primary winding and a rotatable secondary winding. The primary winding is attached to the primary structure and is excited by the 22-volt, 8-kc voltage from 8-kc filter subassembly 1A7A6 in power supply 1A7. The secondary consists of an unexcited winding located within the torquer core, which rotates with the float. Because of its position relative to the excited primary winding, when the float is in its null position, no voltage is induced in the pickoff secondary. However, when the float is rotated in either direction, the relative position between the secondary and the excited primary changes, and the excited primary induces an ac voltage in the secondary. The magnitude of this voltage is proportional to the amount of float displacement and its phase is indicative of the direction of float displacement. This voltage is phase-demodulated by PTSA electronics and quantized and converted to a torquing signal that is fed back to the torquer to return the float to the null position. The current flowing in the torquer, which is attached to the float and rotates with it, sets up a magnetic field that reacts with the field produced by the permanent magnet attached to the primary structure. The reaction between the magnetic field produced in the torquer windings and the magnetic field produced by the permanent magnet creates sufficient torque to return the float to its null position. Current flow in the gyro tends to heat the gyro, which, in turn, increases errors in temperature-sensitive components. These errors appear at the output axis as a displacing force at the input axis. Temperature produced errors are minimized by the forced limit cycle mode.

4.3.3.1.3.3 Forced Limit Cycle. Any periodic motion which always goes to some positive amplitude as a limit and from the positive amplitude limit to some equal negative amplitude limit can be considered as a forced limit cycle. This motion need not be symmetrical; i.e., having identical positive and negative periods, but its total period must always remain constant as must its amplitude with respect to its positive and negative periods. For such a motion, the power dissipated during the total period is always the same, although the average value with respect to the positive and negative amplitudes can change proportionately with the duty cycle.

In waveform A of Figure 4.3.3.1-6, the areas for the positive and negative excursions are 25 units each. Since the areas below and above the reference line are equal, the average value for this periodic waveform is zero. However, the sum of these two areas (without regard for sign) is the total power expended to produce the motion, or 50 units. The duty cycle

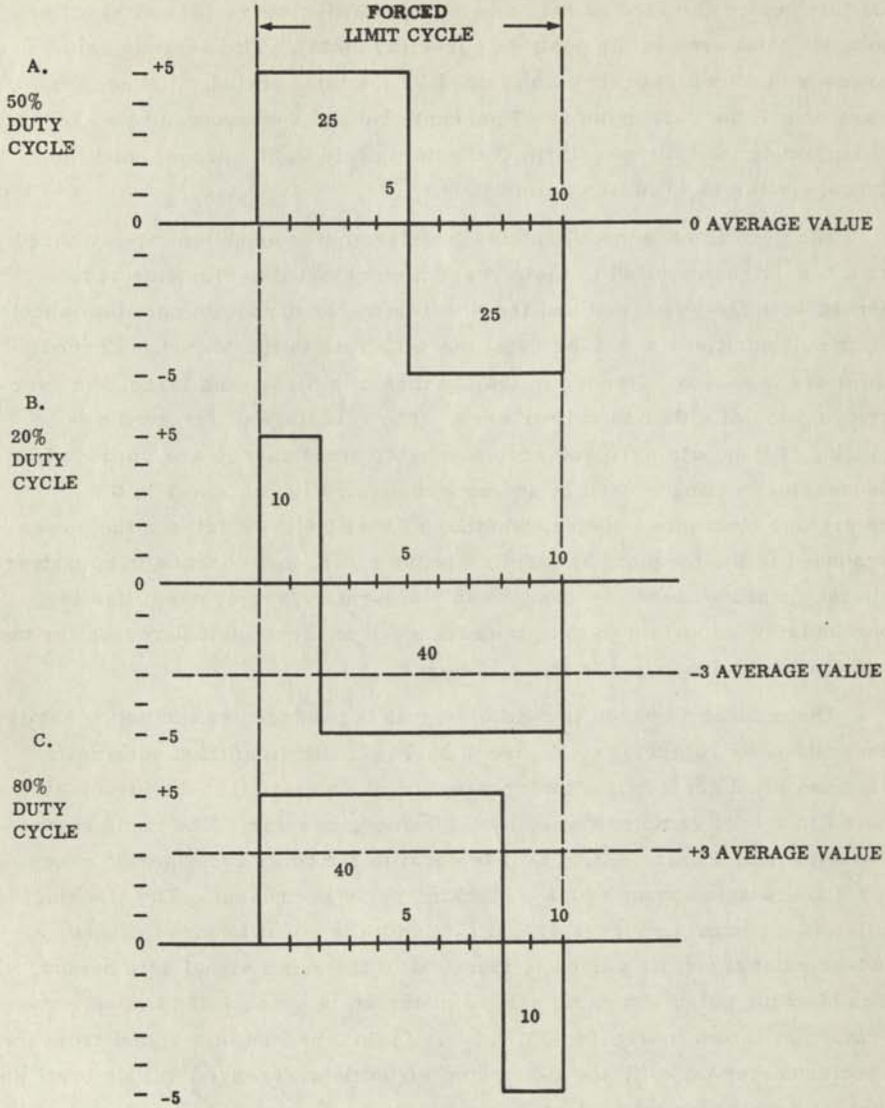


Figure 4.3.3.1-6. Forced Limit Cycle, Timing Diagram

for this period is taken as the area of the positive curve (25 units) to the possible total area of the positive curve (50 units). The average value, however, is the algebraic sum divided by the total period. Hence, for waveform B the duty cycle is 20 percent, but the average value is -3 units of amplitude; and for waveform C the duty cycle is 80 percent, and the average value is +3 units of amplitude.

The gyro float (sensor) and the accelerometer pendulum are dithered by a 1-kc signal applied to their respective retorquing elements and, hence, both the gyro float and the accelerometer pendulum oscillate about their null positions at a 1-kc rate; the 1-kc rate is the forced (enforced) limit of the period. Hence, in the absence of a displacing force, the average position of either is null or zero. The advantage of forced limit cycling is that both gyro and accelerometer performance are improved by decreasing heating caused by the uneven dissipation of power in the retorquing elements. Hence, whether at rest (null) or driven, the power expended in the torquers is identically the same, and changes in operating characteristics caused by changes in temperature are avoided; this is particularly important in gyro operation and is the principal reason for the low drift rate attributed to the ASA gyros.

The manner in which the dither signal is produced can be more easily understood by referring to Figure 4.3.3.1-7, the simplified schematic diagram of an accelerometer torquing loop. Operation of the circuit is keyed to a 1-kc ramp produced by the ramp generator. The ramp period (waveform B, Figure 4.3.3.1-8) is equal to the limit cycle period except for a short time during which a blanking pulse is present. The blanking pulse (waveform A, Figure 4.3.3.1-8) width is equal to three quanta (64-kc pulses) and its period is identical to the ramp signal gate period. The blanking pulse and ramp gate is produced in a precision timing generator not shown in Figure 4.3.3.1-7. In the absence of a signal from the accelerometer pickoff, the 1-kc ramp signal is referenced to a dc level so that its zero corssover (point 2, waveform B, Figure 4.3.3.1-8) occurs at exactly half the limit cycle. The dc level to which the ramp is referenced is established by the summing network. Each time the ramp signal exceeds the zero (or crossover) level, the crossover detector responds

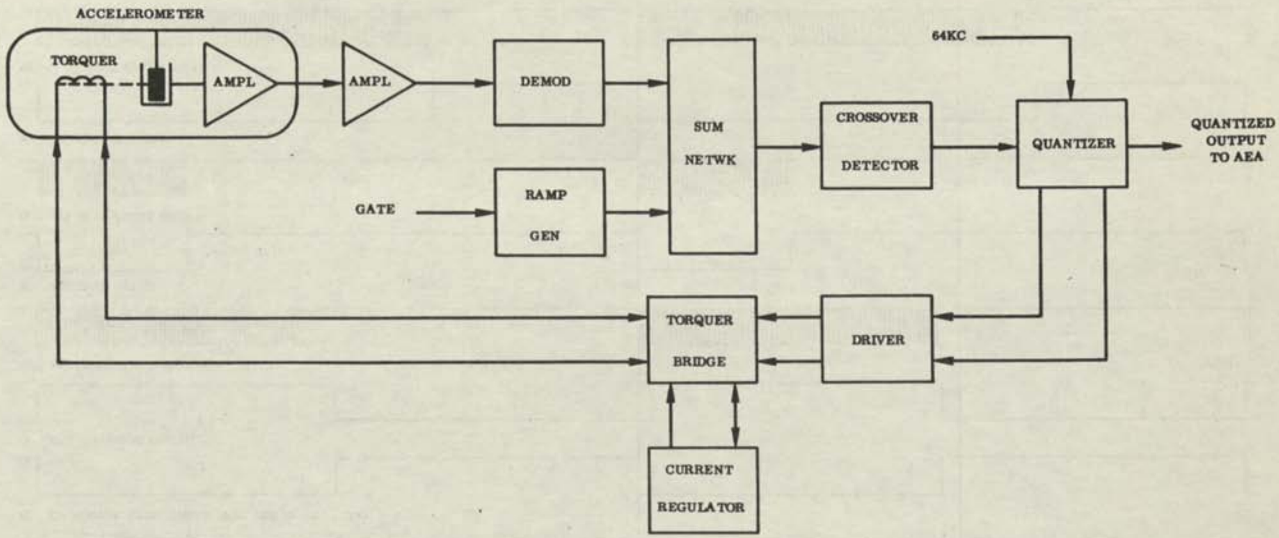


Figure 4.3.3.1-7. Accelerometer Torquing Loop, Block Diagram

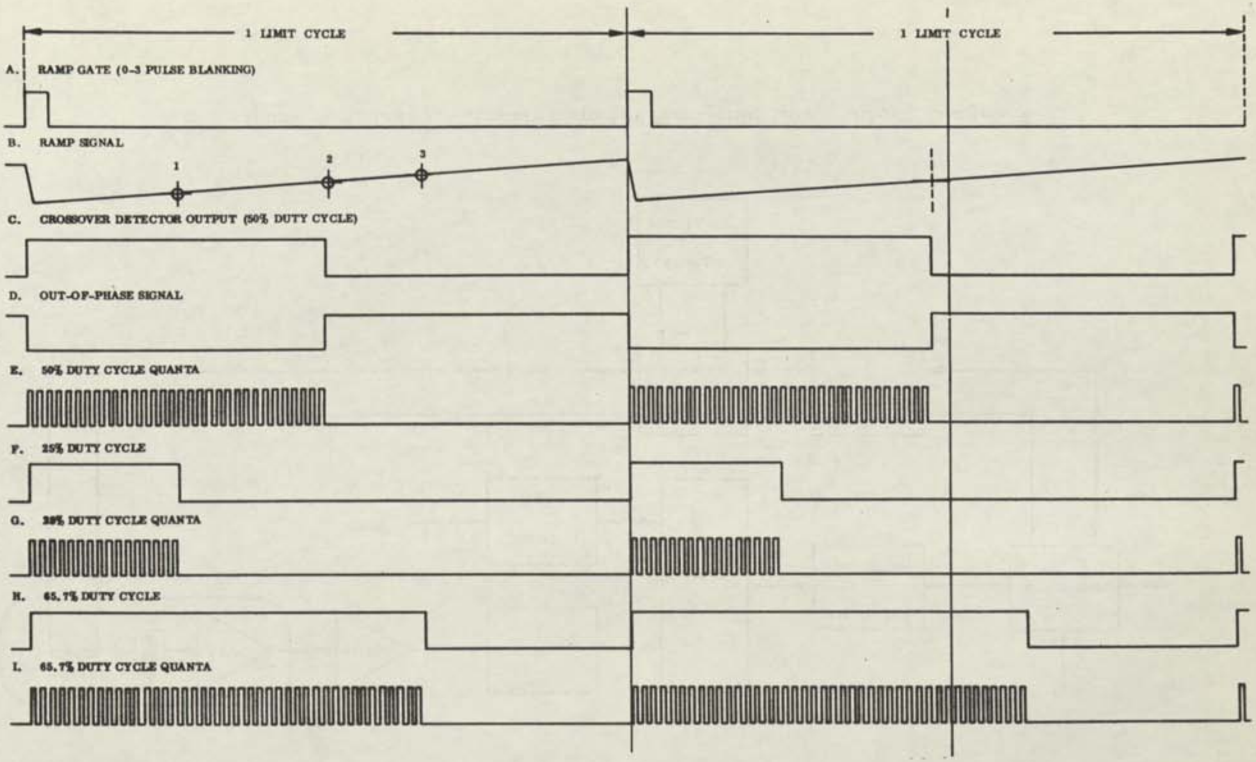


Figure 4.3.3.1-8. Accelerometer Torquing Loop, Timing Diagram

with an output pulse (waveform C), and, with no output from the accelerometer, the period of this pulse corresponds to a 50 percent duty cycle of the forced limit period.

The output of the crossover detector is a pulse whose duty cycle is proportional to the zero crossover of the ramp signal and whose repetition rate is equal to the forced limit cycle. This pulse is applied to the quantizer, Figure 4.3.3.1-7, and is used to gate continuous 64-kc pulses (waveform E, Figure 4.3.3.1-8) through the quantizer for application to the AEA. The quantizer also reproduces the original pulse information from the crossover detector output pulse (waveform C) and the other out of phase (antiphase) with the crossover pulse (waveform D). The reference and antiphase signals from the quantizer are used to control the driver, (Figure 4.3.3.1-7) which, in turn, controls the torquer bridge. Thus, the torquer bridge is switched on and off at the duty-cycle rate. The load for the torquer bridge is the accelerometer torquer as indicated in Figure 4.3.3.1-7. Anomalies in switching transistor characteristics or changes in the supply voltage can cause current variations that appear to the torquer as changes in the detected duty cycle. Such disturbances are minimized by careful control of the current permitted to flow through the torquer. The current regulator senses the switched current flowing through the torquer and feeds back a voltage proportional to the difference between a reference and the sensed current.

Since the quanta are derived from continuous 64-kc pulses, 32 such pulses (waveform E, Figure 4.3.3.1-8) will be gated through the quantizer when the duty cycle is 50 percent; hence, 32 quanta per 1/1000 second are equivalent to zero output (no error signal) from the accelerometer (or gyro). Only whole pulses are generated, since there can be no distinction for partial quanta. However, excess and missing partial pulses tend to average out over many limit cycles. The AEA accumulates pulses over 20 limit cycles to obtain such a self-cancelling average.

Any time an accelerometer pendulous mass (or gyro float) is displaced, the duty cycle of the dither signal changes. When the pendulous mass in the accelerometer is displaced, the accelerometer amplifier produces an 8-kc output signal having a phase and magnitude indicative of the direction and amount of displacement. Considering, for the moment, that such a displacement has occurred and that it is negative (positive and negative polarities and directions are arbitrarily taken for the purpose of this description) the following takes place: The output from the accelerometer amplifier is applied to the preamplifier in the PTSA. The output from this amplifier is compared with the reference 8-kc signal, and a dc voltage proportional to displacement magnitude, whose polarity is negative, is supplied from the demodulator to the summing network. In the summing network, this dc level is algebraically added to the ramp signal that is produced by the precision timing generator. The resultant signal from the summing network is a ramp whose dc reference has changed. This change in the ramp dc level effectively changes the dc crossover level so that it occurs at a lower point on the ramp slope (point 1, waveform B, Figure 4.3.3.1-8). The resultant output pulse from the crossover detector (waveform F) is a pulse having a shorter duty cycle (25 percent for the example shown). A 25 percent duty cycle permits only 16 quanta (waveform G) to be gated through the quantizer for application to the AEA.

The duty cycle of the dither signal has now changed. If the positive-going portion of the crossover detector output pulse is considered as the length of time that current flows negative in the (reference) direction, and the negative-going portion of the same signal is considered as the length of time that current flows in the positive (opposite) direction, it can be seen from waveform F, Figure 4.3.3.1-8, that positive-going current flows 25 percent longer than negative-going current during the 1-kc forced limit cycle. Effectively, this longer-lasting positive-going current produces a torquer force that rebalances the negatively deflected pendulous mass in the accelerometer. When the original displacing force no longer exists, the dither signal reverts to the 50-percent duty cycle. It can be seen from the foregoing example that when the dither signal duty cycle is less than 50 percent, the displacement direction is negative and the number of quanta gated to the AEA during the limit cycle are inversely proportional to the magnitude of the negative displacing force.

When the displacement of the pendulous mass in the accelerometer is positive, the change in ramp dc level effectively changes the crossover point so that it occurs at a higher point on the ramp slope (point 3, waveform B, Figure 4.3.3.1-8). The resultant output pulse from the crossover detector (waveform H) is a pulse having a longer duty cycle (65.7 percent for the example shown). A 67.7-percent duty cycle permits 42 quanta (waveform I) to be gated through the quantizer for application to the AEA.

If the relative directions for positive and negative current flow through the torquer are taken to be the same as previously established, it can be seen from waveform H, Figure 4.3.3.1-8, that negative-going current flows 15.7 percent longer than positive-going current during the 1-kc forced-limit cycle. Effectively, the longer-lasting negative-going current produces a torquer force that rebalances the positive-deflected pendulous mass in the accelerometer. When the original displacing force no longer exists, the dither signal reverts to the 50-percent duty cycle. It can be seen from the foregoing example that when the dither signal duty cycle is greater than 50 percent, the displacement direction is positive and the number of quanta gated to the AEA during the limit cycle are directly proportional to the magnitude of the displacing force.

4.3.3.1.4 Detailed Performance Requirements. The following detailed performance requirements were taken from GAEC LSP-300-37E.

4.3.3.1.4.1.1 Combined Rates. The ASA is capable of following rotational rates up to and including 25 deg/sec consecutively or concurrently about all vehicle axes with a maximum lag of ±0.05 deg/axis.

4.3.3.1.4.1.2 Following Errors.

4.3.3.1.4.1.2.1 Steady-state Following Error. The ASA is capable of following rotational rates up to and including 25 deg/sec about each axis with a maximum steady-state following error of 30 arc seconds (excluding the time delay caused by output signal quantization).

4.3.3.1.4.1.2.2 Dynamic Following Error. The instantaneous dynamic following error (including the time delay as a result of output signal quantization) shall not exceed 180 arc seconds under the following input conditions:

a) All input conditions as specified in Table V of LSP-300-37.

b) An angular acceleration step function of 2 rad/sec^2 driving the input angular rate from an initial value of plus (or minus) 25 deg/sec to a final value of minus (or plus) 25 deg/sec.

4.3.3.1.4.1.3 Drift Rate. The drift rate about each ASA axis shall not exceed the values specified below. In determining drift rate, the time period used for averaging is 260 seconds.

4.3.3.1.4.1.3.1 Gyro Channel Bias. The absolute value of the non-g-sensitive drift rate shall not exceed 2.4 deg/hr.

The bias instability over a 3-day period shall not exceed 0.67 deg/hr (3σ). The bias instability over an 8-day period shall not exceed 0.85 deg/hr (3σ). The bias instability over a period of 120 days shall not exceed 2.8 deg/hr (3σ). Residual bias changes caused by shock and vibration are included. Bias discrepancies as a result of orientation changes and transient changes during vibration shall not be included.

The difference in mean bias determined when the input axis is vertical (up and down) and the mean bias determined when the output axis is vertical (up and down) shall not exceed 0.20 deg/hr. The difference in mean bias determined when the spin axis is vertical (up and down) shall not exceed 0.20 deg/hr. The difference in mean bias determined when the output axis is vertical (up) and the mean bias determined when the output axis is vertical (down) shall not exceed 0.20 deg/hr.

4.3.3.1.4.1.3.2 Gyro Input Mass Unbalance. The absolute value of the input axis mass unbalance shall not exceed 4.0 deg/hr/g. The input axis mass unbalance instability over a period of 120 days shall not exceed 2.3 deg/hr/g (3σ).

4.3.3.1.4.1.3.3 Gyro Spin Axis Mass Unbalance. The absolute value of the spin axis mass unbalance shall not exceed 4.0 deg/hr/g. The input axis mass unbalance instability over a period of 120 days shall not exceed 2.3 deg/hr/g (3σ).

4.3.3.1.4.1.4 Gyro Channel Scale Factor. The scale factor of each gyro channel shall be 2^{-16} radians/pulse, plus or minus 4500 ppm. The scale factor instability over a period of 120 days shall not exceed 0.00087 deg/deg (3σ).

4.3.3.1.4.1.5 Gyro Channel Nonlinearity. For rotation rates from zero deg/sec to 22 deg/sec, the loss of attitude reference about each axis caused by gyro channel nonlinearity shall not exceed 0.00020 deg/deg. For rotation rates from 22 deg/sec to 25 deg/sec, the loss of attitude reference caused by gyro channel nonlinearity shall not exceed 0.00058 deg/deg.

4.3.3.1.4.1.6 Off-null Limitation. The gyro loops shall instantaneously maintain the gyro gimbal rotations about their output axis within 40 arc seconds of null under the following input conditions:

a) All input conditions as specified in Table V of LSP-300-37.

b) An angular acceleration step function of 2 rad/sec^2 driving the input angular rate from an initial value of plus (or minus) 25 deg/sec to a final value of minus (or plus) 25 deg/sec.

4.3.3.1.4.2 Velocity Increment Signals

4.3.3.1.4.2.1 Accelerometer Channel Bias. The absolute value of the bias of each accelerometer channel shall not exceed 600×10^{-6} g. The bias instability over a period of 3 days shall not exceed 113×10^{-6} g (3σ). The bias instability over a period of 120 days shall not exceed 585×10^{-6} g (3σ). The difference in the mean bias determined when the pendulous axis is vertical (up and down) and the mean bias determined when the output axis is vertical (up and down) shall not exceed 100×10^{-6} g. In determining bias the time period used for averaging is 260 seconds.

4.3.3.1.4.2.2 Accelerometer Channel Scale Factor. The scale factor of each accelerometer channel is 0.003125 ft/sec/pulse, plus or minus 0.1 percent. The scale factor instability over a period of 120 days shall not exceed 940×10^{-6} g (3σ).

4.3.3.1.4.2.3 Accelerometer Channel Nonlinearity. For positive and negative input accelerations from zero ft/sec^2 to 32 ft/sec^2, the error in sensed acceleration in each channel shall not exceed 100×10^{-6} g. For positive and negative input accelerations from 32 ft/sec^2 to 90 ft $\times 10^{-6}$, the error in sensed acceleration shall not exceed 100×10^{-6} g/g.

4.3.3.1.4.2.4 Thrust Axis Ternary Conditioner. Outputs from the X accelerometer channel ternary conditioner to the V display ($+\Delta VX$ and $-\Delta VX$) are inhibited for input velocity changes of less than 0.05 ft/sec.

4.3.3.1.4.2.5 Dynamic Following Error. The instantaneous dynamic following error in each channel (including the time delay caused by output signal quantization) shall not exceed 0.5 ft/sec when subjected to an input acceleration ramp function of 0 to 1 g in 50 milliseconds.

4.3.3.1.4.2.6 Limit Cycle Pendulum Motion. The accelerometer pendulum rotations about their output axes, caused by forced limit cycles under steady-state input acceleration, shall not exceed 40 arc seconds.

4.3.3.1.5 Electrical Characteristics. The electrical characteristics of the ASA are presented under input and output signals as described in the following paragraphs.

4.3.3.1.5.1 Input Signals. The following signals are furnished to the ASA.

Signal	Function
a) Clock	Reference for all internal synchronization. (128 kpps; 4V ± 20% = "1"; 0-0.5V = "0")
b) Warmup	To select warmup or standby/operate mode. (Switch closed = warmup; switched open = standby/operate)

4.3.3.1.5.2 Output Signals. Output signals from the gyros and accelerometers are processed and quantized in the PTSA's (pulse torquing servo amplifiers). All PTSA's operate in an identical manner except for the X-axis accelerometer PTSA, which receives the signal conditioner gates. The description of PTSA operation relates specifically to operation of X-axis accelerometer PTSA 1A3, but applied equally to all other PTSA's unless otherwise noted. See Table 4.3.3.1-1 and related figures for further definition of the outputs.

The PTSA (Figure 4.3.3.1-11) comprises four models: (1) amplifier and demodulatr A1, (2) quantizer A2, (3) bridge and driver A3, and (4) current regulator A4; the X-axis accelerometer PTSA 1A3 has a fifth

Table 4.3.3.1-1. ASA Output Signals

Signal	Type of Signal	Load Impedance	Source Impedance	Frequency and Accuracy	Level and Accuracy	Noise and Ripple	Scale Factor	Rise Time (10-90%)	Fall Time (90-10%)	Width (50% Pt.)	Positive Overshoot	Droop	Back-Swing
$\Delta\int P$ Output Pulses	Pulse (1)	*	(2)	64 kpps max	6v ±20% = 1 0 ±0.20 = 0	(3)	2^{-16} rad/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
$\Delta\int Q$ Output Pulses	Pulse (1)	*	(2)	64 kpps max	"	(3)	2^{-16} rad/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
$\Delta\int R$ Output Pulses	Pulse (1)	*	(2)	64 kpps max	"	(3)	2^{-16} rad/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
ΔV_x Output Pulses	Pulse (1)	*	(2)	64 kpps max	"	(3)	0.1/32 ft/sec/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
ΔV_y Output Pulses	Pulse (1)	*	(2)	64 kpps max	"	(3)	0.1/32 ft/sec/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
ΔV_z Output Pulses	Pulse (1)	*	(2)	64 kpps max	"	(3)	0.1/32 ft/sec/pulse	(4)	0.3 μs	1 ± 0.2μs	Less than 1.3v	Less than 40%	(5)
$+\Delta V_x$ to display	Pulse (1)	500 Ω (6)	(2)	0-2 kpps	"	No Req't	0.05 ft/sec/pulse	2μs	2μs	16μs ±30%	Less than 1.3v	Less than 40%	(5)
$-\Delta V_x$ to display	Pulse (1)	500 Ω (6)	(2)	0-2 kpps	"		0.05 ft/sec/pulse	2μs	2μs	16μs ±30%			
Block Temp Sensor	Resistance		(7)	NA	NA		(7)	NA	NA	NA	NA	NA	NA
28 v dc (Prec)	dc Voltage	50 K	10 K ±1%	NA	28v(8) ± 0.05%	5 mv p-p	NA	NA	NA	NA	NA	NA	NA
12 v dc	dc Voltage	50 K	10 K ±1%	NA	12v(8) ± (8)	100 mv p-p	NA	NA	NA	NA	NA	NA	NA
400 ~ 29 v	ac Voltage	100 K	10 K ±1%	400 H_z ± 1	29v(8) ± (8)	100 mv p-p	NA	NA	NA	NA	NA	NA	NA
Power Supply Common (Caution and Warning Return - 2 Connector Pins)					rms								
28 v dc	dc	500 m Ω		NA	28v nominal			NA	NA	NA	NA	NA	NA
ASA Clock	Pulses	(11)	(11)	128 kpps	(11) 4 ± 20% = 1 0-0.5v = 0								
Signal grd	NA	NA	NA	NA	NA			NA	NA	NA	NA	NA	NA

NOTES:
(1) Refer to Figure 4.3.3.1-9 for definition of pulse characteristics.
(2) As required to meet signal characteristics as measured across load.
(3) Such that the input signal is within -11 volts and + 11 volts and noise does not produce ambiquity (i.e. greater than + 2.5 volts = "1") less than + 1.0 volts = "0".
(4) Rise time -0.310 μ sec max for pulse width of 0.8 μ sec; 0.330 μ sec max for pulse width of 1.0 μ sec; 0.350 μ sec max for pulse width of 1.2 μ sec.
(5) Minimum backswing such that the AEA input transformer (57 volt -μ sec) recovers to 90%; maximum backswing -11 volts.
(6) Through a cable whose shunt capacitance is 875 ft maximum.
(7) In accordance with LSP-360-605A.
(8) The characteristics of the caution and warning voltages (28 vdc, 12 vdc and 29 v 400 cps) as called out herein are given under "No-load" conditions.
(9) ± 2% set point accuracy, ± 2% regulation.

*Figure 4.3.3.1-10

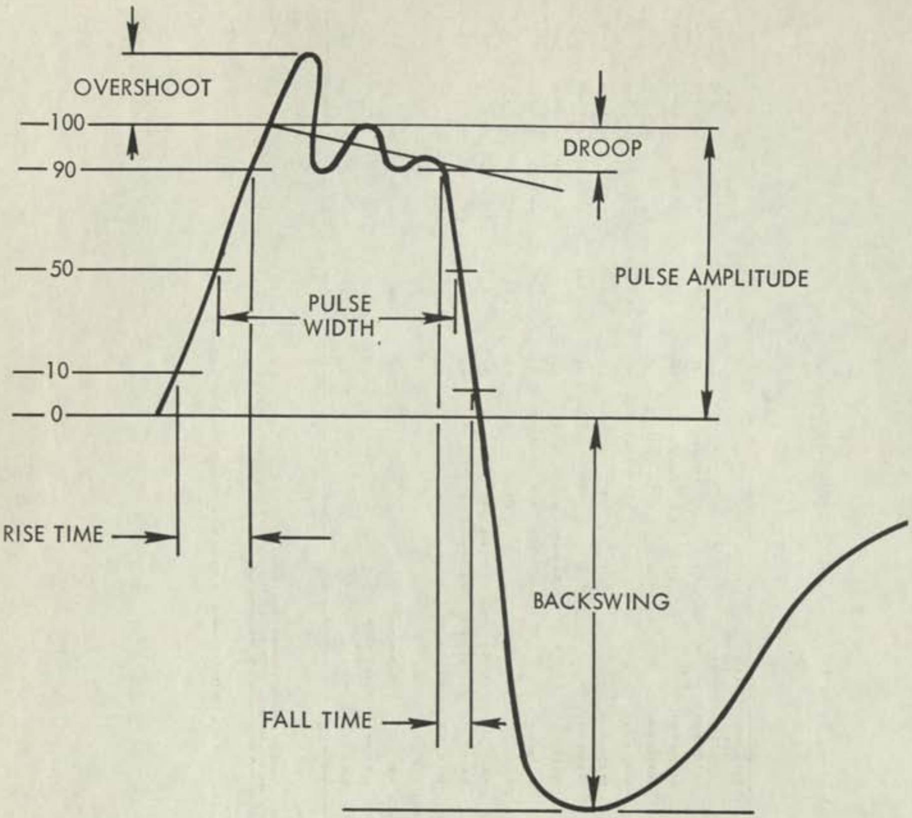
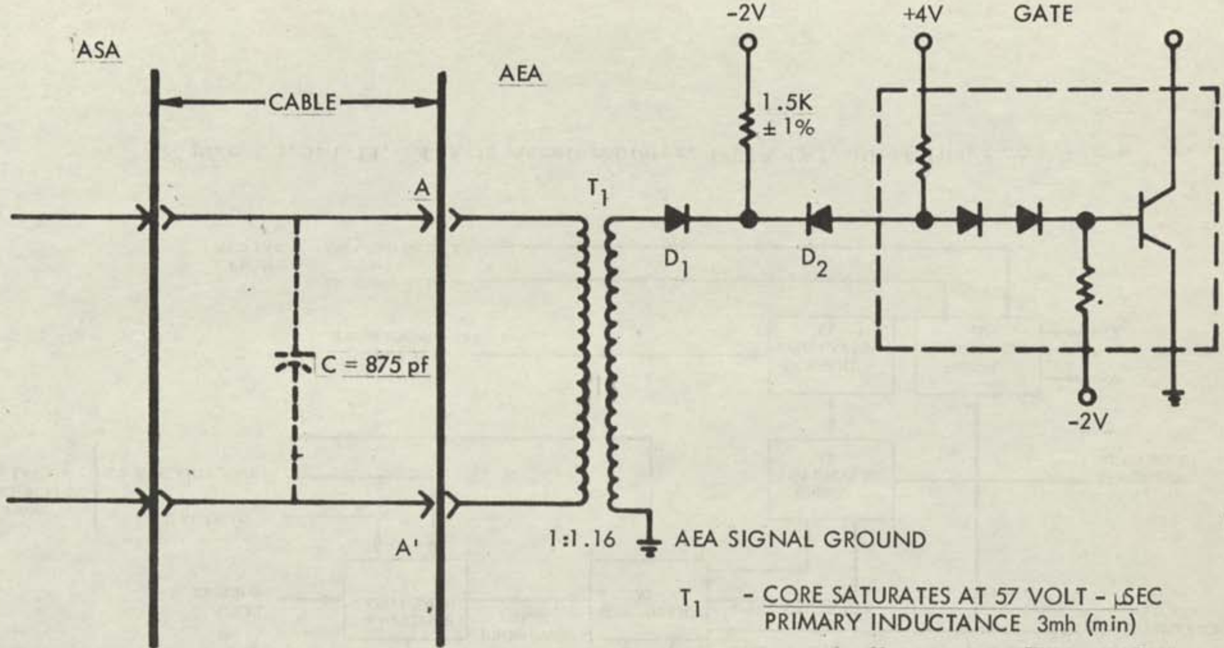


Figure 4.3.3.1-9. Pulse Waveform



T_1 - CORE SATURATES AT 57 VOLT - μSEC
PRIMARY INDUCTANCE 3mh (min)

D_1, D_2 - COMMERCIAL EQUIVALENT - FD6008

GATE - COMMERCIAL EQUIVALENT - SE101

Figure 4.3.3.1-10. Pulse Output Load Definition

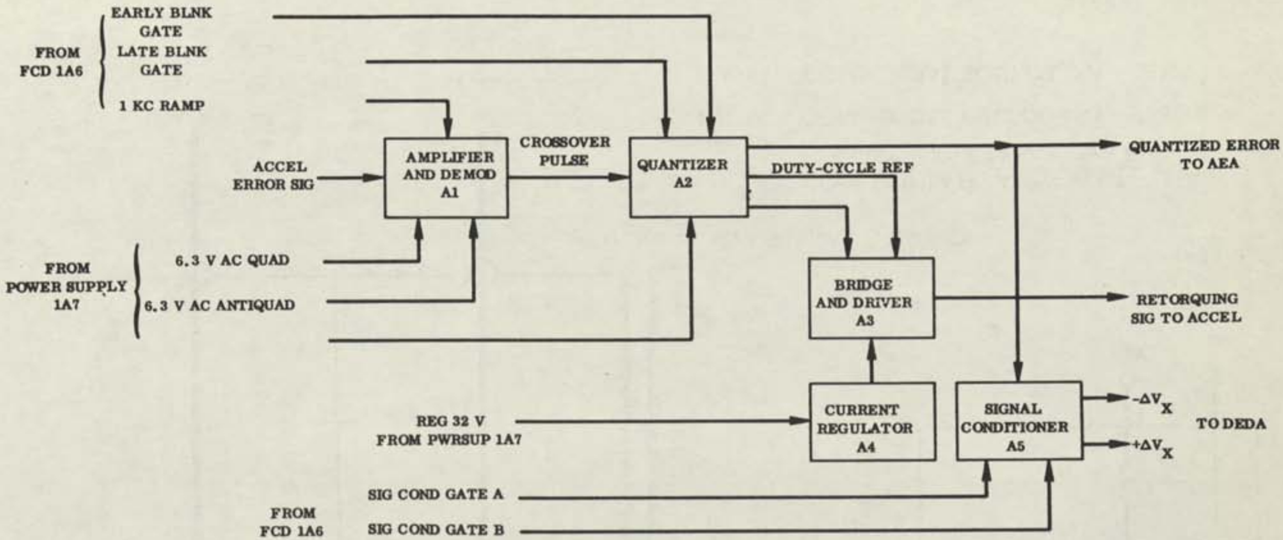


Figure 4.3.3.1-11. X-Axis Accelerometer, PTSA 1A3, Block Diagram

module called signal conditioner A5. Amplifier and demodulator A1 receives four signals: (1) the 1-kc ramp signal (waveform A, Figure 4.3.3.1-12), which fixes the rate at which the variable duty cycle retorquing signal is repeated, (2) the accelerometer (or gyro) error signal, and (3) and (4) two 6.3-volt, 8-kc quadrature and antiquadrature signals that are used to reference the phase demodulator section of the amplifier and demodulator. The gyro PTSA's receive reference and antireference 6.3-volt, 8-kc signals to operate their respective phase demodulators. The output from the amplifier and demodulator is the crossover pulse (waveforms C, F, and H, Figure 4.3.3.1-8) which is used to operate the quantizer and establish the duty cycle width.

The quantizer (Figure 4.3.3.1-11) also receives four signals: (1) input from the amplifier and demodulator, (2) and (3) the early and late blanking gates (waveforms F and G, Figure 4.3.3.1-12, respectively) from frequency countdown 1A6, and (4) 64-kpps from the frequency countdown. The output from the quantizer consists of the quantized digital pulses (waveforms E, G, and I, Figure 4.3.3.1-8) that represent the amount and direction of the error signal sensed by the accelerometer (or gyro). This module also produces two pulses representing the duty-cycle pulse width and rate (waveforms C, F, and H); one of these is a reference and the other is an antireference signal, which is used to operate the bridge and driver A3.

Bridge and driver A3 receives the two duty-cycle pulses and a voltage representing the current through the torquer. The current through the torquer winding must be maintained constant since any change in the current will appear as a change in the duty-cycle, and the duty-cycle alone is the controlling parameter of interest. The bridge and driver output is connected across the accelerometer (or gyro) torquer.

Current regulator A4 insures that the current switched through bridge and driver A3 will remain constant. This constant current is derived from the 32-volt regulated power source in power supply 1A7 and switched into the torquer load by the switching bridge that controls the direction in which current flows through the torquer coil and the length of time this current is permitted to flow.

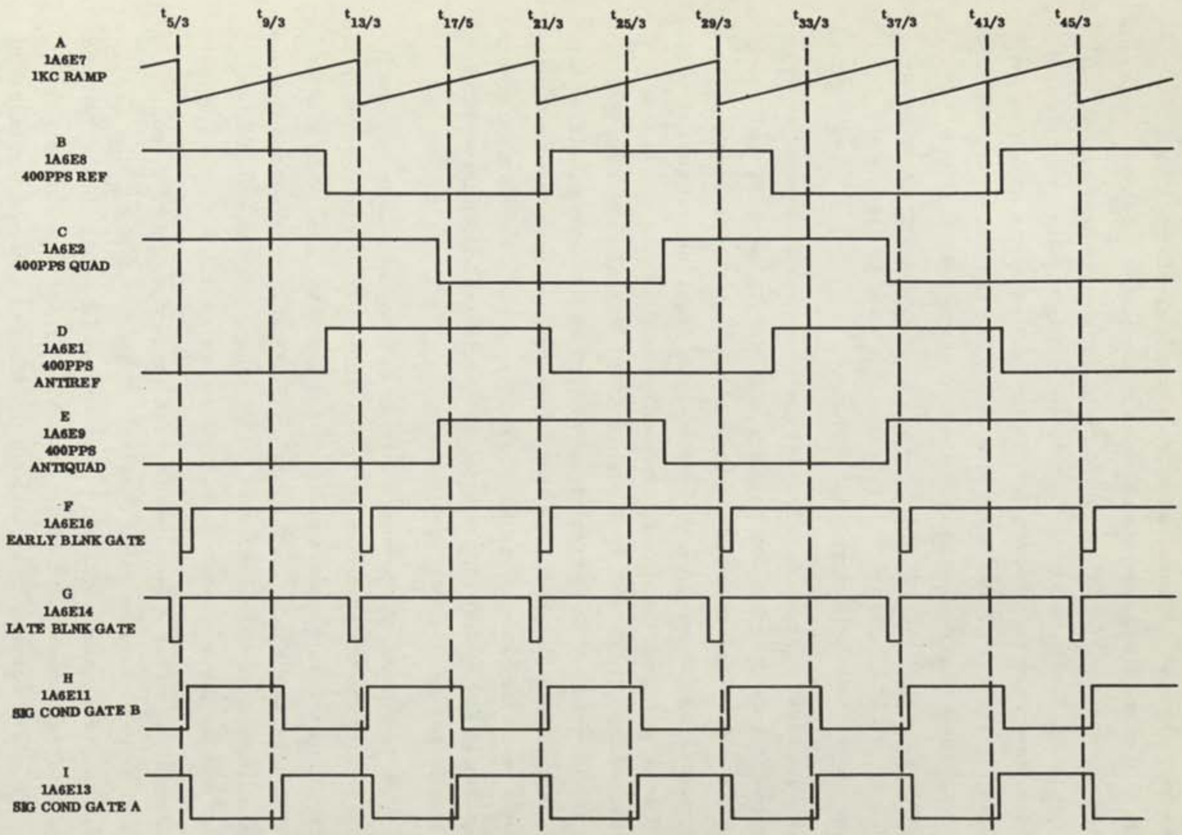


Figure 4.3.3.1-12. Frequency Countdown 1A6, Timing Diagram

In the X-axis accelerometer PTSA, signal conditioner A5 is used to change the scale factor of the 64kpps quanta from 0.003125/ft/sec/pulse to 0.05/ft/sec/pulse.

4.3.3.1.5.2.1 Output Signal Conditioning. For ASA 001, all input and output signals are processed through four signal conditioners called interface electronics subassemblies. These subassemblies condition output and input signals so that the ASA distribution and power circuits are isolated from other LM circuits; these four subassemblies also provide electromagnetic interference (EMI) isolation between the ASA and external LM circuits. The four interface electronic subassemblies are:

a) Interface electronics subassembly No. 1 (1A8)

b) Interface electronics subassembly No. 2 (1A15)

c) Interface electronics subassembly No. 3 (1A16)

d) Interface electronics subassembly No. 4 (1A17)

For ASA 002, all input and output signals are processed through three interface electronic subassemblies. These subassemblies also condition the output and input signals so that the ASA distribution and power circuits are isolated from other LM circuits; these three interface electronic subassemblies are:

a) Interface electronics subassembly No. 1 (1A15)

b) Interface electronics subassembly No. 2 (1A16)

c) Interface electronics subassembly No. 3 (1A17)

The major difference between these subassemblies is the deletion of one interface electronics subassembly; interface electronics subassembly No. 1 (1A8) and gross design changes.

4.3.3.1.6 ASA Temperature Control. The accuracy of the inertial data outputs from the ASA is temperature dependent. The ASA temperature control system is capable of maintaining the internal ASA temperature at 120 degrees Fahrenheit ± 1 degree Fahrenheit for external temperatures between +30 degrees Fahrenheit and +130 degrees Fahrenheit. Two temperature controllers for fast warmup and fine temperature control are used. During the fast warmup mode, the ASA temperature is capable of rising from 0 degree Fahrenheit to approximately 116 degrees Fahrenheit in 40 minutes with an attendant power consumption of between 190 and 220 watts. The fine temperature controller takes over ASA temperature control after the 116-degree Fahrenheit temperature is reached. This control is capable of providing an ASA temperature rise of 4 degrees Fahrenheit and maintaining this operating temperature within ±2 degrees Fahrenheit stabilization. Fine temperature power consumption is between 50 and 56 watts.

4.3.3.1.6.1 Fast Warmup Controller Subassembly 1A5. The fast warmup temperature controller (Figure 4.3.3.1-13) provides the heater power necessary to bring the ASA up to operating temperature within the required warmup time allotment. The fast warmup controller consists of a thermal sensor which detects the internal ASA temperature and provides a proportional dc analog to the regulation and heater driver electronics that maintain the required heater power constant. Also, controlled turn-on circuitry prevents switching transients from developing, which would otherwise produce electromagnetic interference. The constant heater power requirement permits a uniform temperature rise within the ASA of approximately 3 to 4 degrees Fahrenheit per minute in an earth environment. The fast warmup controller subassembly comprises two flat-pack resistive networks, four transistor matched pairs, and discrete components consisting of two zener and three ordinary diodes. Also part of the fast warmup circuitry, but mounted on the housing subassembly, are three heater control transistors and three heaters.

4.3.3.1.6.2 Fine Temperature Controller Subassembly 1A4. The fine temperature controller (Figure 4.3.3.1-14) provides the stable long-term temperature control required to maintain the ASA operating temperature at a nominal 120 degrees Fahrenheit ±1 degree Fahrenheit nominal, with ±2 degrees Fahrenheit stability, during operation. For maximum operating efficiency, the fine temperature controller is supplied directly with 28-volt power from the LM primary power source. The fine temperature controller has two thermal sensors which detect ASA internal temperatures and provide a proportional dc analog to the detecting electronics.

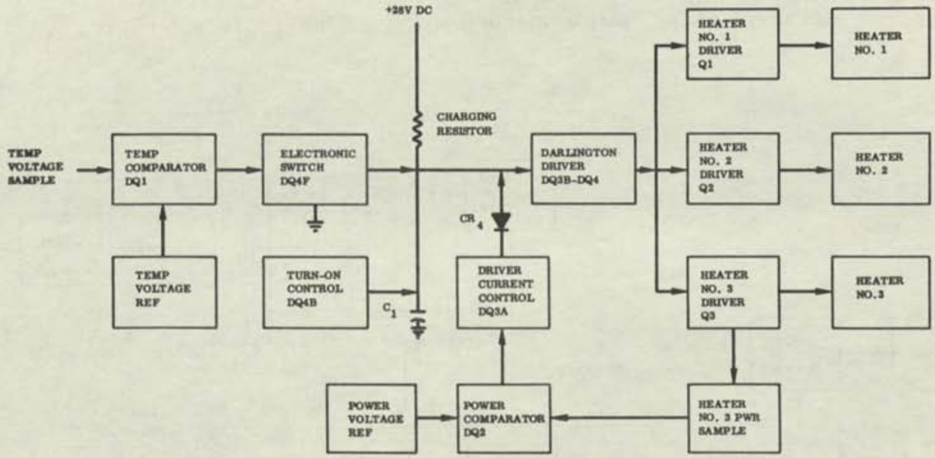


Figure 4.3.3.1-13. Fast Warmup 1A5, Block Diagram

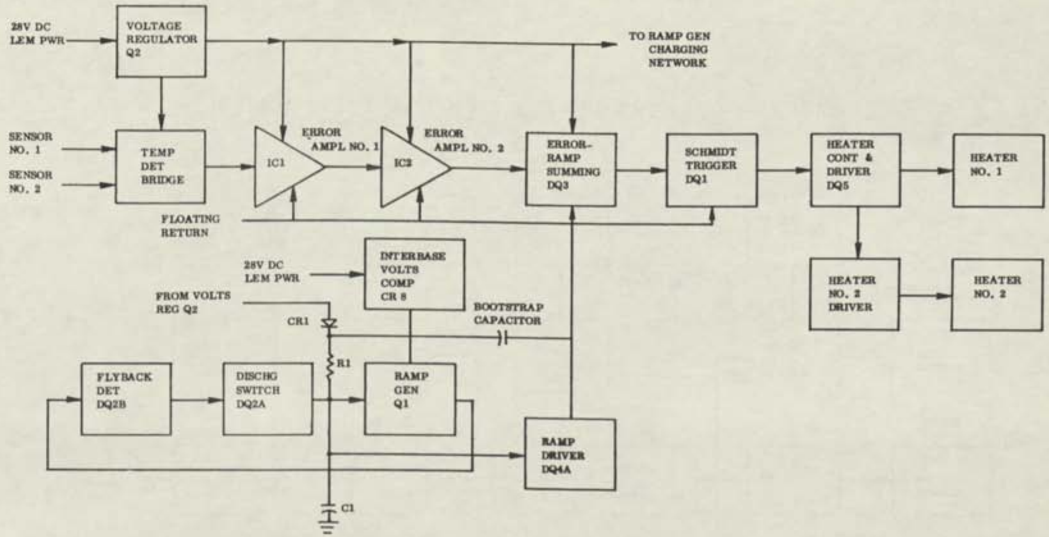


Figure 4.3.3.1-14. Fine Temperature 1A4, Block Diagram

Unlike the fast warmup controller, the fine temperature controller provides switched, rather than continuous, power to the fine temperature heaters. The fine temperature controller electronics changes the temperature-sensed analog into a digital form. This converted analog signal is very much like the variable duty cycle of the switching regulator in power supply 1A7 and controls the on and off time of the fine temperature controller heaters at a variable rate.

4.3.3.1.7 Power Requirements. All power, whether ac or dc, required by assemblies of the ASA is provided by power supply 1A7 (see Figure 4.3.3.1-15). This power supply receives LM 28-volt primary power and synchronizing signals from the frequency countdown and provides the following dc and ac voltages:

a) +32 volts dc

b) +12 volts dc

c) +4 volts dc

d) -12 volts dc

e) -6 volts dc

f) -2 volts dc

g) 6.3 volts, 8 kc, single-phase reference

h) 6.3 volts, 8 kc, single-phase quadrature

i) 6.3 volts, 8 kc, single-phase antireference

j) 6.3 volts, 8 kc, single-phase antiquadrature

k) 22 volts, 8 kc, single-phase reference

l) 1.5 volts, 8 kc, single-phase reference

m) 29 volts, 8 kc, three-phase, gyro spin motor voltage

4.3.3.1.7.1 Direct Current Voltages. The direct-current voltages listed in (a) through (f) are used by subassemblies of the ASA. Although other direct voltages are produced by the power supply, they are used internally

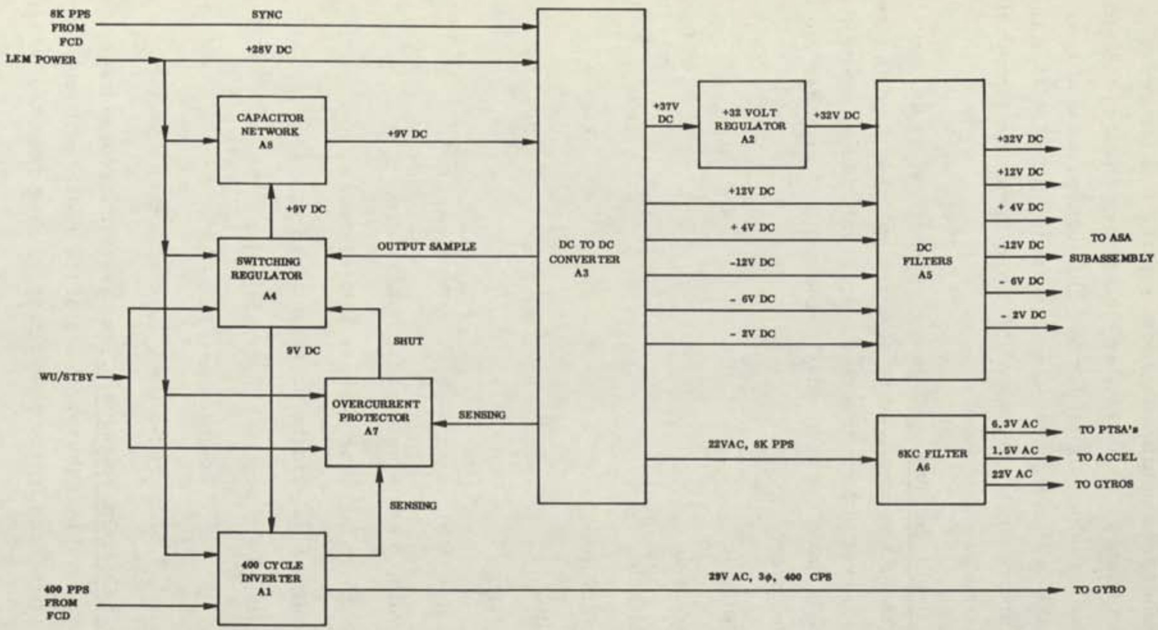


Figure 4.3.3.1-15. Power Supply 1A7, Block Diagram

and are not available to other subassemblies. The subassemblies that use dc power are as follows:

a) The +32-volt power is used by the current regulators in each PTSA.

b) The +12-volt power is used to bias the amplifiers in the accelerometers and by each PTSA to bias integrated circuit amplifiers.

c) The +4-volt power is used in the frequency countdown subassembly and in the PTSA's to bias logic performing integrated circuit modules.

d) The 12-volt power is used in the PTSA's and by the amplifiers in the accelerometers.

e) The 6-volt power is used as bias voltage in the PTSA's.

f) The -2-volt power is used as bias in the frequency countdown subassembly and in the PTSA's to bias logic performing integrated circuit modules.

4.3.3.1.7.2 Alternating-Current Voltages. The alternating-current voltages listed in (g) through (m) of Paragraph 4.3.3.1.7 are used by the accelerometers, gyros, and the PTSA's. The assemblies of the ASA that use ac power are as follows:

a) The 6.3-volt, 8-kc power is used by the demodulators in the PTSA's.

b) The 22-volt, 8-kc power is used to excite the pickoffs in the accelerometers.

c) The 1.5-volt, 8-kc power is used to excite the pickoffs in the accelerometers.

d) The 29-volt, 400-cycle, three-phase power is used to drive the gyro spin motors.

4.3.3.1.7.3 Power Supply 1A7. Power supply 1A7 comprises eight modules that are separated into functional and mechanical parts consisting of the following:

a) 400-cycle inverter A1

b) +32-volt regulator A2

c) dc to dc converter A3

d) Switching regulator A4

e) dc filters A5

f) 8-kc filter A6

g) Overcurrent protector A7

h) Capacitor network A8

Each of the modules listed in (a) through (h) contains one or more discrete components that provide each of the module functions. Capacitor network A8 contains a single capacitor used for filtering the input signal from switching regulator A4; this capacitor is considered a functional part of switching regulator A4.

4.3.3.1.8 Mechanical Characteristics. The ASA is a removable assembly, see Figure 4.3.3.1-3. The housing of the assembly is constructed of cadmium-plated material. The total assembly weight does not exceed 20.7 pounds. The dimensions of the package excluding mounting flanges are 9.00 x 13.50 x 5.10 inches.

4.3.3.2 Abort Electronics Assembly (AEA). A brief description of AEA Component Identification, Function, Mechanization, Electrical Characteristics, Detailed Performance, and Mechanical Characteristics is presented in the following paragraphs.

4.3.3.2.1 Component Identification. The AEA is a compact, high-speed general purpose computer which uses special purpose input/output electronics.

4.3.3.2.2 Function. The function of the AEA is to provide semiautomatic, preprogrammed flight control data to the Control Electronics Assembly in the event of a PGNCS failure. Inertial data for use in abort computations are supplied to the AEA from the Abort Sensor Assembly. The AEA provides the monitoring feature required to facilitate ground support during flight. Separate test connectors are provided to allow determination of AEA inputs and failures without access to the flight connector. The AEA also provides a stored diagnostic routine for self-checks initiated upon command from the DEDA or ACE and on iterative self-check routine during the AEA program.

4.3.3.2.3 Mechanization. The AEA employs a fractional two's complement, parallel arithmetic section, and parallel data transfer. Instruction words are 18 bits in lenght, consisting of a 5-bit order code, an index bit, and a single 12-bit operand address. Data words are 18 bits in length including sign. The cycle time of the memory is 5 microseconds. A block diagram of the AEA is shown in Figure 4.3.3.2-1. As indicated in the block diagram, the AEA is functionally divided into four major sections; the Memory Section, the Central Computer, the Input/Output Section, and the Power Supplies.

4.3.3.2.3.1 Memory. The computer memory is a coincident-current, parallel, random-access, ferrite-core memory with a capacity of 4096 words of 18 bits each. Each core in the coincident memory is subjected to X- and Y-coordinate selection and to an inhibit action as will be explained. The cores are arranged in square planes, one plane for each bit in the word. Each plane consists of 64 rows and 64 columns (4096 cores). The number of words in the memory is equal to the number of cores in a bit plane. An address is specified by an X-coordinate and a Y-coordinate and refers to the cores at that location in all 18-bit planes.

The memory core stack is divided into two sections: (1) soft-wire memory and (2) hard-wire memory. The function of the soft-wire memory is to store replaceable instructions and data. Temporary results may be stored there, and may be updated as necessary. Checkout routines may be loaded during checkout and may then be discarded. The function of the hard-wire memory is to store instructions and constants that are not to be modified during the operation of the system. Using a hard-wire memory prevents inadvertent loss of information. The address of the soft-wire storage locations are 0 through 3777_8, and the address of the hard-wire storage locations are 4000_8 through 7777_8.

In the soft-wire memory each core is threaded by a sense winding, an inhibit winding, and X-selection winding, and a Y-selection winding. All windings except the inhibit are single turn. There are 18 sense and 18 inhibit windings. The sense and inhibit windings thread all cores in a single bit plane. Associated with each bit plane is a sense amplifier, an

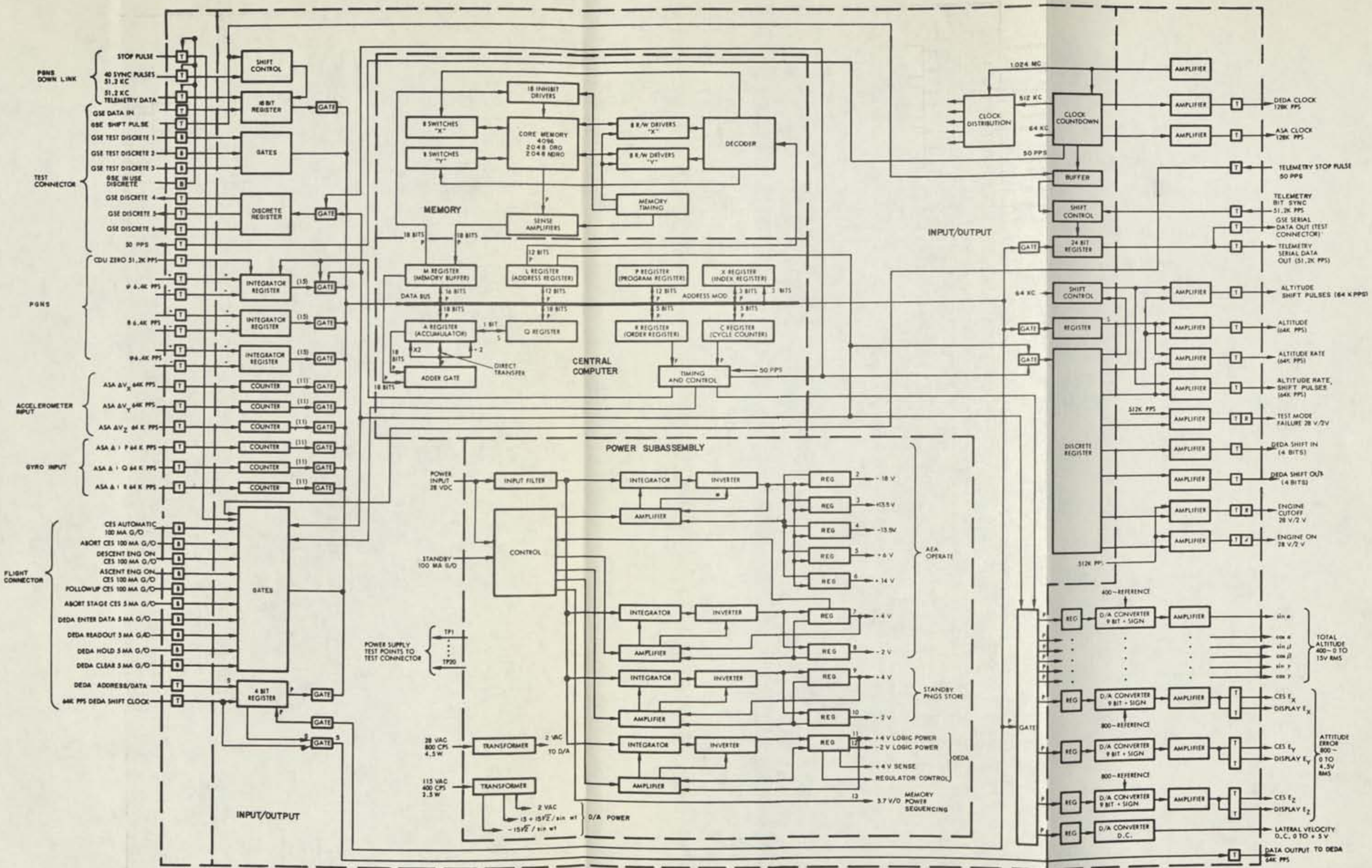


Figure 4.3.3.2-1. AEA Block Diagram

inhibit driver, and a digit storage flip-flop; the digit storage flip-flop receives signals from the sense amplifier and controls the inhibit drivers.

In the hard-wire memory each core is threaded by a sense winding, a Y-selection winding, and an X-selection winding (X-selection winding is present if core is to contain a ZERO and omitted if core is to contain a ONE). The sense winding and Y-selection windings are common to the soft-wire and hard-wire memory. The sense amplifier and digit storage flip-flops are also common to the soft-wire and hard-wire memories.

Reading the binary state of a core is accomplished by applying equal current pulses of $+I_s/2$ to the X- and Y-drive lines that cross the desired bit. This applies an effective full-select current $+I_s$ to the selected core and drives it to the ONE state. All other cores threaded by the pulses X- and Y-drive lines will receive half-select currents $+I_s/2$. While reading, if the selected core is changing from the ZERO state to the ONE state, a signal is present across the sense winding. This signal is interpreted by the sense amplifier as a ZERO. If the core is already in the ONE state at the start of read, no signal appears across the sense winding. The absence of a signal is interpreted by the sense amplifier as a ONE. In the hard-wire memory, a selected core that is threaded by both an X-selection and a Y-selection winding switch from the ZERO state to the ONE state and is, therefore, interpreted as a ZERO. A selected core which is not threaded by an X-selection wire receives only a current pulse of $+I_s/2$ and is, thereby, always interpreted as ONE.

Reading a bit always leaves the core in the ONE state. In order to achieve non-destructive readout, the word read must be written back into the memory. When writing, equal pulses of $-I_s/2$ are applied to the X- and Y-selection lines. These selection currents are of opposite polarity of those used in reading, thus requiring that the X-Y drivers be bi-directional. Applying the currents $-I_s/2$ to an X and $-I_s/2$ to a Y-selection line causes the selected bits to switch to the ZERO state. In the soft-wire memory, if a ONE is to be written, a current of $+I_s/2$ is applied to the inhibit line at the same time as the X- and Y-lines are pulsed. The control of the inhibit is done by the state of the digit storage flip-flop which contains the bit to be written. The inhibit current pulse is made to overlap the X- and Y-selection currents in order to prevent the possibility of partial switching of the selected core to the ZERO state. The system logic is such that only ZEROS can be written into the memory. ONES are written by inhibiting the writing of ZEROS. Therefore, it is necessary to set the cores of the word selected to the ONE state before information can be written into the cores. This core setting is accomplished by performing the read operation (sets all cores to the ONE state) before performing the write operation. In the hard-wire memory no inhibit current is required since all cores which were ZEROS can be restored to ZERO by applying X- and Y-write currents of $-I_s/2$ each. Those cores which were ONES remain in the ONE state, since they were subjected to a current drive of only $-I_s/2$ total.

The memory operates in three modes: (1) Read-restore, (2) read-clear, and (3) clear-write. At the beginning of a memory cycle, the central computer supplies the memory with an initiate memory cycle signal, a mode signal, and an address. Address information is contained in a 12-bit address register located in the central computer. Decoding is performed within the memory. The address register always contains the address at the beginning of a memory cycle. For a read-restore mode, the contents of the location specified by the address register is read from the memory into the central computer M register. These data bits are available to the central computer after the second microsecond of the memory cycle. The data bits are restored during the second two microseconds of the cycle. This mode is used with the instructions ADD, CLA, MPR, SUB, DVP, and MPY and applies to both the soft-wire and hard-wire sections of the memory. The read-clear mode is similar to the read-restore mode except that the information is not restored to the memory. This information is used with the instructions ADZ, CLZ, MPZ, and SUZ and applies only to the soft-wire memory. This mode is used where it is not necessary to restore the information after reading. In the clear-write mode, the contents of the M register are written into the location specified by the address register. The word to be written is transferred into the M register during the second microsecond of the memory cycle.

4.3.3.2.3.2 Central Computer. The central computer consists of eight data and control registers (L, R, P, A, Q, M, X, and C), a data bus, adder gates, and two timing registers. The data and control registers are interconnected via the parallel data bus. Bit positions with respect to this data bus are shown in Figure 4.3.3.2-2. There are, however, some non-data bus parallel paths through the 18-bit parallel adder. Operations are executed by appropriately timed transfers of information between these registers, between memory and the memory register, and between the A register and a specified input or output register.

4.3.3.2.3.2.1 Data and Control Registers. The eight data and control registers are described below.

a) Memory Register - The memory (M) register is an 18-bit static flip-flop register that is loaded via its direct set inputs from the memory sense amplifiers. It is also loaded from the data bus via its clocked set inputs. The memory register must always be reset prior to entry of new data. The memory register is also used to write or rewrite data into the memory for read and restore operations involving the soft-wire and for clear and store operations from the central computer.

b) Accumulator Register - The accumulator (A) register is an 18-bit static flip-flop register that may be loaded from the data bus or which may be loaded from the adder outputs. In the process of loading the A register from the adder outputs, a one-bit left or one-bit right shift may be executed. The A register holds the result of most arithmetic operations and is also the register used to communicate with specified input/output registers.

c) Multiplier-Quotient Register - The multiplier-quotient (Q) register is an 18-bit two-way shift register, that, for some operations, may be logically attached to the low-order end of the A register to form a double length register. The Q register also may be loaded from the data bus. The Q register holds the least significant half of the double length dividend for divide operations. The Q register is also used to hold a dummy transfer order during execution of the transfer and set Q order (to be used subsequently as a program return link).

d) Operation Code Register - The operation code (R) register is a 5-bit flip-flop static register that is loaded from the most significant 5 bits of the data bus and is used to hold the 5-bit orders during the execution.

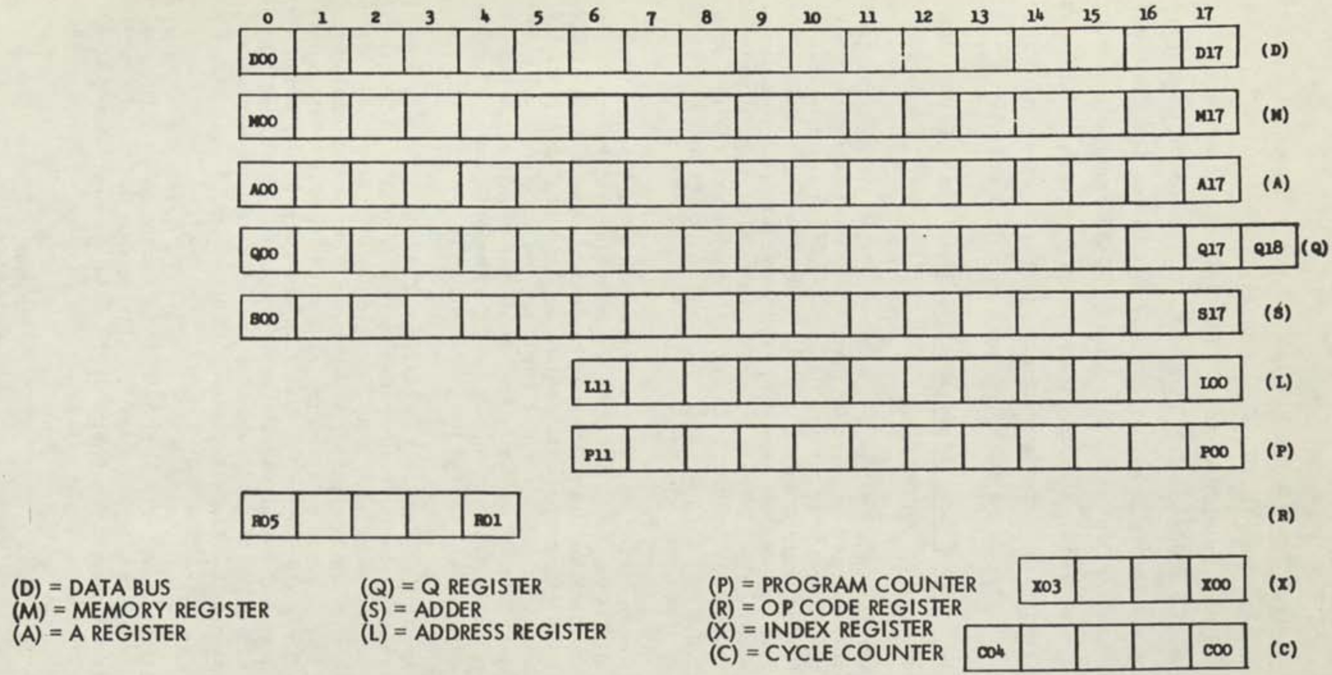


Figure 4.3.3.2-2. Bit Positions With Respect to Data Bus

e) Address Register - The address (L) register is a 12-bit static register that is loaded from the low-order 12 bits of the data bus and is used as the address register by the memory system.

f) Program Counter - The program (P) counter is a 12-bit ripple counter that is loaded via its direct set inputs from the low order 12 bits of the data bus and must be reset prior to loading. It is used to hold and to generate order addresses in sequence.

g) Index Register - The index (X) register is a 3-bit flip-flop ripple counter that is loaded via its direct set input from the low-order 3 bits of the M register via the data bus and must be reset prior to loading. The index register is used for address modification; its 3 bits are OR'd with the low-order 3 bits of an operand being gated into the address register from the data bus to accomplish address modification. The application of address modification to an operand is based on the type of order and whether M5 (the sixth most significant bit of the order) is ONE.

h) Cycle Counter - The cycle (C) counter is a 5-bit flip-flop counter that is loaded via its direct set inputs from the low-order 6 bits of the memory register via the data bus for shift instructions and is alternately loaded or preset to specific constants for certain long orders. It must be reset prior to loading.

4.3.3.2.3.2.2 Data Bus. The data bus is a collection of gates through which data (up to 18 bits in parallel) may transfer from one register to another. For example, to perform a data transfer of 18 bits from A to M, the timing and control logic generates two control signals. The first gates all 18 bits of the A register onto the data bus for 2 microseconds (some data bus transfers take 3 microseconds). At the same time, the second control signal gates the contents of the data bus into the clocked set input of the 18 M register flip-flops; at the end of the second microsecond, the M register is clocked, and the information transfer is complete.

For several transfers from the data bus, both true and complement information are gated into the receiving register. However, for transfers into P, only selected true signals are gated; therefore, this register must always be cleared to ZERO's prior to receiving new information. For transfers into X, C, and M, only selected complement signals are gated;

these registers are cleared to ONE's prior to receiving new information. The transfer of information into most registers is via the clocked set/ reset inputs at the appropriate clock time; however, the three registers connected as ripple counters (C, P, and X) are loaded from the data bus via the flip-flop direct set/reset inputs. The interconnection of registers via the data bus is shown in Figure 4.3.3.2-3.

4.3.3.2.3.2.3 Instruction Word. The computer uses a single-address instruction word consisting of 18 bits, Figure 4.3.3.2-4, 5 bits represent the operation to be executed, 1 bit indicates whether or not address modification is to be used, and 12 bits normally indicate the memory location of the operand ($2^{12} = 4096$). The instruction word is accessed from a memory location and placed in the memory register M. Then the operation code part of the instruction word is transferred from the memory register to the operation code register R, and the address part of the instruction word is transferred from the memory register to the address register L.

4.3.3.2.3.2.4 Operation Decoding. The outputs of the five operation code register flip-flops are decoded and used as operation control lines where needed in the control circuitry. There are 27 different instructions, Table 4.3.3.2-1. The control line associated with each instruction is identified by the three letter mnemonic assigned to that instruction.

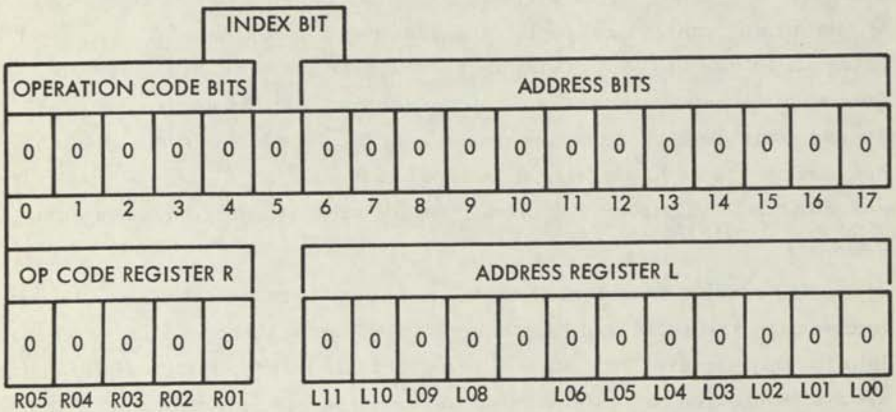


Figure 4.3.3.2-4. Instruction Format for Instruction Registers

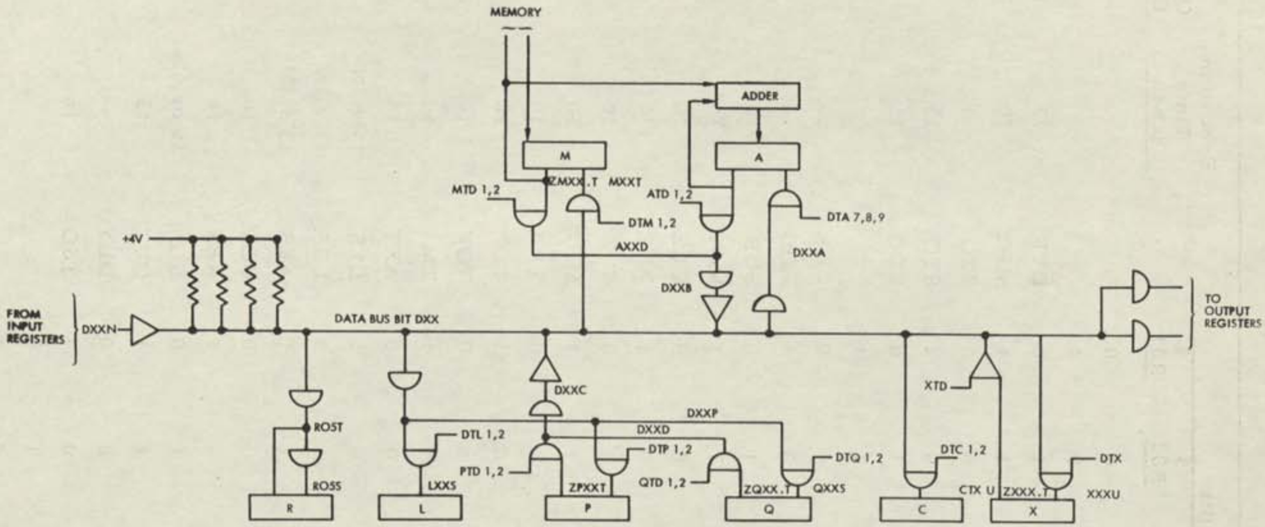


Figure 4.3.3.2-3. Data Bus, Register Interconnection

Table 4.3.3.2-1. Operation Codes

Instruction Bit 0 R05	1 R04	2 R03	3 R02	4 R01		Execution Time μ sec	Octal Code
0	0	0	0	0			
0	0	0	0	1			
0	0	0	1	0	DVP	73	04
0	0	0	1	1	MPY	70	06
0	0	1	0	0	STO	13	10
0	0	1	0	1	STQ	13	12
0	0	1	1	0	LDQ	13	14
0	0	1	1	1			
0	1	0	0	0	CLA	10	20
0	1	0	0	1	ADD	10	22
0	1	0	1	0	SUB	10	24
0	1	0	1	1	MPR	70	26
0	1	1	0	0	CLZ	10	30
0	1	1	0	1	ADZ	10	32
0	1	1	1	0	SUZ	10	34
0	1	1	1	1	MPZ	70	36
1	0	0	0	0	TRA	10	40
1	0	0	0	1	TIX	10	42
1	0	0	1	0	TOV	10	44
1	0	0	1	1	TMI	10	46
1	0	1	0	0	AXT	13	50
1	0	1	0	1	LLS	13 + 3N	52
1	0	1	1	0	LRS	13 + 3N	54
1	0	1	1	1	ALS	13 + 3N	56
1	1	0	0	0	COM	16	60
1	1	0	0	1	ABS	16	62
1	1	0	1	0	INP	16 or 67	64
1	1	0	1	1	OUT	13	66
1	1	1	0	0	DLY	-	70
1	1	1	0	1	TSQ	16	72
1	1	1	1	0			
1	1	1	1	1			

4.3.3.2.3.2.5 Adder. An 18-bit parallel adder adds either true (add) or complement (subtract) information from the M register into the contents of the A register. 3 microseconds are required for the carry to propagate and the correct sum to be available at the adder outputs after the M register information is gated into the adder. At the end of a 3-microsecond period, e.g., T4C, the sum is clocked into the A register and replaces its previous contents augend. A 1-bit shift (either left or right) may be implemented simultaneously by gating the output of the adder into adjacent bits of the A register (i.e., sum bit 13 may be gated into A bit 12, 13, or 14 to effect a shift-left, no-shift, or shift-right, respectively). Note that the data bus is not used to interconnect the adder with either A or M.

4.3.3.2.3.2.6 Timing. There are two timing generation registers (1) an eight flip-flop shift register to generate T1 through T8 signals, and (2) a three flip-flop shift register to generate TA, TB, and TC signals, Figure 4.3.3.2-5. These signals are logically combined (e.g., T3A or T5C) to produce all the timing required for all operations of the central computer. T1 and T3 are each 2-microsecond sequences subdivided as follows: T1A, T1B, T3A, T3B. T2, T4, T5, T7, and T8 are 3-microsecond sequences subdivided as follows: T4A, T4B, T4C, T5A, T5B, etc.

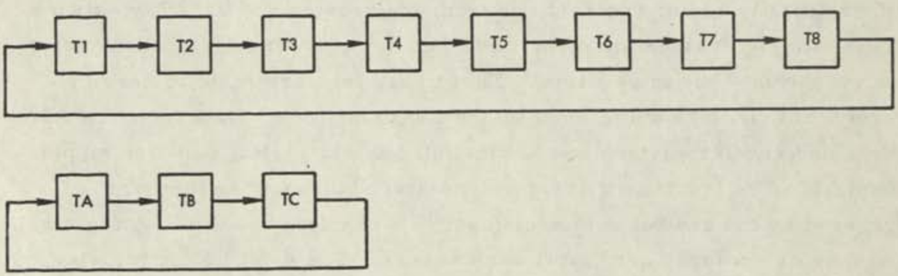


Figure 4.3.3.2-5. Timing Shift Registers

All short orders (10 microseconds) use timing sequences T1 through T4; the longer orders use additional sequences as necessary. Long orders, such as, divide, multiply, or shift use one of the basic timing sequences repetitively under the control of the cycle counter that is initially preset for a given order and then counted down to ONE.

The timing flow portion of Figure 4.3.3.2-6 shows the standard timing chain from T1A through T8C. At times T4C, T5C, T6C, T7C, or T8C, shifting may be terminated. The timing sequences T5 or T6 may be repetitively cycled. The end-of-instruction timing portion of Figure 4.3.3.2-6 illustrates instruction timing chain termination and initiation. The end flip-flop is set at the beginning of the last timing sequence T () A of an instruction timing chain. The END signal then prevents shifting past T () C in the current instruction timing chain, and causes the next instruction timing chain to be initiated.

4.3.3.2.3.3 Input/Output. The input/output equipment consists of numerous special registers, counters, etc., which operate independently of the central computer except when specifically accessed by a particular input or output order. There are basically four types of registers in the input/output section. The PGNCS angles are accumulated in three-integrator registers each of which consists of a 15 flip-flop shift register, a half-adder/subtractor, and the necessary control logic. The registers shift 15-bit positions at 512 kc upon receipt of each new input pulse so as to either serially add or subtract one from the previous count. The shifting or counting is inhibited momentarily when an input transfer to the A register via the data bus is required. These registers are reset to zero by receipt of a 51.2-kc pulse input on the CDU zero line. Data from the ASA gyros and accelerometers are accumulated in six 11-bit, flip-flop ripple counters. The inputs to these counters are inhibited when they are accessed by the central computer (again via the data bus to the A register), and they are reset to zero after each access. The 4-bit DEDA register, the 18-bit input telemetry register, the 24-bit output telemetry register, and the 15-bit register time shared for altitude and altitude rate are all implemented as shift registers. The telemetry registers are asynchronous (shifted by an external shift clock); the other one has associated shift

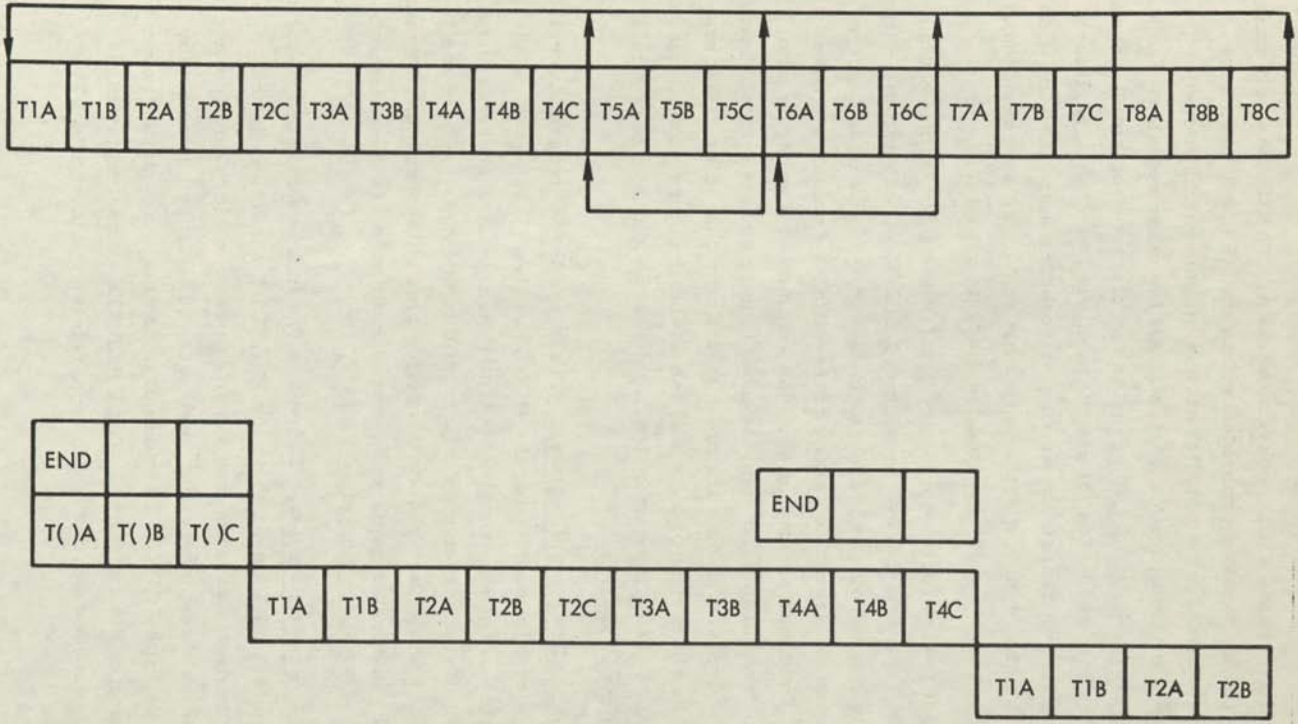


Figure 4.3.3.2-6. Timing Flow and End of Instruction Diagram

controls and generates shift clocks in the AEA. The DEDA 4-bit shift register has shift controls generated within the AEA. In each case, shifting is inhibited during parallel access by the central computer (via the data bus to or from the A register). All the other registers are static flip-flop registers (e.g., D/A converter and discrete register). The D/A registers are loaded in parallel via the data bus from the A register. The discrete register is changed by setting or resetting a specific flip-flop within the register with a specific output order (unique address); the A register is not involved in changing the discrete register.

The addresses (octal number system) of the input/output equipment are given in Tables 4.3.3.2-2 through 4.3.3.2-5. Combinations of certain individual input/output addresses can be made, within an input or output instruction, to facilitate setting or resetting a group of registers or discretes. Only addresses that have the same most significant octal character can be combined. To combine one or more addresses, add numerically each address excluding the most significant character. For example, combining the addresses of all six of the 11-bit counters would result in an address of 6177. An input instruction with the address 6177 would reset all of the counters.

4.3.3.2.3.3.1 <u>Integrator Register.</u> There are three integrator registers for the purpose of accumulating the PGNCS angles θ, ϕ, and ψ. The PGNCS angles are represented by pulse train signals. These asynchronous signals have a frequency range from 0 to 6.4 kpps. There are two input lines per integrator register, one for plus pulses and one for minus pulses. The integrator register operates during the standby mode when the central computer logic is off.

4.3.3.2.3.4 <u>Power Supplies.</u> There are two independent power supplies in the computer. One operates in the STANDBY as well as in the OPERATE mode of operation and supplies power to clock countdown circuits and to the three integrating registers. The other power supply operates only in the OPERATE mode of operation and supplies power to the remainder of the computer. The OPERATE power supply turns on its outputs in sequence in order not to compromise the memory. The

Table 4.3.3.2-2. AEA Input Registers

Name	Type of Register	Address
θ (PGNCS)	Integrating Reg.	2001
ϕ (PGNCS)	Integrating Reg.	2002
ψ (PGNCS)	Integrating Reg.	2004
ΔV_x	11 Bit Counter	6020
ΔV_y	11 Bit Counter	6040
ΔV_z	11 Bit Counter	6100
$\Delta \int g$	11 Bit Counter	6002
$\Delta \int p$	11 Bit Counter	6010
$\Delta \int r$	11 Bit Counter	6004
Downlink Telemetry	18 Bit Shift Register	6200
Disc. Inp. Word 1	8 Bits	2020
Disc. Inp. Word 2	7 Bits	2040
DEDA	4 Bit Shift Register	2200

Table 4.3.3.2-3. AEA Output Registers

Name	Type of Registers	Address
sin θ	9 Bits plus Sign	2001
cos θ	9 Bits plus Sign	2002
sin ϕ	9 Bits plus Sign	2004
cos ϕ	9 Bits plus Sign	2010
sin ψ	9 Bits plus Sign	2020
cos ψ	9 Bits plus Sign	2040
E_x	9 Bits plus Sign	6001
E_y	9 Bits plus Sign	6002
E_z	9 Bits plus Sign	6004
Lateral Velocity	8 Bits plus Sign	6020
Altitude, Altitude Rate	14 Bits plus Sign	6010
Output Telemetry*	24 Bit Shift Register	
Word 1	Bits 0-17	6200
Word 2	Bits 6-23	6100
DEDA	4 Bit Shift Register	2200

*Only Word 1 is used for GSE data output.

Table 4.3.3.2-4. AEA Discrete Outputs

Name	Set	Reset
Ripple Carry Inhibit	2410	3010
Altitude	2420	3040
Altitude Rate	2440	3040
DEDA Shift In	2500	--
DEDA Shift Out	2600	--
GSE Discrete 4	6401	7001
GSE Discrete 5	6402	7002
GSE Discrete 6	6404	7004
Test Mode Failure	6410	7010
Engine Off	6420	7020
Engine On	6440	7040

Table 4.3.3.2-5. AEA Discrete Inputs

Bit Position	Discrete Word 1	Discrete Word 2
1	Downlink Telemetry Stop	GSE Discrete 1
2	Output Telemetry Stop	GSE Discrete 2
3	Follow-up	GSE Discrete 3
4	Automatic	DEDA Clear
5	Descent Engine On	DEDA Hold
6	Ascent Engine On	DEDA Enter
7	Abort	DEDA Readout
8	Abort State	

OPERATE power supply also monitors the input voltage and the output voltages and provides a control signal. PS13 is true if the computer is in the OPERATE mode and the input voltage is at least 20 vdc.

4.3.3.2.3.4.1 Startup. To place the computer in operation, the operator switches from STANDBY to OPERATE (see Figures 4.3.3.2-7 and 4.3.3.2-8, Logic Diagram, Board R). The OPERATE power supply then applies power to the computer in a predetermined sequence. When all OPERATE voltages are at their proper levels, OPERATE power supply signal PS13 remains ZERO for a delay of at least 10 microseconds. During this 10-microsecond period, certain initial conditions are established in the computer by ZBEGIT, where

$$\text{Set BEG1} = \overline{\text{PS13}} * \text{CLK.}$$

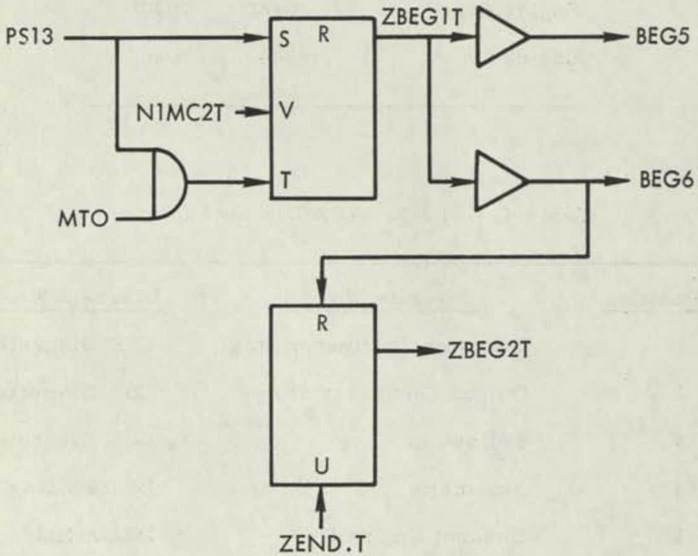


Figure 4.3.3.2-7 Startup Control

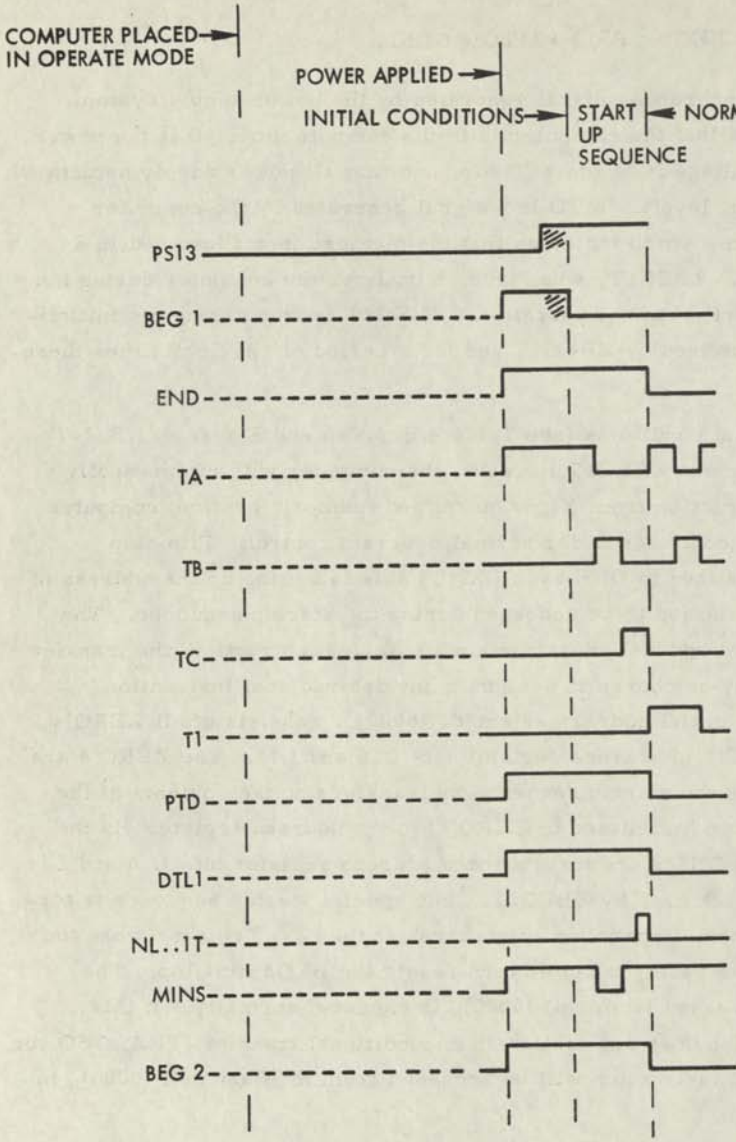


Figure 4.3.3.2-8. Startup Sequence

When PS13 goes to ONE, ZBEG1T is reset by:

Reset BEG1 = PS13 * MTO * CLK.

PS13 is an asynchronous signal generated by the power supply system, which indicates that the computer is in the compute mode, that the power supply input voltage is at least 20 vdc, and that all power supply outputs are at operating levels. MTO is a signal generated by the computer memory system, which indicates that the memory is not involved in a memory cycle. ZBEG1T, when true, initializes the computer during the start (or restart) phase of operation. ZBEG2T is true during the initialization period defined by ZBEG1T and for a period of six clock times thereafter.

The initial conditions (see Table 4.3.3.2-6 and Figure 4.3.3.2-7) are such that, once ZBEG1T is reset, the computer will automatically access an instruction from a predetermined memory location; computer operation then continues under normal program control. Flip-flop ZBEG2T (initialized to ONE by ZBEG1T) aids in setting up the address of the initial instruction to be accessed during the startup sequence. The arbitrary initial address must contain a transfer instruction; the transfer instruction amy be chosen to designate any desired next instruction address. The initial address selected, $(6000)_8$, consists of all ZERO's except for ONE's in address register bits L10 and L11. The ZERO's are inserted during the startup sequence by transferring the contents of the program counter (initialized to ZERO's) to the address register via the data bus. The ONE's are inserted into address register bits L10 and L11 at the same clock time by ZBEG2T. The special startup sequence is terminated and normal operation commences at the next T1B time when the END flip-flop is reset, which in turn resets the BEG2 flip-flop. The instruction accessed from cell $(6000)_8$ is executed normally. If this memory location does not contain an unconditional transfer (TRA, TSQ, or DLY), the next instruction will be accessed from memory cell $(0001)_8$ in the soft-wire.

Table 4.3.3.2-6. Forced Initial Conditions

	ONE	ZERO
END	X	
TA	X	
TB		X
TC		X
T1 - T8		X
R1 - R5		X
P0 - P11		X
A0V		X
SUF		X
ADF		X
BEG2	X	

4.3.3.2.3.4.2 Shutdown. The computer is designed to shut down operation systematically when the power supply input voltage drops below 20 vdc, and to restart automatically when the proper input voltage is restored. The prime consideration in shutting down operation is to do so without losing memory information. If the computer shuts down after reading from memory but before restoring to memory, information is lost. This is prevented by completing the current memory cycle while preventing initiation of additional memory cycles during shutdown. Accordingly, the power supply is designed to store power during normal operation; the stored power is adequate to operate the computer for the length of time necessary to complete one memory cycle. Shutdown is initiated when the power supply input voltage drops below 20 vdc by:

$$\text{Set BEG1} = \overline{PS13} * CLK,$$

where PS13 is a logic signal generated by the power supply. Signal ZBEG1T returns the central computer logic to its initial condition as shown in Figure 4.3.3.2-7. The computer is then ready to go through its startup sequence as soon as power is restored, after the current memory cycle is completed. The necessary conditions to end the initialization period and to begin the startup sequence are:

Reset BEG1 = PS13 * MTO * CLK,

where MTO indicates that the computer has completed any memory cycle previously initiated, and PS13 indicates that the power supply input voltage has returned to its operating level and that power supply outputs are still at operating levels or have returned to operating levels.

4.3.3.2.3.4.3 Power Supply Outputs. The following voltages are furnished by the AEA power supply subassembly:

Voltage (nominal)

a) -2 vdc

b) +4 vdc

4.3.3.2.3.4.4 Normal Load Conditions. The power supply must satisfy all load requirements of the AEA in addition to the following external loads:

To DEDA

Normal	Swing
+4 v, 4 w	400 mw to 8 w
-2 v, 400 mv	5 mw to 1000 mw

Total external output power is limited to 9 watts maximum. See Table 4.3.3.2-7 for a further listing of power output.

4.3.3.2.3.4.5 Maximum dc Voltage Power Consumption. The maximum steady state power consumed by the AEA from the 28-vdc power supply under all normal conditions will not exceed the following limits:

a) Align and operate: 98 w
This limit includes an external load of 8 watts.

b) Standby: 12 w
No external power will be supplied by the AEA during Standby.

Table 4.3.3.2-7. Power Output to DEDA

Function	Nominal Voltage (v)	Max Avg. Power (Watts) Requirement	No. of Lines	Notes
+4 v Logic Power	+4 v to +5 v	8.2	2	Max line resistance AEA to DEDA = 0.33 ohms
+4 v Return			2	
-2 v Logic Power	-2 v to -3 v	0.4	1	Max line resistance = 1 ohm
-2 v Return			1	
+4 v Sense	+4 v to +5 v	0.1	1	
Return			1	
10 v (nominal) Control	10 v	0.1	1	
Return			1	

4.3.3.2.4 Electrical Characteristics. The electrical characteristics of the AEA are presented under input signals, and output signals as described in the following paragraphs.

4.3.3.2.4.1 Input Signals. The following paragraphs describe the AEA inputs. See Table 4.3.3.2-8 for further definition of the inputs.

4.3.3.2.4.1.1 DEDA Inputs. The DEDA inputs will:

a) Provide mode selection for the AEA, except standby mode

b) Provide address and data entry into the AEA

c) Command readout of various AEA parameters for display by supplying a readout signal

4.3.3.2.4.1.2 Instrumentation Subsystem. The IS will furnish the AEA:

a) Basic AEA clock (1.024 mc)

b) Telemetry control signals for extracting information from the AEA. (AGS downlink "stop" pulse. AGS downlink bit sync pulses)

c) Control signals for inserting PGNCS downlink data into the AEA (PGNCS downlink "stop" pulse, PGNCS downlink bit sync pulses)

4.3.3.2.4.1.3 ASA Inputs. The ASA will furnish the AEA with:

a) Gyro inputs ($\Delta\int p$, $\Delta\int q$, $\Delta\int r$)

b) Accelerometer inputs (ΔVx, ΔVy, ΔVz)

4.3.3.2.4.1.4 CES Inputs. The CES will furnish the AEA with the following input signals:

a) Abort

b) Abort stage

c) Ascent engine on

d) Descent engine on

e) Automatic

f) Follow-up

g) 800 cps reference

Table 4.3.3.2-8. Input Signals

Signal	Basic Characteristics: Load	Basic Characteristics: Source	Signal Characteristics
Descent Engine-on Ascent Engine-on	100 ma (max): ground applied 0 ma: open circuit	Relay closure	Ground or open circuit Ground: "1" state Open "0" state
1024 kc AEA Clock	100 ohms transformer coupled	100 ohms ± 10%	Freq: 1.024 mc (accurate to 15 parts/million during 135-hr mission) (Implemented when AEA gets internal clock) Pulse train: 50 ± 10% duty cycle' Amplitude: 3v ± 0.5v Short term stability: 0.1% of bit period
Accelerometer Input Pulses ΔV_x, ΔV_y, ΔV_z	As required, pulse transformer coupled	As required, pulse transformer coupled	Pulse train: 64 pulses/ms - max 0 pulses/ms - min Amplitude: 6 volts ± 20% Pulse width: 1 ± 0.2 μsec, nominal
DEDA Enter DEDA Readout DATA DEDA Clear DEDA Hold	As required	Switch closure to AEA signal ground	Switch closed: "1" state, 5 ma - max Switch closed: "0" state, 0 ma
DEDA Address/Data	Pulse transformer coupled, 900 ohm, min	Pulse transformer coupled, 50 ohms ± 20%	Freq: 64 kpps, 36 bit bcd word Amplitude: 4v ± 1v Pulse width: 2μsec ± 1.4μsec
DEDA Shift Pulses	Pulse transformer coupled, 900 ohm, min	Pulse transformer coupled, 50 ohms ± 20%	Freq: 64 kpps Amplitude: 4.5v ± 1v Pulse width: 2μsec ± 1.4μsec
PGNCS Downlink "Stop" Pulse	200 ohm ± 20%, pulse transformer coupled	100 ohms, max	Freq: 50pps Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1 μsec

Table 4.3.3.2-8. Input Signals (Continued)

Signal	Basic Characteristics: Load	Source	Signal Characteristics
PGNCS Downlink Bit sync Pulses	200 ohms ± 20%, pulse transformer coupled	100 ohms max	Freq: 51.2 kc Pulse train: 40 pulses per burst Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1
PGNCS Downlink "Data"	500 ohms ± 10%, pulse transformer coupled	100 ohms max	Freq: 51.2 kc Pulse train: "1" = presence of pulse, "0" = absence of pulse, with shift pulse Amplitude: 7v ± 3.5 v Pulse width: 4 μsec ± 1
800 cps Reference Signal	2.0 watts	less than 10 ohms	Freq: 800 cps ± 1% Amplitude: 28v rms ± 0.5 v Harmonic distortion: 3% max
$\Delta\int p$, $\Delta\int q$, $\Delta\int r$	As required, pulse transformer coupled	As required, pulse transformer coupled	Freq: 3 pulses/msec, min Pulse train: 2^{-16} radians per pulse, max Amplitude: 6v ± 20% Pulse width: 1 μsec ± 0.2
PGNCS θ, Ψ and ϕ ($+\Delta\theta$, $-\Delta\theta$, $+\Delta\Psi$, $-\Delta\Psi$, $+\Delta\phi$, $-\Delta\phi$)	500 ohms ± 10%, pulse transformer coupled	100 ohms, max pulse transformer coupled	Freq: 6.4 kpps, max Pulse train: $\frac{360 \text{ deg/pulse}}{2^{15}}$
400 ~ Power for Total Attitude Signals	As required	1 ohm, max	Freq: 400 cps ± 10 cps, 3.5w Amplitude: 115v rms ± 2.5% Voltage waveform: Per MIL-STD-704 para. 5.1.3.5
CDU Zero	500 ohms ± 10% pulse transformer coupled	100 ohms, max, pulse transformer coupled	Duration: 300 msec, min Pulse train: 51.2 kpps, "on" Amplitude: 7v ± 3.5 v Pulse width: 3 μsec ± 1 μsec

Table 4.3.3.2-8. Input Signals (Continued)

Signal	Basic Characteristics: Load	Source	Signal Characteristics
Standby	100 ma, max: ground applied, on an open circuit	Relay or switch contact closure to AEA power ground	Ground: "1" state Open circuit: "0" state
Follow-up Automatic Abort Abort Stage	5 ma, max: ground applied, 0 ma; open circuit input	Relay contact closure or switch closure to AEA signal ground	Ground: "1" state Open circuit: "0" state
AGS Downlink "Stop" Pulse	200 ohms, ± 20% pulse transformer coupled	100 ohms, max	Freq: 50 times per second Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1 μsec
AGS Downlink Bit sync Pulses	200 ohms ± 20%, pulse transformer coupled	100 ohms, max	Freq: 50 times per second Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1 μsec

4.3.3.2.4.1.5 PGNCS Inputs. The PGNCS will furnish the following input signals:

a) $\pm\Delta\theta$, $\pm\Delta\psi$, and $\pm\Delta\phi$ (for IMU alignment)

b) CDU zero

The PGNCS θ, ψ, and ϕ attitude signals are accumulated in the AEA following the receipt of the CDU zero signal. These signals are accumulated during all modes of operation.

c) PGNCS downlink data - The AEA, upon command from the DEDA decodes PGNCS downlink data and extract position, velocity, and epoch information for updating navigation

The AEA accepts the first sixteen bits of each PGNCS 40-bit downlink word. The first bit will be one, if a data word follows, or zero if an ID word follows. Data words will consist of the sign bit followed by fourteen bits of data, the most significant bit of which follows the sign bit. ID words will consist of five zeros followed by a 10-bit binary data index, the most significant of which follows the last of the five zeros.

During AEA initialization by the PGNCS, no interrupts will occur to the state vector or ID words, and CDU zero will be received by the AEA. The number of times that PGNCS data are repeated to AGS is ten.

Position, velocity, and time will be double precision words, the most significant half transmitted prior to the least significant half. The bit weights are as follows:

Position: Most significant bit of the most significant half of word = 2^{22} ft

Velocity: Most significant bit of the most significant half of word = 2^{12} ft/sec

Time: Least significant bit of least significant half of word = 10 ms

Note: Data will be in a 2's complement binary form.

The format of 38 sequential words occurring at a basic rate of 50 words per second is shown in Table 4.3.3.2-9.

Table 4.3.3.2-8. Input Signals (Continued)

Signal	Basic Characteristics: Load	Source	Signal Characteristics
Standby	100 ma, max: ground applied, on an open circuit	Relay or switch contact closure to AEA power ground	Ground: "1" state Open circuit: "0" state
Follow-up Automatic Abort Abort Stage	5 ma, max: ground applied, 0 ma; open circuit input	Relay contact closure or switch closure to AEA signal ground	Ground: "1" state Open circuit: "0" state
AGS Downlink "Stop" Pulse	200 ohms, ± 20% pulse transformer coupled	100 ohms, max	Freq: 50 times per second Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1 μsec
AGS Downlink Bit sync Pulses	200 ohms ± 20%, pulse transformer coupled	100 ohms, max	Freq: 50 times per second Amplitude: 4.5v ± 1v Pulse width: 4 μsec ± 1 μsec

Table 4.3.3.2-9. LGC Format for AGS Initialization

(a)	Data Index (0)	ID WORD
(b)	LP_x (MS)	
(c)	LP_x (LS)	DATA WORDS
(d)	LP_y (MS)	
(e)	LP_y (LS)	
(f)	Data Index (1)	ID WORD
(g)	LP_z (MS)	
(h)	LP_z (LS)	DATA WORDS
(i)	LV_x (MS)	
(j)	LV_x (LS)	
(k)	Data Index (2)	ID WORD
(l)	LV_y (MS)	
(m)	LV_y (LS)	DATA WORDS
(n)	LV_z (MS)	
(o)	LV_z (LS)	
(p)	Data Index (3)	ID WORD
(q)	LTE (MS)	
(r)	LTE (LS)	DATA WORDS
(s)	CP_x (MS)	
(t)	CP_x (LS)	
(u)	Data Index (4)	ID WORD
(v)	CP_y (MS)	
(w)	CP_y (LS)	DATA WORDS
(x)	CP_z (MS)	
(y)	CP_z (LS)	
(z)	Data Index (5)	ID WORD
(aa)	CV_x (MS)	
(bb)	CV_x (LS)	DATA WORDS
(cc)	CV_y (MS)	
(dd)	CV_y (LS)	

Table 4.3.3.2-9. LGC Format for AGS Initialization (Continued)

(ee)	Data Index (6)	ID WORD
(ff)	CV_z (MS)	
(gg)	CV_z (LS)	DATA WORDS
(hh)	CTE (MS)	
(ii)	CTE (LS)	
(jj)	Data Index (7)	ID WORD
(kk)	TA (MS)	
(ll)	TA (LS)	DATA WORDS
(mm)	XXX	
(nn)	XXX	

Abbreviations

L--------LM

C--------CSM

P--------Position

V--------Velocity

T--------Time

A--------Absolute

XXX-----Data not pertinent to AGS operation

4.3.3.2.4.1.6 GSE Inputs. For information on this subject refer to LSP-300-33E, Section 3.3.1.4.5.1.

4.3.3.2.4.2 Output Signals. The following paragraphs describe the AEA outputs. See Table 4.3.3.2-10 for further definition of the outputs.

4.3.3.2.4.2.1 DEDA. The AEA provides readout (via the DEDA data output signal line) of any preselected parameter by the DEDA (including total velocity, V_T) upon receipt of a readout data signal. The maximum number of parameters that may be preselected for output to the DEDA 777 (octal notation) is equivalent to 511 (decimal notation). The AEA provides a DEDA data shift pulse signal.

4.3.3.2.4.2.2 Instrumentation Subsystem. The AEA supplies an AGS downlink signal which is a 24-bit digital word. The word format shall be as follows:

a) The six most significant bits are used for identification.

b) The 18 remaining bits are data.

c) The word is transmitted with the most significant bit first. The AEA uses the telemetry control signals given in Table 4.3.3.2-8 to start, stop, and shift the 24-bit word to the instrumentation subsystem.

d) The output rate shall be 50 words per second.

e) The word bit is as per Table 4.3.3.2-11.

4.3.3.2.4.2.3 ASA. The AEA provides the ASA with a 128-kpps synchronizing signal.

4.3.3.2.4.2.4 CES. The AEA provides the CES with steering signals (E_x, E_y, E_z) and engine commands (engine-on command and engine cutoff command) during any abort.

4.3.3.2.4.2.5 Displays. The AEA continuously provides the display subsystem with:

a) Total attitude (sine α and cosine α, sin B and cos B, sin γ and cos γ) in all modes except standby

b) Attitude error (E_x, E_y, E_z) during the inertial reference mode

Table 4.3.3.2-10. Output Signals

Signal	Basic Characteristics: Load	Source	Signal Characteristics
Engine Cut-off Command Engine-on Command	15K ohms, min, returned to 28 volts ± 12 volts dc	On: 2 K ohms Off: 100 K ohms minimum	Saturated transistor switch closure
Switching Ground to GSE	100 ohms, max	As required	NA
Test Mode Fail	Equivalent circuit of 14 volt battery in series with 300 K ohms resistor	Transistor switch closure	On: 2 volts, max Off: 12 volts, min
Shift 4 Bits In Shift 4 Bits Out	Pulse transformer coupled, 900 ohms, min	Pulse transformer coupled, 50 ohms ± 20%	Freq: Single pulse Amplitude: 5.5v ± 20% Pulse width: 0.5 to 2.0μsec
DEDA Clock	Pulse transformer coupled 900 ohms, min	Pulse transformer 50 ohms ± 20%	Freq: 128 kpps Amplitude: 5.5v ±20% Pulse width: 1.7 to 2.7 μsec
DEDA Data	Pulse transformer coupled 900 ohms, min	Pulse transformer 50 ohms ± 20%	Freq: 64 kpps; 36-bit bcd word Amplitude: 5.5v ± 20% Pulse width: 1.3 to 3.5μsec
Lateral Velocity	47.5 K ohms to 105 K ohms	As required	0 to±5 vdc Dynamic range: 0 to ±200 ft/sec
Altitude	170 ohms, min	50 ohms ± 20% pulse transformer coupled (during pulse)	Freq: 5 per sec; 15-bit word (msb first) Amplitude: 6v ± 35% Pulse width: 3 μsec ± 1 μsec
Altitude Rate	170 ohms, min	50 ohms ± 20% pulse transformer coupled (during pulse)	14 bit word +1 sign bit. Sign bit first followed by most significant bits. Sign-magnitude notation. Freq: 5 per sec Amplitude: 6v ± 35% Pulse width: 3 μsec ± 1 μsec

Table 4.3.3.2-10. Output Signals (Continued)

	Basic Characteristics*		
Signal	Load	Source	Signal Characteristics
Shift Pulses to Altitude	170 ohms, min	50 ohms pulse transformer coupled	Freq: 64 kpps Amplitude: 6v ± 35%
Shift Pulses for Altitude Rate	170 ohms, min	50 ohms pulse transformer coupled	Freq: 64 kpps Amplitude: 6v ± 35%
AGS Downlink	500 ohms ± 10%	50 ohms nominal	Freq: 51.2 kc, max Pulse train: 50 words/sec; 24 bits/word Amplitude: 4v, -0.5v + 2v Pulse width: 4 μsec ± 1μsec
ASA Clock	Pulse transformer coupled 200 ohms, min	Pulse transformer coupled, 50 ohms ± 20%	Freq: 128 kpps Amplitude: 4v ± 1v Pulse width: 1 μsec ± 20%

Table 4.3.3.2-10. Output Signals (Continued)

Signal	Basic Characteristics: Load	Source	Signal Characteristics	Scale Factor
Switching Ground to Descent Engine Control Assembly	Switch closure	AEA signal ground		
Standby Reference to AGS Status Switch	Switch closure	AEA Power ground		
Switching Ground to DEDA	100 ma, max	AEA signal ground		
Switching Ground to CES	100 ma, max	AEA signal ground		
Attitude Error Signals (Ex, Ey, Ez) to CES Attitude Error Signals (Ex, Ey, Ez) to Displays	10 K ohms, min (Displays) 33 K ohms, min (CES)	Transformer coupled, 100 ohms, max	0 to 4.5 vrms nominal, 800 cps sinusoidal	300 mv/deg ± 5% with range of ± 15 degrees
Total Attitude Signals sin α sin β sin γ	2400 ohms, 75 deg ± 10 deg, min	100 ohms, max	$E_o \sin \theta_A \sin (2\pi ft + \zeta_1)$: 0-15 rms, 400 cps	granulation: 30 mv/step
cos α cos β cos γ	2400 ohms, 75 deg ± 10 deg, min	100 ohms, max	$E_o \cos \theta_A \sin (2\pi ft + \zeta_2)$: 0-15 rms, 400 cps	granulation: 30 mv/step

Table 4.3.3.2-11. AGS Downlink Signal Word Bit

Word	Symbol	Name
1 2 3	x y z	Compensated incremental components of vehicle rotation about the body axes. (Corrected and scaled gyro output.)
4 5 6	V_x V_y V_z	Compensated incremental velocity components accumulated along the body axes. (Corrected and scaled accelerometer output.)
7 8 9 10 11 12	A_{11} A_{21} A_{31} A_{13} A_{23} A_{33}	Direction cosines of the angles between the body axes and the first and third inertial axes, respectively.
13	S_3	Auto engine ON CMD, Auto engine OFF
14		CMD Mode word (AGS function selection).
15		DEDA address
16		DEDA data word
17		DEDA clear
18		DEDA readout
19	B_1, B_2, B_3, B_4, B_5, B_6	External inputs (CES discretes)
20 21 22 23 24 25	X_{LM} Y_{LM} Z_{LM} $\dot{X}_{LM}$ $\dot{Y}_{LM}$ $\dot{Z}_{LM}$	LM State Vector
26 27	t_{MSB} t_{LSB}	Absolute time
28 29 30 31 32 33	X_{CSM} Y_{CSM} Z_{CSM} $\dot{X}_{CSM}$ $\dot{Y}_{CSM}$ $\dot{Z}_{CSM}$	CSM State Vector

Table 4.3.3.2-11. AGS Downlink Signal Word Bit (Continued)

Word	Symbol	Name
34	T_o	Rendezvous time (from now) for chosen direct transfer orbit.
35	V_G	First impulsive velocity to be gained
36	V_O	Total impulsive velocity to be gained
37	V_L	ΔV capability remaining in LM
38	V_{yb}	Inertial velocity component along the y body axis.
39	$\delta 19$	Equation select, direct rendezvous/ parking orbit.
40 41 42	$\underline{A}bd(x)$ $\underline{A}bd(y)$ $\underline{A}bd(z)$	Components of the desired pointing direction (used to compute attitude error signals)
43	q	Pericynthion of LM orbit.
44	T_B	Time to engine burnout.
45	rf	Predicted burnout altitude.
46	h	Altitude.
47	$\dot{h}$	Altitude rate.

c) Lateral body velocity (V_y) (inertial reference mode only)

d) Altitude (h) (inertial reference mode only)

e) Altitude rate ($\dot{h}$) (inertial reference mode only)

4.3.3.2.4.2.6 GSE Outputs. For information on this subject refer to LSP 300-33E, Section 3.3.1.4.6.6.

4.3.3.2.5 Detailed Performance. The AEA has the following performance capabilities:

a) Drift Rate. All the errors resulting from the AEA processing limitations add up to an attitude reference drift rate of not more than 0.2 deg/hr.

b) Total Attitude Signal Accuracy. The total attitude signals are accurate to ±0.5-deg maximum per axis, except when within ±0.5 deg of either of the two attitude points of singularity. In this case the maximum error in the angle 'B' is less than 0.5 deg and the maximum error in the sum of the angles $\alpha + \gamma$ is less than 0.5 deg.

c) Acceleration Range. The AEA is capable of processing accelerometer range of ±100 ft/sec^2 with a velocity increment of 0.05 ft/sec/pulse or less.

d) Initial Reset. Upon application of power to the AEA, a signal is generated which resets all flip-flops and clears all registers which are required to be initially in the zero state. The reset function is completed within 20 seconds after the application of power. The PGNCS accumulators (θ, Ψ, and ϕ) reset only with a CDU zero signal.

e) Navigation outputs. From the navigation computations the following variables are computed and displayed.

1) Altitude and altitude rate

	Altitude	Altitude Rate
Scale factor:	2.34 ft/bit	0.5 ft/sec/bit
Range:	15 bits	14 bits plus sign
Output rate requirements:	5 times/sec	5 times/sec

2) Y-axis velocity

Range: ±200 ft/sec

Output rate requirements: Once per 2 sec

4.3.3.2.6 Mechanical Characteristics. The outline dimensions of the AEA are shown in Figure 4.3.3.2-9. The AEA weight does not exceed 32.5 pounds. This weight includes connectors but does not include mating plugs and cables or mounting hardware.

4.3.3.3 Data Entry and Display (DEDA). A brief description of DEDA Component Identification, Function, Mechanization, and Mechanical Characteristics is presented in the following paragraphs.

4.3.3.3.1 Component Identification. The DEDA consists of two main assemblies which are as follows:

a) Control Panel - The numerical display and pushbuttons are mounted on the front panel, and a switch matrix network is potted in place on the rear of the panel. A service cable loop permits the front panel to be separated from the logic enclosure.

b) Logic Enclosure - The enclosure is a hermetically sealed box filled with an inert gas designed to withstand one atmosphere of internal pressure. The enclosure houses nine multilayer circuit boards, which are bolted together. The circuit board stack is mounted to the enclosure with heat dissipating wedges. DEDA circuitry is composed of macro flat-packs and thin film networks.

4.3.3.3.2 Function. The DEDA enables manual control of the AGS modes of operation, manual insertion of data into the AEA, and the capability to command the contents of the accessible AEA memory to be displayed on the numerical displays. Of the 4096 AEA words, 452 are accessible via the DEDA.

4.3.3.3.3 Mechanization. The DEDA is mechanized with three distinct modes of operation consisting of the off mode, on mode, and operate mode.

a) Off Mode - In the off mode, no power is applied to the DEDA.

b) On Mode - The on mode is initiated when lighting power is applied. In this mode, the panel and keyboard shall be illuminated; however, the electroluminescent (EL) numerics shall remain extinguished.

c) Operate Mode - The DEDA is in the operate mode when the AGS is placed in the operate mode. At this time logic power is supplied to the DEDA.

See Figure 4.3.3.3-1 for a block diagram of the DEDA.

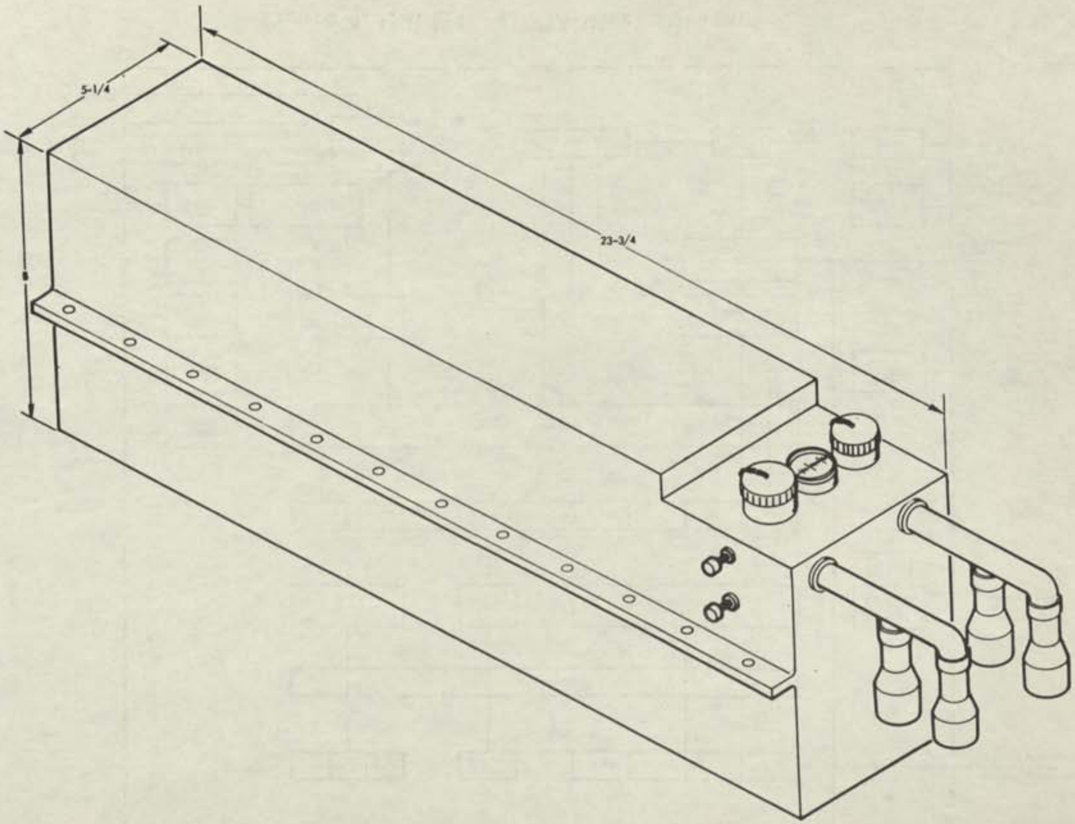


Figure 4.3.3.2-9. Abort Electronics Assembly

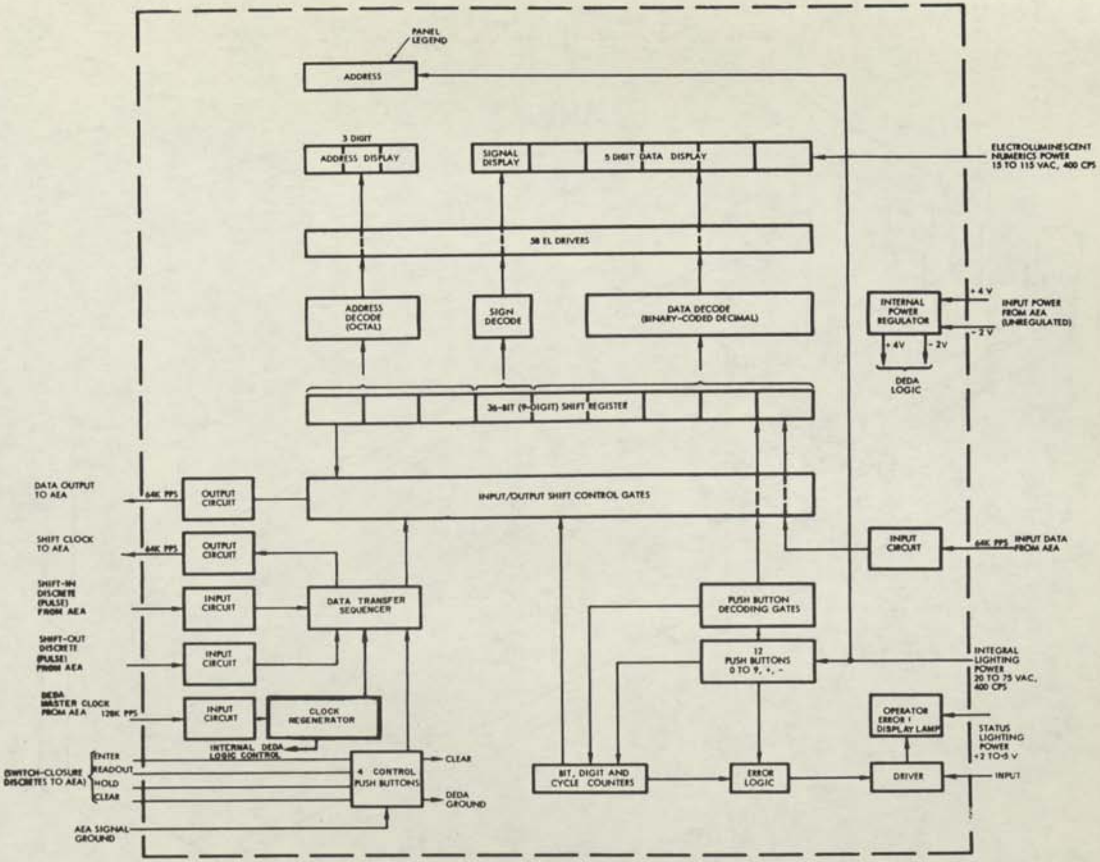


Figure 4.3.3.3-1. DEDA Block Diagram

4.3.3.3.3.1 General Operation. The DEDA operation is initiated by depression of the CLEAR pushbutton which enables entry from the digit pushbuttons. As each digit pushbutton is depressed, its code is placed in the register for display in the order that it is depressed (most significant digit first). When the appropriate number of digits have been entered, the ENTER or READOUT pushbutton can be depressed without generating an operator error. When entering information, nine digit pushbuttons must be depressed prior to depression of the ENTER pushbutton. When requesting a READOUT, three pushbuttons must be depressed prior to depression of the READOUT pushbutton.

The AEA computer recognizes the ENTER or READOUT signal and sends the DEDA a SHIFT-IN discrete. The DEDA then provides a serial sequence of a data pulse and a shift pulse that is repeated four times and results in the transferal of a digit code to the AEA computer. The computer issues further SHIFT-IN discretes and the process is repeated until all digits have been transferred. If the READOUT pushbutton had been depressed, following the entering process, the DEDA will receive a SHIFT-OUT discrete from the computer. The DEDA will then provide a sequence of four shift pulses and receive four bits of data from the computer in return. The computer issues further SHIFT-OUT discretes, and the process is repeated until the entire word is received. The information is displayed until the AEA computer updates the word or the CLEAR pushbutton is depressed, which terminates the readout operation. Upon depressing the HOLD pushbutton, the AEA computer will stop sending SHIFT-OUT discretes after the next complete word has been received. Depression of the READOUT pushbutton in this mode will cause the AEA computer to resume updating the displayed data.

4.3.3.3.3.1.1 Display Decoding. (Figure 4.3.3.3-2) The display segments for each digit are activated via drivers by outputs from the decoding logic associated with the respective 4-bit digit positions of the shift register. The three most significant digit positions (1, 2, and 3) of the shift register are decoded to form the address display. The next position (4) is decoded to form the sign display. The remainder of the digit positions (5, 6, 7, 8, and 9) are decoded to form the data display. The digit decode for each numeric display provides an output for each segment of the display. The segments are activated in such a manner as to form the decimal numbers. Two outputs are provided by the fourth digit decode to form the sign. The all one's code in any digit position of the shift register deactivates the decode outputs blanking the digit display.

1 DIGIT DECODE	2 DIGIT DECODE	3 DIGIT DECODE	4 DIGIT DECODE	5 DIGIT DECODE	6 DIGIT DECODE	7 DIGIT DECODE	8 DIGIT DECODE	9 DIGIT DECODE
ADDRESS			SIGN	DATA				

Figure 4. 3. 3. 3-2 Display Digit Decode Positions

4.3.3.3.3.1.2 Shift Register Function. The DEDA shift register accumulates and holds all information loaded from the pushbuttons or transferred from the AEA computer for display. The shift register contains 36 bits, comprising nine digits. Codes presented from the pushbuttons are loaded directly into the four least significant bits of the register and positioned for display. Address and data bits transferred from the AEA computer are loaded into the least significant bit position of the register. Address and data bits to be transferred to the AEA computer are taken from the most significant bit position of the register. The digit positions of information in the Shift Register are shown in Figure 4. 3. 3. 3-3.

4.3.3.3.3.1.3 Bit Counter Function. The DEDA bit counter is a 6-bit ripple counter. The counter accumulates the number of clock pulses gated to the shift register. It is used in conjunction with the digit counter to control positioning of digit codes in the shift register as they are loaded from the pushbuttons. The bit counter also is used to control the transfer sequence as information is transferred either to or from the AEA computer.

4.3.3.3.3.1.4 Digit Counter Function. The digit counter is a 5-bit shift register counter that is incremented on the final count of the bit counter. It is used to control positioning of digit codes in the shift register as they are loaded from the pushbuttons. In addition, the digit counter is used to detect operator error conditions.

MSB			SHIFT REGISTER					LSB
1	2	3	4	5	6	7	8	9
ADDRESS DIGITS			SIGN DIGIT	DATA DIGITS				

Figure 4.3.3.3-3 Digit Positions in Shift Register

4.3.3.3.3.1.5 Operator Error Indication. An operator error condition arises when pushbuttons are depressed out of sequence while loading the shift register. A flip-flop is set and the status indicator is illuminated.

4.3.3.3.3.1.6 Data Transfer Sequencer Function. The data transfer sequencer consists of a two step counter and miscellaneous flip-flops that buffer control and data pulses transferred from the AEA computer. The sequencer controls the timing of shift pulses and information bits transferred to or from the AEA computer and provides control for shifting the contents of the Shift Register during the transfer.

4.3.3.3.3.2 Functional Operation. Functional operation of the DEDA is described in the following paragraphs.

4.3.3.3.3.2.1 Pushbutton Load. To initiate loading from the pushbuttons, the CLEAR pushbutton must be depressed. This action sets the entire shift register blanking the display. In addition, the digit and bit counters, cycle control, and operator error logic are reset. Also, a flip-flop (ZAU3) is set which enables the keyboard control logic. This flip-flop is reset when either the ENTER or READOUT pushbutton is depressed.

As the pushbuttons are depressed, the encoded (BCD) signals are fed to gates which in parallel set or reset the four least significant flip-flops of the shift register. These gates are controlled by the pushbutton strobe, a delayed pulse generated in the keyboard control logic. The pushbutton strobe also sets the bit counter enable flip-flop (ZBCE) and the cycle control flip-flop (ZDCY).

The bit counter enable flip-flop opens a gate which feeds clock pulses to the shift register. These clock pulses are counted by the bit counter. When the bit counter reaches state 31, a gate (PBCC/T) is opened which resets the bit counter enable flip-flop on the next clock, thus terminating the shift register clocks. This same gate enables a clock pulse that is fed to the digit counter, incrementing the counter by one.

Digit codes loaded into the shift register are positioned in their proper location by the cycle control as the register is clocked. The first digit is positioned in the most significant end of the shift register, the second digit in the next most significant digit location and so on until the ninth (and last) digit is loaded into the least significant digit location. The cycle control positions the digit codes by opening gates which cause the information in the shift register to circulate out of the most significant end into the eighth digit position in the register while holding the current pushbutton input in the least significant digit position (see Figure 4.3.3.3-4). At the appropriate time, determined by the current states of the digit counter and bit counter, a gate is opened which resets the cycle control flip-flop allowing the current input to shift through the register behind the information previously loaded.

When loading the first digit, the cycle control flip-flop is not set so that the first digit is immediately shifted 32 places to appear in the most significant digit location. When loading the ninth digit, the bit counter enable flip-flop is not set as the input information is already in the correct

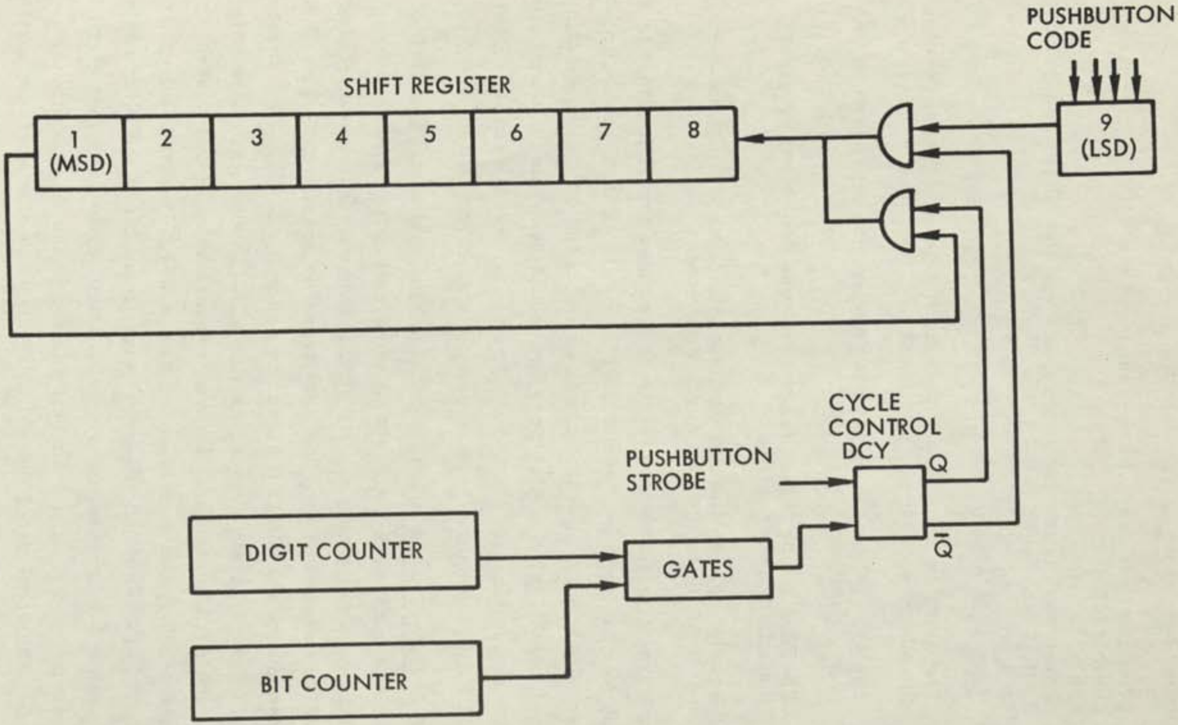


Figure 4.3.3.3-4. Pushbutton Load Cycle Control

position. An operator error indicator is illuminated by setting a flip-flop (ZOE1) under the following conditions:

a) Nine pushbuttons not depressed and the ENTER pushbutton depressed

b) Three pushbuttons not depressed and the READOUT pushbutton depressed

c) More than nine pushbuttons depressed

d) The eight or nine digit pushbuttons depressed when loading the address portion of the word

e) The plus (+) or minus (-) pushbuttons not depressed in the fourth position

f) The plus (+) or minus (-) pushbuttons depressed in other than the fourth position

g) Receipt of a shift-in or shift-out discrete from the computer while in the keyboard entry mode

The operator error flip-flop feeds a gate that causes the data output to the AEA to appear as all ones. If the ENTER or READOUT pushbutton depression resulted in the operator error or occurred during an existing operator error condition, the AEA program would recognize the error by detecting ones in positions normally constrained to contain zeros. Depression of the CLEAR pushbutton resets the operator error flip-flop.

4.3.3.3.3.2.2 Data Transfer. Upon loading nine digits into the shift register from the pushbuttons and visual verification that an operator error condition does not exist, the ENTER pushbutton may be depressed. Flip-flop ZAU3 is reset enabling the data transfer sequencer. The AEA computer responds to the ENTER signal by sending a SHIFT-IN discrete to the DEDA. This discrete sets a flip-flop (ZAU2) which initiates data transferal to the AEA computer. Data are transferred in a two-step operation controlled by the data transfer sequencer. During the first step (S1) of the SHIFT-IN sequence, the most significant bit of the shift register is gated with a 2-microsecond wide clock and fed to an output circuit. During the second step (S2), a shift pulse is fed to the AEA computer. Also during S2, the contents of the shift register are shifted 1-bit position and the bit counter is incremented by one.

Each SHIFT-IN discrete results in four cycles of the data transfer sequencer or the transferal of one digit. The computer continues to send SHIFT-IN pulses until the entire word is transferred. As the contents of the register are shifted off the most significant end, one's are shifted into the least significant end, resulting in a blank display after all digits have been transferred.

Because of the four cycle operation of the data transfer sequencer, the address digits, which are coded in three bits (octal), and the sign digit, which is coded in one bit, are allotted four bits in the shift register. The extra bits are used in providing redundant codes for blanking the respective displays or detection of an operator error by the computer.

If it is desired to display a word within the AEA computer, the three octal digit address of the word is loaded into the shift register and the READOUT pushbutton depressed. The computer responds by sending SHIFT-IN discretes resulting in the transferal of the address digits to the computer. The computer then sends a series of SHIFT-OUT discretes. Each SHIFT-OUT discrete similarly sets ZAU2 which initiates the data transfer sequence. In addition, the SHIFT-OUT discrete sets a control flip-flop (ZAU1) which enables shift register loading of bits transferred from the computer.

During the first step (S1) of the SHIFT-OUT sequence, a shift pulse is fed to the AEA computer which transmits a data bit back to the DEDA, where it is stored in buffer flip-flop ZBUF. During the second step (S2), the buffered data bit is shifted into the least significant bit of the shift register as the register contents are shifted one bit position. Simultaneously, the bit counter is incremented by one. As for SHIFT-IN discretes, the data transfer sequencer cycles four time for each SHIFT-OUT discrete so that the computer must send a SHIFT-OUT discrete prior to transferal of each digit.

The AEA computer may continually update the DEDA display by sending repeated sequences of none SHIFT-OUT discretes with accompanying data. Upon depression of the HOLD pushbutton, the computer will stop sending SHIFT-OUT discretes after the next full word has been transferred to the DEDA. The contents of the shift register will remain unchanged. Upon depression of the READOUT pushbutton, the computer will resume sending SHIFT-OUT discretes and the contents of the shift register changes accordingly. In this mode of operation, depression of the READOUT pushbutton will not cause an operator error condition.

4.3.3.3.3.3 DEDA Logic. Logic of the DEDA is described in the following paragraphs:

4.3.3.3.3.3.1 Clock Regenerator. The DEDA logic operates at a clock rate of 128 kc per second. The master clock is supplied by the AEA, which enters the DEDA via a 372 isolation circuit and is fed to the clock regenerator. Two clocks of different pulse widths are supplied by the clock regenerator, which is composed of two one shots triggering consecutively. The first clock, N2C./T, has a pulse width of 2 μ seconds and is used in gating signals from the DEDA to the AEA. The second clock, N4N..T, is 400 nano seconds wide and is used for internal DEDA logic clocking. To provide sufficient drive, the clock N4N..T is supplied by a clock driver circuit which is fed by the second one shot. Clock signal N2C./T is supplied by the first one shot.

4.3.3.3.3.3.2 Digit Display Decoder. The display decoder provides seven outputs for each 4-bit numeric digit position of the shift register. Two outputs are provided for the sign digit. Each output activates an electroluminescent segment of the associated digit display (via a segment driver), when it is logically true. The segment numbers correspond to those shown in Figure 4.3.3.3-5. The address digits are numbered 1, 2, and 3, beginning with the most significant digit. The sign digit is 4 and the data digits are numbered 5, 6, 7, 8, and 9. The display decoder outputs are associated with the segments and digits by name. Thus, output signal P1DS3T feeds the driver of segment 3, digit 1, and signal P5DS7T feeds the driver of segment 7, digit 5, etc.

Decoding of each digit is accomplished in two levels. The first level combines codes that will activate two or more segments. The second level combines the first level outputs to activate a particular segment. Table 4.3.3.3-1 indicates the segments activated for all possible codes of a digit in the address, sign, or data position of the shift register. For the code consisting of all ones, no segments are activated in any digit position.

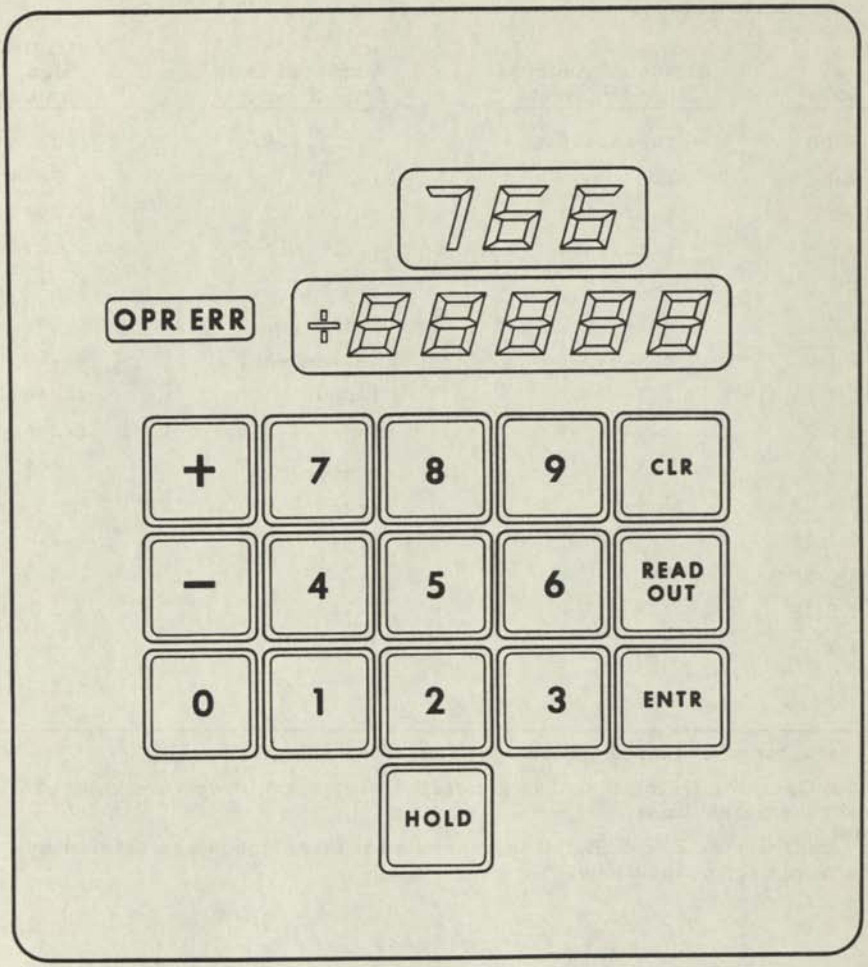


Figure 4.3.3.3-5. DEDA Front Panel

Table 4.3.3.3-1. Segments Activated for all Possible 4-bit Codes in Different Digit Positions of the Shift Register

Code	Activated Address Digit Segments	Activated Data Digit Segments	Activated Sign Segments
0 0 0 0	1-2-3-4-5-6	1-2-3-4-5-6	1-2
0 0 0 1	1-2	1-2	2***
0 0 1 0	1-3-4-6-7	1-3-4-6-7	---**
0 0 1 1	1-2-3-6-7	1-2-3-6-7	---**
0 1 0 0	1-2-5-7	1-2-5-7	1-2**
0 1 0 1	2-3-5-6-7	2-3-5-6-7	2**
0 1 1 0	2-3-4-5-6-7	2-3-4-5-6-7	---**
0 1 1 1	1-2-6	1-2-6	---**
1 0 0 0	2-3-4-6**	1-2-3-4-5-6-7	1-2**
1 0 0 1	2-3-6**	1-2-3-5-6-7	2**
1 0 1 0*	3-4-6	1-3-4-6	---
1 0 1 1*	---	1	---
1 1 0 0*	2	1-2-5	1-2
1 1 0 1*	2-3-6	2-3-6	2
1 1 1 0*	3-4-6	3-4-6	---
1 1 1 1	---	---	---

*These codes cannot be generated from the pushbuttons.

**An Operator Error Signal is generated when these codes are entered from the pushbuttons.

***An Operator Error Signal is generated if these codes are entered by the numbered pushbuttons.

The digits held in the shift register are encoded in 8-4-2-1 code. However, the address digits, which are allotted four flip-flops in the register, are in octal code. Therefore, the most significant bit of these digits is normally zero. If the most significant bit is not zero, the output of the particular address digit decoder will result in a test pattern of activated segments. Likewise, if a code of greater value than nine is contained in a data digit position, a test pattern is generated.

The fourth digit decoder supplies two outputs to activate the sign digit segments. P4DS1T activates the two vertical segments. P4DS2T activates the horizontal segment. To maintain data transfer sequence uniformity, four flip-flops of the shift register are assigned to the sign digit position; although, only two are used as inputs to the decoder. The plus sign is formed whenever the first two bits are reset (nominally coded 0000). The minus sign is formed whenever the least significant bit of the four is one and the second bit is zero (nominally coded 0001).

4.3.3.3.4 Mechanical Characteristics. The front panel of the DEDA is shown in Figure 4.3.3.3-5. The DEDA weight does not exceed 8.4 pounds. This weight does not include mating plugs and cables or mounting hardware but does include the external cable and connector.

4.3.4 Control Electronics Section

The Control Electronics Section is comprised of the attitude and translation controller assembly (ATCA), descent engine control assembly (DECA), rate gyro assembly (RGA), thrust/translation controller assembly (T/TCA), attitude controller assembly (ACA), and the gimbal drive actuator assembly (GDA). The gimbal angle sequence transformation assembly (GASTA), flight director attitude indicator (FDAI), engine sequencer and the ascent engine latching device (AELD), and the orbital rate drive electronics for Apollo and Lunar Module (ORDEAL) are not part of the CES: however, these units are discussed in conjunction with the CES for clarity.

4.3.4.1 Attitude and Translation Controller Assembly (LSP-300-14)

A brief description of ATCA Component Identification, Component Description, Function, Modes of Operation, Mechanization, and Outline Drawing is presented in the following paragraphs.

4.3.4.1.1 Component Identification. The ATCA includes the following principal subassemblies:

a) Three analog subassemblies

b) Four output subassemblies

c) One power supply subassembly

d) One parameter trim board

e) One elapsed-time meter

A block diagram of ATCA subassembly interconnections is shown in Figure 4.3.4.1-1.

4.3.4.1.2 Component Description. Each analog subassembly includes the following principal components:

a) Input transformers

b) Signal limiter (attitude error channel)

c) Demodulator-summer

d) Wide deadband

e) dc gain amplifier

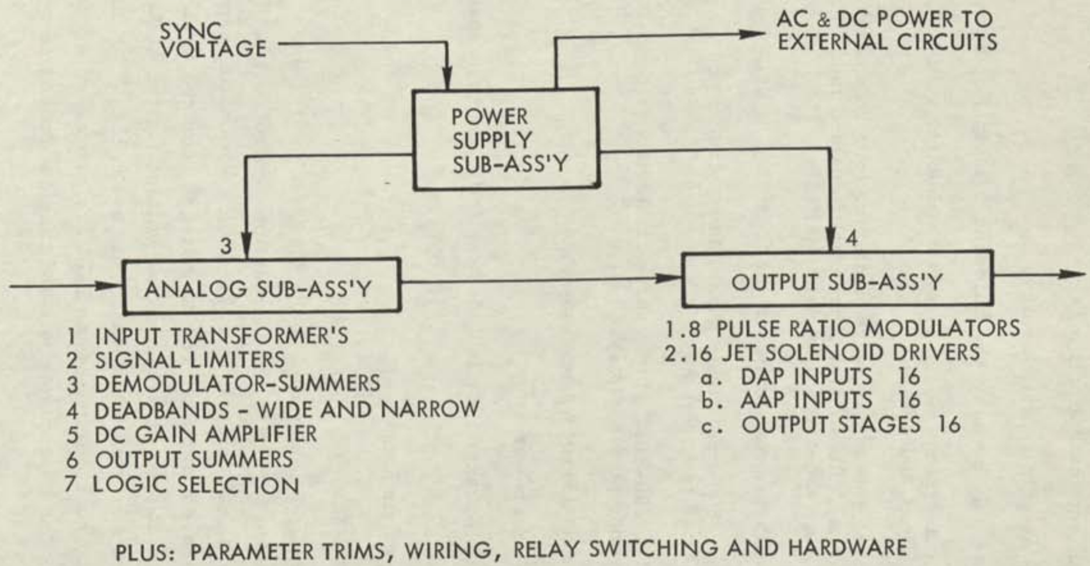


Figure 4.3.4.1-1. Attitude and Translation Control Assembly Block Diagram Assembly Breakdown

f) Narrow deadband

g) Logic circuits (includes four optimally located thruster logic circuits)

h) Output summing amplifiers

i) Mode or gain selection relays

Each output subassembly includes the following principal components:

a) Two pulse ratio modulators (with absolute value amplifiers and gates)

b) Four jet solenoid drivers (including four computer or digital autopilot input preamplifiers and four ATCA or abort autopilot preamplifiers)

The power supply subassembly includes the following principal components:

a) 26 vac, 800-cps, three phase, four wire, wye connected power supply

b) 28 vac, 800-cps, single phase, two wire, reference power supply

c) +15 and -15-vdc power supply

d) +4.3-vdc power supply

The parameter trim subassembly includes the following principal components:

a) Circuit elements

b) Mounting board

4.3.4.1.3 Function. The function of the ATCA is to (1) control the logical time and sequence for firing the 16 reaction-control jets both for attitude and translational control, and (2) provide automatic trim signals for the gimbaled descent engine. The dc voltages required in all of the CES subassemblies and the synchronized ac voltage for the RGA, AGS, and controllers are also generated within this assembly. The input signals from the AGS, hand controllers and the RGA are processed in the ATCA and are directed to the appropriate jets and the descent engine control assembly.

In PGNCS operation, only the LGC preamplifiers are enabled by the guidance select switch. LGC jet commands are amplified by the preamplifiers and jet drivers and sent to the primary solenoids of the RCS.

In AGS operation, only the abort preamplifiers are enabled and jet signals from the pulse ratio modulators are amplified and drive the RCS primary solenoids. Pitch, roll, and yaw signals are processed in three similar channels. Attitude error signals from the AGS are applied to the ATCA in automatic mode and attitude hold mode with the attitude controllers in detent. In narrow deadband operation the error signals are first passed through limiters to prevent excessive vehicle rates. The error signals are next summed with rate damping signals from the RGA. In attitude hold mode with an ACA out of neutral, proportional attitude rate commands are summed with the RGA rate damping signals. The gains of the summing amplifiers are changed when the LM is staged to account for the change in vehicle moment of inertia. The summed ac signals are changed to dc by keyed demodulators and filters. In the roll and pitch channels the demodulator output is sent to the DECA to control the trim of the gimballed descent engine. The deadband circuitry provides for either a wide deadband (±5 deg) or a narrow deadband (±0. 3 deg) depending on the mission phase. In automatic and attitude hold modes, the signals from the deadband circuits are applied to the jet select logic. In pulse mode, dc voltages from the ACA's are sent directly to the jet select logic circuitry. The jet select logic combines the attitude and translation inputs, determines whether two or four jets will fire to control rotation and x-axis translation, and applies the resultant signals to eight-pulse ratio modulators. The pulse ratio modulators generate pulses which vary in width and frequency depending on the input voltage in order to control the duty ratio of the RCS jets. Each pulse ratio modulator controls two opposing jets. The jet to be fired is determined by the sign of the pulse ratio modulator input voltage. (For RCS jet thruster commands refer to the applicable section of LSP-370-3).

4. 3. 4. 1. 4 Modes of Operation. The ATCA operating modes in the pitch, roll, and yaw channels are as follows:

a) Power on

b) Primary mode

c) Abort mode (normal)

1) Automatic mode

2) Attitude hold mode

3) Open loop modes (pulse and direct)

Abort mode (normal) or open-loop selection in each of three axes (pitch, roll, or yaw) is controlled by the three-mode select relays. Each of the three axes, or any combination thereof, may be in the open-loop mode with the remaining axis or axes in an abort or closed-loop mode.

4.3.4.1.4.1 Power On Mode. The application of power to the ATCA to furnish power for all internal and external circuitry is described in the following paragraphs.

4.3.4.1.4.2 Primary Mode. The PGNCS output consisting of sixteen incoming lines of dc level pulses drives a set of redundant jet solenoid driver preamplifiers (primary preamps). Power is furnished to the ATCA power supply and is distributed as follows:

a) 3 phase and 1 phase, 800 cps: to the rate gyros

b) 1 phase, 800 cps: to the control panel (CP) and internal circuitry

c) Regulated dc: to the descent engine control assembly (DECA) and control panel (CP)

d) Filtered 28 vdc (vehicle power): to the ATCA primary driver preamps, as enable power

4.3.4.1.4.3 Abort Guidance Mode. Attitude error signals from the abort guidance section terminate at the ATCA attitude error inputs culminating in jet driver command signals via the ATCA controlled jet driver channels. Rate command capability is also externally enabled. (See Figure 4.3.4.1-2 through 4.3.4.1-5.) Twenty-eight vdc jet drivers enable power is furnished to the ATCA controlled driver preamplifiers (abort preamps) instead of the primary preamplifiers. Other power is distributed as in the primary mode. The abort guidance mode operations are as follows:

4.3.4.1.4.4 Automatic Mode. Figure 4.3.4.1-2, and 4.3.4.1-3 show the function block diagrams for the ATCA in the automatic mode for

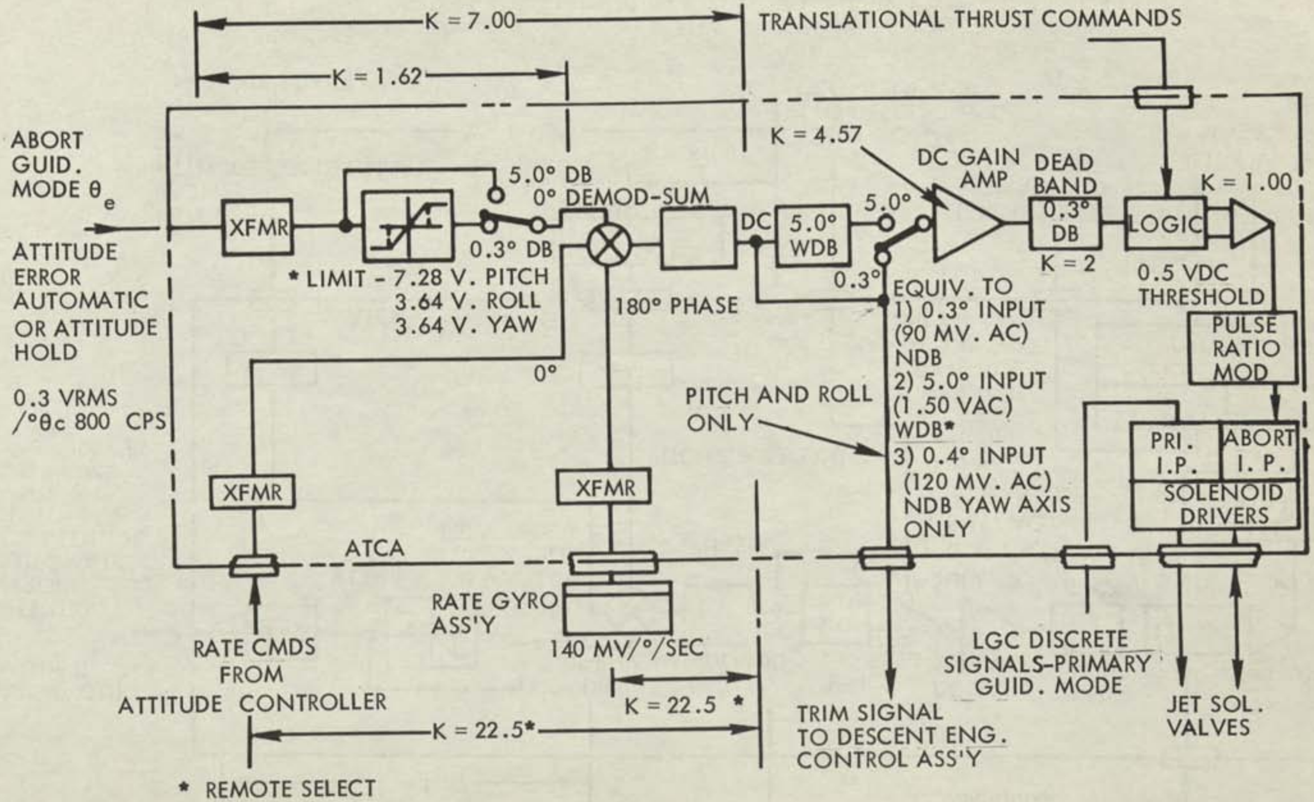


Figure 4.3.4.1-2. Attitude and Translation Control Assembly Pitch, Roll, or Yaw Block Diagram Mission Descent Phase - Normal Mode

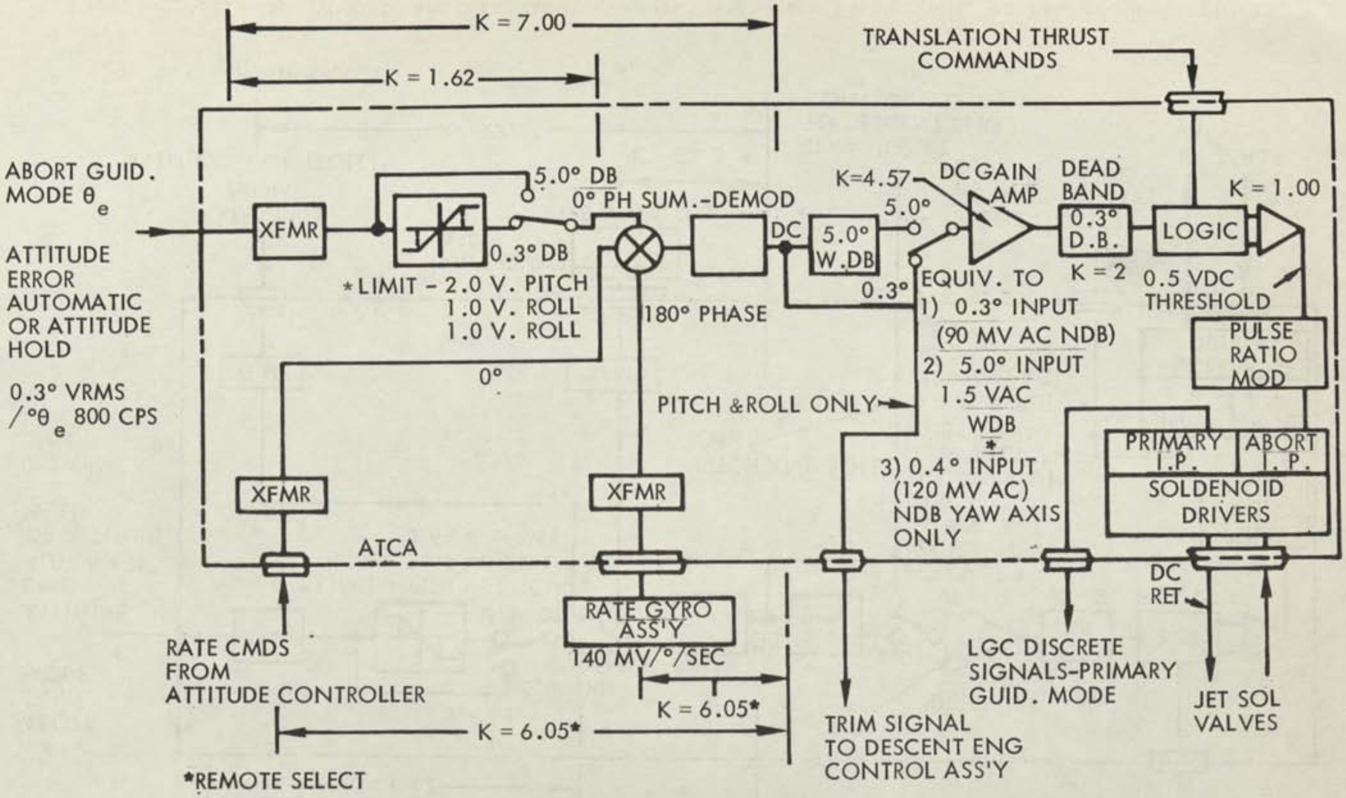


Figure 4. 3. 4. 1-3. Attitude and Translation Control Assembly Pitch, Roll, or Yaw Block Diagram Mission Ascent Phase - Normal Mode

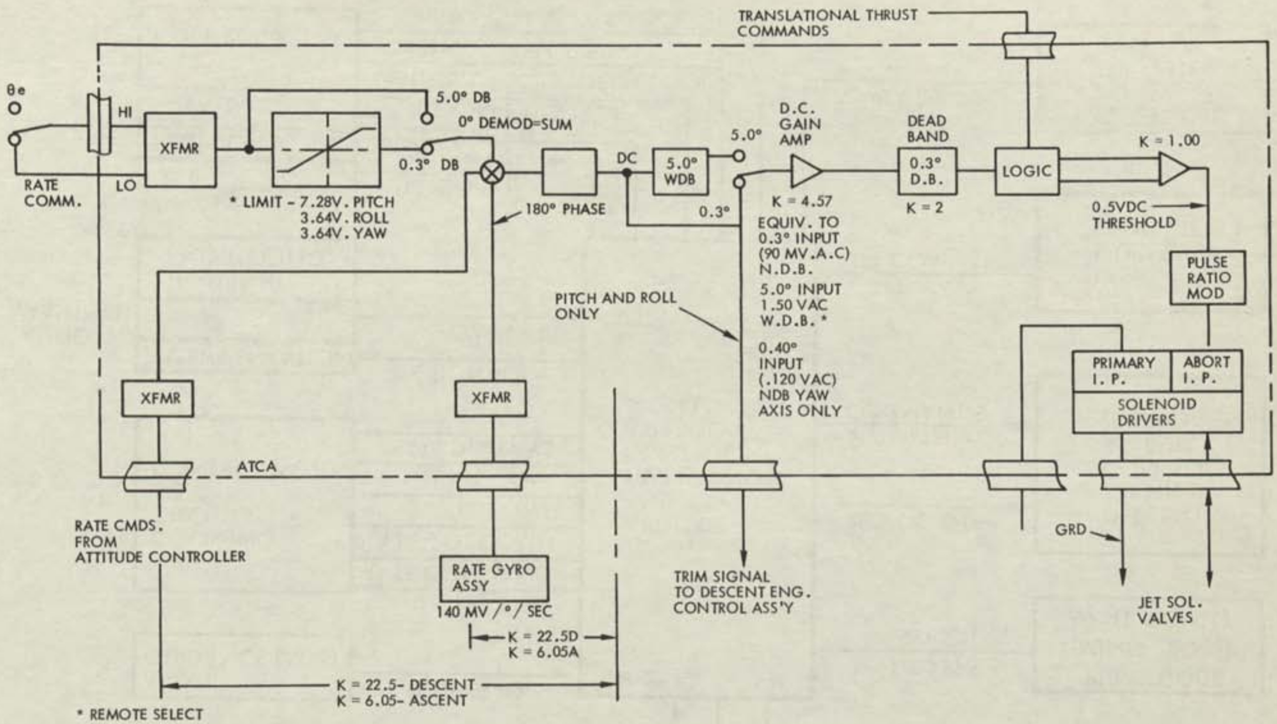


Figure 4. 3. 4. 1-4. Attitude and Translation Control Assembly Pitch, Roll, or Yaw Block Diagram Ascent-Descent Phase - Abort Mode

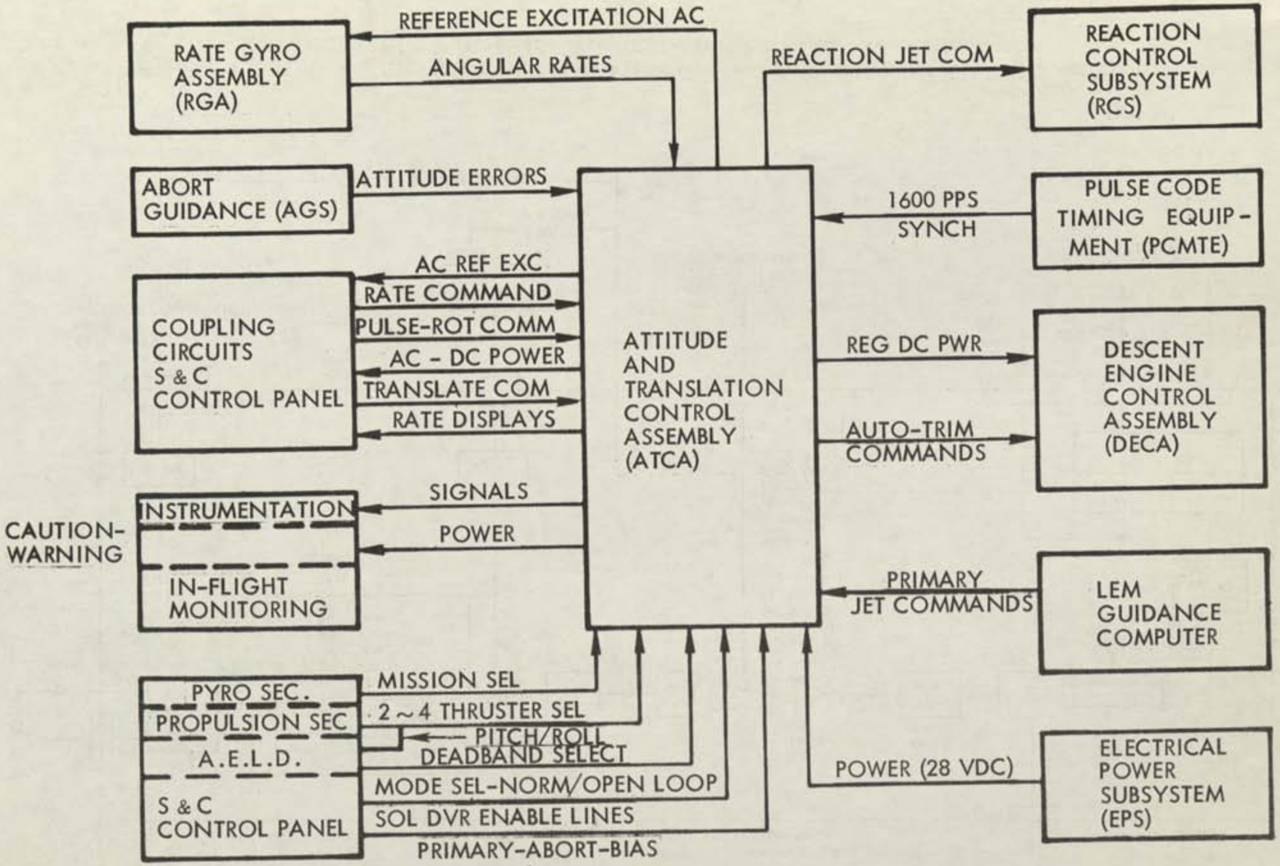


Figure 4.3.4.1-5. ATCA Interface Block Diagram

mission descent and ascent phases, respectively, on a single-axis basis. The three axes are identical except for the automatic trim commands which are derived in the pitch and roll axes only. The following signal inputs govern the operation:

a) Attitude errors: signals derived from the abort guidance section

b) Rate gyro feedback signals: inputs from vehicle rate transducers provide damping control

c) Translation input commands: translation thrust command to the logic from the thrust/translation controller assembly (T/TCA)

4.3.4.1.4.5 Attitude Hold Mode. This mode is externally activated. It provides manual rate command capability and holds vehicle attitude when no rate command is applied. Signal inputs are as follows:

a) Attitude errors: Signals are derived from the AGS while in the attitude hold mode.

b) Rate gyro input signals

c) Rate command input signals: Initiated by the attitude controller and summed with the rate gyro signal. When the controller is in its neutral position, the inputs will be at null.

d) Translation inputs: Translational thrust commands shall be from the controller.

4.3.4.1.4.6 Open-loop Modes. (see Figure 4.3.4.1-6)

4.3.4.1.4.6.1 Pulse Mode. Internal mode relays, remotely actuated by a ground, disconnect analog dc outputs and apply external fixed amplitude dc signals from the attitude controller assembly directly to the logic. The resultant pulse train controls jet operation. Each of the three axes may be controlled independently. Translational thrust commands are via the external thrust/translation controller assembly.

4.3.4.1.4.6.2 Direct Mode. The ATCA is in the pulse mode configuration during direct operation, except the pulse input signal is externally

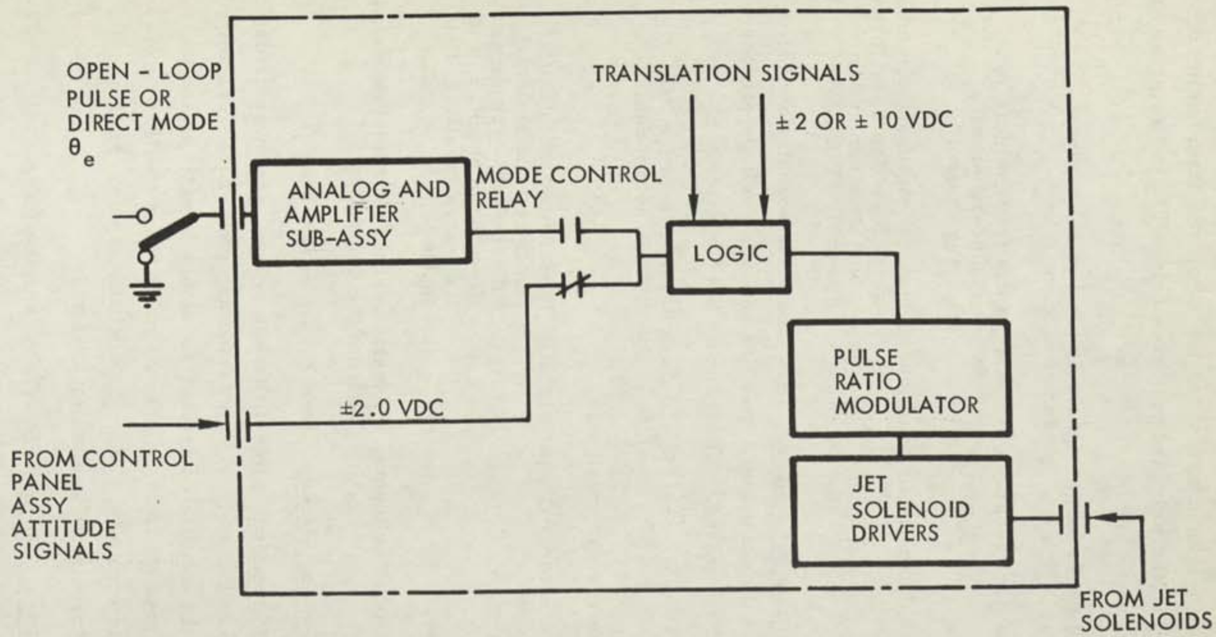


Figure 4.3.4.1-6. ATCA Block Diagram Open Loop Modes Pitch, Roll, or Yaw; X, Y, or Z

grounded. The ATCA is externally bypassed in this mode.

4.3.4.1.5 Mechanization

4.3.4.1.5.1 Details of ATCA Performance

4.3.4.1.5.1.1 ATCA Gains. The ATCA gains from input of the ATCA to the demodulator output in each of the three axes are as follows:

a) Attitude gain: 7.0 ± 5 percent vdc/vrms

b) In phase rate gain (feedback or command):

   1) Descent: 22.5 ± 5 percent vdc/vrms

   2) Ascent: 6.05 ± 5 percent vdc/vrms

4.3.4.1.5.1.2 ATCA Offset and Null

a) With the attitude and rate signal inputs shorted, the demodulator output is less than ±0.050 vdc (offset with input shorted),

b) With the signal inputs at zero in-phase 56 mv total rms noise with harmonics and guadrature of 70 percent or less and an attitude error source impedance of 1,000 ohms, the demodulator output is less than ±0.050 vdc for either rate or attitude input channels.

4.3.4.1.5.1.3 ATCA Resolution. The maximum allowable change in ATCA input without a corresponding change in dc amplifier output for signals in excess of deadband is 3 mv for either attitude or rate input.

4.3.4.1.5.1.4 ATCA Threshold. Threshold of all dc inputs into the logic subassembly is ±0.50 ±0.05 vdc. The minimum input signals required for pulse modulator operation are as follows:

4.3.4.1.5.1.4.1 Narrow Deadband Operation. For narrow deadband operation the threshold in each of the three axes are:

a) Attitude inputs: (with no rate inputs) 0.090 ± 11 percent vrms for pitch and roll signals 0.120 ± 11 percent vrms for yaw signals (P).

b) Rate inputs: (with no attitude inputs)

1) Descent phase: 0.028 ± 11 percent vrms for pitch and roll rate signals
0.037 ± 11 percent vrms for yaw rate signals (P)

2) Ascent phase: 0.105 ± 11 percent vrms for pitch and roll rate signals
0.140 ± 11 percent vrms for yaw rate signals

4.3.4.1.5.1.4.2 Wide Deadband Operation. For wide deadband operation the thresholds in each of the three axes are:

a) Attitude inputs: (with no rate inputs) 1.50 ± 10 percent vrms for pitch and roll signals
1.53 ± 10 percent vrms for yaw signals (P)

b) Rate inputs: (with no attitude inputs)

1) Descent phase: 0.47 ± 10 percent vrms for pitch and roll rate signals
0.48 ± 10 percent vrms for yaw rate signals (P)

2) Ascent phase 1.75 ± 10 percent vrms for pitch and roll rate signals
1.78 ± 10 percent vrms for yaw rate signals (P)

4.3.4.1.5.2 Analog Subassembly

4.3.4.1.5.2.1 Input Transformers

a) Input signal: attitude error at 0.30 vrms/deg

b) Input impedance: 33,000 ohms minimum at 800 ± 8 cps

c) Signal amplitude: 10 vrms maximum; 4.5 vrms operating range

4.3.4.1.5.2.2 Signal Limiter. The signal limiter limits the input signals to maintain the maximum rate command at ±10 degrees per second in pitch and ±5 degrees per second in roll and yaw during descent phase or ascent phase with narrow deadband only. With wide deadband the limiting action is eliminated.

4.3.4.1.5.2.3 Rate Input Signals. In each rotational channel both gyro feedback and command shall be transformer isolated and summed at the input amplifier to the demodulator. Required ascent-descent rate gains of both rate feedback and command shall be transformer isolated and summed at the input amplifier to the demodulator. Required ascent-descent rate gains of both rate feedback and command signals are obtained by the appropriate switching of input summing resistors at the demodulator summer. Proper polarities are observed for nulling of either rate command or attitude error signals.

a) Gyro rate feedback input circuit isolated:

Signal gradient:	0.140 vrms/deg/sec for each of the three rate commands
Input impedance:	10K ohms minimum 15K ohms maximum
Phase:	signals at zero or 180 ± 10 degrees maximum with reference 1 phase, 800 cps external excitation.
Maximum input:	4.80 vrms (2.80 vrms proportional working range)
Source impedance:	1100 + j 1200 ohms nominal
Rate gyro monitor display signal (to external circuits):	for yaw, pitch and roll rate circuits gradients (0.40 volts/deg/sec (into a 40K ohm load).

b) Rate command input isolated:

Signal gradient:	0.140 vrms/deg/sec (ordered)
Input impedance:	10K ohms minimum 15K ohms maximum
Phase:	signal at zero or 180 ± 10 degrees with reference 1 phase, 800 cps external excitation (180 degrees out of phase with rate feedback)
Maximum input:	4.80 vrms. (2.80 vrms proportional range)

Source impedance: less than 2K ohms

4.3.4.1.5.3 Amplifier Section

4.3.4.1.5.3.1 Demodulator - Summer

a) Input signals

1) Attitude error from limiter circuit

2) Rate gyro feedback signal (180 degrees out of phase with respect to attitude error or rate command signal).

3) Rate command signal from input transformer

b) Output signals (per channel)

1) dc signals to wide deadband circuit (wide deadband operation)

2) dc signals to dc gain amplifier (narrow deadband operation)

3) dc pitch and roll signals to descent engine control assembly (DECA external). (Minimum DECA impedance 20K ohms)

4) Maximum output ±14 vdc (12 vdc maximum linear)

c) Gain: see "ATCA Gains" above

d) Linearity: one percent of full output

e) Polarity: positive dc output for in-phase (0 degree with Reference 1 phase voltage excitation) input

f) Filtering: frequencies above 18 cps ± 20 percent shall be attenuated at 12 db per octave

4.3.4.1.5.3.2 Wide Deadband Circuit. The wide deadband circuits in each of the three axes are simultaneously bypassed by remote switching during narrow deadband operation. The total wide deadband including the thresholds of the narrow deadband circuit and the pulse modulator are equivalanet to ±5.0 degrees ±10 percent of attitude error input (with no rate signal) in each of the three axes. For wide deadband operation the threshold of the wide deadband circuit is 9.87 ±3 percent vdc.

a) Input signals: dc signals from demodulator (±12 volts linear maximum 14 vdc maximum saturation)

b) Output signals: signals to dc gain amplifier

c) Gain - (for signals in excess of deadband): 4. 57 ± 6. 25 percent including dc amplifier

4. 3. 4. 1. 5. 3. 3 DC Gain Amplifier

a) Input signals:

1) Wide deadband operation: the output of the wide deadband citcuit

2) Narrow deadband operation: the output of the demodulator

b) Output signals: ±12 vdc maximum, to the narrow deadband

c) Gain: 4. 57 ± 0. 057 vdc/vdc

d) Bandwidth: 0 to 100 cps minimum

4. 3. 4. 1. 5. 3. 4 Narrow Deadband Circuit. The total narrow deadband including the threshold of the pulse modulator is equivalent to ±0. 3 degree ±11 percent of the attitude error input (with no rate signal) in pitch and roll axes and ±0. 4 degree ±11 percent in yaw. For narrow deadband operation, the threshold of the deadband circuit is ±2. 63 vdc ±8. 5 percent for pitch and roll and ±3. 59 vdc ±8. 5 percent for yaw.

a) Input signals: dc signals from the dc gain amplifiers (±12 vdc maximum)

b) Output signals: through mode select relays to logic

c) Gain (for signals in excess of deadband): 2. 0 ±5 percent vdc (at signals of 3. 63 vdc input and up for pitch and roll channels, and 4. 59 vdc input and up for yaw ehannel)

4. 3. 4. 1. 5. 4 Logic Circuits. The logic circuit selects the pulse modulators, solenoid drivers, and associated reaction control thrusters to be operated for both rotational and translational control. Figure 4. 3. 4. 1-7 shows the locations of the 16 reaction control thrusters and defines the

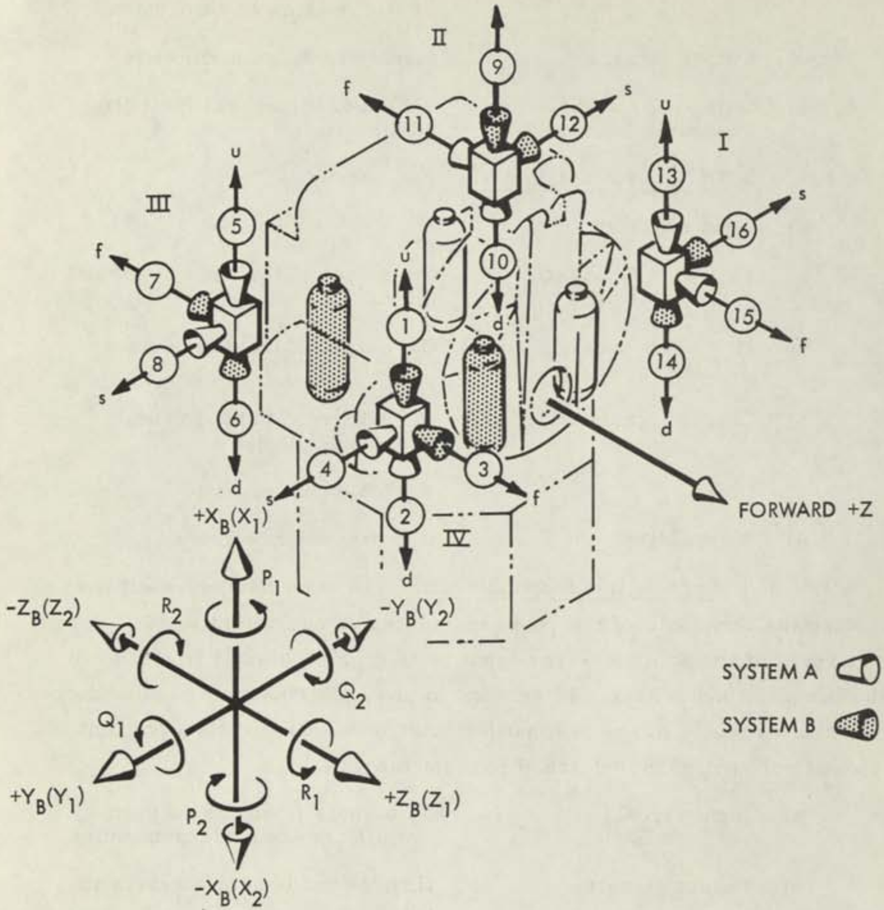


Figure 4.3.4.1-7. RCS Thruster Location and Identification

axes of rotation and translation. The logic circuit functional block diagrams are shown in Figures 4.3.4.1-8 and 4.3.4.1-9. Rotational and translational signals are combined in analog fashion prior to pulse modulation. Each of the eight pulse modulators control a pair of opposing reaction control thrusters through dual channel outputs.

4.3.4.1.5.4.1 Logic Inputs. Input impedance for all logic input signals shall be at least 5000 ohms.

a) Three rotational commands:

1) Normal modes:	three dc analog signals from the narrow deadband circuits
2) Pulse mode:	on-off signals ±2 vdc ±5 percent from attitude controller switches for pulse train operation

b) Three translational commands:	on-off signals at ±11 vdc plus 2 minus 0 vdc for positive or negative translation modes
c) Three mode select signals:	pulse mode is selected by an externally supplied ground to the mode control relays
d) Two jet selected inputs:	two thruster operation for Y-Z rotation or X translation is selectable by external switch closures with returns to the ATCA. Four-thruster operation for X rotation (P-yaw) is always closed within the ATCA. (See Figures 4.3.4.1-8 and 9 and Tables 4.3.4.1-1 and 2). Two jet rotations (above) are valid only with the presence of single axis rotation or X translation signals.

4.3.4.1.5.4.2 Logic Outputs. The logic output consists of eight dc analog signals to the pulse modulators of the output subassembly. Each of these eight signals is the summed net positive or negative output of the final summers. The proportional range for each summar input is 0 to 10 vdc. The gain of each summing amplifier is 1.0 ±3 percent for each input.

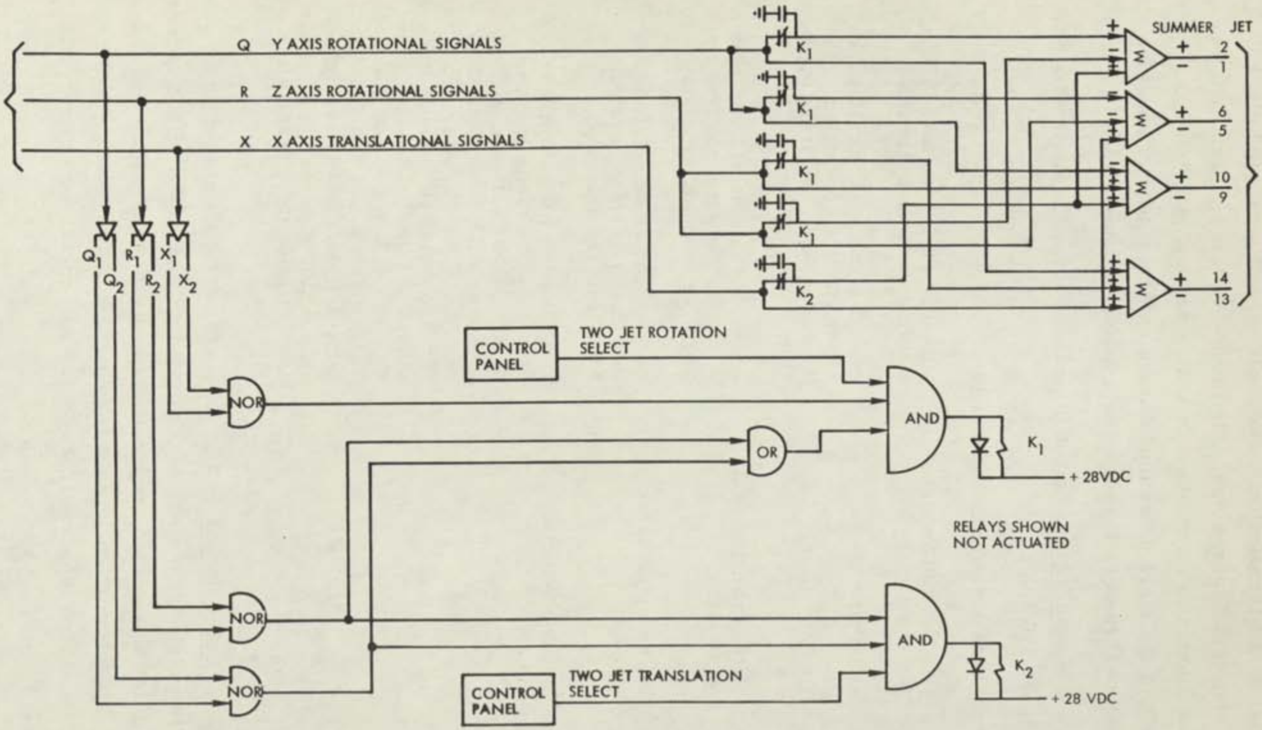


Figure 4.3.4.1-8. Symbolic Logic RCS Vertical Jet Selection ATCA

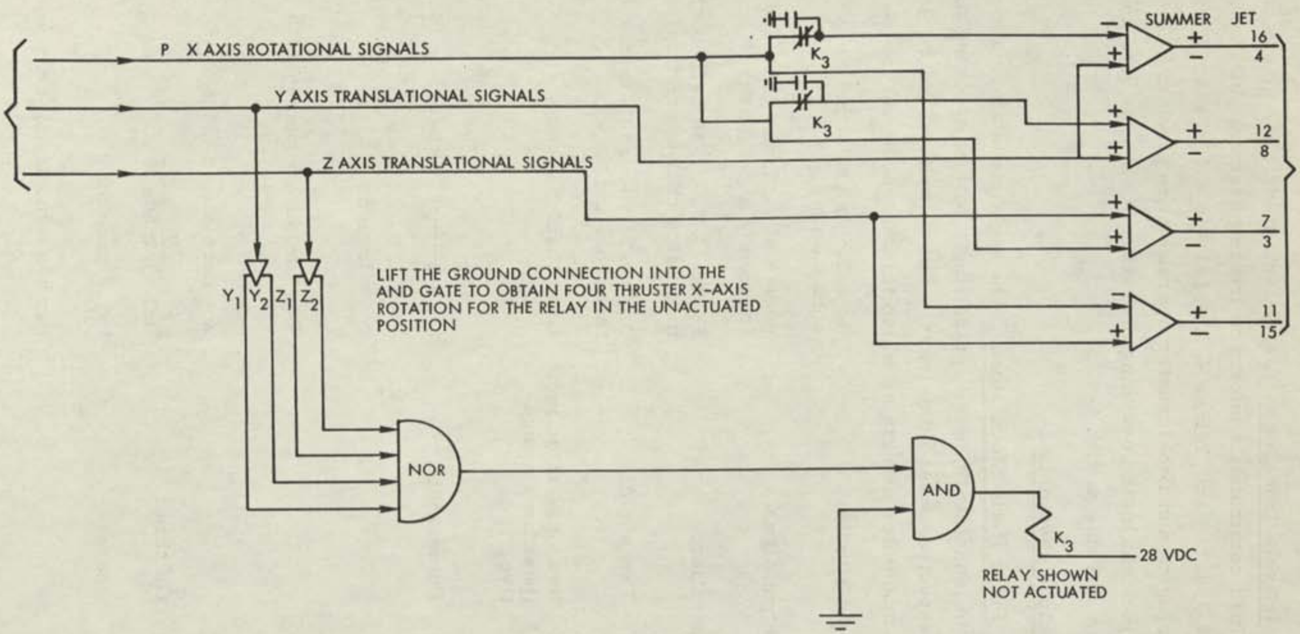


Figure 4.3.4.1-9. Symbolic Logic RCS Horizontal Jet Selection ATCA

4.3.4.1.5.4.3 Jet selection Logic. Jet selection logic for single command and combined commands conform to the requirements of Figures 4.3.4.1-8 and 4.3.4.1-9 and Tables 4.3.4.1-1 and 4.3.4.1-2. Equations for combination logic, and two-thruster operation are given in Table 4.3.4.1-3. Coincident logic governing operation of opposite jets and drivers is shown in Table 4.3.4.1-4.

4.3.4.1.5.5 Output Assembly

4.3.4.1.5.5.1 Pulse Ratio Modulators. The eight modulators are non-linear pulse ratio modulators, each controlling a pair of opposing thrusters through their respective solenoid drivers. The polarity of the dc input signals to each modulator determine which thruster is fired.

a) Input signils: dc analog signals from logic subassembly

b) Output signals: pulses of variable width and frequency to jet drivers

c) Operation: Figure 4.3.4.1-10 defines the modulator characteristics

   1) Threshold: 0.5 ±0.05 vdc inputs between threshold and 10 vdc.

   2) Response to dc input (positive or negative): Less than 25 μ sec

   3) Pulse width: $t_{on} = \frac{T_m}{1 - X}$, where:

      t_{on} = time on

      T_m = minimum pulse width= 13 msec

      X = normalized input

   4) Off time: $t_{off} = \frac{Tm}{\text{lambda } X}$

      where: t_{off} = time off

      lambda = modifying constant 0.1

Table 4.3.4.1-1 Vertical Thruster Ignition for Q, R and X Signals

Type of Signal			Thruster Response	Mode
Single Axis				
Q1			(9, 14)	Two-jet
Q2			(10, 13)	Rotation
R1			(5, 10)	
R2			(6, 9)	
Q1			(2, 5, 9, 14)	Four-jet
Q2			(1, 6, 10, 13)	Rotation
R1			(1, 5, 10, 14)	
R2			(2, 6, 9, 13)	
X1			(6, 14)	Two-jet
X2			(5, 13)	Translation
X1			(2, 6, 10, 14)	Four-jet
X2			(1, 5, 9, 13)	Translation
Q	R			
2	1		(1, 6, 10, 13) (1, 5, 10, 14)	
2	2		(1, 6, 10, 13) (2, 6, 9, 13)	
1	1		(2, 5, 9, 14) (1, 5, 10, 14)	
1	2		(2, 5, 9, 14) (2, 6, 9, 13)	
Q	X			
2	1		(1, 6, 10, 13) (2, 6, 10, 14)	All
2	2		(1, 6, 10, 13) (1, 5, 9, 13)	
1	1		(2, 5, 9, 14) (2, 6, 10, 14)	Modes
1	2		(2, 5, 9, 14) (1, 5, 9, 13)	
R	X			
2	1		(2, 6, 9, 13) (2, 6, 10, 14)	
2	2		(2, 6, 9, 13) (1, 5, 9, 13)	
1	1		(1, 5, 10, 14) (2, 6, 10, 14)	
1	2		(1, 5, 10, 14) (1, 5, 9, 13)	
Q	R	X		
2	2	1	(1, 6, 10, 13) (2, 6, 9, 13) (2, 6, 10, 14)	
2	2	2	(1, 6, 10, 13) (2, 6, 9, 13) (1, 5, 9, 13)	
2	1	1	(1, 6, 10, 13) (1, 5, 10, 14) (2, 6, 10, 14)	
2	1	2	(1, 6, 10, 13) (1, 5, 10, 14) (1, 5, 9, 13)	
1	2	1	(2, 5, 9, 14) (2, 6, 9, 13) (2, 6, 10, 14)	
1	2	2	(2, 5, 9, 14) (2, 6, 9, 13) (1, 5, 9, 13)	
1	1	1	(2, 5, 9, 14) (1, 5, 10, 14) (1, 5, 9, 13)	
1	1	2	(2, 5, 9, 14) (1, 5, 10, 14) (1, 5, 9, 13)	

Table 4.3.4.1-2. Horizontal Thruster Ignition for P, Y, and Z Signals

Type of Signal			Thruster Response	Mode
Single Axes				
	P1		(4, 7, 12, 15)	Four-jet Rotation
	P2		(3, 8, 11, 16)	
P	Y			
2	1		(3, 8, 11, 16) (12, 16)	All Modes
2	2		(3, 8, 11, 16) (4, 8)	
1	1		(4, 7, 12, 15) (12, 16)	
1	2		(4, 7, 12, 15) (4, 8)	
P	Z			
2	1		(3, 8, 11, 16) (7, 11)	
2	2		(3, 8, 11, 16) (3, 15)	
1	1		(4, 7, 12, 15) (7, 11)	
1	2		(4, 7, 12, 15) (3, 15)	
Y	Z			
2	1		(4, 8) (7, 11)	
2	2		(4, 8) (3, 15)	
1	1		(12, 16) (7, 11)	
1	2		(12, 16) (3, 15)	
P	Y	Z		
2	2	1	(3, 8, 11, 16) (4, 8) (7, 11)	
2	2	2	(3, 8, 11, 16) (4, 8) (3, 15)	
2	1	1	(3, 8, 11, 16) (12, 16) (7, 11)	
2	1	2	(3, 8, 11, 16) (12, 16) (3, 15)	
1	2	1	(4, 7, 12, 15) (4, 8) (7, 11)	
1	2	2	(4, 7, 12, 15) (4, 8) (3, 15)	
1	1	1	(4, 7, 12, 15) (12, 16) (7, 11)	
1	1	2	(4, 7, 12, 15) (12, 16) (3, 15)	

P1 = yaw left
P2 = yaw right

Table 4.3.4.1-3. ATCA Logic Equations for Signals to Pulse Modulators

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A. <u>Logic Relay Actuation Equations</u>

K_1 = (Two-jet Rotation) $\bullet$ $(Q_1^{1} \cdot Q_2^{1} + R_1^{1} \cdot R_2^{1}) \cdot X_1^{1} \cdot X_2^{1}$

K_2 = (Two-jet Translation) $\bullet$ $x \cdot Q_1^{1} \cdot Q_2^{1} \cdot R_1^{1} \cdot R_2^{1}$

$K_3 = Y_1^{1} \cdot Y_2^{1} \cdot Z_1^{1} \cdot Z_2^{1}$

Code:
1. Subscripts 1 and 2 represent positive and negative motion commands, respectively.
2. Superscript^1 represents a "not" signal.
3. Connecting · represents the logical "and".
4. Connecting + represents the logical "or".

B. <u>Modulator Electrical Input Equations</u>

E(2, 1)	=	$K_1^{1}Q$	$-K_1^{1}R$	$+K_2^{1}X$
E(6, 5)	=	$-K_1^{1}Q$	$-R$	$+X$
E(10, 9)	=	$-Q$	$+R$	$+K_2^{1}X$
E(14, 13)	=	$+Q$	$+K_1^{1}R$	$+X$
E(16, 4)	=	$-K_3^{1}P$	$+Y$	
E(7, 3)	=	$+P$	$+Z$	
E(11, 15)	=	$-P$	$+Z$	

Code: E (M, N) represents the analog signal input to the modulator that outputs to jet thrusters M or N.

Table 4.3.4.1-4. Coincident Logic for Opposed Jet Drivers

	Type Signal		Condition for Jet No.	Condition for Jet No.
Vertical Jets	Q	R	Q > R	Q < R
	2	1	13, 6	14, 5
	2	2	1, 10	2, 9
	1	1	2, 9	1, 10
	1	2	14, 5	13, 6
	Q	X	Q > X	Q < X
	2	1	1, 13	2, 14
	2	2	10, 6	9, 5
	1	1	9, 5	10, 6
	1	2	2, 14	1, 13
	R	X	R > X	R < X
	2	1	13, 9	14, 10
	2	2	2, 6	1, 5
	1	1	1, 5	2, 6
	1	2	14, 10	13, 9
Horizontal Jets	P	Y	P > Y	P < Y
	2	1	8	12
	2	2	16	4
	1	1	4	16
	1	2	12	8
	P	Z	P > Z	P < Z
	2	1	3	7
	2	2	11	15
	1	1	15	11
	1	2	7	3

NOTE: Above conditions shown allow opposite drivers to operate in proportion to the signal magnitude difference.

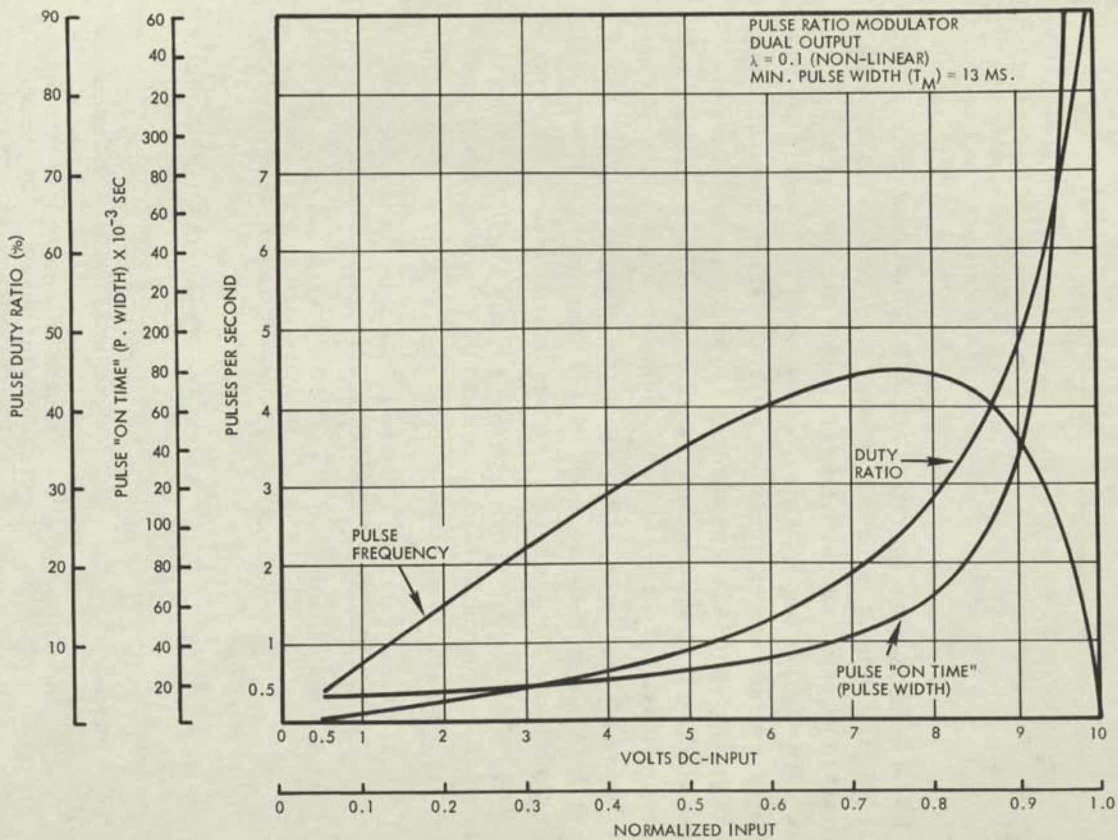


Figure 4.3.4.1-10. Pulse Ratio Modulator Dual Output

5) Duty ratio: $\dfrac{t_{on}}{t_{on} + t_{off}}$

6) Pulse frequency: $\dfrac{\text{duty ratio}}{\text{pulse width}}$

d) Failure operation: Failure of one of the two output channels in any pulse ratio modulator does not affect operation of the other channel.

4.3.4.1.5.5.2 Jet Solenoid Driver. Each of 16 jet solenoid drivers has two stages (i.e., an input and an output stage which is switched on by the input stage) to provide a ground return for the reaction jet fuel and oxidizer solenoids thus firing their associated jets. Each driver has two input stages; one primary preamplifier (external signal controlled), and one abort preamplifier (ATCA controlled). Primary preamplifiers are activated with 28-vdc enable power in the primary guidance mode and the abort preamplifiers (ATCA signal controlled) in the abort mode. The outputs of the two preamplifiers are coupled to one output stage. Any power failure of the ATCA up to the common point of the primary and abort input stages will not cause a failure of the primary input channel to the output stage. The converse is true for malfunctions in the 16 externally (LM primary guidance) controlled input channels. (See Figure 4.3.4.1-11 for the symbolic representations of these circuits).

4.3.4.1.5.5.3 Characteristics of Primary Externally Switched Preamplifier (or Input) Stage

a) B + enable power: 23-32 vdc externally filtered to no more than 10 volts above 32 vdc nor more than 20 volts below + 23 volts

b) Bias power: as required to insure positive operation to the source impedance level changes

c) Interface: signal levels of 16 primary inputs

1) Amplitude "1": source impedance less than 3K ohms (greater than 2K ohms). I_C = 5 ma max JSD output fires and provides ground for valve solenoid load.

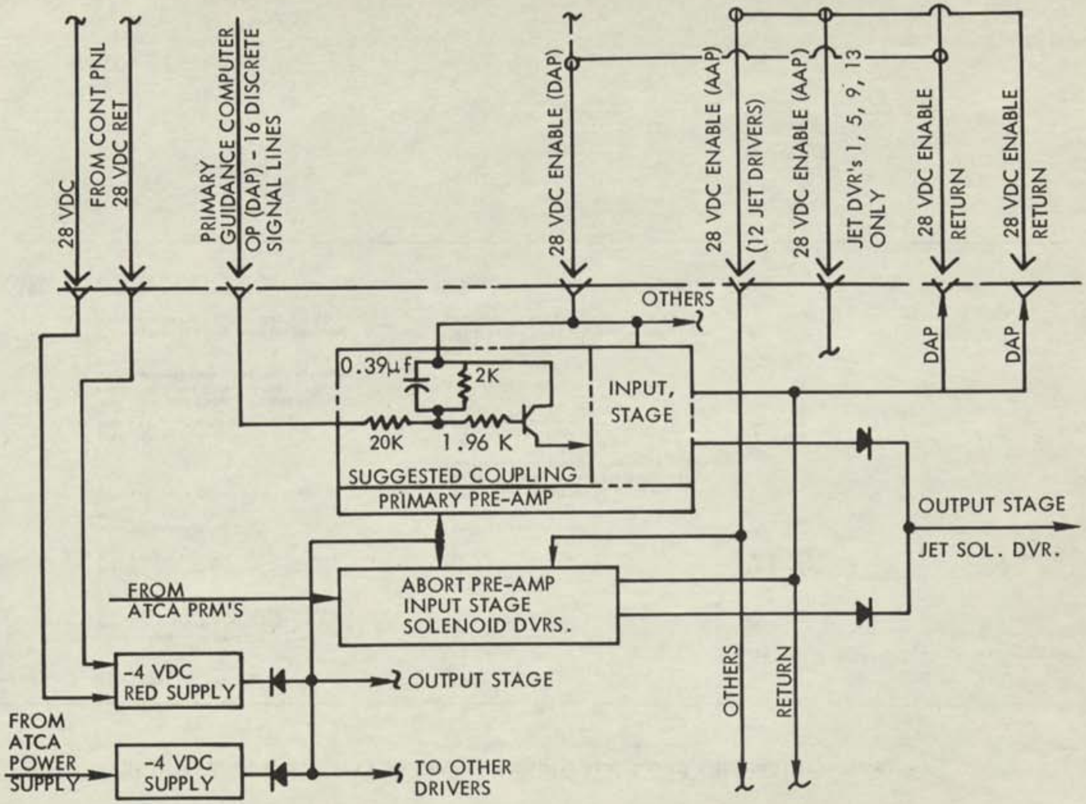


Figure 4. 3. 4. 1-11. Jet Driver Input Interfaced (Typical for ATCA)

Table 4.3.4.1-5. ATCA Power Supply and Load Characteristics

Power Supplied	3ϕ800 cps	1ϕ800 cps	dc	dc
Normal Voltage	26 vac L to L 15.0 vrms L to neutral 3ϕ, 4 wire, wye connected, phase rot. ABC, rate gyro or equivalent load connected	28 vrms Isolated 2 wire	+15 vdc, -15 vdc	+4.3 vdc
Steady State Regulation	Max 27.5 vrms L to L Min 24.5 vrms L to L	Max 28.5 v Min 27.5 v	+0.2 v Dev. -0.2 v Dev.	+0.1 v Dev. -0.1 v Dev.
Transient Regulation	Max 29 vrms L to L Min 23 vrms L to L 95% recovery time, 50 millisec	Max 30 vrms Min 26 vrms 95% recovery time, 50 millisec Interval between transient regulation disturbances 500 milliseconds minimum		
Transient Spike	Max +44 peak L to L Min -44 peak L to L 95% recovery time, 10 microseconds max. Interval between transient spike disturbances 500 milliseconds minimum.	Max 40 volts peak Min 16 volts peak	+1 v Dev. -1 v Dev.	±0.4 v Dev.
Frequency	With external sync: in synchronism without external sync: 800 ± 8 cps	With sync: in synchronism with input Without external sync: 800 ± 8 cps	dc	dc
Steady State Line Voltage Unbalance	Difference between L to L Voltages = 1.0 vrms max			

Table 4.3.4.1-5. ATCA Power Supply and Load Characteristics (Continued)

	3ɸ 800 cps	1ɸ 800 cps	dc	dc
Electrical Phase Angle	120 ± 2 degrees nominal between phases with phase A zero crossover points constant with the leading edge of the synchronization timing signal (Ref. 4.2.4.1.5.6.1 Stability of the sine wave zero crossovers shall not exceed ± 7 microseconds from the offset value.	Sine wave zero crossover points at 0 ± 7 microseconds steady state with the leading edge of the synchronization timing signal. Stability of the sine wave crossovers shall not exceed ± 7 microseconds from the offset value.	N. A.	N. A.
Waveform	Nominal sine wave. Max total distortion 5% Max distortion due to any one harmonic 4%	Nominal sine wave maximum total distortion rms 3%	N. A.	N. A.
Load Characteristics	3 phase 4 wire, wye connected balanced load Peak transient: 18 watts max Steady state: 8.5 watts Load power factor between 0.38 and 0.43 Load power unbalance: 15% max phase difference Nominal phase load impedance: (ref. $Z_{AN} = Z_{BN} = Z_{CN} = 40 \angle \underline{66}$ degrees) Phase relationship of spin power to the 1 ɸ supply is constant.	Total external load will be up to 8 watts maximum (internal as required) at 0.8 ± 0.1 power factor steady state for the various applications employed.	A. External to ATCA 1. To control panel 2. To DECA B. Internal to ATCA C. Max load requirements 1. External +15 vdc, 5 w -15 vdc, 5 w 2. Internal +15 vdc, as -15 vdc, required	A. External to ATCA 1. Control panel 2. DECA B. Internal to ATCA C. Max load requirements 1. External: 3.4 watts 2. Internal: as required
Other	Under normal conditions steady state load will be within 15% of the nominal value except peak power periods. Load can vary from full nominal to 50% of nominal. Duration of peak power periods will be no more than 60 seconds for each on-off cycle.		ac ripple: Within steady state regulation value. Long term drift within regulation limits.	ac ripple: Within steady state regulation value. Long term drift within regulation limits.

2) Amplitude "0": source impedance greater than 500K ohms. V (input to 28 volts enable return) = 40 vdc maximum

4.3.4.1.5.5.4 Characteristics of Abort (ATCA Controlled) Preamplifier Stages

a) B + enable power: 23-32 vdc externally filtered to no more than 10 volts above 32 vdc nor more than 20 volts below +23 vdc.

b) Bias voltage: as required to insure positive operation to pulse ratio module input pulses

c) Operation: solenoid ground returns are switched on with positive rise of pulse input and off with pulse decay of the pulse ratio modulator output signal.

4.3.4.1.5.5.5 Output Stage (Jet Solenoid Driver)

a) B + switch power: +28 vdc (low-side of solenoid load).

b) Bias voltage: as required per common usage circuit.

c) Operation: provides ground return to fuel and oxidizer solenoids upon signal level application to input switching (preamplifier) under either primary or abort operation conditions

4.3.4.1.5.6 Power Supply Subassembly. The power supply subassembly provides the power required for external loads as specified in Table 4.3.4.1-5. In addition, the power supply provides for all ATCA power requirements.

4.3.4.1.5.6.1 Three Phase and Single Phase Synchronization. The three-phase and single-phase power supply is synchronized to an externally supplied 1600-pulse-per-second synchronizing timing signal having the following characteristics:

a) Format. The timing signal is a square wave of 50 ± 1 percent duty cycle.

b) Frequency. The frequency of the timing signal is 1600 pps.

c) Output Impedance. The output source impedance is 100 ohms + 10 percent.

d) Load Impedance. The isolated ATCA load impedance is 100 ohms ± 10 percent.

e) Imput Level. The voltage levels of the input timing signals are:

1) $0.0 \begin{matrix} +0.5 \text{ volts} \\ -0.0 \text{ volts} \end{matrix}$

2) +3.0 ±0.5 volts

f) Amplitude Stability. The amplitude stability of the timing input voltage levels is ±5 degrees.

g) Frequency Stability. The frequency stability is within two parts for each million.

h) Rise and Fall Times. The rise and fall times of the input timing signal is less than 0.3 microsecond between the 10 and 90 percent amplitude points.

4.3.4.1.5.6.2 Free Running Frequency. The power supply "free runs" at a frequency of 800 ± 8 cps without a timing signal.

4.3.4.1.5.7 Parameter Trim Capability. The parameter trim elements are capable of changing the operating characteristics of various essential circuits within the ATCA. This is accomplished by interchanging certain key elements mounted at accessible points on subassembly modules. As a minimum, the following capabilities are provided:

Circuit	Parameter	Change Capability		
a) Limiters (3)	Level	Pitch - Descent	+7.5%	-50%
		Ascent	± 50%	
		Roll and Yaw Descent	± 50%	
		Ascent	+ 50%	-10%
b) Rate Gradient	Gain	Descent - Pitch	+ 15%	-50%
		Roll	+ 100%	-50%
		Yaw	+ 100%	-50%
		Ascent - All	+ 100%	-25%
c) Demodulator-Summer (3)	Gain		±10 %	
d) Deadbands (3)	Value			
WDB = 5^{o}		WDB	+ 0%	-50%
NDB = 0.3^{o}		NDB	+ 150%	-40%
e) Gain Amplifiers (3)	Gain		+ 200%	-50%
f) Drive Amplifiers (3) (Part of Narrow Deadband)	Gain		+ 100%	-50%

4.3.4.1.5.8 Control Relay Operation

a) Mode Control. Each of the three mode control relays independently controls normal and open-loop mode switching as follows:

   1) Normal mode: absence of a coil ground signal input, switches "control error signals."

   2) Open loop mode: presence of a coil ground signal input; dc signal for minimum pulsing to logic circuit and pulse modulators (see Figure 4.3.4.1-1).

b) Gain Control. The three gain control relays control high-gain and low-gain switching as follows:

   1) High gain: presence of a coil ground in the ATCA via a remote switch closure during descent phase.

   2) Low gain: absence of a coil ground in ATCA applied during ascent phase via a remote switch open.

c) Deadband Control. Each of the three relays control narrow deadband and wide deadband operation as follows:

   1) Narrow deadband: absence of a coil ground in the ATCA via remote switch or line open.

   2) Wide deadband: presence of a coil ground in the ATCA applied by a remote switch closure.

4.3.4.1.5.9 Electrical Interface

4.3.4.1.5.9.1 Input Signals. See Table 4.3.4.1-6 for input signal voltages and Table 4.3.4.1-7 for input signal types.

4.3.4.1.5.9.2 Output Signals. See Table 4.3.4.1-8 for output signals.

4.3.4.1.6 Outline Drawing. The outline of the Attitude and Translation Control Assembly is shown in Figure 4.3.4.1-12. The total weight of the ATCA does not exceed 27 pounds.

Table 4.3.4.1-6 ATCA Interface Voltage Inputs

Signal	Voltage	Gradient	Carrier Freq.	Modulating Frequency Max.
Attitude Error	0-4.5 vrms E_{in} = Emax sin (wt + θc) Emax = 10 vrms	.3 v/degree	800 cps synchronous with synchronizing Timing Signal	0-10 cps
Rate Gyro	0-2.80 vrms per gyro	140 mv/deg/sec	800 cps synchronous with synchronizing Timing Signal	0-10 cps
Attitude Rate Command (Controller Output)	0-2.80 vrms	140 mv/deg/sec command	800 cps synchronous with synchronizing Timing Signal	
Pulse Mode Commands	2.0 v (Controller)		dc	On-Off
Translation Commands	0-10 vdc		dc	0-10 cps
Synchronizing Timing Signal	3 volt level		1600 pps, 50% duty cycle square wave	Constant Constant

Table 4. 3. 4. 1-6. ATCA Interface Voltage Inputs (Continued)

Signal	Voltage	Gradient	Carrier Freq.	Modulating Frequency Max.
Power and Relay Control	28 vdc		dc	
Deadband	28 vdc (open)		dc	Open or ground
Mission Select	28 vdc (open)		dc	Open or ground
2-thruster commands	4. 3 vdc (open)		dc	Open or ground
16 DAP Commands	28 vdc (open		dc pulse train	On or Off
Primary Enable Power	28 vdc		dc	On or Off
Abort Enable Power	28 vdc		dc	On or Off

Table 4.3.4.1-7. Input Signal Types

Vehicle Rotation Command	Attitude Error Phase With Reference Signal Excitation	Type of Signal
+ Pitch (Y-axis)	0 deg (Nom)	Q_1
- Pitch (Y-axis)	180 deg (Nom)	Q_2
+ Roll (Z-axis)	0 deg (Nom)	R_1
- Roll (Z-axis)	180 deg (Nom)	R_2
+ Yaw (X-axis)	0 deg (Nom)	P_1
- Yaw (X-axis)	180 deg (Nom)	P_2
Translational Signals		
+ Y (v ref.)	dc +	Y_1
- Y (v ref.)	dc -	Y_2
+ Z (w ref.)	dc +	Z_1
-Z (w ref.)	dc -	Z_2
+ X (u ref.)	dc +	X_1
- X (u ref.)	dc -	X_2

Table 4.3.4.1-8. ATCA Interface - Output

Type	Voltage	Gradient	Frequency	Remarks
Solenoid Drivers	28 v (time off) (time on)	Nonlinear output relationship	0-4.5 pps approx. to 0.	13 ms to continuous pulse width
Rate Gyros				
Spin. Voltage Supply	26 vrms L to L ±1.5 vrms	NA	800 cps synchronous w. synch timing signal	3 ϕ 4 wire Wye connected. Power supply free runs at 800 ± 8 cps without external synch. Opposite "Ref. Signal Excitation," insert: 1 ϕ, sine wave zero crossovers at 0 ± 7 microseconds with leading edge of synchronizing timing signal.
Ref. Signal Excitation	28 vrms ± 0.5 vrms	NA	800 cps synchronous w. synch timing signal	Power supply free runs at 800 ± 8 cps without external synch.
Translation and Attitude Controllers Excite	28 ± 0.5 vrms	NA	800 cps	800 ± 8 cps with or without external synch voltage.

Table 4.3.4.1-8. ATCA Interface - Output (Continued)

Type	Voltage	Gradient	Frequency	Remarks
Amplifier Voltages	+ and - 15 v ± 0.2 vdc 4.3 v ± 0.1 vdc		dc	To Control Panel Assembly Descent Engine Control Assembly
Automatic DECA Trim	0-14.0 vdc	2.1 vdc/deg θe Att. gain	dc variable	Roll (ψe) and Pitch axes (θe) only for Gimbal control.
Rate Gyro Monitor Signals: Yaw, Pitch, and Roll	0-4.80 vrms (max.) -180 deg or +180 deg phase angle	0.040 v/deg/sec each (with 4OK ohm load)	800 cps carrier	Summed rate signals for displays.

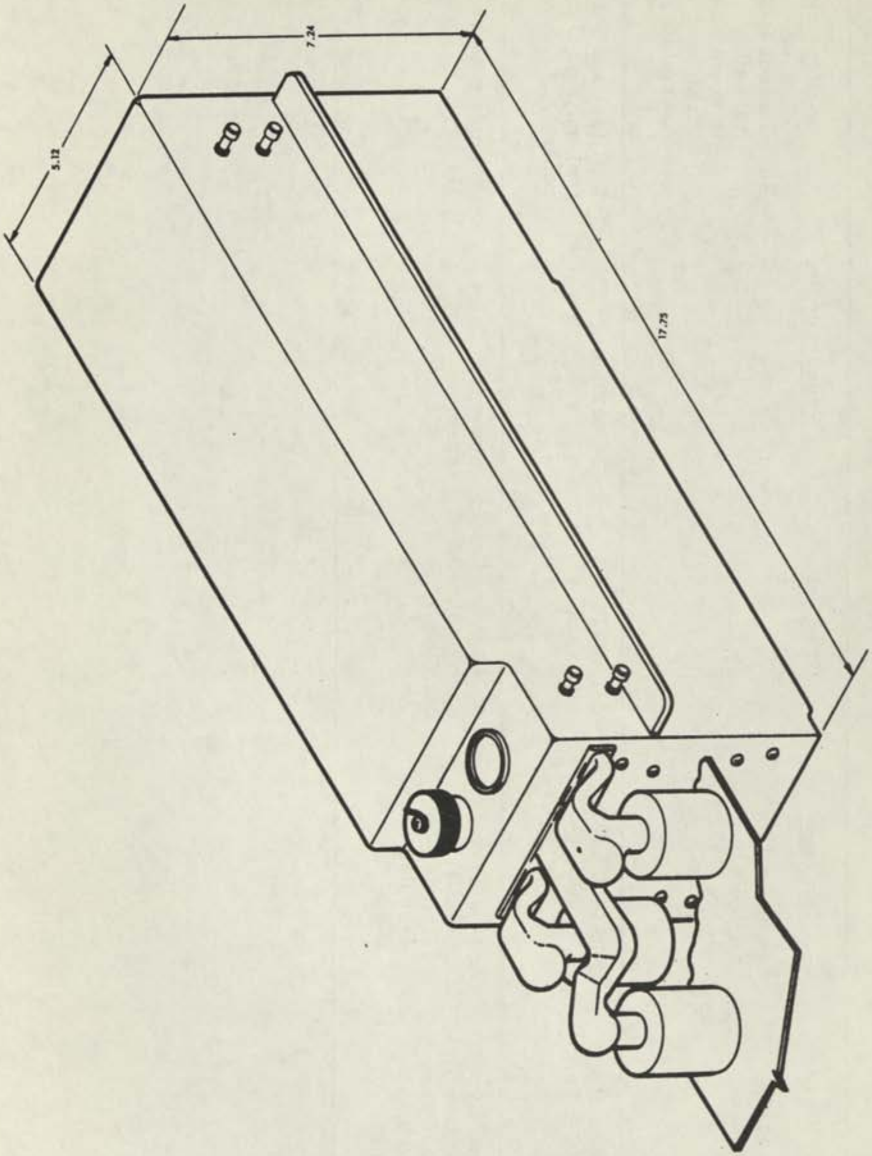


Figure 4.3.4.1-12. Attitude and Translation Control Assembly

4.3.4.2 <u>Descent Engine Control Assembly (LSP-300-13)</u>. A brief description of DECA Component Identification, Function, Mechanization, Mechanical Characteristics, and Electrical Characteristics is presented in the following paragraphs:

4.3.4.2.1 <u>Component Identification</u>: The DECA consists of the following subassemblies:

a) Two trim-error subassemblies, each of which includes:
   1) Signal-level detectors
   2) Trim mode select switching
   3) 400-cycle power switches
b) Two trim-error malfunction-detection subassemblies, each of which includes:
   1) Signal-level detectors
   2) Malfunction logic
   3) 400-cps demodulator
   4) Operational amplifiers
c) One auto-throttle subassembly, consisting of an integrator
d) One manual throttle subassembly, consisting of the following:
   1) A rectifier
   2) A filter
e) One power-switching subassembly, consisting of descent engine power switches
f) One elapsed-time indicator
g) Three electrical connectors

4.3.4.2.2 <u>Function.</u>

The DECA contains circuitry for controlling three functions of the descent engine. Engine throttling commands from the PGNCS and crew are processed and sent to the descent engine. PGNCS or ATCA signals are processed for positioning the Gimbal Drive Actuators to control engine trim. On-off commands are processed in the DECA to control descent engine ignition and shutdown.

Manual throttle commands can be applied to the descent engine with the throttle control switch on the control panel in either the manual or automatic position. Manual throttle commands consist of ac voltages, proportional to T/TCA X-axis displacement, applied to the DECA. The DECA demodulates and filters the signal and sends a proportional dc throttle voltage to the descent engine.

Automatic throttle signals can only be applied to the descent engine when the throttle control switch is in the automatic position. The LGC sends automatic throttle commands to the DECA in the form of pulses on one of two lines. Pulses on one line advance a forward-backward counter while pulses on the other line decrease the count. A dc voltage proportional to the count is generated and sent to the descent engine for automatic throttle control.

In PGNCS operation automatic trim control is provided by the LGC. When the LGC determines descent engine trim is required, it sends a trim command to the DECA on either a trim positive or a trim negative line for the roll or pitch axis. The trim signal is routed through a malfunction logic circuit to a power switching circuit. The power switching circuit applies 400 cycle 115 volts power to the proper gimbal drive actuator. The GDA rotates the descent engine and sends a 400-cycle signal proportional to its position back to the DECA. The DECA demodulates and differentiates the feedback signal and applies it back to the malfunction logic circuit. If the GDA is not moving in the commanded direction, the malfunction logic generates a trim failed signal and sends TRIM FAILED signals to the LGC and instrumentation.

In abort guidance operation, roll and pitch trim signals are received from the roll and pitch error channels in the ATCA. A comparator and threshold circuit in the DECA generates trim negative or trim positive commands and sends them to the malfunction logic controls the power switching circuits the same as in PGNCS operation.

4.3.4.2.3 Mechanization

4.3.4.2.3.1 Trim-error Subassemblies. The DECA, by sensing trim-error discrete or analog signal amplitude, controls the on/off excitation of 115-volt, 400-cycle ac power to external actuator loads. Each of two external roll actuator loads is controlled by separate trim-error signals and independent electronic and electromechanical circuits. Circuitry which affects pitch and roll analog trim-error threshold values is designed so that the threshold can be set between the values of 200 and 1000 mvdc without major modification. With analog trim-error signal magnitudes between the specified threshold magnitude and 20 vdc. the trim-error circuitry saturates. The DECA opens or closes ac power paths within 10 msec after the respective T_{Pos} or T_{Neg} states occur.

A simplified block diagram of the trim-error subassembly is shown in Figure 4.3.4.2.1. This figure is typical for the roll or pitch channel. In the discrete trim mode (K13 deenergized), the circuit logic is as follows:

LGC Command	400 cycle power
T_p	(1) ac power on line A phase referenced zero degrees
	(2) ac power on line B phase referenced 90 degrees
Neither T_p or T_N	(1) Line A and B open circuit
T_N	(1) ac power on line A phase referenced 90 degrees
	(2) ac power on line B phase referenced zero degrees

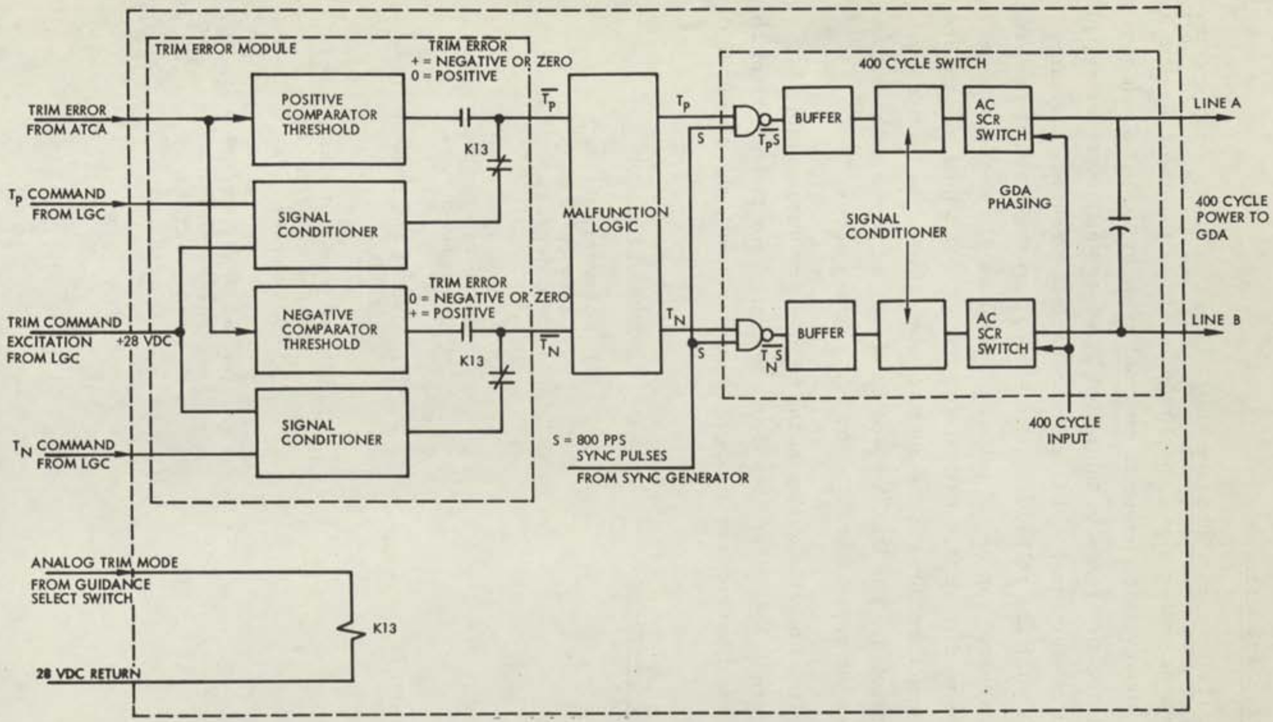


Figure 4.3.4.2-1. DECA Trim Error Subassembly Logic

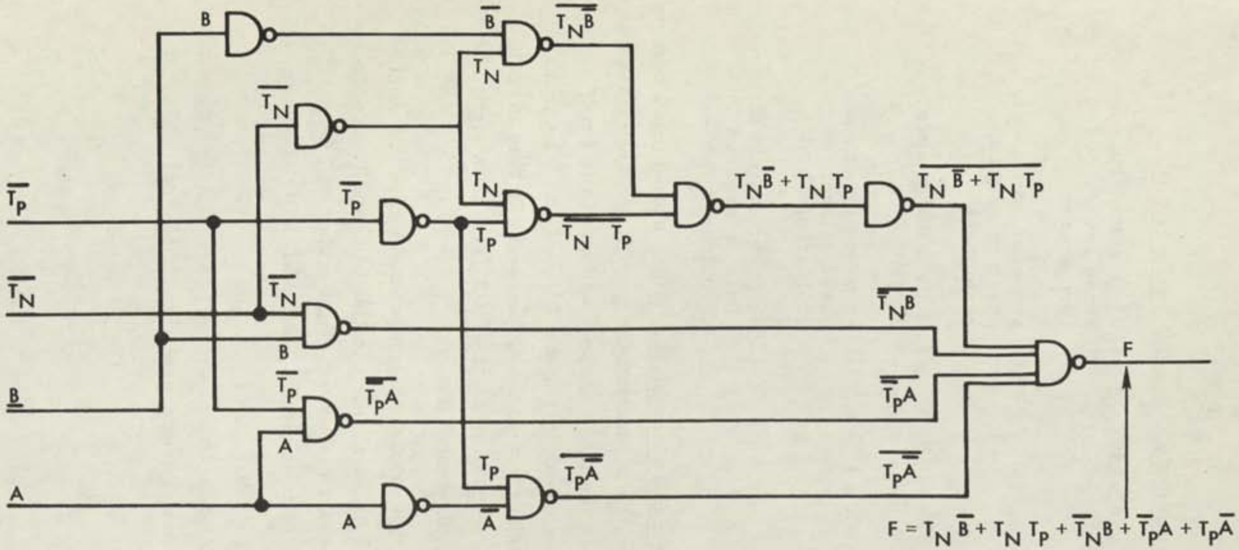
In the analog trim mode (K13 energized), the circuit logic is as follows:

ATCA Trim Error Amplitude	400 Cycle Power
Threshold> + 0.7875 vdc ± 15%	(1) ac power on line A phase referenced zero degree
	(2) ac power on line B phase referenced 90 degrees
-0.7875 vdc to + 0.7875 vdc	(1) lines A and B open circuit
Threshold < -0.7875 vdc ± 15%	(1) ac power on line A phase referenced 90 degrees
	(2) ac power on line B phase referenced zero degree

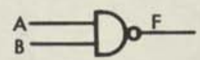
4.3.4.2.3.2 Malfunction Logic Subassemblies. This subassembly detects DECA switching or external load malfunctions in the GDA. There are two identical subassemblies; one in the pitch channel and one in the roll channel. The logic diagram is shown in Figure 4.3.4.2-2. The truth table for this logic circuit is shown in Table 4.3.4.2-1. In this circuit the "A" or "B" "1" state is generated when the GDA feed back signal is greater than ±0.36 vrms second, respectively.

If a failure is detected, the DECA shall respond within 0.5 second and transmit separate TRIM FAIL discretes to the LGC and the control panel by TRIM FAIL relay contact closures. This status will be maintained until the DECA is disarmed or until a TRIM SHUTDOWN command is received.

TRIM SHUTDOWN commands from control panel switches open circuit 400-cps power to the GDA and deenergize the TRIM FAIL relay in each channel.



T_N = NEGATIVE TRIM COMMAND
T_P = POSITIVE TRIM COMMAND
B = NEGATIVE POSITION GRADIENT > 0.36 VRMS/SEC
A = POSITIVE POSITION GRADIENT > 0.36 VRMS/SEC



LOGIC GATE: IF EITHER INPUT IS ABSENT, THERE WILL BE AN OUTPUT

Figure 4.3.4.2-2. DECA Malfunction Logic Diagram

Table 4.3.4.2-1. Trim Error Malfunction Logic - Truth Table

T_N	T_P	A	B	F
0	0	0	0	0
0	0	0	1	1
0	0	1	0	1
0	0	1	1	-
0	1	0	0	1
0	1	0	1	1
0	1	1	0	0
0	1	1	1	-
1	0	0	0	1
1	0	0	1	0
1	0	1	0	1
1	0	1	1	-
1	1	0	0	-
1	1	0	1	-
1	1	1	0	-
1	1	1	1	-

When F = 0 DECA continues monitoring trim error and load feedback signals

When F = 1 DECA generates a TRIM FAIL signal

$$F = T_P T_N + T_P \overline{A} + \overline{T}_p A + \overline{T}_N B + T_N \overline{B}$$

4.3.4.2.3.3 Auto-throttle Signal-conditioning Subassembly. The DECA provides an analog auto-thrust signal which varies between 0 and 12 vdc. These signals increase with an auto-throttle increase signal, decrease with an auto-throttle decrease signal, or remain constant when no auto-throttle command signal is present. The rate of change of the auto-throttle output signal is 14 mvdc per four pulses of auto-throttle increase signal and -14 mvdc per four pulses of auto-throttle decrease signal. Polarity is + for throttle increase. The net input pulse count is accumulated subject to the 0- to 12-vdc limits. Integration rate circuits allow changes in integration rate without major modification.

Prior to receipt of the engine-arm signal, the auto-thrust output signals are at signal ground potential. The auto-thrust output signals return to signal ground potential after the engine arm signal is removed. Upon receipt of the auto-thrust disable signal, the DECA returns the integrator and auto-thrust signals to the initial conditions until the auto-thrust disable signal is removed.

A simplified block diagram for the automatic throttle subassembly is shown in Figure 4.3.4.2-3.

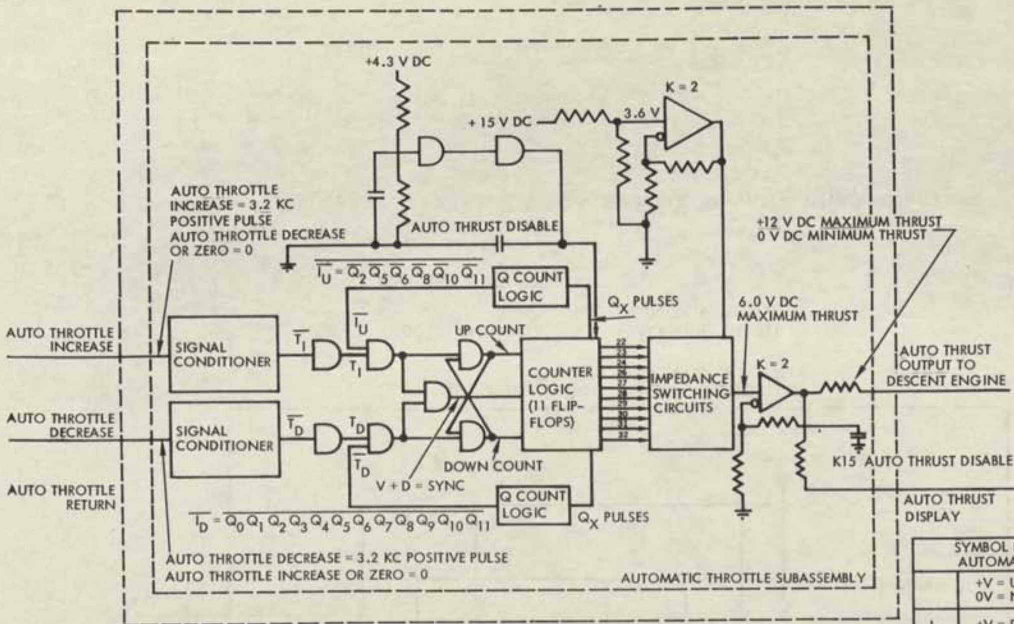
4.3.4.2.3.4 Manual Throttle Signal-conditioning Subassembly. The DECA receives an 800-cps manual throttle input signal. The DECA demodulates and provides a dc analog manual thrust signal proportional to the input. The manual thrust signal varies in amplitude from 2.6 to 14.6 vdc. A block diagram of this subassembly is shown in Figure 4.3.4.2-4.

The smallest manual throttle signal that will cause a detectable change in manual thrust output signal will be 0.05 v rms.

4.3.4.2.3.5 Power Switching Subassembly. Upon receipt of the DESCENT ENGINE ARM signal, the DECA will switch on the 28-vdc, ± 15-vdc, + 4.3-vdc, and 400-cps power supplies required for operation of the descent engine and its control packages.

This assembly also controls the manual and automatic modes of descent engine start/stop.

The manual descent-engine start and manual descent-engine stop signals are received from the AELD. The mechanization of this circuit



SYMBOL DEFINITIONS AUTOMATIC THROTTLE	
	+V = UP COUNT INHIBIT 0V = NO UP COUNT INHIBIT
I_D	+V = DOWN COUNT INHIBIT 0V = NO DOWN COUNT INHIBIT
	+V = AUTO THROTTLE INCREASE PULSES 0V = NO AUTO THROTTLE INCREASE PULSES
	+V = AUTO THROTTLE DECREASE PULSES 0V = NO AUTO THROTTLE DECREASE PULSES
Q_X	0V = COMMANDS 2^X SWITCH CLOSURE +V = COMMANDS 2^X SWITCH OPENING
U	0V = UP COUNT PULSE +V = DOWN OR NO UP COUNT PULSE
D	0V = DOWN COUNT PULSE +V = UP OR NO DOWN COUNT PULSE

Figure 4.3.4.2-3. Block Diagram DECA Automatic Throttle

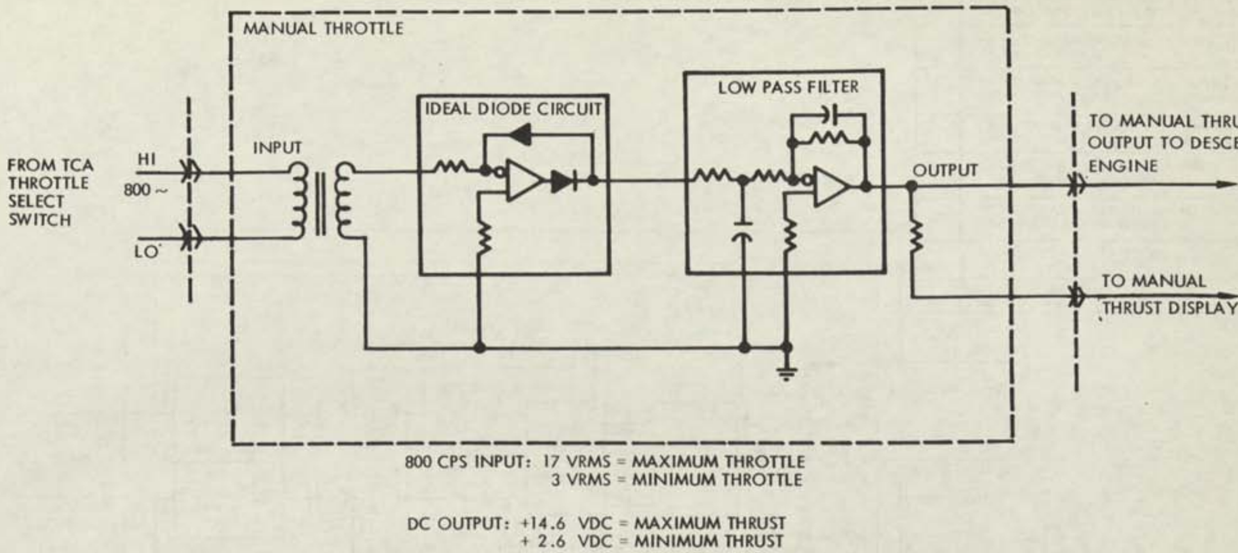


Figure 4. 3. 4. 2-4. DECA Manual Throttle Block Diagram

is illustrated by Figure 4. 3. 4. 2-5. The manual start signal closes a +28-vdc line in parallel with the circuit controlled by the auto descent engine-on signal. Either source will provide a +28-vdc start signal to the descent engine when armed.

The manual descent engine stop signal will interrupt the manual start or auto-on commands to the descent engine by deenergizing K7 and K6, the manual start, and auto-on relays. The stop signal takes precedence over any combination of auto descent engine or manual start signals.

The auto descent engine on signal in the absence of auto descent engine off and manual descent engine stop signals will energize K6 and supply a 28-vdc start signal to the descent engine. The logic diagram for the automatic mode of the descent engine control is shown in Figure 4. 3. 4. 2-6. The logic table for this circuit is shown in Table 4. 3. 4. 2-2.

4. 3. 4. 2. 4 Mechanical Characteristics

4. 3. 4. 2. 4. 1 Outline. The outline of the DECA package is as shown in Figure 4. 3. 4. 2-7. For more details refer to Grumman drawing #LSC-300-130.

4. 3. 4. 2. 4. 2 Weight. The weight of the DECA package is 7. 25 pounds or less.

4. 3. 4. 2. 4. 3 Reference. For details on mechanical design, configuration, and packaging, refer to Grumman Specification LSP-300-13, Sections 3. 3. 1 and 3. 3. 3.

4. 3. 4. 2. 5 Electrical Characteristics

4. 3. 4. 2. 5. 1 Input Signal Characteristics. The characteristics of DECA input signals are listed in Table 4. 3. 4. 2-3 and in Figure 4. 3. 4. 2-8.

4. 3. 4. 2. 5. 2 Output Signal Characteristics. The characteristics of DECA output signals are listed in Table 4. 3. 4. 2-4. Details of thrust output signals are shown in Figure 4. 3. 4. 2-9.

Table 4.3.4.2-2. Auto Descent Engine Control Logic

Descent Engine On	Descent Engine Off	State of K6A, Engine Start Relay
0	0	In state previously held (see note 1)
1	0	Closed circuit
0	1	Open circuit
1	1	In state previously held

NOTES:

(1) Prior to the receipt of the engine arm signal, K6A is always open circuit.

(2) 1 = Respective external contacts closed circuit.

(3) 0 = Respective external contacts open circuit.

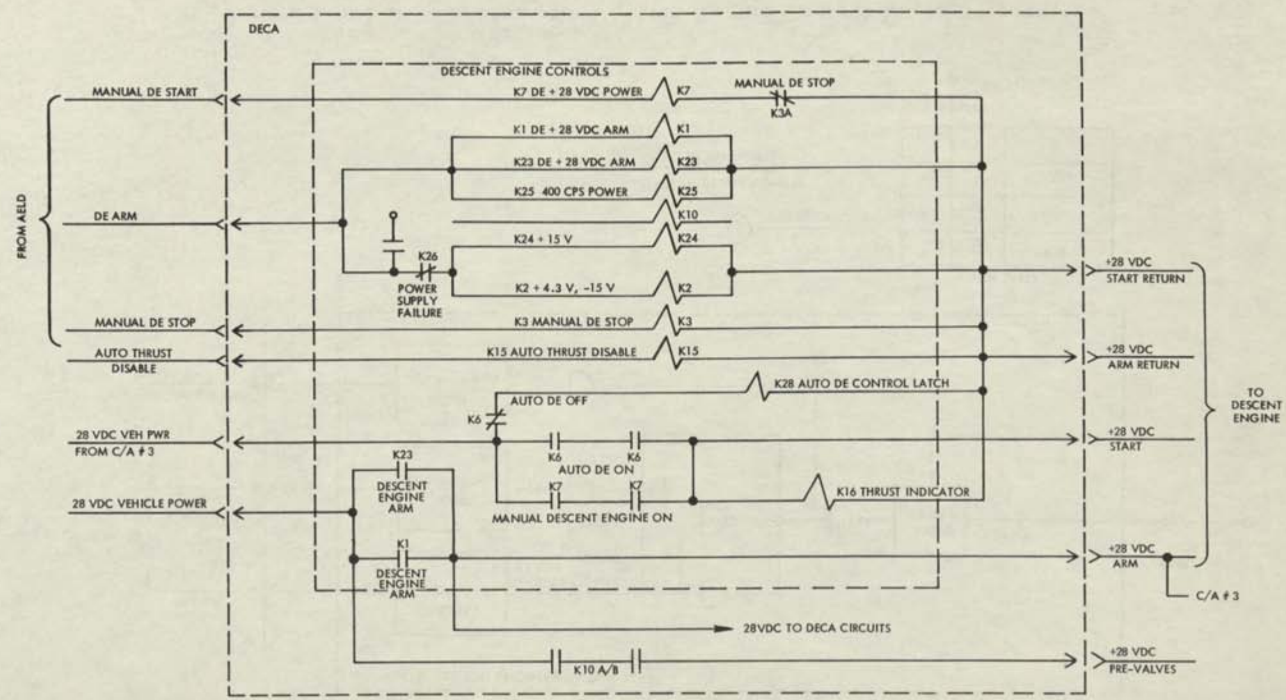
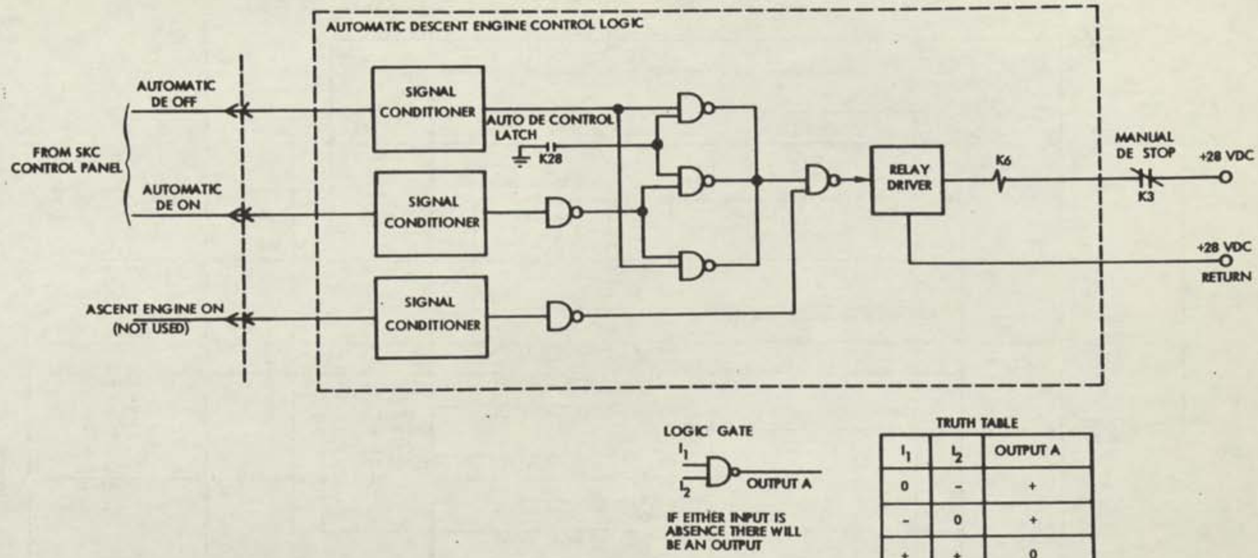


Figure 4. 3. 4. 2-5. DECA Power Switching and Engine Control



TRUTH TABLE

I_1	I_2	OUTPUT A
0	-	+
-	0	+
+	+	0

0 = 0 VDC
+ = 4 VDC
- = DON'T CARE 0 OR 4

Figure 4. 3. 4. 2-6. Automatic Descent Engine Control Logic Diagram

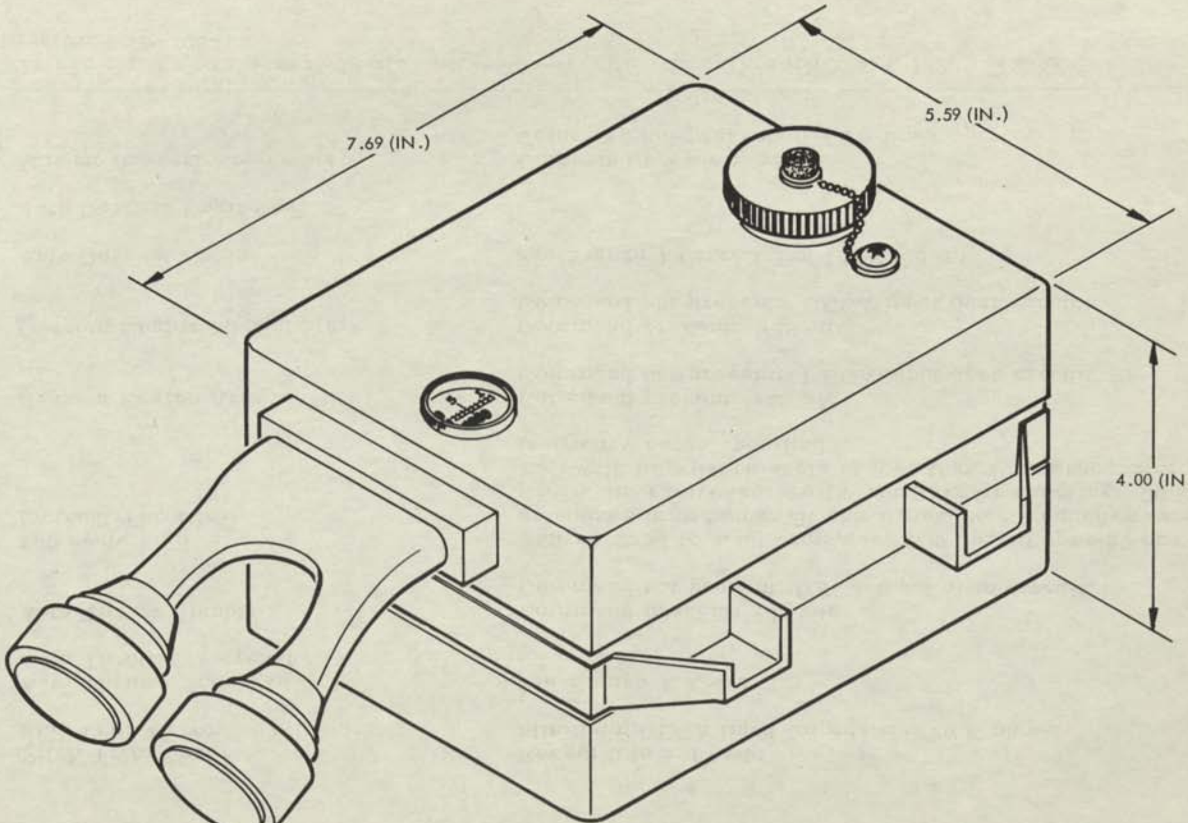


Figure 4. 3. 4. 2-7. DECA Outline

Table 4. 3. 4. 2-3. DECA Input Signal Characteristics

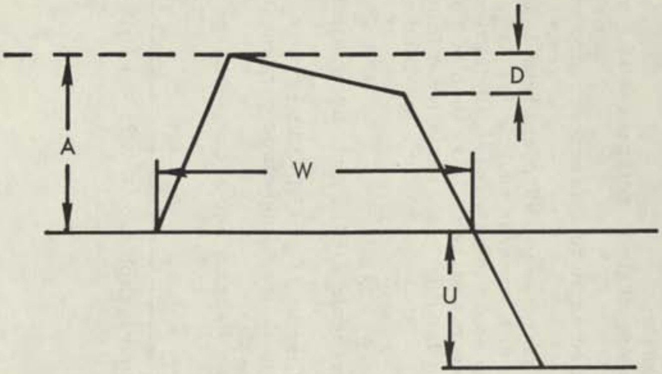
(1) (2)	Pitch Trim Error Roll Trim Error	Range: 0 to ± 13 vdc Minimum DECA input impedance: 20 K ohms.
(3) (4)	Auto Throttle Increase Auto Throttle Decrease	See Figure 4. 3. 4. 2-8
(5)	Auto Thrust Disable	Command present: +28 vdc* Command not present: DECA input open circuit.
(6)	800 cycle high (Manual Throttle)	Range: 0. 54 to 3. 00 vrms, ac, 800 cps +8 cps -80 cps ac source impedance: 2K ohms max over frequency range DECA input impedance with transformer coupling: 20 K ± 5% with impedance angle of less than 10 degrees over the frequency range specified
(7)	Descent Engine Arm	Command present: +28 vdc* Command not present: DECA input open circuit
(8)	Descent Engine Manual Start	Command present: +28 vdc* Command not present: DECA input open circuit
(9)	Auto Descent Engine On	See Figure 1 Circuit 3 of LSP-300-13
(10)	Auto Descent Engine Off	
(11)	Manual Descent Engine Stop	Command present: +28 vdc* Command not present: DECA input open circuit

*All +28 vdc signals are power signals and are as specified in paragraph 3. 3. 4. 1. 2. 3 of GAEC Specification LSP-300-13.

Table 4.3.4.2-3. DECA Input Signal Characteristics (Continued)

(12)	Pitch Position	400 cycle ac signals:
(13)	Roll Position	Zero degrees phase angle to reference: 15 vrms to Zero vrms
		180 degrees phase angle to reference: zero vrms to 15 vrms
		Signal rate of change when 400-cycle trim loads are actuated: 0.5 ± 0.1 vrms/second
		Phase angle tolerance: ± 10 degrees
		Source resistance: Less than 500 ohms
		DECA loading: 10K resistive minimum
		Null voltage: 50 mv rms maximum
(14)	Analog Trim Mode	Command present: +28 vdc*
		Command not present: DECA input open circuit
(15)	Pitch - T_N Command	See Figure 1, Circuit #1 of LSP-300-13
(16)	Pitch - T_p Command	"1" = Source impedance <3000 ohms current (maximum) = 5 miliamperes
(17)	Roll - T_N Command	"0" = Source impedance > 500 K ohms voltage (maximum) = 40 vdc
(18)	Roll - T_p Command	

*All +28 vdc signals are power signals and are as specified in paragraph 3.3.4.1.2.3 of GAEC Specification LSP-300-13.



Repetition Rate 3200 PPS

Source Impedance - 100 Ohms

1. Amplitude A - 3.5 ± 1.5 volts
2. Pulse Width W - 1 to 5 usec.
3. Undershoot U - Less than 4 volts
4. Droop - D - Less than 20% of A.
5. Signal plus noise during transmission of pulse train shall remain within the range of amplitudes defined.

   Noise during absence of pulse train - ripple or spikes shall be less than 0.2 volts peak to peak.
6. Loading - 1000 ohms ±10%

Figure 4.3.4.2-8. DECA Automatic Throttle Input Signal Description

Table 4.3.4.2-4. DECA Input Signal Characteristics

	Signal	Description
	(1) Auto Thrust	Range: 0 to +12 vdc. DECA output impedance: 1K ohm maximum Load impedance: 22K ohms minimum Simulation of all noise: 15 mv maximum
	(2) Manual Thrust	Range: 2.6 to 14.6 vdc. Load impedance: 22K ohms minimum DECA output impedance: 1K ohm Summation of all noise: 15 mv maximum
	(3) Pitch Trim Fail (4) Roll Trim Fail	Output is relay closure through 2000 ohm current limiting resistor. For details see Figure #1, Circuit #2 of LSP 300-13 "1" = 28 ± 11 vdc "0" = 0 ± 2 vdc
	(5) Engine Thrust Verification (6) Descent Engine Thrusting	DECA supplies a relay closure for an externally supplied voltage. Switching requirements: Current less than 100 ma 35 vdc maximum
INSTRUMENTATION	(7) Pitch Trim Fail (8) Roll Trim Fail	DECA provides a switch closure for an externally supplied voltage. Switch requirements: Current: less than 10 ma 35 vdc maximum
INSTRUMENTATION	(9) Auto Thrust Disp. (10) Manual Thrust Disp.	Load impedance - 50 K minimum DECA isolation resistance = 10K ± 1%. Those loads in parallel with items (1) or (2) above, as applicable.
INSTRUMENTATION	(11) Descent Engine Arm. Relay Closure K1 and K23	See Figure #1, Circuit #4 of LSP-300-13
INSTRUMENTATION	(12) Pitch GDA Position (13) Roll GDA Position	Modulated 400 cycles from GDA Transducer to DECA. External loading 250K. DECA isolation = 5K ± 1%
INSTRUMENTATION	(14) 115 vrms High (15) 115 vrms Low	External loading: 125K DECA isolation: 10K ± 5%

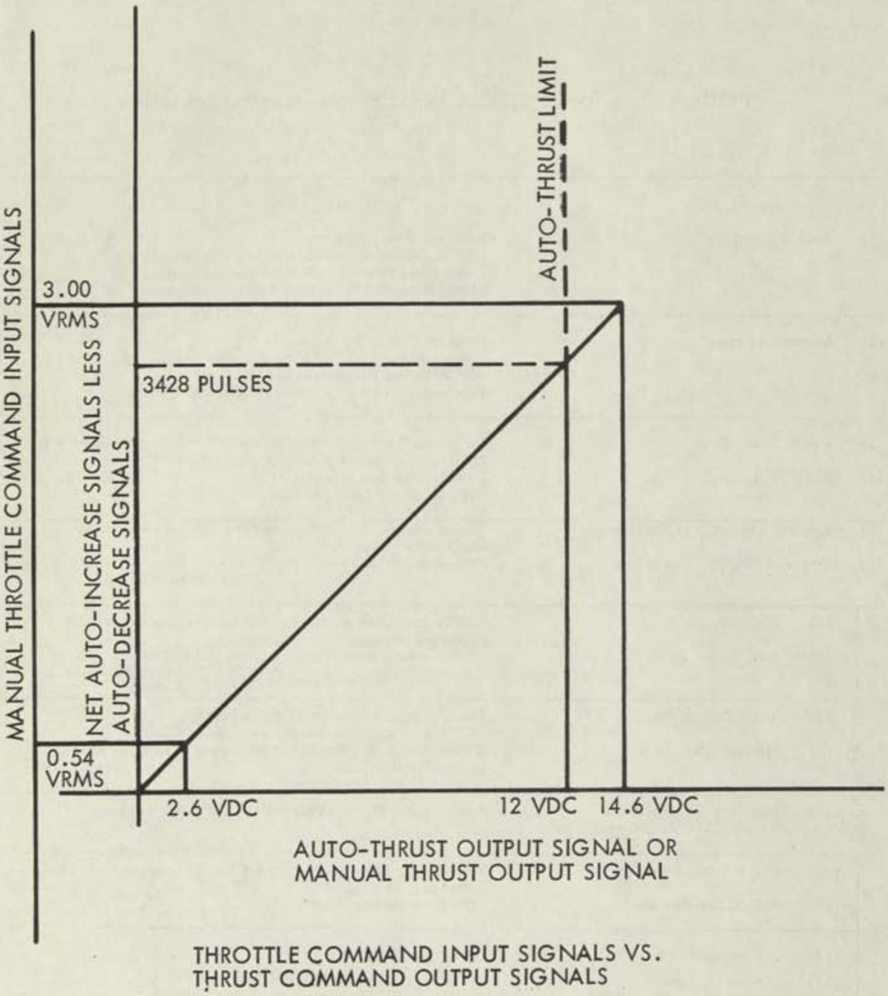


Figure 4.3.4.2-9. DECA Thrust Output Signal

4.3.4.3 Rate Gyro Assembly (LSP-300-11)

A brief description of RGA Component Identification, Component Description, Function, Monitoring and Self Test Requirements, Electrical Interface, and Outline Drawing is presented in the following paragraphs.

4.3.4.3.1 Component Identification. The RGA consists of the following principal components:

a) Three rate gyros

b) One assembly block

c) One electrical connector

4.3.4.3.2 Component Description. The rate gyro assembly consists of three sub-minature single-degree-of-freedom rate gyros mounted so as to sense vehicle roll, pitch, and yaw rates, respectively. Each gyro is capable of measuring input rates up to ±25 deg/sec and contains self-test features consisting of a spin motor rotation detector and a self-test torquer. Each rate gyro senses a rate of turn about its input axis, that is perpendicular to both the spin and output axis. In response to this input, the gyro precesses about its precession or output axis. Accordingly, the gimbal rotates against the torque restraint (torsion bar) until the input torque is equal to the restraining torque. Hence, the balanced gimbal position at this point is a measure of rate of turn. An electrical analog output (rate of turn) is obtained from the microsyn pickoff which is positioned by the precession of the gimbal.

Three-phase 26-volt, 800-cps power, generated in the ATCA, is applied to the spin motor of each gyro. The flywheels run at synchronous speed of 24,000 rpm. Pickoff excitation of 28 v 800 cps, generated in the ATCA, is applied to the primary of microsyn pickoff T1. The secondary of T1 will pickoff an 800 cps voltage proportional to the precession of the gyro and hence proportional to vehicle rate about the gyro input axis. An in phase pickoff voltage indicates a positive rate and a 180 degree phase pickoff voltage indicates a negative rate. The pickoff voltage is sent to the rate meters for display and to the ATCA for rate damping in AGS guidance mode.

The gimbal torquer or self test torquer is used to apply a torque on the movable gimbal. When 28 vdc is applied to winding L1 - L2 the secondary of T1 picks off a voltage equivalent to 5 degrees per second rotation. The direction of torquing is reversible by changing the polarity of the dc voltage.

Each gyro has two permanent magnets attached to the flywheel. These magnets induce pulses into winding L1 - L2. At synchronous speed the pulse frequency is 1, 600 pulses per second. See figure 4. 3. 4. 3-1 for a simplified drawing of a rate gyro.

4. 3. 4. 3. 2. 1 Rate Gyro. The linear input range for each gyro is ±25 deg/sec. Stop settings may vary from 26 to 33 deg/sec at 800 cps, three-phase power. Nominal gyro gain is 140 mv/deg/sec.

4. 3. 4. 3. 2. 2 Assembly Block. The assembly block provides a structural mounting and a heat-sink surface.

4. 3. 4. 3. 3 Function. The RGA senses angular velocity about each of the mutually orthogonal vehicle axes. Gyro output voltages are connected to the ATCA input summing amplifiers and the FDAI's. (See Figure 4. 3. 4. 3-2 for RGA schematic.) Specific functions and gyro limitations are described in the following paragraphs.

4. 3. 4. 3. 3. 1 Threshold. The minimum input that will cause a detectable change in output shall not exceed 0. 01 deg/sec.

4. 3. 4. 3. 3. 2 Resolution. The minimum change in input that will cause a detectable change in output (for inputs greater than threshold) shall not exceed 0. 01 deg/sec.

4. 3. 4. 3. 3. 3 Hysteresis. One-half of the algebraic difference of the two outputs obtained, when zero input is approached from each end of the maximum range, shall not exceed 0. 02 deg/sec.

4. 3. 4. 3. 3. 4 Linearity. The maximum output deviation from the best straight line (fitted by method of least squares) through Input-output data shall not exceed 0. 5 percent of full scale range up to one-half scale and shall not exceed 2. 0 percent of the full scale range from half to full scale and shall not exceed 5 percent of the full scale range between full scale and the stops.

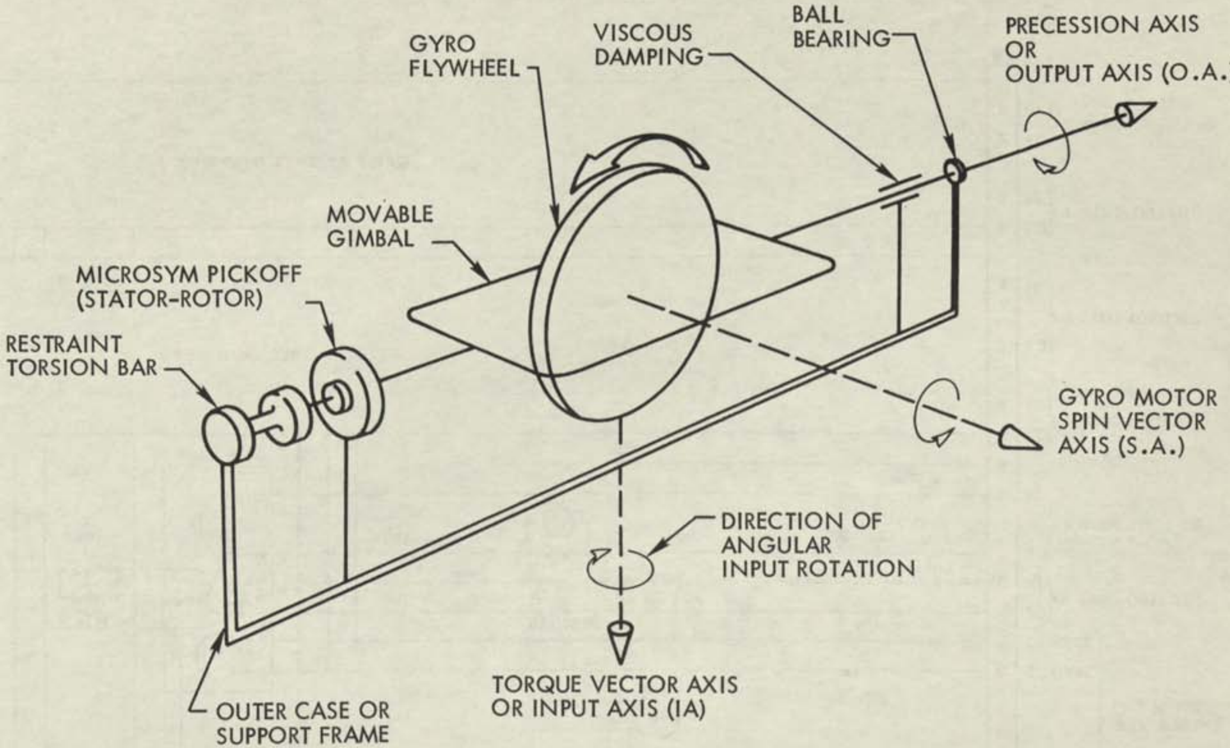


Figure 4. 3. 4. 3-1. Rate Gyro-Simplified Drawing

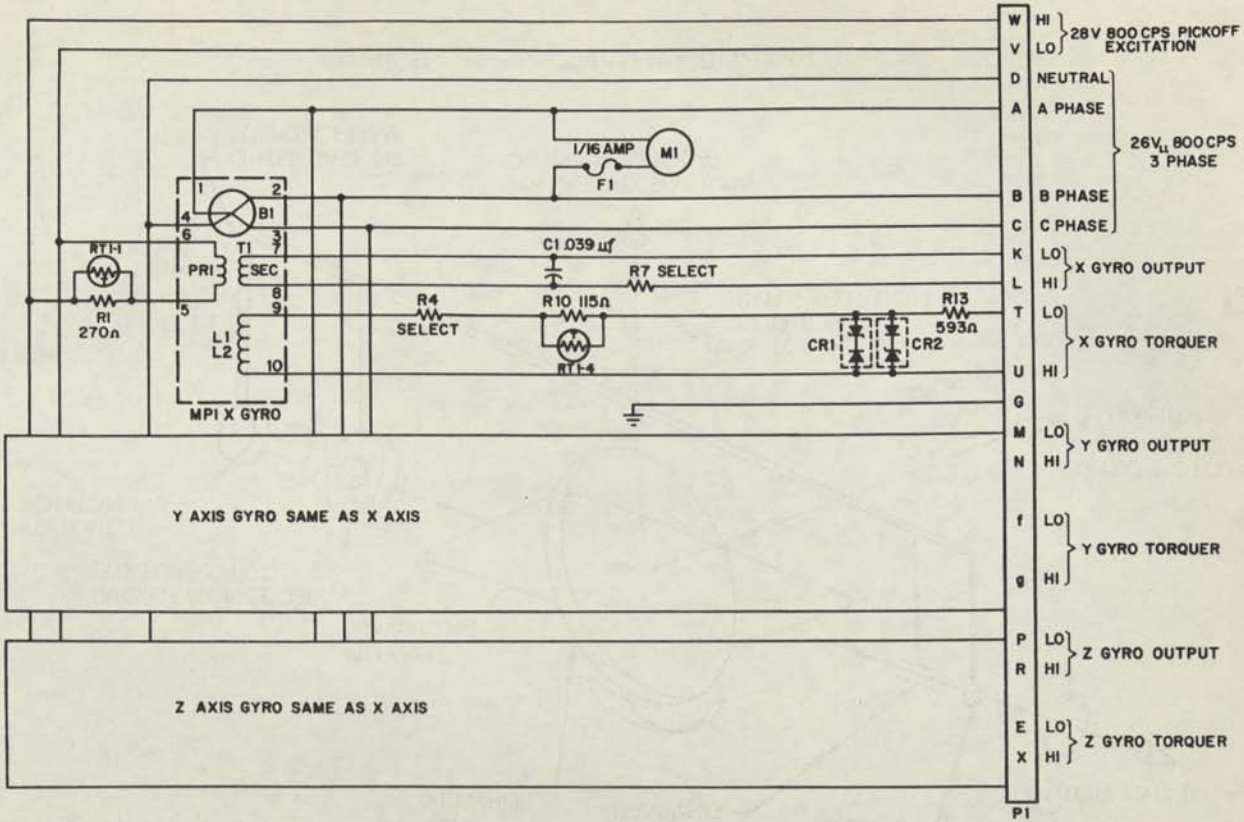


Figure 4.3.4.3-2. Rate Gyro Assembly Schematic

4.3.4.3.3.5 Zero Offset. The average of the two outputs obtained at zero input when zero input is approached from each end of the maximum range shall not exceed the value indicated by the following expression.

$$|Z.0| = 0.04 + 0.0006 \; |T_s - 75| \; + 0.0001 \; |T_a - 75|$$

where

$|Z.0|$ = absolute value of zero offset

T_s = temperature of heat sink surface in degrees Fahrenheit

T_a = ambient temperature in degrees Fahrenheit

4.3.4.3.3.6 Linear Acceleration Sensitivity. Gyro output sensitivity to linear accelerations, applied in any direction relative to the gyro case, shall not exceed 0.050 deg/sec/g.

4.3.4.3.3.7 Angular Acceleration Sensitivity. Gyro output sensitivity to angular acceleration about the output axis shall not exceed 0.05 deg/sec/rad/sec^2.

4.3.4.3.3.8 Damping Ratio. The ratio of actual damping to critical damping shall be within the limits indicated in Figure 4.3.4.3-3.

4.3.4.3.3.9 Natural Frequency. The input rate frequency at which the gyro output signal lags the sinusoidal rate input by 90 degrees shall be 20 ± 4 cps.

4.3.4.3.3.10 Cross Axis Coupling. Gyro construction and mounting alignment (with respect to the package mounting reference) shall be such that cross-axis coupling sensitivity does not exceed 0.005 deg/sec/deg/sec for input rates less than 1 deg/sec. For input rates up to 10 deg/sec (causing corresponding gimbal deflection) cross-axis coupling sensitivity shall not exceed 0.025 deg/sec/deg/sec.

4.3.4.3.3.11 Scale Factor. The slope of the best straight line (fitted by the method of least squares) through Input-output data shall be 140 ± 5 mv/deg/sec (at 800 cps excitation frequency and load impedance of 10 K ohms).

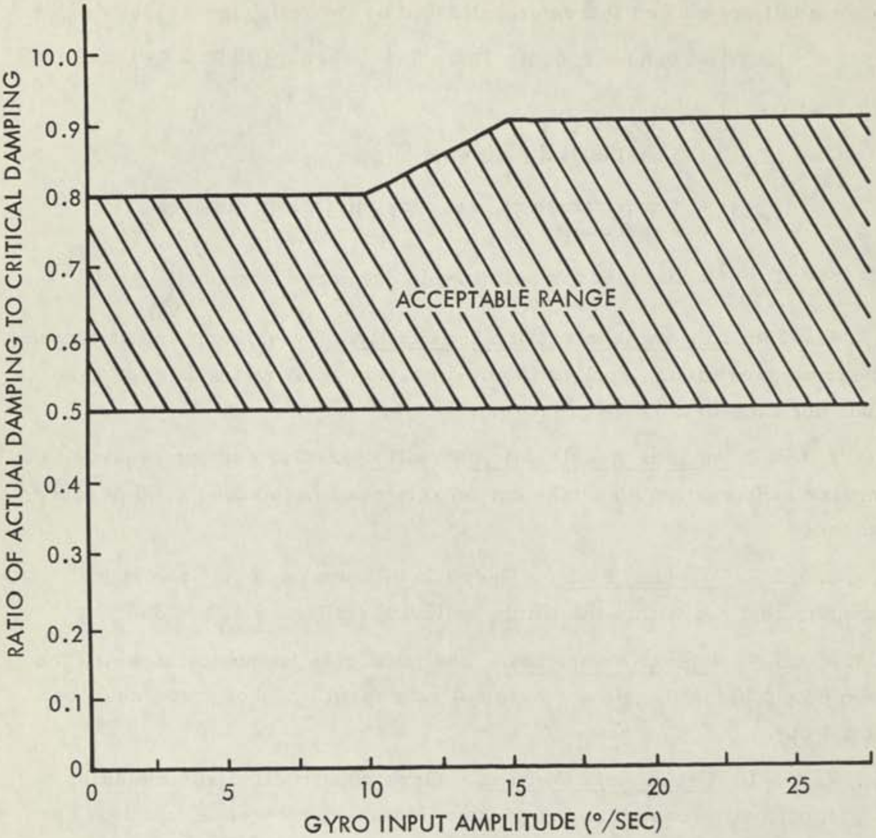


Figure 4. 3. 4. 3-3. Gyro Damping Ratio versus Output Amplitude

4.3.4.3.3.12 Synchronization Time. The time required for the gyro spin motor to accelerate from zero to synchronous speed shall not exceed 25 seconds.

4.3.4.3.3.13 Maximum Input Rate. Gyros shall withstand without damage damage, input rates up to 600 deg/sec.

4.3.4.3.3.14 Gimbal Displacement. The nominal gimbal displacement (at 800 cps motor excitation) shall be 0.09 deg/deg/sec of input rate.

4.3.4.3.4 Monitoring and Self-test Requirements. The RGA incorporates monitoring and self-test features to facilitate ground handling and inflight tests, and to permit telemetry monitoring of performance during flight.

4.3.4.3.4.1 Self-test Torquer. Each gyro incorporates a self-test torquer. A 28-vdc excitation signal applied to the self-test torquer is equivalent to an input rate of +5 ±0.1 deg/sec, (at 800-cps motor excitation). Direction of torquing is reversible by changing polarity. Torquing may be accomplished simultaneously (i.e., all three gyros torqued at once) or individually.

4.3.4.3.5 Electrical Interface.

4.3.4.3.5.1 Input Signals. See Table 4.3.4.3-1 for input signal characteristics and Table 4.3.4.3-2 for input signal load requirements.

4.3.4.3.4.2 Output Signals. See Table 4.3.4.3-3 for a tabulation of RGA angular rate output signal requirements.

4.3.4.3.6 Outline Drawing. The outline of the rate gyro assembly is shown in Figure 4.3.4.3-4. The total weight of the RGA does not exceed 2 pounds.

Table 4.3.4.3-1. Input Signal Characteristics

Input Signal	Self-Test Torquing Signal
Number of lines	6
Signal Characteristics	dc voltage
Normal Voltage	28 vdc
Steady State Regulation	31.5 vdc max. to 25 vdc min. No restriction on amplitude, time or frequency characteristics of voltage changes within regulation limits.
Transient Regulation	34 vdc max. to 20 vdc min. 95% recovery time for each excursion beyond steady state regulation limits 0.1 sec. Minimum interval between transient excursion 500 milliseconds for a period not exceeding 5 seconds.
Transient Spikes	+80 vdc max. to -52 vdc min. 95% recovery time for each excursion beyond steady state regulation limits 10 micro seconds. Minimum interval between transient spikes 50 milliseconds.
Ripple	Peak to peak amplitude of any ripple present will not cause instantaneous voltage to exceed steady state regulation limits. No restriction on ripple frequency.

Table 4.3.4.3-2. Input Signal Load Requirements

Input Signal	Self-test Torquing Signal
Input impedance to load (at RGA input terminal)	max. 1350 ohms min. 650 ohms
Open circuit dc voltage at load input terminal	0 ± 0.25 vdc*
Open circuit rms voltage at load input terminal	max. 0.50 vrms*
Transient spike peaks and noise generated by lead on input line	max. +0.25 volts peak** min. -0.25 volts peak**

*To be determined with an open circuit signal source.

**To be determined with a 1200 ohms passive resistive signal source.

Table 4.3.4.3-3. Output Signal Requirements

Output Signal	Angular Rate Signals
Number of Lines	6
Signal Characteristic	modulated 800 cps voltage vm sin (2π ft ± 10 deg)
Frequency of 800-cps Reference Voltage (f)	800 cps$^{+0}_{-80}$ cps
Phase Shift with Respect to 800 cps Reference Voltage	± 10 electrical deg
Nominal Range	0 to 3.5 vrms ($vra/\sqrt{2}$)
Maximum Range	0 to 4.8 vrms ($vm/\sqrt{2}$)
Phase (Positive Angular Rate Signals)	vm sin (2π ft ± 10 deg)
Phase (Negative Angular Rate Signals)	vm sin (2π ft ±180 ± 10 deg)
Quadrature Voltage	max 20 millivolts rms
Total Null (Quadrature, Harmonics, Noise)	max 30 millivolts rms
Load Impedance (Normal Conditions)	max: open circuit min: 10 K ohms
Load Impedance (Emergency Conditions)	max: open circuit min: zero

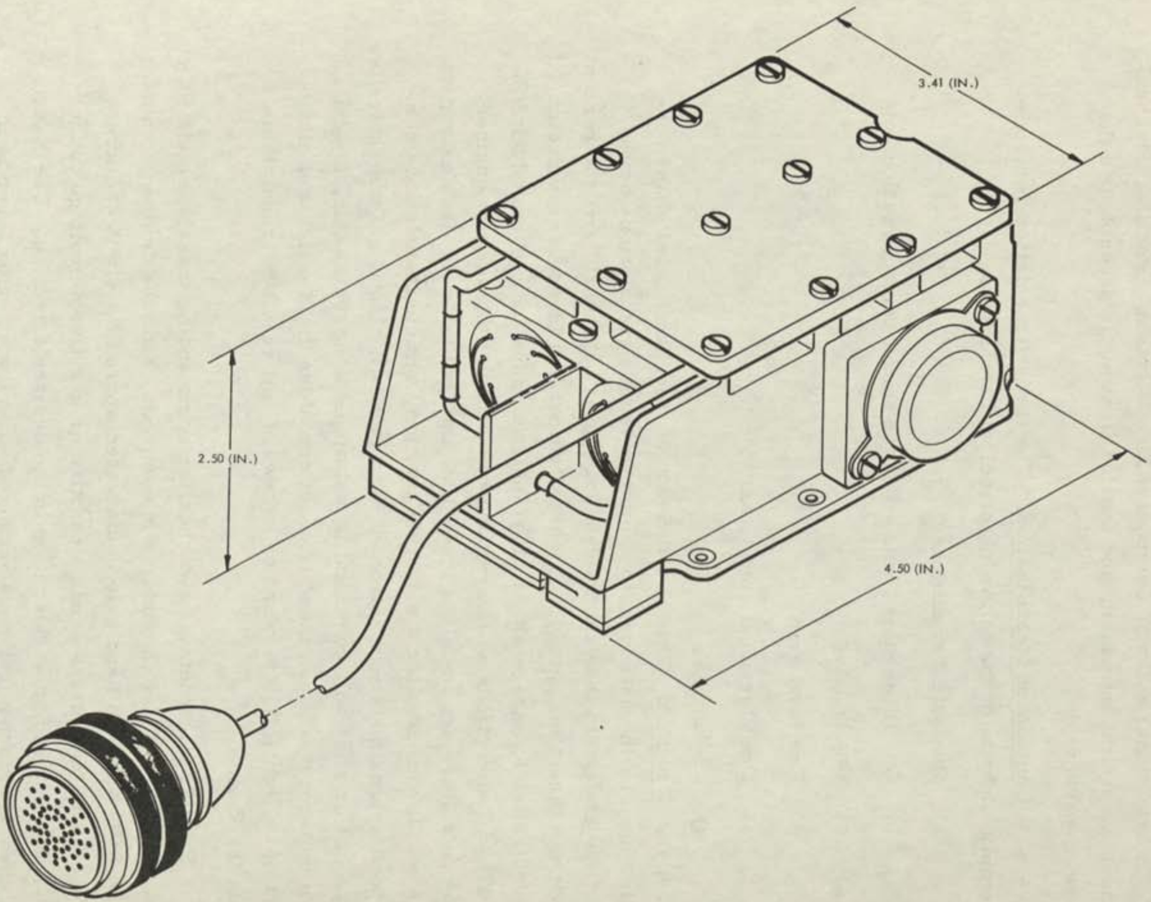


Figure 4. 3. 4. 3-4. Rate Gyro Assembly

4.3.4.4 Thrust/Translation Controller Assembly (LSP 300-180A)

A brief description of Component Identification, Function, Mechanization, Electrical Interface, and Outline Drawing is presented in the following paragraphs.

4.3.4.4.1 Component Identification. The thrust/translation controller assembly consists of the following principal components:

a) One output transducer

b) One three-axis translation and thrust control mechanism

c) One housing

d) One hand grip

e) One electrical connector

f) 13 switches

4.3.4.4.2 Function. This is a three-axis, tee-handle, left-hand controller used by the astronaut to command vehicle translations by the reaction jets and to throttle the descent engine between 10 and 100 percent maximum thrust magnitude. A manually operated lever is provided on the controller to engage either all translation capability or translation in the Y and Z directions with throttling capability of the descent engine. Two controllers are installed in the LM, one available for each astronaut. As a result of controller's mounting position, vehicle translations correspond approximately to astronaut hand motions. Motion of the controller to the left or right will command translations along the Y-axis, up and down movement will command translations along the X-axis, and movement of the controller forward or backward will result in translations along the Z-axis.

Each T/TCA contains switches for commanding translation via RCS jets. One pair of switch contacts in each axis sends the desired command to the LGC when the hand grip is displaced approximately 0.25 inches. Another pair of contacts sends a positive or negative dc voltage to the ATCA to command an acceleration in the desired direction. The X-axis switches are mechanically disconnected when the selector lever is in the throttle position.

When the T/TCA is used for throttle control, 800-cps power is applied to the primary of the position transducer. Depending on the position of the manual throttle switch, either the Commander or Systems Engineer will be able to generate manual throttle commands. The secondary winding of the transducer will pick up a voltage depending on the X-axis displacement of the hand controller. This voltage is sent to the DECA where it is used for controlling the thrust of the descent engine.

4.3.4.4.3 Mechanization. The T/TCA is mechanized as described in the following paragraphs. See Figure 4.3.4.4-1 for wiring diagram.

4.3.4.4.3.1 Modes of Operation. The modes of operation are:

a) Translation Mode - Motion of the handgrip sufficient to cause detent switch closure applies a low-level voltage to command translation along the vehicle axis corresponding to the direction of the motion of the handgrip. Translation along the X-, Y-, and Z-axis may be commanded independently or in combinations.

b) Throttle Control Mode - In this mode, handgrip motion along the X-axis provides a proportional 800-cps output for descent engine throttle control. The other two axes remain operable as in the translation mode. The throttle control (X-axis) has an adjustable friction device that can be controlled by the operator.

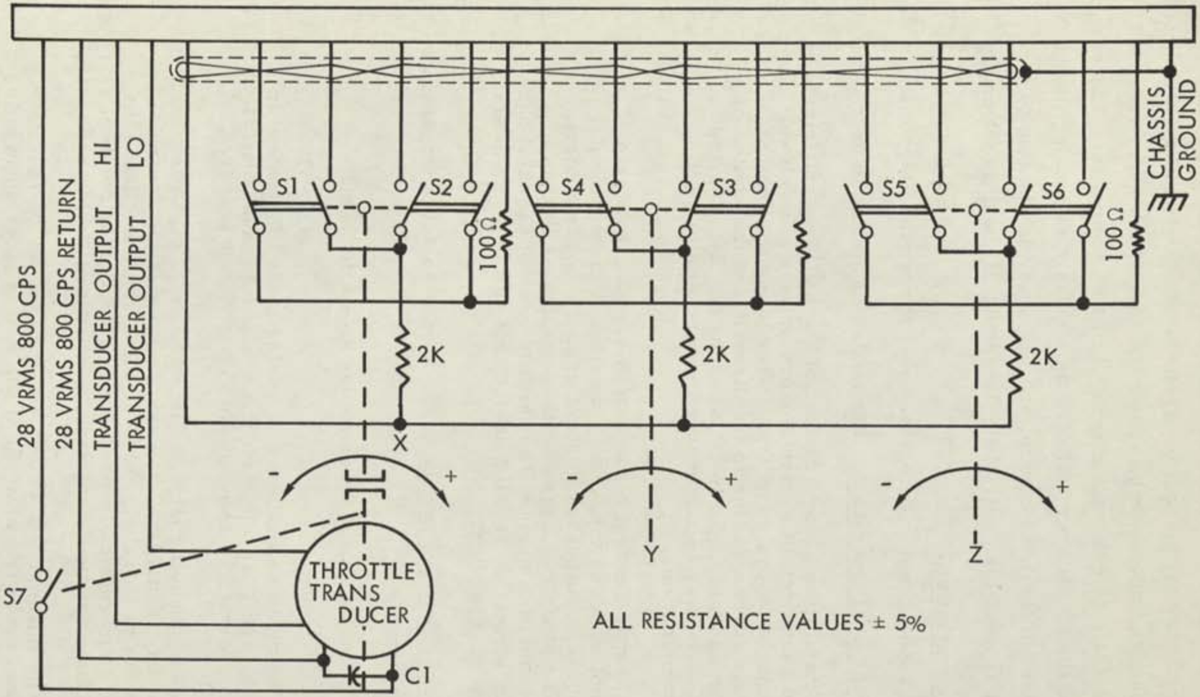
4.3.4.4.3.2 Mode Select Function. The modes select lever provides the following capability:

a) The mode select lever moves along the vehicle X-axis.

b) The mode select lever is spring loaded (detented) in a locked position in each mode and a visual indication is provided to show in which position it is locked.

c) Mode changing is possible with the T/TCA positioned at any thrust level. When the translation control mode is selected, the controller shaft and handgrip are automatically positioned to the center position. There are no detent switch actuations in the transition.

d) The mode select switch actuates within the last 25 percent of travel of the mode select lever when going from the detented translation control mode position to the detented throttle control mode position.



DETENT SWITCHES - S1 THRU S6
MODE SWITCH S7 (SHOWN IN TRANSLATION CONTROL MODE)
C1 - SELECT FOR POWER FACTOR CORRECTION

Figure 4. 3. 4. 4-1. Wiring Diagram - Thrust/Translation Control Assembly

4.3.4.4.3.3 Outputs. As the displacement of the handgrip is increased in either direction along the X-, Y-, or Z-axis away from its center position, the following events occur in the sequence shown:

a) Translation Control Mode

1) A detent switch closes providing an external signal to the vehicle control system

2) The hard stop is contacted

3) As the displacement of the handgrip is decreased, the events occur in the reverse order.

4) The T/TCA is capable of providing individual commands in all three axes or as many as three commands simultaneously.

b) Throttle Control Mode

1) Motions in the Y- and Z-axis result in events identical to those outlined in (1) through (4) above.

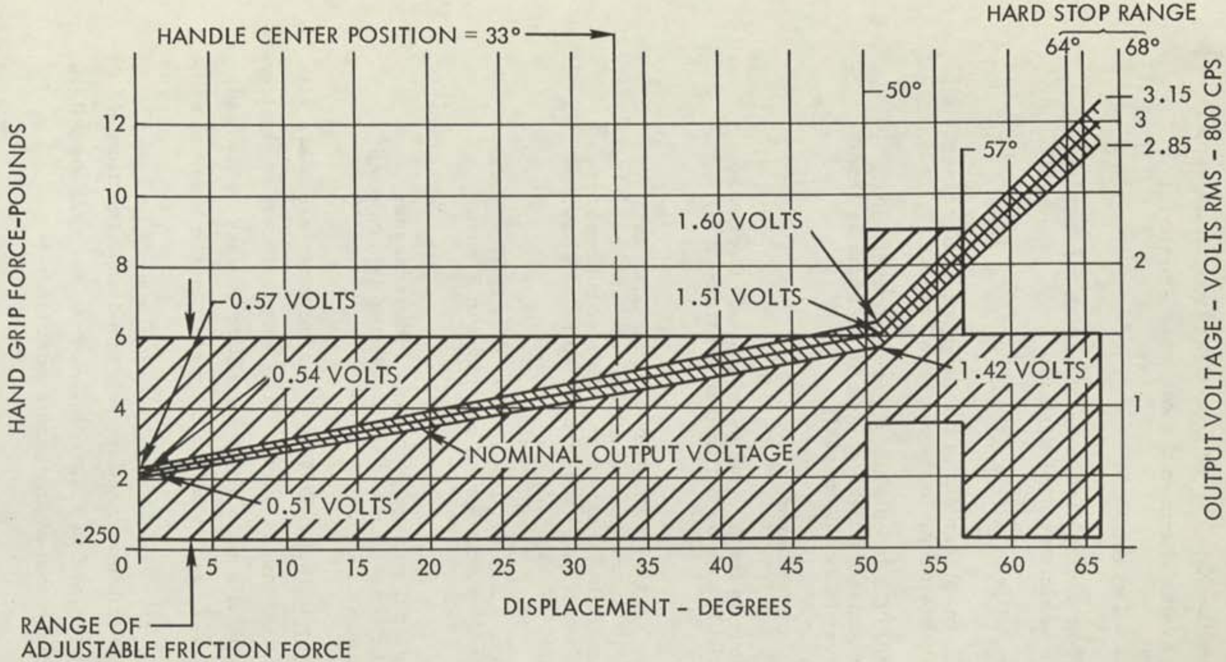
2) In the X-axis or throttle-control axis, an 800-cps sinusoidal output appears at the output of the X-axis transducer and increases with increasing handgrip position, as shown in Figure 4.3.4.4-2.

3) A force spike occurs between 50 and 57 degrees of handle displacement as shown in Figure 4.3.4.4-2.

4) The output voltage increases with increasing handgrip position and the hard stop is contacted as shown in Figure 4.3.4.4-2.

4.3.4.4.3.4 Transducer Output. Voltage measurements are made with a 20 K ohm load on the transducer output. The output voltage of the transducer is a linear function of handle displacement as defined by the end points in Figure 4.3.4.4-2. and is within ±5 percent of the nominal value of any handle position.

The output voltage of the transducer is a non-linear function of handle displacement as defined in Figure 4.3.4.4-2. and shall be within ±5 percent of the nominal value at any handle position.



NOTES:

1. THE MINIMUM JUMP IN FORCE WHEN GOING THRU THE 50° TO 57° POSITION OF THE HAND GRIP IN EITHER DIRECTION SHALL BE 3 POUNDS.
2. THE OUTPUT VOLTAGE SHALL BE WITHIN ±5% OF NOMINAL AT THAT POSITION, THE NOMINAL BEING DEFINED BY THE ENDPOINTS AND THE BREAKPOINT AT 51°.
3. THE RESOLUTION OF THE TRANSDUCER OUTPUT SHALL BE AT LEAST 0.8%.

Figure 4.3.4.4-2. Handgrip Force and Output Voltage versus Displacement

4.3.4.4.3.5 Handgrip Travel. The normal travel of the handgrip from the null or neutral position along or about the axes centered at the handle pivot point is as indicated in the performance characteristics of Table 4.3.4.4-1 and Figures 4.3.4.4-2, 4.3.4.4-3 and 4.3.4.4-4.

4.3.4.4.3.6 Forces and Displacements. Forces and displacements are measured at the intersection of the centerlines of the handgrip and the controller shaft. The displacements and forces are as shown in Table 4.3.4.4-1 and Figures 4.3.4.4-2, 4.3.4.4-3, and 4.3.4.4-4.

4.3.4.4.3.7 Frictional Hysteresis. When in the translation control mode, the friction forces on the handle do not prevent the handle from returning to its neutral position from any other position.

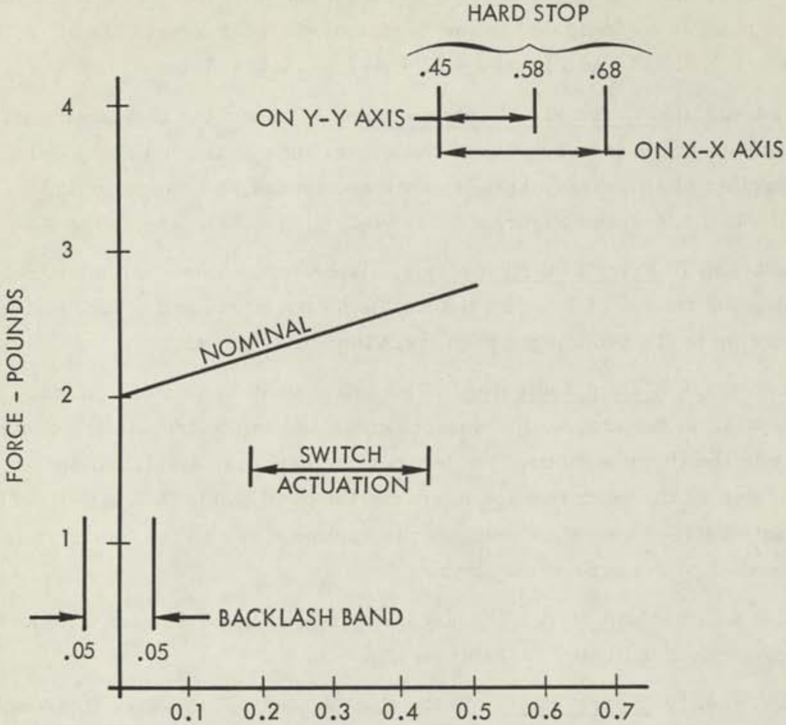
4.3.4.4.3.8 Detent Switching. The detent switches actuate in each axis, except X, in the translation control mode and in the translation control axes in the throttle control mode when the handgrip displacement as indicated in the performance characteristics of Table 4.3.4.4-1. The detent switches remain closed as the applied force to the handgrip is increased to the structural limit.

4.3.4.4.3.9 Hard Stop. The hard stop is contacted in each axis at the displacement indicated in Table 4.3.4.4-1.

4.3.4.4.3.10 Force Spike. In the throttle control mode, a force spike occurs between 50 and 57 degrees of handle displacement. The torque required to go through this position in either direction is shown in Figure 4.3.4.4-2.

4.3.4.4.3.11 Friction Adjustment. In the throttle control mode it is possible to adjust the friction level within the limits shown on Figure 4.3.4.4-2 while wearing an Apollo spacesuit glove. For a fixed friction level the static friction shall not exceed the running friction by more than 10 percent. It is not possible, under any conditions, to adjust the friction to a level that will cause seizure of the throttle handle.

4.3.4.4.3.12 Damping. Internal damping is such that unrestrained release of the handgrip from any translation control position does not result in overshoot of the center position of sufficient magnitude to actuate the opposite control switch.

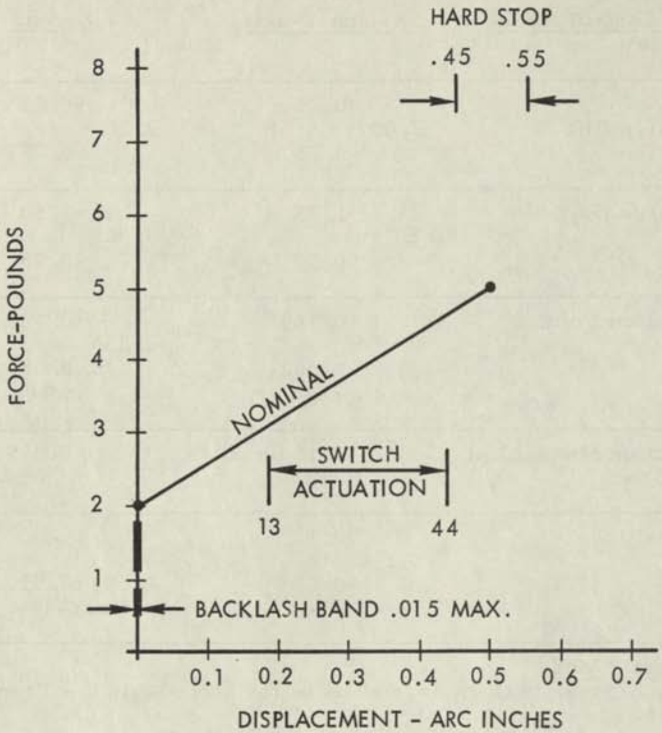


DISPLACEMENT - ARC INCHES
(AS MEASURED ON HAND GRIP 6" FROM HANDLE PIVOT POINT)

DISPLACEMENTS ARE ALONG ±X AXIS WHEN USED FOR TRANSLATION CONTROL AND ALONG THE ±Y AXIS WHEN USED IN TRANSLATION AND THROTTLE CONTROL MODES.

SWITCH "ON" ACTUATION SHALE OCCUR AT A MINIMUM OF 0.05 ARC INCHES BEFORE HARD STOP POSITION.

Figure 4.3.4.4-3. Force and Switch Actuation versus Displacement (X-Y)



DISPLACEMENTS ARE ALONG THE ±Z AXIS WHEN USED IN TRANSLATION AND THROTTLE CONTROL MODES.

SWITCH "ON" ACTUATION SHALL OCCUR AT A MINIMUM OF 0.05 INCHES BEFORE HARD STOP POSITION.

Figure 4.3.4.4-4. Force and Switch Actuation versus Displacement (Z)

Table 4. 3. 4. 4-1. T/TCA Performance Characteristics

Translation Control Mode (in each axis)	* X- and Y-axis	Z-Axis
Breakout Force (lb)	$2.00^{+0.75}_{-0.75}$ **	$2.00^{+0.75}_{-0.25}$
Hardstop Force (lb)	$2.75^{+1.75}_{-0.50}$ **	$5.00^{\pm 3.50}_{-0.75}$ **
Switch Actuation Point (On-off)	$0.250^{+0.188}_{-0.062}$ arc inches	$0.250^{+0.188}_{-0.062}$ inches
Neutral Position Mechanical Backlash Band	0. 10 (max) arc inches	0. 015 (max) inches
Hardstop Position	$0.5^{+0.18}_{-0.05}$ arc inches	0. 50 ±0. 05 inches

Throttle Control Mode
In the Y-and Z-axes, performance shall be the same as in the Translation Control Mode.

In the X-axis, the throttle control characteristics shall be as show in Figure 4. 3. 4. 4-2.

*As measured at a point on the control handle 6 inches from its pivot.

**Force limits include forces caused by switch actuation.

4.3.4.4.3.13 Dynamic and Static Balance. The T/TCA will not experience inadvertent switch closure or changes in throttle control output voltage caused by the environmental conditions specified.

4.3.4.4.3.14 Mechanical Backlash Band. The mechanical backlast is less than ±0.05 arc inch in each axis.

4.3.4.4.3.15 Pressure Transfer. A filter device is employed to permit air pressure transfer of at least 0.21 psi/sec.

4.3.4.4.4 Electrical Interface.

4.3.4.4.4.1 Input Signals. See Table 4.3.4.4-2 for electrical power requirements.

4.3.4.4.4.2 Output Signals. See Table 4.3.4.4-3 for output signal requirements.

4.3.4.4.5 Outline Drawing. See Figure 4.3.4.4-5 for an outline drawing of the T/TCA. The total weight of the T/TCA does not exceed 5.25 pounds.

4.3.4.5 Attitude Controller Assembly (LSP-300-19). A brief description of ACA component identification, function, mechanization, electrical interface, and outline is presented in the following paragraphs.

4.3.4.5.1 Component Identification. The attitude controller assembly consists of the following principal components:

a) Three linear output transducers

b) One three-axis rotation control mechanism

c) One hand grip

d) Forty-three switches

e) Two electrical connectors and cables

f) One housing

4.3.4.5.2 Function. Vehicle attitude changes can be commanded manually by the use of either one of the two three-axis, pistol-grip, right-hand attitude controllers. One attitude controller is used by each astronaut and is spring-restrained toward a center position. Each axis contains an 800-cps transducer and detent, pulse/direct, and manual override switches,

Table 4.3.4.4-2. Electrical Power Requirements

	1Ø Voltage Reference Input Normal Conditions, 800 $^{+0}_{-80}$ cps	1Ø Voltage Reference Input Abnormal Conditions, 800 $^{+0}_{-80}$ cps
Normal Voltage	28 v rms..	
Steady State Regulation	Max. 28.5 v rms. Min. 27.5 v rms.	Max. 30 v rms. Min. 0 v rms.
Transient Regulation	Max. 30 v rms Min. 26 v rms 95% recovery time 50-millisecond max. interval between transient regulation disturbances 500-millisecond max.	Max. 32 v rms. 95% recovery time 50-millisecond max. interval between transient regulation disturbances 50-millisecond max.
Transient Spikes	Max. 40 v peak Min. 16 v peak 95% recovery time for 10-microsecond interval (max.)	Max. 100 v peak Min. -100 v peak 95% recovery time 10 microsecond max. interval between transient spike disturbances 1 millisecond min.
Waveform	Nominal sine wave max. Total distertion 1%	Sine wave max. distortion: 10% Max. 400 cps Subharmonic: 10%
dc Component		Max. +28 v dc Min. -28 v dc

Table 4.3.4.4-3. Output Signal Requirements

No. of lines	21
Signal Characteristics	(a) 800 cps voltage Vm sin (2π ft) (b) Switch closures
Frequency (f), of 800 cps reference voltage	800 cps $^{+0}_{-80}$ cps
Nominal Range	Reference Test Plan
Phase (Throttle Command Signals)	Vm sin (2π ft)
Source Impedance of transducer	Less than 2000 ohms at 800 cps

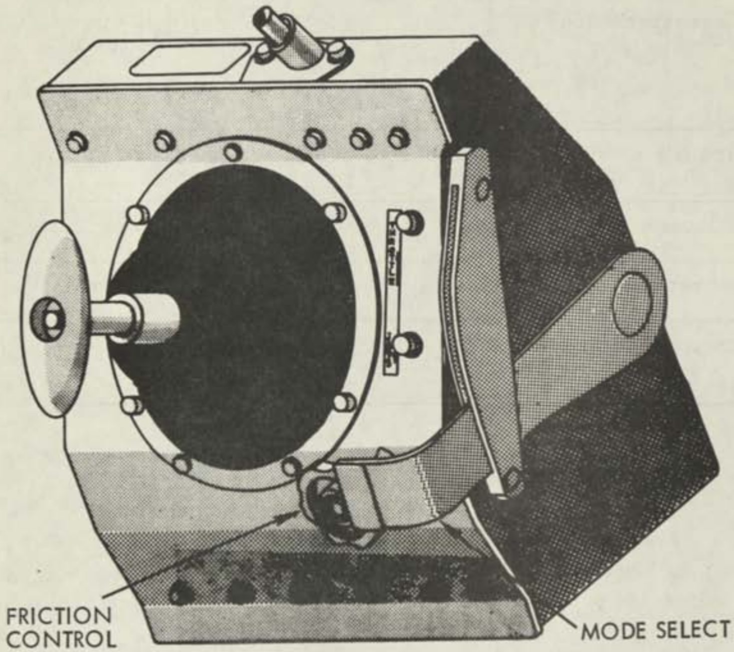


Figure 4. 3. 4. 4-5. T/TCA Outline Drawing

all of which are actuated by torque applied through the handgrip to perform the following functions:

a) Detent Switch - The purpose of the detent switch is to initiate follow-up in attitude-hold mode.

b) Pulse/Direct Switch - The purpose of the pulse/direct switches is to command a pulse train in the pulse mode, or two-jet direct operations in the direct mode.

c) Manual Override Switch - The purpose of the manual override switches is to command four-jet direct action in all modes.

d) Communications Switch - A press-to-talk trigger is mounted on the front near the top of the handgrip.

e) Transducers - The transducers provide 800-cps three-axis proportional rate commands for changes of vehicle attitude.

The attitude controller is installed with its X-axis approximately parallel to the LM X-axis so that vehicle rotations will correspond to astronaut hand movements. Clockwise or counterclockwise rotation of the controller handle about its X-axis will command vehicle yaw right or yaw left rotation about the X-axis. Forward or backward movement of the controller will command vehicle pitch-down or pitch-up rotation about the Y-axis. Movement of the controller to the left or right will command a roll-left or a roll-right rotation about the Z-axis. The ACA is used for closed loop proportional rate commands in attitude hold mode and open-loop acceleration commands in pulse, direct, and hardover operation.

Twenty-eight v 800 cps is applied to each of three proportional transducers. The secondary voltage of each transducer is proportional to hand grip displacement about its input axis, i.e., one transducer each of roll, pitch, and yaw axis. When hand grip displacement exceeds a nominal value of 1.25 degrees, the out-of-detent switch closes. One contact sends an out-of-detent signal to the LGC and the AGS. The other contact sends the proportional secondary voltage to either the LGC or the ATCA. The output voltage is 0.28 volts per degree hand displacement from 1 to 10 degrees displacement. Positive rotation commands will generate in phase voltage and negative commands 180 degrees out-of-phase voltage.

When pulse operation is selected in PGNCS operation, the LGC common, applied to a pair of ACA switches, is sent back to the LGC when the hand controller is displaced by more than 2.5 degrees (nominal). When pulse operation is selected in AGS operation for an axis, plus and minus dc voltages are applied to a pair of ACA switches. When the hand grip is displaced 2.5 degrees (nominal) or greater from neutral, either a positive or negative dc voltage is sent to the ATCA. A positive voltage commands a positive acceleration (pulsed) about the axis and a negative voltage commands a negative acceleration.

When direct mode is selected, +28 vdc is applied to two RCS secondary solenoids when the hand controller is displaced a minimum of 0.5 degree past the soft stops (10, +2, -0), the hardover switches will send +28 vdc to four RCS secondary solenoids.

The ACA is capable of providing proportional outputs in either one, two, or three axes, simultaneously. Input torques, motions, and outputs in any one axis are independent of any inputs to either or both of the other two axes.

4.3.4.5.3 Mechanization

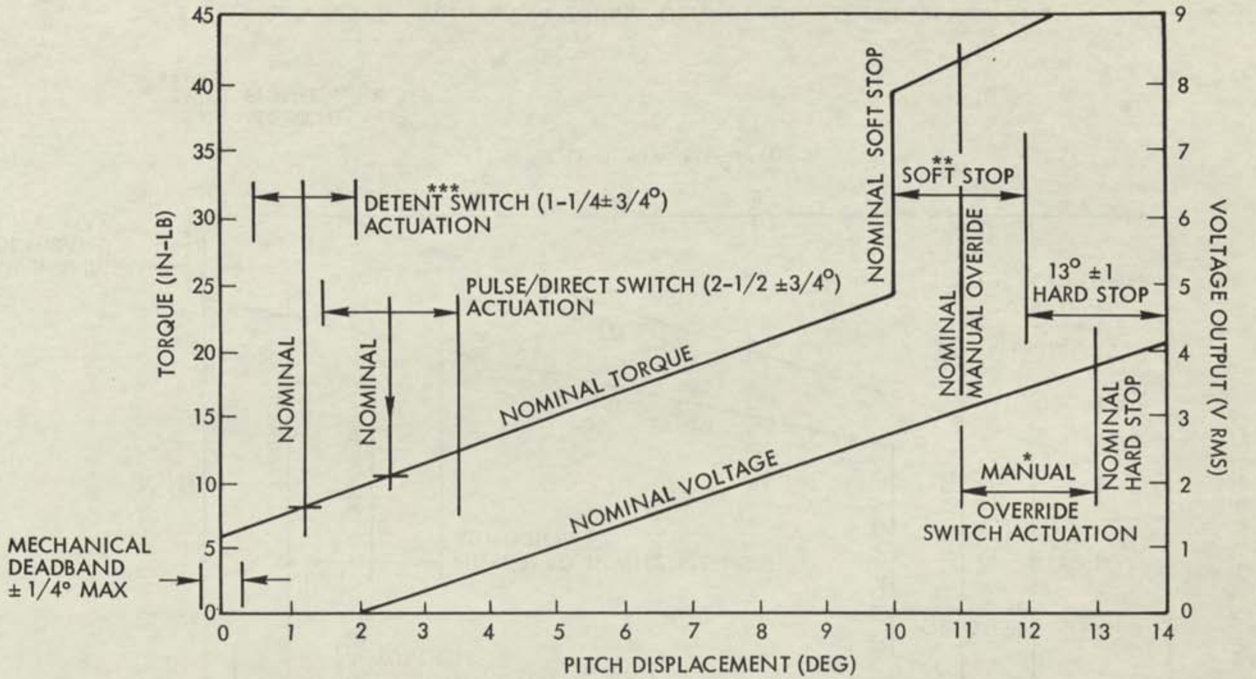
4.3.4.5.3.1 Torque Response. Application of an increasing torque to the hand grip in any direction will result in the following sequence of events.

a) Initial motion of the grip begins when the breakout torque level, as shown in Figures 4.3.4.5-1, 4.3.4.5-2, and 4.3.4.5-3 is exceeded. Mechanical dead zone is limited to ±1/4 degree or less.

b) The detent switch closes and remains closed at a nominal displacement of 1.25 degrees.

c) The transducer provides a linear electrical output signal in accordance with the voltage curves of Figures 4.3.4.5-1, 4.3.4.5-2, and 4.3.4.5-3 until the soft stop is reached. The signal is in-phase with the input voltage for positive handgrip displacement and out-of-phase with the input voltage for negative handgrip displacement.

d) At a nominal displacement of 2.5 degrees the pulse/direct switches close and remain closed.

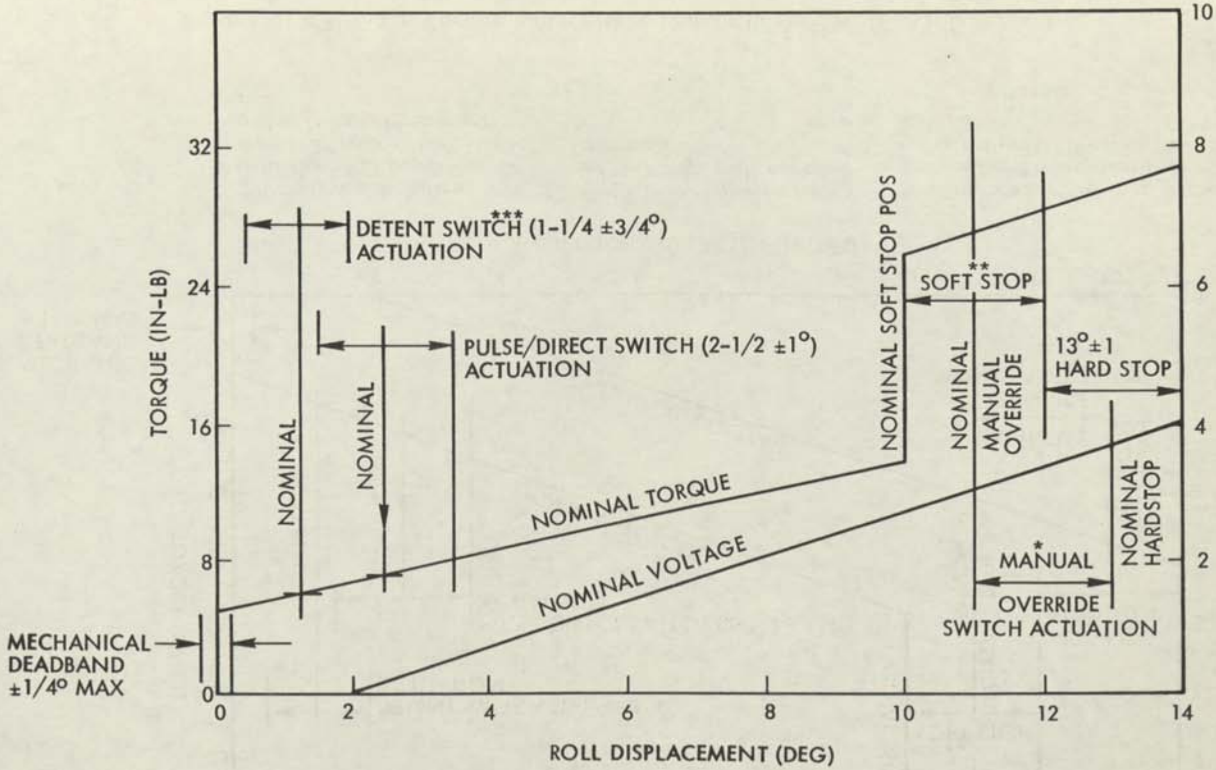


* Manual override switch actuation shall be no less than 1/2° prior to hard stop and at least 1/2° after soft stop

** Displacement between soft and hard stops shall not be less than 2°

*** Displacement between mechanical deadband and detent switch actuation shall not be less than 1/2°

Figure 4.3.4.5-1. Torque and Voltage Output versus Roll Displacement



*, **, *** } SEE NOTES FOR FIGURE 4.3.4.5-1

Figure 4.3.4.5-2. Torque and Voltage Output versus Roll Displacement

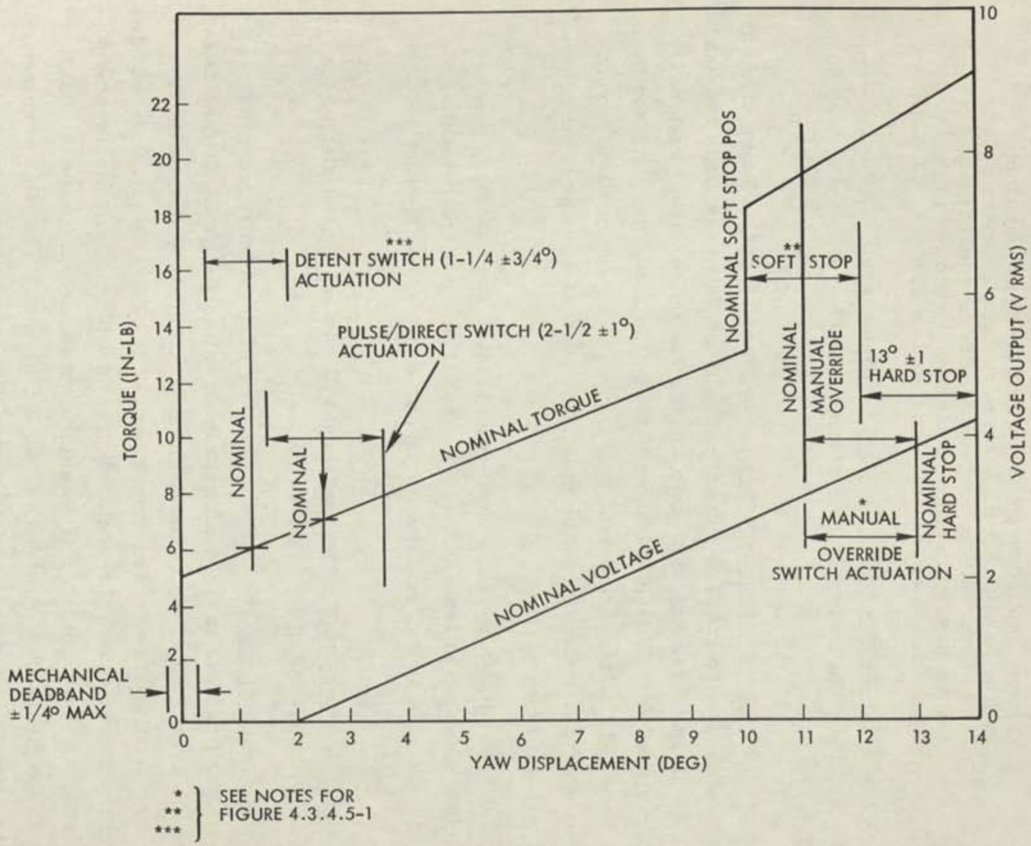


Figure 4.3.4.5-3. Torque and Voltage Output versus Yaw Displacement

e) At a point beyond the soft stop and prior to contracting the hard stop, the manual override switches are actuated.

f) The hard stop is contacted at the displacement specified in Table 4. 3. 4. 5-1. Decreasing the torque applied to the hand grip from its position at the hard stop will reverse the above switching and signal voltage sequence. The nominal travel of the hand grip from the center position in each direction, in each axis, is listed in Table 4. 3. 4. 5-1.

4. 3. 4. 5. 3. 2 Transducer Null Voltage. With no torque applied to the hand grip, the total voltage output of the transducer will not exceed 30 mv rms.

4. 3. 4. 5. 3. 3 Pulse/Direct Switches. The pulse direct switches close at a nominal displacement of 2. 5 degrees from the null position of the hand grip. With an increasing hand grip torque, the pulse/direct switches remain closed for all displacements between the actuation point and the hard stop. With decreasing hand grip torque from any position beyond the pulse/direct switch actuation point, the switchs open at 2. 5 ± 1. 0 degrees.

4. 3. 4. 5. 3. 4 Soft Stop. With increasing torque, the hand grip passes through a soft stop at least 0. 5 degree less than the displacement of the manual override switch actuation point but at a displacement of not less than 11 degrees.

4. 3. 4. 5. 3. 5 Hard Stop. The hand grip will contact a hard stop at the displacement specified in Table 4. 3. 4. 5-1.

4. 3. 4. 5. 3. 6 Damping. The internal damping of the ACA is such that unrestrained release of the hand grip from any initial position results in minimum overshoot of the center neutral position consistent with the following settling times. The time required for the grip to reach and stay within the detent switch threshold when released from any displacement within the soft stops in the roll axis does not exceed 1. 0 second, and in the pitch and yaw axes does not exceed 0. 5 second. The time required for the stick to reach and stay within the detent switch threshold when released from any other position does not exceed 1. 2 seconds in the roll axis, and shall not exceed 0. 6 second in the pitch and yaw axes.

Table 4.3.4.5-1. ACA Performance Characteristics*

Axis	Breakout Torque (in. -lb)	Lower Limit of Soft Stop Torque (in. -lb)	Maximum Upper Limit Of Soft Stop (in. -lb)	Minimum Step in Torque at Soft Step (in. -lb)	Torque at Nominal Hard Stop Position (in. -lb)
Roll	5.0 ± 1	14 ± 2.8	31.2	10	30 ± 6.0
Pitch	6.5 ± 1.3	23 ± 5.8	46.8	13	46 ± 9.2
Yaw	5.0 ± 1	13 ± 2.6	21.6	4.5	23 ± 4.6

In each axis:	1) Mechanical slop band	± 0.25 deg max
	2) Detent switch actuation	1.25 ± 0.75 deg
	3) Pulse/direct switch actuation	2.5 deg ± 1.0 deg
	4) Soft stop position	10 deg + 2 deg - 0 deg
	5) Transducer output at soft stop	2.8 volts + 0.14 - 0
	6) Manual override switch actuation	11 deg + 2 deg - 0 deg
	7) Hard stop position	13 deg ± 1 deg

*Reference, LSP 300 - 19

4.3.4.5.3.7 Manual Override Switching. With increasing torque, the manual override switches close at a hand grip displacement of not less than 0.5 degree after the soft stop, but not less than 0.5 degree prior to contacting the hard stop. With increasing hand grip torque, the manual override switches remain closed for all displacements between the actuation point and the hard stop. With decreasing hand grip torque from any point beyond the manual override switch actuation point, the switches open before the hand grip passes through the soft stop.

4.3.4.5.3.8 Switch Characteristics.

a) Detent Switches. Each of the six detent switches is a SPDT switch normally open, capable of switching 500 ma at 28 vdc (inductive load).

b) Pulse/Direct Switches. Each of the twelve pulse/direct switches is a SPDT switch, normally open, capable of switching 1 amp at 28 vdc (contact load R = 28.8 ± 0hm, L = 176 ± 18 mh, series connected

c) Manual Override Switches. Each of the 24 manual override switches is a SPDT switch, normally open, capable of switching 1 amp at 28-vdc (contact load R = 28.8 ± 1 ohm, L = 176 ± 18 mh, series connected).

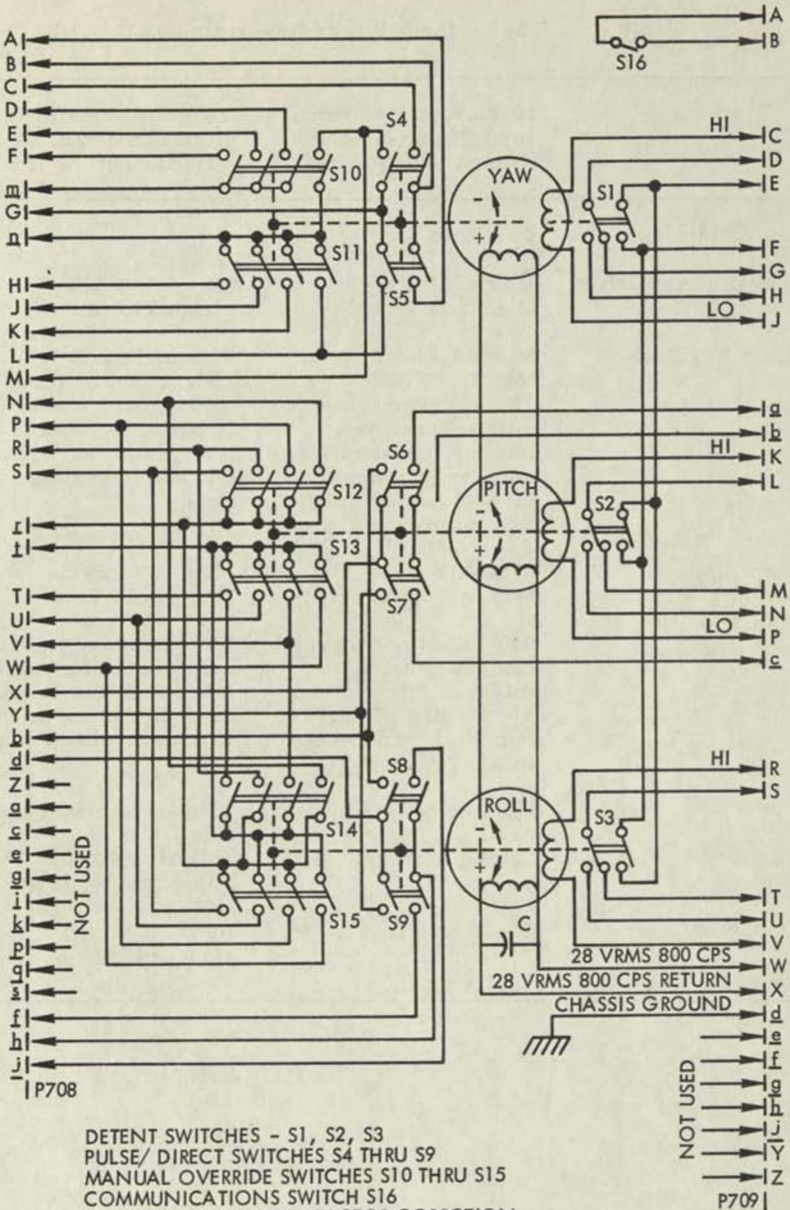
d) Communications Switch. The communications switch is a SPDT switch, normally open, capable of switching 500 ma at 28 vdc (resistive load)

4.3.4.5.4 Electrical Interface. See Figure 4.3.4.5-4 for a schematic diagram of the ACA.

4.3.4.5.4.1 Input Signals. See Table 4.3.4.5-2 for input power requirements.

4.3.4.5.4.2 Output Signals. See Table 4.3.4.5-3 for output signal requirements.

4.3.4.5.5 Outline Drawing. An outline drawing of the ACA with approximate dimensions and articulation is shown in Figure 4.3.4.5-5. The total weight of the ACA does not exceed 4.75 pounds including cables and connectors.



DETENT SWITCHES - S1, S2, S3
PULSE/ DIRECT SWITCHES S4 THRU S9
MANUAL OVERRIDE SWITCHES S10 THRU S15
COMMUNICATIONS SWITCH S16
C - SELECT FOR POWER FACTOR CORRCTION

Figure 4. 3. 4. 5-4. ACA Schematic

Table 4.3.4.5-2. Input Power Requirements

	1Ø Voltage Reference Input Normal Conditions, 800 +0 -80 cps	1Ø Voltage Reference Input Abnormal Conditions, 800 +0 -80 cps
Normal Voltage	28 VRMS	
Steady State Regulation	Max 28.5 VRMS Min 27.5 VRMS	Max 30 VRMS Min 0 VRMS
Transient Regulation	Max 30 VRMS Min 26 VRMS 95% recovery time 50 millisecond max. interval between transient regulation disturbances 500 millisecond max.	Max 32 VRMS 95% recovery time 50 millisecond max. Interval between transient regulation disturbances 50 millisecond max.
Transient Spikes	Max 40V Peak Min 16V Peak 95% recovery time 10 microsecond max. interval between transient spike disturbances 1 millisecond min.	Max 100V Peak Min -100V Peak 95% recovery time 10 microsecond max. interval between transient spike disturbances 1 millisecond min.
Waveform	Nominal sine wave max. total distortion 1%	Sine wave max distortion 10% Max 400 cps Subharmonic 10%
dc Component		Max. +28 vdc Min. -28 vdc

Table 4.3.4.5-3. Output Signal Requirements

Output Signals	Angular Rate Signals
No. of lines	52
Signal Characteristics	(a) 800 cps voltage vm sin (2 pi ft ± 10 deg) (b) Switch closures
Frequency of 800 cps reference voltage (f)	800 cps $^{+0}_{-80}$ cps
Phase Shift with respect to 800 cps reference voltage	± 10 electrical degrees
Nominal Range	0 to 2.8 vrms
Maximum Range	0 to 4.5 vrms
Phase (positive angular rate command signals)	Vm sin (2 pi ft ± 10 deg)
Phase (negative angular rate command signals)	Vm sin (2 pi ft +180 deg ± 10 deg)
In Phase (or out of phase) (between detent switch positions)	Max. 20 millivolts rms
Quadrature Voltage (between detent switch positions)	Max. 20 millivolts rms
Total Null between detent switch positions (in phase, quadrature, harmonics, noise)	Max. 30 millivolts rms
Source Impedance	Less than 2000 ohms
Load Impedance (normal)	10 K ± 10%
Load Impedance (emergency)	Max: open circuit Min: zero
Maximum Switch Contact Resistance	25 milliohms

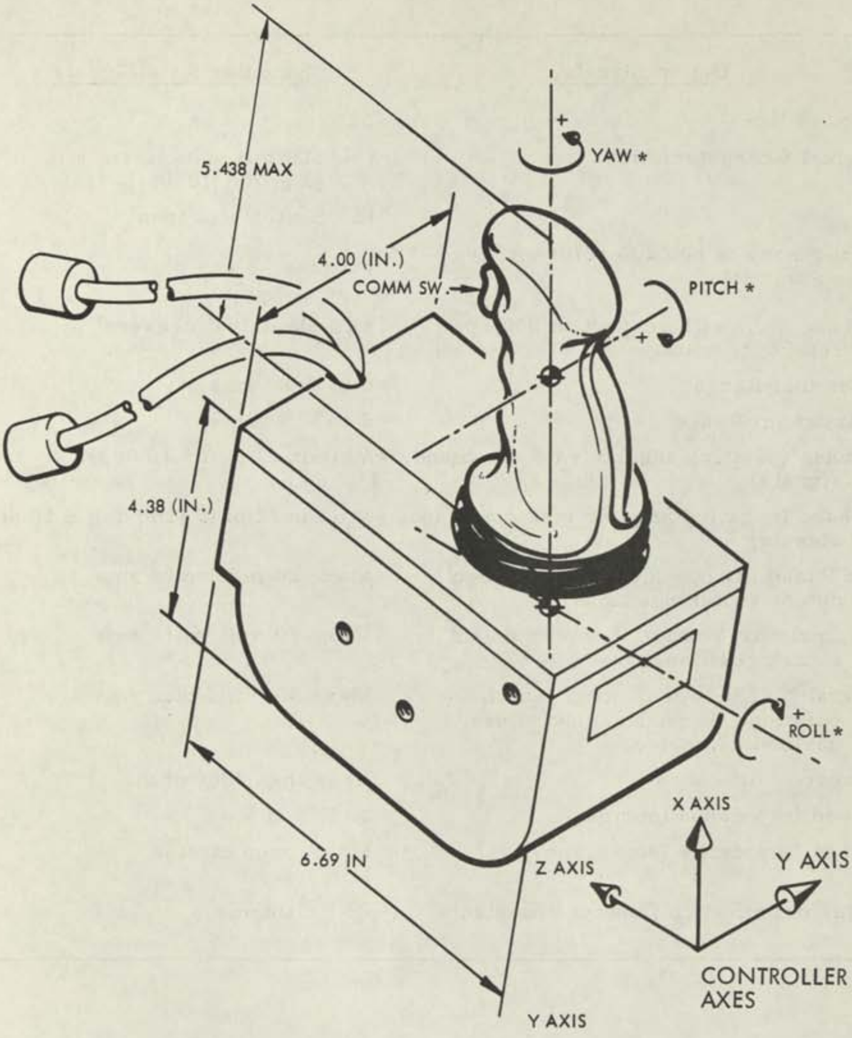


Figure 4.3.4.5-5. ACA Outline

4.3.4.6 Gimbal Drive Actuator Assembly (GDA) (LSP-300-17)

A brief description of GDA Component Identification, Function, Mechanization, and Outline Drawing is presented in the following paragraphs:

4.3.4.6.1 Component Identification. The GDA is composed of the following subassemblies:

a) A single-phase motor (single-phase, 400-cps input)

b) Gearing

c) Rod end fittings

d) Screwjack (linear and irreversible)

e) Housing

f) Mechanical stops

g) Induction potentiometer (for position and actuation data)

h) Electrical connector (all electrical connections are made through one 19-pin connector)

4.3.4.6.2 Function. The two GDA's are low speed electromechanical devices each containing a 115V 400 cps ac motor driving a screw jack and a position transducer. Under control of the DECA the GDA's rotate the descent engine about the pitch and roll axes so that the thrust vector goes through the LM center of gravity. One GDA controls the pitch gimbal and the other controls the roll gimbal. Each GDA is capable of extending or retracting 2 inches from the mid-position and will gimbal the descent engine a maximum of plus or minus 6 degrees about the LM Y-and Z-axes.

The position transducer receives 115V 400 cps from the DECA on its primary winding. The secondary voltage is 400 cps, 7.5 volts per inch travel from the mid-position; in phase when extended and 180 degrees out-of-phase when retracted past the mid-position. The transducer output is sent back to the DECA malfunction logic.

4.3.4.6.3 Mechanization.

a) Stroke. The stroke of the GDA is ±2.0 inches ±5 percent.

b) Rate of Travel. During steady state operation the constant rate of travel of the GDA stroke is 0.0662 inch per second ±10 percent with an aiding or opposing load from 0 to 250 pounds.

c) Equivalent time constant. The equivalent time constant of the loaded GDA is compatible with the starting and stopping (overtravel) requirements.

d) Start. The steady-state rate of travel of the GDA is attained in less than 0.1 second.

e) Overtravel. When electrical power is removed, the GDA stops within 0.01 inch with an aiding or opposing load from 0 to 250 pounds.

f) Reversing. Reversing time does not exceed 0.2 second

g) Stall Condition. The GDA is capable of withstanding continuous stall at full voltage without damage.

h) Angular Rate. The angular rate of the GDA is 0.198 degree per second.

i) Irreversibility. The GDA is irreversible up to the ultimate static load in tension and compression without use of the motor brake.

j) Induction Potentiometer. (See Figure 4.3.4.6-1.) Gimbal position signals are transmitted from the actuator output shaft or gear train by an induction potentiometer. The potentiometer possesses the following characteristics:

Output:	15 ± 0.2 vrms
Linearity:	2.0 percent
Maximum linear range:	±2.00 in.
Residual null voltage (total)	50 mv
Sensitivity:	7.5 v/in.
(1) 15 volts:	+2.00 in.
(2) 15 volts (phase shifted 180 deg)	-2.00 in.
(3) Zero volts:	Midstroke setting
Input:	115 volts, single-phase 400 cps

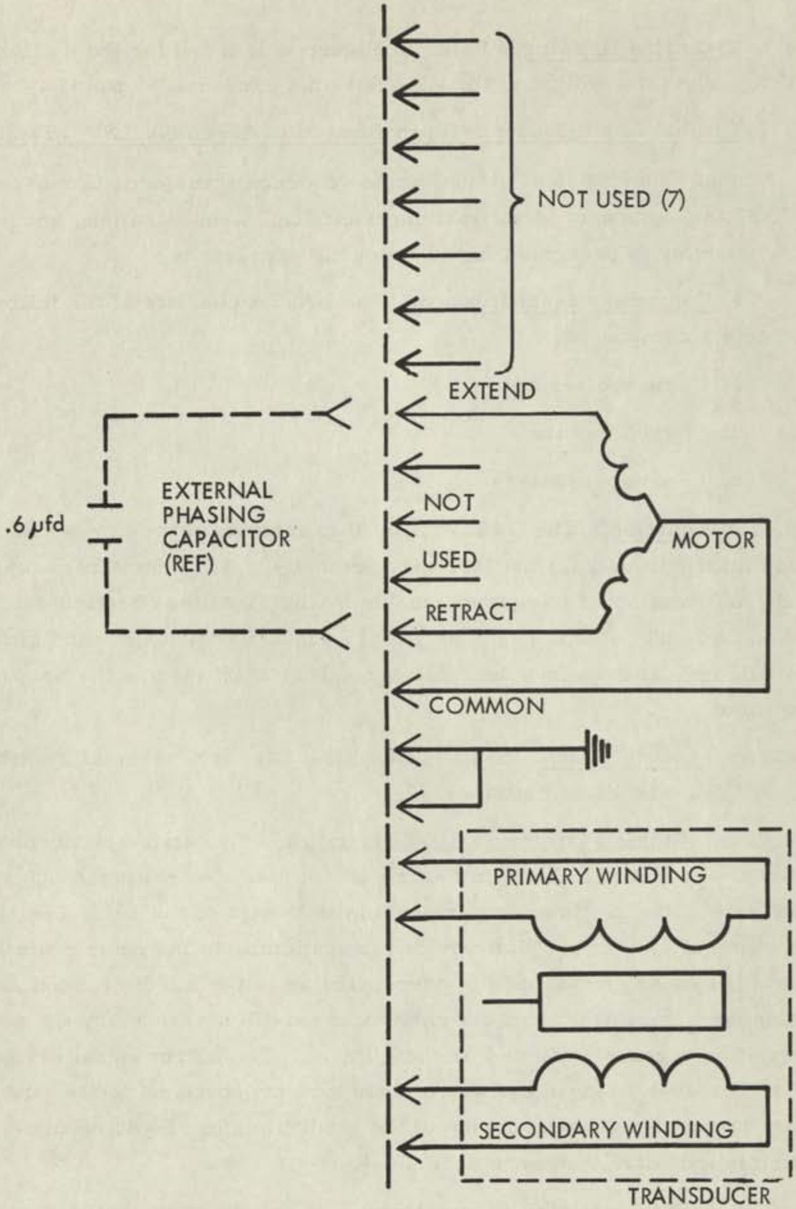


Figure 4.3.4.6-1. AGA Schematic

4.3.4.6.4 Outline Drawing. Refer to Figure 4.3.4.6-2 for the outline of the GDA. The total weight of the GDA does not exceed 3.55 pounds.

4.3.4.7 Gimbal Angle Sequence Transformation Assembly (LSP-350-302B.)

A brief description of gimbal angle sequence transformation assembly (GASTA) Component Identification, Function, Mechanization, and Outline Drawing is presented in the following paragraphs.

4.3.4.7.1 Component Identification. The GASTA consists of the following principal components:

a) Synchro resolvers

b) Servo motors

c) Servo amplifiers

4.3.4.7.2 Function. The GASTA is used to transform the three-axis attitude information from the IMU (see Section 4.3.1.1) into three-axis attitude information in a sequence usable by the FDAI (see Section 4.3.4.8). The GASTA must determine the necessary angular position of each gimbal in the ball indicator so that the indicator sphere may assume the proper orientation.

4.3.4.7.3 Mechanization. See Figure 4.3.4.7-1 for a schematic of the IMU, GASTA, and FDAI functions.

4.3.4.7.3.1 Stable Platform (IMU) Orientation. The stable platform and its gimbal sequence is shown in Figure 4.3.4.7-2. Note that the outer gimbal axis of the platform is parallel to the X-axis of the LM. The middle gimbal axis of the platform is perpendicular to the outer gimbal axis and the inner gimbal axis is perpendicular to the middle gimbal axis. Rotation of each gimbal about its particular axis is measured by the output of a synchro resolver mounted on that gimbal. The output signal of each synchro resolver is given by two voltages; one proportional to the sine and one proportional to the cosine of the gimbal angle. Positive angle rotation is indicated in Figure 4.3.4.7-1.

4.3.4.7.3.2 Ball Indicator Orientation. The sphere in the ball indicator and its gimbal sequence are also shown in Figure 4.3.4.7-1. Note that

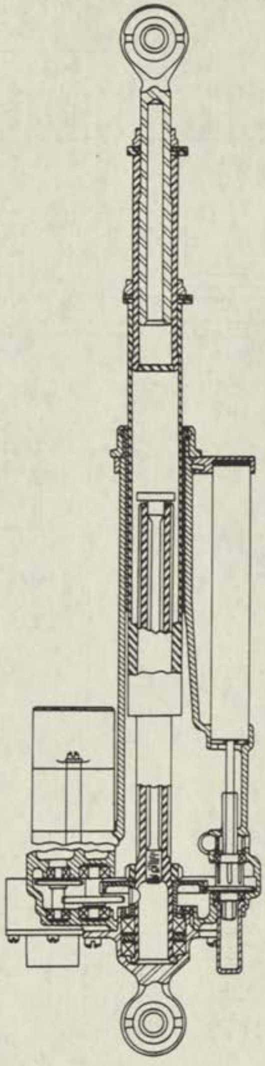


Figure 4. 3. 4. 6-2. Gimbal Drive Actuator Assembly Outline

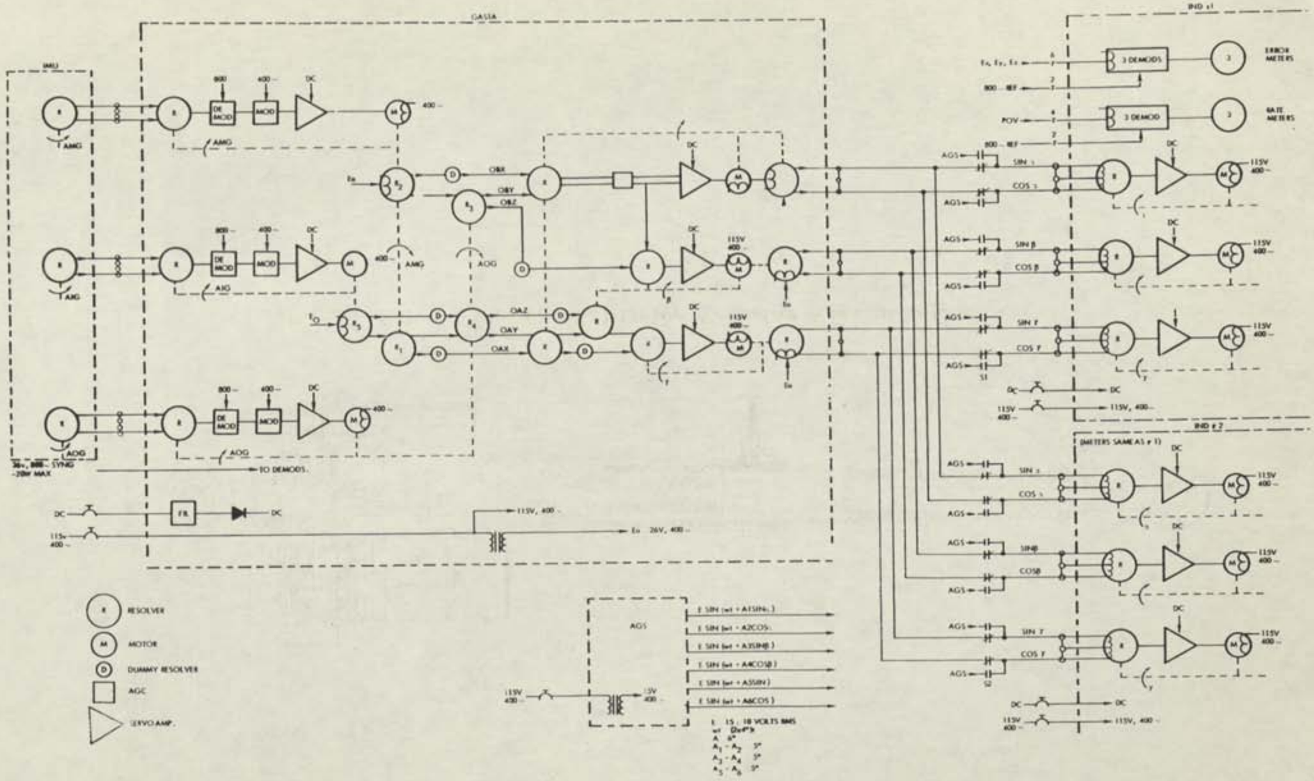
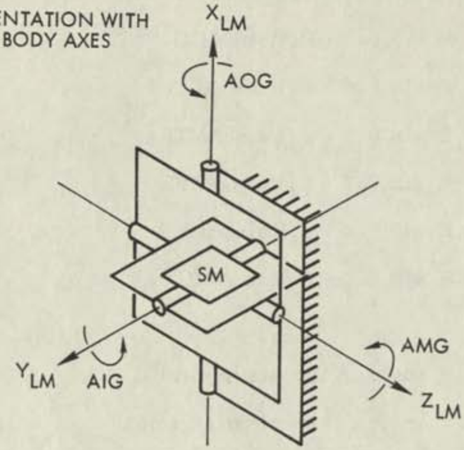


Figure 4. 3. 4. 7-1. IMU to FDA Transformation Mechanization

THE FOLLOWING RELATIONSHIPS HOLD BETWEEN BODY ANGLES AND CORRESPONDING PLATFORN AND BALL GIMBAL ANGLES

BODY	PLATFORM MECHANICAL	PLATFORM ELECTRICAL	BALL
+ YAW	-AOG	+ AOG	$+\beta$
+ ROLL	-AMG	+AMG	$-\alpha$
+PITCH	-AIG	+AIG	$+\gamma$

PLATFORM ORIENTATION WITH RESPECT TO LM BODY AXES



INDICATOR ORIENTATION WITH RESPECT TO LM BODY AXES

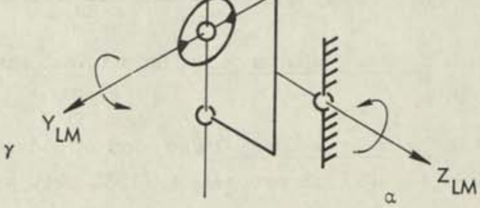


Figure 4. 3. 4. 7-2. IMU and FDAI Gimbal Sequence and Orientation

the outer gimbal axis of the sphere is parallel to the LM Z-axis, the middle gimbal axis is perpendicular to the outer gimbal axis, and the inner gimbal axis is perpendicular to the middle gimbal axis. Positive angle rotation is shown in Figure 4.3.4.7-2.

4.3.4.7.3.3 Input Signals. The input signals to the GASTA are derived from the IMU as voltages proportional to the sine and cosine of the IMU gimbal angles. These voltages are as follows:

$$E_1 = E \sin(\omega t + \theta_1) \sin AOG$$

$$E_2 = E \sin(\omega t + \theta_2) \cos AOG$$

$$E_3 = E \sin(\omega t + \theta_3) \sin AMG$$

$$E_4 = E \sin(\omega t + \theta_4) \cos AMG$$

$$E_5 = E \sin(\omega t + \theta_5) \sin AIG$$

$$E_6 = E \sin(\omega t + \theta_6) \cos AIG$$

Where:

AOG = angle of the outer gimbal

AMG = angle of the middle gimbal

AIG = angle of the inner gimbal

$E \sin \omega t$ = IMU resolver excitation voltage

θ = phase shift from the excitation voltage

$\theta_1 - \theta_2 \leq 0.5$ deg

$\theta_3 - \theta_4 \leq 0.5$ deg

$\theta_5 - \theta_6 \leq 0.5$ deg

$$\text{Scale factor near null} = \frac{26 \text{ vrms} \pm 10 \text{ percent/deg}}{57.3}$$

4.3.4.7.3.3.1 Source Impedance. The nominal source impedance is 190 + j 150 ohms.

4.3.4.7.3.3.2 Load Impedance. The load impedance on both sine and cosine windings is 415 (±15 percent) + j 1950 (±10 percent) ohms.

4.3.4.7.3.3.3 <u>Signal Phase.</u> The total attitude signal phasing is defined by one stability (or body) axis at a time using positive rotations measured from IMU gimbal zero position as shown in Figure 4.3.4.7.1.

a) A positive rotation is defined as a clockwise rotation of the vehicle about an axis viewed from the inside of the vehicle out along the positive direction of the axis.

b) IMU gimbal zero position is defined as IMU gimbal resolver electrical zero with the sine winding at minimum coupling and the cosine winding at maximum coupling.

c) For positive vehicle rotations from 0 degree to 90 degrees, both E_1 and E_2 are in phase with the resolver excitation voltage. For positive rotations from 90 degrees to 180 degrees, E_1 is in phase and E_2 is out of phase with the resolver excitation voltage.

4.3.4.7.3.4 <u>Output Signals.</u> GASTA output signals are voltages proportional to the sine and cosine of the required indicator gimbal angles. The roll, yaw, and pitch gimbal axis angles are designated α, β, and γ, respectively. The signal for each axis is supplied on three lines, and one line for the common return. The GASTA computes the following angles:

$$\alpha_A = \tan^{-1}\left(\frac{-\sin AMG}{\cos AOG \cos AMG}\right)$$

$$\beta_A = \tan^{-1}\left(\frac{\sin AOG \cos AMG}{-\sin AMG \sin\alpha + \cos AOG \cos AMG \cos\alpha}\right)$$

$$\gamma_A = \tan^{-1}\left[\frac{-(\sin AOG \cos AIG + \cos AOG \sin AMG \sin AIG)\sin\alpha + (\sin AIG \cos AMG)\cos\alpha}{(\sin AIG \sin AOG - \cos AIG \cos AOG \sin AMG)\sin\alpha + (\cos AIG \cos AMG)\cos\alpha}\right]$$

4.3.4.7.3.4.1 <u>Output Signal Characteristics.</u> The output signal characteristics are given as:

$$\left.\begin{array}{ll} \text{a)} & E_1 = E \sin(\omega t + A_1) \sin\alpha \\ \text{b)} & E_2 = E \sin(\omega t + A_2) \cos\alpha \end{array}\right\} \text{Roll}$$

$$\left.\begin{array}{ll} \text{c)} & E_3 = E \sin(\omega t + A_3) \sin\beta \\ \text{d)} & E_4 = E \sin(\omega t + A_4) \cos\beta \end{array}\right\} \text{Yaw}$$

$$\left.\begin{array}{ll} \text{e)} & E_5 = E \sin(\omega t + A_5) \sin\gamma \\ \text{f)} & E_6 = E \sin(\omega t + A_6) \cos\gamma \end{array}\right\} \text{Pitch}$$

Where:

$E \sin \omega t$ = signal carrier from the 400-cycle power source

$E = 15 \pm 10$ percent vrms

$\omega t = (2\pi\ 400)t$

A = phase shift from the 400-cycle power source (phase shift ± 15 deg or less)

4.3.4.7.3.4.2 Output Impedance. The maximum output impedance of the GASTA is 100 $\underline{/20 \text{ deg} \pm 10 \text{ deg}}$ ohms.

4.3.4.7.3.4.3 Load Impedance. The GASTA load is the FDAI (see Section 4.3.4.8). The minimum load impedance presented by one FDAI is 4800 $\underline{/75 \text{ deg} \pm 10 \text{ deg}}$ ohms.

4.3.4.7.3.4.4 Static Accuracies. The maximum error in the GASTA output signals are:

$$E\alpha \equiv \left| \alpha_A - \tan^{-1} \frac{E \sin \alpha \cos A_1}{E \cos \alpha \cos A_2} \right| \leq 0.75 \text{ deg} \qquad \text{Roll}$$

$$E\beta \equiv \left| \beta_A - \tan^{-1} \frac{E \sin \beta \cos A_3}{E \cos \beta \cos A_4} \right| \leq 1.1 \text{ deg} \qquad \text{Yaw}$$

$$E\gamma \equiv \left| \gamma_A - \tan^{-1} \frac{E \sin \gamma \cos A_5}{E \cos \gamma \cos A_6} \right| \leq 1.75 \text{ deg} \qquad \text{Pitch}$$

α_A, β_A, and γ_A are computed per paragraph 4.3.4.7.3.4.

4.3.4.7.3.4.5 Slew Rate. The minimum slew rate of each of the GASTA output angles, individually or concurrently, is 45 degrees per second.

4.3.4.7.4 Outline Drawings. Refer to GAEC LSC 350-30200. The total weight of the GASTA does not exceed 7 pounds.

4.3.4.8 Flight Director Attitude Indicator (LSP-350-301).

A brief decription of FDAI component Identification, Function, Mechanization, and Outline Drawing is presented in the following paragraphs.

4.3.4.8.1 Component Identification. The attitude indicator includes the following:

a) A gimbaled, servo-driven sphere which is free to rotate through 360 degrees in each of three mutually perpendicular axes. The sphere position in all three axes is read against a fixed reference centered on the face of the indicator.

b) Six meter-driven pointers, three of which are center-zeroed pointers in front of the sphere face. The remaining center-zeroed pointers are located to the right, above, and below the sphere, and translate against fixed scales.

4.3.4.8.2 Function. Two LM indicators in each vehicle provide an integrated display of vehicle attitude, attitude error (steering error), and vehicle angular rates.

Since the gimbal axes of the IMU and the indicator are not parallel, it is not possible to slave the indicator gimbals directly to the IMU gimbals. A transformation mechanism (resolver chain) is required between the IMU and the indicator. This transformation is accomplished by the gimbal angle sequence transformation assembly. The abort guidance input to the FDAI may be selected rather than the PGNCS input. This is a direct input to the FDAI servos and need not be resolved since the AEA calculates the correct indicator gimbal angles digitally and converts its digital signals to analog signals via ladder networks.

Also displayed on the FDAI are the LM vehicle's roll, pitch, and yaw rates. These indications result from signals generated by the three rate gyros associated with the CES. The roll rate pointer is at the tip of the FDAI, pitch at the right, and yaw at the bottom.

LM vehicle attitude errors are indicated on the FDAI by the three attitude error needles located between the outer edges of the sphere and the reticle. These indicators are located identical to the rate pointer; roll error at the tip, pitch error at the right, and yaw error at the bottom. Shaft and trunnion angles can be displayed on the pitch and yaw error needles, respectively, if the proper control switching is made.

4.3.4.8.3 Mechanization

4.3.4.8.3.1 Gimbal Set. The FDAI ball is positioned by three gimbals whose axes are mutually perpendicular. The gimbal sequence is roll, yaw, and pitch from the outer to the inner gimbal. Sphere rotation is illustrated in Figure 4.3.4.8-1. When the ball of the FDAI is displaying a condition of zero roll, zero yaw, and zero pitch, the gimbal positions are as follows:

a) Roll Gimbal. The roll or outer gimbal axis is perpendicular to the face of the indicator.

b) Yaw Gimbal. The yaw or middle gimbal axis is at right angles to the roll gimbal and vertically oriented.

c) Pitch Gimbal. The pitch or inner gimbal axis is at right angles to the yaw gimbal axis and oriented horizontally.

4.3.4.8.3.2 Indicator Sphere. The indicator sphere contains the inner gimbal servo, which drives two sphere caps for pitch information. The sphere has a fixed segment (with respect to the pitch axis) around the outer surface of the sphere, and two caps with a narrow separation band covered by the fixed segment. The left and right sections of the sphere are the pitch caps which rotate with respect to the fixed segment about the pitch axis. The sphere as a whole rotates about the yaw and roll axes with continuous freedom in either direction.

4.3.4.8.3.3 Attitude Readout. Pitch attitude is read as ball pitch displacement from the horizontal reference bar. Yaw attitude is read as ball displacement from the vertical fixed reference bar. Roll attitude is obtained qualitatively from the inclination of the horizon line relative to the horizontal bar of the fixed reference and quantitatively by reading the intersection of a yaw graduation, through the center of the fixed reference, with the roll mask at the gray end of the sphere. The α (roll attitude if β and γ are zero) is read as the position of the α bug against the roll mask. Both roll and α are read as counterclockwise rotation from the zero diamond at the top of the roll mask.

4.3.4.8.3.4 Polarity. The attitude display is an "inside-out" display which adheres to this convention; for positive vehicle roll (right wing down), the sphere will rotate counterclockwise as viewed by the operator; for positive vehicle pitch (pitch up) the sphere will rotate downward as

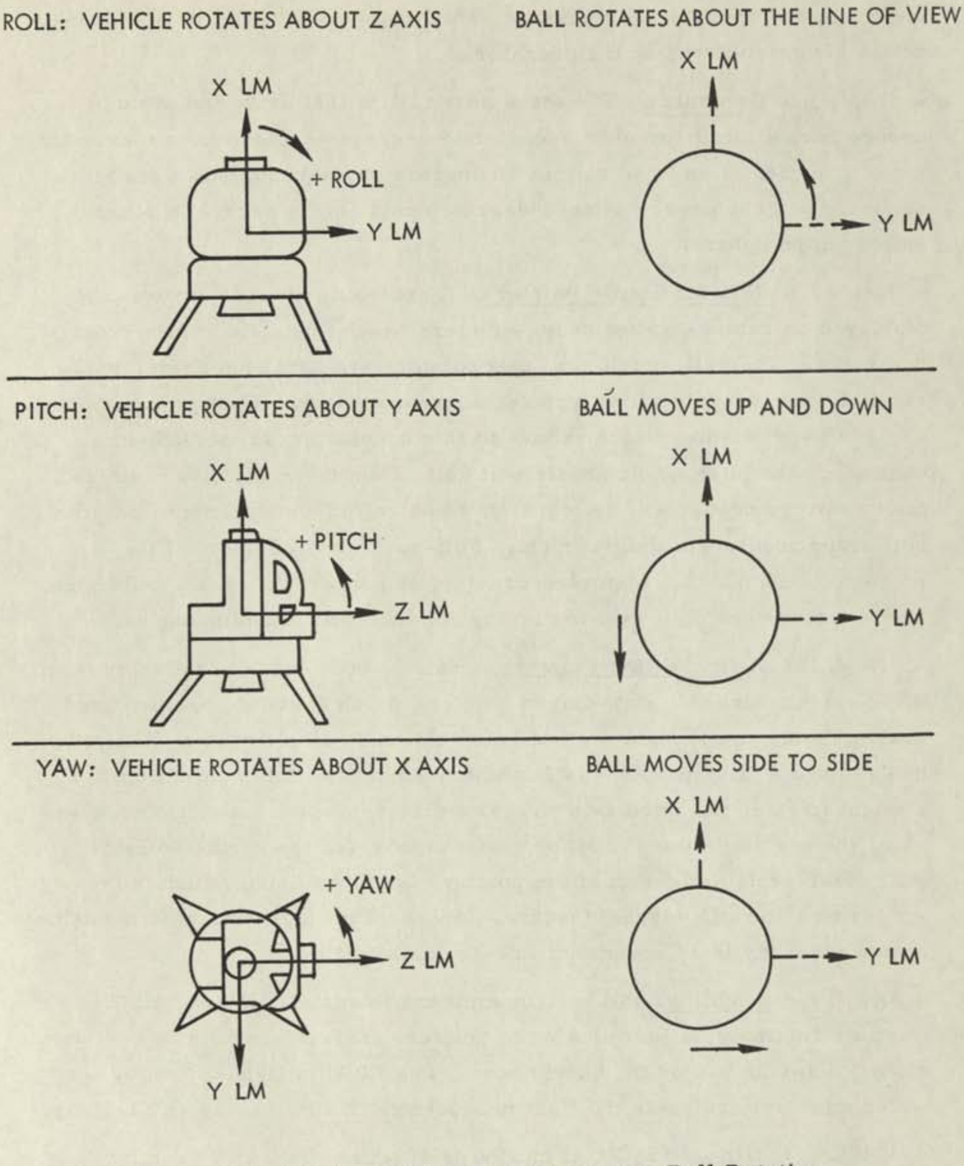


Figure 4.3.4.8-1. Vehicle Rotation versus Ball Rotation

viewed by the operator; for positive vehicle yaw (yaw left), the sphere will rotate right as viewed by the operator.

4.3.4.8.3.5 Dynamics. The three servo loops that drive the attitude sphere have a minimum slew speed of 45 degrees per second, a maximum lag of 1 degree at an input rate of 10 degrees per second, and a static accuracy of ±0.5 degree at zero-degree inputs and ±1 degree at other points on the sphere.

4.3.4.8.3.6 Attitude Error Pointers. Three-axis attitude errors are displayed on center-zeroed cross-pointers which translate from a fixed reference. The roll, pitch, and yaw pointers are arranged above, to the right of, and below the fixed reference, respectively. The error needles are a "fly-to" display which adhere to this convention: for a pitch-up command, the pitch error needle will deflect up; for a yaw left command, the yaw error needle will deflect left, for a roll clockwise command, the roll error needle will deflect right. Full-scale displacement of the pointers is ±0.8 inch. Meter accuracy is ±3 percent full-scale deflection. These meters are also used to display RR shaft and trunnion angles.

4.3.4.8.3.7 Attitude Rate Pointers. Vehicle body rates are displayed on three center-zeroed meter-driven pointers which translate against fixed scales above, to the right of, and below the main display face. The rate needles are a "fly-to-null" display which adheres to this convention: for a pitch-up rate, the pitch rate pointer deflects down; for a roll clockwise rate, the needle deflects left; for a yaw right rate, the needle deflects left. Full-scale deflection of the pointers is ±0.940 inch, which corresponds to ±5 or ±25 degrees/second, dependent upon scaling switch setting. Meter accuracy is ±3 percent of full-scale deflection.

4.3.4.8.3.8 Lighting. All instruments are integrally lighted with EL lamps. The integral illumination of pointers and bars results in a silhouetted pointer or bar on the background. The FDAI is lighted by a wedge which uniformly refracts the light produced on the display by an EL lamp.

4.3.4.8.4 Outline. The FDAI outline is specified by GAEC Specification No. LSP-350-301. See Figure 4.3.4.8-2 for a front panel view.

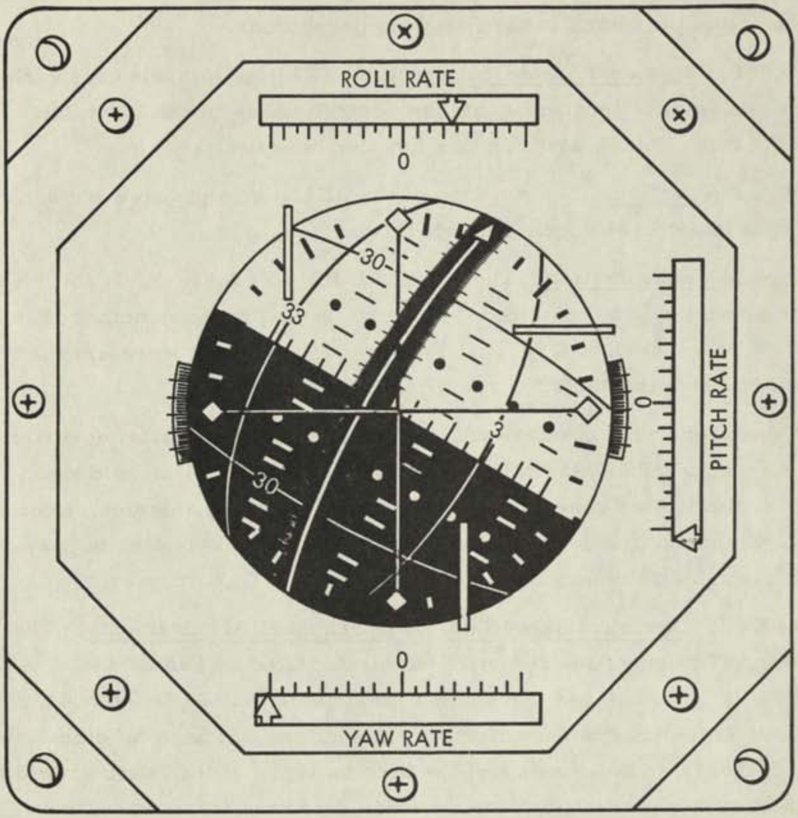


Figure 4.3.4.8-2. Flight Director Attitude Indicator

4.3.4.9 Ascent Engine Latching Device and Engine Sequencing

A brief description of the ascent engine latching device and engine sequencing is presented in the following paragraphs.

4.3.4.9.1 Component Identification. The AELD is comprised of various relays and devices located behind the control panels in the LM cabin, consequently a "black box" for this function does not exist.

4.3.4.9.2 Function. The function of the AELD is to process and activate the applicable engine sequencing operations.

4.3.4.9.3 Mechanization. The AELD, LGC, ATCA, AGS, DECA, and controls and displays combined determine the engine sequencing. Tables 4.3.4.9-1 through 4.3.4.9-7 and Paragraph 4.3.4.9.3.1 present the operational requirements for specific engine conditions.

NOTE: Refer to the displays and controls section for a brief description of the DES ENG CMD OVRD switch (1-S-15). During the non-abort state zone (Dead Man Zone) of the lunar mission, prior to landing, the descent engine redundant feed is applied to prevent engine shutdown in the event of a loss of DECA power.

4.3.4.9.3.1 Descent Engine Shut Off Operational Requirements. The DE will shut off if any of the command elements listed in Tables 4.3.4.9-1, 4.3.4.9-2, or 4.3.4.9-3 appear in a state other than in the state they appeared in the previous equations.* In addition, the auto off signal will shut off the DE in the absence of the auto on signal if the latter caused the engine to turn on initially. Stop override operation is described in Table 4.3.4.9.3-4. Either engine may also be stopped by disabling its respective power lines by pulling off their circuit breakers.

*For purposes of this statement, an auto "off" signal is considered to be the opposite state of the auto "on" signal.

Table 4. 3. 4. 9-1. Normal Descent Engine Turn On Operational Requirements

Descent Engine On Command = DEarm • $\overline{\text{AEarm}}$ • Auto On Signal • $\overline{\text{Abort Stage}}$ • $\overline{\text{Stop Override}}$

Command Elements	Primary Functions	Special Functions
DE Arm Commanded (Engine Select Switch)	Makes dc power available for final "on" command upon receipt of auto "on" signal.	Resets flip-flip in Descent Engine automatic signal channel to await presence of the auto engine on signal.
	Provides dc power for DECA and DE electronics (Servo Amp).	Enables fire override command to reach the Descent Engines
	Notifies LGC of engine arm status (see Note a).	
Auto "On" Signal (See Note b)	Sets flip-flop in the DE automatic command channel which causes the start relay to transmit power from the arm relay to the descent engine isolation valves. This operation will also occur in the simultaneous presence of the auto "on" and auto "off" command signals.	The automatic "on" signal causes the "on" power to be routed to the Descent or Ascent Engine as follows: Descent Engine a) In the presence of the DE arm command. b) In the event an abort is initiated. Ascent Engine a) In the presence of the AE arm command. b) In the event an abort stage is initiated.

Note a: LGC will receive a closure when either the DE or AE are armed through the engine select switch. The abort and abort stage functions also arm the DE and AE respectively; however, when either of these are actuated, the LGC shall not receive an engine arm status signal.

Note b: AGS or LGC will generate an auto "on" and an auto "off" command.

Table 4.3.4.9-2. Emergency Descent Engine Turn on Operational Requirements

Descent Engine On Command = Abort • $\overline{\text{AE arm}}$ • Auto On Signal • $\overline{\text{Abort Stage}}$ • $\overline{\text{Stop Override}}$

Command Elements	Primary Functions	Special Functions
Abort Command	Arms the Descent Engine by making dc power available for final "on" command upon receipt of auto "on" signal and providing dc power for DECA and DE electronics (Servo Amp). Notifies PGNCS and AGS to begin an abort program and to subsequently send an engine "on" signal.	Reset flip-flop in Descent Engine automatic signal channel to await presence of the auto engine on signal.
Auto "On" Signal	Sets flip-flop in the DE automatic command channel which causes the start relay to transmit power from the arm relay to the descent engine isolation valves. This operation will also occur in the simultaneous presence of the auto "on" and auto "off" command signals.	The automatic "on" signal causes the "on" power to be routed to the Descent or Ascent Engines as follows: Descent Engine a) In the presence of the DE arm command b) In the event an abort is initiated. Ascent Engine a) In the presence of the AE arm command b) In the event an abort stage is initiated

Table 4. 3. 4. 9-3. Manual Descent Engine Turn on Operational Requirements

Descent Engine On Command = DE arm • Fire Override • $\overline{\text{Abort Stage}}$ • $\overline{\text{AE Arm}}$ • $\overline{\text{Stop Override}}$

Command Elements	Primary Functions	Special Functions
DE Arm Commanded (Engine Select Switch)	Arms DE by making dc power available for final "on" command upon receipt of auto "on" signal, providing dc power for DECA and DE electronisc (Servo Amp) and notifying LGC of engine arm status (See Note a of Table 4. 3. 4. 9-1). Enables the fire override command to reach the DE.	Resets flip-flop in Descent Engine automatic signal channel to await presence of the auto engine on signal.
Fire Override Commanded	Causes the start relay to transmit power from the arm relay to the DE.	The fire override command is routed to the Descent or Ascent Engine Latching Device via the engine select switch.

Table 4.3.4.9-4. Description of Manual Descent Engine Shutoff Command Element

Command Elements	Primary Functions	Special Functions
Stop Override	Simultaneously prevents the descent engine and ascent engine from firing in the normal and emergency automatic modes. In addition, the DE is prevented from firing in the manual mode.	

Table 4. 3. 4. 9-5. Normal Ascent Engine Turn On Operational Requirements

Ascent Engine On Command = AE arm • Auto on Signal • $\overline{\text{Stop Override}}$ • $\overline{\text{DE Arm}}$

Command Elements	Primary Functions	Special Functions
AE Arm Commanded (Engine Select Switch)	Makes dc power available for final "on" command upon receipt of the automatic "on" signal.	Enables the fire override command to reach the AE Resets flip-flop in ascent engine automatic signal channel to await presence of the auto engine on signal. Permits the auto "on" relay to be activated by the auto "on" signal and inhibits the DE auto on channel. This insures proper routing of the auto "on" signal to the ascent or descent engines and prevents a single failure from causing simultaneous AE or DE ignition.

Table 4. 3. 4. 9-5. Normal Ascent Engine Turn on Operational Requirements (Continued)

Ascent Engine On Command = AE arm • Auto on Signal • $\overline{\text{Stop Override}}$ • $\overline{\text{DE Arm}}$

Command Elements	Primary Functions	Special Functions
Auto On Signal	Sets flip-flop in the AE automatic command channel which causes the start relay to transmit power to the ascent engine isolation valves. This operation will also occur in the simultaneous presence of the auto "on" and auto "off" command signals.	The automatic "on" signal causes the "on" power to be routed to the descent or ascent engine as follows: Descent Engine a) In the presence of the DE arm command b) In the event an abort is initiated Ascent Engine a) In the presence of the AE arm command b) In the event an abort stage is initiated.

Table 4. 3. 4. 9-6. Emergency Ascent Engine Turn on Operational Requirements

Ascent Engine On Command = Abort Stage • Auto On Signal • $\overline{\text{Stop Override}}$

Command Elements	Primary Functions	Special Functions
Abort Stage Command	Arms the ascent engine through a time delay* by making dc power available for final "on" command upon receipt of the automatic "on" signal. Permits the auto "on" relay to be activated by the auto "on" signal and inhibits the DE auto on channel. This insures proper routing of the auto "on" signal to the ascent engine and prevents a single failure from causing simultaneous AE or DE ignition. Pressurizes the AE fuel and oxidizer tanks Notifies PGNCS and AGS to begin an abort stage program and to subsequently send an engine "on" signal. Enables the fire override command to reach the AE.	Stops the DE by: a) Actuating the DE stop override relay b) Disabling the DE auto on channel c) Opens the DE fire override. Circuit path

*The magnitude of the time delay is in the order of 200 msec to insure DE shutoff prior to AE ignition. (Thrust Decoy)

Table 4. 3. 4. 9-6. Emergency Ascent Engine Turn on Operational Requirements (Continued)

Ascent Engine On Command = Abort Stage • Auto On Signal • $\overline{\text{Stop Override}}$

Command Elements	Primary Functions	Special Functions
Auto "On" Signal	Sets flip-flop in AE automatic command channel which causes the start relay to transmit power to the ascent engine isolation valves. This operation will also occur in the simultaneous presence of the auto "on" and auto "off" command signals.	The automatic "on" signal causes the "on" power to be routed to the Descent or Ascent Engines as follows: Descent Engine a) In the presence of the DE arm command b) In the event an abort is initiated Ascent Engine a) In the presence of the AE arm command b) In the event an abort stage is initiated

Table 4. 3. 4. 9-7. Manual Ascent Engine Turn on Operational Requirements

Ascent Engine On Command = AE arm • Fire Override

Command Elements	Primary Functions	Special Functions
Ascent Engine Arm Command (Engine Select Switch)	Enables the fire override command to reach the AE.	Resets flip-flop in ascent engine automatic signal channel to await presence of the auto engine on signal.
		Permits the auto "on" relay to be activated by the auto "on" signal and inhibits the DE auto on channel.
		This insures proper routing of the auto "on" signal to the ascent or descent engines and prevents a single failure from causing simultaneous AE or DE ignition.
Fire Override Command	Causes power to reach the AE.	The fire override command is routed to the descent or ascent engine latching device via the engine select switch.

4.3.4.10 Orbital Rate Drive Electronics for Apollo and Lunar Module (ORDEAL).

A brief description of ORDEAL Component Identification, Function, Mechanization and General Characteristics is presented in the following paragraphs:

4.3.4.10.1 Component Identification. The ORDEAL is an electromechanical device capable of rotating a pair of self-contained resolvers at a variable rate. The ORDEAL is located in the lower side console.

4.3.4.10.2 Function. The ORDEAL provides signal insertion and processing between the FDAI and its normal pitch attitude drive signals from the selected drive source. The processed signal produces an FDAI display of the computed local vertical attitude (open-loop torquing) as an inflight selectable alternative to the inertial attitude display.

4.3.4.10-3 Mechanization. The ORDEAL is mechanized to permit manual setting of altitudes from 10 to 310 nautical miles, calibrated in 5-nautical mile steps. A positive locking mechanism is provided to allow locking of the altitude knob in any desired position. See Figure 4.3.4.10-1 for a simplified block diagram of the ORDEAL. Refer to Figure 4.3.4.10-2 for a simplified diagram of the ORDEAL operation. Given point A in a circular lunar orbit, the IMU and the LM vehicle are aligned to the local horizontal; i.e., the FDAI indicates 0 degree pitch. At point B, in order to align the vehicle to the local horizontal the vehicle must be pitched down (negative pitch). When the vehicle pitches down, the FDAI moves up and displays θ degree on the black portion of the ball. To obtain the desired display of zero pitch (alignment with local horizontal), the ORDEAL cancels θ causing the ball to move down by θ degrees.

4.3.4.10.3.1 Orbit Rate. The resolver rates associated with the altitude position settings may be acquired from the equation of orbital period.

$$T = K_1 \left(\frac{a}{R}\right) \frac{3}{2}$$

Where R = Planetary radius
a = Orbit semimajor axis
K = Planetary constant
T = Orbital period in hours

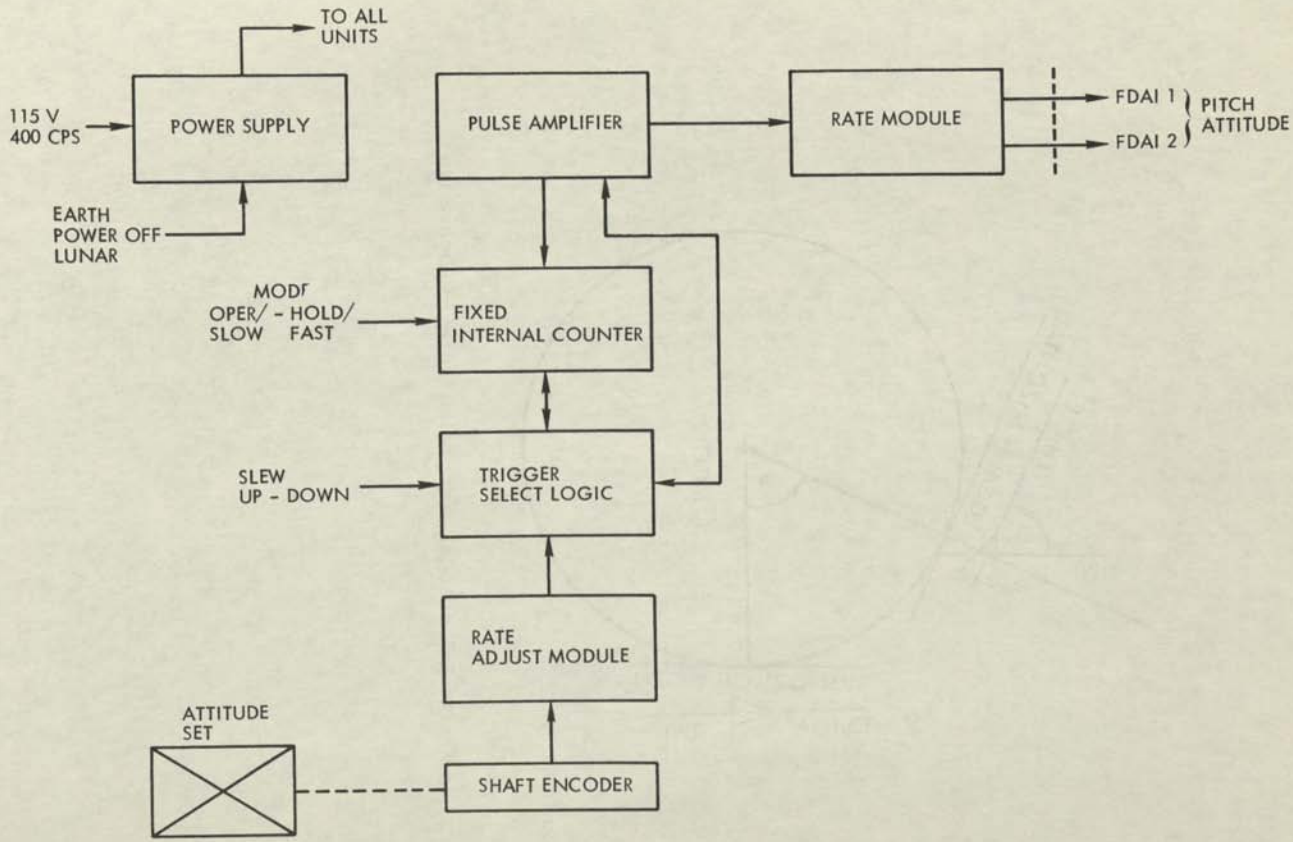


Figure 4. 3. 4. 10-1. Simplified ORDEAL Block Diagram

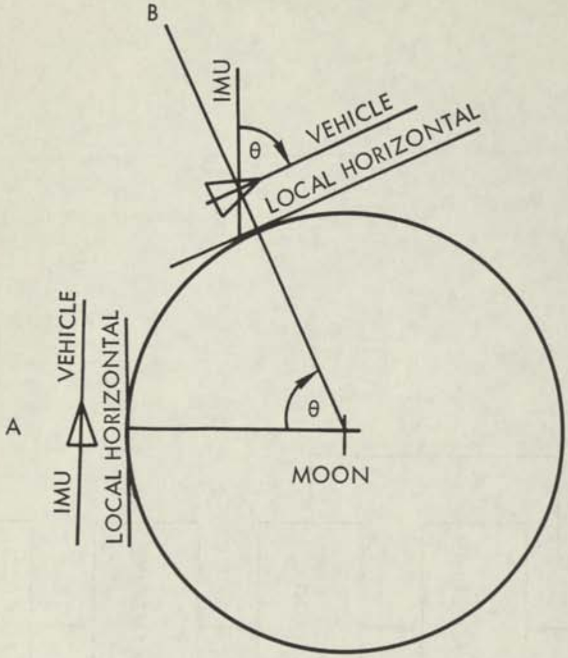


Figure 4. 3. 4. 10-2. Simplified ORDEAL Operation

For the purpose of calculations, "a" corresponds to the orbit radius, since a circular orbit is the basic assumption for ORDEAL operation.

Substituting:

$$\frac{a}{R} = \frac{R + h}{R} \qquad \text{Where} \quad h = \text{Orbit Altitude}$$

The orbital rate:

$$B' = 2nf = \frac{360^\circ}{T}$$

B' = Orbital rate

f = Frequency

T = Orbital period hours

Since

$$f = \frac{1}{T}$$

To obtain degrees per second

$$B' = \frac{360^\circ}{3600T} = \frac{0.1^\circ}{T}/\text{sec}$$

$$B' = \frac{.1}{K_1}\left(1 + \frac{h}{R}\right) - 1.5$$

Since for lunar orbit $K_1 = 1.81$ and $R = 933$ nautical miles

Therefore:

$$B' = \frac{0.1}{1.81}\left(1 + \frac{h}{R}\right) - 1.5^\circ/\text{sec}$$

4.3.4.10.4 General Characteristics. The ORDEAL weights approximately 6.5 pounds. The total system accuracy is 2 degrees per hour. The maximum dimensions are 4 x 4.5 x 9 inches. See Figure 4.3.4.10-3 for a front panel view of the ORDEAL.

4.3.5 Landing Point Designator (LPD)

Discussion of the LPD is presented in this section independent of the other four major elements previously presented under Identification of System Elements. This breakdown was brought about by the fact that

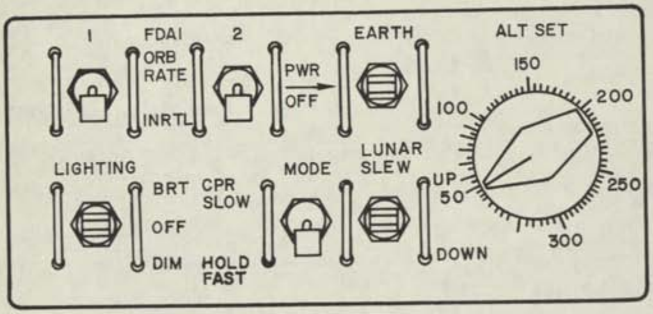


Figure 4. 3. 4. 10-3. ORDEAL Front Panel

the LPD is not an integral part of the Guidance and Control System; however, it is used directly in support of the landing phase of the Lunar Module. A brief description of LPD Identification, Function and Mechanization is presented in the following paragraphs:

4.3.5.1 Identification.

The LPD is located in the left-side crew station window of the LM vehicle (looking toward +Z-axis). See Figures 4.1.3-1 and 4.2.3-11 for equipment location and window markings.

4.3.5.2 Function. The function of the LPD is to provide a visual aid to the astronaut during the lunar landing phase. By using the LPD, the astronaut may more accurately determine and thus maneuver the LM vehicle to a more desirable landing point.

4.3.5.3 Mechanization. Implementation of the LPD is mechanized through programs in the LGC and through use of the Y-and Z-axis "Pulse Direct" switches in the attitude controller circuitry. Fore and aft motion of the attitude controller increments the landing site fore and aft about the Y-axis. Left and right motion of the attitude controller increments the landing site left and right about the Z-axis. For the function of the LPD, the LGC is programed to ignore the "Pulse Direct" contact closures while in the automatic mode prior to arrival at hi-gate and to accept these signals to increment changes to the landing point from hi-gate through the remainder of the descent. During the LPD implementation period of the trajectory, the LGC is programed to ignore the X-axis override capability for the attitude controller.

When using the LPD, one pulse will increment the landing point 2 degrees right or left or 0.5 degree fore or aft.

4.4 LM G&C INTERFACE DATA

This section of the LM G&C Data Book is for convenience and information reference only. An attempt will be made to maintain a comprehensive summary of the G&C ICD'S and sufficient details to describe the G&C interfaces. The up-to-date status on LM ICD'S is to be found in the latest revision of GAEC document LSR-540-XX, LM Interface Control Documentation Status Report.

The principal PGNCS interface requirements, ICD numbers, and interfacing equipments are shown in Figure 4.4-1, LM PGNCS Master Interface Requirements.

4.4.1 LM PGNCS Electrical and Functional Interfaces

The information in this section relates primarily to the interfaces of the guidance system with other LM systems and subsystems. Detailed information on this interface is contained in LIS 370-10004, LGC-LM Electrical Interface and in LIS 540-10001, LEM-PGNCS Functional Interface Requirements. This interface is represented in block diagram form in Figure 4.4-2, PGNCS and LM Functional Interface Diagram.

The following list tabulates all of the electrical and functional interfaces between the MIT guidance equipment and LM spacecraft. A short description is given of the nature and use of each interface.

The listing uses the following format:

a) Number assigned arbitrarily by this submittal

b) The designation (I) or (O) depending upon whether the interface is an input to or output from the guidance equipment

c) Interface name in capital letters

d) Pertinent ICD number. For LM ICD'S, the 1000 preceeding the last number is omitted as conveying no information. For LGC interfaces listed in LIS 370-10004, the signal number of this ICD is also given in parentheses.

e) The nature of the interface, i.e., dc, ac, pulse, etc.

f) Short description of the interface and its use

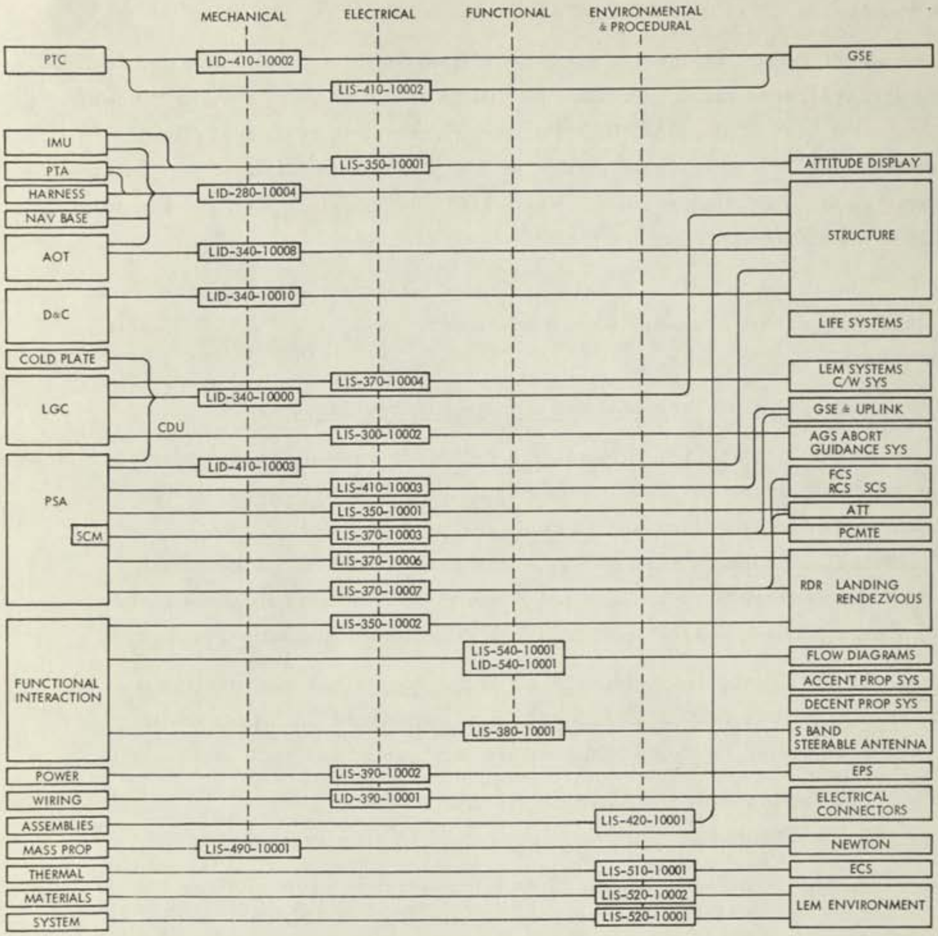


Figure 4.4-1. LM PGNCS Master Interface Requirements

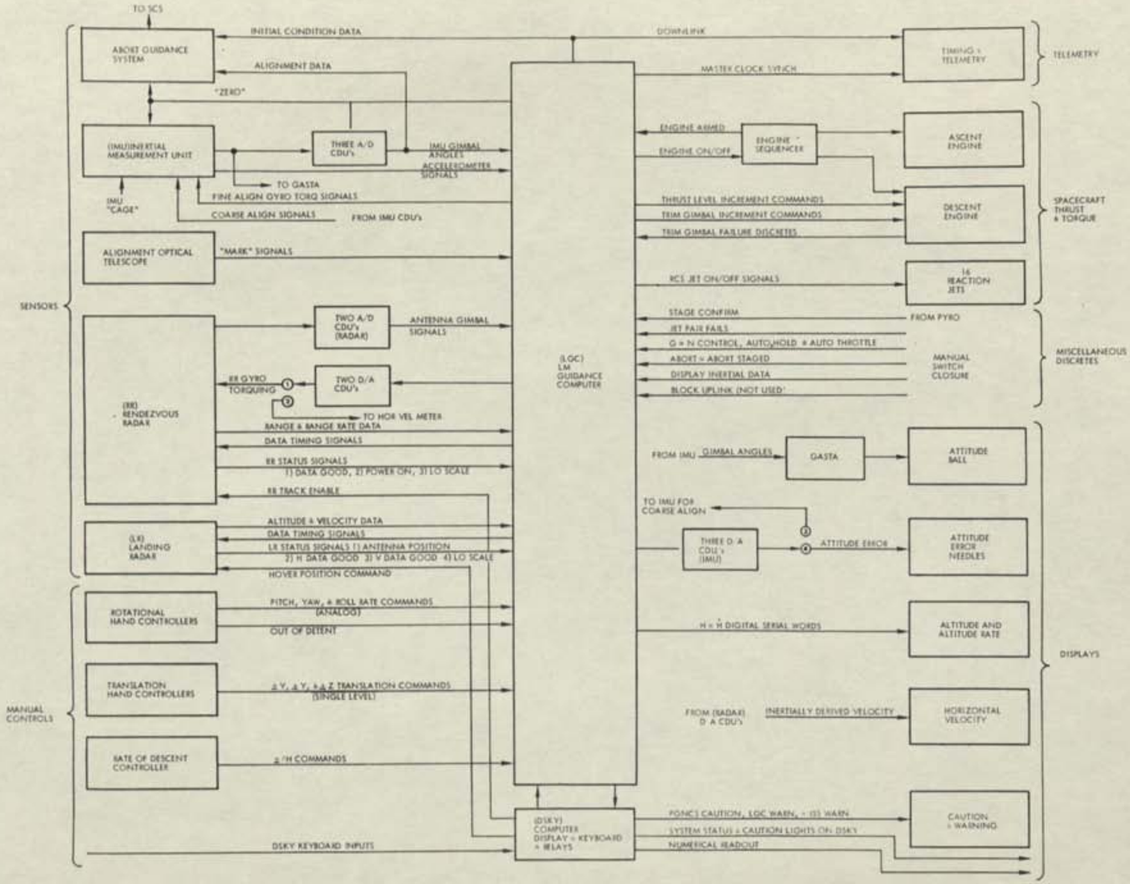


Figure 4.4-2. PGNCS Functional Interface Diagram

The information in this section is indexed as follows:

4.4.1.1 Mode and Status

4.4.1.1.1 (I) DIGITAL AUTOPILOT IN CONTROL (A07) (LIS 370-4) Contact Closure. Indicates astronaut has chosen primary PGNCS control of LM spacecraft rather than SCS or Abort Guidance System.

4.4.1.1.2 (I) ATTITUDE HOLD MORE (A02) (LIS 370-4) Contact Closure. This signal from control panel to LGC indicates astronaut has selected rate-command attitude-hold mode of attitude control where the vehicle responds to hand controller inputs and maintains a fixed inertial attitude in the absence of commands.

4.4.1.1.3 (I) AUTOMATIC MODE (A01) (LIS 370-4) Contact Closure. The signal from control panel to LGC indicates astronaut has selected the automatic mode of attitude control whereby the spacecraft attitude is determined and controlled by G&N system.

4.4.1.1.4 (I) IN AUTO THROTTLE (A08) (LIS 370-4) Contact Closure. The signal from control panel to LGC indicates that the G&N system should command the throttle of the descent engine.

4.4.1.1.5 (I) STAGE VERIFY (A23) (LIS 370-4) Contact Closure. Removes the STAGE STATUS signal from pyro to LGC at stage separation.

4.4.1.1.6 (I) ENGINE ARMED (A06) (LIS 370-4) Contact Closure. Indicates to G&N that the engine (ascent or descent) is ready to be ignited.

4.4.1.1.7 <u>(I) START ABORT PROGRAM</u> (A04) (LIS 370-4) Contact Closure. The signal from control panel to LGC to indicate astronaut command for abort, using the descent engine.

4.4.1.1.8 <u>(I) START ABORT/STAGE PROGRAM</u> (A05) (LIS 370-4) Contact Closure. The signal from control panel to LGC to indicate astronaut command for abort with ascent stage only.

4.4.1.1.9 <u>(I) IMU CAGE</u> (LIS 350-1) Contact Closure. Signal from cover protected spring-loaded main panel switch commands the G&N to drive all IMU gimbal angles to zero. Release of switch releases IMU to inertial stabilization at an attitude within 1 degree of that at the time of switch release.

4.4.1.1.10 <u>(I) DISPLAY INERTIAL DATA</u> (A12) (LIS 370-4) Contact Closure. The signal from control panel to LGC to display altitude, altitude rate, forward, and lateral velocity during lunar landing and inhibit RR antenna angular error from LGC to the RR.

4.4.1.1.11 <u>(I) THRUSTER 4D/4S FAIL</u> (A15)
4.4.1.1.12 <u>(I) THRUSTER 3U/3S FAIL</u> (A16)
4.4.1.1.13 <u>(I) THRUSTER 4U/4F FAIL</u> (A17)
4.4.1.1.14 <u>(I) THRUSTER 3D/3F FAIL</u> (A18)
4.4.1.1.15 <u>(I) THRUSTER 1D/1S FAIL</u> (A19)
4.4.1.1.16 <u>(I) THRUSTER 1U/1F FAIL</u> (A20)
4.1.1.1.17 <u>(I) THRUSTER 2U/2S FAIL</u> (A21)
4.1.1.1.18 <u>(I) THRUSTER 2D/2F FAIL</u> (A22)

(LIS 370-4) Contact Closure

Eight signals from manual panel switches to indicate status of RCS jet pairs to the LGC. With this information the LGC will modify appropriately the jet select program in recognition of failed jets.

4.4.1.1.19 <u>(I) PITCH GIMBAL OFF</u> (D09)
4.4.1.1.20 <u>(I) ROLL GIMBAL OFF</u> (D10)

(LIS 370-4) Contact Closure

Two signals from automatic failure detection circuitry to indicate status of the descent engine from gimbals to the LGC. With this information the LGC will modify the descent engine gimbal commands appropriately.

4.4.1.2 RCS Jet Discretes

4.4.1.2.1 (O) RCS JET 4D (+X) (E01)

4.4.1.2.2 (O) RCS JET 3D (+X) (E02)

4.4.1.2.3 (O) RCS JET 2D (+X) (E03)

4.4.1.2.4 (O) RCS JET 1D (+X) (E04)

4.4.1.2.5 (O) RCS JET 3U (-X) (E05)

4.4.1.2.6 (O) RCS JET 2U (-X) (E06)

4.4.1.2.7 (O) RCS JET 4U (-X) (E07)

4.4.1.2.8 (O) RCS JET 1U (-X) (E08)

4.4.1.2.9 (O) RCS JET 2S (+Y) (E09)

4.4.1.2.10 (O) RCS JET 1S (+Y) (E10)

4.4.1.2.11 (O) RCS JET 4S (-Y) (E11)

4.4.1.2.12 (O) RCS JET 3S (-Y) (E12)

4.4.1.2.13 (O) RCS JET 2F (+Z) (E13)

4.4.1.2.14 (O) RCS JET 3F (+Z) (E14)

4.4.1.2.15 (O) RCS JET 4F (-Z) (E15)

4.4.1.2.16 (O) RCS JET 1F (-Z) (E16)

(LIS 370-4) (See Figure 4.3.4.1-7)

Sixteen signals from the G&N LGC to each of the SCS reaction jet drivers. The presence of a low impedance path to zero volts through the G&N indicates "jet-on" command.

4.4.1.2.17 (O) -ROLL GIMBAL TRIM (D08)

4.4.1.2.18 (O) +ROLL GIMBAL TRIM (D07)

4.4.1.2.19 (O) -PITCH GIMBAL TRIM (D06)

4.4.1.2.20 (O) + PITCH GIMBAL TRIM (D05)

(LIS 370-4) dc voltage

Four signals from the LGC to drive descent engine trim gimbal at the fixed rate of 0.2 degree per second.

4.4.1.2.21 <u>(O) INCREASE THROTTLE</u> (D01) 4.4.1.2.22 <u>(O) DECREASE THROTTLE</u> (D02)	(LIS 370-4) Pulse Train

Two signals from the LGC to descent engine throttle servo. Signals are transformer coupled 3200-pps pulse trains which indicate increments of thrust of 3 pounds each increment.

4.4.1.2.23 <u>(O) ENGINE</u> ON (D03) 4.4.1.2.24 <u>(O) ENGINE OFF</u> (D04)	(LIS 370-4) dc voltage

Two signals from the LGC controls ascent and descent engine on and engine off. Engine on is signalled after receipt of "Engine Armed," 4.4.1.6 and after the LGC has performed proper ullage. Engine off is signalled based upon computer guidance program. Engine off at touchdown is initiated by the crew and not the LGC.

4.4.1.3 <u>Angle Data Signals</u> 4.4.1.3.1 <u>(O) PITCH ERROR</u> 4.4.1.3.2 <u>(O) YAW ERROR</u> 4.4.1.3.3 <u>(O) ROLL ERROR</u>	(LIS 350-1) 800 cps voltage

Three separate signals from the G&N IMU D/A CDU'S to the FDAI attitude error needles. Each singal is a ± 5.4 volts rms 800 cps suppressed carrier voltage analog of each axis of the G&N indication of spacecraft attitude error at a scaling of 3.3 degrees per volt. Reference 800 cps is transmitted as signal 4.4.1.8.10. These signals are available anytime the IMU is operating and aligned.

4.4.1.3.4 <u>(O) SINE INNER GIMBAL ANGLE</u> 4.4.1.3.5 <u>(O) COSINE INNER GIMBAL ANGLE</u> 4.4.1.3.6 <u>(O) SINE MIDDLE GIMBAL ANGLE</u> 4.4.1.3.7 <u>(O) COSINE MIDDLE GIMBAL ANGLE</u> 4.4.1.3.8 <u>(O) SINE OUTER GIMBAL ANGLE</u> 4.4.1.3.9 <u>(O) COSINE OUTER GIMBAL ANGLE</u>	(LIS 350-1) ac voltage

Six separate signals from the PGNCS IMU to the total attitude FDAI display via the coordinate conversion of the GASTA. Each signal is an 800-cps suppressed carrier voltage analog. Reference 800-cps is transmitted as signal 4.4.1.8.10.

4.4.1.3.10	(O) SHAFT RATE COMMAND	(LIS 370-6) ac voltage
4.4.1.3.11	(O) TRUNNION RATE COMMAND	

Two 800-cps signals from the radar D/A CDU's to transmit rate commands to the gyros on the rendezvous radar for radar antenna pointing by the PGNCS. Antenna stabilization is accomplished by the radar gimbal gyros. Designation pointing loop closure by the LGC uses the position feedback signals, 4.4.1.3.4 through 4.4.1.3.9, and 4.4.1.3.12 through 4.4.1.3-19 and LGC program to generate these rate commands.

4.4.1.3.12	(I) 1X SIN SHAFT ANGLE	(LIS 370-6) ac voltages
4.4.1.3.13	(I) 1X COS SHAFT ANGLE	
4.4.1.3.14	(I) 1X SIN TRUNNION ANGLE	
4.4.1.3.15	(I) 1X COS TRUNNION ANGLE	
4.4.1.3.16	(I) 16X SIN SHAFT ANGLE	
4.4.1.3.17	(I) 16X COS SHAFT ANGLE	
4.4.1.3.18	(I) 16X SIN TRUNNION ANGLE	
4.4.1.3.19	(I) 16X COS TRUNNION ANGLE	

Eight 800-cps supressed carrier signals, four 1-speed and four 16-speed, from resolver transmitters on the rendezvous radar gimbals representing the sine and cosine of the shaft and trunnion angles. Received by the radar A/D CDU's in the PGNCS. Reference 800-cps is signal 4.4.1.8.10, which is used to excite the radar resolver transmitter.

4.4.1.4 Maneuver Command Signals

4.4.1.4.1	(I) +X TRANSLATION COMM. (B01)	(LIS 370-4) Contact Closure
4.4.1.4.2	(I) -X TRANSLATION COMM. (B02)	
4.4.1.4.3	(I) +Y TRANSLATION COMM. (B03)	
4.4.1.4.4	(I) -Y TRANSLATION COMM. (B04)	

4.4.1.4.5 (I) +Z TRANSLATION COMM. (B05) (LIS 370-4) Contact Closure

4.4.1.4.6 (I) -X TRANSLATION COMM. (B06) (LIS 370-4) Contact Closure

Six separate signals from the translation controller to the LGC. Each signal is a voltage derived from a contact closure on the controller. When in the "PGNCS guidance" mode, 4.4.1.1.1, the LGC respond to these signals appropriately by sending jet commands. The resulting spacecraft response will be acceleration in the commanded direction.

4.4.1.4.7 (I) OUT OF DETENT (A03) (LIS 370-4) Contact Closure. Signal from the rotational hand controller to the LGC indicating the rotational hand controller is displaced from the center detent position and attitude rate commands are being sent on signals 4.4.1.4.8 through 4.4.1.4.10.

4.4.1.4.8 (I) PITCH RATE COMMAND (C01)

4.4.1.4.9 (I) ROLL RATE COMMAND (C02)

4.4.1.4.10 (I) YAW RATE COMMAND (C03)

(LIS 370-4) 800-cps Signal

Three analog 800-cps signals from the rotational hand controller to the LGC indicating commanded attitude rates. Spacecraft response is attitude rate proportional to signal and with polarity determined by phase of signal with respect to PGNCS 800 cps reference 4.4.1.8.10. Signal is 2.8 volts rms at 11 degree hand controller position. Scaling is 7.15 degrees per second per volt rms.

4.4.1.4.11 (I) RATE OF DESCENT DISCRETE (+) (A10)

4.4.1.4.12 (I) RATE OF DESCENT BAR (A09)

4.4.1.4.13 (I) RATE OF DESCENT DISCRETE (-) (A11)

(LIS 370-4) Contact Closure

Three discrete signals indicating position of the left-hand rate of descent command controller. Each plus or minus discrete occurring after a reset discrete (at center position of controller) indicates a command increment of local vertical rate of descent to the LGC. The LGC will recognize and respond to these commands only when the crew has selected the "Attitude Hold" mode, 4.4.1.1.2, and only during the landing approach and final approach phases by sending appropriate commands to the descent engine throttle, 4.4.1.2.21 and 4.4.1.2.22.

4.4.1.5 Timing Signals and Telemetry

4.4.1.5.1 (O) SYNC SIGNAL (LIS 370-4) (H05) Pulse Rate. Signal from the LGC which synchronizes the LM master timing equipment with the LGC. The signal is a 1024 kpps waveform transmitted from an LGC output transformer.

4.4.1.5.2 (O) SERIAL DIGITAL DATA (LIS 370-4) (H04) Pulse Train. Signal from the LGC to the PCM telemetry for transmitting to the ground for downlink and AGS initialization. Data are transmitted as a serial train of pulses representing ones, and lack of pulses represent zeros.

4.4.1.5.3 (I) START PULSE (LIS 370-4) (H01) Pulse Train. Signal from the PCM to the LGC to indicate to the LGC to start sending serial data.

4.4.1.5.4 (I) STOP PULSE (LIS 370-4) (H02) Pulse Train. Signal from the PCM to the LGC to indicate at the end of LGC downlink word transmission.

4.4.1.5.5 (I) BIT SYNC PULSES (LIS 370-4) (H03) Synchronizes serial bit transmitted through downlink between start and stop pulses.

4.4.1.5.6 (I) UPLINK ZERO (J14) }
4.4.1.5.7 (I) UPLINK ONE (J13) } (LIS 370-4) Pulses

Two signals provide the LGC the capability to receive uplink data. Used for ground test prelaunch communication to the computer and for the LM mission programmer in unmanned flight. See 4.4.1.15.13 and 4.4.1.15.14.

4.4.1.5.8 through 4.4.1.5.61 (O) ANALOG TELEMETRY (LIS 370-3) Analog Voltage. Fifty-four analog instrumentation signals from the PGNCS signal conditioner.

4.4.1.6 Caution and Warning Signals

4.4.1.6.1 (O) LGC POWER FAILURE WARNING (I01) (LIS 370-4) Contact Closure. Also called "CGC Warning" and "Computer Warning." Contact Closure within the main panel DSKY to the main panel master caution and warning lights indicating the detection of a computer power loss or computer circuit failure.

4.4.1.6.2 <u>(O) INRTL WARNING</u> (I03) (LIS 370-4) Contact Closure. Also called "Inertial Subsystem Warning." Contact closure within the main panel DSKY under LGC program control to the main panel master caution and warning lights indicating the detection of an ISS error. The LGC can be programmed to indicate on the DSKY whether the detected error is from the accelerometers, the gyro stabilization loop, or the inertial CDU's.

4.4.1.6.3 <u>(O) PGNCS FAILURE CAUTION</u> (LIS 370-4) (I02) Contact Closure. Contact closure within the main panel DSKY under LGC program control to the main panel master caution and warning lights indicating the detection of IMU temperature out of limits, gimbal lock caution, tracker caution, computer restart, or program caution.

4.4.1.7 <u>Unused Capacity</u>

4.4.1.7.1 <u>(O) MONITOR</u> (+) (K01)
4.4.1.7.2 <u>(O) MONITOR</u> (-) (K02)
(LIS 370-4) Pulses

Two signals for possible use in monitoring lunar landing. Not connected to spacecraft display.

4.4.1.7.3 <u>(O) CROSSLINK "O" OUT</u> (M01)
4.4.1.7.4 <u>(O) CROSSLINK "1" OUT</u> (M02)
4.4.1.7.5 <u>(I) CROSSLINK "O" IN</u> (M03)
4.4.1.7.6 <u>(I) CROSSLINK "1" IN</u> (M04)
(LIS 370-4) Pulses

Four signals providing capability for LM computer to communicate with another LGC or AGC. (Not Used)

4.4.1.7.7 <u>(I) 4 JET SELECT</u> (A24) (LIS 370-4) Contact Closure. Input discrete which would allow astronaut to command four-jet operation in X translation and pitch and roll rotation. (Not used)

4.4.1.7.8 <u>(I) BLOCK UPLINK</u> (A14) (LIS 370-4) Contact Closure. Signal from panel switch indicating the computer should block uplink data. (Not used)

4.4.1.8 <u>Power, Reference, and Lighting</u>

4.4.1.8.1 <u>(I) LGC POWER BUS</u> (LIS 390-2) 28 vdc. 28 vdc power to energize LGC.

4.4.1.8.2 <u>(I) IMU OPERATE POWER BUS</u> (LIS 390-2) 28 vdc. 28 vdc power to energize inertial subsystem operating loads including the gimbal system and the IMU and Radar CDU's.

4.4.1.8.3 <u>(I) IMU STANDBY POWER BUS</u> (LIS 390-2) 28 vdc. 28 vdc power to provide standby heat to the IMU.

4.4.1.8.4 <u>(I) AOT HEATER POWER</u> (LIS 390-2) 28 vdc. 28 vdc power to energize the AOT heater.

4.4.1.8.5 <u>(I) DSKY KEYBOARD AND PANEL POWER</u> (N02) (LIS 370-4) 115 volts nominal and 400 cps power for the DSKY keyboard and panel illumination. Voltage varied by GAEC dimming control 20 to 75 volts.

4.4.1.8.6 <u>(I) DSKY STATUS POWER</u> (N03)
4.4.1.8.7 <u>(I) DSKY CAUTION POWER</u> (N04)
} (LIS 370-4) Voltage

Two separate two-line 5-volt nominal power for DSKY status and caution indicators. Voltage is varied for dimming 2 to 5 volts by GAEC dimming control.

4.4.1.8.8 <u>(I) DSKY EL NUMERIC DIMMER</u> (N01) (LIS 370-4) Resistance. Variable resistance on GAEC dimming shaft to vary the DSKY numerics brightness.

4.4.1.8.9 <u>(I) AOT RETICLE</u> (LIS 390-2) 400 cps. 115 volts rms 400 cps power from the spacecraft to the AOT dimmer for illumination of the AOT reticle.

4.4.1.8.10 <u>(O) AC POWER OUT</u> (LIS 370-7) 800 cps. 28 volts rms 800 cps power from the PGNCS to provide power and reference to the GASTA, FDAI, autopilot, hand controllers, and rendezvous radar.

4.4.1.9 <u>Rendezvous Radar</u>

4.4.1.9.1 <u>(I) RDR IN 'O' (DATA FLOW)</u> (F02)
4.4.1.9.2 <u>(I) RDR IN '1' (DATA FLOW)</u> (F01)
} (LIS 370-4) Pulse

Digital data from RR to computer (range, range rate)

4.4.1.9.3 <u>(O) RR RANGE CATE STROBE</u> (F04)
4.4.1.9.4 <u>(O) RR RANGE RATE GATE STROBE</u> (F05) } (LIS 370-4) Pulse

Pulses from computer to gate info into high-speed counter.

4.4.1.9.5 <u>(O) RR RESET STROBE</u> (F03) (LIS 370-4) Pulse. Reset

4.4.1.9.6 <u>(O) RR COUNTER READOUT</u> (F06) (LIS 370-4) Pulse. Sync pulses from high-speed computer to counter.

4.4.1.9.7 <u>(I) RR DATA GOOD</u> (F07) (LIS 370-4) Contact Closure. Indicated RR Range Tracker and Frequency Tracker locked on.

4.4.1.9.8 <u>(I) RR POWER ON AND AUTO MODE</u> (F08) (LIS 370-4) Contact Closure. Indicates the radar in ON and in AUTO mode.

4.4.1.9.9 <u>(O) RR AUTO TRACK ENABLE</u> (F10) (LIS 370-4) Contact Closure. Closure removed to allow RR to lock on a return signal.

4.4.1.9.10 <u>(I) RR RANGE LO SCALE</u> (F09) (LIS 370-4) Contact Closure. Indicates the scaling of data has changed to LO scaling.

4.4.1.10 <u>Landing Radar</u>

4.4.1.10.1 <u>(I) LR IN</u> 'O' (G08)
4.4.1.10.2 <u>(I) LR IN</u> '1' (G07) } (LIS 370-4) Pulses

Two signals providing digital data from LR to computer of three components of velocity in antenna coordinates and range along the altitude beam.

4.4.1.10.3 <u>(I) LR RANGE DATA GOOD</u> (G09) (LIS 370-4) Contact Closure. Indicates LR range tracker has locked on. (Valid range data requires that velocity trackers be locked on).

4.4.1.10.4 <u>(I) LR ANTENNA POSITION 1</u> (G10) (LIS 370-4) Contact Closure. Indicates antenna in position 1 (descent).

4.4.1.10.5 <u>(I) LR ANTENNA POSITION 2</u> (G11) (LIS 370-4) Contact Closure. Indicates antenna in position 2 (Hover).

4.4.1.10.6 <u>(I) LR VELOCITY DATA GOOD</u> (G12) (LIS 370-4) Contact Closure. Indicates velocity tracker has locked on.

4.4.1.10.7 (O) LR X_A VELOCITY GATE STROBE (G03)
4.4.1.10.8 (O) LR Y_A VELOCITY GATE STROBE (G04)
4.4.1.10.9 (O) LR Z_A VELOCITY GATE STROBE (G05)
4.4.1.10.10 (O) LR RANGE GATE STROBE (G02)

(LIS 370-4) Pulses

Four signals providing pulses to gate data into radar high-speed counter.

4.4.1.10.11 (O) LR RESET STROBE (G01) (LIS 370-4) Pulse. Continuous pulse train to reset range and velocity gates.

4.4.1.10.12 (O) LR COUNTER READOUT COMMAND (G06) (LIS 370-4) Pulse. Syncs pulses from computer to radar high-speed counter for readout.

4.4.1.10.13 (O) LR ANTENNA POSITION COMMAND (G14) (LIS 370-4) Contact Closure. Contact closure in DSKY which provides signal to move antenna from position 1 to position 2.

4.4.1.10.14 (I) LR RANGE LOW SCALE FACTOR (G13) (LIS 370-4) Contact Closure. Signal which indicates scaling of altitude data to computer is at LO.

4.4.1.11 Display Data

4.4.1.11.1 (O) ALTITUDE '1' (L02)
4.4.1.11.2 (O) ALTITUDE '0' (L01)
4.4.1.11.3 (O) ALTITUDE RATE '1' (L04)
4.4.1.11.4 (O) ALTITUDE RATE '0' (L03)

(LIS 370-4) Pulses

Four signals providing digital data to tape-drive meters on panel to display altitude and altitude rate. Data computed and transmitted only on receipt of "Display Inertial Data" discrete 4.4.1.1.10.

4.4.1.11.5 (O) ALTITUDE METER SYNC (L05) (LIS 370-4) dc Voltage. Signal providing for synchronization of digital display data. (Not Used)

4.4.1.11.6 (O) FORWARD VELOCITY
4.4.1.11.7 (O) LATERAL VELOCITY

(LIS 350-2) dc Voltage

Two dc signals from the (" radar") D/A CDU's transmitting local horizontal velocity data to the panel cross pointer display meter. These data will be transmitted on receipt of the "Display Inertial Data" discrete 4.4.1.1.10. Signals cover ± 200 feet per second of velocity display with no scale shifting in transmitted voltage.

4.4.1.12 Abort Guidance System Initialization

4.4.1.12.1	(O) A_{IG} PLUS	(LSP 300-2) Pulses
4.4.1.12.2	(O) A_{IG} MINUS	
4.4.1.12.3	(O) A_{MG} PLUS	
4.4.1.12.4	(O) A_{MG} MINUS	
4.4.1.12.5	(O) A_{OG} PLUS	
4.4.1.12.6	(O) A_{OG} MINUS	

Six signals from the IMU A/D CDU's providing the AGS with the IMU gimbal angle increments that are simultaneously sent to the LGC so that the AGS and LGC hold the same IMU angle data in their counters. The AGS uses these data to initialize the AGS attitude sensing.

4.4.1.12.7 (O) ZERO CDU (LIS 300-2) Pulse Train. Signal to the AGS to provide for simultaneously zeroing the angle counters in the AGS, CDU, and LGC. When this signal is removed the CDU registers increments up to the present IMU gimbal angles while sending the same increments to the LGC and AGS counters, 4.4.1.12.1 through 4.4.1.12.6.

4.4.1.12.8 (O) DIGITAL DATA (AGS INITIALIZATION) (M05) (LIS 370-4) Pulses. Second buffered output in parallel with "Downlink Data" 4.4.1.5.2 and containing at specified times the state vector initialization of the Abort Guidance System from the LGC.

4.4.1.13 Astronaut Interface Displays

4.4.1.13.1 (DSKY) COMP ACTY (MH01-01388) Green E. L. Signifies computer activity in processing data.

4.4.1.13.2 (DSKY) PROG (MH01-01388) Green E. L. Two digit (octal) 00 to 77 signifies major program mode of computer activity.

4.4.1.13.3 (DSKY) VERB (MH01-01388) Green E. L. Two digit (octal) 00 to 77 signifies verb chosen by astronaut or signaled to astronaut.

4.4.1.13.4 (DSKY) NOUN (MH01-01388) Green E. L. Two digit (octal) 00 to 77 signifies noun chosen by astronaut or signaled to astronaut.

4.4.1.13.5 (DSKY) X DATA (MH01-01388) Green E. L. Five digits (decimal) and sign ± 00000 to ±99999 displaying X variable data.

4.4.1.13.6 (DSKY) Y DATA (MH01-01388) Green E. L. Five digits (decimal) and sign ± 00000 to ±99999 displaying Y variable data.

4.4.1.13.7 (DSKY) Z DATA (MH01-01388) Green E. L. Five digits (decimal) and sign ±00000 to ±99999 displaying Z variable data.

4.4.1.13.8 (DSKY) UPLINK ACTY (MH01-01388) White I. L. Uplink activity indicates the computer is receiving data from up-telemetry.

4.4.1.13.9 (DSKY) NO ATT (MH01-01388) White I. L. Signifies that the ISS is not suitable for use as an attitude reference.

4.4.1.13.10 (DSKY) STBY (MH01-01388) White I. L. Indicates that the computer is in the power-saving standby mode.

4.4.1.13.11 (DSKY) KEY REL (MH01-01388) White I. L. Keyboard release indicates that the internal program has attempted to display information on DSKY but found display system program busy (Flashing).

4.4.1.13.12 (DSKY) OPP ERR (MH01-01388) Yellow I. L. Operator error, keyboard and display program has detected improper use of keyboard by astronaut (Flashing).

4.4.1.13.13 (DSKY) TEMP (MH01-01388) Yellow I. L. Indicates IMU temperature in excess of ±5 degrees Fahrenheit from set point.

4.4.1.13.14 (DSKY) GIMBAL LOCK (MH01-01388) Yellow I. L. Indicates an IMU middle gimbal angle in excess of ±70 degrees and there is, consequently, a possibility of reaching gimbal lock.

4.4.1.13.15 (DSKY) PROG (MH01-01388) Yellow I. L. Indicates a program internal check has failed.

4.4.1.13.16 (DSKY) RESTART (MH01-01388) Yellow I. L. Internal detection in computer signifies abnormal operation.

4.4.1.13.17 (DSKY) TRACKER (MH01-01388) Yellow I. L. Indicates star tracker is in acquisition status but star not acquired. (Used to signal rendezvous radar is in acquisition status but CM target is not acquired.)

4.4.1.13.18 Counter RETICLE ROTATION (No ICD). Counter on AOT, reading reticle rotation set by control 4.4.1.14.5. Readout is over full revolution to nearest 0.01 degree by interpolation.

4.4.1.14 Astronaut Interface Controls

4.4.1.14.1 Button MARK X (A 25)
4.4.1.14.2 Button MARK Y (A26)
4.4.1.14.3 Button REJECT MARK (A28)
} (LIS 370-4)

Three buttons on the computer control and reticle dimmer on the AOT protection frame to be used by the astronaut during IMU alignment with the AOT. Provides computer with discrete indicating the instant of time target crosses reticle lines.

4.4.1.14.4 Knob AOT POSITION (No ICD). On AOT provides for rotating the objective end of the AOT into the detented viewing and stowage positions.

4.4.1.14.5 Knob AOT RETICLE ROTATION (No ICD). On AOT provides means of rotating reticle in field of view of AOT. Angle is read out on counter display.

4.4.1.14.6 Knob AOT RETICLE DIMMER (No ICD). On AOT computer control and reticle dimmer provides dimming of the reticle.

4.4.1.14.7 DSKY BUTTONS (MH01-01388)

4.4.1.14.7.1 DSKY VERB (ICD MH01-01388). Button signifies next two digits are verb code.

4.4.1.14.7.2 DSKY NOUN (ICD MH01388). Button signifies next two digits are noun code.

4.4.1.14.7.3 DSKY +(PLUS) (ICD MH01-01388). Button signifies next five digits are positive decimal numbers.

4.4.1.14.7.4 DSKY -(MINUS) (ICD MH01-01388). Button signifies next digits are negative decimal numbers.

4.4.1.14.7.5 through 4.4.1.14.7.14 DSKY DIGITS (ICD MH01-01388). Button enters digits 0 through 9.

4.4.1.14.7.15 DSKY CLR (ICD MH01-01388). Button clears data in numeric display.

4.4.1.14.7.16 DSKY STBY (ICD MH01-01388). Button puts computer in power-saving standby mode, and takes it out of standby.

4.4.1.14.7.17 DSKY KEY REL (ICD MH01-01388). Button releases keyboard from manual operation for display of program data or requests.

4.4.1.14.7.18 DSKY ENTR (ICD MH01-01388). Button enters keyboard data into program.

4.4.1.14.7.19 DSKY RSET (ICD MH01-01388). Button resets appropriate lights under program control.

4.4.1.15 LM Mission Programmer

4.4.1.15.1 through 4.4.1.15.8 (O) COMMAND WORD (J01 through J08) (LIS 370-4) dc Voltage. The first 8 bits of the normal DSKY output lines from the LGC used in unmanned operation as command signals from the LGC to the Program Coupler Assembly of the LMP.

4.4.1.15.9 through 4.4.1.15.12 (O) ENABLE WORD (J09 through J12) (LIS 370-4) dc Voltage. Bits 12 through 15 of the normal DSKY output lines. In unmanned LMP use, the presence of all "ones" indicates an LMP command word.

4.4.1.15.13 (I) UPLINK DATA BINARY "O" (J14) } (LIS 370-4) Pulses
4.4.1.15.14 (I) UPLINK DATA BINARY "1" (J13) }

Two signals used during unmanned LMP operation to send information from the digital command assembly into the LGC. (See 4.4.1.5.6 and 4.4.1.5.7.)

4.4.1.15.15 (O) INRTL REF (WARNING) (J15) (LIS 370-4) Contact Closure. Warning signal to the LMP programmer coupling assembly indicating PGNCS detected failure in the inertial subsystem.

4.4.1.15.16 (O) LGC (WARNING) (J16) (LIS 370-4) Contact Closure. Warning signal to the LMP programmer coupling assembly indicating PGNCS detected failure in the LGC.

4.4.2 Contractor Interfaces

This section defines the interfaces between contractors, provides a brief status report and also furnishes an ICD matrix, key, and description.

4.4.2.1 ICD Status

For current status of all LM ICD's, refer to latest issue of GAEC LM Interface Control Documentation Status Report LSR-540-XX.

4.4.2.2 ICD Matrix

The ICD for a particular interface can be determined by selecting ICD key numbers from the ICD matrix shown in Figure 4.4-3. The ICD key number should then be referenced to following paragraph for complete description and number.

4.4.2.3 ICD Key and Description

Matrix No.	Title	ICD No.
1	IMU-AOT-NVB-PTA Installation	LID-280-10004
2	Abort guidance section electrical interface with PGNCS	LIS-300-10002
3	LGC-PSA Installation	LID-340-10000
4	AOT Field of View	LID-340-10008
5	DSKY Installation	LID-340-10010
6	Total attitude signals, electrical interface between the PGNCS and the GASTA	LIS-350-10001
7	LM-PGNCS Lateral and Forward Velocity - Electrical Interface	LIS-350-10002
8	LM-PGNS Measurements List Interface Provisions	LIS-370-10003
9	LGC LM Electrical Interface	LIS-370-10004
10	PGNCS to Rendezvous Radar Angle Electrical Interface	LIS-370-10006
11	LM-PGNCS 800 cps Excitation Signal Electrical Interface	LIS-370-10007

		GAEC																								
		G & C		S & C																						
				CES													PROPULSION		OTHER LEM SYSTEMS							
		WIRING	ELECTRICAL	ATCA	DECA	ENG SEQ	T/TCA	ACA	GDA	GASTA	FDAI	CONTROL PANEL	AGS	RCS	LR	RR	AE	DE	INSTRUMENTATION	POWER	THERMAL	ENVIRONMENT	STRUCTURE	WEIGHT	PYROTECHNICS	GSE-ACE
MIT	LGC	13	9	9,25 24	25 24,9	9	9,25 24	25 24	9		6,11	1,25,7 9,24	9,24 25,9	9,24 25	9,25 24	25	24 25	9,24 25	8,25 12,24	14	21	22	3,14	20	9,25 24	18
	CDU's										6,11	6,7	2			10				14	21	22	19	20		18
	PSA/SCA	13					11	11		11	6,11	6,11	2		10 11	10 11			24,9 25,8	14	21	22	3,19	20		18
	IMU	13	16							6,25 24		6	24 25							14 16	21	22	1,19	20		16 18
	AOT																				21	22	1 4,19	20		
	DSKY														9	9,10			8,12	14	21	22	5,19	20		
	D AND C																			14	21	22	19	20		
	PTA																			14	21	22	1,19	20		
	NVB																					22	1,19	20		
	MATERIALS																					23				

Figure 4.4-3. ICD Matrix

Matrix No.	Title	ICD No.
12	S-band steerable antenna functional interface with PGNCS	LID-380-10001
13	LM-PGNCS wiring electrical interface	LIS-390-10001
14	G&N prime power requirements and characteristics	LIS-390-10002
15	Inertial components temperature system mechanical interface	LID-410-10002
16	Inertial components temperature System interface	LIS-410-10002
17	PSA adapter module mechanical interface	LID-410-10003
18	PSA adapter module electrical interface	LIS-410-10003
19	PGNCS installation fixtures and procedures	LIS-420-10001
20	Weight requirements for GFE/PGNCS equipment	LIS-490-10001
21	G&N thermal dissipation and cooling requirements	LIS-510-10001
22	LM design environment	LIS-520-10001
23	GAEC-MIT materials compatibility requirements	LIS-520-10002
24	G&N functional interface flow diagram	LID-540-10001
25	LM-PGNCS functional interface requirements	LIS-540-10001

The following ICD's are no longer valid (some have been cancelled and others have been incorporated into other ICD's):

LIS-290-10001

LIS-290-10002

LIS-290-10003

LIS-290-10005

Matrix No.	Title	ICD No.
		LIS-290-10006
		LIS-300-10003
		LIS-340-10009
		LID-370-24001
		LID-370-24001
		LIS-410-10001
		LIS-480-10002
		LIS-480-10006
		LIS-540-10003
		LIS-540-10004

4.5 WEIGHT DATA

For total weight status of LM vehicle refer to latest LM Mass Property Report GAEC Document LED-490-XX.

4.5.1 Weight Budget

4.5.1.1 SCS

The maximum allowable weight for each SCS subassembly is shown in Table 4.5.1.1-1.

4.5.1.2 G&N

The design load weights for the G&N system are as appears in ICD LIS-490-10001 as modified by NASA letter EG 43-422-65-766, 27 October 1965. These weights are tabulated in Table 4.5.2.1-2, Weight Status of LM PGNCS.

4.5.2 Weight Status

4.5.2.1 SCS (See LED-490-XX).

4.5.2.2 G&N

For weight status of LM PGNCS, refer to Table 4.5.1.1-2. For mass properties data for LM PGNCS, refer to Table 4.5.1.1-3.

Table 4.5 1.1-1: SCS Weight Budget

Item Item	Weights (max) (lb)
ATCA	27.0
DECA	7.3
RGA	2.0
T/TCA (2)	5.25 each
GASTA	7.0
ACA (2)	4.75 each
DEDA	8.4
ASA	20.7
AEA	32.5
GDA (2)	3.55 each
FDAI (2)	6.96 each

Note: These data extracted from procurement specification for each individual assembly. For current total SCS Weight Budget and Status, see GAEC Report LED-490-XX.

Table 4.5.1.1-2. Weight Status of LM PGNCS

LM PGNCS Equipment	Previous Status	Change	Current Status	Design Load Weight*
IMU	42.4 (M)	0.0	42.4 (M)	80.0
AOT (including eyepiece and bellows)	23.1 (E)	0.0	23.1 (M)	
NVB	5.1 (E)	0.0	5.1 (M)	
HARNESS "B" Supported by the NVB	0.6 (E)	0.0	0.6 (E)	
HARNESS "B" Supported by the PTA	0.8 (E)	0.0	0.8 (E)	21.0
HARNESS "B" Supported by the structure	3.1 (E)	0.0	3.1 (E)	
PTA	14.3 (M)	0.0	14.3 (M)	
HARNESS "A"	14.6 (E)	0.0	14.6 (M)	22.0
LGC (with six rope modules - mag trays)	69.0 (E)	+0.5	69.5 (M)	65.0
DSKY	17.5 (E)	0.0	17.5 (M)	20.0
AOT Control Unit (CCRD)	1.6 (M)	0.0	1.6 (M)	2.0
CDU	37.2 (M)	+0.3	37.5 (M)	37.0
PSA	17.5 (E)	0.0	17.5 (E)	28.2
SCA (Operational Flights)	7.2 (E)**	0.0	7.2 (E)	
TOTAL	254.0	+0.8	254.8	---
The reported total weight exceeds the 245.0 pound total control weight by 9.8 pounds.				
Bare Guidance Systems - IMU, LGC, IMU portions of the CDU's and IMU Support Electronics			167.3	

* Design Load Weights are taken from ICD LIS-490-10001 as signed by R. A. Gardner (NASA/MSC) on 29 March 1966.

**The weight of a qualification flight signal conditioner assy is 9.2 (E) pounds.

†The Total Control Weight is specified in NASA letter EG-26-233-66-565, 18 August 1966.

Table 4.5.1.1-3. Lunar Module PGNCS Mass Property

LEM GN&C EQUIPMENT	Center of Gravity - Inches				Moment of Inertia - Slug-ft^2			
	$\overline{x}$	$\overline{y}$	$\overline{z}$	±Error	I_x	I_y	I_z	±Error
IMU	307.0	0	49.9	5%	22.90	886.76	863.97	1%
Navigation Base	309.0	0	54.4	5%	2.57	3.39	0.86	1%
AOT								
AOT Control Unit (CCRD)								
PTA								
Harness "B"								
DSKY	254.0	0	58.6	10%	12.98	256.47	243.50	1%
LGC	266.0	0	-22.9	5%	8.24	1034.57	1027.81	1%
CDU	252.2	0	-22.8	10%	4.69	553.63	549.28	2%
PSA (not integral unit) / Signal Conditioner Assy	240.0	0	-22.8	5%	3.04	308.63	305.88	5%
Harness "A"								
TOTAL								

*Updated per MIT Status Report E-1142, September 1966

4.6 POWER DATA

Detailed power data for a typical lunar mission can be obtained in the following document:

> GAEC Report LED-390-10000, Latest Revision, LEM Electrical Load Analysis.

This document contains power requirements, load profiles, budgets, energy requirements, etc., for SCS, G&N and other LM systems.

4.6.1 Power Budget

(Refer to LED-390-10000).

4.6.2 Power Requirements by Unit

4.6.2.1 SCS

(Refer to GAEC Report LED-390-10000, Latest Revision.

4.6.2.2 G&N

Electrical power and energy reporting is based upon the inflight spacecraft sequence of events for the Design Reference Mission as developed by the Apollo Mission Planning Task Force (AMPTF). (Reference GAEC Report Volume III - LED-540-12)

Figure 4.6-1 presents the magnitude and distribution of G&N power dissipated on a subsystem level. It is assumed that power is drawn from the spacecrafts' primary +28 vdc supply and 400 cps - 115 vac single-phase inverter.

Intermittent power peaks can exist, particularly during operation of displays and controls at random times. The energy content in these peaks is considered negligible.

All values (except those mentioned above) are actual expected levels of power. No margin factor has been applied to protect against possible differences between actual loads which will be experienced and the calculated levels quoted. Thus, these values should not be taken as "not to exceed" extremes.

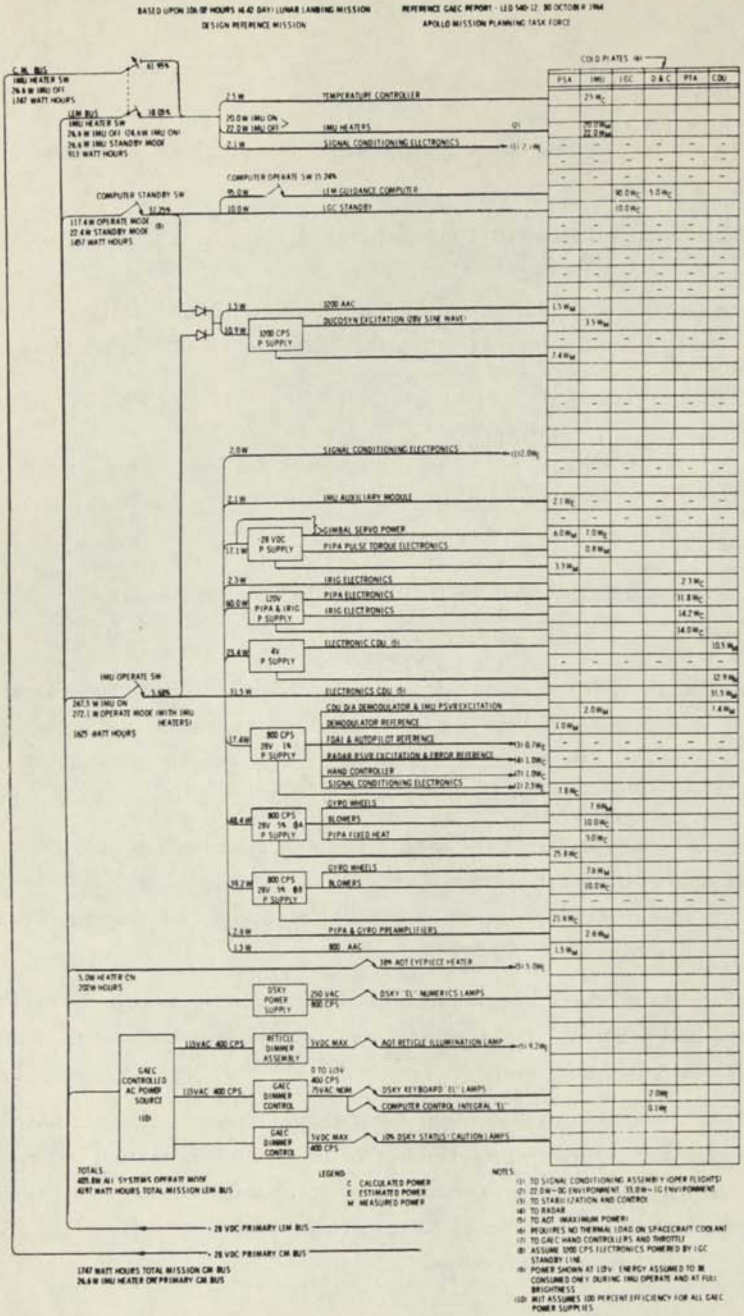


Figure 4.6-1. Lunar Module Guidance and Navigation Load on +28 VDC Primary

The following interface control document serves as the guideline for reporting power figures:

LIS-390-10002 "PGNCS Prime Power Requirements and Characteristics."

4.6.3 Power Profiles

Detailed power profiles are available in LED-390-10000.