

LUNAR MODULE RADARS

STUDY GUIDE

For Training Purposes Only

February 1967

Prepared by William Barber  
LM Training



### Preface

This document is designed to support  
the LM Radar Familiarization Course  
30417.



## TABLE OF CONTENTS

	<u>Page</u>
PREFACE	i
CONTENTS	ii
LIST OF FIGURES	iii
CHAPTER I Introduction to the LM Primary Guidance and Navigation Section	
CHAPTER II Rendezvous Radar	
CHAPTER III Transponder	
CHAPTER IV Landing Radar	
CHAPTER V Controls and Displays	



# LIST OF FIGURES

<u>Title</u>	<u>Figure</u>	<u>Page</u>
Photo of Lunar Module	1-1	25
Lunar Module	1-2	26
LM Subsystems Interface	1-3	27
Guidance, Navigation and Control Subsystem Basic Block Diagram	1-4	28
Guidance, Navigation and Control Subsystem Block Diagram	1-5	29
Functions of PGNS	1-6	30
PGNS Requirements	1-7	31
PGNS Block Diagram	1-8	32
PGNS Equipment Location	1-9	33
ISS Equipment	1-10	34
Alignment Optical Telescope	1-11	35
CSS Equipment	1-12	36
Profile of LM Descent to Lunar Surface	1-13	37
LM Powered Descent Profile	1-14	38
Nominal Descent Trajectory	1-15	39
Profile of LM Ascent from Lunar Surface	1-16	40
LM Rendezvous Radar	2-1	54
RR Electronic Assembly	2-1A	55
RR Antenna Assembly	2-2	56
AA Location of Parts	2-3	57
RR Antenna Modes	2-4	58
RR Angular Coverage	2-5	59
RR AA Heater Circuit	2-6	60
RR Block Diagram	2-7	61
RR Displays and Controls	2-8	62
RR Antenna Control Block Diagram	2-9	63
RR Interface	2-10	64
Transponder Electronics Assembly	3-1	73
Location of Transponder in CSM	3-2	74
Transponder Block Diagram	3-3	75
Transponder Controls and Displays	3-4	76
LR Electronic Assembly	4-1	94
LR Antenna Assembly	4-2	95
LR Concepts	4-3	96
LR Antenna Tilt Positions	4-4, 5, 6,	97, 8, 9
LR Antenna Beam Configuration	4-7	100
LR Block Diagram	4-8	101
LR Displays and Controls	4-9	102
LR Data Good Logic	4-10	103
LR Interface	4-11	104
LM Displays and Controls Photo	5-1	114



# LIST OF FIGURES (Cont.)

<u>Title</u>	<u>Figure</u>	<u>Page</u>
IM Displays and Controls	5-2	115
Panel I	5-3	116
Panel II	5-4	117
Panel III	5-5	118
Panel XI	5-6	119
Panel XVI	5-7	120
FDAI	5-8	121
X-Pointer	5-9	122
ALT/Range Indicator	5-10	123
RR Displays and Controls	5-11	124
LR Displays and Controls	5-12	125
PGNS Displays and Controls	5-13	126
PGNS and Radar Composite Controls and Displays	5-14	127



## CHAPTER I

### INTRODUCTION TO THE LM PRIMARY GUIDANCE AND NAVIGATION SECTION

One of the major subsystems of the Lunar Module is the Guidance, Navigation and Control Subsystem. This is comprised of the Primary Guidance and Navigation Section (PGNS), Control Electronics Section (CES) and the Abort Guidance Section (AGS).

This chapter describes the Primary Guidance and Navigation Section requirements for the Lunar Module (LM), the interfaces of the PGNS with other subsystems, the PGNS equipment, and the PGNS functions as related to the LM mission.

#### 1. PGNS REQUIREMENTS FOR THE LM MISSION

The PGNS provides the data measuring equipment, data processing capabilities and control functions necessary to accomplish descent, lunar landing, ascent, rendezvous and docking with the Command Service Module (CSM). To accomplish these goals the PGNS performs the following functions: inertial guidance, navigation, and flight control. The PGNS also provides a means for establishing an inertial reference by use of an optical instrument.

A. **INERTIAL GUIDANCE.** The inertial guidance portion of the PGNS employs three (3) accelerometers mounted on a gyroscopically stabilized three degree of freedom, gimbal-mounted platform and a central data processing device. The inertial guidance portion senses acceleration and attitude changes instantaneously and provides incremental velocity and attitude information to the LM Guidance Computer (LGC) for the generation of attitude change and thrust commands.

B. **NAVIGATION.** The optical portion of the PGNS has one function: to provide data for establishing an inertial reference. This requires the use of an Alignment Optical Telescope (AOT) to provide accurate line of sight angles to celestial bodies.

The navigation performed by the PGNS is dependent upon the "software" capability of the central data processing device with data obtained from other portions of the PGNS and other LM systems.

As an aid to navigation the rendezvous radar is used to track a transponder on the CSM or a transponder on the lunar surface. The radar interfaces with the LGC for automatic antenna positioning towards the CSM and data transfer of radar tracking angles, range and range rate.



C. FLIGHT CONTROL. The PGNS consists of the sensing and monitoring devices necessary to determine the LM vehicle position, velocity, and attitude and provides output signals necessary to control or correct the vehicle's motion. Using inputs from the inertial guidance portion of the PGNS and landing radar, the LM Guidance Computer calculates solutions to guidance equations during the lunar landing operation and generates steering signals to maintain the vehicle on a satisfactory trajectory. During descent for lunar landing, the PGNS controls the descent motion of the LM by issuing engine on-off, throttle, and trim commands to the descent engine and translation and rotation commands to the reaction control jets. During ascent, control of the LM is accomplished by the PGNS using data from its various inputs and issuing engine on-off, steering, and attitude commands. Rotation and translation commands are also provided during the rendezvous and docking mission phases utilizing inertial, radar and manual inputs to the PGNS.

## 2. PGNS INTERFACE

The PGNS interfaces directly with three LM subsystems: electrical power subsystem, environmental control subsystem and instrumentation subsystem. The PGNS interfaces indirectly with the communications subsystem and also issues commands to the propulsion subsystem and reaction control subsystem through the control electronics section.

A. LM SUBSYSTEMS INTERFACE. A description of the seven LM subsystems and their relationship to the PGNS is contained in the following paragraphs.

Electrical Power Subsystem (EPS). The EPS provides 28 vdc prime power to the PGNS. This power originates from six silver-zinc batteries, four in the descent stage and two in the ascent stage. The descent batteries are rated at 400 ampere-hours each. The ascent batteries are rated at 300 ampere-hours each.

Environmental Control Subsystem (ECS). The ECS is used to sustain life in space by providing breathable atmosphere, acceptable temperatures, food and water, and waste disposal. In addition to these functions, the ECS circulates an ethylene glycol-water coolant about the temperature sensitive electronics equipment in the PGNS and other LM subsystems to provide thermal stability. The inertial measurement unit (IMU) has coolant circulated through its case while the other major components of the PGNS are mounted on cold plates through which the coolant is circulated to provide temperature control.



Instrumentation Subsystem. The instrumentation subsystem senses physical data, monitors the various subsystems of the LM during manned phases of the mission, performs inflight and lunar surface checkout, prepares LM status data for transmission to earth, provides timing frequencies, and stores voice data when the LM is unable to transmit to earth. This system contains sensors, the signal conditioning electronics assembly (SCEA), the caution and warning electronics assembly (CWEA), the pulse code modulation and timing electronics assembly (PCMTEA), and the data storage electronics assembly (DSEA).

The scientific instruments which will be used by the astronauts during their lunar stay are also included in this subsystem.

Control Electronics Section (CES) and Abort Guidance Section (AGS) of GN & CS. The CES provides data processing for flight control data during all phases of the LM mission. The AGS provides semi-automatic, preprogrammed flight control data to the CES in case a PGNS malfunction causes a mission abort to be initiated.

The CES consists of a control panel, an attitude and translation control assembly (ATCA), a descent engine control assembly (DECA), two gimbal drive actuator assemblies (GDA), two thrust translation controller assemblies (TTCA), two attitude controller assemblies (ACA), and a rate gyro assembly (RGA). The CES provides signals to fire any combination of the 16 thrusters in the reaction control subsystem to stabilize the LM by controlling vehicle attitude and translation during all phases of the mission.

The AGS provides abort capability from any point in the LM mission and serves the function of increasing crew safety by acting as a backup system to the PGNS. The AGS consists of three main assemblies: abort sensor assembly (ASA), abort electronics assembly (AEA), and data entry and display assembly (DEDA).

Communications Subsystem. The communications subsystem provides the link between the astronauts, the orbiting CSM, and manned space flight network (MSFN) stations on earth. The system has two radio frequency sections, one operating in the VHF range and one operating in the UHF (S-band) range. The RF sections provide two-way voice transmission between the LM and CSM, LM and MSFN, and LM and extravehicular astronaut. In addition, the UHF portion receives and transmits tracking and ranging signals from earth and transmits television, telemetry, biomedical, and suit environment data to earth.



Propulsion Subsystem. The LM utilizes separate descent and ascent propulsion subsystems, each of which is complete and independent of the other. Each subsystem consists basically of a liquid propellant rocket engine with its propellant storage, pressurization, and feed components.

The descent propulsion subsystem is contained within the vehicle descent stage and utilizes a throttleable, gimballed engine that is ignited to inject the LM into the descent transfer orbit and subsequently used in the final descent trajectory to control the rate of descent of the LM to the lunar surface. The descent engine develops 10,500 pounds of thrust in a vacuum at full throttle and 1050 pounds at minimum thrust and is capable of being gimballed  $\pm 6$  degrees in two directions. The PGNS issues the "ON-OFF" commands for the descent engine and also provides signals controlling thrust magnitude and gimbal position.

The ascent propulsion subsystem utilizes a fixed, constant-thrust engine installed along the centerline of the ascent stage midsection and includes the associated propellant feed tanks and pressurization components. The engine develops 3500 pounds of thrust in a vacuum, sufficient to launch the ascent stage from the lunar surface and place it in orbit. The PGNS issues the "ON-OFF" commands for the ascent engine.

Reaction Control Subsystem (RCS). The RCS provides rocket thrust impulses to stabilize the LM during descent and ascent and to control the vehicle attitude and translation about or along all axes. The RCS consists of 16 thrust chambers supplied by two separate and independent propellant feed and pressurization sections. The thrust chambers are mounted in clusters of four on outriggers that are arranged diametrically opposite and equally spaced around the LM ascent stage. In each cluster, two thrust chambers are aligned with respect to the vehicle vertical axis, facing in opposite directions; the other two are spaced 90 degrees apart, parallel to the vehicle's Y and Z axes. The RCS utilizes the same fuel as the ascent engine. In the event of RCS fuel depletion, the remaining ascent engine fuel can be used for RCS. The RCS can be operated in any of three modes: manual, automatic, or semiautomatic. The PGNS supplies "ON-OFF" signals, through the CES, to the valves on the desired thrust chambers during the automatic and semiautomatic modes. The fully automatic mode is normally used to provide attitude control during all mission phases except where manual control is required. It is possible to select manual control in one or two axes and retain automatic control in the other axis during all mission



phases. Semiautomatic control combines attitude hold with manual control about each axis through the astronauts' manual attitude controller, and is used primarily to control the LM during lunar landing and docking phases of the mission. In the manual modes of operation, all control commands originate from the manual attitude controller, including manual control of the thrust duration.

Automatic translation commands originate in the PGNS and are routed to the RCS similar to the attitude control signals.

B. ASTRONAUT INTERFACES. The astronaut is an active and controlling element of the PGNS. He is able to monitor information to and from the PGNS and can manually duplicate PGNS control functions. The astronaut activities are classified into the following five types:

a. Initiate Optical Sighting: The astronauts initiate optical sighting programs and identify and select celestial objects on which optical sightings are taken. These sightings are used to align the inertial measurement unit (IMU).

b. Monitor System Performance: Mode, condition, and failure lights are monitored by the astronauts to determine system status. As calculations are performed by the computer, the results are displayed for astronaut evaluation and verification with ground calculated data.

c. Load Data: Data is loaded into the LGC through the computer display and keyboard (DSKY) by the astronaut. The data includes such information as star selection, positional data, and system angles.

d. Select Mode of Operation: The astronaut, by means of displays and controls, can select the equipment modes of operation for the various spacecraft systems required to perform a particular function. In some cases, the sequencing of modes is controlled by the computer. The program being performed by the computer can be selected by the astronaut or initiated internally by the computer.

e. Provide System Backup: In case of PGNS failure, the astronaut provides manual control of the LM and performs those functions normally performed by the PGNS.



### 3. PGNS EQUIPMENT

The PGNS equipment consists of a navigation base (NB), inertial measurement unit (IMU), rendezvous radar, alignment optical telescope (AOT), power and servo assembly (PSA), pulse torque assembly (PTA), signal conditioner (SC), LM guidance computer (LGC), coupling data unit (CDU), display and keyboard (DSKY), displays and controls, and landing radar (LR). The navigation base is mounted to the upper structure at the front of the LM cabin and is used as a holding fixture for the IMU and AOT which are attached and precisely aligned to this base. The LGC and CDU are mounted on coldplates located on the upper portion of the rear compartment wall. The PSA and signal conditioner are also mounted on coldplates and are located below the LGC and CDU. The PTA is mounted on a coldplate to the rear wall of the IMU compartment. The LGC DSKY, along with other controls and displays, is located on the front wall of the LM cabin such that the astronaut can operate them while confined in his harness.

### 4. PGNS SUBSECTIONS

The PGNS is functionally divided into three major subsections: inertial, optical and radar, and computer. The subsections, or combinations of subsections, with assistance from the astronaut using displays and controls, can perform the following functions:

- a. Establish an inertial reference which is used for measurements and computations.
- b. Provide a means to align the inertial measurement unit by entering celestial body optical sighting data via the DSKY to the LGC and through interface with the Command Service Module PGNS.
- c. Provide a means to calculate the position and velocity of the LM by inertial techniques.
- d. Provide a means of accomplishing a LM and Command Service Module rendezvous by radar tracking and inertial techniques.
- e. Generate attitude control and thrust commands to maintain the LM on a satisfactory trajectory.
- f. Control throttling and gimbal trim of descent engine during lunar landing.
- g. Display pertinent data related to guidance status.



A. INERTIAL SUBSECTION. This section describes the components that comprise the inertial subsection (ISS).

#### ISS EQUIPMENT

The ISS consists of the following equipment:

- a. Inertial measurement unit (IMU)
- b. Navigation base (NB)
- c. Portions of the coupling data unit (CDU)
- d. Pulse torque assembly (PTA)
- e. Power and servo assembly (PSA)
- f. Signal conditioner (SC)

#### ISS FUNCTIONS AND INTERFACE

The ISS maintains a platform fixed with respect to a pre-selected inertial reference frame, senses acceleration along the coordinate system defined by the platform orientation and provides the computer with incremental velocity data and the orientation of the platform with respect to the spacecraft.

The ISS platform (stable member) alignment must be re-established during flight since the stable member may deviate from its alignment due to gyro drift. The CSM inertial reference or star LOS data is used for coarse alignment of the stable member. The tracking of two stars is required for fine alignment. The stable member is aligned after the computer manipulates angular data received as a result of the star tracking, combines that with the known spacecraft and IMU angles, and supplies alignment signals to the IMU.

The ISS measures acceleration along the axes of the stable member coordinate system and supplies incremental velocity data to the computer. The computer calculates the spacecraft's present position and velocity by adding the components of velocity due to gravitational field accelerations with velocity due to thrust (from ISS) and the last known position and velocity.



The ISS also generates signals which are used to control spacecraft attitude. Gimbal-mounted resolvers measure the alignment of the stable member axes with respect to the spacecraft axes. This angular displacement is fed to the computer and displayed to the astronaut on the FDAI. The astronaut or the computer can command changes to the angular displacement by controlling the firing of the reaction control system jets.

#### NAVIGATION BASE (NB).

The NB provides a rigid support for the IMU and the AOT which are precisely aligned on the NB to provide a known alignment between them.

#### Inertial Measurement Unit

The inertial measurement unit (IMU) is the primary inertial sensing device of the LM. Three rate integrating gyroscopes and three pendulous accelerometers are mounted on the innermost gimbal of a three-degree-of-freedom gimbal system. The innermost gimbal is held nonrotating with respect to the inertial space by the three gimbal servos, which derive their input error signals from the three gyroscopes. The orthogonal input axes of the three accelerometers represent the LM X, Y, and Z axes. Velocity changes along these axes are sensed by the accelerometers, which produce signals that represent incremental changes in velocity and apply them to the LGC for the calculation of total velocity. Signals proportional to the attitude of the LM are generated by the inner, middle, and outer gimbal angle resolvers. These signals are applied to the LGC and to the displays. The LGC calculates gimbal angles that would result from bringing the LM to a desired attitude and compares these with the actual gimbal angles supplied by the gimbal angle resolvers. Any difference between the actual and calculated gimbal angle results in generation of steering-error signals, which are applied to the descent engine trim gimbal system during descent and to the reaction control subsystem (RCS) jet drivers during all mission phases, to correct the attitude of the LM.

#### Coupling Data Units

The coupling data units (CDU's) convert and transfer angular information between major elements of the GN & CS. The PGNS uses five CDU's: one for the shaft axis and one



for the trunnion axis of the RR antenna, and one for each of the three gimbals of the IMU.

The two CDU's used with the RR antenna provide interfaces between the antenna and the LGC. The LGC calculates digital antenna position commands before acquisition of the CSM transponder. These signals are converted into analog form by the CDU's and applied to the antenna drive mechanism to aim the antenna toward the CSM. Tracking angle information is digitized by the CDU's and applied to the LGC.

The three CDU's used with the IMU provide interfaces between the IMU and the LGC, and between the LGC and the AGS. Each IMU gimbal angle resolver provides its CDU with analog gimbal angle signals that represent the attitude of the IM. The CDU converts these signals to digital form and applies them to the LGC. The LGC uses these signals to calculate necessary attitude or translation commands and routes the commands through the CES to the proper RCS jets. The CDU converts the steering-error signals to 800 cps analog signals and applies them to the flight director attitude indicator. IMU coarse and fine align commands, generated by the LGC are coupled to the IMU thru the CDU's.

#### Power and Servo Assembly

The power and servo assembly (PSA) provides a central mounting place for most of the PGNS amplifiers, modular electronic components, and power supplies. The PSA consists of the following subassemblies: gimbal servo align and power amplifiers, CDU electronics, power diodes and signal conditioners, and power supplies. A cold plate, through which water-glycol coolant from the ECS flows, is mounted under the PSA subassemblies to dissipate heat.

#### PULSE TORQUE ASSEMBLY (PTA)

The PTA consists of a group of modular electronic assemblies related to the IMU inertial components. The PTA is mounted close to the IMU to avoid degradation of low-level signals.

#### SIGNAL CONDITIONER (SC)

The signal conditioner converts PGNS signals to levels acceptable for telemetry transmission and provides isolation



between the component or circuit being monitored and the telemetry system. Six different modules are used in the signal conditioner to perform these functions.

#### SUMMARY OF ISS

The ISS maintains an inertial reference from which spacecraft attitude and velocity changes are measured. The ISS has the capability of controlling spacecraft attitude and velocity changes.

The IMU consists of a stable member with three degrees of freedom. Mounted on the IMU stable member are: three stabilization gyros, three accelerometers, and certain critical preamplifiers. The coupling data unit monitors the IMU gimbal angles and provides the angles to the LGC. The power and servo assembly (PSA), and pulse torque assembly (PTA), contain the support electronics for the ISS. The signal conditioner (SC) conditions PGNS parameters for downlink telemetry transmission.

B. OPTICAL AND RADAR SUBSECTION. This section describes the purpose of the optical subsection (OSS) and the OSS hardware. A description of the radar section will follow in subsequent chapters.

#### ALIGNMENT OPTICAL TELESCOPE (AOT)

The optical subsection consists of the AOT mounted on the navigation base. This provides a mechanical alignment and a common reference between the AOT and IMU. The AOT which is similar to a periscope is mounted in the forward part of the LM with the shaft axis parallel to the LM X axis and the center of the field of view approximately  $45^\circ$  from the X axis in the  $X_{NBZ_{NB}}$  plane or one of the two detent positions  $60^\circ$  to either side of the  $X_{NBZ_{NB}}$  plane. A pressure strain seal is used to seal the LM cabin from the environment of space. The purpose of the AOT is to provide angular data from star sightings for use in stable member alignment via the LGC to a reference coordinate system.

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. Reticle angle and shaft detent position for the star sightings are then 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 capability of adjusting the brightness of the AOT reticle light 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 discrete be sent to the LGC.

#### MECHANICAL CONSTRUCTION

The AOT is a unity power telescope with a 60 degree field-of-view. In construction, it is essentially an "L" shaped device approximately 36 inches in length consisting of the perpendicular intersection of two cylinders each of which is a major assembly, the 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.

#### FIELD OF VIEW

The AOT line of sight may be rotated about the Z axis to three viewing positions, forward, left, or right. Three shaft detent positions at  $0^\circ$ ,  $+60^\circ$ , and  $-60^\circ$  provide this capability. A fourth detent position at  $180^\circ$  is normally used for storage. This position reverses the prism so that its viewing face may be protected from space particles by the shield. These detent positions permit the astronaut to select a field of view of the celestial sphere free of obstructions such as the lunar surface, the vehicle structure or the sun.

#### SUMMARY OF OSS

The alignment optical telescope (AOT) is a unity power 60 degrees field-of-view optical instrument similar to a periscope. The AOT is operated manually by the astronauts with mark commands initiated by the astronauts while viewing stars through the AOT. The AOT line-of-sight may be rotated to one of three viewing positions with a fourth position used for storage.

The AOT is used for alignment of the IMU stable member while in flight or on the lunar surface. 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. When on the lunar surface, the Archimedes spiral is used for alignment sightings.

C. COMPUTER SUBSECTION. This section describes the CSS equipment and functional operation.

#### CSS PURPOSE

The computer subsection is the control and computational center of the PGNS. It performs the following functions:

- a. Solves the guidance and navigation problems for all mission phases.
- b. Provides control information to the PGNS, as well as other spacecraft systems.
- c. Displays pertinent information to the astronaut and routes data to the downlink telemetry system.
- d. Provides a means by which the astronaut can directly communicate with the PGNS.
- e. Provides direct on-off control for the reaction control jets, and ascent and descent engines.
- f. Provides gimbaling and throttling control for the descent engine.
- g. Monitors its own operation and other PGNS operations.

#### CSS EQUIPMENT

The CSS consists of the LM Guidance Computer (LGC) and a display and keyboard (DSKY). The LGC is located on the rear bulkhead of the LM ascent stage crew cabin. The DSKY is located in the center of the display and control panels below the astronauts' center panels.

#### LGC PHYSICAL DESCRIPTION.

The LGC measures approximately 24 x 12.5 x 6 inches and weighs approximately 60 pounds. It consumes 10 watts of power during standby operation and 100 watts of power



during normal operation. The LGC is mounted on a coldplate base which is cooled by a water glycol solution flowing through it.

#### LGC USE

The LGC is a core memory, digital computer with two types of memory: fixed and erasable. The fixed memory permanently stores navigation tables, trajectory parameters, programs and constants. The erasable memory stores intermediate information.

The LGC processes data and issues discrete control signals, both for the PGNS and the other spacecraft subsystems. It is a control computer with many of the features of a general purpose computer. As a control computer, the LGC aligns the stable platform of the inertial measurement unit (IMU) in the inertial subsection, and issues control commands to the spacecraft ascent and descent engines and the RCS jets. As a general purpose computer, the LGC solves guidance problems required for the spacecraft mission. In addition, the LGC monitors the operation of the PGNS.

The LGC stores data pertinent to the ascent and descent flight profiles that the spacecraft must assume in order to complete its mission. This data, consisting of position, velocity, and trajectory information, is used by the LGC to solve the various flight equations. The results of various equations can be used to determine the required magnitude and direction of thrust required. Corrections to be made are established by the LGC. The spacecraft engines are turned on at the correct time, and steering signals are controlled by the LGC to re-orient the spacecraft to a new trajectory, if required. The inertial subsection senses acceleration and supplies velocity changes to the LGC for calculating the total velocity. Drive signals are supplied from the LGC to the coupling data unit (CDU) and stabilization gyros in the inertial subsection to align the gimbal angles in the IMU. Error signals are also supplied to the CDU to provide steering capabilities for the spacecraft. CDU position signals are fed to the LGC to indicate changes in gimbal angles, which are used by the LGC to keep cognizant of the gimbal positions. The LGC receives angular information automatically inserted by the rendezvous radar during tracking of the CSM. This information is used by the LGC to calculate present position and orientation and is used to refine trajectory information. The LGC receives



velocity and range data with respect to the lunar surface during descent from the Landing Radar.

#### DSKY PHYSICAL DESCRIPTION

The DSKY measures approximately 8 x 8 x 7 inches and weighs about 17.5 pounds. Mounted on the back of the DSKY are six interchangeable driver modules, a 91 pin connector, a filler valve, and a power supply. The driver modules are used in switching of the data displayed on the DSKY. The connector provides the interface between the DSKY and the LGC. The filler valve is used to pressurize the DSKY to one atmosphere. The power supply module supplies the voltage to light the DSKY indicators.

#### DSKY USE

The DSKY provides a two-way communications link between the astronaut and the LGC. Through this communications link the following functions can 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 in the LGC.
- d. Display of the LGC modes of operation.

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. 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 PGNS status.

#### SUMMARY OF CSS.

The CSS is the control and computational center of the PGNS. It solves guidance and navigation problems; displays information; provides control information to the PGNS, CES, RCS, and propulsion systems; and monitors PGNS operation including its own.



The CSS consists of the LGC and the DSKY. The LGC is an automatic, digital, electronic computer which is a combination general purpose and control computer. The DSKY provides a two-way communication interface between the astronaut and the LGC.

## 5. DISPLAY AND CONTROL INTERFACES

The primary displays and controls of the PGNS consists of the computer DSKY, the attitude and thrust/translation controllers, and portions of the astronauts' center panels. The DSKY allows the astronaut to communicate with the LGC and perform a variety of tasks such as testing the computer, entering voice link data, and commanding IMU mode switching. The controllers permits manual or computer-aided manual changes in attitude or translation. The thrust/translation controllers provide manual control of descent engine thrust level.

The PGNS acts as a digital autopilot in controlling the LM throughout its mission. To accomplish this, the system contains various sensors and a data processing unit which performs computations and issues flight control signals. The PGNS is primarily an inertial system aided by radar and optical equipment. The PGNS function is backed-up by the abort guidance system and by the astronaut himself. These integrated equipments allow the astronaut to operate the vehicle in a fully automatic mode or in several semi-automatic modes. Because these various flight control capabilities do exist, the PGNS interfaces with equipments not directly involved in its own digital autopilot function. These interfaces include appropriate displays of vehicle attitude and velocity, displays of radar data, and commands from the astronauts' manual controllers. In addition to those displays driven by the PGNS, the astronaut views displays of attitude change rates, caution and warning lights, total velocity change information, clocks, and other meters that aid him in his manual flight control operation.

## 6. LM PGNS FUNCTIONS AS RELATED TO THE LM MISSION

The PGNS operates during nine phases of the LM mission:

- (1) separation of LM from CSM, (2) transfer orbit insertion,
- (3) descent coast, (4) powered descent and landing,
- (5) lunar stay, (6) powered ascent, (7) ascent coast,
- (8) terminal rendezvous, and (9) docking.



## SEPARATION OF LM FROM CSM.

Prior to LM/CSM separation, the astronauts perform various tasks to prepare the LM PGNS for the descent to the lunar surface. A voice communications link and LGC time reference are established. The PGNS is turned on and the IMU is brought to an operate condition. The IMU is coarse aligned using the known CSM inertial attitude.

The astronauts check the system displays and verify system parameters with the CSM and/or the earth. The RCS is then manually fired to accomplish the LM/CSM separation.

## TRANSFER ORBIT INSERTION

During the coast period between LM/CSM separation and transfer orbit insertion, the IMU is fine aligned. During IMU fine alignment the LGC controls vehicle attitude by using the coarse aligned IMU as a reference. The astronaut must use the AOT to sight on at least two stars in order to accomplish IMU fine alignment.

Both the LM and the CSM computers are used to determine the desired time and velocity for transfer orbit injection. If the results compare favorably, when the calculated time is reached the LGC commands descent engine ignition, trim, and throttling. IMU accelerometer loop outputs in the form of velocity increments are fed directly to the LGC which monitors the insertion maneuver and determines the required thrust commands to achieve the desired insertion conditions.

## DESCENT COAST

The descent coast phase begins at the end of transfer orbit insertion (thrusting). This phase will last approximately one hour until the LM reaches pericynthion altitude of 50,000 feet. During this period, the LM will be tracking the CSM with the rendezvous radar. The primary activity will be on board determination of the LM orbit, with the CSM making a separate determination and the ground stations doing the same when the LM comes into view. Also, fine alignment of the IMU platform will again be performed and preparation for powered descent will begin. Up to this time, the LM has been leading the CSM in phase. At pericynthion, the phase lead is about  $10^\circ$  but once powered descent begins, the CSM will start to catch up with the LM and finally go ahead during the latter portion of landing.



POWERED DESCENT. Powered descent is divided into three distinct portions, or sub-phases, called the braking phase, the final approach phase and the landing phase. The total trajectory covers approximately 250 nautical miles.

1. Braking Phase. The braking phase lasts about 450 seconds and covers about 244 nautical miles during which the altitude is reduced from 50,000 feet to about 8,600 feet at the "HI-GATE" position. The main objectives of the braking phase are:

- a. To reduce most of the orbital velocity for acceptable landing approach magnitudes.
- b. Maintain efficient use of propellants by keeping the thrust vector along the line of the flight path.
- c. To reach a predetermined state vector at the "HI-GATE" position.

The PGNS is continuously calculating altitude, altitude rate and forward and lateral velocity. Also, within the LGC, calculations are being made in order to provide the proper attitude and thrust control for targeting the LM to the specified "HI-GATE" position and the specified state vector. The Landing Radar altitude information is used for updating the LGC altitude calculations from 25,000 feet. During the entire braking phase, the pitch of the LM is near the 90° position such that the crew cannot observe down range to the landing sight. This position is changed at the "HI-GATE" point where the braking phase terminates.

2. Final Approach Phase. The final approach phase begins at "HI-GATE" at an altitude of 8,600 feet, covers about 5 1/2 nautical miles, lasts about 105 seconds and ends at the "LO-GATE" position at an altitude of 500 feet. The objectives of this phase are:

- a. To provide crew visibility of the landing site and adequate time to assess the landing site.
- b. To provide enough time for assessing the flight safety of the trajectory before commitment to landing is made.
- c. To provide a stable craft for accomplishing a. and b., i.e., maneuvering will be kept to a minimum.



At "HI-GATE" the LM will pitch up to about  $46^\circ$  from the vertical in order to provide a limited view of the landing site. At this point the LM window becomes an important part in selection of the landing site. The PGNS will be targeting the LM along a particular trajectory for a particular landing site and continuously calculating thrust requirements from altitude and velocity information to obtain the proper trajectory. The Landing Radar will supply LM X, Y, Z velocity data to the LGC from Hi Gate to touchdown. The pilot will know where to look to find the landing point in which the PGNS is taking him by information displayed on the DSKY. This information is given in relation to grid co-ordinates located on the LM window (Landing Point Designator). The pilot looks beyond the given point on the window to see the landing site. If the pilot does not want to go where the PGNS is bringing him, he can "re-designate" a landing site by movement of his attitude hand controller in a pitch (forward change) or roll (lateral change) motion. The LGC will accept these inputs and provide the proper thrusting for the new landing site. The final approach phase will blend in with the landing phase at the "LO-GATE" position in which the pitch of LM is about  $10^\circ$  from the vertical at this time.

3. Landing Phase. The landing phase covers the remaining distance to the landing site (about 1,200 feet) with forward motion from the "LO-GATE" (altitude 500 feet) to an altitude of 100 feet at which time vertical descent begins. The objectives of the landing phase are:

- a. Allow detail assessment and final selection of the landing site.
- b. Allow maneuvering capability for landing point adjustment.

Shaping of the trajectory continues during this phase with PGNS controlling. At the start of this phase, the forward velocity is about 50 ft/sec and the vertical velocity about 15 ft/sec. The approach is such that forward and lateral velocity will be zero at 100 ft. at which time the altitude rate will be about 5 ft/sec. A hover maneuver can be initiated at this time for further evaluation of the landing spot. The vertical descent can be an automatically controlled maneuver, but most likely will be assisted by the pilot with descent rate changes being given to the LGC via a manual control. The final vertical descent rate will be about  $3 \frac{1}{2}$  ft/sec. The total time from "LO-GATE" to nominal touch down will take 80 seconds.



## LUNAR STAY

The astronauts will first perform a complete checkout of all equipment required for ascent. When this checkout is complete one astronaut or both will leave the LM to gather lunar samples, take pictures, and perform experiments. The PGNS may be shut down to a standby configuration during the lunar stay.

Prior to powered ascent from the lunar surface it is necessary to return the PGNS to an operate condition and perform IMU coarse and fine alignments. The IMU may be coarse aligned in one of two ways. For one method the IMU gimbal angles which were observed after lunar landing but prior to PGNS shutdown may be entered into the LGC via the DSKY and used as a coarse align reference. For the second method CSM parameters could be used as a coarse reference. Regardless of which coarse align method is used, two star sightings will be required to accomplish IMU fine alignment.

When the astronauts return from their last exploration period, prelaunch checkout is initiated where all LM systems necessary for ascent are activated and checked out. During or just prior to this time, the CSM will make a final adjustment, if necessary, in its orbit to bring the vehicle over the landing site at nominal lift off time. Determination of the CSM's orbital parameters will be made by the ground stations on earth and possibly the CSM itself. This information will be entered in the LGC. This is an essential part of prelaunch activities. Additional activities will be performed as follows:

- a. IMU platform alignment will be performed utilizing the updated CSM orbital parameters as a reference.
- b. The rendezvous radar will be given a check by tracking the CSM on a previous pass.
- c. Lift off time will be calculated.
- d. The Abort Guidance System attitude measurements will be referenced to the new IMU alignment.
- e. The LM and CSM state vectors will be updated.

The LM crew then prepares the ascent propulsion system and initiates the launch.



ASCENT. The launch window for lunar ascent will be about 5 1/2 minutes long. For the purpose of describing a nominal LM ascent from launch to rendezvous, it will be assumed that launch will occur at the beginning of this window.

At the time of engine ignition, the CSM position is such that the CSM leads the LM in phase of about  $9.2^\circ$ . From this time to rendezvous, the CSM will not lag the LM in phase. The launch period, or powered ascent, is typically 6 1/2 minutes, of which 12 seconds is a period of vertical rise. During the launch period, the PGNS will command the LM attitude to an optimum profile designed to boost the LM to orbital velocity. The launch trajectory will end at 50,000 feet with an overspeed that results in a coast trajectory (after ending cut-off) defining an elliptical orbit with an apocynthion of 30 nautical miles and a pericynthion of 50,000 feet. This will be considered a safe trajectory which will not impact the moon if subsequent burns are not performed and will allow CSM active rescue. The PGNS will control the launch such that at the end of ascent engine burn, the LM will have translated into the CSM orbital plane. The PGNS will also provide monitoring data to the crew via various displays. The abort guidance system has the capability to monitor PGNS control during the launch, and cross checking by the crew can be accomplished via the attitude display.

After engine cut-off, the LM will coast along the established elliptical path for approximately 30 minutes. During this period, the LM will acquire and track the CSM with the rendezvous radar in order to determine its own orbit with respect to the CSM. This tracking will continue during all subsequent phases and maneuvers. Also, major targeting for the next and subsequent maneuvers will be determined during this coast period based upon the updated tracking data and LGC guidance calculations.

CONCENTRIC FLIGHT SEQUENCE. The flight plan to be used for a nominal rendezvous with the CSM consists of a sequence of maneuvers. These maneuvers are designed to place the LM in an orbit concentric or co-elliptic, with that of the CSM for the purpose of:

- a. Providing a period of a constant differential altitude coast for monitoring the PGNS and preparation for the critical transfer to the CSM orbit.
- b. Insuring that this transfer will occur at maximum operational convenience (lighting, ground station monitoring, etc.).



The planned concentric orbit is 15 nautical miles below the CSM which is in a circular orbit 80 nautical miles from the moon. The concentric flight sequence maneuvers consist of the Concentric Sequence Initiation, Constant Delta Height, Terminal Phase Initiation, and Intercept.

Concentric Sequence Initiation (CSI). The CSI maneuver begins 30 minutes after the end of powered ascent. The thrusting will be accomplished with the RCS jets as commanded by the PGNS. The thrusting will be about 50 seconds and parallel to the CSM orbital plane. The maneuver is designed to change the existing orbit to one that has a apocynthion altitude consistant with that for proper phasing with the CSM. Ideally, this altitude will be at 65 nautical miles. This maneuver also results in a pericynthion altitude higher than that of the existing safe orbit. The magnitude of thrust and the direction for this maneuver is a function of the best determination of the CSM and LM orbital parameters prior to CSI.

For the next 50 minutes the LM will coast along its new trajectory, tracking the CSM with the rendezvous radar and preparing for the next event. The PGNS continues to monitor the flight and perform targeting calculations in preparation for the next maneuver.

Constant Delta Height (CDH). The CDH maneuver will be initiated when the LM reaches the high point in its new orbit or at some constant time determined prior to CSI for keeping the operational time line intact. In general, the CDH maneuver is a function of the current knowledge of the LM and CSM orbital parameters and is designed to place the LM orbit at a minimum differential height (circularization) prior to transferring to the CSM orbit. Time for RCS burn is about 60 seconds and probably will occur behind the moon, out of sight of earth. Distance to the CSM, at this time, is expected to be approximately 50 N.M. After CDH burn is complete, the LM will coast in its new orbit (65 N.M., circular) for 20 or 30 minutes, continuously tracking the CSM and preparing for the final trajectory change for intercept.

Terminal Phase Initiation (TPI). The TPI maneuver is designed to place the LM on an intercept trajectory with the CSM about 140° from initiation and to cancel the execution errors incurred by the CSI and CDH maneuvers. The transfer trajectory will be initiated by thrusting along a set elevation angle. This angle is the elevation angle of the line of sight of the LM to the CSM referenced to the current LM horizontal plane. The LGC will



perform calculations for predicting when this line of sight will occur and issue command to the RCS for the burn at this point. The burn will be that required for the subsequent 140° intercept. Sometime after the TPI maneuver (10-15 min) a mid-course correction may be performed for a final correction to the required trajectory. After this has been accomplished, the LM will coast for about 35 minutes to a range of approximately 3 N.M. from the CSM.

At the 3 N.M. range point, braking will be initiated in order to reduce the closing rate. Braking will consist of several short RCS burns, enough to reduce the closing rate to near zero at intercept. The rendezvous radar will be providing the PGNS and displays with range and range rate information and the PGNS will provide commands to the RCS for the amount of braking required.

#### DOCKING

The final LM mission phase is hard docking of the LM to the CSM. This is a computer-aided manual operation. The astronaut uses both the attitude controller and the thrust/translation controller to achieve hard docking. Incremental translation commands from the thrust/translation controller are routed to the LGC. Proportional rotation commands from the attitude controller are also routed to the LGC. The LGC uses these inputs to generate RCS on-off commands to move the LM to a hard dock with the CSM. Following docking the lunar samples are transferred to the CSM along with selected scientific equipment. The two astronauts then enter the CSM, the LM is jettisoned, and CSM checkout preparatory to return to earth is initiated.

#### 7. SUMMARY OF PGNS

The PGNS performs three basic functions: navigation, inertial guidance, and flight control. Navigation is accomplished by using computer "software", the AOT and Radar data. The data is inserted into a computer which performs the calculations necessary for space navigation.

Inertial guidance is accomplished by the use of an inertially stabilized platform. Three stabilization gyros are mounted on a three degree of freedom platform to provide inertial stabilization. Three accelerometers, also mounted in the platform, and their associated electronics provide the computer with incremental changes in velocity. These changes in velocity are accumulated in the computer. The computer, using these velocity changes, time, and certain constants,



then calculates the spacecraft position in space. The stable platform of the inertial measurement unit is also used for an inertial reference in space. From this reference, spacecraft attitude and velocity can be controlled through the control portion of PGNS.

The PGNS interfaces with the astronaut and six LM subsystems: The Electrical Power Subsystem (EPS), the Environmental Control Subsystem (ECS), the Communications Subsystem, Instrumentation Subsystem, the Reaction Control Subsystem (RCS) and the Propulsion Subsystem. The PGNS receives prime power from the EPS, cooling from the ECS and ground information from the Communications and Instrumentation Systems. The PGNS provides steering and thrusting commands to the Propulsion Subsystem and RCS for attitude, thrust, and steering control.

The astronaut:

- a. Loads data into the computer.
- b. Monitors the systems operations.
- c. Can control the mode selection of the system.
- d. Provides system backup.
- e. Provides optical sightings.

The PGNS includes the following major components:

- a. Navigation Base (NB)
- b. Inertial Measurement Unit (IMU)
- c. Rendezvous Radar (RR) and Landing Radar (LR)
- d. Alignment Optical Telescope (AOT)
- e. Coupling Data Unit (CDU)
- f. Power and Servo Assembly (PSA)
- g. Display and Keyboard (DSKY)
- h. LM Guidance Computer (LGC)



- i. Pulse Torque Assembly (PTA)
- j. Signal Conditioner (SC)

The PGNS equipment is divided into three subsections: inertial subsection (ISS), optical and radar subsection (OSS) and computer subsection (CSS).

The PGNS is required to perform various functions during a lunar mission. The LM mission phases are listed below:

- a. Separation of LM from CSM.
- b. Transfer orbit insertion.
- c. Descent coast.
- d. Power descent and landing.
- e. Lunar stay.
- f. Powered ascent.
- g. Ascent coast and concentric flight sequence maneuvers.
- h. Terminal rendezvous.
- i. Docking.



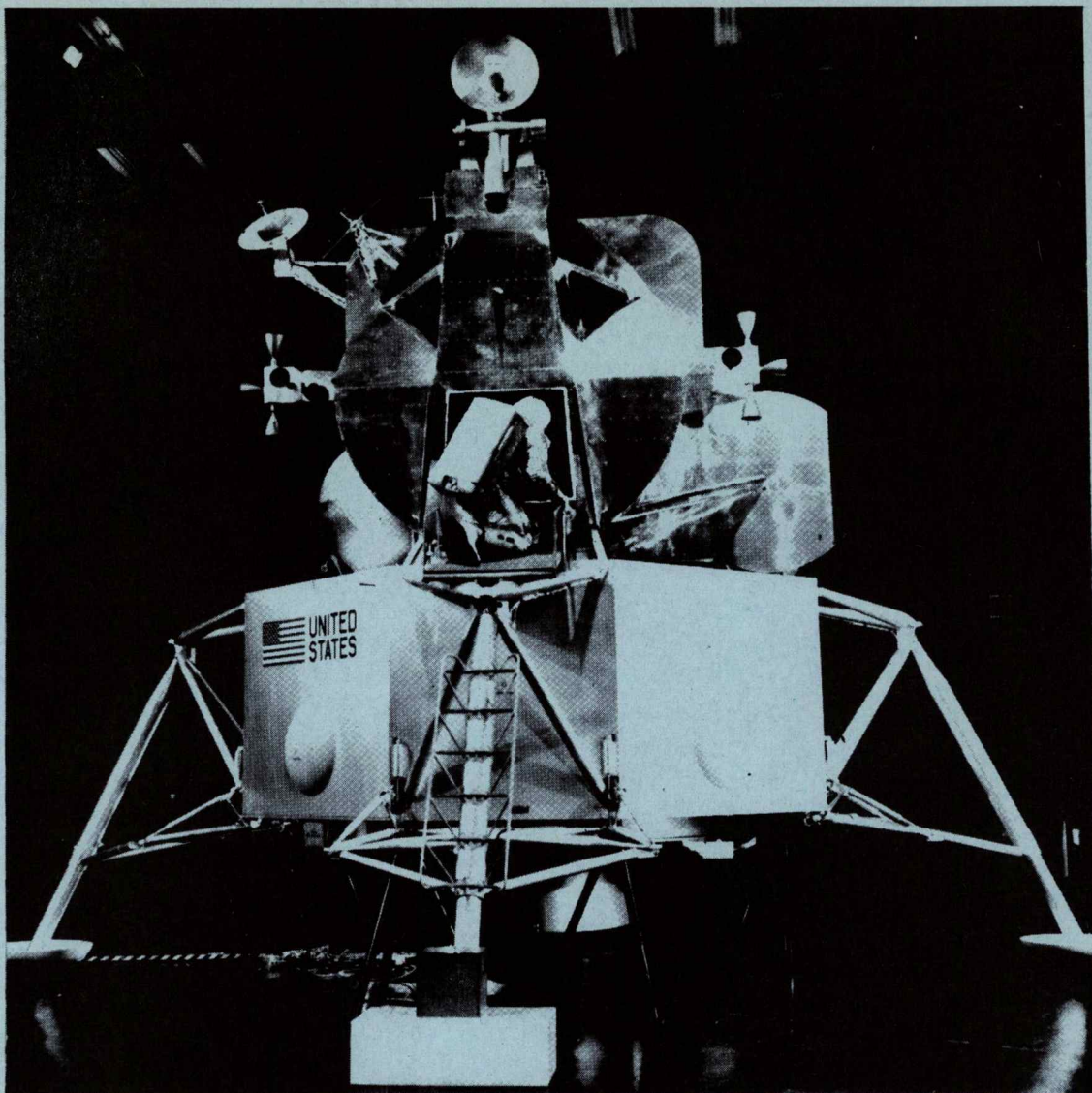
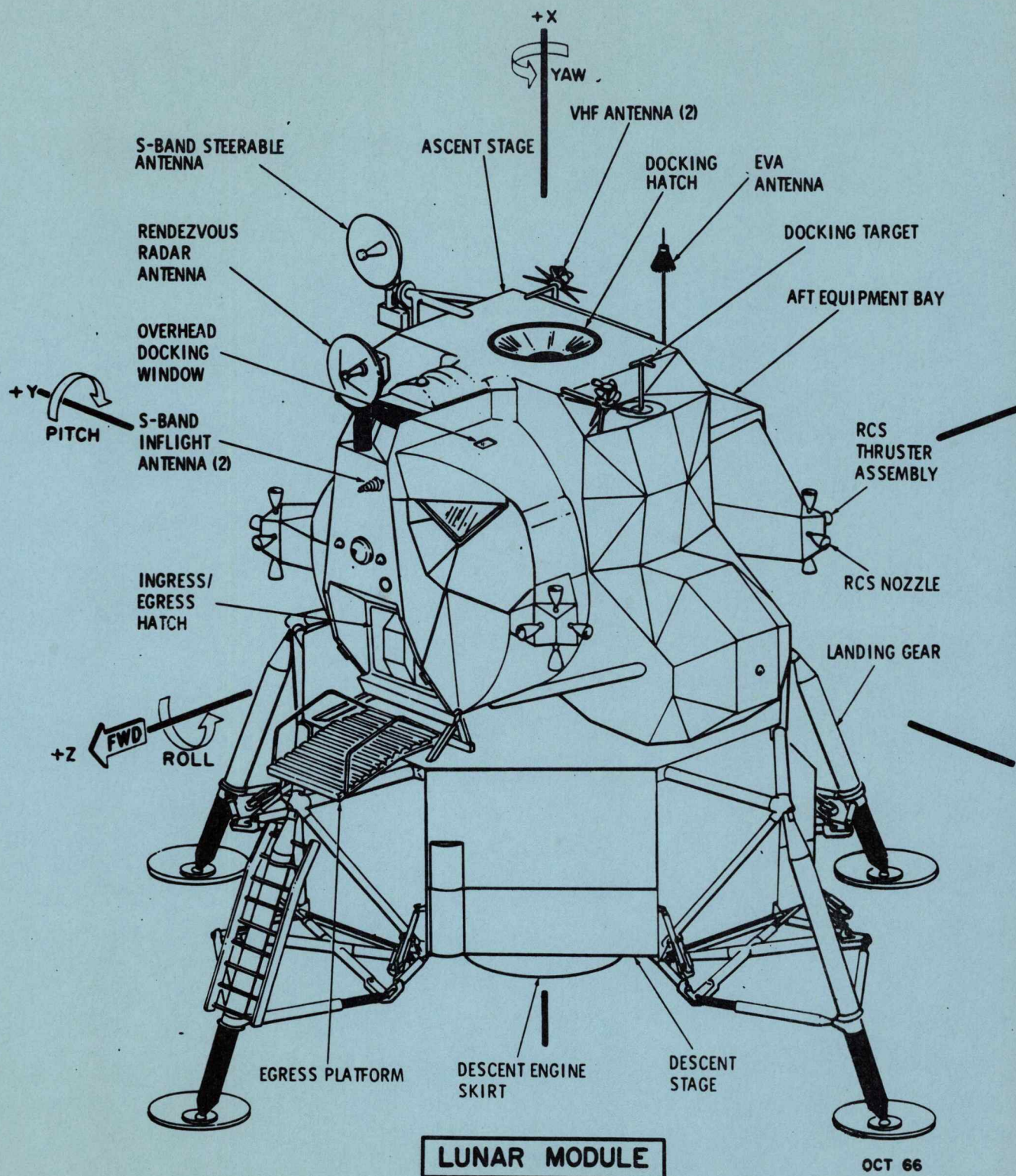


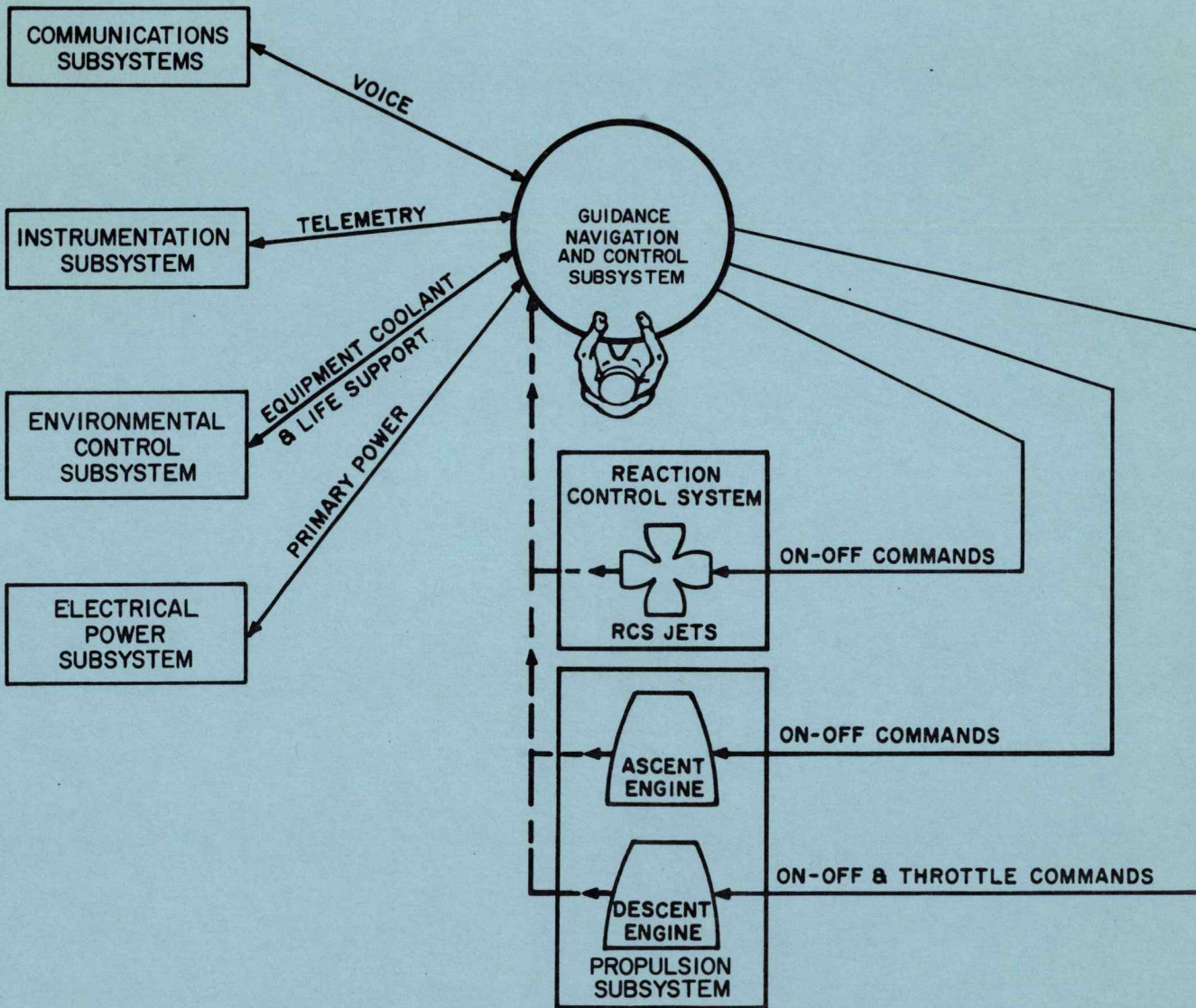
PHOTO OF LUNAR MODULE

T30405-114  
OCT 66





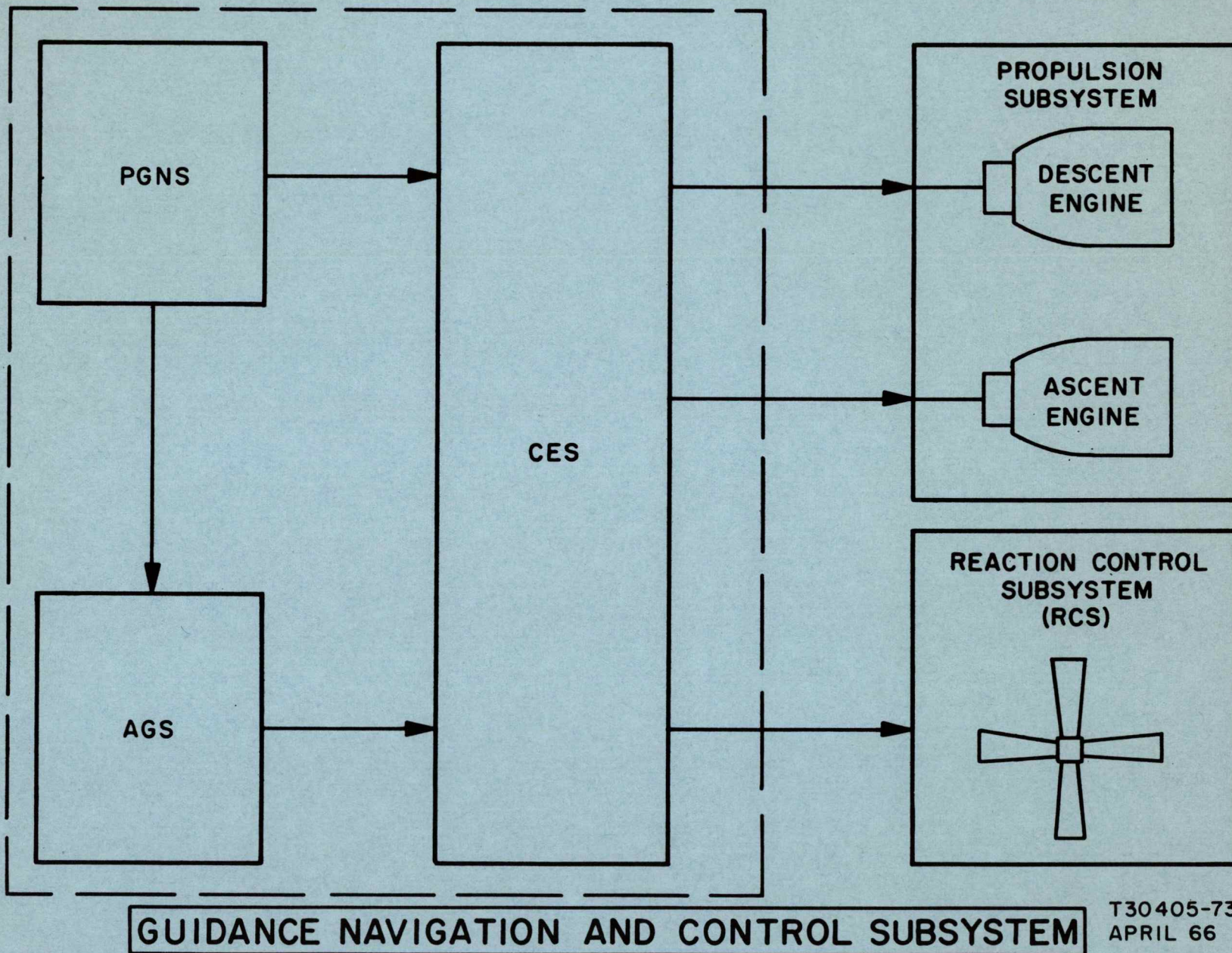




LM SUBSYSTEMS INTERFACE

T30405-45  
OCT 66



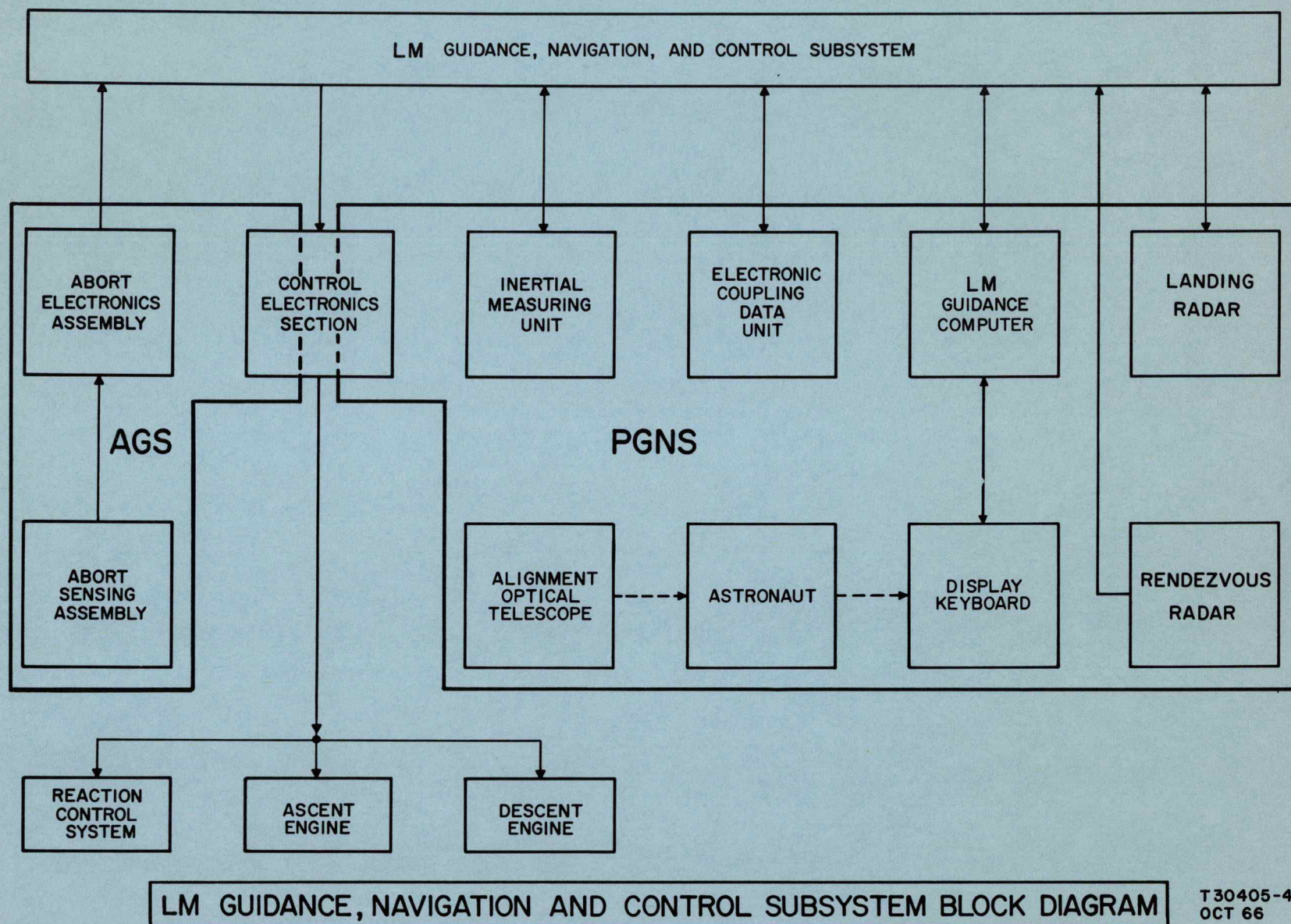


28

Figure 1-4

T30405-73  
APRIL 66







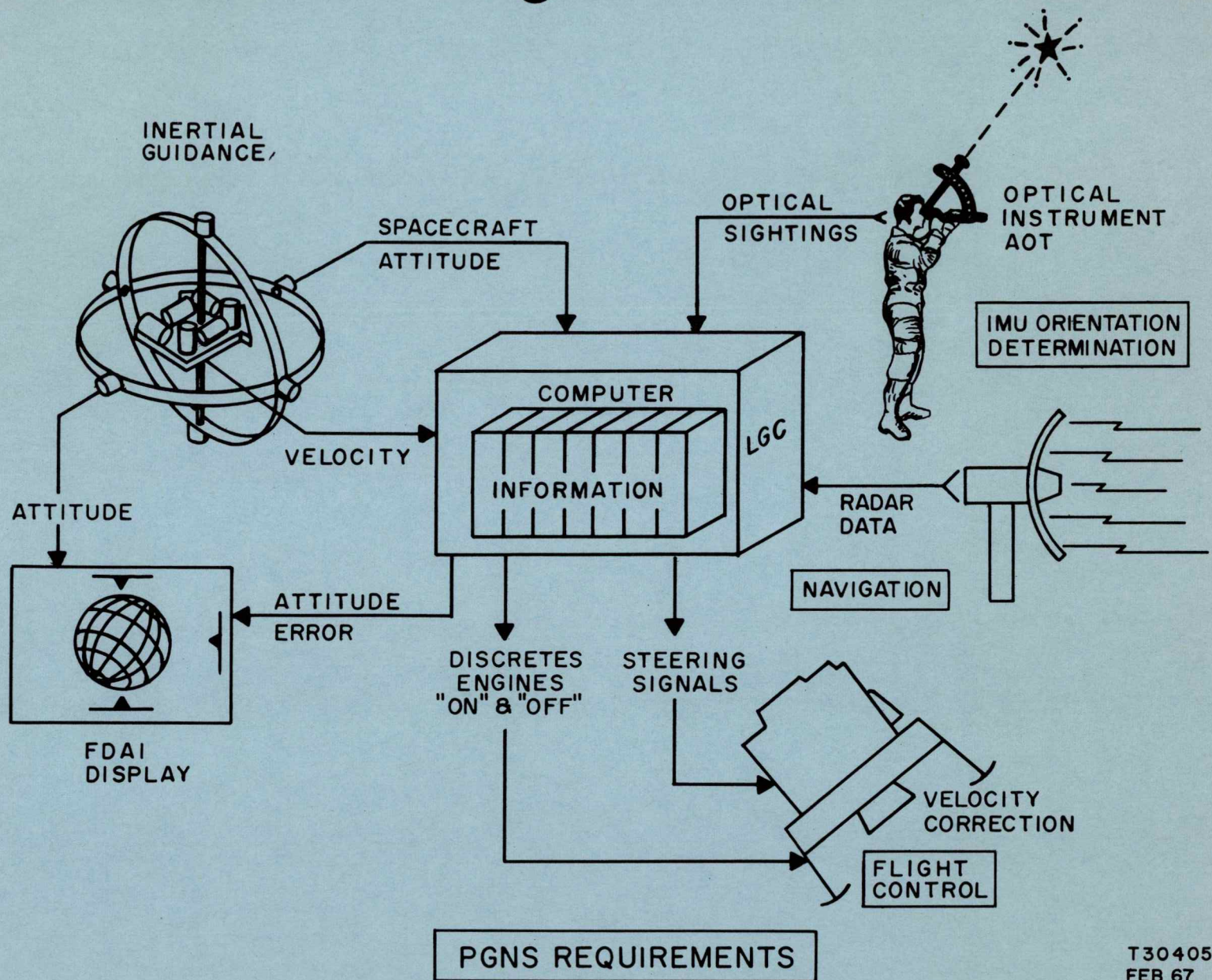
## FUNCTIONS OF THE PGNS

1. SEPARATION OF LM FROM CSM.
2. INSERTION INTO DESCENT ORBIT.
3. DESCENT COAST.
4. POWERED DESCENT AND LANDING.
5. LUNAR STAY.
6. POWERED ASCENT.
7. ASCENT COAST AND CONCENTRIC SEQUENCE MANEUVERS.
8. TERMINAL RENDEZVOUS.
9. DOCKING.

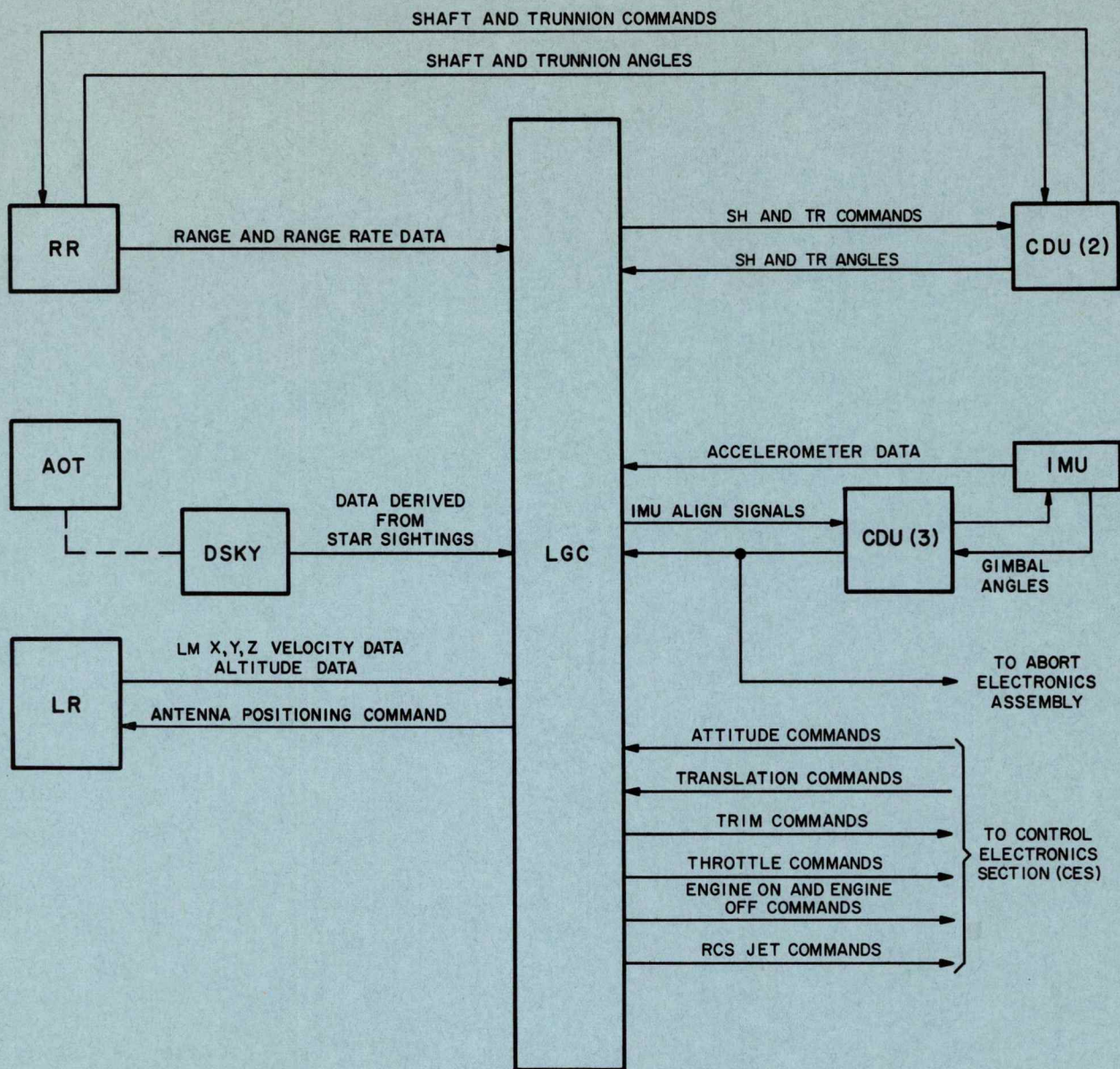
FUNCTIONS OF PRIMARY GUIDANCE  
AND NAVIGATION SECTION (PGNS)

T30405-97  
FEB 67





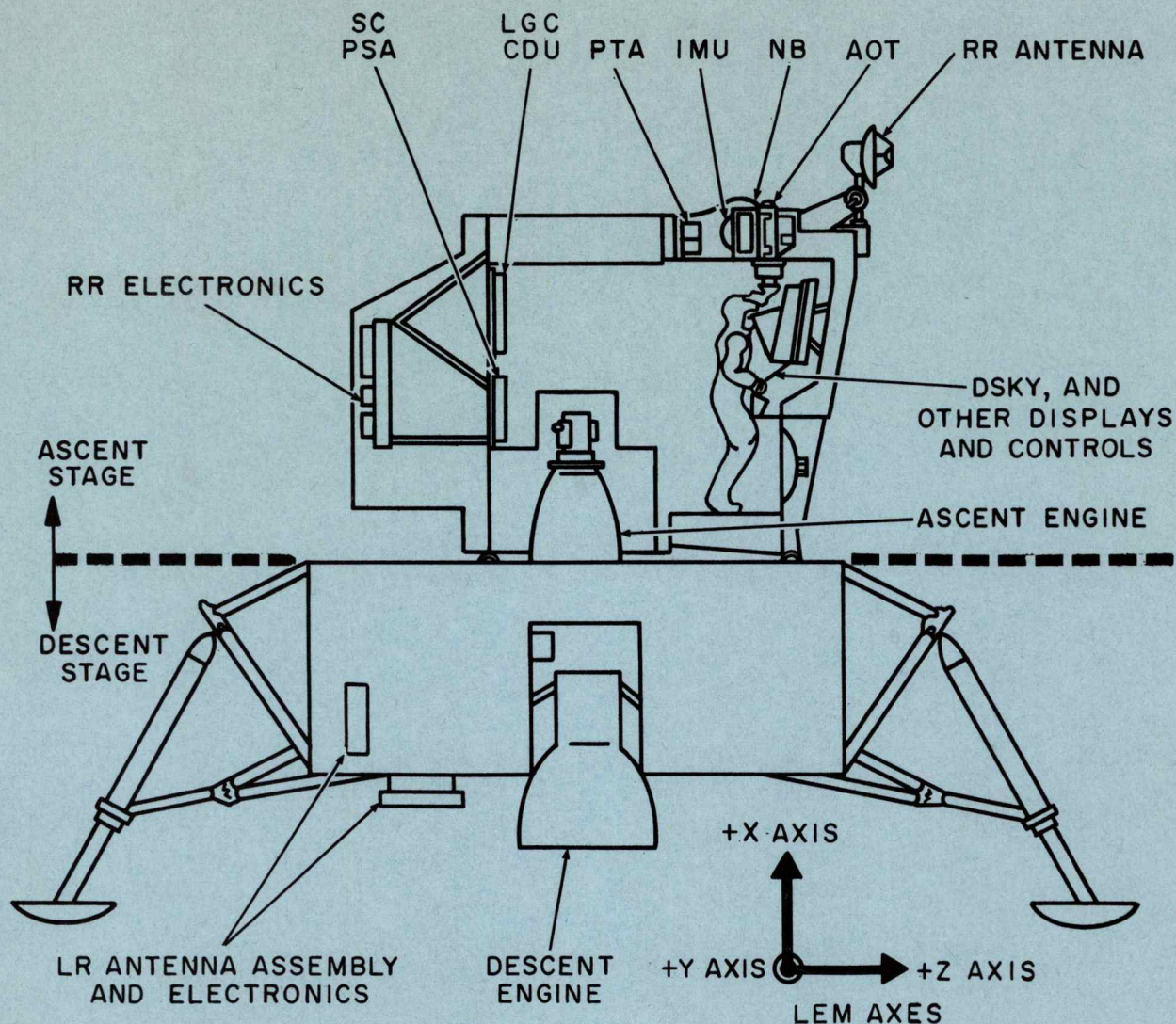




PRIMARY GUIDANCE AND NAVIGATION SECTION — BLOCK DIAGRAM

T30405-98  
FEB 67



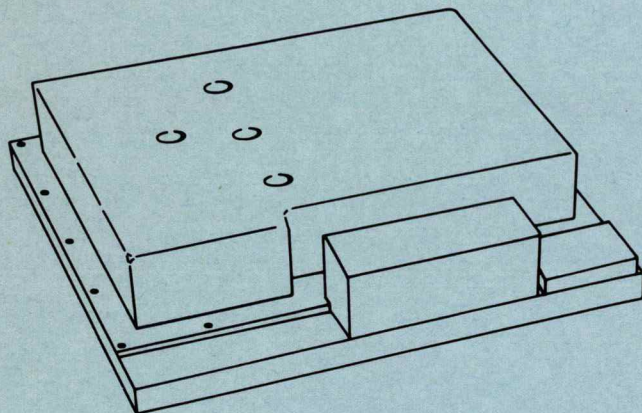


#### LEGEND

NB	NAVIGATION BASE	LGC	LM GUIDANCE COMPUTER
AOT	ALIGNMENT OPTICAL TELESCOPE	IMU	INERTIAL MEASUREMENT UNIT
CDU	COUPLING DATA UNIT		
DSKY	DISPLAY AND KEYBOARD	SC	SIGNAL CONDITIONER
LR	LANDING RADAR	PSA	POWER AND SERVO ASSY
RR	RENDEZVOUS RADAR	PTA	PULSE TORQUE ASSY

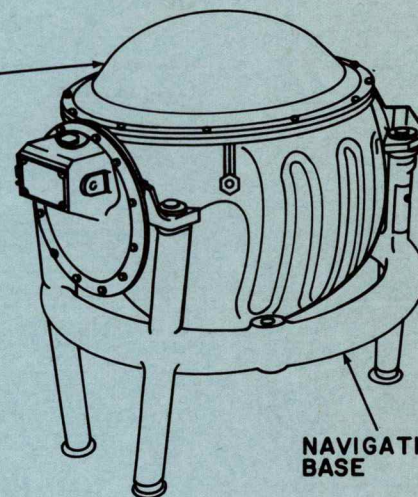
### LM PRIMARY GUIDANCE AND NAVIGATION SECTION EQUIPMENT LOCATION



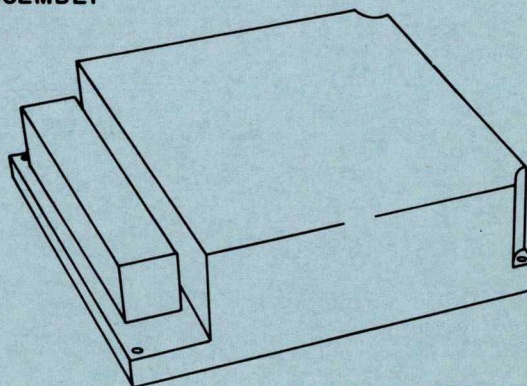


PULSE TORQUE ASSEMBLY

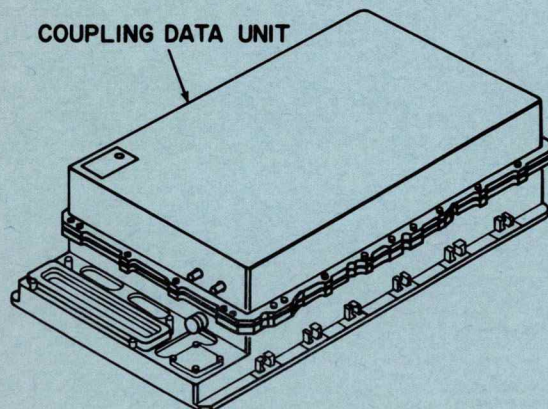
INERTIAL  
MEASUREMENT  
UNIT



NAVIGATION  
BASE

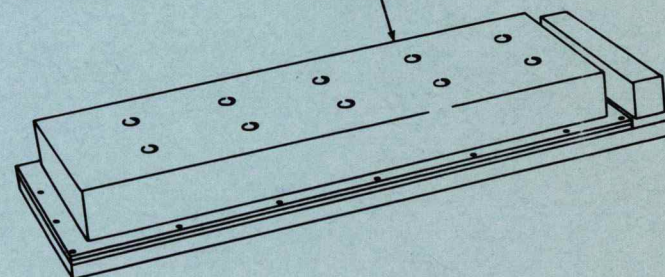


SIGNAL CONDITIONER



COUPLING DATA UNIT

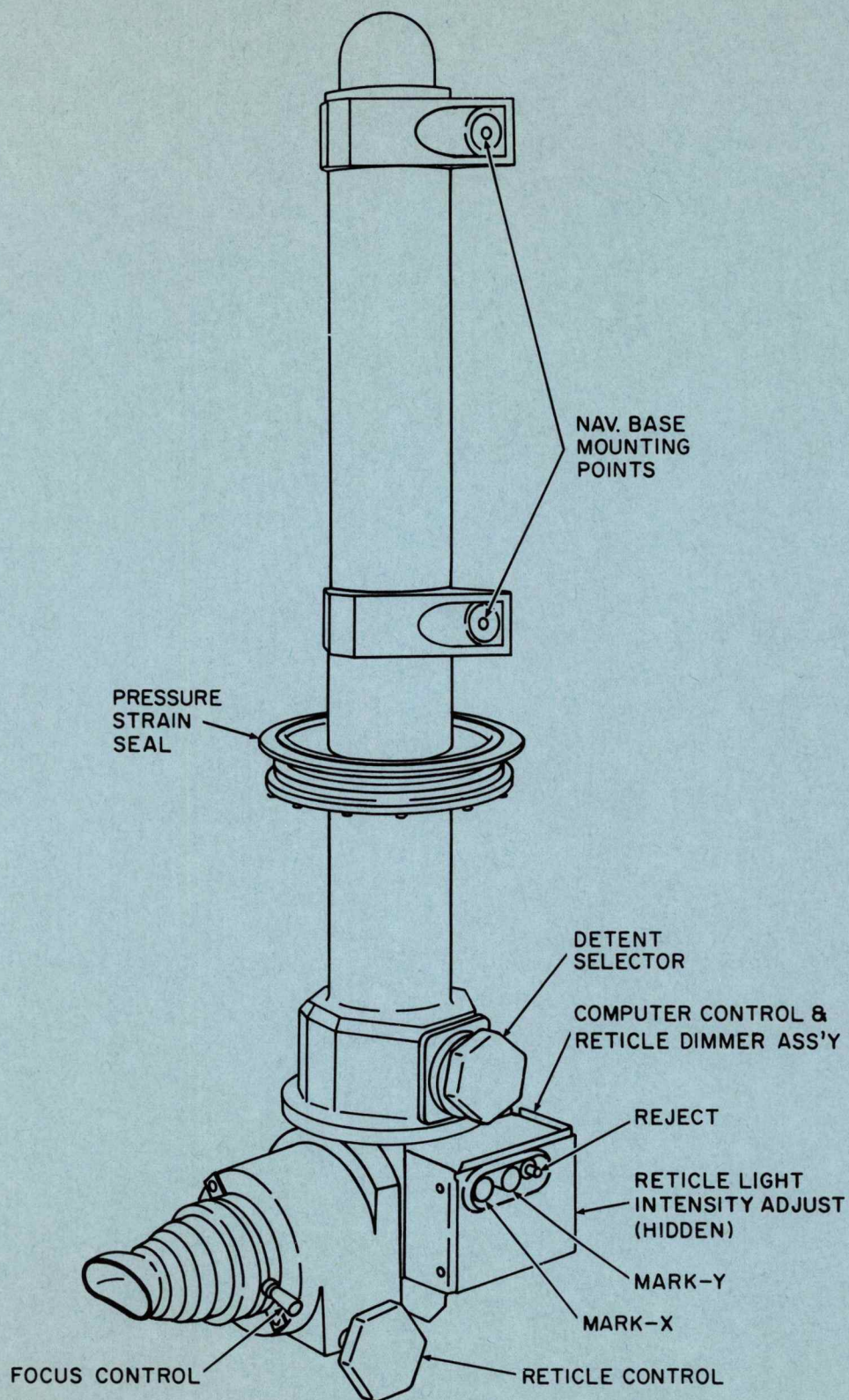
POWER AND SERVO  
ASSEMBLY



ISS EQUIPMENT

T30405-140  
OCT 66

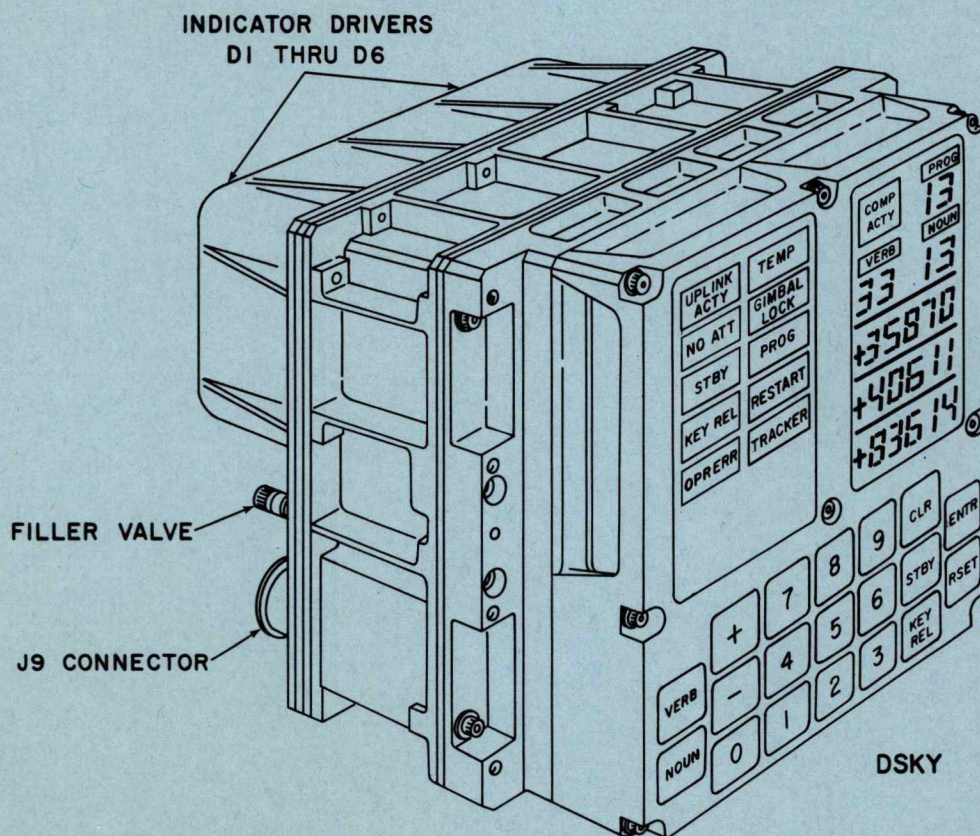
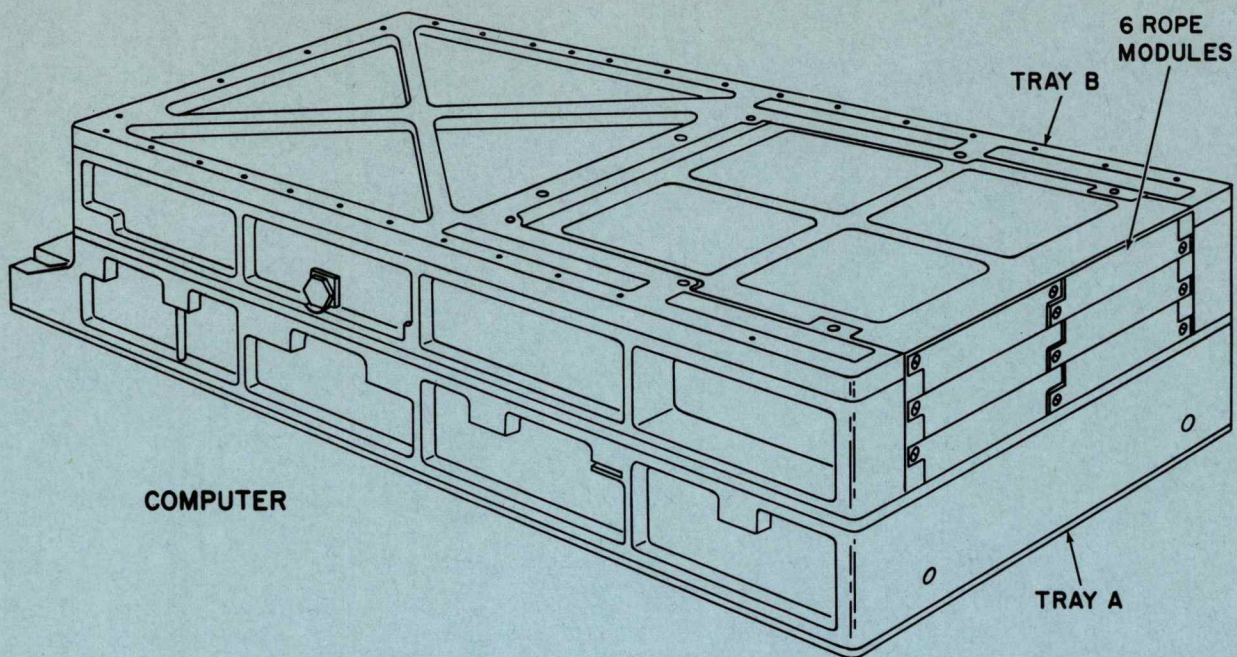




ALIGNMENT OPTICAL TELESCOPE

T30405-141  
OCT 66

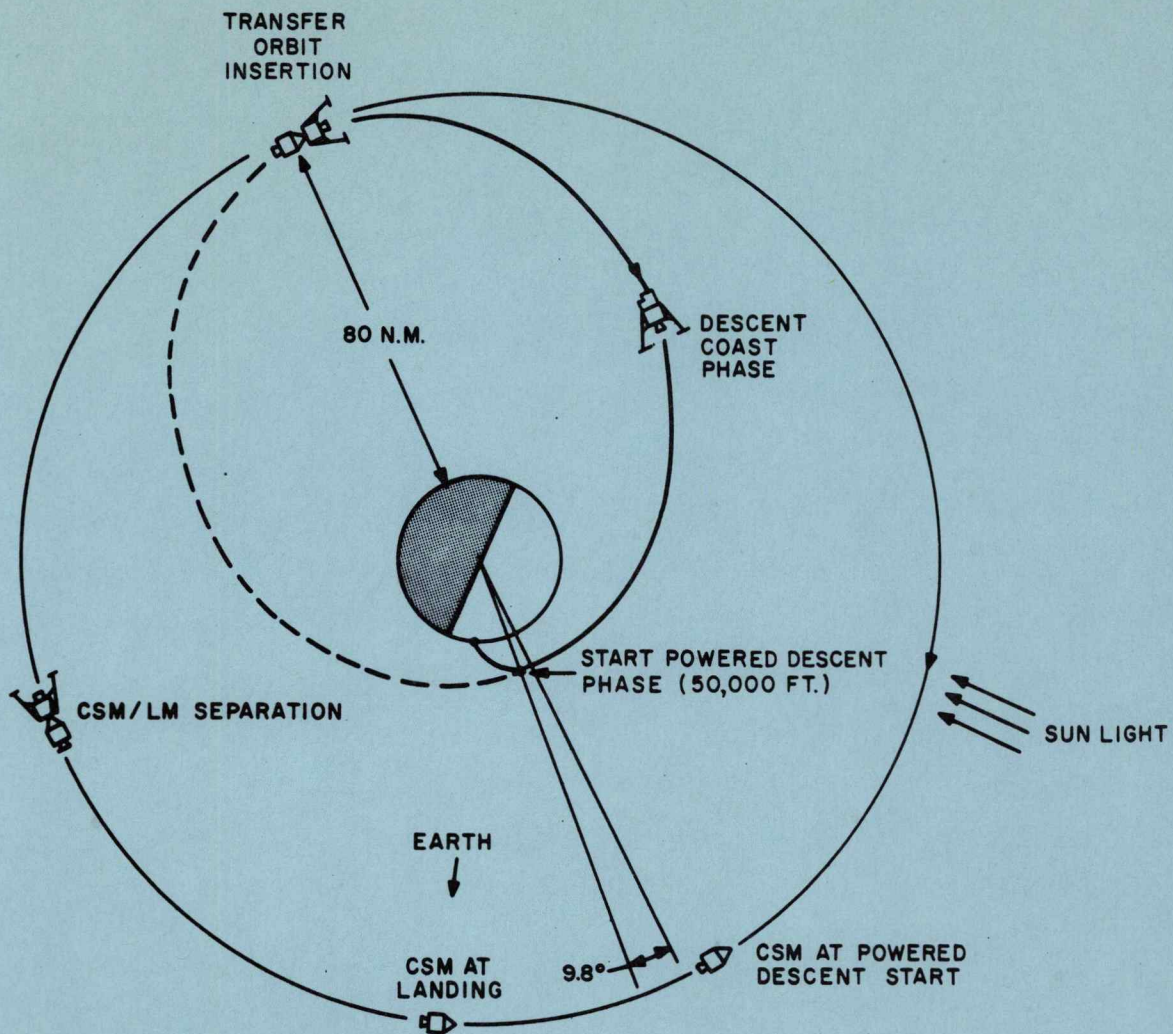




**CSS EQUIPMENT**

T30405-139  
FEB 67



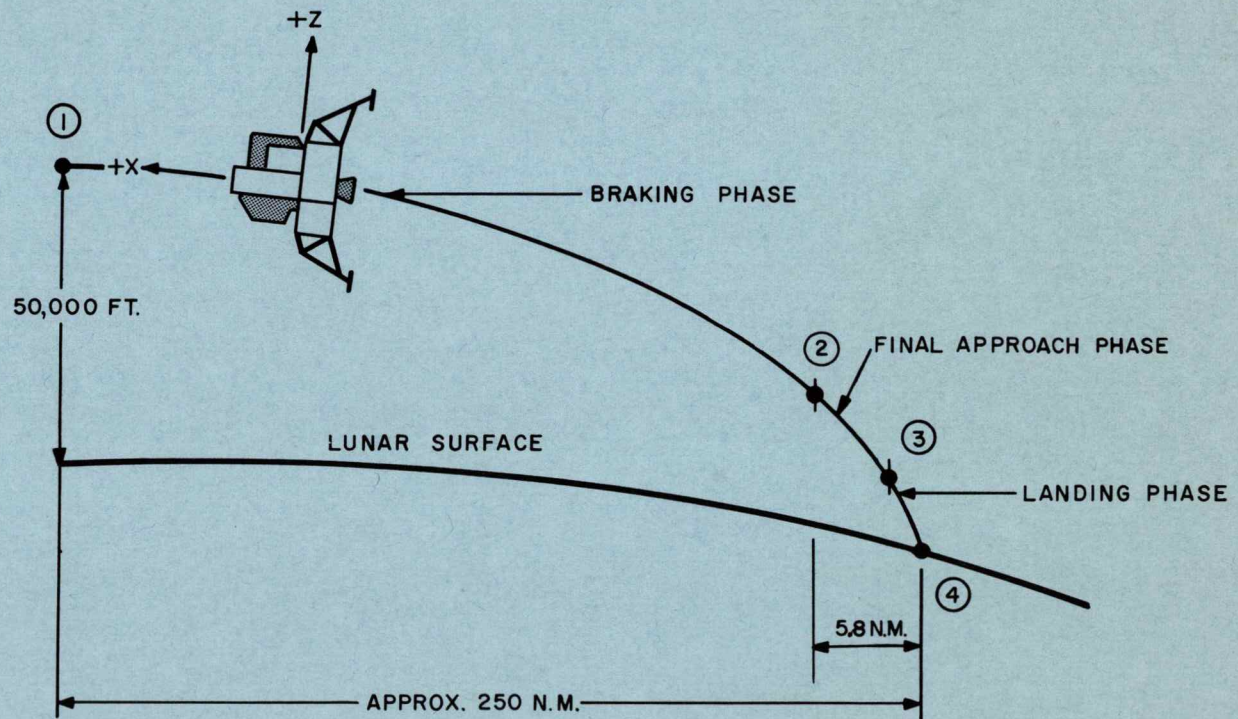


**LM DESCENT PROFILE**

T30405-151  
FEB 67



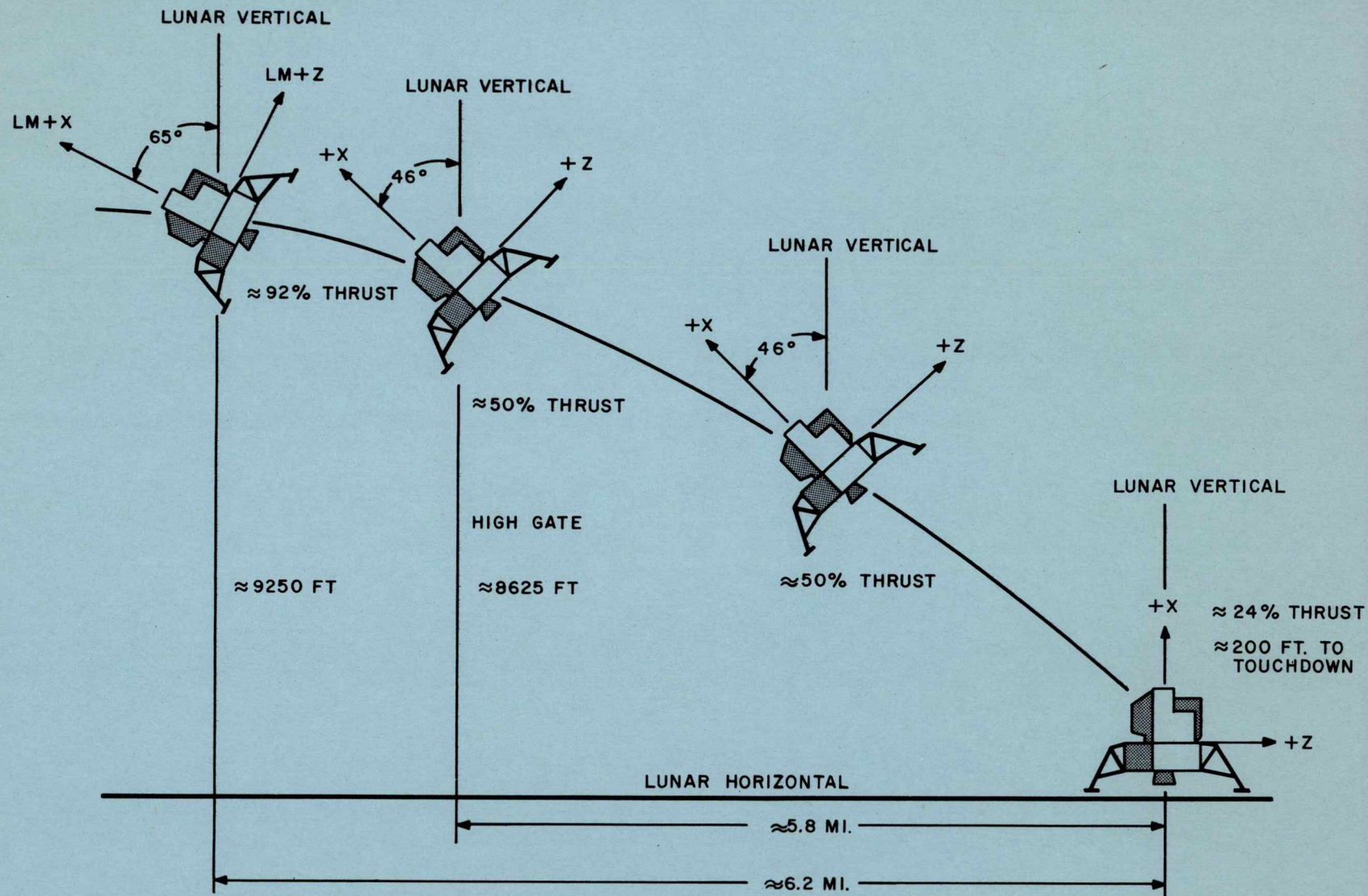
- ① — INITIATE POWERED DESCENT
- ② — HIGH GATE  $\approx 8600$  FT.
- ③ — LOW GATE  $\approx 500$  FT.
- ④ — TOUCH DOWN



LM POWERED DESCENT PROFILE

T30405-152  
FEB 67

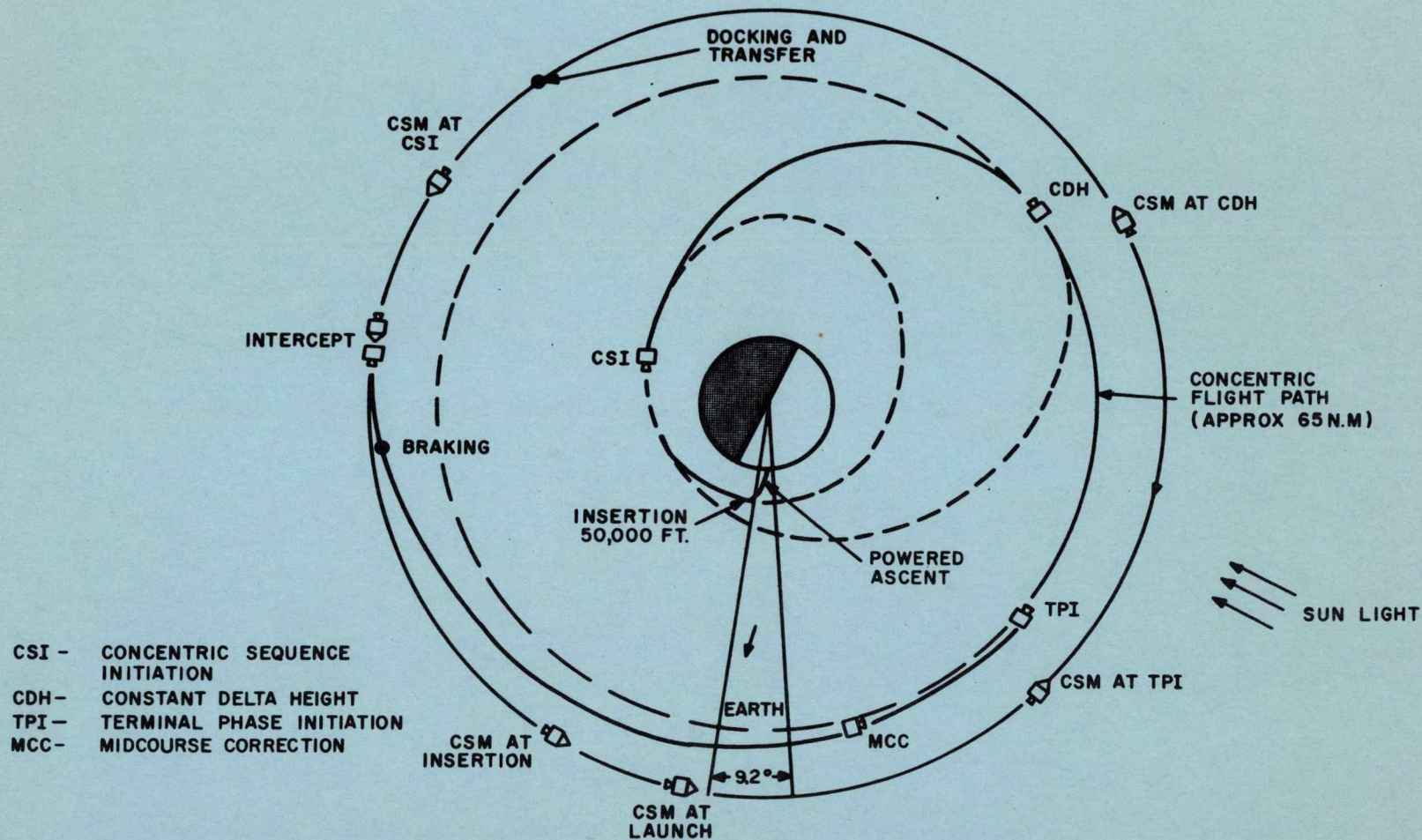




**NOMINAL DESCENT TRAJECTORY  
FROM HIGH GATE TO TOUCHDOWN**

T30405-153  
FEB 67





**LM NOMINAL ASCENT PROFILE  
(CONCENTRIC RENDEZVOUS FLIGHT PLAN)**

T30405-154  
FEB 67



## CHAPTER II

### RENDEZVOUS RADAR

#### 1. General

The Rendezvous Radar/Transponder (RR/T) is a space-stabilized, continuous-wave tracking radar designed for the Apollo Mission. This light-weight, highly reliable, accurate radar will function under the hostile operational environments encountered on the earth, in space, and on the moon. The Rendezvous Radar (RR) is an all solid-state, coherent, tracking radar used in the Lunar Module (LM) for accomplishing rendezvous with the Command Service Module (CSM) in an 80 nautical-mile orbit around the moon. Operating in conjunction with a Transponder (T) located in the CSM, the Rendezvous Radar measures range, range rate, angle, and angle rate with respect to the CSM. Both Rendezvous Radar and Transponders use 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 (los) 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 bore-sight 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 measurements.

The Rendezvous Radar comprises an Antenna Assembly (AA) and an Electronic Assembly (EA). 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 which 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.

A. RR ELECTRONIC ASSEMBLY. The Electronic Assembly consists of a receiver, frequency synthesizer, frequency tracker, range tracker, servo electronics, signal data converter, self-test circuitry, and a power supply.



## Receiver

The return signal is first processed in a highly-stable, three-channel, triple-conversion superheterodyne receiver having intermediate frequencies of 40.8 Mc/s, 6.8 Mc/s, and 1.7 Mc/s. The bandwidth of the first and second IF amplifiers is approximately 3 Mc/s and the bandwidth of the third IF amplifier is approximately 1 kc/s.

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 agc 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/s lower than the incoming 40.8-Mc/s doppler-shifted frequency. At the second mixer, the doppler frequency shift is removed, and all subsequent signal processing is accomplished at fixed carrier frequencies.

## Frequency Synthesizer

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

## Frequency Tracker

The frequency tracker is used to phase-lock the vco with the incoming narrow-line spectrum. A sweep circuit is used to vary the vco frequency across the expected doppler frequency range of  $\pm 100$  kc/s. 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/s reference signal supplied by the frequency synthesizer. The error signal drives the vco to a



frequency which, when mixed with a 27.2-Mc/s 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/s filters. These filters have a bandwidth of 1 kc/s.

#### Range Tracker

The range tracker determines the range to the Transponder by measuring the phase angle between the transmitted and received tones. The 6.8-Mc/s signal received from the Transponder is demodulated in a coherent product detector using a 6.8-Mc/s 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 or backward until phase null is achieved in each of three phase detectors.

#### Servo Electronics

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, forms an inner and outer closed servo loop for each axis. The inner, or stabilization loop, keeps the antenna boresight axis fixed in inertial space in the presence of IM body motions. The outer, or tracking loop, keeps the antenna boresight 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 supplies an automatic track-enable signal for the Rendezvous Radar when within  $1^{\circ}$  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 inertial 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 after acquiring the CSM Radar Transponder.

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

#### Signal Data Converter

The signal data converter accepts range and range-rate data from the range and frequency trackers and converts this data to the 15-bit serial format required by the guidance computer. Data is shifted out to the computer on range or range-rate 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.

#### Self-test Circuits

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. Single values of range and range rate are inserted to permit quantitative checking via the displays. 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 N.M., the Range Rate Meter reads -500 fps and the antenna oscillates in both axes.

#### Power Supply

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/s 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 de-activate the 20 kc/s chopping oscillator for a preset period in order to prevent damage due to a short circuit condition. If the overload has been removed after this time, normal operation is resumed; otherwise, the de-activation cycle will continue until the overload is removed.

#### Mechanical Configuration

The RR EA packaging configuration provides high component densities through the use of cordwood modules. Cordwood packaging permits a reduction in size over the usual transistor circuit packaging methods. Shielding of hf receiver circuits is provided by an eggcrate case design. Conductive cooling of the electronic parts is achieved by sealing the bottom of a cordwood board to the subassembly web with a high thermal conductivity compound. Where required, beryllium ceramic heat sinks are used to conduct excessive heat to the subassembly web. Thermal design is based on the use of conductive heat flow to the cold rails on which the assembly flanges are bolted.

#### B. RR ANTENNA ASSEMBLY

In addition to the microwave radiating and gimbaling elements, the Antenna Assembly includes internally-mounted gyros and resolvers, a multiplier chain, modulator, and mixer-preamplifier components which avoid the necessity of microwave rotary joints and permit the use of flexible coaxial cables between the outboard Antenna Assembly and the inboard Electronics Assembly. A flexible cable wrap-up system is used at each of the rotary bearing points. The AA has two axes. The trunnion (azimuth) axis lies parallel to the LM X axis. The shaft (elevation) axis lies parallel to the LM Y axis. When the trunnion and shaft angles are zero degrees the antenna boresight is parallel to the LM +Z axis.

##### Antenna.

The antenna is a four-horn, amplitude-comparison, monopulse type aperture. 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 servo motor which is driven by the servo electronics.

### Gyros and Resolvers

Four rate-integrating gyros (two active and two on standby) provide line-of-sight space stabilization and line-of-sight angle-rate measurements. These 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 the 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.

### Electronic Components

The multiplier chain, phase-modulator, 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 side-step 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.

## 2. Mission Requirements

The mission of the Rendezvous Radar (RR) is to obtain present position and velocity of the CSM with respect to the LM. This data is required during various phases of the LM mission when the CSM and LM are separated.

The functions performed by the RR during a LM mission from separation through lunar landing and lunar launch to terminal rendezvous, are as follows:

### Separation

After separation, at a distance of 500 to 1000 feet between the LM and CSM, a range and range rate measurement affords the crew a first independent check on the operation of the LM Primary Guidance and Navigation Section (PGNS) and Abort Guidance Section



(AGS) by comparing the Rendezvous Radar range and range rate displays to the outputs of the other two sections.

#### Coasting Descent

During the Coasting Descent Phase, a range rate measurement made right after insertion into the coasting descent determines the applied insertion DV, and provides an independent check on orbit safety by giving an estimate of the expected pericynthion altitude. This may be used again to check the operation of the PGNS and AGS. Early malfunction detection permits early abort initiation using the subsystems which are operative.

Tracking the CSM transponder provides range, range rate, and LOS tracking angle data to the LM Guidance Computer (LGC) for orbit determination.

#### Powered Descent

By tracking the CSM transponder, range and range rate data are obtained which can be compared to data derived in the PGNS. This constitutes a means for monitoring and early malfunction detection.

During the powered descent phase the RR can obtain range and range rate data with respect to the lunar surface which can be compared to data derived in the PGNS if a transponder has been previously placed near the designated landing site. This could be a backup to the Landing Radar.

#### 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.

#### Powered Ascent

Rendezvous Radar range, range rate and LOS tracking angles can be measured with respect to the CSM to aid the LGC in orbit determination.

#### Coasting Ascent

When in the coasting ascent phase, CSM tracking furnishes data to the PGNS for orbit determination and for computation of mid-



course correction maneuvers. Range and range rate data can be fed manually into AGS to be used in orbit determination and computation of midcourse correction maneuvers. This data is 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 manually slewed to zero degrees.

### Rendezvous

During Rendezvous phase the Rendezvous Radar data fed into the PGNS is used to compute rendezvous maneuvers. Rendezvous Radar 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.

### 3. RENDEZVOUS RADAR/TRANSPONDER ACQUISITION SEQUENCE

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 - c is 4.5 seconds.
- d. The computer transmits "auto track enable" when antenna LOS arrives within  $1^{\circ}$  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 c 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. Support and Maintenance Plan

##### a. General

Electronic equipment designed and built for the Lunar Module have many distinctive aspects and features which necessitate utilizing a maintenance approach different from that of normal aircraft. Among the foremost of these aspects are:

Reliability requirements imposing severe standards on workmanship, which require complete traceability of all electrical parts, and the selection of only those parts with the highest life expectancy for the particular application.

Operational standards which require specifications to call out rigid circuit tolerances. Minimum size, weight, and power consumption have demanded new packaging techniques, such as the multi-layer circuit board and integrated circuits. Where these could not be used, solid state circuitry is packaged into cordwood components. This requires parts to be selected not only for high reliability, but for minimum size, weight, and power dissipation. Thus, a requirement exists for special handling techniques and special test equipment.

Environmental operating and storage specifications require that the LM equipment be designed to operate and withstand more severe environmental conditions of shock, vibration, temperature, and radiation than almost any present military or commercial avionics equipment. No maintenance will be performed by the flight crew, therefore, no spares are carried on board the LM. The flight crew will use displays and self-test equipment to make a gross in-flight check of usability of the outputs.

##### b. Pre-Launch Maintenance

Pre-Launch maintenance is that maintenance performed on the radars and transponder after they have been installed in the LM and CSM, respectively. This maintenance is necessary to verify radar operational status or to enable replacement of the antenna assembly or electronic assembly if a fault should be noted during verification testing.

Operational checks will be accomplished at the launch site prior to lift off using Pre-Launch Test Equipment (PLTE) which will be situated on the ground in a mobile van. The PLTE will be connected through a microwave junction box by



waveguides and cables to the radars and the transponder. The PLTE will also be connected to the control computer located inside the blockhouse. During the radar checks, the PLTE will send radar operational information in a digital format to the control computer. Since other operational checks will be made on the vehicle concurrent with the radar checkouts, RF interference will be a problem. In order to minimize this type of interference, the PLTE will use radiation hats designed to cover the antennas during checkout.

c. Bench Maintenance

Bench maintenance is a more detailed type of support. It will be used by GAEC technicians to ascertain the operational status of the radars and the transponder. Bench Test Maintenance Equipment (BTME) will be used to verify the performance of the uninstalled radar assemblies. These functional checks are quantitative determinations that the hardware being tested is operating within specified limits. The BTME will be able to simulate the radar assembly input and output environment.

5. Basic Block Diagram Signal Flow

The RR is packaged in two assemblies: Antenna Assembly and Electronic Assembly. The assemblies in turn, consist of a number of sub-assemblies. The Control Assembly is discussed separately from the other two assemblies in the Controls and Displays chapter.

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 modulated signal is applied to the antenna for radiation through the circulator. The radiated energy incident upon the CSM Transponder is processed such that the tones are removed, the frequency is side stepped, the tones are replaced, and the energy is returned to the RR Antenna.

The return energy from the transponder is received by four feedhorns arranged in a monopulse configuration. 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 thru the circulator to the



monopulse 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, and Y coordinate pointing error is generated. Similarly, with the RR line-of-sight on either side of the transponder, an X coordinate pointing error is generated. The preamplifiers contain three channels: the X (trunnion) channel, the Y (shaft) channel, and the Sum (reference) channel. The X channel input is the difference in received energy between the vertically adjacent feedhorns  $(A+B)-(C+D)$ . The Y 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 (VCO) 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 is 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 X and Y receiver channels) for 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 X and Y 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.

6. The major radar parameters are listed below:

Antenna Feed	Cassegrainian
Antenna Reflector main sub	Parabolic Dish 24" Diameter Hyperbolic 4.65" 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° - 4.0°
Radiation frequency	9832.8 mc
Receiver frequency (from transponder)	9792 mc
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 nautical miles
Range Rate	$\pm 1$ fps to $\pm 4900$ fps
Angular accuracy:	

<u>Range (nmi)</u>	<u>Maximum Error (mr)</u>	
	<u>Random</u>	<u>Bias</u>
400	4	8
300	3	8
200	2	8
5	2	8
2	5	8
1	10	8
(80 ft)	10	8

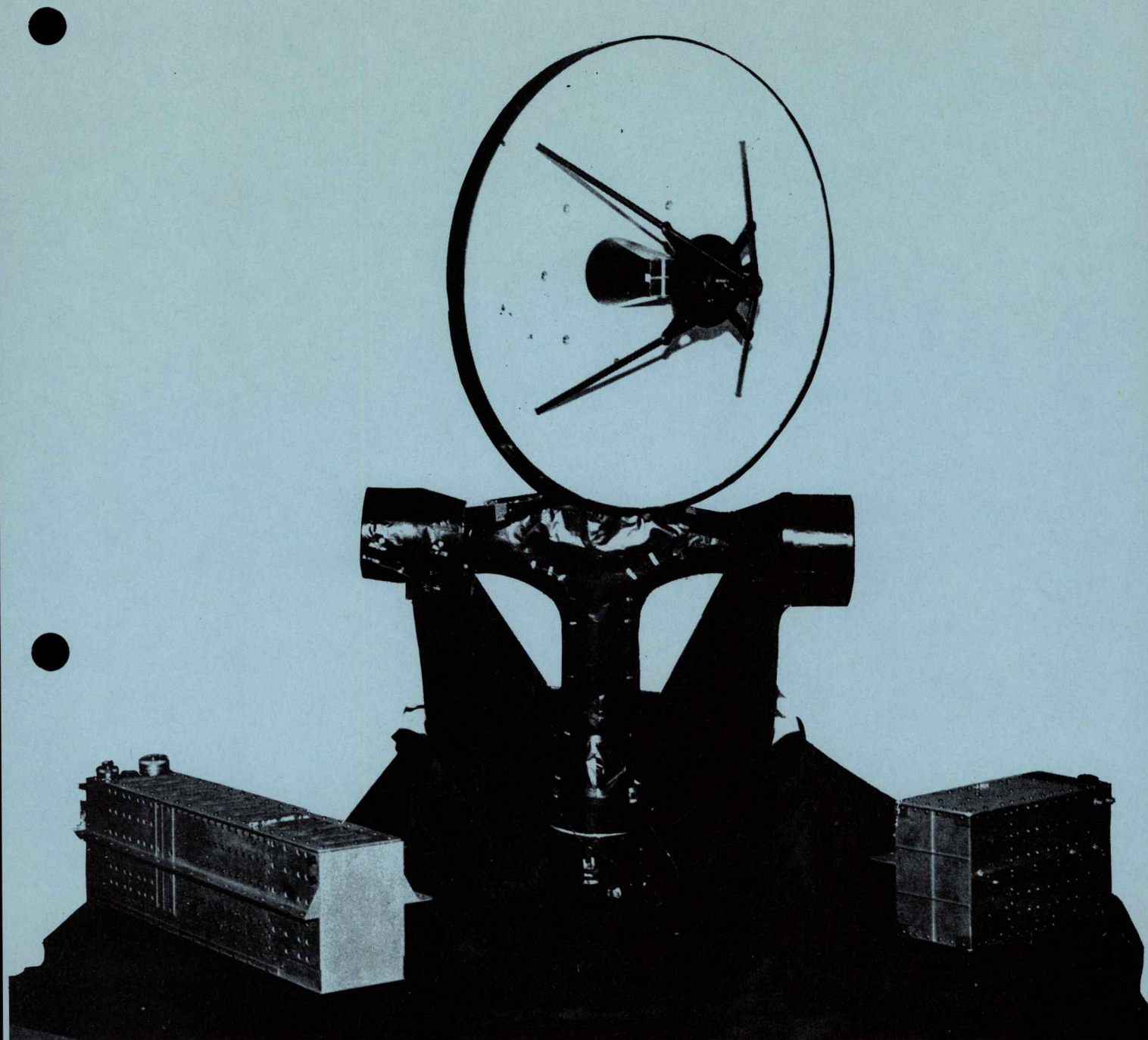
Angle Rate Accuracy:

<u>Range</u>	<u>Maximum Error</u>
400 nm	0.4 mr/sec
200 nm to 1000 ft	0.2 mr/sec

Range Accuracy:

<u>Range (nmi)</u>	<u>Random Error</u>	<u>Bias Error</u>
400 - 5	1/4% or 300 feet	$\pm 50.8$ nmi $\pm 500$ feet
5 nm - 80 ft	1% or 80 feet	$\pm 50.8$ nmi $\pm 80$ feet
Range Rate Accuracy	$\pm 1$ fps	$\pm 1$ fps

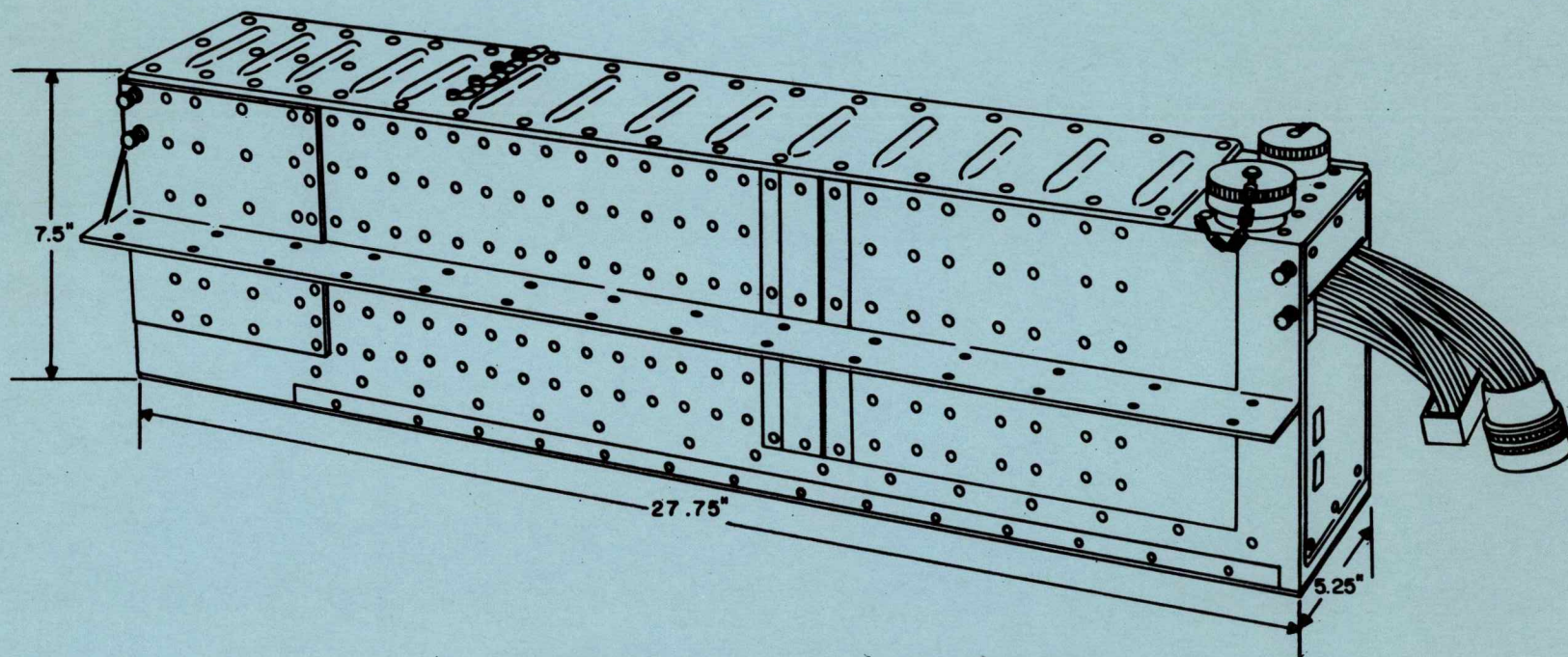




LM RENDEZVOUS RADAR

T30405-159  
FEB 67

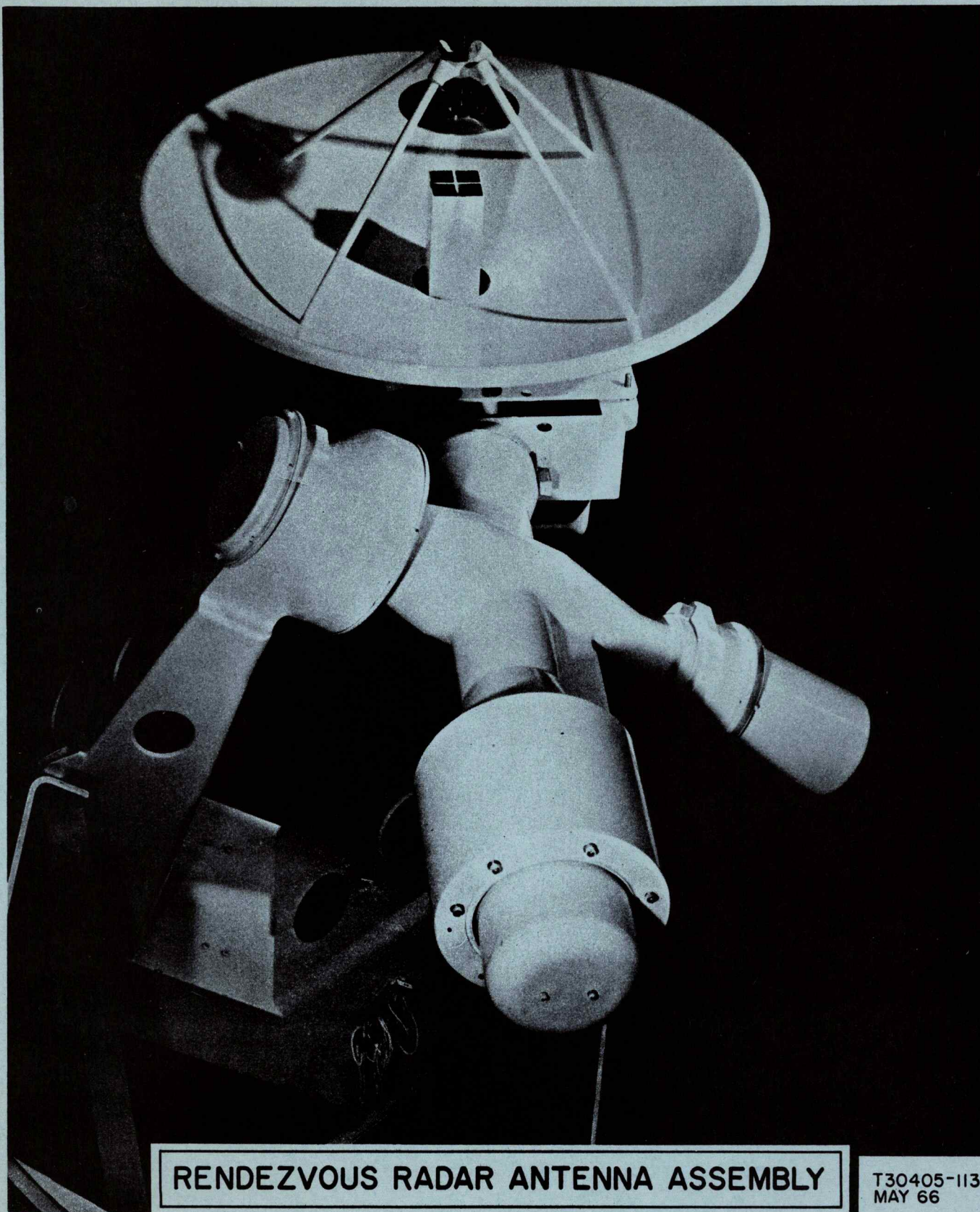




RENDEZVOUS RADAR ELECTRONIC ASSEMBLY

T30405-142  
FEB 67

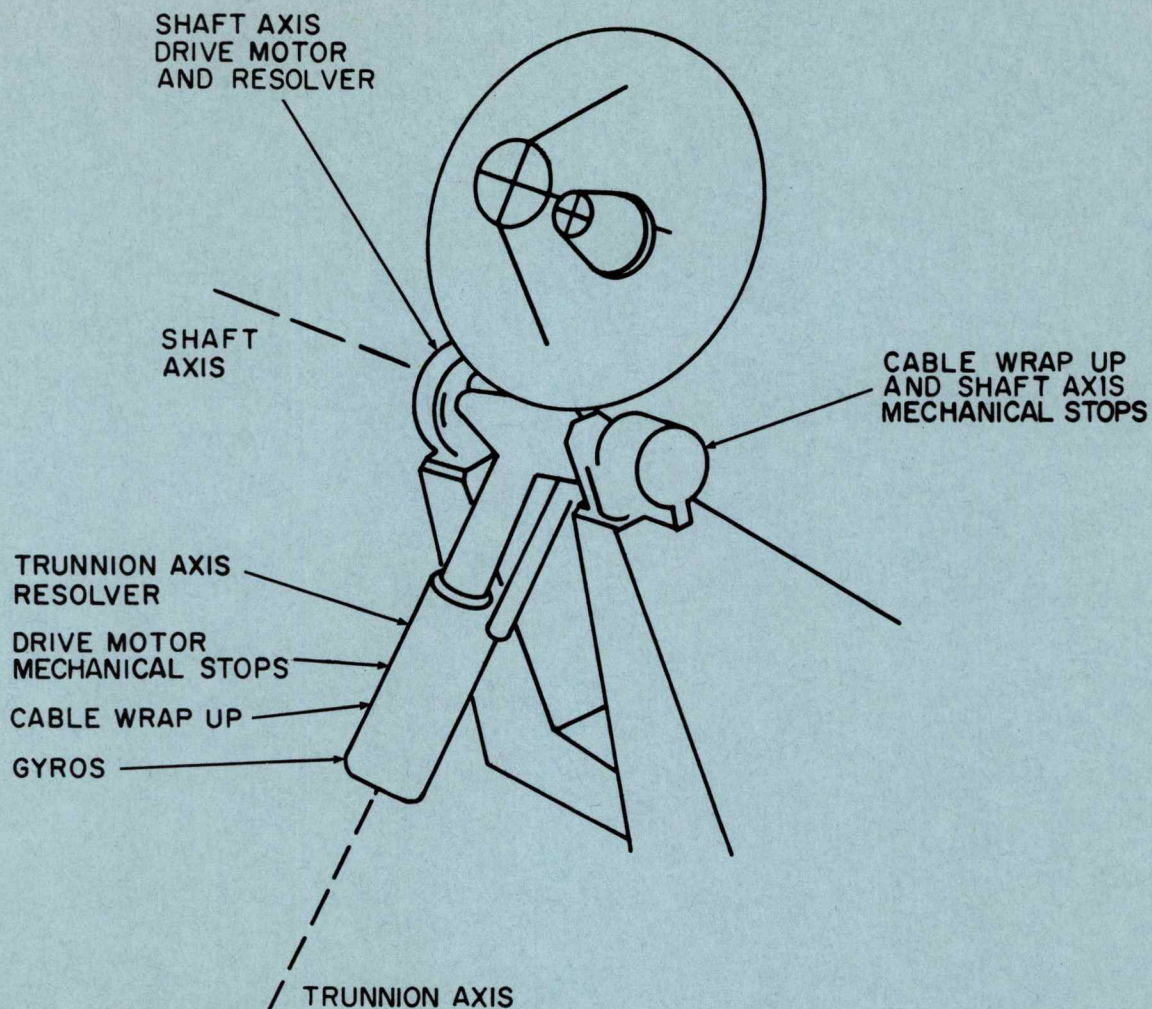




**RENDEZVOUS RADAR ANTENNA ASSEMBLY**

T30405-113  
MAY 66

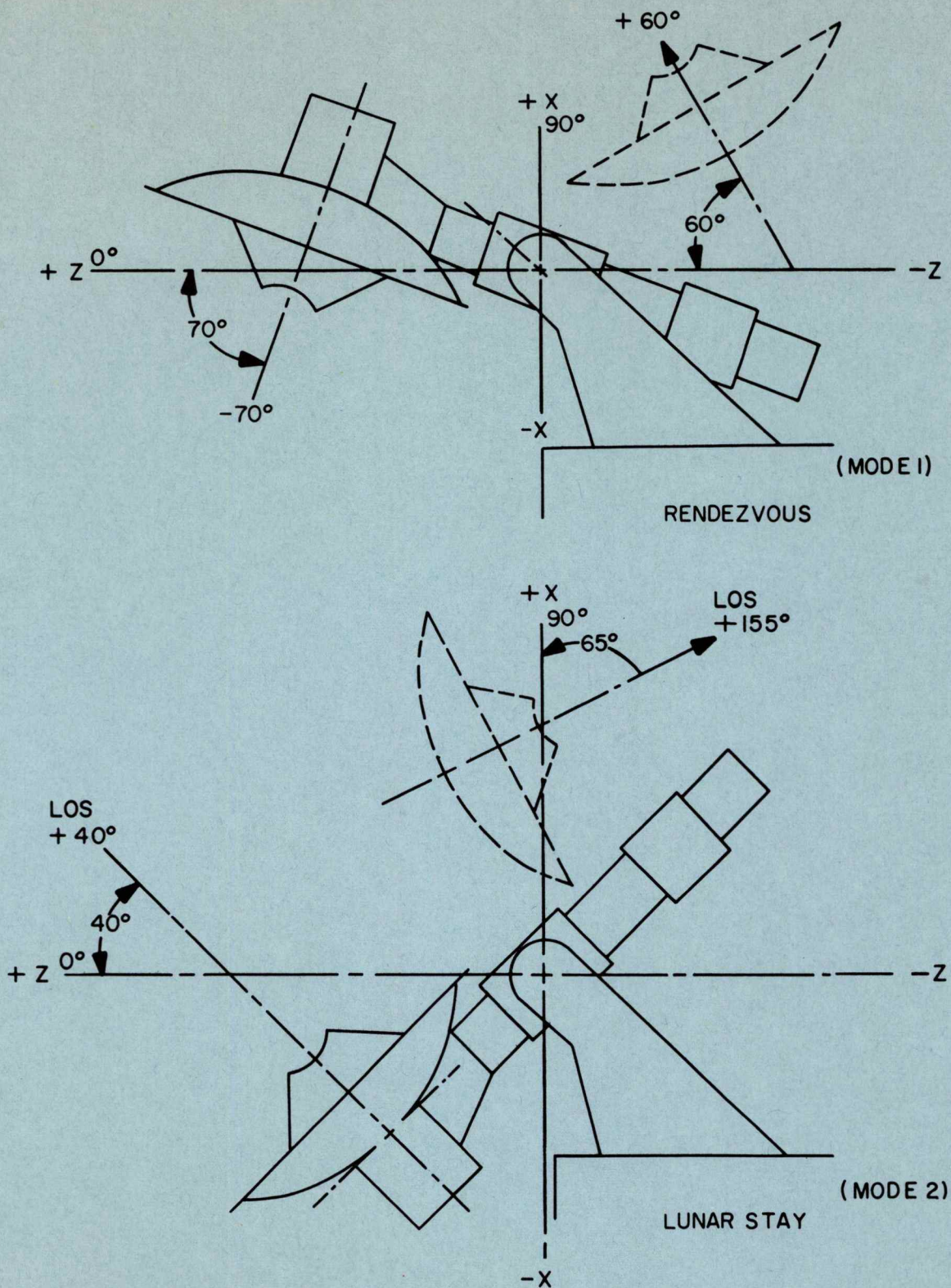




ANTENNA ASSEMBLY,  
RELATIVE LOCATION OF PARTS

T30405-143  
FEB 67



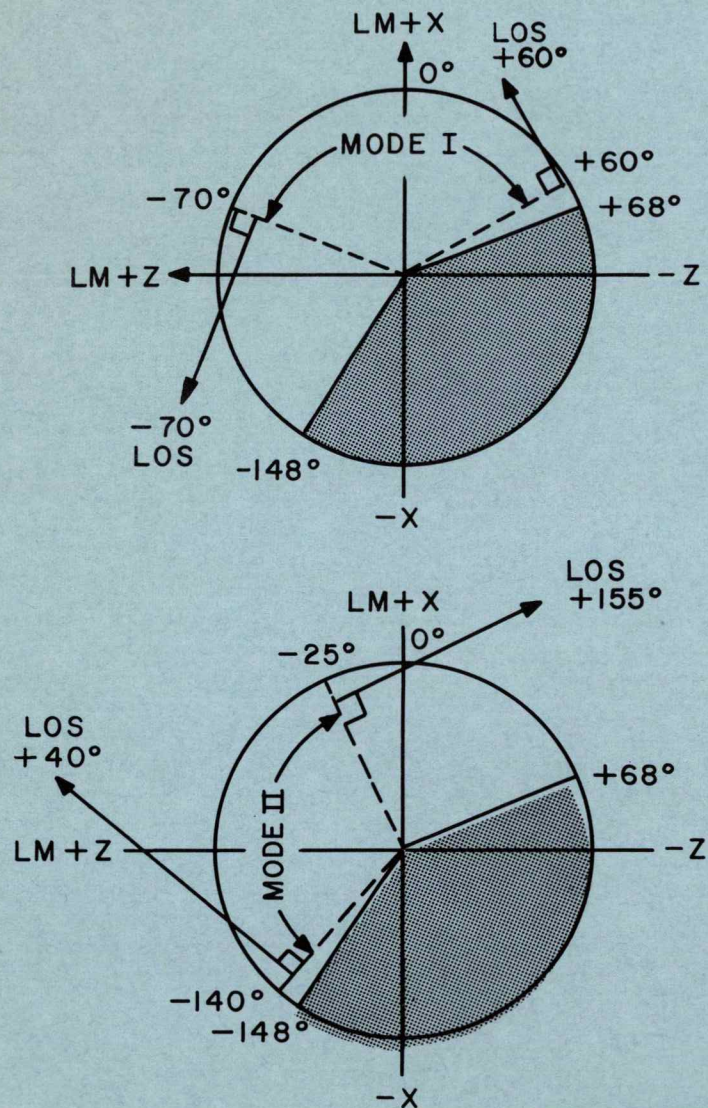


# RADAR ANTENNA MODES OF OPERATION

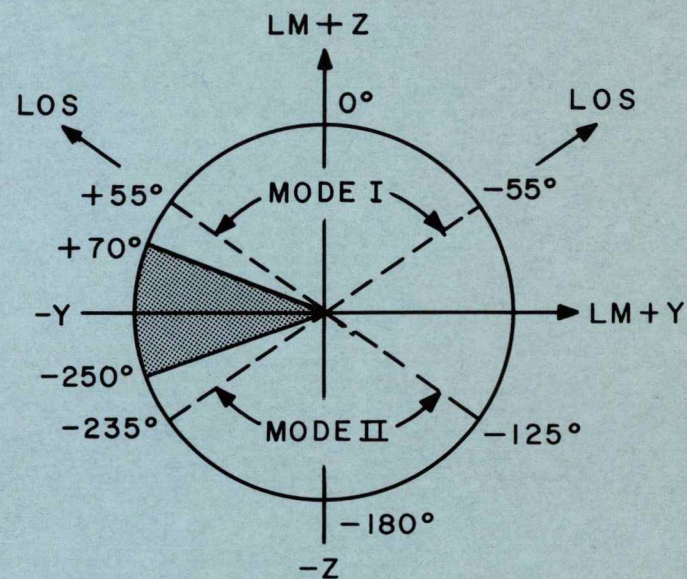
T30405-144  
FEB 67



# SHAFT



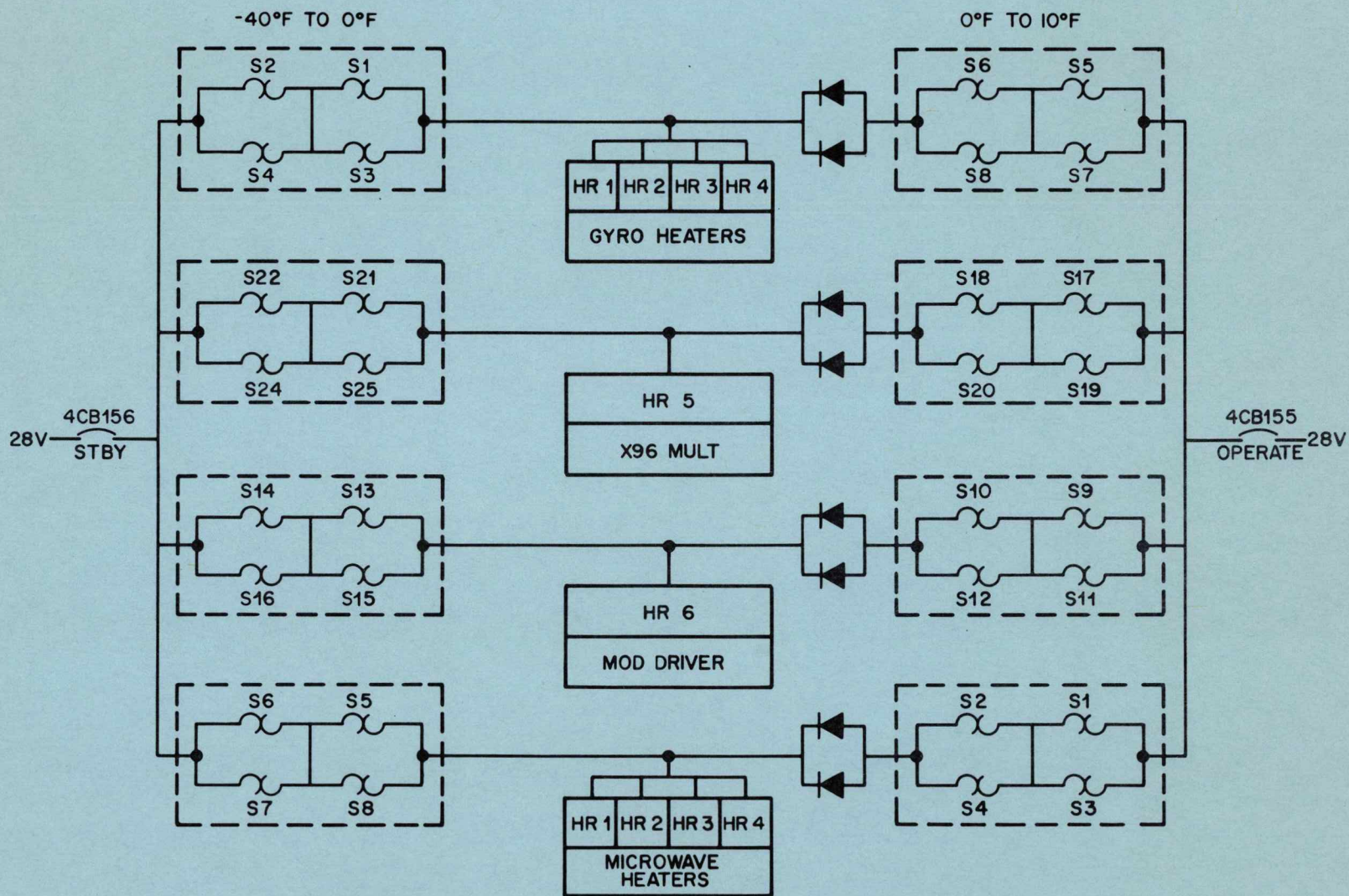
# TRUNNION



RR ANGULAR COVERAGE

T30405-155  
FEB 67

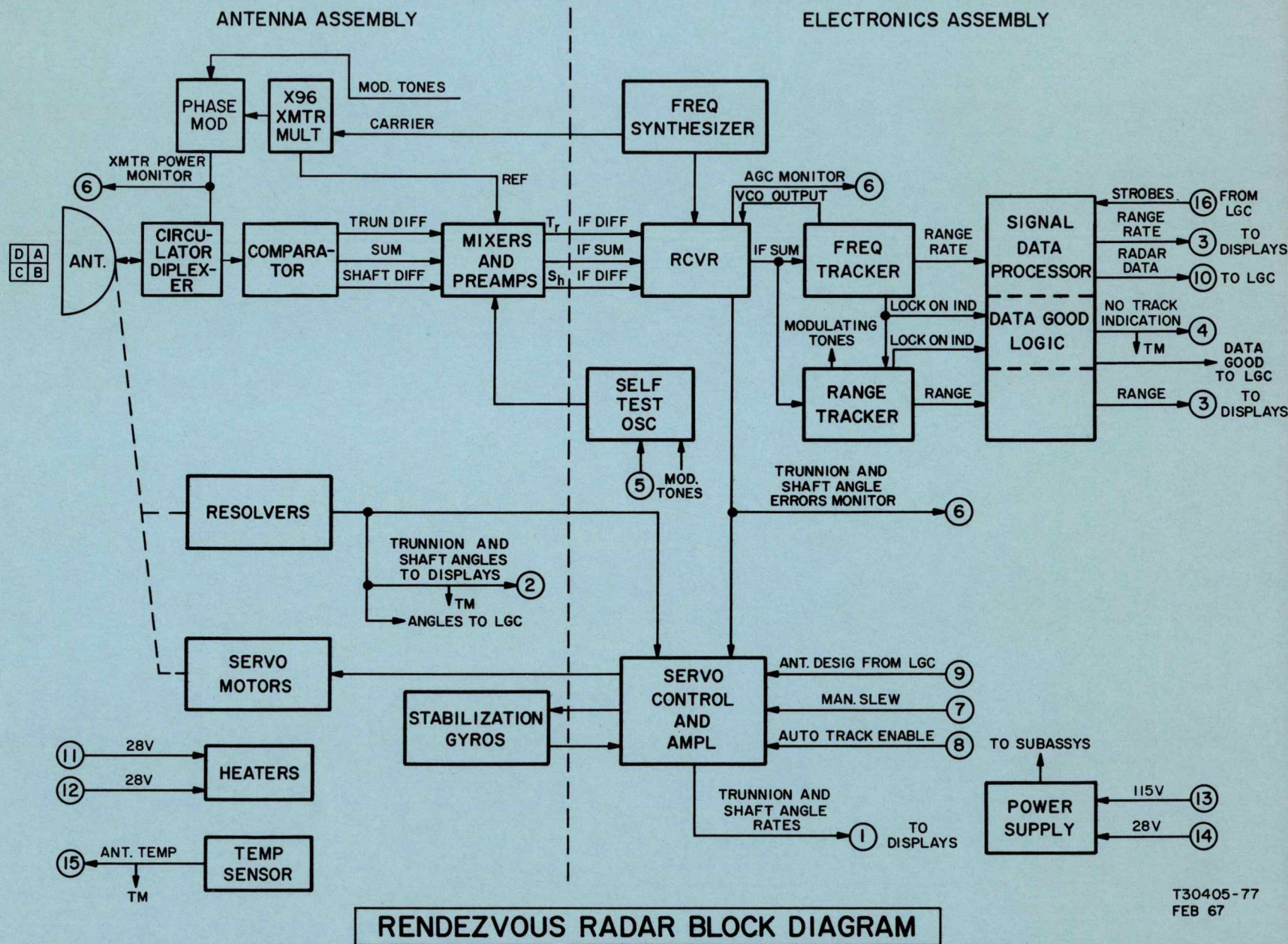




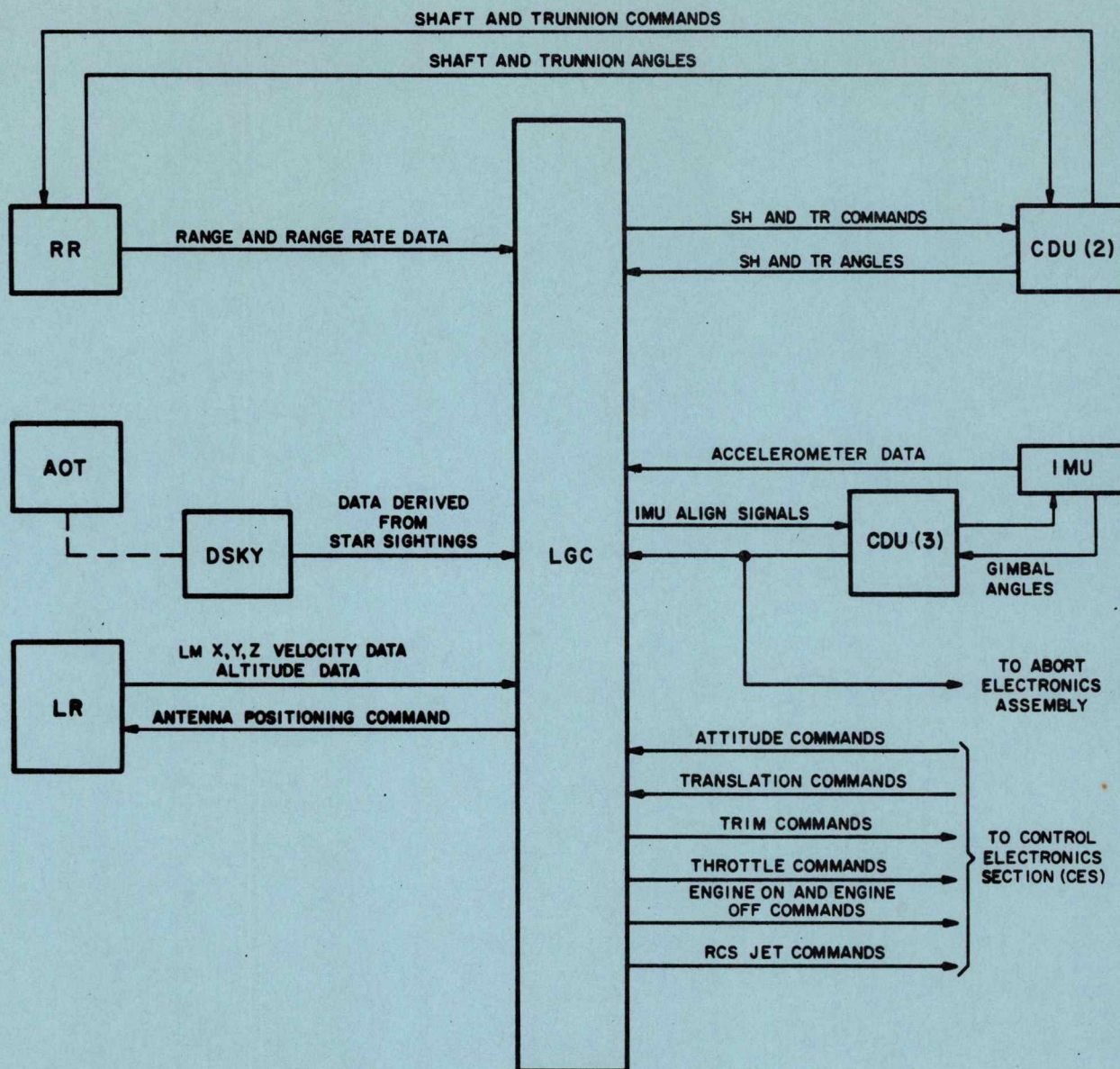
RENDEZVOUS RADAR ANTENNA ASSEMBLY  
HEATER CIRCUIT

T30405-148  
JAN 67





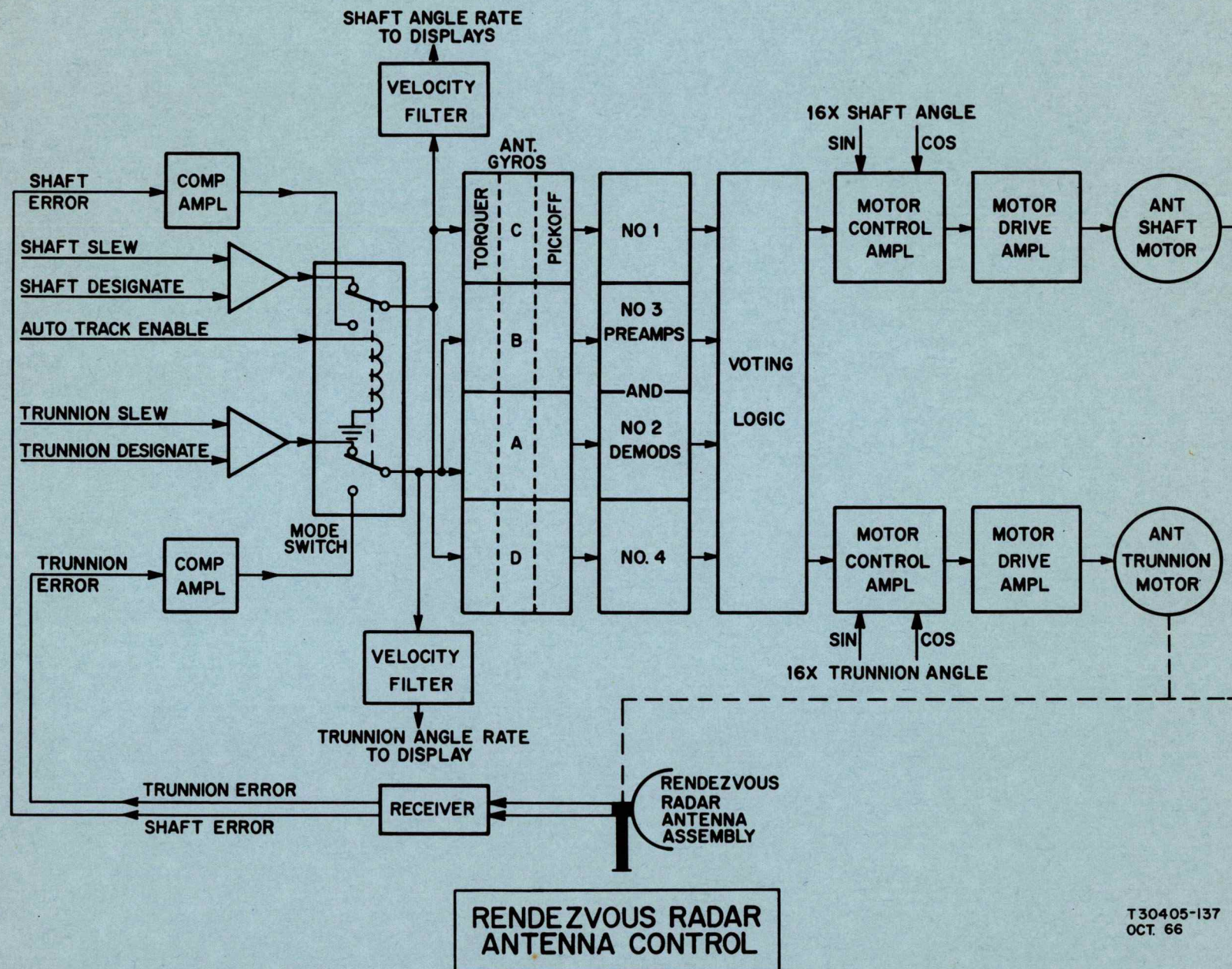




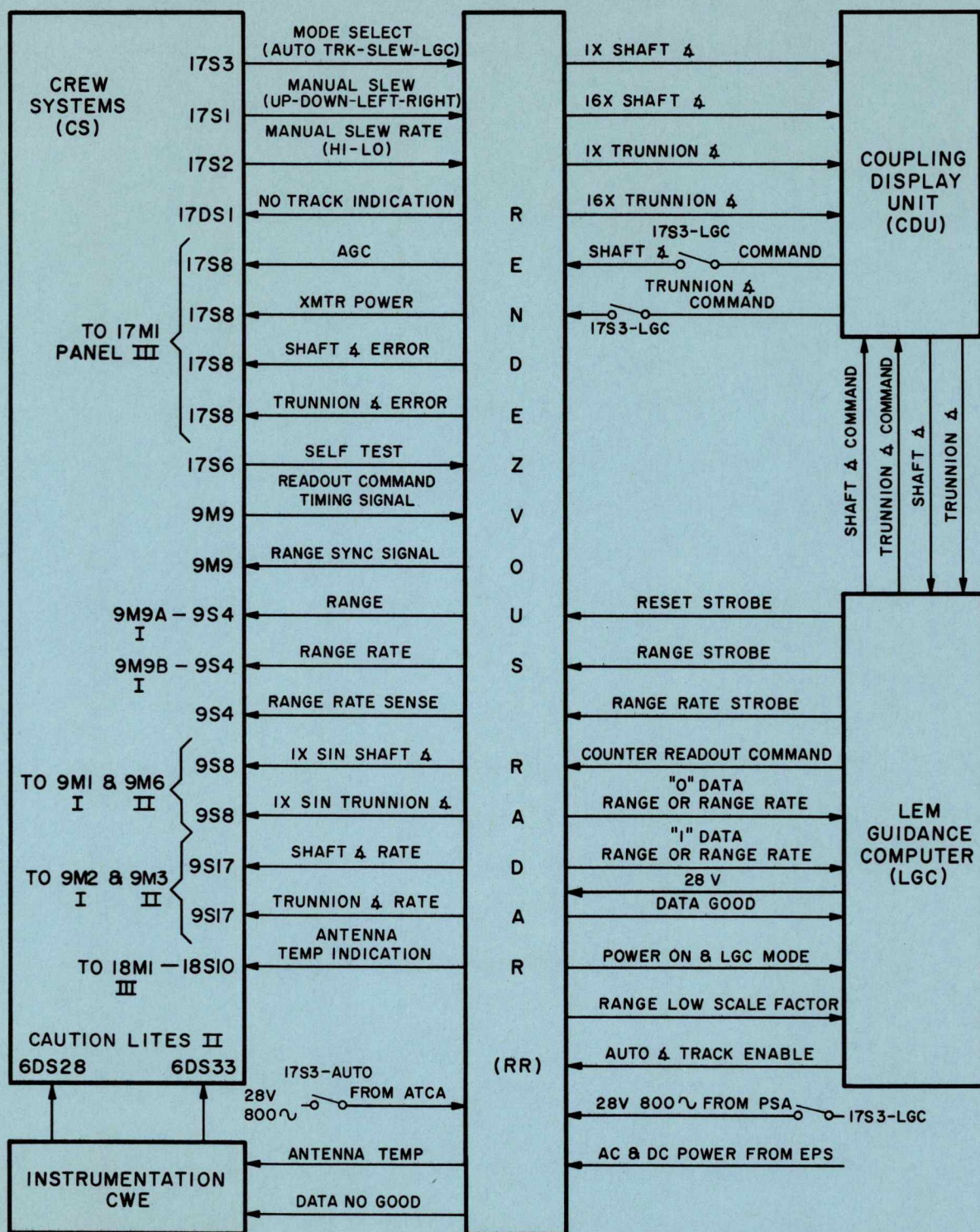
**PRIMARY GUIDANCE AND NAVIGATION SECTION — BLOCK DIAGRAM**

T30405-98  
FEB 67









## RENDZVOUS RADAR INTERFACE

T30405-90  
FEB 67



## CHAPTER III

### RENDEZVOUS RADAR TRANSPONDER

#### 1. INTRODUCTION

The RR Transponder is located in the Command and Service Module (CSM) and performs the function of receiving the Rendezvous Radar (RR) X Band CW Signal illuminating the CSM, and retransmitting (back to the RR) a phase coherent return signal.

The return signal is offset in fundamental carrier frequency from the received signal by 40.8 mc and contains the same modulation components phase related with respect to the received signal.

The Transponder is a part of the LM Radar Section which consists of a Rendezvous Radar in the LM, the Transponder in the CSM, and a Landing Radar mounted in the descent stage of the LM.

During the descent to the lunar surface, the LM and CSM maintain continuous radar contact through the Rendezvous Radar - Transponder link.

At the end of the lunar stay, the Rendezvous Radar is used to track the Transponder in the orbiting CSM to obtain orbital parameters, which are used to calculate the launching of the LM into a rendezvous trajectory.

In the rendezvous phase, the LM and CSM again maintain radar contact to obtain information needed for mid-course correction, rendezvous, and docking operations. By accepting the weak Rendezvous Radar transmitted signal, as discussed above, and by retransmitting (back to the RR) the phase coherent return signal, the range capabilities of the RR are greatly increased as power losses are reduced from fourth power to second power.

#### 2. PHYSICAL AND OPERATIONAL CHARACTERISTICS

##### Type of System

Electronics Assembly - Solid State coherent radar Transponder

Antenna Assembly - Four inch "Y" Horn and Twelve inch interconnecting waveguide between EA and Antenna.



## Performance Characteristics

Range - Operates with the Rendezvous Radar (RR) in a closed loop tracking system at LOS range between 80 feet and 400 nmi.

Range Accuracy - The Transponder will retransmit each of the range tones received from the Rendezvous Radar at the following maximum phase shifts.

Tone Frequency	Max. Degrees Phase Shift	Corresponding Maximum Ranging error
200 cps	$\pm 0.69^\circ$	1.78 mi
6.4 kcs	$\pm 1.0^\circ$	480 feet
204.8 kcs	$\pm 3^\circ$	45 feet

Range Rate Accuracy -  $1/4$  percent or 1 foot per second (whichever is greater).

Angular Coverage - Angular Coverage over a solid angle of  $160^\circ$  by  $105^\circ$ .

### Transponder Modes:

Signal Search Mode - In the Signal search mode the Transponder transmits a CW signal. The frequency is swept  $\pm 104$  kc about the nominal transmit frequency of 9792 mc with a sawtooth function at a rate of approximately 200 kc.

Transponder Mode - The Transponder provides a CW, sinusoidally phase modulated signal, at a frequency equal to that received from the Rendezvous Radar, multiplied by  $240/241$ . The phase modulation is at frequencies of 204.8 kc, 6.4 kc and 200 cps  $\pm 0.0015\%$ .

Self Test Mode - Enables astronaut to test transponder without RR signal.



## Mode Activation:

Signal Search Mode - The Transponder will be in the Signal Search Mode at all times that the Transponder is in an "ON" condition and no signal is being received from the Rendezvous Radar.

Transponder Mode - The Transponder will be in the Transponder Mode at all times that a signal is being received from the Rendezvous Radar and the power is on. Signals equal to or greater than -123 dbm which fall within the Transponder frequency range are automatically detected and acquired by the Transponder.

Self Test Mode - The Transponder will be in the Self Test mode when the self test enable signal is applied to the Transponder Assembly. The astronaut can Self Test the Transponder by placing System Test switches on system test Panel 101 to the proper positions.

## Monitor Outputs to the Control Display Panel

AGC Monitor	0 to 4.5 volts
-------------	----------------

Frequency Lock up	
-------------------	--

Search Mode	$0 \pm 0.4$ volts dc
-------------	----------------------

Transponder Mode	4.5 volts dc $\pm 0.5$ v
------------------	--------------------------

Transmitter Power - 2.5 volts dc  $\pm 0.4$  volts dc for specified minimum output power of 240 mw.

Telemetry Outputs - AGC Monitor, Frequency Lock Up, Transmitter Power. (TM outputs available but not presently used).

## Design and Construction Characteristics

### Size:

Electronic Assembly

12.75 x 8.531 x 7.40 inches (L x W x H)

### Weight:

approximately 14 pounds.



### 3. EQUIPMENT CONFIGURATION

#### a. Electronics Assembly

The Electronics Assembly (EA) of the Transponder is a solid state coherent radar transponder.

The Transponder Electronics Assembly is made by RCA and interfaces with the Transponder Antenna Assembly made by GAEC to form the Transponder Section.

The Transponder Electronics Assembly (TEA) is made up of the following subassemblies.

- A1 Cabling Subassembly
- A2 Carrier Lock Loop Subassembly
- A3 Receiver Subassembly
- A4 Tone Modulator Detector Subassembly
- A5 Power Supply Subassembly
- A6 Microwave Self Test Subassembly
- A7 Wiring Subassembly
- A8 Converter, Instrumentation Subassembly

The entire Electronics Assembly is housed in an Aluminum case, fabricated with exterior mounting flanges, for mounting to the cold rail in the Service Module of the CSM.

### 4. SUPPORT AND MAINTENANCE PLAN

The support and maintenance plan for the transponder follows that for the Rendezvous Radar. For a discussion of this plan refer to the appropriate section of the RR chapter.

### 5. BASIC BLOCK DIAGRAM - SIGNAL FLOW

Transmitted signals from the Rendezvous Radar (RR) are first received by the Transponder Antenna Assembly (AA) and then routed to the Microwave Circulator in the Microwave subassembly (A6).



The signal enters the input port of the Circulator and is routed CW to the output port, then to the first Mixer. A 102 mc signal from the Carrier Lock Loop (A2), is multiplied by 96 in the Frequency Multiplier Chain to the Transponder transmitting frequency, then routed through a Directional Coupler and applied to the first Mixer.

The 40.8 mc output of the Mixer (first IF frequency) is amplified by the pre-amplifier in the Microwave Subassembly and then routed to the Receiver Subassembly (A3) where it is applied to the First IF Amplifier.

The IF signal is amplified in the first IF amplifier and applied to the second mixer where it is mixed with a 34 mc signal from the carrier lock loop to produce a second IF signal of 6.8 mc.

The second IF signal is split and sent in several directions. One output is sent directly into the Threshold detector in the Carrier Lock Loop Subassembly (A2) as the Wideband Signal input. The second output is routed to the AGC section of the Receiver.

One output goes thru an 8 kc Filter and is sent out as the Narrow Band Signal to the Carrier Lock Loop Subassembly (A2) where it is applied to the Balanced Modulator and the Threshold Detector. The AGC signal is distributed separately to the IF preamplifier in the Microwave Subassembly (A6) and to amplifiers in the Receiver Subassembly (A3).

The Transponder transmits continuously. In the absence of an input received signal from the Rendezvous Radar, the Voltage Controlled Crystal Oscillator (VCXO) in the Carrier Lock Loop Subassembly (A2) sweeps over a narrow frequency range of approximately  $\pm 75$  cycles about a center frequency of 6.8 mc. After multiplication by 1440 this VCO output becomes the Xmtr output of 9792 mc  $\pm 104$  kc. The VCXO output is amplified and sent to the Frequency Multiplier, the Receiver and the coherent detector.

The Threshold Detector in the Carrier Lock Loop compares the 6.8 mc Narrow Band Signal with the Wideband Signal to produce a logic level to the Acquisition Logic and Sweep Generator. When no signal is present at the Narrow Band input, the Wideband input will cause the output of the Threshold Detector to be a logic zero which causes the Sweep Generator to sweep the VCXO.

This condition will exist until a signal is received from the Rendezvous Radar and there is a Narrow Band input to the Threshold Detector and also to the Balanced Modulator.



Again the Threshold Detector compares the Narrow Band input with the Wideband input. When the Narrow Band energy rises above the Wideband, the Threshold Detector output is a logic one and stops the VCXO sweep. The VCXO continues to oscillate at a stable frequency.

A portion of the VCXO output is sent to the Balanced Modulator. The modulator uses this input signal as a reference with respect to the Narrow Band signal received from the 8 kc Filter in the Receiver (A3).

When the Rendezvous Radar is transmitting, and there is a NB signal output from the 8 kc Filter to the Balanced Modulator, the NB signal and reference are compared in the modulator. The Balanced Modulator operates to maintain an identical frequency and phase relationship or phase locking between the VCO signal and the 6.8 mc IF narrow band signal. Any error results in an output from the Balanced Modulator which drives the VCO to a new value until the error is eliminated.

The X Band output of the X96 Multiplier chain is used both as the Transponder transmitter signal and as the Local Oscillator (LO) signal from the first mixer. The chain output going to the First Mixer is reduced by the 20 db Directional Coupler in the Microwave Subassembly.

The X Band energy output from the chain used for transmission is coupled through the Directional Coupler to the Waveguide Phase Modulator where the three tones from the originally received signal from the Rendezvous Radar are reintroduced, modulating the transmitted signal.

The modulated output signal from the Waveguide Phase Modulator is coupled to the Antenna Waveguide through the Circulator. The modulated transmitter energy enters one port of the Circulator and is rotated CW to the output port and then to the Antenna.

The Circulator in the Waveguide subassembly permits the dual use of the Antenna for both transmitting and receiving without significant leakage of the transmitted signal into the receiver circuits.

A Power Monitor detector at the output of the Microwave Subassembly provides a dc output to permit measurement of the transmitted power.



Referring back to the Second Mixer output, in addition to the signal going to the AGC and the Carrier Lock Loop circuits, the second IF output is distributed to the Coherent Detector in the Tone Modulation Detector (A4).

The Coherent Detector uses the 6.8 mc signal from the VCO as a reference to establish frequency coherence to the Rendezvous Radar transmitted signal.

The output of the Coherent Detector is sent to three parallel Bandpass Filters and Amplifier circuits, one for the 200 cps, one for the 6.4 kc, and the third for the 204.8 kc transmitted tones from the Rendezvous Radar. The two higher frequency tones are located in an oven controlled to  $160^{\circ} \pm 1^{\circ}\text{F}$  for stability.

Each circuit filters out one of the three tones, amplifies and routes it to the Phase Modulator in the Microwave subassembly (A6). In the Phase Modulator the three tones are phase modulated onto the newly generated carrier prior to transmission back to the Rendezvous Radar.

During initial operation, no tones modulate the signal received by the Transponder. Following initial phase locking in the Transponder Carrier Lock Loop and subsequent locking in the Rendezvous Radar Frequency Tracker, Logic in the RR permits the three tones to phase modulate the RR transmitted signal. After a brief interval of time which allows the RR Range Tracker Counter to be slewed by the 200 cps error signal, Logic in the RR switches off the low tone and the remaining two tones modulate the RR transmitted signal. As long as coherence is maintained in the Transponder the two tones (after initial locking) modulating the received signal are processed by the Transponder and are retransmitted back to the RR on the newly generated CW carrier.

Located in the Microwave Subassembly (A6) is the Self-Test Oscillator. A Self-Test enable voltage of 12 vdc is applied to the Oscillator through a switch contact closure in the Control Assembly to enable the oscillator. When enabled, the oscillator provides a 40.8 mc test output which is then applied to the preamp in the Microwave Subassembly for self-testing of the Receiver.

Voltage outputs are available to indicate Phase Locking (Frequency Lock Up), AGC action and transmitter power level.

The Power Supply Subassembly (A5) provides all the secondary electrical power of +25 vdc, +12 vdc and -12 vdc required by the Transponder Electronics Assembly. Primary power of +28 vdc is supplied through a CSM circuit breaker and the Transponder Power Switch to the power supply.

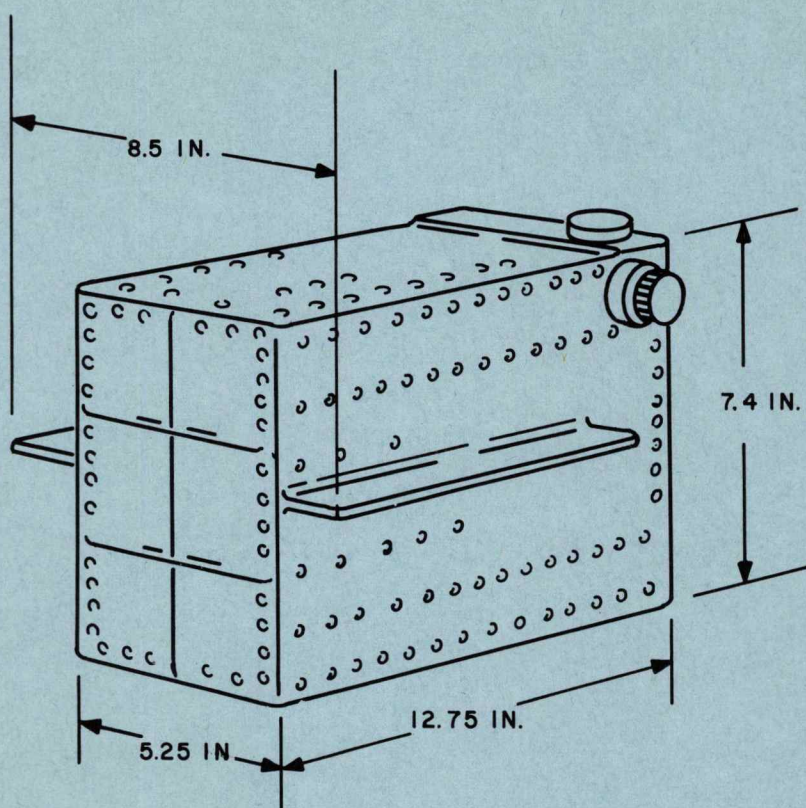


Not shown on the Functional Block Diagram of the Transponder Electronics Assembly is the Cabling Subassembly (A1), and the Wiring Subassembly (A7). The Wiring Subassembly provides the necessary wiring interconnection between the subassemblies in the Transponder. The Cabling Subassembly provides the external cabling interconnection between the Transponder Electronics Assembly and the Control Assembly in the CSM.

The Instrumentation Telemetry Subassembly TM Slice (A8) converts the Frequency Lock on signal, the AGC Monitor signal, and Power Monitor signal to usable dc signals for the CSM Telemetry and display. Presently, there are no telemetry points utilized or contemplated although the transponder hardware retains the capability.

All subassemblies of the Transponder are physically located inside the Aluminum Transponder case, thereby making them inaccessible. The Operational Connector and the GSE connector are located on the outside of the case to provide interface with the CSM or to the BTC for testing procedures, and for supplying input/output operating voltages and signal levels to the CSM telemetry and display circuits.

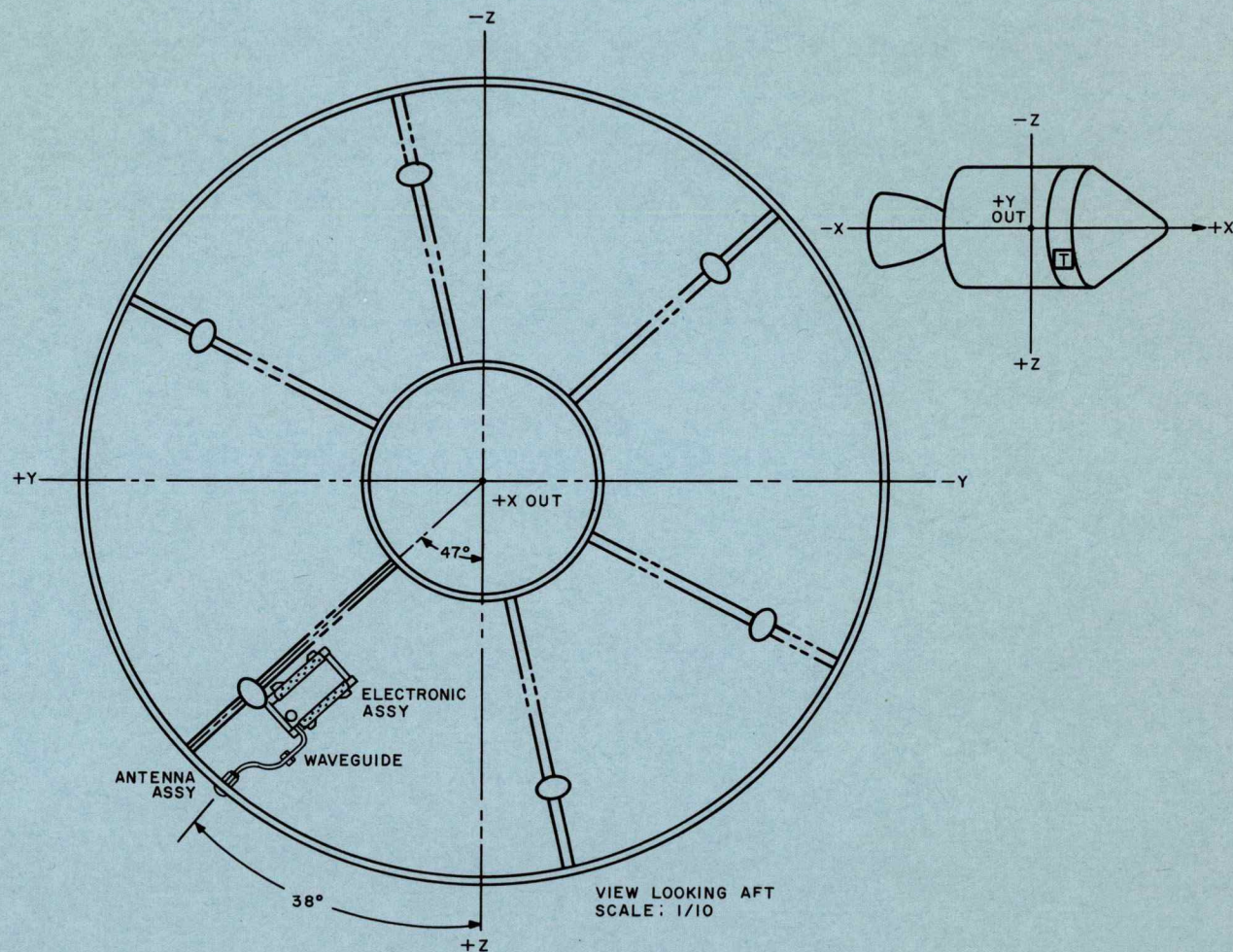




TRANSPONDER ELECTRONICS  
ASSEMBLY

T30405-158  
FEB 67

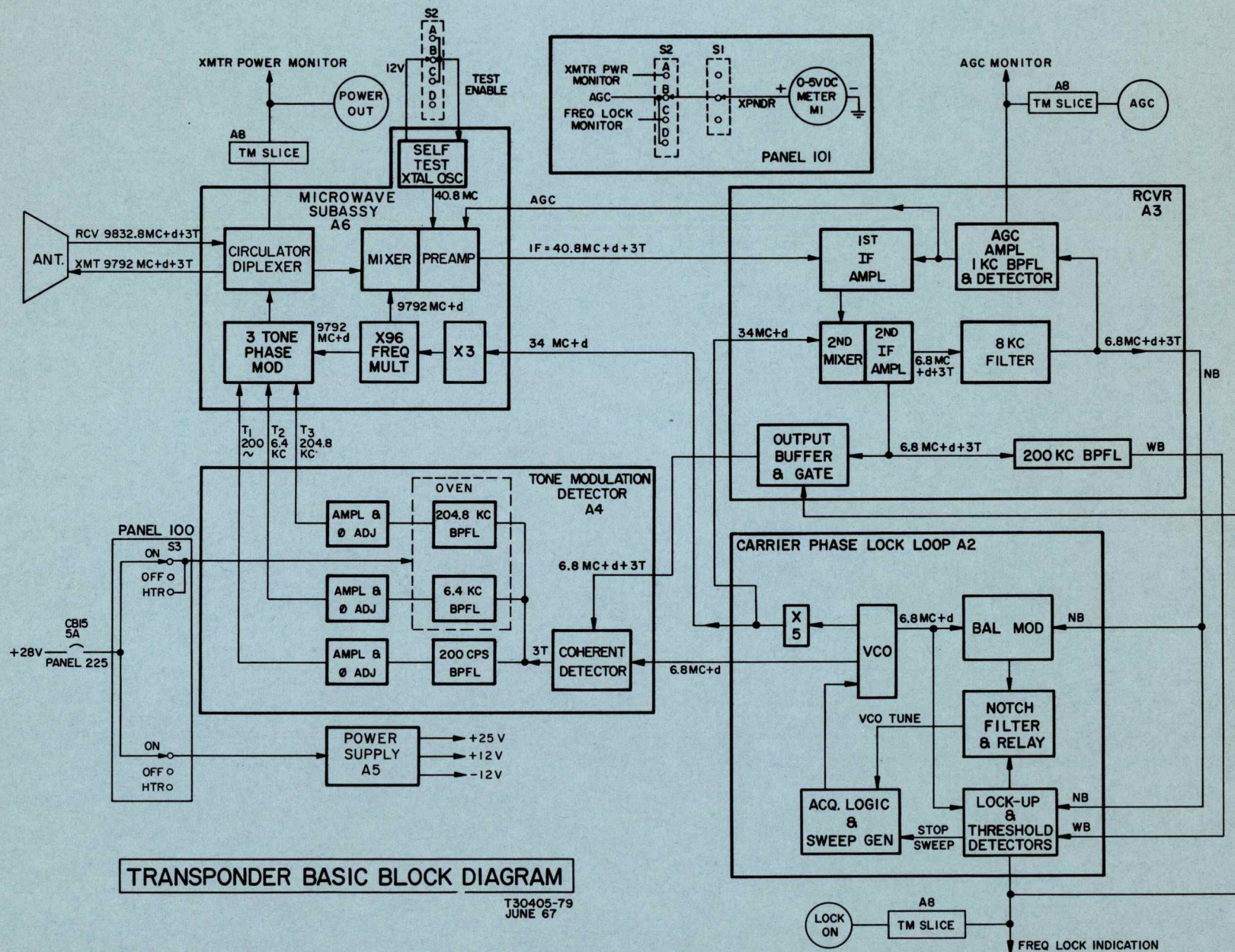




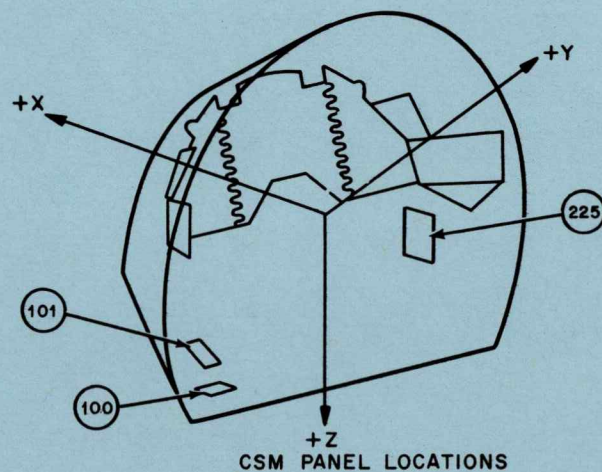
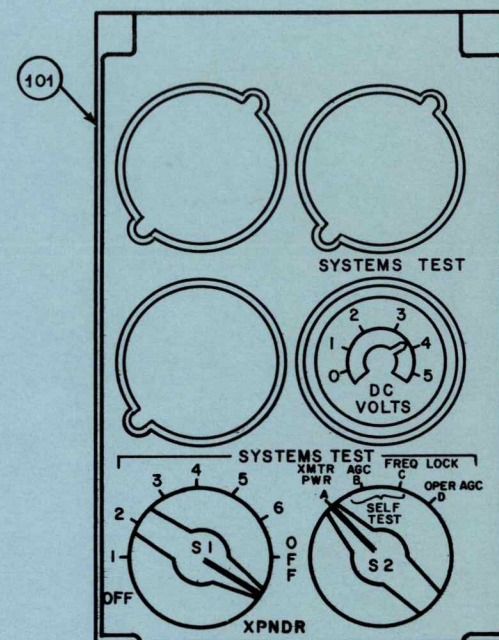
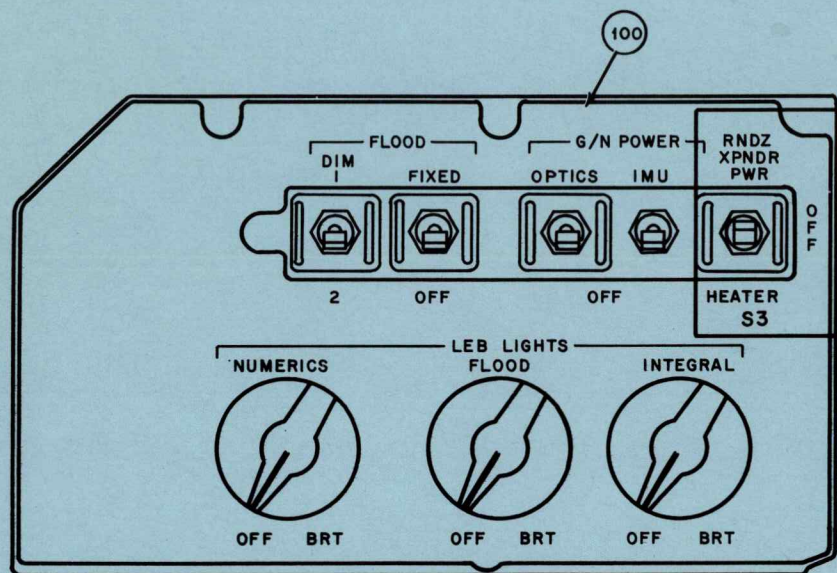
LOCATION OF TRANSPONDER IN CSM

T30405-157  
FEB 67









RR TRANSPONDER CONTROLS AND DISPLAYS

T30405-156  
FEB 67



## CHAPTER IV

### LANDING RADAR

#### 1. INTRODUCTION

The Landing Radar senses the velocity and range of the Lunar Module 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 Antenna Assembly (AA) serves to form, direct, transmit and receive four narrow microwave beams. To perform this function the AA 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 pre-amplifiers, two solid-state microwave transmitters, an FM modulator, and an antenna pedestal tilt mechanism. The Electronic Assembly (EA) 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.

This data obtained by direct microwave contact with the lunar surface is used by the computer and/or the astronauts to achieve a controlled rate of descent, a hover at low altitude to permit selection of a landing site, and a soft landing at the selected site. To accomplish these goals, the computer and/or astronauts must know the distance from the LM to the lunar surface and the velocity of the LM with respect to the lunar surface.

The cw microwave energy from the velocity sensor solid state transmitter is radiated toward the moon by the transmitting antenna. Reflected energy is received by three separate receiving antennas. The received doppler shifted energy, split in such a manner to form quadrature pairs, is mixed with a portion of the transmitted energy by microwave diodes, functioning as balanced mixers. The output of the balanced mixers is the difference frequency between the received and transmitted signals. The difference is the doppler shift which is directly proportional to velocity between the LM and the lunar surface.



The altimeter transmitter is frequency modulated by the Frequency Modulator in a sawtooth manner. The repetition rate of the sawtooth is 130 cps. The altimeter microwave energy is applied to the altimeter transmitting antenna. The received energy from the receiving antenna is split to form a quadrature pair and is coupled to balanced microwave 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. This doppler shift is undesired and must be compensated for in the range computer.

The quadrature outputs of the three velocity sensors and the altimeter balanced mixers are applied to the four Audio Frequency Amplifiers. Each of these amplifiers contains logic that switches the gain of the amplifier from 87 db to 55 db as the signal level increases to prevent distortion of the signal.

The broadband signals at the AF Amplifier outputs are applied to the inputs of frequency trackers which are located in the Electronics Assembly. The frequency trackers search for the signal over the expected frequency range with a narrow band tracking filter, and once acquired, track the signal with a high order of accuracy. The tracker output is an average frequency equal to the frequency corresponding to the center of power of the doppler or doppler and range signal spectrum plus a 614.4 kc reference. The tracker also provides a dc step voltage to indicate tracker lock.

The tracker outputs are applied to Velocity and Range Data Converters where beam velocity information indicated at the tracker outputs is resolved into velocity components. The coordinate system is referenced to the body coordinates of the antenna and is denoted  $V_{xa}$ ,  $V_{ya}$ , and  $V_{za}$ . The velocity information, as well as range information, is generated for use in the LGC and in the LM Displays. DC analog voltages are provided for  $V_{ya}$  and  $V_{za}$ , and pulse train information is given for range and  $V_{xa}$  (range rate) to drive the displays. A dc discrete signal is provided to indicate the sense of the  $V_{xa}$  pulse train. In antenna position number two the velocity components yield true LM axes velocities.

The velocity data in the body coordinates of the antenna are given in pulse train form superimposed on a 153.6 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 Indication and Range Scale Factor are provided to the LGC in the form of discrete relay contact closures.

The power supply for the Landing Radar is located in the Electronic Assembly. The power supply provides regulated voltages of +25, -25, +4, and -2 volts at 1% regulation and with the low ripple over an input voltage range of 25 to 31.5 vdc.

## 2. MISSION REQUIREMENTS

The Landing Radar Heater Circuit Breaker (4CB157 on Panel XI) is placed on at KSC launch pad and left on until lunar touchdown. The Landing Radar (LR) is first turned on and self-checked for five (5) minutes during LM checkout prior to the LM separation from the CSM. The self-check is accomplished by first manually activating the LR by closing the power circuit breaker 4CB105 on Panel XI, then after a 1 to 2 minute "warm-up" period, positioning the "Radar Self Test" switch (17S6) on Panel III to "LDG". This will provide a contact closure to the radar, which is used as a BIT activate command that permits the generation of internal self-test signals. These self-test signals will represent simulated "known" doppler returns, which after processing through the radar are made available to the LGC, and presented to the Astronaut's displays, where they are represented as range and velocities and may be monitored for presence and accuracy. An additional self check is made by positioning the meter selector switch (17S8 on Panel III) to the LR "ATM" or "VSOM", position and monitoring the "signal strength meter" for a relative output power reading from each transmitter. A half scale reading will indicate the minimum power output level.

If the above self-check is successful, 4CB105 and 17S6 are returned to their "off" position.

The LR is again self tested, in the same manner as described above, prior to the start of LM powered descent, at an altitude of approximately 70,000 feet.

The Landing Radar is turned "ON" (4CB105) at start of Powered Descent (50,000 feet above the lunar surface), and is left "ON" until touch-down approximately 11 minutes later.



LR beams #1, 2 and 4 should "acquire" the lunar surface from an altitude of approximately 25,000 feet. When this occurs the astronaut will be presented slant range on his "range and range rate display" (meter 9M9). Range data will also be available to the LGC from this point thru touchdown. The display slant range data is corrected by a .96 multiplication to give the astronaut a better indication of his true altitude to the lunar surface.

Due to the vehicle altitude, relative beam angles, and presently assumed lunar back-scatter coefficient, beam #3 is not required to "acquire" the lunar surface until 18,000 feet. When beam #3 "acquires", velocity data is made available to the LGC, and velocities  $V_{za}$  and  $V_{ya}$  are displayed to the astronauts. At approximately 8600 feet (Hi Gate) the LR antenna is tilted to the second of the two positions. This serves the purpose of making the antenna group beam center parallel with the LM X axis. At this point the LGC will strobe for and accept the velocity data from the LR for updating its calculations. Velocity and range data will be strobed for sequentially.

The Control and Display Assembly, located in the cockpit of the LM Ascent Stage, is prominently visible to the astronauts during descent maneuvers. The purpose of the assembly is to present to the astronauts in readily understandable form the LR operational status, slant range, and velocity information. In addition, the assembly will permit the astronauts to turn the LR on and off, to self-check the velocity sensors and the altimeter, and to place the antenna in either one of two fixed positions. In antenna position one and two the velocities are referenced to the antenna group beam center. In antenna position two the group beam center is parallel to the LM X axis and the computed velocities are therefore LM velocities. Range is always the range along the altimeter beam.

Antenna position No. 1 and beam angles are selected to optimize range accuracy at 15,000 ft. Antenna position No. 2 is selected to optimize range accuracy and velocity accuracy from 8600 feet thru touchdown.

When the spacecraft reaches 200 ft. altitude, LM makes a final pitch maneuver to orient the X axis perpendicular to the lunar surface. All velocity vectors should be near zero ft.sec. This is done to permit the astronauts to make final, visual selection of the most desirable landing site, and to manually control the soft let-down at the selected site. During this final and most critical phase of the landing part of the mission, the astronauts obtain altitude and velocity data from the LR via the cockpit displays. The capability exists for a completely automatic landing or a manually controlled landing as selected by the astronaut.



When the let-down is accomplished, the LR mission is over. The LR is then deactivated (all LR circuit breakers off) and left on the lunar surface in the LM Descent Stage when the LM Ascent Stage takes off the rendezvous with the CSM.

### 3. MAINTENANCE CONCEPT

The support and maintenance plan for the LR follows that of the RR. For a discussion of the plan refer to the appropriate section in the RR chapter.

### 4. BASIC BLOCK DIAGRAM SIGNAL FLOW

As previously stated, the Landing Radar serves the two primary functions of radar velocity sensor and radar altimeter. This section provides a signal flow type description of each of these two functions. First, however, there are certain circuits in the Landing Radar that serve both the Velocity Sensor and Altimeter and therefore do not logically belong to either. These separate or supporting functions (circuits) include the following:

- a. Antenna Pedestal tilt mechanism.
- b. Antenna Heating Elements.
- c. Low Voltage DC Power Supply.
- d. Reference oscillator (located in Velocity Converter).
- e. Signal Data Converter.
- f. BIT circuits.
- g. Cabling Subassembly.
- a. Antenna Pedestal - The Antenna Pedestal serves as both electrical and mechanical interface between the Antenna Assembly and the spacecraft. In addition it serves to position the antenna in either of its two operating positions (Position 1 - Descent (from 50K feet to 8600 feet), and Position 2 (Descent from 8600 feet to touchdown). The antenna positioning is performed by a tilt actuator (reversible DC motor) controlled manually by positioning the LDG ANT switch 17S5 on the astronaut's control panel III to the desired position or automatically by an LGC command if 17S5 is placed in the Auto position. The Auto position



is the normal mission position and will allow the LGC to switch the antenna from POS 1 (Descent) to POS 2 (Hover) at approximately 8600 feet. A position is selected by the application of +28 vdc on the appropriate control line. Power is removed from the actuator through the opening of a limit switch when the antenna reaches the selected position. Travel time in either direction is approximately 10 seconds. It is not practical to make this time shorter because of the relatively high pre-load used to prevent the antenna from vibrating off its stops. Nominal power drain during the 10 second travel time is 10 watts.

Antenna position indication is provided by a separate set of switch contacts which are located on the Antenna Pedestal. The discrete indication provided by the switch is used to generate Antenna Position 1 and Position 2 discrete indications to the LGC and Instrumentation.

- b. Antenna Heater - The design temperature range of the Landing Radar is from 0°F to +185°F. The LM surface temperature (where the Antenna is mounted) may vary between -260°F and +260°F during the long translunar and lunar orbital periods. The Antenna is partially protected from these extremes by thermal blankets and non-absorptive finishes. The low temperature extremes are however not adequately compensated for by these means when the radar is non-operating. To provide the additional protection needed during non-operate periods heating elements (cal-rods) have been imbedded in the transmitting array support structure. The heating elements control the antenna non-operating temperature between 0° and +20°F, and together with its control circuits represent a maximum power drain of 65 watts. Spacecraft power at +28 vdc is supplied to the control circuits and heating elements via contacts of a relay which is located in the cabling subassembly. This relay energizes and disables the heaters whenever power is applied to the LR Low Voltage DC Power Supply. The heater control circuit employs quadruple (X4) redundancy in switch drivers and thermistors to assure reliability. These switch driver circuits enable the power amplifier (which acts as a +28 vdc switch) when the internal antenna temperature reaches 0°F, and disables the power amplifier (open the switch) when the antenna temperature reaches +20°F.



- c. Low Voltage DC Power Supply - The Low Voltage DC Power Supply is mechanized in two subassemblies; the Inverter Subassembly and the Regulator Subassembly. Independent of packaging the power supply consists functionally of an input EMI filter, a pre-regulator used to boost input 28 v voltage to 33.5 vdc, a 10 kc AC to DC converter, a power transformer, rectifiers, post-regulators and a 25V interlock used to insure removal of both the +25 vdc and the -25 vdc if one or the other should fail. This interlocking is required in order to prevent excessive back-biasing of transistor junctions and electrolytic capacitors.

Total unregulated input power to the supply is 120 watts, with approximately 78 watts of regulated power delivered to loads, and the remainder, 42 watts, dissipated as heat in the power supply. The regulated outputs are +25.3 vdc for the ATM and VSOM, and for general use +25 vdc, -25 vdc, +4 vdc and -2 vdc.

- d. Reference Oscillator - The reference oscillator located in the Velocity Converter provides all the sub-carriers used in the LR for signal processing, and mathematical constants used in velocity coordinate and range computation. These are derived by dividing and mixing the output of a 2.4576 mc oscillator.  $F_c$  is 614.4 kc and is the basic reference signal. Other signals are usually expressed as a function of  $F_c$ , for example,  $F_c/4$ ,  $F_c/32$ , and  $F_c/64$ .

Landing Radar accuracy is largely dependent on Reference Oscillator accuracy. Therefore, the 2.4576 mc oscillator is a phase-lock oscillator that is synchronized with the 3.2 kc LGC clock reference.

- e. Signal Data Converter - The basic function of the SDC is to convert the radar outputs into the 15 bit format used by the LGC. To accomplish this, the SDC uses a 15 bit counter, transfer gates, and a 15 bit shift register. The LGC provides the timing signals which sequentially select the PRF input to be counted, determine the counting time, and provide the Data Readout Command required to shift out the contents of the shift register. The resulting 15 bit range or velocity word is shifted out "most-significant-bit-first" (MSBF) on two lines (two twisted shielded pairs). This permits a parity check to be made by the LGC, on the transfer functions, since both lines can never simultaneously be high or low.



- f. BIT Circuits - The BIT (built-in-test) or Self Test circuits generate fixed predetermined doppler and range signals which are fed to the signal processing channels of the Velocity Sensor and the Altimeter and produce known outputs on the display. A contact closure in the astronaut's Radar control panel III ("Self Test Switch" to LDG position) gates out  $F_c/64$  (9.6 kc) to the Velocity Sensor, and  $F_c/32$  (19.2 kc) to the Altimeter.

These signals go to phase splitters and scalars located in the V.S. and ALT Amp. Subassemblies of the Antenna Assembly. These circuits provide sine and cosine functions of the self test signals scaled to a level of 10 db above minimum acquisition level. The test signals are then applied to the 55 db pre-amplifiers and processed through the Velocity Sensor and Altimeter as if they were Lunar returns. The X-Pointers and the Alt/Range Meter are driven to set valves.

- g. Cabling Subassembly - The Cabling Subassembly contains all of the LR input/output wiring going to the operational connector. The subassembly also contains relays, relay drivers and pulse drivers. The relays are primarily those required to obtain the "discretes" for LGC and instrumentation, but include the Heater Power and 25 vdc interlocks. The pulse drivers shape and drive the Vxa and Range PRF outputs over the 45 ft. cable between the LR EA and the Display/Control Assembly.

#### Radar Velocity Sensor Functions

##### Transmitter

The CW Radar Velocity Sensor operates in the X-band (10.51Gc) using a single solid state transmitter in conjunction with a three-beam, electronically tilted transmitting array antenna. The reflected X-band energy in each beam is shifted in frequency in proportion to the relative velocities and beam angles of the LM and the reflecting surface.

$$F_d = \frac{2v}{c} \cdot F_t \cos \phi$$

Where:

$F_d$  = The doppler shift

$v$  = Relative velocity (total)



$F_t$  = Transmitted Frequency

$c$  = Velocity of Light

$\phi$  = Angle between velocity vector (total) and center of beam

#### Receiving Antenna Array

A portion of the reflected energy is picked up by the three Velocity Sensor receiving arrays, one array for each of the three beams. The receiving antennas are broadside array antennas, tilted mechanically for maximum efficiency in the direction of the corresponding transmitter beam.

#### Mixers

The received energy on each beam is routed through hybrids (signal splitting devices) to quadrature pair balanced mixers. In the balanced mixer, the received energy is combined with energy directly from the solid state transmitter to accomplish a direct RF to AF conversion. Double balanced (Quadrature pair) mixers are used after the hybrids to permit retention of doppler signal sense. The signal in one channel of the balanced mixers leads or lags the signal in the other channel of the balanced mixers by  $90^\circ$  depending on the sense of doppler which results from an opening or a closing velocity. A closing velocity results in a positive doppler and a  $90^\circ$  lead signal.

#### AF Preamplifiers

The AF output of the balanced mixers is a doppler spectrum resulting from the finite width (approximately  $4^\circ$  in the fore-aft direction) of the transmitted and received beams. This doppler spectrum is amplified and processed in a Dual AF Preamplifier, (55 db and 33 db ampl.) then routed to a Frequency Tracker. The Preamplifiers employ a switched AGC. At high altitudes, where signal strengths are low, the Preamplifiers use two stages of amplification: 55 db and 33 db. The output of each Dual AF Preamplifier is sampled by a threshold detector. When the output signal level rises to approximately 300 mv (indicating a strong input signal to the preamplifiers) a solid state switch is enabled and a low gain indication is given the respective Frequency Tracker. When the solid state switch is thus enabled, the 33 db amplifier is by-passed, and the doppler signal gets only 55 db of amplification.



## Frequency Tracker

The Frequency Trackers acquire and track the center of each doppler spectrum. In this process the doppler return is averaged and smoothed to obtain the doppler frequency  $D'$  representative of the doppler shift corresponding to the center of the RF beams. The smoothing tends to reduce effects of random noise and fluctuations resulting from uneven or different quality of reflecting surfaces. In addition the AF doppler is mixed with a 614.4 kc subcarrier used in the further processing of the signal in the Velocity Data Converter.

The Doppler Velocity Sensor (DVS) Frequency Trackers consist of two functional loops: the Tracking Loop and the Acquisition Loop. The Tracking Loop is basically a narrow band AFC circuit. Its purpose is to smooth and track the center of the doppler spectrum. Doppler sense is retained. The output is a pure sinusoid at the frequency of  $F_c + D'$ .

The basic functions of the Acquisition Loop are to sweep the Tracking Loop to within its acquisition (pull-in) capability and to prevent the Tracking Loop from locking on and tracking a false return. In addition, the Acquisition Loop provides a tracker lock signal (TL) used to determine LR operational status.

Tracker lock, or acquisition, is determined based on a signal to noise power comparison in the Doppler Signal Threshold Detector. A S/N ratio of 3 db is required before the tracking loop is permitted to lock on, and start tracking the doppler return.

When all three frequency trackers are locked the Data Good Logic issues a Velocity Data Good (VDG) discrete to the LGC and to Instrumentation.

## Velocity Data Converter

The Velocity Data Converter contains three identical doppler processing circuits, referred to here as the D1, D2 and D3 converter. These circuits extract the subcarrier  $F_c$ , synchronize and multiply the doppler frequency by four, indicate doppler sense, and based on doppler sense provide an output to the Velocity Data Computer (part of same block as Velocity Converter).

In addition to the three converters, the Velocity Data Converter contains the Velocity Sensor Bandwidth Control Circuit (VSBCC). This circuit is basically a frequency comparator used to switch each DVS Frequency Tracker to its Narrow Band-Pass Mode when (1)



its output frequency becomes less than 1.47 kc ( $D_1 < 1.47$  kc), and (2) to switch all of the DVS Frequency Trackers to the Narrow Band Pass Mode when altitude is less than 2500 ft. as indicated by the discrete signal from the Range Computer.

The Velocity Data Computer converts the beam doppler products,  $4D_1$ ,  $4D_2$  and  $4D_3$  into the antenna velocity coordinate outputs;  $V_{xa}$ ,  $V_{ya}$ ,  $V_{za}$ . The relationships computed are as follows:

Antenna Coordinate Outputs to LGC and Displays

$$V_{xa} = \left( \frac{D_1 + D_3}{2} \right) K$$

$$V_{za} = \left( D_3 - D_2 \right) K$$

$$V_{ya} = \left( D_1 - D_2 \right) K$$

The scale factors (K) used in interpreting the above velocity data specified as discrete numbers are derived from the appropriate trigonometric relationships of the beams associated with each velocity coordinate.

The three antenna coordinate computers each consists of an "adder, subtractor, divider", and a pulse shaper and driver. The "adder, subtractor, and dividers" form the velocity coordinate terms, and then combine this term with an  $F_c/4$  subcarrier in order to retain velocity coordinate **sense**. The resulting PRF modulated pulse train is applied to a pulse shaper and driver.

There are three display coordinate computers located in the Velocity Data Computer Sub-assembly. The outputs of these computers are dc voltages directly proportional to the measured velocity and are used to drive the velocity displays. The pulse train is applied to the "Gate and Averaging Circuit" (Digital to Analog Converter), where it is averaged to obtain the analog outputs for  $V_{YA}$  and  $V_{ZA}$ . The  $V_{XA}$  pulse train is applied directly to the range rate meter with a sense signal (open relay contact for closing range rate).



## Radar Altimeter Functions

### Transmitter

The FM/CW Radar Altimeter is very similar to the Doppler Velocity Sensor in circuitry and in operation. The Altimeter solid state transmitter is very similar to the DVS transmitter, however, it does not contain an internal oscillator. Instead it receives an input from the Modulator. This input is frequency modulated in a sawtooth manner, at a repetition rate of 130 cps. The Altimeter X-band (9.58 GC) microwave energy is applied to the altimeter transmitter antenna, a single-beam, electrically tilted array antenna.

A continuous sample is taken of the transmitted energy and mixed with the return energy. The resultant difference of the mixing process is called the range frequency and is proportional to the sawtooth frequency deviation rate, and two times the distance to the reflecting surface.

$$fr = \frac{2a \cdot dr}{c}$$

where

fr = Range Frequency

a = Distance to the reflecting surface

$$dr = \text{Deviation rate} = \frac{40 \text{ mc}}{7000 \text{ Msec}} (\text{Hi}) \quad \text{or} \quad \frac{8 \text{ mc}}{7000 \text{ Msec}} (\text{Lo})$$

(depending on Altitude  
or doppler signals received)

c = Velocity of Light

In addition to this frequency difference, (fr), there will be an additional frequency component (doppler) due to any relative velocity along the altimeter beam.

This doppler component, (called  $D_4$ ), is proportional to velocity along the altimeter beam, and adds to or subtracts from the range frequency (fr) depending on opening or closing velocity. A closing velocity results in a positive doppler signal. The signal processed by the Altimeter circuits can therefore be expressed as:



$$fr + D_4 = \left[ \frac{2a \cdot dr}{c} \right] + \left[ \frac{2v \cdot ft}{c} \cos \phi \right]$$

#### Mixer

Energy from the receiving antenna is split to form a quadrature pair of returns. These returns are coupled to microwave mixers along with a sample of the transmitted signal. The output is the difference frequency which is proportional to the time difference of the transmitted and received modulated energy, plus the doppler shift, as expressed above.

#### AF Preamplifier and Frequency Tracker

The quadrature pair output,  $(fr + D_4)$ , is routed to an processed by a Dual AF Preamplifier in the same manner as was done in the Velocity Sensor. After amplification,  $(fr + D_4)$  is routed to the Altimeter Frequency Tracker, where it is processed in a like manner as previously discussed for the doppler signals in the DVS Frequency Trackers. The main difference in operation is that the Altimeter Frequency Tracker contains a gate used to disable the tracking loop during flyback of the sawtooth modulation. This is accomplished with a 700 usec. blanking gate from the Oscillator/Modulator.

The altimeter frequency tracker has two outputs; the tracker lock indication (RTL) routed to the Data Good Logic and  $F_c + (fr + D_4)$  which is routed to the Range Converter for processing and doppler compensation (removal of the term  $D_4$ ).

In the Data Good Logic the term RTL is logically combined with the tracker lock indications TL1 and TL2 to provide the Range Data Good (RDG) signal used as a discrete indication of radar altimeter operation to the LGC and to Instrumentation.

#### Range Converter

Processing of the term  $F_c + (fr + D_4)$  consists of removing the term  $(F_c)$  and sending the signal to the Adder/Subtractor, where the term  $D_4$  is removed through logical combination with  $\frac{15}{32} (D_1 + D_2)$ .



The velocity compensated range frequency (fr), a negative going pulse train, is routed to a frequency comparator, called the "Reference Signal Generator" and to a pulse shaper and gating circuit.

The Reference Signal Generator circuit functions as a frequency comparator, in that it compares (fr) against reference frequencies derived from the Reference Oscillator. The reference frequencies are selected so as to provide a change in the output logic states (M) and ( $\bar{M}$ ) when slant range equals 2500 ft. on a descending trajectory, and 2600 ft. on an ascending trajectory. The logic state outputs are used as "mode control signals", and as "gating signals". As mode control signals, ( $\bar{M}$ ) (Altitude  $<$  2500 feet) is routed to the Velocity Sensor Bandwidth Control circuit, where it is used to simultaneously switch all DVS frequency trackers to the Narrow Bandwidth Mode whenever slant range is less than 2500 ft. ( $\bar{M}$ ) is also used to switch the ALT. Frequency Tracker to the Narrow Bandwidth Mode, and to accomplish the Oscillator/Modulator deviation rate change. Mode Control (M), is also called "Range Low Scale Factor", and as such it is routed to a driver and relay in the cabling subassembly, where it is converted to a range scale discrete signal for use by the LGC. When M is present, the altitude is greater than 2500 feet, a relay is energized to remove the low scale factor discrete from the LGC. At 2500 feet M is not present and the relay contacts close for the discrete indication.

The range pulse train (fr) is routed to the displays and to the SDC where it is processed by the 15 BIT counter and Shift Register in the same manner as previously described for Vxa, Vza and Vya. The primary difference is that (fr) is not riding a bias frequency.

## 5. LANDING RADAR

### Physical and Operational Characteristics

#### Type of System

Velocity Sensor	CW, 3-beam
Radar Altimeter	FM/CW

#### Altitude Capability

Velocity	5 to 18,000 feet
Altimeter	10 to 25,000 feet



Velocity Capability	<u>Hi Alt Mode</u>	<u>Lo Alt Mode</u>
$V_{xa}$ :	-2000 to + 500 fps	-200 fps to 300 fps
$V_{ya}$ :	-500 to + 500 fps	-200 fps to +300 fps
$V_{za}$ :	-500 to + 3000 fps	-200 fps to +300 fps

#### Weight

Specified Weight 27.75 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 132 watts dc max

#### Altimeter Antenna

Type Planar array, space duplexed

Gain (Two-way) 50.4 db

Beamwidth (Two-way) 3.9 degrees E plane Fore-Aft  
6.5 degrees H plane Port-Stbd

#### Velocity Sensor Antenna

Type Planar array, space duplexed

Gain (Two-way) 49.2 db

Beamwidth (Two-way) 3.7 degrees E plane  
7.3 degrees H plane



## Transmitters

Type Solid State

### Frequency

Velocity Sensor 10.51 gc

Radar Altimeter 9.58 gc

### Output Power

Velocity Sensor 200 mw (3 beam total)

Altimeter 100 mw

Altimeter Modulation Sawtooth FM

Modulation Frequency 130 cps

Deviations  $\pm 4$  mc (Lo)  $\pm 20$  mc (Hi)

Alt > 2500ft Alt < 2500ft

### Outputs

#### Serial Binary to Computer

<u>Parameter</u>	<u>Scale Factor</u>
$V_{xa}$	0.5148 fps/count
$V_{ya}$	1.002 fps/count
$V_{za}$	0.6857 fps/count
Range > 2500 ft	2.158 ft/count
Range < 2500 ft	0.4317 ft/count

### Display Outputs

#### Analog

$V_{ya}$  (-200 to +200 fps) 25 mv/fps ( $\pm 5$  volts max.)

$V_{za}$  (-200 to +200 fps) 25 mv/fps ( $\pm 5$  volts max.)



## Pulse Trains

$V_{xa}$	(-500 to +500 fps)	19.45 pps/fps
Range	(2500 - 25,000 ft)	2.32 pps/ft
Range	(10 - 2500 ft)	11.6 pps/ft

Status Indications (Contact Closures referenced to  $28 \pm 11$  V)

### LGC:

Velocity Data Good

Range Data Good

Range Low Scale Factor

Antenna Pos 1

Antenna Pos 2

Displays: (Isolated Contact Closures)

$V_{xa}$  Sense (closed contact for ascent)

Instrumentation: (Contact Closures)

Range Data No Good

Velocity Data No Good

Antenna Pos 2/Not Pos 2

LR Power On

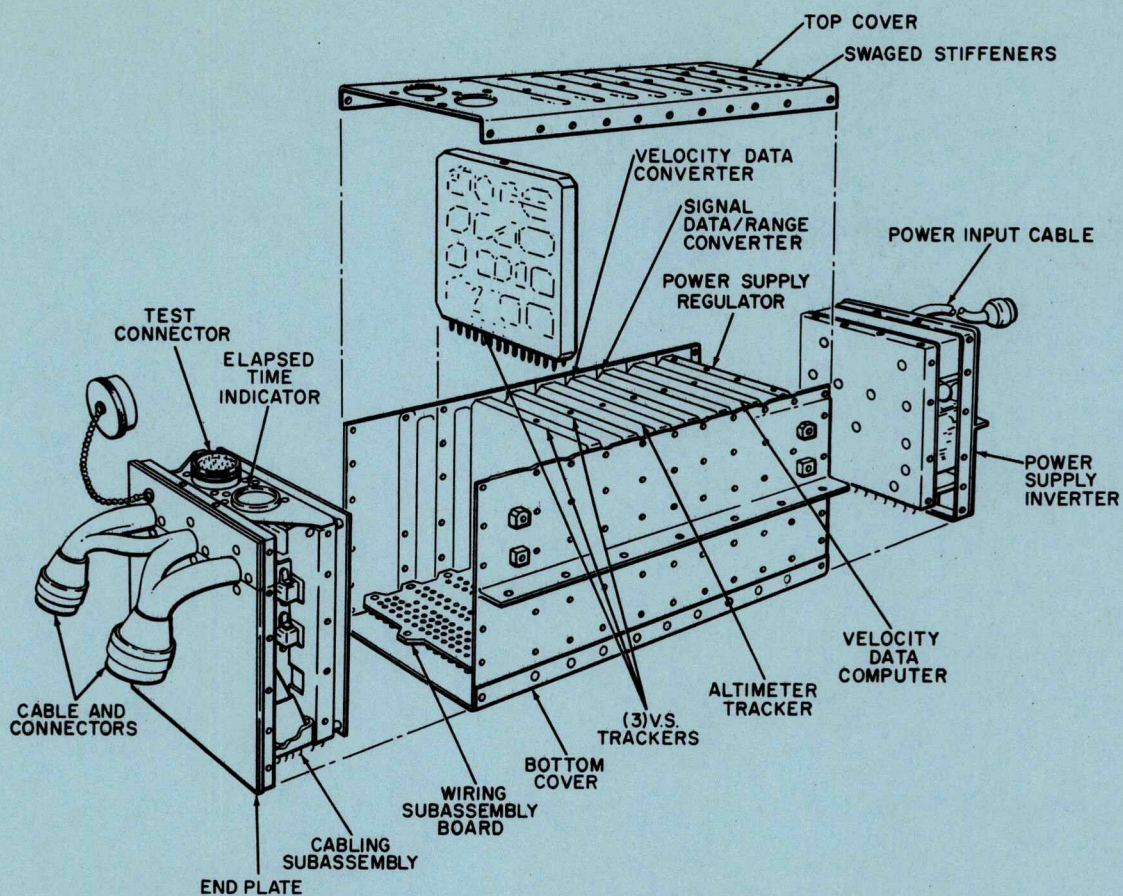
### Accuracy

#### RANGE

Component	25K to 2K ft.	2K to 200 ft.	200 ft to 5 ft.
$V_{xa}$	1.5% or 1.5 ft/sec.	1.5% or 1.5 ft/sec.	1.5% or 1.5 ft/sec.
$V_{ya}$	2% or 2 ft/sec.	3.5% or 3.5 ft/sec.	2% or 1.5 ft/sec.
$Y_{za}$	2% or 2 ft/sec.	3% or 3 ft/sec.	2% or 1.5 ft/sec.
Range	1.4% + 15 ft	1.4% + 5 ft	1.4% + 5 ft

Use % or ft/sec., whichever is greater.

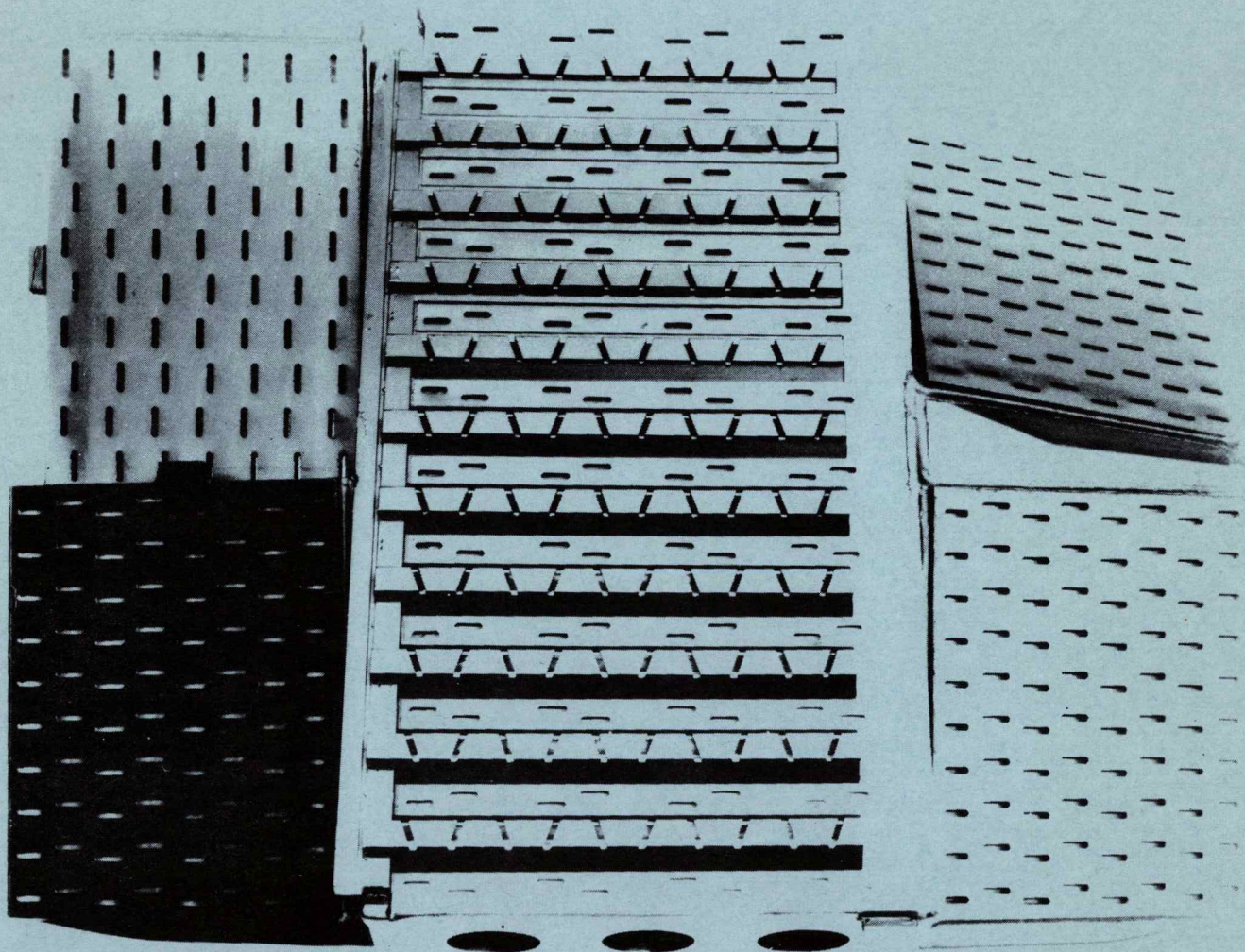




**LANDING RADAR ELECTRONIC ASSEMBLY**

T30405-121  
JULY 66



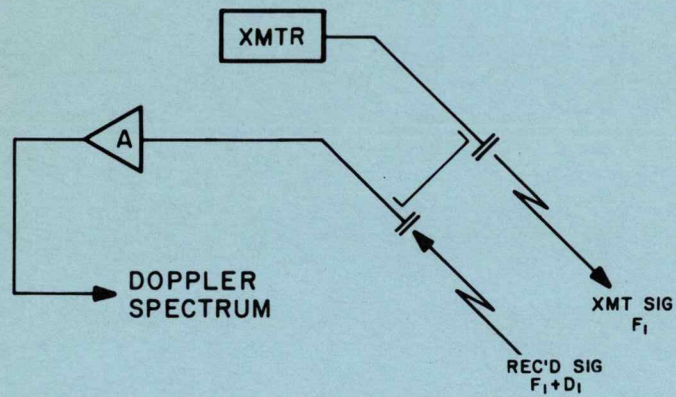


LANDING RADAR ANTENNA ASSEMBLY

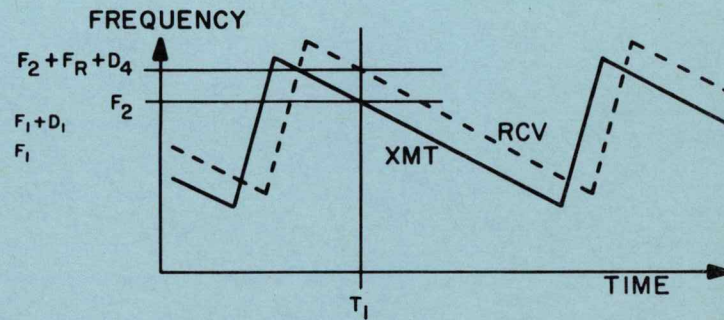
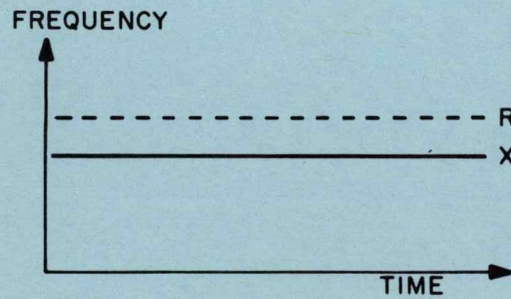
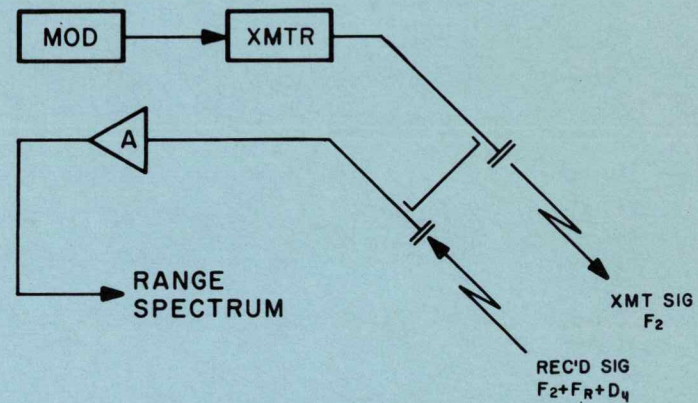
T30405-81  
OCT 66



# VELOCITY SENSOR (CW)



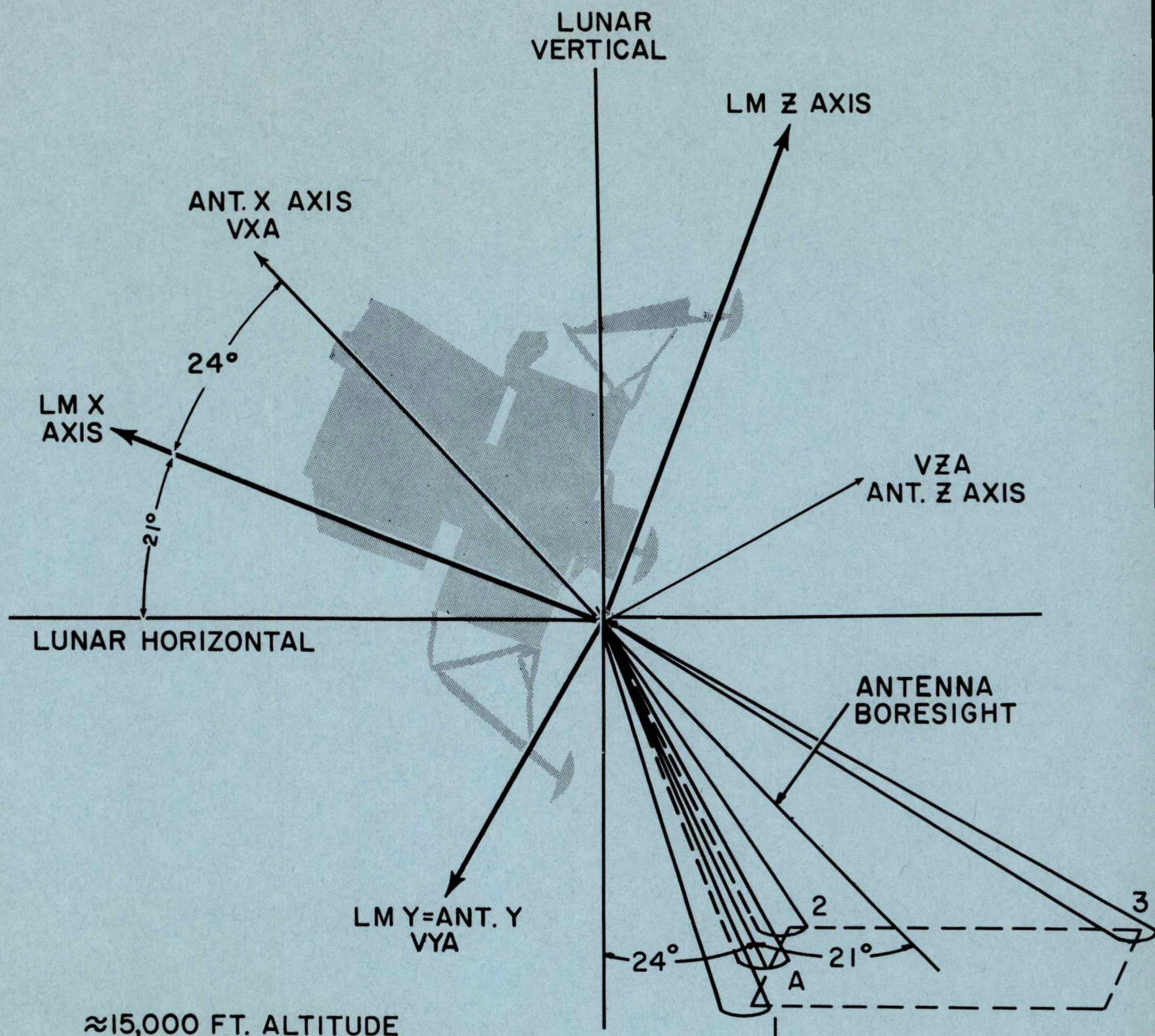
# ALTIMETER (FM/CW)



## LANDING RADAR CONCEPTS

T30405-25  
OCT 66

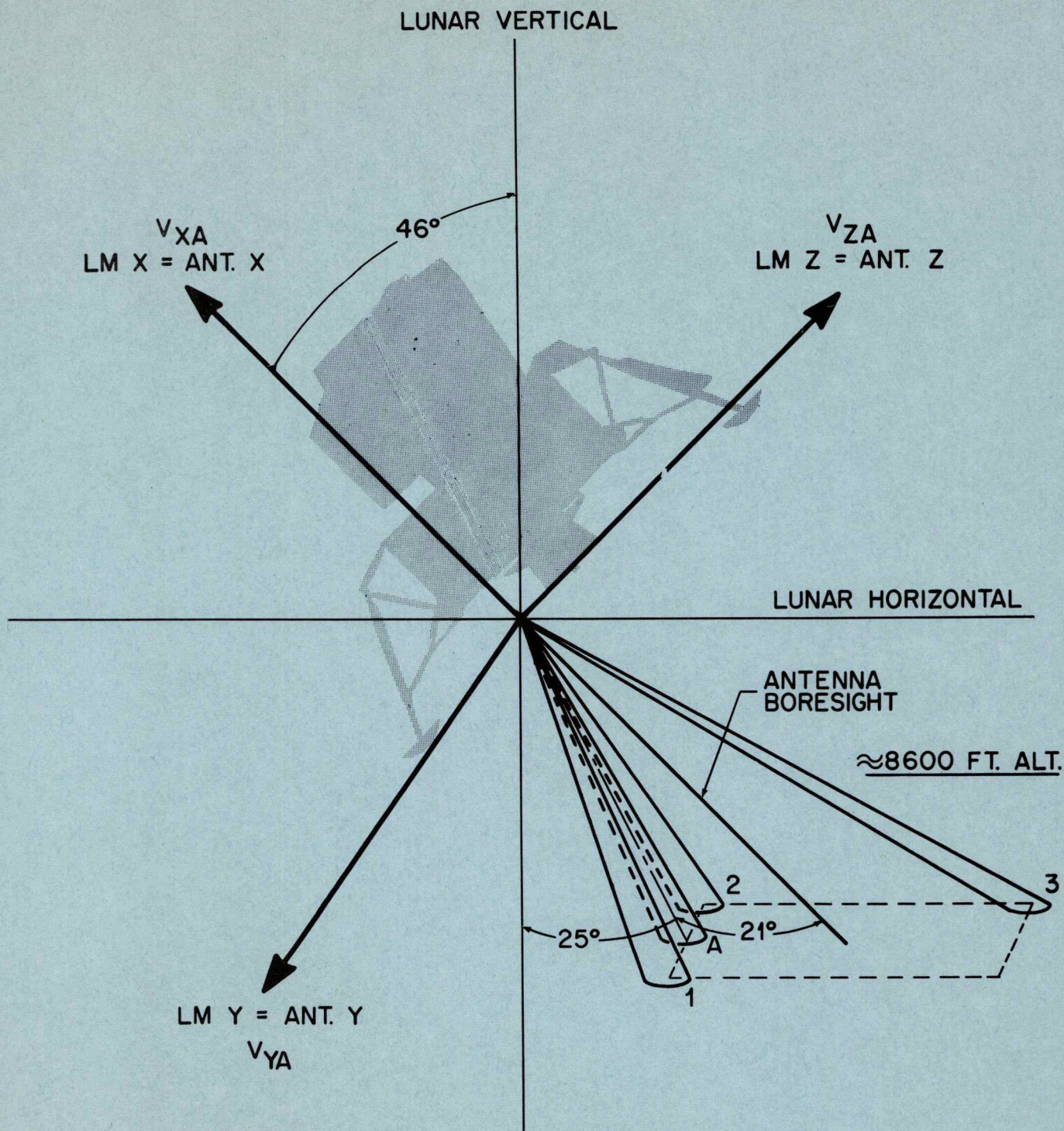




# LR BEAM CONFIGURATION DURING POWERED DESCENT

T30405-40A  
FEB 67

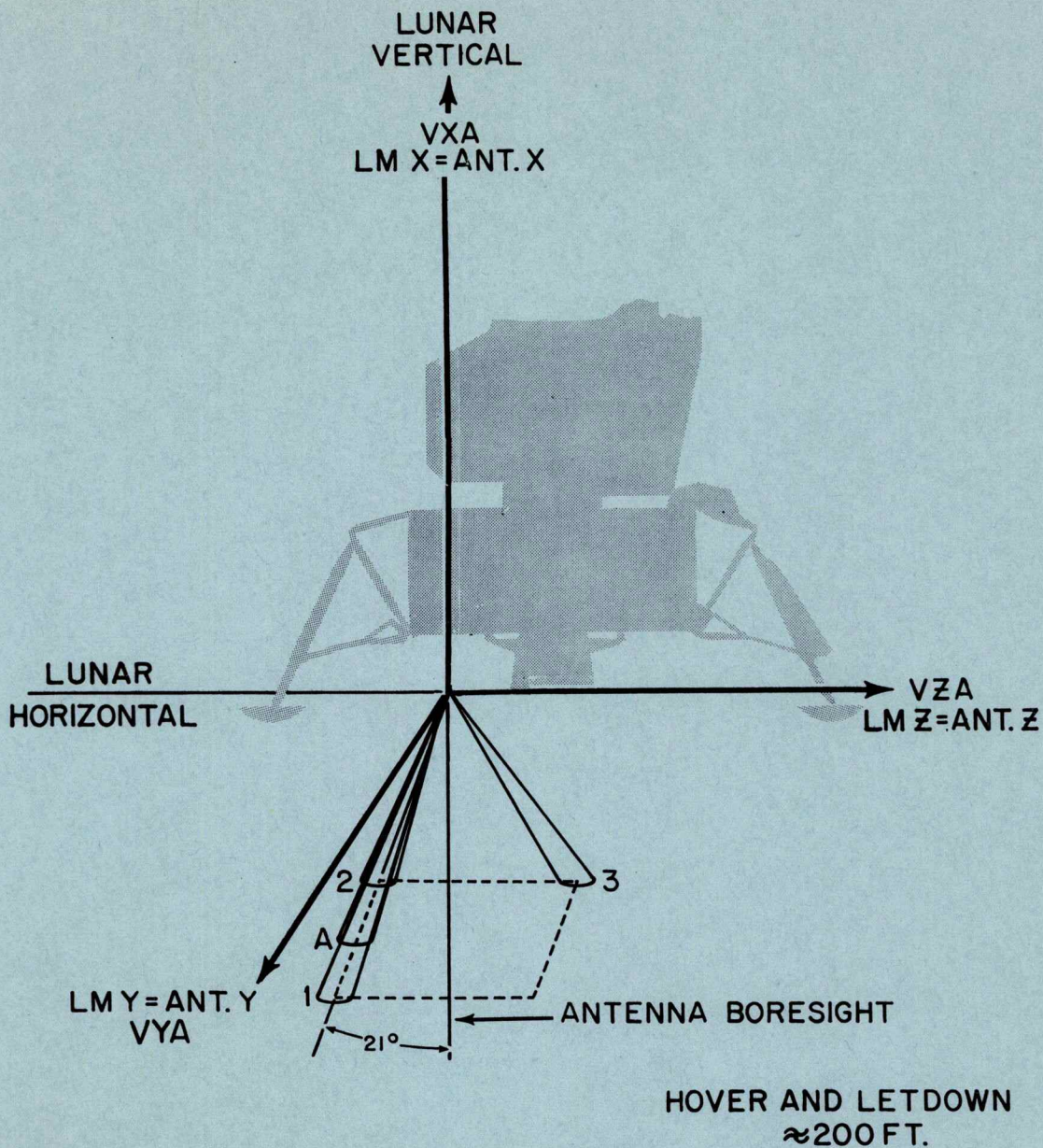




LR BEAM CONFIGURATION  
AT HIGH GATE

T30405-40B  
FEB 67

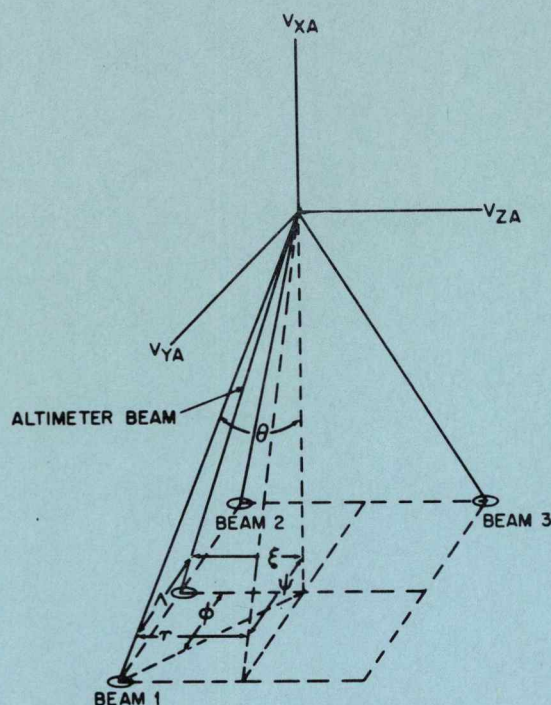




# LR BEAM CONFIGURATION AT HOVER

T30405-40C  
FEB 67





BEAM CONFIGURATION

VELOCITY EQUATIONS

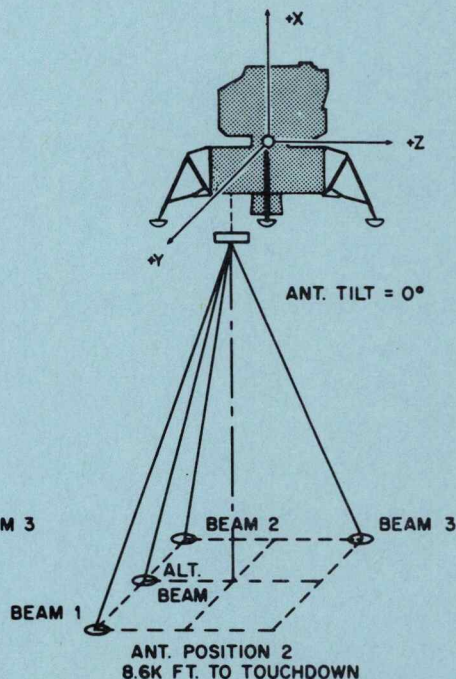
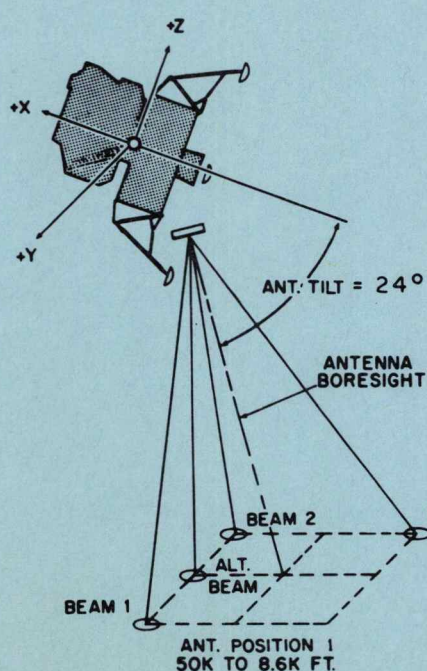
$$V_{xa} = \frac{\lambda}{2 \cos \Lambda \cos \xi} \frac{(D_1 + D_3)}{2}$$

$$V_{ya} = \frac{\lambda}{4 \sin \Lambda} (D_1 - D_2)$$

$$V_{za} = \frac{\lambda}{4 \cos \Lambda \sin \xi} (D_3 - D_2)$$

BEAM ANGLES

$\gamma$ GAMMA	19°	58'	56"
$\theta$ THETA	24°	27'	36"
$\Lambda$ LAMBDA	13°	31'	17"
$\xi$ XI	20°	34'	38"
$\phi$ PHI	34°	22'	47"
$\psi$ PSI	14°	24'	20"



ANTENNA TILT ANGLES

LR ANTENNA BEAM CONFIGURATION AND TILT ANGLES

T30405-135  
FEB 67



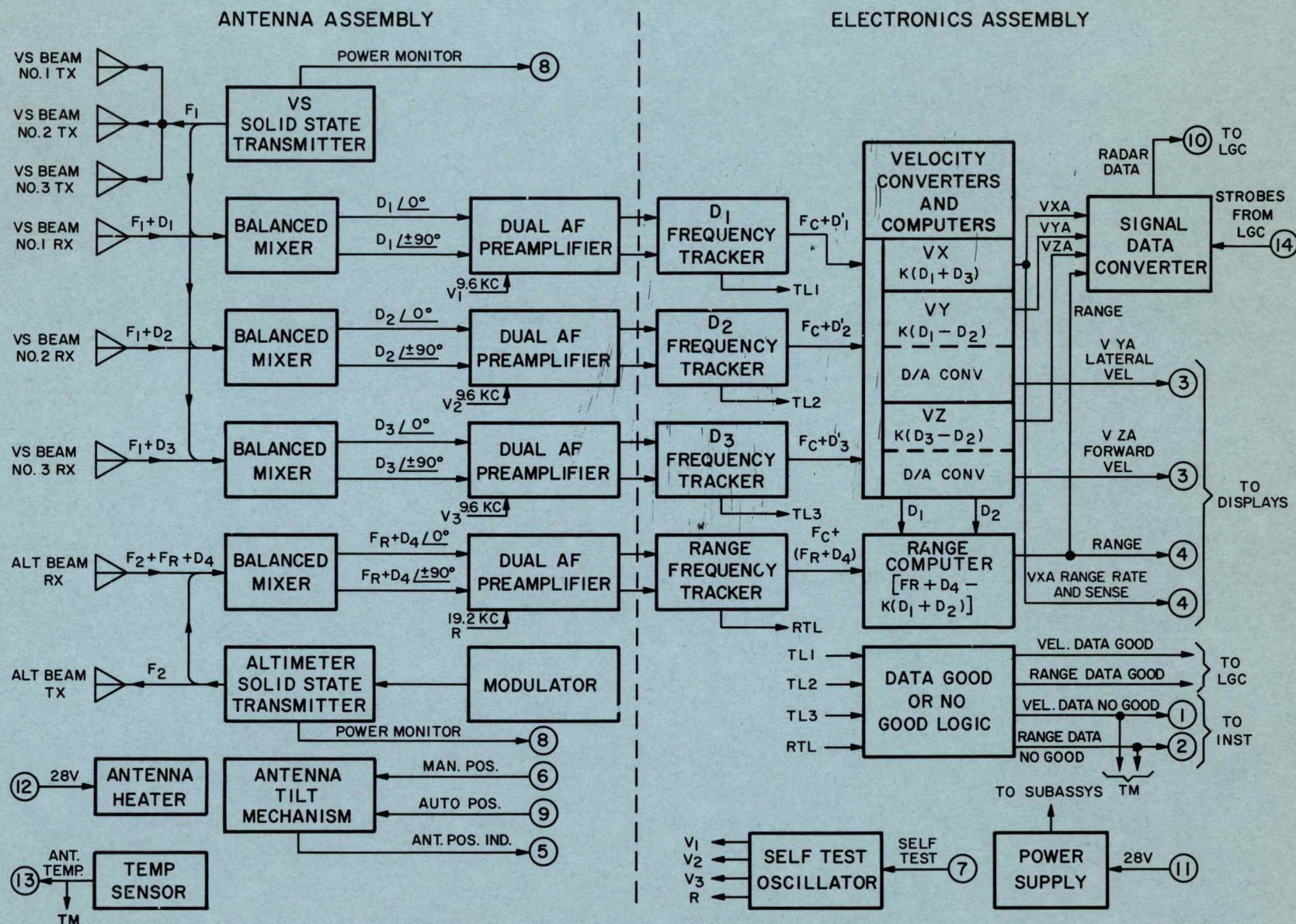






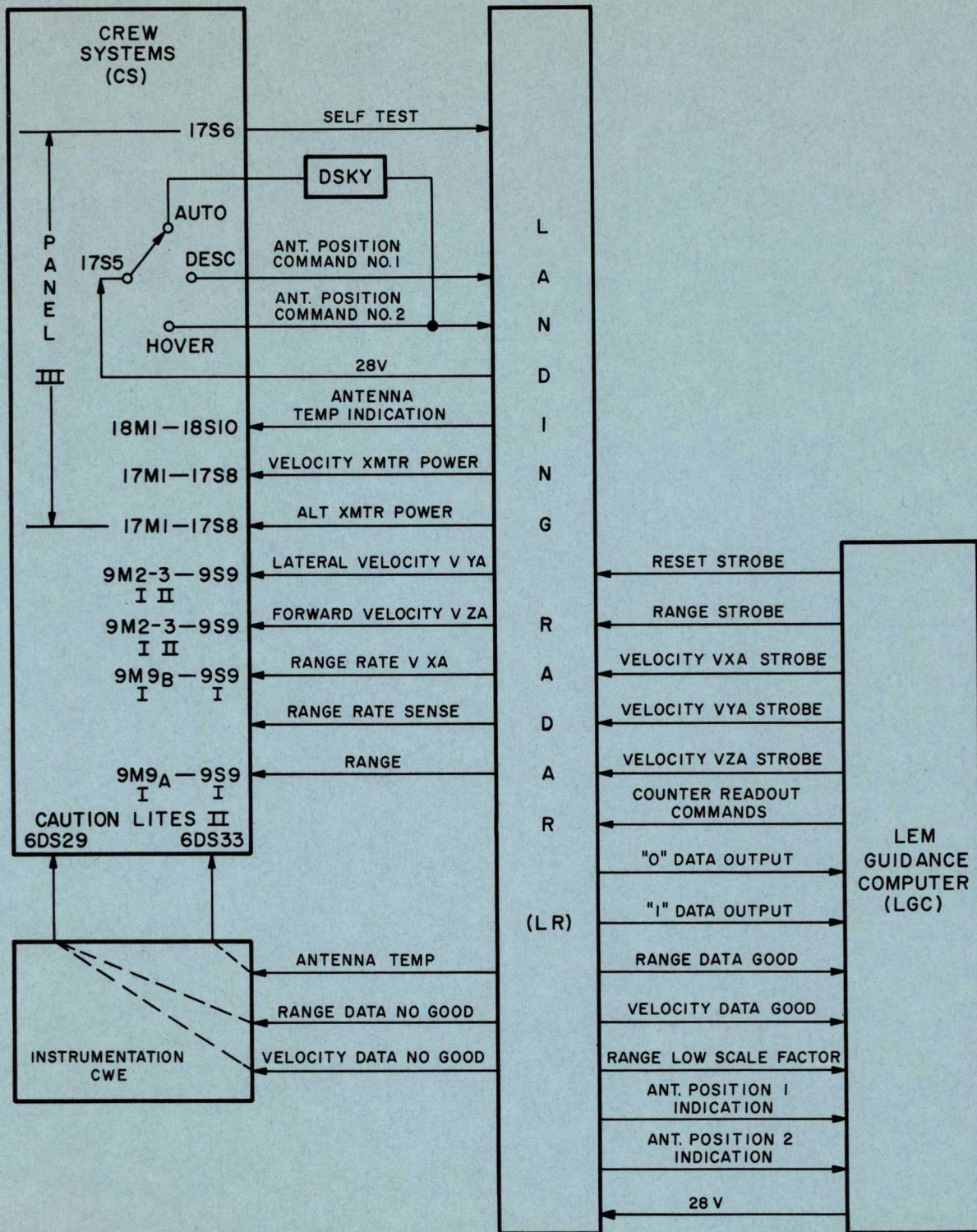
Figure 4-9

T30405-87  
FEB 67









LANDING RADAR INTERFACE

T30405-92  
FEB 67



## CONTROLS AND DISPLAYS

Since the LM is manned, the crew must be furnished data relative to the status of the subsystems, be warned of malfunctions, be provided with guidance and navigational readouts, and once supplied this information, be able to take the required action.

Many considerations and principles went into the layout and design of the controls and displays. Controls and displays relating to a particular function are generally grouped together. Thought was given to the 0 and 1/6 "G" environments to be encountered, as well as to space suit limitations. As details are discussed in more depth, the philosophy of design and placement will be better understood.

Displays

Displays may be described as devices used to exhibit information. Several types are used in the LM. Only those displays used in conjunction with the Radar Section will be covered in this chapter.

Controls

Controls are selected to suit their function. For instance, a common toggle switch may be found in various configurations. There are two, three, and four position toggles. Each position may be a "Maintain" (MAIN), in which the toggle will stay in the position selected, a "Momentary" (MOM), where the lever is spring loaded to return to its original position, or a "LOCK", where a mechanical restraint must be released in order to move the lever.

Panels

Displays and Controls are mounted on panels. Each panel is identified by its location and by a Roman numeral designator.

Alpha Numeric Designators

Every display and control is assigned an alpha numeric designator. Below is an explanation of the system used.

FIRST DESIGNATOR

- 1 = ENGINE/THRUST CONTROLS
- 2 = EXPLOSIVE DEVICE (PYROTECHNICS)



- 3 = GUIDANCE AND NAVIGATION
- 4 = ELECTRICAL POWER
- 5 = TO BE DETERMINED
- 6 = CAUTION/WARNING
- 7 = ENVIRONMENTAL CONTROL
- 8 = REACTION CONTROL
- 9 = FLIGHT CONTROL
- 10 = PROPULSION
- 11 = STABILIZATION AND CONTROL
- 12 = ABORT GUIDANCE
- 13 = COMMUNICATIONS
- 14 = COMMUNICATIONS ANTENNAS
- 15 = AUDIO CONTROL
- 16 = LIGHTING CONTROLS
- 17 = RADAR CONTROLS
- 18 = HEATER CONTROLS

SECOND DESIGNATORS

- S - SWITCH
- M - DISPLAY (METER-DIGITAL DISPLAY-FLIGHT INDICATOR, ETC.)
- DS - CAUTION/WARNING, COMPONENT CAUTION INDICATOR, OR LUNAR CONTACT/LIGHT
- R - THUMBWHEEL OR CONTINUOUS ROTARY SWITCH
- CB - CIRCUIT BREAKER
- FL - STATUS



- T - AUDIBLE WARNING TONE
- R - THUMBWHEEL OR CONTINUOUS ROTARY SWITCH
- CB - CIRCUIT BREAKER
- FL - STATUS FLAG
- T - AUDIBLE WARNING TONE

#### Commander's LH Center Panel

The Commander's LH Center Panel, or Panel I, is used by the commander to perform most of his primary tasks. Below is a chart which gives the name and a brief description of the function of each control and display associated with the Radar Section.

The Warning Light array, found on the upper center portion of the panel consists of lights which indicate a malfunction or failure of a critical component or subsystem. Whenever a failure is sensed, one of these red lights comes on, the Master Alarm Light (6S2) illuminates, and an audible tone (T-1) is introduced into the headsets of both crew members. The tone may be stopped and the Master Alarm Light extinguished by depressing the Master Alarm pushbutton, but the appropriate Warning light will remain on until the trouble is cleared up.

A Lunar Contact light is located in the center of this panel.

#### COMMANDER'S L.H. CENTER PANEL

##### PANEL I

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
9M1	<u>FLIGHT DIRECTOR ATTITUDE INDICATOR</u>  Special Flight Instrument	Displays total vehicle attitudes, attitude rates, and attitude errors, or vehicle attitude, attitude rate, and rendezvous radar antenna shaft and trunnion angles.
9M2	<u>LINE OF SIGHT RATE/ HORIZONTAL VELOCITY INDICATOR</u>  Crosspointers	A flight indicator whose crosspointers indicate either forward and lateral velocities, lateral velocities only, or RR LOS elevation and azimuth angular rates.



# COMMANDER'S L.H. CENTER PANEL (cont)

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
9M9	<u>ALTITUDE/RANGE INDICATOR</u>  Dual Vertical Meter	The left scale displays, in miles or feet, the range or altitude furnished by the radars, PGNS, or AGS. The right scale shows range rate or altitude rate supplied by these units in feet per second.
9S2	<u>RATE/ERROR MONITOR SWITCH</u>  2 pos toggle (MAIN-MAIN)	This two-position toggle (RNDZ RADAR - LDG RDR/COMPTR), 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 and Elevation rates are displayed in the X-Pointer display, (illuminating AZ RT and ELEV RT). In the "LDG RDR/COMPTR" position, vehicle attitude errors (PGNS or AGS as selected by 9-S-3), are displayed on the FDAI error needles. Forward and Lateral Velocities from the landing radar, PGNS or AGS as selected by 9-S-9 are displayed on the X-pointer display (9-M-2)
9S3	<u>ATTITUDE MONITOR SWITCH</u>  2 pos toggle (MAIN-MAIN)	In "PGNS", data from PGNS will be displayed on FDAI. "AGS" selects AGS data.
9S4	<u>ALTITUDE/RANGE MONITOR SWITCH</u>  2 pos toggle (MAIN-MAIN)	Selects either Altitude/Altitude Rate or Range/Range Rate to be displayed on 9M9.
9S6	<u>GUIDANCE CONTROL SWITCH</u>  2 pos toggle (LOCK-LOCK)	Selects use of "PGNS" or "AGS" to control LM.



# COMMANDER'S L.H. CENTER PANEL (cont)

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
9S8	<u>SHAFT/TRUNNION SWITCH</u> 2 pos toggle (MAIN-MAIN)	Selects which scale ( $\pm 5^\circ$ or $\pm 50^\circ$ ) is to be used for angular RR antenna information displayed on the FDAI pitch and yaw indicators.
9S9	<u>MODE SELECT SWITCH</u> 3 pos toggle (MAIN-MAIN-MAIN)	Selects data source for FDAI or X-Pointer. Either "LDG RADAR", "PGNS", or "AGS" may be chosen.
9S17	<u>CROSS POINTER SCALE SWITCH</u> 2 pos toggle (MAIN-MAIN)	Selects the scale multiplier for the Velocity Indicator 9M2. For RR, Hi Mult selects $\pm 20$ mr/sec scale for Az Rt and El Rt. Lo Mult changes scale to $\pm 2$ mr/sec. For LR, Hi Mult selects $\pm 200$ fps scale for Fwd and Lat Vel. Lo Mult selects $\pm 20$ fps scale.

## Systems Engineer's (LM Pilot) RH Center Panel

Like Panel I, the System Engineer's RH Center Panel (Panel II) mounts those controls and displays which are of primary importance to the SE in the accomplishment of his normal duties.

All of the Caution Lights are mounted on Panel II. These yellow lights, when illuminated, indicate a failure or malfunction in one of the less critical systems. As with the Warning Lights, the Caution Light sensors trigger the Audible Tone and the Master Alarm Light. Caution lights may only be extinguished by clearing the malfunction or failure, except in the case of 6DS33.

## SYSTEMS ENGINEER'S R.H. CENTER PANEL

### PANEL II

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
6DS25	<u>PGNS FAILURE</u> Caution Light	The (PGNS) Caution Light is illuminated if any of the following conditions are sensed: (1) Middle



# SYSTEMS ENGINEER'S R.H. CENTER PANEL (cont)

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
6DS25 (cont)		roll gimbal angle is greater than $\pm 60^\circ$ ; (2) deviations in IMU temperature are greater than $\pm 5^\circ$ from normal; (3) the "AND" of RNDZ RADAR in LGC MODE and RADAR CDU FAIL; (4) the "AND" of RNDZ RADAR in AUTO Track Mode and No Track; (5) the "AND" of LDG RADAR DATA NOT GOOD and the landing radar antenna in the "DESCENT" or "HOVER" position; (6) an ambiguous LR antenna position signal; (7) failure of some program check; (8) restart; (9) PIPA failure during periods when not thrusting. The light is extinguished either by selecting the "AGS" position on 9-S-6 or actuating the DSKY RESET pushbutton.
6DS28	<u>RNDZ RDR FAILURE</u>  Caution Light	Enabled only if RR Mode Switch (17S3) is in the AUTO TRACK position. This indicator lights if an RR data-no-good condition occurs. Extinguished when a data-good condition is restored, or when the RR is turned off.
6DS29	<u>LDG RDR FAILURE</u>  Caution Light	Indicator lights if: <ol style="list-style-type: none"> <li>1. LR range data-no-good condition occurs.</li> <li>2. LR velocity data-no-good condition occurs.</li> </ol> <p>Extinguished when the data-good condition is restored, or when the LR is turned off.</p>
6DS33	<u>HEATER FAILURE</u>  Caution Light	Stimuli: Out of Tolerance Temperature condition in: <ol style="list-style-type: none"> <li>1. RCS Thruster Quads</li> <li>2. S Band Antenna Assembly</li> <li>3. RR Antenna Assembly</li> <li>4. LR Antenna Assembly</li> </ol> <p>Positioning 18S10 puts caution light out to indicate which one of 1,2,3, 4 above</p>

FOR TRAINING PURPOSES ONLY



# SYSTEMS ENGINEER'S R.H. CENTER PANEL (cont)

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
9M3	<u>LINE OF SIGHT RATE/ HORIZONTAL VELOCITY INDICATOR</u>	Same as 9M2, Panel I
	Crosspointers	
9M6	<u>FLIGHT DIRECTOR ALTITUDE INDICATOR</u>	Same as 9M1, Panel I
	Special Flight Indicator	
9S7	<u>RATE/ERROR MONITOR SWITCH</u>	Same as 9S2, Panel I. Selects inputs to 9M3 and 9M6 on Panel II
	2 pos toggle (MAIN-MAIN)	
9S10	<u>ATTITUDE MONITOR SWITCH</u>	Same as 9S3, Panel I. Selects inputs to 9M6 on Panel II.
	2 pos toggle (MAIN-MAIN)	

## Commander's and Systems Engineer's Lower Center Panel

The Commander's and Systems Engineer's Lower Center Panel (Panel III) contains controls and displays for which both crew members will find use while performing normal duties. Easily accessible to both, this panel is sectionalized with D/C for radars, stabilization/control, attitude control, heaters, and lighting. A Lunar contact light is mounted on the right side of the panel.

## COMMANDER'S AND SYSTEMS ENGINEER'S LOWER CENTER PANEL

### PANEL III

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
9S18	<u>CROSS POINTER SCALE SWITCH</u>	Selects the scale multiplier for the Velocity Indicator 9M3. For RR, Hi Mult selects $\pm 20$ mr/sec scale for Az Rt and El Rt. Lo Mult changes scale to $\pm 2$ mr/sec. For LR, Hi Mult selects $\pm 200$ fps scale for Fwd and Lat Vel. Lo Mult selects $\pm 20$ fps scale.
	2 pos toggle (MAIN-MAIN)	



COMMANDER'S AND SYSTEMS ENGINEER'S LOWER CENTER PANEL (cont)

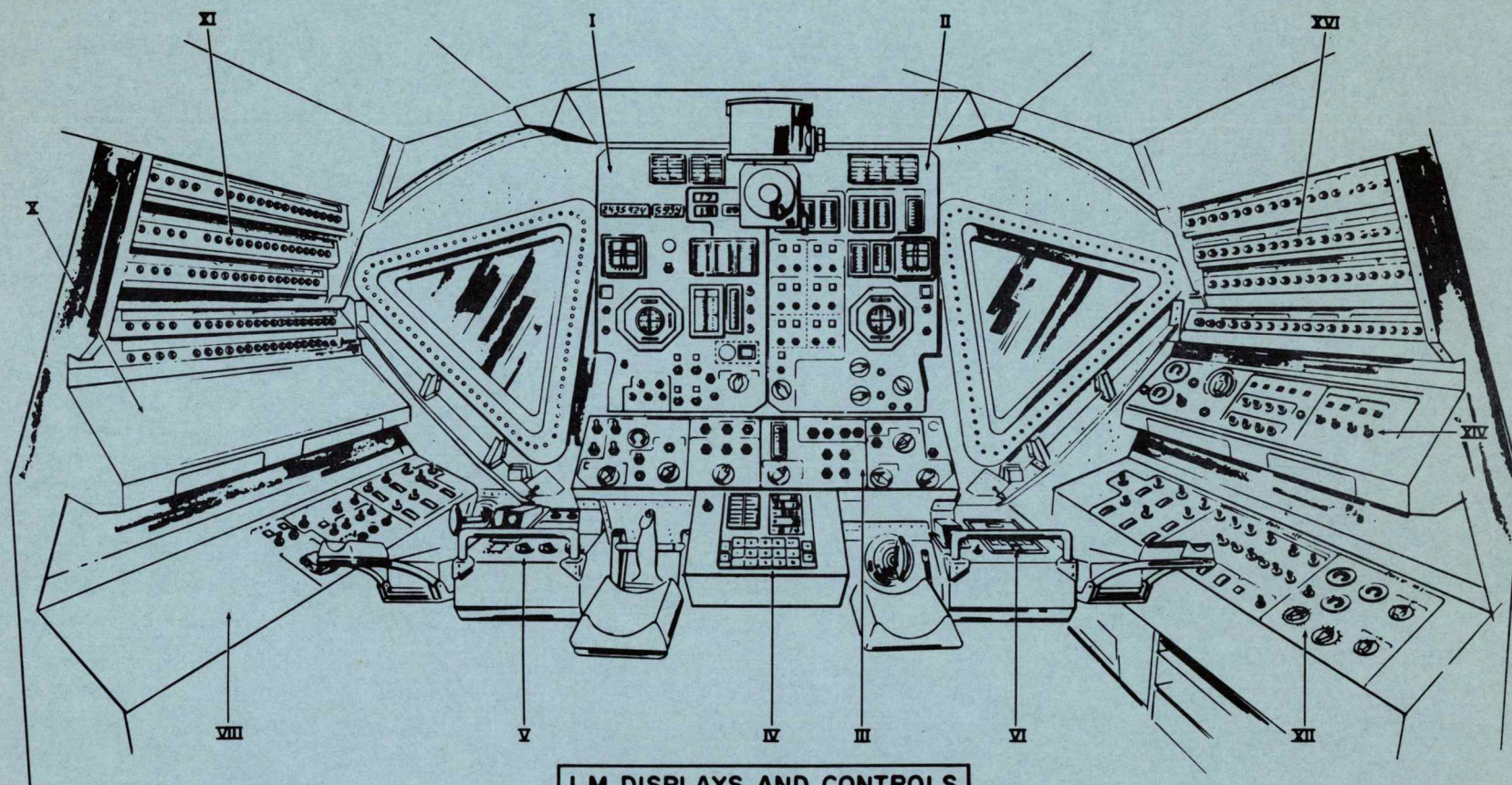
<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
17DS1	<u>NO TRACK LIGHT</u>  Component Caution Light	Indicates failure of the RR to supply a data good signal during any mode.
17M1	<u>SIGNAL STRENGTH METER</u>  Ammeter	Displays the relative strength of LR or RR signals as selected by 17S8.
17S1	<u>MANUAL SLEW SWITCH</u>  4 way toggle, center return loaded	Used to control RR antenna movement in any of 4 directions, "UP", "DOWN", "LEFT", and "RIGHT".
17S2	<u>SLEW RATE SWITCH</u>  2 pos toggle (MAIN-MAIN)	Selects either a "HI" (9°/sec) or "LO" (1-1/3°/sec) slew rate for the RR antenna.
17S3	<u>RENDEZVOUS RADAR MODE SWITCH</u>  3 pos rotary	Selects the method by which the RR antenna will be controlled. "AUTO TRACK", incoming signals will aim the antenna (after lock-on). "SLEW", control is manual, using the slew switch, 17S1. In "LGC", the computer will steer the antenna until the RR locks on to the CSM transponder.
17S5	<u>LANDING ANTENNA SWITCH</u>  3 pos toggle (LOCK-LOCK-LOCK)	Selects Landing antenna position. In "AUTO", LGC will select antenna position two automatically. The "DES" and "HOVER" positions allow crew members to select antenna position manually.



COMMANDER'S AND SYSTEM ENGINEER'S LOWER CENTER PANEL (cont)

<u>Control/ Indicator</u>	<u>Nomenclature and Type</u>	<u>Function</u>
17S8	<u>TEST/MONITOR SWITCH</u>  6 pos rotary	Selects one of the following for display on 17M1.  <ol style="list-style-type: none"> <li>1. ALTXMTR - Altitude transmitter power output. (LR)</li> <li>2. VELXMTR - Velocity transmitter power output. (LR)</li> <li>3. AGC - Displays RR AGC loop signal strength.</li> <li>4. XMTR PWR - Shows transmitter output power. (RR)</li> <li>5. SHAFT ERR - Shows relative pointing error from direct LOS to CSM for RR antenna shaft angle.</li> <li>6. TRUN ERR - Shows relative pointing error from direct LOS to CSM for RR antenna trunnion angle.</li> </ol>
18M1	<u>TEMPERATURE INDICATOR</u>  Single Scale Vertical Meter	Displays in degrees Fahrenheit, the temperature of the radars, RCS quads and S Band Antenna Assembly selected by 18S10. (-100° to +200°F)
18S10	<u>TEMPERATURE MONITOR SELECT SWITCH</u>  7 pos rotary	Selects the temperature to be displayed on 18M1. Positions are: "RNDZ", "LDG", "QUAD 1", "QUAD 2", "QUAD 3", "QUAD 4", and "S BAND".

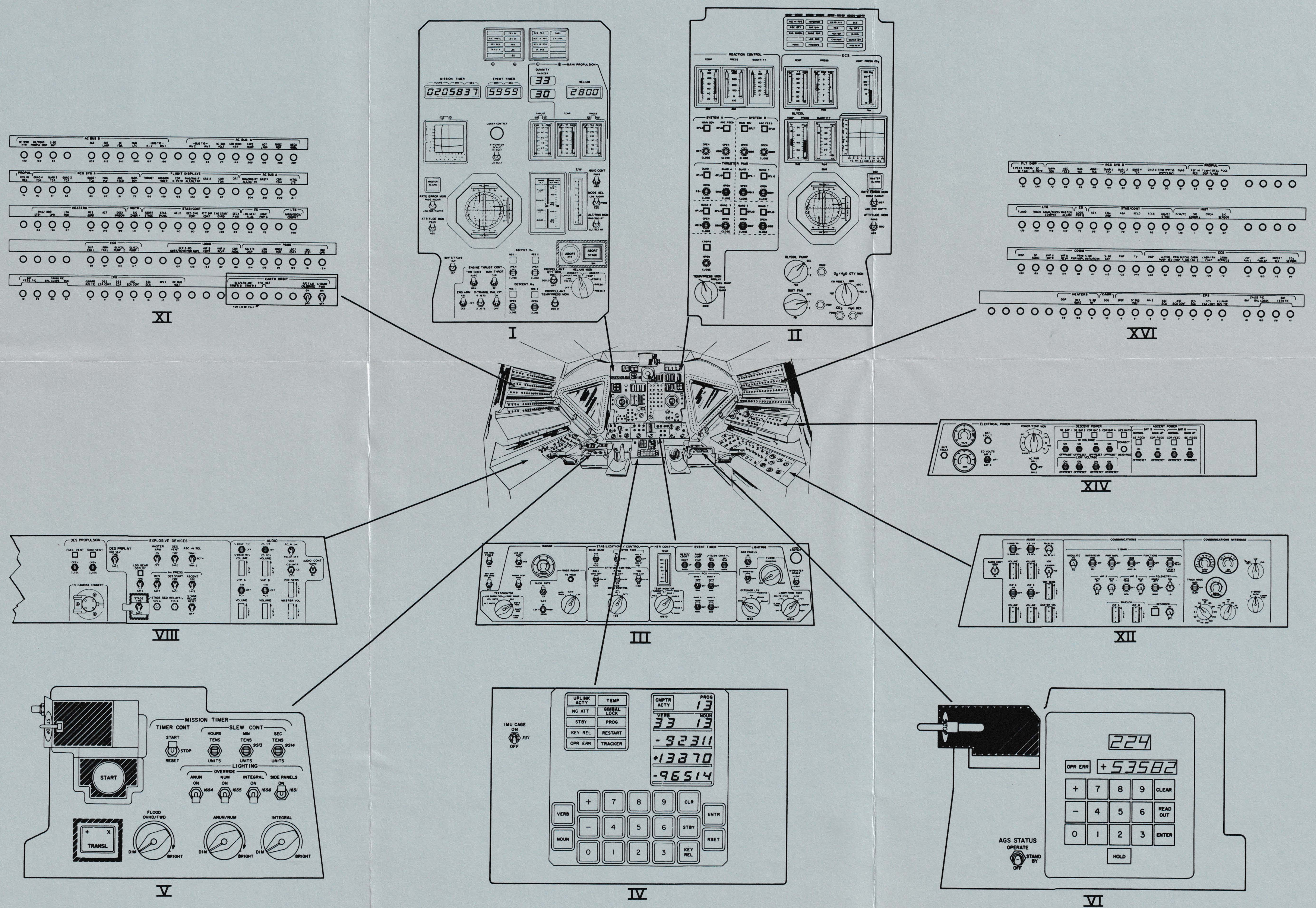




LM DISPLAYS AND CONTROLS  
LM-3 CONFIGURATION

T30005-28  
DEC 66

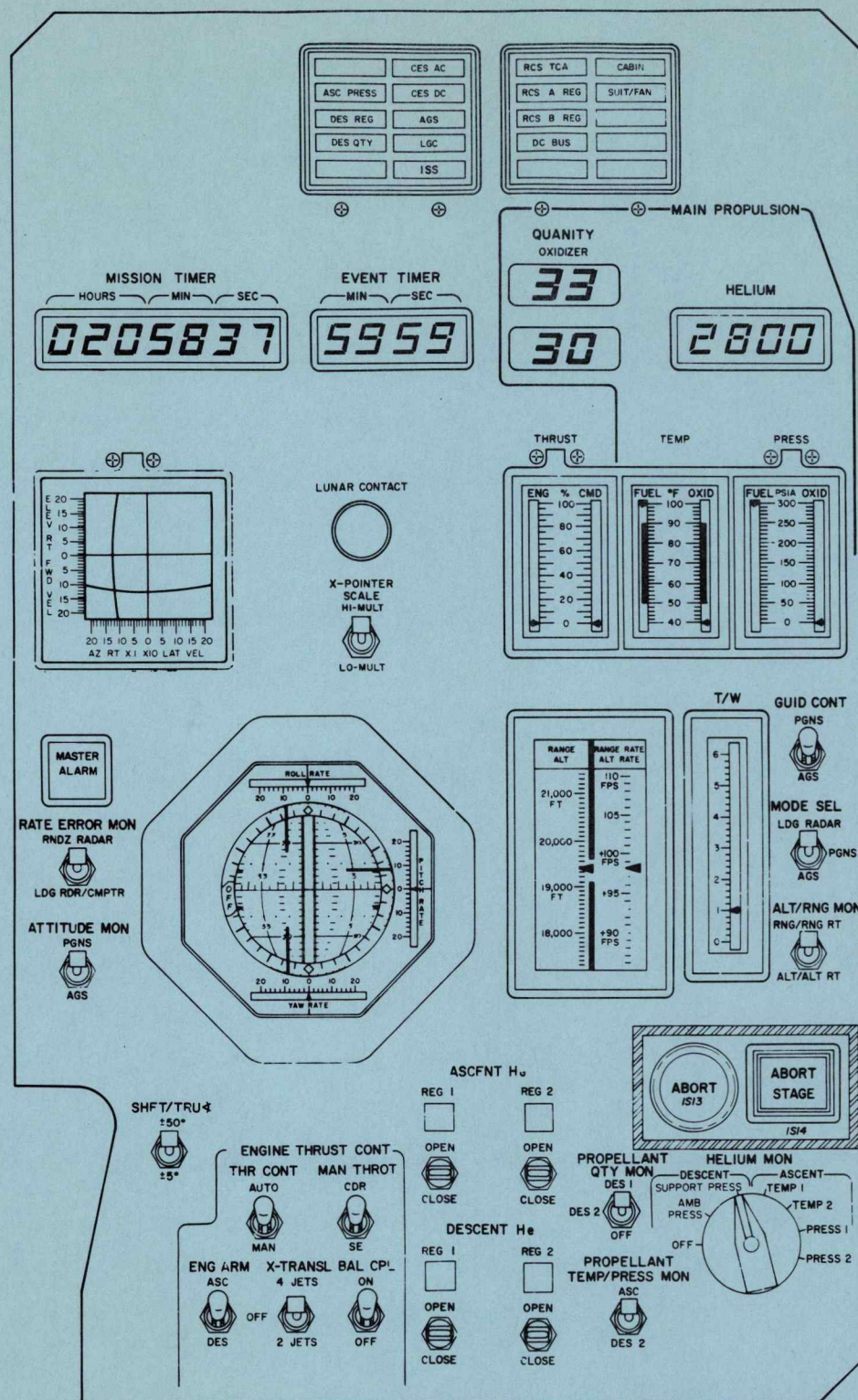




# LM DISPLAYS AND CONTROLS

T30005-28  
DEC 66

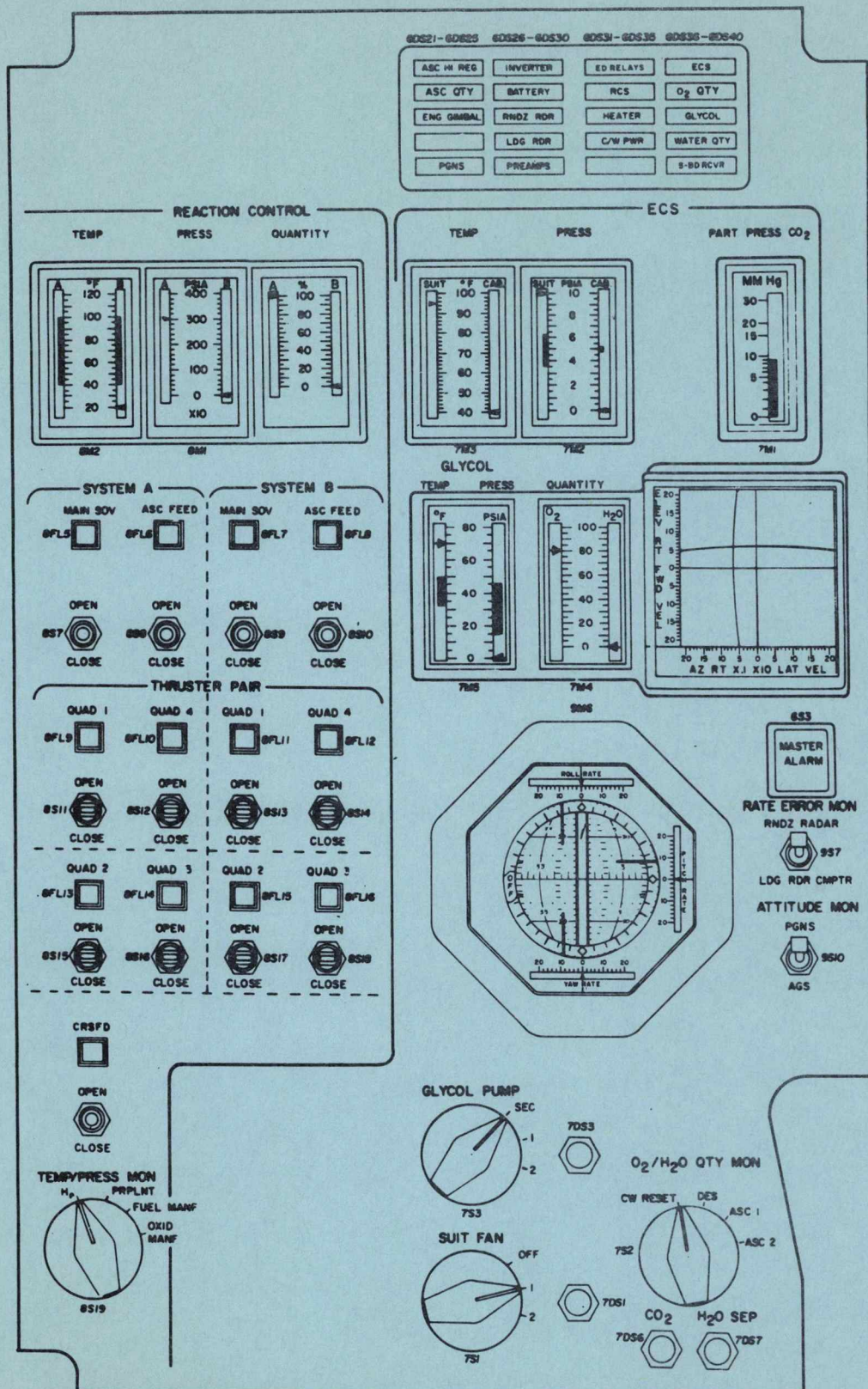




PANEL I

T30900-1  
DEC 66

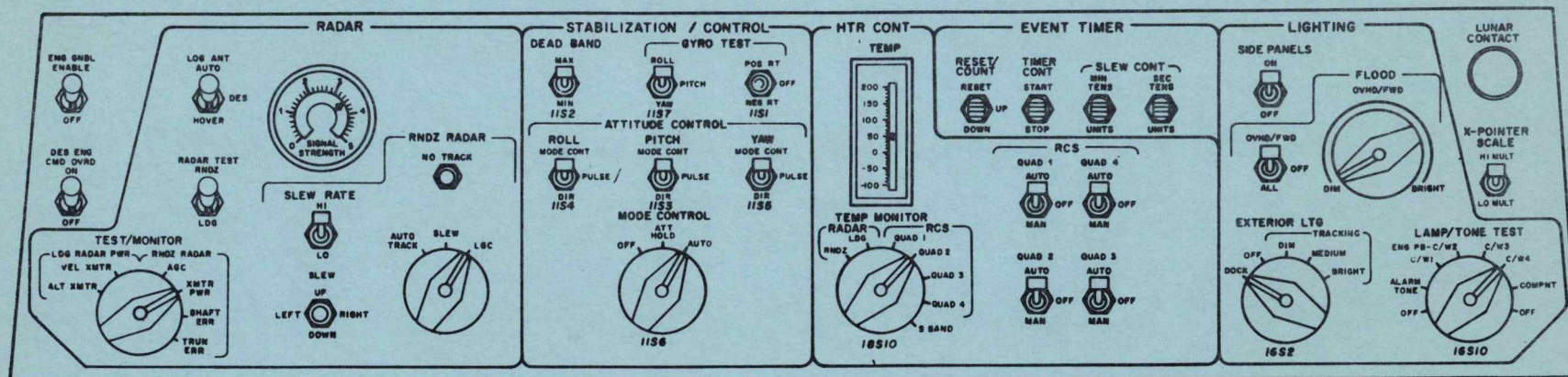




PANEL II

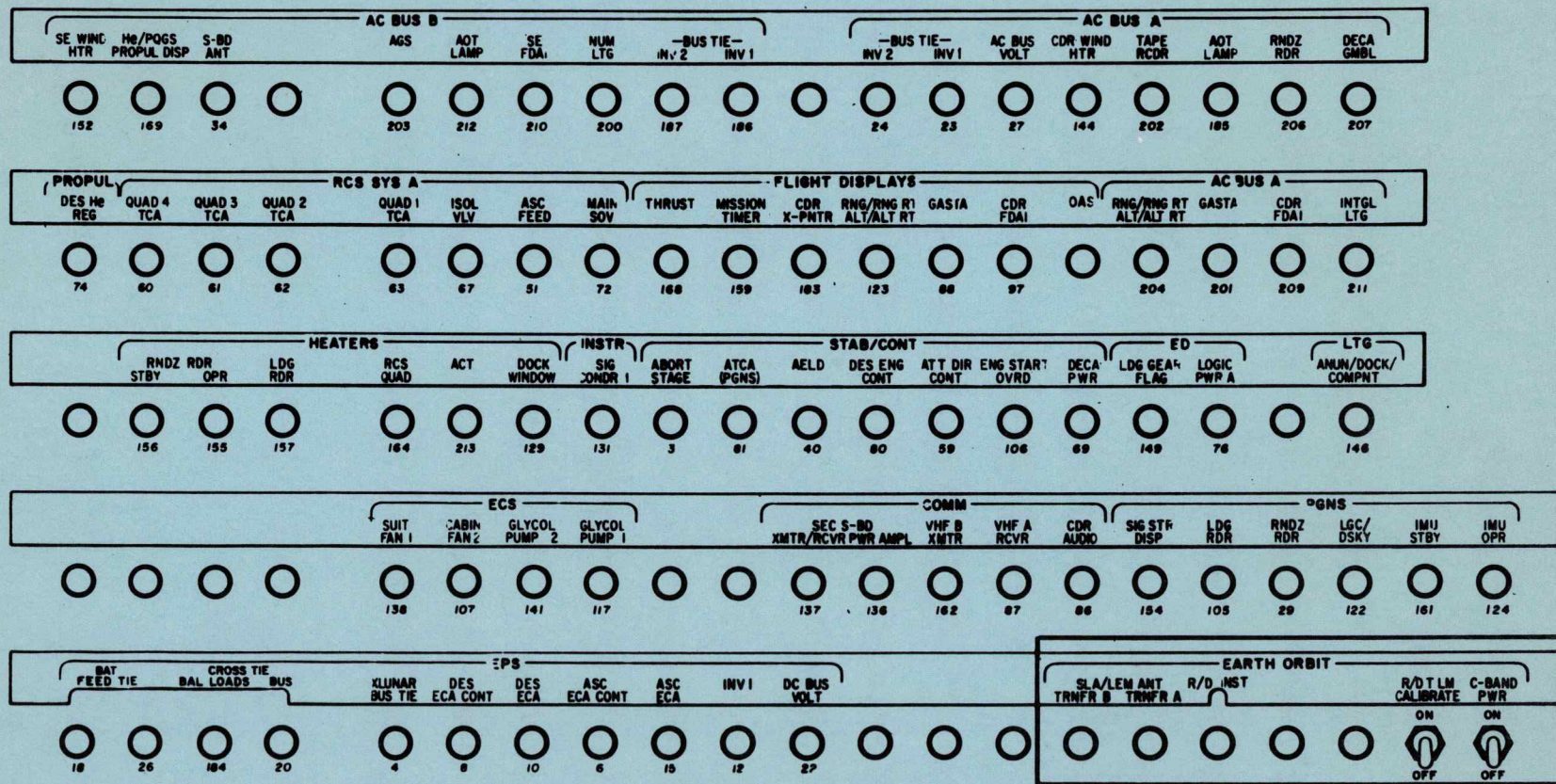
T30900-2  
DEC 66





PANEL III

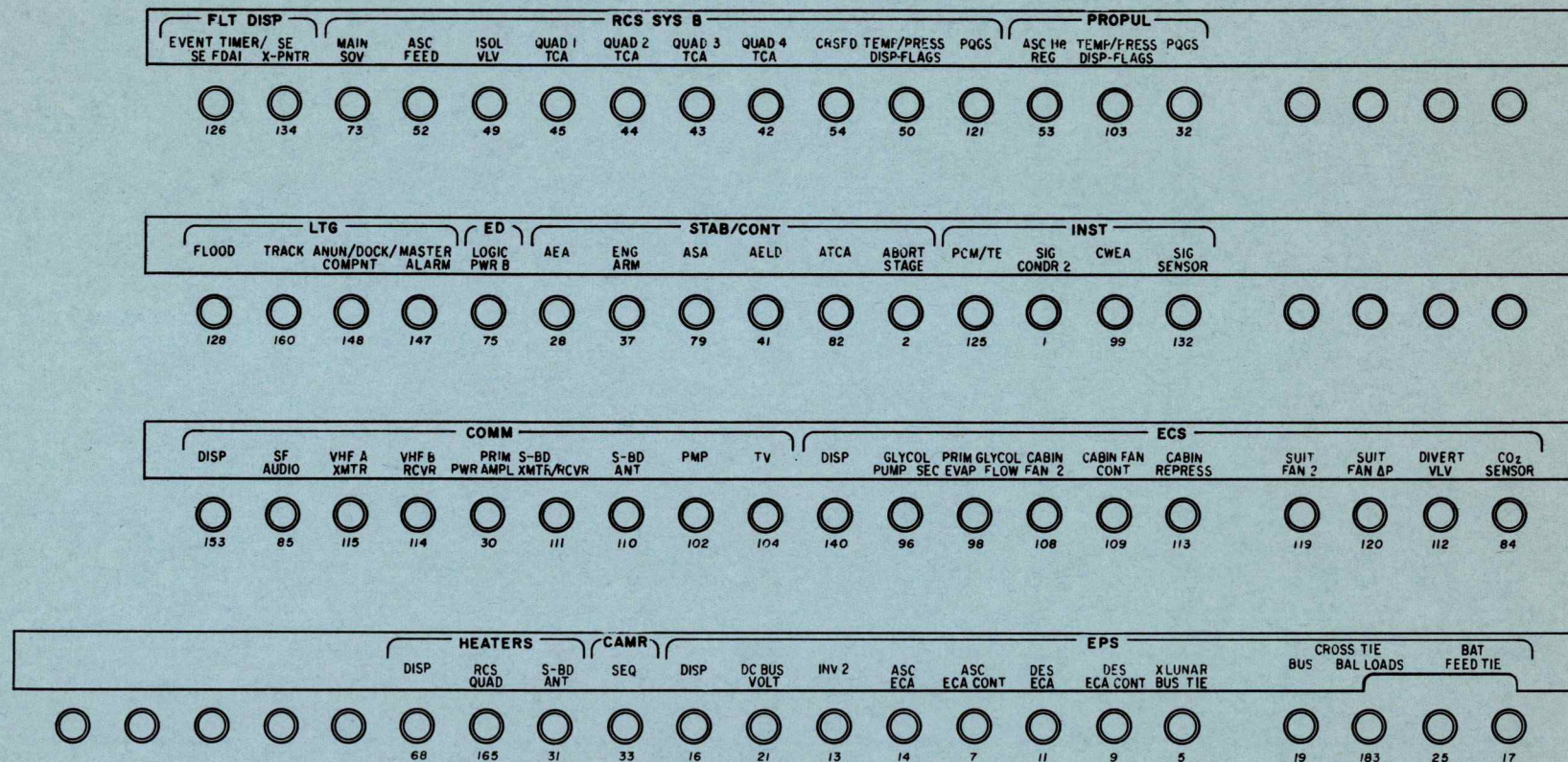




PANEL XI

T 30900-20  
NOV 66

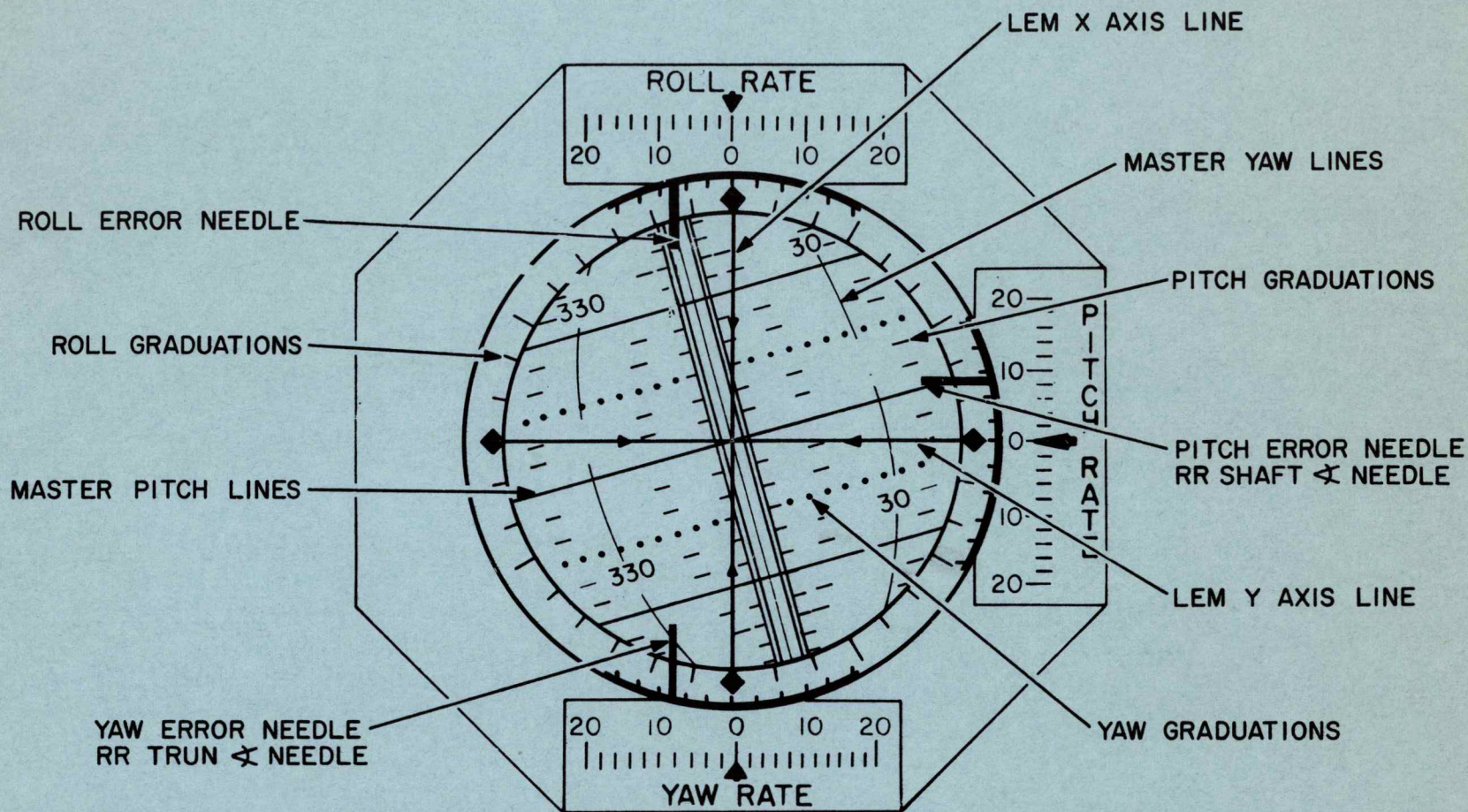




PANEL XVI

T30900-21  
NOV 66

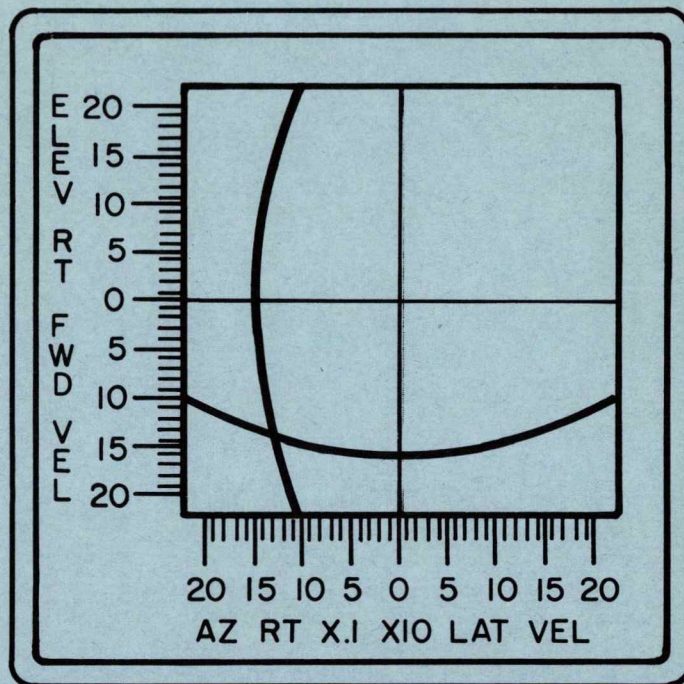




FLIGHT DIRECTOR ATTITUDE INDICATOR  
9MI & 9M6

T30405-85  
FEB 67

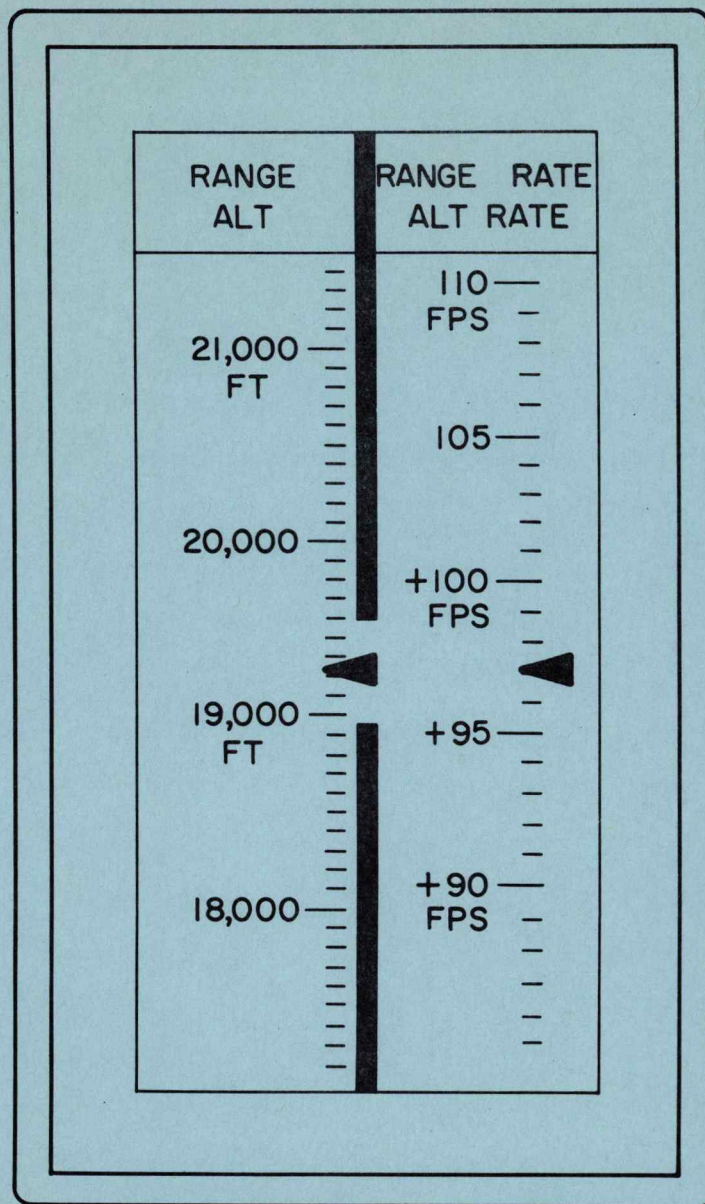




**X-POINTER INDICATOR  
9M2 AND 9M3**

T30405-96  
FEB 67

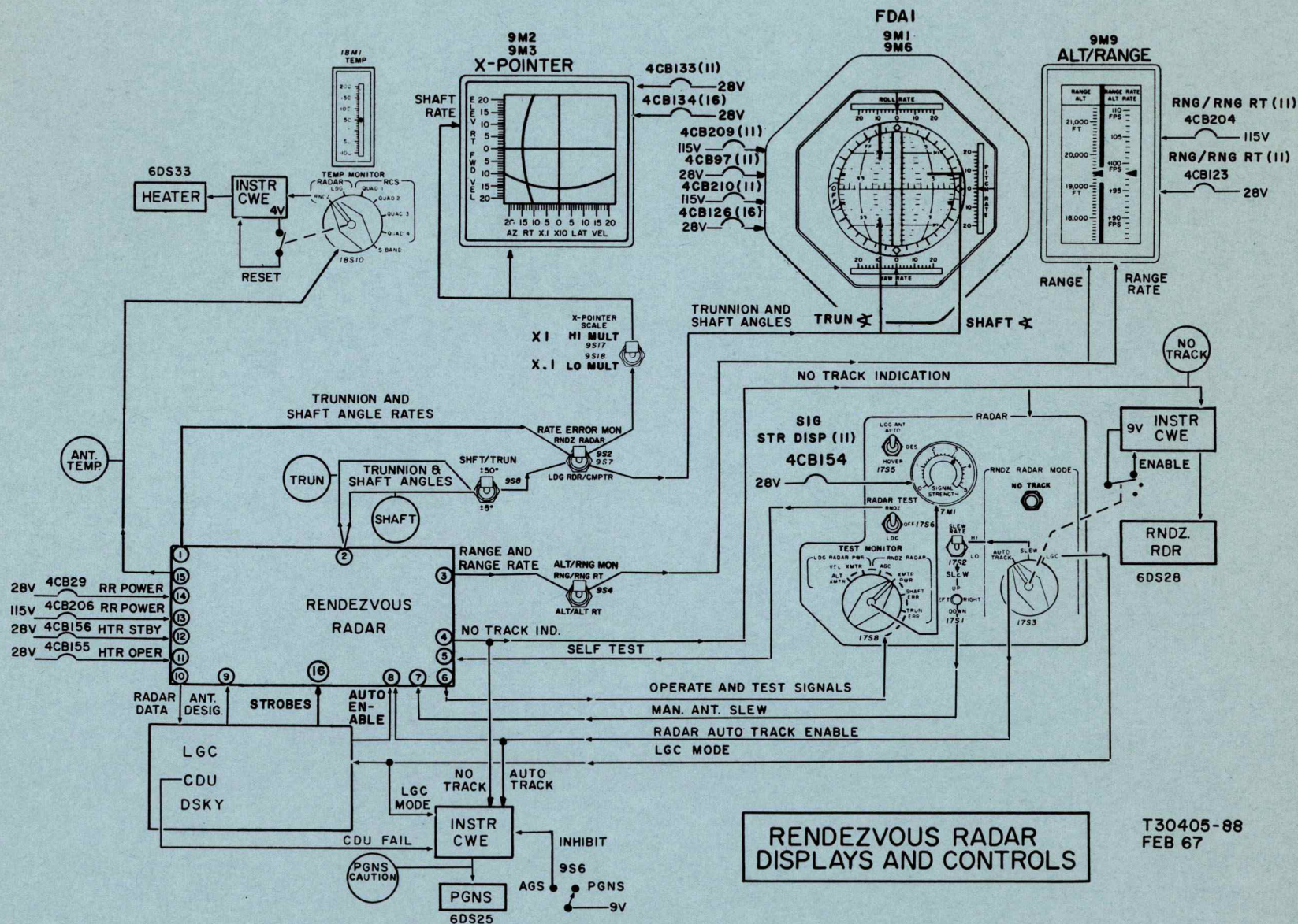




ALTITUDE / RANGE INDICATOR 9M9

T30405-112  
MAY 66



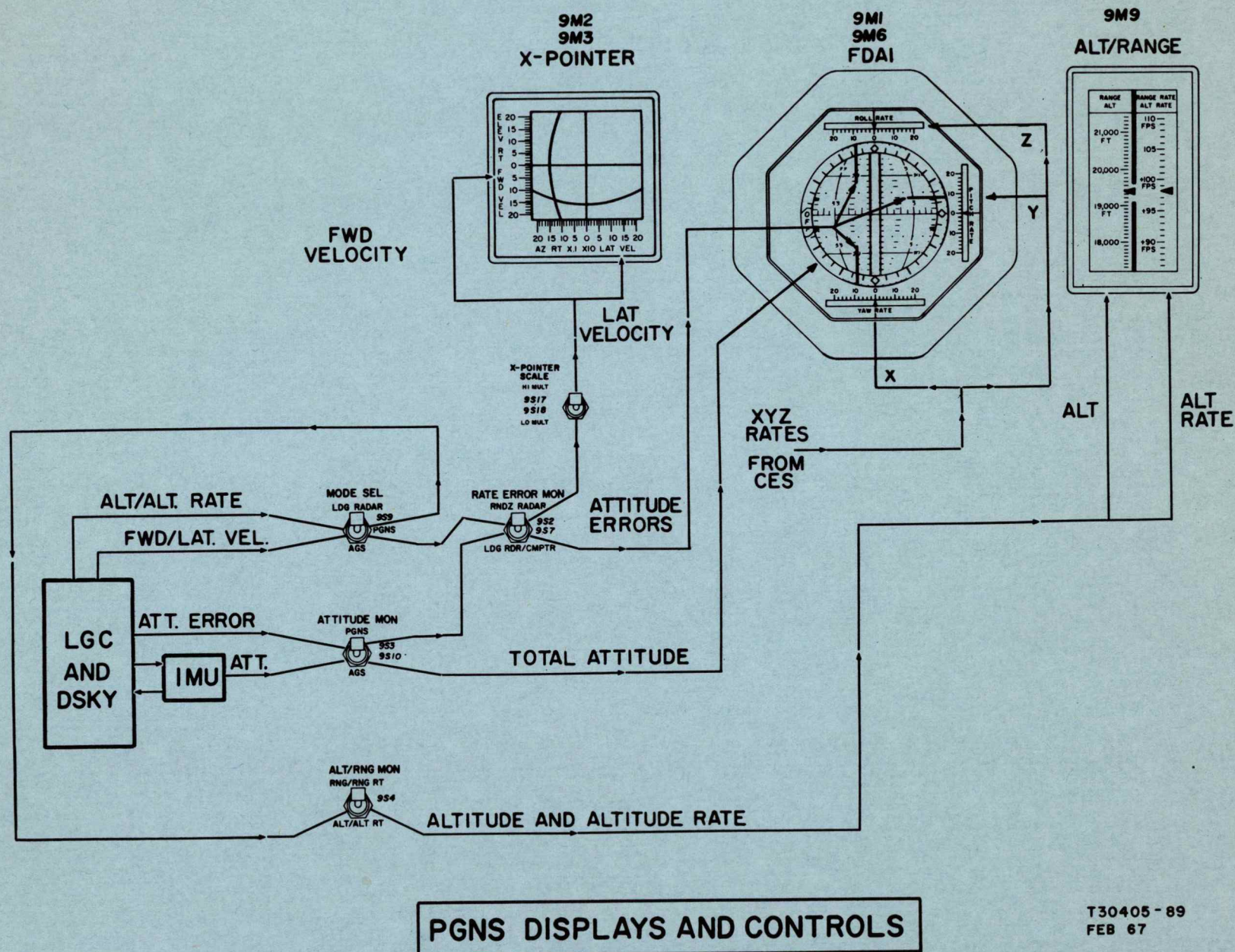


T30405-88  
FEB 67









T30405-89  
FEB 67



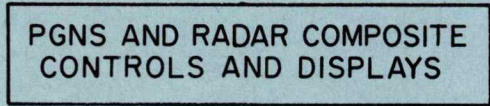


Figure 5-14