

# R E N T R Y

AN ORBITAL SIMULATOR

LUNAR MODULE

FLIGHT MANUAL

# REENTRY

AN ORBITAL SIMULATOR  
LUNAR MODULE FLIGHT MANUAL

DRAFT

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## I. INTRODUCTION



# I. INTRODUCTION

## 1. ABOUT

**Project Apollo for Reentry** is one of the modules available for the space simulator “Reentry – An Orbital Simulator” by Wilhelmsen Studios. It comes with a study level implementation of the spacecrafts used in the Apollo program (with some simplifications). The module includes the Apollo Command Module spacecraft and the Lunar Module spacecraft. This document will cover the Lunar Module.

The goal of Project Apollo for Reentry is to create a gamified and immersive experience on how it was to be an Apollo astronaut, what procedures the real astronauts had to follow, and learn about the systems onboard.

The spacecraft is modelled after the Apollo Operations Handbook, Lunar Module, LM 10/11 and Subsequent Manuals (LMA790-3-LM10/11-and-Subsequent), Volume 1 and Volume 2. The differences between the various Lunar Module versions created in real life are minor. The term “SE” was changed to “LMP” on the panels, as well as having an extra Lunar battery available for the Lunar stay, etc. The reason for basing my implementation on the above reference is that this configuration of the spacecraft has all the systems developed for the Lunar Module, and can fly all the real and fictional scenarios I wish to implement. Many of the figures used in this manual is taken from the Lunar Module News Reference guide (<https://www.hq.nasa.gov/alsj/LMNewsRef-Boothman.html>) and the LM handbooks indicated below.

All training needed to fly the capsule is available in this manual and in-game. If you want to study the spacecraft down to the lowest details, I highly recommended to read the manual by NASA. You can find the manual here: <https://www.hq.nasa.gov/alsj/alsj-LMdocs.html>

Volume 1: <https://www.hq.nasa.gov/alsj/LM10HandbookVol1.pdf>

Volume 2: <https://www.hq.nasa.gov/alsj/LM11HandbookVol2.pdf>

Lunar Module News Reference: <https://www.hq.nasa.gov/alsj/LMNewsRef-Boothman.html>

### NOTE

Not all of the components described in this document is simulated. Some might have been simplified or is a placeholder for a future update, while some will never be implemented. They are described because they are needed to complete the descriptions of systems and its operation, and for historical accuracy. This is a computer game meant for the general user, so some simplifications have been made to make it better suited for a computer game.

### GET THE GAME

The game can be purchased from <https://reentrygame.com/buy> - the Project Apollo for Reentry module is included in this package.

### JOIN THE COMMUNITY

An important aspect of virtual space flight is the community – learning to operate these crafts yourself can be very complex. I recommend you join the official “Reentry – An Orbital Simulator” server on Discord, accessible from the in-game menus or <http://discord.gg/reentrygame>! Ask for help, find multiplayer sessions, get roles for your game progress, share clips, screenshots and meet fellow virtual astronauts and mission controllers.

### WHAT IS THIS MANUAL?

This manual contains most of the information you need to successfully master the Command Module Spacecraft in Reentry. This manual is specific to the Command Module spacecraft. For generic Reentry information, please see the **Reentry – An Orbital Simulator: User Manual**.

### DONATE TO SUPPORT THE DEVELOPMENT OF THE GAME

If you wish to support the development of this game, or if you enjoy playing it, please consider giving a small donation. Creating a game like this is a lot of fun, but also takes up a lot of my spare time and my limited resources to fund it.

Any donations will help me cover costs for development, assets, server hosting, and coffee for staying up late.

You can donate from the Main Menu of the game, or online using PayPal on the following page:  
<http://reentrygame.com/donate>

**From one space enthusiast to another, thank you again for considering giving a donation!**

### LEGAL

Images and information in the manual, as well as in the **Project Apollo for Reentry** module is based on information made public by NASA and related documents. Images and references from various NASA documents are used.

The images in this guide and game are using public domain images from NASA.

<https://www.jsc.nasa.gov/policies.html#Guidelines>

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The information described here is tailored to the simulation and my implementation of the spacecraft for Reentry – An Orbital Simulator. Some systems are simplified or made differently due to being used in a computer software, and for gamifying the experience.

Both public documents released by NASA and Wikipedia have been used as a reference in my implementation of Project Apollo, as well as writing the education material for the game, including this manual, in-game academy, and mission flow.

This module is subject to change and/or removal at any time.

## 2. A BRIEF HISTORY



The Lunar Module was part of the Apollo Program. It was operational between 1969 and 1972, and was a dedicated spacecraft used to land two astronauts on the Lunar Surface. It was the first manned spacecraft that exclusively operated in the airless vacuum of space. It was built by Grumman Aircraft, and 10 Lunar Modules were launched to space, where 6 landed on the Moon.

The Lunar Module had two stages. One descent stage, and one ascent stage. The ascent stage contained the pressurized cabin, where the astronauts were living and operating the craft. The Lunar Module was inserted into the SIV-B stage during launch and ascent, and extracted by the Apollo Command/Service Module spacecraft shortly after the TLI maneuver. It was then attached to the Apollo Command and Service module all the way to the Lunar orbit.

The Lunar Module was not designed to reenter or operate on Earth, and was discarded after the Lunar ascent.

## 3. INTERACTING WITH THE COCKPIT

The Lunar Module is working in a similar fashion to the other spacecrafts in Reentry. You will not be able to enter the cockpit before the Lunar Module is docked with the Apollo spacecraft, and that the entry hatches are opened. Once the Lunar Module is accessible, you can use F4 to switch between the two active spacecrafts, even after separation. This means you can both fly the Lunar Module and the Apollo CSM in the same session.

To change cockpit cameras when the Lunar Module is the active spacecraft, you use F5 to F12, or the View menu by pressing [V] on the keyboard.

There are multiple controls you can interact with in the cockpit, as well as joystick(s) to orient and translate the craft. This section will briefly explain how you can use the different types of controls available.

### CONTROLS



#### Switch

Multiple switches are used to configure various onboard systems. A label is usually describing the function of the switch and what positions it can be set to. A switch can either go in a vertical direction, or a horizontal direction.

Vertical switches are pushed upwards using a single left mouse click, and downwards using a single right click.

Horizontal switches are pushed left using the left mouse button, and right using the right mouse button.

A switch can have two or three positions. When a three-position switch occurs, the same logic applies.



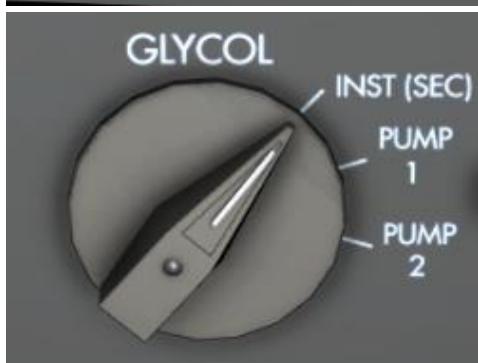
### **Push Buttons**

The push button can be clicked using the left mouse button to push it down. This will trigger the related action to the push button



### **Circuit Breaker**

Circuit breakers can be closed using left mouse button, and opened using the right mouse button. A closed circuit breaker (cb) means that the electrical loop is closed and functional. An open cb means the electrical loop is open and disabled. An open cb is identified by being further out than closed ones, as well as a white ring on the inner side.



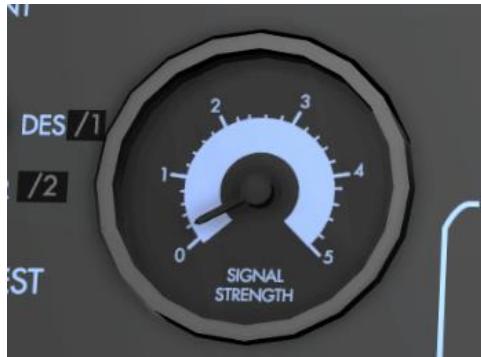
### **Selector**

A selector can be rotated to configure a system or select the source sensor an indicator will use.

If can rotate both left and right:

- Right click moves the selection rightwards (counterclockwise).
- Left click moves it leftwards (clockwise).

A selector is usually identified by a label with marks showing what the selector is configured to.



### Gauges

Gauges are used to show the status or signal from sensors located throughout the capsule. Circular gauges and vertical gauges exist in the Gemini craft.

A gauge can consist of one or multiple needles showing the current signal, typically the amount of fuel left, oxygen levels, and pressure and temperature of various onboard systems.

Some gauges can be controlled by a selector, where the selector chooses the input/source of the gauge.



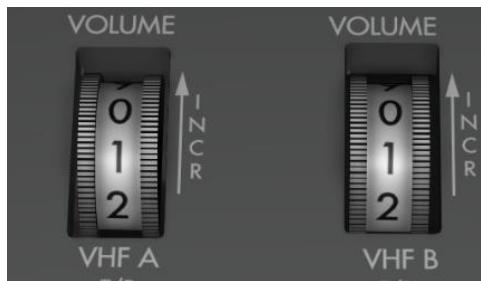
Vertical gauges are used to read the state of a system or a quantity.

Some of them have a light above or below, illuminating if it's not receiving power.



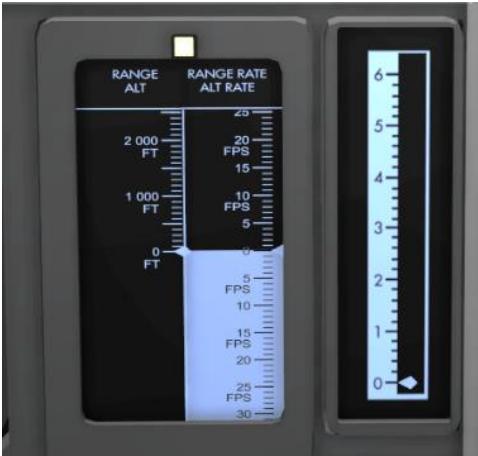
### Talkbacks

Talkbacks can show gray or barberpole (bp). They indicate if a valve is open or closed, or if a system or flag is enabled or disabled. Gray usually means it is enabled, while barberpoled means it is disabled.



### Thumb Wheel

A thumb wheel can be increased using right mouse button, and decreased using left mouse button. They can have a number used as an indicator, or simply be a unitless wheel.



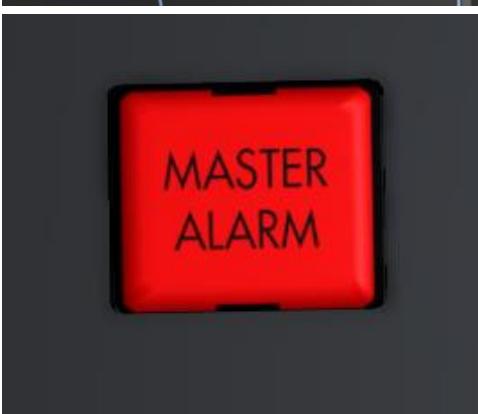
### Tape Instruments

Tape instruments use a tape to move the background of the instrument. Using a fixed pointer, the value is displayed by moving the tape to the correct location, instead of the pointer.



### Digital Number

Numerical digits are used to show a value, either on the computer or mission time.



### Master Alarm

The Master Alarm switch shows when a critical warning or issue is detected. An alarm will sound. The alarm can be disabled by clicking on it.



### Caution & Warning Lights

Caution and warning lights are used to highlight systems that require attention, either due to issues, out of tolerance values or warnings.

Extinguished lights means that everything is OK, while illuminated lights means that the system needs attention.



### Indicators / Component Lights

Light indicators are used to show the status of a system, similar to the caution and warning light array.



### Computer Buttons

The computer buttons function in the same was as a push button and is used to operate the onboard computer.



### Computer Display

The computer display shows the current running program, verb, noun and data in three register displays (each with a green line above it). A dedicated caution light array for the computer is used to communicate its state.

## 4. KEYBOARD CONTROLS

### MANEUVERING

Maneuvering is done using the keyboard or joysticks. The input is configured through the Reentry – An Orbital Simulator settings dialogue.

### ORIENTATION

Maneuvering is done using the keyboard or joysticks. The input is configured through the Reentry – An orbital simulator settings dialogue.

W: Pitch down

S: Pitch up

A: Yaw left

D: Yaw right

Q: Roll left

E: Roll right

## TRANSLATION

U: Forward

O: Backwards

I: Upwards

K: Downwards

J: Leftwards

L: Rightwards

## TOOLS

F4: Switch to the Lunar Module/Command Module

T: Flashlight (move it around by using the mouse)

C: Show/Hide Radio Communication menu (both UI and circular cockpit buttons)

M: Show/Hide Mission Pad

V: Show/Hide View Selector

ESC: Show/Hide in-game menu

## FUNCTION-BUBBLE BUTTONS

### Cockpit

Circular buttons (bubbles) are used to trigger or toggle various functions used during cockpit operation. These are usually short radio commands, or toggling various equipment.

Radio Check is used to verify the radio.

Request weight data is used to request the current weight of the Command Module and the Lunar Module (if available) in lbs. This data is used with the computer.

Hold/Resume countdown is used to hold or resume the countdown before a launch.

Toggle Hatch is used to open the forward or the overhead hatch. One button dedicated to a specific hatch exists in front of them. Please note that it is not possible to open the hatch unless

the pressure is equal on both sides of the hatch. If you wish to exit the Lunar Module to walk on the Moon, you will need to configure the Environment Control System into Egress mode (covered later).

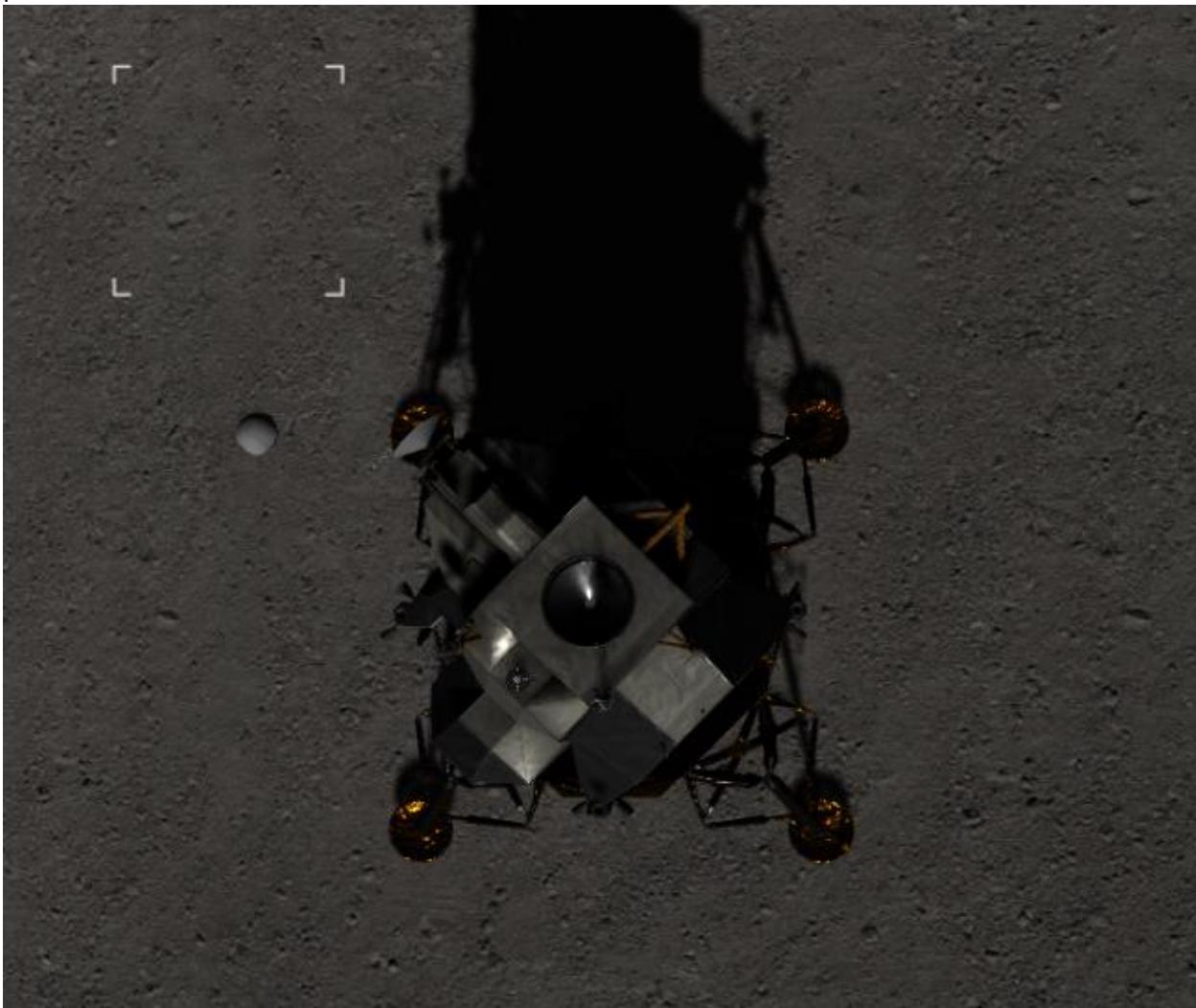
Exit Lunar Module on the forward hatch will make the astronaut leave the Lunar Module to perform a Lunar EVA.

### **Lunar EVA**

Some dedicated function-bubble buttons exist during Lunar EVA.

Returning inside will make the astronaut climb back into the Lunar Module.

Assemble Lunar Rover Vehicle (LRV) will make the astronaut assemble the Lunar Rover Vehicle. The camera will switch to a top-down view and a cursor will let you select where you wish to place the LRV.



Enter LRV will make the astronaut climb into the Lunar Rover.

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When in the LRV, pressing C will open a contextual menu. This will let the astronaut exit the vehicle:

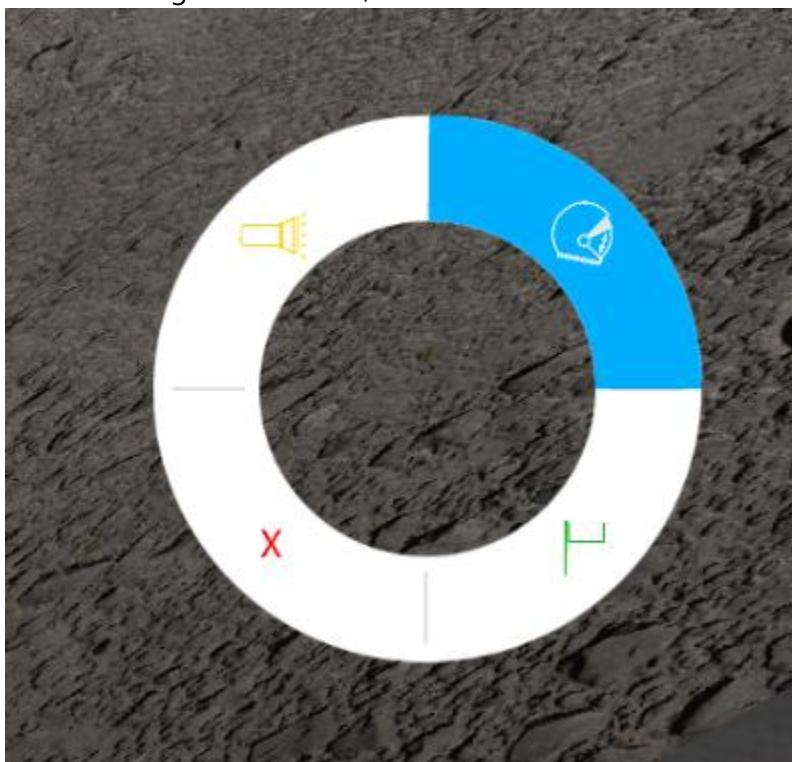


The other toggles are to open/close the visor, close the menu or toggle the flashlight.

### **Walking on the Moon**

When walking on the moon, you use the arrow keys to walk and the mouse to look around/direction.

When walking on the moon, another contextual menu exists:



This will let you toggle a flashlight, toggle the visor, place a flag, and close the menu.

Placing the flag:

Pressing the flag button will place the flag in the direction you face in your current position.  
Every time you press the flag button, the flag will be moved to the new position.

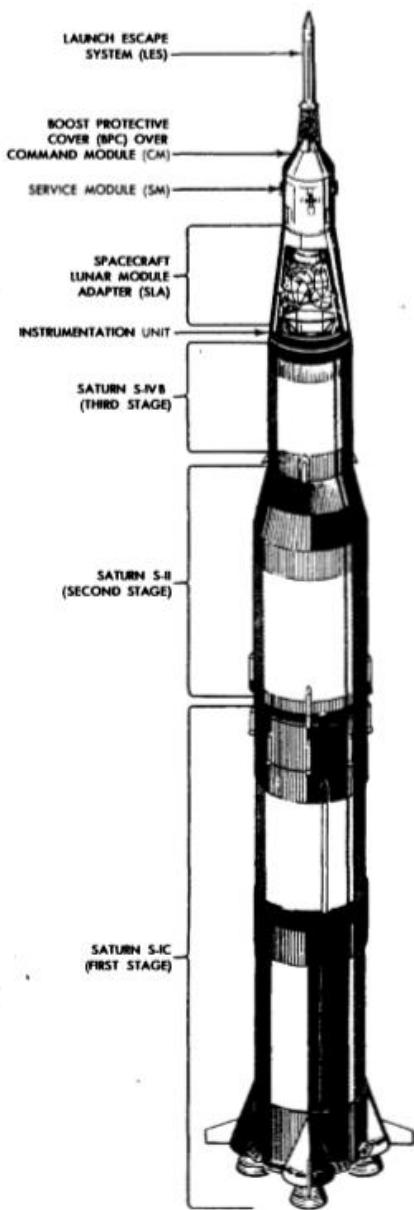
## II. MAJOR COMPONENTS



## II. MAJOR COMPONENTS

### 1. GENERAL

The Lunar Module (LM) is designed for manned lunar landing missions. It consists of an ascent stage and a descent stage; the stages are joined together at four interstage fittings by explosive nuts and bolts. Subsystem continuity between the stages is accomplished by separable interstage umbilicals and hardline connections.



Both stages function as a single unity during lunar orbit, until separation is required. The descent stage is used primarily for descending to the Lunar surface, where its powerful descent engine is used to reduce the LM's velocity.

The descent stage is also used as a launch pad for the ascent stage, when the lunar stay is complete and the astronauts are ready to get back to the Command Module. The ascent stage is then functioning as a single unit to accomplish rendezvous and docking with the CSM.

The LM is located inside the S-IVB stage during launch and ascent. The landing gear is in a retracted position to make it more compact during storage.

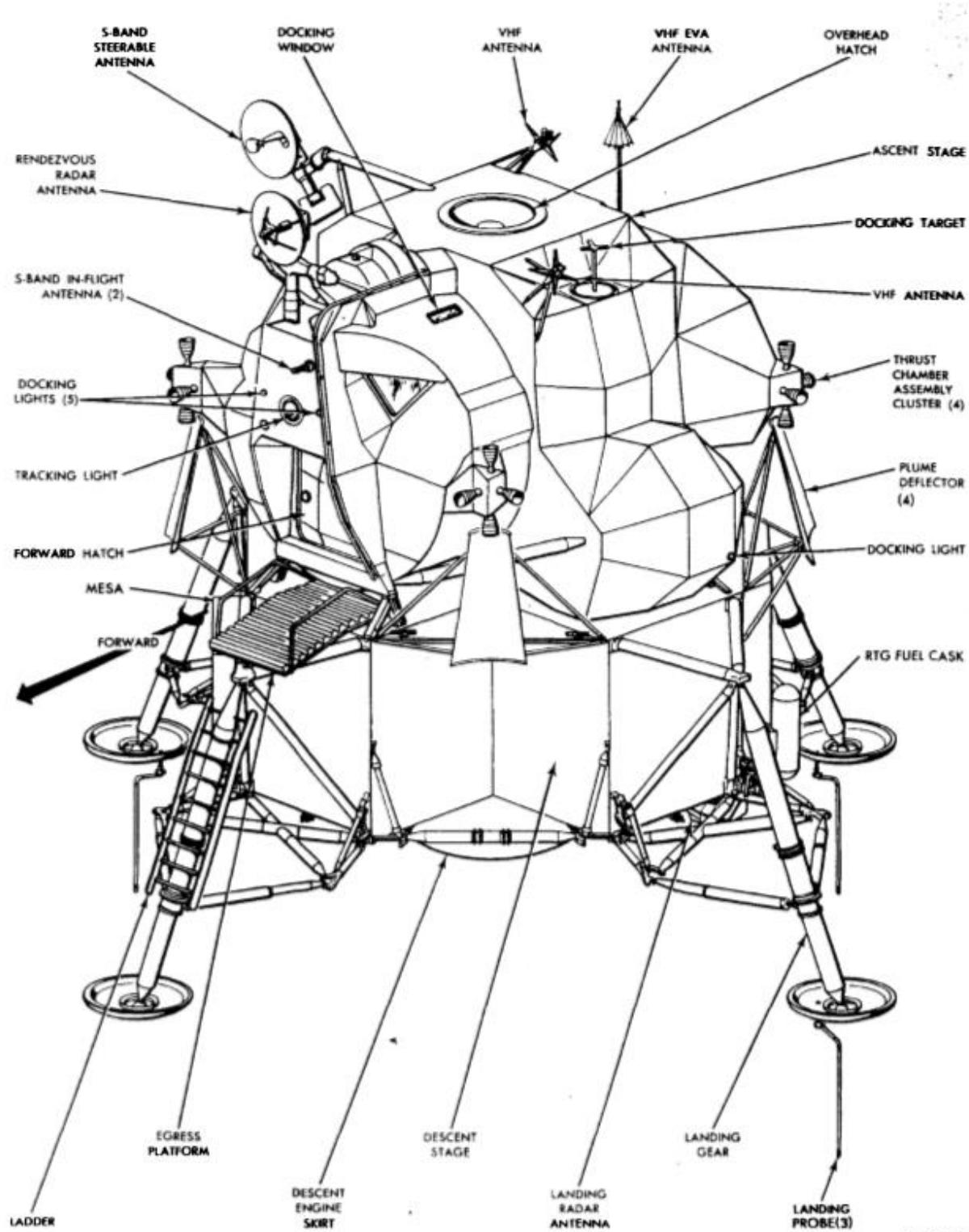
The S-IVB has four panels that protect the LM; the Spacecraft Lunar Module Adapter (SLA). During S-IVB separation, the SLAs are separated, exposing the Lunar Module docking port for extraction.

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A-300LM10-2

## 2. ASCENT STAGE

The ascent stage is the control center and the manned portion of the LM. It accommodates two astronauts, and compromises of three main sections: the crew compartment, midsection, and aft equipment bay. The crew compartment and midsection make up the cabin, and is pressurized to 4.8 psig. The cabin is the only pressurized area of the LM.

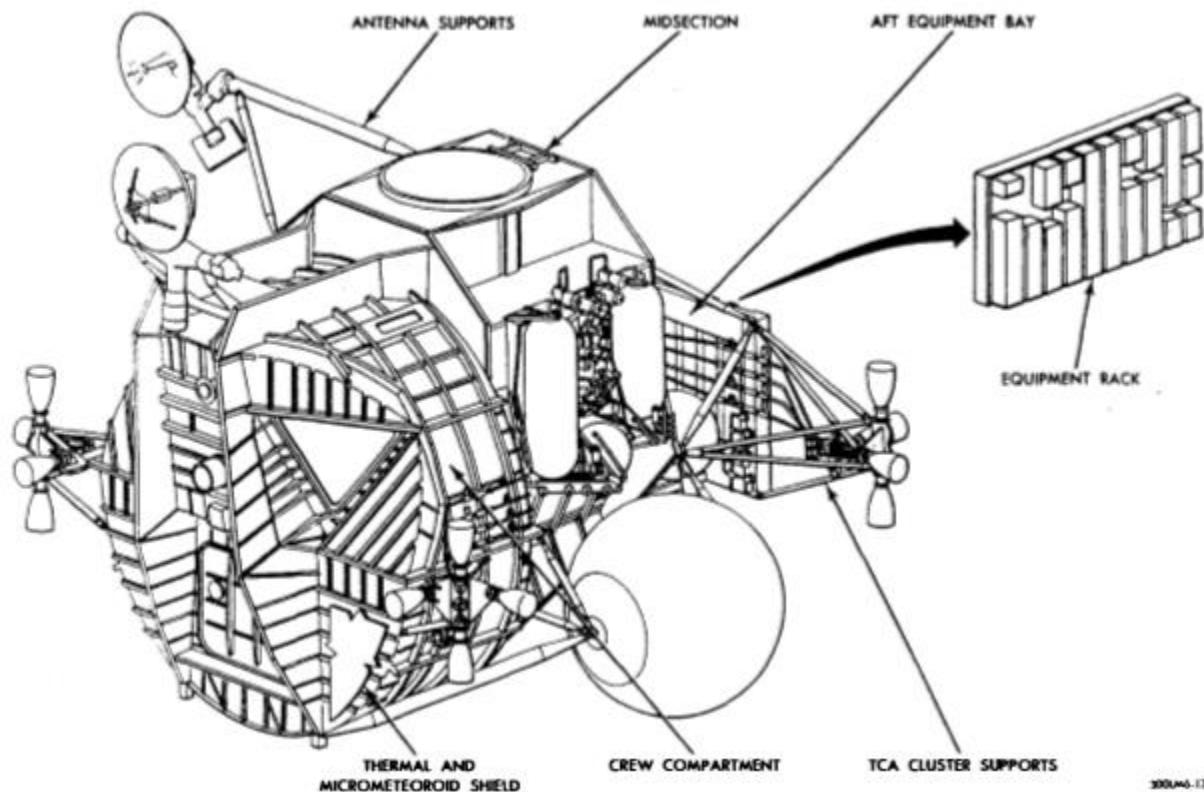


Figure 2.1 - The ascent stage

The ascent stage has two hatches. The forward hatch is in the front-face assembly, just below the display panels. It is only possible to open this hatch when the cabin is completely depressurized, and is used to perform lunar EVAs. The overhead hatch is in front of the docking tunnel when docked with the CSM. It is used by the astronauts to move between the LM and the

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CSM, and can only be opened when the pressure between the CSM and LM is equalized.

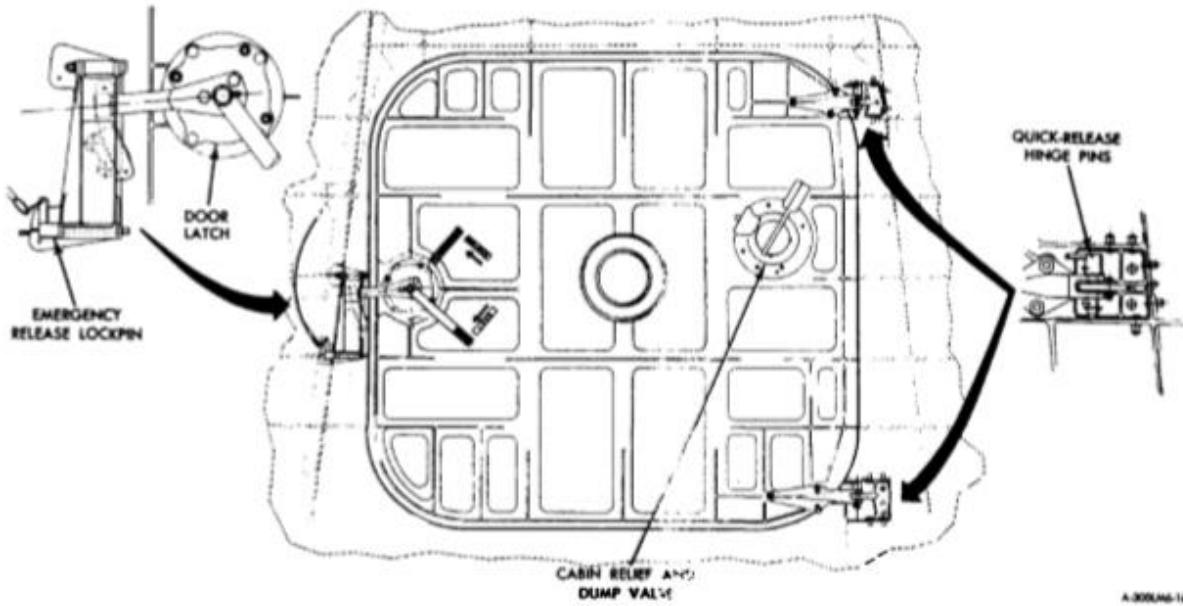


Figure 2.2 - The forward hatch

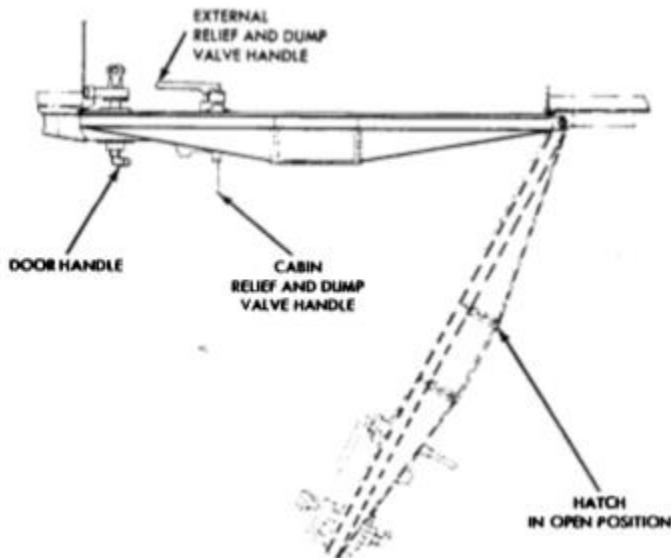


Figure 2.3 - The overhead hatch

The crew compartment is the frontal area of the ascent stage, and is the flight station area. It has control and display panels, armrests, body restraints, landing aids, two triangular front windows, a docking window, a COAS, and an alignment optical telescope (AOT).

There are 12 control and display panels used to operate the spacecraft. Panel 1 and Panel 2 are located on each side of the front face assembly centerline, at eye level. They contain warning

lights, flight indicators, propellant quantity indicators, Reaction Control Subsystem (RCS) controls, and Environmental Control Subsystem (ECS) indicators and controls.

Panel 3 spans the width of Panel 1 and 2, and is located just below them. It contains radar antenna temperature indicators and engine, radar, spacecraft stability, event timer, RCS and lighting controls.

Panel 4 contains the attitude controller assembly (ACA) and thrust translation controller assembly (TTCA) controls, inertial indicators, and LM guidance computer (LGC) indicators and controls.

Panel 5 and 6 contains lighting and mission timer controls, engine start and stop pushbuttons, and the abort guidance controls.

Panel 8 is at the left of the Commander's station, and contains the Explosive Devices controls, audio controls, and heater controls.

Panel 11 is above Panel 8 and contain the Commander's circuit breakers.

Panel 12 is at the right of the LM Pilot's station and contains audio and communications controls.

Panel 14 is above Panel 12 and contains the Electrical Power Subsystem (EPS) controls and displays.

Panel 16 is above Panel 14 and contains the LM Pilot's circuit breakers.

The ORDEAL is located above Panel 8 and contains the controls for orbiting LM attitude, with respect to a local horizontal, from the LGC.

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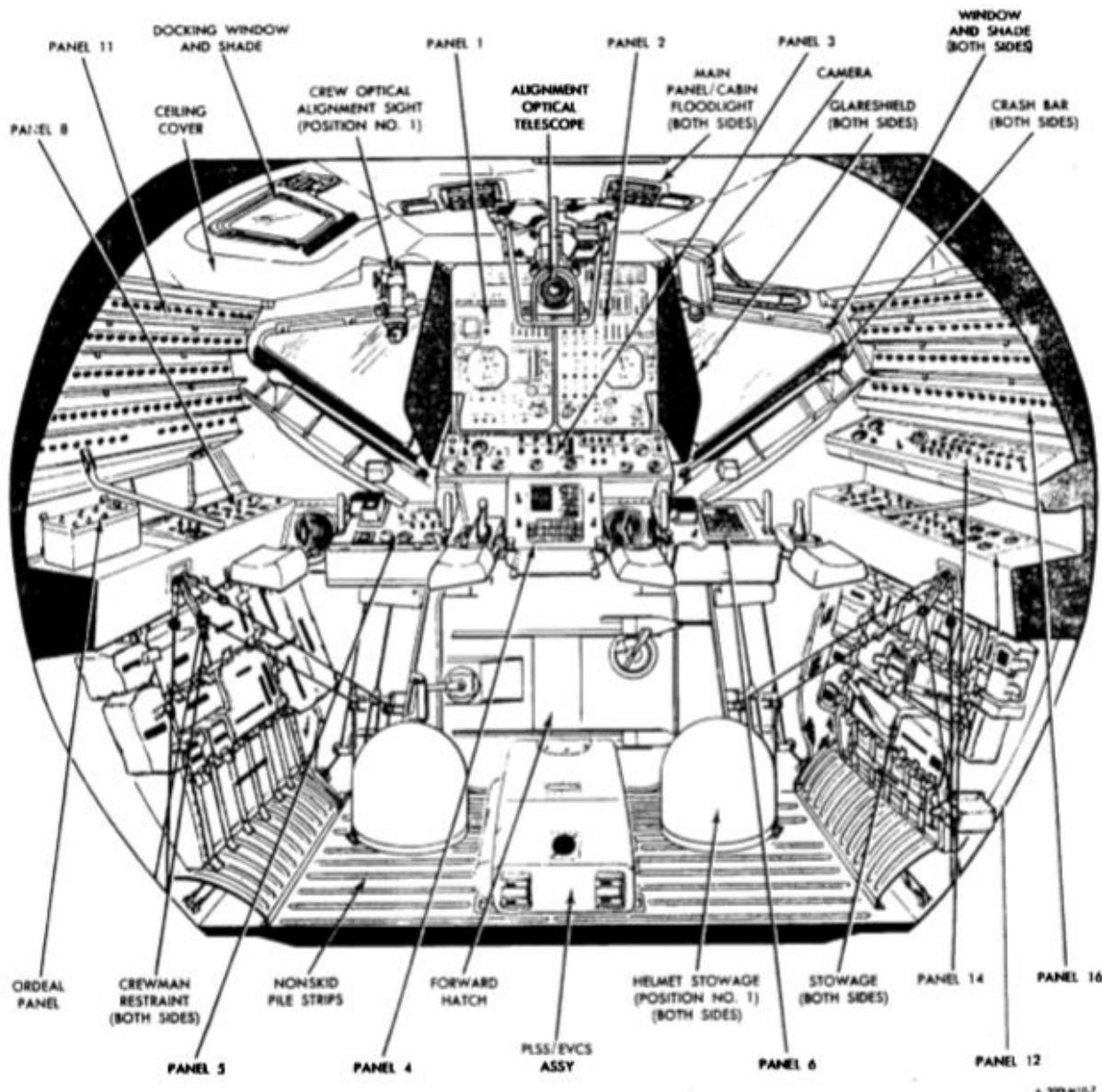


Figure 2.4 - The crew compartment

Two triangular windows in the front face assembly provide visibility during descent, ascent, and the rendezvous and docking phase of the mission. Both windows are canted down to the side to permit adequate peripheral and downward visibility. A third docking window is located above the Commander's flight station and provides visibility for docking maneuvers.

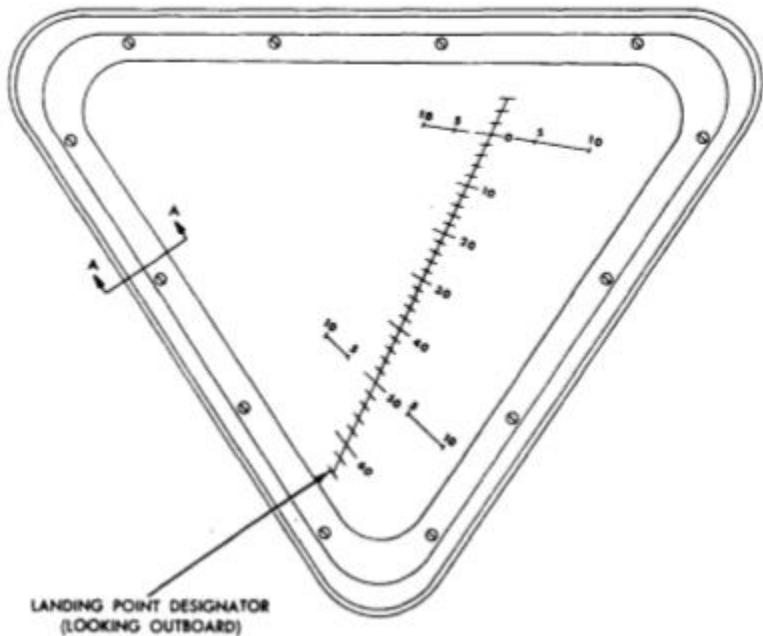


Figure 2.5 - The Commander's front window

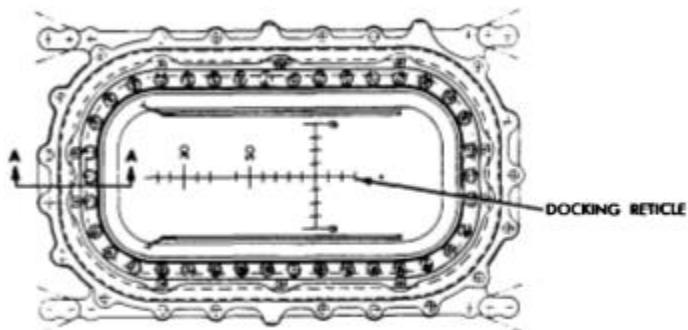


Figure 2.6 - The Docking Window

The mid-section is the area behind the two astronaut stations and control panels. The right side of the mid-section consist of the Environmental Control System (ECS) controls and most of the heat transport section and water glycol plumbing. The valves for operating the ECS equipment are readily accessible from the crew compartment. The left side of the mid-section contains flight data file, a portable life support system (PLSS), and other crew provisions stowage. Guidance, Navigation and Control Subsystem (GN&CS) electric units that don't require access by the astronauts are located on the midsection aft bulkhead.

A ring at the top of the ascent stage is used to dock with the CM, and is covered by the overhead hatch.

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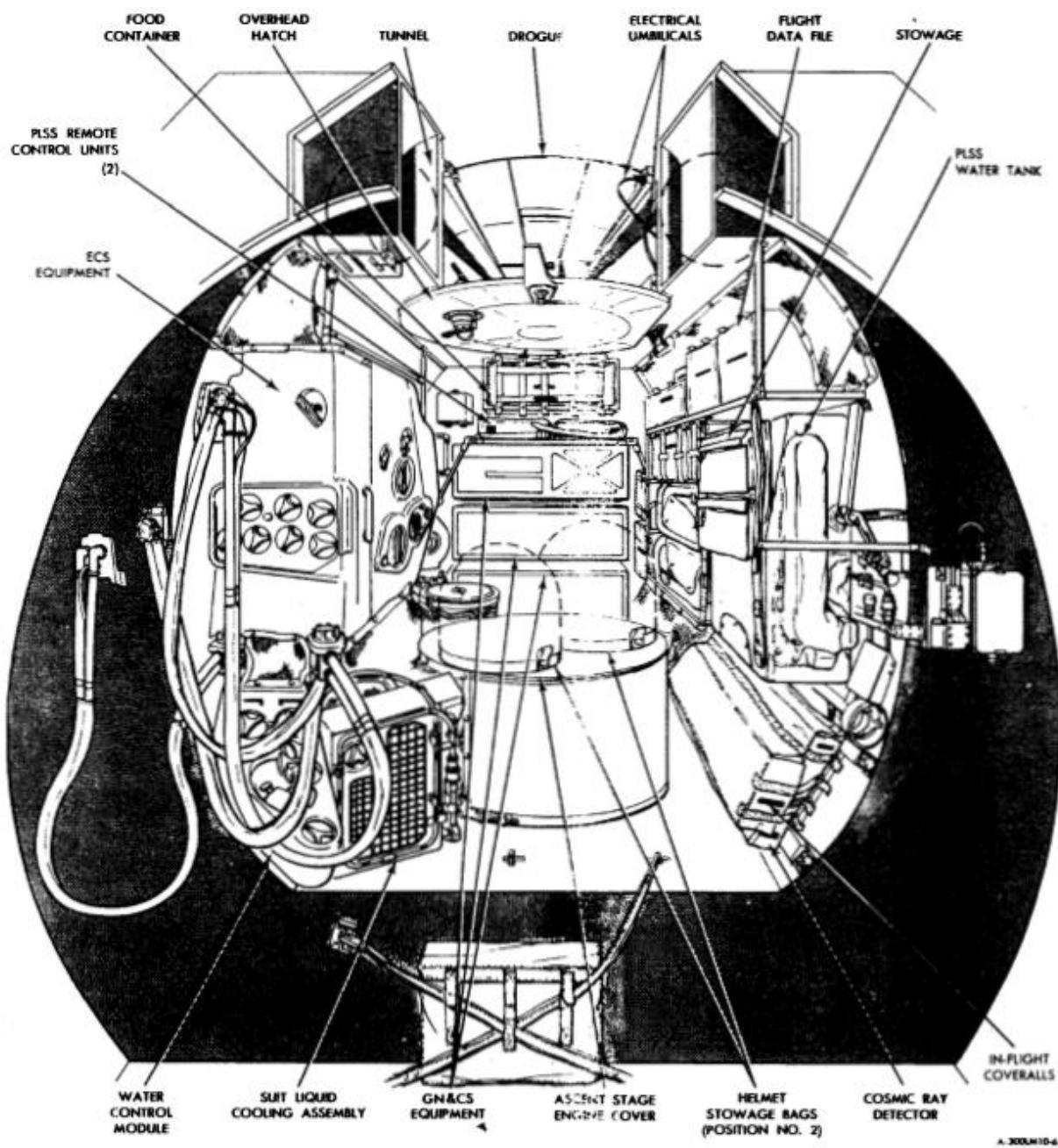


Figure 2.7 – The midsection compartment

The Aft Equipment Bay is unpressurized and has two oxygen tanks and two gaseous helium tanks, and the aft equipment rack. Two of the Thrust Camber Assembly clusters are located here, while the other two are in the front. These are used to control attitude and translational motion using the Reaction Control System (RCS).

### 3. DESCENT STAGE

The descent stage is the unmanned portion of the LM. Compartments inside the descent stage house equipment required by LM subsystems, the descent stage engine, and the landing gear.

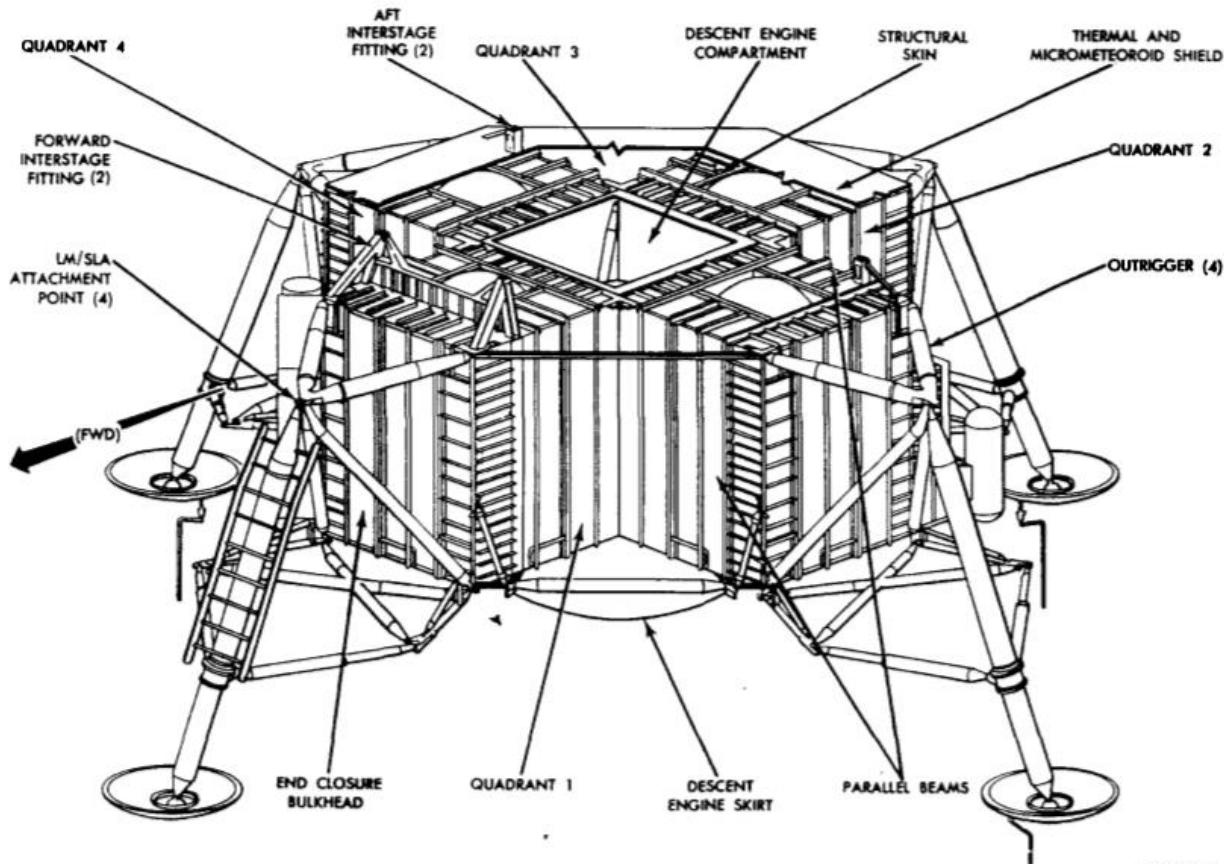
The center compartment houses the descent engine, and its two oxidizer tanks are stored in the forward and aft compartments, and its fuel tanks in the left and right-side compartments. The triangular areas between the main compartments are named quadrants.

Quadrant 1 has the mounting provisions for a Lunar Rover Vehicle (LRV), or Mobile Equipment Transport System (METS) and a high-pressure oxygen disconnect.

Quadrant 2 houses an ECS water tanks and the Apollo lunar surface experiment packages (ALSEP), a cosmic ray experiment package, a fuel cask for use with the ALSEP.

Quadrant 3 houses supercritical helium and ambient helium tanks, the descent engine control and assembly of the GN&CS, an ECS gaseous oxygen tank and interstage hardline disconnects.

Quadrant 4 houses Explosive Devices Subsystem (EDS) components, an ECS water tank and gaseous oxygen tanks, waste management container, and the propellant quantity gaging (PQGS). The five EPS batteries are mounted on the bulkhead.



### *Figure 2.8 – The descent stage*

The landing gear provides the impact attenuation required to land the LM on the lunar surface, prevents tip over, and supports the LM during lunar stay and launch. Landing impact is attenuated to load levels that preserve the structural integration of the LM.

At the initial Apollo Saturn V launch, the landing gear is stowed in a retracted position, and remains there until the commander operates to LDG GEAR DEPLOY switch on panel 8 in the Lunar Module. A Lunar-Surface sensing probe is attached to the left, right and aft landing gear footpads. These energize the LUNAR CONTACT lights to advise the crew to shut off the descent engine.

## 4. LM – SLA – S-IVB CONNECTIONS

At earth launch, the LM is within the SLA, which is connected to the S-IVB booster. The SLA has an upper section and a lower section. The outriggers, to which the landing gear is attached, provide attachment points for securing the LM to the SLA lower section. The upper section of the SLA is explosively separated into four segments, which fold back upon release and are then forced away from the SLA by spring thrusters. The LM is then explosively released from the lower section on command by the crew.

## 5. LM-CSM INTERFACES

A ring at the top of the ascent stage provides a structural interface for joining the LM to the CM. The drogue portion of the docking mechanism is secured below this ring. The drogue is required during docking operations to mate with the CM-mounted probe.

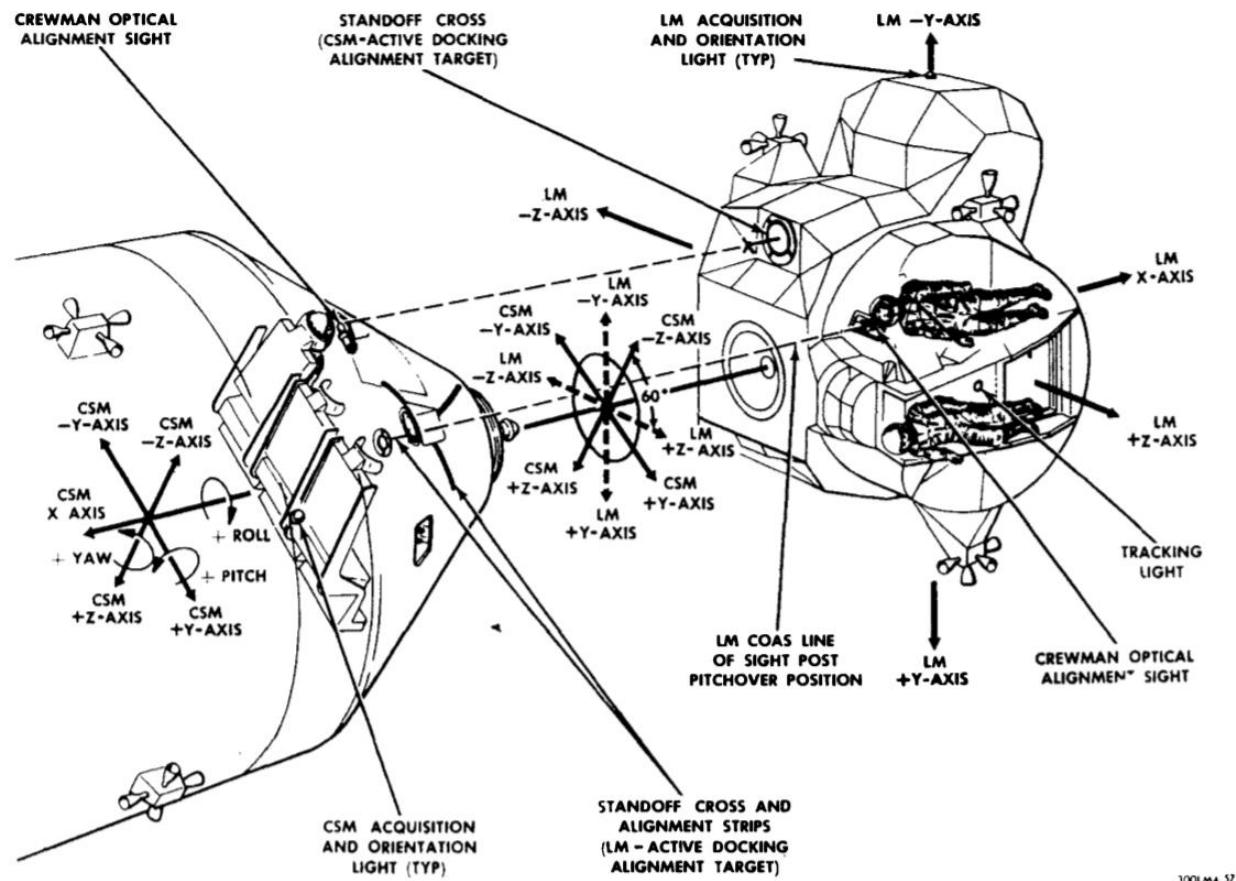
The crew transfer tunnel (LM-CM interlock area) is the passageway created between the LM overhead hatch and the CM forward pressure hatch when the LM and the CSM are docked. The tunnel permits inter-vehicular transfer of the crew and equipment without exposure to space. The tunnel and the LM are normally pressurized from the CM.

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Figure 2.9 – LM-CSM interfaces and reference axes

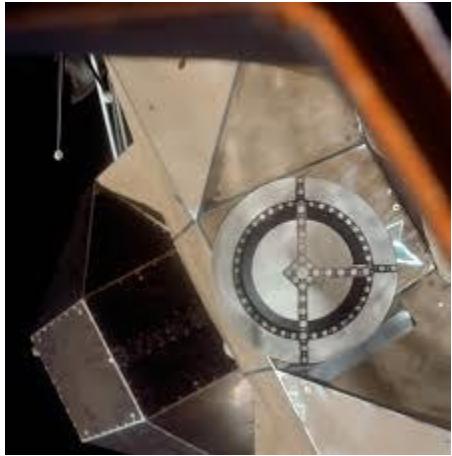
Twelve latches are spaced equally about the periphery of the CM docking ring. They are placed around and within the CM tunnel so they do not interfere with docking probe operation. When secured, the latches and the docking drogue ensure structural continuity and pressurization between the LM and the CM, and seal the tunnel interface.

An electrical umbilical, in the LM portion of the tunnel, is connected by an astronaut to the CM. This happens automatically in Reentry, when docking.

The drogue assembly is a conical structure with provisions for mounting the LM portion of the crew transfer tunnel. The drogue may be removed from either end of the crew transfer tunnel, and may be temporarily stowed inside the CM or the LM, during Service Propulsion System (SPS) burns. This happens automatically in Reentry, and you only need to pressurize the tunnel, and remove/insert/open/close the hatches manually.

## 6. DOCKING INTERFACES

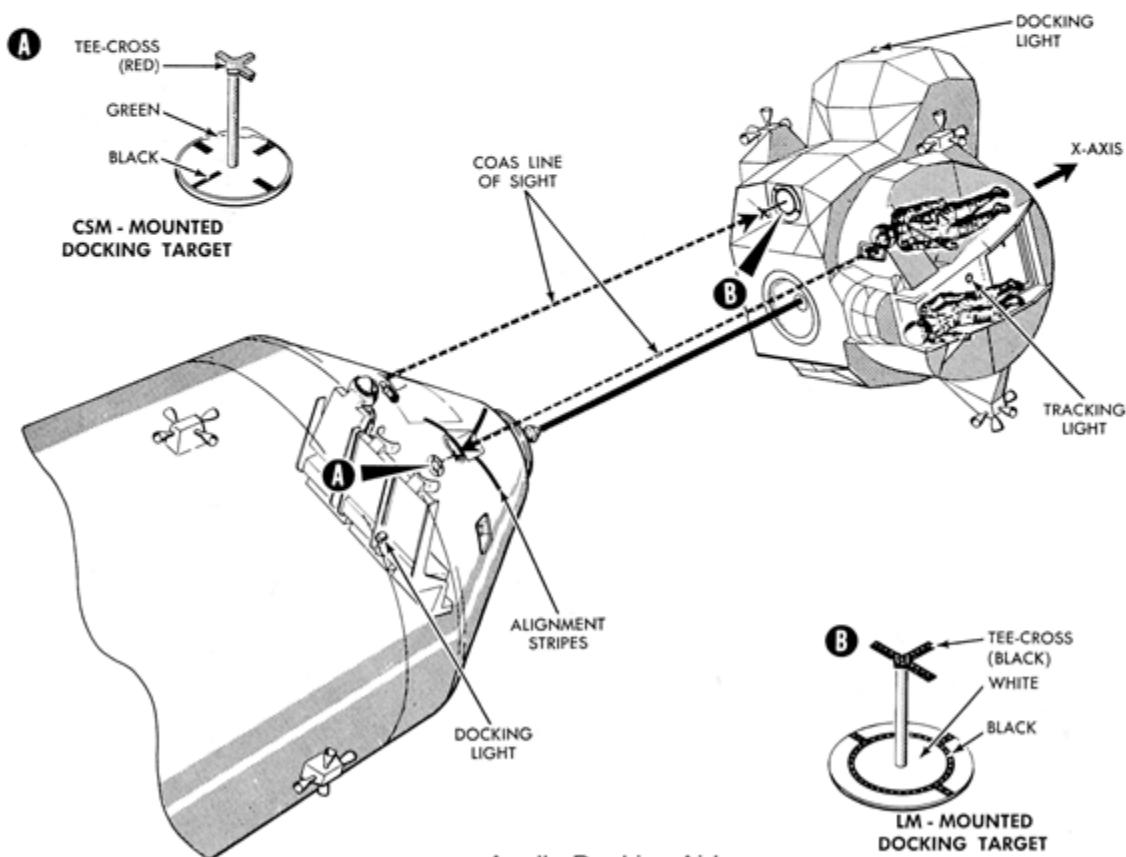
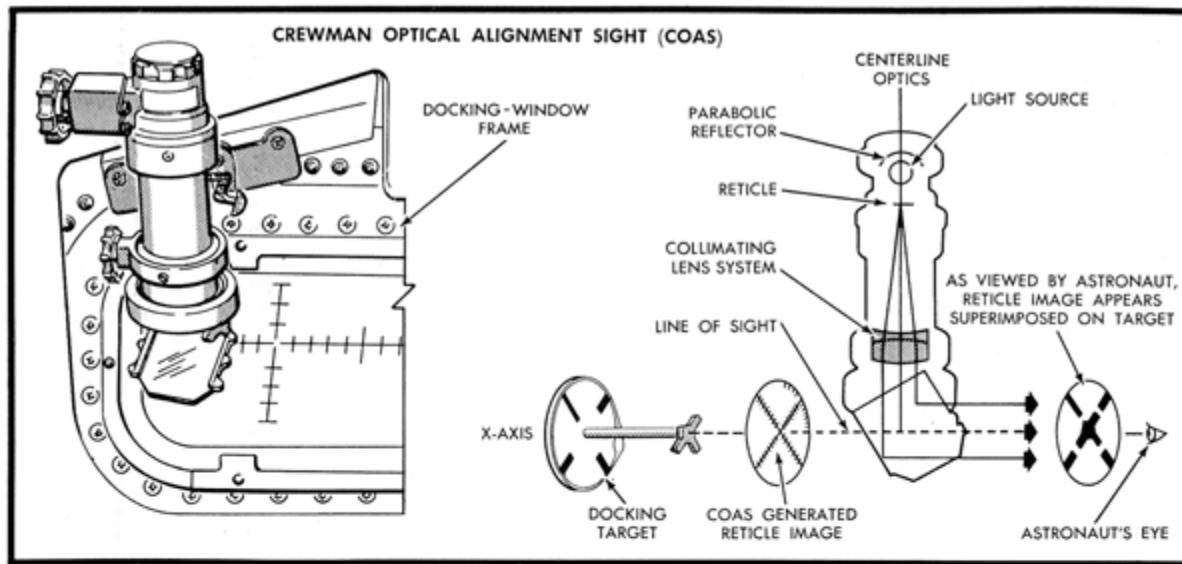
Two docking targets are available as aid during either LM to CM docking, or CM to LM docking (depending on who performs the active docking part).



An alignment target is recessed into the LM. The target has a standoff cross, with illuminated spherical markers. It has two layers, one background disk, and a standoff cross that extrudes out from the disc. Crossmembers on the standoff cross will be aligned with the orientation indicators and centered within the target circle of the COAS, when viewed through the commanders' window of the Command Module.

The CM docking target is simpler, but similar logic.

*Figure 2.10 – The LM docking target*



**Apollo Docking Aids**  
Ref Grumman Apollo News Reference

(from *Historic Space Systems at space1.com*)

Figure 2.11 – The Docking Target and COAS logic

As figure 2.11 indicates, the Tee-cross in A (CSM side) or the Tee-cross in B (LM side) is used to align the docking between the two spacecrafts, depending on who performs the active part of the docking.

# III. GUIDANCE & CONTROL



## III. GUIDANCE & CONTROL

### 1. GENERAL

The primary function of the Guidance, Navigation, and Control Subsystem (GN&CS) is accumulation, analysis, and processing of data to ensure that the vehicle follows a planned trajectory. The GN&CS provides navigation, guidance, and flight control to accomplish the specific guidance goal. To accomplish guidance, navigation, and control, the astronauts use controls and indicators that interface with the various GN&CS equipment. Functionally, this equipment is contained in a primary guidance and navigation section (PGNS), an abort guidance section (AGS), and a control electronics section (CES).

The PGNS provides the primary means for implementing inertial guidance and optical navigation for the vehicle. When aided by either the rendezvous radar (RR) or the landing radar (LR), the PGNS provides for radar navigation. The PGNS, when used in conjunction with the CES, provides automatic flight control. The astronauts can supplement or override automatic control, with manual inputs. The PGNS acts as a digital autopilot in controlling the vehicle throughout the mission.

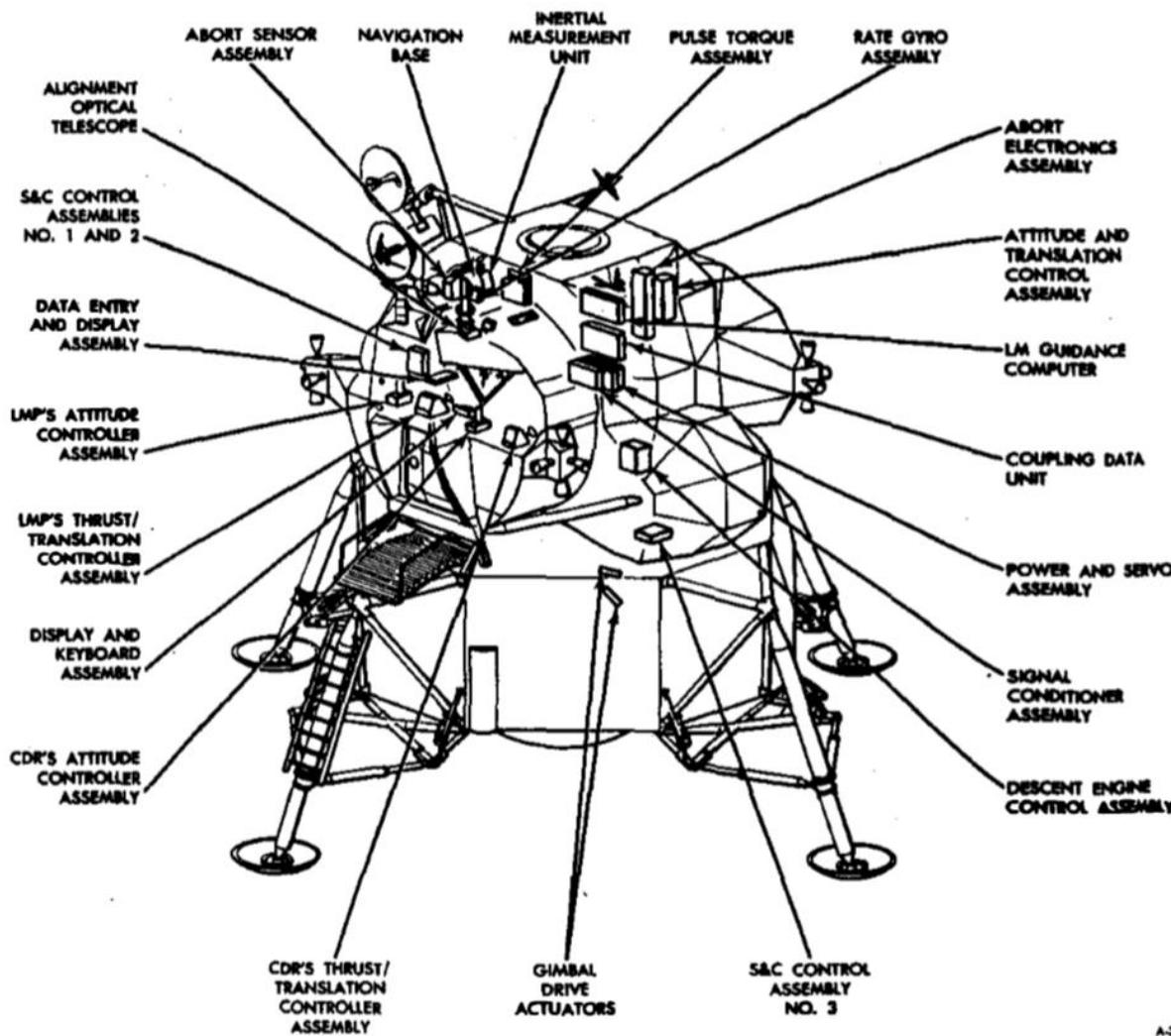
Normal guidance requirements include transferring the vehicle from a lunar orbit to its descent profile, achieving a successful landing at a preselected or crew-selected site, and performing a powered ascent maneuver that results in terminal rendezvous with the CSM.

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The PGNS provides the navigational data required for vehicle guidance. These data include line-of-sight (LOS) data from an alignment optical telescope (AOT) for inertial reference alignment, signals for initializing and aligning the AGS, and data to the astronauts for determining the location of the computed landing site.

The AGS is primarily used only if the PGNS malfunctions and is not implemented in Reentry.

The CES processes Reaction Control Subsystem (RCS) and Main Propulsion Subsystem (MPS) control signals for vehicle stabilization and control. To stabilize the vehicle during all phases of the mission, the CES provides signals that fire any combination of the 16 RCS thrusters. These signals control attitude and translation about or along all axes. The attitude and translation control data inputs originate from the PGNS during normal automatic operation, or from two hand controllers during manual operations.

The CES also processes on and off commands for the ascent and descent engines and routes automatic and manual throttle commands to the descent engine. Trim control of the gimbaled

descent engine is also provided to assure that the thrust vector operates through the vehicle center of gravity (automatic in Reentry).

These integrated sections (PGNS, AGS, and CES) allow the astronauts to operate the vehicle in fully automatic, several semiautomatic, and manual control modes.

### 1.2. PRIMARY GUIDANCE AND NAVIGATION SECTION

The PGNS includes three major subsections similar to the Apollo Command Module: inertial, optical, and computer.

#### 1.2.1. INERTIAL SUBSECTION

The inertial subsection (ISS) establishes the inertial reference frame that is used as the central coordinate system from which all measurements and computations are made. The ISS measures attitude and incremental velocity changes, and assists in converting data for computer use, onboard display, or telemetry. Operation is started automatically by the guidance computer or by an astronaut using the computer keyboard. Once the ISS is energized and aligned to the inertial reference, any vehicle rotation (attitude change) is sensed by a stable platform. All inertial measurements (velocity and attitude) are with respect to the stable platform. These data are used by the computer in determining solutions to the guidance problems.

The IMU is the primary inertial sensing device of the vehicle. It is a three-degree-of-freedom, stabilized device that maintains an orthogonal, inertially referenced coordinate system for vehicle attitude control and maintains three accelerometers in the reference coordinate system for accurate measurement of velocity changes. The IMU contains a stable platform, gyroscopes, and accelerometers necessary to establish the inertial reference. The stable platform serves as the space-fixed reference for the ISS. It is supported by three gimbal rings (outer, middle, and inner) for complete freedom of motion.



You can think of it as a device that has its own body-axis, and tracks the orientation of the Lunar Module relative to a set stable and fixed platform. It provides the attitude of the spacecraft relative to this platform. The attitude is seen on the attitude indicator on panel 1 and 2.

### 1.2.2. OPTICAL SUBSECTION

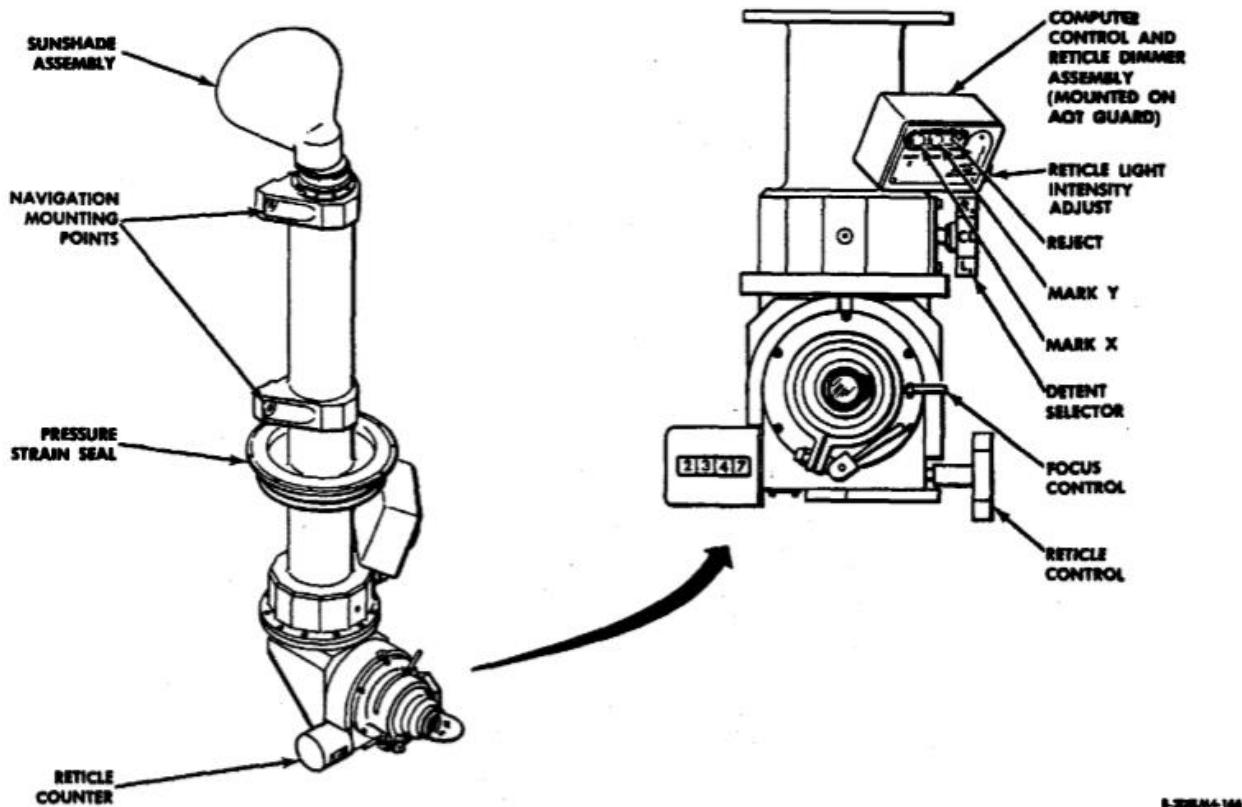
The optical subsection (OSS) is used to determine the position of the vehicle using a catalog of stars stored in the computer and celestial measurements made by an astronaut. The identity of celestial objects is determined before earth launch. The AOT is used by the astronaut to take direct visual sightings and precise angular measurements of a pair of celestial objects. The computer subsection (CSS) uses this data, along with prestored data, to compute position and velocity and to align the inertial components. The OSS consists of the AOT and a computer control and reticle dimmer (CCRD) assembly.

The optics are most commonly used to remove drift error in the IMUs. Unlike the Command Module, these optics have not yet been implemented in Reentry and all IMU drift-errors are automatically removed.

The AOT, an L-shaped periscope, is used by the astronaut to take angular measurements of celestial objects. These angular measurements are required for orienting the platform during certain periods while the vehicle is in flight and during prelaunch preparations while on the lunar

surface. Sightings taken with the AOT are transferred to the LGC by the astronaut, using the CCRD assembly. This assembly also controls the brightness of the telescope reticle pattern.

The computer control and reticle dimmer (CCRD) is used to perform the markings. The MARX X and MARX Y pushbuttons are used by the astronauts to send discrete signals to the LGC when star sightings are made. The REJECT pushbutton is used if an invalid mark has been sent to the LGC. A thumbwheel on the assembly adjusts the brightness of the telescope reticle lamps.



### 1.2.3. COMPUTER SUBSECTION

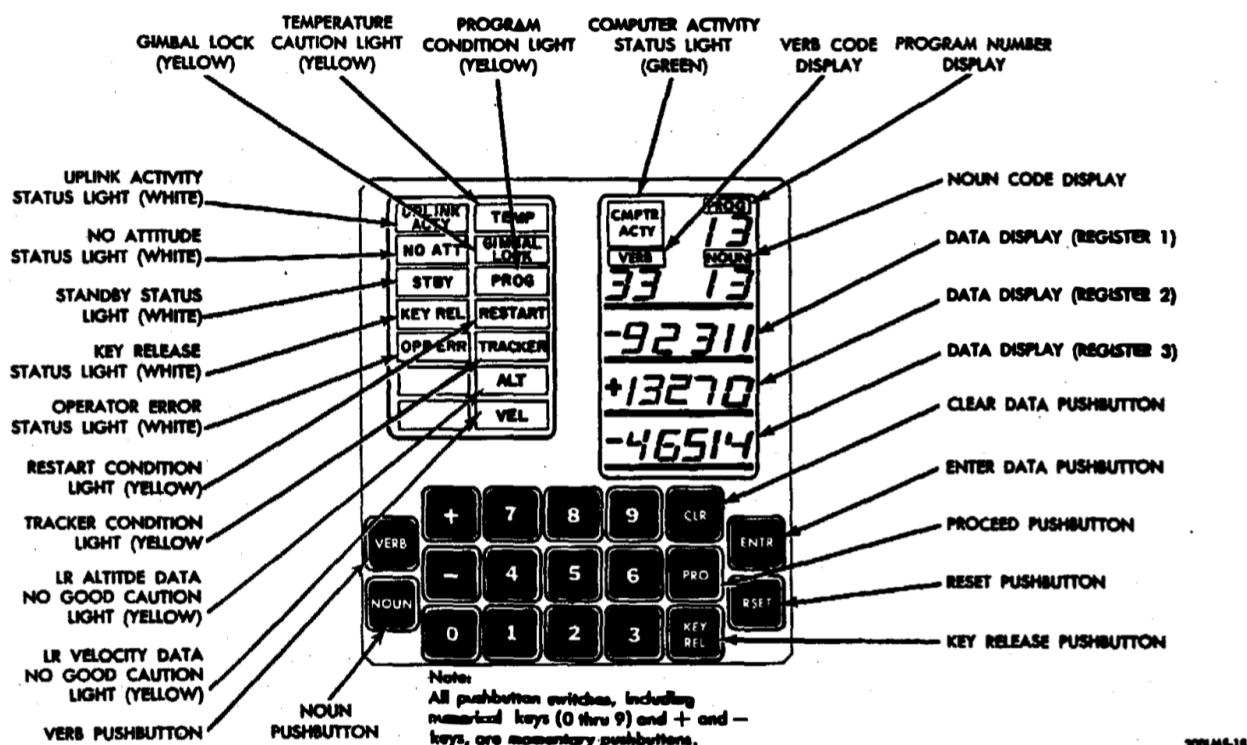
The computer subsection (CSS) is described in the Lunar Guidance Computer chapter but it's important to understand how it influences the PGNS, so the introduction is given here as well.

It is the control and data-processing center of the vehicle that performs all the guidance and navigation functions necessary for automatic control of the flight path and attitude of the vehicle. For these functions, the GN&CS uses a digital computer. The computer is a control computer with many of the features of a general-purpose computer. As a control computer, it aligns the stable platform, and positions both radar antennas. It also provides control commands to both radars, the ascent engine, the descent engine, the RCS thrusters, and the vehicle cabin displays. As a general-purpose computer, it solves guidance problems required for the mission. The CSS consists of a LM guidance computer (LGC) and a display and keyboard (DSKY), which is the computers control panel.

The LGC is used to solve flight equations. The results are used to determine the required magnitude and direction of thrust. The LGC establishes corrections to be made. The vehicle engines are turned on at the correct time, and steering commands are controlled by the LGC to orient the vehicle to a new trajectory, if required. The ISS senses acceleration and supplies velocity changes, to the LGC, for calculating total velocity. The LGC is communicating with the ISS for attitude information.

The display and keyboard (DSKY) is used to load information into the LGC, retrieve and display information contained in the LGC, and initiate any program stored in memory. The astronauts can also use the DSKY to control the modeing of the ISS. The exchange of data between the astronauts and the LGC is usually initiated by an astronaut; however, it can also be initiated by internal computer programs.

The DSKY is located on panel 4, between the Commander and LM Pilot and above the forward hatch. The upper half is the display portion; the lower half comprises the keyboard. The display portion contains five caution indicators, six status indicators, seven operation display indicators, and three data display indicators. These displays provide visual indications of data being loaded in the LGC, the condition of the LGC, and the program being used. The displays also provide the LGC with a means of displaying or requesting data.



### 1.3. ABORT GUIDANCE SECTION

*Note: The AGS is not yet implemented in Reentry – An Orbital Simulator.*

The AGS consists of an abort sensor assembly (ASA), abort electronics assembly (AEA), and a data entry and display assembly (DEDA). The ASA performs the same function as the IMU; it establishes an inertial reference frame, used by the AGS. The AEA, a high-speed, general-purpose digital computer, is the central processing and computational device for the AGS. The DEDA is the input-output device for controlling the AEA.

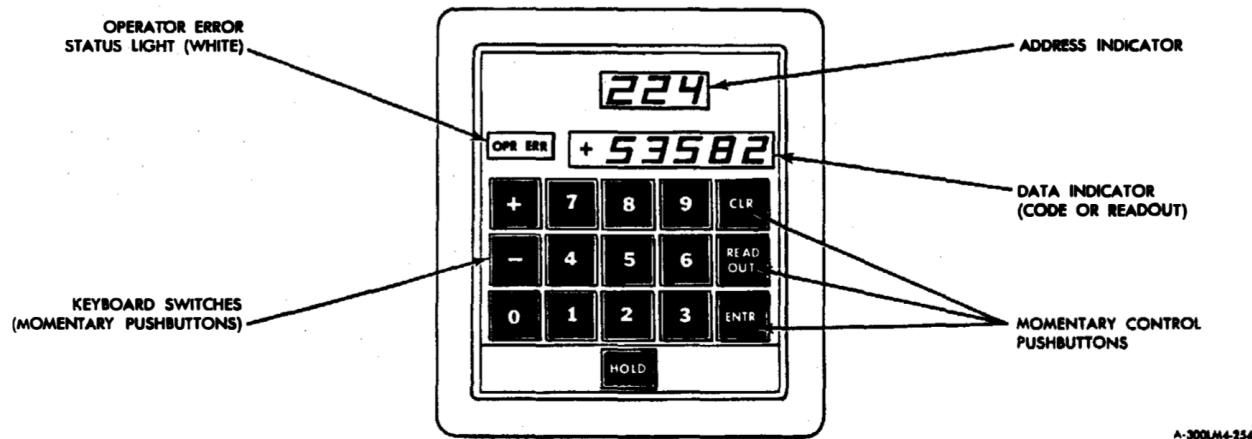
Navigation is performed by the AGS through integration of the equations of motion and substitution of instantaneous LM velocity for the variables. The AGS decodes the PGNS downlink data to establish LM and CSM position, velocity, and associated time computations. This information is used to initialize or update the AGS navigational computations upon command from the DEDA. The AGS solves the guidance problems of five distinct guidance routines: orbit insertion, coelliptic sequence initiate, constant delta altitude, terminal phase initiate, and change in LM velocity (external dV).

The AGS provides steering commands for three steering submodes: attitude hold, guidance steering, and acquisition steering. The attitude hold submode maintains the vehicle attitude that exists when the submode is entered. In the guidance steering submode, the AEA generates attitude commands to orient the LM X-axis so that it lies along the direction of the thrust vector. In the acquisition steering submode, the AEA generates attitude commands to orient the LM Z-axis along the estimated line of sight (LOS) between the LM and CSM.

The AGS outputs an engine-on or engine-off command during all thrusting maneuvers. If the PGNS is in control, the command is a follow-up of the signal produced by the PGNS. If the AGS is in control, the engine-on command can be routed only after the appropriate switches are set and the ullage maneuver has been performed. When proper velocity-to-be-gained are achieved, an engine-off command is issued.

The AGS uses RR angle information and accepts range and range-rate information from the RR for updating LM navigation so that the LM Z-axis is toward the CSM, or for midcourse correction. These data are manually inserted into the AEA by the astronaut by using the DEDA.

The DEDA (panel 6) is used by the astronauts to select the desired mode of operation, insert the desired targeting parameters, and monitor related data throughout the mission. Essentially, the DEDA consists of a control panel to which electroluminescent displays and data entry pushbuttons are mounted and a logic enclosure that houses logic and input-output circuits.



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#### 1.4. CONTROL ELECTRONICS SECTION

The CES processes attitude and translation signals when operating in the primary guidance mode or the abort guidance mode. It's mainly used to command the Reaction Control System.

The CES converts RCS commands to the required electrical power to operate the RCS solenoid valves. The CES accepts discrete (on and off) descent engine gimbal commands and, upon receipt of an on command, causes the descent engine to move about its gimbal axis. The CES accepts LGC and manual engine on and off commands and routes them to the MPS in fire or stop the descent or ascent engine. The CES accepts LGC and manual thrust commands to throttle the descent engine (10% to 92. 5% of maximum thrust). The CES also provides manual attitude and translation commands to the LGC.

The CES comprises two attitude controller assemblies (ACA's), two thrust/translation controller assemblies (TTCA's).

##### 1.4.1. ATTITUDE CONTROLLER ASSEMBLIES

The ACA's are right-hand pistol grip controllers, which the astronauts use to command changes in vehicle attitude. Each ACA supplies attitude rate commands proportional to the displacement of its handle, to the LGC and the ATCA; an out-of-detent discrete each time the handle is out of its neutral position; and a follow-up discrete to the AGS each time the controller is out of detent.

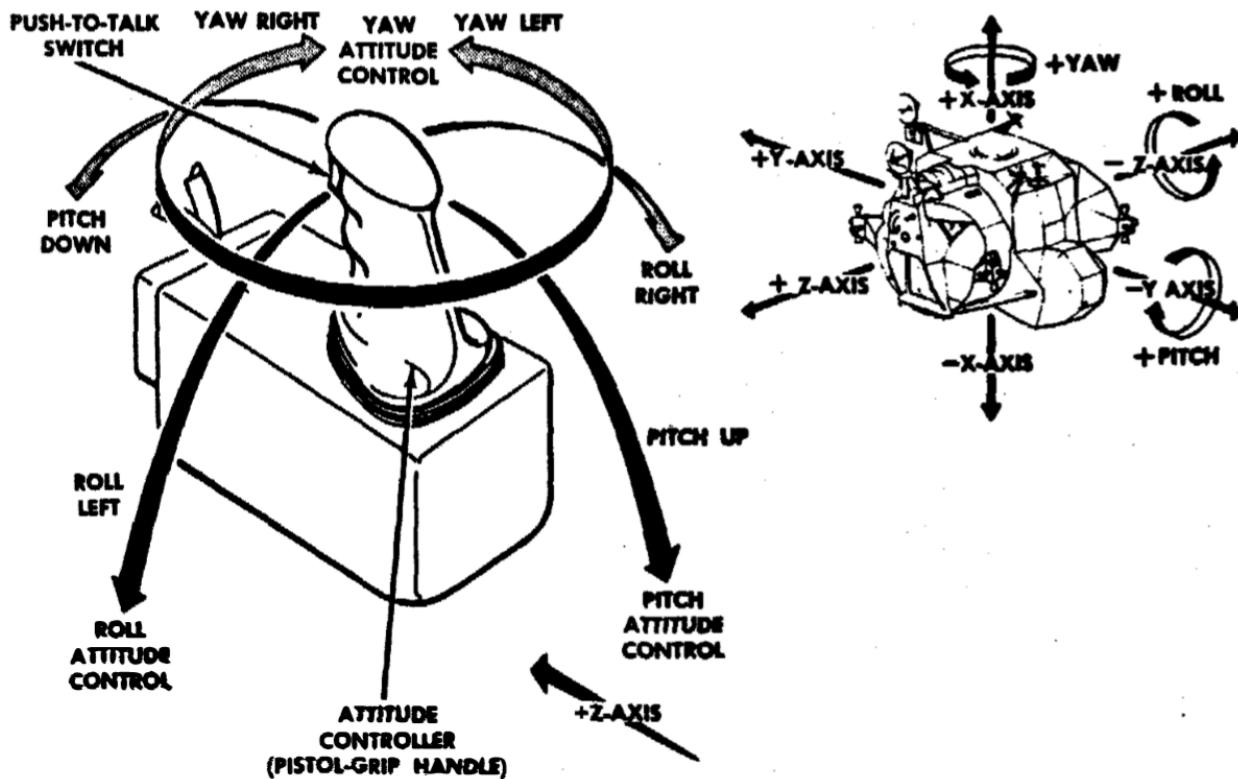
As the astronaut uses the ACA, the hand movements are analogous to vehicle rotations. Clockwise or counterclockwise rotation of the controller commands yaw right or yaw left, respectively. Forward or aft movement of the controller commands vehicle pitch down or up, respectively. Left or right movement of the controller commands roll left or right, respectively.

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The ACA's are also used in an incremental landing point designator (LPD) mode, which is available to the astronauts during the final approach phase. In this mode, the angular error between the designated landing site and the desired landing site is nulled by repetitive manipulation of an ACA. LPD signals from the ACA are directed to the LGC, which issues commands to move the designated landing site incrementally along the Y-axis and Z-axis.

#### 1.4.2. THRUST/TRANSLATION CONTROLLER ASSEMBLIES

The TTCA's control LM translation in any axis; they are functionally integrated translation and thrust controllers. The astronauts use these assemblies to command vehicle translations by firing RCS thrusters and to throttle the descent engine between 10% and 92. 5% thrust magnitude. The controllers are three-axis, T-handle, left-hand controllers, mounted with their longitudinal axis approximately 45° from a line parallel to the LM Z-axis (forward axis).

A lever on the right side of the TTCA enables the astronaut to select either of two control functions: (1) translation control in the Y-axis and Z-axis, using the RCS thrusters, and descent engine throttling to control X-axis translation and (2) translation control in all three axes, using the RCS thrusters.

Due to the TTCA mounting position, vehicle translations correspond to astronaut hand movements when operating the controller. Moving the T-handle to the left or right commands

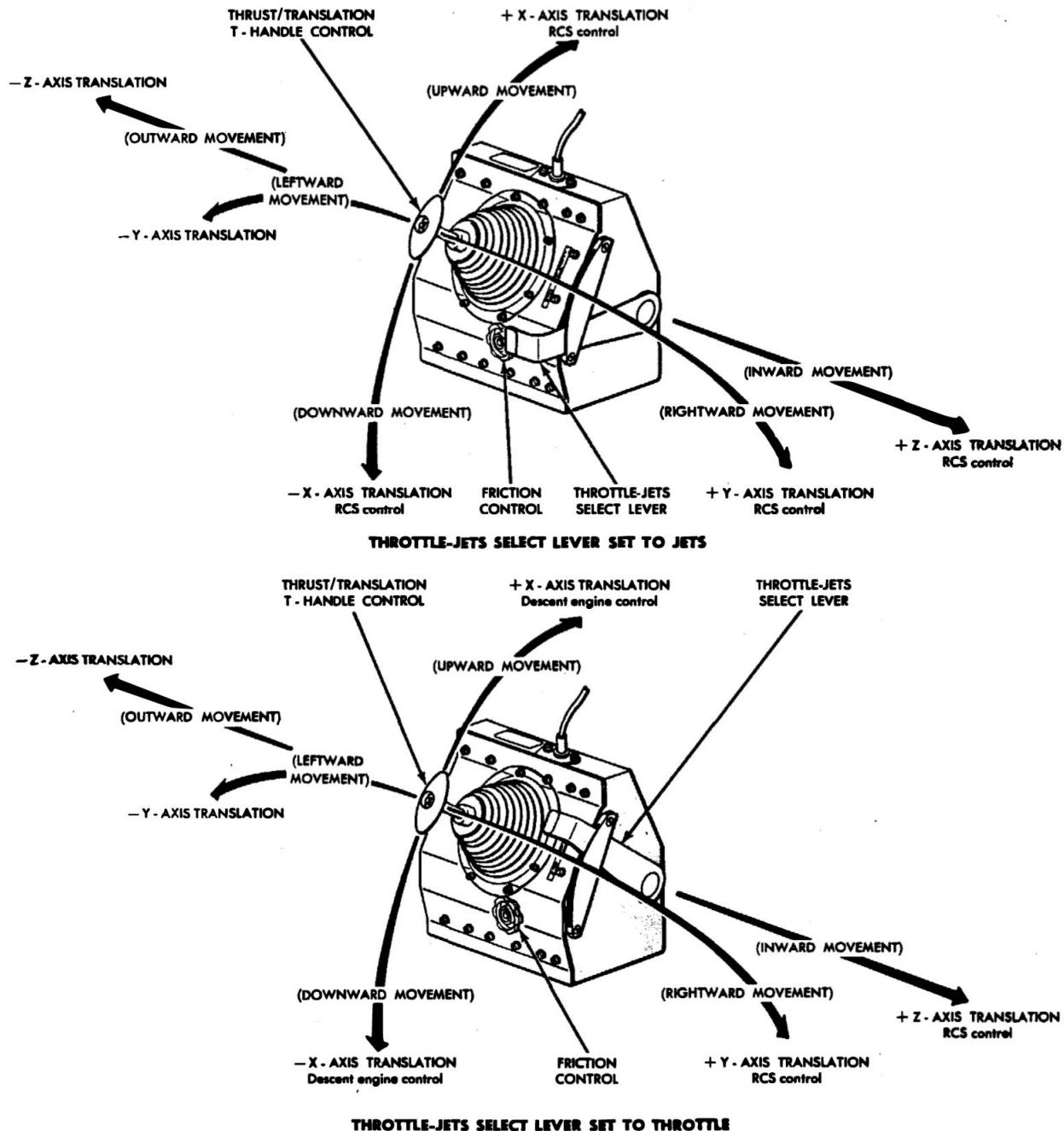
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translation along the Y-axis. Moving the T-handle inward or outward commands translation along the Z-axis. Moving the T-handle upward or downward commands translation along the X-axis, using the RCS thrusters, when the selected lever is in the down position. When the lever is in the up position, upward or downward movement of the TTCA increases or decreases, respectively, the magnitude of descent engine thrust. Regardless of the select lever position selected, the TTCA can command translation along the Y -axis and Z-axis, using the RCS thrusters.



### 1.5. ORDEAL

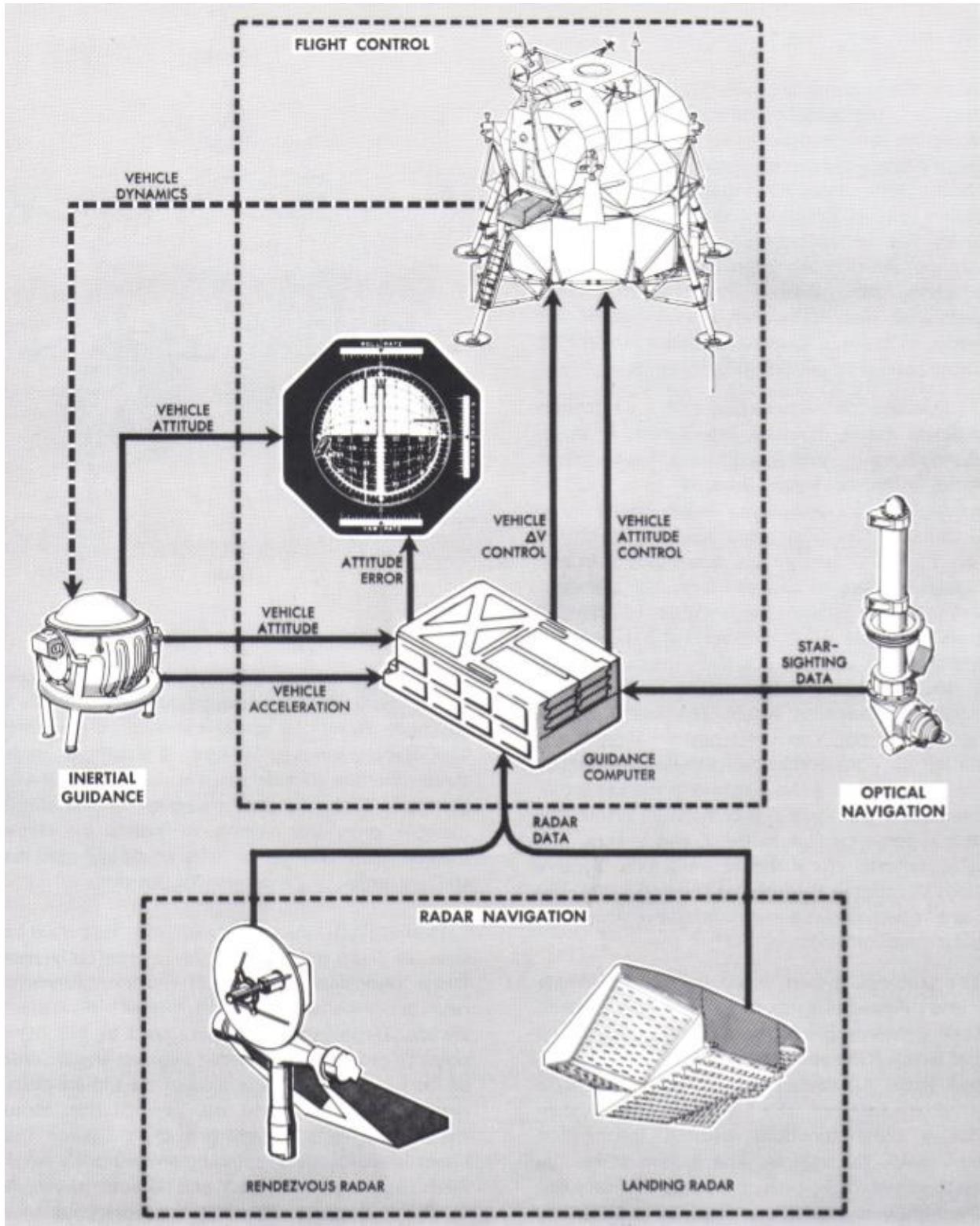
The Orbital Rate Display - Earth and Lunar (ORDEAL) provides an alternative to the attitude display. When selected, the ORDEAL produces an FDAO display of computed local vertical attitude during circular orbits around the earth or the moon. This has the same functionality as the one in the Apollo Command Module.

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*PGNS interfaces*

## 2. SUBSYSTEM INTERFACES

The PGNS interfaces with a lot of the systems in the Lunar Module. This section will cover the major interfaces.

### 2.1 GN&CS – MPS INTERFACES

The GN&CS provides a sequence of commands to the Main Propulsion Subsystem (MPS) to control the ascent and descent engines. For ignition to occur, the engine must first be armed. Normally, this involves setting the ENG ARM switch to the desired position. Depending on the switch setting, a discrete is generated in the CES to enable the START pushbutton (panel 5) for ascent engine operation or to operate actuator isolation valves for descent engine operation. Under abort or emergency conditions, the ABORT and ABORT STAGE pushbuttons (panel 1) are used to perform the arming function.

When the PGNS is in control, on and off commands are generated automatically by the LGC under program control, or manually with the START pushbutton (panel 5) and stop pushbuttons (panels 5 and 6). The on and off commands actuate pilot valves, which hydraulically open or close the fuel and oxidizer shutoff valves. Under emergency conditions, the ascent engine ignition sequence may also be automatically completed through use of the ABORT STAGE pushbutton. If the ascent engine-on command from either computer is lost, a memory circuit in the CES keeps issuing the command to the ascent engine.

The descent engine receives on and off commands, throttle commands, and trim commands from the Descent Engine Control Assembly (DECA). The ignition sequence commands for the descent engine are generated automatically or manually in a manner similar to that of the ascent engine. On and off commands are routed from either computer (dependent on the guidance mode selected), or the START and stop pushbuttons, through the DECA to actuate the descent engine pilot valves.

Throttle commands to the descent engine are generated automatically by the LGC under program control, or manually with a TTCA. The TTCA can be used to override LGC throttle commands.

The GN&CS generates trim commands to tilt the descent engine to control the direction of the thrust vector. The descent engine is tilted about the LM Y-axis and Z-axis to compensate for the offset of the center of gravity due to fuel depletion during descent engine operation.

### 2.2 GN&CS – RCS INTERFACES

The GN&CS provides on and off commands to the 16 TCA's (referred to as thrusters or jets) to control LM attitude and translation. In the primary mode of operation (PGNS in control), the LGC generates the required commands and sends them to the proper jet drivers in the CES. The jet drivers send selected on and off commands to the RCS primary solenoids. The Attitude and

Translation Control Assembly (ATCA) in the CES uses these inputs to select the proper thruster for attitude and translation control.

The thrusters are controlled manually with an ACA and a TTCA. The ACA provides attitude commands and the TTCA provides translation commands to the LGC. The ACA can fire the thrusters directly during the pulse, direct, and hardover modes, bypassing the LGC or AEA, and the ATCA. The four downward-firing thrusters may be fired by pressing the +X TRANSL pushbutton (panel 5). The on and off commands supplied to the thruster take the form of a step function. The duration of the signal determines the firing time of the selected thruster, which ranges from a pulse (less than 1 second) to steady-state (1 second or longer).

Each thruster contains an oxidizer solenoid valve and a fuel solenoid valve which, when open, pass propellant through an injector into the combustion chamber, where ignition occurs. Each valve contains a primary (automatic) solenoid and a secondary (direct) solenoid, which open the valve when energized. On and off commands from the ATCA are applied to the primary solenoids; the direct commands are applied to the secondary solenoids.

### 2.3 GN&CS – EPS INTERFACES

The Electrical Power Subsystem (EPS) supplies primary d-c and a-c power to the GN&CS. This power originates from seven silver-zinc batteries (five in the descent stage and two in the ascent stage). The descent batteries feed power to the buses during all operations, before staging. Immediately before staging occurs, ascent battery power is switched on and descent battery power is terminated. A deadface relay circuit deadfaces the descent batteries when normal staging occurs. Under emergency conditions, when the ABORT STAGE pushbutton is pressed, a power switchover command, which initiates deadfacing automatically, is routed to the EPS. The 28-volt d-c battery power is routed through an inverter to provide 115-volt, 400-cps ac to the GN&CS equipment.

### 2.4 GN&CS – ECS INTERFACES

The Environmental Control Subsystem (ECS) provides thermal stability for the temperature sensitive electronic equipment of the GN&CS. The electronic equipment (except the IMU) is mounted on cold plates and rails through which ECS coolant (ethylene glycol-water solution) is routed to remove heat. To cool the IMU, the coolant flows through its case. The heat that is removed from the equipment is vented overboard by the ECS sublimators.

### 2.5 GN&CS – EDS INTERFACES

The GN&CS interfaces with the Explosive Devices Subsystem (EDS) by supplying a descent engine on signal to the supercritical helium explosive valve and an ascent engine on signal,

which initiates the staging sequence. When the descent engine is operated for the first time, the MASTER ARM switch (panel B) is set to ON so that the supercritical helium explosive valve is blown when the descent engine-on signal is issued, and a time delay has been met. All other normal pressurization and staging sequences are initiated by the astronauts.

During an emergency situation, the ABORT STAGE pushbutton when pushed, shuts down the descent engine and pressurizes the APS, blowing the helium tank explosive valves that are selected by the ASC He SEL switch (panel B). After a time delay, the GN&CS generates an ascent engine-on signal which initiates the staging sequence as the ascent engine begins to fire. Upon completion of staging, a stage status signal is routed from the EDS deadface switch to the ATCA and to the LGC. This signal automatically selects the power deadband for RCS control during ascent engine operation.

### 3. FUNCTIONAL DESCRIPTION

The GN&CS comprises two functional loops, each of which is an independent guidance and control path. The primary guidance path contains elements necessary to perform all functions required to complete the lunar mission.

#### 3.1 PRIMARY GUIDANCE PATH

The primary guidance path comprises the PGNS, CES, Landing Radar, Rendezvous Radar, and the selected propulsion section required to perform the desired maneuvers. The CES routes flight control commands from the PGNS and applies them to the descent or ascent engine, and/or the appropriate thrusters.

The IMU, which continuously measures attitude and acceleration, is the primary inertial sensing device of the vehicle. The Landing Radar senses slant range and velocity. The Rendezvous Radar coherently tracks the CSM to derive Line-Of-Sight (LOS) range, range rate, and angle rate. The LGC uses AOT star-sighting data to align the IMU when needed (automatic in Reentry). Using inputs from the LR, IMU, RR, TTCA's, and ACA's, the LGC solves guidance, navigation, steering, and stabilization equations necessary to initiate on and off commands for the descent and ascent engines, throttle commands and trim commands for the descent engine, and on and off commands for the thrusters.

Control of the vehicle, when using the primary guidance path, ranges from fully automatic to manual. The primary guidance path operates in the automatic mode or the attitude hold mode. In the automatic mode, all navigation, guidance, stabilization, and control functions are controlled by the LGC. When the attitude hold mode is selected, the astronaut uses his ACA to bring the vehicle to a desired attitude. When the ACA is moved out of the detent position, proportional attitude-rate or minimum impulse commands are routed to the LGC. The LGC then

calculates steering information and generates thruster commands that correspond to the mode of operation selected via the DSKY. These commands are applied to the primary preamplifiers in the ATCA, which routes the commands to the proper thruster. When the astronaut releases the ACA, the LGC generates commands to hold this attitude. If the astronaut commands four-jet direct operation of the ACA by going to the hard over position, the ACA applies the command directly to the secondary solenoids of the corresponding thruster.

In the automatic mode, the LGC generates descent engine throttling commands, which are routed to the descent engine via the DECA. The astronaut can manually control descent engine throttling with his TTCA. The DECA sums the TTCA throttle commands with the LGC throttle commands and applies the resultant signal to the descent engine. The DECA also applies trim commands, generated by the LGC, to the GDA's to provide trim control of the descent engine. The LGC supplies on and off commands for the ascent and descent engines to the S&C control assemblies. The S&C control assemblies route the ascent engine on and off commands directly to the ascent engine, and the descent engine on and off commands to the descent engine via the DECA.

In the automatic mode, the LGC generates +X-axis translation commands to provide ullage. In the manual mode, manual translation commands are generated by the astronaut, using his TTCA. These commands are routed, through the LGC, to the ATCA and on to the proper thruster.

Figure 3.3.1.1 and 3.3.1.2 shows a simplified and detailed diagram of this path.

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

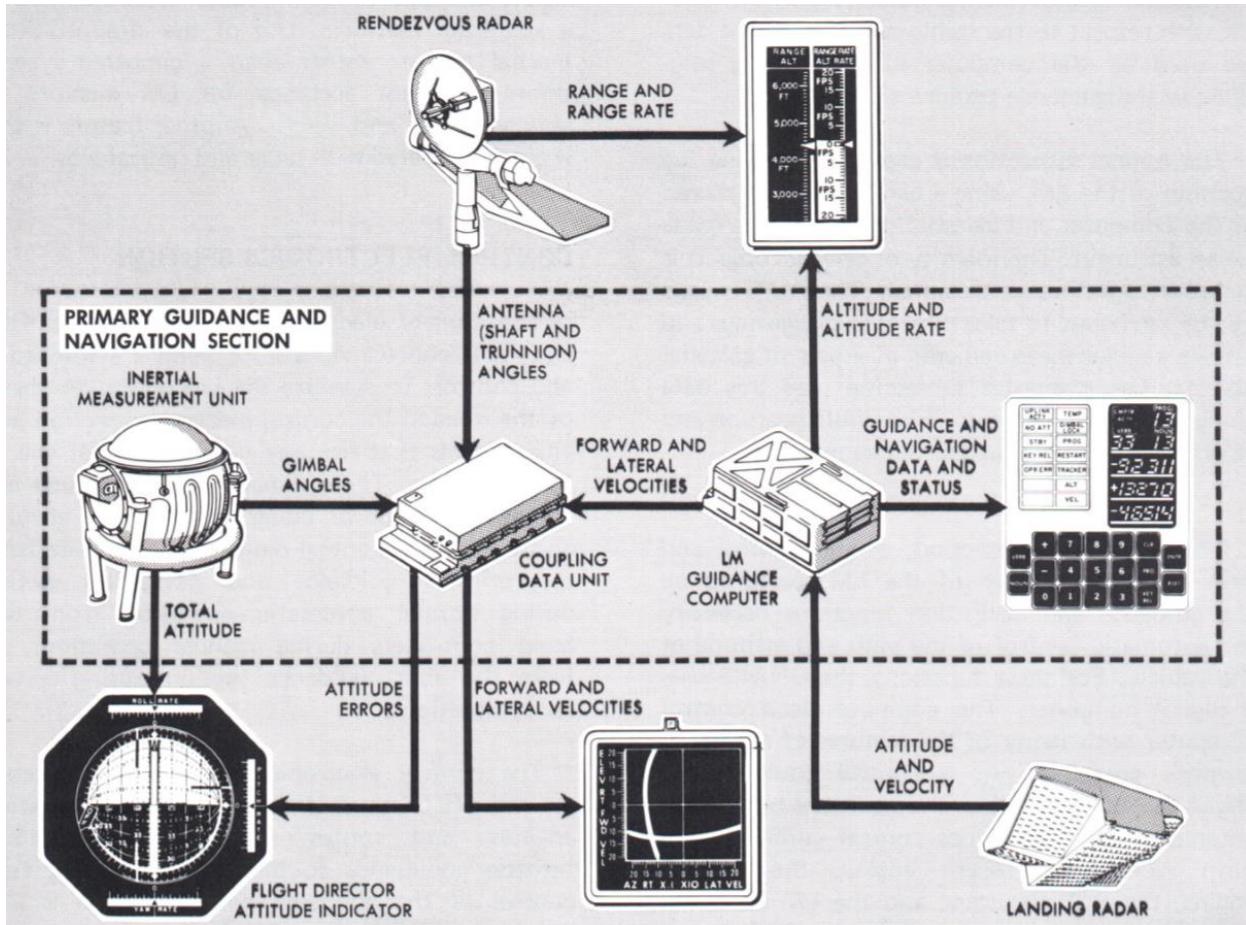


Figure 3.3.1.1 – Simplified PGNS path

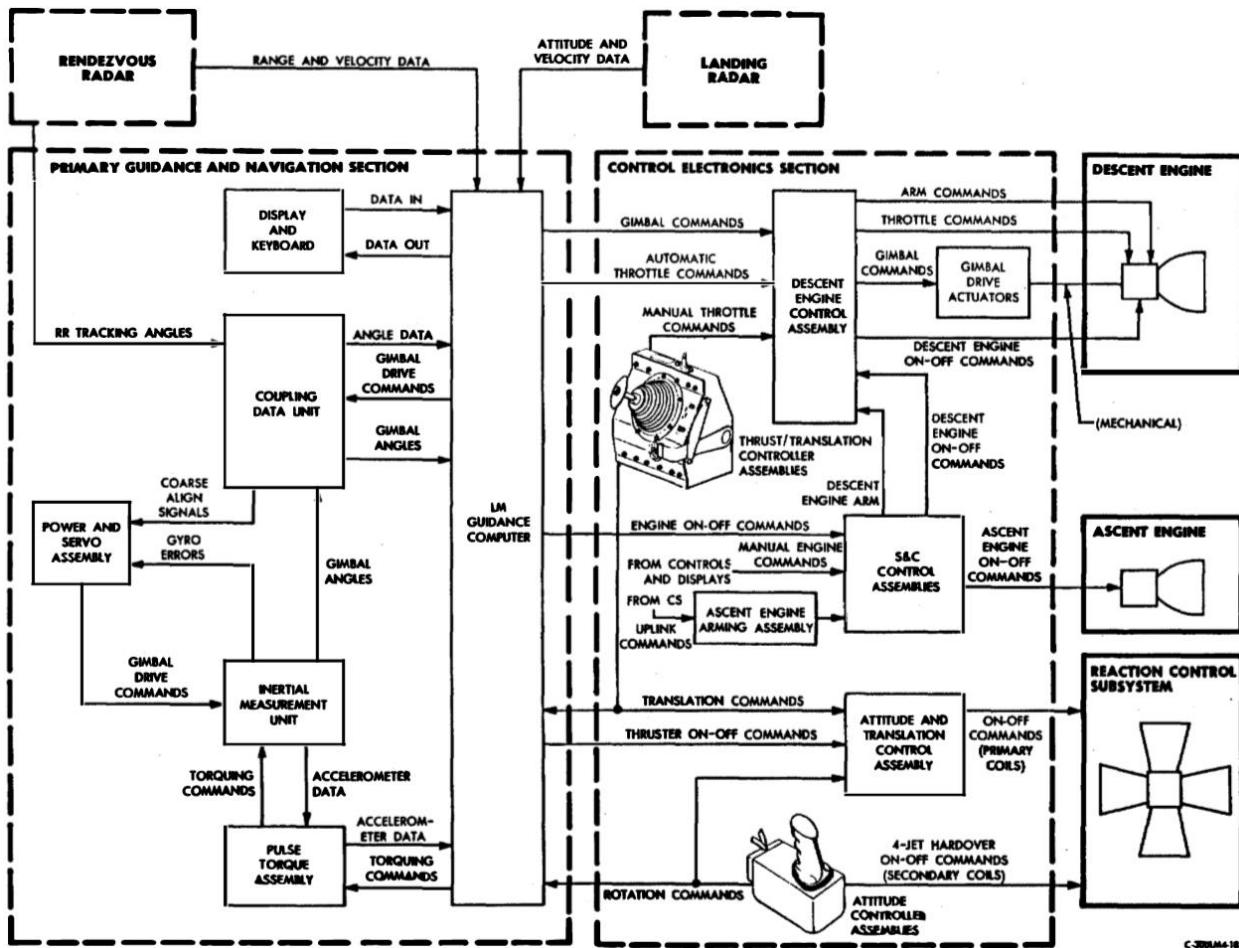


Figure 3.3.1.2 – Detailed PGNS path

### 3.2 ABORT GUIDANCE PATH

Note: AGS is not implemented in Reentry.

## 4. OPERATING THE PGNS

This discussion of PGNS operation is limited to astronaut interface with the PGNS, because PGNS operation is dependent upon the LGC program in process and upon the mission phase. The astronaut can perform optical sightings, monitor subsystem performance, load data, select the mode of operation, and, with the aid of the PGNS, manually control the LM. The program to be performed by the LGC is selected by the astronaut or initiated by the LGC.



PGNS is activated when the GUID CONT switch on Panel 1 is set to PGNS [A].

The DSKY enables the astronaut to communicate with the LGC and perform a variety of tasks. The hand controllers permit manual changes or computer-aided manual changes in attitude or translation. The PGNS flight data that are displayed to assist the astronaut during various phases of the mission are as follows: total LM attitude, attitude errors, altitude and altitude rate, forward and lateral velocities, and percentage of descent engine thrust commanded by the LGC.

Total attitude is generated from the IMU gimbal angles. With the ATTITUDE MON [B] switch (panel 1/2) set to PGNS, the IMU gimbal angles are converted to a proper configuration of total attitude, and routed to the FDD [C] sphere for direct astronaut readout. The FDD also displays roll, pitch, and yaw rates and errors. The FDD rate indicators monitor the rate of change of angular position.

When the RATE/ERR MON [D] switch (panel 1) is set to LDG RDR/CMPTR, the FDD error indicators indicate the deviation between the programmed and actual attitude.

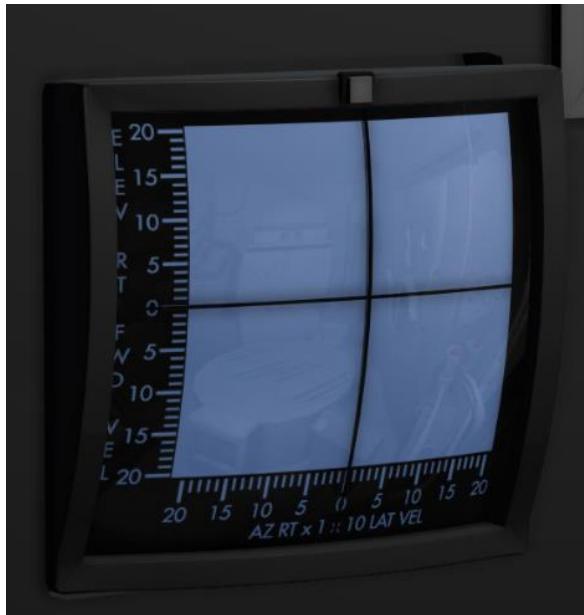
The astronauts can select the LR, PGNS, or AGS as the source for the altitude and altitude rate parameters. When the MODE SEL [E] switch is set to PGNS, the LGC calculates altitude and altitude rate, but issues signals for display of either altitude or altitude rate. Altitude and altitude rate are not displayed simultaneously. These signals are routed through the RNG/ALT MON switch [F] (panel 1) to the ALT and ALT RATE indicators [G] (panel 1).

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Forward and lateral velocities are displayed on the X-pointer indicator.



The indicator receives velocity signals from the LGC when the MODE SEL switch is set to PGNS. The LGC calculates the velocities from its stored information and from information received from the Landing Radar. The LGC feeds the calculated data to the CDU for digital-to-analog conversion before display. The X-POINTER SCALE switch (panel 3) selects the scale of the indicator. The type of velocity and the scale selected are indicated by illuminated placarding on the borders of the X-pointer indicator.

The amount of descent engine throttling, as commanded by the LGC, is routed to the CES.

The CES sends this command to the THRUST

indicator (panel 1) and to the descent engine.

The THRUST indicator also displays the amount of thrust sensed at the engine thrust chamber, so that a comparison can be made.

PGNS vehicle control includes interfacing for attitude and translation control and for propulsion control (descent and ascent engine). Commands from the LGC are routed through the CES to the RCS thrusters and to the ascent and descent engines for proper flight control.

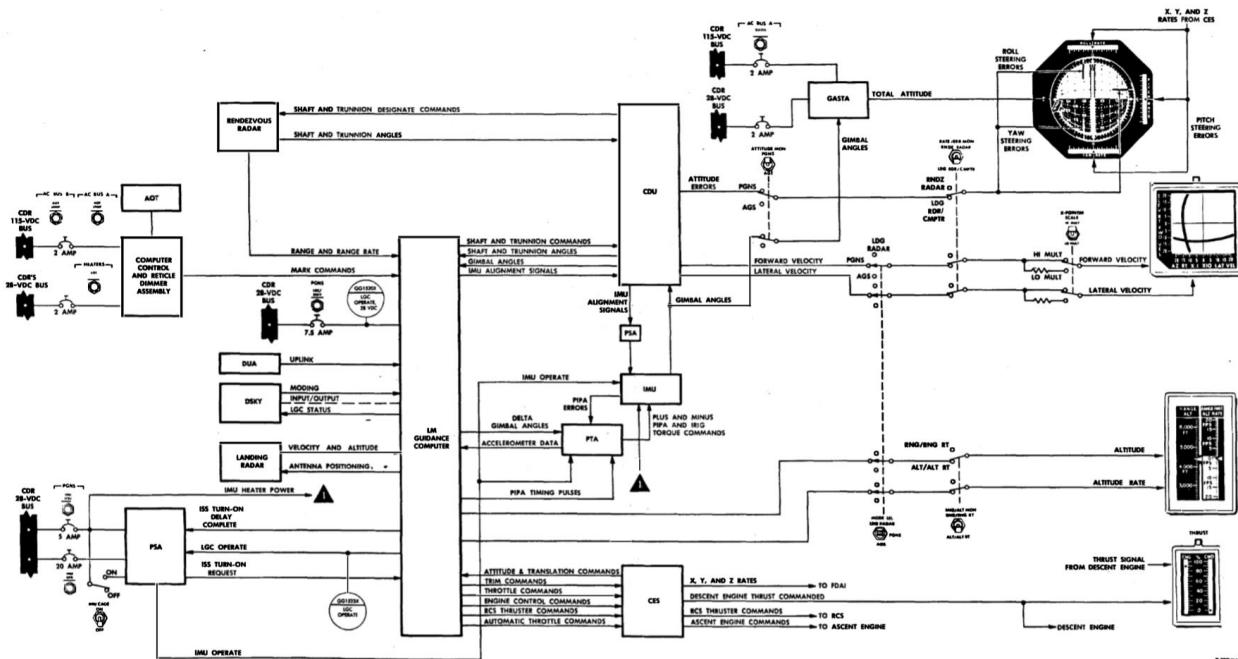


# LUNAR MODULE

## FLIGHT MANUAL

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AN ORBITAL SIMULATOR



## 5. OPERATING THE AGS

*Note: AGS is not yet implemented*

## 6. OPERATING THE CES

The PGNS, in conjunction with the CES, provides automatic control of LM attitude, translation, and descent or ascent propulsion maneuvers. Automatic control can be overridden by the astronauts, with manual inputs. As backup for PGNS control, the AGS, supplemented by manual inputs, can be used if the PGNS malfunctions. The following contains a summary of the CES modes of attitude control:

Mode	Switches and Positions	Guidance Signals	Manual Attitude Control	Manual Translation Control	Attitude Damping	Engine Gimbal Control	Remarks
Automatic (PGNS control)	MODE CONTROL: PGNS sw - AUTO GUID CONT sw - PGNS ATTITUDE CONTROL: ROLL, PITCH, and YAW sw - MODE CONT (normally)	Automatic steering and translation are performed by LGC commands to jet drivers.	N/A (See remarks for manual override)	Linear translation of LM by on-and-off firing of thrusters when TTCA is moved out of detent	Accomplished in LGC	Pitch and roll gimbal commands from LGC applied to DECA	All thruster commands from LGC go directly to primary preamplifiers. Attitude control function is overridden by operating ACA to hardover position, thereby commanding on-and-off four-jet operation through secondary coils of thruster solenoid valves. +X-axis translation is obtained by commanding four-jet operation direct to RCS secondary coils, by pressing +X TRANSL pushbutton on panel 5.
Attitude hold (PGNS control)	MODE CONTROL: PGNS sw ATT HOLD GUID CONT sw - PGNS ATTITUDE CONTROL: ROLL, PITCH, and YAW sw - MODE CONT (normally)	Stabilization is accomplished by LGC commands to jet drivers.	Attitude rate commands are proportional to ACA displacement. LM attitude is held to value when ACA is	Linear translation of LM by on-and-off firing of thrusters when TTCA is moved.	Accomplished in LGC	Pitch and roll gimbal commands from LGC applied to DECA	Same as for automatic mode (PGNS control). Minimum impulse mode is made available by entering command into DSKY. In this mode, LGC commands one RCS pulse each time ACA is moved past 2.5° nominally from detent.

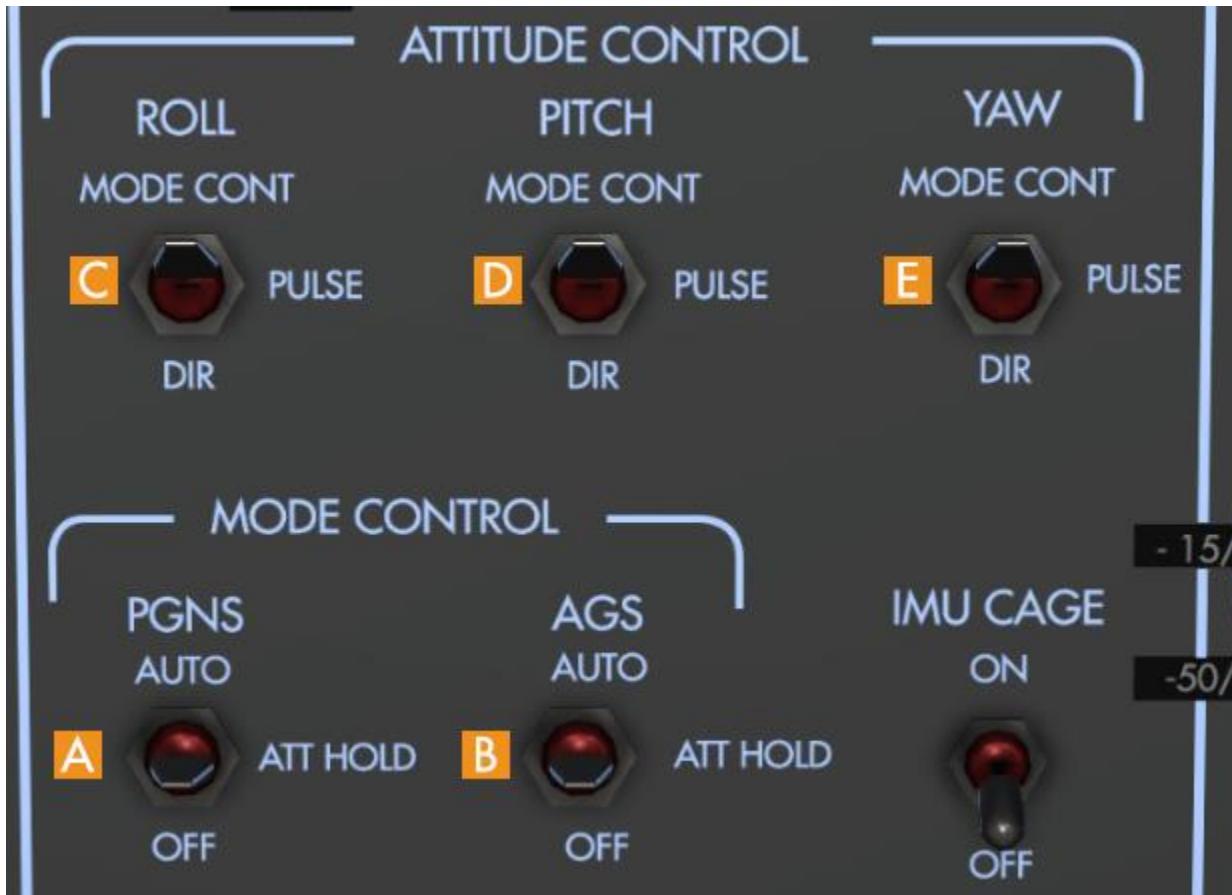
Pulse	MODE CONTROL: AGS sw - AUTO or ATT HOLD GUID CONT sw - AGS  ATTITUDE CONTROL: ROLL, PITCH, and YAW sw - PULSE (selected on individual-axis basis)	Abort guidance signals interrupted on individual-axis basis.	Astronaut commands angular acceleration through low-frequency pulsing of thrusters (two jets)	Translation commands along LM axes by on-and-off firing of thrusters when TTCA is moved out of detent.	No rate damping in axis selected.	No AGS control	Same as for automatic mode (AGS control)
Direct	MODE CONTROL: AGS sw - AUTO or ATT HOLD GUID CONT sw - AGS  ATTITUDE CONTROL: ROLL, PITCH, and YAW sw - DIR (selected on individual axis basis)	Abort guidance signals interrupted on individual-axis basis.	Astronaut commands angular acceleration through on-and-off firing of thrusters (two-jet operation direct to secondary coils)	Translation commands along LM axes by on-and-off firing of thrusters when TTCA is moved out of detent.	No rate damping in axis selected.	No AGS control	Same as for automatic mode (AGS control), except that attitude commands for selected axis are directly applied to RCS secondary coils

## 6.1. ATTITUDE CONTROL

LM attitude is controlled by X, Y, and Z axes. There are five modes of attitude control:

- Automatic
- Attitude hold
- Pulse
- Direct
- Hardover (manual override)

The image below shows the main attitude control switches, followed by a description of each.



The automatic and attitude hold modes are selected with the MODE CONTROL: PGNS or AGS switch [A]/[B]; the pulse and direct modes, with the ATTITUDE CONTROL: ROLL [C], PITCH [D], and YAW [E] switches.

Some of the rules below will only be applied when using a joystick, as the keyboard commands will only trigger to on or off command.

#### 6.1.1. AUTOMATIC MODE

The automatic mode provides fully automatic attitude control. It is set when MODE CONTROL [A] is set to AUTO, and the individual ROLL [C], PITCH [D], and YAW [E] switches set to MODE CONT.

During PGNS control, the LGC generates the required thruster commands and routes them to the ATCA. The jet drivers in the ATCA provide thruster on and off commands to selected RCS primary solenoids for attitude changes. The jet select logic determines which jets fire to correct the attitude errors. The astronaut can override attitude control about all three LM axes by initiating hardover commands with the ACA.

### 6.1.2. ATTITUDE HOLD MODE

This is a semiautomatic mode, in which either astronaut can command an attitude change at an angular rate proportional to ACA displacement. LM attitude is held when the ACA is in the detent (neutral) position.

It is set when the ROLL [C], PITCH [D], and YAW [E] switches is set to MODE CONT, and MODE CONTROL [A] is set to ATT HOLD.

In this mode, rate commands proportional to ACA displacement are sent to the LGC. The LGC operates on these commands and provides signals to the jet drivers in the ATCA to command rotation rates by means of the thrusters. When the ACA is returned to the neutral position, LM rotation stops and the LGC maintains the new attitude.

The LGC controls the signals, and can be set to either work in Minimum Impulse Command mode by keying V76 or Rate Command and Attitude Hold mode by keying V77. V76 uses pulses to move and hold attitude, while V77 uses rate command to move and hold attitude.

### 6.1.3. PULSE MODE

For minimum impulse control, the LGC commands a minimum impulse burn for each movement of the ACA beyond 2. 5° of the detent position. The ACA must be momentarily returned to the detent position between each impulse command; The maximum rate at which minimum impulses can be commanded is approximately five per second. In this mode, the astronaut performs rate damping and attitude steering. It is selected by setting the ROLL [C], PITCH [D], and YAW [E] switches is set to PULSE.

To change attitude in this mode, the ACA must be moved past 2. 5° from detent; this commands acceleration about the selected axis. To terminate LM rotation, an opposite acceleration about the same axis must be commanded.

### 6.1.4. DIRECT MODE

The direct mode is also an open-loop acceleration mode. It is selected on an individual-axis basis by setting the appropriate ATTITUDE CONTROL switch (ROLL [C], PITCH [D], or YAW [E]) to DIR.

Direct commands to two thrusters are routed to the RCS secondary solenoids when the ACA is displaced 2. 5°. The thrusters fire continuously until the ACA is returned to the detent position.

#### 6.1.4. HARDOVER MODE

In an emergency, the ACA can be displaced to the maximum limit (hardover position) to command an immediate attitude change about any axis. This displacement applies signals directly to the RCS secondary solenoids to fire four thrusters. This maneuver can be implemented in any attitude control mode. This is only possible if you are using a joystick as the input devices, and not the Keyboard.

#### 6.2. TRANSLATION CONTROL

Automatic and manual translation control are available in all three axes, using the RCS.

Automatic control consists of thruster commands from the LGC to the jet drivers in the ATCA. These commands are used for translations of small velocity increments and for ullage settling before ascent or descent engine ignition after coasting phases. Manual control in the primary guidance mode consists of on and off commands from a TTCA, through the LGC, to the primary preamplifiers.

Control consists of *on* and *off* commands from a TTCA to the jet selected logic in the ATCA. RCS thrust (+X-axis) is available when the +X TRANSL pushbutton is pressed. The secondary solenoids of the four downward-firing thrusters (B1D, A2D, B3D, A4D) are energized as the +X TRANSL pushbutton is pressed.

#### 6.3. DESCENT ENGINE CONTROL

Descent engine control accomplishes major changes in LM velocity.

##### 6.3.1. IGNITION AND SHUTDOWN

Descent engine ignition is controlled by the PGNS and the astronaut through the CES. Before ignition, the engine must be armed by setting the ENG ARM switch (panel 1) to DES. This action sends an engine arm discrete to the LGC and to the S&C control assemblies. Engine-on commands from the LGC are routed to the DECA through the S&C control assemblies. When it receives the engine arm and start discrete from the S&C control assemblies, the DECA commands the descent engine on. The engine remains on until an engine-off discrete is initiated with either stop push button located on panel 5 and 6. An engine-off discrete is generated when the  $\Delta V$  reaches a predetermined value. The astronauts can command the engine on or off, using the engine START (panel 5) and STOP pushbuttons.

### 6.3.2. THROTTLE CONTROL

Descent engine throttle (thrust) can be controlled by the PGNS and/or the astronauts. Automatic throttle (increase/decrease) signals from the LGC are sent to the DECA. The analog output of the DECA controls descent engine thrust from 10% to maximum thrust (92. 5%).

In the automatic mode (THR CONT switch set to AUTO), the astronauts can use the TTCA's to increase descent engine thrust. When the THR CONT switch is set to MAN, the astronaut has complete control over descent engine thrust. If a TTCA is used for throttle control, X-axis translation capability is disabled, as it is used to increase/decrease the engine throttle.

### 6.3.3. TRIM CONTROL

Descent engine trim is automatically controlled to compensate for center-of-gravity offsets during descent engine operation. The LGC routes trim on and off signals in two directions, for each gimbal axis, to the DECA.

## 6.4. ASCENT ENGINE CONTROL

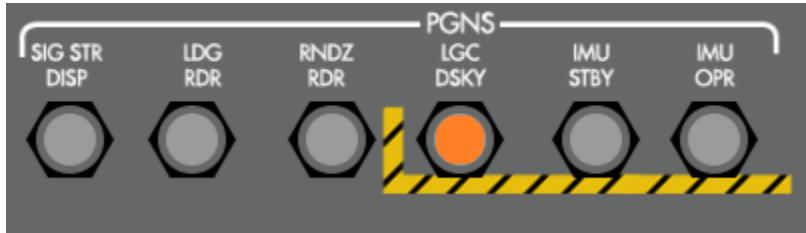
Ascent engine ignition and shutdown can be initiated by the PGNS or the astronaut. Automatic and manual commands are routed to the S&C control assemblies. These assemblies provide logically ordered control of LM staging and engine on and off commands. The S&C control assemblies provide a positive command for fail-safe purposes if the engine-on command is interrupted. In the event of an abort stage command while the descent engine is firing, the S&C control assemblies provide a time delay before commanding LM staging and ascent engine ignition. The time delay ensures that descent engine thrusting has completely stopped before the LM is staged.

## 7. POWER DISTRIBUTION

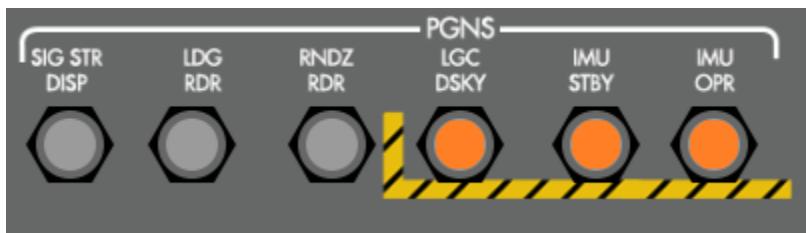
Each section of GN&CS receives its power independently of the other sections, from the CDR's and the LMP's buses through the circuit breakers on panels 11 and 16, respectively. The flight displays associated with the GN&CS receive power from CDR's a-c and d-c buses. When power is supplied to a particular display, a power-on indicator is energized. For the X-pointer, THRUST, RANGE, and RANGE RATE indicators, the power-on indicator is a lamp; for the FDAl's, talkbacks are used. The MISSION TIMER and the EVENT TIMER do not have power-on indicators.

### 7.1. PGNS POWER DISTRIBUTION

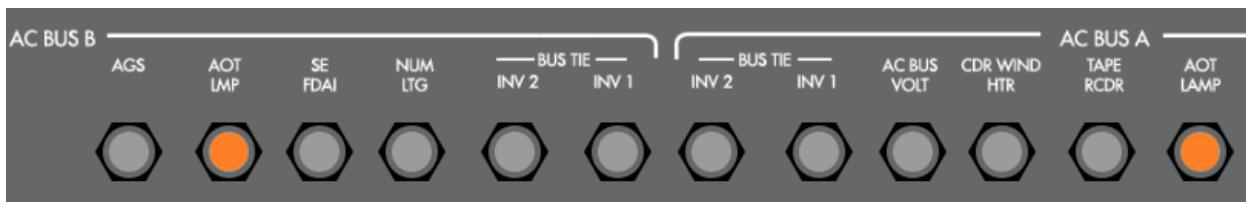
The LGC receives 28 volt d-c primary power from the PGNS: LGC/DSKY circuit breaker. When the LGC is put into standby mode, it is put into a low-power mode, where only the LGC timer and a few auxiliary assemblies are operative.



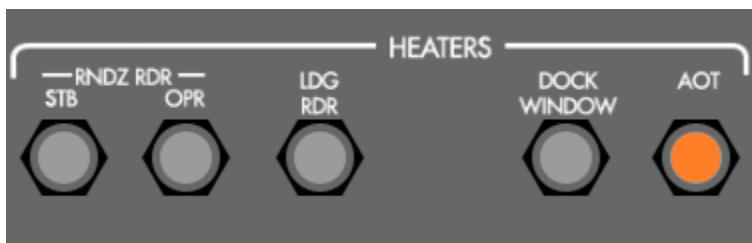
The power and servo assembly (PSA) and the IMU receives input power from the PGNS: LGC/DSKY, IMU STBY, and IMU OPR circuit breakers.



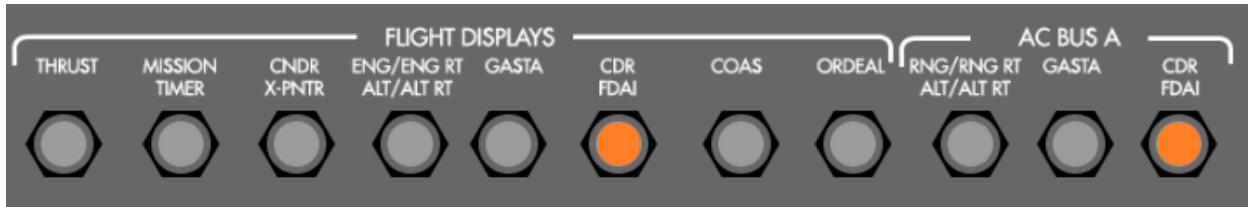
The AOT receives 115 volts a-c for illumination of the reticle, from the AC BUS A and the AC BUS B: AOT LAMP circuit breakers.



The heaters in the AOT receive power from the CDR's d-c bus through the HEATERS: AOT circuit breaker.

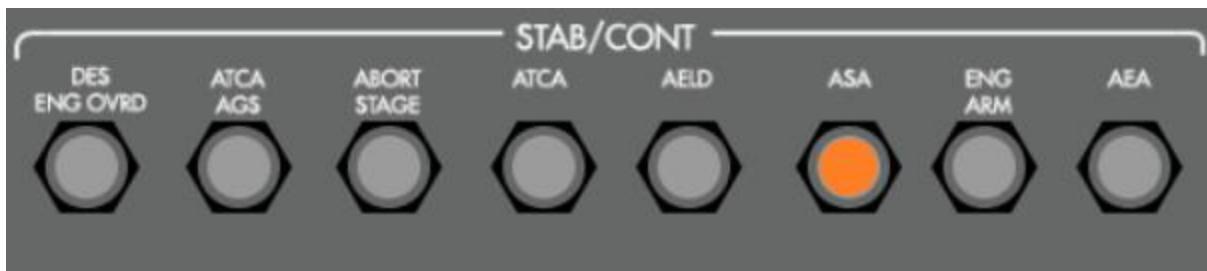


The Commanders FDAO receives power when the AC BUS A: CDR FDAO and FLIGHT DISPLAYS: CDR FDAO is closed.



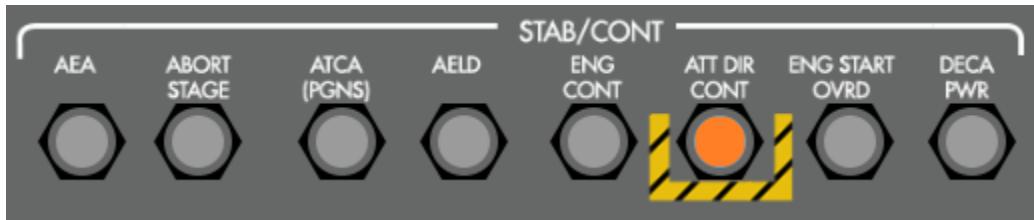
### 7.2. AGS POWER DISTRIBUTION

All power (ac and dc) required by the ASA is provided by the ASA power supply, which receives 28-volt d-e power from the STAB/CONT: ASA circuit breaker (panel 16).



### 7.3. CES POWER DISTRIBUTION

The CDR's and LMP's 28-volt d-c buses and the CDR's 115-volt a-c bus supply power to the CES. The ACA receives 28-volt d-c power from the CDR's bus for two-jet direct control through the STAB/CONT: ATT DIR CONT circuit breaker.



The ATCA primary power supply receives 28 volts d-c from the LMP's bus through the STAB/CONT: ATCA circuit breaker.



When the ATCA/AGS circuit breaker is on and GUID CONT switch is set to AGS and the MODE CONTROL: AGS switch is set to ATT HOLD or AUTO, the thruster drivers are enabled. When the BAL CPL switch is set to ON, the 28 volts from the circuit breaker enables the four upward-firing thrusters.

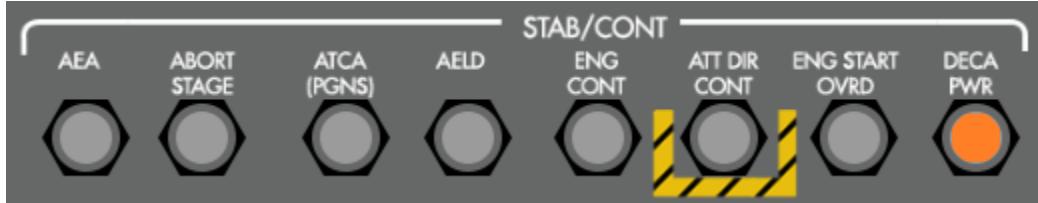
# LUNAR MODULE

## FLIGHT MANUAL

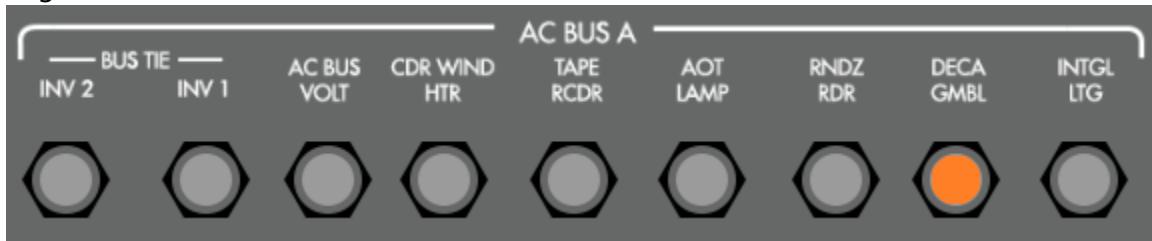
R E N T R Y

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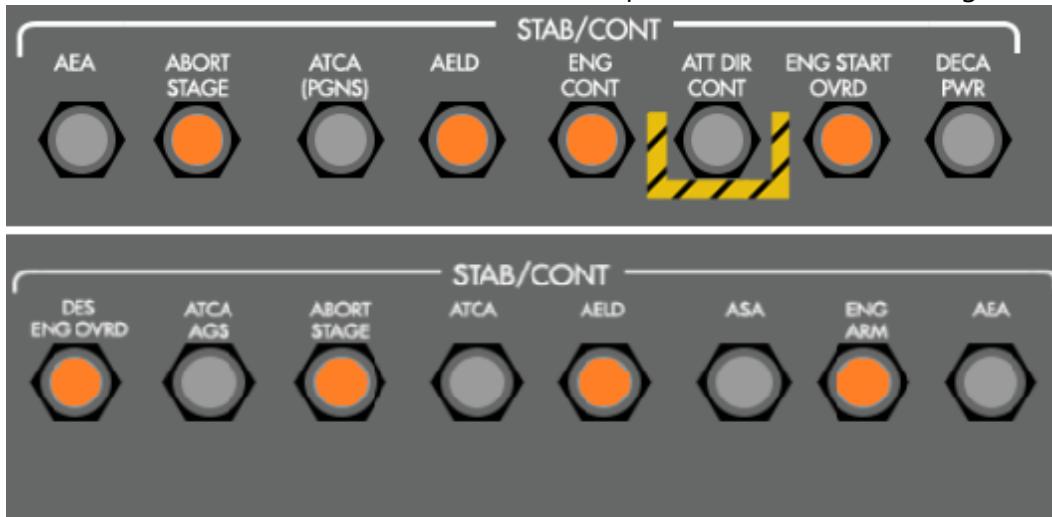
The STAB/CONT: DECA PWR circuit breaker supplies +28 volts d-c to the descent engine control circuit in the DECA. When the descent engine is armed, this input power is routed to the actuator isolation valves of the descent engine. The power supply of the DECA consists of a reference power supply and an auxiliary power supply.



The auxiliary power supply receives 400-cps power from the CDR's a-c bus through the AC BUS A: DECA GMBL circuit breaker. During an ATCA power failure, the auxiliary power supply provides +6 volts dc to the descent engine control circuit and enables full thrust of descent engine.

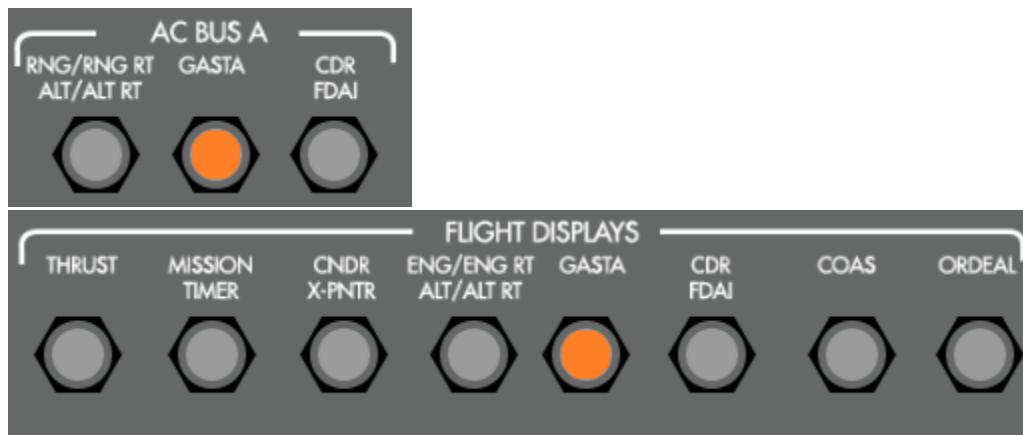


Power from the STAB/CONT: ENG CONT circuit breaker enables the engine control circuits in the DECA. This power is interrupted when the ABORT STAGE pushbutton is used or when the ABORT or STOP pushbuttons are used. The STAB/CONT: ENG START OVRD, AELD (2), ABORT STAGE (2), ENG ARM, and DES ENG OVRD circuit breakers are used in conjunction with the relay logic of the DECA and S&C control assemblies to accomplish ascent or descent engine control.



The Gimbal Angle Sequencing Transformation Assembly (GASTA) is used to translate the signals for engine gimbaling, and receives 115 volts ac from the CDR's a-c bus through the AC BUS A:

GASTA circuit breaker and 28 volts d-c from the CDR's d-c bus through the FLIGHT DISPLAYS: GASTA circuit breaker. These two Inputs energize the computer servo in the GASTA.



### 7.4. ORDEAL POWER DISTRIBUTION

The ORDEAL receives 115 volts ac from the CDR's a-c bus through the AC BUS B: ORDEAL circuit breaker and 28 volts de from the CDR's d-c bus through the FLIGHT DISPLAYS: ORDEAL circuit breaker.



## 8. PROCEDURES AND OPERATION

In the previous sections, we have learned how the systems operate from a functional perspective. In this section, we will dive deeper into how some of the major components are operated by the astronauts, focusing on procedures.

## 8.1. PGNS – INERTIAL SUBSECTION

### 8.1.1 GENERAL

ISS operation can be initiated automatically by the LGC, or manually by the astronaut using DSKY entries to command the LGC to select the various operating modes. The ISS status or mode of operation can be displayed on the DSKY, as determined by a computer program. The IMU furnishes the inertial reference; it consists of a stable member with three degrees of freedom, stabilized by three integrating gyros. The stable member must be aligned with respect to the reference coordinate system each time the ISS is powered up. The stable member must be realigned during flight because it may deviate from its alignment. Also, the crew may desire a new stable member orientation. The alignment orientation may be that of the CSM or that defined by the thrusting programs within the LGC. The stable member is aligned after the LGC processes sighting data that have been combined with the known IMU angles and supplies gyro-torqueing signals to the IMU.

Once the ISS is energized and aligned to an inertial reference, LM rotation is about the gimbaled stable member, which remains fixed in space. Resolvers, mounted on the gimbal axes, act as angle-sensing devices and measure LM attitude with respect to the stable member. These angular measurements are displayed to the astronaut by the flight director attitude indicator (FDI), and angular changes of the inertial reference are sent to the LGC.

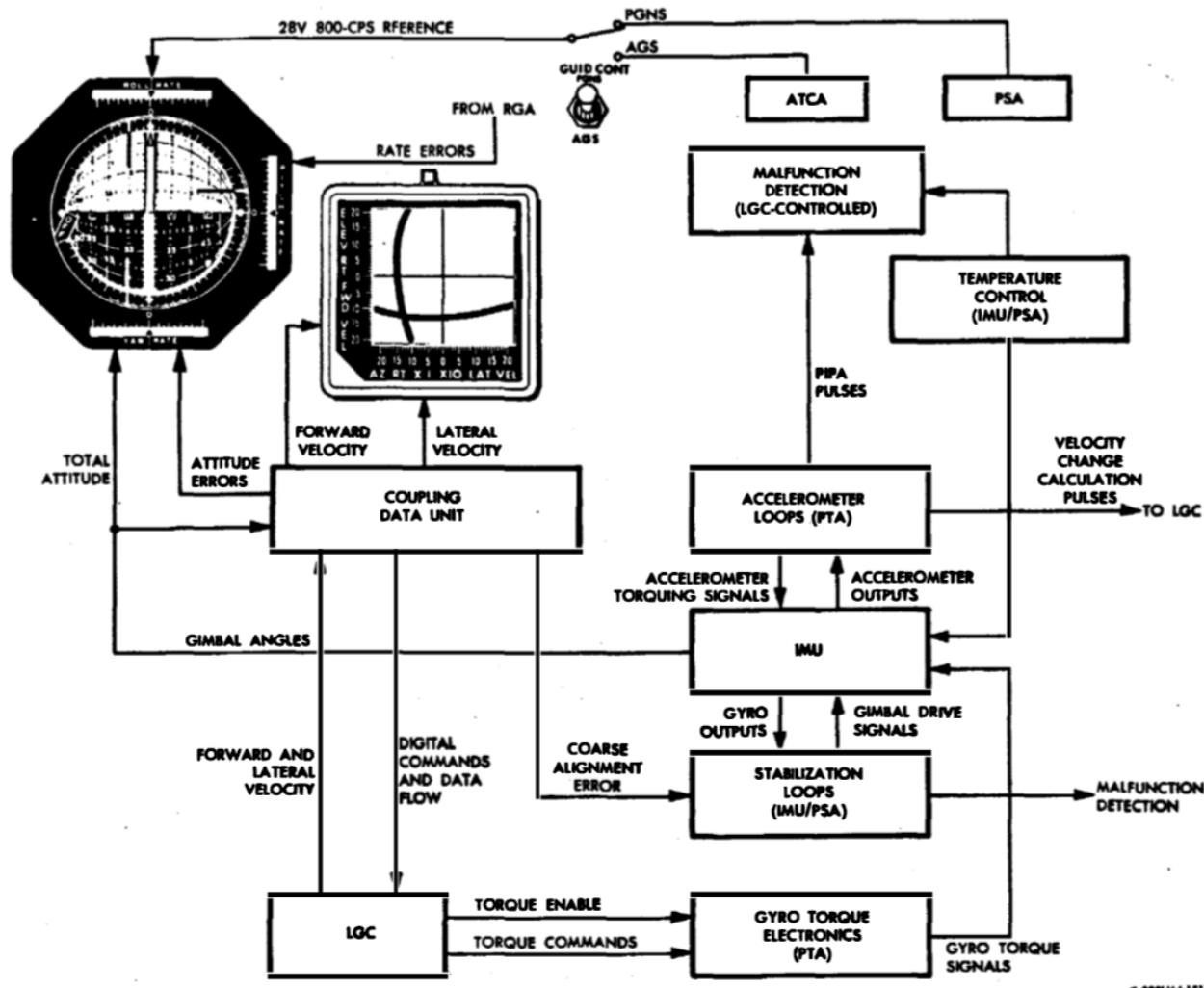
Desired LM attitude is calculated in the LGC and compared with the actual gimbal angles. A difference between the actual and calculated angles results in generation of attitude error signals, by the ISS channels of the CDU, which are sent to the FDI for display. These error signals are used by the digital autopilot program in the LGC to activate RCS thrusters for LM attitude correction.

LM acceleration due to thrusting is sensed by three PIPA's, which are mounted on the stable member with their input axes orthogonal. The resultant signals (velocity changes) from the accelerometer loops are supplied to the LGC, which calculates the total LM velocity.

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### 8.1.2. INERTIAL MEASUREMENT UNIT

The IMU uses three Apollo 25-inertial reference integrating gyros (IRIG's) to sense changes in stable member orientation, and three 16-pulse integrating pendulous accelerometers (PIPA's) to sense velocity changes. A temperature alarm circuit monitors the temperature control assembly. The alarm circuit contains an alarm thermostat or high-temperature sensing ( $+134^\circ \pm 0.2^\circ$  F), an alarm thermostat for low-temperature sensing ( $+126^\circ \pm 0.2^\circ$  F), and a temperature alarm module that provides a discrete to the LGC during normal-temperature operation ( $+126^\circ$  to  $+134^\circ \pm 0.2^\circ$  F). When an out-of-limit temperature occurs on the DSKY, a warning light goes on. The heat control is based on the glycol loop.

The IMU can be caged to nullify and reset the gimbals, holding them in place on the 0,0,0 position.

### 8.1.3. MODES OF OPERATION

Except for the IMU cage and inertial reference modes, the modes are controlled by the CDU as commanded by the LGC. The IMU cage mode is initiated when the IMU CAGE switch (panel 1) is set to ON. The inertial reference mode is entered automatically whenever the ISS is not in another mode.

#### 8.1.3.1. IMU TURN-ON MODE

The IMU turn-on mode initializes ISS operation by driving the IMU gimbals to zero and clearing and inhibiting the CDU read and error counters. The IMU turn-on mode (program controlled) is initiated by applying IMU operate power to the ISS. The LGC issues the two discretes required for this mode: CDU zero and coarse align enable. The LGC also issues the turn-on delay complete discrete to the ISS after 90 seconds.

When IMU power is applied to the ISS, the LGC receives an ISS power-on discrete and a turn-on delay request. The LGC responds to the turn-on delay request by issuing the CDU zero and coarse-align enable discretes to the CDU. To prevent PIPA torqueing for 90 seconds during the IMU turn-on mode, the IMU is automatically cage during this time-delay.

After the 90-second delay, the LGC program removes the CDU zero and coarse-align enable discretes, allowing the ISS to go to the inertial reference mode (coarse-alignment relay deenergized), or it can remove the CDU zero discrete and provide an error-counter enable discrete while maintaining the coarsealign enable discrete. The latter combination of discretes defines the coarse-alignment mode of operation.

#### 8.1.3.2. COARSE-ALIGNMENT MODE

The coarse-alignment mode enables the LGC to align the IMU rapidly to a desired position, with limited accuracy. The gimbals then drive in the direction commanded by the LGC. It is used to quickly set the coarse attitude relative to the platform. This operation usually takes up to 10 seconds, and is initiated by the LGC.

#### 8.1.3.3. FINE-ALIGNMENT MODE

The fine-alignment mode allows the LGC to position the IMU accurately to a predetermined gimbal angle. In this mode, the LGC is issuing fine-alignment pulses to drive minor changes in the IMU. When the LGC is not issuing fine-alignment pulses, the stable member can be considered inertially referenced. This operation can take a few minutes.

#### 8.1.3.4. INERTIAL REFERENCE MODE

Inertial reference is considered a mode of ISS operation during any period after IMU turn-on is completed and the stabilization loops are closed (coarse-alignment relay deenergized) without any gyro-torqueing occurring. The IRIG's hold the stable member inertially referenced, and the reference can be displayed on the FDAO. The ISS is considered to be in the inertial reference mode of operation during any period after IMU turn-on is completed during which the ISS is not in any other of its modes.

#### 8.1.3.5. IMU CAGE MODE

The IMU cage mode is an emergency mode that enables the astronauts to recover a tumbling IMU by setting the gimbals to zero, and to establish an inertial reference. This mode can also be used to establish an inertial reference when the LGC is not activated. The IMU cage mode is initiated by setting the IMU CAGE switch to ON to allow the IMU gimbals to settle at the zero position. The IMU gimbal zeroing can be observed on the FDAO.

If the mode is commanded to recover a tumbling IMU after the IMU turn-on mode is completed or to establish an inertial reference with the CSS in standby or off, setting the IMU CAGE switch to ON drives the IMU gimbals to zero. When the switch is released, the ISS enters the inertial reference mode.

The IMU cage mode should not be used indiscriminately. The mode is intended only as an emergency recover function for a tumbling IMU.

### 8.2. PGNS – OPTICAL SUBSECTION

*Note: There is no need to use the Lunar Module optics in Reentry yet, as the IMUs are not generating drift errors. I will leave this section here as a reference. When a new IMU platform is selected from P52 or in-game events, all drift-errors are removed, and the attitude is updated in the IMU.*

#### 8.2.1 GENERAL

The AOT is used by the astronaut to take direct visual sightings and precision angular measurements of a pair of celestial objects. These measurements are transferred to the LGC. The LGC uses this angular information along with the prestored data to compute the LM position and velocity and to perform the fine alignment of the IMU stable member.

### 8.2.2 ALIGNMENT OPTICAL TELESCOPE

The AOT, mounted on the navigation base to provide a mechanical alignment and a common reference between the AOT and IMU, is a unity-power, L-shaped periscope-type device with a 60° conical field of view. The AOT has a movable shaft axis (parallel to the LM X-axis) and a line-of-sight axis (approximately 45° from the X-axis).

The eyepiece optics section is the assembly through which the astronaut views the images of the stars on the reticle. The AOT reticle pattern consists of crosshairs and a pair of Archimedes spiral lines. The vertical crosshair, an orientation line designated the Y-line, is parallel to the LM X-axis when the reticle is at the 0° reference position. The vertical crosshair (upper quadrant) is a pair of radial lines that facilitate accurate superimposition of target stars between them. The horizontal crosshair, designated the X-line, is perpendicular to the orientation line. The pair of spiral lines are one-turn spirals, originating from the center of the reticle and terminating at the top of the vertical crosshair. Stars will appear white, reticle imperfections, red.

### 8.2.3 OPERATING THE OPTICAL SUBSECTION

The OSS is used for manual star sightings, which are necessary for accurate determination of the inertial orientation of the IMU stable member. These star sightings are required during certain periods while the LM is in flight. There are two methods for using the OSS.

#### 8.2.3.1 IN-FLIGHT SIGHTINGS

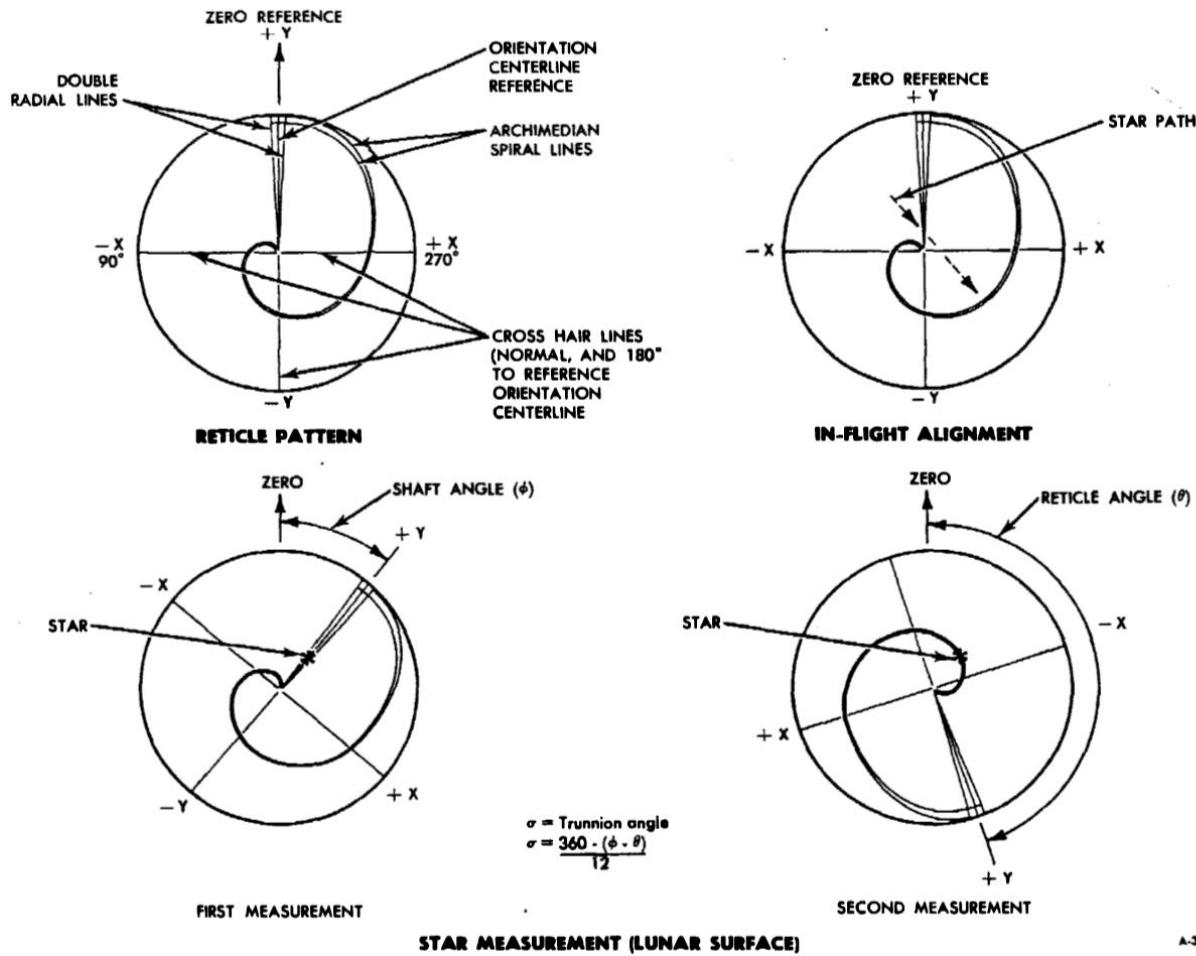
For in-flight operation, the CSS and the ISS are turned on, the AOT counter is zeroed, a detent position is selected, and the LM is maneuvered using the RCS system to obtain a selected star in the AOT field of view, near the center. The specific detent position code and selected star code are entered into the LGC via the DSKY. The LM is then maneuvered so that the star image crosses the reticle crosshairs. When the star image is coincident with the Y-line, the astronaut presses the MARK Y pushbutton; when it is coincident with the X-line, he presses the MARK X pushbutton. The astronaut may do this in either order and, if desired, he may erase the latest mark by pressing the REJECT pushbutton. When the MARK X or MARK Y pushbutton is pressed, a discrete is sent to the LGC. The LGC then records the time of mark and the IMU gimbal angles at the instant of the mark. Crossing of a reticle crosshair line by the star image defines a plane containing the star. Crossing of the other reticle crosshair line defines another plane containing the same star. The intersection of these planes forms a line that defines the direction of the star. To define the inertial orientation of the stable member, sightings on at least two stars are required. Each star sighting requires the same procedure. Multiple reticle crossings and their corresponding marks can be made on either or both stars to improve the accuracy of the sightings. Upon completion of the second star sightings, the LGC calculates the orientation of the stable member with respect to a predefined reference coordinate system.

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### 8.2.3.2 LUNAR SURFACE SIGHTINGS

On the lunar surface, the LM cannot be maneuvered to obtain a star-image crossing on a reticle crosshair line. The astronaut, using the manual reticle control knob, adjusts the reticle to superimpose the target star between the two radial lines on the reticle. The angle (star shaft angle,  $A_s$ ) displayed on the AOT counter is then inserted into the LGC by a DSKY entry. The astronaut next rotates the reticle until the same target star is superimposed between the two spiral lines on the reticle. This provides a second angular readout (reticle angle,  $A_r$ ), which is inserted into the LGC by a DSKY entry. The AOT detent position and the star code numbers are also inserted into the LGC. The LGC can now calculate the angular displacement of the star from the center of the field of view by computing the difference between the two counter readings. Due to the characteristics of the reticle spiral, this angle ( $A_r - A_s$ ) is proportional to the distance of the star from the center of the field of view. Using this angle and the proportionality equation, the LGC can calculate the trunnion angle ( $A_t$ ). At least two star sightings are required for determination of the inertial orientation of the stable member.

### 8.3. PGNS – COMPUTER SUBSECTION

#### 8.3.1 GENERAL

The Computer Subsection (CSS) is the control and processing center of the PGNS. It consists of the LGC and the DSKY. The CSS processes data and issues discrete outputs and control pulses to the PGNS, AGS, CES, and to other LM subsystems.

#### 8.3.2 LM GUIDANCE COMPUTER

The LGC contains a timer, sequence generator, central processor, priority control, an input-output section, and a memory. The main functions of the LGC are implemented through execution of programs stored in memory. Programs can be selected by the astronauts, and manual operation is mainly done through the DSKY. The computer uses VERBs and NOUNs to select the mode of operation, and is similar to the Apollo Guidance Computer when it comes to operations.

#### 8.3.3 OPERATING THE COMPUTER

The operator of the DSKY can communicate with the LGC by pressing a sequence of push buttons on the DSKY keyboard. The LGC can also initiate a display of information or request the operator for some action, through the processing of its program. The Lunar Guidance Computer chapter will cover this in-depth.



## IV. LUNAR GUIDANCE COMPUTER

## IV. LUNAR GUIDANCE COMPUTER

### 1. GENERAL

The computer is the control and data-processing center of the vehicle that performs all the guidance and navigation functions necessary for automatic control of the flight path and attitude of the vehicle. As a control computer, it aligns the stable platform, and positions both radar antennas. It also provides control commands to both radars, the ascent engine, the descent engine, the RCS thrusters, and the vehicle cabin displays. As a general-purpose computer, it solves guidance problems required for the mission. The CSS consists of a LM guidance computer (LGC) and a display and keyboard (DSKY), which is a computer control panel.

The LGC and its programming help meet the functional requirements of the mission. The functions performed in the various mission phases include automatic and semiautomatic operation that are implemented mostly through the execution of the programs stored in the LGC memory.

Through the DSKY, the astronauts can load information into the LGC, retrieve and display information contained in the LGC, and initiate any program stored in memory.

The DSKY is located on panel 4, between the Commander and LM Pilot and above the forward hatch. The upper half is the display portion; the lower half comprises the keyboard. The display portion contains five caution indicators, six status indicators, seven operation display indicators, and three data display indicators. These displays provide visual indications of data being loaded in the LGC, the condition of the LGC, and the program being used. The displays also provide the LGC with a means of displaying or requesting data.

In short, the Lunar Guidance Computer is a control computer located in the Lunar Module, and is operated through a Display and Keyboard panels (DSKY). The computer is generally referenced to as the Lunar Guidance Computer (LGC).

The LGC can run one major mode (mission program) at a time. It is controlled by using something called VERBs and NOUNs. A VERB is an action that is to be performed, like changing the mission program, monitor data, change data etc. A NOUN is the location or register the VERB (action) is being performed. For example, if the flight crew wish to run major mode 63, they enter VERB 37 that means ACTION: CHANGE PROGRAM (MAJOR MODE), and NOUN 63 that means TO PROGRAM 63. This will basically run program 63, the program used to initialize lunar descent.

The LGC contains a timer, sequence generator, central processor, priority control, an input-output section and a memory. The main functions of the LGC are implemented through execution of programs stored in memory. Programs are written in a machine language called

basic instructions. A basic instruction can be an instruction word or a data word. All words for the LGC are 16 bits long.

## 2. DISPLAY AND KEYBOARD

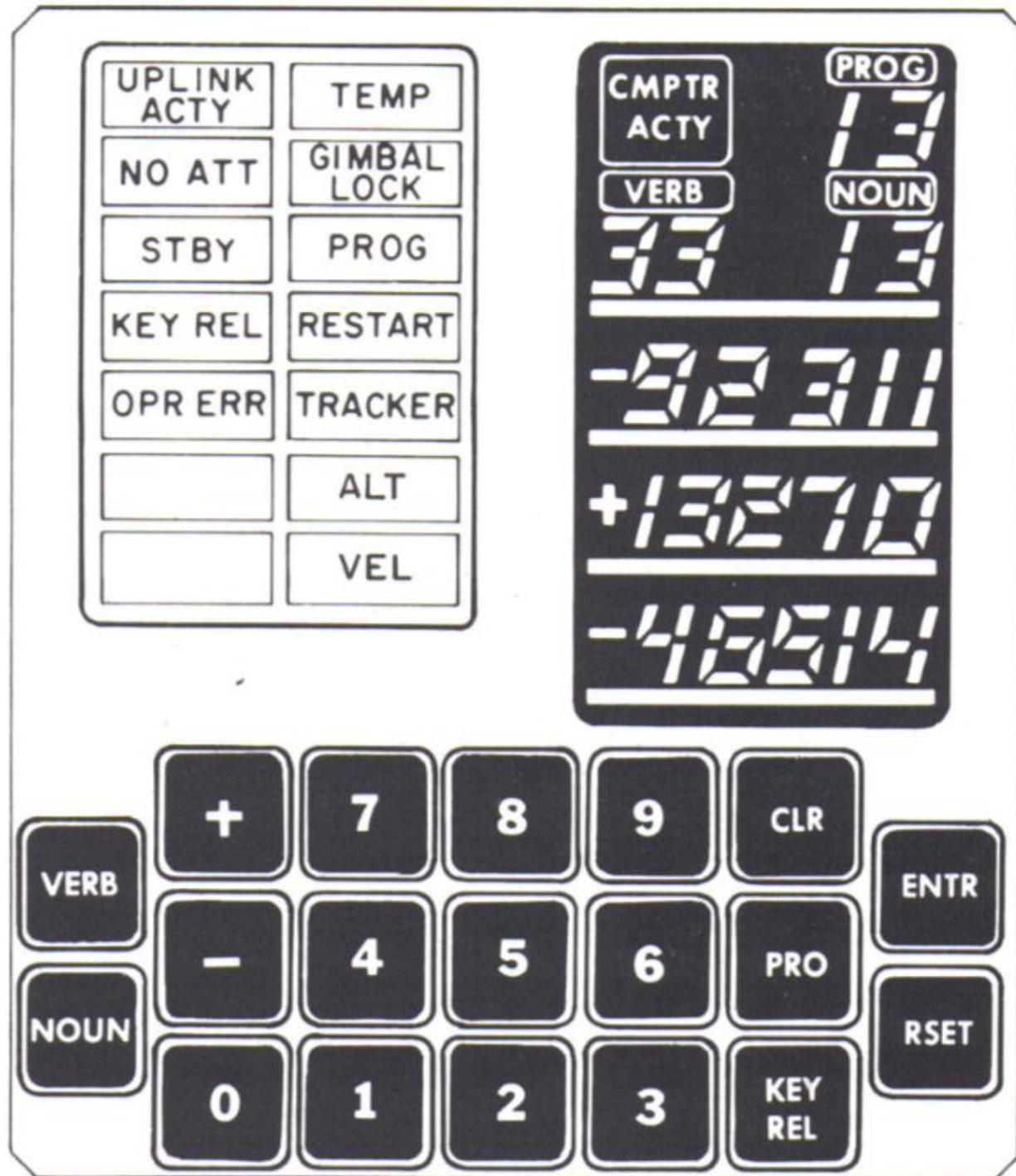
The DSKY facilitate intercommunication between the flight crew and the LGC.

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The DSKYs have buttons for input, warning lights and a display. Below is a table with a short description of each component.

DISPLAY

COMP ACTY light	When illuminated, the CMC is occupied with an internal sequence.
PROG display	Shows the current major mode (mission program) running on the computer.
VERB display	Shows the current VERB being performed.
NOUN display	Shows the current NOUN where the action (VERB) is being performed.
REGISTER 1, 2 and 3	Three registers are used to display data from a register or memory location. Different VERBs and NOUNs are used to show content in the registers.

## INDICATORS

UPLINK ACTY light	Shows when the LGC received data remotely from for example mission control.
NO ATT light	Illuminated when the LGC don't have an attitude from the IMU. This usually happens when the IMU is in a coarse align mode.
STBY light	On when the LGC is in standby mode
KEY REL light	Illuminates when the astronaut is using the DSKY, the computer enters a mode where it won't interrupt the input. If the LGC desires to display data during this, they KEY REL light will illuminate/flash. This means the LGC has data to show, but is waiting for you to complete your session.
OPR ERR light	Illuminates when an input error has happened, like improper sequence of key depressions.
TEMP light	Illuminated when the IMU is outside its operational temperature range of 126.3 to 134.3°F
GIMBAL LOCK	Illuminated when the middle gimbal (yaw) exceed +/- 70 degrees from its zero position. This can require a realignment of the stable platform.
PROG light	Illuminated when the internal program detects computational difficulties.
RESTART light	Illuminates when the CMC detects a temporary hardware or software failure.
TRACKER light	Illuminated when the LGC receives a signal from the optics indicating a failure.

ALT  
VEL

## KEYBOARD

Numeric keys	Ten numeric keys (0 – 9) is used to enter numbers.
+ and -	Sets the sign before a numerig register can be entered.
NOUN	The next two numeric inputs will be for a NOUN.
VERB	The next two numeric inputs will be for a VERB.
CLR	Clears the current active input field. Used if a numeric input mistake was made. If NOUN is active, depressing this will clear the NOUN fields and position the input position to the beginning, same with a REGISTER etc.
ENTR	Used to ENTER data into the registers, NOUN or VERB fields. Will execute the requested function.
PRO	Proceed in a program, with or without data, and go to the next program step.
KEY REL	Will release the display from the flight crew to the computer when the computer wish to display data (KEY REL light)
RSET	Extinguishes the DSKY caution indicators OPR, ERR, PROG, RESTART, STBY and UPLINK ACTY).

## 3. VERBS

Verbs and Nouns are used to talk with the computer, where each word is a digit. A verb code decides what action is to be taken.

## A DESCRIPTION OF EACH VERB

VERB CODE	DESCRIPTION
01	Display Octal Component 1 in R1
04	Display Octal Components 1, 2 in R1, R2
05	Display Octal Components 1, 2, 3 in R1, R2, R3

- 06 Display decimal in R1 or R1, R2 or R1, R2, R3
- 11 Monitor Octal Component 1 in R 1
- 14 Monitor Octal Components 1, 2 in R 1, R2
- 15 Monitor Octal Components 1, 2, 3 in R1, R2, R3
- 16 Monitor decimal in R1 or R1, R2 or R1, R2, R3
- 21 Load Component 1 into R1
  - Allows the flight crew to enter data into register 1. The noun decides what memory location is bound to this register. ENTR will commit the data.
- 22 Load Component 2 into R2
  - Allows the flight crew to enter data into register 2. The noun decides what memory location is bound to this register. ENTR will commit the data.
- 23 Load Component 3 into R3
  - Allows the flight crew to enter data into register 3. The noun decides what memory location is bound to this register. ENTR will commit the data.
- 24 Load Component 1,2 into R1, R2
  - Allows the flight crew to enter data into register 1 and 2. The noun decides what memory location is bound to this register. ENTR will commit the data.
- 25 Load Component 1,2,3 into R1, R2, R3
  - Allows the flight crew to enter data into register 1, 2, and 3. The noun decides what memory location is bound to this register. ENTR will commit the data.
- 34 Terminate function
- 35 Test lights
- 36 Request FRESH START
- 37 Change program (major mode)
- 48 Request DAP Data Load routine ( R03)
- 49 Request Crew Defined Maneuver routine (R62)
- 50 Please perform
- 56 Terminate tracking (P20 and P25)
- 57 Permit Landing Radar updates
- 58 Inhibit Landing Radar updates
- 60 Display vehicle attitude rates on FDAO error needles
- 61 Display DAP following attitude errors
- 62 Display total attitude errors with respect to N22
- 75 Backup liftoff
- 76 Minimum Impulse Command mode
- 77 Rate Command and Attitude Hold mode
- 80 Enable LM state vector update
- 81 Enable CSM state vector update
- 95 No update of either state vector allowed (P20 or P22)
- 96 Terminate integration and go to POO
- 99 Please enable Engine Ignition

## 4. NOUNS

A noun refers to locations, registers, devices or informations used/needed by the verb. Registers are used to display information related to the noun. A noun can be made up of up to three components. These are displayed in the three registers. A verb can be used to request an action to enter data into each or all of the registers.

### A DESCRIPTION OF EACH NOUN

<b>NOUN CODE</b>	<b>DESCRIPTION</b>	<b>FORMAT</b>
01	Specify address (fractional)	XXXXX fractional .XXXXX fractional .XXXXX fractional
06	Option code ID	Octal
	Option code	Octal
	Data code	Octal
09	Alarm codes	
	First	Octal
	Second	Octal
	Last	Octal
18	Desired automaneuver FDAO ball angles	R XXX.XX deg P XXX.XX deg Y XXX.XX deg
20	Present ICDU angles	OG XXX.XX deg IG XXX.XX deg MG XXX.XX deg
22	Desired ICDU angles	OG XXX.XX deg IG XXX.XX deg MG XXX.XX deg
25	CHECKLIST (used with N25)	XXXXX.
33	Time of ignition (GETI)	00XXX. h 000XX. min 0XX.XX s
40	Time from ignition/cutoff (TFI/TFC)	XXbXX min/s
	VG	XXXX.X ft/s
	Delta V (accumulated)	XXXX.X ft/s
42	Apocenter altitude	XXXX.X nmi
	Pericenter altitude	XXXX.X nmi
	Delta V (required)	XXXX.X ft/s
43	Latitude	XXXX.X nmi
	Longitude	XXXX.X nmi
	Altitude	XXXX.X ft/s

44	Apocenter altitude	XXXX.X nmi
	Pericenter altitude	XXXX.X nmi
	TFF	XXbXX min/s
45	Marks	XXXXX. marks
	Time from ignition of next burn	XXbXX min/s
	Middle gimbal angle	XXX.XX deg
46	DAP configuration	Octal
	Switch function fail code	Octal
47	LM weight	XXXXX. lb
	CSM weight	XXXXX. lb
49	Delta R	XXXX.X nmi
	Delta V	XXXX.X ft/s
	Radar data source code	0000X
54	Range	XXX.XX nmi
	Range rate	XXXX.X ft/s
	Theta	XXX.XX deg
60	Forward velocity	XXXX.X ft/s
	Altitude rate	XXXX.X ft/s
	Computed altitude	XXXXX. ft
61	Time to go in braking phase	XXbXX min/s
	Time from ignition	XXbXX min/s
	Crossrange distance	XXXX.X nmi
62	Absolute value of velocity	XXXX.X ft/s
	Time from ignition	XXbXX min/s
	Delta V (accumulated)	XXXX.X ft
63	Delta Altitude (+LR > LGC)	XXXXX. ft
	Altitude rate	XXXX.X ft/s
	Computed altitude	XXXXX. ft
64	Time left for redesignations (TR)/LDP	XXbXX seconds/deg
	Altitude rate	XXXX.X ft/s
	Computed altitude	XXXXX. ft
65	Sampled LGC time (fetched in interrupt)	00XXX. h 000XX. min 0XX.XX s
72	RR trunnion angle	XXX.XX deg
	RR shaft angle	XXX.XX deg
74	Time from ignition	XXbXX min/s
	Yaw after vehicle rise	XXX.XX deg
	Pitch after vehicle rise	XXX.XX deg
76	Desired downrange velocity	XXXX.X ft/s
	Desired radial velocity	XXXX.X ft/s
	Crossrange distance	XXXX.X nmi
81	Delta VX (LV) (+Fwd)	XXXX.X ft/s

	Delta VY (LV) (+Rt)	XXXX.X ft/s
	Delta VZ (LV) (+Dn)	XXXX.X ft/s
83	Delta VX (body) (+Up)	XXXX.X ft/s
	Delta VY (body) (+Rt)	XXXX.X ft/s
	Delta VZ (body) (+Fwd)	XXXX.X ft/s
85	VGX (body) (+Up)	XXXX.X ft/s
	VGY (body) (+Rt)	XXXX.X ft/s
	VGZ (body) (+Fwd)	XXXX.X ft/s
92	Percent of full thrust (10,500 lb)	00XXX %
	Altitude rate	XXXX.X ft/s
	Computed altitude	XXXXX. ft
94	VGX (LM) (+Up)	XXXX.X ft/s
	Altitude rate	XXXX.X ft/s
	Computed altitude	XXXXX. ft

## 5. OPERATION

The VERB and NOUN combinations are used to operate the computer. Referring to the lists above, you can start talking with the computer using its language.

For example, if you wish to display a decimal in the register(s) from a noun, you use Verb 06. If you wish to montito a decimal in the register(s), use Verb 16. If you wish to set up the DAP configuration, insert Verb 48 and press ENTER (V48E).

To let the computer know your intention, you first depress the VERB button, then followed by two numerics, 0 and 6. Then you depress NOUN button, followed by two numerics, 1 and 8. When you are ready to execute, you depress the ENTR button.

The input sequence above is then the following:

VERB 0 6 NOUN 1 8 ENTR

Once ENTR is pressed, register 1 will show the ROLL, register 2 will show the PITCH and register 3 will show the YAW. Each register is either positive or negative, and will always consist of five digits. You will always see the sign and the five digits, no matter what the format of the number really is. In this case, the format is that each register is a decimal number with two decimals:

XXX.XX, so if register 1 (R1) reads +04510, the current roll of the spacecraft is +045.10.

Each noun controls the format of the register, and the format each register within a given noun can be different. The first register in a noun can for example be XXX.XX, the second XXXXX (whole number, integer), and the third can be XXXX.X. Time can sometimes be displayed using the following format: XX0XX where the first two XX is the minutes, and the last two XX is the

seconds. So if for example a NOUN wants to display the time for a burn in register 1, R1 can look like this: 16045, meaning the burn is 16 minutes and 45 seconds away.

When you tell the computer you wish to enter data into the registers using Verb 21 to 25, or change the NOUN or VERB using the pushbuttons, the active input field will blank.

It is normal for checklists to use a shortened form to communicate with the computer. Looking at the above example:

VERB 0 6 NOUN 1 8 ENTR

This is normally shortened to V06N18E.

When a verb-noun combination or program wants you to input data, the verb and noun fields are usually flashing, meaning you can change the data using V21 to V25, or proceed without changes.

## 6. MAJOR MODES

There are many major modes that the computer can run. There are different major modes for the different phases of the mission, so it is also normally referred to as a mission program.

A major mode can for example be used to prepare and execute the lunar descent, set up an a maneuver or burn, monitor and handle setup, ascent, and much more.

The major modes each follow a program much like a normal executable on a PC. A major mode can request the astronaut to validate/change data, and use this as input to calculation and routines. A major mode is using verb-noun combinations to go through its intended flow, as well as internal routines (functions) for calculations.

One example of a major mode is to prepare for descent. Lunar descent is a complex process and requires a lot of setup and calculations. The entire landing sequence consists of many major modes, usually referred to as the 60's. Major Mode 63 (program 63, P63) will ask for descent parameters used for calculations, followed by program 64, 65 and 66.

PHASE	PROGRAM	TITLE
Service	00	Idle program
	06	LGC Power Down
Ascent	12	Powered Ascent
Coast	20	Rendezvous Navigation
Pre-Thrusting	30	External Delta V
Thrusting	40	DPS
	41	RCS
Alignments	52	IMU Realign

Descent	63	Braking Phase
	64	Approach Phase
	66	Landing Phase (ROD)
	68	Landing Confirmation

## 7. CHECKLIST REFERNECE (V50N25)

Verb 50 means Please Perform: and V 25 means Checklist. V50N25 will show the checklist that needs to be performed in R1. The following is a table contains all the checklists:

<b>CODE</b>	<b>CHECKLIST</b>
00062	Switch LGC power down (P06)
00203	Switch guidance control to PGNS, mode to Auto, thrust control to Auto (P63)
00500	Switch LR antenna to Position 1 (P63)

## 8. ALARM CODES (V05N09)

If an alarm code is present, relevant alarm code is visible in V05N09. Below is a list of what they mean.

<b>CODE</b>	<b>PURPOSE</b>	<b>SET BY</b>
00213	IMU not operating with turn-on request	T4RUPT

## 9. PROGRAMS

The following section will go into each program in detail. This includes the purpose and assumptions of the program, as well as the sequence of events.

## P00 – LGC IDLING PROGRAM

Purpose:

1. To maintain the LGC in a condition of readiness for entry into other programs.
2. To update the CSM and LM state vectors every four time steps.

Assumptions:

1. This program is automatically selected by V96E, which may be done during any program. State vector integration is permanently inhibited following V96E. Normal integration functions will resume after selection of any program or extended verb. P00 integration will resume when P00 is reselected. Usage of V96 can cause incorrect W-matrix and state vector synchronization.
2. Program changes are inhibited during integration periods and program alarm 1520 will occur if a change is attempted when inhibited.

Sequence of Events:

V37E00E

V06N38E

Optional Display.

V06N38

Time of State Vector Being  
Integrated

00XXX h

000XX min

0XX.XX s

## P06 – LGC POWER DOWN PROGRAM

Purpose:

1. To transfer the LGC from the Operate to the Standby program.

Assumptions:

1. If the computer power is switched off, the LGC Update program (P27) would have to be performed to update the LM and CSM state vectors and computer clock time.
2. The LGC is capable of maintaining an accurate value of ground elapsed time (GET) for only 23 hours when in the Standby mode. If the LGC is not brought out of the standby condition to the running condition at least once within 23 hours, the LGC value of GET must be updated.
3. Once the program has been selected, the LGC must be put in standby. When P06 appears, the LGC will not honor a new program request (V37EXXE), a terminate (V34E), or an enter in response to the request for standby.

Sequence of Events:

V37E06E

	Flashing	
V50N25	Checklist Code	0062
	Power Down LGC	CB(11) IMU Operate-Open
PRO	Standby light – on	(No DAP light on)
PGNS Turn On		CB(11) IMU Operate-Close
	Standby light – on. PRO until Standby light off. No Att light – on (90 seconds)	
V37E00E	Program P00 Chosen	

## P12 – POWERED ASCENT PROGRAM

Purpose:

1. To control the PGNS during countdown, ignition, thrusting, and thrust termination of PGNS controlled APS powered ascent maneuver from the lunar surface.

Assumptions:

1. The LGC has stored injection values which define an ascent trajectory that will result in an orbit coplanar with the CSM orbit and an apolune of 30 nmi. These values at orbit insertion are altitude, distance between the LM and CSM orbital planes, LM vertical ( $V(R)$ ), LM horizontal ( $V(Y)$ ), and LM downrange ( $V(Z)$ ) velocities. All altitudes are measured with respect to the LGC stored landing site vector.
2. The PGNS will control the LM ascent maneuver such that the LM injection velocity is in the CSM orbital plane or parallel to it at a distance specified by the astronaut inserted crossrange. The injection conditions can be modified by changing the nominal downrange and radial velocities displayed.

Crossrange should not be specified so that the ascent trajectory crosses through the CSM orbital plane.

3. Engine ignition may be slipped beyond TIG (AS) if desired by the crew or if the state vector integration cannot be completed in time. Variation of the time of ascent ignition (TIG(AS)) changes the relative phasing of the ascent trajectory with respect to the CSM and alters the resultant LM orbit.
4. The initial period of the ascent trajectory consists of two phases:
  - a. Vertical Rise Phase. From TIG until the LM radial velocity ( $V(R)$ ) exceeds 40 ft/s. During this phase, the PGNS holds the LM attitude with the +X axis parallel to the LM position vector at TIG. At TIG, the PGNS commands the LM around its X axis (yaw) until the LM +Z axis points downrange.
  - b. Pitchover Phase. When  $V(R)$  exceeds 40 ft/s and the LM X axis is within 5 degrees of the desired attitude. During this phase, the PGNS commands the LM to pitch down (about the Y axis) an amount defined by the guidance equations.
5. IMU aligned at a known orientation.
6. The inertial velocity Y axis will be displayed on the lateral velocity cross pointer and the forward velocity cross pointer will be zeroed during ascent .
7. The X-axis override option provides the crew with the ability to exercise manual control about the LM X axis with the attitude controller even though the PGNS Attitude Control mode is Auto. When the controller returns to detent, the DAP damps the yaw rate, stores the yaw attitude when the rate is damped, and then maintains that attitude.

- This option is inhibited from TIG (AS) until 10 seconds after V® equals 40ft/s and the LM yaw attitude is within 5 degrees of the desired pitch over initiation.
8. Either the Load DAP routine (R03) or the Landing Confirmation program has been performed prior to selection of this program. The DAP will be energized when the PGNS Control mode and the Auto Attitude or Attitude Hold Control mode have been selected. If this occurs prior to the PGNS autocheck in this program, the attitude errors will be zeroed and the attitude deadband will be set to the value specified by P68 (5 degrees) or R03 (astronaut defined), whichever occurred more recently. Immediately prior to the PGNS autocheck, this program will set the attitude deadband to 1 degree.
  9. If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Auto, the PGNS controls the total vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FDAO. The crew may exercise control about only the yaw axis with the ACA (X-axis override) provided the X-axis override capability is permitted (see Assumption 7).
- If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS holds the vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FDAO. The crew may exercise manual control about all vehicle axes with the ACA using either the Rate Command or Minimum Impulse mode. Not recommended.
10. Deleted
  11. The PGNS generates two types of errors for display on the FDAO as selected by the astronaut:
    - a. Mode 1 - Selected by Verb 61. Autopilot following errors used as a monitor of the DAP's ability to track automatic steering commands.
    - b. Mode 2 - Selected by Verb 62. Total attitude errors used to assist the crew in manually maneuvering the vehicle.
  12. This program is selected by the astronaut at least 5 minutes prior to ignition.

## Sequence of Events:

V37E12E

Flashing	Ascent Time of Ignition	00XXX h
V06N33		000XX min
		0XX.XX s

V25 Load New TIG

PRO

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing V06N76	Desired Downrange Velocity Desired Radial Velocity Crossrange	XXXX.X ft/s XXXX.X ft/s XXXX.X nmi
V25 Load New Crossrange		

PRO

Flashing V50N25	Checklist Code	00203
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PRO Switch Guidance Control – PGNS  
Mode Control – Auto

V06N74	Time from Ignition FDAL Yaw – After Vertical Rise FDAL Pitch – After Vertical Rise	XXbXX min/s XXX.XX deg XXX.XX deg
--------	--	---

TFI counts down until TIG -35 seconds, when DSKY blanks for 5 seconds.  
V06N74 display returns until TIG -5 seconds.

Flashing V99N74	Time from Ignition FDAL Yaw – After Vertical Rise FDAL Pitch – After Vertical Rise	XXbXX min/s XXX.XX deg XXX.XX deg
--------------------	--	---

PRO Astronaut okays ignition. TIG occurs.

V06N94	VGX Altitude Rate Computed Altitude	XXXX.X ft/s XXXX.X ft/s XXXXXX. ft
--------	---	--

N76E

V06N76	Desired Downrange Velocity Desired Radial Velocity Crossrange	XXXX.X ft/s XXXX.X ft/s XXXX.X nmi
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V16N77E

V16N77	Time to Engine Cutoff Velocity Normal to CSM Plane Absolute Value of Velocity	XXbXX min/s XXXX.X ft/s XXXX.X ft/s
--------	---	---

N85E

# LUNAR MODULE

## FLIGHT MANUAL

# R E N T R Y

AN ORBITAL SIMULATOR

V16N85	Velocity to be Gained (X Body)	XXXX.X ft/s
	Velocity to be Gained (Y Body)	XXXX.X ft/s
	Velocity to be Gained (Z Body)	XXXX.X ft/s

Display chosen at velocity to be gained = 500 ft/s. VG (X Body) monitored to enable APS fuel to be fed to RCS thrusters by astronaut.

Null residual velocities

### KEY REL

Flashing	VGX	XXXX.X ft/s
V16N94	Altitude Rate	XXXX.X ft/s
	Computed Altitude	XXXXXX. ft

### PRO

V16N85	Velocity to be Gained (X Body)	XXXX.X ft/s
	Velocity to be Gained (Y Body)	XXXX.X ft/s
	Velocity to be Gained (Z Body)	XXXX.X ft/s

### V82E

Flashing	Apocenter Altitude	XXXX.X nmi
V16N44	Pericenter Altitude	XXXX.X nmi
	Time from Phase	XXbXX min/s

### KEY REL

Flashing	(Same as above)
V16N85	

### PRO

Select New Program.

## P20 – RENDEZVOUS NAVIGATION

Purpose:

1. To control the LM attitude and the Rendezvous Radar (RR) to acquire and track the CSM with the RR while the LM is in flight.
2. To update either the LM or CSM state vector (as specified by the astronaut by DSKY entry) on the basis of RR tracking data or to track the CSM without updating either vehicle state vector.
3. To point the LM optical beacon at the CSM.

Assumptions:

1. The CSM is maintaining a preferred tracking attitude that correctly orients the CSM radar transponder for RR tracking by the LM.
2. At the beginning of the program, the state vector update option is automatically set to the LM. This option may be changed at any time later by one of the following manual entries.
  - a. VSOE-Update LM state vector.
  - b. V81 E-Update CSM state vector,
  - c. V95E-No state vector update.
3. The initialization of the W matrix is enabled by:
  - a. A manual DSKY entry (V93EI).
  - b. Computer Fresh Start (V36E),
  - c. State vector update from the ground (P27I) (Except for update of Landing Site vector when the LM is on the lunar surface).
  - d. The powered ascent program (P12) invalidates the W matrix used by P22 and causes P20 to reinitialize the W matrix when selected.
4. The RR tracking mark counter counts the number of RR marks processed by the LGC. This counter is zeroed by:
  - a. Manual selection of P20/22 (V37E20/22EI •
  - b. Completion of the Target Delta V program (P76I).
  - c. Selection of a new program from a program which ha\_Q\_ turned on Average G \_
  - d. Initialization of the W matrix.
  - e. Completion of RR search routine (R24) in P20.
5. The crew may manually adjust the LGC-stored values of RR shaft and trunnion bias by a direct load of four registers. However, unless the RR has been jarred, the LGC bias estimate should be more accurate than that from another source.

6. The selection and termination of P20, P22, and P25 are subject to special operating procedures different from all other programs:
  - a. Selection
    - (1) Always by V37EXXE.
    - (2) If any other program IS running at the time of P20/22/25 selection the new program will replace the old. This includes P20/22/25 selection whenever either P20, 22, or 25 is running.
    - (3) If P20 or P25 is running, selection of any program other than POO or P22 will result in P20 or P25 continuing and the new program also operating with its number in the DSKY program lights.
    - (4) If P20 or P25 is running, selection of POO or P22 will result in the termination of the old program and operation of the new.
  - b. Termination
    - (1) By selection of POO, V56E, or by V34E.
    - (2) POO selection will terminate P20, 22, and P25 and any other program in process, and establish POO.
    - (3) V56E selection will select the Terminate Tracking routine (R56) which will terminate only P20 or P25 if either of these programs is running in conjunction with another program. In all other cases R56 will select ROO. V56E may be performed any time during P20, 22, or 25 operation.
    - (4) The LGC will act upon V34E only in response to a flashing verb-noun. If this display was originated by P20, 22, or 25, V34E will result in an identical LGC response to that of V56E; that is, selection of R56. If this display was not originated by P20, 22, or 25 (such as P32, while running with P20), the LGC will go to ROO; however, the program in the background will continue. The new program selected follows the selection rules above.
7. The RR Manual Acquisition routine ( R23) may be selected only if P20 is not running in conjunction with another program.
8. When P20 is selected any time prior to the landing phase in the lunar mission, this program must be operated in the no update mode to prevent modifying a precision state vector for landing.
9. The RMS position and velocity errors computed from the W matrix are available by Extended Verb (V67E). Based upon values in this display and the details of the mission, the astronaut can elect to stop or continue the current navigation procedure or to reinitialize the W matrix and continue navigating. The capability to reinitialize the W matrix is also provided via V67 E.
10. State vector integration may be permanently inhibited by V96E. This entry will terminate all present programs and select the LGC Idling program (POO) with the POO automatic state vector integration permanently inhibited until selection of another program. Use of V96 can cause incorrect W-matrix extrapolation since state vector synchronization is not maintained.

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

### Sequence of Events:

V37E20E

V80E or V81E or V95E

State Vector Option

V80E - LM, V81 E - CSM. V95E - None

RR Mode Switch - in LGC

Flashing

RR Trunnion Angle

XXX XX deg

### PRO

If RR locked on and tracking. No Track light out, DSKY blanks RA taking marks.

V16N78E

Range	XXX.XX nmi
Range Rate	XXXXXX ft/s
Time from Ignition	XXbXX min/s

### KEY REL

Flashing	Delta R	XXXX.X nmi
V06N49	Delta V	XXXXXX ft/s
	Data Source Code	0000X

X= 1 – Range X = 2 - Range Rate X = 3 - Shaft Angle

X= 4 - Trunnion Angle

V32E                                    Reject partial mark

V34E                                    Reject total mark

PRO                                    Update with mark

Flashing	(see above display)
V06N49	

To terminate: V56E or V37EOOE or V34E during a flashing display. To keep P20 running in the background: V37E XXE.

If RR not locked on and pointing angle greater than 15 degrees,

Flashing	Desired Automaneuver to F DAI Ball	R	XXX.XX deg
V50N18	Angles	P	XXX.XX deg
		Y	XXX.XX deg

Automaneuver: Guidance Control – PGNS  
                   : Mode Control – PGNS Auto

## PRO

V06N18	Desired Automaneuver to FDAO Ball Angles	R P Y	XXX.XX deg XXX.XX deg XXX.XX deg
--------	---	-------------	--

Monitor maneuver to attitude.

Manual Maneuver: Mode Control – PGNS Attitude Hold, then maneuver.

## PRO

Flashing V50N18	Desired Automaneuver to FDAO Ball Angles	R P Y	XXX.XX deg XXX.XX deg XXX.XX deg
--------------------	---	-------------	--

When maneuver is complete, by either method, select mode of RR acquisition of CSM.

## ENTER

Manual RR acquisition. RR Mode switch: Auto or Slew

Flashing V50N25	Checklist Code	00201
--------------------	----------------	-------

## ENTER

Choose RR acquisition mode.

Flashing V50N25	Checklist Code	00205
--------------------	----------------	-------

Perform manual acquisition of CSM with RR. Slew RR for lockon. RR Mode switch – LGC.  
 No Track light is off. Wait 10 seconds.

## PRO

Flashing V50N72	Trunnion Angle Shaft Angle	XXX.XX deg XXX.XX deg
--------------------	-------------------------------	--------------------------

Verify main lobe lockon.

## PRO

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

### PRO

DSKY blanks; No Track light is out; RR taking marks.

Flashing	Delta R	XXXX.X nmi
V06N49	Delta V	XXXX.X ft/s
	Data Source Code (see above)	0000X

V32E	Reject partial mark.
V34E	Reject total mark.
PRO	Update with mark.

Flashing	See above display and response options.
V06N49	

Automatic RR acquisition.

RR Mode switch – LGC.

Flashing	Trunnion Angle	XXX.XX deg
V50N72	Shaft Angle	XXX.XX deg

### PRO

No Track light is on.

Flashing	Alarm Code	00503
V05N09		

RR data no good for 30 seconds or Designate fails.

### V32E

Redesignate to new V50N72 display.

### PRO

Start Search mode.

Flashing	Data Indicator	00000 -Search (42 seconds/scan)
V16N80	Angle Between LOS and LM +Z Axis	11111 – Lockon XXX.XX deg

### PRO

When lockon occurs automatically, DSKY blanks; No Track light out; RR taking marks after PRO.

Flashing	Delta R	XXXX.X nmi
V06N49	Delta V	XXXX.X ft/s
	Data Source Code (see above)	0000X
V32E	Reject partial mark.	
V34E	Reject total mark.	
PRO	Update with mark.	

To terminate: V56E or V37E00E, or V34E during a flashing display.

To keep P20 running in the background: V37EXXE.

## P30 – EXTERNAL DELTA-V

Purpose:

1. To accept targeting parameters obtained from a source(s) external to the LGC and compute therefrom the required velocity and other initial conditions required by the LGC for execution of the desired maneuver. The targeting parameters inserted into the LGC are the time of ignition (TIG) and the impulsive dV along LM local vertical axes at TIG.

Assumptions:

1. The target parameters (TIG and Delta V(LV)) may have been loaded from the ground during a prior execution of P27.
2. The External Delta V flag is set during this program to designate to the thrusting program that External Delta V steering is to be used.
3. The ISS need not be on to complete this program unless the Rendezvous Radar is to be used for automatic state vector updating by the Rendezvous Navigation program (P20).
4. The Rendezvous Radar may or may not be used to update the LM or CSM state vectors for this program. If radar use is desired, the ISS should be in operation and the radar should have been turned on and locked on the CSM by previous selection of P20. Radar sighting marks will be made automatically approximately once a minute when enabled by the Track and Update flags.
5. This program is applicable in either earth or lunar orbit.

Sequence of Events:

V37E30E

Flashing	Time of Ignition	00XXX. h
V06N33		000XX. min
		0XX.XX s

V25E Load New TIG

PRO

Flashing	Components of $\Delta V$ (LV)	X XXXX.X ft/s
V06N81		Y XXXX.X ft/s
		Z XXXX.X ft/s

V25E Load Desired  $\Delta V$

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

PRO

Flashing	Apogee/Apolune Altitude	XXXX.X nmi
V06N42	Perigee/Perilune Altitude	XXXX.X nmi
	Magnitude of Delta V at TIG	XXXX.X ft/s

PRO

Flashing	Marks	XXXXX marks
V16N45	Time Until Next Burn	XXbXX min/s
	Middle Gimbal Angle	XXX.XX deg

PRO Middle gimbal set to 00002 if REFSMMAT flag is not set.

PRO Select New Program.

## P40 – DPS

### Purpose:

1. To compute a preferred IMU orientation and a vehicle attitude for a LM DPS thrusting maneuver and to maneuver the vehicle to that attitude.
2. To control the PGNS during countdown, ignition, thrusting, and thrust termination of a PGNS controlled DPS maneuver.

### Assumptions:

1. The target parameters have been calculated and stored in the LGC by prior execution of a prethrusting program.
2. The required steering equations are identified by the prior prethrust program, which either reset ("ASTEER") or set (External Delta V) the External V flag. For External Delta V steering, VG is calculated once for the specified time of ignition. Thereafter both during DPS thrusting and until the crew notifies the LGC that RCS trim thrusting has been completed, the LGC updates VG only as a result of accelerometer inputs. For steering control when using "ASTEER", the velocity required is calculated from the most recent intercept trajectory semimajor axis. The Lambert routine periodically recomputes the intercept trajectory semimajor axis for the "ASTEER" calculations. The interval between Lambert solutions is controlled by an erasable load value (UT).
3. Engine ignition may be slipped beyond the established TIG if desired by the crew, or if state vector integration cannot be completed in time.
4. If a thrusting maneuver is performed with the Guidance Control Switch in PGNS and the Mode Control Switch in Auto, the PGNS controls the total vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise control about only the yaw axis with the ACA (X-axis override) provided the X-axis override capability is permitted.

If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS holds the vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise manual control about the vehicle axes with the A.C.A. using either the Rate Command or Minimum Impulse mode. However, it is strongly recommended that powered flight not be attempted in the Minimum Impulse mode.

- If the Guidance Control switch is changed from PGNS to AGS during a thrusting maneuver, the LGC continues computation of position, velocity, desired thrust vector, and desired attitude errors.
5. The PGNS generates two types of errors for display on the FOAI as selected by the astronaut.
    - a. Mode 1 - Selected by Verb 61. Autopilot following errors used as a monitor of

- the DAP's ability to track automatic steering commands.
- b. Mode 2 - Selected by Verb 62. Total attitude errors used to assist the crew in manually maneuvering the vehicle.
6. The X-axis override option provides the crew with the ability to exercise manual control about the LM X axis with the attitude controller even though the PGNS Attitude Control mode is Auto. When the controller is returned to detent the DAP damps the yaw rate, stores the yaw attitude when the yaw rate is damped, and then maintains that attitude.

The X-axis override option is always available to the crew. However, it should not be exercised when the LGC is specifying a desired yaw attitude; that is, during the attitude maneuver to the thrusting attitude.

7. When the thrust/translation controller is set to minimum thrust position and the LGC throttle command is zero, the DPS will start at 10 percent thrust.
8. The Load OAP Data routine (R03) has been performed prior to selection of this program and the DPS engine gimbal has been previously driven to the correct trim position. If this burn is of sufficient duration that vehicle transients at ignition due to CG/thrust do not affect accomplishment of maneuver aim conditions, then the gimbal need not be driven to the trim position before TIG. Driving the gimbal to the trim position in worst case conditions could require 2 minutes.
9. During DPS burns only, the pitch-roll RCS jet autopilot (U and V jets) may be disabled by (V65) or enabled by (V75). This capability is intended to be used to prevent LM and descent stage thermal constraint violations during CSM-docked OPS burns (P40). The capability exists during P63 and P70 also. Performance of F R ESH START (V36E) will always enable the capability in the autopilot.
10. The LGC will neither designate nor read the Rendezvous Radar (RR) during this Program.
11. This program should be selected by the astronaut by DSKY entry at least 5 minutes before the estimated time of ignition.
12. The value of Delta V required will be stored in the local vertical coordinate system and is available during this program by keying V06 N81E.

#### Sequence of Events:

V37E40

Flashing	Desired Automaneuver to FDAO Ball	R XXX.XX deg
V50N18	Angles	P XXX.XX deg
		Y XXX.XX deg

Automaneuver: Guidance Control – PGNS  
Mode Control – PGNS Auto

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

### PRO

Monitor automatic maneuver to attitude.

V06N18	Desired Automaneuver to FDAO Ball Angles	R XXX.XX deg P XXX.XX deg Y XXX.XX deg
--------	---	--

Manual Maneuver: Guidance Control – PGNS  
Mode Control – PGNS Attitude Hold

Maneuver to V50N 18 displayed angles.

### ENTER

Flashing V50N25	Checklist Code	00203
Please switch to:	Guidance Control – PGNS Attitude Control – Auto Throttle Switch – Auto	

### ENTER

V06N40	Time from Ignition (TFI) Magnitude of Velocity to be Gained $\Delta V$ (accumulated)	XXbXX min/s XXXX.X ft/s XXXX.X ft/s
--------	--	---

TFI counts down until TIG -35 seconds, when DSKY blanks for 5 seconds. V06N40 display returns until TIG -5 seconds.

TIG - 15 seconds. R3 should be less than 00005.

TIG - 7.5 seconds. Verify +X ullage.

TI G -5 seconds

Flashing V99N40	Time from Ignition Magnitude of Velocity to be Gained $\Delta V$ (accumulated)	XXbXX min/s XXXX.X ft/s XXXX.X ft/s
--------------------	--	---

PRO Astronaut okays ignition. TIG occurs.

V06N40	Time from Engine Cutoff Velocity to be Gained $\Delta V$ (accumulated)	XXbXX min/s XXXX.X ft/s XXXX.X ft/s
--------	--	---

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing	Time from Engine Cutoff	XXbXX min/s
V16N40	Velocity to be Gained	XXXX.X ft/s
	$\Delta V$ (accumulated)	XXXX.X ft/s

Turn off DPS engine:

- Push ENG STOP
- Switch ENG ARM to OFF

PRO

Flashing	VGx (body)	XXXX.X ft/s
V16N85	VGy (body)	XXXX.X ft/s
	VGz (body)	XXXX.X ft/s

Null residual velocities.

PRO

Flashing	Select New Program
V37	

## P41 – RCS

Purpose:

1. To compute a preferred IMU orientation and a vehicle attitude for an RCS thrusting maneuver find to perform the vehicle maneuver to that attitude.
2. To provide suitable displays for manual execution of the thrusting maneuver in the Attitude Hold mode.

Assumptions:

1. The target parameters have been calculated and stored in the LGC by prior execution of a prethrusting program.
2. The required steering equations are identified by the prior prethrust program, which either reset ("ASTEER") or set (External Delta V) the External Delta V flag. For External Delta V steering, VG is calculated once for the specified time of ignition. Thereafter until the crew notifies the LGC that RCS thrusting has been completed, the LGC updates VG only as a result of accelerometer inputs.

For steering control when using "ASTEER," the velocity required is calculated from the most recent intercept trajectory semimajor axis. The Lambert routine periodically recomputes the intercept trajectory semi major axis for the "ASTEER" calculations. The interval between Lambert solutions is controlled by an erasable load value (UT).

3. RCS ignition may be slipped beyond the established TIG if desired by the crew, or if state vector integration cannot be completed on time.
4. If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS holds the vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise manual control about all vehicle axes with the ACA using either the Rate Command or Minimum Impulse mode. However, it is strongly recommended that powered flight not be attempted in the Minimum Impulse mode.
5. The PGNS generates two types of errors for display on the F DAI as selected by the astronaut.
  - a. Mode 1 - Selected by Verb 61. Autopilot following errors used as a monitor of the OAP's ability to track automatic steering commands.
  - b. Mode 2 - Selected by Verb 62. Total attitude errors used to assist the crew in manually maneuvering the vehicle.

6. The X-axis override option provides the crew with the ability to exercise manual control about the LM X axis with the attitude controller even though the PGNS Attitude Control mode is Auto. When the controller is returned to detent, the OAP damps the yaw rate, stores the yaw attitude when the yaw rate is damped, and then maintains that attitude.

The X-axis override option is always available to the crew. However, it should not be exercised when the LGC is specifying a desired yaw attitude; that is, during the attitude maneuver to the thrusting attitude.

7. The Load DAP Data routine ( R03) has been performed prior to selection of this program.  
 8. The LGC will neither designate nor read the Rendezvous Radar (RR) during this program.  
 9. This program should be selected by the astronaut by DSKY entry at least 5 minutes before the estimated time of ignition.  
 10. The value of Delta V required will be stored in the local vertical system and is available in this program until Average G turns on by keying in V06N81E.

#### Sequence of Events:

V37E41E

Flashing	Desired Automaneuver to FDA I Ball	R XXX.XX deg
V50N18	Angles	P XXX.XX deg
		Y XXX.XX deg

Automaneuver:      Guidance Control -PGNS  
                           Mode Control – PGNS Auto

PRO

V16N18	Desired Automaneuver to FDA I Ball	R XXX.XX deg
	Angles	P XXX.XX deg
		Y XXX.XX deg

Monitor automatic maneuver to V06N18 displayed values.

Manual Maneuver: Guidance Control – PGNS  
                           Mode Control - PGNS Attitude Hold

Maneuver to V50N 18 values.

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

### ENTER

V16N85	VGx (body)	XXXX.X ft/s
	VGy (body)	XXXX.X ft/s
	VGz (body)	XXXX.X ft/s

Mode Control: Attitude Hold

At TIG - 35 seconds, the DSKY blanks until TIG + 30 seconds and V16N85 display returns.

At TI G - 00 seconds.

Flashing	VGX (body)	XXXX.X ft/s
V16N85	VGy (body)	XXXX.X ft/s
	VGz (body)	XXXX.X ft/s

Null components of velocity, when satisfied.

### PRO

Flashing	Select New Program.
V37	

## P52 – IMU REALIGN

Purpose:

1. To align the IMU from a "known" orientation to one of four orientations selected by the astronaut using sightings on two celestial bodies with the AOT or a backup optical system.

- a. Preferred Orientation (Option 00001). An optimum orientation for a previously calculated maneuver. This orientation must be calculated and stored by a previously selected program.
- b. Landing Site Orientation (Option 00004)  
 $XSM = \text{Unit} ( RLS )$   $Y SM = \text{Unit} ( ZSM \times XSM )$   $ZSM = \text{Unit} ( HCSM \times XSM )$   
where:

The origin is the center of the moon.  $R_{LS}$  = The position vector of the LM on the lunar surface at a landing site and a time  $T(\text{align})$  selected by the crew.

$HcsM$  = The angular momentum vector of the CSM (  $RcsM \times V_{CSM}$  ).  
A special case of the landing site orientation occurs when  $T(\text{align})$  is defined as the time of lunar landing  $T(\text{land})$ . This case occurs only if  $T(\text{land})$  has been defined by the MSFN, transmitted to the crew, and the crew has then defined  $T(\text{Align})$  to be  $T(\text{land})$  in this program.

- c. Nominal Orientation (Option 00002)  
 $XSM = \text{Unit} ( R )$   $Y SM = \text{Unit} ( V \times R )$   $ZSM = \text{Unit} ( XSM \times Y SM )$   
where:  
  
 $R$  = The geocentric (earth orbit) or selenocentric (lunar orbit) radius vector at time  $T(\text{align})$  selected by the astronaut.  
 $V$  = The inertial velocity vector at time  $T(\text{align})$  selected by the astronaut.
- d. REFSMMAT (Option 00003). A known orientation stored in the LGC at a previous time.

Assumptions:

1. The configuration may be docked (LM/CSM) or undocked (LM alone). The present configuration should have been entered into the LGC by completion of the OAP Data Load routine (R03).
2. There are no restraints upon the LM attitude control modes until a PGNS controlled maneuver is called by a program or the crew wishes to manually maneuver the vehicle. The Guidance Control switch may be at PGNS or AGS and, if

at PGNS, the mode may be Auto or Attitude Hold. Prior to PGNS controlled maneuvers the LGC will request the correct mode if it is not in effect. For manually controlled maneuvers the crew must select the correct modes.

3. This program makes no provision for an attitude maneuver to return the vehicle to a specified attitude. Such a maneuver, if desired, must be done manually. An option is provided however to allow pointing of the AOT at astronaut or LGC selected stars either manually by the crew or automatically by an LGC controlled attitude maneuver.
4. An option is provided to realign the IMU to the preferred, nominal, or landing site orientation without making celestial body sightings.
5. Extended verbs should not be exercised during this program because of possible interference with the AOT Mark routine (R53).

#### Sequence of Events:

V37E52E

Flashing	Option Code ID	00001 IMU
V04N06		Alignment Option
		0000X
( 1 -Preferred, 2-nominal, 3-REFSMMAT, 4-landing site)		

V22E. Reload desired option.

Landing Site option.

PRO

Flashing	Time og Landing	00XXX. h
V06N34		000XX. Min
		0XX.XX s

V25E. Reload desired landing time.

PRO

Flashing	Designated Landing Site Latitude	XX.XXX deg (+ north)
V06N89	Designated Landing Site Longitude/2	XX.XXX deg (+ east)
	Designated Landing Site Altitude	XXX.XX nmi

V25E. Load corrected landing site coordinates.

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

PRO Go to Preferred option.

Nominal option

Flashing

V06N34

Time of Alignment

00XXX. h

000XX. Min

0XX.XX s

V25E. Load desired T ALIGN

PRO Go to Preferred option.

Preferred, Nominal, or Landing Site options continue from this display.

PRO

Flashing

V06N22

IMU Gimbal Angles at Desired

Orientation

OGA

XXX.XX deg

IGA

XXX.XX deg

MGA

XXX.XX deg

PRO

Flashing

V50N25

Checklist Code

00013

Gyro Torque Only

Mode Control: PGNS – Attitude Hold, V76E – minimum impulse, No DAP light on.

ENTER

V16N20

Present ICDU Angles

OGA

XXX.XX deg

IGA

XXX.XX deg

MGA

XXX.XX deg

when torquing complete

Flashing

V50N25

Checklist Code

00014

ENTER

No fine alignment desired.

Flashing

Select New Program

V37

Normal alignment and realignment

PRO No Attitude light-on – then off

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing  
V50N25

Checklist Code

00015

Select star acquisition mode.

PRO or V32E for  
Cursory Spiral  
Marking

Flashing  
V01N70

Code

00CDE

C-AOT Detent

0 -COAS calibration (not allowed), 1-front left,  
2-front center, 3-front right, 4-right rear,  
5-rear center, 6-rear left, 7-backup optical  
system – COAS

DE-Celestial Body Code

00-Planet, 01/45-star from code list, 46-sun,  
47-earth, 50-moon.

V21E. Load desired star code and detent.

PRO If C=7, COASnto be used.

Flashing      Backup Optics LOS Azimuth  
V06N78      Backup Optics LOS Elevation

XXX.XX deg  
XXX.XX deg

V24E. Load correct data.

+E, +E for forward window  
+E, +9000E, overhead window

If DE=00

Flashing      Components of Celestial Body Unit  
V06N88      Vector

.XXXXXX  
.XXXXXX  
XXXXXX

V25E. Load desired vector components.

If Cursor and  
Spiral Option

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing		
V06N79	Cursor Angle	XXX.XX deg
	Spiral Angle	XXX.XX deg
	Detent Code	+0000X

V32E

Recycle to Flashing V01N70 display above.

or

PRO

Flashing		
V01N71	Code	00CDE
C-AOT Detent		
0-COAS calibration (not allowed), 1-front left, 2-front center, 3-front right, 4-right rear, 5-rear center, 6-rear left, 7-backup optical system-COAS		
DE-Celestial body code		
00-Planet, 01/45 star from code list, 46-sun, 47-earth, 50-moon.		

If DE = 00

Flashing		
V06N88	Components of Celestial Body	.XXXXX
	Unit Vector	.XXXXX
		.XXXXX
Flash		
V52N71	Code	00CDE
	Mark X/Cursor Counter	.XXXXX
	Mark Y/Spiral Counter	.XXXXX

Position cursor, Mark X, or ROD switch pushed.

PRO

Flashing	Cursor Angle	XXX.XX deg
V21N79	Spiral Angle	XXX.XX deg
	Detent Code	+0000X

(Definition of detent code, same as above in C position of N71 display)

Enter current value of cursor angle or V22; enter current value of spiral angle.

Flashing		
V06N79	Cursor Angle	XXX.XX deg
	Spiral Angle	XXX.XX deg
	Detent Code	+0000X

PRO

Flashing	Code	00CDE
V53N71	Mark X/Cursor Counter	XXXXXX
	Mark Y/Spiral Counter	XXXXXX

Position spiral, Mark X or Mark Y, or ROD switch pushed.

PRO

Flashing	Cursor Angle	XXX.XX deg
V22N79	Spiral Angle	XXX.XX deg
	Detent Code	+0000X

Enter current value of spiral angle or V21; enter current value of cursor angle.

PRO

Program recycles to Flashing V52N71 or V53N71 display as above.  
ENTER will alternate V52N71 or V53N71 displays at this point.

After second star marking is finished

PRO

Flashing		
V06N05	Display defined below.	

PRO

Flashing	Desired Automaneuver to FDAO Ball	R XXX.XX deg
V50N18	Angles	P XXX.XX deg
		Y XXX.XX deg

Manual Maneuver

Mode Control: PGNS – Attitude Hold. Do manual maneuver.

PRO

Flashing	Desired Automaneuver to FDAO Ball	R XXX.XX deg
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# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

V50N18	Angles	P XXX.XX deg
		Y XXX.XX deg

ENTER

After maneuver Complete

Flashing	Mark X or Y or ROD
V54N71	Display same as V01N71 above.

MARK X

V53N71	Mark Y or ROD. Display same as V01N71 above.
--------	--

MARK Y

V52N71	Mark X or ROD. Display same as V01N71 above.
--------	--

PRO Display same as V01N71 above.

After second star

Flashing	Star Angle Difference	XXX.XX deg
V06N05		

V23E

Recycle to start og program with flashing V50N25 display.  
(RE FSMMAT option)

or

PRO

Flashing	Gyro Torque Angle	X XX.XXX deg
V06N93		Y XX.XXX deg
		Z XX.XXX deg

PRO

Flashing	Select New Program.
V37	

V32E

Recycle to part of program with flashing V50N25 display.  
(REFSMMAT option)

## P63 – BRAKING PHASE

Purpose:

1. To calculate the required time of OPS ignition (TIG) and other initial conditions required by the LGC for a PGNS-controlled, DPS-executed, braking phase of the powered landing maneuver.
2. To provide option to fine align the IMU to an existing REFSMMAT.
3. To align the LM to the thrusting ignition attitude.
4. To control the PGNS during countdown, ignition, and thrusting of the powered landing maneuver until HI gate.
5. To indicate to the crew that HI gate has been reached by automatic selection of the Approach Phase program (P64).

Assumptions:

1. The LM is on a descent coast orbit (Hohmann transfer) approaching the braking ignition point which is nominally 50,000 feet above the lunar radius at the designated landing site. The descent coast orbit is approximately coplanar with the CSM orbital plane. If the designated landing site is not in the descent coast plane at the nominal time of landing the plane change will be accomplished by the powered landing maneuver (Braking program, P63, and Approach program, P64).
2. The CSM is in a near-circular orbit around the moon at a nominal altitude of 60 nautical miles. The CSM is maintaining a preferred tracking attitude for optical tracking of the LM.
3. The IMU is on and aligned to a landing site orientation defined for the designated landing site and the nominal time of landing ( $T_{(land)}$ ), but should be fine aligned to this orientation as closely as possible prior to DPS ignition. The LM has not yet been aligned to the correct attitude for ignition for the powered landing maneuver.
4. The landing Radar (IR) was energized, checked out, and made ready at IR Position No. 1 prior to selection of this program. Radar data will not be incorporated into the IM state vector until the astronaut sets the IR permit flag via V57E indicating he is satisfied with the quality of the data. V58E will reset the LR permit flag.
5. The landing Analog Displays routine (R10) is enabled at DPS ignition and is terminated upon termination of Average G. The Powered Flight Designate routine (R29) is not enabled during the lunar descent.
6. The entire powered landing maneuver (braking, approach, and landing) will be accomplished using the DPS engine.
7. The aim conditions for braking phase are stored in the LGC.
8. The following parameters required by this program have been stored by the LGC since IGC initialization by erasable load.
  - a. The LM and CSM state vectors. The LGC has updated these as required. No

further state vector updates from any external source other than the LR will be accepted by this program.

- b. The nominal landing time at the designated landing site T(land) and the position RIS. Corrections to the landing site position RLS may be made by keying V21 through V25 N69 and entering the appropriate correction.
9. The DPS is not throttles over the whole range (0 to maximum). It must be operated either at maximum throttle or over a specific throttle range of lower settings. These throttle settings are total throttle settings; that is, the sum of the manual setting (whose minimum is about 10 percent) and the PGNS commanded setting.

This program assumes the Throttle Control switch to be in Auto (the DPS receives the sum of the manual and PGNS commanded settings) and the manual throttle to be set at minimum for ZOOMTIME seconds of thrusting, and thereafter at a level less than that required by the IGC. The value ZOOMTIME is in erasable storage, having been loaded prior to launch or by P27.

Due to the region of forbidden throttling, thrust command logic in conjunction with the interim terminal conditions assures that the commanded throttle remains at maximum until the guidance equations first require it to be within the allowable throttle range. Thereafter it should remain within the allowable throttle range.

- Furthermore, the DPS must be started in the following sequence: (1) +X axis 2-jet ullage for 7.5 seconds, (2) ignition at minimum throttle, (3) ullage off 0.5 seconds after ignition, (4) ZOOMTIME seconds at minimum thrust, and (5) maximum throttle. The throttle setting then becomes controlled by the guidance equations.
10. During the powered landing maneuver, the LGC will monitor the presence or absence of the Auto Stabilization discrete. This discrete is issued to the LGC when the Mode Control switch is in the Auto position.

The LGC will also monitor the presence or absence of the Auto Throttle discrete. This discrete is issued to the LGC when the Thrust Control switch is in the Auto position.

Should either of these discretes be interrupted during the powered landing maneuver, the LGC assumes that it no longer has complete automatic control of the maneuver.

The monitor and the associated LGC logic is included in the Landing Auto Modes Monitor routine (R 13) which will be called by this program.

The LGC can be forced to ignore the absence of the Auto Throttle discrete and continue issuing normal throttle commands by setting the CHANBACKUP location

- (0374) in the computer to 0001 X8. This location can only be set by astronaut or
11. The X-axis override option provides the crew with the ability to exercise manual control about the LM X axis with the attitude controller even though the PGNS Mode Control switch is in Auto. When the controller is returned to detent the PGNS damps the yaw rate, stores the yaw attitude when the yaw is damped, and then maintains that attitude.

The X-axis override option is available to the crew (until the estimated altitude is below 30,000 feet); however, it should not be exercised when the LGC is specifying a desired yaw attitude; that is, during the attitude maneuver to the thrusting attitude. The option is inhibited by this program from midway in the program to the end.

12. The LGC specifies LM attitude during the powered landing maneuver based upon the requirements of thrust vector control, landing site visibility, and LR orientation. After DPS ignition, thrust vector control is required through the remainder of this program. The landing site becomes visible at the beginning of the approach phase.

Thrust vector control does not constrain the LM orientation about the thrust axis (yaw attitude). Rotation about the LM Y and LM Z axes is used to point the measured thrust vector along the desired thrust vector.

The first restraint upon the LM yaw attitude to occur is that of LR orientation. The LGC will not attempt to use LR data until the LGC estimation of altitude is 50,000 feet. Automatic X-axis override lockout and yaw attitude specification by the LGC will not occur until the LGC estimated altitude is 30,000 feet. Before this time, the astronaut must maneuver to a roughly-window-up yaw orientation to prevent subsequent loss of S-band lock-on. The LGC will then command the vehicle to the LGC-specified yaw attitude.

Subsequent to X-axis override lockout, control of the vehicle about the LM X axis is governed by LR orientation requirements during this program. The landing site becomes visible to the command pilot if the "look" angle (the angle between the LM-X axis and the LOS to the landing site) is greater than 25 degrees and the LOS is in or near the LM X/Z plane.

At any time during P63 or P64, the magnitude of the look angle and the orientation of the look angle plane (that plane containing the LOS and the LM X axis) are defined by the inertial orientation of the LM X axis and the position of the LM with respect to the landing site.

13. The crew has the capability to display LGC calculated values of forward velocity, lateral velocity, altitude, and altitude rate on certain LM meters during this program. The calculations of these parameters is under the control of the Landing Analog Displays routine.

14. The crew can select a display of the LGC computed throttle setting by keying V16 N92.
15. The Rate of Descent (ROD) mode is not enabled during this program.
16. An abort from the lunar descent may be required at any time during the descent orbit injection, the descent coast, or the powered descent (P63), (P64I, or (P66).

For aborts after DPS ignition for the powered landing maneuver, time is critical. During this period an abort is nominally commanded by pushing one of two buttons in the LM. The abort may be commanded to use the descent stage (Abort button) or the ascent stage (Abort Stage button). If the descent stage is selected, and the DPS propellant approaches exhaustion, control must be switched to the ascent stage by the crew by ascent stage selection (Abort Stage button).

During the powered landing maneuver, the LGC will continuously monitor the AbOrt and Abort Stage discretes, and upon receipt of either will terminate the program in process and call the appropriate abort program IOPS Abort program (P70) or APS Abort program (P71). Both abort programs will guide the LM to an acceptable orbit.

The monitor of the Abort and Abort Stage buttons is controlled by the Abort Discretes Monitor routine (R11) which will be enabled by this program.

This step can be locked out by setting the CHANBACKUP location (0374) in the computer to OOOX18. This location can only be set by astronaut or ground loading and is not changed by Fresh Start or Restart.

17. If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Auto, the PGNS controls the total vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FDAI. The crew may exercise control about only the yaw axis with the ACA (X-axis override) provided the X-axis override capability is permitted.

If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS holds the vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise manual control about all vehicle axes with the ACA using either the Rate Command or Minimum Impulse mode. However, it is strongly recommended that powered flight not be attempted in the Minimum Impulse mode.

During a thrusting maneuver in the PGNS/ Attitude Hold mode the astronaut is responsible for maintaining small enough attitude errors to achieve guidance objectives.

18. Control of LM DPS, RCS, and APS is transferred from PGNS to the Abort Guidance System (AGS) by changing the Guidance Control switch from PGNS to

AGS.

The AGS will be capable of taking over control of the LM during any portion of the lunar descent or ascent or during either of the abort programs (P70 or P71). The AGS will guide the LM to a safe orbit.

The AGS may be initialized by the LGC at any time by manual selection of the AGS Initialization routine (R47). However, it is not recommended that the AGS be initialized during powered flight because OAP attitude control is interrupted during the CDU zero part of the routine.

In the event that the Guidance Control switch is changed from PGi-IS to AGS during a thrusting maneuver, the LGC will continue computation of position and velocity, the desired thrust vector, and the desired attitude errors.

19. The PGNS generates two types of errors for display on the FOAI as selected by the astronaut.
  - a. Mode 1-Selected by Verb 61. Autopilot following errors used as a monitor of the DAP's ability to track automatic steering commands.
  - b. Mode 2 - Selected by Verb 62. Total attitude errors used to assist the crew in manually maneuvering the vehicle.
20. The event timer was set prior to selection of this program to co4bt to zero at T BRAK based on a time from ignition provided by the ground.
21. The Load DAP Data routine (R03) has been performed prior to selection of this program. At that time the DPS engine gimbal should have been driven to the correct trim position.
22. During DPS burn only, the pitch-roll RCS jet autopilot (U and V jets) may be disabled (V65E) or enabled (V75) by Extended Verb as shown. This capability is intended to be used to prevent LM and descent stage thermal constraint violations during CSM-docked DPS burns (P40). The capability exists during P70 also. Performance of FRESH START (V36E) will always enable the pitch-roll jets.
23. This program is selected by the astronaut by DSKY entry. It should be selected at least 20 minutes before the nominal time of ignition for the powered landing maneuver (T BRAK).
24. Engine ignition may be slipped beyond the established TIG if desired by the crew or if state vector integration cannot be completed in time.
25. Two alarm conditions may be originated by the PGNS powered landing equations:
  - a. If subroutine ROOTSPRS in the RG/VG calculation fails to converge in 8 passes the LGC will turn on the Program Alarm light, store Alarm Code 1406, and go immediately to the final Automatic Request routine (ROO). This alarm can occur only in P63 or P64.
  - b. If an overflow occurs anywhere in the landing equations the LGC will turn on the Program Alarm light, store Alarm Code 1410, stop all vehicle attitude rates, and continue. This alarm can occur only in P63, P64, or P66.

26. This program allows manual control of LM attitude and the selection of P66. During P63 (P64) the astronaut can display the PGNS total guidance error on the FDAO error needles (Attitude Monitor switch in PGNS) by having keyed in V62E through the DSKY. He can then steer out the PGNS P63 attitude errors with the PGNS manually (Guidance Control switch in PGNS and the PGNS Mode Control switch in Attitude Hold); or automatically (PGNS Mode Control switch in Auto); or with the AGS manually (Guidance Control switch in AGS and the AGS Mode Control switch in Attitude Hold).

NOTE: If the astronaut hits the ROD (Rate of Descent) switch while the PGNS Mode Control switch is in Attitude Hold, the LGC will irrevocably transfer him out of the automatic guidance program modes (P63 and P64) into the ROD program (P66).

#### Sequence of Events:

V37E63E

Flashing	Time to Go in Braking Phase	XXbXX min/s
V06N61	Time from Ignition	XXbXX min/s
	Crossrange Distance	XXXX.Xnmi
		(+Landing Site north of S/C)

N33E

Flashing	Time of Ignition	00XXX. h
V06N33		000XX. min
		0XX.XX s

KEY REL

PRO

Flashing	Desired Automaneuver FDAO Ball	R XXX.XX deg
V50N18	Angles	P XXX.XX deg
		Y XXX.XX deg

Automaneuver: Guidance Control – PGNS  
Mode Control – PGNS, Auto

PRO

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

V06N18	Desired Automaneuver to FDAO Ball Angles	R XXX.XX deg P XXX.XX deg Y XXX.XX deg
--------	---	--

Monitor maneuver to previous angles displayed.

Flashing V50N18	Desired Automaneuver to FDAO Ball Angles	R XXX.XX deg P XXX.XX deg Y XXX.XX deg
--------------------	---	--

ENTER

V06N62	Absolute Value of Velocity Time from Ignition Delta V accumulated	XXXX.X ft/s XXbXX min/s XXXX.X ft/s
--------	---	---

Time from ignition keeps counting down until TIG -35 seconds. DSKY blanks for 5 seconds and V06N62 display returns at TIG -30 seconds.

V06N62	Absolute Value of Velocity Time from Ignition Delta V accumulated	XXXX.X ft/s XXbXX min/s XXXX.X ft/s
--------	---	---

TIG -5 seconds.

Flashing V99N62	(Same as above display)
--------------------	-------------------------

PRO Astronaut okays ignition.

Flashing V06N63	Delta Altitude Altitude Rate Computed Altitude	XXXX.X ft XXXX.X ft/s XXXXXX. ft
--------------------	--	--

At approximately 42,000 feet computed altitude, ALT and VEL lights – Off.

V57E At approximately 40,000 feet computed altitude.

V06N63	Delta Altitude Altitude Rate Computed Altitude	XXXX.X ft XXXX.X ft/s XXXXXX. ft
--------	--	--

At approximately ignition +9:30, P64 automatically entered and P64 displayed.

## P64 – APPROACH PHASE

Purpose:

1. To control the PGNS during the thrusting of the powered landing maneuver between HI gate and LO gate.
2. To control the DPS thrust and attitude between HI gate and LO gate.
3. To provide the crew with the capability of redesignating the landing site to which the PGNS is guiding the LM.

Assumptions:

1. The LM is on the powered landing descent between HI gate and LO gate.
2. The CSM is in a near circular orbit around the moon at a nominal altitude of 60 nautical miles. The CSM is maintaining a preferred tracking attitude for optical tracking of the LM.
3. The Landing Radar (LR) is on, checked out, and should have been providing to the LGC velocity and range information with respect to the moon. This information should have been incorporated into the LM state vector. The LGC/LR operation is under the control of the Descent State Vector Update routine (R 12) which is already in process.
4. The entire powered landing maneuver (braking, approach, and landing) will be accomplished using the OPS engine.
5. The aim conditions (LO gate) for the approach phase are stored in the LGC.
6. The LM state vector has been stored in the LGC since initialization by ERASAB LE register load. The LGC has updated this as required during thrusting. No further state vector updates from any source other than the LR will be accepted by this program.
7. The DPS is not throttles over the whole range from 0 to maximum. It must be operated either at maximum throttle or over a specific throttle range of lower settings. These throttle settings are total throttle settings; that is, the sum of the manual setting (whose minimum is about 10 percent) and the PGNS commanded setting.

This program assumes the Throttle Control switch to be in Auto (the DPS receives the sum of the manual and PGNS commanded settings) and the manual throttle to be set at a level less than that required by the LGC.

Nominally, if the Approach Phase program is completed without any redesignation of the landing site (see Assumption 10), the throttle will remain within the allowable throttle range throughout the phase. Excessive target redesignations during this program, however, may result in required throttle excursions outside the allowable range. In such cases the LGC will command maximum throttle for at least 2 seconds,

- and until the required throttle setting returns to the permitted throttle region.
8. During the powered landing maneuver, the LGC will monitor the presence or absence of the Auto Stabilization discrete. This discrete is issued to the LGC when the Mode Control switch is in the Auto position.

The LGC will also monitor the presence or absence of the Auto Throttle discrete. This discrete is issued to the LGC when the Thrust Control switch is in the Auto position.

Should either of these discretes be interrupted during the powered landing maneuver, the LGC assumes that it no longer has complete automatic control of the maneuver.

The monitor and the associated LGC logic is included in the Landing Auto Modes Monitor routine (R13) which is already in process.

The LGC can be forced to ignore the absence of the Auto Throttle discrete and continue issuing normal throttle commands by setting the CHANBACKUP location (0374) in the computer to 0001 X8. This location can only be set by astronaut or ground loading and is not changed by Fresh Start or Restart.

- 9.
10. During most of the approach phase, the LGC provides the crew with the option to redesignate the landing site to which the PGNS is guiding the LM. This option is called the Landing Point Designator (LPD) mode. The PGNS Mode Control switch must be in Auto for the ACA to function as a landing site redesignator.

The landing point redesignation, if exercised, is based upon visual assessment of the lunar terrain with respect to the presently designated landing site. During the LPD mode the present landing site is displayed on the DSKY in terms of coordinates on the LPD sighting grid on the left hand LM window (LPD angle). Landing site redesignations are manually put into the computer via the attitude controller on an incremental basis; that is, a limit switch actuation in the attitude controller causes the LGC to redesignate the landing site at a fixed angular increment (1 degree in elevation, 1 degree in azimuth) from the present LM/landing site. The applicable attitude controller polarities are:

- a. -Pitch Rotation gives -LPD Elevation (new site beyond present site).
  - b. +Pitch Rotation gives +LPD Elevation (new site short of present site).
  - c. +Roll Rotation gives +LPD Azimuth (new site to right of present site).
  - d. -Roll Rotation gives -LPD Azimuth (new site to left of present site).
11. The initial maneuver of the approach phase is the LM attitude transition from the LM attitude at the start of P64 to a satisfactory attitude for landing site visibility. After the completion of this maneuver the LM attitude is constrained by thrust

pointing requirements and is controlled about the thrust axis so as to maintain the current landing site in the LM X-Z plane. The conditions achieved at the start of P64 should be such that the thrust pointing requirements of the approach phase will yield satisfactory visibility and radar orientations.

The landing site becomes visible to the command pilot if the "look" angle (the angle between the -X LM axis and the LOS to the landing site) is greater than 25 degrees and the LOS is in or near the LM X-Z plane.

At any time during P63 or P64, the magnitude of the look angle and the orientation of the look angle plane (that plane containing the LOS and the LM X axis) are defined by the inertial orientation of the LM X axis and the position of the LM with respect to the landing site.

The inertial orientation of the LM X axis is controlled by requirements of thrust vector control. The orientation of the LM windows with respect to the look angle plane is controlled by rotation of the vehicle about the LM X axis.

12. The crew has the capability to display LGC calculated values of forward velocity, lateral velocity, altitude, and altitude rate on certain LM meters during this program. The calculation of these parameters is under control of the Landing Analog Display routine which is already in process.
13. The Rate of Descent (ROD) mode is not enabled during this program.
14. An abort from the lunar descent may be required at any time during the descent orbit injection, the descent coast, or the powered descent (P63), (P64), or (P66).

For aborts after DPS ignition for the powered landing maneuver, time is critical. During this period an abort is nominally commanded by pushing one of two buttons in the LM. The abort may be commanded to use the descent stage (Abort button) or the ascent stage (Abort Stage button). If the descent stage is selected, and the DPS propellant approaches exhaustion, control must be switched to the ascent stage by the crew by ascent stage selection (Abort Stage button).

During the powered landing maneuver the LGC will continuously monitor the Abort and the Abort Stage discretes, and upon receipt of either will terminate the program in process and call the appropriate abort program (DPS Abort program (P70) or APS Abort program (P71)). Both abort programs will guide the LM to an acceptable orbit.

Monitoring the Abort and Abort Stage buttons is controlled by the Abort Discretes Monitor routine (R 11) which is already in process.

This step can be locked out by setting the CHANBACKUP location (0374) in the computer to OOOX18. This location can only be set by astronaut or ground loading

- and is not changed by Fresh Start or Restart.
15. If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Auto, the PGNS controls the total vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise control about only the yaw axis with the ACA (X-axis override) provided the X-axis override capability is permitted.
- If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS holds the vehicle attitude and generates either Mode 1 or Mode 2 attitude errors for display on the FOAI. The crew may exercise manual control about all vehicle axes with the ACA using either the Rate Command or Minimum Impulse mode. However, it is strongly recommended that powered flight not be attempted in the Minimum Impulse mode.
16. Control of the LM DPS, RCS, and APS is transferred from the PGNS to the Abort Guidance System (AGS) by placing the Guidance Control switch from PGNS to AGS.
- The AGS will be capable of taking over control of the LM during any portion of the lunar descent or ascent or during either of the abort programs (P70 or P71). The AGS will guide the LM to a safe orbit.
- The AGS may be initialized by the LGC at any time during this program by manual selection of the AGS Initialization routine (R47). However, it is not recommended that the AGS be initialized during powered flight because DAP attitude control is interrupted during the CDU zero part of the routine.
- In the event that the Guidance Control switch is changed from PGNS to AGS during a thrusting maneuver, the LGC will continue computation of position and velocity, the desired thrust vector, and the desired attitude errors. However, the PGNS will not be responsible if register overflows occur within the LGC.
17. The PGNS generates two types of errors for display on the F DAI as selected by the astronaut:
- Mode 1 -Selected by Verb 61. Autopilot following errors are used as a monitor of the DAP's ability to track automatic steering commands.
  - Mode 2-Selected by Verb 62. Total attitude errors used to assist the crew in manually maneuvering the vehicle.
18. The Load OAP Data routine (R03) has been performed prior to the start of the powered landing maneuver and should not be required during this program.
19. This program is automatically selected by the Braking Phase program (P63) at the completion of the P63 aim conditions.
20. Two alarm conditions may be originated by the PGNS powered landing equations:

- a. If Subroutine ROOTSP RS in the RG/VG calculation fails to converge in 8 passes the LGC will turn on the Program Alarm light, store Alarm Code 1406, and go immediately to the Final Automatic Request routine (ROO). This alarm can occur only in P63 or P64.
- b. If an overflow occurs anywhere in the landing equations the LGC will turn on the Program Alarm light, store Alarm Code 1410, stop all vehicle attitude rates, and continue. This alarm can occur only in P63, P64, or P66.
21. This program allows manual control of the LM attitude. If manual control is desired, put the PGNS Mode Control switch in Attitude Hold and use the ACA to control the LM attitude.
- If P66 is desired, click the ROD switch while the PGNS Mode Control switch is in Attitude Hold. The ACA does not redesignate the landing site while the Mode Control switch is in Attitude Hold. To use the ACA to redesignate the landing site, put the Mode Control in Auto and rotate the ACA in the desired direction.
- NOTE : Landing Site Redesignation must be completed before P66 is selected because P64 cannot be reentered once it has been exited.
22. The crew can select a display of the LGC computed throttle setting by keying V16 N92.

#### Sequence of Events:

Flashing V06N64	Time Left for Redesignations/LPD Angle Altitude Rate Computed Altitude	XXbXX deg XXXX.X ft/s XXXXX ft
--------------------	--	--------------------------------------

#### Manual Throttle Control

TTCA – Advance until thrust = 10%, throttle control – MAN

V16N92E

Flashing V16N92	Percent of Full Thrust (10,500 lb) Altitude Rate	00XXX% XXXX.X ft/s XXXXX ft
--------------------	---	-----------------------------------

To return to auto throttle

Throttle Control – AUTO

TTCA – minimum position

KEY REL

Flashing      Same display as above  
V06N64

Manual Attitude Check  
Mode Control (PGNS – Attitude Hold)

To use Landing Point designator  
Verify Mode Control PGNS – AUTO

PRO

V06N64	Time Left for Redesignations/LPD Angle	XXbXX s/deg
	Altitude Rate	XXXX.X ft/s
	Computed Altitude	XXXXXX ft

Redesignate landing site as described (+ pitch redesignates landing site toward LM by 1 degree) (+roll redesignates new site to right of present site by 1 degree in azimuth.) V06N64 changes the elevation LPD angle accordingly.

Manual Rate of Descent Control

PGNS - Attitude Hold. Activate ROD switch. Automatic Transfer to P66.

Automatic Transfer to ROD Control

When time remaining is zero - Automatic transfer to P66 occurs.

## P66 – LANDING PHASE

Purpose:

1. To modify the rate of descent of the LM (with respect to the lunar surface) in response to astronaut originated inputs via the LM Rate of Descent (ROD) switch to the LGC.
2. To modify the inertial attitude of the LM in response to astronaut originated inputs via the attitude controller only if the Mode Control switch is in Attitude Hold.
3. To null the forward and lateral surface velocities of the LM when the Mode Control switch is in Auto and still respond to the Rate of Descent (ROD) switch inputs.
4. To update the LM state vector with vehicle acceleration and Landing Radar (LR) Data.

Assumptions:

1. The LM is in the late stages of landing, with a low inertial velocity.
2. The Landing Radar (LR) is on, checked out, and providing to the LGC velocity and range information with respect to the moon. This information has been incorporated into the LM State Vector. The LGC/LR operation is under the control of the Descent State Vector Update routine (R 12) which is already in process.
3. The entire powered landing maneuver (braking, approach, and landing) will be accomplished using the DPS engine.
4. The LM State Vector has been stored in the LGC since initialization by erasable register load. The LGC has updated this as required during thrusting. No further state vector updates from any source other than the LR will be accepted by this program.
5. The DPS is not thottlable over the whole range from 0 percent to maximum. It must be operated either at maximum throttle or over a specific throttle range of lower settings. These throttle settings are total throttle settings; that is, the sum of the manual setting (whose minimum is 10 percent) and the PGNS commanded setting.

This program assumes the Throttle Control switch to be in Auto (the DPS receives the sum of the manual and PGNS commanded settings) and the manual throttle to be set at a level less than that required by the LGC.

- Nominally the throttle will remain within the allowable throttle range through this program.
6. During the powered landing maneuver, the LGC will monitor the presence or absence of the Auto Stabilization discrete. This discrete is issued to the LGC when the Mode Control switch is in the Auto position.

In the Auto Stabilization mode, the PGNS will operate to null the forward and

lateral surface velocities by controlling the inertial attitude of the spacecraft.

In the Attitude Hold mode, the LGC will hold an inertial attitude. However, the attitude may be changed by manual control via the attitude controller.

7. The LPD option is not provided to the crew during this program.
8. The crew can display LGC calculated value of forward velocity, lateral velocity altitude, and altitude rate during this program. The calculation of these parameters is under the control of the Landing Analog Displays routine ( R 10 ) which is already in process.
9. During this program the LGC monitors the output of the Rate of Descent (ROD) switch in the LM. This switch is operated by the astronaut in response to his assessment of the present LM rate of descent based on out-of-window references and LM/DSKY displays.

Switch operation is on an incremental basis: + (increase ROD) or - (decrease ROD). Each command results in an LGC-commanded change of "ROD SCALE" in LM rate of descent. (ROD SCALE is a value loaded into erasable storage prior to flight. Presently 1 foot per second.)

10. An abort from the lunar descent may be required at any time during descent coast or powered descent (P63, P64, or P66).

For aborts after DPS ignition for the powered landing maneuver, time is critical. During this period an abort is commanded by pushing one of two buttons in the LM. The abort may be commanded to use the descent stage (Abort button) or the ascent stage (Abort Stage button). If the descent stage is selected, and the DPS propellant approaches exhaustion, control must be switched to the ascent stage by the crew by ascent stage selection (Abort Stage button).

During the powered landing maneuver, the LGC will continuously monitor the Abort and Abort Stage discretes, and upon receipt of either will terminate the program in process and call the appropriate abort program (DPS Abort program (P70) or APS Abort program (P71)). Both abort programs will guide the LM to an acceptable orbit.

Monitoring the Abort and Abort Stage buttons is controlled by the Abort Discretes Monitor routine ( R 11 ) which is already in process.

This step can be locked out by setting the CHANBACKUP location (0374) in the computer to OOOX18. This location can only be set by astronaut or ground loading and is not changed by Fresh Start or Restart.

11. If a thrusting maneuver is performed with the Guidance Control switch in PGNS and the Mode Control switch in Attitude Hold, the PGNS will hold the vehicle attitude and will generate either Mode 1 or Mode 2 attitude errors for display on the FOAI.

The crew may exercise manual attitude control about all vehicle axes with the ACA in either the Rate Command or Minimum Impulse mode. It is strongly recommended that powered flight not be attempted in the Minimum Impulse mode.

The LGC is not permitted to compute body rates via R60 during this program. The attitude will always be available for astronaut display so that they are aware of the impending S/C motion when switching from Attitude Hold to Auto.

12. Control of the LM DPS, RCS, and APS is transferred from the PGNS to the Abort Guidance System (AGS) by placing the Guidance Control switch from PGNS to AGS.

The AGS will be capable of taking over control of the LM during any portion of the lunar descent or ascent or during either of the abort programs (P70 or P71). The AGS will guide the LM to a safe orbit.

The AGS may be initialized by the LGC at any time by manual selection of the AGS Initialization routine (R47). However, it is not recommended that the AGS be initialized during powered flight because DAP attitude control is interrupted during CDU Zero in that routine.

In the event the Guidance Control switch is changed from PGNS to AGS during a thrusting maneuver, the LGC will continue computation of position and velocity, the desired thrust vector, and the desired attitude errors; however, the PGNS will not be responsible if register overflows occur within the LGC.

13. The Load DAP Data routine (R03) has been performed prior to the start of the powered landing maneuver and should not be required during this program.
14. This program is automatically selected by the Landing Auto Modes Monitor routine (R13) during the powered landing maneuver when:
  - a. The targeted conditions for P64 are met (either automatically or astronaut flown).
  - b. When the Rate of Descent (ROD) switch is activated by the astronaut after P63 throttle up in Attitude Hold.

Once this program has been selected it is no longer possible to return to the completely automatic powered landing programs (P63 or P64).

15. The crew has the capability to select a display of the LGC computed throttle setting by keying in V16N92.

Sequence of Events:

# LUNAR MODULE

## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing	Forward Velocity	XXXX.X ft/s
V06N60	Altitude Rate	XXXX X ft/s
	Computed Altitude	XXXXXX ft

Use ROD switch as desired.

To manually null forward and lateral velocities

Mode Control: PGNS - Attitude Hold

Forward (pitch) and lateral (roll ) cross pointers

Manual Throttle

TTCA - Advance until thrust = 10%

Throttle Control – Manual

V16N92E

Flashing	Percent of Full Thrust	00XXX%
V16N92	Altitude Rate	XXXX.X ft/s
	Altitude	XXXXXX ft

To return to auto throttle

Throttle Control - Auto

TTCA - minimum position

KEY REL

Flashing	Forward Velocity	XXXX.X ft/s
V06N60	Altitude Rate	XXXX.X ft/s
	Computed Altitude	XXXXXX ft

At height actual = 5.6 ft., lunar contact light - ON

ENGINE STOP-PUSH

PRO

ENGINE ARM-OFF

## P68 – LANDING CONFIRMATION

Purpose:

1. To terminate landing program and DAP functions.
2. To initialize the LGC for lunar surface operation.
3. To permit the astronaut to prevent RCS jet firings on the lunar surface.

Assumptions:

1. This program is selected by the astronaut by DSKY entry. It is to be selected only after the LM has landed on the lunar surface (Program P66)
2. V37E68E selection of P68 will terminate Average G and command the engine off (see ROO)
3. The selection of this program places the DAP in the Minimum Impulse mode. As long as the astronaut keeps the mode control in Attitude Hold, RCS jet firings will not occur, even while the platform is being torqued (in P57).
4. This program will not shut off the DAP. However, the attitude errors are zeroed and the maximum deadband is set. No jet firings should result until one of the following occurs in sufficient magnitude to cause the attitude errors to exceed the deadband:
  - a. The moon rotates.
  - b. The LM shifts on the lunar surface
  - c. The IMU gyros are torqued for alignment by P57,
  - d. The IMU drifts.

The DAP may be shut off by setting the Mode-Control-PGNS switch to Off.

Sequence of Events:

V37E68E

Flashing	Latitude	XXX.XX deg (+north)
V06N43	Longitude	XXX.XX deg (+east)
	Altitude	XXXX.X nmi

PRO

V76E, Mode Control (PGNS) – Attitude Hold, No DAP light on.

# LUNAR MODULE

FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

Flashing      Select New Program  
V37

## 10. PROCEDURES

### 10.1 RCS HOT/COLD FIRING TEST

This section will cover what you need to know to perform the RCS HOT/COLD FIRING TEST in the Subsystems Activation checklist.

#### 10.1.1 RCS JETS & LGC CHANNELS

The computer can be used to control the automatic and semi-automatic flight modes of the Lunar Module. This section will go through these details, but it's good to understand how the computer works before diving into this.

All of the RCS jets has logical names assigned to them. The LGC can send commands to these to either turn them on, or turn them off.

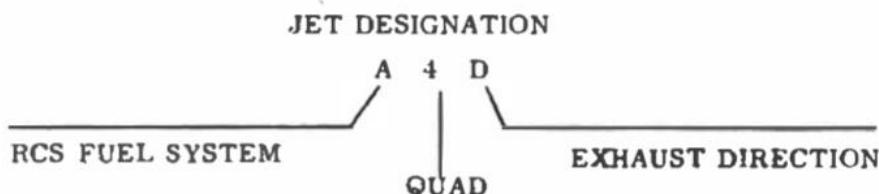
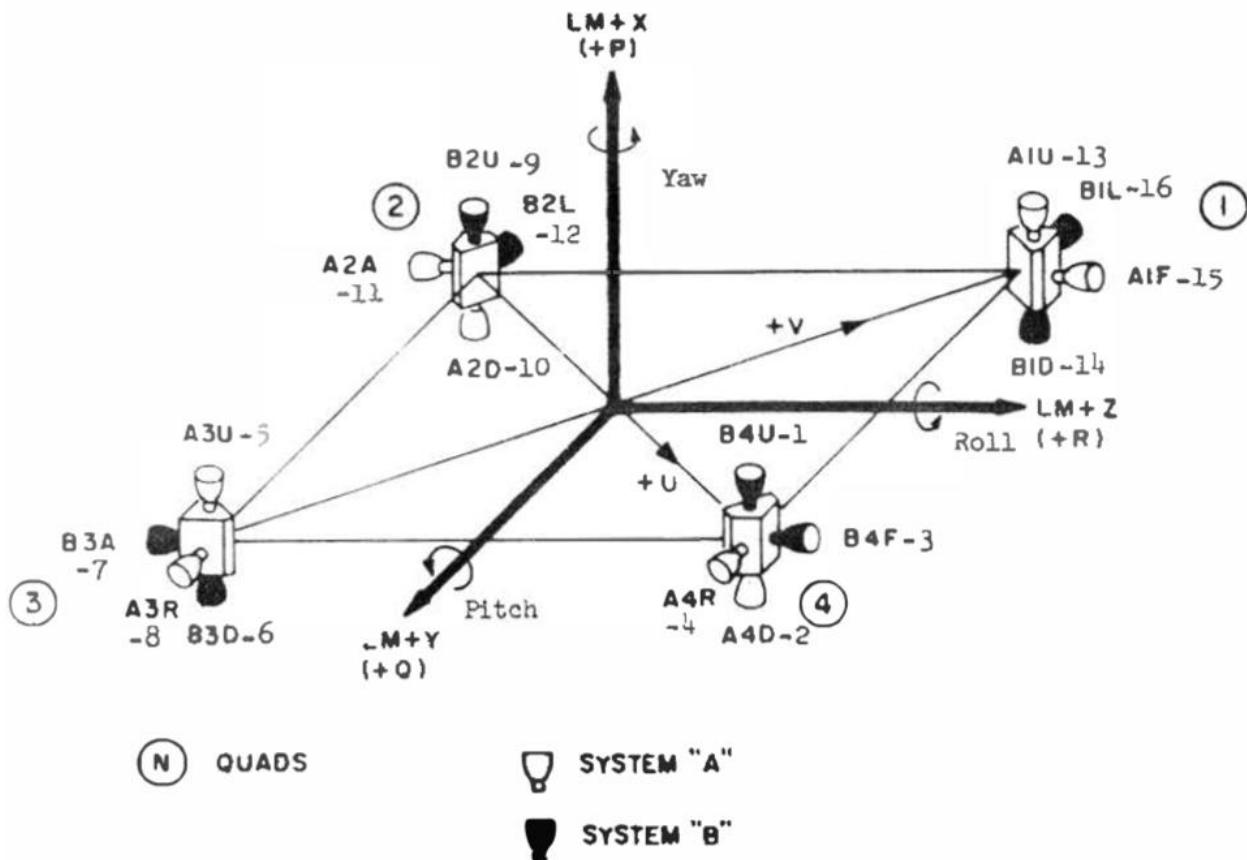
# LUNAR MODULE

FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

## LM RCS JET CONTROL AXES



Two output channels in the LGC is used for this:

Output channel No. 5: LGC jet-on commands (up and down firing), and UV attitude

Output channel No. 6: LGC jet-on commands (right, left, forward, and aft firing), and P attitude.

These channels are a sequence of binary numbers, where bits 1 to 8 is used.

## OUTPUT CHANNEL 5

BIT	JET DESIGNATION	JET NO.	ROTATION EFFECT	(RCS JET FIRINGS) TRANSLATION EFFECT
1	B4U	1	+V	-X
2	A4D	2	-V	+X
3	A3U	5	+U	-X
4	B3D	6	-U	+X
5	B2U	9	-V	-X
6	A2D	10	+V	+X
7	A1U	13	-U	-X
8	B1D	14	+U	+X

Bits 9 - 15 not used

## OUTPUT CHANNEL 6

BIT	JET DESIGNATION	JET NO.	ROTATION EFFECT	(RCS JET FIRINGS) TRANSLATION EFFECT
1	B3A	7	+P	+Z
2	B4F	3	-P	-Z
3	A1F	15	+P	-Z
4	A2A	11	-P	+Z
5	B2L	12	+P	+Y
6	A3R	8	-P	-Y
7	A4R	4	+P	-Y
8	B1L	16	-P	+Y

These channels can be monitored using the LGC, and is a part of the HOT/COLD RCS FIRE TEST of the Subsystems Activation checklist. Since these values are in binary, we can easily translate this into the octal system, to be able to visualize this in the registers of the LGC.

To monitor these channels, you can key V11 N10E 5E or 6E on the DSKY.

The above means the following:

V11: Verb 11, Monitor Octal Component 1 in R1

N10: Noun 10, Channel to be specified

ENTER

As Noun 10 needs another digit to know what channel to monitor, you can now enter the channel. It will appear in register 3 on the LGC display.

Key 5 and ENTER to monitor channel 5, or 6 and ENTER to monitor channel 6.

Channel 31 is used to monitor the minimum impulse in-bits, and has the following table:

## INPUT CHANNEL 31

## BIT

1	+EL (LPD), + PMI
2	-EL (LPD), - PMI
3	+ YMI
4	-YMI
5	+AZ (LPD), +RMI
6	-AZ (LPD), -RMI
7	+X Translation
8	-X Translation
9	+Y Translation
10	-Y Translation
11	+Z Translation
12	-Z Translation
13	Attitude Hold
14	Auto Stabilization
15	Attitude Control Out of Detent

## NOTE:

All of the input signals in Channel 31 are inverted; that is, a ZERO bit indicates that the discrete is ON.

Channel No. 31. This input channel has 15 bit positions and uses inverted logic.

- Bit positions No. 1 and 2 indicate positive and negative pitch manual input commands, respectively, from the ACA. These bits are used for elevation changes when the landing point designator (LPD) is used.
- Bit positions No. 3 and 4 indicate positive and negative yaw manual input commands, respectively, from the ACA.
- Bit positions No. 5 and 6 indicate positive and negative roll manual input commands, respectively, from the ACA. These bits are used for azimuth changes when the LPD.
- Bit positions No. 7 through 12 indicate positive and negative X-, Y -, and Z-translation commands from the TTCA. These signals command LM translation by on-and-off firing of the thrusters, under LGC control.
- Bit position No. 13 indicates that the CES if operating in the attitude hold mode.
- Bit position No. 14 indicates that the C ES is operating in the automatic mode.
- Bit position No. 15 informs that LGC that the ACA is out of detent.

This can be read using keying **V11 N10E 31E**.

## 10.1.2 MINIMUM IMPULSE COMMAND MODE

Key V76 to enter the Minimum Impulse Command Mode. This requires that the RCS is set up, and that the control mode is set to Att Hold PGNS.

### 10.1.3 RATE COMMAND AND ATTITUDE HOLD MODE

Key V77 to enter the Rate Command and Attitude Hold. This requires that the RCS is set up, and that the control mode is set to Att Hold PGNS.

## 10.2 THE DIGITAL AUTOPILOT SETUP

The Digital Autopilot (DAP) is used by the LGC to aid in attitude control. It's quite simple, and allows you to set limits on rate and deadbands etc. It is set up using Verb 48. To start the setup, enter V48E into the computer.

This will start the DAP Setup routine. The first setup is to enter the DAP configuration string. Once you enter V48E, the verb 4 and noun 46 is flashing, and showing data in register 1 and register 2.  
FL V04 N46

Register 1 has the content of a five-digit number, where each number represents a logical mode, and Register 2 has a two-digit number. The position of each number can be read like this:

R1: ABCDE

R2: 000IJ

Below is a list indicating what each of the digits represents:

**A: What is the stage configuration:**

A = 0 - Ascent stage only

A = 1 - Ascent & Descent stages

A = 2 - Ascent & Descent stages docked with CSM

**B: Jet selection**

B = 0 - Two-jet translation & roll/pitch minimum impulse (RCS system A)

B = 1 - Two-jet translation & roll/pitch minimum impulse (RCS system B)

B = 2 - Four-jet translation (Minimum impulse/RSC sys A)

B = 3 - Four-jet translation (Minimum impulse/RCS sys B)

**C: Limit**

C = 0 - Fine scaling ACA - 4 deg/sec (LM only) -

0.4 deg/sec (LM & CSM)

C = 1 - Normal scaling ACA - 20 deg/sec (LM only)

2.0 deg/sec (LM & CSM)

**D: Attitude deadband**

D = 0 - Attitude deadband 0.3 degrees

D = 1 - Attitude deadband 1.0 degrees

D = 2 - Attitude deadband 5.0 degrees

**E = KALCHANU rate**

0 - 0.2 deg/sec

1 - 0.5 deg/sec

2 - 2.0 deg/sec

3 - 10.0 deg/sec

The next register has two digits, where each digit is representing the following:

I = 0 - Test channel 30, bit 5 for AUTO THROTTLE discret & act accordingly

I = 1 - Ignore AUTO THROTTLE discrete & assure auto throttle is desired.

J = 0 - Test channel 30, bits 1 & 4 for ABORT & ABORT STAGE discrete& & act accordingly.

J = 1 - Ignore ABORT & ABORT STAGE discretes.

To modify, use the normal modification method on the LDC:

Key V21E to change register 1, remember to set the sign first, then follow with the 5 digits.

Key V22E to change register 2, remember to set the sign first, then follow three zeros, and then with the 2 digits- Load desired DAP data code and/or channel 30 back-up code

If you are happy with the setup, you can accept it.

Accept: Key PRO

When accepted, the next screen expects you to enter the weights, received by MSFN.

FL V06 N47

R1 LM weight XXXXX lb

R2 CSM weight XXXXX lb

R3 ----

Accept: Key PRO - If DPS has been staged, exit RO3

Reject: Key V24E - Load desired parameters If descent stage is attached, proceed to step 4.

# V. MAIN PROPULSION



# V. MAIN PROPULSION

## 1. GENERAL

The Main Propulsion Subsystem (MPS) consists of the descent propulsion section (DPS) and the ascent propulsion section (APS). Each section is complete and independent of the other and consists of a liquid-propellant rocket engine with its own propellant storage, pressurization, and feed components. The DPS provides the thrust to control descent to the lunar surface. The APS provides the thrust for ascent from the lunar surface. In case of mission abort, the APS and/or DPS can place the LM into a rendezvous trajectory with the CSM from any point in the descent trajectory; there is a deadman zone immediately above the lunar surface, where abort cannot be accomplished. The choice of engine to be used depends on the cause for abort, on how long the descent engine has been operating, and on the quantity of propellant remaining in the descent stage.

Both propulsion sections use identical hypergolic propellants: a 50-50 mixture, by weight, of hydrazine ( $N_2H_4$ ) and unsymmetrical dimethylhydrazine (UDMH) as the fuel; nitrogen tetroxide ( $N_2O_4$ ), as the oxidizer. The injection ratio of oxidizer to fuel is approximately 1.6 to 1, by weight.

Basic operation of the two propulsion sections is similar. In each section, gaseous helium forces the propellants from their tanks, through propellant shutoff valves, to the engine injectors. The DPS uses supercritical helium for propellant pressurization; the APS uses ambient gaseous helium. The primary reason for using supercritical helium is the weight saving. Both the descent and ascent engine assemblies consist of a combustion chamber, where the propellants are mixed and burned; an injector that determines the spray pattern of the propellants injected into the combustion chamber; and propellant control valves and orifices that meter, start, and stop propellant flow to the engine upon command. The descent engine, which is larger and produces more thrust than the ascent engine, is throttleable for thrust control and is gimbaled for thrust vector control. The ascent engine is neither throttleable nor gimbaled. Redundancy of vital components in both propulsion sections provides a high reliability factor.

Before starting the descent or ascent engine, proper propellant settling must be established. This is accomplished by moving the LM in the +X-direction to cause the propellants to settle at the bottom of the tanks. As the propellants are consumed, tank ullage increases and more propellant settling time is required for each subsequent engine start. The +X-translation is accomplished by operating the Reaction Control Subsystem (RCS) downward-firing thrust chamber assemblies (TCA's). Two or all four downward-firing TCA's can be selected, depending upon whether RCS propellant conservation (two TCA's) or a shorter RCS firing time (four TCA's) is the major consideration.

## 2. DESCENT PROPULSION

### 2.1 DESCENT PROPULSION INTERFACES

The DPS receives 28-volt d-c and 115-volt a-c primary power through the Commander's and LM Pilot's buses of the Electrical Power Subsystem (EPS). The outputs of the DPS pressure and temperature transducers and liquid-level sensors are processed in the Instrumentation Subsystem (IS) and are transmitted via the Communications Subsystem (CS) to MSFN. The IS also processes the DPS caution and warning and display signals. The Explosive Devices Subsystem (EDS) opens explosive valves in the DPS to enable propellant tank pressurization and venting.

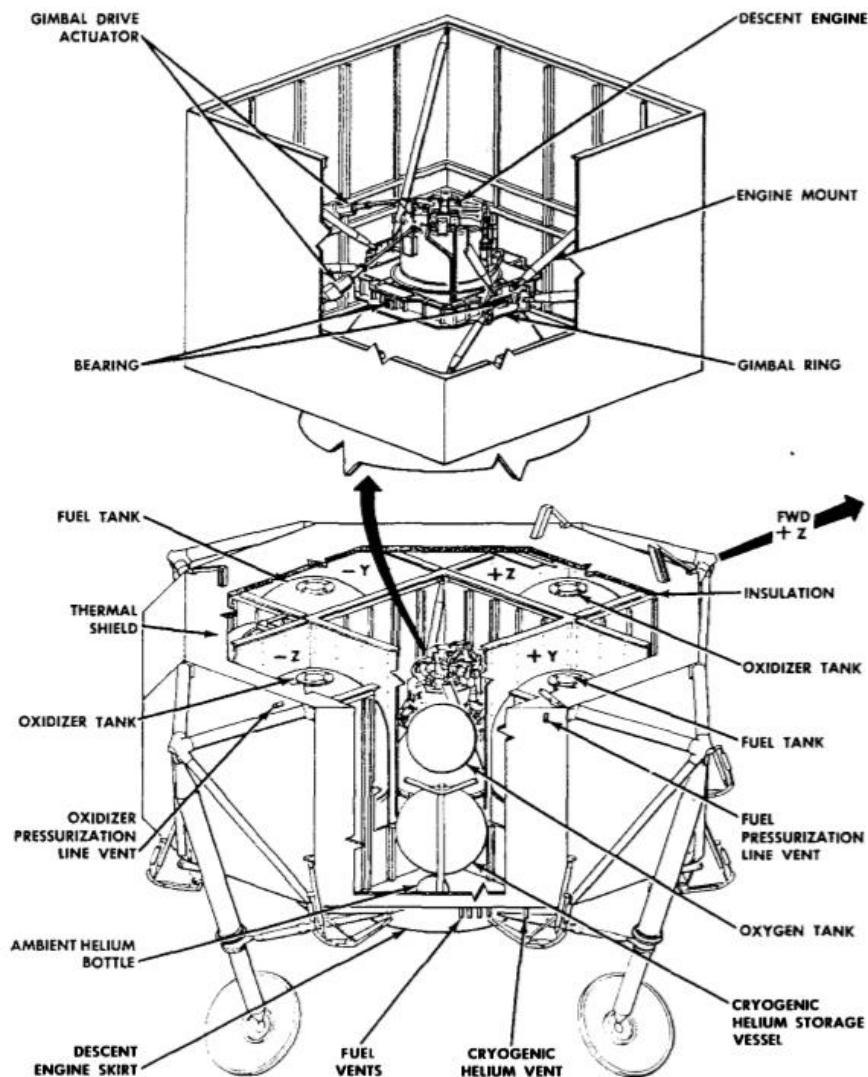


Figure 2.1.1 – Descent Propulsion Engine

The Guidance, Navigation and Control Subsystem (GN&CS) issues automatic on and off commands, gimbal drive actuator commands, and thrust level commands to the descent engine. The automatic on and off commands and thrust level commands can be overridden manually.

Descent engine arming and ignition are controlled by automatic guidance equipment, or by the astronauts through the stabilization and control (S&C) control assembly and the descent engine control assembly (DECA). A descent engine arm signal is sent to the S&C control assembly when an astronaut sets the ENG ARM switch (panel 1) to DES or when he presses the ABORT pushbutton (panel 1) preparatory to starting a mission abort program, using the descent engine.

Engine-on signals from the LM guidance computer (LGC) or abort guidance section (AGS) are sent to the DECA through the S&C control assembly. The DECA turns the descent engine on upon receiving the arm and the engine-on signals. If DECA power fails, the DES ENG CMD OVRD switch (panel 3), in the ON position, will supply an alternate voltage source to keep the engine firing. The engine remains on until the engine-off command is received from the automatic guidance equipment. The astronauts can also generate engine on and off commands manually; these commands override the automatic commands. A manual start is accomplished (after propellant tank pressurization with ambient helium) by arming the descent engine and pressing the START pushbutton (panel 5). Either astronaut can shut off the descent engine by pressing his STOP pushbutton (panels 5 and 6) or by pressing the ABORT STAGE pushbutton (panel 1). An abort-stage command results in immediate descent engine shutdown, automatically followed by ascent propellant tank pressurization, and enabling of circuitry for stage separation and ascent engine firing. Stage separation and ascent engine firing occurs when the ascent engine-on command is issued.

Descent engine throttling is controlled by the LGC or the astronauts. The throttling-range limitations are from minimum thrust (approximately 10% of 10,500 pounds) to approximately 65% and full throttle (approximately 92.5%). The range between 65% and 92.5% is a transient region that cannot be used for extended periods because excessive engine erosion occurs in this zone. Under normal conditions, the engine cannot be operated in the transient region because automatic throttle commands above 65% automatically produce a full throttle output. Only in case of malfunction can inadvertent throttling occur in the transient region, in which case manual correction must be made. Automatic throttle increase and decrease signals from the LGC are sent to an integrating counter in the DECA. The analog output of the DECA controls descent engine thrust. In the automatic mode, the thrust/translation controller assemblies (TTCA's) can be used by the astronauts to increase descent engine throttle (overriding the automatic throttle command); the TTCA's cannot be used, however, to decrease the throttle command. In the manual throttle mode, the astronauts have complete control over descent engine thrust.

The primary guidance and navigation section (PGNS) of the GN&CS, or the AGS, automatically controls descent engine gimbal trim, to compensate for center-of-gravity offsets during descent

engine firing. In PGNS operation, the LGC sends trim on and off signals in two directions, for each gimbal axis, to the DECA. These signals operate power control circuitry, which drives the two gimbal drive actuators (GDA's). In AGS operation, Y- and Z-axis error signals from the attitude and translation control assembly (ATCA) are sent to the DECA to drive the GDA's. The GDA's tilt the descent engine along the Y-axis and Z-axis a maximum of 6° from the center position. The ENG GMBL switch (panel 3) permits removing GDA power to interrupt the tilt capability if the ENG GMBL caution light (panel 2) goes on, indicating a malfunction.

## 2.2 DESCENT PROPULSION FUNCTIONAL DESCRIPTION

The DPS consists of an ambient helium bottle and a cryogenic helium storage vessel with associated helium pressurization components; two fuel and two oxidizer tanks with associated feed components; and a pressure-fed, ablative, throttleable rocket engine. The engine can be shut down and restarted, within operational limitations and restrictions, as required by the mission. At the fixed full-throttle position, the engine develops a nominal thrust of 9,870 pounds; it can also be operated within a nominal range of 1,050 to 6,800 pounds of thrust.

The engine is mounted in the center compartment of the descent stage cruciform; it is suspended, at the throat of the combustion chamber, on a gimbal ring that is part of the engine assembly. The gimbal ring is pivoted in the descent stage structure, along an axis normal to that of the engine pivots. The engine can be tilted up to +6° or -6°, by means of the GDA's, to ensure that the thrust vector passes through the LM center of gravity.

Functionally, the DPS can be subdivided into a pressurization section, a propellant feed section, and an engine assembly.

### 2.2.1 PRESSURIZATION

Before earth launch, the propellant tanks are only partly pressurized, so that the tanks will be maintained within a safe pressure level under the temperature changes that can occur between the time the tanks are loaded and launch. Before initial engine start, the ullage space in each propellant tank requires additional pressurization. This initial pressurization (prepressurization) is accomplished with ambient helium. (Supercritical helium cannot be used because the helium circulating through the fuel/helium heat exchanger may freeze the fuel before fuel flow is established.)

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A pressure transducer at the outlet port of the ambient helium bottle supplies a signal through the HELIUM MON selector switch (panel 1), when set to AMB PRESS, to the HELIUM indicator (panel 1) to enable the astronauts to check the status of the bottle before initial engine start.



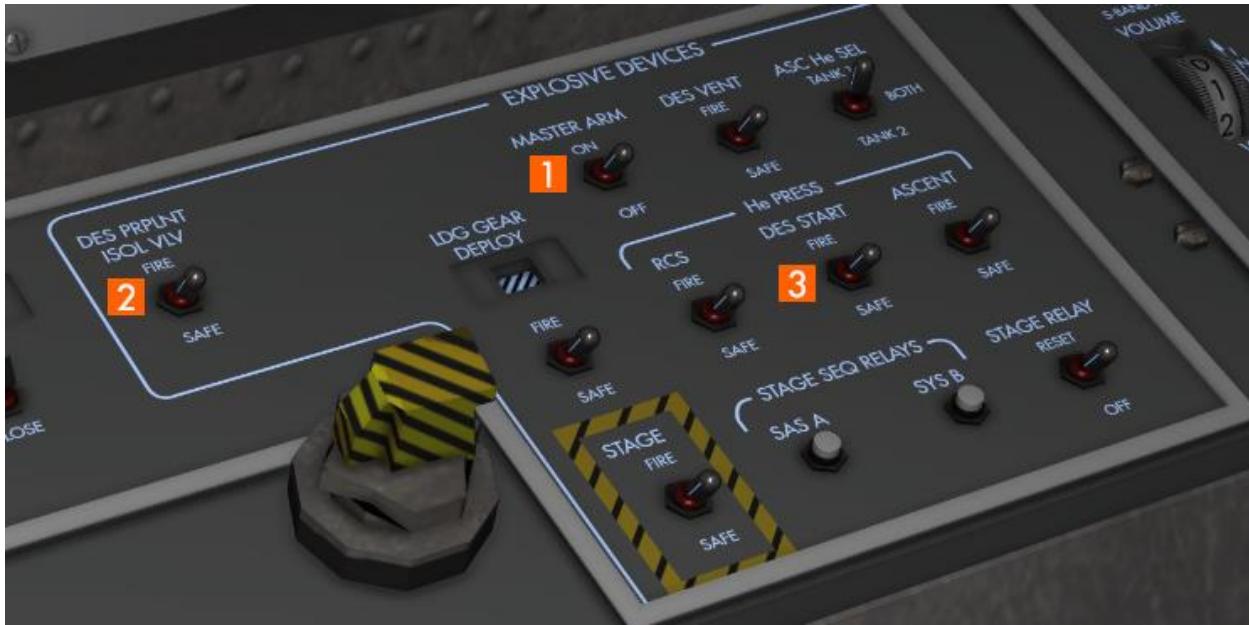
The propellant tanks are prepressurized by opening explosive valves in the ambient helium line and in the lines leading to the fuel and oxidizer tanks. The valve in the ambient helium line prevents helium flow from the storage bottle before prepressurization. The compatibility valves in the lines leading to the fuel and oxidizer tanks prevent propellant vapors from degrading the upstream components due to prolonged exposure before pressurization.

# LUNAR MODULE

## FLIGHT MANUAL

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AN ORBITAL SIMULATOR



After setting the MASTER ARM switch (panel 8) to ON [1], the DES PRPLNT ISOL VLV switch (panel 8) is set to FIRE [2] to open the fuel and oxidizer compatibility valves. The DES START He PRESS switch [3] (panel 8) is then set to FIRE, opening the ambient helium isolation explosive valve. Ambient helium flows from the storage bottle through the open explosive valve and through a filter, where debris from the explosive valve is trapped. The ambient helium enters the main pressurization line downstream of the normally closed helium isolation solenoid valve and flows through the secondary pressure regulator where pressure is reduced to approximately 245 psi. The regulated ambient helium then enters the propellant tanks to provide the normal ullage pressure.

After pressurization with ambient helium, the supercritical helium from the cryogenic storage vessel is used to maintain ullage pressure. Supercritical helium is stored at a density approximately eight times that of ambient helium. Because heat transfer from the outside to the inside of the cryogenic storage vessel causes a gradual increase in pressure (approximately 6 to 10 psi per hour, depending on ambient conditions). Only a component malfunction, a significant time slippage, or a change in the predetermined burn profile can cause the supercritical helium to approach, or exceed, the limits of the pressure/time envelope. An out-of-limit condition may result in rupture of the dual burst disks which may cause the descent engine to operate in a blowdown mode. (The period of engine operation in the blowdown mode depends upon the amount of ullage volume present.)



The cryogenic storage vessel is isolated by an explosive valve, which is fired automatically after an engine-on command has been given. With the ENG ARM switch [A] (panel 1) set to DES, the engine-on command can be given manually by pressing the START pushbutton (panel 5), or it can be given automatically by the LGC or AGS. In either case, the MASTER ARM switch must be in the ON position to fire the explosive valve. A delay circuit causes a 1.3-second

delay between opening of the propellant shutoff valves and firing of the supercritical helium isolation explosive valve. This time delay prevents the supercritical helium from entering the fuel/helium heat exchanger until fuel flow is established so that freezing of the fuel in the heat exchanger cannot occur.

The supercritical helium initially passes through the first loop of the two-pass fuel/helium heat exchanger. Here it absorbs heat from the fuel that is routed from the fuel tanks through the heat exchanger, before ultimate delivery to the engine. The helium is warmed to approximately -200° F and routed back through the helium/helium heat exchanger inside the cryogenic helium storage vessel. The -200° F helium transfers heat to the remaining supercritical helium in the vessel, causing an increase in pressure in the vessel that ensures continuous expulsion of helium throughout the entire period of operation. After passing through the helium/helium heat exchanger, where it is cooled to approximately -300° F, the helium is routed back through the second loop of the fuel/helium heat exchanger and heated to approximately +350 F before delivery to the pressure regulators.

With the supercritical helium pressurization system operating, the pressure in the cryogenic helium storage vessel varies between 400 and 1,750 psia. The pressure is monitored on the HELIUM indicator when the HELIUM MON selector switch is set to SUPCRIT PRESS. The cryogenic storage vessel is protected against overpressurization by a dual burst disk assembly. If an excessive heat transfer through the vessel wall increases the internal pressure above approximately 1,900 psia, the burst disks rupture and the entire helium supply is lost. A normally open vent relief valve between the two burst disks protects against back pressurization of the upstream burst disk if it develops a small leak. If a large leak develops, the vent relief valve closes and the downstream burst disk protects the storage vessel. A thrust neutralizer at the outlet of the downstream burst disk prevents generation of unidirectional thrust if the burst disks rupture.

Downstream of the fuel/helium heat exchanger, the helium flow continues through a filter that traps debris from the explosive valve, then the pressurization line divides into two parallel legs. A normally open, latching solenoid valve and a pressure regulator are in series in the primary leg; a normally closed, latching solenoid valve is in series with a pressure regulator in the secondary leg. The pressure of the helium flowing through the primary leg is reduced by the pressure regulator to the nominal pressure (245 psia) required to pressurize the propellant tanks. If this regulator fails open or closed, pressure at the helium manifold increases or decreases accordingly beyond acceptable limits (rises above 260 psia or drops below 220 psia) and the DES REG warning light (panel 1) goes on. When a caution or warning light goes on, a signal is routed from the caution and warning electronics assembly (CWEA) in the IS to light the MASTER ALARM pushbutton/lights (panels 1 and 2) and to provide a warning tone. Pressing either MASTER ALARM pushbutton turns off both lights and terminates the tone, but has no effect on the caution or warning light. (The DES REG warning light is inhibited before initial descent engine arming. It will go off when normal pressure is restored when the CWEA circuit breaker is cycled, or when the ascent and descent stages separate.)



Under regulator failure conditions, the astronauts must close the solenoid valve in the malfunctioning leg and open the solenoid valve in the redundant leg, to restore normal propellant tank pressurization. The normally open (primary) solenoid valve is closed by momentarily setting the DESCENT He REG 1 switch (panel 1) to CLOSE; the DESCENT He REG 1 talkback above the switch then provides a barber-pole display. The normally closed (secondary) solenoid valve is opened by momentarily setting the DESCENT He REG 2 switch to OPEN; the DESCENT He REG 2 talkback above the switch then provides a gray display. (Both solenoid valves may be closed during the coast periods of descent, to prevent inadvertent tank overpressurization due to possible helium leakage through the pressure regulators and to inhibit leaks downstream of the latching valves.)

The primary and secondary helium flow paths merge downstream of the regulators to form a common helium pressurization manifold. Transducers monitor the manifold pressure; they provide continuous telemetry signals to MSFN, and signals that cause the DES REG warning light to go on when the sensed pressure exceeds 260 psia or drops below 220 psia. The manifold routes the helium into two flow paths: one path leads to the oxidizer tanks; the other, to the fuel tanks. Each path has a quadruple check valve assembly in a series-parallel arrangement. The quadruple check valves isolate the upstream components from the corrosive propellant vapors and prevent hypergolic action, as a result of backflow from the propellant tanks, in the helium pressurization manifold. After passing through the compatibility explosive valves, the helium flows into the top of the fuel and oxidizer tanks. Diffusers at the top of the tanks uniformly

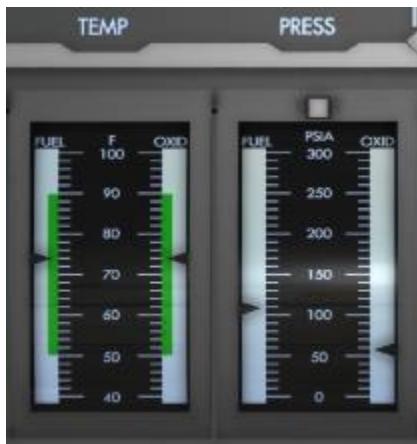
distribute the helium throughout the ullage space. Helium crossover lines maintain a balanced ullage pressure in the tanks containing the same propellants.

Immediately upstream and in parallel with the propellant tanks, each helium flow path contains a relief valve assembly to protect the propellant tanks against overpressurization. The assemblies (a burst disk in series with a relief valve) vent pressure in excess of approximately 275 psia and reseal the flow paths after overpressurization is relieved (254 psia). Thrust neutralizers eliminate unidirectional thrust generated by the escaping gas. To prevent leakage through single point relief valves during normal operation, the burst disks are located upstream of the relief valves. The burst disks rupture at a pressure between 260 and 275 psi; the relief valves open fully at 275 psi to pass the entire helium flow from a failed-open regulator preventing damage to the propellant tanks.

Two vent lines, in parallel with the relief valve assemblies, include an explosive valve in series with a normally open solenoid valve for each propellant tank. The vent lines are intended for planned depressurization of the tanks after lunar landing, when temperature rise of the supercritical helium and heat soak-back from the engine (after shutdown) causes pressure buildup in the tanks. The planned venting arrangement protects the astronauts against untimely venting of the tanks through the relief valve assemblies. The fumes are vented overboard, through the relief valve thrust neutralizers at the fuel and oxidizer pressurization line vents. If the helium pressurization line is open, the supercritical helium in the cryogenic storage vessel will be vented together with the propellant tanks. The supercritical helium will vent rapidly until pressure drops to approximately 350 psia, then the pressure remaining in the cryogenic storage vessel will decrease with the decreasing propellant tank pressures. To open the vent lines, the MASTER ARM switch is set to ON and the DES VENT switch (panel 8) is set momentarily to FIRE, opening both explosive vent valves simultaneously. The MASTER ARM switch is then set to OFF. Venting of the lines is monitored by setting the PRPLNT TEMP/PRESS MON switch (panel 1) to DES 1 and the HELIUM MON selector switch (panel 1) to SUPCRIT PRESS. When the OXID PRESS indicator indicates less than 20 psia, the HELIUM MON selector switch is set to OFF, the OXID VENT switch is set to CLOSE, and the OXID VENT talkback will change to a barber-pole display. When the FUEL PRESS indicator indicates less than 8 psia, the FUEL VENT switch is set to CLOSE, causing the FUEL VENT talkback to provide a barber-pole display.

### 2.2.2 PREOPELLANT FEED

Each pair of propellant tanks (containing like propellants) is manifolded into a common delivery line. Balanced propellant flow is maintained by trim orifices in all propellant lines downstream of the tanks.



Helium pressure in the propellant tanks is monitored on the FUEL and OXID PRESS indicators (panel 1), propellant temperature in the tanks is monitored on the FUEL and OXID TEMP indicators. The PRPLNT TEMP/PRESS MON switch selects the set of fuel and oxidizer tanks (No. 1 or No. 2) for monitoring. Each propellant tank has its own temperature transducer to supply temperature signals to the indicator. One pressure transducer in the fuel pressurization line and one in the oxidizer pressurization line supply pressure signals to the indicators. Therefore, the pressure reading remains constant regardless of whether tank No. 1 or 2 monitored. Propellant quantity remaining in the tanks is monitored on the OXIDIZER and FUEL QUANTITY indicators (panel 1). The PRPLNT QTY MON switch selects the set of fuel and oxidizer tanks (No. 1 or 2) for monitoring.



Pressurized helium, acting on the surface of the propellant, forces the fuel and oxidizer into the delivery lines through a propellant retention device that maintains the propellant in the delivery lines during negative-g acceleration (up to acceleration in excess of -2g). The oxidizer is piped directly to the engine assembly; the fuel circulates through the fuel/helium heat exchanger before it is routed to the engine assembly. A small bypass permits some fuel to reach the engine without flowing through the heat exchanger. This protects against a pressure buildup should the fuel in the heat exchanger be frozen. Each delivery line contains a trim orifice and a filter. The trim orifices provide engine interface pressure of approximately 222 psia at full throttle position for proper propellant use. The filters prevent debris, originating at the explosive valves or in the propellant tanks, from contaminating downstream components.

### 2.2.2.1 PREOPPELLANT QUANTITY GAGING SYSTEM

The propellant quantity gaging system (PQGS) enables the astronauts to continuously monitor the quantity of propellants remaining in the four tanks. The PQGS is of the capacitance type. It consists of four quantity-sensing probes with low-level sensors, a control unit, two QUANTITY



indicators, the PRPLNT QTY MON switch, and the DES QTY warning light. During a lunar-landing mission, the PQGS will be turned on approximately 10 seconds before engine ignition and shut off approximately 10 seconds after engine shutdown. The continuous PQGS power-on time is limited to 45 minutes. This limitation safeguards the thermal capability of the electronic components which, if exceeded, could result in



erroneous indications. The PROPUL: PQGS circuit breaker (panel 16) is used to apply or remove PQGS power. The PRPLNT QTY MON switch selects a set of propellant tanks (fuel and oxidizer tanks No. 1 or 2) to be monitored on the FUEL and OXIDIZER QUANTITY indicators. With the PRPLNT QTY MON switch set to OFF, the QUANTITY indicators remain lit; however, the digital readouts on the indicators blank out. With the

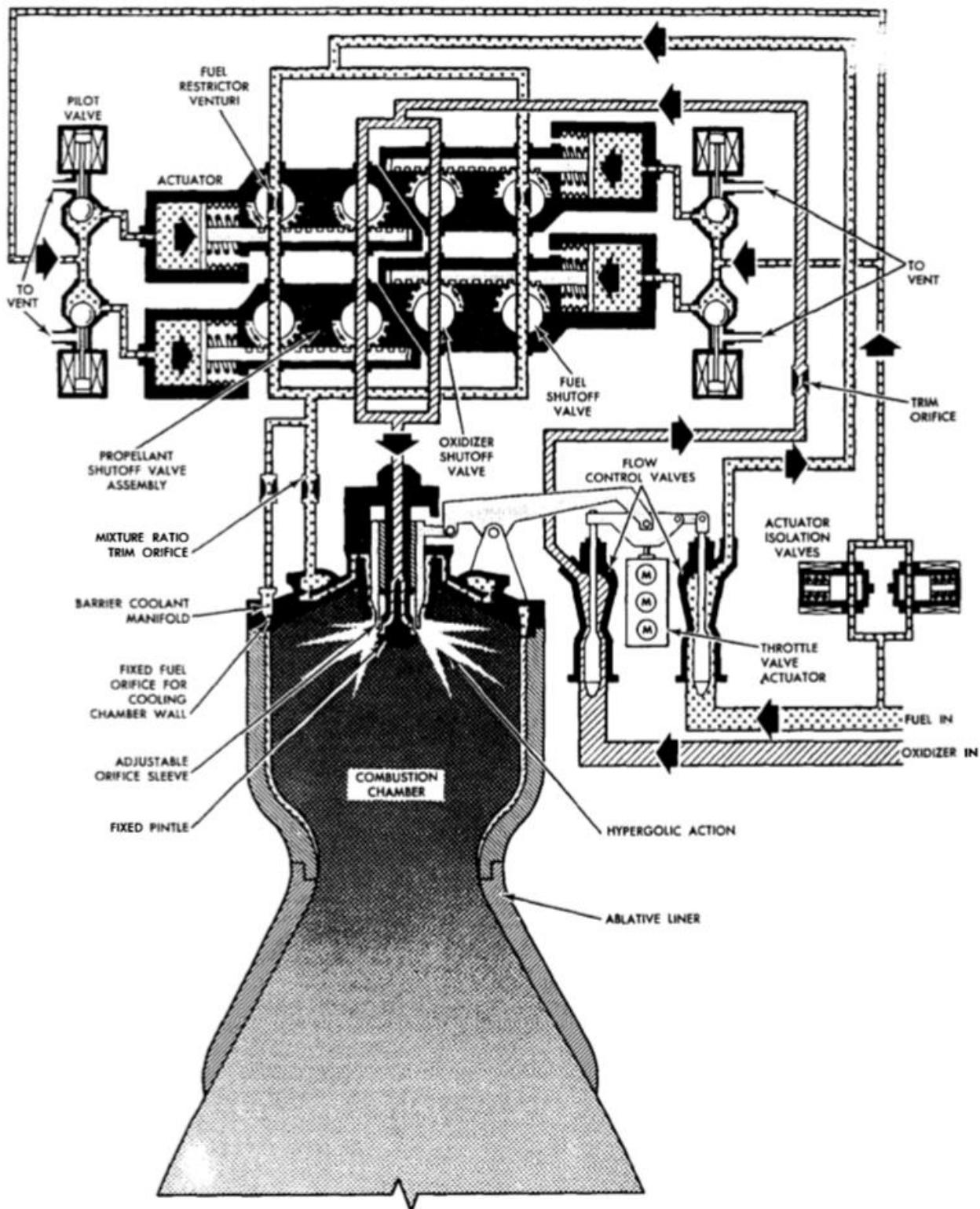
PRPLNT QTY MON switch set to DES 1 or DES 2 and the descent engine shut off, the QUANTITY indicator readings remain stable until a zero-g condition develops, at which time the readings drift and become indeterminate.

The low-level sensors provide a discrete signal to cause the DES QTY warning light to go on when the propellant level in any tank is down to 9.4 inches (equivalent to 5.6% propellant remaining, or sufficient for 116 seconds of engine burn at hover thrust (approximately 25%)). The MASTER ALARM pushbutton/light and the warning tone are not activated when the DES QTY warning light is energized to prevent distraction of the astronauts during the most critical phase of the lunar landing mission. The PQGS has an estimated uncertainty tolerance of 1.3% of full tank capacity for cabin display and telemetry transmission. This tolerance is reduced to 1% in the 8% to 25% propellant quantity range where the PQGS performs at a higher accuracy.

### 2.2.3 ENGINE ASSEMBLY

Fuel and oxidizer enter the engine assembly through interface flanges on opposite sides of the engine. The fuel line has a tap-off branch (pilot valves actuation line) that leads through two actuator isolation valves (arranged in parallel for redundancy) to the four solenoid-operated

pilot valves. The fuel in this line is routed, through the pilot valves, to the actuators, where it is used as actuation fluid to open the propellant shutoff valves. The main fuel and oxidizer flow is routed through respective flow valves, then each flow path splits into two parallel paths that route the propellants through the redundant propellant shutoff valves. The propellant shutoff valve assemblies are in a series-parallel arrangement. The series redundancy prevents open failure; the parallel redundancy prevents closed failure. The valves open simultaneously to permit propellant flow to the engine while it is operating; they close simultaneously to terminate propellant flow at engine shutdown. At the two upstream fuel shutoff valves, venturis restrict the fuel flow so that the oxidizer reaches the injector between 40 and 50 milliseconds before the fuel. This precludes the possibility of a fuel lead, which would result in rough engine starts. Downstream of the propellant shutoff valves, the parallel paths merge to form one fuel and one oxidizer path. The fuel passes through a final trim orifice and enters the variable-area injector manifold, where a concentric annulus of fuel flow is formed. The oxidizer is routed, through the center element of the injector, to the combustion chamber, where it mixes with the fuel for combustion.



Before initial engine operation and during engine shutdown, the solenoid-operated actuator isolation valves (pre-valves) are closed to prevent possible fuel loss in the pilot valve actuation line due to leakage at the pilot valves. The actuator isolation valves are opened by setting the

ENG ARM switch to DES. This enables the actuation fuel to flow to the pilot valves just before the pilot valves are opened.

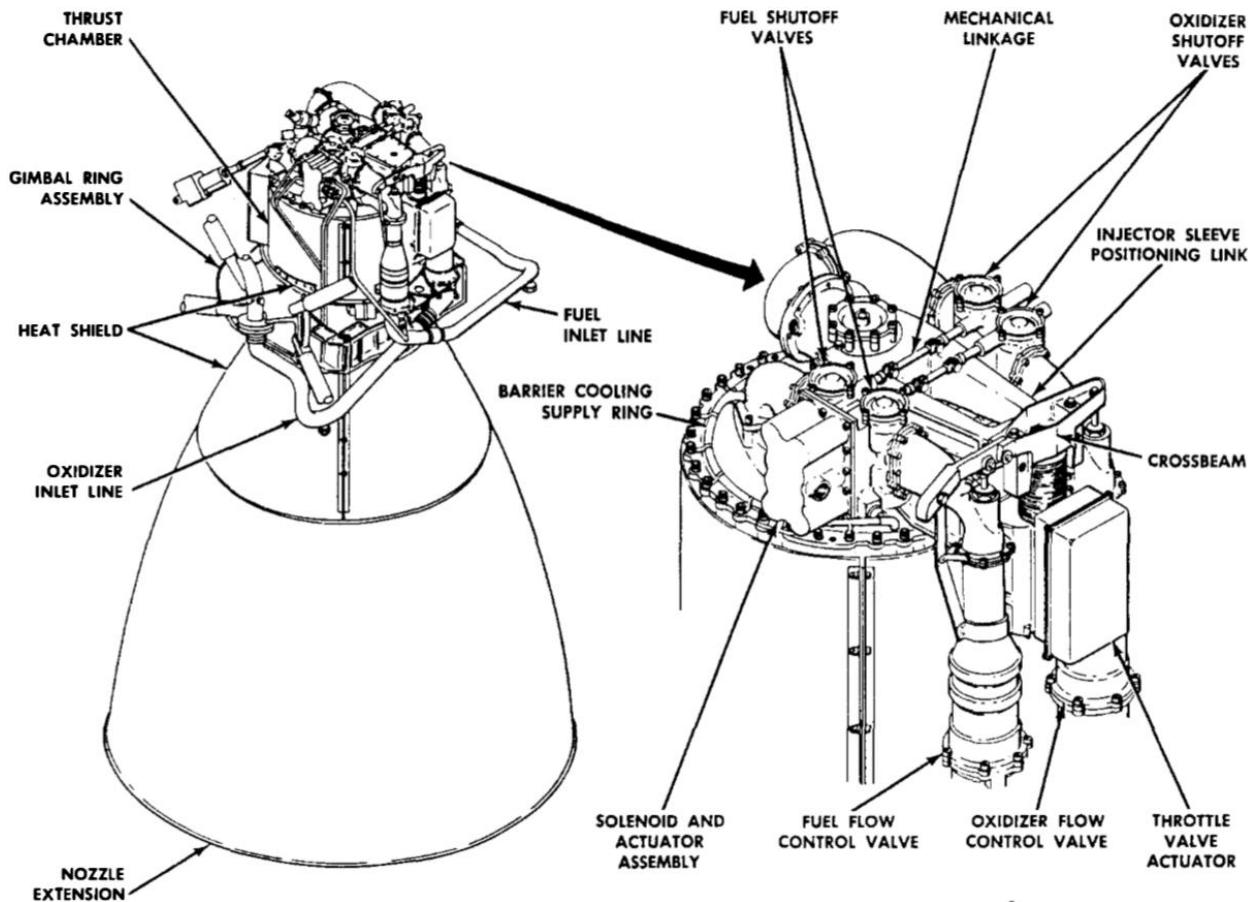


When the START pushbutton is pressed (or an engine-on command initiated), the four solenoid-operated pilot valves open simultaneously, permitting the actuation fuel to open the propellant shutoff valves, thus routing fuel and oxidizer to the combustion chamber. During the start, the solenoids in the pilot valves unseat the caged balls from the inlet ports and seat them against the overboard vent ports; fuel enters the actuator cavities. The actuator pistons are connected to rack-and-pinion linkages that rotate the balls of the shutoff valves 90° to the open position to permit propellant flow to the injector. The series-parallel redundancy in the valve arrangement provides for positive start and shutdown. During shutdown, the solenoids in the pilot valves are deenergized and the vent ports are open. The spring-loaded actuators close the shutoff valves. Residual actuation fuel is vented overboard through four separate lines that lead to vent ports at the bottom of the descent stage.

The propellant in the main fuel and oxidizer lines flows through cavitating-venturi-type flow control valves that control propellant flow to the engine below the 65% throttle setting. Transition from cavitation to noncavitation occurs between 70% and 80%. At full throttle, and during momentary transition through the full throttle to 65% range, engine throttling takes place primarily in the pintle assembly of the injector and in the flow control valves. At approximately 70% of maximum thrust, cavitation commences in the throats of the flow control valves, causing the valves to function as cavitating venturis down to minimum thrust. Once

cavitation begins, the propellant-metering function is entirely removed from the injector; flow is controlled entirely by the flow control valves.

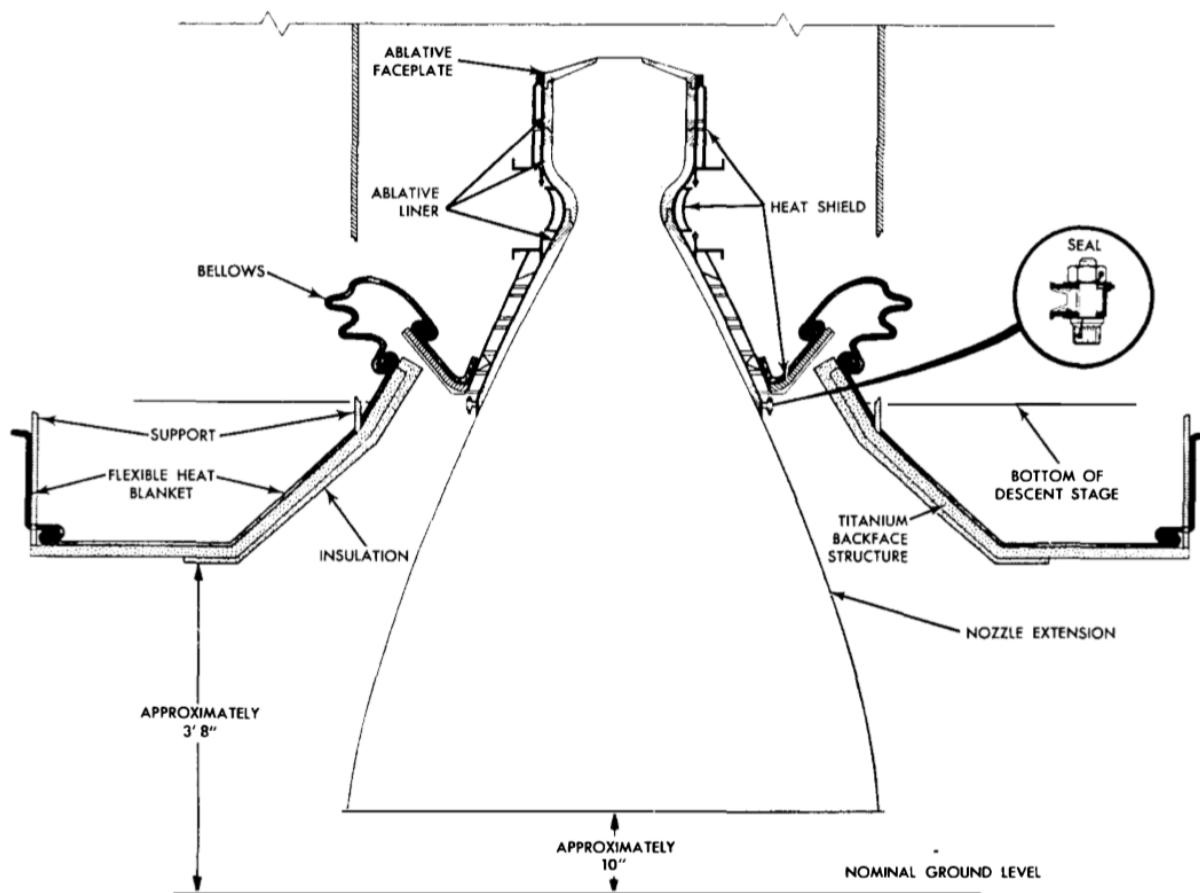
The throat area of the flow control valves is regulated by a close-tolerance, contoured, metering pintle that is linked directly to the injector sleeve. The linkage is operated by a single actuator so that movement of the actuator simultaneously adjusts the pintles in the flow control valves and the movable sleeve in the injector. The fuel and oxidizer are injected at velocities and angles compatible with variations in weight flow. At full throttle, engine operation is conventional. As the engine is throttled down, the pintles in the flow control valves are moved to decrease the flow control area in the venturis so that the pressure drop across the valves balances out the differential between engine and injector inlet pressures. At the same time, the injector orifice areas are decreased so that the injection velocities and impingement angles of fuel and oxidizer are maintained at near-optimum condition for combustion efficiency.



The injector consists of a faceplate and fuel manifold assembly with a coaxial oxidizer feed tube and an adjustable (metering) orifice sleeve. Oxidizer flows through the center tube and out between a fixed pintle and the bottom of the sleeve; the fuel orifice is an annular opening between the sleeve contour and the injector faceplate. The fuel flows behind the face of the injector to cool the faceplate. Some fuel, tapped off the fuel manifold is used for barrier cooling

of the wall. The fuel is emitted in the form of a thin cylindrical sheet; the oxidizer, in a series of individual sprays. The oxidizer sprays break up the fuel stream and establish the injector pattern at all thrust settings.

The mechanical linkage that connects the pintle of the flow control valves and the injector sleeve is pivoted about a fulcrum attached to the injector body. Throttling is controlled by the throttle valve actuator, which positions the linkage in response to electrical input signals. At maximum thrust, the actuator positions the linkage to fully open the flow control valves and injector apertures. As commanded thrust is reduced, the actuator reduces the flow at the flow control valves and moves the injector orifice sleeve to reduce the apertures. As the adjustable orifice sleeve moves upward, the area of the propellant orifices increases.



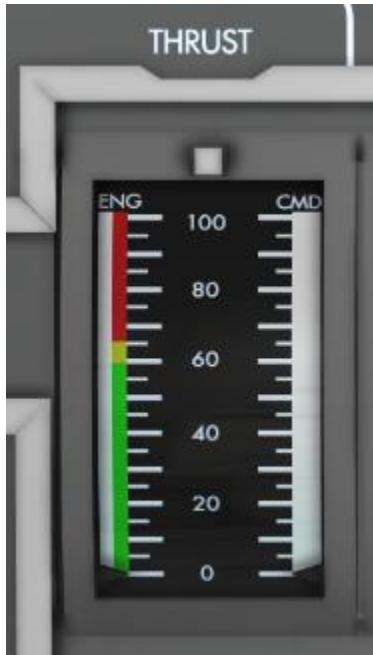
#### 2.2.3.1 DESCENT ENGINE CONTROL

After the engine is manually armed by setting the ENG ARM switch to DES, it can be fired automatically or manually.



Under manual control, the engine can be started and stopped by the Commander by pressing the START and STOP pushbuttons on panel 5; it can be stopped by the LM Pilot by pressing the STOP pushbutton on panel 6. The mode of thrust control is determined by the THR CONT switch [B] (panel 1). When this switch is set to AUTO, engine thrust is controlled by the LGC. When the switch is set to MAN, the Commander's or LM Pilot's TTCA (depending on the setting of

the MAN THROT switch [C], panel 1) controls the engine thrust. In the automatic mode, the TTCA is still operational. It is normally set against its hard (low) stop, where it supplies a 10% thrust command that is summed with the LGC command, resulting in a combined thrust command to the descent engine. For example, if the required thrust is 50%, the LGC commands 40%, which is augmented by the 10% obtained from the TTCA. If the TTCA is moved from the hard stop, it supplies a greater portion of the combined command and the LGC command decreases accordingly. Thus, for the required 50% thrust, the TTCA may now command 20%; the LGC, 30%. If the TTCA is moved to a setting such that it commands more than the required thrust, it overrides the automatic command (the LGC portion becomes zero) and descent engine thrust is determined entirely by the TTCA setting.



The dual-scale CMD THRUST and ENG THRUST indicator (panel 1) displays commanded manual or automatic thrust on the CMD scale and actual engine thrust on the ENG scale. The ENG scale input is derived from a pressure transducer in the combustion chamber, because thrust is proportional to chamber pressure. At full throttle position the ENG scale reads 92.5% (actual full-throttle-position thrust) while the CMD scale reads between 92.5% and 100%. At all other throttle settings (10% to 65% throttling range) the ENG and CMD scales normally display identical readings of the actual engine thrust. Display of dissimilar readings indicates that the engine is not following the thrust commands or that transfer from automatic to manual throttle control is in process. As shown in the example given previously, if the TTCA is displaced from the hard stop in the automatic mode, for 50% required thrust, the TTCA may command 20% while the LGC contributes 30%. The ENG scale of the THRUST indicator will read 50%; however, the CMD scale (where LGC command is summed with a 10% bias) will read 40%.

As the TTCA is moved to increasing throttle settings, the CMD scale readings decrease. When the CMD scale reading has dropped to 10%, the LGC no longer supplies any portion of the thrust command and the TTCA is in control. At this point, a smooth transfer from the automatic mode to the manual mode is accomplished by setting the THR CONT switch to MAN. The CMD and ENG scales will now indicate identical readings. (For the preceding example, both pointers will align at 50%.) Very slight deviations between CMD and ENG scale readings may occur as engine operating time increases. The deviations are due to combustion chamber erosion, which causes chamber pressure to decay slightly.

The T/W (thrust/weight) indicator (panel 1) is used primarily to monitor X-axis acceleration during lunar landing and lift-off. The T/W indicator is a self-contained accelerometer that displays instantaneous X-axis acceleration in lunar-g units (1 lunar g = 5.23 ft/sec<sup>2</sup>). In as much as a given throttle setting provides a specific acceleration when the vehicle has a given mass, the T/W indicator can be used as backup for the THRUST indicators to monitor engine performance.

### 2.3 EXPLOSIVE DEVICES OVERVIEW

The diagram below (Figure 2.3.1) shows the explosive devices described in the preceding sections. The main switches used for operating the descent engine is the following:

#### EXPLOSIVE DEVICES SUBSYSTEM



The three switches, excluding the ARM switch, are used to operate the substances.

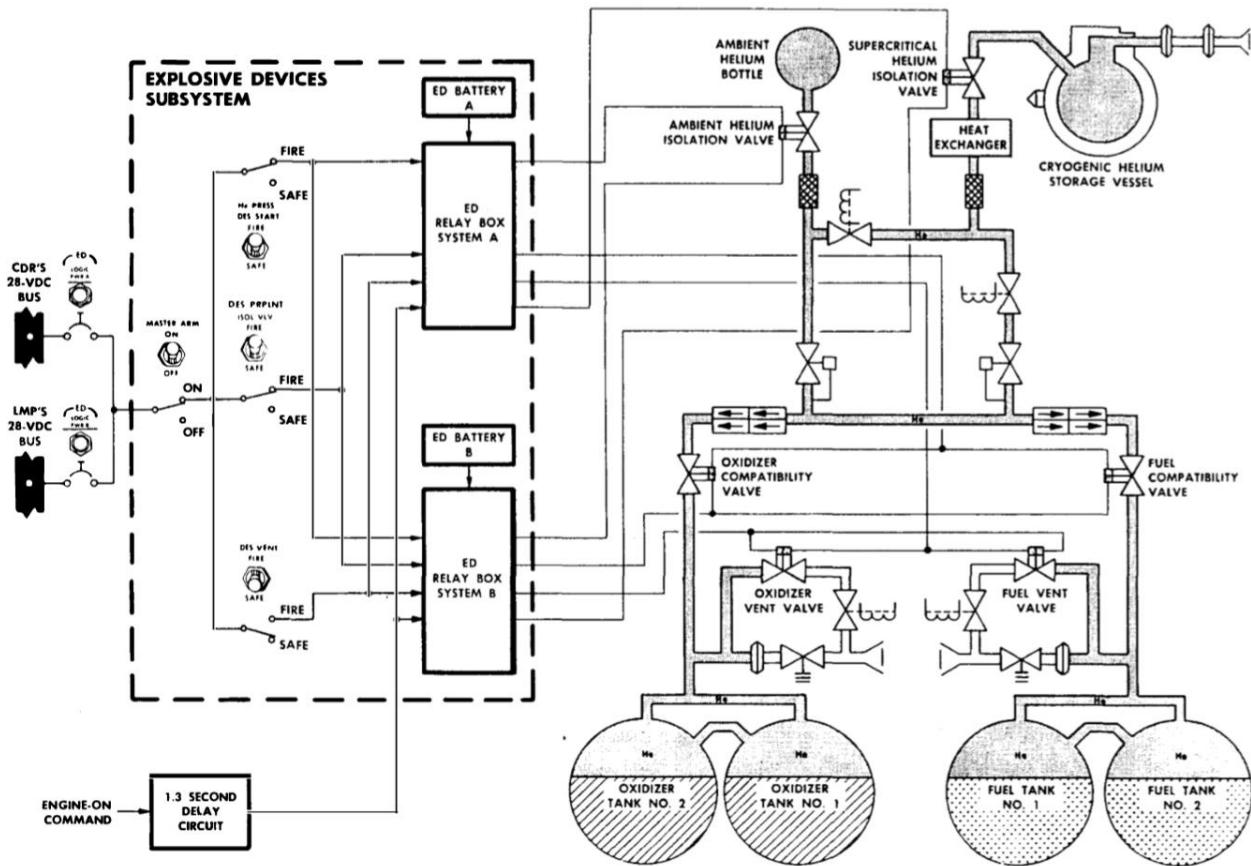


Figure 2.3.1 – Explosive Devices overview

## 2.4 DESCENT PROPULSION SECTION OPERATIONAL LIMITATIONS AND RESTRICTIONS

The operational limitations and restrictions for the DPS are as follows:

- The landing gear must be deployed before descent engine firing. If not deployed, the landing gear would be in the path of the descent engine plume and would be damaged.
- Before descent engine starts, the RCS +X-axis thrusters must be fired to settle the propellants under weightless conditions. Unsettled or insufficiently settled propellants will result in rough or erratic starts that could lead to engine failure.
- Propellant bulk temperatures before descent engine start must be between +50° and +90° F. If temperature limits are exceeded, the resultant rough combustion may adversely affect engine-component performance, and ullage pressure limitations may be exceeded.

- The descent engine must not be operated for prolonged periods in the throttling range of 65% to 92.5%. In this range, operation of the cavitating venturis of the flow control valves becomes unpredictable and may cause an improper fuel-oxidizer mixture ratio, which will result in excessive engine erosion and early combustion chamber burn-through.
- The DPS must not remain pressurized longer than 3.5 days before the anticipated termination of use. If this limit is exceeded, the pressurization section components may fail to operate because of the corrosive nature of the propellants.
- The descent engine combustion chamber must not be subjected to more than 1,100 seconds of engine operation. Exceeding this limitation will cause the engine to operate with a severely charred combustion chamber, possibly resulting in burn-through.

## 3. ASCENT PROPULSION

### 3.1 ASCENT PROPULSION INTERFACES

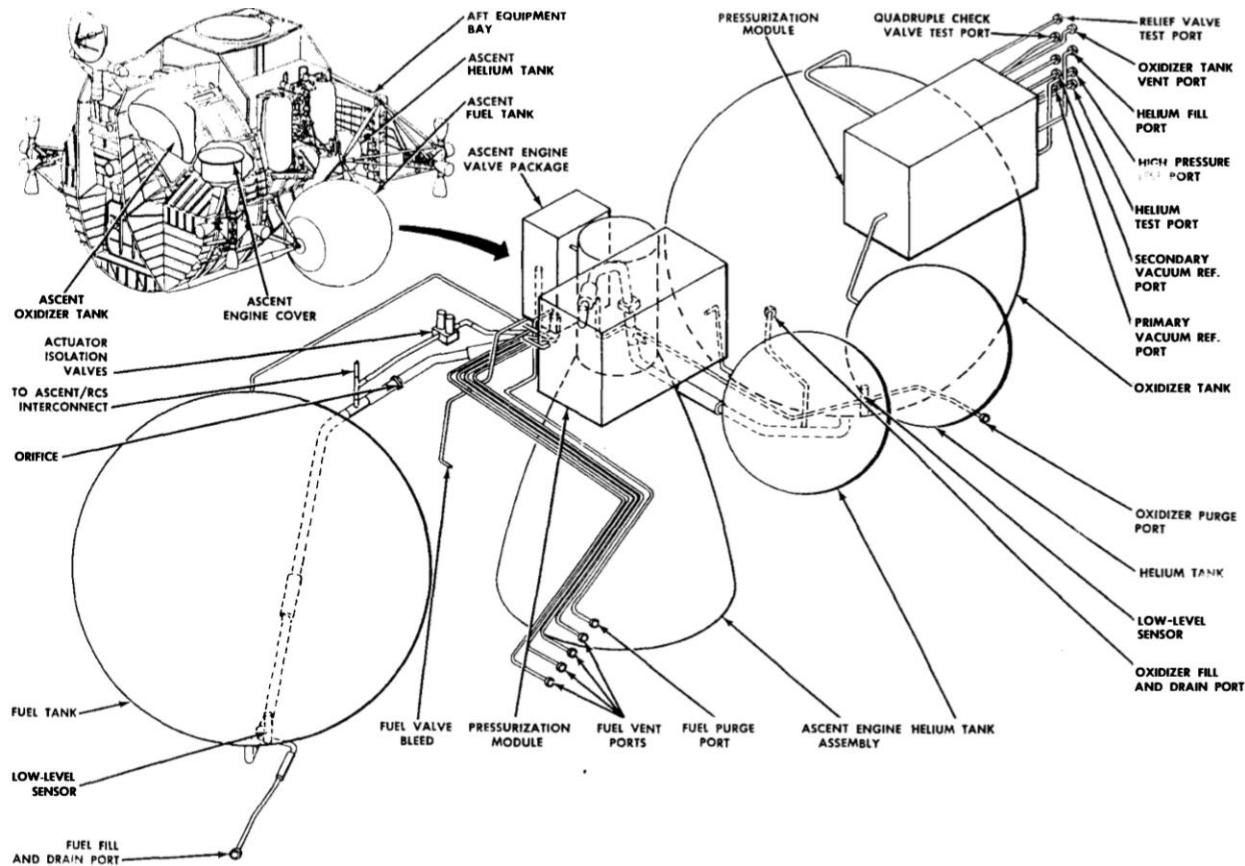
The APS receives 28-volt d-c and 115-volt a-c primary power through the Commander's and LM Pilot's buses of the EPS. The outputs of the APS pressure and temperature transducers, low-level sensors, and valve position indicator switches are processed in the IS and are transmitted via the CS to MSFN. The IS also processes the APS caution and warning signals. The EDS opens explosive valves in the APS to enable propellant tank pressurization. Interconnect plumbing between the APS propellant tanks and the RCS thrust chamber assembly feed lines permits the RCS to use APS propellants during certain mission phases, thereby conserving the RCS propellant supply.

# LUNAR MODULE

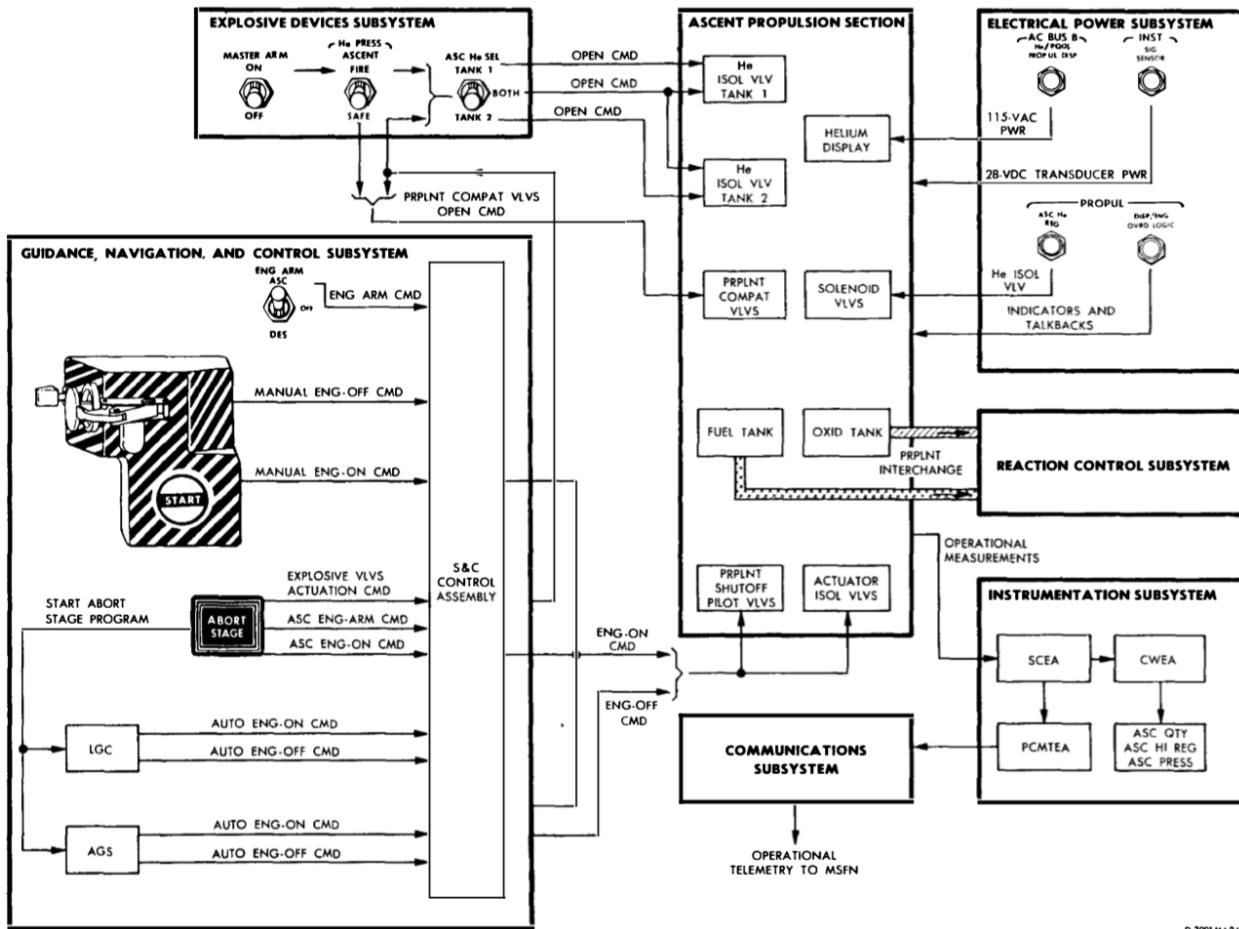
## FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR



The GN&CS issues automatic on and off commands to the ascent engine. Ascent engine arming, ignition, and shutdown can be initiated by automatic guidance equipment (PGNS or AGS) or by the astronauts. The automatic and manual commands are sent to the S&C control assemblies, which provide sequential control of LM staging and engine on and off commands. The ascent engine is armed manually by setting the ENG ARM switch to ASC or pressing the ABORT STAGE pushbutton. (Either action automatically shuts off the descent engine if it is firing.) After setting the ENG ARM switch to ASC (and after initial propellant tank pressurization), the ascent engine can be started manually by pressing the START pushbutton and stopped by pressing either STOP pushbutton. In the event of an abort-stage command while the descent engine is firing, the S&C control assemblies provide a time delay before commanding LM staging and ascent engine firing. This delay ensures that the descent engine has stopped thrusting before staging occurs. To stop the ascent engine after an abort-stage start, the ABORT STAGE pushbutton must be reset (pressed a second time) to release the switch (This procedure is necessary because the ABORT STAGE pushbutton disables the STOP pushbutton.). The manual commands override the commands issued by the automatic guidance equipment.



D-300LM4-243

### 3.2 ASCENT PROPULSION FUNCTIONAL DESCRIPTION

The APS consists of a constant-thrust rocket engine that is not gimbaled, two propellant tanks, two helium tanks, and associated propellant feed and helium pressurization components. The engine is installed in the midsection of the ascent stage; it is canted so that the center line is tilted 1.5° from the X-axis, in the +Z-direction. The engine develops 3,500 pounds of thrust in a vacuum, sufficient to launch the ascent stage from the lunar surface- and place it into a predetermined lunar orbit. The engine can be shut down and restarted, within operational limitations and restrictions, as required by the mission.

Functionally, the APS can be subdivided into a pressurization section, a propellant feed section, and an engine assembly.

#### 3.2.1 PRESSURIZATION

The propellants are pressurized by gaseous helium, supplied from two identical tanks and routed through redundant flow lines into the propellant tanks. The helium is stored at a nominal

pressure of 3,050 psia at a nominal temperature of +70° F. A pressure transducer at each tank outlet port is connected to the HELIUM indicator via the HELIUM MON selector switch. Before staging, the pressure transducers also supply a signal to the IS that will cause the ASC PRESS warning light to go on when the pressure in either helium tank is less than 2,773 psia. This alerts the astronauts to identify and isolate the faulty tank. The HELIUM MON selector switch selects the tank pressure to be displayed. When a caution or warning light goes on, a signal is routed from the CWEA to light the MASTER ALARM pushbutton/lights (panels 1 and 2) and to provide a warning tone in the astronaut headsets. Pressing either MASTER ALARM pushbutton turns off both lights and terminates the tone, but has no effect on the caution or warning light. If the ASC PRESS warning light goes on because pressure in either helium tank is less than 2,773 psia, the light goes off upon separation of the descent and ascent stage.

Before initial ascent engine operation, the helium isolation explosive valves prevent the helium from leaving the tanks. These valves can be opened individually or simultaneously. The propellants are not pressurized until shortly before initial ascent engine start. To accomplish initial pressurization, the helium isolation explosive valves and the fuel and oxidizer compatibility explosive valves (upstream of the propellant tanks) are opened simultaneously. Normally, propellant pressurization is initiated by setting the ASCENT He SEL switch (panel 8) to TANK 1. If the HELIUM indicator shows a leak (zero or decaying pressure) in one of the tanks, the ASCENT He SEL switch is set so that only the explosive valve leading from the nonleaking tank opens, thus preventing helium loss through the leaking tank via the helium interconnect line (downstream of the explosive valves). The MASTER ARM switch is then set to ON and the ASCENT He PRESS switch is set to FIRE and released, firing the explosive valves in the APS. A filter in each helium flow path traps debris from the explosive valve.

Downstream of the filter, each helium flow path has a normally open latching solenoid valve and two series-connected pressure regulators. The downstream regulator is set to a slightly higher output pressure than the upstream regulator; the regulator pair in the primary flow path produces a slightly higher output than the pair in the secondary (redundant) flow path. This arrangement causes lockup of the regulators in the redundant flow path after the propellant tanks are pressurized, while the upstream regulator in the primary flow path maintains the propellant tanks at their normal pressure of 184 psia. If either regulator in the primary flow path fails closed, the regulators in the redundant flow path sense a demand and open to pressurize the propellant tanks. If an upstream regulator fails open, control is obtained through the downstream regulator in the same flow path. Because the downstream regulator normally does not control the output pressure, an open failure of this regulator has no effect. If both regulators in the same flow path fail open, pressure in the helium manifold increases above the acceptable limit of 220 psia, causing the ASC HI REG caution light (panel 2) to go on. This alerts the astronauts to the fact that the failed-open regulators must be identified and the helium isolation solenoid valve in the malfunctioning flow path must be closed so that normal pressure can be restored. The regulator outlet pressure is sensed by redundant pressure transducers that supply

inputs to the IS, for telemetry to MSFN. One of these pressure transducers supplies the input signal to the ASC HI REG caution light. Excessive pressure is vented by the relief valve assemblies in parallel with the propellant tanks. The solenoid valve is closed by setting the ASCENT He REG 1 or REG 2 switch (panel 1) to CLOSE; the talkback above the switch will change to a barber-pole display.

The primary and secondary helium flow paths merge downstream of the regulators to form a common helium manifold. The manifold routes the helium into two flow paths: one path leads to the oxidizer tank; the other, to the fuel tank. A quadruple check valve assembly, a series-parallel arrangement in each path, isolates the upstream components from corrosive propellant vapors. The check valves also safeguard against possible hypergolic action in the common manifold resulting from mixing of propellants or fumes flowing back from the propellant tanks. Two parallel combustibility explosive valves, downstream of each quadruple check valve assembly, seal off the propellant tank inlets, isolating the fuel and oxidizer (liquid and vapor) before initial ascent engine start. This reduces contamination problems involving helium components and prolongs the life of the pressure regulators. The fuel and oxidizer compatibility explosive valves are opened simultaneously with the helium isolation explosive valves before initial engine start. The four compatibility valves are arranged so that two fuel compatibility valves and two oxidizer compatibility valves, in parallel paths, lead to their propellant tanks. One fuel and one oxidizer compatibility valve have dual cartridges, the other two are fired by single cartridges.

Immediately upstream of the fuel and oxidizer tanks, each helium path contains a burst disk and relief valve assembly to protect the propellant tanks against overpressurization. This assembly vents pressure in excess of approximately 245 psia (relief valve cracking pressure) and reseals the flow path after overpressurization is relieved. A thrust neutralizer eliminates unidirectional thrust generated by the escaping gas. To prevent leakage through a faulty relief valve during normal operation, the burst disk is located upstream of the relief valve. The burst disk ruptures at a pressure slightly below the relief valve setting. The relief valves can pass the entire helium flow from a failed-open pair of regulators, preventing damage to the propellant tanks.

### 3.2.2 PROPELLANT FEED

The APS has one oxidizer tank and one fuel tank. Each tank has a temperature transducer that supplies propellant temperature signals to the FUEL and OXID TEMP indicators, and a pressure transducer that supplies ullage pressure signals to the FUEL and OXID PRESS indicators. (The same TEMP and PRESS indicators are used for the APS and DPS.) APS data are displayed when the PRPLNT TEMP/PRESS MON switch is set to ASC. A low-level sensor in each propellant tank causes the ASC QTY caution light (panel 2) to go on when the remaining propellant in either tank is sufficient for only 10 seconds of engine operation. The ASC QTY caution light is inhibited when the ascent engine is not operating.

Helium flows into the top of the fuel and oxidizer tanks. Diffusers at the top of the tanks uniformly distribute the helium throughout the ullage space. The outflow from each propellant tank divides into two paths. In the primary path, each propellant flow through a trim orifice to the propellant filter in the engine assembly, and then to the isolation and bipropellant valve assemblies (propellant shutoff valves). The trim orifice provides an engine interface pressure of 170 psia for proper propellant use. The secondary path connects the ascent propellant supply to the RCS. This interconnection, at the normally closed ascent feed solenoid valves (part of the RCS), permits the RCS to burn ascent propellants, providing the APS is pressurized and the ascent or descent engine is operating when the RCS thrusters are fired. A line branches off the RCS interconnect fuel flow path and leads to two parallel actuator isolation solenoid valves. From there, this line routes fuel to the engine pilot valves that actuate the propellant shutoff valves in the engine assembly. The normally closed actuator isolation valves prevent possible fuel loss through a leaking pilot valve before initial engine operation or during engine shutdown. The actuator isolation valves, and the propellant shutoff pilot valves in the engine assembly, are opened and closed simultaneously by engine-on and engine-off commands.

### 3.2.3 ENGINE ASSEMBLY

The ascent engine is a fixed-injector, restartable, bipropellant rocket engine with an ablative combustion chamber, throat, and nozzle extension. Fuel and oxidizer enter the engine assembly through inlet ports at the interface flanges and are routed through the propellant filters and the propellant shutoff valves (isolation and bipropellant valve assemblies) to the injector; a separate fuel path (actuator pressure line) leads to the pilot valves, where fuel pressure actuates the propellant shutoff valves.

Propellant flow to the engine combustion chamber is controlled by a valve package assembly, trim orifices, and an injector assembly. The valve package assembly consists of eight propellant shutoff ball valves that make up the two fuel-and-oxidizer-coupled isolation valve assemblies and the two fuel-and-oxidizer coupled bipropellant valve assemblies, four actuators, and four solenoid-operated pilot valves. Inside the valve package assembly, the fuel and oxidizer passages divide into dual flow paths, with two series ball valves in each flow path. The paths rejoin at the valve package outlet. The propellant shutoff valves are arranged in fuel-oxidizer pairs; each pair is operated from a single crankshaft assembly by an individual fuel-pressure-operated actuator. Shaft seals and vented cavities prevent fuel and oxidizer from coming into contact with each other due to seepage along the shafts.

After the ascent propellants have been pressurized, the ascent engine can be started manually by setting the ENG ARM switch to ASC and by pressing the START pushbutton. Automatic starts are initiated by LGC or AGS engine-on commands. At engine start, the two actuator isolation valves in the propellant feed section, and the four pilot valves, are opened simultaneously, routing fuel into the actuator feed line and to the four pilot valves. When the solenoids of the

pilot valves are energized, the pilot valve spools slide away from the fuel inlet ports and block the overboard vent ports. Fuel enters the actuator chambers and extends the actuator pistons, cranking the propellant shutoff valves 90° to the fully open position. The propellants now pass through the shutoff valves and trim orifices directly to the injector. The orifices determine the thrust level of the engine and the mixture ratio of the propellants by trimming the pressure differentials of the fuel and oxidizer. The physical characteristics of the injector establish an oxidizer lead of between 40 and 50 milliseconds. This precludes the possibility of fuel lead, which would result in rough engine starts.

At engine cutoff, the pilot valve solenoids are deenergized, opening the actuator ports to the overboard vents so that residual fuel in the actuators is vented into space. With the actuation fuel pressure removed, the actuator pistons are moved back by spring pressure, causing the propellant shutoff valves to turn 90° to the closed position.

The ascent engine assembly has transducers and valve position indicator switches for sensing fuel and oxidizer inlet pressures, thrust chamber pressure, and propellant shutoff valve positions. The transducer outputs are converted to telemetry data in the IS. These data are transmitted to MSFN, where they are used to monitor the performance of the ascent engine.

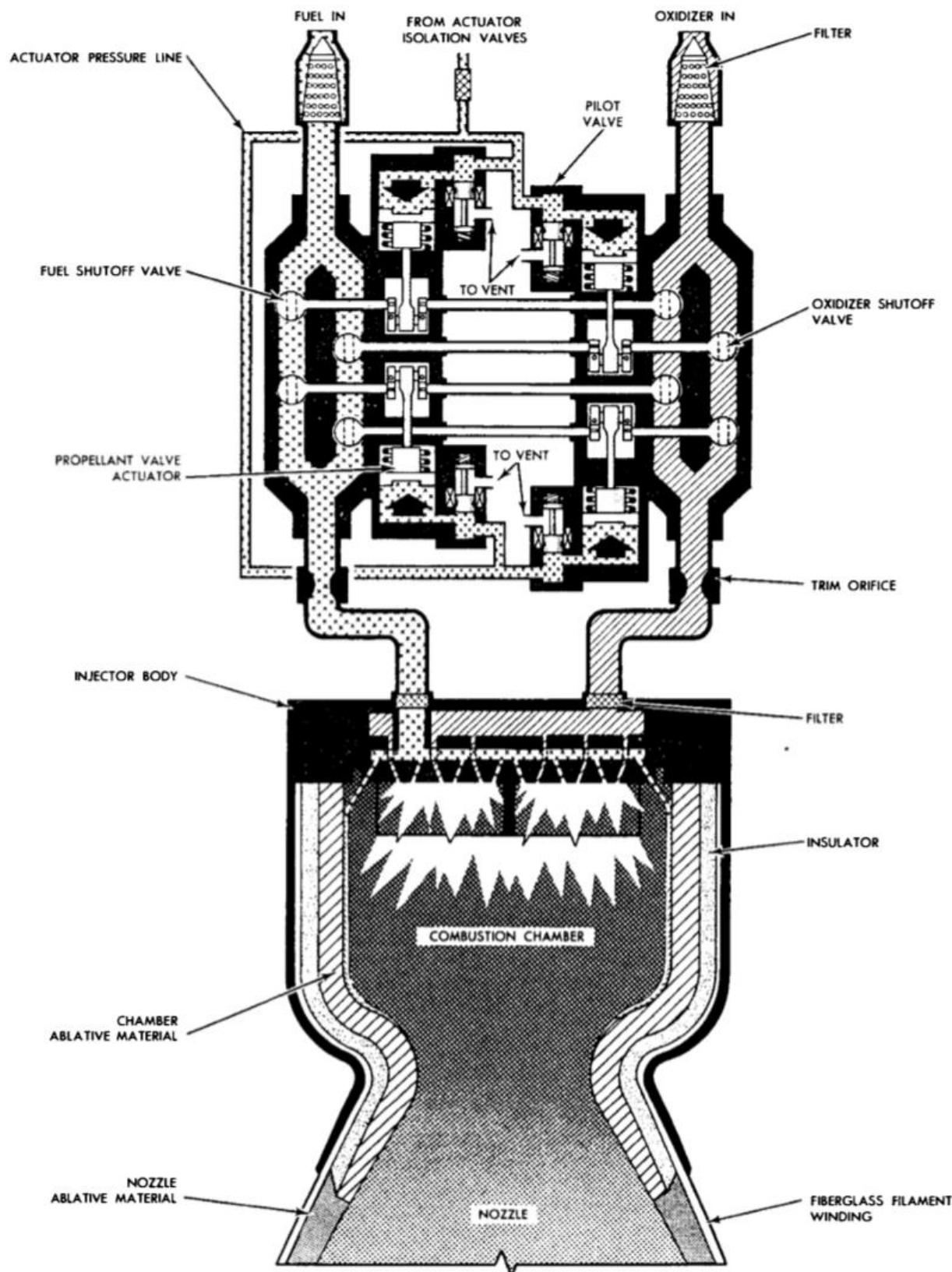


Figure 3.2.3.1 – Engine Assembly Logic

A detailed Engine Assembly image with its components can be seen below, in figure 3.2.3.2.

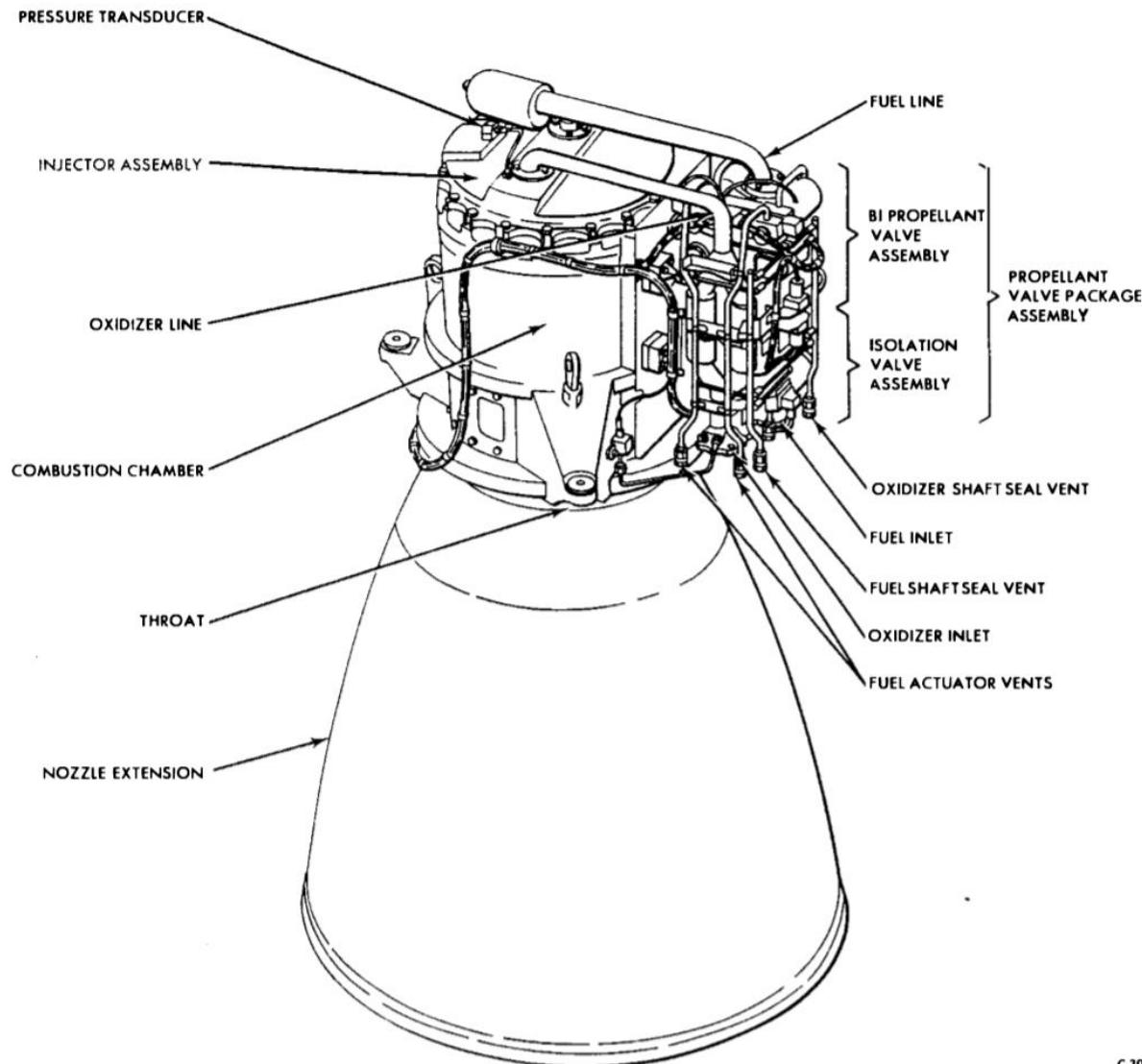


Figure 3.2.3.2 – Engine Assembly

### 3.3 EXPLOSIVE DEVICES OVERVIEW

The diagram below (Figure 3.3.1) shows the explosive devices described in the preceding sections.

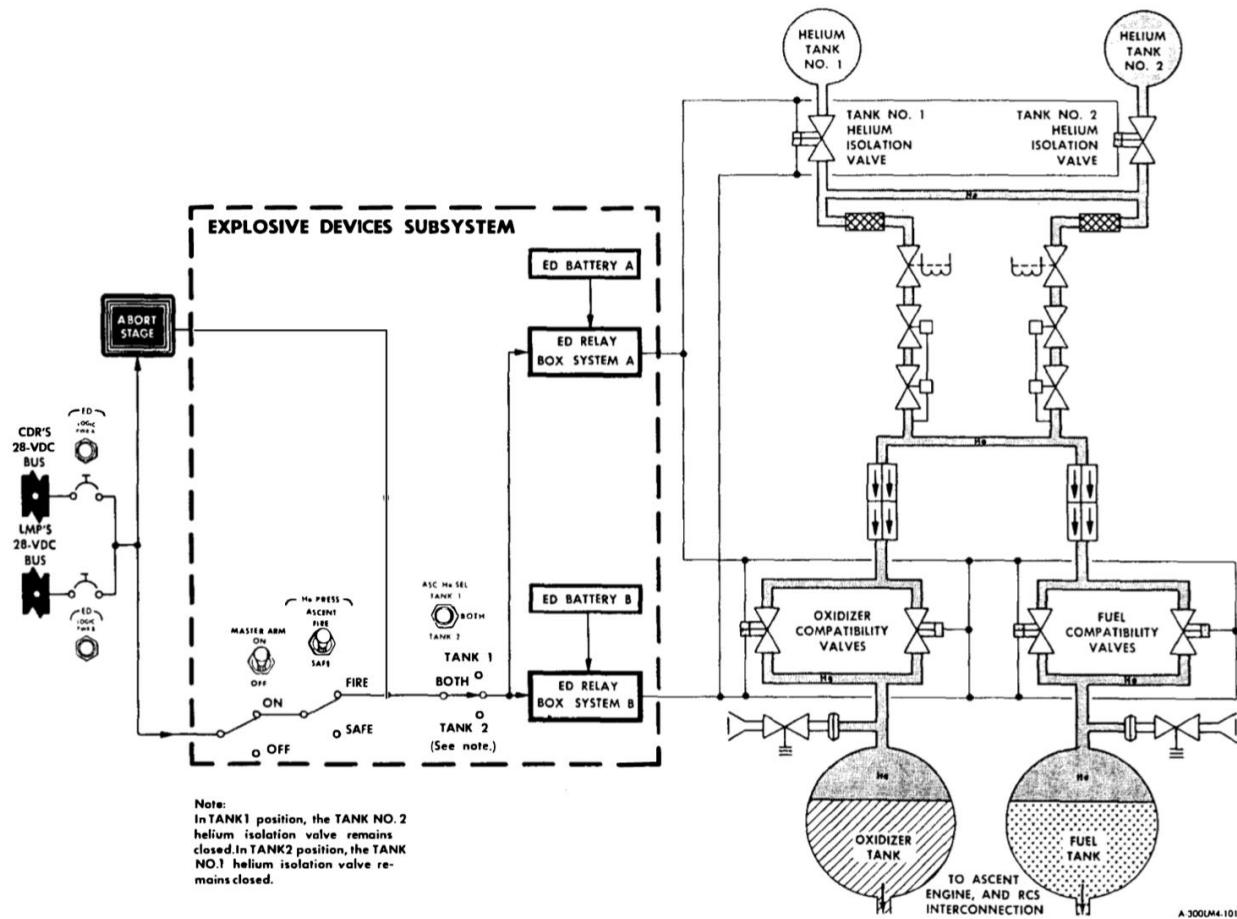


Figure 3.3.1 – Explosive Devices overview

### 3.4 ASCENT PROPULSION OPERATIONAL LIMITATIONS AND RESTRICTIONS

The operational limitations and restrictions for the APS are as follows:

- Propellant tank pressure before pressurization must be between 62 and 205 psia. If these limits are exceeded during dynamic loading, structural failure in the propellant tanks may result, causing loss of the LM.
- Propellant bulk temperature before ascent engine start must be between +50° and +90° F. If the temperature limits are exceeded, the engine performance will be degraded.
- Before ascent engine starts (except FITH with DPS burn), the RCS +X-axis thrusters must be fired to establish proper propellant tank ullage, to settle the propellants under weightless conditions, and to prevent helium from entering the RCS interconnect lines. Unsettled or insufficiently settled propellants may result in rough or erratic starts that could lead to engine failure.

- The APS must not remain pressurized longer than 24 hours before anticipated termination of use. If this limit is exceeded, the pressure regulator assemblies will exceed their qualified propellant exposure time.
- In the blow-down mode, it is not desirable that the ascent engine be operated when chamber pressure has decayed to less than 112 psia. At a pressure of less than 114 psia, firing should be terminated unless engine operation is critical to mission success.

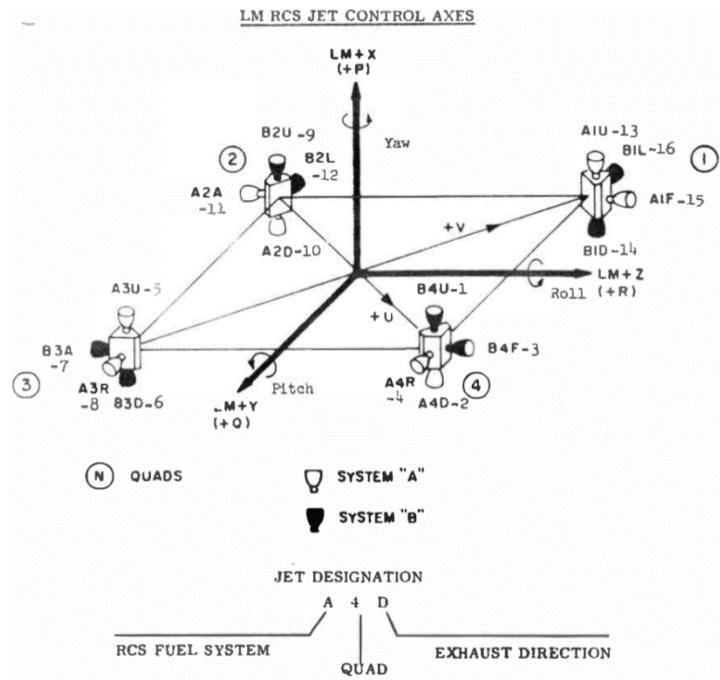
# VI. REACTION CONTROL



# VI. REACTION CONTROL

## 1. GENERAL

The Reaction Control Subsystem (RCS) stabilizes the LM during descent, helps to maintain the desired trajectory during ascent, and controls LM attitude and translation about, or along, all axes during hover, rendezvous, and docking maneuvers. It also provides acceleration for propellant settling in the descent and ascent propellant tanks.



Attitude and translation are controlled with 16 RCS thrust chamber assemblies (TCA's), which are fed by the RCS propellant supplies or by propellants from the ascent propulsion section (APS) of the Main Propulsion Subsystem (MPS). The 16 TCA's are arranged in clusters of four, mounted on four outriggers that are equally spaced around the LM ascent stage.

Four plume deflectors are attached to the descent stage, extending upward to the nozzle of each downward-firing TCA. These deflectors are shields that prevent the downward exhaust plume from damaging the LM structure.

The RCS consists of two parallel, independent systems (A and B), which, under normal conditions, function simultaneously. Each system has its own pressurized propellant supply that feeds eight TCA's (two in each cluster). The arrangement of the TCA's is such that either system, functioning alone, can provide complete control in all axes, with some translation effects. Moreover, if the propellant supply of one system is depleted or fails, a cross feed capability permits routing propellants from the operative system to all 16 TCA's.

Fuel and oxidizer are loaded into bladders within the propellant tanks and into the manifold plumbing that extends from the tanks through the normally open main shutoff valves up to the TCA solenoid valves. Before separation of the LM from the CSM, the TCA's are heated to their operating temperature and the explosive valves are fired. Gaseous helium, reduced to a working pressure, enters the propellant tanks and forces the fuel and oxidizer to the TCA's. Here, the propellants are blocked by normally closed fuel and oxidizer valve assemblies until a thruster-on command is issued. As the selected TCA receives a thruster-on command, its fuel and oxidizer valve assemblies open to route the propellants through the TCA injector into the combustion

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chambers, where they impinge and ignite. Switches on panel 2 generate signals for the LGC, telemetry, and caution and warning talkbacks to indicate the status of the thrusters.)

The RCS propellant tanks are pressurized immediately before separation of the LM from the CSM, by firing of explosive valves by the Explosive Devices Subsystem (EDS).

Interconnect plumbing between the TCA feed lines and the APS propellant tanks permits the RCS to use APS propellants during certain phases of the mission, thereby conserving the RCS supply. The RCS receives 28-volt d-c primary power through the Commander's and LM Pilot's buses of the Electrical Power Subsystem (EPS).

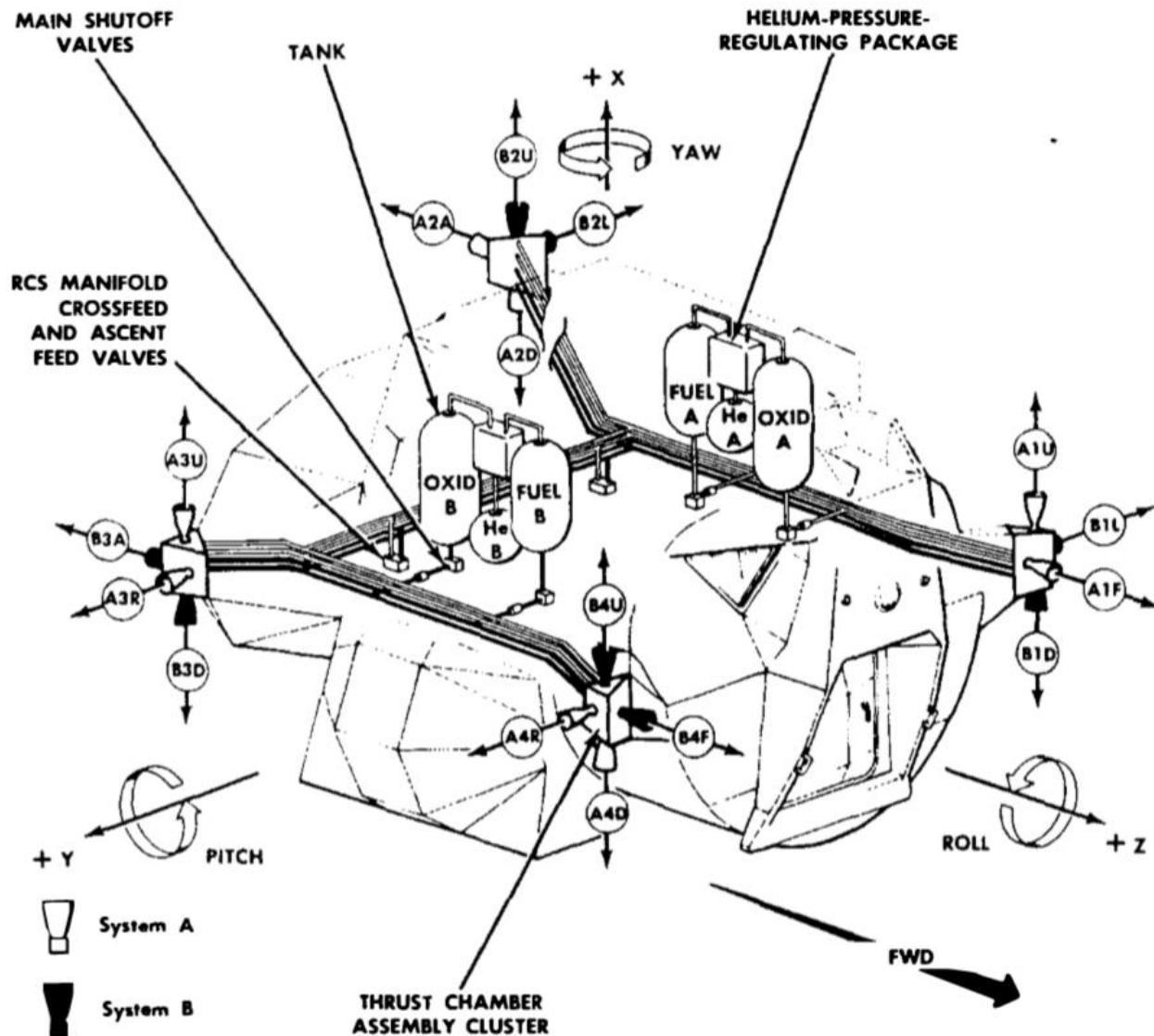


Figure 6.1.1.1 – Jet locations and their tie to System A/B

The Guidance, Navigation, and Control Subsystem (GN&CS) provides commands that select and fire TCA's for durations ranging from a pulse of less than 1 second to steady-state operation.

The TCA's can be operated in an **automatic mode**, an **attitude hold mode**, or a **manual override** mode. RCS operation can be controlled by the primary guidance and navigation section (PGNS) or by the abort guidance section (AGS).

Normally, translation and attitude are controlled by the PGNS in the automatic mode, in which all navigation, guidance, stabilization, and steering functions are controlled by the LM guidance computer (LGC). Under AGS control, the abort electronics assembly (AEA) and the attitude and translation control assembly (ATCA) take the place of the LGC.

The attitude hold mode provides semiautomatic operation. In this mode, either astronaut can manually determine attitude changes by displacing his attitude controller assembly (ACA), and three-axis translation changes by displacing his thrust/translation controller assembly (TTCA). When the ACA is displaced, an impulse proportional to the amount of displacement is routed to the LGC. The LGC uses this impulse to perform steering calculations and to generate a thruster-on command. The thruster-on command is routed to appropriate jet drivers in the ATCA, firing selected TCA's. A display and keyboard (DSKY) input to the LGC determines whether the LGC commands an angular rate change proportional to ACA displacement, or a minimum impulse for each ACA displacement. When the ACA is returned to the detent position, the LGC sends a command to hold the attitude. For translation, displacement of the TTCA sends a discrete to the LGC, which sends a thruster-on command to selected TCA's. When the TTCA is returned to neutral, the TCA's are turned off.

In the attitude hold mode, under AGS control, the AEA generates attitude errors that are summed, in the ATCA, with proportional rate commands from the ACA and a rate-damping signal from the rate gyro assembly (RGA). The ATCA then performs the steering calculations and generates the thruster on and off commands. Two or four X-axis TCA's for translation maneuvers and a manual override for attitude control in each axis (2-jet direct) can be selected. The four upward-firing TCA's can be inhibited to conserve RCS propellants during the ascent engine thrust phase.

The manual override mode, under PGNS or AGS control, overrides the automatic mode. The four-jet hardover command from the ACA is a manual override command, which is applied directly to the TCA's. The hardover output fires four TCA's simultaneously.

For MPS propellant-settling maneuvers, two or four downward-firing TCA's can be selected. Under PGNS control, the selection is manually keyed into the DSKY for routing to the LGC. Under AGS control, the selection is made by setting the ATT/TRANSL switch (panel 1) to the 2 JETS or 4 JETS position. Under manual control, pressing the +X TRANSL pushbutton (panel 5) provides a command to operate the four downward-firing TCA's, which continue firing until the pushbutton is released. Firing two TCA's conserves RCS propellants; however, it requires a longer firing time to settle the MPS propellants. Before staging, firing of four TCA's may be necessary

because thermal impingement on the descent stage skin limits operating time for the downward-firing TCA's to 40 seconds.

## 2. FUNCTIONAL DESCRIPTION

Functionally, each RCS system (A or B) can be subdivided into a helium pressurization section, a propellant section, a propellant quantity measuring device, and eight TCA's. All RCS components are located in the ascent stage. Because RCS systems A and B are identical, only one system is described.

### 2.1. HELIUM PRESSURIZATION

Approximately 1 pound of gaseous helium, at a nominal pressure of 3,050 psia at +70° F, is stored in the helium tank. Flow from the tank separates into two parallel paths. Each flow path contains a normally closed explosive valve that isolates the helium tank from the downstream components before RCS pressurization. A sensor (part of the propellant quantity measuring device) in the helium tank senses the pressure-temperature ratio of the helium. The sensor output is conditioned and routed to the A or B QUANTITY indicator (panel 2), which indicates the combined (fuel and oxidizer) percentage of propellants remaining.

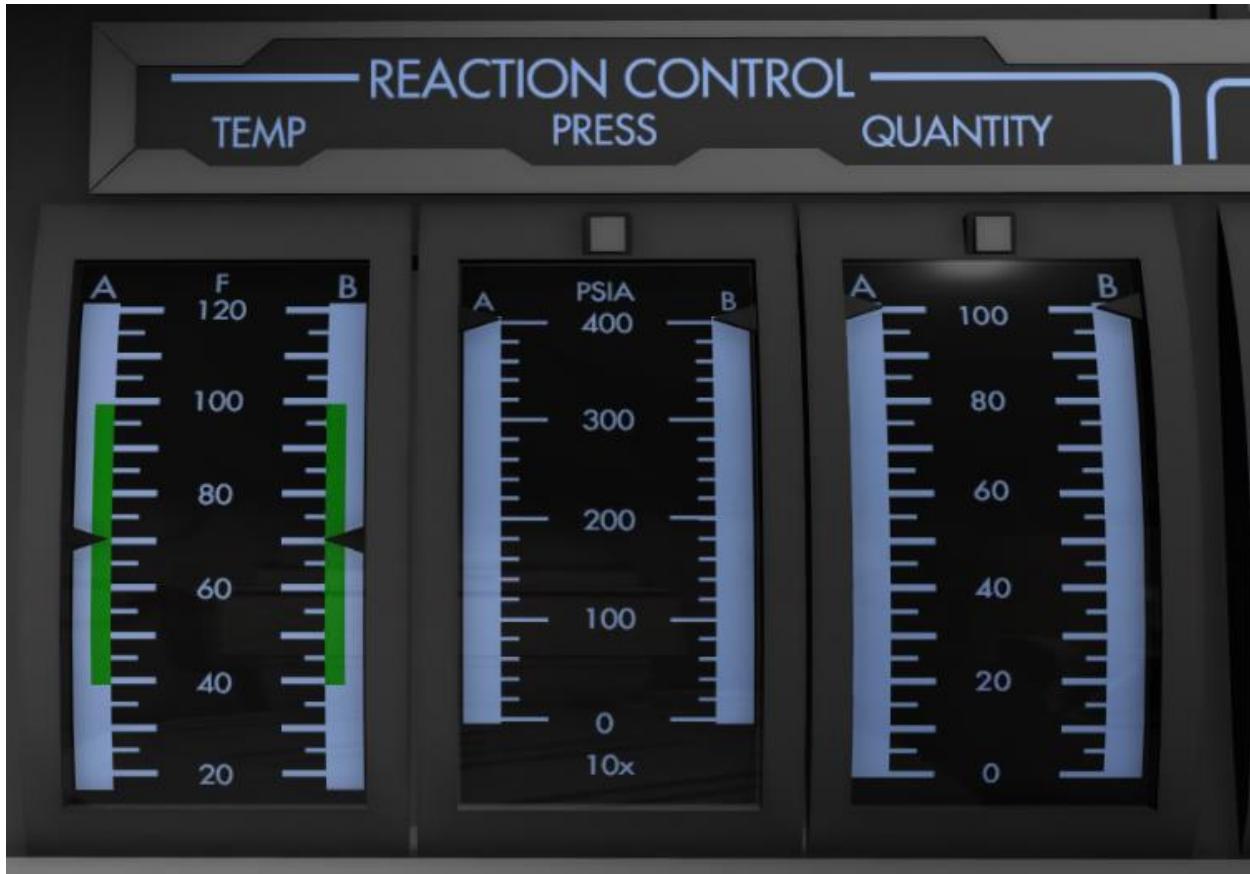
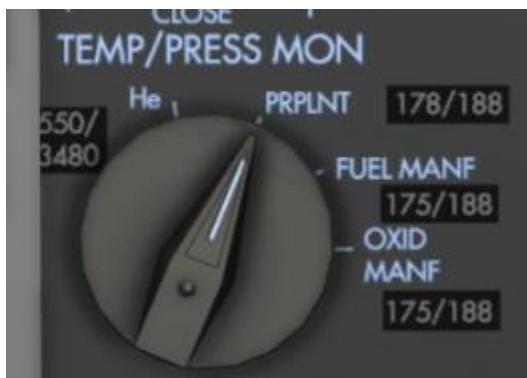


Figure 6.2.1.1 – RCS monitoring

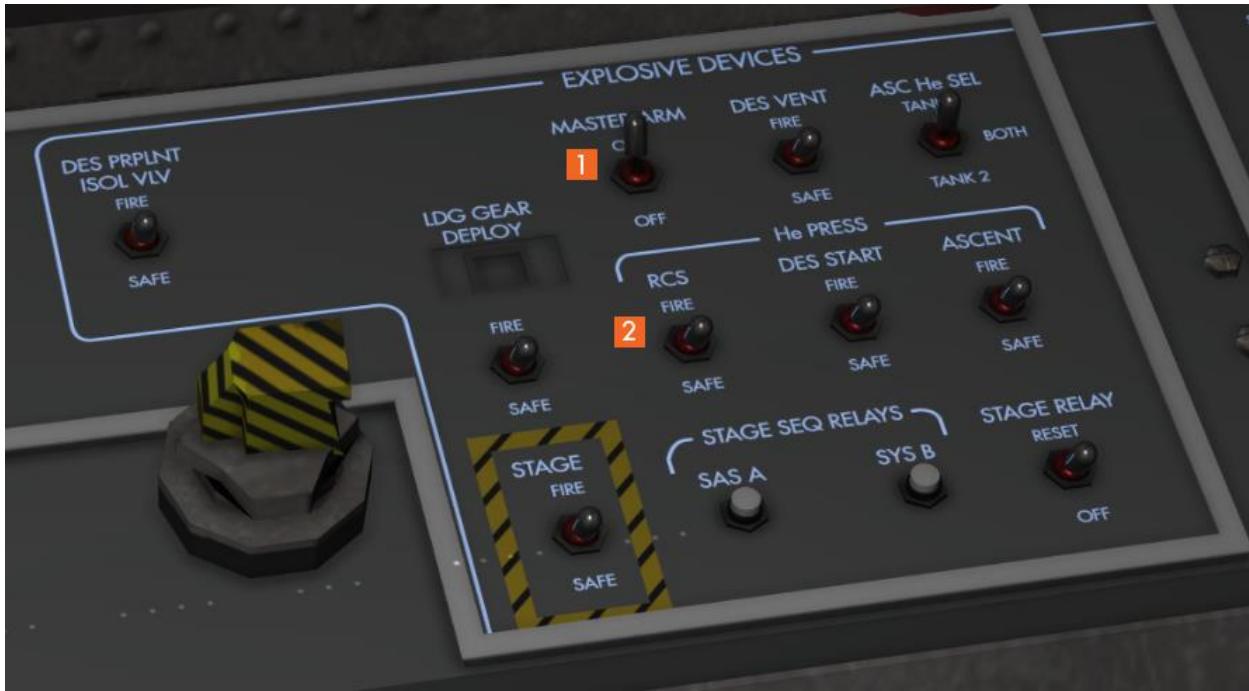


A pressure transducer at the outlet port of the helium tank monitors the helium pressure. It supplies a signal to the PRESS indicator (panel 2) when the TEMP/PRESS MON selector switch (panel 2) is set to He (when the switch is set to He, the PRESS indicator goes on to indicate that the values displayed must be multiplied by 10.).

The pressure transducer also supplies a signal that causes the RCS caution light (panel 2) to go on, when

helium pressure at the tank outlet port drops below 1,700 psia. When a caution or warning light goes on, a signal is routed from the caution and warning electronics assembly (CWEA) in the IS to light the MASTER ALARM pushbutton/lights (panels 1 and 2) and to provide a tone. Pressing either MASTER ALARM pushbutton/light turns off both lights and terminates the tone, but has no effect on the caution or warning light.

When the MASTER ARM switch (panel 8) is set to ON ([1] in figure 6.2.1.2) and the RCS He PRESS switch is set to FIRE ([2] in figure 6.2.1.2), the explosive valves open simultaneously to pressurize the RCS.



*Figure 6.2.1.2 – RCS pressurization on the EDS*

Because of the redundant paths, failure of one explosive valve does not affect pressurization of the propellant tanks. Downstream of the explosive valves, the two flow paths merge and the helium flows through a filter that prevents contamination of downstream components by trapping debris generated by firing the cartridges in the explosive valves. A restrictor orifice, downstream of the filter, dampens the initial helium surge, thereby minimizing the possibility of rupturing the burst disk in the downstream pressure relief valve assemblies.

Downstream of the restrictor, the flow path contains two series-connected pressure regulators. The primary (upstream) regulator is set to reduce pressure to approximately 181 psia. The secondary (downstream) regulator is set for a slightly higher output (approximately 185 psia). In normal operation, the primary regulator provides proper propellant tank pressurization. A pressure transducer senses the pressure at the output of the regulators and provides an input to the PRESS indicator via the TEMP/PRESS MON selector switch (PRPLNT position). If one regulator fails closed, or if both regulators fail open, the downstream pressure decreases or increases beyond acceptable limits (minimum pressure of 165 psia; maximum pressure of 218.8 psia) and the RCS A REG or RCS B REG warning light (panel 1) goes on. (If the main propellant shutoff valves are closed, this warning light is inhibited and does not go on, regardless of the pressure at the helium regulator manifold.) Downstream of the pressure regulators, a helium manifold divides the flow into two paths: one leads to the oxidizer tank; the other, to the fuel tank. Each flow path has quadruple check valves in a series-parallel arrangement to prevent backflow of propellant vapors into the helium manifold if seepage occurs in the propellant tank bladders. The helium flows through the check valves into the propellant tanks. A relief valve assembly at the inlet port of each propellant tank protects the propellant tank against over

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pressurization. If pressure in the helium lines exceeds 220 psia, a burst disk in the relief valve assembly ruptures. When the pressure reaches 232 psia, helium is vented overboard through the relief valve vent port. When the pressure drops below 212 psia, the relief valve closes, permitting normal system operation.

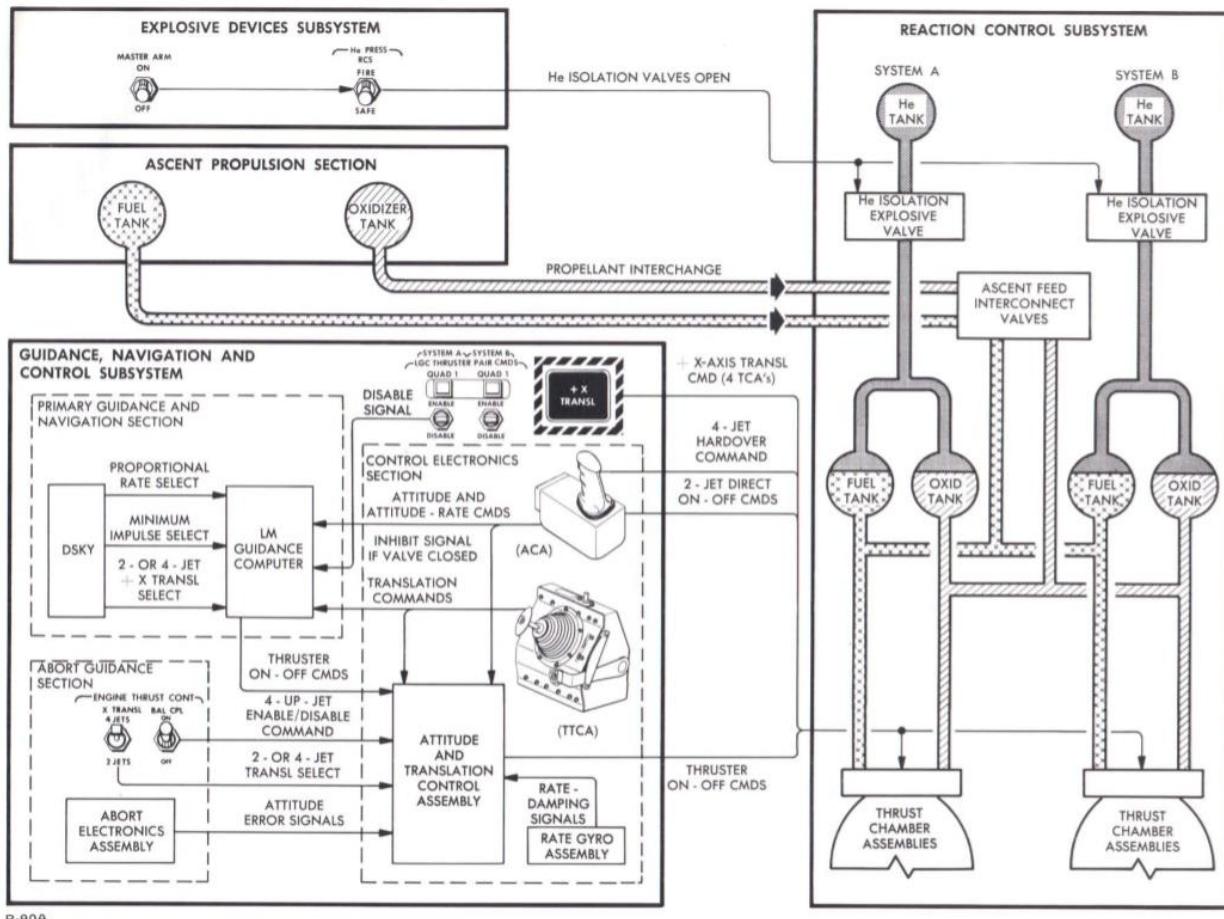


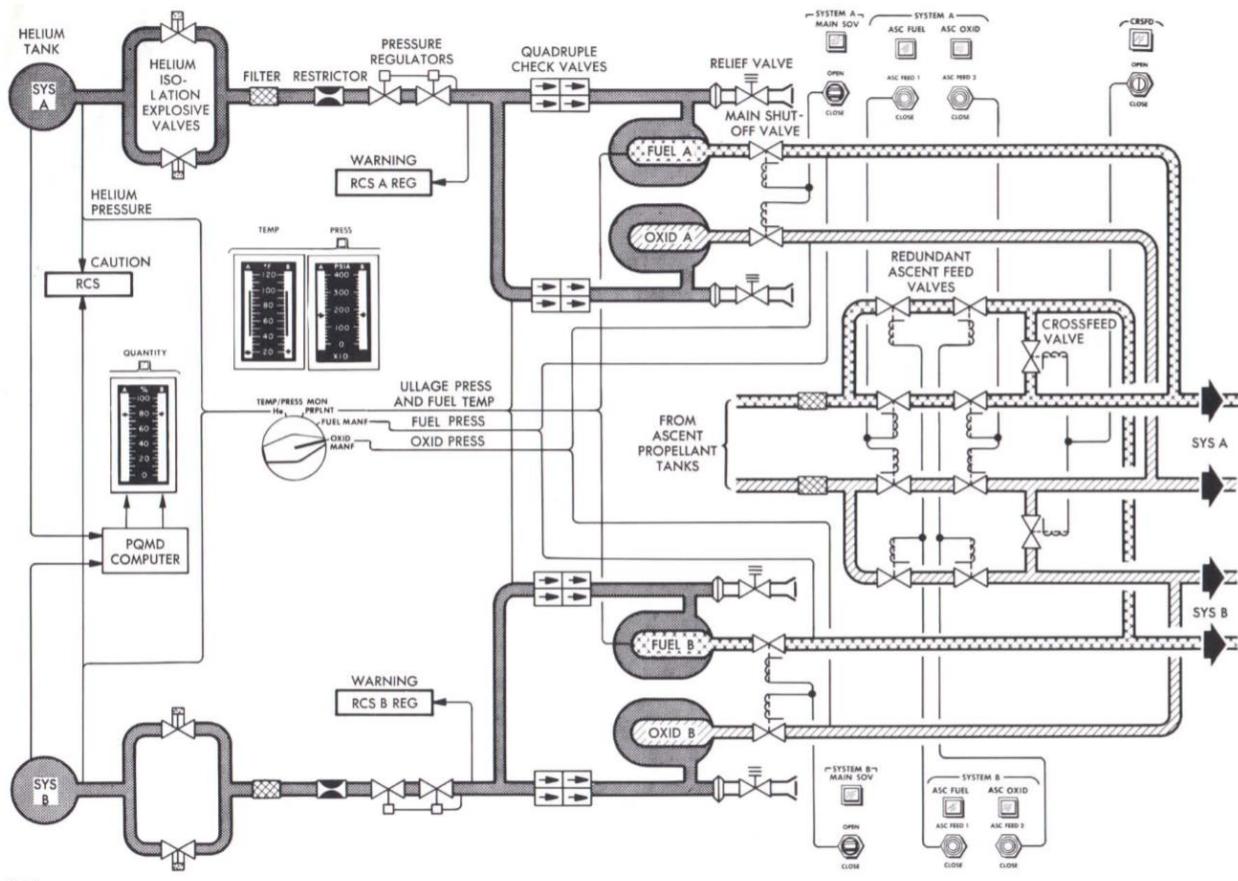
Figure 6.2.1.3 – Helium and Propellant feed lines

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Figure 6.1.1.3 – Helium Pressurization and Propellant Feed Section Flow Diagram

## 2.2. THRUST CHAMBER ASSEMBLIES

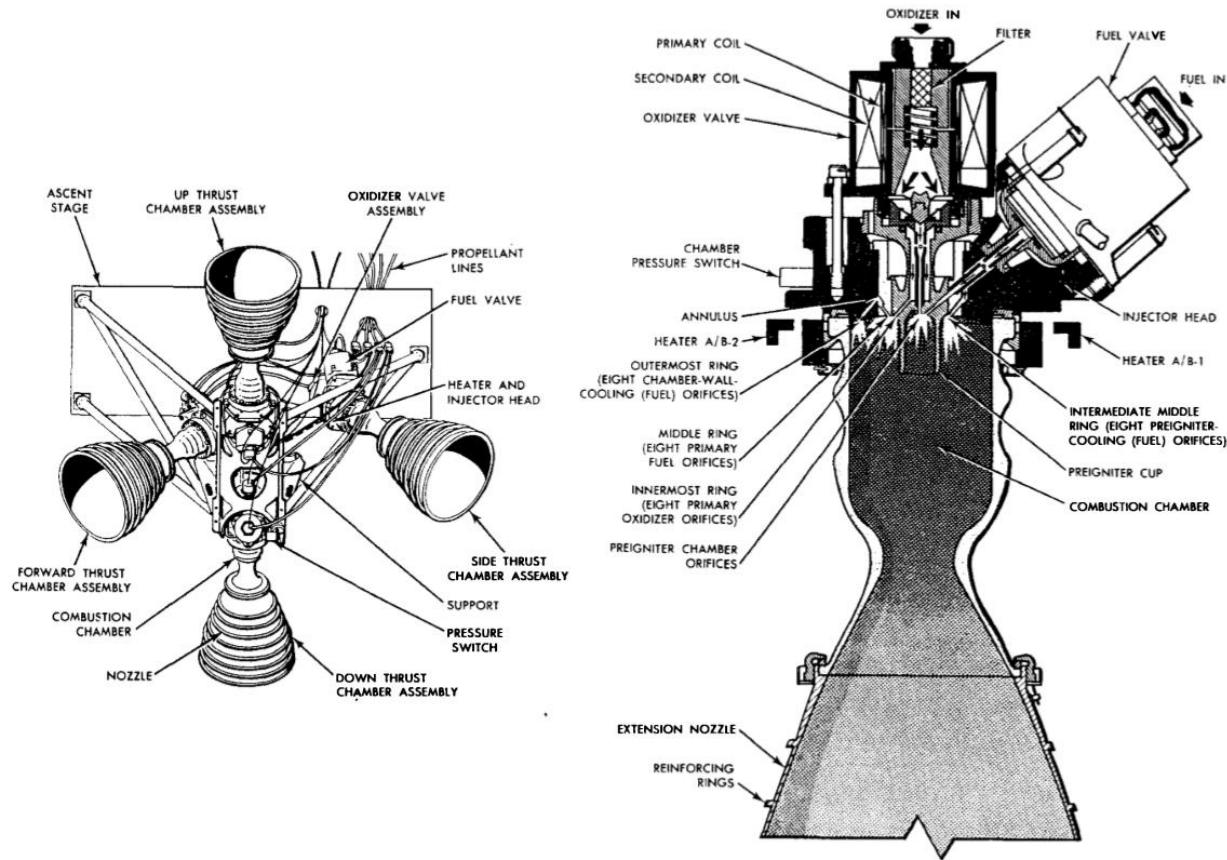
The TCA clusters are enclosed in thermal shields, with the TCA combustion chamber outboard of the second rib and the nozzle protruding from the shield. The thermal shields aid in maintaining a temperature-controlled environment for the propellant lines from the ascent stage to the TCA's, minimize heat loss, and reflect radiated engine heat and solar heat.

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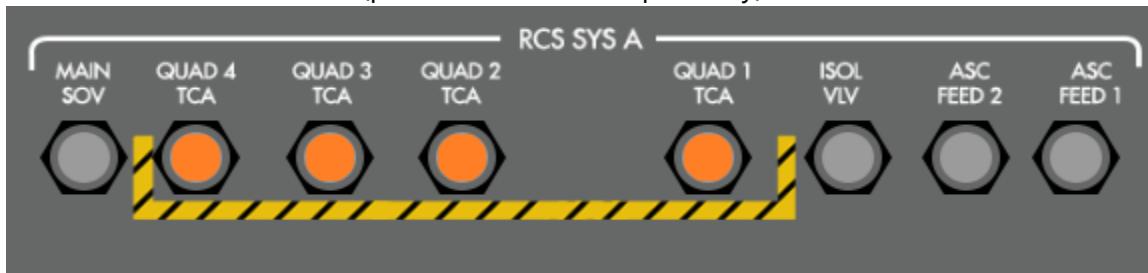
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*Figure 6.2.2.1 – TCA cluster*

Each TCA contains an oxidizer solenoid valve and a fuel solenoid valve, which, when open, pass propellant through an injector into the combustion chamber, where ignition occurs. Each solenoid valve contains a primary (automatic) coil and a secondary (direct) coil, either of which, when energized, opens the valve.

Primary power (28 volts) is applied to the TCA primary coils via the RCS SYS A and RCS SYS B: QUAD TCA circuit breakers (panels 11 and 16, respectively).



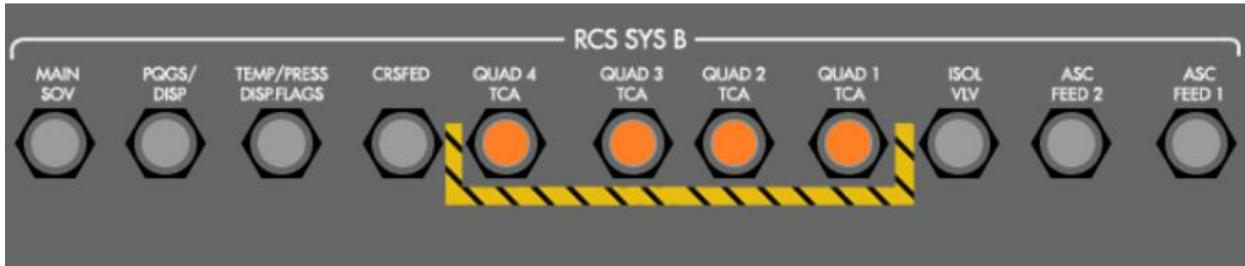
*RCS SYS A circuit breakers on Panel 11.*

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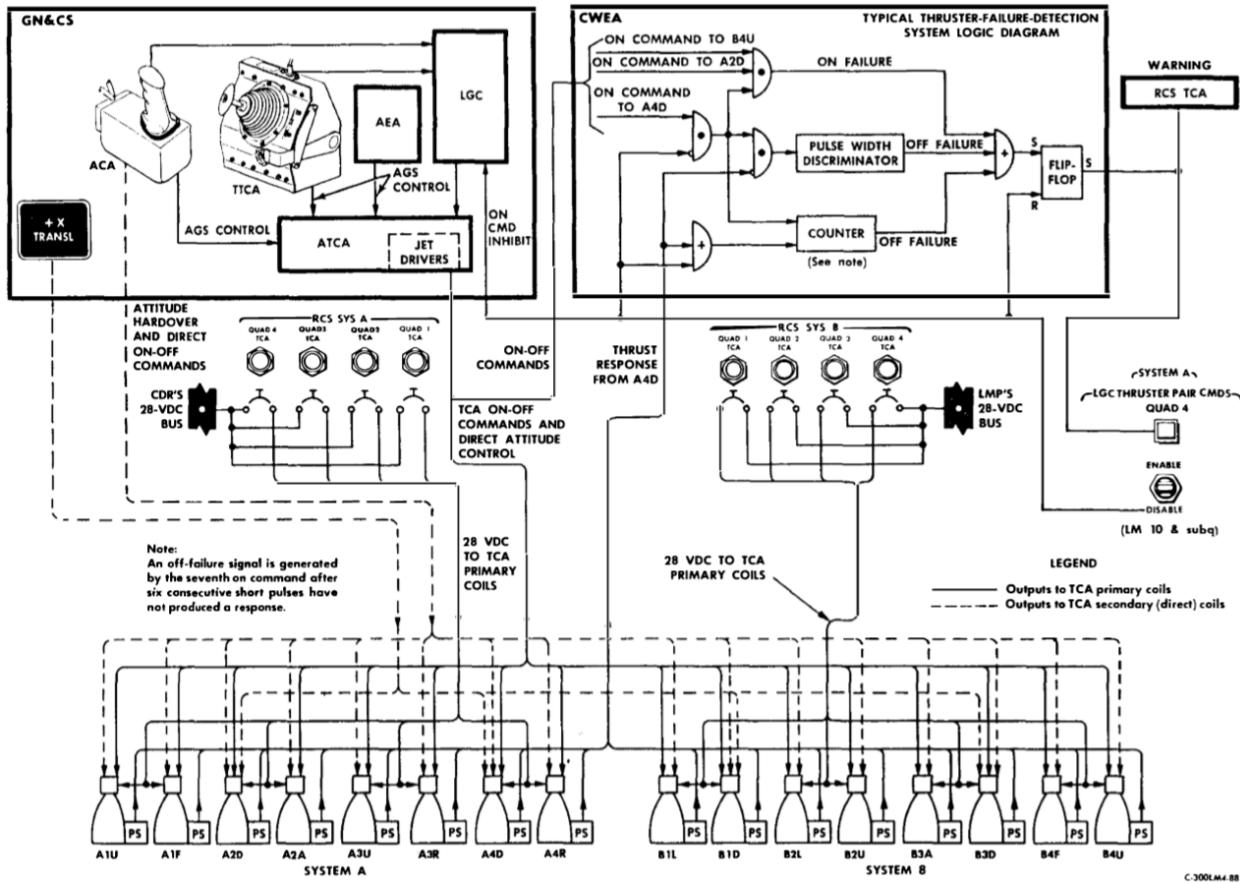
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*RCS SYS B circuit breakers on Panel 16.*

Each circuit breaker controls power to the primary coils of two TCA's in each quad. In the automatic mode, the on and off commands are applied to the primary coils of the TCA's. When a manual override command (such as attitude hardover or +X-translation) is issued, the command is applied directly to the secondary coils of the TCA's.



*Figure 6.2.2.2 – TCA logic*

For attitude control, the duration of TCA firing ranges from a pulse to steady-state operation. The pulse mode is used for small attitude changes in the selected axis. In this mode, TCA operation lasts for less than 1 second: the minimum pulse for TCA firing (at 21 volts) is 13 milliseconds. In the steady-state mode, operation lasts for 1 second or longer. For a short-

duration pulse (figure 6.2.2.3), engine thrust is just beginning to rise when the shutdown command is given, and only a very small amount of propellant is injected into the combustion chamber. Under these conditions of minimum impulse, the full 100 pounds of thrust cannot be achieved and TCA operation is comparatively inefficient.

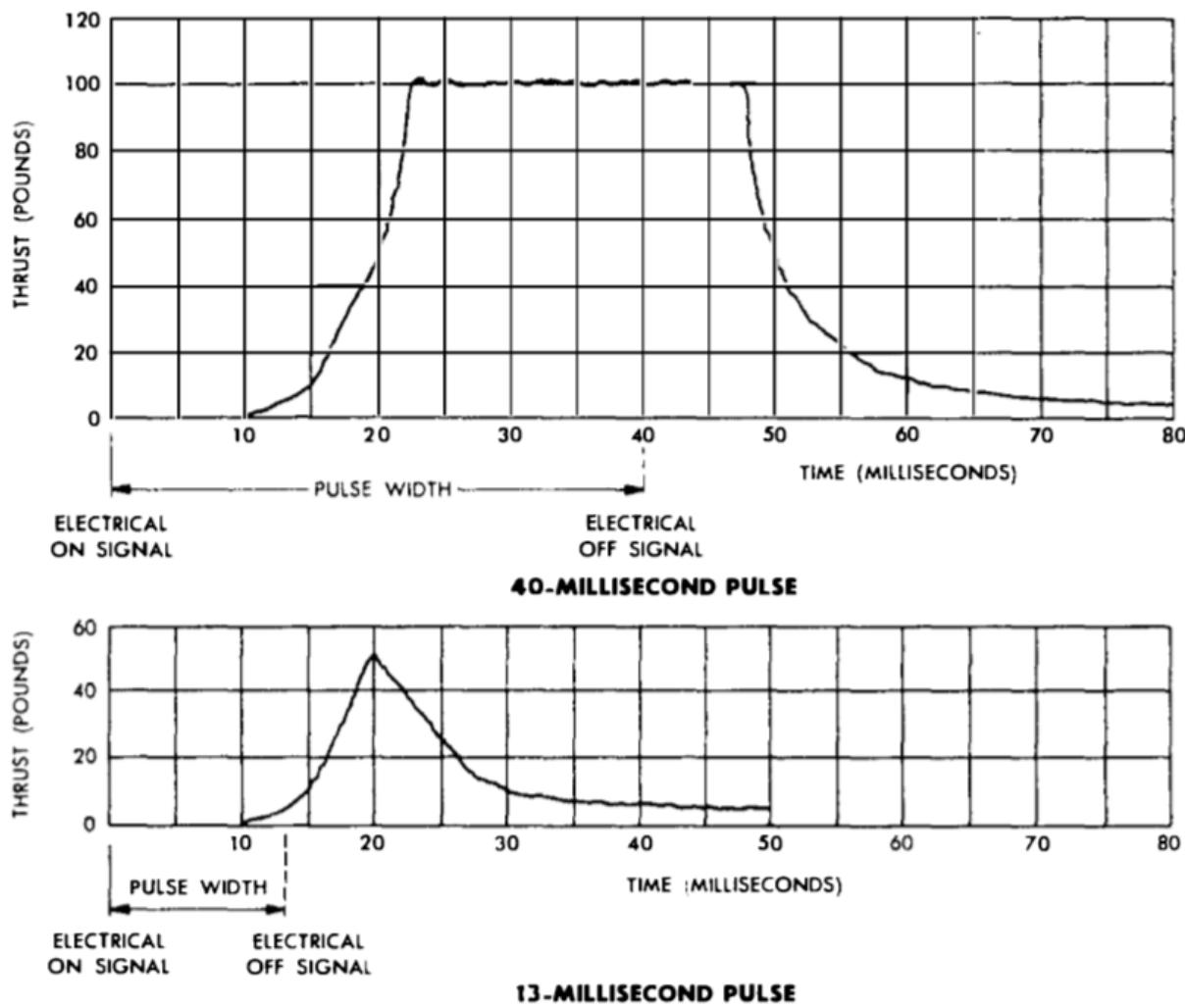


Figure 6.2.2.3 – TCS Firing – Thrust versus time

If the pressure regulator in the helium pressurization section fails closed, TCA operation continues briefly, until helium pressure drops below the acceptable limit and propellants are no longer injected into the TCA combustion chamber.

If both pressure regulators fail open, the relief valves in the pressurization section vent the helium overboard; however, TCA operation continues until near depletion of the helium. In either type of pressurization failure, use of the eight TCA's of the affected system is lost until the main shutoff valves are closed and the crossfeed valves are opened to allow the operative system to supply propellants to the TCA's of the malfunctioning system.

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The TCA's of System A and System B are activated on Panel 2.



The valves can be opened by setting the switch to OPEN, and closed by setting the switch to CLOSE. Springs will push the switch back to center when released. A talkback indicator shows if the thruster pair is active or not.

The main shut off valve needs to be OPEN for the thrusters to work. The MAIN SOV switches are used to open or close the valve. A talkback indicator shows gray when the valves are open, and barber poled with closed.



A cross feed valve exists, allowing System A and System B to share the reactant supplies. In addition, the ascent fuel and oxidizer reactant feed lines can be connected to the RCS systems, consuming reactants from this system as well, saving reactants from the RCS systems.



Below you can see a diagram showing the feed lines reaching the thrusters.

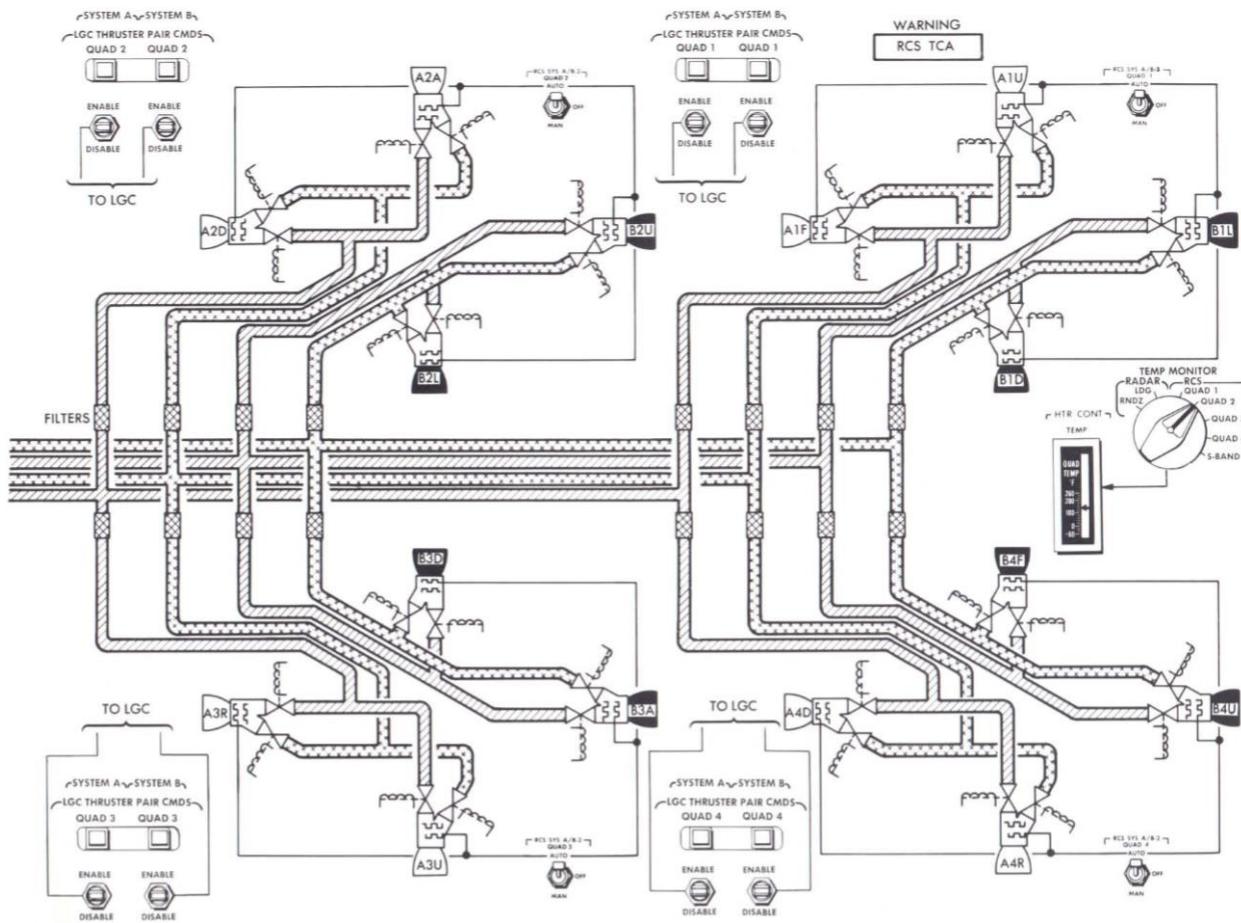
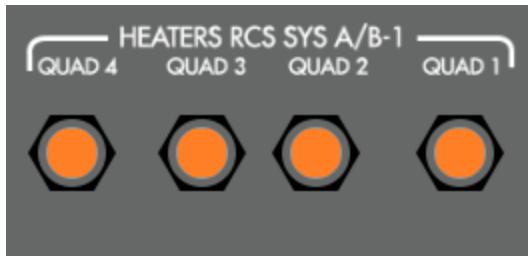


Figure 6.2.2.4 – Propellant lines and Thruster Flow Diagram

### 2.3. HEATERS

Thirty-two heaters are used to heat the 16 TCA's; two heaters are attached to each TCA flange. The heaters are fed by a 28-volt d-c input; each heater consumes 17.5 watts at 24 volts. One heater of each TCA receives power from the SYS A/B-1 circuit breaker of its respective cluster; the other heater, from the SYS A/B-2 circuit breakers.



*Heater circuit breakers on P11*

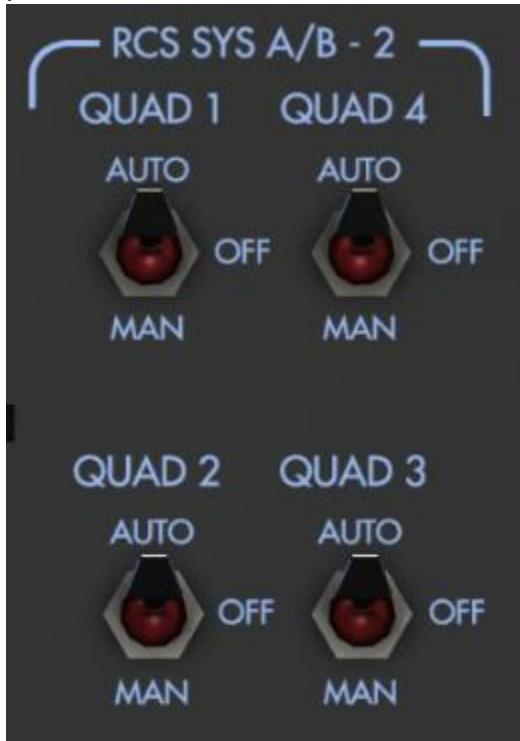


*Heater circuit breakers on P16*

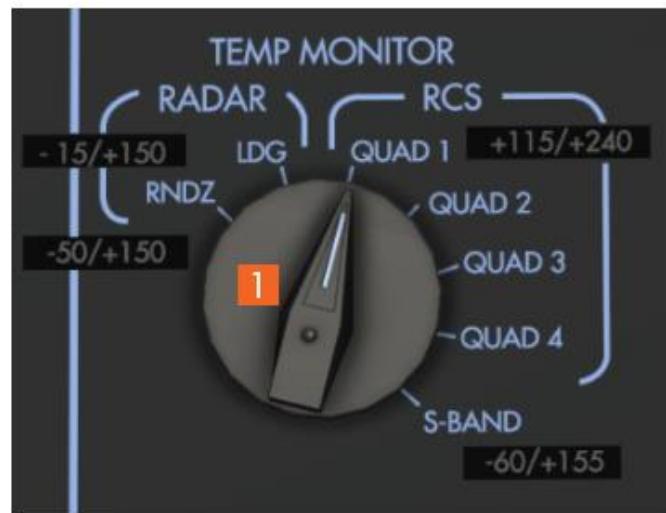
In effect, two redundant, independently operating heating systems are used to heat each TCA cluster. The heaters normally operate in an automatic mode, in which redundant thermal switches (two connected in parallel for each heater) sense the TCA injector head temperature and turn the heaters on and off, as required, to maintain TCA temperature within the required range. The primary thermal switches are set to maintain TCA temperature at  $+140^{\circ}\pm 8^{\circ}$  F; the redundant thermal switches are set for  $+147^{\circ}\pm 7^{\circ}$  F. A 2° deadband permits the temperature to drop 2° below the turn-on temperature.

The heaters of system A/B-1 are connected directly to their circuit breakers and cycle on and off, as required, unless the appropriate circuit breaker (HEATERS RCS SYS A/B-1: QUAD circuit breakers on panel 11) is opened. The heaters of system A/B-2 are connected to their circuit breakers (HEATERS: RCS SYS A/B-2 QUAD circuit breakers on panel 16) through four three-

position switches (one for each cluster) on Panel 3.



Normally, these switches (RCS SYS A/B-2 QUAD 1 through QUAD 4 on panel 3) are set to AUTO to cycle the heaters on and off to maintain the desired temperature. With the switch set to OFF, the circuit is interrupted and the heaters in the particular cluster are off. With the switch set to MAN, the thermal switches are bypassed and the heaters remain on regardless of the temperature.



The astronauts can determine the temperature of each cluster by using the TEMP MONITOR selector switch and TEMP indicator (panel 3). The inputs to the TEMP indicator originate at four temperature transducers, one for each quad. The heaters of the off-temperature cluster can be disconnected from the electrical power supply by opening the appropriate heater QUAD circuit breaker.

# VII. ELECTRICAL POWER



## VII. ELECTRICAL POWER

### 1. GENERAL

The Electrical Power Subsystem (EPS), principal source of electrical power for the LM, consists of a d-c section and an a-c section. Both sections supply operating power to respective electrical buses, which supply all LM subsystems through circuit breakers.

The EPS is controlled through panel 14:



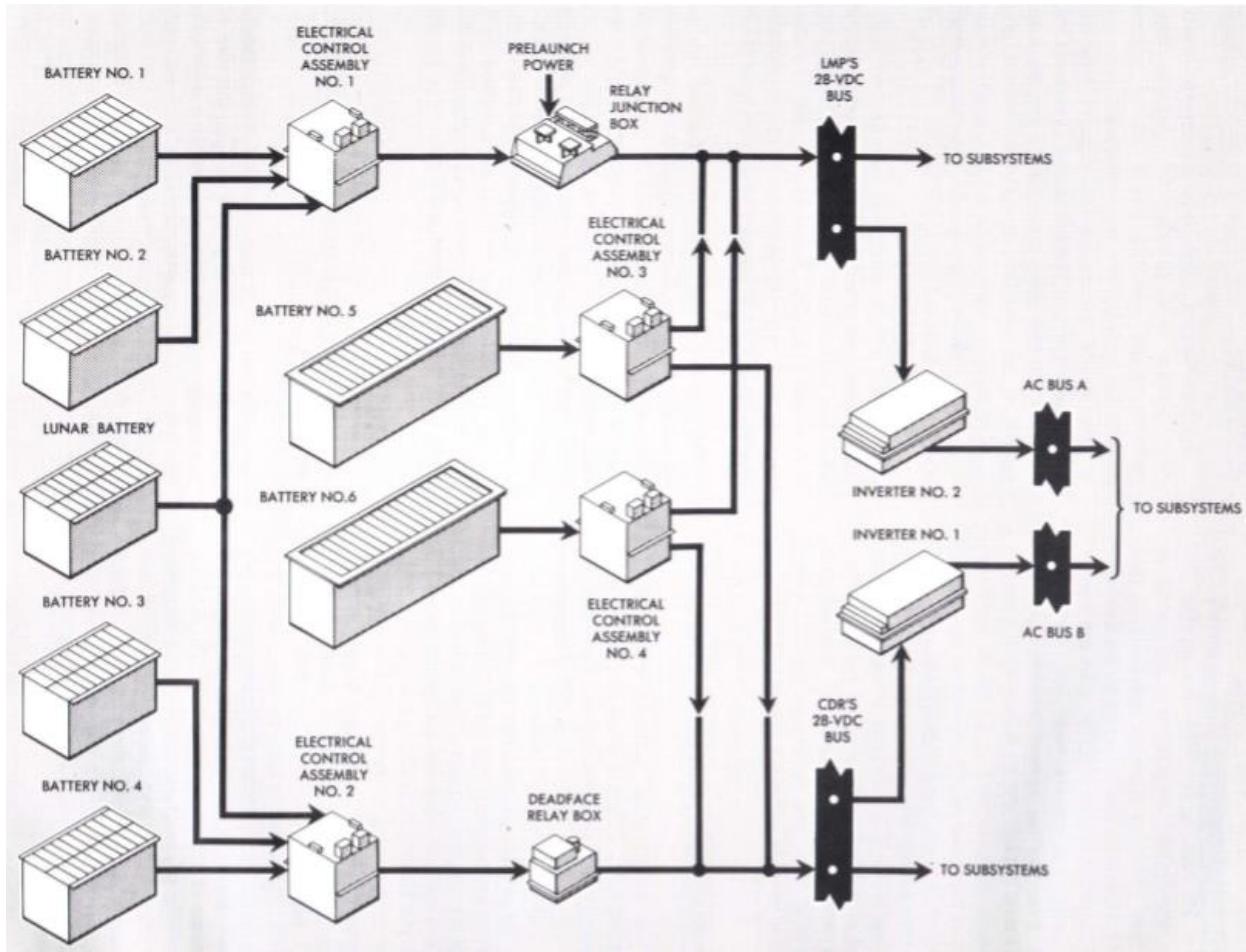
Electrical power is supplied by five batteries (one, a lunar battery) in the descent stage and two in the ascent stage. Two descent stage batteries (No. 1 and 4) power the LM from T -30 minutes until after transposition and docking, at which time the LM receives electrical power from the CSM. After separation from the CSM, during the powered descent phase of the mission, four descent stage batteries are paralleled with the ascent stage batteries. Paralleling the batteries ensures minimum required voltage for all LM operations. During lunar stay, specific combinations of the five descent stage batteries can be paralleled to provide LM power.

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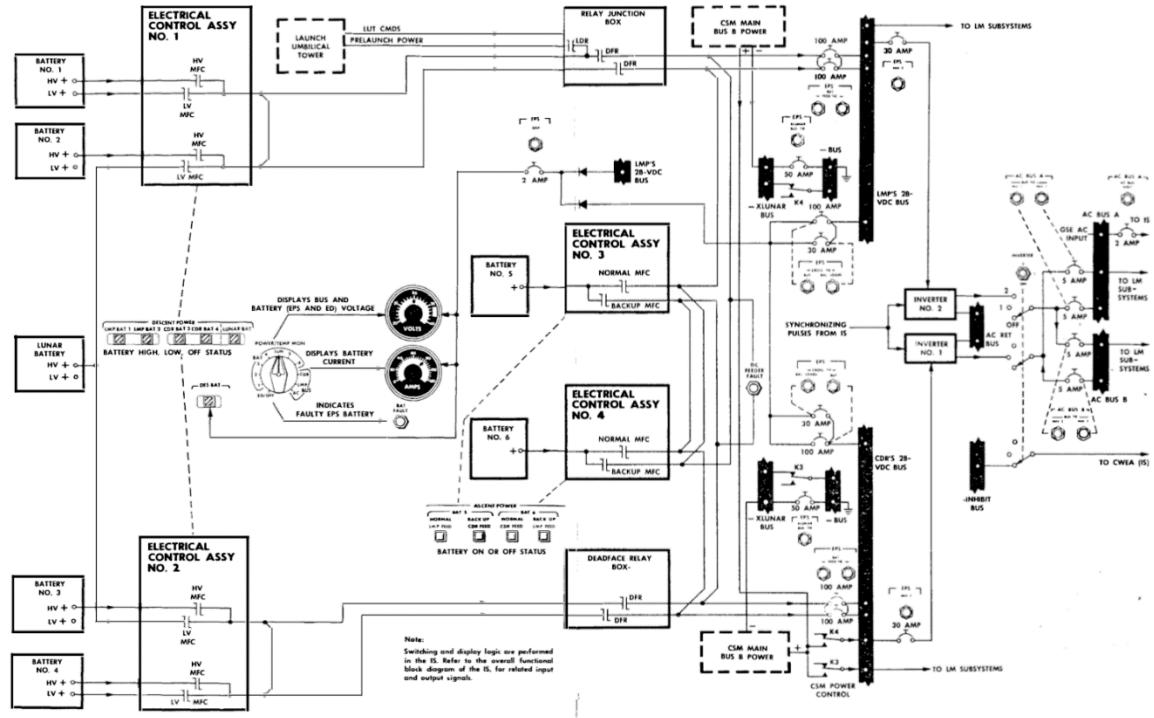


Before lift-off from the lunar surface, ascent stage battery power is introduced, descent power is terminated, and descent feeder lines are deadfaced and severed. Ascent stage battery power is then used until after final docking and astronaut transfer to the CM.

The EPS batteries are controlled and protected by electrical control assemblies, a relay junction box, and a deadface relay box, in conjunction with the control and display panel. A battery control relay assembly adapts the lunar battery to the four descent battery control circuits. Primary a-c power is provided by two inverters, which supply  $115 \pm 2.5$ -volt,  $400 \pm 0.4$ -cps (synced condition), a-c power to LM subsystems. The operating frequency of the inverters is  $400 \pm 10$  cps, in the non-synced, free-running condition.

Electrical power is distributed to LM subsystems via the LM Pilot's d-c bus, the Commander's d-c bus, and a-c buses A and B. Secondary d-c power distribution is provided by two sensor-power fuse assemblies, which supply d-c power to sensors of the Environmental Control Subsystem (ECS), Reaction Control Subsystem (RCS), and Main Propulsion Subsystem (MPS).

Batteries in the Explosive Devices Subsystem (EDS) provide primary power to trigger LM explosive devices. Other batteries supply power to operate the portable life support system (PLSS) and scientific equipment. These batteries are in addition to those in the EPS.



### 1.1. EXTERNAL POWER

The EPS provides the primary source of electrical power for the LM during the mission and is the distribution point for externally generated power during prelaunch and docked operations. The EPS interfaces with all functional subsystems of the LM. The Commander's and LM Pilot's buses supply 28 volts d-c; a-c buses, A and B, supply 115-volt, 400-cps ac. Prelaunch d-c and a-c power is initially supplied to the EPS from external ground power supplies: d-c power to the Commander's and LM Pilot's buses and a-c power to a-c bus A. (The a-c power can be routed through circuit breakers to a-c bus B.) Approximately 7 hours before launch, ground-supplied power is discontinued and d-c power is connected from the launch umbilical tower (LUT) to the LM Pilot's bus via the relay junction box (RJB). The EPS distributes internally generated d-c power from launch until LM-CSM docking, at which time LM power is shut down and the CSM supplies 28 volts d-c to the Commander's bus. (The d-c power can be routed through circuit breakers to the LM Pilot's bus.) Ground return is effected via the floating translunar bus. (Ground return in the LM is generally accomplished through the metal structure, whereas ground return in the CSM is isolated from the CSM structure.) Before LM-CSM separation, all LM internally supplied electrical power (a-c and d-c) is restored.

Circuits in electrical control assemblies (ECA's) of the EPS monitor and control LM electrical power. The circuits route discrete output signals if operating limits are exceeded. The output signals are fed to the Instrumentation Subsystem (IS) for conditioning before indicator, talkback, or component caution light display and telemetering. The EPS monitoring signals are representative of malfunctioning batteries, battery currents and terminal voltages, and feed line/battery status. The conditioned analogs of these signals are then routed from electronic replaceable assemblies (ERA's) No. 1 and 2 in the IS to the ELECTRICAL POWER portion of panel 14 for related display. In addition, primary bus-monitoring signals are connected from the LM Pilot's and Commander's 28-volt buses and a-c bus A to the IS for conditioning and subsequent voltage display on panel 14.

Synchronization pulses are supplied from the pulse-code-modulation and timing electronics assembly (PCMTEA) of the IS to the inverters to accurately control a -c output frequency. An inverter inhibit signal path is completed in the caution and warning electronics assembly (CWEA) of the IS when the INVERTER switch (panel 14) is set to OFF.

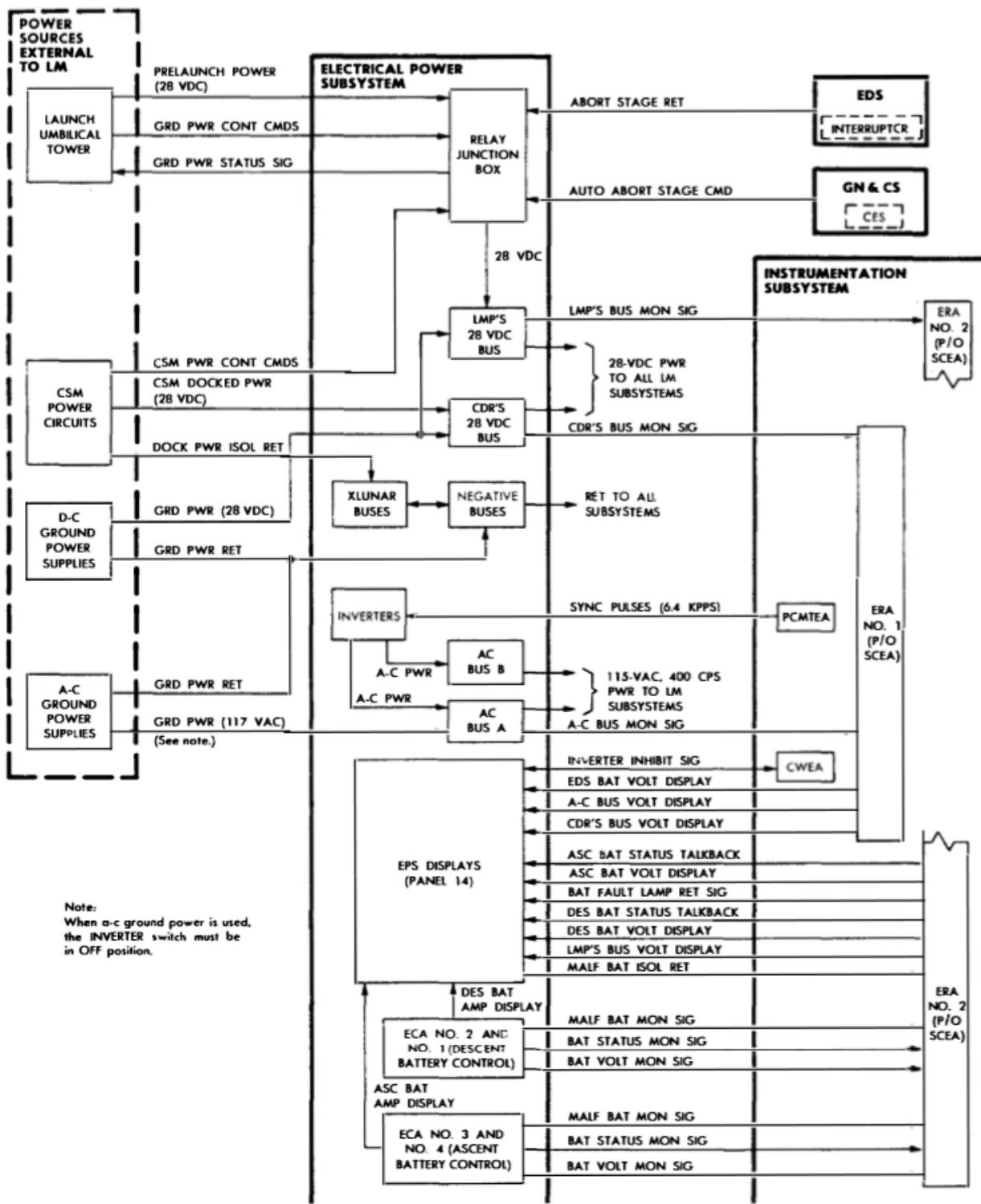
Pressing the ABORT STAGE pushbutton (panel 1) supplies an automatic abort stage command from the Guidance, Navigation, and Control Subsystem (GN&CS) to the RJB, with a return path via the EDS circuit interrupters. Abort staging initiates a switchover from descent d-c power to ascent d-c power before stage separation.

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## 2. FUNCTIONAL DESCRIPTION

### 2.1. GENERAL

The output of each battery is applied to an ECA. The descent stage ECA's provide an independent control circuit for each descent battery. The ascent stage ECA's provide four independent battery control circuits, two control circuits for each ascent battery.

Reverse-current (R/C) and overcurrent (O/C) conditions are monitored by the ECA's.

Temperature is monitored within each battery, the overtemperature (O/T) signal is enabled by the ECA's. Each ECA battery control circuit can detect a bus or feeder malfunction and operate a main feeder contactor (MFC) associated with the malfunctioning battery to remove the battery from the EPS distribution system (overcurrent only). Only the ascent power backup contactors do not have overcurrent protection.

Descent stage battery power is controlled and monitored by the descent ECA's, which switch and apply the power to the RJB and the deadface relay box (DRB). The power feeders of descent batteries No. 1 and 2 are routed through the RJB deadface relay to the LM Pilot's circuit breaker panel (16). The power feeders of batteries No. 3 and 4 are routed through the DRB deadface relay to the Commander's circuit breaker panel (11). The power from the lunar battery can be routed through the RJB or DRB deadface relays to the LM Pilot's or Commander's circuit breaker panel. In all instances, this circuit arrangement provides for deadfacing all descent battery power in the event of normal staging or abort staging situations. Ascent stage battery is controlled and monitored by the ascent ECA's, where the power is switched and applied, through power feeders, to terminal junction points at the RJB and DRB.

Ascent and descent battery main power feeders are routed through circuit breakers to the d-c buses. From the d-c buses, d-c power is distributed through circuit breakers to all LM subsystems.

Two inverters, which make up the EPS a-c section power source, supply all a-c power requirements of the LM. The INVERTER switch connects either inverter to the circuit breakers that supply the a-c buses. Closing the appropriate AC BUS A and AC BUS B circuit breakers on panel 11 connects the selected inverter to feed the a-c buses. Either inverter can supply the LM a-c load requirements.

### 2.2. D-C SECTION

The EPS d-c section consists of the four descent stage batteries, two descent stage ECA's, two relay switching boxes, two ascent stage batteries, two ascent stage ECA's, a lunar battery, a battery control relay assembly, a control panel, two circuit breaker panels, and two sensor power fuse assemblies.

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After ground facility power is disconnected, descent stage batteries No. 1 and 4 are connected at their LV (low voltage) taps. These batteries are under light load (less than 200 watts) at this time; therefore, the LV taps (at the 17th cell of the 20-cell batteries) are used to facilitate discharge of the inherent initial high voltage of the battery. During normal EPS operation, beginning with subsystem activation, descent batteries No. 1 and 4 HV (high voltage) taps and batteries No. 2 and 3 are selected with switches on panel 14.



As indicated in the picture above, switches are used to connect the batteries, and a talkback indicator shows if they are on line or not. Barber poled (LMP BAT 2) means it is off line, and gray means it is on line.

Power is then fed from the descent stage batteries through their associated ECA's (batteries No. 1 and 2 through ECA No. 1, batteries No. 3 and 4 through ECA No. 2, and, the lunar battery through ECA No. 1 or 2) to the ascent stage via the RJB and DRB, terminating at the LM Pilot's and Commander's d-c buses.



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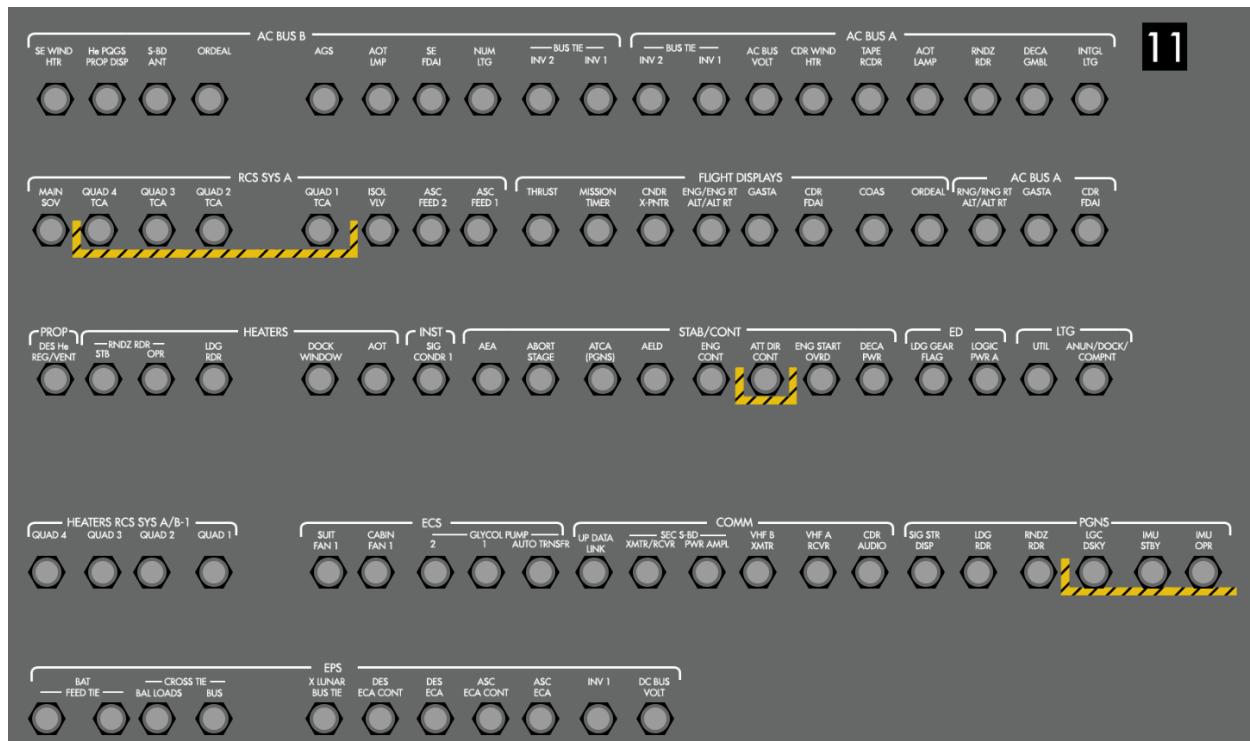
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The battery feeders are connected to the d-c buses through the 100-ampere EPS: BAT FEED TIE circuit breakers (panels 11 and 16). Each of the CDR and the LMP bus has the BAT FEED TIE A and B fuses. A is to the left, and B to the right on P16.

From the d-c buses, d-c power is then distributed to the LM subsystems and to the EPS inverters through circuit breakers on panels 16 and 11.

The following fuses are located on Panel 11.



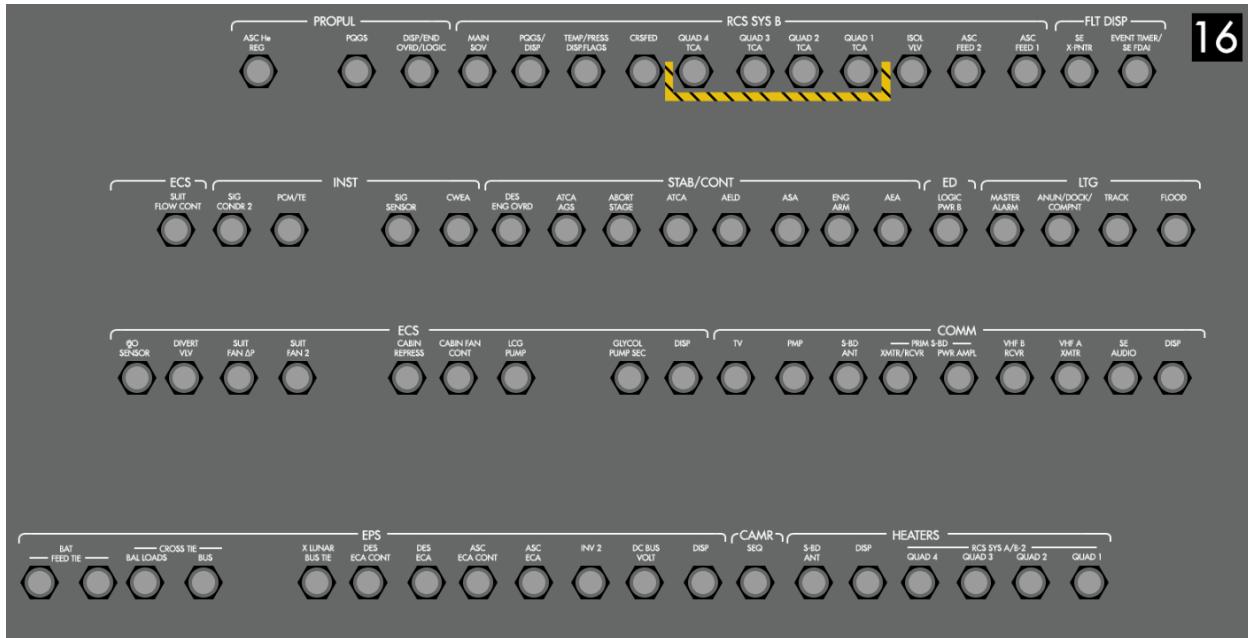
The following fuses are located on Panel 16.

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During non-critical phases of normal operation, the 30-ampere EPS: CROSS TIE BAL LOADS circuit breakers (panels 11 and 16) are closed to distribute unbalanced loads between buses so that the batteries discharge evenly. The ascent stage batteries supply d-c power to the LM Pilot's and Commander's buses in essentially the same manner as the descent stage batteries. The batteries are selected with the ASCENT POWER switches (panel 14). The corresponding descent and ascent stage batteries (by respective bus) are paralleled during DPS operation, and the EPS: CROSS TIE circuit breakers (panels 11 and 16) are opened, thereby isolating the Commander's and LM Pilot's d-c buses.

The lunar battery supplies power to input power feeders in ECA No. 1 and 2. Switches on panel 14 select the ECA that feeds the associated bus (ECA No. 1 for the LM Pilot's d-c bus, ECA No. 2 for the Commander's d-c bus). The battery control relay assembly logic, in conjunction with ECA No. 1 and 2, inhibits placing the lunar battery and battery No. 2 simultaneously, on the LMP bus, and battery No. 3 and the lunar battery simultaneously on the CDR bus. The battery control relay assembly logic also inhibits placing the lunar battery on both buses simultaneously.

Descent batteries No. 1 and 2 supply power to a power feeder in ECA No. 1. Descent batteries No. 3 and 4 are connected to separate power feeders in ECA No. 2 and are jumpered to the power feeder of the other batteries at the output of the ECA and at the 100-ampere EPS: BAT FEED TIE circuit breakers.

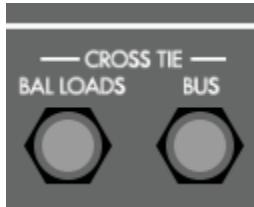


# LUNAR MODULE

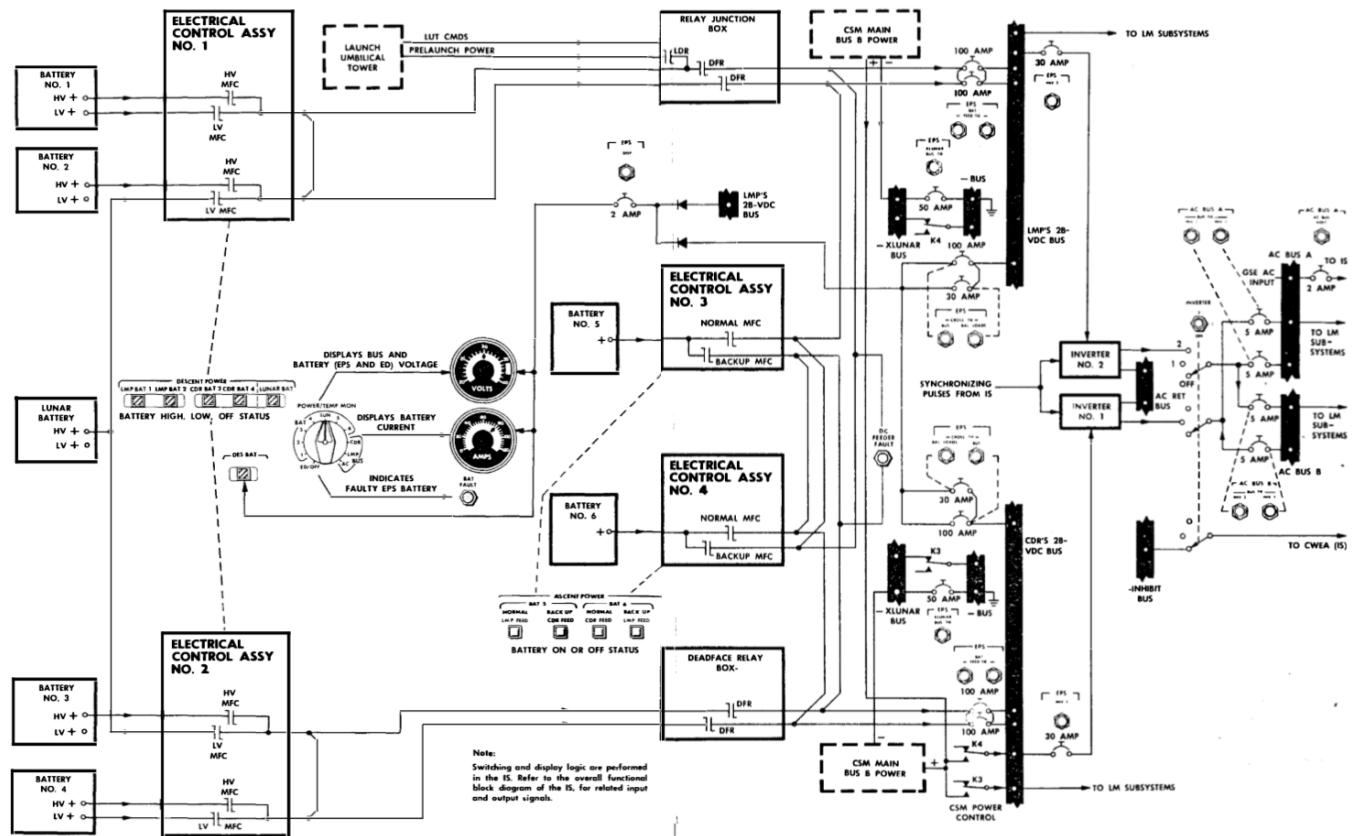
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Failure of battery No. 3 or 4 therefore permits the other batteries to supply both power feeders. A power feeder short while operating on descent battery power is detected by an overcurrent-sensing circuit in the respective ECA, which disconnects the one or two descent batteries associated with the shorted power feeder. The d-c buses are isolated from the shorted feeder by opening the respective EPS: BAT FEED TIE circuit breakers. During an emergency, either operating d-c bus can supply power to the inoperative isolated bus if the EPS: CROSS TIE BUS circuit breakers (panels 11 and 16) are closed.

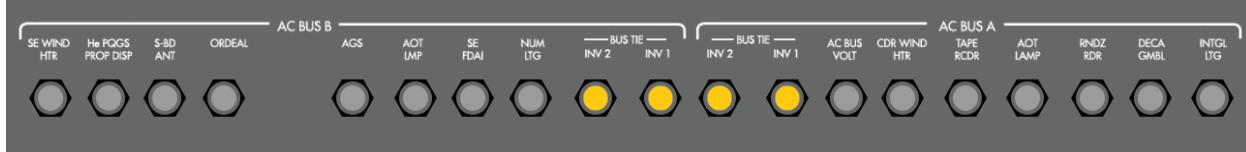


A similar condition, whereby a shorted feeder is automatically isolated, exists when operating on both ascent batteries. However, the battery associated with the shorted feeder can be connected manually to the power feeder of the other ascent battery by setting the respective BACK UP FEED switch (panel 14) to ON, thereby permitting continued use of the battery.



### 2.3. A-C SECTION

A-C power is provided to LM subsystems by either of two identical, redundant inverters; it is controlled by the EPS: INV 1 and INV 2 circuit breakers (panel 11 and 16, respectively), the INVERTER switch, and the AC BUS A: BUS TIE or AC BUS B: BUS TIE circuit breakers on panel 11.



The EPS: INV 1 or INV 2 circuit breaker supplies 28 volts d-c from the LM Pilot's or Commander's 28-volt d-c bus to the chosen normally synced inverter, where the d-c is changed to 115-volt, 400-cps, a-c power (350 voltamperes, steady-state).

The INVERTER switch on Panel 14 selects the output of either inverter and routes it to the a-c buses via the respective AC BUS: BUS TIE circuit breakers. Normally, inverter No. 2 is energized when the LM subsystems are first activated and connected to the a-c buses.

Inverter No. 1 functions as backup during the mission, except that it is the operating inverter during DPS and APS engine burns. An a-c bus voltage and frequency readout signal is supplied from a-c bus A, via the AC BUS A: AC BUS VOLT circuit breaker (panel 11), to the IS for telemetry and caution light display.

An out-of-tolerance frequency (less than 398 cps or more than 402 cps) or a low-voltage condition (less than 112 volts ac) causes the INVERTER caution light (panel 2) to go on. The astronaut determines the cause of the malfunction and performs corrective action. The INVERTER caution light goes out when the malfunction is remedied. (When the INVERTER caution light goes on, the MASTER ALARM pushbutton/light on panels 1 and 2 goes on and a tone is generated for the astronaut headsets. Pressing either MASTER ALARM pushbutton/light extinguishes the pushbutton/lights and terminates the tone.)

When set to AC BUS, the POWER/TEMP MON selector switch (panel 14) selects a-c bus A for voltage display on the VOLTS indicator (panel 14); the reading on the AMPS indicator has no significance.

## 3. POWER MONITORING

### 3.1. GENERAL

The primary a-c and d-c voltage levels, d-c current consumption, and the status of all main power feeders must be monitored periodically to ensure availability of proper power for all LM subsystems, throughout the mission. This monitoring is normally the responsibility of the LM Pilot, who controls the ELECTRICAL POWER portion of panel (14). This panel has talkbacks that indicate main power feeder status, indicators that display battery and bus voltages and battery

current, and component caution lights that are used for detecting shorted buses or main power feeder lines and for isolating a malfunctioning battery. Availability of redundant a-c and d-c power permits the astronaut to disconnect, substitute or reconnect batteries, feeder lines, buses, or inverters to assure a continuous electrical supply.

### 3.2 BATTERY STATUS

Talkbacks on panel 14 indicate the status of the descent and ascent batteries (on or off the line).

A DESCENT POWER talkback barber-pole display signifies that the related battery is disconnected from its main power feeder; a gray display, showing LO, that the battery low-voltage tap is connected to the power distribution system via the main power feeder; a gray display, that the high-voltage tap is connected to the power distribution system.



A DES BATS talkback gray display signifies that the descent battery outputs are connected to the ascent stage; a barber-pole display, that these power lines are disconnected and the ascent stage is isolated electrically from the descent stage. The ASCENT POWER talkbacks indicate that their related ascent battery is connected (gray) or disconnected (barber pole) from the respective normal feed or backup feed main power feeder lines. The LUNAR BAT talkback CDR, LMP, and barber-pole displays indicate that the lunar battery is on the CDR bus, the LMP bus, or is off-line.

When a descent battery is disconnected from its main feeder line, its related talkback provides a barber-pole display. Setting a high-voltage (or low-voltage switch (panel 14)) to ON energizes

the related high-voltage (or low-voltage) MFC in the respective ECA. D-C power, supplied to the buffer stage from the LM Pilot's d-c bus, passes to the talkback upon arrival of the ECA output signal. Output closure signals of the high-voltage (or low-voltage) signal buffer stages are connected to individual coils (in the talkback assembly), which, when energized, activate talkbacks to display the status of the batteries. In addition, receipt of an ECA output signal at the signal buffer stages causes an event signal to be routed to the PCMTEA, for telemetry.

When the lunar battery is disconnected from its main feeder line in both ECA's, the LUNAR BAT talk back provides a barber-pole display. Setting the LMP LUNAR BAT switch (panel 14) to ON, energizes the related MFC relay in ECA No. 1 and activates the LUNAR BAT talkback to display LMP. When the CDR LUNAR BAT switch is set to ON, the related MFC relay in ECA No. 2 is energized and the LUNAR BAT talkback displays CDR.

Three conditions are associated with batteries No. 1 and 4: high-voltage tap on the power-feeder line, low-voltage tap on the power-feeder line, and battery completely off the power-feeder line. Two conditions are associated with batteries No. 2 and 3: voltage on or off the power feeder line. Three conditions are associated with the lunar battery: voltage on the LMP or CDR power feeder line, or voltage completely off the power feeder lines.



Four conditions are associated with each ascent battery: battery on or off the normal power feeder line (for that particular battery) and battery on or off the backup power feeder line. These

conditions are determined by the setting of the NORMAL FEED and BACK UP FEED switches (panel 14).

Operation of the ASCENT POWER talkbacks and generation of the activating signals are similar to that of the DESCENT POWER talkbacks, except that each signal buffer stage output closure signal is routed to an individual talkback and each talkback provides only two displays (gray or barber-pole). Telemetry signals are also generated within the PCMTEA for ascent battery monitoring.

### 3.3 BATTERY VOLTAGE & CURRENT

The VOLTS and AMPS indicators (panel 14) enable monitoring the terminal voltage and current draw of each descent battery (high-voltage tap for voltage only) and ascent battery. In addition, the VOLTS indicator, when used in conjunction with the POWER/TEMP MON selector switch and the ED VOLTS switch (panel 14), monitors the terminal voltage of ED batteries A and B of the EDS. Except that a descent ECA controls two batteries and an ascent ECA controls one battery, the voltage- and current-monitoring circuits of each EPS battery are essentially the same. Therefore, the voltage- and current-monitoring circuits of only one EPS battery are described.



The positive terminal of the EPS battery is connected through a fuse (in the respective ECA) to an attenuator in ERA No. 2 of the IS. The attenuator input signal is in the range of 0 to 40 volts dc. The resultant attenuator output (0 to 5 volts dc) is an analog of the input signal. This output

is routed directly to its assigned terminal of the POWER/TEMP MON selector switch. When the switch is set to the battery position (BAT 1 through BAT 6, and LUN), the analog is routed to the positive terminal of the VOLTS indicator and displayed. (The indicator scale is graduated from 20 to 40 volts dc; although the input signal is of considerably smaller magnitude.) Only the high-voltage positive tap of each descent battery is monitored in this manner. Each ascent battery has only one positive terminal.

Battery current flowing through the main feeder line (normal or backup feeder for ascent batteries, high- or low-voltage feeder for descent batteries) is sensed by the related current monitor coil in its respective ECA. Each battery has an individual current monitor circuit; the lunar battery uses current monitor circuits of battery No. 2 for the LMP bus; and current monitor circuits of battery No. 3, for the CDR bus. The monitor senses the magnitude of current flow through the associated main feeder line and provides a representative analog output. This analog is routed to a specific terminal (assigned by battery number) on a deck of the POWER/TEMP MON selector switch, as are the current analogs of all other ascent and descent batteries. When the switch is set to the battery position, the analog is routed to the positive terminal of the AMPS indicator and displayed. (The indicator scale is graduated from 0 to 120 amperes, although its input signal is of considerably smaller magnitude). Ascent battery current is read directly from the indicator; for descent battery current, the indicator reading must be halved.

The positive and negative terminals of the two ED batteries are connected to respective contacts of the ED VOLTS switch. When the switch is set momentarily to BAT A or BAT B, that particular battery is connected to signal conditioner (SC) No. 1 in the IS. The SC output is a conditioned analog (0 to +5 volts dc) that represents the actual ED battery terminal voltage (0 to +40 volts dc), which is routed to the ED/OFF terminal of the POWER/TEMP MON selector switch, for voltage monitoring.

### 3.4 PRIMARY BUS VOLTAGE

The Commander's d-c bus, the LM Pilot's d-c bus, and a-c bus A are monitored periodically for the proper voltage level, to ensure application of adequate power to all LM subsystems. Monitoring is accomplished by selecting the bus, with the POWER/TEMP MON selector switch, and reading the associated voltage on the VOLTS indicator. The readings on the AMPS indicator have no significance. D-C bus voltages are read directly from the indicator scale, which is graduated from 20 to 40 volts dc; for a-c readings, conversion is required. The d-c scale is equivalent to a range of 62.5 to 125 volts ac, when monitoring the a-c bus. The indicator green band is equivalent to the nominal a-c range of 112 to 118 volts.

The Commander's d-c bus is connected via the EPS: DC BUS VOLT circuit breaker (panel 11) to an attenuator in ERA No. 1 of the IS. The attenuated signal is routed to the CWEA and through an isolation amplifier. The 0- to 5-volt attenuated d-c signal is an analog, which is fed to the PCMTEA for telemetry and to the CDR terminal of the POWER/TEMP MON selector switch.

Except that it is processed through ERA No. 2, the LM Pilot's d-c bus analog is connected to the LMP terminal of the switch in like manner. The attenuated analogs of the Commander's and LM Pilot's d-c bus voltages are fed through buffer amplifiers and an OR gate. If the d-c voltage on either bus drops below 26.5 volts, a CWEA relay is energized; its contacts close to provide a ground return to the DC BUS warning light (panel 1), causing it to go on. The warning light goes off when the bus voltage is again within limits.

A-C bus A (115 volts) is connected via the AC BUS A: AC BUS VOLT circuit breaker to a frequency-to-dc stage and an ac-to-dc stage in ERA No. 1. The d-c output signals are fed to the CWEA and to isolation amplifiers. The amplifier outputs are fed to the PCMTEA for telemetry. A 0- to 5-volt d-c analog signal from the ac-to-dc stage isolation amplifier is sent to the AC terminal of the POWER/TEMP MON selector switch. Selecting this switch position routes the analog signal to the positive terminal of the VOLTS indicator. The output signals of the frequency-to-dc and ac-to-dc converters are also fed through buffer amplifiers and an OR gate to an AND gate in the CWEA. An inhibit signal is routed to this AND gate when the INVERTER switch is set to OFF, thereby inhibiting signal passage. Setting the switch to 1 or 2 removes the inhibit signal. If the a-c bus A voltage is less than 112 volts, or if the a-c bus frequency is less than 398 cps or more than 402 cps, a CWEA relay is energized; its contacts close to provide a ground return to the INVERTER caution light (panel 2), causing it to go on. The INVERTER light goes off when the a-c bus voltage and frequency are again within limits.

### 3.5 MALFUNCTIONING BATTERY ISOLATION

The BATTERY caution light (panel 2) goes on when there is an overtemperature, overcurrent, or reverse-current condition in any EPS battery. The malfunction isolation circuitry associated with each battery, within its respective ECA, is essentially the same. Only the isolation circuitry of one battery in ECA No. 1 is discussed.

D-C power from the related SC in the IS is supplied to the paralleled array of malfunction relay contacts within ECA No. 1. Thermal sensors, in parallel within the battery, close if any cells overheat; a reverse-current or overcurrent condition causes the respective relays in the ECA to become energized and related relay contacts close. If either of these conditions exists, the d-e power from the SC flows through the respective set of contacts back to the SC and to a CWEA OR gate. The OR-gate output energizes a CWEA relay that provides a ground return to the BATTERY caution light, causing it to go on. The caution light goes off when the malfunction that caused it to go on is eliminated.

The d-e signal fed back to the SC from the ECA activates an event gate, which provides a channeled signal to the pulse code modulator of the IS, for telemetry. This same d-e signal also activates a contact closure gate, completing a circuit from the BAT FAULT component caution light (panel 14) to the respective battery terminal of the POWER/TEMP MON selector switch. Setting the switch to this battery position applies a ground return to the light, causing it to go on. The malfunctioning battery is identified by the placarded switch position. The light goes off when the malfunction is corrected, or the battery is removed from its main power feeder line by momentarily setting the respective switch (panel 14) to OFF/RESET.

## 4. EPS OPEREATION

### 4.1. GENERAL

The EPS operates in seven basic stages, as follows:

- GSE-vehicle ground power supply (VGPS) to LM until T-7 hours (ac and dc)
- GSE-launch umbilical tower (LUT) power to LM from T-7 hours until T-30 minutes (dc only)
- Descent stage power from T-30 minutes through CSM rendezvous and docking (two descent batteries on low-voltage taps)
- Lunar orbit - CSM/LM power
- Descent stage power (two descent batteries manually switched to high-voltage taps and remaining descent batteries switched on)
- Ascent stage power (on normal main feeders)
- Staging (normal or abort)

### 4.2. GSE-SUPPLIED PRELAUNCH POWER

Before earth launch, a-c and d-c power are supplied to the LM by the VGPS. An umbilical cable or a portable power supply is carried onboard the LM and connected to two GSE connectors, one under each center side console. At approximately T-7 hours, the VGPS power is replaced by LUT power. The SLA/LM assembly is now connected to the LUT by umbilical cable. LUT power is supplied to the LM Pilot's bus through the RJB. LUT remote control includes LUT deadface relay (LDR) set and reset commands, and LM high- and low-voltage off and low-voltage on signals. The LDR supplies power through the LM Pilot's main power feeders to the LM Pilot's d-c bus. D-C power is supplied to the Commander's bus through the EPS: CROSS TIE circuit breakers.

### 4.3. DESCENT STAGE POWER (LOW-VOLTAGE TAPS)

At T-30 minutes, the LUT resets the LDR and connects the EPS descent battery low-voltage taps to the respective feeder lines. The descent batteries supply limited power to certain critical equipment from T-30 minutes until completion of CSM transposition and docking.

### 4.4. LUANR ORBIT – CSM/LM POWER

On completion of CSM rendezvous and docking, an astronaut connects power and control umbilical cables from the LM to the CSM. The CSM deactivates LM power and then supplies power to the LM critical equipment, using the LM translunar negative bus. CSM/LM control logic prohibits CSM power to the LM power distribution system while the LM descent stage batteries are on the main power feeder lines.

CSM/LM control logic is as follows:

- CSM deactivates LM power (low voltage off)
- CSM initiates CSM power to LM
- CSM deactivates CSM power to LM
- CSM initiates LM power on (low voltage on).

### 4.5. DESCENT STAGE POWER (HIGH-VOLTAGE TAPS)

Batteries No. 1 and 4 have HI-V and LO-V switches, and talkbacks, on panel 14. Batteries No. 2 and 3 have only ON-OFF/RESET switches, and talkbacks. The lunar battery has an ON-OFF/RESET switch for the CDR and LMP bus, and a talkback that indicates which bus is selected.

Descent stage battery low-voltage and off/reset control lines are routed through the RJB to the appropriate ECA; high-voltage control lines are routed directly to the respective ECA. Additional low-voltage and off/reset control is effected within the RJB through internal logic control signals. Descent battery No. 2 supplies electrical power to the LM Pilot's d-c bus via ECA No. 1 and the RJB. Descent batteries No. 3 and 4 supply the Commander's d-c bus via ECA No. 2 and the DRB.

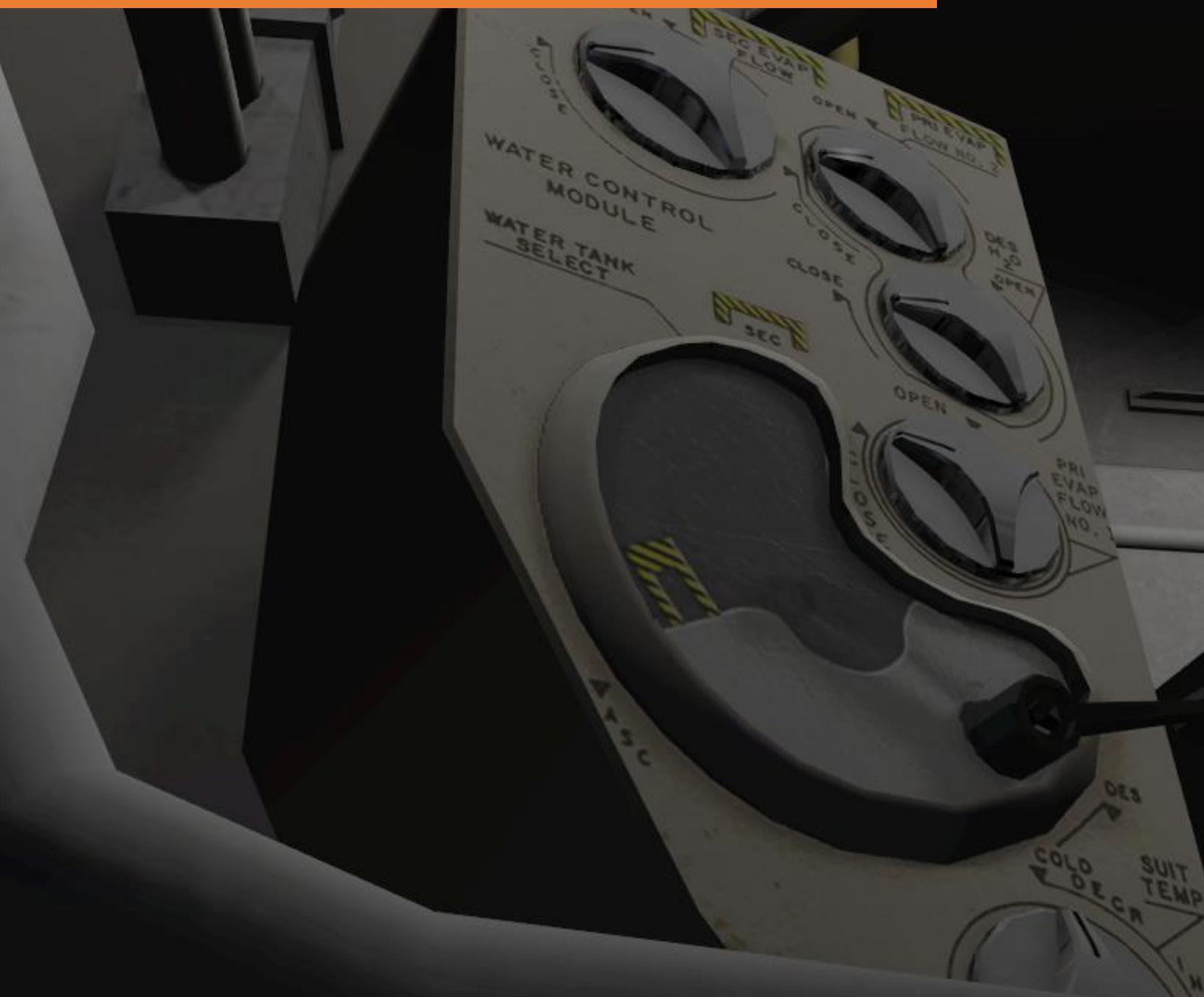
### 4.6. ASCENT STAGE POWER

Each of the two ascent batteries has a NORMAL and BACK UP feeder switch and related talkbacks on panel 14. Battery No. 5 normally supplies power to the LM Pilot's d-c bus via ECA No. 3. Battery No. 6 normally supplies power to the Commander's d-c bus via ECA No. 4.

#### 4.7. STAGING – NORMAL OR ABORT

Before stage separation, the LM d-c bus loads are transferred from the descent stage batteries to the ascent stage batteries; d-c power to the bus power distribution system is maintained uninterrupted. Automatic transfer is initiated by pressing the ABORT STAGE pushbutton (panel 1); this applies 28 volts dc to the EPS abort stage relays in the RJB. These relays close the normal MFC's of each ascent ECA (if the ECA's have not been selected previously in the backup feed mode). After the ascent stage battery MFC's have placed the ascent stage batteries on the line, additional abort stage relay contacts supply an off/reset signal through the RJB, to each descent stage battery and to the deadface relays (DFR's). Transfer from descent stage battery power to ascent stage battery power can also be initiated manually. In manual transfer, the ascent batteries must be placed on the line before initiating transfer with the DES BATS switch (panel 14). This is accomplished by setting the ASCENT POWER switches (panel 14) to ON, setting appropriate DESCENT POWER switches (panel 14) to OFF/RESET, then setting the DES BATS switch to DEADFACE. Auxiliary DFR contacts in the RJB and DRB provide direct control of the DES BATS talkback for monitoring the deadface operation.

## VIII. ENVIRONMENTAL CONTROL



# VIII. ENVIRONMENTAL CONTROL

## 1. GENERAL

### 1.1 INTRODUCTION

The ECS system is under heavy development, and its operation can be skipped, although some of the systems are fully functional.

The Environmental Control Subsystem (ECS) provides a habitable environment for two astronauts while the LM is separated from the CSM, and permits the astronauts to decompress and repressurize the cabin in accordance with mission requirements. It also controls the temperature of electronic equipment and provides water for drinking, cooling, fire extinguishing, and food preparation.

The major portion of the ECS is in the cabin. Part of the equipment-cooling loop and two oxygen tanks are in the aft equipment bay. Two water tanks are in the upper midsection, two larger oxygen tanks and two larger water tanks are in the descent stage. (See figures 2.6-1 and 2.6-2.) Power for the ECS is supplied from the Electrical Power Subsystem. (See figures 2.6-3 and 2.6-5.) The ECS (figure 2.6-4) comprises an atmosphere revitalization section (ARS), an oxygen supply and cabin pressure control section (OSCPGS), a water management section (WMS), a heat transport section (HTS), and provisions for supplying oxygen and water to the portable life support system (PLSS).

### 1.2 ATMOSPHERE REVITALIZATION SECTION

The ARS consists of a suit circuit assembly, suit liquid cooling assembly, and steam flex duct leading to a vent. The suit circuit assembly is a closed-loop recirculation system that cools and ventilates the pressure garment assemblies (PGA's); maintains a desirable level of carbon dioxide in the atmosphere; removes odors, particles, noxious gases, and excessive moisture; enables control of the temperature of oxygen flow to the PGA's; and, if required, automatically isolates the PGA's from the system. The suit liquid cooling assembly circulates water and controls its temperature in the liquid-cooled garment, circulates cabin air, when required, and removes lunar dust from the cabin during ascent from the lunar surface. The steam flex duct carries steam discharged from the suit circuit water sublimator, when operating, to ambient vacuum.

Oxygen circulation is maintained by centrifugal fans. Heating and cooling are accomplished by passing the oxygen through heat exchangers, where heat is either surrendered to, or picked up from, the coolant of the HTS. Water separators remove excess moisture after the cooling phase.

### 1.3 OXYGEN SUPPLY AND CABIN PRESSURE CONTROL SECTION

The OSC PCS stores gaseous oxygen and maintains cabin and suit pressure by supplying oxygen to the ARS. This replenishes losses due to crew metabolic consumption and cabin or suit leakage. There are four oxygen supplies: two, in the descent stage, provide oxygen during the descent and lunarstay phases of the mission; two, in the ascent stage, during the ascent and rendezvous phases of the mission.

### 1.4 WATER MANAGEMENT SECTION

The WMS supplies water for drinking, cooling, firefighting, food preparation, and for refilling the PLSS cooling water tank. It also provides for delivery of water from ARS water separators to HTS sublimators and from the water tanks to ARS and HTS sublimators. The water tanks are pressurized before launch, to maintain the required pumping pressure in the tanks. The two descent stage tanks supply most of the water required until staging occurs. After I staging, water is supplied by the two ascent stage tanks. Pressure-regulated water from the tanks, along with ARS water, is delivered to the HTS I sublimators via manually controlled shutoff valves. A manual disconnect and shutoff valves are provided ' for control and use of ascent and descent tank water for drinking, food preparation, fire extinguishing, and P LSS refills.

### 1.5 HEAT TRANSPORT SECTION

The HTS consists of a primary coolant loop and a secondary coolant loop. The secondary loop serves as a backup loop and functions in the event that the primary loop fails. A water-glycol solution circulates through each loop. The primary loop provides temperature control for batteries, electronic equipment that requires active thermal control, and for the water that circulates through the LCG and the oxygen that circulates through the cabin and PGS' s. The batteries and electronic equipment are mounted on cold plates and rails through which coolant is routed to remove waste heat. The cold plates used for equipment that is required for mission abort contain two separate coolant passages; one for the primary loop and one for the secondary loop. The secondary coolant loop, which is used only if the primary loop is inoperative, serves only these cold plates. In-flight waste heat rejection from both coolant loops is achieved by the primary and secondary sublimators, which are vented overboard. A coolant pump recirculation assembly contains all the HTS coolant pumps and associated check and relief valves. Coolant flow from the assembly is directed through parallel circuits to the cold plates for

the electronic equipment, the oxygen-to-glycol heat exchangers, and the water-to-glycol heat exchanger in the ARS.

## 2. SUBSYSTEM INTERFACES

An absolute pressure transducer in each descent oxygen tank feed line generates an output proportional to tank pressure. The output is routed to the Instrumentation Subsystem (IS), where it is conditioned to provide a telemetry signal, a caution indication, and through the O2 /H2O QTY MON 1 selector switch (panel 2), a display on the O2 QUANTITY indicator. When descent tank 1 pressure drops 1 below 135 psia (approximately 5% of capacity), a signal is routed to the O2 QTY caution light. Momentarily setting the O2 /H2O QTY MON selector switch to C/W RESET extinguishes the light.

A pressure transducer in the fill line of each ascent oxygen tank generates an output that is also conditioned in the IS to provide a telemetry signal, a caution indication, and a display on the O2 QUANTITY indicator. The quantity of oxygen remaining in the tanks is read on the indicator by setting the selector switch to ASC 1 or ASC 2, as applicable. The O2 QTY caution light goes on if, before staging, the pressure in either ascent oxygen tank is less than 684 psia (less-than-full condition). After staging, the signal that causes this indication is inhibited; instead, the light goes on when pressure in ascent tank No. 1 is less than 100 psia. The caution light is extinguished by setting the selector switch to C/W RESET.

## 3. FUNCTIONAL DESCRIPTION

The functional description of each of the four major ECS sections is supported by a functional flow diagram which, to reduce complexity, does not contain electrical circuitry, Figure 2.6-5 shows all ECS electrical circuits and all ECS interfaces with the Electrical Power Subsystem (EPS) and IS.

### 3.1 ATMOSPHERE REVITALIZATION SECTION

The ARS is a recirculation system that conditions oxygen by cooling or heating, and dehumidifying and deodorizing it for use within the PGS's and cabin; it also circulates water through the liquid-cooled garment to provide cooling during peak heat loads; during ascent from the lunar surface, it removes dust from the cabin atmosphere. The major portion of the ARS is within the suit circuit assembly.

In normal operation the SUIT ISOL valve is set to SUIT FLOW and conditioned oxygen flows to the PGS's and is discharged through the return umbilical to the suit circuit. Suit circuit pressure, sensed at a point downstream of the suits, is referenced to the oxygen regulators that control

pressure by supplying makeup oxygen to the suit circuit. The suit circuit relief valve protects the suit circuit against overpressurization, by venting to cabin ambient.

The CABIN position of the SUIT GAS DIVERTER valve is used during pressurized-cabin operation, to divert sufficient conditioned oxygen to the cabin to control carbon dioxide and humidity levels. Pulling the valve handle selects the EGRESS position to isolate the suit circuit from the cabin. The E GRESS position is used for all unpressurized cabin operations as well as closed suit mode with pressurized cabin. An electrical solenoid override automatically repositions the valve from CABIN to EGRESS when cabin pressure drops below the normal level or when the PRESS REG A or PRESS REG B valve is set to EGRESS.

With the SUIT GAS DIVERTER valve set to CABIN, cabin discharge oxygen is returned to the suit circuit through the CABIN GAS RETURN valve. Setting the CABIN GAS RETURN valve to AUTO enables cabin pressure to open the valve. When the cabin is depressurized, differential pressure closes the valve, preventing suit pressure loss.

A small amount of oxygen is tapped from the suit circuit upstream of the PGA inlets and fed to the carbon dioxide partial pressure sensor, which provides a voltage to the PART PRESS C02 indicator (panel 2).

The primary and secondary carbon dioxide (C02) and odor removal canisters are connected to form a parallel loop. The primary canister contains a LM-size cartridge; the secondary canister, a PLSS-size cartridge. A debris trap in the primary canister cover prevents particulate matter from entering the cartridge. A relief valve in the primary canister permits flow to bypass the debris trap if it becomes clogged. Oxygen is routed to the C02 and odor removal canisters through the C02 CANISTER SEL valve. The carbon dioxide level is controlled by passing the flow across a bed of lithium hydroxide (LiOH); odors are removed by absorption in activated charcoal. When carbon dioxide partial pressure reaches or exceeds 7. 6 mm Hg, as indicated on the PART PRESS C02 indicator, the C02 component caution light and ECS caution light go on. The C02 CANISTER SEL valve is then set to SEC, placing the secondary canister onstream. This unlocks the cover on the PRIM C02 CANISTER. The PRIM C02 CANISTER VENT pushbutton is pushed to release pressure from the canister before the canister cover is removed. The primary cartridge is replaced, the canister cover is installed, and the C02 CANISTER SEL valve is set to PRIM, placing the primary canister back onstream. The ECS caution light is extinguished by the lowering of the carbon dioxide partial pressure below 7. 6 mm Hg. The C02 component caution light is on when the C02 level is above 7. 6 mm Hg or C02 CANISTER SEL valve is in the SEC position. A flow limiter in the primary cartridge makes the flow resistance equivalent to a secondary cartridge and maintains suit circuit oxygen flowrates compatible with the water separator requirements. The flow limiter is removable in flight should additional oxygen flow be desired.

From the canisters, conditioned oxygen flows to the suit fan assembly, which maintains circulation in the suit circuit. Only one fan operates at a time. The ECS: SUIT FAN circuit breaker

is closed and the SUIT FAN selector switch is set to 1 to initiate suit fan operation. At startup, a fan differential pressure sensor is in the low position (low P), which, through the fan condition signal control, energizes the ECS caution light and SUIT FAN component caution light. The lights remain on until the differential pressure across the operating fan increases sufficiently to cause the differential pressure sensor to move to the normal position. If the differential pressure drops to 6.0 inches of water or less, the lights go on and switchover to fan No. 2 is required. The ECS caution light goes off when No. 2 fan is selected and the SUIT /FAN warning light goes on. The SUIT FAN component caution light goes off when fan No. 2 comes up to speed and builds up normal differential pressure. The SUIT/FAN warning and SUIT FAN component caution lights go off when fan No. 2 differential pressure reaches 9.0 inches of water. The fan check valve permits air to pass from the operating fan without backflow through the inoperative fan.

From the check valve, the conditioned oxygen passes through a sublimator to the cooling heat exchanger. The sublimator cools the oxygen under emergency conditions. Normally, the sublimator is inoperative; it is placed in operation when the secondary evaporator flow (SEC EV AP FLOW) valve of the water control module is opened because a failure in the primary heat transport loop renders the cooling heat exchanger inoperative. Heat transfer to the coolant in the heat exchanger reduces gas temperature and causes some condensation of water vapor.

The conditioned oxygen is next routed to two, parallel-redundant water separators through the water separator selector (WATER SEP SEL) valve. One separator, selected by pushing (SEP 1) or pulling (SEP 2) the WATER SEP SEL valve handle, is operated at a time. The separator is driven by the gas flowing through it. Moisture removed from the oxygen is discharged under a dynamic head of pressure sufficient to ensure positive flow from the separator to the WMS. A water drain carries some water from the separators to a surge (collection) tank outside the recirculation system.

The conditioned oxygen from the water separator is mixed with makeup oxygen from the OSCPCS to maintain system pressure. The mixture flows through the regenerative (heating) heat exchanger, where the temperature may be increased, to the SUIT ISOL valves. The SUIT TEMP valve on the water control module controls the flow of coolant through the regenerative heat exchanger. Setting the valve to INCR HOT increases oxygen temperature; setting it to DECR COLD reduces the temperature.

If a PGA tears while the SUIT ISOL valves are set to SUIT FLOW, the valves are automatically repositioned to SUIT DISC when suit circuit pressure drops to 2.9 psia minimum (they can close between 2.9 and 3.4 psia), actuated by the suit circuit pressure switch. This action isolates the PGA's from the ARS. At the same time, the CABIN REPRESS valve automatically opens, repressurizing the cabin, if the CABIN REPRESS circuit breaker is closed.

### 3.1.1 SUIT LIQUID COOLING ASSEMBLY

The suit liquid cooling assembly assists in removing metabolic heat by circulating cool water through the liquid-cooled garment (LCG). A pump maintains the flow of warm water returning from the LCG through the water umbilical hoses. An accumulator in the assembly compensates for system volumetric changes and leakage. A mixing bypass valve controls the quantity of water that flows through the water-glycol heat exchanger for removal of heat; the remaining water is bypassed around the heat exchanger. This bypassed (warm) water, mixed with the cool water downstream of the heat exchanger, flows through the water umbilical hoses back to the LCG. A fan circulates cabin air through the lunar dust filter.

### 3.2 OXYGEN SUPPLY AND CABIN PRESSURE CONTROL SECTION

The ECS descent stage oxygen supply hardware consists of the following: two descent oxygen tanks, two high-pressure fill couplings, high-pressure oxygen control assembly, interstage flex line, and descent stage disconnect. The descent tanks pressure transducers, part of the IS, generate an output proportionate to tank pressure. The second descent stage oxygen tank connects to the common oxygen feed line and branches off into a high pressure PLSS fill line that has its own pressure regulator, overboard relief valve, interstage flex line, and descent stage disconnect. Two check valves prevent oxygen flow from one tank to the other.

The ascent stage oxygen supply hardware consists of the following: ascent stage disconnects, interstage flex line, oxygen module, two ascent oxygen tanks, low pressure P LSS oxygen disconnect (GFE), and the cabin pressure switch. In addition, the high pressure PLSS fill has its own interstage flex line, ascent disconnect, oxygen shutoff valve, high pressure PLSS oxygen disconnect, and a PLSS hose. Two automatic pressure relief and dump valves, one in each hatch, protect the cabin from overpressurization. Two ascent stage tank pressure transducers and a selected oxygen supply transducer, part of the IS, operate in conjunction with OSCPCS.

The OSCPCS stores gaseous oxygen, replenishes the ARS oxygen, and provides refills for the PLSS oxygen storage tank. Before staging, oxygen is supplied from the descent stage oxygen supply. After staging, or if the descent supply is depleted, the ascent stage oxygen tanks supply oxygen to the oxygen control module. The high-pressure assembly in the descent stage, and the oxygen control module in the ascent stage, contain the valves and regulators necessary to control oxygen in the OSCPCS. Two cabin relief and dump valves vent excess cabin pressure.

Descent tank pressure, approximately 2690 psia, is reduced to a level that is compatible with the components of the oxygen control module, approximately 900 psig, by a high-pressure regulator. A series-redundant bypass relief valve protects the descent oxygen tanks against overpressurization. In addition, a series-redundant overboard relief valve protects the oxygen control module against excessive pressure caused by a defective regulator or by flow through the bypass relief valve. If the pressure on the outlet side of the regulator rises to a dangerous

level, the burst diaphragm assembly vents the high-pressure assembly to ambient. A poppet in the burst diaphragm assembly reseats when pressure in the high-pressure assembly is reduced to approximately 1, 000 psig. Descent oxygen flows through the inter-stage disconnect to the oxygen control module and is controlled with the descent oxygen shutoff (DES 02) valve. The interstage disconnect acts as a redundant seal to prevent loss of oxygen overboard after staging.

When ascent stage oxygen is required, the ascent oxygen shutoff (#1 and #2 ASC 02) valves are used to select their respective tank. A mechanical interlock prevents the valves from being opened unless the descent oxygen shutoff valve is closed. The mechanical interlock may be overridden (if the descent oxygen shutoff valve cannot be closed and the ascent oxygen shutoff valves must be opened) by pressing the INTLK OVRD pushbutton on the oxygen control module.

From the oxygen shutoff valves, oxygen is routed to oxygen demand regulators (PRESS REG A and PRESS REG B valves), a low pressure PLSS FILL valve, and a cabin repressurization and emergency oxygen (CABIN REPRESS) valve. The oxygen demand regulators maintain the pressure of the suit circuit at a level consistent with normal requirements. Both regulators are manually controlled with a four-position handle; both are ordinarily set to the same position. The CABIN position is selected during normal pressurized-cabin operations, to provide oxygen at  $4.8 \pm 0.2$  psia. Setting the regulators to EGRESS maintains suit circuit pressure at  $3.8 \pm 0.2$  psia. The DIRECT 02 position provides an unregulated flow of oxygen into the suit circuit. The C LOSE position shuts off all flow through the regulator. In the CABIN and E GRESS positions, the regulator is internally modulated by a reference pressure from the suit circuit. The demand regulators are redundant; either one can fulfill the ARS oxygen requirements.

Descent tank No. 2 pressure, approximately 2 690 psia, is reduced to approximately 1400 psia by a high pressure PLSS regulator. The oxygen is then routed to a high pressure PLSS FILL valve, which connects to the P LSS oxygen tank through a flex hose. A check valve in the P LSS disconnect is opened when the hose is connected. The valve automatically closes when the hose is disconnected.

If cabin pressure drops to 3.7 to 4.45 psia, the cabin pressure switch energizes the CABIN REPRESS valve, and oxygen flows through the valve into the cabin. If cabin pressure builds up to 4.40 to 5.0 psia, the cabin pressure switch deenergizes the valve solenoid, shutting off the oxygen flow. The valve can maintain cabin pressure at 3.5 psia for at least 2 minutes following a 0.5 -inch-diameter puncture of the cabin. It responds to signals from the cabin pressure switch during pressurized-cabin operation. Manual override capabilities are provided.

The two cabin relief and dump valves are manually and pneumatically operated. They prevent excessive cabin pressure and permit deliberate cabin decompression. The valves automatically relieve cabin pressure when the cabin-to-ambient differential reaches 5.4 to 5.8 psid. When set

to AUTO, the valves can be opened manually with their external handle. Each valve can dump cabin pressure from 5.0 to 0.08 psia in 180 seconds without cabin inflow. In addition to relieving positive pressure, the valves relieve a negative cabin pressure condition.

To egress from the LM, the PRESS REG A and PRESS REG B valves are set to EGRESS, turning off the cabin fan and closing the SUIT GAS DIVERTER valve; the CABIN GAS RETURN valve is set to EGRESS, the CABIN REPRESS circuit breaker is opened; and cabin pressure is dumped by setting the cabin relief and dump valve to OPEN. When repressurizing, the CABIN REPRESS circuit breaker is closed, the cabin relief and dump valve is set to AUTO, the PRESS REG A and PRESS REG B valves are set to CABIN, and the CABIN GAS RETURN valve is set to AUTO. The CABIN warning light goes on when the regulators are set to CABIN, it goes off when the cabin reaches the actuation pressure of the cabin pressure switch.

### 3.3 WATER MANAGEMENT SECTION

The WMS stores water for metabolic consumption, evaporative cooling, fire extinguishing, and PLSS water tank refill. It controls the distribution of this stored water and the water reclaimed from the ARS by the water separators. Reclaimed water is used only for evaporative cooling, in the ECS sublimators. Water is stored in two tanks in the descent stage and two identical smaller tanks in the upper midsection of the ascent stage. All water tanks are bladder-type vessels, which are pressurized with nitrogen before launch. The controls for the WMS are grouped together on the water control module located to the right rear of the LM Pilot's station.

Water from the descent stage water supply is fed through a manually operated shutoff (DES I H<sub>2</sub>O) valve, and a check valve, to the P LSS H<sub>2</sub>O DISCONNECT. Water quantity is determined by a pressure transducer with a range of 0- to 60-psia. The output is displayed on the H<sub>2</sub>O QUANTITY indicator (panel 2) after the O<sub>2</sub>/H<sub>2</sub>O QTY MON selector switch 1 is set to DES 1 or DES 2. A nomograph is supplied to convert the readings on the H<sub>2</sub>O QUANTITY indicator into actual percentage of water indications. FULL and 1 6% FULL marks are affixed to the indicator. When the DES H<sub>2</sub>O valve is set to OPEN, high-pressure water is available for drinking, food preparation, PLSS fill, and fire extinguishing.

When the vehicle is staged, the descent interstage water feed line is severed by the interstage umbilical guillotine and water is supplied from the ascent stage water tanks. Water quantity in each ascent stage tank is measured and displayed in the same manner as descent water quantity; the O<sub>2</sub>/H<sub>2</sub>O QTY MON switch is set to ASC 1 or ASC 2, as required. The water quantity measuring system for ascent stage water is similar to the one for descent stage water. The H<sub>2</sub>O QUANTITY indicator marks are affixed for FULL and 95% FULL indications. Water from ascent stage water tank No. 1 is fed to the PLSS H<sub>2</sub>O DISCONNECT through the ascent water (ASC H<sub>2</sub>O) valve for drinking, food preparation, PLSS fill, and fire extinguishing.

Water from the ascent and descent water tanks enters the WATER TANK SELECT valve which consists of two water diverting spools. Setting the valve to DES or ASC determines which supply is on-line.

When using the descent supply, water is routed to the primary pressure regulators by setting the WATER TANK SELECT valve to DES. The water flows through the series primary pressure regulators, which control water discharge pressure in response to suit circuit gas reference pressure, at 0.5 to 1.0 psi above this gas pressure. With the PRI EVAP FLOW #1 valve set to OPEN, the water is routed through a flow limiter to the primary sublimator. The flow limiter limits the water flow rate during the sublimator startup period. Discharge water from the water separator is routed through the secondary spool of the selector valve and joins the water from the primary pressure regulators. Setting the WATER TANK SELECT valve to ASC routes water from the ascent tanks through the primary pressure regulators and, with PRI EVAP FLOW #1 valve set to OPEN, to the primary sublimator. Water flow from the water separators is not changed by selection of the ASC position. If the primary pressure regulators fail an alternative path to the primary sublimator is provided by setting the PRI EVAP FLOW #2 valve to OPEN. Water then flows directly from the ascent water tanks through the secondary pressure regulator, and the PRI EVAP FLOW #2 valve through the flow limiter to the primary sublimator.

Under emergency conditions (failure of the primary HTS loop), water from the ascent tanks is routed through the secondary manifold (secondary pressure regulator) by setting the SEC EVAP FLOW valve to OPEN. This allows for flow of water to the second in series pressure regulator through flow limiters, to the secondary sublimator and the suit circuit sublimator. Discharge water from the water separators is also diverted to these sublimators. To divert water, the WATER TANK SELECT valve is set to SEC.

### 3.4 HEAT TRANSPORT SECTION

The HTS consists of two closed loops (primary and secondary) through which a water-glycol solution is circulated to cool the suit circuits, cabin atmosphere, and electronic equipment. Coolant is continuously circulated through cold plates and cold rails to remove heat from electronic equipment and batteries. For the purpose of clarity, the primary and secondary coolant loops, and the primary and secondary coolant loop cold plates and rails are discussed separately in the following paragraphs. When necessary, the primary loop is also a heat source for suit loop oxygen and a cooling source for the liquid-cooled garment. Heat is removed by absorption and is rejected to space by sublimation.

### 3.4.1 PRIMARY COOLANT LOOP

The primary coolant loop is charged with coolant at GSE fill points and is then sealed. The glycol pumps force the coolant through the loop. The glycol accumulator maintains a constant head of pressure (5.25 to 9 psia, depending on coolant level) at the inlets of the primary loop glycol pumps. Coolant temperature at the inlets is approximately +40 °F. A switch in a low-level sensor trips when only  $10 \pm 5\%$  of coolant volume remains in the accumulator. When tripped, the switch provides a telemetry signal and causes the GLYCOL caution light (panel 2) to go on.

The coolant is routed to the pumps through a filter, which has a bypass to maintain flow if the element becomes clogged. Each pump can provide a minimum of 3.7 pounds per minute of coolant flow at 40 °F with a pressure rise of 30 psid when subjected to a voltage of 28 volts dc. They are started by closing the ECS: GLYCOL PUMP 1 and 2 circuit breakers and setting the GLYCOL selector switch to PUMP 1 or PUMP 2. If the operating pump does not generate a minimum differential pressure ( $\Delta P$ ) of  $7 \pm 2$  psi (or if the pump  $\Delta P$  drops below 3 psid after primary pump operation has been initiated), the  $\Delta P$  switch generates a signal to energize the ECS caution light and the glycol pump (GLYCOL) component caution light. Manually selecting the other pump deenergizes the lights when the onstream pump develops a minimum  $\Delta P$  of 5.0 to 9.0 psid. If both pumps fail, the secondary loop is activated by setting the WATER TANK SELECT valve to SEC, setting the GLYCOL switch to INST (SEC), and closing the ECS: GLYCOL PUMP SEC circuit breaker. Automatic transfer from primary pump No. 1 to primary pump No. 2 is initiated by closing the ECS: GLYCOL PUMP AUTO TRNFR circuit breaker and setting the selector switch to PUMP 1. When transfer is necessary, the ECS caution light and the GLYCOL component caution lights go on, the transfer is accomplished, and the ECS caution light goes off. The GLYCOL component caution light remains on.

If primary loop  $\Delta P$  exceeds 33 psia, the appropriate pump bypass relief valve opens and routes the coolant back to the pump inlet, relieving the pressure. These relief valves start to open at 33 psia, are fully open at a maximum of 39 psia, and reseat at a minimum of 32 psia. Check valves prevent coolant from feeding back through an inoperative primary pump.

Part of the coolant leaving the recirculation assembly flows to the suit circuit heat exchanger to cool the suit circuit gas of the ARS. The remainder of the coolant flows to electronic equipment mounted on cold plates. The flow paths then converge and the coolant is directed to the liquid-cooled garment water-glycol heat exchanger to cool suit water as required. The coolant then flows through the aft equipment bay cold rails. A portion of the warmer coolant flow returning from the aft equipment bay cold rails can be diverted to the suit circuit regenerative heat exchanger through the SUIT TEMP valve to increase suit inlet gas temperature. The diverted flow returning from the heat exchanger, combined with the bypassed coolant, is routed to the primary sublimator.

The sublimator decreases the temperature of the coolant by rejecting heat to space through the sublimation of water, followed by the venting of the generated steam through an overboard

duct. A thrust deflector located above the duct exit port diffuses the exhaust steam, thereby decreasing the thrust effect on the vehicle. Pressure regulators in the sublimator water feed line maintain water outlet pressure at 0.4 to 1.0 psid above the referenced suit circuit pressure. This regulated water outlet pressure will be above 4.0 psia and will normally be below 7.0 psia. During the suit pressure integrity check, the regulated water outlet pressure can be as high as 10 psia for a short time. The sublimator inlet and outlet temperatures are sensed by temperature transducers, which provide telemetry signals. Coolant from the sublimator flows through the ascent and descent battery cold rails, then returns to the recirculation assembly.

Self-sealing disconnects upstream and downstream of the glycol pumps permit servicing and operation of the HTS by GSE. Interstage disconnects are installed in coolant lines that connect to the descent stage. Before staging, coolant flows through the ascent and descent stage battery cold rails. After staging, the interstage disconnects separate, the lines are sealed by the interstage disconnects and the full coolant flow enters the ascent stage battery cold rails.

### 3.4.2 SECONDARY COOLANT LOOP

The secondary (emergency) coolant loop provides thermal control for those electronic assemblies and batteries whose performance is necessary to effect a safe return to the CSM. Cooling is provided by the secondary sublimator. The procedure for starting up the secondary loop is discussed in paragraph 3.4.1.

The secondary loop is also charged with coolant at GSE fill points and is sealed. A secondary accumulator identical with the primary accumulator maintains a positive secondary pump inlet pressure. As with the primary pump, the secondary pump is protected by a filter which has a bypass to maintain flow if the element becomes clogged. A pump bypass relief valve relieves excessive pressure by routing coolant back to the pump inlet. A check valve at the discharge side of the glycol pump prevents coolant flow from bypassing the HTS during GSE operation. The coolant from the pump passes through the check valve, to the secondary passage of the cold plates and cold rails of the electronics and batteries section. Waste heat is absorbed by the coolant. The warm coolant then flows to the secondary sublimator.

The secondary sublimator operates in the same manner as the primary sublimator in the primary coolant loop. Water for the sublimator is provided when the SEC EVAP FLOW valve is set to OPEN. The coolant returns to the pump for recirculation.

### 3.4.3 COLD PLATES AND RAILS

Equipment essential for mission abort is mounted on cold plates and rails that have two independent coolant passages, one for the primary loop and one for the secondary loop.

**Primary Coolant Loop Cold Plates and Rails.** The cold plates and rails in the primary coolant loop are arranged in three groups: upstream electronics, aft equipment bay, and batteries.

Coolant from the recirculation assembly flows into parallel paths that serve the upstream electronics group. In this group, the data storage equipment assembly (DSEA) is cooled by cold rails; the remainder of the group, by cold plates. The cold plates are located in the pressurized and unpressurized areas of the LM. The flow rates through the parallel paths are controlled by flow restrictors, installed downstream of each cold plate group. The first upstream electronics flow path cools the suit circuit heat exchanger. The second flow path cools five cold plates mounted on the pressurized side of the equipment tunnel back wall. The third path serves the integrally cooled inertial measurement unit (IMU) and the rate gyro assembly (RGA) cold plate, both located in the unpressurized area (on the navigation base). All the plates for the fourth path are in the unpressurized area above the cabin; the ASA is on the navigation base. The fifth path serves the tracking light electronics (TLE), gimbal angle sequencing transformation assembly (GASTA), lighting control assembly (LCA), and data storage electronics assembly (DSEA) plates : one in the unpressurized area in front of the cabin, a second one in the control and display panel area, a third one below the cabin floor, and another one on the left wall of the cabin.

The aft equipment bay is cooled by eight cold rails; the flow is in parallel. The batteries are cooled by cold rails. The ascent batteries are in the center section of the aft equipment bay. The descent batteries are on the -Z- bulkhead. During the descent phase, the coolant flow is split between the descent batteries and the ascent batteries; the ascent batteries are not used during this time. When the stages are separated, quick-disconnect break the coolant lines and seal the ends; all coolant then flows through the ascent battery cold rails.

**Secondary Coolant Loop Cold Plates and Rails.** The secondary coolant loop is for emergency use. Only cold plates and cold rails that have two independent passages (one for the primary loop and one for the secondary loop) are served by this loop.

In the upstream electronics area, the secondary coolant flow is split between three cold plates (RGA, ASA, and TLE) in parallel. The flow rate is controlled by flow restrictors downstream of the TLE and RGA. After these three plates, the secondary loop cools the ascent battery cold rails and the aft equipment bay cold rails in a series -parallel arrangement. The coolant first flows through three ascent battery cold rails in parallel, then through eight aft equipment bay cold rails in parallel.

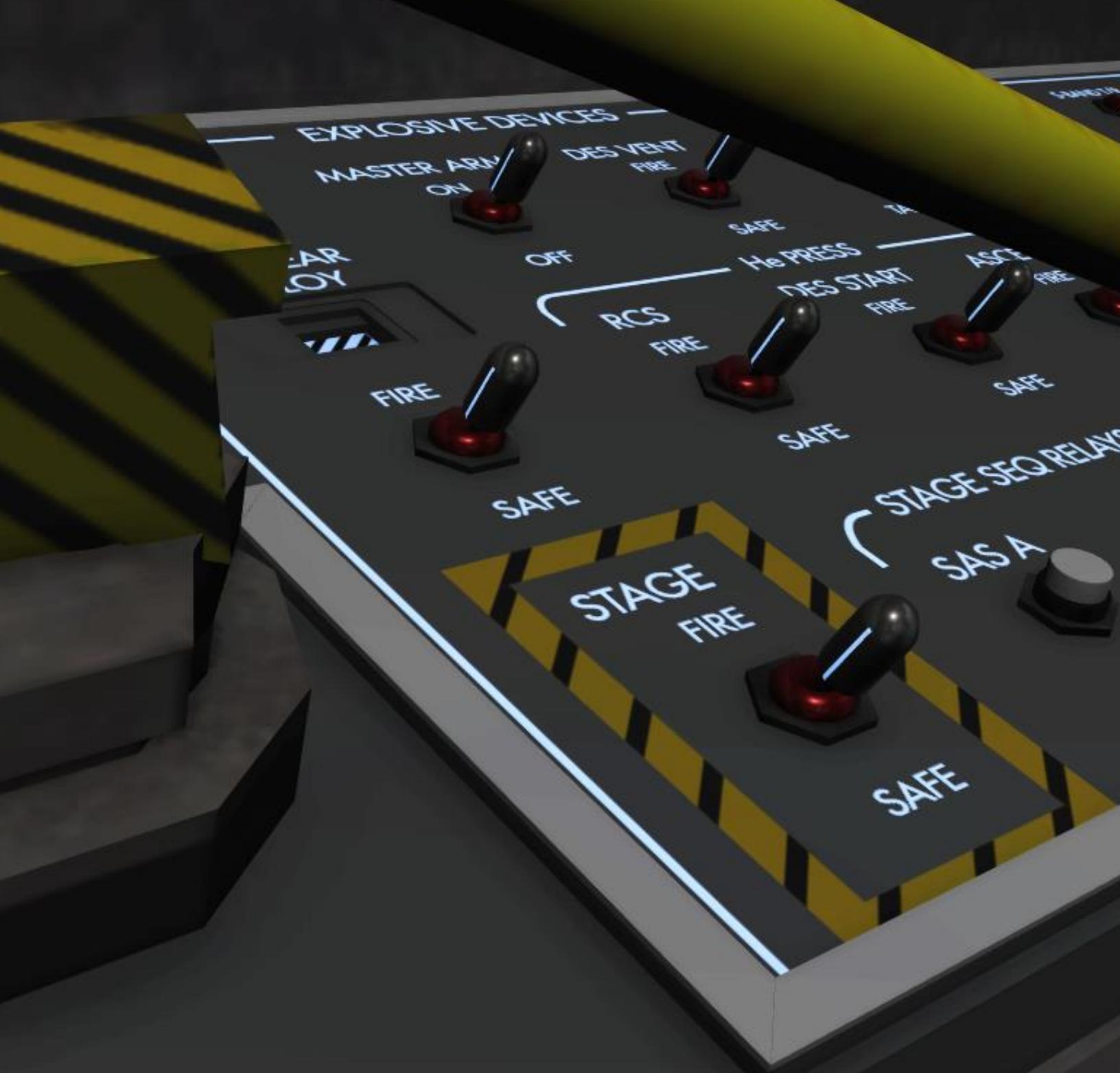
# LUNAR MODULE

FLIGHT MANUAL

R E N T R Y

AN ORBITAL SIMULATOR

# IX. EXPLOSIVE DEVICES



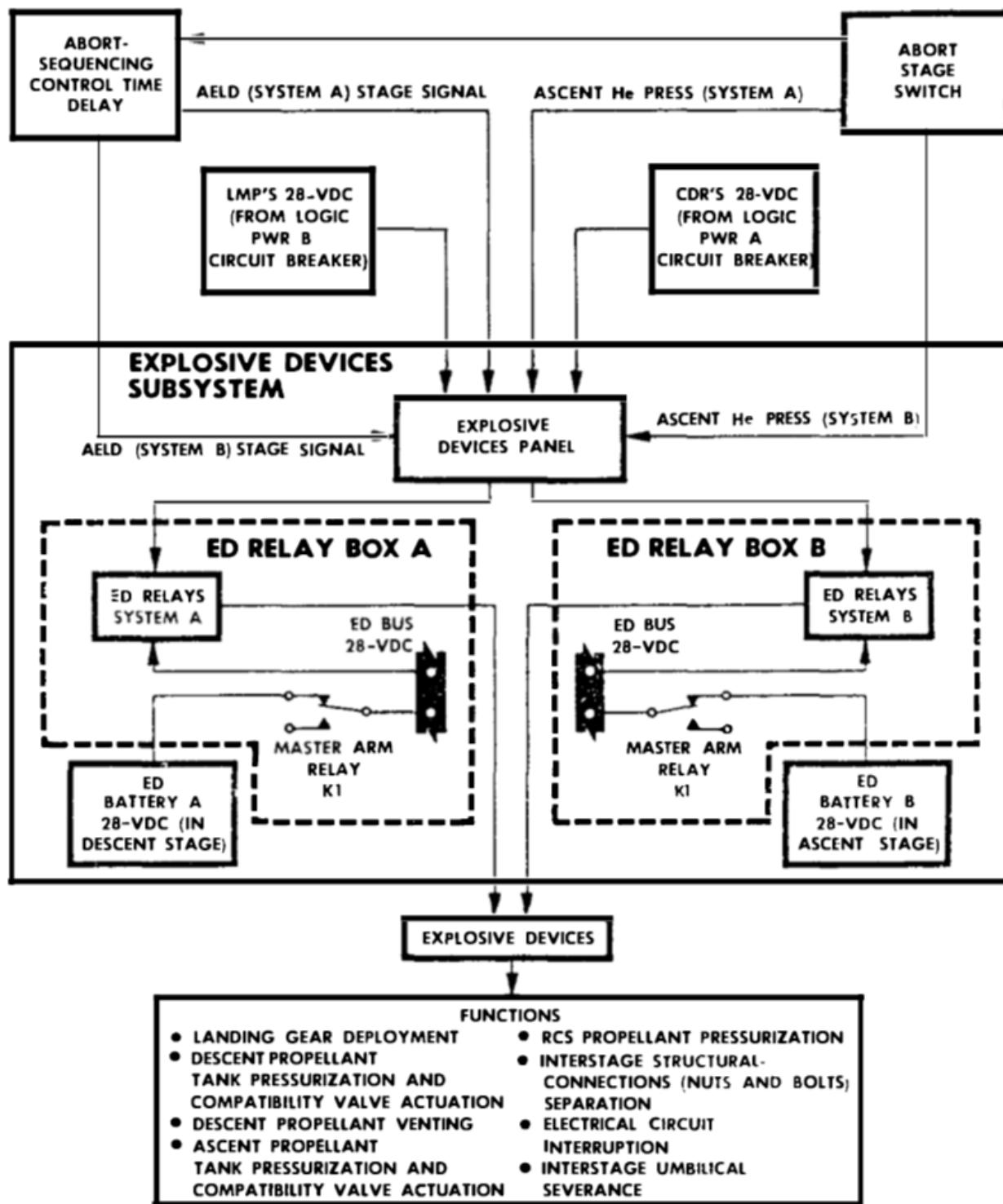
# IX. EXPLOSIVE DEVICES

## 1. GENERAL

The Explosive Devices Subsystem (EDS) permits the astronauts to operate or enable various LM equipment, using explosive devices. The EDS controls deployment of the landing gear; pressurization of the descent, ascent, and reaction control propellant tanks; venting of descent propellant tanks; and separation of the ascent and descent stages.

Electrical power to activate the components in both systems is supplied through circuit breakers on the Commander's circuit breaker panel (11) and the LM Pilot's circuit breaker panel (16). Switches on the EXPLOSIVE DEVICES portion of panel 8 route this power to relays in the ED relay boxes. The relay contacts route ED battery current to explosive cartridges located in LM subsystems. The explosive devices actuated by the cartridges perform the EDS functions.

There are two separate systems (A and B) in the EDS. The systems operate in parallel and provide complete redundant circuitry; each system has its own 37.1-volt d-e battery, relays, time-delay circuits, fuse-resistors, buses, and explosive cartridges. Two separate cartridges are provided for each EDS function. Each cartridge uses power from a separate ED bus and can perform its task without the other cartridge.



EDS relays are actuated by 28 volts de received through the Commander's and LM Pilot's buses of the Electrical Power Subsystem (EPS). Contacts of these relays route ED battery current to associated explosive devices in the Main Propulsion Subsystem (MPS) and the Reaction Control

Subsystem (RCS) and to explosive devices that enable landing gear deployment, interstage structural connection separation, electrical circuit interruption, and interstage umbilical severance. In the MPS, explosive devices enable descent propellant tank pressurization and venting, and ascent propellant tank pressurization. In the RCS, explosive devices enable RCS propellant tank pressurization.

The EDS receives two commands from the control electronics section (CES) of the Guidance, Navigation, and Control Subsystem (GN&CS). After the descent engine is fired, a delayed descent engine-on command is received from the CES. This command actuates the ED relay associated with the explosive device that enables descent propellant tank pressurization (with supercritical helium). An automatic stage command is received from the CES when the ascent engine is fired. This command I actuates the ED staging sequence relays.

ED relay status and malfunction signals are routed to the Instrumentation Subsystem (IS), which processes them for telemetering to MSFN. If the master arm relay or the staging sequence relays malfunction, a signal is routed to the failure-detection circuits in the caution and warning electronics assembly (CWEA) of the IS, which causes the ED RELAYS caution light (panel 2) and the appropriate STAGE SEQ RELAYS SYS A or SYS B component caution light (panel 8) to go on. Landing gear deployment is monitored by sensors in the landing gear assemblies. A composite signal from these sensors is sent to the IS, where it is processed for telemetering to MSFN. The signal causes a gray LDG GEAR DEPLOY talkback display (panel 8).

## 2. FUNCTIONAL DESCRIPTION

The redundant circuits and components of the EDS operate simultaneously, as two independent systems (A and B). Either system can perform the EDS functions. Each function is completed before initiation of the next function. The EDS functions are performed in the following sequence:

- Landing gear deployment
- Reaction Control Subsystem (RCS) propellant tank pressurization
- Descent propellant tank prepressurization (ambient helium)
- Descent propellant tank pressurization (supercritical helium)
- Descent propellant tank venting
- Ascent propellant tank pressurization
- Interstage nut-and-bolt separation and ascent stage deadfacing (occur simultaneously)
- Interstage umbilical severance

EDS systems A and B are operated from the EXPLOSIVE DEVICES portion of panel 8. Power (28 volts de) from the EPS, to actuate the relays in the system A and system B relay boxes, is routed through the ED: LOGIC PWR A circuit breaker (panel 11) and the ED: LOGIC PWR B circuit breaker (panel 16), respectively. Power for the cartridges is supplied by the ED batteries. When

the MASTER ARM switch is set to ON, each battery supplies power to an identical redundant bus in the ED relay boxes, arming the firing circuits and enabling all explosive devices of the LM. Firing of any explosive device is controlled by its respective switch on panel 8, except that the descent propellant supercritical helium valves are fired on command from the CES (refer to paragraph 2. 8. 3. 4). Automatic initiation of the staging sequence can be generated by the CES, and EDS arming and ascent propellant tank pressurization can be initiated by pressing the ABORT STAGE pushbutton (panel 1).

Inadvertent operation is indicated by a master alarm, an ED RELAYS caution light, and the appropriate STAGE SEQ RELAYS component caution light. When the MASTER ARM switch is set to ON, both component caution lights, SYS A and SYS B, go on to indicate that both ED buses are armed. MSFN is provided with a telemetry signal for monitoring the status of the master arm and stage sequence relays. The RESET switch on Panel 8 will reset the component caution lights after arming the ED. If there are any issues, they will illuminate again. The affected EDS system should be deactivated by opening the ED: LOGIC PWR A or ED: LOGIC PWR B circuit breaker. A subsequent EDS failure may jeopardize crew safety.

### 3. OPERATIONAL

#### 3.1. LANDING GEAR DEPLOYMENT

The landing gear, stowed in the retracted position at launch, remains retracted until just prior to LM-CSM separation. The LDG GEAR DEPLOY switch is used to deploy the landing gear assemblies.

Once triggered, the four legs will extend and lock into a deployed position. A LDG GEAR DEPLOY talkback will turn gray when the operation is complete, and if the LDG GEAR FLAG circuit breaker on panel 11 is closed. Opening the circuit breaker will make the talkback barber poled.

#### 3.2. DESCENT PROPELLANT TANK PREPRESSURIZATION

An ambient helium isolation valve is used when pre-pressurizing the descent propellant tanks. The valves are normally closed. Setting the MASTER ARM switch to ON arms the ED buses. Momentarily holding the DES START He PRESS switch to FIRE routes power from the EPS to both EDS systems, and to the cartridges in the ambient helium isolation valve, opening the valve and allowing ambient helium to flow to the descent propellant tank **compatibility valves**.

Momentarily holding the DES PRPLNT ISOL VLV switch to FIRE routes EPS power to the ED battery and will open the **compatibility valves**. Ambient helium then flows freely to the descent engine fuel and oxidizer tanks, pressurizing them. The increase in propellant tank pressure is observed on the FUEL and OXID PRESS indicators (panel 1); the drop in ambient helium tank pressure on the HELIUM indicator (panel 1).

Ambient helium pre-pressurization permits fuel to flow through the fuel/helium heat exchanger before cryogenic helium flow through the helium/helium heat exchanger is initiated.

With the MASTER ARM switch set to ON and the descent engine fired, the ED relay boxes receive a signal from the CES after a 1.3-second delay. ED battery power is then routed to the supercritical helium isolation valve, opening the valve. Helium from the cryogenic helium storage vessel pressurizes the descent propellant tanks in place of the expended ambient helium. Descent helium pressure malfunctions light the DES REG warning light (panel 1).

### 3.3. STAGING

Stage separation involves simultaneous operation of the circuit interrupters (deadfacing) and separation of the interstage structural connections (nuts and bolts), then umbilical severance (cable cutting). This is accomplished by cartridges fired by operation of the STAGE switch or a signal from the CES when the ascent engine is fired. The signal from the CES bypasses the STAGE switch, which is not normally used.

Setting the MASTER ARM switch to ON arms the ED buses; setting the STAGE switch to FIRE, or the signal from the CES when the ascent engine is fired, applies 28 volts dc from EPS systems A and B to the ED stage-sequencing circuits (RC time-delay circuits) to the ED buses. This energizes the separation of nuts and bolts, and deadfacing, and energizes a time delay. A time delay is necessary to ensure removal of all power from the interstage umbilical before it is cut. After approximately a 15- to 20-millisecond delay (during which the nuts and bolts separate), guillotine detonators are fired, completing the stage separation.

The staging sequence can also be initiated by pressing the ABORT STAGE pushbutton. This arms the ED buses (bypassing the MASTER ARM switch) and initiates an automatic ascent engine fire sequence in the CES. The CES fire signal initiates the ED staging sequence. Contacts of K3 ensure that the deadface relay has removed electrical power from the umbilical before the guillotine detonators are fired.

### 3.4. ASCENT PROPELLANT TANK PRESSURIZATION

Two helium isolation valves, each with two cartridges and two sets of parallel redundant fuel and oxidizer compatibility valves are used when pressurizing the ascent propellant tanks. One fuel compatibility valve and one oxidizer compatibility valve use two cartridges; the other two

compatibility valves a single cartridge each. The ASC He SEL switch provides the astronauts with the option of using either one or both helium tanks to pressurize the system.

The valves are normally closed; they open when the cartridges are fired, depending on the setting of the ASC He SEL switch. In addition, the control relays for the helium isolation valves and the compatibility valves can be energized with a direct signal from the ABORT STAGE pushbutton, which bypasses the ASCENT He PRESS switch. Normally, the ASC He SEL switch is set to BOTH so that both helium tanks are used for ascent propellant tank pressurization.

Setting the MASTER ARM switch to ON, arms the ED buses. Momentarily holding the ASCENT He PRESS switch to FIRE, routes 28 volts dc from the EPS to the ED battery power to the fuel and oxidizer compatibility valves cartridges. With the ASC He SEL switch in TANK 1 position, both ED systems are energized, routing ED battery power to the helium tank No. 1 isolation valves. With the switch in the TANK 2 position, both ED systems are energized, routing ED battery power to the isolation valves of helium tank No. 2. In the BOTH position, both ED systems are energized, firing the valves in both tanks. The astronauts are alerted to ascent helium pressure malfunctions by the ASC PRESS warning light (panel 1).

### 3.5. RCS PROPELLANT TANK PRESSRIZAION

The RCS has two helium fuel, and oxidizer tanks. Each helium tank pressurizes one fuel tank and one oxidizer tank. Two parallel helium isolation valves are used in conjunction with each helium tank when pressurizing the RCS propellant tanks. Each valve has one cartridge, which is fired by one of the EDS systems.

The valves are normally closed; their firing circuits are controlled by relays actuated by the RCS He PRESS switch. Setting the MASTER ARM switch to ON arms the ED buses. Momentarily holding the RCS He PRESS switch to FIRE routes 28 volts de from EPS systems A and B to the cartridges in the helium isolation valves and the ED buses. Box A route ED bus voltage to helium isolation valves in RCS systems A and B. The other helium isolation valve in RCS systems A and B receives its ED bus voltage through box B. Opening either valve in system A and B permits helium to pressurize the RCS fuel and oxidizer tanks. The RCS A REG and RCS B REG warning lights and the RCS caution light alert the astronauts to malfunctions of RCS helium line pressure and tank pressure.

### 3.6. RCS PROPELLANT TANK PRESSRIZAION

The descent fuel and oxidizer helium pressurization venting systems each contain an ED-controlled valve in series with a normally open latching valve. The latching valves are normally open; activating the ED valves vents the systems. Setting the MASTER ARM switch to ON arms the ED buses. Momentarily setting the DES VENT switch to FIRE routes EPS power to the

cartridges of the vent valves, opening them. The helium vents through the normally open fuel and oxidizer vent valves. Setting the FUEL VENT and OXID vent switches (panel 8) to CLOSE, closes the vent valves and causes the talkbacks to change from gray to barber-pole displays.

# X. INSTRUMENTATION



## X. INSTRUMENTATION

### 1. GENERAL

The

# XI. LIGHTING



## XI. LIGHTING

### 1. GENERAL

The

## XII. EQUIPMENT



## XII. EQUIPMENT

### 1. GENERAL

The

# XIII. RADAR



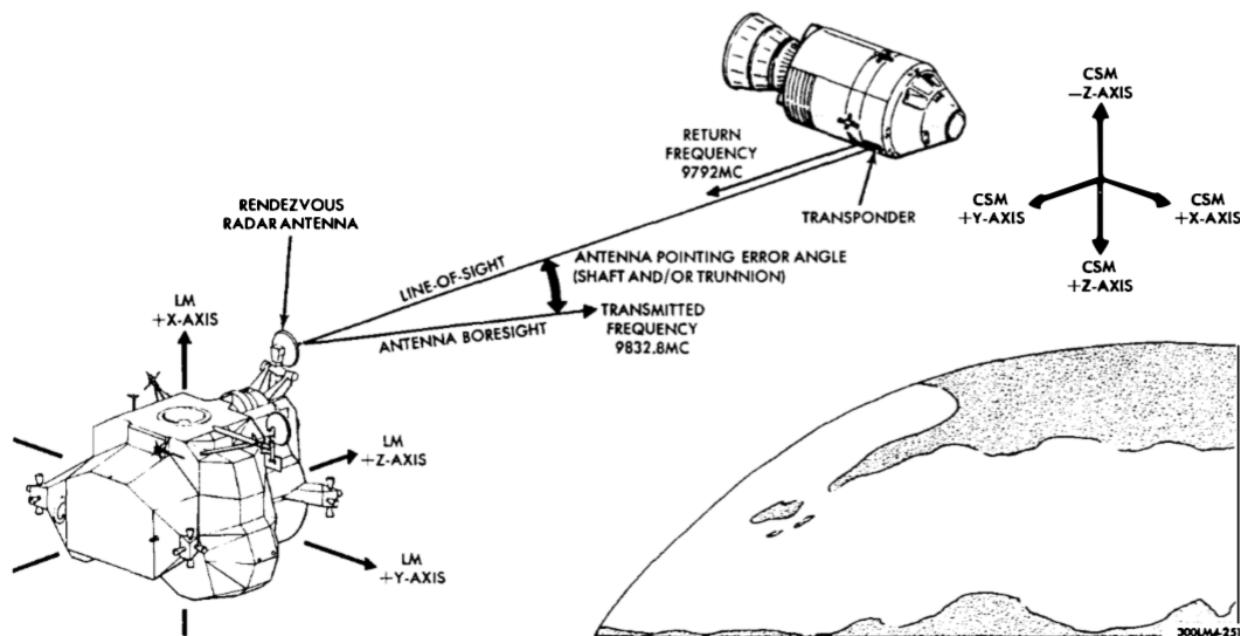
# XIII. RADAR

## 1. GENERAL

During the landing phase and subsequent rendezvous phase, the LM uses radar navigational techniques to determine distance and velocity. Each phase uses a radar designed specifically for that phase (rendezvous radar (RR), landing radar (LR)). Both radars inform the astronaut and the computer concerning position and velocity relative to acquired target. During lunar landing, the target is the surface of the moon; during rendezvous, the target is the Command Module.

### 1.1. RENDEZVOUS RADAR

The Rendezvous Radar (RR), operated in conjunction with a RR transponder in the CSM, is used to acquire and track the CSM before and during rendezvous and docking.



When the primary navigation and guidance section (PGNS) is in control (normal operation), the RR provides the PGNS and cabin displays with line-of-sight (LOS) range and range rate data, and LOS angle and angle rate data with respect to the CSM. If PGNS fails, the astronauts furnish these data to the abort guidance section (AGS). The RR LOS is defined as the line-of-sight direction between the CSM target and the RR antenna. The RR also tracks the CSM target during the coasting and descent engine burn phases of the mission to supply tracking data for any required abort maneuver.

## 1.2. LANDING RADAR

The Landing Radar (LR) provides the PGNS and the astronauts with slant range and velocity data for control of descent to the lunar surface. Based on these data, the PGNS calculates control signals for LM rate of descent, hovering at low altitudes, and soft landing at the selected lunar site. The LR is activated at approximately 50,000 feet above the lunar surface and remains activated until touchdown. In the nominal descent trajectory, slant range data are available to the PGNS at approximately 40,000 feet (no less than 25,000 feet); velocity data, at approximately 35,000 feet (no less than 18,000 feet).

## 2. RR - FUNCTIONAL DESCRIPTION

The RR provides LOS range, range rate, and antenna angular data to the LM guidance computer (LGC) and to the LM cabin displays. By entry via the data entry and display assembly (DEDA), the abort electronics assembly (AEA) is supplied with range and range rate data for a computer-controlled (LGC or AEA) rendezvous of the LM and CSM.

The RR, when search-sweeping or tracking and locked on to the RR transponder in the CSM, is a continuous-wave coherent system, which uses phase-lock techniques in the radar and in the transponder. When operating with the LGC, the RR derives range, range rate, and shaft and trunnion angle data and routes these data to the LGC. The AEA state vector is automatically updated by range and range rate data from the LGC or by manually inserting this data via the DEDA.

Range is determined by measuring the time delay between the transmitted signal and the received signal. Range rate is determined by measuring the two-way Doppler shift of the carrier signal received from the transponder. The RR has an unambiguous range capability from 80 feet to 400 nm and a range rate capability from -4900 to +4900 fps.

Range and range rate data are accepted by the LGC only upon program demand. The CDU continuously receives the shaft and trunnion angles, converts them to digital form, and sends these data to the LGC upon program demand. In addition to the RR outputs to the LGC, range and range rate are displayed in the cabin. The shaft and trunnion angles and angle rates are indicated as elevation and azimuth angles by the FDAO and as pitch and yaw rates by the X-pointer indicator on panels 1 and 2. The azimuth indication is derived from the trunnion angle; the elevation indication, from the shaft angles. The angle rates are derived from the gyro torquer currents.

The RR consists of an antenna assembly and an electronics assembly. In free flight, the LGC computes the LOS angle to the CSM and then points the LM +Z-axis to the CSM target. The LGC designates the RR antenna angles, pointing the RR antenna to the target, so that the transmitter signal can be acquired by the transponder.

## 2.1 MODES OF OPERATION

The RR operates in three modes: automatic tracking, slew (manual), and LGC control. The RENDEZVOUS RADAR selector switch (panel 3) is used to select the mode of operation.



Automatic Tracking Mode: The automatic tracking mode is selected by setting the RENDEZVOUS RADAR selector switch to AUTO TRACK. This mode enables the RR to track the CSM automatically after it has been acquired; tracking is independent of LGC control. When this mode is selected, tracking is maintained by comparing the received signals from the shaft and trunnion channels with the sum channel signal. The resultant error signals drive the antenna, thus maintaining track.

Slew Mode: The slew (manual) mode is selected by setting the RENDEZVOUS RADAR selector switch to SLEW. This mode enables an astronaut to slew the antenna manually to a position for acquisition of the CSM. The controls to slew the Antenna can be change in the Input Settings menu, and is set by default to CTRL+ Arrow Keys.

LGC Control Mode: The LGC control mode is selected by setting the RENDEZVOUS RADAR selector switch to LGC. In this mode, the LGC automatically controls antenna positioning, initiates automatic tracking once the CSM transponder is acquired, and controls change in

antenna orientation. When this mode is selected, the RR sends a RR power-on/LGC discrete to the LGC. Automatic control of RR search and acquisition is provided by the PGNS, which transmits LGC-derived slewing commands to position the RR antenna.

## 2.2 OPERATING THE RR

### 2.2.1 – LGC MODE

The LGC positions the radar antenna at an angle such that its transmitted continuous-wave radiation can be received at the transponder. When the antenna is within  $0.5^\circ$  of the LGC-estimated direction to the transponder (a total maximum antenna pointing error of  $2.5^\circ$ ), the LGC provides an autotrack-enable signal that initiates the tracking function. The antenna boresight may be as much as  $2.5^\circ$  off the target LOS due to cumulative mechanical, electrical, and LGC programming errors. After the frequency tracker has locked up with the transponder carrier frequency, the RR angle-tracks the transponder target. During tracking, the RR supplies shaft and trunnion angles to the LGC via the CDU analog-to-digital converters.



The shaft and trunnion resolver outputs are routed through the SHFT/TRUN switch ([3] on panel 1) and the RATE/ERR MON [1] switch to the FDAO error needles [4]. With the RATE/ERR MON switch set to RNDZ RADAR, the shaft angle signals are applied to the pitch error needle and the trunnion angle signals are applied to the yaw error needle. The SHFT/TRUN switch provides the desired scaling factor ( $\pm 5^\circ$  or  $\pm 50^\circ$ ) for the resolver outputs.

With the RATE/ERR MON switch set to RNDZ RADAR, the shaft and trunnion rates, corresponding to RR bore sight angular rates, are routed through the X-POINTER SCALE switch

(panels 1 and 3) for display on the X-pointer indicator (panels 1 and 2): the ELEV RT and AZ RT placarding on the indicators are illuminated. The setting of each X-POINTER SCALE switch determines the scaling and whether the X.1 or X1.0 placarding on the associated indicator is illuminated.



The RR angle-tracking loop cannot be completed until the transponder is phase locked to the RR signal and the RR receiver is phase locked to the transponder signal. After range and range rate lock-on are completed, the RR issues a data-good discrete to the LGC. The NO TRACK light (panel 3) is extinguished. Range and range rate data are now available to the LGC and to cabin displays.

If the target is lost during LGC mode tracking, the NO TRACK light goes on, indicating that the RR is not locked on to the CSM. A no-track signal supplied to the caution and warning electronics assembly (CWEA) causes the RNDZ RDR caution light (panel 2) to go on. The PGNS takes control and positions the antenna to regain acquisition and automatic tracking. The acquisition sweep generator in the frequency tracker begins to sweep and continues sweeping until acquisition is achieved and angle tracking is restored. When ranging circuits in the electronics assembly achieve lock-on, the data-good discrete is restored, allowing range and range rate from the electronics assembly to be accepted by the LGC and routed through the RNG/ALT MON switch (set to RNG/RNG RT) to the RANGE [5] and RANGE RATE [6] tape indicators (panel 1).

### 2.2.1 – SLEW & AUTO TRACK MODE

With the RENDEZVOUS RADAR selector switch set to SLEW, the RR is operated in the slew mode, using the SLEW and SLEW RATE switches (panel 3). With the SLEW RATE switch set to HI, the slew rate is a nominal 7° per second; with the switch set to LO, the slew rate is a nominal 1.3° per second. The astronaut can slew the antenna in the desired direction by holding the SLEW buttons on the KEYBOARD (Ctrl+Arrow Keys by default) and using the proper slew rates for the periods of time required. When automatic tracking is desired, the RR must be placed in the automatic-tracking mode or LGC mode. You usually slew it until the NO TRACK light is extinguished. When this happens, set the selector to AUTO TRACK.

## 2.3 ELECTRICAL POWER

The RR is powered through the CDR DC bus, by inserting the PGNS: RNDZ RDR circuit breaker.

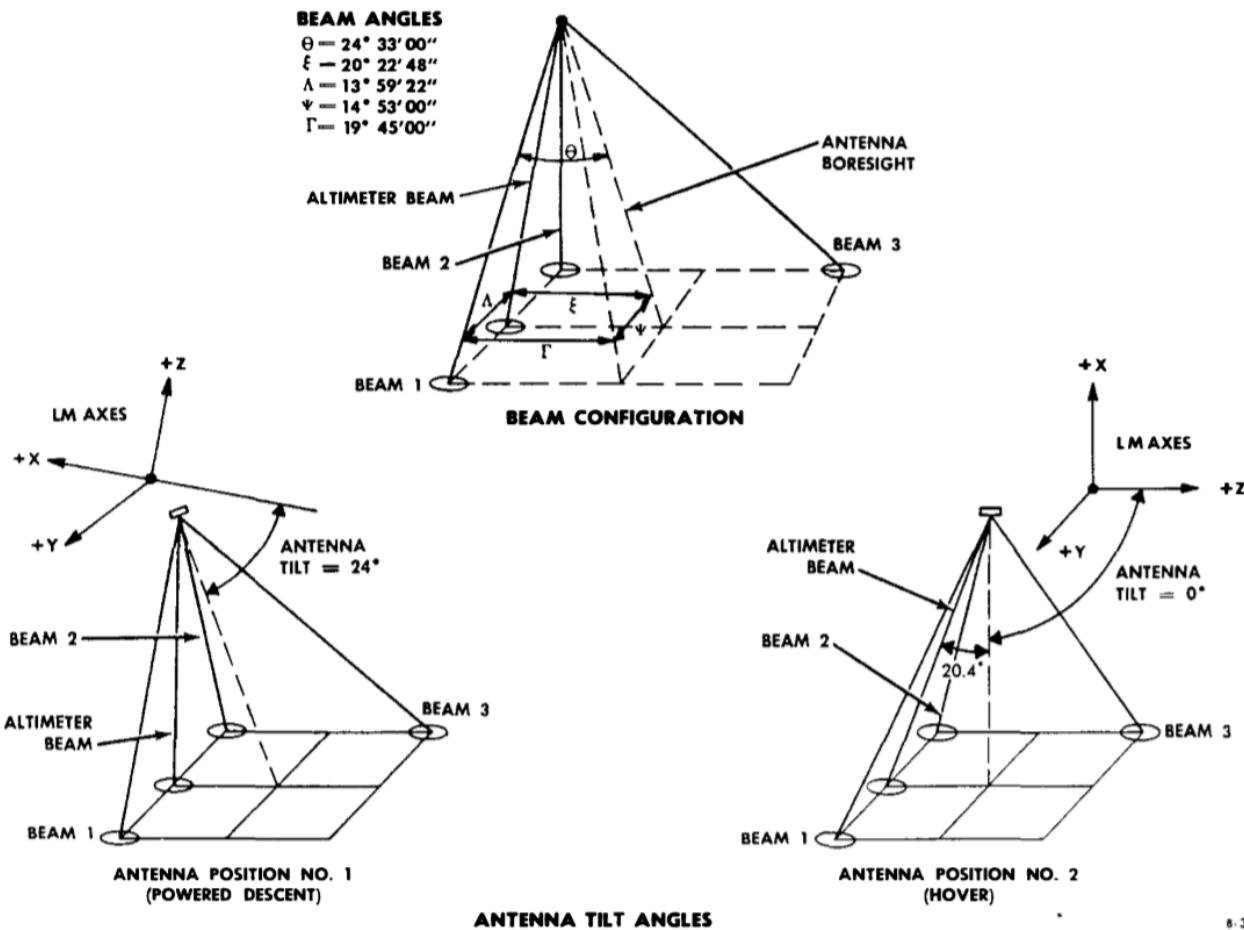


## 3. LR - FUNCTIONAL DESCRIPTION

The LR senses the velocity and slant range of the LM relative to the lunar surface by means of a three-beam Doppler velocity sensor and a radar altimeter. Coordinate velocities are calculated from the measured Doppler shift along three beams of microwave energy, as is the Doppler shift of the radar altimeter beam. The radar altimeter is Doppler corrected and the output slant range is used by the LGC to derive LM altitude from the lunar surface. Slant range is calculated from the frequency deviation difference between the altimeter transmitted and received signals, resulting from the frequency modulation of the transmitted signal and the time delay between the transmitted and received signals. Velocity and range data are made available to the LGC. Forward and lateral velocity data and the slant range and vertical velocity (range rate) data is displayed on the LM panels.

The LR, located in the LM descent stage, consists of an antenna assembly and an electronics assembly. The antenna assembly contains two interlaced phased arrays which transmit four velocity and two altimeter beams. Four broadside arrays receive the reflected energy of the three desired velocity beams and the one desired altimeter beam; there is one broadside array for each desired received radar beam. The electronics assembly searches, tracks, processes,

converts, and scales the Doppler and CW /FM returns, which provide the velocity and slant range information for the LGC and the LM controls and displays.



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### 3.1 OPERATING THE LR

Using the LM controls and indicators, the astronauts can monitor the LM velocities and LM altitude, and temperature; apply power to heat the antenna and power to the LR; and place the antenna in position No. 1 (descent) or 2 (hover).

**THE HEATERS:** LDG RDR circuit breaker (panel 11) is closed at the launch pad and left closed for the mission duration. This circuit breaker activates an antenna temperature control, which applies power to antenna heaters to protect antenna components against the spatial cold when power is not applied to the LR.

LR antenna temperature is monitored by setting the TEMP MONITOR selector switch to LDG and observing the TEMP indicator for an indication within the operating temperature range of  $+50^\circ$  to  $+150^\circ\text{F}$ .

LR slant range data is available to the LGC and on the ALT indicator at or above 25,000 feet. At or above 18,000 feet, the forward (Vza) and lateral (Vxa) velocities are available to the LGC and on the X-pointer indicators.

When the LM reaches an altitude of 500 feet or less, the astronauts initiate manual control (P66) for the remainder of the descent to the lunar surface. The LR may drop lock during this phase of the mission, due to a zero Doppler. At a preselected minimum altitude (usually 50 feet), LR data to the LGC is inhibited. During this critical phase of the mission, the altitude and velocity data on the LM displays is inertially derived by the LGC from the state vector which has been updated by LR data.

LR raw data can be displayed as a cross-check and/or if a problem occurs in the PGNS coordinate transformation. When touchdown is accomplished, the LR is deactivated by opening all LR circuit breakers.

### 3.1.1 – ANTENNA POSITIONS

The LR antenna is oriented to position No. 1 (descent) or 2 (hover). Position No. 1 orients the antenna boresight 24° with respect to the LM X-axis. In position No. 2, the antenna boresight is parallel to the LM X-axis. The antenna pivots on its Y-axis which is at a 6° angle with respect to the LM Y -axis. The angle between the antenna bore sight and the altimeter beam is 20.4°.

In the nominal mission, the LDG ANT switch (panel 3) is set to AUTO and the LGC controls the antenna position. For test or override purposes, the LR antenna position may be controlled manually. Antenna position No. 1 is selected by setting the LDG ANT switch to DES; antenna position No. 2, with the switch set to HOVER. With the switch set to AUTO, the LGC commands the antenna to position No. 2 at the LM pitchover point on the descent trajectory, at about 7800 feet.

## XIV. ABORT GUIDANCE



## XIII. AGS

### 1. GENERAL

The Abort Guidance System is not implemented in Reentry.